Electric Sail Propulsion Modeling and Mission Analysis

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Abstract: The electric sail[1,2] is a new propulsion concept that uses the solar wind momentum flux for producing thrust. Like the more conventional solar sail, it allows a spacecraft to perform high-energy orbit transfers without a need for reaction mass. The electric sail could accelerate small (10-100 kg) payloads to substantial final speeds, larger than that are possible with conventional (either chemical or electric) propulsion systems. It could also provide a lightweight propulsion alternative for larger payloads (100-1000 kg). With reference to the latter choice, in this paper we provide quantitative estimates for optimized Earth-Mars transfers for both an electric sail and a solar electric propulsion (SEP) system. To facilitate the comparison we use the same specific power assumption for both systems. For this case study it is found that the electric sail and SEP performance are rather similar. The electric sail has higher payload fraction than SEP, whereas SEP tends to have somewhat shorter mission times.

Nomenclature

α	= angle between thrust and the Sun-spacecraft line (coning angle)
β_0	= initial specific power of SEP system (kg/W)
e	= electron charge
ϵ_0	= vacuum permeability
K	= numerical coefficient (nominally 3.09)
λ_{De}	= solar wind Debye length
m_e	= electron mass
m_p	= proton mass
m_f	= spacecraft final mass
m_0	= spacecraft initial mass
n_0	= solar wind plasma density
r_0	= two times solar wind Debye length
r_w	= wire radius
r_w^*	= effective electric radius of tether
T_e	= solar wind electron temperature in energy units
θ	= angle between radial and tether direction
v	= solar wind speed

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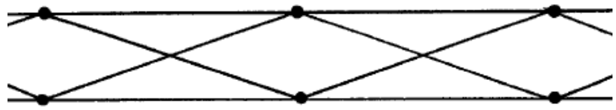

Figure 1. Four-line Hoytether geometry.

I. Introduction

The electric sail[1,2] uses long and thin conducting tethers which are kept at a high positive potential by an onboard (typically solar-powered) electron gun. The positively charged tethers repel the solar wind protons, thereby extracting momentum from them. As a byproduct they also attract electrons, but since the solar wind plasma and also the resulting electron sheath surrounding the tether is highly collisionless, the number of electron actually hitting the tether is small provided the tether is thin. Typically the tether is composed of four 20-μm diameter interconnected redundant wires (Fig. 1) for micrometeoroid-resistance and the tethers are kept at $\sim$ 20 kV potential, which requires $\sim$ 500 W of electric power for 50-100 tethers of 20-30 km length.

In current scenarios the tethers are kept stretched by the centrifugal force, i.e. by rotating them. Electric means of attitude control of the tether spinplane are available via potentiometers that exist between the spacecraft and each tether. In fact, because the force acting upon the tether depends on its potential, each force can be individually controlled using the potentiometers. The angular momentum needed for spinning up the system can be obtained from conventional propulsion units placed at ends of booms or directly from the solar wind ("windmill" approach). The windmill approach requires that periodic changes are made in the tether length during the deployment phase, i.e. one must also partly reel in the tethers at some points. It is also possible to initiate the spin using two identical spacecraft spun in opposite directions and connected by an axis where a torque is applied using a small electric motor ("Siamese Twins" approach).

The thrust per unit length of an electric sail tether is given by

$$\frac{dF}{dz} = \frac{K m_p n_0 v^2 r_0}{\sqrt{\exp\left[\frac{m_p v^2}{eV_0}\log(r_0/r_w^*)\right] - 1}} \tag{1}$$

where K is a numerical constant (nominally $K = 3.09$), m_p is the proton mass, n_0 is the solar wind plasma density, v is the solar wind speed, e is the electron charge, V_0 is the tether potential, r_w^* is the effective electric radius of the tether and r_0 is twice the solar wind Debye length, that is

$$r_0 = 2\lambda_{De} = 2\sqrt{\frac{\epsilon_0 T_e}{n_0 e^2}} \tag{2}$$

where ϵ_0 is the vacuum permeability and T_e is the solar wind electron temperature in energy units. Typically, on average, solar wind conditions at 1 AU distance from the Sun and with $V_0 \approx 20$ kV correspond to $dF/dz = 50 - 100$ nN/m. The effective electric radius of the tether depends on the physical wire radius r_w, the spacing between the wires and the tether geometry. An order of magnitude estimate is $r_w^* \sim \left(h^2 r_w\right)^{1/3}$ where h is the spacing between the parallel wires.

Only the component of the solar wind which is perpendicular to the tether produces thrust, the flow which is parallel to the tether has no effect. For this reason, a simple geometrical consideration shows that for a set of spinning tethers inclined at an angle θ with respect to the solar wind flow, the net thrust is directed at an angle (the coning angle) $\alpha = \theta/2$. Accordingly, the thrust vector can be varied from the radial solar wind direction up to (probably) $\approx$ 30 degrees. Coning angles which are significantly larger than this are likely to be impractical because of the thrust reduction at high α and also, at least conceivably, due to some mechanical instabilities.

