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Electric Propulsion for Satellite Collision Avoidance and Orbit Maintenance

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Abstract

The increasing adoption of electric propulsion technology for orbital control in commercial satellites is driven by its high specific impulse (Isp), which enables reduced propellant consumption and extended mission lifetimes compared to chemical propulsion systems. This paper explores the application of electric thrusters for collision avoidance (COLA) maneuvers, focusing on the development of a continuous-thrust maneuver designed for space situational awareness and conjunction risk mitigation. The proposed methodology utilizes the impulse response characterization of the orbital dynamics to formulate an optimized control strategy, in which the thruster operates in a discrete on/off mode. The key control parameters optimized in this study include: (i) time to closest approach (TCA), (ii) burn duration, (iii) thrust vector orientation, and (iv) thrust magnitude as a percentage of the maximum available thrust. The objective is to analyze miss distance and collision probability as functions of these control parameters while ensuring compliance with satellite operational constraints. To provide a comprehensive feasibility assessment, the guidance and control method takes into account power consumption, battery discharge characteristics, and propellant usage based on the efficiency and specific impulse of the electric thruster. Case study simulations validate the proposed approach and demonstrate the practicality of electric thruster-based collision avoidance for operational satellite missions.

Keywords: electric propulsion, CubeSats, electric thruster, Hall thruster, collision avoidance, orbital mechanics, conjunction data message, collision avoidance maneuver

Acronyms/Abbreviations

CDM	Conjunction data message.
COLA	Collision avoidance.
MPD	Magneto plasma dynamic Thruster.
GIT	Gridded ion thruster.
18 SDS	18th Space Defense Squadron.
TCA	Time of closest approach.
EP	Electric propulsion.
HET	Hall-effect thrusters.
PoC	Probability of collision,

Nomenclature

Ah	=	amp-hour requirement for a given burn time.
Isp	=	Specific impulse.
f_t , $\boldsymbol{t}_{\text{vec}}$	=	Thrust force and thrust vector.
$\dot{m}_p$	=	Propellant mass flow rate.
P	=	Electric power required for an electric thruster.
V	=	Satellite bus voltage.
η	=	Efficiency of the electric propulsion.
ω	=	Orbit angular velocity.
h	=	Satellite altitude above the Earth's surface.
P_c	=	Value of the probability of collision.
$\mathbf{r}_{\text{miss}}$, $\mathbf{r}^+_{\text{miss}}$	=	Miss-distances before and after COLA maneuver.
t_{b}	=	Burn time of the electric thruster.
$t2\text{tca}$	=	Time to TCA.
$\mathbf{Q}$	=	Covariance matrix of the position vector.

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1. Introduction

Electric thrusters, also known as electric propulsion (EP) systems, employ various technologies to achieve exceptionally high exhaust velocities, thereby minimizing the total propellant burden and reducing the corresponding launch mass of spacecraft. Due to their high specific impulse (Isp) and efficient propellant utilization, they are widely used in satellite applications. A system-level design for a small satellite equipped with electric thrusters for a specified mission is described in [1,23]. This study examines key trade-off parameters critical to mission lifetime, including mass, power availability, trajectory options, thrust duration, and thermal requirements. While electric propulsion systems generate lower thrust compared to chemical propulsion, they offer significant advantages for in-space propulsion. By decoupling energy from the propellant, they enable higher energy densities, making them highly efficient for long-duration missions [2,3]. In fact, the concept of electric propulsion for satellites has been explored since the late 1960s [4] and has since matured, as demonstrated by the recent adoption of all-electric thrusters for various satellite operations. These applications include geostationary station-keeping [5,6,7], drag compensation for small satellites and constellations [8], efficient transfers from geostationary transfer orbit (GTO) to geostationary Earth orbit (GEO) [9], formation flights [10], and the controlled de-orbiting of satellites at the end of their operational life [11]. For instance, [8] proposes the use of electric thrusters to maintain a given low Earth orbit as an alternative to traditional methods that rely on frequent ground-based control actions to compensate for atmospheric drag and other perturbative forces. Electric thrusters are particularly well-suited for small satellites, as they provide significantly higher exhaust velocities compared to chemical propulsion. This allows satellites to achieve the required delta-v with minimal propellant, which is crucial for extending mission life, especially for spacecraft with limited onboard resources. Moreover, the low-thrust nature of electric propulsion reduces dynamic disturbances, making it particularly beneficial for satellites carrying delicate instruments such as telescopes or high-precision sensors.

