Directed-Energy Propulsion Architecture for Deep-Space Missions with Characteristic Velocities of Order 100 km/s

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This paper investigates the feasibility of a propulsion system architecture that may enable missions with characteristic velocities in the range 100 to 200 km/s. The conceptual architecture is based on the use of a kilometer-scale, space-based, phased-array laser with an output power of hundreds of megawatts that beams power over distances up to 40 au to a receiving vehicle. The conceptual receiving vehicle is equipped with a very lightweight, 100-m-scale photovoltaic array (with an areal density of 100 to 200 g/m^2) that converts the laser power into electrical power with cells tuned to the laser frequency at an efficiency of 50%. The photovoltaic array outputs this power at 6 kV to directly drive a lithium-fueled, gridded ion thruster. At 6 kW with lithium propellant the gridded ion propulsion system produces a specific impulse of 40,000 s. The power and propulsion systems for receiving vehicle are projected to have a total specific mass of less than 1 kg/kW for a maximum input power to the electric propulsion system of 10 MW. A preliminary assessment of pointing and navigation requirements suggests no major show stoppers. If such a system could be developed, robotic rendezvous missions to the outer planets, Uranus, Neptune, and Pluto may be possible with flight times of 2, 3 and 4 years, respectively. Human-scale, roundtrip missions to Jupiter and Saturn may also be possible with flight times of 2.8 and 4.0 years, respectively, including 180-day stay times at the target bodies.

Nomenclature

c	=	speed of light (m/s)
D_L	=	laser aperture diameter (m)

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d_{LEV}	=	diameter of LEV's photovoltaic array (m)
d_s	=	diffraction-limited laser spot diameter (m)
f	=	power density amplification factor relative to sunlight
g	=	gravitation acceleration at the Earth's surface (m/s^2)
I_{sp}	=	specific impulse (s)
P_{EP}	=	power to the electric propulsion subsystem (W)
P_{laser}	=	laser output power (W)
R	=	distance from laser (m)
R_S	=	distance from the Sun (m)
R_0	=	distance from the Sun to Earth (m)
T_{EP}	=	thrust of an electric propulsion system (N)
T_{sail}	=	thrust of a solar sail system (N)
ΔV	=	characteristic velocity (m/s)
ε_{LPC}	=	efficiency of laser power converter
ε_r	=	photon reflection coefficient
η	=	efficiency of an electric propulsion subsystem
λ	=	laser wave length (m)
ρ_{laser}	=	average power density in the laser spot (W/m^2)
ρ_{Sun}	=	power density of solar insolation (W/m^2)
ρ_0	=	power density of solar insolation at 1 au (W/m^2)

I. Introduction

This paper describes a propulsion system architecture with the potential to provide spacecraft ΔVs in the range 100 to 200 km/s. The current record for propulsive ΔV in deep space is held by NASA's Dawn spacecraft where its ion propulsion system provided a spacecraft velocity change of 11.5 km/s.[1] This is more than a factor of two greater than the previous record of 4.5 km/s by NASA's Deep Space 1 vehicle[2]. It is clear that even spacecraft equipped with high-specific-impulse electric propulsion systems have historically demonstrated ΔV's nowhere near 100 km/s.

The ability to provide ΔV's in the range 100 to 200 km/s would enable rapid transportation throughout the solar system. An example of the types of missions potentially enabled by this capability are given in Table 1. The first four missions in this table assume the delivery of robotic spacecraft with a mass of 600 kg. The last two rows indicate potential round trip missions to Jupiter, and Saturn with human-scale payloads (50 to 60 metric tonnes) and 180-day stay times. For all the examples shown in Table 1, the mass of the power and propulsion subsystem is not included in the "Payload Mass." Flight times shorter than those in this table would require even higher total ΔV's. Since the ΔV's in Table 1 are roughly ten to fifteen times that demonstrated by Dawn, a propulsion system with a specific impulse roughly ten to fifteen times that of the Dawn ion thrusters would be required to keep the propellant mass manageable.

It is well known that rapid in-space transportation using electric propulsion requires a vehicle with a very low specific mass. For example, specific masses of < 1 kg/kW are needed for one-way flight times to Mars of less than 40 days[3] with a ΔV of about 50 km/s. To put this specific mass in perspective, the mass of the Dawn spacecraft is divided into four parts in Table 2: 1) the power subsystem; 2) the propulsion subsystem; 3) the mass of the spacecraft that isn't power and propulsion; and 4) the propellant. The specific masses in this table are simply the indicated mass divided by 2.5 kW, the maximum input power to the ion propulsion subsystem. The total Dawn spacecraft specific mass (dry) of 299 kg/kW is three hundred times that needed for a 40-day flight time to Mars. To achieve the ΔV's indicated in Table 1, specific masses of < 1 kg/kW are needed. One potential way to achieve this is the Directed-Energy Electric Propulsion (DEEP) system architecture described in this paper.

The key to any transportation system designed to go fast is to apply a lot of power to a relatively small mass. In our DEEP architecture this is achieved through the use of four advanced technologies: laser-beamed power; very lightweight photovoltaic arrays that have cells tuned to the laser frequency to convert the laser power to electrical power; direct-drive to eliminate the mass of power electronics needed to condition the power for the electric thrusters; and ultra-high specific impulse (40,000 to 50,000 s) electric thrusters to minimize the required propellant mass. The resulting architecture is depicted in Fig. 1 which shows a large, high-power, space-based Laser Transmit Vehicle (LTV) beaming power to a Laser Electric propulsion Vehicle (LEV) that collects a fraction of the laser power, converts it to electrical power at an output voltage of 6 kV to directly drive a lithium-fueled, gridded ion propulsion subsystem that produces a specific impulse of ~40,000 s.

Table 1. Examples of very high ΔV missions.

Mission	Total Time of Flight (years)	Payload Mass* (kg)	ΔV (km/s)
Uranus Rendezvous	1.8	600	175
Neptune Rendezvous	3.2	600	172
Pluto Rendezvous	4.4	600	171
Solar Gravity Lens Focus at 550 au	13	600	191
Round Trip to Jupiter (with 180-day stay at Jupiter)	2.8	53,000	122
Round Trip to Saturn (with 180-day stay at Saturn)	4.0	59,000	147

*Not including the power and propulsion subsystems.

Table 2. Dawn specific mass breakdown.

Subsystem	Mass (kg)	Specific Mass (kg/kW)***	Maximum Vehicle Acceleration (m/s^2)
Power*	177	71	---
Ion Propulsion	136	54	---
Spacecraft sans power and propulsion**	480	192	---
Propellant	425	170	---
Total (wet)	**1218**	**487**	**$7.5x10^{-5}$ (at 1 au)**
Total (dry)	**748**	**299**	**$1.2x10^{-4}$ (at 1 au)**

*Not including energy storage, i.e., the batteries are bookkept with the Spacecraft in this accounting.
**Includes the science instruments and 45.5 kg of hydrazine propellant.
**Maximum input power to the ion propulsion system is 2.5 kW which produces a thrust of 0.091 N at an Isp of 3100 s.

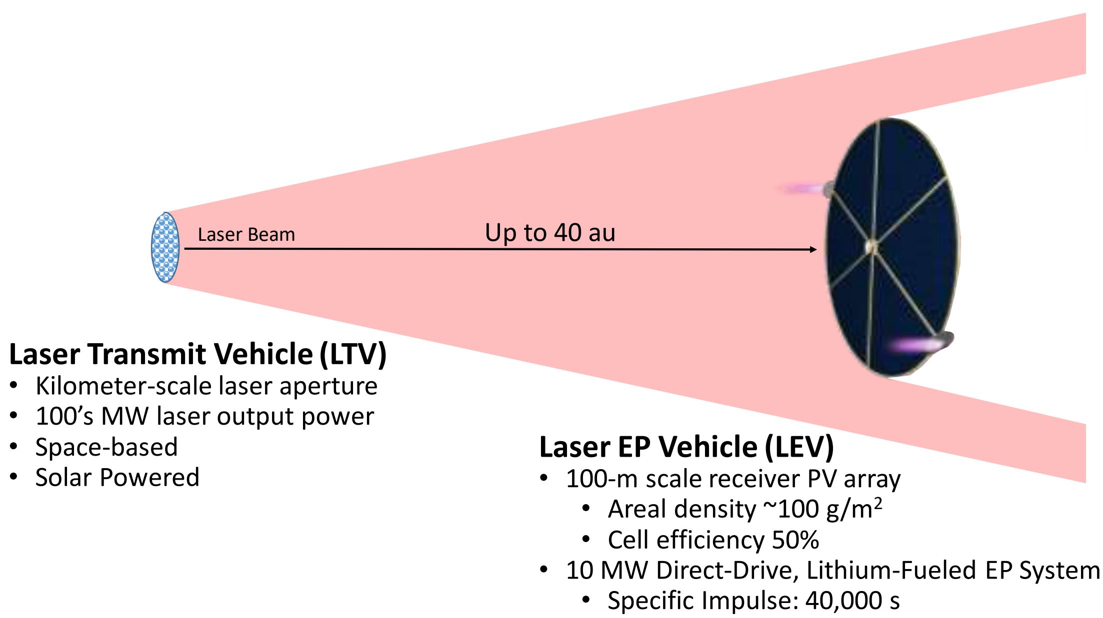

Figure 1. Depiction of a Directed-Energy Electric Propulsion (DEEP) propulsion architecture potentially capable to enabling missions with characteristic velocities of between 100 and 200 km/s. The architecture consists of a laser transmit vehicle (LTV) that is a high-power, large-aperture, space-based laser transmitter, and a laser electric propulsion vehicle (LEV) that collects a fraction of the laser power to operate an ultra-high specific impulse electric propulsion system.