The electric sail thrust can be easily controlled (throttled) by modifying the electron gun current or voltage. For example, coast arcs in orbital planning can be implemented simply by turning the electron gun off. Throttling could be used also to counteract the intrinsic variations of the solar wind. For each solar wind

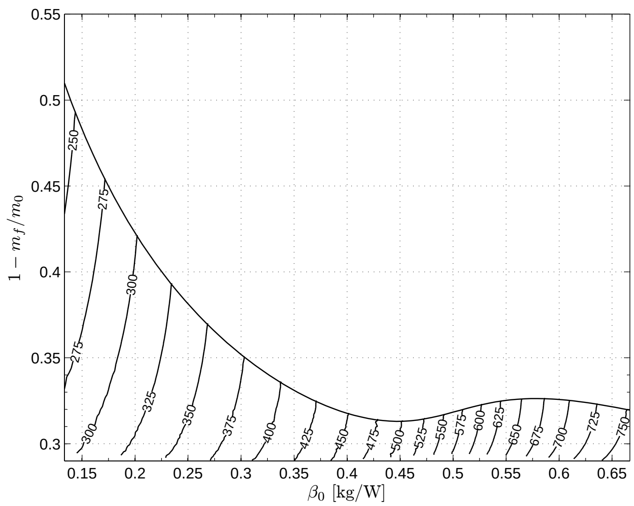

Figure 2. Iso-contour lines of mission time (in days) for SEP as a function of β_0, i.e., the initial mass versus maximum 1 AU power of the SEP system, and propellant fraction $1-m_f/m_0$. The calculation is for Earth-Mars circular, coplanar transfer.

condition, the electron gun hardware and power processing unit set some maximum thrust which cannot be exceed. The thrust can be throttled between zero and the maximum at will, however.

The purpose of this paper is to characterize the electric sail performance and analyze the capabilities of this propulsion system in performing interplanetary missions. As a case study we consider an Earth-Mars two-dimensional transfer between circular and coplanar orbits and make use of an explicit comparison with respect to a classical solar electric propulsion (SEP) system.

II. Results

A. SEP

We start the discussion with a model orbital calculation made for a SEP spacecraft. Figure 2 shows various iso-contour lines (representing the mission time in days) as a function of β_0, i.e., the ratio between the initial total spacecraft mass and the maximum available power (in contrast to the e-sail in this case, of course, the value of β must be calculated at the initial time because the spacecraft mass varies during the mission). The enveloping line corresponds to the minimum-time trajectories. Here m_f is the final spacecraft mass, i.e. the dry mass, and m_0 is the initial (dry plus propellant) mass. For example, for a SEP technology level corresponding to a specific power $\beta_0 = 0.3$ kg/W, the minimum-time Earth-Mars transfer takes 375 days, requiring 35% propellant fraction.

Figure 3 shows the dry mass fraction m_f/m_0 for a specific value of $\beta_0 = 0.25$ kg/W which corresponds to the overall specific power value of SMART-1. With this value the minimum-time trajectories for an Earth-Mars missions have been simulated and the corresponding results have been summarized in Fig. 3. The black point represents the condition of minimum transfer time and is compatible with the results shown in Fig. 2. Note that if one accepts to increase the mission time, there is a corresponding gain in the ratio between m_f and m_0. However there is an upper limit for this gain, corresponding to a mission time greater than about 390 days. This means that $m_f/m_0 = 0.7217$ is the maximum value of dry mass obtainable for this transfer (this value corresponds to the minimum required propellant mass).

B. Electric sail

For the electric sail case we have calculated elsewhere[3] the average acceleration $a_\oplus$ at 1 AU, corresponding to the same specific power $\beta = 0.25$ kg/W as assumed above for the SEP and using a tether wire material density of 4000 kg/m^3 and a 4-fold Hoytether structure. The voltage was optimized so that the total mass (spacecraft body plus tethers) was minimized. A functional fit was performed to the result, giving the

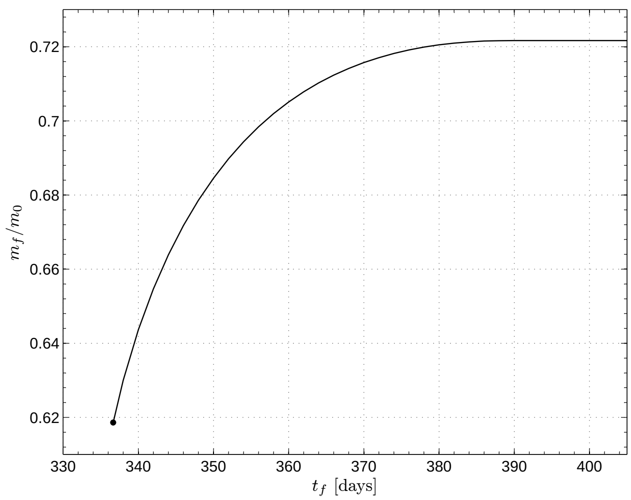

Figure 3. Dry mass fraction for SEP corresponding to specific power $\beta_0 = 0.25$ kg/W as function of mission time for minimum-time Earth-Mars circular, coplanar transfer.

