300-kW Solar Electric Propulsion System Configuration for Human Exploration of Near-Earth Asteroids

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The use of Solar Electric Propulsion (SEP) can provide significant benefits for the human exploration of near-Earth asteroids. These benefits include substantial cost savings – represented by a significant reduction in the mass required to be lifted to low Earth orbit – and increased mission flexibility. To achieve these benefits, system power levels of 100's of kW are necessary along with the capability to store and process tens of thousands of kilograms of xenon propellant. The paper presents a conceptual design of a 300-kW SEP vehicle, with the capability to store nearly 40,000 kg of xenon, to support human missions to near-Earth asteroids.

I. Introduction

Small body rendezvous missions have long been recognized as a class of missions for which electric propulsion provides significant benefits relative to chemical propulsion. It is no accident that three of the four deep-space missions using electric propulsion (Deep Space 1[1], SMART-1[2], Hayabusa[3], and Dawn[4]) have been to small bodies. The use of electric propulsion on Dawn reduced the cost of the mission from flag ship class (>$1B) or a New Frontiers class (>$650M) to a Discovery class (~$400M). This savings primarily manifests itself in the ability of missions to use a smaller, less expensive launch vehicles. It is therefore, natural to ask if electric propulsion can provide similar benefits for human exploration of near-Earth asteroids (NEAs). NASA's Human Exploration Framework Team (HEFT) asked exactly that question in the summer of 2010 and concluded that the use of a high-power (of order 300-kW) solar electric propulsion (SEP) system could cut in half the number of heavy lift launch vehicles required for a human mission to a "hard-to-reach" NEA.[5] This is consistent with the benefits identified in the "electric path" concept developed by Strange and Landau[6-8]. The HEFT study also concluded that the use of high-power SEP makes the system architecture significantly less sensitive to mass growth in the other in-space elements; improves mission flexibility; provides more graceful propulsion system failure modes; makes substantial power available at the destination and during coast periods; and has the potential to be reusable.

This paper looks at a candidate configuration for a 300-kW SEP vehicle and provides an estimate of its size and mass. There are many possible ways to configure a high-power SEP vehicle (see Refs. 9-11 for example). Our approach was to configure a system that minimized the development cost. While cost estimates for different technical alternatives are not included in this paper, the approach we took was to minimize the use of new technology where ever possible. If there was a choice between two or more approaches to meeting a particular requirement we selected the approach which we believed was the easiest to implement as a proxy for cost, even if it resulted in a higher system mass.

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II. Driving Requirements

The driving requirements for the high-power SEP vehicle were developed from the functional requirements of a Design Reference Mission (DRM) illustrated in Fig. 1. Given the preliminary nature of the exploration campaign it is quite likely, perhaps even a certainty that these requirements will change in the future. For example, the derived solar array power level of around 300-kW could easily change significantly. It could be as low as 200 kW or as high as 700 kW depending on particulars of the ultimate mission implementation. For the purposes of this paper we have selected 300 kW as the target minimum power level input to the electric propulsion subsystem throughout the mission with a corresponding solar array power capability of > 350 kW at the beginning of the mission. This power level provides attractive mission performance for the current human spaceflight architecture DRM to a very interesting but hard to reach NEA.

A. Design Reference Mission (DRM)

We consider the following DRM illustrated in Fig. 1 in order to determine the driving requirements for the SEP vehicle which we'll refer to as the "SEP Freighter" in this paper. This DRM is a human exploration mission of a "hard" Near-Earth Asteroid (NEA) and requires the use one SEP Freighter and two heavy-lift launch vehicles with a 105-metric-ton to low Earth orbit (L)EO capability. The SEP Freighter is launched with a Deep Space Habitat (DSH) and a Space Exploration Vehicle (SEV) with a combined mass of about 35 metric tons. The SEP Freighter transports both of these in-space elements from LEO to a High Earth Orbit (HEO), with a perigee of approximately 60,000 km and apogee at lunar distance, in about 700 days. Once the vehicles are above the Van-Allen radiation belts the option exists for additional crewed missions that can outfit, perform check out operations, and build confidence in the DSH systems. Once at the High Perigee (HP) HEO staging location and phased correctly the SEP+SEV+DSH perform a lunar close approach to lower the perigee of the orbit to a LEO rendezvous altitude or Low Perigee (LP) HEO designed for Earth Departure. This maneuver is used for NEA targets with interplanetary departure asymptote declinations of less than 30 degrees with respect to the Earth-Moon plane. For greater declinations the HP-HEO is not in the Earth Moon plane and the SEP Freighter must lower the perigee over several orbits at an additional cost of approximately 330 m/s. A second heavy-lift launch uses a cryogenic Chemical Propulsion Stage (CPS) to transfer the crew in a Multi-Purpose Crew Vehicle (MPCV) from LEO to LP-HEO in

