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December 2011

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## Abstract

Human exploration beyond low Earth orbit (LEO) will require enabling capabilities that are efficient, affordable and reliable. Solar electric propulsion (SEP) has been proposed by NASA's Human Exploration Framework Team as one option to achieve human exploration missions beyond Earth orbit because of its favorable mass efficiency compared to traditional chemical propulsion systems. This paper describes the unique challenges associated with developing a large-scale high-power (300-kWe class) SEP vehicle and design concepts that have potential to meet those challenges. An assessment of factors at the subsystem level that must be considered in developing an SEP vehicle for future exploration missions is presented. Overall concepts, design tradeoffs and pathways to achieve development readiness are discussed.

## 1.0 Introduction

Since early summer 2010, NASA has been systematically evaluating human exploration missions and architectures consistent with direction provided by the Executive and Legislative branches of the United States government. By the end of 2010, initial top-level studies were completed by NASA's Human Exploration Framework Team (HEFT). These studies evaluated the capabilities needed to perform a variety of human exploration missions of interest. Possible pathways for developing the required mission architectural elements and associated technologies were identified with an overall objective of achieving affordable and sustainable human exploration. Missions of great interest included deep space missions to Near Earth Objects (NEOs). One outcome of the HEFT studies was identification of Solar Electric Propulsion (SEP) stages as having potential to be the most effective solution to perform deep space transfer for human missions to NEOs.

In the HEFT studies, top level functional, performance, and cost features of an SEP stage were identified. Associated risk reduction plans were also identified. Following HEFT, further architectural study and risk reduction project planning has occurred. This paper first describes the SEP stage concept as it relates to human exploration, and then both the challenges and promising solutions for realizing such a stage. It also provides a brief introduction to near term steps to reduce SEP stage development risk by demonstrating the ability to solve the system challenges.

## 2.0 System Concept and Mission Applications

SEP has been used both as primary propulsion in NASA science missions, first on Deep Space-1 in 1998 and most recently on the Dawn Spacecraft now in orbit around the asteroid Vesta (Ref. 1), and routinely for station-keeping on communication satellites in geostationary orbit (GEO). However, the use of an SEP vehicle as a stage or a "tug" to move cargo and humans has only been studied at the conceptual design level. These conceptual designs have shown that SEP is an efficient way of moving large masses from low Earth orbit (LEO) to GEO and beyond, although at the expense of trip time.

All SEP system concepts feature an electric power system converting sunlight to electricity via photovoltaic solar arrays which power electric propulsion (EP) thrusters, typically either gridded ion engines or Hall-effect thrusters, usually running on xenon propellant. Since EP inherently produces low thrust, and therefore requires long thrusting periods as compared with chemical propulsion, trajectories beginning in LEO ending anywhere higher are typically spirals. The LEO portion of a transfer mission has frequent insolation and eclipse periods resulting in operational complexities between the electric power system, electric propulsion system, attitude control system and the guidance, navigation and control system. In fact, compared to a chemical stage, the design of an SEP vehicle is more complicated in that the key systems are highly coupled, with the electric power system driving much of the vehicle's design and concept of operations.

NASA science missions have successfully employed SEP systems at or under 10 kWe. There is interest within and beyond NASA for systems at moderate power levels (10 to 30 kWe) and within NASA, for very high power levels (200 to 400 kWe and higher). Throughout this paper, the SEP power levels are quoted as the total power input to the EP thrusters. This means at beginning-of-life for a "300-kWe-class" vehicle, the solar array power generation capability will actually be on the order of 400 to 450 kWe. Mission applications for the moderate power levels include commercial and government orbit raising and advanced planetary missions (Ref. 2). Very large power level missions involve human spaceflight, either for pre-positioning of cargo at destinations prior to a crew's arrival (e.g., Earth-Moon L1, Lunar orbit or Mars orbit) or, as in the recent HEFT study activity, for moving humans as well.

## 3.0 Past Studies

In the last two decades, NASA has conducted several SEP conceptual design studies for human and robotic mission applications. Most notable of these studies include the "all-solar" (versus nuclear) option for a human Mars exploration Design Reference Mission (DRM) SEP vehicle (Ref. 3) and the human exploration Lunar Gateway OASIS study (Ref. 4). These conceptual SEP vehicles are depicted in Figures 1 and 2, respectively. Several more SEP conceptual vehicles and mission descriptions can be found in Reference 5.

Recent studies performed by NASA's HEFT activity concluded that SEP is a "big enabler" reducing launch mass by 50 percent (factor of two) and mass growth sensitivity by 60 percent (Ref. 6). A decreased sensitivity to mass growth is particularly important given the likelihood that element masses grow as the architecture matures. Exploration architectures utilizing SEP for in-space transportation are particularly robust with respect to mass growth because of the high specific impulse of their propulsion systems.

