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# Engineering Notes

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## Engine-Switching Strategies for Interplanetary Solar-Electric-Propulsion Spacecraft

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### Introduction

SINCE the Deep Space 1 spacecraft successfully demonstrated the first use of solar electric propulsion (SEP) for interplanetary travel,[1] low-thrust SEP has been considered a viable option for future interplanetary missions. Previous SEP mission analyses, such as Williams and Coverstone-Carroll,[2] demonstrated the mass benefits gained by using SEP instead of conventional chemical propulsion. Many preliminary mission analyses utilize simple models for the SEP system and its operation; however, some of the finer points and operational details of SEP need to be considered as the mission design is refined to create an accurate mass budget. Engine switching and power allocation is one such SEP operational detail. For example, the NASA Evolutionary Xenon Thruster (NEXT)[3] can only be operated within an acceptable input power range, and therefore it provides limited thrust. Consequently, multiple NEXT engines are needed to increase the total thrust acceleration level that is required for high-energy interplanetary missions in order to reduce the transfer time or deliver more payload mass. Furthermore, because the total power generated by the solar array depends on the distance from the sun, SEP spacecraft will require oversized arrays for trajectories to high-energy (or distant) interplanetary targets. The ensuing trajectory can exhibit periods where the spacecraft is near the sun (perhaps performing a Venus gravity assist) and far from the sun, and therefore available solar power will fluctuate. For these mission scenarios, operating multiple engines results in a new question: how many engines should be turned on as available solar-array power changes? The engine-switching (or power-allocation) strategy must be considered, and its effect on the trajectory and payload mass must be determined.

In this Note, we consider the effects of engine-switching schemes for multiple-engine operation. In particular, two simple engine-switching schemes are investigated. A Pluto-flyby mission scenario using SEP and four NEXT engines is used to demonstrate the effects of engine switching on payload mass. Numerical results which show that the two engine-switching schemes exhibit different payload performance, are presented.

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### Ion Thruster Operation

#### Engine-Switching Schemes

The total mass-flow rate of a SEP spacecraft operating $n$ engines can be expressed by

$$\dot{m} = -T/gI_{\mathrm{sp}} \tag{1}$$

where

$$T = \sum_{i=1}^{n} T_i$$

is the total thrust amplitude, $T_i$ is the thrust amplitude generated by the $i$th engine, $m$ is the spacecraft mass, $g$ is the sea-level Earth gravitational acceleration, and $I_{\mathrm{sp}}$ is the specific impulse. (We assume all $n$ engines operate with the same specific impulse.) The thrust amplitude for each engine onboard is modeled as

$$T_i = 2d\eta P/gI_{\mathrm{sp}} \tag{2}$$

where $\eta$ is the thruster efficiency, $P$ is the input power to the $i$th engine, and $d$ is the so-called duty cycle that accounts for coasting periods required for navigation and orbit determination.[1] Duty cycle is assumed to be a constant 0.92 in the thrust computation in this work. This duty-cycle constant approximates the relatively short coast arcs interspersed during long powered arcs (i.e., 0.92 means thrust is off for 8% of the powered phase). Equations (1) and (2) also indicate that operating multiple engines always occurs at the same input power, specific impulse, and thruster efficiency for each engine. Therefore, we do not use the subscript $i$ for $P$, $I_{\mathrm{sp}}$, or $\eta$; however, the last two parameters typically vary with input power. An inverse-square relation is used to model the total available power

$$P_a = P_0 \big/ r^2 \tag{3}$$

where $P_0$ is the total power at 1 astronomical unit (AU) available to the EP system and $r$ is the distance from the sun to the spacecraft (in AU). The available power has an upper limit of $2P_0$ as $r$ decreases below 0.7071 AU.

A typical ion thruster (such as the NEXT engine) can only operate when the input power is between fixed upper and lower bounds defined by $P_{\max}$ and $P_{\min}$, respectively. It is obvious that all $n$ engines on an SEP spacecraft should be simultaneously burned if $P_a > nP_{\max}$, and no engine can be operated if $P_a < P_{\min}$. Two simple engine-switching schemes are proposed when $P_a$ is between these two extremes.

