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Low-Thrust Control Strategies for Earth-to-Mars Trajectories

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Abstract

Recent advances in electric propulsion systems have demonstrated that these engines have the potential to be used for long-duration travels, with applications such as cargo and human transportation for interplanetary voyages. The Variable Specific Impulse Magnetoplasma Rocket (VASIMR) is an example of this type of engine, possessing the ability to operate at a wide range of specific impulse levels. This chapter presents the results of a study comparing three different thrust control strategies for Earth-Mars trajectories, using the VASIMR engine at a power of 150 kW. These are constant thrust trajectories, trajectories with coasting periods, and trajectories with variable specific impulse, resulting in variable thrust. To achieve this, an optimization tool was created using spherical coordinates to model the dynamics of the spacecraft, optimal control theory to setup the optimization problem, and a differential evolution algorithm to minimize the cost function. A novel approach to model variable specific impulse and coast-arcs in the trajectories for spherical coordinates is presented as well. The optimization tool was utilized to find optimal trajectories from Earth to Mars orbit, and it was concluded that using variable thrust reduces propellant consumption for a variety of trajectories, when compared to the other two methods.

Keywords: low-thrust trajectories, high power electric propulsion, global optimization

1. Introduction

The National Aeronautics and Space Administration (NASA) announced in 2015 its partnership with commercial industry to develop 12 key technologies that will allow space and human exploration to deep-space destinations, such as the Moon and Mars [1]. The Next Space Technologies for Exploration Partnerships (NextSTEP) include concepts in advanced

propulsion, habitation, and small satellites. Among these, three companies developing high power electric propulsion systems were selected to develop engines in the 50–300 kW range, with high specific impulse (2000–5000 s) and efficiency (greater than 60%). The purpose of the development of these engines is to obtain propulsion systems that can operate continuously for long periods, to enable deep space transportation using highly efficient propulsion.

The selected companies for NextSTEP are:

- Ad Astra Rocket Company of Webster, Texas
- Aerojet Rocketdyne Inc. of Redmond, Washington
- MSNW LLC of Redmond, Washington

Although all three companies are working on electric propulsion systems, these engines operate under different principles. Ad Astra Rocket Company's Variable Specific Impulse Magnetoplasma Rocket (VASIMR) uses radio waves to ionize and energize a propellant, converting it to a plasma state, and a magnetic field to guide and expel the plasma, producing thrust [2]. Aerojet Rocketdyne is working on a high power Hall thruster, which uses electrons trapped in a magnetic field to ionize propellant and accelerate the propellant to produce thrust, while neutralizing the plume to avoid the spacecraft from acquiring a charge [3]. The electrodeless Lorentz force (ELF) thruster developed by MSNW LLC, is a pulsed propulsion system that generates a high density and magnetized plasmoid, known as a field reversed configuration (FRC), using radio waves to produce a rotating magnetic field (RMF) [4]. These FRC sources are pulsed devices where the plasmoid evolves from neutral gas injection and ionization, to plasmoid growth and acceleration, and finally to plasmoid ejection.

If the parameters specified by NASA for engine performance are reached, these propulsion systems could be powered by solar energy for interplanetary flight. These type of systems are called solar electric propulsion (SEP) and would require approximately 10 times less propellant to operate than the typical chemical propellant that are currently operating [5]. Furthermore, SEP systems with thrust control could provide even more propellant savings compared to continuous thrust system. The main motivation of this study is to test whether this thrust strategy is indeed more efficient in terms of propellant consumed for interplanetary travel.

This chapter aims to find the optimal low thrust control strategy for transfers from Earth to Mars using three different thrust control strategies: (1) constant thrust trajectories, (2) trajectories with coasting periods, and (3) trajectories with variable specific impulse, resulting in variable thrust. To achieve this goal, an optimization tool was created to compute the optimal trajectory, given a fixed time of flight, for each thrust control strategy. The optimal trajectory was selected based on propellant consumption for each transfer. The engine used for the study is the VASIMR, given its ability to operate at a wide range of specific impulse values, and therefore thrust levels. Section 2 presents a description of this engine, while Section 3 presents the optimization tool created for this study. The results of the analysis are presented in Section 4, leading to the conclusions presented in Section 5.