II. Power Subsystem

For deep-space missions with electric propulsion, the power subsystem characteristics are the primary drivers of the overall system performance. All deep space missions with electric propulsion to date have used solar arrays as their source of power. So in the quest for very low specific mass systems it is appropriate to begin with solar array technology

A. Solar Arrays

The rigid-panel solar arrays on Dawn had a specific mass of about 13 kg/kW[4]. State-of-the-art, flexible blanket solar arrays are in the 7 to 10 kg/kW range at 1 au. For comparison, radioisotope thermoelectric generators (RTGs) have a specific mass of ~250 kg/kW. The 1-kW version of the Kilopower[5] fission reactor currently under development has a target specific mass of 400 kg/kW and about 150 kg/kW for the 10 kW version. The fission reactor system for the Jupiter Icy Moons Orbiter (JIMO) mission was projected to have a specific mass of about 40 kg/kW. Dankanich, et al[3], have asserted that it is unlikely that the specific mass of a turbo-Brayton fission reactor system will fall below 10 kg/kW, even at very high power levels (>> 1 MW). Therefore, at 1 au, solar arrays are by far the lightest power system available, which explains their dominance for spacecraft operating at solar ranges of 1 au or less. At some solar range beyond 1 au, the specific mass of a solar array will exceed that of nuclear power systems. A 7 kg/kW solar array at 1 au will continue to be less massive than the 1-kW Kilopower reactor out to a solar range of about 7.5 au (i.e., well past the orbit of Jupiter) for the same output power.

New ultralight solar array structures (see [6] for example) coupled with emerging thin-film solar cell technology such as ultrathin, flexible and lightweight perovskite solar cells with high specific power characteristics[7] have the potential to reduce solar array specific masses by an order of magnitude, to around 0.6 kg/kW at 1 au. If such a solar array could be developed it would be lighter than RTGs out to a solar range of 20 au (i.e., the orbit Uranus) for the same power, and lighter than the 1-kW Kilopower reactor out to 26 au (or nearly the orbit of Neptune).

E. Gdoutos et al. have demonstrated the development of a lightweight 1.7 m x 1.7 m prototype solar array structure with areal density of 150 g/m^2. This array structure design[6] is scalable up to 60 m x 60 m and is expected to have an areal density of 50 g/m^2 at this size.

Perovskite solar cells have the potential to be lightweight and flexible. Ultrathin (3 μm), highly flexible solar cells with a specific power as high as 23 kW/kg (specific mass of 0.043 kg/kW) under terrestrial conditions have been reported[7]. Moreover, high radiation tolerance of perovskite solar cells under high-energy particles for space environment have been demonstrated[8,9]. Perovskite photovoltaics is actively investigated and the current record for a perovskite solar cell efficiency is 20.9%[10]. The specific mass of 0.043 kg/kW for perovskite solar photovoltaic cells corresponds to an areal density of 5.2 g/m^2. High radiation tolerance of perovskite solar cells may eliminate the need of coverglass. Combining this technology (areal density of 5.2 g/m^2) with the light weight structures from Gdoutos et al (areal density of 50 g/m^2 at the 60-m x 60-m size) may enable the development of photovoltaic arrays with a specific mass of 55 g/m^2. For the performance projections in this paper we assume a photovoltaic array on the LEV with an areal density 100 g/m^2.

B. Space-based Laser for Power Beaming

Laser efficiency has improved dramatically over the past several years to the point where 40% efficiency is common place and lasers that are 50% efficient may be expected in the near future[11]. This increase in efficiency makes it reasonable to consider power beaming with lasers to enable rapid transportation throughout the solar system.

1. Laser Scaling

The first question is how big does the laser need to be? For rapid transportation throughout the solar system, we want the photon flux from the laser to be some factor, f, greater than solar insolation at any solar range. The diffraction-limited spot size of the laser, d_s, at a distance R is given by,

$$d_s = \frac{2\lambda}{D_L} R \tag{1}$$

where λ is the laser wavelength and D_L is the laser aperture. The power density of solar insolation at a distance R_s is,

$$\rho_{Sun} = \rho_0 \left(\frac{R_0}{R_S}\right)^2 \tag{2}$$

where ρ_o is the value at R_o = 1 au. The average power density in the laser beam is given by,

$$\rho_{laser} = \frac{4P_{laser}}{\pi d_s^2} \tag{3}$$

where P_{laser} is the output power of the laser. We want to size the laser such that the average power density in the beam is a factor, *f*, greater than sunlight at any solar range, i.e.,

$$\rho_{laser} = f\rho_{Sun} \tag{4}$$

Combining these four equations gives the desired relationship between the size and output power of the laser in order to increase the power density on the LEV's photovoltaic array by the factor *f*,

$$P_{laser} = \frac{f\pi\rho_0\lambda^2 R_0^2}{D_L^2}\left(\frac{R}{R_S}\right)^2 \tag{5}$$

Assuming a space-based laser at a location 1 au from the sun, then $R/R_S \approx 1$ when the LEV is far from the sun. With this approximation, Eq. (5) becomes,

$$P_{laser} \approx (f\pi\rho_o\lambda^2 R_0^2)D_L^{-2} \tag{6}$$

Equation (6) is plotted in Fig. 2 for a laser wavelength of λ = 1064 nm and values of f = 10, 25, 50, and 100. Any combination of laser output power and laser aperture size on a curve for a fixed value of *f* would provide the same vehicle performance. For example with *f* = 100 (blue curve in Fig. 2), a laser with an output power of 700 MW and a 4 km aperture provides the same performance as a 100 MW laser with a 10 km aperture providing the ability to trade laser output power with the laser aperture size.

The key feature of the curves in Fig. 2 is the vast scale of the laser required to significantly increase the power density of photons relative to the sun. Even for an amplification factor of *f* = 10 (orange curve in Fig. 2), a kilometer-scale laser with an output power of order 100's of megawatts is necessary. The feasibility of kilometer-scale lasers is discussed by Lubin[11].

To determine the amplification factor necessary to achieve the desired capability of ΔV's in the range 100 to 200 km/s, a number of assumptions regarding the characteristics of the LEV to be accelerated must be made. These assumptions are captured in Table 3. The maximum EP system power is assumed to be 10 MW. This power level along with the total efficiency of 0.97 in Table 3 results in a relatively impressive thrust level of 49 N. This thrust magnitude is more impressive given the very high specific impulse—40,000 s—of the system. How to develop a propulsion system with these characteristics is discussed in Section IV.

The cell efficiency for the photovoltaic array on the LEV is assumed to be relatively high at 50% because the cells are assumed to be tuned to the laser frequency as discussed in Section IIC.

Table 3. Characteristics of the Laser EP Vehicle.

Characteristic	Value
Maximum thrusting distance	40 au
PV receiver array diameter	140 m
PV receiver array cell efficiency	0.50
PV receiver array areal density	100 g/m^2
Maximum EP system input power	10 MW
Specific impulse	40,000 s
EP system efficiency	0.97
Dry spacecraft specific mass (sans PV array)	0.42 kg/kW

Using these assumptions the spacecraft speed as a function of distance from the laser assuming operating in field-free space is given in Fig. 3. These curves were generated by selecting a laser output power of 800 MW and then finding the corresponding laser aperture size from Fig. 2. For example, at 800 MW, the laser aperture diameter for *f* = 10 in Fig. 2 is 1.16 km. The resulting curves in Fig. 3 indicate that for an amplification factor of 10, a final spacecraft speed of 120 km/s may be achieved, i.e., ΔV = 120 km/s. An amplification factor of 100 is required to get a ΔV approaching to 200 km/s and requires a laser aperture diameter of 3.68 km (from Fig. 2 at 800 MW). For the low amplification cases, i.e., $f = 10$, about 93% of the acceleration is done over the first 10 au. For $f = 100$, this drops to ~70%, with ~30% of the acceleration accomplished between 10 and 40 au.

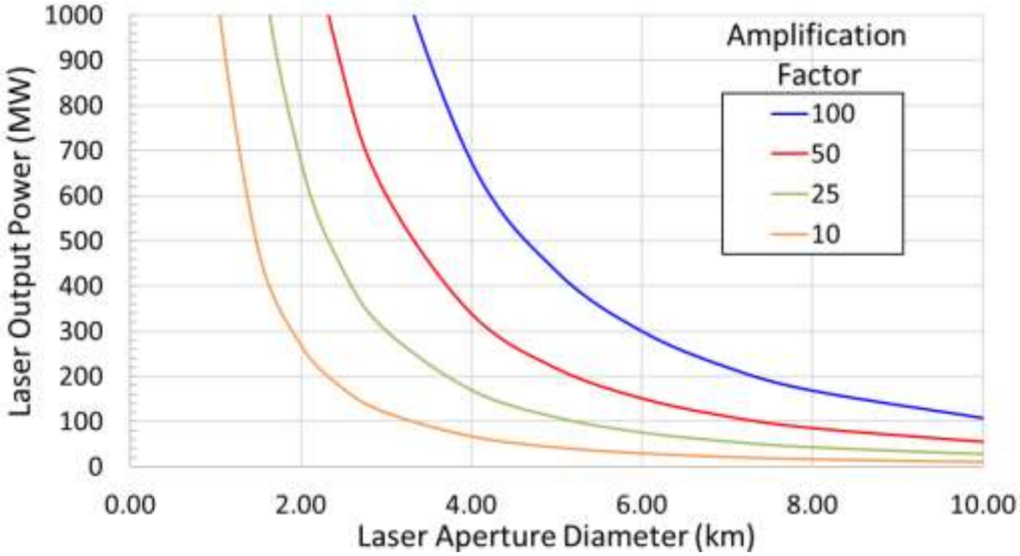

Figure 2. Plot of Eq. (6) showing that a very large, very high power laser is required to significantly increase the power density of photons relatively to solar insolation (for a laser wavelength of 1064 nm).