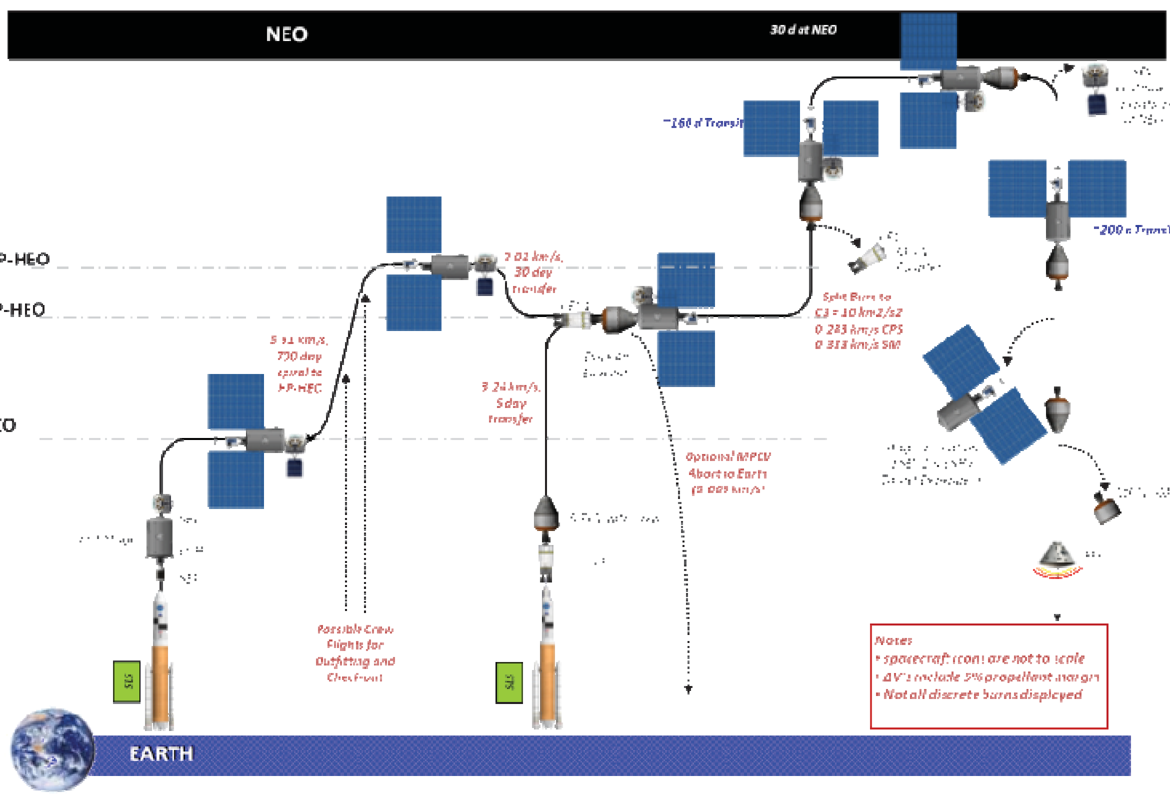

Fig. 1. Design Reference Mission (DRM) for a "hard-to-reach" near-Earth asteroid.

about four days. The MPCV and CPS are mated with the rest of the vehicle (SEP+SEV+DSH) in LP-HEO (see Fig. 2). The CPS and MPCV are used to provide an Earth-departure burn resulting in a *C3* of about 10 km^2/s^2, while moving the crew quickly through the Van Allen belts. This burn results in a maximum acceleration (thrust-to-weight ratio) of the entire vehicle of about 0.1 g. The CPS is staged once its propellant is expended allowing the MPCV service module to complete the Earth Departure maneuvers. The rest of the heliocentric transfer to and rendezvous with the NEA is performed by the SEP Freighter. The combination of the MPCV+SEP+SEV+DSH is referred to as the Deep Space Vehicle (DSV). The transfer to the NEA takes approximately 160 days for the NEA identified in the DRM. After a 30-day stay at the NEA the SEP Freighter is used to transfer the DSV minus the SEV, which remains at the NEA, back to Earth. This transfer takes approximately 210 days ending with a direct entry at Earth by the crew capsule. The other in-space elements are discarded. To perform these functions the SEP vehicle uses about 37,000 kg xenon. Re-capture of the SEP+DSH stack is possible through the expenditure of additional xenon propellant, however these maneuvers are not included in this DRM, but may be included future studies to reduce costs through reuse of the in-space elements where possible.

Fig. 2. Crew Rendezvous with the Deep Space Vehicle **(DSV)**

B. SEP Vehicle Driving Requirements

The driving requirements for the SEP vehicle in the above DRM were derived from the Mission Functional Requirement and the Mission Design Requirements as indicated in Table 1. Some of the unique and key driving requirements are discussed below.

1. Solar Array Requirements

One of the most significant requirements is the need for an autonomously deployable solar array with a total cell area of approximately 800 m^2. Assuming 33% efficient solar cells (see IIIA below), this area translates into approximately 350 kW at 1 AU. In addition to the large area, the solar array must be capable of withstanding a maximum g-loading of 0.2-g (0.1-g CPS burn thrust-to-weight times a dynamic amplification factor of 2 to account for the near step-change in CPS thrust at engine cut-off) when fully deployed, may need a first-mode natural frequency greater than 0.1 Hz and a stowed specific power density of greater than ~70 kW/m^3. The beginning-of-life (BOL) specific mass of the solar array should be less than 5 kg/kW (specific power of > 200 W/kg). The solar array configuration should have an aspect ratio of approximately 1-to-1 to minimize its moment of inertia around the roll axis of the spacecraft and to lower solar array and solar-array-gimbal bending moments to the extent possible while still meeting wing geometry restrictions to minimize EP plume impingement. The SEP vehicle must also provide the capability to articulate the solar array around at least one axis. The solar array development to meet these requirements represents the major technology advance necessary for SEP vehicle.

2. Electric Propulsion Subsystem Requirements

The electric propulsion subsystem has an input power of 300 kW. This is assumed to be divided equally among seven thrusters operating simultaneously. The driving requirements for the electric propulsion subsystem are divided into the major components of the subsystem: the Power Processor Units (PPU); the electric thrusters and thruster-gimbals; and the xenon storage and propellant management. The PPUs must be capable of processing up to 43 kW with input voltages over the range of 250 to 350 V and an efficiency of ≥95% at the maximum flight allowable operating temperature of 60C. The PPUs must have a specific mass of < 1.8 kg/kW.

Each thruster must be capable of operating at up to 41 kW at a specific impulse of 2000 s and a thruster efficiency of 60%. This combination of power and specific impulse are best provided by Hall thrusters. The thrusters must be capable of processing > 5000 kg of xenon with a low risk of wear-out failure. Each Hall thruster must have a specific mass of < 1.9 kg/kW and thruster gimbal shall have a mass not to exceed 50% of the thruster mass.