Figure 1: Artistic rendering of WildFireSat (courtesy of CSA)

While the use of electric thruster technology on satellites is widely discussed in the literature for various orbital control applications, relatively few studies have explored its potential for collision avoidance. The increasing population of both operational and defunct satellites, along with space debris, poses a growing risk of collisions, threatening critical space assets. Therefore, satellites equipped with electric thrusters for orbital maintenance should leverage the same onboard propulsion capability for collision avoidance maneuvers. This is essential not only for spacecraft safety but also for the long-term sustainability of the space environment. In fact, without effective mitigation strategies, collisions can trigger a cascading effect known as the Kessler syndrome [12], where debris from one collision leads to further collisions, exponentially increasing the amount of space debris. Collision alerts issued by organizations such as the 18th Space Defense Squadron (18 SDS) provide early warnings of potential collisions by tracking satellites and space debris larger than 10 cm [13]. When a satellite is equipped with a propulsion system, it can adjust its trajectory through a COLA maneuver, using impulsive thrust to avoid predicted close encounters [14,15,19,20]. From an operational perspective, a collision risk assessment and mitigation system is essential for ensuring satellite safety. These systems process Conjunction Summary Message (CDM), e.g., from 18th SDS and determine whether a COLA maneuver is necessary, executing maneuvers when a high-probability conjunction is detected.

This work is motivated by the need for collision avoidance maneuvers for satellites equipped with electric propulsion systems, such as the WildFireSat mission—a constellation of seven microsatellites dedicated to daily wildfire monitoring across Canada, Figure 1. This mission will provide critical data to fire managers, enabling them to track fire behavior, assess high-risk wildfires, and make informed decisions to protect Canadians, particularly in remote and northern communities. We present a method for designing a continuous-thrust collision avoidance maneuver. The key control parameters in the trade space include: (i) time to the time of closest approach (TCA), (ii) burn duration, (iii) thrust vector, and (iv) percentage of maximum thrust. The electrical energy and battery amp-hours required for the maneuver are computed based on the efficiency and specific impulse of the electric thruster. The system then calculates the resulting miss distance, probability of collision, battery amp-hour consumption, and propellant usage necessary to execute the maneuver.

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2. Satellite Collision Avoidance using Electric Thrusters

2.1 Modelling of Electric Thrusters

$$f_t = \dot{m}_p g_0 I_{sp}, \tag{1}$$

where f_t is the thrust force, $\dot{m}_p$ is the propellant mass flow rate, $g_0 = 9.81$ m/s ^{2} is the gravitational acceleration at Earth's surface, and I_{sp} is the specific impulse in seconds. Notably, I_{sp} quantifies propulsion efficiency and is related to the exhaust velocity v_e by

$$I_{sp} = \frac{v_e}{g_0}. \tag{2}$$

It is worth noting that electric thrusters typically achieve specific impulse values ranging from 1,000 to 10,000 seconds, significantly higher than the approximately 300 seconds characteristic of chemical thrusters. Then, the electric power required for the electric thruster unit can be calculated from

$$P = \frac{g_0 I_{sp}}{2\eta} f_t \tag{3}$$

where P is the eclectic power used by the thruster in Watt, η is the efficiency of the electric thruster with which the thruster converts its input electric power to thrust power [16]. Typically, thruster efficiency ranges between 0.4 and 0.7, depending on the specific thruster type. The performance parameters of various electric thruster types [3] are specified in Table 1.

Table 1: The performance parameters of different types of electric thrusters.

