Preliminary Design of Low-Thrust Interplanetary Missions

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Introduction

For interplanetary missions, highly efficient electric propulsion systems can be used to increase the mass delivered to the destination and/or reduce the trip time over typical chemical propulsion systems.[1,2] This technology is being demonstrated on the Deep Space 1 mission[3] – part of NASA's New Millennium Program validating technologies which can lower the cost and risk and enhance the performance of future missions. With the successful demonstration on Deep Space 1, future missions can consider electric propulsion as a viable propulsion option.

Electric propulsion systems, while highly efficient, produce only a small amount of thrust. As a result, the engines operate during a significant fraction of the trajectory. This characteristic makes it much more difficult to find optimal trajectories. The methods for optimizing low-thrust trajectories are typically categorized as either indirect or direct. Indirect methods are based on calculus of variations, resulting in a two-point boundary value problem that is solved by satisfying terminal constraints and targeting conditions.[4] These methods are subject to extreme sensitivity to the initial guess of the variables – some of which are not physically intuitive. Adding a gravity assist to the trajectory compounds the sensitivity. Direct methods parameterize the problem and use nonlinear programming techniques to optimize an objective function by adjusting a set of variables. A variety of methods of this type have been examined with varying results.[5,6,7,8] These methods are subject to the limitations of the nonlinear programming techniques.

In this paper we present a direct method intended to be used primarily for preliminary design of low-thrust interplanetary trajectories, including those with multiple gravity assists. Preliminary design implies a willingness to accept limited accuracy to achieve an efficient algorithm that executes quickly.

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Approach

Trajectory Structure

The trajectory is divided into legs which begin and end at *control nodes*. (See Figure 1.) Typically, the control nodes are associated with planets or small bodies, but they can be free points in space. On each leg is a single *match point*, and the trajectory is propagated forward in time from the leg's earlier control node to the match point and backward from the leg's later control node to the match point.

Continuous thrusting is modeled as a series of impulses. The legs are subdivided into *segments* with an impulsive ΔV in the middle of each segment. When modeling low-thrust propulsion systems, the magnitude of the impulse is limited by the amount of ΔV that could be accumulated over the duration of the segment.

The propagation between impulses and nodes is according to a two-body model with the Sun as the primary body. Flybys of planets are modeled as instantaneous changes in the direction of the $\mathbf{V}_\infty$ (relative velocity vector).

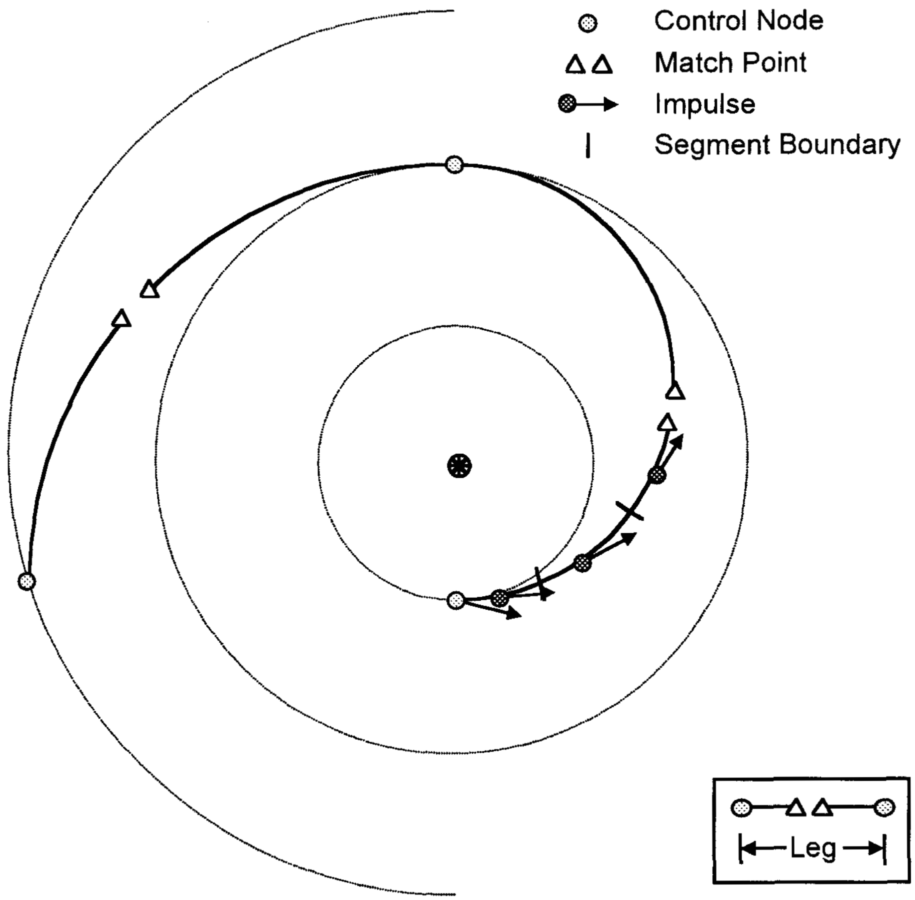

Figure 1. Trajectory Structure

Optimization

This structure results in a constrained, nonlinear optimization problem which we solve using the nonlinear programming software SNOPT.[9] The potential set of independent variables includes the state (position, velocity, and mass) of the spacecraft at each control node and the corresponding epoch. If a control node is associated with a solar system body, the position of the spacecraft is the same as the body and therefore is not independent. Two independent variables are available for an intermediate flyby, and there are variables representing the impulsive ΔVs on the segments. Depending on the optimization objective function and engine model, the solar array reference power and engine specific impulse can be independent variables. We normally try to maximize final spacecraft mass or net mass (final spacecraft mass – propulsion system mass), but other objective functions are possible.