The xenon storage system must be capable of storing up to 40,000 kg of xenon with a tankage fraction of < 0.04. The xenon storage system must be capable of reducing residuals to < 1%. The propellant management system shall control the flow rate of xenon to less than ±3% (3-sigma)

3. Thermal Subsystem Requirements

The key requirement for the thermal management subsystem is to be able to reject the waste heat from the PPUs. Even with a PPU efficiency of 95%, this requires the ability to radiate about 15 kW at a maximum temperature of 60C. In addition, the thermal subsystem must radiate the 5 kW of power allocated to the non-electric propulsion

Table 1. Requirements Traceability Matrix.

Mission Functional Requirements	Mission Design Requirements	Flight System Requirements	Driving Component Requirements
Transport cargo from LEO to a high-perigee HEO (HP-HEO) - 40,000 kg of cargo - Trip time of < 2 years - LEO is 407 km x 407 kg altitude - HEO is 60,000 km x 400,000 km altitude	**Design a low-thrust trajectory to transport the flight system from LEO to HP-HEO based on:** - Launching any time of year. - Utilizing an unmanned, heavy-lift launch vehicle (HLV) with a 105 metric ton capabilty to LEO. - Maintaining the sun on the solar panels to within 0.1 rads (5.7 degrees) when not in eclipse (when thrusting and not thrusting). - Starting thrust with the electric propulsion system within TBD minutes after exiting eclipse. - Being capable of thrusting with a duty cycle of greater than 95% (mission design assumes duty cycle of 95%). - Designing an orbit transfer trajecory that minimizes the exposure of the flight system to the Earth's radiation belts. - Autonomous Earth-orbit transfer requiring contact with the ground no more than once per week during normal operations. - Thrusting with a minimum thrust level of 18 N with a specific impulse of 2000 s at 1 AU. - No shadowing of the SEP solar arrays by the payload. - The power system shall provide a margin of at least 10% for the power to the electric propulsion system.	- Provide a single-fault-tolerant flight system. - Provide a minimum power of 300 kW to the EP system at 1 AU. - Provide a minimum of 5 kW to the SEP vehicle non-electric propulsion loads. - Provide a minum of 15 kW to the payload at 1 AU. - Package inside the HLV along with the other in-space elements. - Maintain 3-axis attitude control with the EP system while thrusting. - Maintain 3-axis attitude control during eclipse periods and any time when not thrusting with the EP system. - Be power positive and thermally safe after launch and before deployment of the large solar arrays. - Minimize the roll-moment of inertia of the solar arrays. - Provide a single-axis of articulation of the solar arrays. - Provide TBD Amp-hours of battery capacity. - Maintain flight system components within their flight allowable temperature limits when in Earth-orbit. - Be capable of storing at least 39 t of xenon (including a 10% propellant margin). - Maintain flight system components within their flight allowable temperature limits when in heliocentric space. - The SEP vehicle must structurally interface with the other in-space elements. - The vehicle configuration shall reduce the direct impingement of the EP thruster exhaust on the solar arrays to a negligible level. - The vehicle configuration and thrsuter technology shall reduce the contamination of the solar arrays and other sensitive spacecraft surface to a neglible level.	- Solar array must be autonomously deployed. - Provide at least 800 m^2 of solar cell area with cells that are at least 33% efficient. - Each solar array wing outboard of the SADA must have a specific power of > 200 W/kg BOL at 1 AU. - Each solar array wing must have a 1st mode natural frequency > 0.1 Hz. - Each solar array wing must be capable of operating a peak-power voltage of 300 V at 1 AU. - The solar array shall have a stowed specific power of > 70 kW/m^3 with the longest stowed dimension less than 6.5 m. - Each SADA shall be capable of transferring a minimum of 700 A at a voltage of 300 V. - The SADA shall not allow arcing during operation with solar array voltages up to 400 V. - Each solar array wing shall have an aspect ratio of approximately 1-to-1 not including the yoke. - The solar array power shall not degrade by more than 10% over the mission. - The high-voltage power distribution unit shall have a spcific mass of < TBD kg/kW and operate over the voltage range 250 V to 400 V. - The high-voltage power distribution unit shall have the capability to disconnect each PPU from the bus at any time. - Xenon tanks must have a tankage fraction < 4%. - The xenon residuals must be < 1% of the flight load. - The thermal control system shall be capable of rejecting the waste heat from the PPUs, which could be as much as 20 kW, at 60C. - The thermal control system for the PPUs shall have a specific mass of < 25 kg/kW. - Each electric thruster shall be capable of operating at up to 41 kW and processing > 5000 kg of xenon, and shall have a specific mass of < 1.9 kg/kW - Each PPU shall be capable of operating with an input voltage in the range 250 V to 350 V at an input power of 43 kW and an efficiency of 95% with a baseplate temperature of 60C. - Each PPU shall have a specific mass of < 1.8 kg/kW - The mass of each thruster gimbal shall be less than half the thruster mass.
Transport cargo from HP-HEO to Low-Perigee HEO (LP-HEO) - Trip time of ≤ 1 month - LP-HEO is 407 km x 400,000 km altitude			
Transport Crewed Vehicle to/from a NEA - 65,000 kg of payload - Round trip flight time of ≤ 1 year	- Operate over the solar range from 0.80 AU to 1.7 AU. - Autonomous heliocentric orbit transfer requiring contact with the ground no more than once per week during normal operations. - Be capable of thrusting with a duty cycle of > 90% (mission design assumes a duty cycle of 90%).		
Loiter at HP-HEO - Maintain attitude control for up to 120 days	- Develop a mission design that allows attitude control with the SEP system during this loiter.		
Staging at LP-HEO -Maintain stable attitude control for docking	- SEP vehicle shall be passive during docking with the other in-space elements.	- Provide 6-DOF balanced thrusters for attitude control during docking.	-RCS system with 24 thrusters.
Earth-Departure Burn with a Cryogenic Propulsion Stage (CPS) - Survive a maximum g-loading of 0.25 g with the solar arrays	- SEP vehicle shall remain powered, but perform no attitude control or thrusting during the CPS departure burn.	- Solar array and the Solar Array Drive Assembly (SADA) must withstand a 0.25-g loading.	- Solar array and the Solar Array Drive Assembly (SADA) must withstand a 0.25-g loading.

loads on the SEP vehicle. Finally, the thermal subsystem must maintain the 39,000 kg of xenon onboard within its flight allowable temperature limits.