Switching scheme 1:

$$\begin{cases} k \quad \text{engines operated if} \quad kP_{\min} \le P_a \le (k+1)P_{\min} \\ \qquad\qquad\qquad\qquad\qquad k = 1, 2, \ldots, n-1 \\ n \quad \text{engines operated if} \quad nP_{\min} \le P_a \end{cases} \tag{4}$$

Switching scheme 2:

$$\begin{cases} k \quad \text{engines operated if} \quad (k-1)P_{\max} \le P_a \le kP_{\max} \\ \qquad\qquad\qquad\qquad\qquad k = 2, 3, \ldots, n \\ 1 \quad \text{engine operated if} \quad P_{\min} \le P_a \le P_{\max} \end{cases} \tag{5}$$

For switching scheme 1, the total available power is equally distributed to as many engines as possible; for switching scheme 2 the total power is equally distributed to as few engines as possible. For example, consider the SEP vehicle where $n = 3$, $P_{\min} = 1$ kW, and $P_{\max} = 5$ kW. If the available power at some point in the trajectory is $P_a = 8$ kW, then switching scheme 1 commands three engines (each using 2.667 kW of input power), and switching scheme 2 commands two engines (each using 4 kW of power).

**NEXT Thruster Model**

The minimum and maximum input power ($P_{\min}$ and $P_{\max}$) for a single NEXT engine is 1.485 and 6.055 kW, respectively. Like many ion thrusters, both $I_{\text{sp}}$ and $\eta$ vary with respect to the input power. However, the NEXT thruster exhibits an envelope for $I_{\text{sp}}$ and $\eta$ over the range of input power, as shown in Fig. 1. Therefore, the acceptable values of $I_{\text{sp}}$ and $\eta$ are defined by

$$I_{\text{sp}} = I_{\text{sp},L} + a(I_{\text{sp},U} - I_{\text{sp},L}) \tag{6}$$

$$\eta = \eta_L + a(\eta_U - \eta_L) \tag{7}$$

where $I_{\text{sp},L}$ and $I_{\text{sp},U}$ are the lower and upper limits on $I_{\text{sp}}$, and $\eta_L$ and $\eta_U$ are the lower and upper limits on $\eta$, as shown in Fig. 1. The variable $a$ can be thought of as a throttle parameter for both $I_{\text{sp}}$ and $\eta$ because it determines the specific impulse and efficiency settings between their respective envelopes for a particular input power. (Our NEXT model uses a single throttle parameter.) Parameter $a$ is constrained to be between 0 and 1.

## Numerical Example

A Pluto-flyby mission is used to demonstrate the utility of the two engine-switching schemes. This mission scenario uses a Venus gravity assist (VGA) and an SEP spacecraft equipped with four NEXT engines. The launch date is constrained within the year 2008, and the Pluto flyby date is to be no later than 29 August 2019 in order to follow the ground rules for a New Horizons class mission profile. (Therefore, total trip time must be less than 12 years.) A direct, shooting method[4] is used to obtain the optimal trajectory that maximizes the payload mass at the Pluto flyby. The direct method converts the optimal control problem into a finite-dimensional parameter optimization problem that is solved by employing nonlinear programming techniques. Sequential quadratic programming (SQP)[5] is used here to solve the constrained optimization problem. A precise planetary ephemeris[6] is used in this Note.