Type	Thrust (mN)	I_{sp} (s)	η	Propellant
Ion Thrusters	1-250	1000-10000	40%-80%	Xe, Kr, Ar
Hall-effect Thrusters (HET)	1-500	1500-2500	45%-55%	Xe, Kr, Ar
Magnetoplasmadynamic Thruster (MPD)	0.1-2	500-5000	30%-80%	Xe, Kr

Let t_b denote the burn time of the electric thruster and V represent the satellite bus voltage. Then, the propellant consumption Δm_p and the amp-hour requirement for a given burn time can be derived from (1) and (3) by integrating the thrust force:

$$\Delta m_p = \frac{1}{g_0 I_{sp}} \int_0^{t_\mathrm{b}} f_t \; dt \tag{4}$$

$$Ah = \frac{g_0 I_{sp}}{7200\eta V} \int_0^{t_\mathrm{b}} f_t \; dt. \tag{5}$$

If the thrust remains constant over the burn duration, (4) and (5) simplify to

$$\Delta m_p = \frac{t_\mathrm{b}}{g_0 I_{sp}} f_t \tag{6}$$

$$Ah = \frac{g_0 I_{sp} t_\mathrm{b}}{7200\eta V} f_t \; . \tag{7}$$

2.2 Orbital Mechanics and Collision Avoidance Maneuver Using Continuous Thrust

For a primary spacecraft in a circular orbit, orbital angular velocity is given by

$$\omega = \sqrt{\frac{\mu}{(R_e + h)^3}} \tag{8}$$

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where $R_e = 6.371 \times 10^6$ m represents Earth's mean radius, h is the spacecraft altitude above the Earth's surface, and $\mu = 3.986 \times 10^{14}$ m ^{3}/s ^{2} is Earth's gravitational parameter. The relative motion of a secondary object with respect to the primary spacecraft is encapsulated by the state vector:

$$\mathbf{x}^T = [\delta\mathbf{r}^T \quad \delta\dot{\mathbf{r}}^T], \tag{9}$$

where $\delta\mathbf{r}$ and $\delta\dot{\mathbf{r}}$ denote position and velocity changes. The time evolution of relative motion under perturbative effects follows the Clohessy–Wiltshire (CW) equations [17, 18]:

$$\frac{d}{dt}\mathbf{x} = \mathbf{A}\mathbf{x} + \mathbf{B}\mathbf{u}(t), \tag{10}$$

where

$$\mathbf{A} = \begin{bmatrix} \mathbf{0} & \mathbf{I} \\ \mathbf{L}(\omega) & -2[\boldsymbol{\omega}\times] \end{bmatrix}, \qquad \mathbf{B} = \begin{bmatrix} \mathbf{0} \\ \mathbf{I} \end{bmatrix}, \tag{11}$$

$\mathbf{L}(\omega) =$ diag $(3\omega^2, 0, -\omega^2)$, and $\mathbf{u}(t)$ is the acceptation vector due to the thrust. If the thruster burn duration is t_b and the vector force of the electric thruster is $\boldsymbol{f}_t$, then the acceleration input can be described by

$$\mathbf{u}(t) = \begin{cases} (f_t\ /m)\ \boldsymbol{t}_\mathrm{vec} & \text{if} \quad t < t_\mathrm{b} \\ \mathbf{0} & \text{otherwise} \end{cases} \tag{12}$$

Where $\boldsymbol{t}_\mathrm{vec}$ is thrust vector (unit vector) and m is the satellite mass, i.e.,

$$m(t) = m_i - \frac{1}{g_0 I_{sp}} \int_0^t f_t \ \ \mathrm{d}t \tag{13}$$

and m_i being the satellite's initial mass prior to the burn. The system response at the time to the Time of Closest Approach (TCA), denoted as $t2$tca, corresponds to the state solution of equation (10), given as follows:

$$\mathbf{x}(t2\mathrm{tca}) = \mathbf{x}(0) + \int_0^{t2\mathrm{tca}} \boldsymbol{\Phi}(t,\tau)\mathbf{B}\mathbf{u}(\tau)d\tau.$$

where $\boldsymbol{\Phi}(t,\tau) = e^{\mathbf{A}(t-\tau)}$ is the state transition matrix of the dynamics system (10). Assuming a negligible change in the satellite's mass and constant thrust during the burn time, i.e., $\boldsymbol{f}_t =$ const , the convolution integral above can be solved to yield the miss-distance variation in the following closed-form expression:

$$\delta\boldsymbol{r} = \frac{f_t}{m\omega^2}\begin{bmatrix} \cos(\omega\Delta t) - \cos(\omega t_{t2\mathrm{tca}}) & 2\sin(\omega\Delta t) - 2\sin(\omega t_{t2tca}) + 2\omega t_b & 0 \\ 2\sin(\omega t_{t2\mathrm{tca}}) - 2\sin(\omega\Delta t) - 2\omega t_b & 4\cos(\omega\Delta t) - 4\cos(\omega t_{t2\mathrm{tca}}) + \frac{3}{2}\omega^2 {t_b}^2 - 3\omega^2 t_{t2\mathrm{tca}} t_b & 0 \\ 0 & 0 & \cos(\omega\Delta t) - \cos(\omega t_{t2\mathrm{tca}}) \end{bmatrix} \boldsymbol{t}_\mathrm{vec} \tag{14}$$

where $\Delta t = t_{t2\mathrm{tca}} - t_b$. Updating the miss distance based on the change caused by the thrust burn, we get:

$$\mathbf{r}^+_\mathrm{miss} = \mathbf{r}_\mathrm{miss} + \delta\boldsymbol{r} \tag{15}$$

where $\mathbf{r}_{\mathrm{m}iss}$ and $\mathbf{r}^+_{\mathrm{m}iss}$ represent the miss-distances before and after the COLA maneuver. The revised probability of collision is then computed from the covariances of the position vectors of the primary and secondary satellites in the new miss distance calculated in (15). The probability of spacecraft collision at TCA, $P_c = P_c(\mathrm{t2tca})$, can be described by the following equation of probability of collision.

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$$P_c = \frac{1}{(2\pi)^{\frac{2}{3}}\sqrt{\det(\mathbf{Q})}} \int\int\int_V e^{-\frac{1}{2}\delta\mathbf{r}^T\mathbf{Q}^{-1}\delta\mathbf{r}}\,\mathrm{d}x\mathrm{d}y\mathrm{d}z, \tag{16}$$

where $\mathbf{Q}$ is the position covariance matrix of the two objects. Notice that the integration is over volume V, which is a sphere centered at the location of miss distance $\delta\mathbf{r}(\text{tca})$ and with the radius of R_c equals to the had-body radius of the primary and the secondary objects [20,21,22]. Notice that R_c is a combined hard-body radius of the primary and secondary objects. It has also been shown that the covariances associated with the position vectors of the primary and secondary satellites can be combined into a new covariance matrix within the encounter plane [21,22]. This matrix represents the sum of the two individual covariances after an appropriate rotational transformation in the following form:

$$\mathbf{\Sigma} = \begin{bmatrix} \sigma_x^2 & \kappa\sigma_x\sigma_z \\ \kappa\sigma_x\sigma_z & \sigma_z^2 \end{bmatrix}. \tag{17}$$

The probability of collision, as given by the integral in (16), can then be approximated by

$$P_c = \frac{R_c^2}{2\sigma_x\sigma_{zeq}} \exp\left(-\frac{\|\mathbf{r}_{miss}^{+}\|^2}{2\sigma_{zeq}^2}\right) \tag{18}$$

where $\sigma_{zeq} = \sigma_z\sqrt{1-\kappa^2}$ represents the equivalent position error in z-direction with zero correlation [21].

2.1 Optimal COLA Maneuver

The matrix equation in (14) can be expressed in the following compact form, as a function of the burn time and thrust vector

$$\delta\mathbf{r} = \mathbf{M}(t_b)\mathbf{t}_{\text{vec}} \tag{19}$$

The objective now is to determine the minimum burn time t_b and an appropriate thrust vector such that the post-maneuver miss distance exceeds the acceptable threshold r_{th}. This problem can be formulated as a linear optimization program subject to quadratic equality and inequality constraints:

$$\begin{aligned} \text{minimize} \quad & t_b \\ \text{subject to:} \quad & \| \mathbf{M}(t_b)\mathbf{t}_{\text{vec}} + \mathbf{r}_{\text{miss}} \| \geq r_{th} \\ & \| \mathbf{t}_{\text{vec}} \| = 1 \\ & t_{t2tca} \geq t_b \geq 0 \end{aligned} \tag{20}$$

The above problem can be solved in terms of the design parameters $\{t_b, \mathbf{t}_{vec}\}$.