4. Attitude Control Requirements

The attitude control subsystem must maintain 3-axis control of the spacecraft, point the solar arrays at the sun and the thrust vector in the desired direction (nominally along the velocity vector). The very large flexible solar arrays and the relatively short orbital period corresponding to a 407-km circular low-Earth orbit make this more challenging. The SEP vehicle must also maintain attitude control during eclipse periods when not thrusting with the electric propulsion subsystem in order to provide thrust within a few minutes of exiting shadow. Finally, the SEP vehicle must be capable of maintaining sufficient attitude stability to enable docking with other in-space elements as well as station-keeping with or orbiting at the NEA.

5. Structural Requirements

The primary structural loads on the SEP vehicle will occur during launch. The configuration shown in Fig. 3 suggests that the SEP Freighter must support the the 28,000 kg of the DSH during launch. There are launch configurations under consideration in which the SEP vehicle's structure would not have to support the DSH so that the structure mass required to support this DSH does not have to be transported to the NEA and back. This is what is assumed for the purposes of this paper. Preliminary calculations show that attitude control system (ACS) chemical thruster plume loading of the SEP solar array wings during in-space element (i.e., the CPS) docking events could lead to deployed g-loading approximately equivalent to that experienced during the CPS burn main engine cut-off event. Further analyses are required to refine the loads associated with docking vehicle ACS plume loading of SEP tug solar array wings.

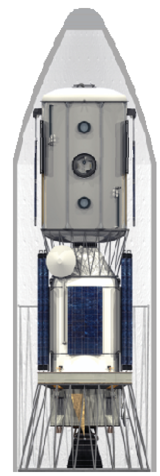

Fig. 3 Illustration of launch configuration with the DSH supported by the SEP vehicle.

III. Vehicle Configuration

Based on the driving requirements listed above and in Table 1, a conceptual design of an SEP vehicle that meets these requirements was developed. The key features of this conceptual design are given below including an estimate of the SEP Freighter mass.

A. Solar Arrays

The 350-kW solar arrays are the dominant feature of the SEP Freighter and represent a significant extension of the state-of-the-art. The highest-power SEP vehicle ever flown in deep-space is the Dawn spacecraft with a 10.4-kW solar array BOL at 1 AU. The highest-power commercial communication satellites have BOL power levels of about 24 kW. The international space station (ISS) has about 260 kW of solar array power with an active cell area of about 1680 m^2. Our SEP Freighter requires an active cell area of about 780 m^2 assuming the use of inverted metamorphic solar cells[12] that are expected to have an efficiency of 33%. (Note, the total area of the array will be greater depending on the packing factor of the selected array configuration.)

While 350 kW sounds like a huge solar array, it should be noted that solar power in space has increased dramatically since the Vanguard spacecraft, with 1 W of solar power, was launched in 1959. The power level corresponding to the highest power spacecraft launched each year is plotted in Fig. 4 as a function of its launch year. These data (tabulated in the Appendix) indicate that the maximum solar power onboard a spacecraft has doubled approximately every four years for the last 50 years. There are two notable exceptions. SERT II, the Space Electric Rocket Test II launched in 1970 with 1,000 W of solar array power is well above the curve, suggesting that it was well ahead of its time. This is reflective of the power-intensive requirements of electric propulsion. The other point above the curve is Skylab launched in 1974 with just under 10-kW of solar array power (after the arrays were fixed). Significantly, the two deep-space missions with electric propulsion launched by NASA, Deep Space 1 in 1998, and Dawn in 2007, are well below the curve. This suggests that by the late 1990's solar array capability had "caught up" to the power-intensive requirements of electric propulsion. It is interesting to note that the small, low-cost, Discovery-class Dawn spacecraft has more solar array power than the Skylab space station from 34 years earlier. With respect to this study, a mission with a 350-kW solar array to be launched sometime after the year 2020 will also be well below the curve.

Our SEP Freighter, however, needs more from the solar array than just an increase in size. It also needs to be lightweight, stiff enough to have a first-mode natural frequency of at least 0.1 Hz, and it must be capable of

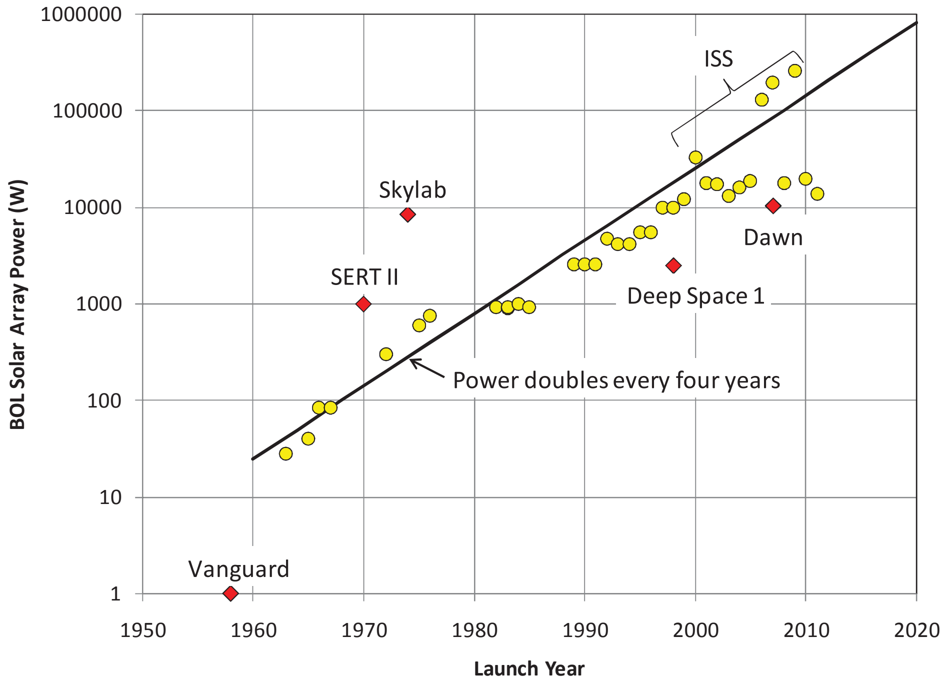

Fig. 4. History of space solar power.

