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Optimal trade studies of interplanetary electric propulsion missions

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Abstract

The aim of this paper is to investigate various aspects of an interplanetary electric propulsion mission. To this end, the problem of minimum-time and minimum-fuel-time-fixed interplanetary rendezvous are first discussed. A circle-to-circle two-dimensional transfer is assumed and the problem is solved using an indirect approach. Key features of the analysis are the use of a realistic thruster model. In particular, a second-order polynomial approximation is used for modelling both the thrust and the propellant mass flow rate as a function of the thruster input power. As the mission time and the propellant mass required to reach the target orbit are two classical conflicting requirements, a number of simulations have been performed with different values of mission times and maximum input power supplied by the power processing unit. The numerical examples show that trade-off studies can be conducted with relative ease and with moderate computational effort. The simulation data have then been collected in plots and graphs that are useful for preliminary mission analysis. They are of particular importance for allowing the designer to know exactly what technological level is required before a mission is feasible.

Keywords: Solar electric propulsion; Mission analysis; Trajectory optimization; Solar system exploration

1. Introduction

The striking advances in solar electric propulsion (SEP) technology have much reduced the costs and risks of using ion thruster as a primary propulsion system. The potential benefits offered by high specific impulse thrusters, when compared to traditional (chemical) propulsion, include a significant reduction in the amount of propellant required to reach distant destinations from Earth [1–6]. This may substantially reduce the overall launch mass, thereby enabling robotic solar system exploration missions.

It is known that the weakness of SEP technology is in the low levels of acceleration it provides and in the reduced solar irradiance available for photovoltaic power generation at the outer reaches of the solar system. Nevertheless, these drawbacks can be circumvented by a suitable design that allows the SEP system to operate efficiently for long periods using a wide range of input powers.

SEP technology is especially interesting for those missions requiring large changes in orbital energy [3,4]. Also, SEP missions to comets and asteroids may be accomplished without using complex gravity assisted trajectories as those needed for ballistic missions [5].

The problem of SEP based mission design is particularly attractive also from a theoretical viewpoint, because the thrust produced is very small and the engines are required to operate during most of the trajectory.

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This characteristic makes it a difficult task to find optimal trajectories and justifies the number of papers dedicated to the subject [7–10].

The formulation of an optimal control problem for a SEP based mission requires the inclusion of an adequate thruster model. Kechichian [11] discusses the minimum-fuel orbit transfer with variable, bounded specific impulse I_{sp}. Other contributions and extensions may be found in Carter and Pardis [12], Vadali et al. [13], Nah et al. [14] and the references therein. All these papers are based on the simplified assumption of constant thruster efficiency.

Only in a few cases realistic SEP throttle performance has been incorporated into the trajectory optimization model. Williams and Coverstone-Carroll [3,4] and Sauer [6] used a polynomial approximation to describe thrust and propellant mass flow rate as a function of the thruster input power to the power processing unit (PPU); however, they do not provide an analytical expression for the optimal control law.

A more refined solution has been recently presented by Mengali and Quarta [15] who, using a best fit polynomial to model the variation law of thruster efficiency with I_{sp}, have solved the problem of minimum-fuel interplanetary orbit transfer and derived an analytical expression for the corresponding optimal control law. The approach is based on a three-dimensional formulation and, as such, is not well suited for preliminary mission analysis and trade-off studies. In fact, mission designers need a quick and accurate way to estimate transfer times and propellant requirements for space missions whose design parameters are constantly in flux. Therefore, the only practical approach is to resort to a two-dimensional problem. Usually, this analysis is performed under the simplified assumption of a constant thrust. On the contrary, in this paper a more accurate thrust model is employed. More precisely, we investigate a circle-to-circle two-dimensional transfer problem, assuming a second-order polynomial approximation for thrust and propellant mass flow rate as a function of the thruster input power.

The aim of this paper is twofold: On one side we study the minimum-time problem and the minimum-fuel-time-fixed rendezvous transfer. In both cases, using an indirect approach, we are able to find the optimal control law. The second paper's aim is to address trade-off studies. The idea of generating simple plots for preliminary mission analysis is by no means new, even if there has been limited focus on SEP trade-off studies [16,17]. Therefore, we have systematically performed a number of simulations with different values of mission times and maximum input power supplied by the PPU. The simulation data have then been collected in plots and graphs that are useful for preliminary trade-off mission analysis.

2. Problem formulation

In this paper we consider a SEP spacecraft whose performance, in terms of available thrust T and propellant mass flow rate $\dot{m}_p$, are a function of the total electric thruster input power P which is supplied to the thruster by the PPU. Both the thrust and the propellant mass flow rate depend on P, where P, by assumption, is a control variable that can be continuously adjusted in the range $P \in [P_{min}, P_{max}]$. The endpoints of the power variation interval depend on the thruster operational characteristics. The assumption concerning the continuity of P approximates the actual thruster behavior which allows the input power to be set at a finite, albeit large, number of operating points (usually on the order of one hundred). We assume that the maximum total electric thruster input power is available from the power processing unit at any time instant during the mission. This amounts to stating that the solar array are designed using the worst case conditions in terms of electric power generation (that is, using the maximum distance from the sun and the end-of-life conditions). Therefore, unlike the models discussed in Refs. [17,18], here P_{max} has not been optimized. Nevertheless the effect of different maximum input power values on mission performance is actually taken into account at the end of the paper when some trade-off studies are performed.

The relationships $T = T(P)$ and $\dot{m}_p = \dot{m}_p(P)$ are obtained through a best-fit of experimental data and are expressed using a suitable polynomial approximation. For a preliminary mission analysis a second-order polynomial approximation is adequate, that is [19]:

$$T = f_2 P^2 + f_1 P + f_0 \tag{1}$$

$$\dot{m}_p = g_2 P^2 + g_1 P + g_0 \tag{2}$$

where f_i and g_i (with $i = 0, 1, 2$) are constant coefficients. Table 1 shows the coefficients corresponding to the PPS-1350-G Hall-effect thruster, mounted on the ESA SMART-1 spacecraft [19,20]. Assuming that all of the PPU design input power is always available, the boundary values of P are $P_{min} = 0.46\,\mathrm{kW}$ and $P_{max} = 1.5\,\mathrm{kW}$. Note that, as long as $P \in [P_{min}, P_{max}]$, the polynomials in Eqs. (1) and (2) take strictly positive values. Accordingly, the thrust T cannot be zeroed by simply setting the input power to zero, as suggested, for example, in the model discussed in Ref. [21]. Therefore, we introduce a further control, the switching parameter