2. Variable Specific Impulse Magnetoplasma Rocket

The VASIMR is an electric thruster of the electromagnetic kind. It uses magnetic fields to guide plasma through an exhaust, producing thrust in the process. The concept was created by Dr. Franklin Chang Díaz during his time as a graduate student at the Massachusetts Institute of Technology (MIT) and has been developed since the late 1970s [2]. During the 1990s, development of the engine took place in the Advanced Space Propulsion Laboratory (ASPL) at NASA's Johnson Space Center. The experimental engine tested at the laboratory operated at 10 kW and was later upgraded to a 50 kW version producing 0.5 N of thrust. Ad Astra Rocket Company was then created as a spin-off of the NASA laboratory and the engine has seen a significant development in technology during the company's lifespan. The most recent version of the engine (VX-200 or VASIMR eXperimental 200) runs at 200 kW and produces a maximum thrust of approximately 6 N at an specific impulse of 5000 s.

Currently, researchers are improving the engine to operate at steady state. In 2015, Ad Astra Rocket Company was awarded a 3-year, \$9 million contract from NASA to develop the maturity of the VX-200 engine [6]. Specifically, by the end of the contract, company must demonstrate that the engine is able to operate at a power level of 100 kW for 100 h. Ad Astra is currently on schedule with this goal, and has successfully completed a NASA review after its second year of contract. Currently, the engine has operated for a total 10 h and there have been considerable changes to the vacuum chamber where the VX-200 operates. These modifications are necessary, so the engine can handle the thermal load produced by the engine. After demonstrating successful steady-state operations, a flight version of the engine called the VASIMR Flight 200 (VF-200) is planned to be constructed and tested in space.

Figure 1 presents a schematic of the VASIMR and its operating principles. The propellant (in gaseous form) enters the first stage of the engine and is converted to plasma by a helicon radio frequency (RF) generator. This was established in nuclear fusion experiments and consists of ionizing the gas. The plasma is guided forward using a magnetic field created by superconducting magnets. It then advances to the second stage where it is energized using ion cyclotron resonance heating (ICRH). The high-energy plasma is then exhausted using a magnetic nozzle, creating thrust. One unique feature of this engine is a technique

Figure 1. VASIMR operating principles.

called constant power throttling (CPW) [2]. This means that the engine can vary its thrust and specific impulse using constant power settings. The throttling is possible by controlling the amount of power that goes to each stage: if more power is directed to the first stage, more plasma is created generating more thrust, but at a lower specific impulse. If more power is directed to the second stage, less plasma is created but it will have a higher exhaust velocity (higher specific impulse), since it gets a greater energy boost from the ICRH. This variation in thrust and specific impulse is a great advantage since the engine can fit many mission profiles due to its flexibility. Additionally, the VASIMR can be scaled up in power (theoretically to MW capability), enabling crewed interplanetary flights using electric propulsion [7].

3. VASITOS

A low-thrust spacecraft trajectory optimization tool, called the Variable Specific Impulse Trajectory Optimization Software (VASITOS), was created to analyze the optimal thrust strategy. This section presents the software environment in which it was created, the propagation scheme used to model the dynamics of the spacecraft, and the global optimization algorithm incorporated to compute optimal low-thrust trajectories.

3.1. Software environment

The software environment in which VASITOS was developed consists of two sections: Spyder and PyGMO. The former was used to model the propagation of the orbit, while the latter was used for optimization. Spyder is an integrated development environment (IDE) that combines various open source packages written in Python [8]. These include some for scientific computing (NumPy and SciPy) and other for plotting (Matplotlib). It offers several advantages over other programs for scientific computing, mainly that it is an open source and that it is written in Python, a language, which is quite intuitive.