One HEFT DRM examined the utility of SEP to support a human mission to a near Earth asteroid. In this mission application, two 300- kWe-class SEP vehicles (see Fig. 3) would be used. The first vehicle would be brought to LEO on a heavy lift launch vehicle and used in a cargo mission, spiraling to the Earth-Moon L1 point with a large Cryogenic Propulsion Stage (CPS) as the payload. The next heavy lift launch brings a second SEP vehicle to LEO with a Deep Space Habitat (DSH), a Space Exploration Vehicle (SEV) and a Propulsion Kick Stage. This vehicle would also spiral out to L1 where it would dock to the CPS brought by the first SEP vehicle. A third heavy lift launcher and second CPS would bring a Crew Transfer Vehicle (CTV) to L1 where it would dock to the DSH and the crew would then transfer to the DSH attached to the SEP stage. The unique aspect of this DRM is that the second SEP stage would then be used to bring the crew, the DSH, SEV and CTV out to a near Earth asteroid and back again within a year. This would be the first instance of SEP being used in a human spaceflight mission. The advantage of this DRM is that the entire mission can be accomplished with only three heavy lift launches, half the number needed for a DRM using all-chemical propulsion.

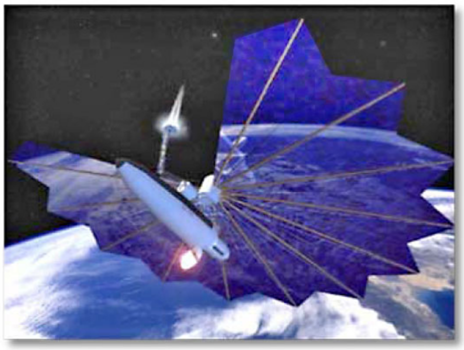

Figure 1.—SEP Tug for DRM 3.0 Human Mars Mission studies 1997-1999.

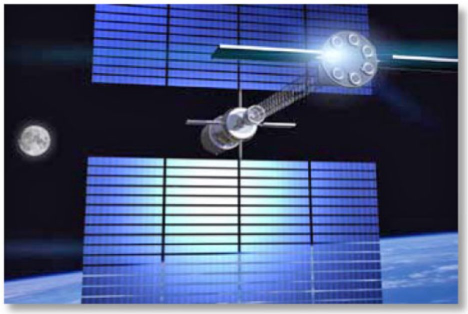

Figure 2.—2001 Lunar Gateway OASIS SEP Tug.

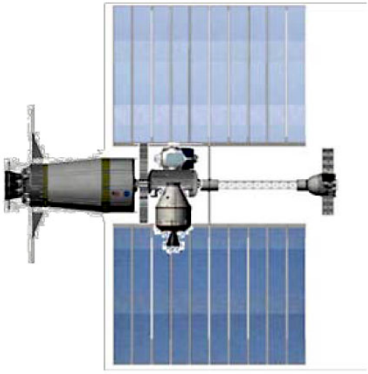

Figure 3.—2010 HEFT SEP vehicle.

## 4.0 Challenges

There are a number of well known significant technical challenges associated with SEP that arise in every conceptual design study. There are also a number of other technical challenges that arise when the details of an SEP vehicle and its operation are more carefully considered. While specific technological advancements would certainly enhance an SEP vehicle of any size, small to moderately sized SEP vehicles (less than 30 kWe or so) will not necessarily require or be enabled by any particular new technology development. However, engineering development is necessary to reduce the risk for the integrated operation of the vehicle for the spiral trajectory from LEO through the radiation belts. For large-scale SEP vehicles, the most significant challenges are feasibility and affordability, with the drivers being the design, construction, integration and testing of the large autonomously deployable solar arrays and the overall vehicle qualification, integration and testing.

## 4.1 Familiar Challenges

The well-known technical challenges center on the large, high power deployable solar arrays, power management and distribution voltage and architecture, multi-thruster operation and life, and optimized low thrust guidance and vehicle attitude control.

Also typically considered is a “direct drive” approach that eliminates a traditional power processing unit (PPU) and instead directly couples the solar array output to the electric thrusters. The mass savings associated with eliminating the PPUs can be significant, although system complexity and operational stability then become considerations.

## 4.2 Less Familiar Challenges

Lesser known challenges not adequately addressed during many of the previous SEP conceptual design studies are uncovered when mission operations are contemplated in more detail. Solar array deployed strength and natural frequency are extremely critical and must be analyzed in detail before a solar array concept can be termed feasible. In some mission scenarios, the deployed solar array must withstand loads due to impulsive chemical thruster firings as well as docking loads. For the HEFT near-Earth asteroid mission, the 400-kWe class solar array must withstand a 0.2 g deployed load (includes a dynamic amplification factor of two) corresponding to a cryogenic propulsion stage main engine burn cut-off. To avoid impacting the vehicle’s attitude control system, a goal, if not hard requirement, is for the deployed solar array wing structure to have a natural frequency no less than 0.1 Hz. Experience shows that making extremely large, low-mass, deployable solar arrays that stiff can prove to be very difficult.