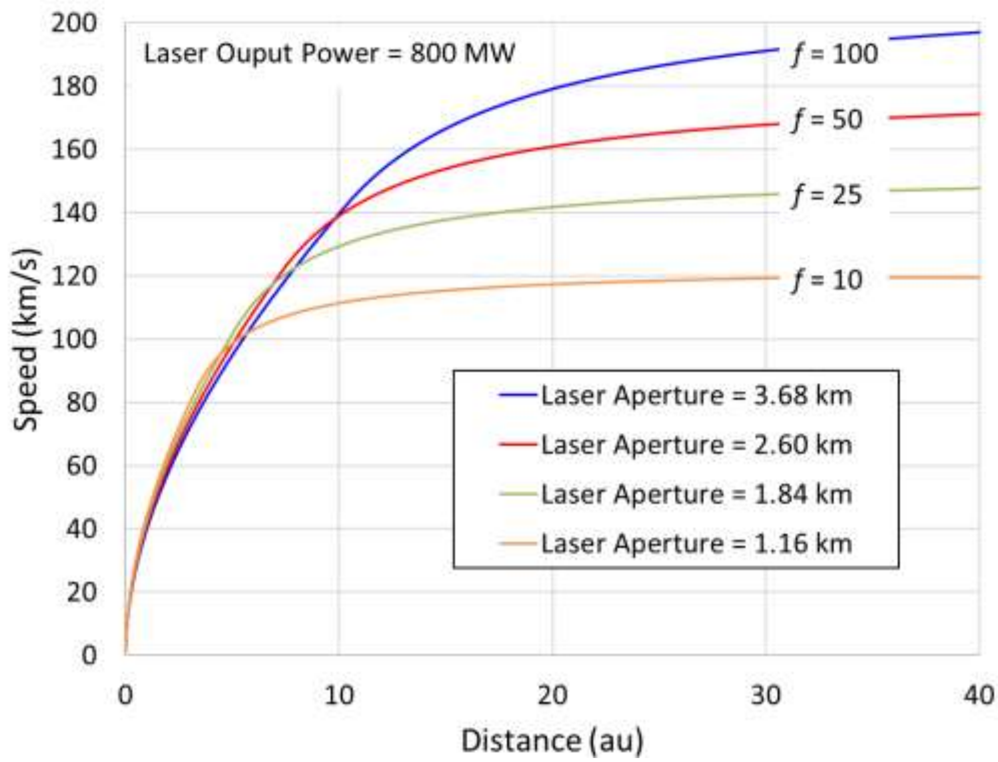

Figure 3. Amplification factors, f, of between 10 and 100 with a laser output power of 800 MW at 1064 nm are required to achieve final spacecraft speeds of between 100 and 200 km/s.

The distance from the transmitting laser over which full power operation of the electric propulsion system on the receiving vehicle is possible may be determined by recognizing that the power available to the EP system is the laser output power times the ratio of the area of the receiving PV array to the area of the laser spot size times the conversion efficiency photovoltaic cells tuned to the laser frequency. We refer to such cells as laser power converters (LPCs) and denote their efficiency as ε_{LPC} so that,

$$P_{EP} = \varepsilon_{LPC} P_{laser} \left(\frac{d_{LEV}}{d_s}\right)^2 \tag{7}$$

Using Eqs. (1) and (5), this can be rewritten as,

$$R_S = \frac{d_{LEV} R_0}{2} \left(\frac{f \pi \rho_0 \varepsilon_{LPC}}{P_{EP}}\right)^{1/2} \tag{8}$$

which gives the maximum solar range at which the LEV's EP system can be operated at full power. Equation (8) is plotted in Fig. 4 for three different diameters of the LEV's PV array, 70 m, 140 m, and 275 m.

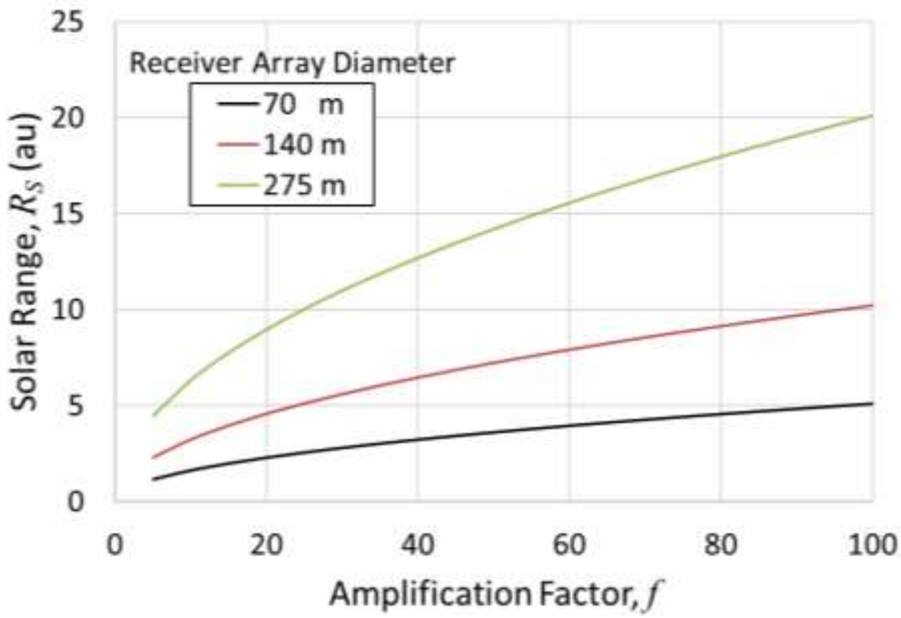

Figure 4. Solar range at which operation at the maximum EP system power can be sustained for three sizes of the PV array on the LEV. For an amplification factor of 100, full power operation at 10 MW could be maintained at 5 au (Jupiter) with a 70-m PV receiver array, 10 au (Saturn) with a 140-m array, and 20 au (Uranus) with a 275-m array.

LEV acceleration as a function of distance from the LTV, for the cases in Fig. 3, are plotted in Fig. 5. The LEV acceleration initially increases as the propellant is consumed at constant power to the EP system. The peaks in these curves represent the range at which the maximum input power to the EP system can be maintained. Beyond these peaks the LEV's PV array intercepts a smaller fraction of the laser beam power as the laser beam spreads out and the power to the EP system decreases accordingly. Prior to these peaks the laser beam is defocused so that the PV array only collects the power required to operate the EP system at full power (and to power the rest of the spacecraft).

2. Laser Location

A preliminary, qualitative assessment was made of where to place the laser transmit vehicle is given in Table 4. Based on this rough assessment there are five locations that may be approximately equivalent: Earth-Moon L2, Sun-Earth L1 or L2, and Earth Leading or Trailing orbits. All of these locations are at solar ranges of approximately 1 au. Assuming the transmit laser is solar powered, the Sun-Earth L1/L2 and Earth Leading/Trailing orbits require the solar array for the transmit laser to rotate once per year to remain pointed at the sun while the laser itself is pointed at the receiver vehicle as indicated in Fig. 6.

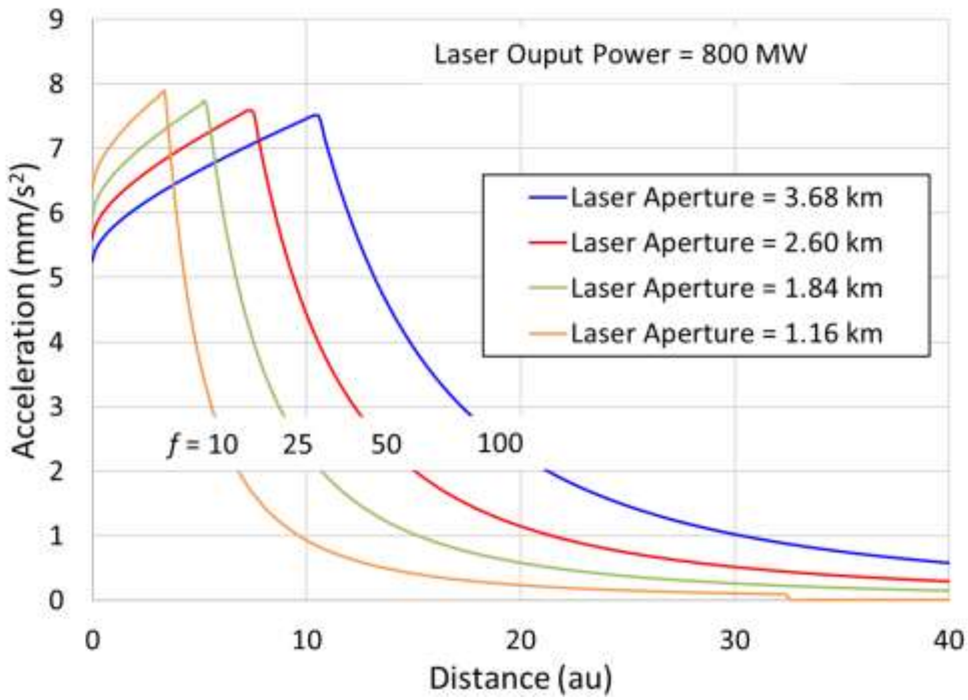

Figure 5. Example: LEV accelerations peak at between 7 and 8 mm/s^2 and then decrease at larger ranges as the laser beam spreads out and the LEV's photovoltaic array intercepts a smaller fraction of the beamed power.

Table 4. Qualitative assessment of possible locations for the transmit laser.

