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FACTORS INFLUENCING SOLAR ELECTRIC PROPULSION VEHICLE PAYLOAD DELIVERY FOR OUTER PLANET MISSIONS

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Systems analyses was performed for missions utilizing solar electric propulsion systems to deliver payloads to outer-planet destinations. A range of mission and systems factors and their affect on the delivery capability of the solar electric propulsion system was examined. The effect of varying the destination, the trip time, the launch vehicle, and gravity-assist boundary conditions was investigated. In addition, the affects of selecting propulsion system and power systems characteristics (including primary array power variation, number of thrusters, thruster throttling mode, and thruster Isp) on delivered payload was examined.

Introduction

An array of analysis factors governs prediction of Solar Electric Propulsion System[1,2] (SEPS) payload delivery to an outer-planetary destination. This paper summarizes these factors into two primary groups; viz. mission analysis factors and systems analysis factors. The ultimate value these mission and system factors take closely associates with science goals stemming from the space science community. The engineering community then couches these science goals within practical engineering requirements as mission goals. As mission and systems analysis efforts unfold, an optimized set of vehicle performance metrics (cost, reliability, and payload to destination) evolve into a set of vehicle requirements. Primarily, this paper focuses on the payload to destination metric, while not addressing critical cost and system/subsystem reliability metrics. Assumed for the purposes of this paper, the science and mission goals have been determined, and these mission/science goals will not be further examined. In summary, this paper provides a parametric survey of a set of mission/system factors that affect the payload metric for SEPS vehicles utilized for unmanned outer planet science missions.

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Primary Factors Affecting Payload Delivery to Destination

Mission Factors. The mission analysis factors examined in this paper include payload to destination, launch vehicle to place the SEPS vehicle and payload on the initial earth escape trajectory, and interplanetary trajectory characteristics (including interplanetary transfer time, transfer geometry characteristics, and gravity assist flyby opportunity). Each of these factors plays a significant role in the resultant value of the optimized SEPS performance metrics. In particular, the payload that will be delivered to a destination directly relates to choices in destination, launch vehicle, and trajectory class.

Destination. The destination of the payload relates to mission difficulty by possessing inherent distance above or below the earth reference point in the solar gravity well. The farther the destination is from the Earth reference point in the solar gravity well, the more difficult the mission, which in turn relates to interplanetary transfer delta velocity. For example, given equivalent mission and systems assumptions, a Saturn destination has a lower delta-v than a Neptune destination, and thus the delivered payload at Saturn would be higher than Neptune. Another payload issue related to destination includes the capture method employed at destination. For equal mission and systems assumptions (except for capture), an aerocapture generally can deliver greater payload to a set destination (assuming an attained "small enough" aerocapture mass fraction) than a chemical mission. Yet early tradeoff results with a chemical capture would probably indicate lower system reliability and higher system development cost for aerocapture. This paper examines the destinations[3, 4] of Saturn and Neptune.

Launch Vehicle. The choice of launch vehicle significantly impacts mission cost, mission reliability, and payload placement performance. This factor is explicitly brought to the foreground given that the trajectory optimization process adopted for this analysis ties the launch vehicle delivery capability directly into the optimization process. Generally, with all other assumptions equivalent, the larger launch vehicle will deliver the greater payload to a given destination. Usually, the mission goal assumes to deliver the reference payload to the destination for a minimum cost. Thus, an optimization process must ultimately be undertaken to find the best compromise between launch vehicle cost and delivery of reference payload. This paper does not examine this optimization process, but does look at the question of predicted payload delivery over a range of transfer time and for several current launch vehicles. This paper examines Delta-IV[5] medium and Atlas-V[6] medium launch vehicles.

Trajectory. SEPS vehicle trajectory optimization can be a complicated process that requires specialized trajectory generation and optimization tools, significant skill at utilizing the tools, and intuitive insight into the complex SEPS mission analysis process[7]. A trajectory tool useful for generating optimized SEPS trajectories must provide the capability to solve complex non-linear trajectory problems. For this paper, these problems encompass determining the maximum payload that can be delivered to destination for a prescribed transfer time, for a vehicle with a non-constant power source, and for a vehicle with a low thrust propulsion system. The computer code SEPTOP[8], a two-body, sun-centered low thrust trajectory optimization program, performed all of the trajectory optimization results shown in this paper. The Jet Propulsion Laboratory used SEPTOP to perform Deep Space 1 mission analysis. The primary SEPTOP inputs required to perform trajectory generation includes the following elements:

1) Solar array model providing power generated by the array versus sun distance
2) Propulsion system definition including:
    - Maximum and minimum power into IPS

- Maximum and minimum number of thrusters powered during transfer
- Thruster performance envelope in thrust versus power
- Thruster performance envelope in mass flow rate versus power

3) Launch vehicle model providing the mass delivered versus C3[9]
4) Planetary destination
5) Transfer time
6) Gravity assist planet specification, if needed.

The primary data derived from the mission analysis activities consist of data necessary to compute the system stack weight, including the SEPS vehicle, adaptors, and payload. Some of the most relevant trajectory derived data is defined in Table 1.

Table 1 Some Relevant Data Derived from the Trajectory Optimization Process

Data From Mission Analysis	Vehicle System Weight Derived
Mass Delivered by Launch Vehicle to Escape Trajectory	Defines total mass from which the systems analysis derives the vehicle component masses, including structures.
Mass of SEPS Propellant	Sizes tanks, propellant management system, and concomitant structures.
Vehicle Time Dependent State Vector	Determines environment variables that relate to insulation, debris shielding, thermal protection, etc.
Propulsion System Time Dependent States	Determines max engine-on time versus power and, in conjunction with propellant mass, engine throughput (related to engine life).
Power and Propulsion Definition	Determines power system weight and propulsion system weights, including PPU weight and PPU radiator weight.