The Parallel Global Multiobjective Optimizer (PaGMO) is an optimization toolbox created by the Advanced Concepts Team at the European Space Agency (ESA) to solve complex optimization problems [9]. It is available for C++ and Python (the Python version is called PyGMO). The software features the generalized island model (GIM), which allows parallel computing in order to reduce computation time. PyGMO includes several optimization algorithms and global optimization problems, such as the genetic algorithm (GA), differential evolution (DE), particle swarm optimization (PSO), and adaptive simulated annealing (ASA), among others. The parallel computing scheme was implemented in the software and optimization simulations were performed in a Lenovo U410 with an Intel Core i5. This has multithreading, which means the operating system can identify up to four CPUs. Therefore, four islands were included in the parallel computing scheme.

To operate VASITOS, the user will input the initial and target orbit into the tool, along with the thruster specifications. VASITOS will run simulations until the end condition specified for the optimization algorithm is met. For example, for GA and DE, one must define the number of generations required in the simulation. The output will be the optimal path, propellant mass

consumed, time of flight, and the offsets. These are defined as the difference between the target state and the final simulated state. If the results meet the mission requirements, then the user will process them further by creating plots and analyzing which trajectory is best based on mission needs.

3.2. Propagation

Spherical coordinates were the preferred method of modeling for this project since it has been successfully used for first-order mission analysis of interplanetary trajectories, resulting in an efficient computation time [10]. The position of the spacecraft in the two-dimensional Euclidean space is defined by the radius vector and the angle θ . The x-y coordinate system is centered at the main body (Sun for interplanetary trajectories). At the center of the satellite, there is another coordinate system defined, consisting of the radial axis and the θ axis. The velocity vector, originated at its center of mass, defines the velocity of the spacecraft. Another vector that starts at the same position is the thrust vector. The angle between the θ axis and the thrust vector is called the pitch angle (α). It is one of the control parameters in the optimization problem (further explained in the following chapter). The radial and tangential acceleration components due to thrust are defined as:

$$a_{r,T} = \frac{T}{m} \sin \alpha \quad (1)$$

$$a_{\theta,T} = \frac{T}{m} \cos \alpha \quad (2)$$

where m is the mass of the spacecraft. The state can then be defined using four parameters: r , θ , v_r , and v_θ , where the last two parameters are the radial and tangential velocity, respectively. The mass of the spacecraft must be included as well, since it is using propellant to transfer from one orbit to the other. Therefore, the final state $\mathbf{X}$ is defined as:

$$\mathbf{X} = [r, \theta, v_r, v_\theta, m]^T \quad (3)$$

Once the state parameters were selected, the following step is to define their rate of change. This is essential to compute the future state. They are defined as [11]:

$$\dot{r} = v_r \quad (4)$$

$$\dot{\theta} = v_\theta \quad (5)$$

$$\dot{v}_r = \frac{\mu - rv_\theta^2}{r^2} + \frac{T}{m} \sin \alpha \quad (6)$$

$$\dot{v}_\theta = \frac{v_r v_\theta}{r} + \frac{T}{m} \cos \alpha \quad (7)$$

$$\dot{m} = -\frac{2\eta P}{(g_0 I_{sp})^2} \quad (8)$$

where μ is the gravitational parameter of the central body and T is the thrust of the low-thrust system. Most variables in equation $\dot{m}$ are engine specifications: η is its efficiency, P is the power, and I_{sp} is the specific impulse. The parameter g_0 is the standard acceleration due to gravity. The thrust magnitude is defined as:

$$T = \frac{2\eta P}{g_0 I_{sp}} \quad (9)$$

The equation shows that the thrust magnitude and specific impulse are inversely proportional, meaning that if one is increased, then the other is decreased. For this study, it is assumed that the engine efficiency and power are constant, so the specific impulse is an independent variable while the thrust is the dependent one. This will be important when selecting the former variable as a control parameter. Once the initial state of the system is defined, it can be combined with this system of equations to compute the state of the spacecraft at future times using an integrator.