withstanding a 0.2-g loading while fully deployed. These requirements are not entirely independent and an improvement in one parameter will impact the ability of the array to meet the others. At a high level, solar arrays can be divided into two main parts: the blanket assembly that includes the solar cells, the substrate to which they're mounted, and font & back cover glass; and the structure that deploys and supports the blanket assembly. For advanced, light-weight solar array designs more than 70% of the total solar array mass may be in the blanket assembly. Consequently, one of the big drivers for the blanket mass is the thickness of the solar cells. For our blanket design we assumed the use of the exquisitely thin IMM cells. These cells are only 10-micron thick and promise an efficiency of 33%. We assume they are mounted to a 5-micron thick kapton substrate. Font and back glass covers, each 125-micron thick, complete the blanket assembly except for the wiring. The 125-micron thick cover glass is used to reduce the total radiation dose on the cells resulting from spiraling through the Earth's radiation belts. The end result is that the blanket assembly is mostly glass.

As the solar array size and power level increases it is necessary to increase the operating voltage of the array in order to keep the mass of the array harness from increasing too rapidly. At 100 V a 175-kW solar array wing will produce a current of 1,750 A. At 300 V this current is reduced to 583 A for the same power. We have estimated the effect of operating voltage on the solar array mass. The results were then incorporated into an overall estimate of the SEP vehicle dry mass, so that the ripple effect of the array mass on other spacecraft subsystems (structure, propellant, tankage, etc.) could be accounted for. The results are shown in Fig. 5. These data indicate that increasing the solar array voltage from 100 V to 300 V reduces the SEP vehicle dry mass by about 1,250 kg and the wet mass by 2,200 kg.

There are many concepts for large, deployable solar array structures, see for example Ref. 9-11. It is not clear which solar array structure will turn out to be the best choice for the SEP Freighter. It is clear, however, that concentrating solar array concepts with high concentration ratios will significantly increase the difficulty of the array development because of their added requirement for tight angular pointing in at least one axis. For the purpose of creating conceptual drawings of the SEP vehicle and to make mass estimates we have assumed the use of the Mega-ROSA (Roll-Out Solar Array) concept under development by Deployable Space Systems[12] as a proxy for the final solar array configuration.

B. Hall Thrusters

Hall thrusters using xenon propellant have been operated at up to 100 kW.[13] For the 300-kW SEP vehicle we assumed an electric propulsion subsystem with eight Hall thrusters in which seven are operated simultaneously with

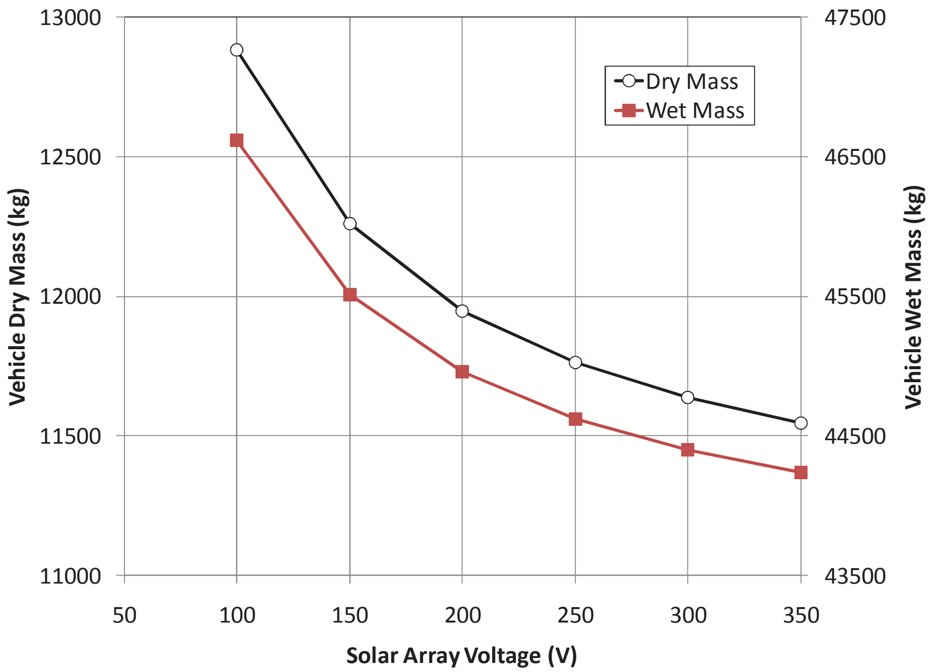

Fig. 5. Effect of solar array operating voltage on the SEP vehicle dry and wet masses.