Spiraling up from LEO brings out other under-appreciated issues. To maximize power to the thrusters, the solar arrays must be kept pointed at the Sun. However, in LEO this must be done while maintaining the thruster pointed along the velocity vector. This can be accomplished with a two-axis solar array gimbal, which adds mass and complexity and may not be able to prevent spacecraft self-shadowing. An alternative method is to use typical single-axis array gimbals and to employ a roll steering maneuver which uses the orientation of the spacecraft to help point the solar arrays. The roll occurs twice per orbit, for non-zero solar beta angles (the angle between the solar vector and the orbit plane—see Fig. 4). While this can be hard to visualize, it is a significant issue that affects the solar array design and the guidance, navigation and control system as well as the attitude control system that must execute this maneuver.

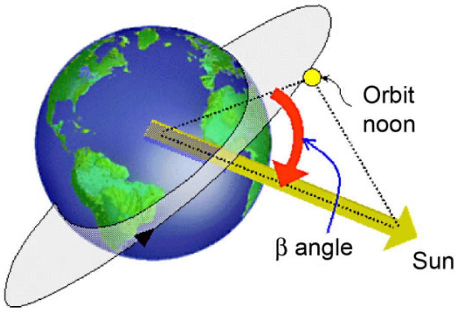

Figure 4.—Solar β angle illustrated.

At a solar beta angle of 0°, a single-axis solar array drive actuator simply rotates 360° during an orbit to maintain normally incident sunlight. At positive or negative beta angles, the orbit is inclined to incident sunlight. In order for a single-axis gimbal to track the Sun, the spacecraft must roll. This is most easily visualized at orbit noon. The highest roll rates occur at the midpoints between orbit noon and orbit midnight for small, non-zero solar beta angles. So the solar arrays must be designed to withstand this maneuver structurally and the attitude control system must be sized to handle this non-trivial event.

If roll steering is not accommodated, the solar arrays will experience off-pointing errors equal to the solar beta angle. This would require the already large solar arrays to be made even larger to compensate for the cosine-law power loss in order to maintain similar mission durations.

Another under-appreciated impact of operating in LEO involves the operation of the electric propulsion thrusters. Given that it is mass-prohibitive to include batteries to provide 300 kWe of power to the EP thrusters during the ~35 min. LEO eclipse period, consideration must be given to the propulsion system design in order to accommodate frequent power cycling over an extended period of time. Specifically, thought must be given as to what should be turned off and what can be kept on, what the impact is on the life of these components, what the consequences are for stack floating potential and arcing and ultimately the impact on the low-thrust trajectory.

Finally, the fidelity of many preliminary SEP conceptual design studies is insufficient to properly size primary and secondary structures for realistic launch loads. Initial estimates based on ratios or other simplified “rule-of-thumb” techniques can result in deceptively optimistic estimates for the mass of primary and secondary structures. Further, standard multiplicative factors applied to these unrealistically low estimates may be insufficient to provide the proper desired margin.

## 5.0 Addressing the Challenges

SEP conceptual design study teams have all suggested options to deal with the familiar challenges stated in the previous section, at least in a preliminary nature. Most, however, have not dealt with the specific engineering details. So, the ultimate feasibility and affordability of large-scale SEP vehicles still requires further study in order to improve the degree of confidence that the benefits associated with large-scale SEP can be realized. Subsequent to NASA's HEFT activity in the summer and fall of 2010, additional work was performed at GRC to address the challenges in more detail (Ref. 7).

## 6.0 Solar Arrays

Large, high power deployable solar arrays are the most significant challenge for large-scale SEP vehicles. The largest solar electrical power system (EPS) ever flown in space provides electricity for the International Space Station (ISS). The eight photovoltaic wings on ISS have a generation capability of about 256 kWe at 165 Vdc from 262,400 15 percent efficient 8 by 8 cm silicon solar cells. The specific power of these 1980s vintage arrays is about 30 W/kg. While the total deployed solar array wing area is about 3,100 $m^2$, the photovoltaic "blanket" area is 2,512 $m^2$ resulting in an areal density of about 100 W/$m^2$. Solar cell and array technology have improved significantly since the ISS solar array design was "frozen" in the late 1980s. Triple junction solar cells near 29 percent efficiency are the state-of-practice and a number of lightweight, deployable solar array concepts are in development or on the drawing board.

Promising options for large-scale SEP vehicle solar arrays have modular designs with multiple copies of identical "subwings" comprising a full wing, two of which are required to generate the ~400 to 450 kWe needed at beginning of life. Although not directly comparable to ISS solar array wing performance values (due to many different design requirements), detailed analysis shows the SEP vehicle solar array performance at just over 200 W/kg generating 390 kWe at 337 Vdc at beginning-of-life (BOL) with two wings and a total deployed area of 1,400 to 1,500 $m^2$ using 33 percent efficient inverted metamorphic (IMM) solar cells (260 W/$m^2$ areal density).