3.3. Optimization

The rates of change of the state parameters are essential to form the Hamiltonian. In the context of optimal control theory, the Hamiltonian does not possess any physical meaning; it is a parameter derived from calculus of variation, which aids in finding the optimal trajectory. In a recent study, optimal control theory was applied to a spherical system, which only considered the radius, radial velocity, and tangential velocity [12]. Additionally, the only control parameter defined was the pitch angle. This chapter expands on previous work by including the position θ of the spacecraft within the trajectory and the mass of the vehicle. Furthermore, it includes the specific impulse as a control parameter. For the system defined in Section 3.2, the Hamiltonian is expressed mathematically as:

$$H = \lambda_r \frac{dr}{dt} + \lambda_\theta \frac{d\theta}{dt} + \lambda_{v_r} \frac{dv_r}{dt} + \lambda_{v_\theta} \frac{dv_\theta}{dt} + \lambda_m \frac{dm}{dt} \quad (10)$$

where λ 's are the costates of each parameter that makes up the state. These costates represent the cost of changing one parameter relative to another. For example, if one simulates a transfer, where the change in radius is much greater than the change in angle θ , then the costates of the radius and radial velocity will be greater in magnitude than the ones associated with θ . The rate of change of the costates over time can be obtained by using the following property derived from optimal control theory:

$$\dot{\lambda}_i = -\frac{\delta H}{\delta i} \quad (11)$$

This results in the following expressions:

$$\dot{\lambda}_r = \frac{v_\theta^2 \lambda_{v_r} - v_r v_\theta \lambda_{v_\theta}}{r^2} - \frac{2\mu \lambda_{v_r}}{r^3} \quad (12)$$

$$\dot{\lambda}_{\theta} = 0 \quad (13)$$

$$\dot{\lambda}_{v_r} = -\lambda_r + \frac{v_{\theta}\lambda_{v_{\theta}}}{r} \quad (14)$$

$$\dot{\lambda}_{v_{\theta}} = \frac{-2v_{\theta}\lambda_{v_r} + v_r\lambda_{v_{\theta}}}{r} \quad (15)$$

$$\dot{\lambda}_m = \frac{T}{m} \frac{\lambda_{v_r} + \lambda_{v_{\theta}}}{\lambda_m \sqrt{\lambda_{v_r}^2 + \lambda_{v_{\theta}}^2}} \quad (16)$$

Computing the costates is of the utmost importance in optimal control theory since the control parameters depend on them. For this study, there are two of them: the thrust direction and the thrust magnitude. The former is defined as the angle of attack α . The latter is inversely proportional to the specific impulse, meaning that if we control the specific impulse, we control the thrust magnitude. To obtain the profile of both control parameters, we need to use Pontryagin's Minimum Principle, which is expressed mathematically as:

$$\frac{\delta H}{\delta u} = 0 \quad (17)$$

where u is the control parameter. Since we have two control parameters, the resulting equations are:

$$\frac{\delta H}{\delta \alpha} = 0 \quad (18)$$

$$\frac{\delta H}{\delta I_{sp}} = 0 \quad (19)$$

By solving these two equations, we obtain the following control laws:

$$\sin \alpha = -\frac{\lambda_{v_r}}{\sqrt{\lambda_{v_r}^2 + \lambda_{v_{\theta}}^2}} \quad (20)$$

$$\cos \alpha = -\frac{\lambda_{v_{\theta}}}{\sqrt{\lambda_{v_r}^2 + \lambda_{v_{\theta}}^2}} \quad (21)$$

$$I_{sp} = \frac{2m\lambda_m}{\sqrt{\lambda_{v_r}^2 + \lambda_{v_{\theta}}^2}} \quad (22)$$