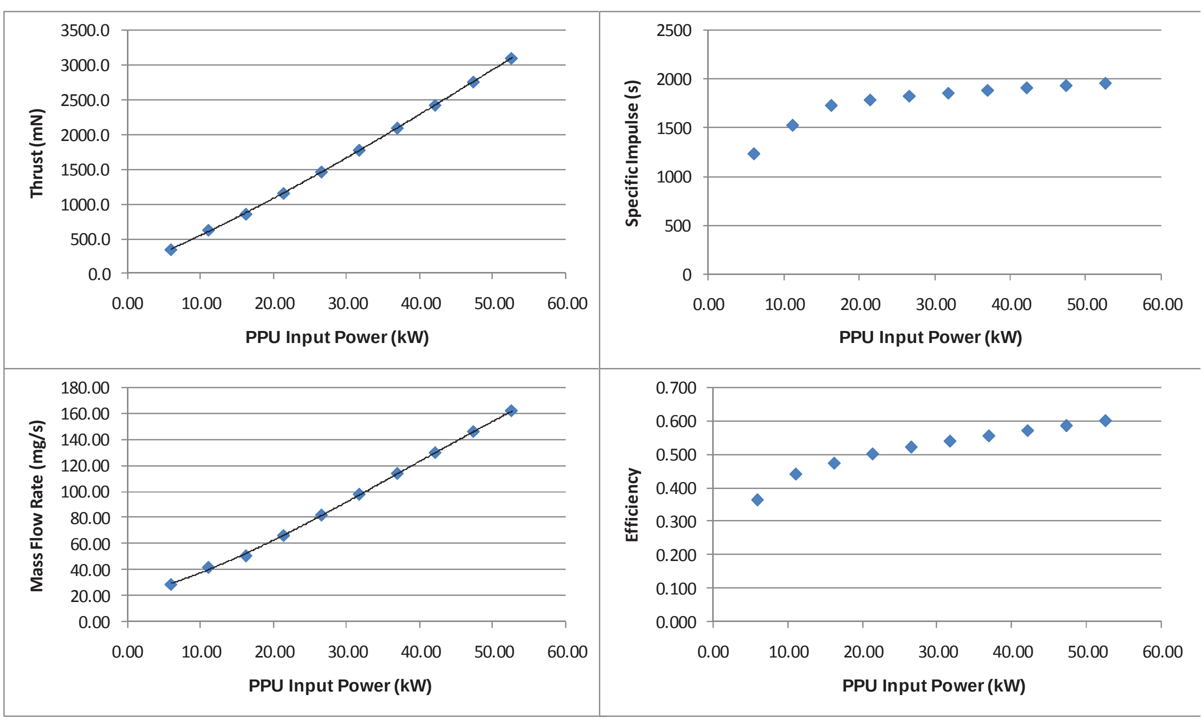

Fig. 6. Assumed throttle curves for the high-power Hall thrusters.

a PPU input power of 43 kW each. The throttle curves used in the trajectory analyses were estimated by Rich Hofer at JPL and are given in Fig. 6. The seven Hall thrusters are assumed to be capable of processing the 37,000 kg of xenon. This means that each thruster must be capable of a propellant throughput of about 5,300 kg with a low risk of wear-out failure.

Mass scaling relationships for the Hall thrusters, PPUs, and xenon feed system components used those developed in Ref. 14. The thruster and conventional PPU mass scaling relations are reproduced below (for the input power, P, in kW).

Thruster Mass (kg): $m_T = 1.8692\ P + 0.7121$

Conventional PPU Mass (kg): $m_{PPU} = 1.7419\ P + 4.654$

C. Thermal

At an efficiency of 95% the waste heat generated by the PPUs is substantial, approximately 15,000 W. This waste heat must be rejected by the thermal subsystem at the relatively low temperature of around 60C. A radiator surface area of about 28 m^2 is required at 60C assuming the radiator does not see any warm bodies (e.g. Sun, Earth, Moon, etc), has an IR emissivity of about 0.86 (white paint), and a fin effectiveness of about 90%. While 28 m^2 sounds like a lot it can readily be accommodated by the SEP vehicle with body mounted radiators and imbedded loop-heat pipes. Our SEP vehicle configuration uses two 14 m^2 radiators mounted to the spacecraft structure in planes that are normal to the axis of rotation of the solar array. This minimizes the sun exposure on the radiators. Each radiator is approximately 4.5-m long x 3.1-m wide. Four PPUs are mounted to each radiator. The vehicle configuration provides room to easily increase the radiator area if necessary. This approach eliminates the need for deployable radiators.

The 15,000 W of power dissipated by the PPUs does not account for any other electronic element dissipation that will have to be rejected. We have allocated 5 kW for the operation of the non-EP loads on the SEP vehicle. The thermal subsystem will have to provide radiator area to accommodate these thermal loads as well.

At 300 kW input to the PPUs each one percentage point decrease in the PPU efficiency increases the amount of waste heat that must be radiated by the thermal subsystem by 3,000 W. This places a premium on PPU efficiency. Direct-drive PPUs with the promise of efficiencies approaching 99% would make the thermal design of the SEP vehicle significantly easier.

D. Direct-Drive

PPU Development for electric propulsion systems has typically been expensive and time consuming. The development of a PPU with the characteristics required for the 300-kW SEP Freighter – 43-kW input power, 250-V to 350-V input voltage, 95% efficiency, 60C baseplate, and a mass of ~80 kg – will certainly be challenging. As indicated in Fig. 4, a high voltage solar array, with a nominal peak-power output voltage of around 300 V, provides a substantial mass reduction for the SEP vehicle relative to a 100-V array. A high-power Hall thruster operating at a specific impulse of around 2,000 s requires an anode voltage of around 300 V, therefore, it is natural to investigate the potential advantages of direct-drive configurations in which the Hall thrusters are operated directly from the high-voltage solar array with a minimum of power processing electronics in between. Direct-drive concepts have been around for a long time[15] and were investigated at low powers (≤ 1 kW) with Hall thrusters over the last two decades.[16-20] Direct-Drive PPUs (DDUs) hold the promise of having significantly higher efficiency, resulting in a slightly smaller solar array and significantly less waste heat, and potentially being much easier to develop.

Following the approach of Ref. 17, we made a preliminary estimate for the mass scaling of a DDU as:

DDU Mass (kg): $m_{DDU} = 0.35\ P + 1.9$.

At an input power of 43 kW the estimated DDU mass is 17 kg, compared to an estimated 80 kg for a conventional PPU. For a system with 8 DDUs this is a mass savings of about 500 kg just in the PPU mass, not counting the corresponding structure mass savings, or the reduction in thermal subsystem mass, or the decrease in solar array size.

The DDU consists mostly of the Heater/Keeper/Magnet (HKM) supplies, control circuitry, and filtering. The solar array is most efficient when providing a DC current at the maximum power point. A Hall thruster, however, operates with a discharge current oscillation that could be as much as 50% to 100% of the DC level. Consequently, filtering is required to make the oscillating Hall thruster load look like a DC load.