The trajectories section below provides an exposition of the mission analysis output. A companion paper[10] includes a more detailed account of the trajectory optimization process and related results.

Systems Factors. Use of high fidelity SEP vehicle synthesis models provided an estimate of the vehicle mass. A graphic of the main system and subsystem elements modeled for this paper is shown in Figure 1. After computing the mass of the electric power (power generation, conditioning and distribution), propulsion (PPU, thrusters, gimbals, actuators), propellant (fluid management and tank thermal conditioning), and structures (bus, adaptors, mechanisms, thruster support, and tank support, component attachment), the remaining mass allocation represents the usable payload delivery capability to the destination. The equation below shows the relationship between the SEPS payload and other principle vehicle masses:

Payload Delivered to Destination = Mass to Earth Departure Condition by Launch Vehicle – Wet Mass of SEPS Heliocentric Transfer Stage

Determining the SEPS vehicle wet mass was a primary task of this study; the discussion that follows focuses on the primary power and propulsion systems.

Power. Large, high efficiency solar photovoltaic arrays provide propulsion power and vehicle housekeeping power (with the exception of some battery power that must be provided for array deployment). An articulation of the arrays, in one axis relative to the sun, provides array feathering to control array temperature and to prevent the solar flux from exceeding a maximum allowable solar flux on the arrays during the high solar intensity portion of the trajectory (e.g. spacecraft at < 1 AU during Venus gravity assist). Prolonged solar array operation at high temperatures and array exposure to solar radiation degrades the efficiency of the photovoltaic cells; for this analysis a cell efficiency degradation factor of 2% average per year was applied. In addition, sizing the array area by 5% larger than required for the 30 kWe array output requirement provided further design margin. Able Engineering[11], a solar array manufacturer, provided Ultra-Flex array modeling characteristics. The Ultra-Flex model represents the present state-of-the-art in lightweight solar array technology.

Propulsion. A propulsion assumption for the SEPS vehicle includes an array of NEXT[12] Xe ion thrusters. SOA power processors[13] (PPU) converts power from the solar array and delivers electrical power at proper voltage and current to the thruster array. The thruster elements consist of a set of thrusters, gimbals, actuators, and support structure.

Other Systems. Obviously other SEPS vehicle subsystems play a critical role in the overall mass breakdown of the spacecraft, and hence affects delivered payload to the destination. This paper does not explicitly examine these subsystem effects. A diagram of the full spacecraft system is depicted in Figure 1.

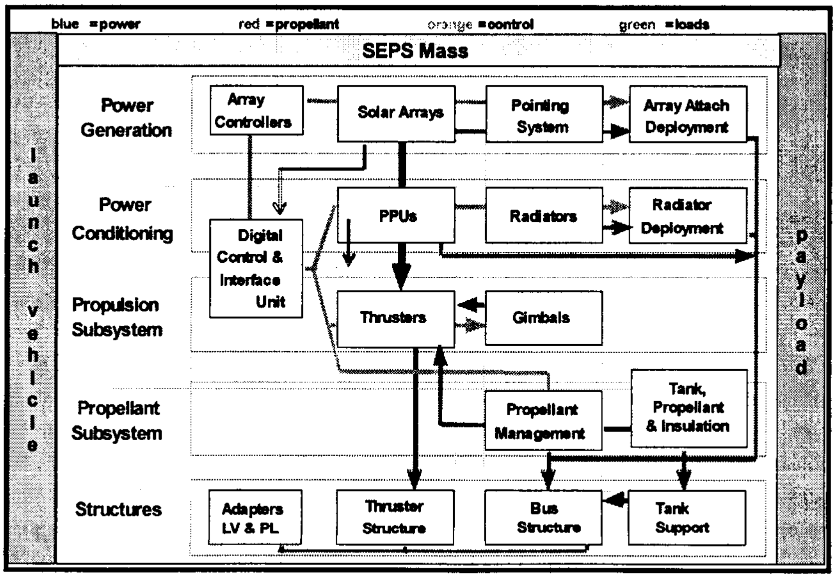

Figure 1 Total System Definition Diagram

For each subsystem block indicated in Figure 1, the mass of each subsystem is computed, and then used to support the systems analysis reported in this paper.

Baseline Mission and Systems Assumptions

To facilitate a systematic examination of SEPS performance, a baseline SEPS vehicle and mission were chosen. The basic performance requirements levied on this system were derived from the NRA-01-OSS-01[14]. Perturbations on the baseline mission and system factors were performed to investigate the sensitivity of the payload metric to the variations in those factors. The baseline mission and vehicle definitions are displayed in Table 2.

Table 2 Baseline Mission and Systems Definition

Mission Definition	
Outer planet destination	Saturn and Neptune
Reference payload	Saturn = 1400 kg, Neptune = 850 kg
Gravity Assist	Venus
Delta-V	≤ 14 km/sec for reference payloads
Launch Vehicles	Delta IV Medium and Atlas V Medium
System Definition	
Power	30 kWe at 1 AU EOL arrays; 25 kWe maximum into Ion Propulsion System; Muti-Junction GaAs arrays; Ultra-Flex design; housekeeping power is assumed to be covered by the 5 additional kWe array reserve.
Thrusters	4 thrusters with 1 spare; 6.1 kWe @ 3900 sec Isp; NEXT design; Molybdenum grids @ 10.2 g/ccm
PPU	4 PPUs with 1 spare; cross strapping PPUs; NEXT design; SOA heat pipe radiators
Tank and Propellant	Tank fraction = 5%; supercritical Xe propellant
Propellant Management	NEXT design

SEPS systems analysis required the following assumptions: mass and power margins, various contingencies, and vehicle system redundancies. These margins, contingencies and redundancy assumptions are provided in Table 3.