The angle of attack is divided into sine and cosine to ensure the right sign (+/-). It is important to use the atan2 function when computing the magnitude and direction of this angle. The optimal specific impulse I_{sp}^* defines the optimal thrust T^* in the following fashion:

$$T^* = \frac{2\eta P}{g_0 I_{sp}^*} \quad (23)$$

The value of the optimal specific impulse will depend on the boundaries defined by the engine specifications. This is expressed mathematically as:

$$I_{sp,L} < I_{sp}^* < I_{sp,U} \quad (24)$$

where $I_{sp,L}$ and $I_{sp,U}$ are the lower and upper boundaries of the specific impulse, respectively. If the user wishes to introduce coast arcs (assuming that the specific impulse of the engine is constant), then the following bang-bang strategy is applied:

$$\text{if } I_{sp} > I_{sp}^* \text{ then } T^* = T \quad (25)$$

$$\text{else if } I_{sp} < I_{sp}^* \text{ then } T^* = 0 \quad (26)$$

Now, there are 10 equations for rate of change of the state and costate parameter (5 equations for states and 5 for costates). We also have the initial and final values for the states, which are defined by the users. The only thing we are missing is the initial values for the costates. These are called the design variables and are stored in the decision vector, which is defined as:

$$\xi = [\lambda_r(0), \lambda_\theta(0), \lambda_{v_r}(0), \lambda_{v_\theta}(0), \lambda_m(0)]^T \quad (27)$$

The goal of the optimization process is to find the decision vector that minimizes the following cost function:

$$J = W_r \Delta r + W_\theta \Delta \theta + W_{v_r} \Delta v_r + W_{v_\theta} \Delta v_\theta \quad (28)$$

where the Δ 's are the offsets (defined as the absolute difference between the final simulated value and target value for selected state parameters) and the W 's represent the weights assigned to each offset. The weights are selected by the user and are modified according to the mission needs. This optimization method is indirect since the function we are minimizing does not include the main parameter to minimize: the time of flight. By obtaining the optimal costate profiles and ensuring the final conditions are met, the time of flight is ensured to be minimized (which is why the method is called indirect). For this project, the optimal decision vector was obtained using a numerical method called differential evolution, which is part of the family of evolutionary algorithms. A detailed description of the algorithm can be found in [13].

4. Thrust control strategies

Electric propulsion systems have considerable potential for interplanetary travel, but to analyze its feasibility, one has to consider not only the spacecraft's optimal path, but thrust strategy. Three strategies are considered in this study:

- Continuous thrust
- Coasting
- Variable thrust

The first one consists of operating at a constant thrust throughout the trajectory, meaning that the engine is operating continuously. The second strategy consists of using “coast arcs,” defined as periods where the engine is not producing thrust. Finally, variable thrust control will be tested given that the VASIMR has the ability to modify this parameter given that it features variable specific impulse.

4.1. Simulation parameters

Each thrust strategy was considered for a transfer from Earth’s orbit to Mars’ orbit in a two-dimensional heliocentric reference frame. Furthermore, it was assumed that the orbits of both planets are circular. The initial and final orbital parameters are displayed in **Table 1**. It can be observed that the final position in the target orbit is not specified, since the aim in these simulations is to reach the orbit, not the planet. The forces acting on the spacecraft are due to

	Initial orbit: Earth	Target orbit: Mars
Radius, r (km)	149.597×10^6	227.937×10^6
Velocity, v (km/s)	29.785	24.130
Position, θ (deg)	0.0	—

Table 1. Initial (Earth) and target (Mars) orbits to test control strategies.

Figure 2. Transfer from Earth to Mars orbit using continuous thrust.

the Sun’s gravity and the thrust produced by the engine. Third body perturbations from the planets on the spacecraft are not considered, nor the position of the planets on arrival and departure of the spacecraft.