The preliminary investigations of direct-drive with Hall thrusters suggest that direct-drive appears to be feasible, but identified a number of important technical issues that need to be addressed. JPL and the Glenn Research Laboratory (GRC), therefore, are working to establish a direct-drive testbed to investigate these issues. This initial testbed, to be located at JPL, will provide at least 10 kW of solar power for 4 hours a day, 8 months of the year (on clear days). Among other things, JPL and GRC will investigate how to operate a single thruster near the peak power point of the solar array, how to start and stop thruster operation, and how to operate multiple Hall thrusters direct-drive from a single array.

For the 300-kW SEP vehicle with direct-drive, the 5-kW spacecraft power for non-electric propulsion loads is assumed to be supplied by an electrically separate segment of the solar array. This segment of the array is assumed to be at whatever input voltage the spacecraft requires in order to eliminate the need for a high-voltage down converter to supply the spacecraft loads.

E. Xenon Tanks

The SEP Freighter needs to store approximately 39,000 kg of xenon. Each one percentage point increase in the tankage fraction will increase the tank mass by nearly 400 kg. This places a premium on affordable technologies that minimize the tankage fraction. We assumed the use of seamless aluminum-lined composite overwrapped pressure vessels (COPV) for the xenon tanks. These tanks are assumed to have a 30-mil aluminum liner with a graphite-epoxy overwrap. Each tank is assumed to be 1-m diameter by 4.5-m long and can store up to 4,900 kg with a maximum design pressure of 1,500 psia. The tanks are estimated to have a tankage fraction of about 3%. We "round up" this to 4% since increasing the diameter of a seamless Al-lined COPV to 1-m represents a significant increase relative to the 0.55-m diameter state of the art. Eight identical tanks are used in the SEP Freighter to store the 39,000 kg of xenon.

F. Configuration

Based on key features described above the vehicle configuration shown in Figs. 7 & 8 was developed. The deployed configuration in Fig. 7 shows, as expected, that the vehicle is dominated by the large solar arrays. Each of the two solar array wings are comprised of 8 "winglets" that are 6.25-m wide x 12.5-m long. Each solar array wing is configured as with an aspect ratio of about 1-to-1 to minimize the moment of inertia around the roll axis of the spacecraft. The central structure houses the 8 cylindrical xenon tanks. The 8 Hall thrusters and the thruster-gimbals are deployed on a ~5-m long boom. The boom length is used to extend the Hall thrusters to an axial location that reduces the impingement of the Hall energetic exhaust on the solar arrays to a negligible level.

To facilitate packaging for launch, the vehicle was configured so that the length of the xenon tanks, the length of the PPU radiators, and the stowed length of the solar array winglets were all comparable. The stowed SEP vehicle configuration, shown in Fig. 8, is consistent with the use of a 5-m diameter shroud. This would enable the SEP

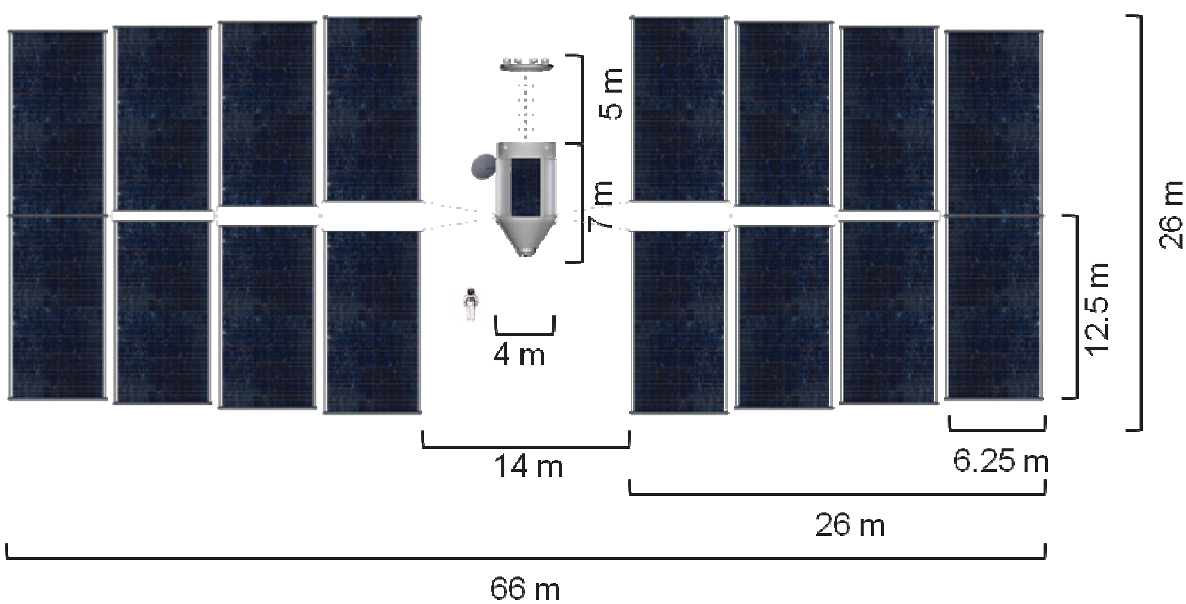

Fig. 7. 300-kW SEP in the deployed configuration.