Table 3 Margins, Contingencies, Redundancies, and Other Assumptions

Contingencies	
Launch Vehicle	10% of nominal capacity
Propellant reserve, residual, navigation and trajectory corrections	10 % of deterministic propellant
Array End-of-Life	14% of baseline power mass
SEPS propulsion duty cycle	92%
Margins	
Dry Mass	30% of non-payload dry mass
Power	5% or baseline power mass
Other	
Redundancy	1 extra thruster, PPU, and DCIU
ACS during low thrust engine operation	Provided by Ion Propulsion System
ACS during engine-off	Provided by RCS

Mission Analysis

Trajectories

Optimization of Trajectories. The SEPTOP trajectory optimization code, including the launch vehicle model as an intimate part of the optimization process, was utilized to perform the optimization of the interplanetary trajectories. An example launch vehicle model is depicted below in Figure 2. The previous figure provides performance data for the Delta-IV Medium 4240 launch vehicle in the form of mass delivered to a particular launch C3. The SEPTOP trajectory optimization tool uses this data to determine the amount of delivered payload mass to the optimal C3 for the particular combination of transfer time, destination, power, and propulsion models.

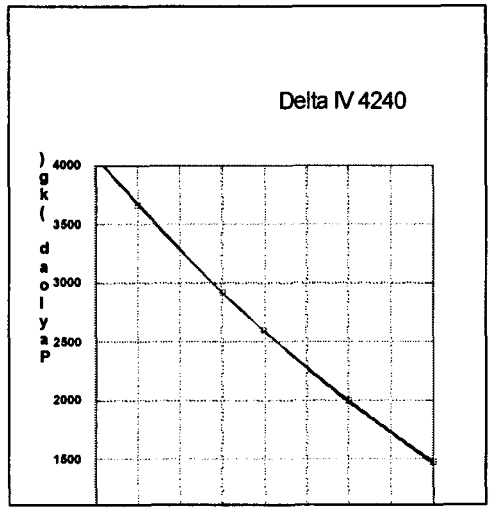

Figure 2 Payload to Escape versus C3 for Delta IV Medium 4240

Topology of Trajectories. Finding optimal trajectories involves a complex process of correctly analyzing the mission space and erudite SEPTOP utilization, potentially fraught with pitfalls. The pitfalls manifest themselves as local optima in the guise of global optima. Thus, the analyst must be very conversant in the optimization process and have significant facility in SEPS mission analysis. This portion of the paper provides results showing the trajectory path with the greatest payload delivery to the destination in a given transfer time. "Trajectory topology" refers to transfer path geometry. The characterization of the trajectory as direct or gravity assist occurs in the first level topology classification implicitly defined. All trajectories shown within this paper include a Venus gravity assist, and hence the paper does not present direct trajectories.

Outer planet missions considered in this analysis contain local optimal that associate with certain launch vehicle C3. This example of C3 optimization, depicted in Figure 3, shows total mass delivered by the SEPS vehicle (including the SEPS vehicle) as a function of transfer time. As shown, high C3 branches of trajectories and low C3 branches of trajectories exist. These various local optimal, delineated in the legend, relate to an energy trade between the launch vehicle and the SEPS propulsion system. The low C3 case generally provides the highest payload, and can be attributed to the higher Isp SEPS performing the maximum delta-v that the given power and propulsion can achieve. The high C3 case generally delivers less payload, and can be attributed to the lower Isp launch vehicle performing a greater share of the transfer delta-v than in the low C3 case. Thus the SEPS vehicle performs less delta-v in the high C3 case, and delivers less payload to the destination.

Trajectory classes include both direct and indirect (via some intermediate body gravity assist). This paper does not examine direct trajectories because without a gravity assist maneuver, direct trajectories require higher power/propulsion levels to place the same payload at an outer-planet destination.

HTTPT : Four Solutions

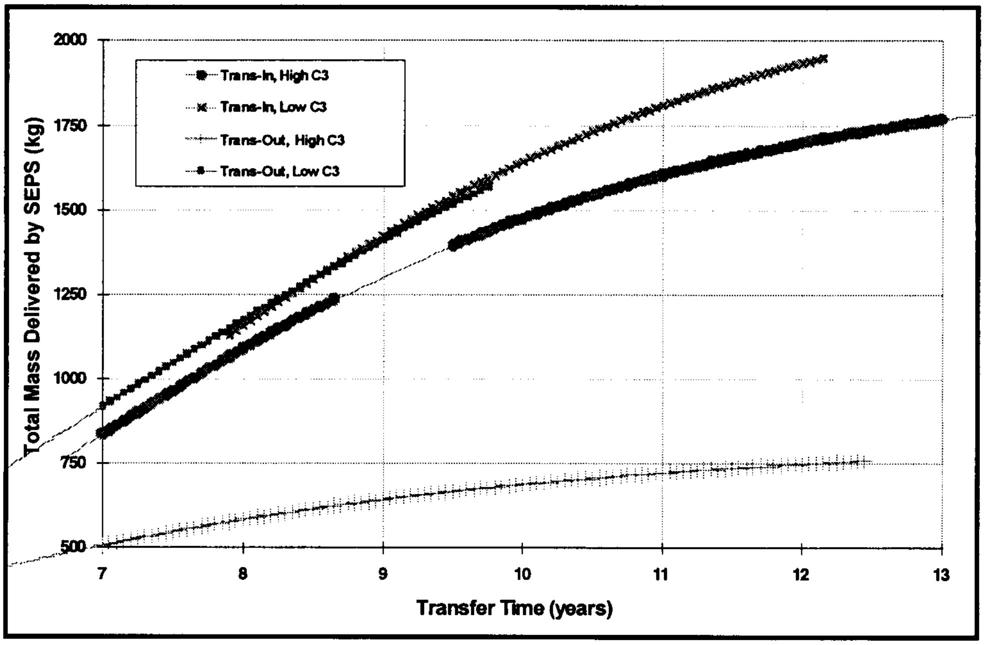

Figure 3 Complex Topography of Mission Space for SEPS Missions

Other trajectory topology factors identified as having potentially large impacts on mission analysis results include the following:

- Gravity assist flyby opportunity
- Direct gravity assist trajectory: launch vehicle inserts SEPS vehicle into a lower total heliocentric energy than Earth
- In-direct gravity assist trajectory: "energy pumping" trajectory
  -Trans-out trajectory: Trajectory initially moves away from sun
  -Trans-in trajectory: Trajectory initially moves toward sun

The flyby opportunity section below examines the issue of gravity assist opportunity. This paper does not examine the direct gravity assist, given that this case generally will not supply the optimum gravity assists (providing maximum payload delivered) for the outer planet missions considered. The next section describes the energy pumping trajectories and this paper focuses examination on this type of trajectory. Note the two sub-classes of energy pumping trajectories identified in Figure 3. The first trajectory, termed 'Trans-out', earned this identity due to an initial path away from the sun then a loop inward to perform the gravity assist. The second trajectory earned the identity 'Trans-in', due to an initial path towards the sun, then moves away and finally loops back inward to perform the gravity assist.

Energy Pumping Trajectories. The optimal gravity assist for the outer planet missions investigated in this paper tend to be of a class termed "energy pumping". This term implies the vehicle expends time in the inner solar system building energy before the Venus gravity assist occurs. The SEPS vehicle requires intense solar flux to provide significant power to the propulsion system. Therefore optimal energy gain occurs within the inner solar system and results from an energy pumping maneuver. After the energy of the vehicle increases to the optimal value, the vehicle performs a Venus gravity assist that provides the remaining transfer energy to reach the destination within the prescribed transfer time. A typical energy pumping trajectory, shown in Figure 4, illustrates a trajectory to a Titan destination. Basically, the vehicle increases its potential energy by looping away from the sun, then upon reaching the optimal energy point, the vehicle dives toward the sun (increasing kinetic energy) and performs the gravity assist. This combination loop, then dive toward Venus, provides a relatively large delta-v at Venus that reaches a value of approximately 4.5 to 5.0 km/sec.

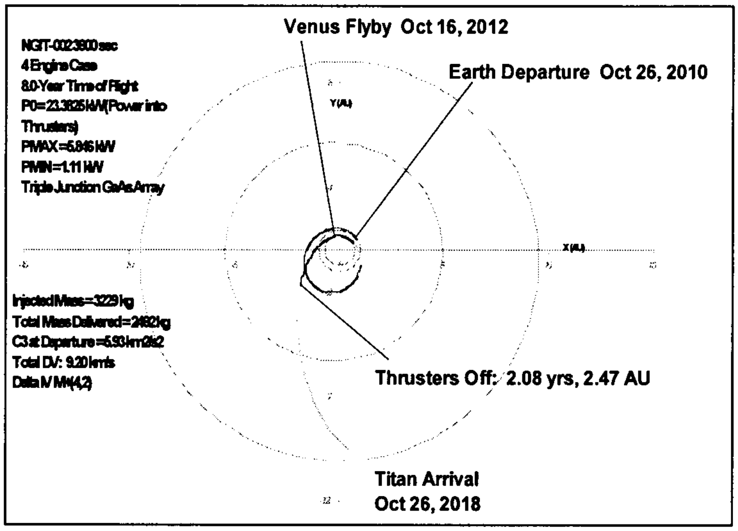

Figure 4 Earth-Venus-Saturn Trajectory

A second case, illustrated in Figure 5, indicates a typical energy pumping trajectory with destination at Neptune.

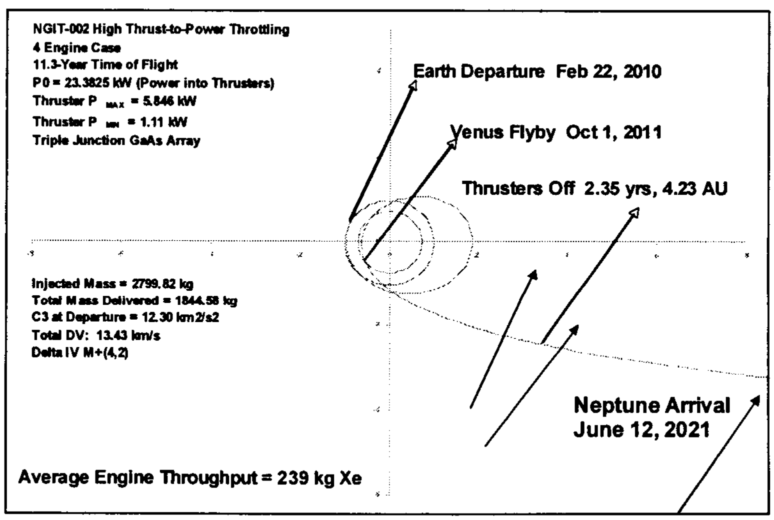

Figure 5 Earth-Venus-Neptune Trajectory

Transfer Time. One of the key factors affecting the delivered mass to the destination is transfer time. Payload delivered as a function of transfer time accounts for the majority of payload results presented in this paper. A typical case of payload delivered to Saturn over a range of transfer time is depicted in Figure 6. Generally, the payload will increase with increasing transfer time up to approximately 1600 kg maximum value, and then the payload will slowly decrease with transfer time. Note a reference payload of 1400 kg can be delivered in approximately 4.6 years. The destination section provides a further exposition of payload dependence on transfer time.