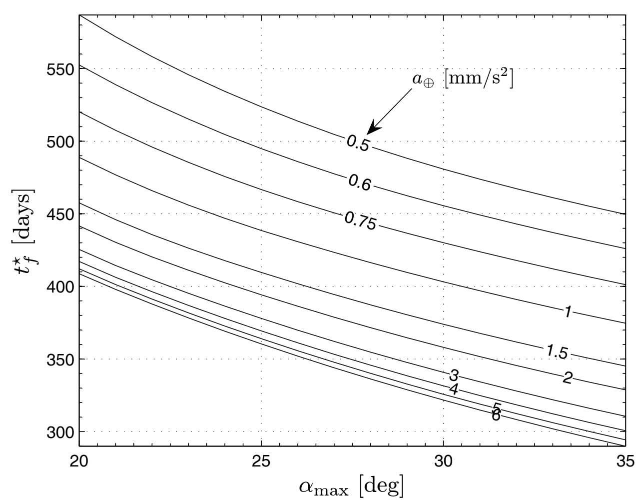

Figure 4. Minimum electric sail Earth-Mars transfer time as function of maximum usable coning angle $\alpha_{\max}$ for different values of acceleration at 1 AU.

following compact formula,

$$a_\oplus = \frac{a_\odot\,(1-\eta)}{c_2\,r_w^2 + c_1\,r_w + c_0} \tag{3}$$

where η is the payload fraction, $a_\odot$ is the solar acceleration at 1 AU ($a_\odot = 5.95$ mm/s^2) and the fitted coefficients are $c_0 = 1.6752 \times 10^{-2}$, $c_1 = 0.28095\mu\text{m}^{-1}$ and $c_2 = 4.568 \times 10^{-3}\mu\text{m}^{-2}$.

Figure 4 shows the minimum Earth-Mars transfer time as a function of the maximum allowed coning angle $\alpha_{\max}$ for different values of electric sail acceleration at 1 AU, $a_\oplus$. In the calculations, a radial thrust dependence $\sim (1/r)^{7/6}$ (which is characteristic of the electric sail) was assumed.

C. SEP-electric sail comparison

For example, selecting $a_\oplus = 0.5$ mm/s^2 and a wire radius $r_w = 10\mu$m, one can obtain from 3 that with the electric sail the payload fraction is $\eta = 0.723$. This is very similar to the maximum dry mass fraction obtainable with a SEP, using the same specific power technology level parameter β. Since this is not yet the payload fraction but the dry mass fraction in the case of SEP, the SEP would then actually have a smaller payload fraction than the electric sail in this case. On the other hand, the electric sail transfer time would be longer (480 days for $\alpha_{\max} = 30^{o}$) than the SEP transfer time (390 days).

III. Conclusions

The electric sail is a potentially revolutionary propulsion technology which may accelerate small (10-100 kg) payloads to 50-100 km/s outgoing speeds. In this paper the aim was, however, to study the suitability of the electric sail for a traditional Earth-Mars transfer problem, using a conservative specific power assumption of 0.25 kg/W based on heritage flight hardware (SMART-1). Because the ability to control the thrust vector coning angle in the electric sail is limited, the Earth-Mars transfer presents a "difficult" case for the electric sail. Nevertheless, our calculations show that the electric sail would perform about equally well or better than a classic SEP system built with the same overall technology level, parameterized by the specific power β. Specifically, the electric sail would tend to achieve a higher payload fraction than SEP, at the expense of somewhat longer transfer time.

A more realistic comparison between SEP and electric sail performance should involve decomposing the mass budgets of each spacecraft into more finely resolved parts. Of these parts some are common to both types of spacecraft (e.g., solar panels, power processing unit), some are unique to SEP (e.g., ion thruster plus valves, xenon tank) and some are unique to the electric sail (e.g., tethers and their reels and the electron gun). For a fair comparison, the common parts should obviously be parameterized with the same technology level as we have done here. The bottom line is that at this stage at least, there is no reason to abandon either of the technologies (SEP and electric sail) for planetary transfer applications.

Acknowledgements

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