The primary constraint on the optimization is that the position, velocity, and mass of the spacecraft must be continuous at the match points. The magnitude of the impulsive ΔVs may be constrained, as previously described, and other constraints can be placed on the trajectory such as total flight time and total propellant mass. In addition, upper and lower bounds can be placed on any of the independent variables.

Results

We have used the method described in this paper to optimize several different types of trajectories. We compare the results to those from SEPTOP, a low-thrust trajectory optimization program using an indirect method. (SEPTOP was used in Ref. 4 and is briefly described there.) The only major difference between SEPTOP and its well-known predecessor VARITOP[1,6,7] is the way in which the engines are modeled. Both programs are the result of a long evolution of low-thrust trajectory optimization software and have been used extensively to design a variety of missions.

We present results from the following three missions in this paper: a flyby of Vesta with a Mars gravity assist, a rendezvous with Tempel 1, and a flyby of Pluto with two gravity assists at Venus and one at Jupiter. The engine models are based on the NSTAR 30-cm ion thruster,[10] a version of which is being flown on Deep Space 1. The initial guesses for the thrust direction and magnitude are crude but simple and have worked well. The direction varies linearly between nodes with the direction at the nodes being perpendicular to the radius vector of the nodes and in the ecliptic plane. The maximum ΔV that could be accomplished on the first segment at a distance of 1 AU from the Sun is used as the starting guess for all segments.

For the Earth-Mars-Vesta mission, we started out by fixing the launch V_∞ and control node epochs and then subsequently released those variables in a series of runs. The final masses from our method and from SEPTOP are shown in Table 1. As can be seen from the table, the final masses agree very closely – well within the accuracy of either method. Our method converged readily; however, SEPTOP did have some trouble optimizing the flyby radius at Mars. In fact, we used the value from our method to get a better solution in SEPTOP. When we freed the Vesta arrival date, SEPTOP had significant trouble converging while our method again converged readily.

Table 1. Earth-Mars-Vesta Flyby

Launch V_∞	Earth Launch Date	Mars Flyby Date	Vesta Arrival Date	Final Mass (kg) Our Method	Final Mass (kg) SEPTOP
fixed	fixed	fixed	fixed	493.76	493.71
free	free	fixed	fixed	503.44	503.39
free	free	free	fixed	504.42	504.22

For the Tempel 1 rendezvous, we examined trajectories for four engine models which assume different levels of technology. The final masses from our method and from SEPTOP are shown in Table 2. The results again agree well, particularly for the first three engine models. The difference in final mass for the fourth engine model arises due to the fact that the spacecraft carries two engines and can operate the engines simultaneously or individually. The engines can operate at a maximum power of approximately 2.5 kW. So for example, if the input power is 3.0 kW, we could either operate one engine at maximum power or two engines at 1.5 kW each. SEPTOP continually checks whether it is optimal to run one or two engines. (The indirect method enables this check, although it can sometimes give results that are not optimal.) Even though the specific impulse of the engines generally decreases as the power level decreases, in most cases one engine operates at maximum power for only a very short interval, if at all. However, the trajectory with engine model 4 is an exception to this rule. Currently, our method uses as much of the available power as possible. So in the example given, it would operate the two engines at 1.5 kW each, using the entire 3.0 kW, instead of operating one engine at 2.5 kW. A more sophisticated algorithm for choosing the number of operating engines may be incorporated into our method in the future.

Table 2. Earth-Tempel 1 Rendezvous

Engine Model	Final Mass (kg) Our Method	Final Mass (kg) SEPTOP
1	754.40	754.30
2	764.76	764.67
3	771.95	771.86
4	760.98	765.02

Even when starting with a converged solution for a given engine model, SEPTOP often has difficulty converging to a solution with a different engine model. Our method has shown promise in being able to handle different engine models more consistently.

Theoretically, the indirect method used by SEPTOP can incorporate an unlimited number of intermediate body flybys; however, because of practical limitations arising from the sensitivity issues, SEPTOP has been programmed to accommodate at most two intermediate body flybys. Hence, SEPTOP cannot optimize the Earth-Venus-Venus-Jupiter-Pluto trajectory in its entirety. To examine such a trajectory, SEPTOP is used to optimize the trajectory to Jupiter, and the Jupiter-Pluto leg is determined by C_3 matching.

Our method can handle any reasonable number of flybys. In fact, using match points as a part of the trajectory structure is intended to reduce the sensitivity to adding intermediate flybys. To compare to SEPTOP, we fixed the Jupiter flyby and Pluto arrival dates, obtaining a final mass of 879.9 kg. Using the procedure described above with SEPTOP resulted in a final mass of 880.4 kg.

Conclusion

We have developed and tested a direct method for preliminary design of low-thrust interplanetary trajectories. This method has been compared to a program using an indirect method, and the results agree very closely. The new method has shown less convergence sensitivity and the ability to handle more intermediate flybys than the indirect method.

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