ORBIT RAISING TRAJECTORY AND SYSTEM ANALYSIS FOR THE MISSION DESTINY

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Abstract: *DESTINY is a JAXA mission candidate planned to be launched in 2018. The design of its orbit raising trajectory is a challenging due to its many revolution low-thrust orbits coupled with multiple mission objectives and constraints. In its early mission design phase, it is of interest to perform analysis on various system designs and assess their corresponding mission performance. In this paper, we perform multi-objective optimization on various cases of initial mass and initial apogee on the DESTINY orbit raising trajectory. Using an analytical averaging technique and a simplified thrust profile, trajectories can be propagated quickly for the optimization task. Preliminary results show that the mission objectives of ion engine operation time, time of flight, and the time in radiation belt increase their values with the increase in initial mass. We demonstrated an technique of assessing different system design parameters on their impact to crucial mission requirements and performance.*

Keywords: *DESTINY, low-thrust, mission design, orbit raising, system analysis.*

1. Introduction

The mission candidate DESTINY (which stands for **D**emonstration and **E**xperiment of **S**pace **T**echnology for **IN**terplanetary vo**Y**age) is proposed as an engineering demonstration mission for ISAS/JAXA's small satellite series.[1] DESTINY is planned to be launched in 2018 by the Epsilon launch vehicleJAXAs next-generation solid fuel rocket. The main objective of DESTINY is to conduct demonstration and experiment on key advanced technology for future deep space missions. For example, the ultra-lightweight solar panel, the large scale ion engine, advanced thermal control system, advanced command and data handling technology, small satellites standard bus, etc. Besides the advanced technologies and components demonstrated in the mission, the trajectory design of DESTINY also covers many aspects in astrodynamics and optimal control, such as multi-revolution low-thrust trajectories, repeating resonant gravity assists, low-energy escape[2], Halo orbit transfer and maintenance[3], etc.

One of the important tasks in preliminary mission planning is to evaluate different design options and its effect on the mission objectives. This can be viewed as a tradeoff in mission cost and potential risk. For example, a conservative design with higher mass and cost versus a system with new technology and higher risk but lower mission cost. Previous work[4] applied an analytical averaging technique to perform multi-objective optimization on DESTINY's orbit raising trajectory. In this paper, we aim to broaden the scope of the study to include more system parameters into considerations that were previously given as constants. Studying different design options can also help us understand the sensitivity of one system parameters to the mission requirements and thus to aid the design team to fine tune the design of the system.

2. Mission Background

Figure 1 illustrates the DESTINY mission profile. The spacecraft will first be placed into a low-Earth elliptical orbit by JAXAs next-generation solid fuel rocket, the Epsilon launch vehicle. Then the ion engine μ20 will be used to raise the orbital altitude to reach the Moon. After that it will be injected into a transfer orbit for L_2 Halo orbit of the Sun-Earth system by using lunar gravity assist.

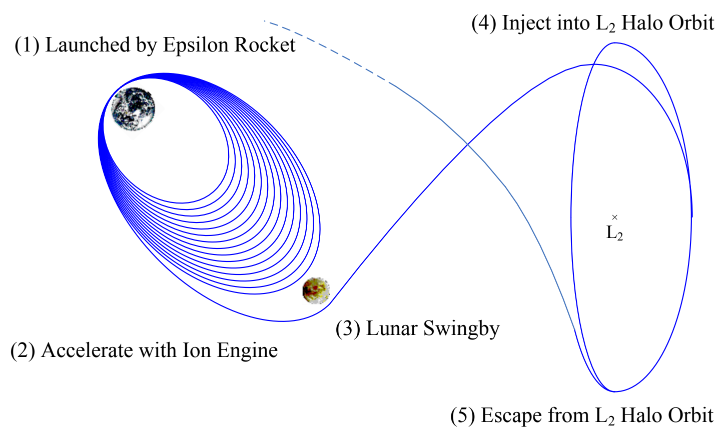

Figure 1. Overview of the DESTINY Mission

On the way to L_2 Halo orbit, DESTINY conducts demonstration and experiment of key advanced technology for future deep space missions. Major items of the technology demonstration are listed as follows. (For details of the technology below, see Ref. [1].)