Location	Duty Cycle	Relative Cost	Coverage
Earth Surface	50%	$	one hemisphere
Low Earth Orbits	50%	$$	one hemisphere
Moon Surface	50%	$$$$$$	E/W hemisphere
Moon Poles	100%	$$$$$$$	N/S hemisphere
Moon Orbits	100%	$$$$	moderate exclusions
Earth-Moon L2	100%	$$$	moderate exclusions
Moon Pole-Sitter Orbit	100%	$$$$$	moderate exclusions
Sun-Earth L2/L1	100%	$$$	small exclusions
Earth Leading/Trailing	100%	$$$	almost 100%
Venus Orbits	100%	$$$$$	small exclusions
Near Earth Asteroid	~100%	$$$$	almost 100%

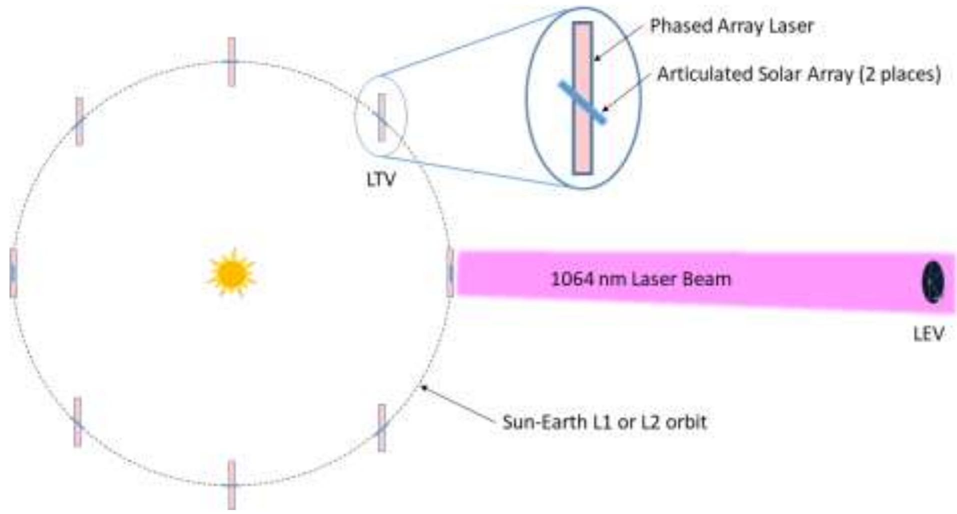

Figure 6. Illustration of the transmit laser at Sun-Earth L1/L2 indicating that the solar array that powers the transmit laser must articulate completely around once per year while the laser remains locked on to the LEV.

The rate at which the attitude of the transmit laser must vary to remain locked on to the LEV is a function of the distance to the LEV. For example, the angular rate for the transmit laser located in space at 1 au is given in Fig. 7 for an LEV at Jupiter. The maximum rate indicated in this figure is less than 0.25 degrees/day. This rate decreases to 0.12 degrees/day to power an LEV at Saturn, 0.06 degrees/day at Uranus, and less than 0.04 degrees/day at Neptune.

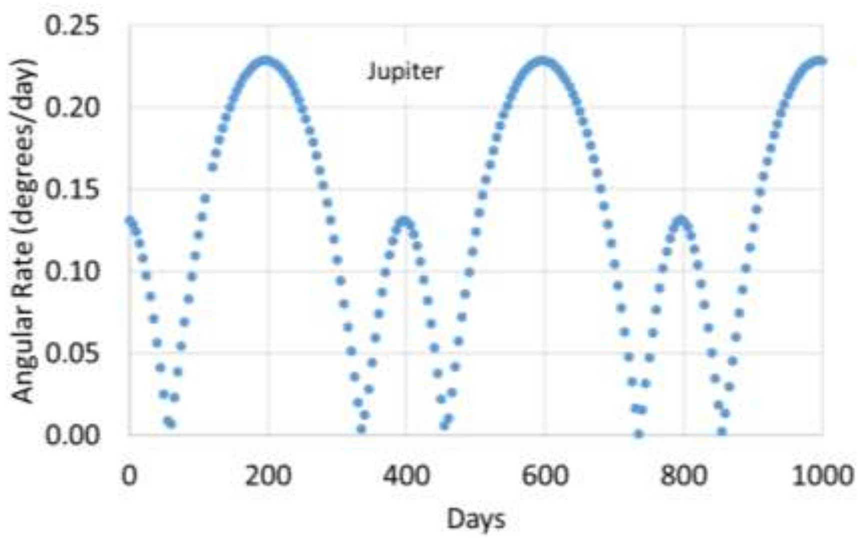

Figure 7. Angular rate required of the transmit laser vehicle to point continuously at an LEV at Jupiter.

3. Phased-array Laser

A phased-array laser is the only viable approach for the development of a kilometer-scale laser[11]. Such a laser is essentially an ideal adaptive optical system with every sub element being electronically phase controlled. The phase control does not require absolute phase matching, rather phase control is only required to within the coherence length of the amplifiers. This is vastly different than a large aperture telescope where absolute path matching is required. This means the phase control is not a mechanical (stiffness) requirement as it would be on telescope but a servo loop control bandwidth issue. This dramatically simplifies the design of the optical structure and significantly reduces the stiffness requirement. Nevertheless, the development of a space-based, large-aperture, phased-array represents substantial technological challenges across multiple fronts including: photonics, optics, structural metrology, and stability.

The kilometer-scale, phased-array laser needed for the DEEP system architecture requires a very lower power per sub element making it ideally suited to benefit from the growth in integrated photonics. Integrated photonics is a semiconductor-based industry that has the potential to drive prices down dramatically as it has for solid state lighting. The exponential growth in performance and price reduction of integrated photonics have doubling times that are currently about 1.5 years (based on the last 25 years) which is similar to "Moore's Law" in the semiconductor electronics industry. The current cost of high power (kilowatt) fiber amplifiers (Yb based at 1.06 microns) is about $50-$100/Woptical in modest quantities. But, this is for power levels that are much higher and coherence lengths much smaller than we need. In contrast, the cost of solid state GaN based LED's is already about $0.1/Woptical in large quantities and dropping. For our propulsion architecture to be practical, low cost fiber and semiconductor based amplifiers with a cost approaching the current solid state lighting costs is necessary. Even at 100x the current solid state lighting costs (or $10/Woptical), the laser amplifier costs for an 800 MWoptical laser would not be the dominant cost. While building a kilometer scale space based array is a formidable challenge, it is likely much more feasible than building a kilometer-scale telescope and has an "exponentially growing photonics engine" behind it.

4. Laser Pointing Accuracy

The best LEV performance is obtained when the maximum power of EP system is less than the output power of the laser. This necessarily so at large distances from the laser when the laser beam diameter is significantly larger than the diameter of the LEV PV array. Assuming a uniform power distribution in the laser beam, the ratio of the EP system power to the laser output power is simply the area of the LEV PV array times the LPC power conversion efficiency divided by the area of the laser spot size at any distance. The worst case pointing requirement occurs at the maximum distance where full power operation of the EP system can still be maintained. This occurs at the peaks of the acceleration curves in Fig. 5. The corresponding pointing accuracy, defined by the angle θ, where,

$$\theta = \tan^{-1}\left(\frac{d_S - d_{LEV}}{R}\right) \tag{9}$$

This is the pointing accuracy necessary to have the LEV be completely inside the laser beam at a distance R. The values of θ are plotted in Fig. 8. These are non-trivial pointing requirements and the implications are discussed in Section VI. At distances less than the maximum acceleration point, the laser is defocused resulting in more relaxed pointing requirements.

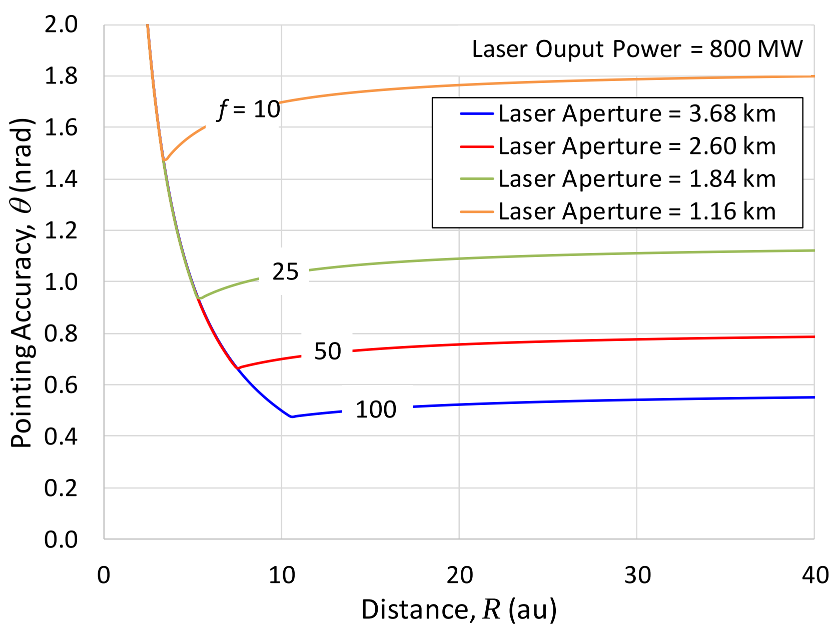

Figure 8. Example for required laser pointing accuracy that varies from 1.5 to 0.5 nrad for amplification factors from 10 to 100.

C. Laser Power Converters (LPCs)

Laser Power Converters (LPCs) convert the optical power beamed from the LTV into electricity. LPCs can achieve high optical to electrical conversion efficiency when the material bandgap is tuned to the laser frequency. Because of the absence of optical absorption in space, we consider materials with high energy bandgaps that can absorb light in the visible or near-infrared with high conversion efficiencies. For example, GaAs has a bandgap that is well matched with an 808 nm laser and InGaAs has a bandgap which is well matched with a 1064 nm laser. Maximum theoretical efficiency calculated with the Shockley and Queisser detailed balance method for a GaAs LPC is around 76% and efficiencies measurements as high as 54% have been reported[12]. Using a 1.55 μm wavelength laser, experimental conversion efficiencicy of 44.6% has been reported for a lattice matched InGaAs/InP LPC[22]. LPCs can be designed to support high power density. Measurements at JPL on a GaAs LPC device show a converted power density of over 11 kW/m^2 which is more than one order of magnitude higher than the incoming light intensity at the surface of Earth.

III. Propellant

The rocket equation dictates that to enable missions with characteristic velocities (ΔV's) up to 200 km/s—roughly 15 times that produced by the ion propulsion system on the Dawn spacecraft—a propulsion technology that produces a specific impulse approximately 15 times that of the Dawn ion thrusters is required. The Dawn ion thrusters produced a specific impulse of 3100 s at full power. Multiplying this by 15 gives a nominal required specific impulse of 46,000 s for the DEEP architecture. To achieve this specific impulse with xenon, the propellant used by the Dawn ion thrusters, would require a net accelerating voltage (aka a "Beam Voltage") of 190 kV. Implementing such an extremely high net accelerating voltage would be very difficult. The use of lithium propellant, however, would require a net accelerating voltage of just 7.5 kV for the same specific impulse. The difference in beam voltages for xenon and lithium are given in Fig. 9. In most of the performance estimates that follow, a specific impulse of 40,000 s is assumed, corresponding to a beam voltage with lithium of 5.7 kV.