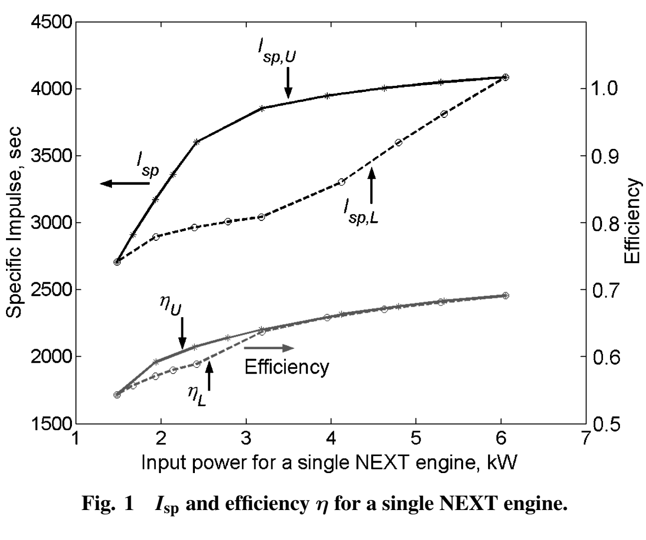


**Fig. 1** $I_{\text{sp}}$ **and efficiency** $\eta$ **for a single NEXT engine.**

The payload mass $m_{\text{PL}}$ is

$$m_{\text{PL}} = m_0 - (1 + K_t)m_{\text{prop}} - \alpha P_0 - m_{\text{EP}} \tag{8}$$

where $m_0$ is the launch mass, $m_{\text{prop}}$ is total propellant mass (including 10% reserves), $K_t$ is the tankage fraction, $\alpha$ is the specific mass of the power system, and $m_{\text{EP}}$ is the mass of the electric propulsion (EP) system (four NEXT thrusters, power-processing units, harness, and cables). A Delta 4450 launch vehicle is used to deliver the spacecraft to heliocentric space, and the launch (or injected) mass is a function of launch energy $C_3$. Launch mass $m_0$ is 90% of the delivered mass predicted by the Delta 4450 launch performance curve in order to account for contingency mass. Tankage fraction is fixed at 13.4%, specific mass is $\alpha = 6.68$ kg/kW, and the EP system mass is fixed at $m_{\text{EP}} = 598.8$ kg.

We use the direct optimization method to obtain trajectories that maximize the payload mass defined by Eq. (8). We assume a flight sequence consisting of an initial powered arc, VGA, second powered arc, and finally a long coast arc to the Pluto flyby. The pitch and yaw thrust-steering angles are determined by linear interpolation among a finite number of discrete control nodes, which are treated as SQP design variables. Other SQP design variables include parameters for the Venus gravity assist (periapsis altitude and flyby plane orientation), time durations of the two burn arcs, time duration for the single coast arc, launch date, launch energy $C_3$, hyperbolic Earth-departure conditions, initial power at 1 AU ($P_0$), and the throttle parameter $a$ in Eqs. (6) and (7) for each powered arc. The optimization problem involves the following constraints: position vector match for the VGA and Pluto flyby, the total SEP propellant mass must be less than 520 kg, the solar-array power is cut off at a maximum radius of 3 AU, and Pluto flyby date must be before 29 August 2019. Finally, engine switching (or power allocation) for the four NEXT engines is dictated by the two engine-switching schemes presented by Eqs. (4) and (5).

Table 1 summarizes the optimal payload solutions for the Pluto-flyby mission with the two different engine-switching schemes. Several optimal mission parameters (such as launch date, VGA date, VGA parameters, and Pluto flyby date) are essentially identical for the two switching schemes. Both switching schemes result in the maximum allowable propellant load ($m_{\text{prop}} = 520$ kg) and the latest possible flyby date. However, switching scheme 2 demonstrates a 24% increase in payload mass when compared to scheme 1 (recall that scheme 2 uses as few engines as possible for the given available power). Note that the throttle parameter $a$ is equal to unity for both burn arcs for scheme 2, which indicates engine operation at the upper limits of the specific impulse and efficiency envelopes presented in Fig. 1. Using switching scheme 2 also results in a larger solar array (increase in $P_0$ of 2.7 kW) when compared to scheme 1, and the slight increase in array mass produces a significant increase in launch mass because some of the launch energy is offloaded to the SEP system. Furthermore, scheme 2 results in maximum power input for each engine with maximum $I_{\text{sp}}$ and efficiency. Maximizing input power to each thruster results in maximum engine efficiency and (in general) minimum propellant consumption. On the other hand, maximum $I_{\text{sp}}$ results in minimum mass-flow rate and thrust magnitude [see Eqs. (1) and (2)], which in turn increases trip time and can lead to undesirable mission scenarios. However, using scheme 2 for a maximum-payload mission does not sacrifice trip time because both solutions show nearly identical transfer times (11.4 years) to the Pluto flyby.