- High energy mission by Epsilon rocket.
- Ultra-lightweight solar panel
- Large scale ion engine μ20
- Advanced thermal control
- Advanced communication system
- Automatic/autonomous onboard operation
- Orbit determination under low thrust operation
- Halo orbit transfer and maintenance

3. Low-Thrust Trajectory Model

To reduce the computational cost in the preliminary design phase, we adopt a fast and reasonably accurate low-thrust model and propagation method adopted in Ref. [4]. In that model, the motion of the spacecraft is propagated by means of an orbital averaging technique, in which the net variation of the orbital elements along a single revolution is computed; then this averaged over the orbit period and the resulting quantity is integrated numerically over the long time periods. In particular the variation of orbital elements along a single revolution due to the thrust is computed by means of an analytical, first-order solution of perturbed Keplerian motion, which has shown to guarantee adequate accuracy at a lower computational cost compared to numerical integration. The contribution of the J_2 erturbation is also included. An extensive description of the analytical formulae and of their accuracy can be found in Refs. [5] and [6].

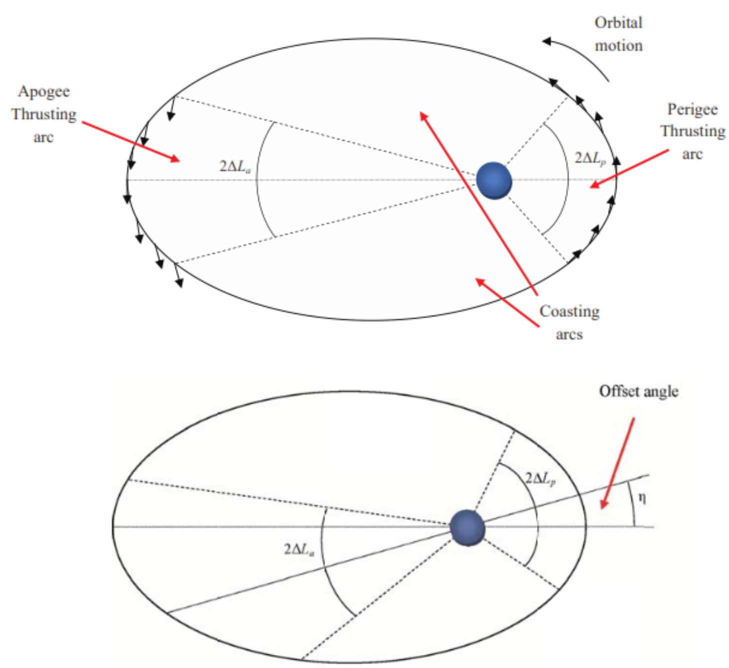

Figure 2. Simplified Low-Thrust Control Scheme: Symmetric (Top) and Asymmetric Thrust Pattern (Bottom).

In order to keep the number of parameters low, a number of assumptions on the thrusting strategy are introduced. First of all, an on/off control is assumed, in the sense that at a given instant, the thrust magnitude can be either zero or the maximum value permitted by engine specifications. Secondly, it is assumed that the thrust direction is purely in plane and directed along the tangential direction, which maximizes the instantaneous variation of orbital energy. Thus, one has to define the timing of the thrust switching. Each revolution is divided in 4 sectors, as shown in Figure 2 (top). A Perigee thrusting arc, an Apogee thrusting arc and two coasting arcs in between. The amplitude of the thrust arcs around perigee and apogee are denoted by ΔL_p and ΔL_a respectively.

The terms $\Delta L_{p,i}$ and $\Delta L_{a,i}$ are defined as a piecewise linear interpolation with respect to time, from N_{nodes} nodal values (for $i = 1, 2, ..., N_{nodes}$), uniformly spaced within the integration boundaries. Following the same setting of previous work [4], we set N_{nodes} to be 8 in our study. To also include the flexibility to employ an asymmetric thrust pattern, e.g. to effectively change the argument of perigee for eclipse avoidance, the angle η_i is included in the control profile as shown in Figure 2 (bottom).