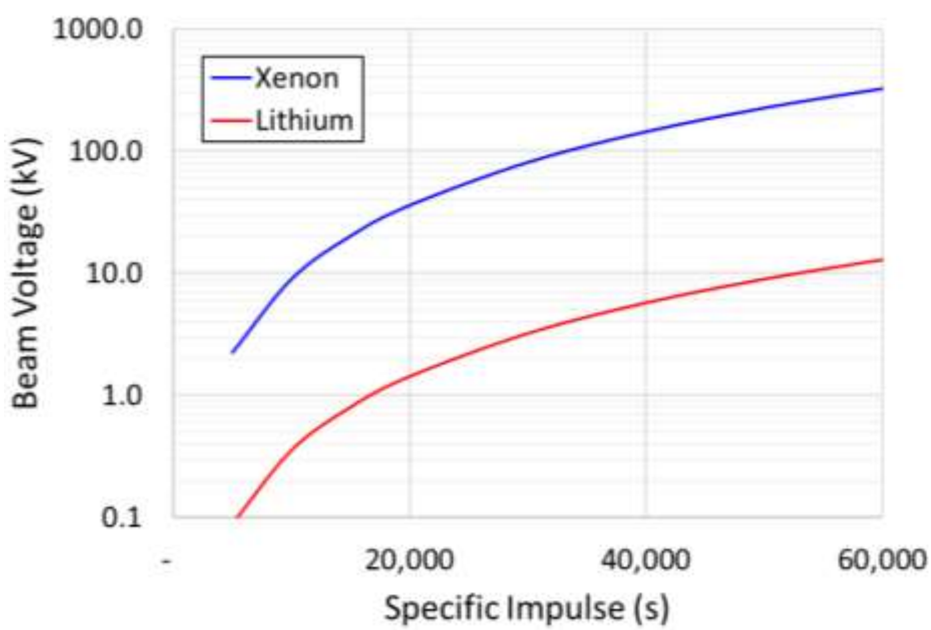

Figure 9. The use of lithium instead of xenon enables gridded ion thruster operation at a specific impulses in the range 40,000 to 60,000 s with beam voltages of 6 to 12 kV. The corresponding beam voltages with xenon would be 140 to 320 kV.

A possible configuration of the LEV would include two groups of five thrusters each, with these thruster groups on opposite sides of the spacecraft. For a maximum input power to the EP system of 10 MW, each group of would have to process 5 MW or 1.0 MW per thruster. The variation of beam current and beam voltage with *Isp* is given in Fig. 10. For an *Isp* of 40,000 s, each thruster must operate with a beam current of 175 A at 5.7 kV. If one thruster fails during the full power phase of the mission, the remaining thrusters on that side would have to operate a 25% higher power to compensate.

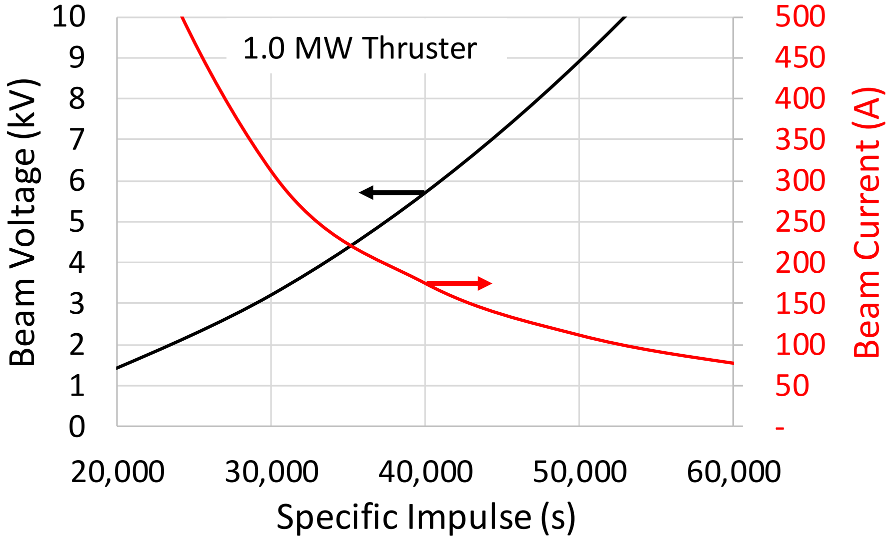

Figure 10. A 1.0-MW, lithium-fueled, gridded ion thruster operating at a specific impulse of 40,000 s would require a beam current of 175 A. While this is much higher than state of the art ion thrusters, the ability to cryopump the lithium exhaust at room temperature makes testing such a thruster feasible.

IV. Propulsion Subsystem

An essential feature for achieving a very high spacecraft speed is to minimize the specific mass of the LEV. We divide the LEV into the following major components and then assess the specific mass of each component: the photovoltaic array, the electric propulsion subsystem (which includes the power distribution system), and the spacecraft structure. These are the most massive components of the vehicle. The rest of the spacecraft subsystems are assumed to be allocated to a mass consistent with a conventionally-sized spacecraft such as New Horizons[23].

A. Electric Propulsion Subsystem Scaling

The electric propulsion subsystem consists of the thrusters, the power processing units (PPUs), the lithium feed system, and the harnessing between the components and to the photovoltaic array. We developed point designs for EP system power levels ranging from 10 to 30 MW and beam voltages of 6 to 12 kV (corresponding to specific impulses of 42000 to 59000 s) and estimated the masses of the individual components in order to determine the total system mass and specific mass over this trade space. The process involved first defining the number, size, and discharge power of the engines in a given system point design, then applying mass models for the components. The assumptions in the mass models and the resulting specific masses are described below.

The engines were sized based on the total power, beam voltage, and reasonable assumptions for the engine design. The total beam current was calculated from the total beam power level and voltage. The total beam area was then calculated assuming an average beam current density of 8.3 mA/cm^2, which was chosen based on an ion optics design using the CEX2D code[13] with an electric field of 2500 V/mm between the grids. This electric field is similar to state-of-the-art thrusters. This current density is about 50% of the maximum current density that can be extracted from the grids, and preliminary erosion estimates indicate this should provide the required lifetime. For the lower subsystem power levels, we assumed the power was processed in ten engines, which resulted in ion beam diameters on the order of 1 to 2 m. For some of the higher power levels the number of active thrusters was increased to as high as 20 so that the beam diameter did not exceed 2 m. This is a size that can reasonably be fabricated and maintain the required grid gap. Discharge power was calculated assuming a beam ion production cost of 200 W/A. This is a conservative value based on a model of a lithium discharge chamber operating at high propellant utilization efficiency[14].

The engine mass was assumed to scale with beam area and the NSTAR, NEXT, and NEXIS ion thrusters[15-18] were used to determine the constant of proportionality. These engines have power levels ranging from 2.5 to 20 kW and beam areas up to 0.25 m^2. The masses of the NSTAR, NEXT, and NEXIS thrusters are well represented by a line with a slope of 146 kg/m^2. We anticipate that the high power lithium engines will work with discharge chamber designs similar to the other thrusters.

The beam power is assumed to be supplied directly by the high voltage array, so the only component required in the PPU for the high voltage is an isolation switch. The power conditioning mass is dominated by the mass of the discharge power converter. The point designs developed for the propulsion system mass scaling require discharge powers of 25 to 50 kW and discharge currents of 1250 to 2500 A for a discharge voltage of 20V (which is based on the lithium discharge chamber model). High power processing units tend to scale with the square root of the discharge power. The mass of the NSTAR thruster discharge power supply[16] (designed for 16 A and 30 V), four high current commercial power supplies built for arcjet thrusters[19], and a 26 kW power supply built for the ESEX high power ammonia arcjet demonstration experiment[20] (130 V and 200 A) were fit to a curve, Eq. (9), that indicates the masses of these power supplies do tend to scale with the square root of power:

$$M_{disch} = -8 + 11.07 P_{disch}^{2} \tag{10}$$

The lithium feed system mass is based on a conceptual design originally developed for a high power lithium electromagnetic thruster[21]. This design uses high temperature valves, filters, pressure transducers, electromagnetic flow sensors and pumps, and heaters and thermal blankets for thermal management. The masses used in this paper are based on similar existing components (valves and filters, for example) or laboratory model versions of less mature components (such as the electromagnetic pumps and flow meters), but the total impact to the specific mass is less than 0.02 kg/kW.

The harness mass for a given thruster consists of the high voltage cable required to carry the beam current, the low voltage input for the discharge power converter (assumed to be supplied by a 100 V segment of the array), and the high current, low voltage PPU output cable. The cables connecting the PV array to the PPU were assumed to be 10 m long and the cables from the PPU to the engine 3 m long. We also assumed that the cable masses would be dominated by the copper conductors and that the insulation mass would be negligible. The conductors were sized so that the voltage drops were only on the order of a few volts and the dissipated power was low enough that it could be self-radiated away from the cables at moderate temperatures. This resulted in cable masses per thruster on the order of 40 kg (0.04 kg/kW for a 1 MW thruster).

The masses of the individual point designs were used to determine the total electric propulsion system mass and the specific mass for the range of system powers and beam voltages. Details for 10 MW systems are given in Table 5. The specific mass decreases with beam voltage because the beam current for a given power level is lower, resulting in smaller engines and discharge power supplies. The specific mass drops slightly with system power because of the nonlinear scaling of some components with power. These results were used with the other subsystem models in system trade studies.

B. 6-kV Output Voltage Feasibility.

The output voltage of the photovoltaic receiver array must be at least 6 kV in order to directly drive the lithium ion propulsion system to produce a specific impulse of 40,000 s. State-of-the art solar arrays are designed for nominal operating voltages of up to 160 V. Flight tests of small arrays have demonstrated operation at 300 V. Ground tests in a plasm environment have demonstrated successful operation at voltages as high as 1 kV[24]. These tests demonstrated that high output voltage operation can be achieved through proper cell separation distances/packing factors and grouting, suggesting that there is a reasonable expectation that operation at 6 kV is feasible.