**Table 1 Optimal Pluto-flyby mission parameters for two engine-switching schemes**

| Switching method | $C_3$, km$^2$/s$^2$ | $m_0$, kg | $m_{\text{prop}}$, kg | $m_{\text{PL}}$, kg | $P_0$, kW | Trip time, yr | Throttle parameter $a$ (1st and 2nd burns) |
|---|---|---|---|---|---|---|---|
| Scheme 1 | 34.4 | 1919 | 520[a] | 624 | 15.96 | 11.43 | 1.0, 0.58 |
| Scheme 2 | 31.7 | 2089 | 520[a] | 776 | 18.64 | 11.44 | 1.0, 1.0 |

[a]Upper bound for propellant mass.

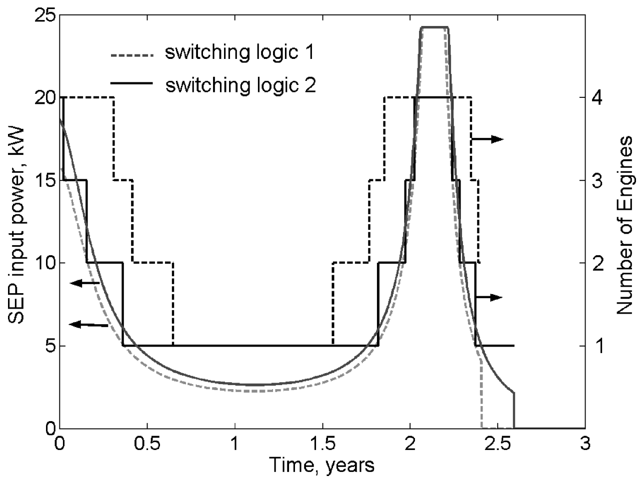


**Fig. 2 Available power and engine-switching sequence for a mission with four NEXT engines.**

Figure 2 presents the time histories for available SEP input power and the number of engines operated during the mission. Figure 2 shows that scheme 2 results in higher input power and fewer operating engines during the powered phase when compared to scheme 1. Maximum input power and maximum engine operation occurs at two years into the mission, near the Venus gravity assist.

## Conclusions

Two simple engine-switching or power-allocation schemes for interplanetary SEP spacecraft have been investigated. The first scheme operates as many engines as possible (minimum input power to each thruster), whereas the second scheme operates as few engines as possible (maximum individual input power) for the given available power. Intuitively, we would expect minimum trip time with scheme 1 and minimum propellant mass with scheme 2. However, intuition cannot predict which switching scheme will maximize the payload mass subject to propellant and trip-time constraints. We obtained optimal payload transfers for a Pluto-flyby mission (using a Venus gravity assist) that utilized an SEP spacecraft equipped with four NEXT ion thrusters. Initial array power is optimized in order to maximize payload. Our analysis shows that switching scheme 2 demonstrates a significant increase in payload mass when compared with scheme 1 without any change in the total trip time. Furthermore, scheme 2 results in higher initial power (larger solar arrays), but this system mass penalty is offset by a much larger increase in launch mass (and hence payload mass) because enhanced use of the electric propulsion system reduces the launch energy. It appears that an engine-switching scheme that maximizes equal power distribution among multiple engines provides maximum payload mass for high-energy interplanetary missions. In addition, a scheme that operates fewer engines might increase mission reliability by providing spares in case of engine failures.

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