4. Problem Description

The design of DESTINY's orbit raising trajectory is formulated as a multi-objective, constrained optimization problem. The following objectives are to be minimized:

Objectives

- Ion Engine System Operation Time (IES)
- Time of Flight (TOF)
- Time spent in the Radiation Belt (t_{belt})
- Maximum Eclipse Time ($t_{ecl,max}$)

with the following set of variables:

Variables

- Initial launch dates and time (t_0)
- Apogee thrust angles ($\Delta L_{a,i}$)
- Perigee thrust angles ($\Delta L_{p,i}$)
- Asymmetric thrust angles (η_i)

where $i = 1, 2, ..., N_{nodes}$.

The terminal condition to be reached at the end of the orbit phase is a radius of 300,000 km at the intersection between the orbit and the current lunar orbital plane. For a given set of control profile and initial condition, the trajectory is propagated until it reaches the target condition OR until the time of flight reaches 580 days (which includes 30 days of initial commission time). If the spacecraft does not reach the target radius, a mismatch in the final target radius Δr_f would appear as a constraint violation in the problem. For the time spent in radiation belt, we simply count the time where the spacecraft is within 20,000 km. On the maximum eclipse time, in the previous design, it should be no longer than 1 hour. However we relax this requirement in this study due to a possible extension in the onboard battery lifetime (benefited from the increase in spacecraft mass).

Constraints

- Violation in the minimum radius Δr_{min}
- Mismatch with the final target radius Δr_f
- Violation in the thrust angles

On the Violation in the thrust angles, because $\Delta L_{a,i}$ and $\Delta L_{p,i}$ are defined as half of the thrust arcs on 1 revolution, their sum cannot exceed π radian. Thus:

$$\Delta L_{a,i} + \Delta L_{p,i} \leq \pi \tag{1}$$

We note that Eq. 1 can easily turn into a linear inequality constraints, which is usually easy to be satisfied in optimization.

4.1. Initial Mass and Orbit

The main purpose of this work is to perform multi-objective optimization on different system design options, which is reflected in the initial spacecraft mass in the low-Earth elliptic orbit. From the design team of the Epsilon Launch Vehicle, the relationship of initial spacecraft mass and the apogee radius of the initial orbit is reported in Table 1. Other orbital elements of the initial orbit and ion engine parameters are summarized in Tables 2 and 3.

Table 1. Initial Mass and Apogee

Initial Mass (kg)	Apogee Radius (km)
400	35,128
410	33,878
420	32,628
430	31,378
440	30,128
450	28,878

Table 2. Initial Orbital Elements in the J2000 Earth Fixed Reference Frame

Perigee Radius	i	Ω	ω	M
$6,528 km$	32^o	21^o	124^o	5^o

Table 3. Ion Engine Parameters

Maximum Thrust	Specific Impulse
40 mN	3800 s

5. Optimization

The design of DESTINY's orbital raising trajectory is formulated as a multi-objective optimization problem, where the objectives presented in the Section **Problem Description** are to be minimized. To account for the nonlinear constraint, we also include the mismatch with the final target radius Δr_f as one of the objective function. That is, the problem is presented as:

$$minimize \quad \mathbf{F}(\mathbf{x}) \tag{2}$$

where

$$\mathbf{F}(\mathbf{x}) = \left[IES \;\; TOF \;\; t_{belt} \;\; t_{ecl,max} \;\; \Delta r_f\right] \tag{3}$$

subjected to the linear constraint in Eq. 1. The optimization variables $\mathbf{x}$ are:

$$\mathbf{x} = [t_0 \;\; \Delta L_{a,i} \;\; \Delta L_{p,i} \;\; \eta_i]_{i=1,2,\ldots,8} \tag{4}$$

We note that the problem has 5 objectives and 25 variables. The launch date t_0 is bounded to be within 1 year in 2018. For the thrust arc angles $\Delta L_{a,i}$ and $\Delta L_{p,i}$, the bounds are 0^o and 180^o; and for the asymmetric angle η, the bounds are -90^o and 90^o.