IMM cells are the leading candidate to become the next generation solar cell technology as they combine increased efficiency with a lower mass than conventional triple junction solar cells. IMM cells, and the processes for integrating them into blankets and arrays that take advantage of their features, will require a few more years of development. As will be discussed in a later section, SEP vehicle development is currently in the early planning stages at NASA, so it is expected there will be sufficient time for IMM cell and solar blanket technology to mature prior to its use in an SEP application. It is important to note that while the IMM solar cell will enhance the SEP vehicle concept, it is not required to enable it. In a recent study performed by the Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) Team at NASA's Glenn Research Center (GRC), the mass savings from assuming 33 percent efficient solar cells rather than 29 percent efficient cells (a 14 percent improvement in efficiency) resulted in a mass savings of less than 8 percent for the EPS and less than 1 percent of the total wet mass of a large-scale SEP vehicle (Ref. 8). It is important to note that this mass delta is from the change in efficiency alone, no additional mass benefits from a lower mass cell providing this increased efficiency were accounted for at this time.

While planar solar arrays just described are attractive due to their simplicity and off-pointing tolerance, concentrator arrays are also being considered since they could reduce the quantity of solar cells needed, albeit at the expense of tighter pointing requirements and more challenging and costly ground test qualification and acceptance test programs.

The masses of any array option must account for stowed and deployed strength and stiffness requirements and the ancillary items like launch packaging and restraints and for yokes that attach the solar arrays to their drive gimbals. The power, and consequent deployed area, must account for all loss mechanisms, environmental, and otherwise. The solar array must be configured to avoid harmful interactions with the EP thruster plumes, such as sputtering and contamination, and must also be designed to avoid or tolerate electrostatic arcing. It must be sized to account for parasitic plasma electron current collection for all ambient plasma environments encountered during the mission (from LEO to heliocentric space) and that plasma environment induced by operating EP thrusters. Solar array materials must be robust to atomic oxygen exposure that necessarily occurs for large SEP vehicle missions that must start in a circular low-altitude LEO (due to launch vehicle up-mass limitations).

Since the feasibility and cost associated with solar arrays are a function of deployed area, there is a strong desire to keep the solar arrays as small as possible. However, it is important to note a nonlinear tradeoff between higher power, higher EP thrust and shorter trip times through the radiation belts, resulting in less solar array degradation and smaller oversizing for radiation losses. NASA GRC is working on a simulation model coupling detailed solar array performance and degradation models, a detailed power system model and an EP thruster model with a detailed low-thrust guidance and attitude control model in order to definitively evaluate this trade. Not only will this model determine an optimum solar array size, it will help determine the optimal vehicle orbital operations concept for the power and propulsion systems as well influence the overall vehicle configuration.

## 7.0 Electric Propulsion

Notwithstanding emerging EP technologies, EP system selection is reduced to a decision between using gridded ion or Hall-effect thrusters (also known as Stationary Plasma Thrusters). For the HEFT DRM to a near Earth asteroid, a second SEP vehicle was required to both spiral from LEO to the Earth-Moon L1 point and then operate in heliocentric space after swinging by Earth to get to the asteroid. For this mission, a specific impulse of approximately 2,000 sec will keep trip times reasonable (about 1 year) while still being fuel efficient, indicating that Hall thrusters would be an appropriate choice. A change in either power level or trip time would cause this conclusion to be re-examined as gridded ion thrusters become more attractive as optimum specific impulse increases beyond 2,500 sec.

The NASA-457M Hall thruster (Figs. 5 and 6) is representative of a device that could meet this DRM's requirements with a modest amount of further development. While the HEFT study assumed eight Hall thrusters operating at 37.5 kWe each, ten Hall thrusters operating at 30 kWe and 300 Vdc would also be an option since this equates to a current of 100 A, the highest current previously demonstrated in the 457 M and results in a propellant flow rate compatible with ground test facilities (e.g., GRC's Electric Propulsion Laboratory's Vacuum Facility VF-5). Key remaining challenges include demonstrating reliable thruster operation over the required lifetime (i.e., propellant throughput) and ensuring no adverse effects from multi-thruster operation.

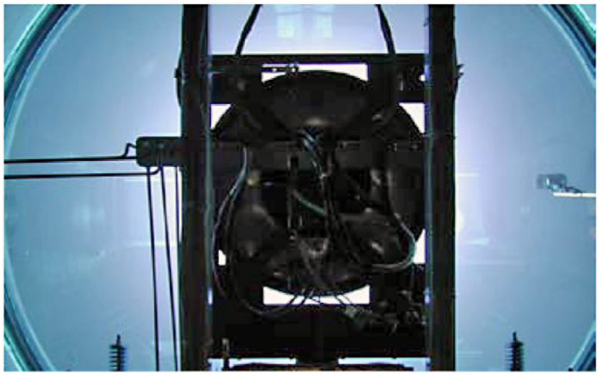

Figure 5.—NASA 457M Hall Thruster operating in GRC Vacuum Facility VF-5 circa 2002.