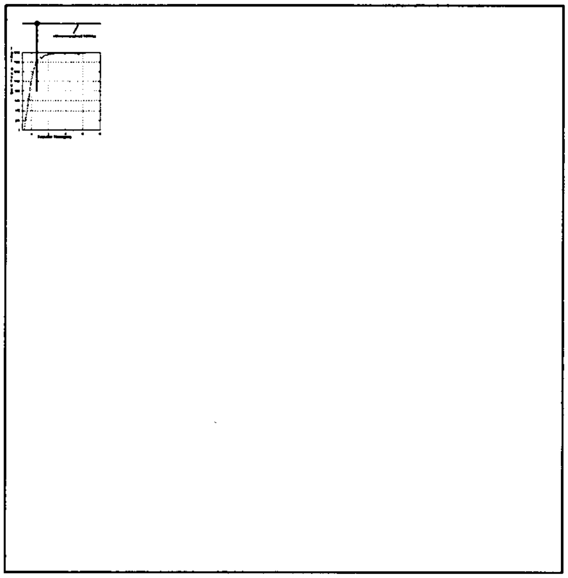

Figure 6 Payload Dependence on Transfer Time for a Saturn Mission

Flyby Opportunity. Two timing aspects of the Venus gravity assist critically affect optimization of the transfer. First, for a particular launch year opportunity (these opportunities do not exist for each year and for each destination) the actual relative geometry between departure, Venus arrival, and destination must be precise for greatest gravity assist advantage. Imprecise arrival geometry at Venus does not allow correct gravity assist to occur, due to inaccurate planetary alignments for the slingshot to be effective. This paper does not examine this aspect of the gravity assist, but note that the trajectory optimization process accounted for the effect.

The second timing aspect of the gravity assist relates to the time after launch date that the gravity assist takes place. Multiple opportunities exist to achieve the gravity assist, and those opportunities occur once each Venus year. An explanation for this involves the relatively swift Venus year as compared to a relatively long outer-planet year. For the late flyby opportunity, see Figure 7, the payload difference is over 200 kg for the 11 year transfer time and about 1.5 years earlier for the reference payload of 850 kg. Thus, the later Venus gravity assist provides a large advantage in performance over the earlier gravity assist opportunity.

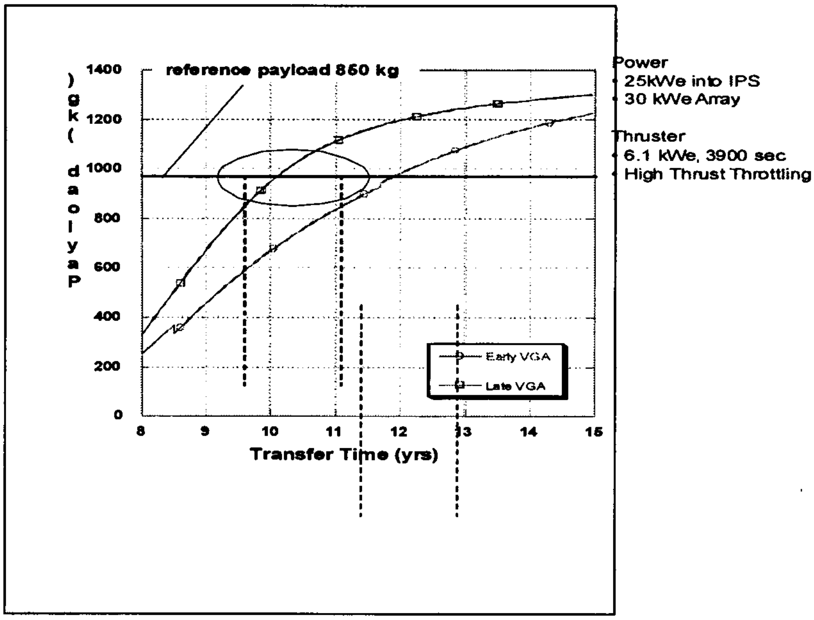

Figure 7 Payload Dependence on Flyby Opportunity for a Neptune Mission

Destination

SEPS vehicle performance depends on the destination, and for this paper the outer planet destinations include Saturn and Neptune. Two factors relate to the final destination: the reference payload to be placed at the destination and interplanetary delta-v required to place that payload at the destination. As an example, for a given payload, power, propulsion, departure date, and transfer time, Neptune is more difficult to reach (has a higher delta-v) than Saturn. For the baseline mission considered in this paper, the reference payloads are 1400 kg for Saturn and 850 kg for Neptune.

Until reaching a certain threshold in transfer time, generally payload rose with increasing transfer time. After this particular transfer time and for the given power and propulsion provided to the vehicle, the increase in transfer time forces the post Venus gravity assist trajectory to increase in a way that wastes some of the propellant rather than provide more payload. For the baseline cases investigated, see Figure 8, the reference Saturn payload of 1400 kg could be placed at the destination in approximately 4.6 years and the reference Neptune payload of 850 kg could be placed at the destination in approximately 10.8 years. Notice the discrepancy in the data in Figure 8 from the previous chart in Figure 7 depicting Neptune payload for the early and late gravity assist. The plot in Figure 7 indicates that the reference payload of 850 kg can be placed at Neptune in 11.1 years and the plot in Figure 8 indicates that the same payload can be delivered to Neptune in 10.8 years. Slight differences in analysis assumptions account for the discrepancy: Analysis related to Figure 8 assumes RTGs in the payload provides the SEPS vehicle housekeeping power and analysis related to Figure 7 assumes the solar arrays provide the SEPS vehicle housekeeping power. This provides some sense of the amount of variability in results that can be derived from subtle and unaccounted for mission and systems analysis differences. This paper illuminates the absolute necessity of the analyst to carefully delineate all mission and system assumptions before reporting to the engineering and science community for comparison of results.