V. Laser-powered Electrically-propelled Vehicle (LEV) Configuration

A conceptual configuration for the LEV shown in Fig. 12. This vehicle is dominated by the large, 110-m-diameter, photovoltaic (PV) receiver array that collects the laser power to operate the lithium-ion propulsion system. The PV array consists of lightweight structure with a PV blanket based on thin-film solar cells. The PV array outputs a voltage of 6 kV necessary to directly accelerate lithium ions to the required specific impulse of 40,000 s. An inflatable structure is shown in the Fig. 11, but this is just one possibility that has the potential to meet the areal density requirements for the PV array. An alternative configuration could use a PV array based on the lightweight structure from Gdoutos[6] as discussed earlier. A complete, standalone spacecraft is located at the center of the PV array. This spacecraft is assumed to be RTG powered to enable it to operate independently of the LEV's power and propulsion subsystems.

The lithium-fueled ion thrusters are located in two pods on opposite sides of the vehicle. Plume shields are included to minimize the deposition of unionized lithium propellant on the PV array. The pods are mounted to one-axis mechanical gimbals that enable the thruster pods to rotate about the line between the two pods. This configuration provides roll and yaw control for the LEV and importantly allow it to perform thrusting maneuvers across the laser beam as well as along it as indicated in Fig. 12. The cross-track maneuver capability is essential for the LEV to control its position within the beam.

Table 5. Specific mass details for 10 MW EP systems.

Total Power, Voltage, and Propellant Mass				
Beam Voltage (V) =	6000	8000	10000	12000
Total Beam Power (MW) =	10	10	10	10
Specific Impulse (s) =	41,000	47,500	53,000	58,000
Total Propellant Mass (kg) =	9000	9000	9000	9000
Thruster Power, Current, Voltage, and Size				
Discharge Loss (W/A) =	200	200	200	200
Discharge Voltage (V) =	20	20	20	20
Discharge Current/Beam Amp (A/A) =	10	10	10	10
Total Beam Current (A) =	1667	1250	1000	833
Total Discharge Current (A) =	16667	12500	10000	8333
Total Discharge Power (kW) =	333	250	200	167
Total Thruster Power (MW) =	10.33	10.25	10.20	10.17
Number of Thrusters =	10	10	10	10
Power per Thruster (MW) =	1.03	1.03	1.02	1.02
Average Beam Current Density (A/cm2) =	0.0083	0.0083	0.0083	0.0083
Beam Current per Thruster (A) =	167	125	100	83
Discharge Current/Thruster (A) =	1667	1250	1000	833
Active Beam Area per Thruster (m2) =	2.01	1.51	1.20	1.00
Grid Diameter (m) =	1.60	1.38	1.24	1.13
Thruster Mass				
Mass (kg) =	293	220	176	147
Specific Mass (kg/kW) =	0.284	0.215	0.172	0.144
Discharge Supply Mass				
Power per Thruster (kW) =	33.33	25	20	16.67
Mass per String (kg) =	55.91	47.35	41.51	37.19
Specific Mass based on total power (kg/kW) =	0.054	0.046	0.041	0.037
Discharge Current Harness (PPU to thruster, per thruster)				
Harness Length (m) =	3	3	3	3
Copper Resistivity (ohm-m) =	1.68E-08	1.68E-08	1.68E-08	1.68E-08
Copper Density (kg/m3) =	8940	8940	8940	8940
Discharge Current (A) =	1667	1250	1000	833
Conductor Diameter (cm) =	2	2	2	2
Anode + CC Resistance (ohm) =	0.0003	0.0003	0.0003	0.0003
Total Voltage Drop (V) =	0.53	0.40	0.32	0.27
Total Power Loss (W) =	890	501	320	223
Total Mass (kg) =	16.85	16.85	16.85	16.85
Beam Current Harness (PV to PPU, PPU to thruster, per thruster)				
Total Harness Length (m)	13	13	13	13
PV to PPU Harness Length (m) =	10	10	10	10
PPU to Thruster Harness Length (m) =	3	3	3	3
Copper Resistivity (ohm-m) =	1.68E-08	1.68E-08	1.68E-08	1.68E-08
Copper Density (kg/m3) =	8940	8940	8940	8940
Beam Current (A) =	167	125	100	83
Conductor Diameter (cm) =	0.5	0.5	0.5	0.5
Anode + NC Resistance (ohm) =	0.0222	0.0222	0.0222	0.0222
Total Voltage Drop (V) =	3.70	2.78	2.22	1.85
Total Power Loss (W) =	617	347	222	154
Total Mass (kg) =	4.56	4.56	4.56	4.56
Discharge Current Harness (PV to PPU per thruster)				
Harness Length (m) =	10	10	10	10
Copper Resistivity (ohm-m) =	1.68E-08	1.68E-08	1.68E-08	1.68E-08
Copper Density (kg/m3) =	8940	8940	8940	8940
Discharge Current (A) =	1667	1250	1000	833
Discharge Voltage (V) =	20	20	20	20
Bus Voltage (V) =	100	100	100	100
Bus Current (A) =	333	250	200	167
Conductor Diameter (cm) =	1	1	1	1
Anode + CC Resistance (ohm) =	0.0043	0.0043	0.0043	0.0043
Total Voltage Drop (V) =	1.42	1.07	0.85	0.71
Total Power Loss (W) =	475	267	171	119
Total Mass (kg) =	14.04	14.04	14.04	14.04
Total Thruster/PPU/PMAD				
Total Mass (kg)	4230	3330	2782	2412
Specific Mass (kg/kW)	0.409	0.325	0.273	0.237

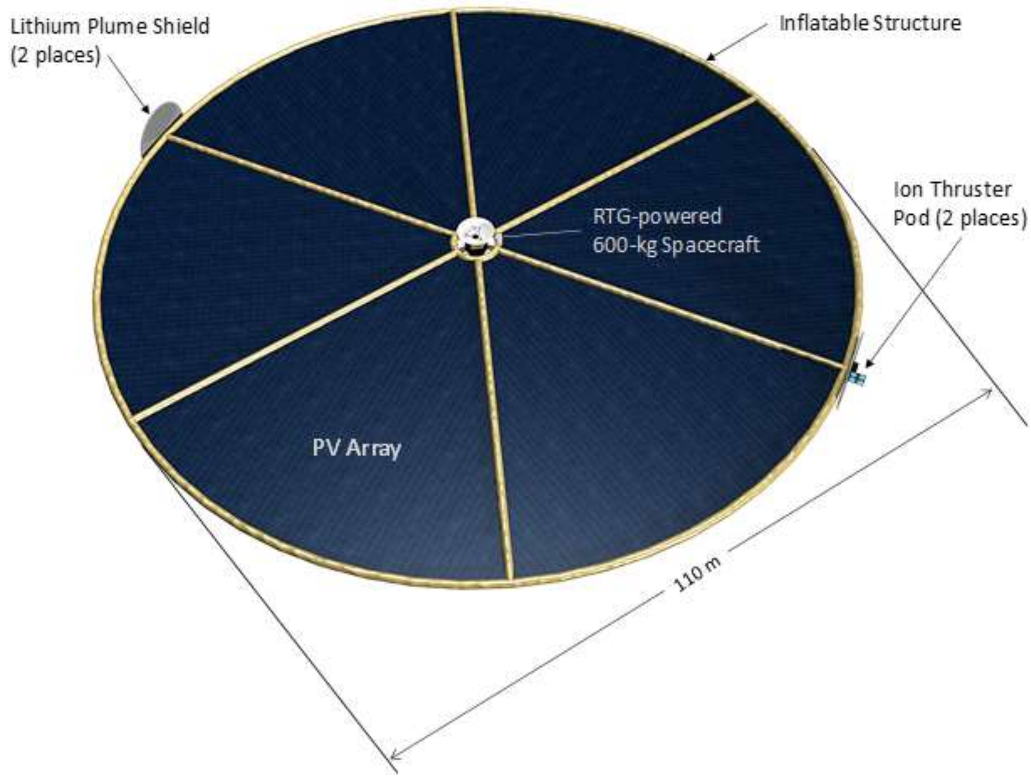

Fig. 11. Configuration of the Laser-powered, Electrically-propelled Vehicle (LEV). The vehicle includes a 110-m-diameter photovoltaic array for receiving the beamed laser power, a 10-MW lithium-fueled ion propulsion system that produces a specific impulse of 40,000 s, and a 600-kg RTG-power spacecraft located at the center of the PV array.

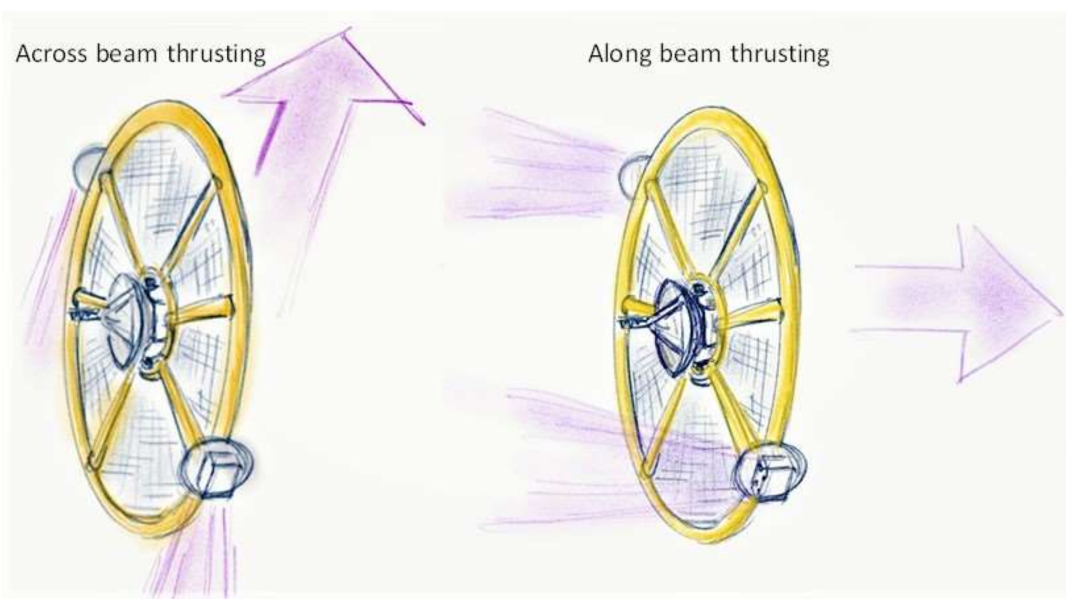

Fig. 12. Illustration of the LEV's ability to thrust across the laser beam (left) or along it (right), assuming the beam is coming in from the left side of the page normal to the surface of the LEV's PV array.