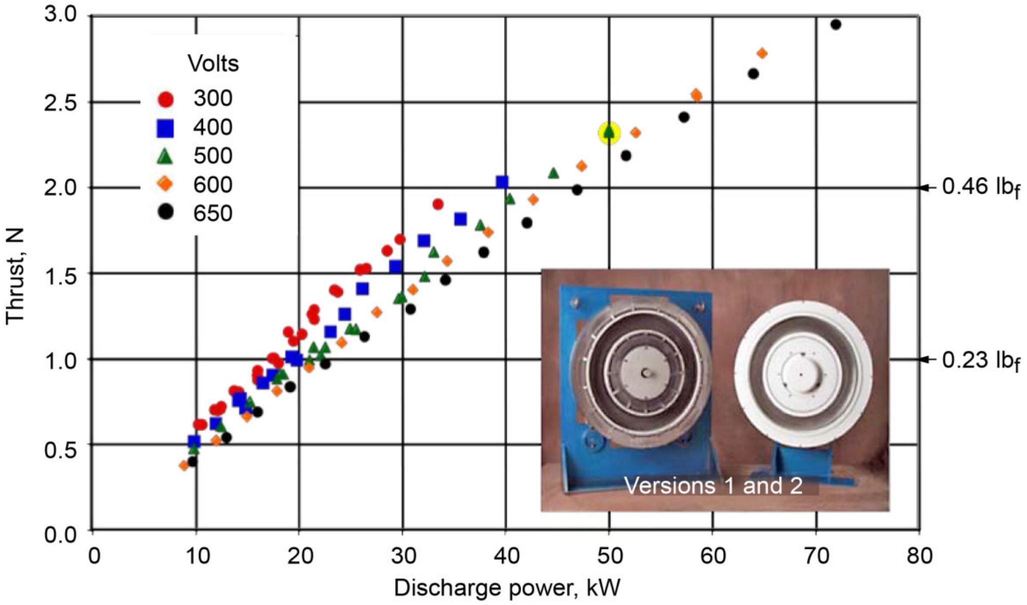

Figure 6.—Operating characteristics of the NASA 457M Hall thruster.

## 8.0 Direct Drive

The operating characteristics of the Hall thruster just described lead to interesting tradeoffs with the solar array and PMAD system. Traditionally, power would be provided to the electric propulsion system at voltages below those required by the Hall thruster. This necessitates the use of a Power Processing Unit (PPU) between the thruster and the PMAD system to convert bus voltage, typically less than 100 V for regulated busses and 80 to 160 V for unregulated busses, to the higher Hall thruster discharge voltage, typically 300 to 400 Vdc.

If a traditional approach was utilized for this application, with a total of 10 thrusters, and therefore 10 PPUs, the mass of the voltage converters in the PPUs and the mass required to reject the waste heat generated within the PPUs, which are typically only 90 to 95 percent efficient becomes significant at these power levels. However, if the solar array and PMAD systems can be operated at the voltages required to operate the Hall thruster discharge, the possibility of eliminating the PPU's voltage conversion exists (Fig. 7). Utilizing an operating mode that couples the solar array output to the thruster discharge with no intermediate conversion is an attractive means of both lowering PPU mass and increasing its efficiency to >99 percent. Estimates for this specific mission indicate a "direct drive" approach decreases PPU mass by greater than 50 percent while also decreasing PPU costs and complexity. Because direct drive couples the solar array output voltage directly to the Hall thruster discharge voltage, there are number of technical challenges related to the stability and utility of this approach, although there have been a number of ground tests that have shown stable direct drive operation with a single Hall thruster (Refs. 9 and 10).

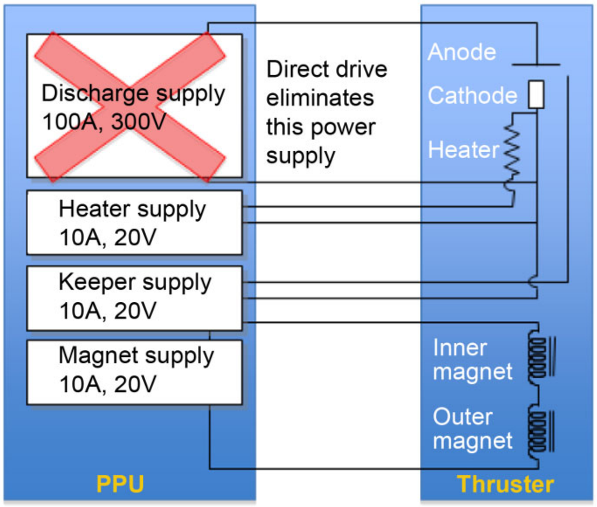

Figure 7.—A direct drive approach eliminates the discharge power supply, increases efficiency and reduces waste heat generation, saving mass.