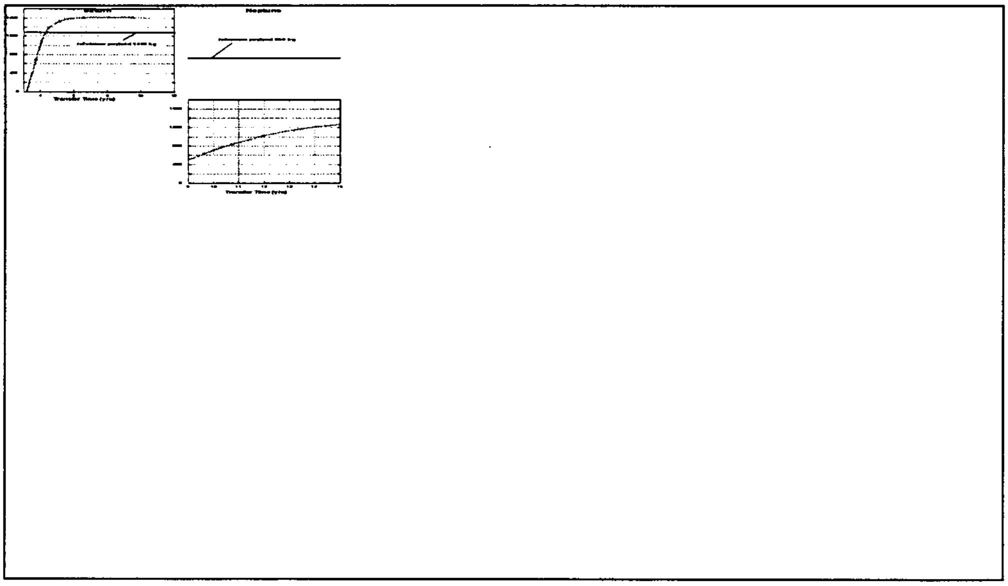

Figure 8 Payload Dependence on Destination

The capture mode at destination also influences the payload that can be delivered. Possibilities include aerocapture, all propulsive, and propulsive with aerobrake assist. This paper assumes an aerocapture mission mode, but includes no aerocapture systems engineering. This paper does not systematically address the issue of capture at the destinations.

Launch Vehicle

An examination of a range of launch vehicles determined the payload variation that could be expected over that set of launch vehicles. For the reference payloads targeted (and for significant payload variations about those reference payloads), a “Medium” class of launch vehicles provided adequate lift capability. For purposes of limiting the length of this study, a set of Boeing Delta-IV Medium and Atlas-V Medium launch vehicles were selected for investigation. The two Delta launch vehicles examined included the Delta-IV Medium 4240 and 4450. The two Atlas cases studied included the Atlas-V Medium 421 and 431. The resulting payloads to Saturn for a 4-engine 3900 sec Isp SEPS vehicle are depicted in Figure 9. Note that the average percent payload increase between consecutive cases averaged from 10% to 12% in mass increase. Another point, making clear the significant impact that margins produced in the systems analysis, includes the 11% percent payload increase for the Delta-IV 4240 case for launch vehicle margins reduced from 10% to 2%. Note that part of this 11% occurred due in part to a change in SEPS duty cycle, yet increase in duty cycle caused the smaller part of the change. This presents the absolute necessity of the analyst to carefully delineate all mission and system assumptions before reporting results for comparison.

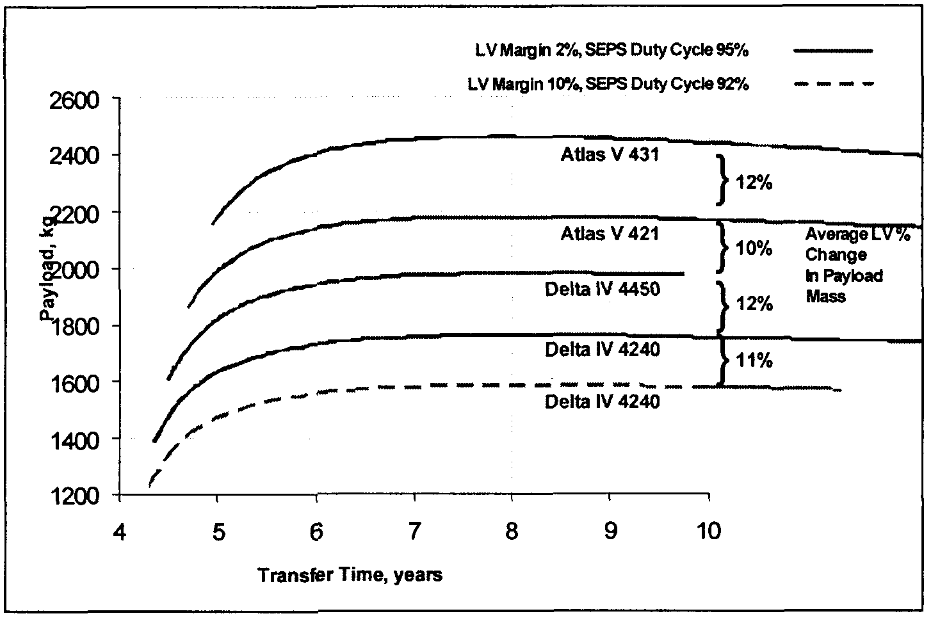

Figure 9 Payload Dependence on Launch Vehicle for Neptune Mission

Vehicle Systems Analysis

Power

The power source for the Xe high Isp engines was modeled in SEPTOP as a solar array of Multi-Junction GaAs technology. This model can be represented as solar flux approximately dropping off as $1/r^2$, yet also includes effects for low intensity light and low temperature[15] (LILLT). The LILLT modeling effects make a significant impact in the overall performance expected from the array. This paper does not further describe the solar array model.