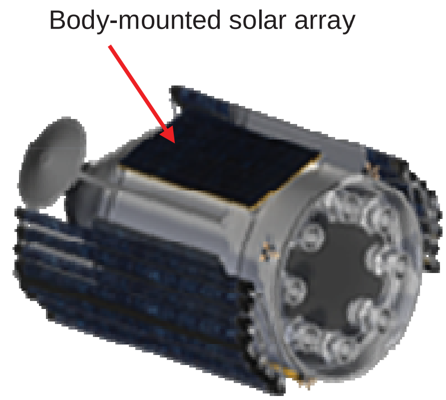

Fig. 8. Stowed SEP

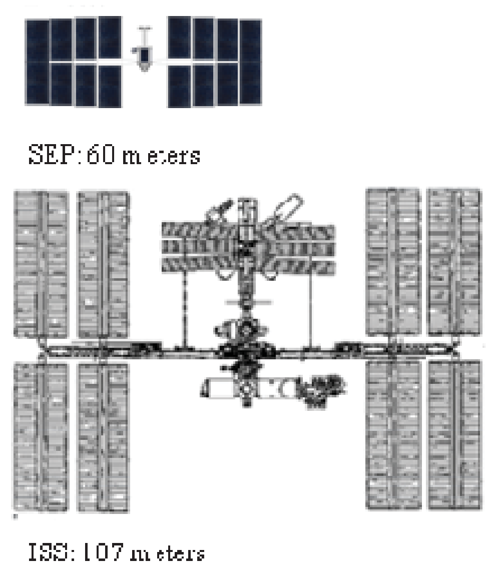

Fig. 9 SEP ISS size comparison.

vehicle to packaged on an Evolved Expendable Launch Vehicle (EELV) with a smaller xenon load, to provide launch flexibility. The solar arrays are packaged on top of the body-mounted radiators. The small, body-mounted solar arrays indicated in this figure are used to provide power to the spacecraft after launch prior to the deployment of the large solar arrays.
The 300-kW SEP vehicle size is compared to the International Space Station (ISS) in Fig. 9. Solar array outputpower levels are roughly similar, the area is reduced for the SEP Freighter due to improved solar cell efficiency.

G. Mass Estimate

High level mass estimates for two conceptual vehicles are given in Table 2, one based on a conventional PPU and the other based on direct-drive. Both of these estimates assume the use of a 300-V solar array. The direct-drive system reduces the vehicle dry mass by about 1.4 metric tons and the vehicle wet mass by 2.6 tons.

IV. Conclusion

To support human exploration missions to hard-to-reach near-Earth asteroids, a solar electric propulsion vehicle with a power level of order 300 kW is required. The use of such a vehicle could cut in half the number of heavy lift launch vehicles required to perform this mission. The key technology required for the SEP vehicle is the development of an autonomously deployable solar array with approximately 800 m^2 of solar cells. For such large, high-power solar arrays, significant mass savings are enable by operating the array at high voltage. A peak-power voltage of 300 V was assumed in the vehicle mass estimates. High-power Hall thrusters, with an input power of approximately 40 kW, that provide a specific impulse of 2,000 s, and can process over 5,000 kg of xenon are also required. Direct-drive systems, in which the Hall thrusters are operated directly from a high-voltage solar array, are projected to provide significant mass savings, substantially simplify the thermal control subsystem, and facilitate the development of the direct-drive PPU.

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Table 2. Estimated masses for the 300-kW SEP Freighter for both Conventional and direct-drive systems.

Subsystem/Component	Conventional PPU	Direct-Drive PPU
	Total Mass with Margin (kg)	Total Mass with Margin (kg)
Structures & Mechanism Subsystem	2535	2305
Ion Propulsion Subsystem (IPS)	4376	3739
Electrical Power Subsystem (EPS)	3391	3286
Reaction Control Subsystem (RCS)	230	230
Command & Data Handling (C&DH)	87	87
Attitude Control Subsystem (ACS)	19	19
Thermal Control Subsystem (TCS)	1049	659
RF Communications (Telecom)	41	41
Spacecraft Harness	365	365
Total Dry Mass	12095	10733
Xenon Mass	40150	39017
Wet Mass	52245	49663

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Appendix

Table A1. List of the Highest Power Spacecraft Launched Each Year.

Satellite Power Data		
Year	BOL Power (W)	Name
1963	28	Syncom
1965	40	Intelsat 1 (Early Bird HS-303)
1966	85	Intelsat II F-1 (HS-303)
1967	85	Intelsat II F-2 (HS-303)
1967	85	Intelsat II F-3 (HS-303)
1972	300	Anik A
1975	600	Intelsat IVA 1 (HS-353)
1976	760	Comstar 1
1976	760	Comstar 2
1982	935	WESTAR IV (HS-376)
1982	935	WESTAR V (HS-376)
1983	900	Anik C-3
1983	917	Telstar 3A (HS-376)
1984	1000	Anik D-2 (HS-376)
1985	917	Telstar 3B (HS-376)
1989	2600	Intelsat VI 2 (HS-389)
1990	2600	Intelsat VI 3 (HS-389)
1990	2600	Intelsat VI 4 (HS-389)
1991	2600	Intelsat VI 1 (HS-389)
1991	2600	Intelsat VI 5 (HS-389)
1992	4800	Intelsat K (AS-5000)
1993	4200	Intelsat-7 1 (LS-1300)
1994	4200	Intelsat-7 2 (LS-1300)
1994	4200	Intelsat-7 3 (LS-1300)
1995	5600	Intelsat-7A 6 (LS-1300)
1996	5600	Intelsat-7A 7 (LS-1300)
1996	5600	Intelsat-7A 8 (LS-1300)
1997	10000	PAS 5 (HS-601HP)
1998	10000	PAS 6B (HS-601HP)
1999	12100	Galaxy II (702HP)
2000	32750	ISS
2001	18000	XM-1 (702HP)
2001	18000	XM-2 (702HP)
2002	17500	Galaxy III-C (702HP)
2003	13000	Thuraya-2 (702HP)
2004	16000	Anik F2 (702HP)
2005	18700	Telstar 8 (LS-1300S)
2006	131000	ISS
2007	196500	ISS
2008	18000	Direct TV 11 (702HP)
2009	262000	ISS
2010	20000	XM-5 (SS/L 1300)
2010	20000	Echostar XV (SS/L 1300)
2011	14000	Sky Terra 2 (702HP)

Special Cases		
Year	BOL Power (W)	Name
1958	1	Vanguard
1970	1000	SERT II
1974	8500	Skylab
1998	2500	Deep Space 1
2007	10400	Dawn