Also, there are missions where the flexibility afforded by a PPU is desired, like those whose trajectories have large variations in distance from the Sun, resulting in large variations in solar array voltage from temperature changes. A direct drive architecture would pass along that voltage variation to the thrusters. This might result in non-optimal performance since the thruster voltage directly relates to specific impulse, which will now vary throughout the mission. A PPU could alleviate that issue and allow operation at a selected specific impulse that optimizes the trajectory. However, in the case of crewed missions to near Earth asteroids, the range from the spacecraft to the Sun is always close to 1 astronomical unit (AU), making direct-drive operation ideal for this type of mission since there will be little variation in the operating voltage of the illuminated array.

## 9.0 High Voltage

Another consequence of the direct drive approach is the need to operate the power system at the 300 Vdc thruster voltage. For high power systems, this turns out to be a great benefit. For example, operating a 400 kWe vehicle at state of practice 100 to 160 Vdc voltages requires distributing as much as 4,000 A of current. At those levels cable mass becomes prohibitive in order to maintain de-rating requirements and to limit voltage drops. Much of that cabling needs to reside on the solar arrays themselves, adding mass that must be handled by the array structures. However, operating at the thruster voltage of 300 Vdc reduces currents by a factor of three and can reduce cable and switchgear mass by an even greater factor. A recent study indicated that operating at 300 Vdc could reduce the mass of this class of vehicle by more than 2,400 kg (Ref. 7). The savings come mostly from the cable and switchgear masses, but also include the benefit from a smaller solar array which is achievable due to reduced voltage drops between the array and the thrusters. Despite the significant mass advantage of higher voltages, challenges remain to ensure that solar array arcing, and spacecraft charging are not issues and that flight rated, radiation hard high voltage EEE parts meeting derating requirements are available.

## 10.0 Trajectory Optimization

Once the EP thruster and its operating point selections are made, along with the solar array and PMAD that will power it, the details of the trajectory and how the combined systems will operate over the life of the mission need to be evaluated. As previously mentioned, the various SEP vehicle subsystems are more tightly coupled than in perhaps any other type of spacecraft. Certainly the trajectory design must account for the time-varying performance of the solar array and electrical power system. There are few, if any, existing analytical models that do this in a detailed manner. For most preliminary

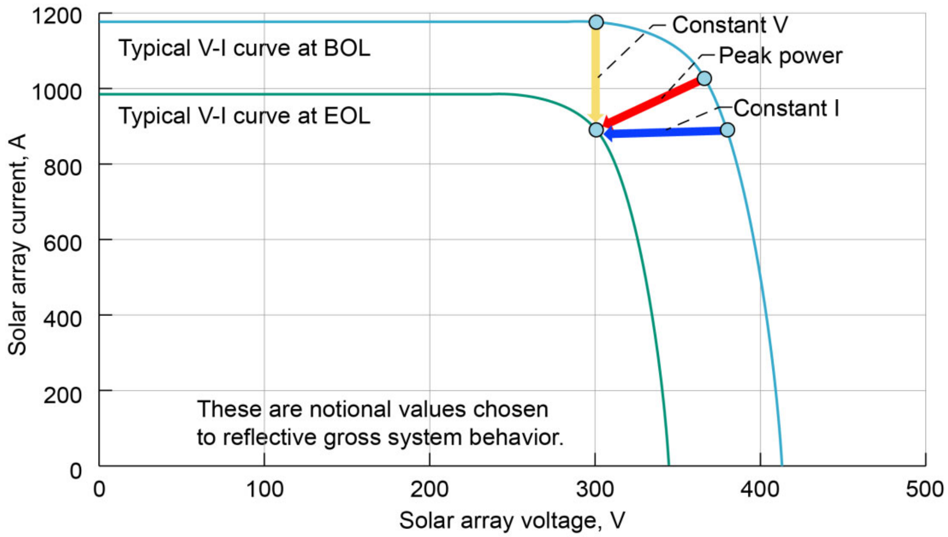

Figure 8.—Solar array current-voltage curves dictate "direct drive" thruster operations.

design studies, the first and easiest assessment of mission feasibility is made by sizing the system for end-of-life, say ten 30-kWe Hall thrusters requiring 300 kWe at 300 Vdc, assuming that this constant power is only what is available throughout the entire mission. This conservative approach makes the mission design much more tractable. However, it does not take advantage of the much higher solar array power available at the beginning of life that can be used to increase the specific impulse (Isp) and/or thrust, thus saving propellant mass and trip time.