VI. Navigation

There are two key issues associated with the navigation of the laser-driven spacecraft and pointing of the laser. These issues are how to start-up the system at any range up to 40 AU (i.e., re-acquire the spacecraft by the laser beam), and how to keep the spacecraft in the beam during thrusting.

A. Startup

For startup, the laser beam must acquire the spacecraft. For the $f = 100$ case in Fig. 8, the required laser pointing accuracy is about 0.5 nrad at 10 au. The current Delta-Differential One-Way Ranging (Δ-DOR) radio navigation capability based on X-band is about 2 nradians,[25] which is not sufficient. To get the required angular position accuracy, we assume the use of a laser beacon on the LEV combined with distributed telescopes on the edges of the LTV to receive the beacon. We also require that the error between the angular measurement of the spacecraft position as determined by the distributed telescopes and the laser beam direction to be small.

The startup process is to measure the spacecraft position based on detection of the laser beacon, hit the LEV with the high-power laser and then allow time for the spacecraft to travel to the center of the beam. This requires that the spacecraft be able to measure its position relative to the laser beam center. The time required for the spacecraft to accelerate and travel from the edge of the laser beam to the center is given in Fig. 13 assuming 100% of the available thrust is used for this purpose. The one-way light time at 10 au is 83 minutes. The time required for the LEV to reach the center of the beam is 5 minutes (from Fig. 13, for $f = 100$). So the time it takes for the LEV to get to the center of the beam is a small fraction of the one-way light time for the transmit laser.

This is also the case even at 40 au where the beam spot size is much larger and the LEV acceleration is much lower. At 40 au the one-way light time is 330 minutes, and it takes about 40 minutes for the LEV to reach the beam center. The entire startup process at 40 au would take about 17 hrs beginning with the command from Earth to resume thrusting. This command would take 5.5 hrs to reach the spacecraft. The LEV would respond by transmitting the beacon laser at the LTV. It would take another 5.5 hrs for the beacon to reach the LTV. The LTV would use the beacon to determine the location of the LEV, phase lock the sub elements, and then transmit the power beam. The power beam from the LTV would take another 5.5 hrs to reach the LEV. Finally, the LEV would determine where it is in the laser beam and take up to 0.67 hrs to traverse to the center.

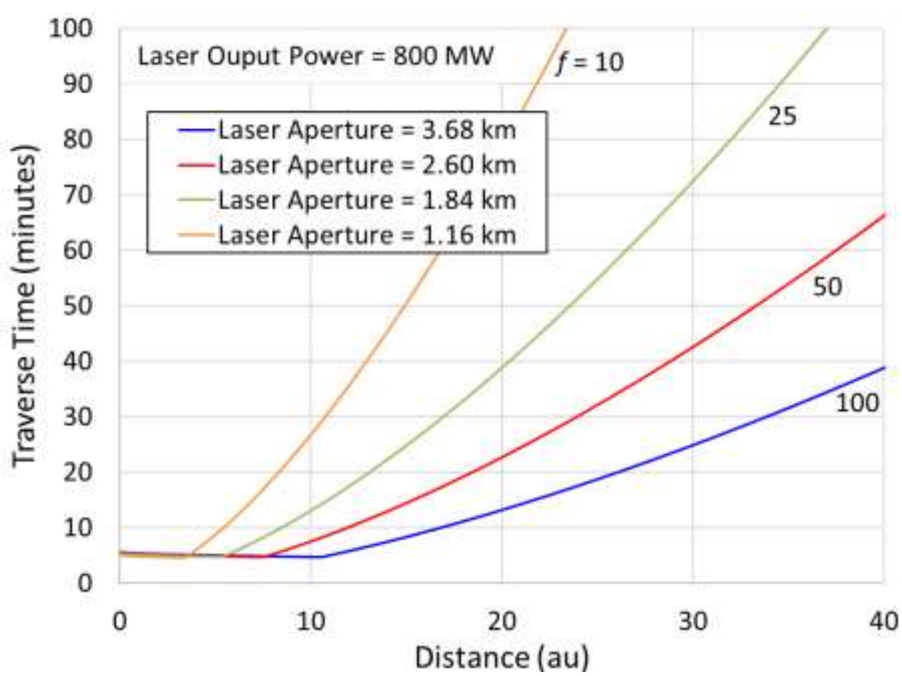

Fig. 13. Example estimates for the time it takes for the LEV to traverse (starting at rest) from the edge of the power beam to the center assuming 100% of the thrust is allocated to this traverse.

B. Thrusting

For normal thrusting three simultaneous control loops are envisioned. The first one is a short loop where the spacecraft chases the laser beam in order to stay centered within the beam. The second is a longer loop where the laser array adjusts the pointing profile with a time constant based on the round trip light time to the spacecraft. The final control loop is a much longer-term one that updates the desired thrusting profile. This final control loop would likely have a time constant of a week or a few weeks.

It is important that the spacecraft be capable of determining where it is inside the laser beam once the beam illuminates it. This will provide even more accurate information regarding the spacecraft position than that provided by the beacon. It is assumed that the LEV provides this information continuously back to the high-power laser. The spacecraft must have sufficient cross-track (cross-beam) thrust capability to stay within the laser beam. Purely cross-track thrusting is probably the most stressing case and sufficient thrust margin would need to be built in to the trajectory to prevent the spacecraft from falling out of the beam.

No missed thrust margin has been built into the performance estimates at this time. Since typical missed thrust margins are of the order 10% or so, this is not a significant impact at this conceptual stage of the concept development. After thrusting ends, however, whether planned or unplanned, it will be necessary to use the laser-array telescope measurements of the spacecraft beacon to reestablish the starting conditions.

C. Navigation Derived Requirements

Based on consideration for startup and navigation of the LEV, the following requirements for the system were developed:

1. The LTV must be capable of measuring the angular position of LEV with the same resolution as the laser beam diameter.
2. The LTV must be capable of directing the power beam with an angular error that is small compared to the measurement error of the spacecraft's angular position.
3. The LEV must provide a laser beacon to the LTV even when not illuminated by the power beam.
4. The LEV must be able to measure its position in the power beam.
5. The LEV must be able to communicate to the LTV during thrusting to provide information regarding its position within the power beam.

VII. Example Mission Capability

Several mission examples are presented below to illustrate the potential of this Directed-Energy Electric Propulsion (DEEP) system architecture. These examples are divided into two categories: Robotic Missions and Human-Scale Missions. The robotic missions are one-way rendezvous trajectories and assume the delivery of a 600-kg RTG-powered, stand-alone spacecraft to the destination. The human-scale missions assume round-trip trajectories with a 40,000 kg spacecraft (not including the laser-driven power and propulsion system), and include a 180-day stay time at the destination.

A. Robotic Mission Examples

For these examples, the LTV is assumed to transmit an 800 MW power beam at a wavelength of 1064 nm. Three different size laser apertures were investigated: 3, 4, and 5 km. The maximum EP system power on the LEV is assumed to be 10 MW with a specific impulse of 40,000 s. The LEV has a dry mass of 4200 kg not including the lithium propellant tanks or the spacecraft structure. A tankage fraction of 5% is assumed for the lithium propellant tanks (i.e., the tank mass is 5% of the total propellant mass stored). The spacecraft structure is assumed to be 4% of the total spacecraft wet mass. These masses are added to the 4200 kg mass. The 4200 kg mass includes the mass of the PV array on the LEV (which is assumed to have a diameter of 140 m with an areal density of 100 g/m^2) and the 600-kg stand-alone spacecraft as mentioned above. All of the example missions are assumed to be launched to a C3 of 0 km^2/s^2. A duty cycle of 90% is used during powered EP operations. A margin of 10% is added to the deterministic propellant mass.

Trajectory results based on these assumptions are given in Fig. 14. The trajectory paths (assuming a 3-km laser aperture) appear to be nearly straight lines between the Earth and the destination. The dashed lines indicate coast periods. The ΔV's for these missions are given in Table 2 and are approximately 170 km/s. The right side of Fig. 14 indicates the flight times as a function of the laser aperture. Rendezvous missions to Uranus with flight times of less than two years appear possible.

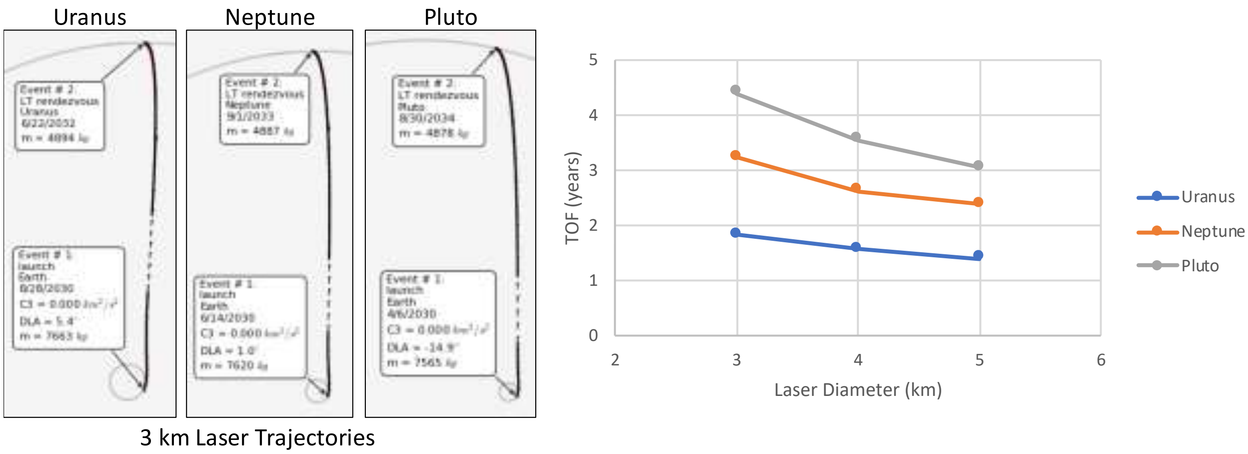

Figure 14 Short flight times for robotic rendezvous missions to the outer planets require ∆V's of approximately 170 km/s and are potentially enabled with the Direct-Energy Electric Propulsion system architecture. The mission performance indicated in this figure requires an 800 MW output power laser at a wavelength of 1064 nm with aperture diameters between 3 and 5 km to power an LEV with a 10 MW direct-drive electric propulsion system operating at a specific impulse of 40,000 s.