We employ a controlled elitist genetic algorithm (a variant of NSGA-II [7]) to solve the multi-objective optimization problem. An linearly feasible initial population size of 20 is set at the beginning, while 100 generations is set as the stopping criteria of the algorithm.

6. Numerical Results and Discussion

Preliminary results of the multi-objective optimization for different initial mass are presented in Figs. 3,4, and 5. Note that only the feasible solutions with nearly zero mismatch on the final target radius are plotted. (i.e. the 5th objective Δr_f). Examples trajectories on initial mass = 450 kg are plotted in Figs. **??** and **??**.

Some statistics of the objective values for various initial mass cases are summarized in Table **??**. We note that the objective values generally increases with initial mass, except for the maximum eclipse time in which its value is more irregular. For the IES, TOF, and t_{belt}, the increase in the objective values with initial mass m_0 is not quite proportional. For example, an increase in 5% of m_0 lead to more than 5% increase in the objectives. This is probably caused by the lower initial apogee on a heavier initial mass, which leads to more propellant expenditure, longer flight time, and longer time in the radiation belt.

Table 4. Summary of the Objective Values from the Results of Multi-Objective Optimization

Objectives	m_0 = 400 kg	m_0 = 420 kg	m_0 = 430 kg	m_0 = 450 kg
min IES (days)	361	415	440	468
mean	381	426	449	477
max	396	441	460	488
min TOF (days)	432	469	485	519
mean	504	518	527	547
max	578	573	577	576
min t_{belt} (hours)	1458	1775	1986	2495
mean	1715	1971	2124	2580
max	2229	2308	2360	2870
min $t_{ecl,max}$ (hours)	1.23	1.46	1.38	1.20
mean	1.85	2.67	2.15	2.38
max	3.38	3.56	2.87	3.06

7. Conclusion

We performed a multi-objective optimization on various cases of initial mass and initial apogee on the DESTINY's orbit raising trajectory. Three out of four objectives in the problems increases their values with the increase in initial mass, but the maximum eclipse time does not. We demonstrated an example of assessing different system design parameters on their impact to crucial mission requirements and performance.