Fundamentally, there are three approaches for utilizing solar array output power with an EP thruster: constant voltage, constant current and peak power tracking (Fig. 8). Constant voltage operation fixes the EP thruster Isp, simplifying the trajectory analysis by removing one of the independent variables. However, it does not take advantage of the higher power levels available near the beginning of the mission, which would enable higher specific impulse operation. In terms of operation, the simplest approach assumes constant current, fixing the propellant mass flow rate and the power channel current rating. In this case, both Isp and thrust varies, complicating trajectory analysis.

Based on preliminary analyses, it appears that the largest propellant savings and shortest trip time result from operating the thruster at the solar array peak power point. This is largely due to the fact that thruster efficiency is modeled such that it varies little through a range of operating voltages. Thus, the peak jet power of the thruster occurs at the specific impulse that corresponds to the peak power voltage of the array.

## 11.0 Roll Steering

As discussed previously, a vehicle with a single-axis tracking solar array drive actuator must perform a “roll steering” maneuver twice per orbit in order to keep the solar arrays normal to incident sunlight. Control Moment Gyroscopes (CMGs) are one means of providing the ability to execute these attitude control maneuvers. Since the roll rates at small solar beta angles can be quite high (Fig. 9), and a large-scale SEP vehicle quite massive, the number of CMGs required to provide the attitude control with redundancy can result in a significant mass for this function.

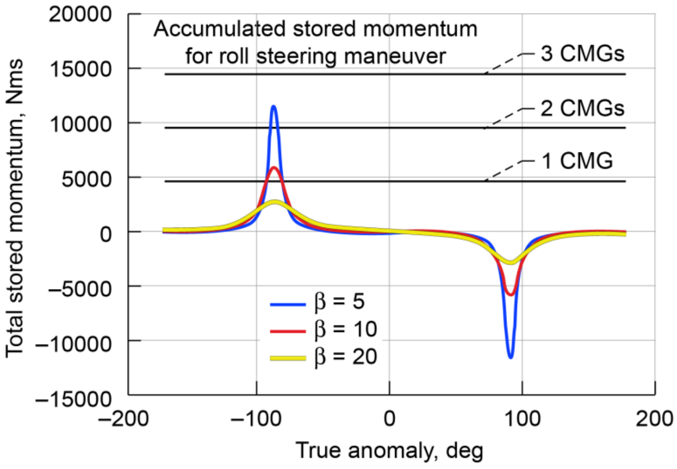

Figure 9.—Momentum compensation required for roll steering maneuvers.

For a large-scale SEP vehicle, the ISS CMGs are representative of what is needed, and they have a mass of about 270 kg each. If three or four are needed, then this subsystem comes in at about one metric ton. This mass could be avoided by simply not performing the roll maneuver and upsizing the solar arrays to account for the resulting cosine power loss due to off pointing. However, given the size of the solar arrays for a large-scale SEP vehicle to begin with, anything that drives their size and corresponding cost to be even greater should be avoided unless there is a beneficial tradeoff, i.e., eliminating CMGs results in lower mass and cost than larger solar arrays, everything else constant.

Another solution would be to employ a two-axis solar array drive actuator. Such gimbals must have the structural, voltage and current handling capabilities in class with that of the ISS solar alpha rotary joint. A two-axis solar array wing gimbal would allow for nearly perfect sun tracking and could minimize solar pointing error power losses and potentially minimize the required solar array deployed area and cost (subject to the interpretation of single-fault tolerance and how this affects the solar array sizing for gimbal failure scenarios).

The drawbacks of this approach include the increased mass of two additional gimbals (each 200 kg class), added stowed volume and more complex stowed configuration of the additional gimbals leading to a greater launch packaging challenge, increased cost of solar array gimballing, increased complexity of solar array gimballing, wing-to-wing asymmetry in performance due to different thermal and self-shadowing environments at moderate to high beta angles (affects power and propulsion channel sizing and operations). The asymmetric solar array configuration complicates vehicle level passive and active thermal control systems design and operations. It also complicates the vehicle communication system design and operations. Lastly, if solar array gimbals are not treated as design-for-minimum-risk items and do not have redundant rotational capability, the solar array must be sized to account for a gimbal failure. In this case, no savings in solar array deployed area (and cost) would be achieved.

# 12.0 Low Earth Orbit Operation

While there are undoubtedly other challenges to be dealt with, the last one to be discussed here deals with SEP operation as it flies through frequent insolation-eclipse-insolation transitions in 90 min. (minimum) intervals during the LEO portion of an upward spiral. The best way to operate EP thrusters coupled to a solar array with insufficient batteries to provide full power through eclipse must be carefully planned as this will occur for thousands of cycles. One possible scenario is outlined below.

Before eclipse, the electric propulsion system power can be ramped down from nominal to zero over a short period of time by simply shutting off the xenon propellant flow. This would likely be preferred to an immediate shutdown based on cutting off electrical power to the thruster to avoid the complications of switching off hundreds of amperes of current. It is, however, desirable to minimize the duration of the shutdown transient since the longer the transient, the lower the orbital average power will be to the EP system (and therefore thrust). Operating the cathodes during eclipse off of 200 We from batteries also helps with charge control on the spacecraft, reducing the possibility of arcing events. Operating the cathodes during the eclipse also maintains the electric propulsion system in a configuration that allows a quick start and return to full power operation once insolation is restored.