Variation in Array Power. SEPS delivery of payload to destination is significantly sensitive to the maximum array power available to the ion propulsion system (IPS), as seen from Figure 10. Assumptions in this analysis include the following: the transfer time is 11.3 years, constant IPS (number of engines = 4, 25 kWe into IPS and other engine/PPU characteristics maintained), and only maximum array power varied. A minimum power level of approximately 21 kWe allows the 850 kg reference payload to be delivered to Neptune in 11.3 years, see Figure 10. The payload delivery mass increased with increasing array power (yet with diminishing returns) until reaching a maximum at approximately 45 kWe. Beyond 45 kWe, the payload delivered begins to decrease due to the SEPS propulsion system maximum power level remaining constant at 25 kWe, but the array, array support structure, and primary bus structure masses all continue to increase. The reference power level of 25 kWe, near the minimum value of 21 kWe, provides little payload (or dry mass) margin. Yet, the margin in payload gained from an increase in array power must be weighed against a significant increase in actual array cost (from inherently high specific cost $/kg), in larger more complex arrays, and in new, large array development risks.

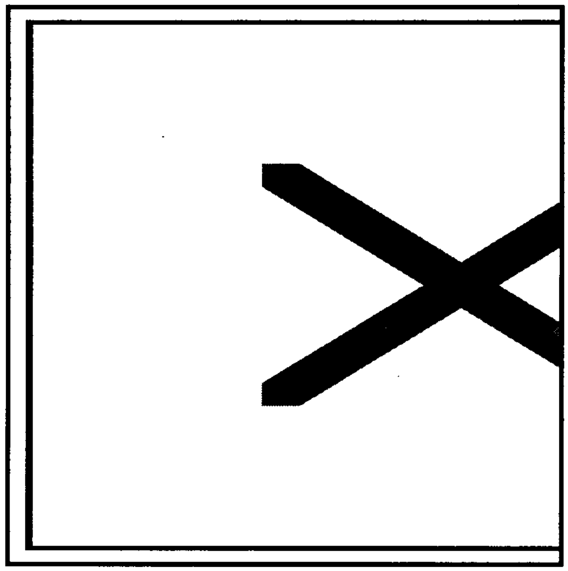

Figure 10 Payload Dependence on Power into the IPS for Neptune Mission

Propulsion

Variation in Number of Thrusters. The SEPS payload versus transfer time plots in Figures 11 and 12 show the payload sensitivity of the SEPS vehicle to number of thrusters. The 'Baseline Mission and Systems Assumptions' section above set the baseline power into the IPS at 25 kWe. In this analysis, the IPS power increased and the number of thrusters increased to utilize the higher IPS power level. Also, the analysis included two launch vehicles; the Delta-IV 4450 and the Atlas 431. From Figure 11, the payload delivered by an Atlas 431 increases with increasing power and increasing number of thrusters, but diminishing returns resulted as the number of thrusters increased to 6. Overall, 5 thrusters appear to be the optimal number of thrusters, whereas 6 thrusters provides a small improvement in transfer times below 5 years. Similar trends hold true for the Delta 4450, but there appears to be no advantage in transfer time for 6 thrusters, see Figure 12.