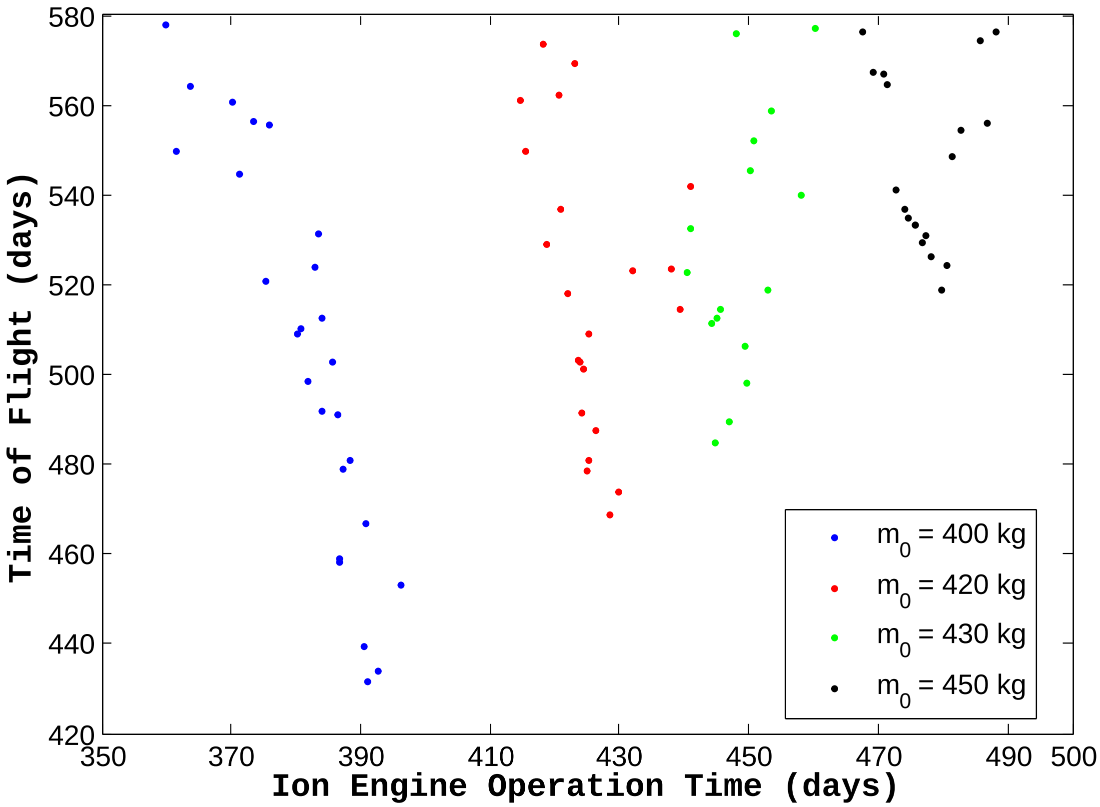

Figure 3. Time of Flight vs Ion Engine Operation Time.

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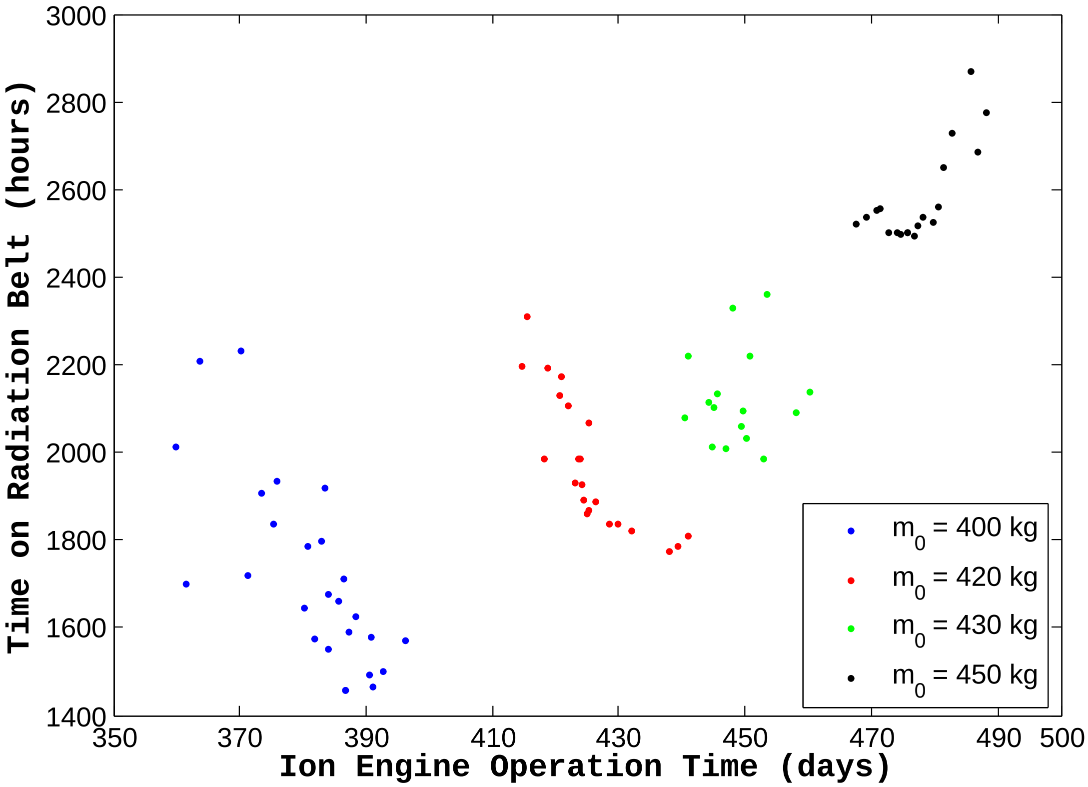

Figure 4. Time in Radiation Belt vs Ion Engine Operation Time.