As the vehicle flies out of eclipse, the thrusters are reactivated by resuming magnet power and anode flow. The discharge will reignite at the open circuit voltage once the flow rate rises above zero. The initial current level will be low as the flow rate ramps up, but full power operation will be achieved quickly. On-off thruster cycling associated with eclipse is critical because maximizing the duration of full power operation during sunlit portions of each orbit maximizes vehicle performance.

# 13.0 Analysis Fidelity

Finally, the lack of fidelity of many preliminary SEP conceptual design studies, especially for large-scale vehicles, can result in optimistic, if not unrealistic vehicle mass and cost estimates. The only way to avoid this is to take the time to perform the detailed analysis to establish credible system sizing estimates or use higher multiplicative factors to account for uncertainty or margin growth allowance. A study by the Aerospace Corporation showed that the average mass and power growth for missions they studied was 43 and 42 percent, exceeding industry guidelines of 30 percent reserves over the current best estimate (Ref. 11). This study also found 76 percent average cost growth, 113 percent over baseline without reserve, far exceeding the typical industry guideline of 30 percent. While many conceptual design studies are done by small teams of people working short timelines, the mass estimates they come up with are quickly overtaken by reality when the detailed design phase gets underway. While it may be tempting to provide optimistic estimates to get a program "sold", the inevitable budget and schedule overruns can be avoided by simply performing more rigorous engineering analysis backed by honest and critical review.

# 14.0 Possible Development Paths

The most significant of the widely-known and less familiar challenges associated with the subsystems comprising a SEP vehicle have been discussed in this paper. These challenges are anticipated to have engineering solutions to be addressed through further design studies, ground demonstrations and technology development. The final remaining systems challenge is demonstrating the integrated operation of an SEP vehicle beginning in LEO and spiraling out to destinations beyond.

Arguably, the top priorities to address in any SEP development program include large, high-power, high-voltage solar arrays; a high-voltage power management and distribution system; and a high power electric propulsion system. Among these, the single greatest need is affordable solar arrays.

A subscale vehicle flight demonstration addressing critical risks will most likely occur before the full development of a 300-kWe class SEP vehicle begins. Since the largest SEP system flown to date is about 10 kWe, it is reasonable to expect a demonstration vehicle to stretch beyond this power level, although it would be unreasonable to expect a demonstration vehicle in the hundreds of kilowatts due to funding limitations, if nothing else.

Analysis and assessment indicates that a 30-kWe-class SEP flight demonstration would be a sound intermediate step. A vehicle near this power level or above would ensure extensibility as it would require subsystem approaches that would be the same as those used in the 300-kWe-class stage but in a lesser number (i.e., one 30-kWe Hall thruster) or via a subsystem module that could be more easily demonstrated in a first flight, but then replicated to reach the ultimate power level (i.e., two 15-kWe modular solar array wings). As a minimum, this mission should demonstrate orbit raising from LEO to as high a destination as funding allows, but at least through the radiation belts. Such a system is large enough to utilize hardware elements that are full scale for 300-kWe class, and also have all the design features and utilize an operations concept that can test out all the coupling issues and operational issues found in the 300 kWe design, thus minimizing cost and risk to pave the way for a large stage.

Recognizing the benefits and utility of SEP vehicles for future missions, NASA is presently formulating a SEP Technology Demonstration project under the auspices of the newly formed Office of the Chief Technologist. While the rate of development will depend on the available funding, the planning and coordination that will eventually bring a SEP vehicle from concept to reality will be in place.

## 15.0 Summary and Conclusions

NASA studies have shown high power SEP in the form of a propulsion stage has the potential to provide the most economical solution for deep space human exploration missions. Prior human exploration studies for human Mars missions have also shown feasible concepts for SEP deep space transfer stages. NASA has been studying various aspects of SEP stage concepts and this paper has outlined the challenges and possible solutions for developing a stage envisioned by the HEFT studies. This work has shown viable pathways to realizing such a stage and identified some near term steps to reduce risk.

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Human exploration beyond low Earth orbit will require enabling capabilities that are efficient, affordable and reliable. Solar electric propulsion (SEP) has been proposed by NASA’s Human Exploration Framework Team as one option to achieve human exploration missions beyond Earth orbit because of its favorable mass efficiency compared to traditional chemical propulsion systems. This paper describes the unique challenges associated with developing a large-scale high-power (300-kWe class) SEP vehicle and design concepts that have potential to meet those challenges. An assessment of factors at the subsystem level that must be considered in developing an SEP vehicle for future exploration missions is presented. Overall concepts, design tradeoffs and pathways to achieve development readiness are discussed.

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