The spacecraft was assumed to have a wet mass of 4500 kg, with a propellant mass of 1500 kg, and a VASIMR engine with 150 W of power and 65% efficiency. The specific impulse ranges from 5000 to 30,000 s, which are the theoretical limits of the engine [14]. When operating at a constant specific impulse, it was assumed that the specific impulse is equal to the lower boundary. The step size defined in the simulation was 24 hours, while the integrator used for propagation was the fourth order Runge-Kutta method. The differential evolution algorithm was set to a population size of 20, running for 500 generations.

Continuous thrust	Coasting	Variable thrust
1303 kg	1267 kg	1267 kg

Table 2. Propellant consumption for Earth to Mars transfer for three different thrust strategies.

Figure 3. Semi-major axis profile for Earth-Mars trajectories using different thrust control methods. Top image displays the complete profile, while the bottom figure displays the profile at mid-flight (blue line = continuous thrust, green line = coasting, and red line = variable thrust).

4.2. Results

Figure 2 presents the results of a transfer from Earth to Mars orbit in a heliocentric reference frame in astronomical units (AU). The dashed inner circle represents Earth's orbit, while the dashed outer circle represents Mars' orbit. The curve represents the spacecraft's trajectory, while the arrows represent the thrust magnitude and direction. This last parameter demonstrates how the thrust direction was controlled to obtain the optimal trajectory. The spacecraft starts thrusting almost normal to the velocity vector and reverses direction at approximately mid-flight until reaching the final orbit. The thrust magnitude is not considered as a control parameter for this simulation since the thrust is assumed to be continuous. The final orbital trajectory results in a time of flight of 185.78 days and a propellant consumption of 1303 kg.

The same transfer was computed for the coasting and variable thrust case. For both of these cases, the time of flight was set to 185.78 days, which was the optimal time for the continuous thrust case. The propellant consumed to achieve the transfer for each case is presented in **Table 2**.

Figure 4. Eccentricity profile for Earth-Mars trajectories using different thrust control methods. Top image displays the complete profile, while the bottom figure displays the profile at mid-flight (blue line = continuous thrust, green line = coasting, and red line = variable thrust).

From **Table 2**, one can observe that using coasting and variable thrust results in a 2.7% reduction in propellant consumption relative to the continuous thrust strategy. To properly understand why this reduction occurs, one must analyze the in-plane orbital elements, as well as the thrust profile for each control method.

Figures 3 and 4 present the semi-major axis and eccentricity profile for the three thrust control methods, respectively. Additionally, **Figure 5** presents the specific impulse for each case. With this figure, the thrust profile can be deduced, given that the specific impulse is inversely proportional to the thrust of the engine. Presenting the specific impulse was favorable to ensure that the engine is operating within its limits. For the coasting case, the specific impulse was set to infinity during periods when the spacecraft is required to coast as dictated by the control law, resulting in zero thrust.

From **Figure 3**, it can be observed that the overall trend in the semi-major axis is an increase throughout the trajectory, except at approximately the halfway point. Here, there exists a considerable decrease in this parameter because the spacecraft performs a radical change in thrust direction: nearly 180° . With the use of a coast arc, the majority of the change of direction is performed without thrust (see **Figure 5**), meaning that there is a smaller change in the semi-major axis during this period, resulting in a lower loss of orbital energy. The strategy is more

Figure 5. Specific impulse profile for Earth-Mars trajectories using different thrust control methods at mid-flight (blue line = continuous thrust, green line = coasting, and red line = variable thrust).

efficient compared to the constant thrust case, given that it requires less propellant. A similar phenomenon is observed when using variable thrust, where the thrust is lowered when performing the change of direction (see **Figure 5**).

Figure 4 explains why the spacecraft performs the rapid change in thrust direction. It is observed that the change in the eccentricity can be divided into two segments: the first one is a uniform increase while the second one is a uniform decrease. The change occurs at the halfway point, where the spacecraft performs the turn. The eccentricity profile is similar to the Hohmann transfer, considered an optimal transfer strategy for chemical rockets. In this type of transfer, the semi-major axis and eccentricity are increased instantly (modeled as an impulsive burn) when the spacecraft enters the transfer orbit and then the former is further increased but the latter return to zero.

