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Optimal low-thrust trajectories for nuclear and solar electric propulsion

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ABSTRACT

The optimization of the trajectory and of the thrust profile of a low-thrust interplanetary transfer is usually solved under the assumption that the specific mass of the power generator is constant. While this is reasonable in the case of nuclear electric propulsion, if solar electric propulsion is used the specific mass depends on the distance of the spacecraft from the Sun. In the present paper the optimization of the trajectory of the spacecraft and of the thrust profile is solved under the latter assumption, to obtain optimized interplanetary trajectories for solar electric spacecraft, also taking into account all phases of the journey, from low orbit about the starting planet to low orbit about the destination one. General plots linking together the travel time, the specific mass of the generator and the propellant consumption are obtained.

1. Introduction

The optimization of low-thrust trajectories is a well known subject, and is performed by using numerical procedures. The two main approaches for optimizing a trajectory, the direct method (the problem is solved by means of gradient-based procedures) and indirect method (the problem is solved by means of shooting procedures) have been dealt with in many studies [1]. Which method is the most expedient is still a controversial topic.

Many algorithms based on direct procedures have been implemented since 1980 [2–5] and they demonstrate the convenience of this method in terms of computational time and robustness, also for complex problems (for example, to analyze the optimal low-thrust trajectories for Earth–Moon transfer for SEP spacecraft [6]).

In particular, Betts [7] emphasizes the use of the direct methods pointing out the main drawbacks of the indirect ones, that is the difficulty to compute analytic expressions for complicated nonlinear dynamics and the necessity to guess values for adjoint variables.

On the other hand, a large variety of complex problems have been successfully solved by indirect procedures, putting in evidence the increased accuracy of the solution. For instance, Kerchichian [8] studied the time-fixed minimum-fuel transfer problem for bounded thrust, Nah and Vadali [9] investigated three-dimensional fuel-optimal Earth-to-Mars trajectories for variable specific impulse, Colasurdo and Casalino [10] studied trajectory optimization problems for non-ideal solar sail spacecraft, introducing a new approach to make less complicated the derivation of the optimal control.

In order to combine the advantages of these two methods, Pastrone and Casalino [11] proposed a mixed optimization to solve a trajectory problem for hybrid rocket motors: the engine design variables were optimized by the direct method and the trajectory by the indirect one.

Abbreviation: ISS, International Space Station; LEO, Low Earth Orbit; LMO, Low Mars Orbit; NEP, Nuclear Electric Propulsion; SEP, Solar Electric Propulsion

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Nomenclature

Symbols

a	ratio between the thrust and the mass of the spacecraft
g	gravitational acceleration on the Earth surface
m	mass of the spacecraft
m_i	initial mass
m_l	payload mass
m_p	propellant mass
m_s	structural mass
m_w	mass of power generator
r_E	radius of the orbit of Earth
r, θ	polar coordinates
r	vector defining the position of the spacecraft
t	time
v_e	exhaust velocity
x, y	Cartesian coordinates
G	gravitational constant
I_s	specific impulse
J	cost function
M	mass of the body producing the gravitational field
P	power
T	final time
T	thrust
U	potential of the gravitational field
α	mass/power ratio of the power generator
γ	optimization parameter
μ	gravitational parameter ($\mu = GM$)

The indirect approach, here followed, leads to a boundary value problem in which the trajectory and the thrust profile are obtained by integrating the equations of motion of the spacecraft. In the general tridimensional case, the problem consists in a set of 12 first order differential equations whose unknowns are the 3 coordinates of the spacecraft and the 3 components of the thrust and their derivatives with respect to time. The boundary conditions are the coordinates and the 3 components of the velocity of the spacecraft at the initial and final instants. The solution of this problem requires the generation of a starting solution which is close enough to the optimized solution to allow the numerical procedure to converge toward the optimized solution. This is the most critical part of the computation, since failure to converge is possible.

Particularly in case of low, continuous thrust systems, the optimization of the trajectory and of the thrust profile is strictly linked with the optimization of the spacecraft. In 2002 an interesting optimization approach was developed by Irving and recalled by Keaton [12], who proposed to separate the spacecraft optimization from the thrust program optimization. The two parts of the problem are linked together by a single parameter, the specific mass of the power generator α. In this approach it is assumed that the power generator works always at full power, with a constant specific mass α, and the thrust level is regulated by suitably changing the specific impulse of the thruster I_s. This clearly implies that a variable specific impulse thruster is used, which is today possible because devices of this type are being developed and, hopefully, will be tested in space. The more advanced of them is the VASIMR® (Variable Specific Impulse Magnetoplasma Rocket), which will hopefully be tested on the ISS [13,14].

Clearly, the actual system may be unable to match the requirements of the theoretically computed trajectory, i.e. it may be unable of reaching the very high values of the specific impulse the optimized trajectory requires but, as specifically mentioned in [12], this may be obviated by switching off the thruster when a specific impulse in excess of the maximum possible value is required, a condition which happens about halfway in an interplanetary transfer. The effect of this maneuver is small, since in these conditions both the thrust and the propellant consumption are very small, and might even be beneficial because it introduces a coasting phase at mid-course in which, in crewed spacecraft, checking and maintenance of the propulsion system can be performed by the crew.

This approach has been followed by the present authors to compute optimal interplanetary trajectories to Mars for nuclear-electric spacecraft [15]. The aims of that paper were to generate starting solutions to proceed with the solution of the boundary value problem and to show that it is easy to optimize the whole journey, consisting of three phases, namely spiraling about the starting planet, interplanetary cruise and then spiraling about the arrival planet.

In that paper the assumption is that the total angle included between the starting and arrival positions of the interplanetary cruise coincides with that of the starting solution. This is an approximation, and does not exclude that a more convenient solution may be obtained by optimizing the solution also with respect to this angle. This will be the subject of a future paper.

These procedures are based on the assumption that the specific mass of the power generator is constant for the whole duration of the journey, a thing that holds, at least as an approximation, in the case of Nuclear Electric Propulsion (NEP), even if surely not exactly. In fact, the specific mass is defined with respect to the power of the propellant jet, which is the power of the generator multiplied by the efficiency of the power converter and the thruster. The efficiency is not exactly constant, since it varies when the specific impulse of the thruster varies.

The situation is more complex in the case of solar electric propulsion. In this case the power generated decreases, at least as a first approximation, with the increase of the square of the distance from the Sun. To extend this approach to solar electric propulsion, the assumption of constant specific mass is substituted by the assumption that the specific mass is a known function of the distance between the spacecraft and the Sun. In the following, this known function is