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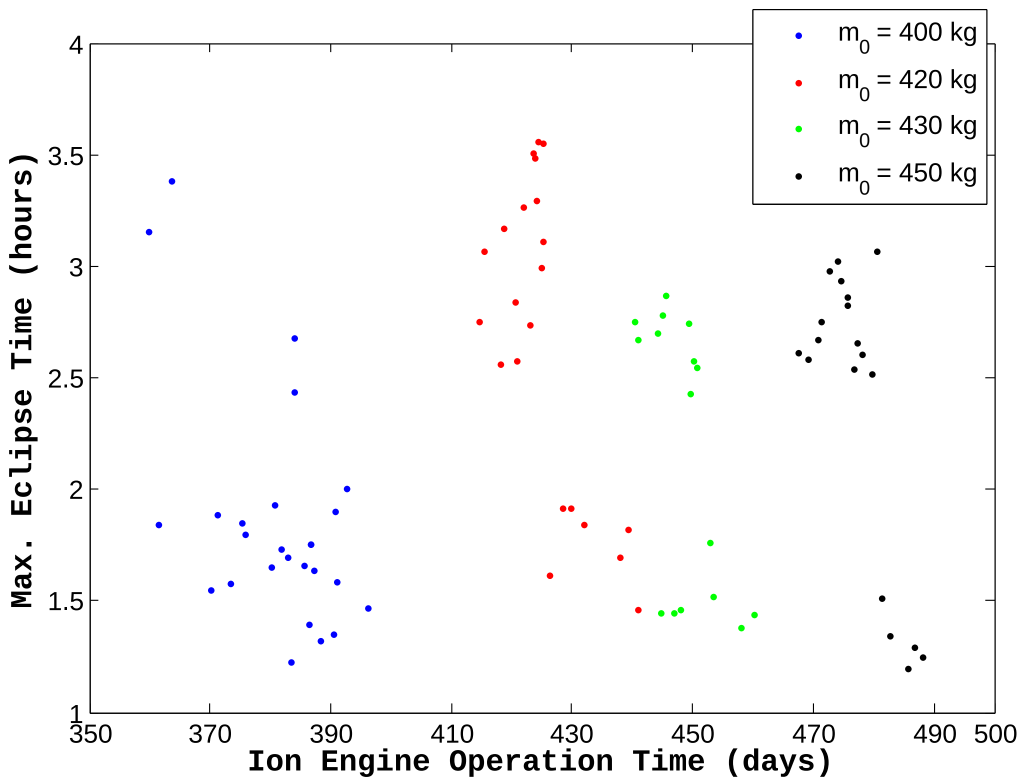

Figure 5. Maximum Eclipse Time vs Ion Engine Operation Time.