B. Human-Scale Mission Examples

Human-scale missions assume the delivery of a 25,000 kg habitat module, logistics of 5,200 kg per year of flight time, and a 13,300 kg spacecraft by the LEV. The LEV power subsystem PV array is assumed to have an areal density of 200 g/m2. The propulsion subsystems has a specific mass of 0.3 kg/kW wiht a specific impulse of 50,000 s. A propellant margin of 10% is added to the deterministic lithium propellant load and the tankage fraction is assumed to be 10%. Half of the logistics mass is assumed to be dropped before the return trip. The LTV laser is sized to provide an amplification factor, f = 100. Trajectory analyses were performed for maximum electric propulsion subsystem input power ranging from 10 to 70 MW.

With these assumptions, a 50-MW EP system in the Direct-Energy Electric Propulsion system architecture would enable a 2.8-year round flight time to Jupiter including a 180-day stay time at Jupiter. The trajectory is indicated in the left side of Fig. 15. The flight times for both the outbound and inbound legs is 1.15 years each. The initial wet mass of the total vehicle is 113,000 kg. The corresponding mission trajectory to Saturn is given in the right side of Fig. 15. This trajectory has a total round trip flight time of 4.0 years for a 50-MW EP system and a total initial wet mass of 141,000 kg.

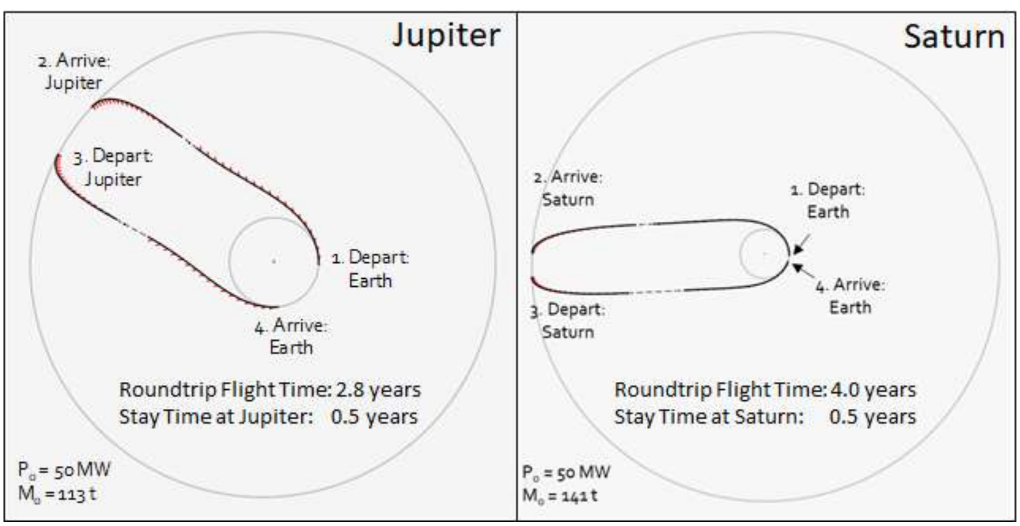

Figure 15 Human-scale missions to Jupiter and Saturn with reasonable roundtrip flight times may be enabled by the Direct-Energy Electric Propulsion (DEEP) system architecture.

C. Comparison with Laser-Sails

The alternative to converting the laser power to electrical power to operate a high-*Isp* EP system is to use the high-power laser to drive a laser sail. One way to evaluate these systems is to compare thrust levels. The thrust of a laser-sail is given by,

$$T_{sail} = \frac{P_{laser}(1+\varepsilon_r)}{c} \tag{11}$$

where P_{laser} is the laser power, ε_r is the reflection (=1 for complete reflection), and c is the speed of light. The thrust for an electric propulsion system is,

$$T_{EP} = \frac{2\eta P_{laser}\varepsilon_{LPC}}{I_{sp}g} \tag{12}$$

where η is the EP system efficiency, assumed to be 0.97 for operation at an $Isp \geq 40{,}000$ s. Combining these two equations gives the ratio of thrusts for a laser-driven electric propulsion system and a laser-drive sail as,

$$\frac{T_{EP}}{T_{sail}} = \frac{2\eta\varepsilon_a}{(1+\varepsilon_r)}\left(\frac{c}{I_{sp}g}\right) \tag{13}$$

Significantly, this ratio does not depend on the laser power, but is primarily a function of the ratio of the EP system specific impulse to the speed of light. This equation is plotted in Fig. 16 assuming $\varepsilon_a = 0.7$, $\varepsilon_r = 1$, and η varies with specific impulse from 0.5 at 2000 s to 0.97 at 60,000 s. This figure indicates that at a specific impulse of 40,000 s the electric propulsion system produces that thrust level that is approximately 500 times greater than that of a laser sail. This means that EP-based vehicles could be 500 times as massive as a laser-sail vehicle for the same acceleration.

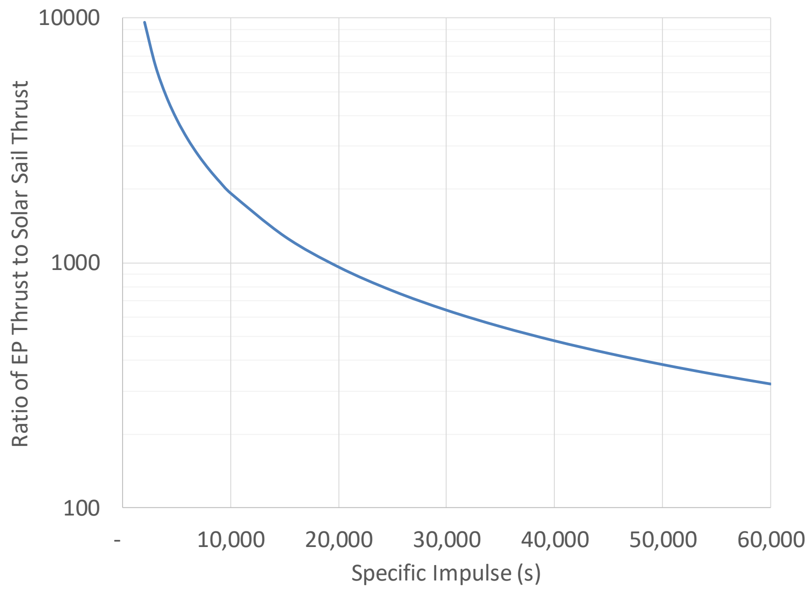

Figure 16. At a specific impulse of 40,000 s, an electric propulsion based vehicle produces approximately 500 times the thrust of a laser-propelled sail for the same laser power.

VIII. Conclusion

The specific masses of current and near-term power sources for electric propulsion systems include ~7 kg/kW for state-of-the-art, flexible-blanket solar arrays; 40 kg/kW for large-scale fission reactors; 150 kg/kW for the 10-kW version of the Kilopower fission reactor; and 250 kg/kW for radioisotope thermoelectric generators (RTGs). But, rapid transportation throughout the solar system needs power and propulsion system specific masses less than 1 kg/kW. The Directed-Energy Electric Propulsion (DEEP) system architecture is a potential approach to overcome the specific mass limitations of existing power source technologies. The DEEP architecture proposes to use a kilometer-scale, space-based laser to beam power over distances up to 40 au in order to increase the power density of photons relative to solar insolation by an amplification factor between 10 and 100. The laser power is collected by a 100-m-scale photovoltaic array on the receiver vehicle whose laser power converters are tuned to the laser frequency with a conversion efficiency of at least 50%. This approach effectively decreases the specific mass of the photovoltaic array by the amplification factor. Advances in deployed structures and thin-film photovoltaic cells suggest that photovoltaic arrays with areal densities in the range 100 to 200 g/m^2 may be possible, which would be an order of magnitude improvement over the current state-of-the-art. Coupling of these advanced photovoltaic systems with laser-driven incident power amplification has the potential to make megawatts of power available onboard the receiver vehicle for electric propulsion. Direct-drive, lithium-fueled, gridded ion thruster systems complete the basic architecture of the receiver vehicle to create a laser electric propulsion vehicle (LEV). The use of direct-drive largely eliminates the need to process the power from the photovoltaic array. This is critical since it eliminates the mass of the power processing hardware and eliminates the mass of the radiators necessary to reject the waste heat from the power conversion inefficiency. Both of these features are necessary to achieve specific masses < 1 kg/kW. Finally, lithium propellant is used in a gridded ion thruster to enable ultra-high specific impulses at reasonable net accelerating voltages. With lithium, a specific impulse of 40,000 s could be achieved at a voltage 5.7 kV. To make the direct-drive system work then, requires the photovoltaic array to output power at this voltage directly to the ion thrusters. This is well beyond the current state-of-the-art for photovoltaic arrays in a plasma environment, but the physics of high-voltage photovoltaic array operation in a plasma environment is relatively well understood. Scaling relations indicate that to achieve amplification factors in the range 10 to 100, a kilometer-scale laser with an output power of 100's of MW is required. With an amplification factor of 10 and the LEV system described above, spacecraft characteristic velocities of up to 120 km/s may be achieved. Amplification factor of 100 are necessary to achieve characteristic velocities up to 200 km/s.

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