4.3. Variable time of flight

In Section 4.2, it was observed that using coasting or variable thrust resulted in a more efficient transfer than using continuous thrust. Another advantage when using these two control strategies is that the mission designer can vary the time of flight to transfer from the initial to the target orbit, to account for the position of the target planet when the spacecraft arrives at its orbit. By varying the time of flight for this transfer, one could also analyze which control strategy would be best for different flight periods. To achieve this goal, both coasting and variable thrust methods were tested for Earth to Mars transfers using fixed time of flights of 195, 205, and 215 days. The results for the propellant consumption for each case are presented in **Figure 6**, along with the case presented in Section 4.2.

Figure 6. Propellant consumption for Earth to Mars transfers for different cases of time of flight using three thrust control strategies (gray dot = continuous thrust, orange line/dot = coasting, and blue line/dot = variable thrust).

Figure 7. Transfer from Earth to Mars orbit using variable thrust (left) and coasting (right).

Figure 8. Specific impulse profile for Earth-Mars trajectories using different thrust control methods (blue line = variable thrust, and red line = coasting).

Figure 6 displays that as the time of flight increases, the variable thrust control strategy is more efficient than coasting in terms of propellant. To properly understand this phenomenon, the trajectory for both strategies was plotted for the case where time of flight was equal to 215 days. For this case, the propellant mass was reduced by 12% when using variable thrust compared to coasting.

Figure 7 shows the trajectory in blue, with the red arrows representing the thrust magnitude and direction. The thrust direction profile is similar to what was computed in Section 4.2, with the main difference being that the trajectory is longer, since the time of flight defined is approximately 30 days greater. The thrust magnitude for the variable thrust control strategy is constant at the beginning, but decreases as the spacecraft starts to change direction. At mid-flight, this parameter reaches its minimum but then starts increasing until it reaches at maximum at the end of the trajectory. For the coasting strategy, it is observed that the thrust is constant until approximately a quarter of the time of flight, when the coasting period begins. The thrust resumes in the opposite direction when there is a quarter of the time of flight remaining. This can also be observed in **Figure 8**, where the profile of the specific impulse is plotted. It is seen that using variable specific impulse creates a more gradual change in the orbit, when compared to the coasting mechanism, resulting in a more efficient transfer with a lower propellant consumption. Additionally, it is observed that the engine operates at its highest specific impulse for approximately 35 days, demonstrating the importance of achieving these high levels of specific impulse for interplanetary orbits.

5. Conclusion

Growing interest in high-power electric propulsion systems motivated the analysis of their performance when used to transfer from Earth to Mars orbits. VASITOS was created to study not only the optimal thrust direction, but the optimal thrust magnitude as well. Three thrust control laws were studied: continuous thrust, coasting, and variable thrust. By using a 150 kW thruster with a specific impulse of 5000 s and an efficiency of 0.65 on a 4500 kg spacecraft, it was computed that the optimal time of flight for the transfer using constant thrust was 185.78 days. Additionally, it was observed that there was a loss in orbital energy mid-way through the transfer. By using a variable specific impulse system (with boundaries of 5000–30,000 s), the propellant consumption was reduced by 2.7% due to the system's ability to throttle down at the point where the energy loss occurred. The coasting strategy resulted in a 2.7% propellant reduction as well since the engine stopped thrusting at the point of energy loss. Further results include the comparison of the coasting and variable thrust strategies for fixed time of flights. As the time of flight was increased, it was observed that the propellant consumption of the former strategy was less than the latter. For example, for a fixed time of flight of 215 days, the propellant consumption of the variable thrust strategy was 12% less. From these simulations, it was concluded that the best thrust control law for Earth to Mars transfers was variable thrust, due to its ability to gradually change the orbit relative to the other methods studied, resulting in a lower propellant consumption.

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