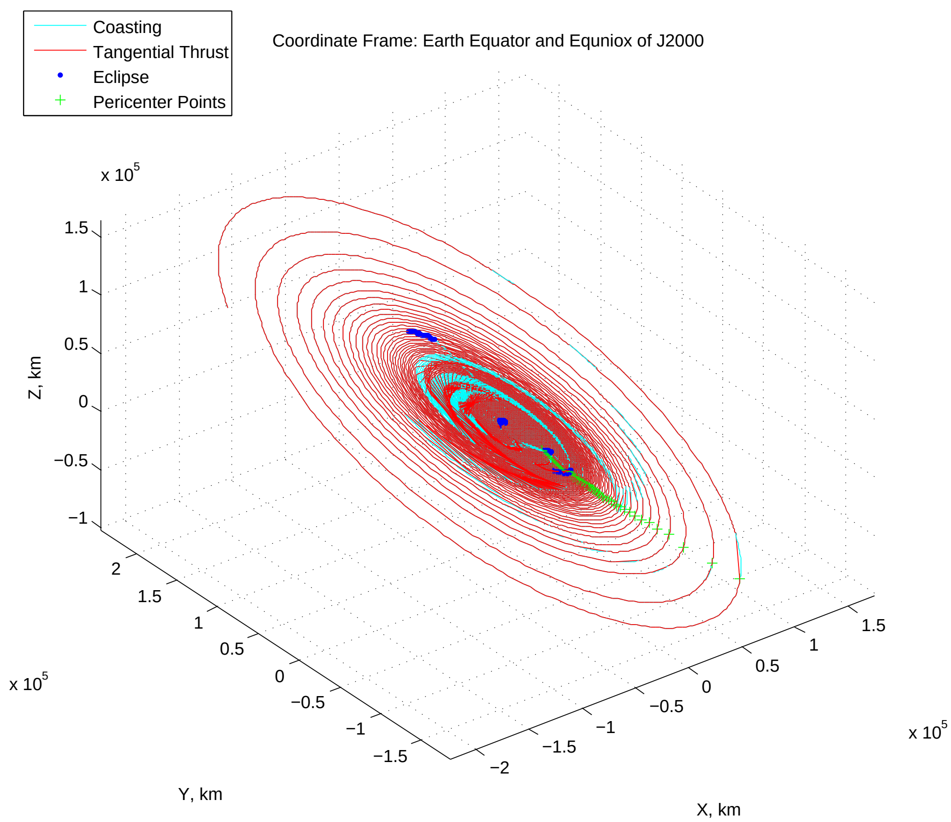

Figure 6. Example Trajectory with minimum IES and m_0 = 450 kg.

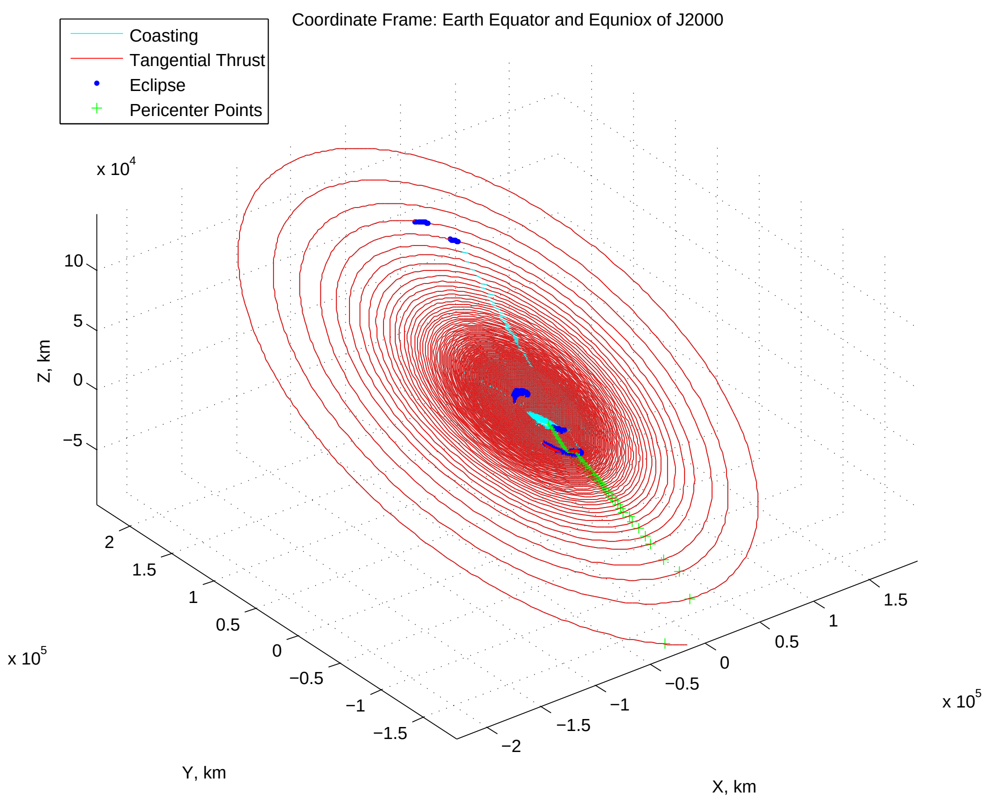

Figure 7. Example Trajectory with minimum TOF and m_0 = 450 kg.