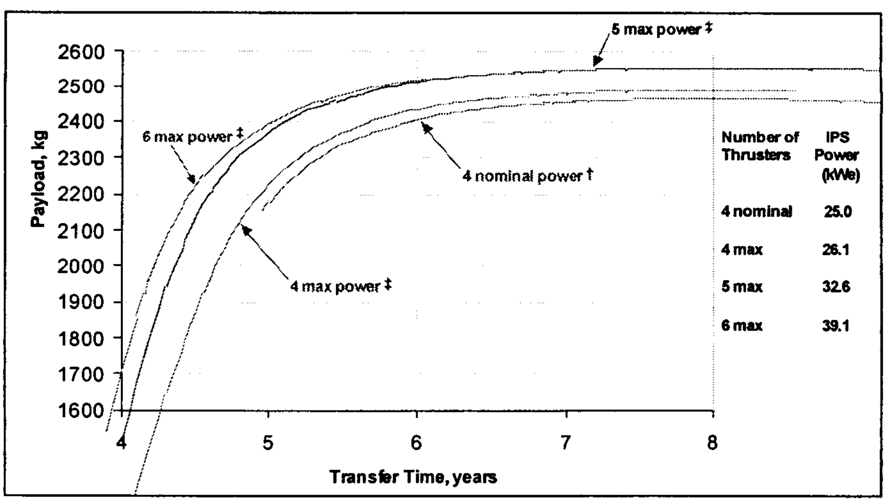

Figure 11 Atlas 431 Payload Dependence on Number of Thruster for Saturn Mission

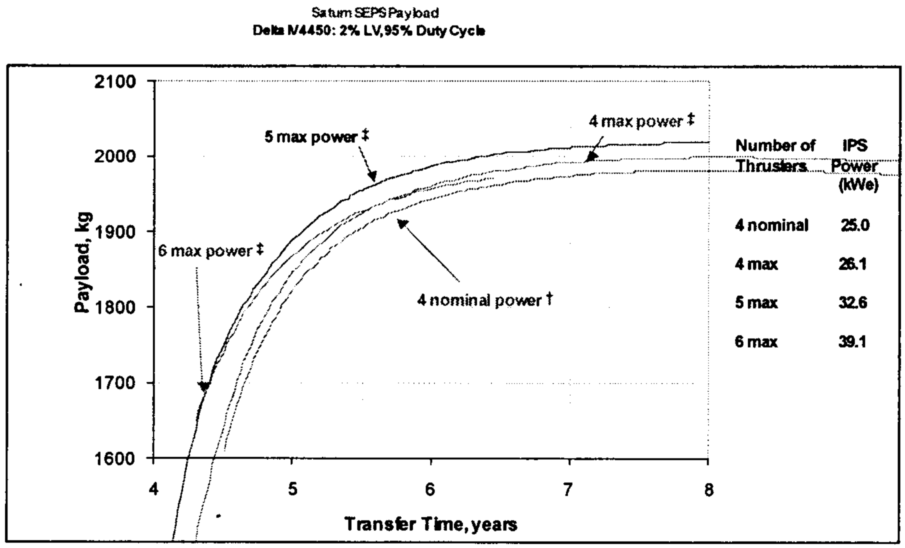

Figure 12 Delta 4450 Payload Dependence on Number of Thruster for Saturn Mission

Thruster Operation. SEPS payload tends to be slightly sensitive to the thruster performance envelope; refer to SEPS payload plotted in Figure 13 as a function of transfer time to Neptune. Using baseline thruster and power levels, a late Venus gravity assist placed the reference payload of 850 kg at Neptune in about 9.6 to 10.1 years. This range of payload variation of approximately 0.5 years can be explained by variations in Isp and thruster throttling mode. For fixed high thrust throttling, an Isp difference of about 150 seconds causes a transfer time difference of approximately 0.3 years. For a fixed Isp of 4050 seconds and a varying throttling mode, the transfer time improvement drops to about 0.2 years for high thrust throttling over low thrust throttling. This total transfer time reduction of about 0.5 years demonstrates sleight SEPS payload delivery sensitivity to thruster operation, yet there does not seem to be dramatic changes to small variation in Isp and the throttling envelope. Note that the optimal payload delivered would be realized for some optimal movement through the thruster performance envelope. This portion of the analysis did not attempt to optimize the throttling within the performance envelope, but simply throttled along the envelope boundaries of high thrust and high Isp. In addition, SEPTOP did find the optimal number of thrusters that are powered-on at any particular time in the trajectory, maximizing payload delivered to destination for the given transfer time and for the given maximum number of thrusters for the case. This paper provided no data showing the variation in thruster number as a function of time in the trajectory.

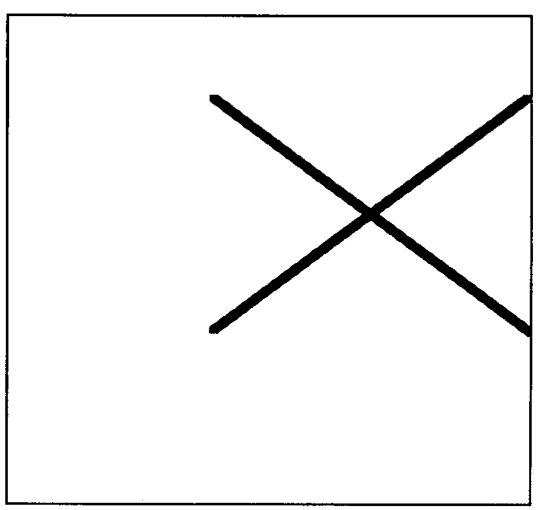

Figure 13 Payload Dependence on Isp and Throttling Modes for a Neptune Mission

Conclusions

Analyses characterized and quantified the effect that a set of mission and systems factors had on prediction of SEPS payload placement capability to outer-planetary destinations. The mission factors examined included transfer time, destination, launch vehicle, and trajectory topology. The system factors examined include power and propulsion.

The optimization of SEPS trajectories entailed a complex function of destination, time of transfer, launch vehicle, gravity assist flyby timing, and the geometrical path the vehicle travels before the gravity assist occurs (trans-in and trans-out). The payload delivered is very sensitive to trajectory optimization and many sub-optimal local minima exist. For the Saturn mission, the reference payload of 1400 kg can be delivered to Saturn in approximately 4.6 years. For the Neptune mission, the reference payload of 850 kg can be delivered to Neptune in less than 11 years for an early Venus gravity assist, but can be delivered to Neptune in 9.6 years utilizing a later Venus gravity assist.

An examination of a range of launch vehicles determined SEPS payload sensitivity to launch vehicle. For consecutively larger launch vehicles, the relative difference in payload ranged between 10% and 12% (average percent difference).

Array power variation impacts payload delivery to destination. For a 4 engine 6.1 kWe maximum thruster power level case, the minimum array power of approximately 21 kWe delivered the reference payload of 850 kg at Neptune in 11.3 years. An increase in payload of approximately 275 kg could be realized for an increase in array power of 45 kWe. As the number of engines increased and array power increased in a one-to-one fashion, the payload increase can be substantial. The optimal number of thrusters is 5 for the Saturn mission launched by a Delta 4450 and Atlas 431. Varying the throttling and Isp of a 4 engine baseline IPS produced the following results: changing the throttling model to high thrust throttling caused a slight difference of 0.2 years in the transfer time for a reference payload of 850 kg to Neptune and reducing the Isp from 4050 sec to 3900 sec resulted in a larger difference of about 0.3 years in transfer time for the same Neptune payload.

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