3. Case Study Scenario

In this case study, we assume the primary satellite operates in a 600 km altitude orbit above Earth's surface, with a mass of 200 kg and a bus voltage of 24V. We then examine a continuous-thrust collision avoidance maneuver to mitigate conjunction risks with a secondary object. The analysis considers two different types of electric thrusters onboard the satellite, as detailed in Table 2, which presents the specifications of the two Hall thrusters [3] used in this study. The key control parameters explored in the trade space include: i) Time to TCA; ii) Burn duration; and iii) Percentage of maximum thrust. In this case study, we also assume a secondary object in conjunction, with a miss distance of 100 m at the TCA and a probability of collision of 2×10^{-3}. In this operation scenario, a COLA maneuver action is required to increase the separation distance beyond the 1000 m threshold and reduce the collision probability below the acceptable limit of 10^{-6}.

Table 2: Parameters of specific electric thrusters considered for the case study.

Name	Type	Thrust (mN)	I_{sp} (s)	Power (W)	Propellant
HT100	Hall thrusters	9	1300	175	Xe
MiXi	GIT	1.4	30	30	Xe

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3.1 COLA analysis

Figure 2 shows plots of the miss-distance and the probability of collision curves against the time to TCA and burn time associated with the given conjunction assuming the primary satellite uses an HT100 Hall thruster for active collision avoidance maneuver; see Table 2. The graphs in Figure 3 shows the required propellant and electric power versus burn duration for the collision avoidance maneuvers. It is evidence from the trade space plots that at least 7 minutes burn is needed to improve the miss distance and PoC parameters to the acceptable level the optimal times to start the burn are 2.6 hours or 4.1 hours prior to TCA. From the plots in Figure 4, it is apparent that the thrust burn for this COLA maneuver requires 0.3 grams of Xe propellant and 0.8 Ahr of battery power. Figures 5 and 6 illustrate the trade space results for the same CDM when the primary satellite is equipped with thruster type MiXi; see Table 2.

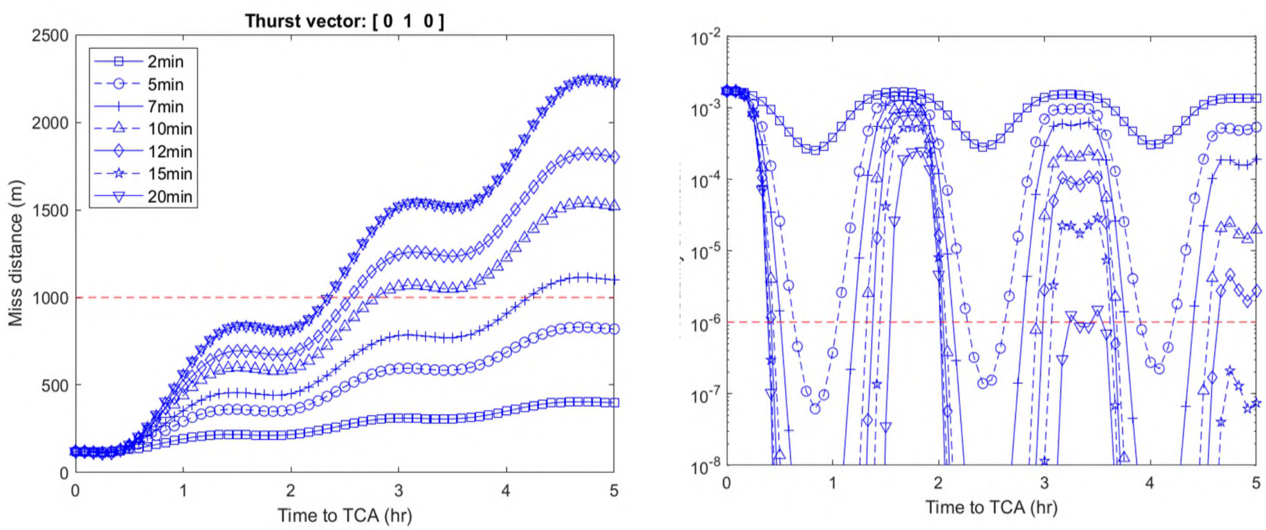

Figure 2: The miss-distance curves (left) and POC curves (right) versus time to TCA and burn time for the HT100 Hall thruster

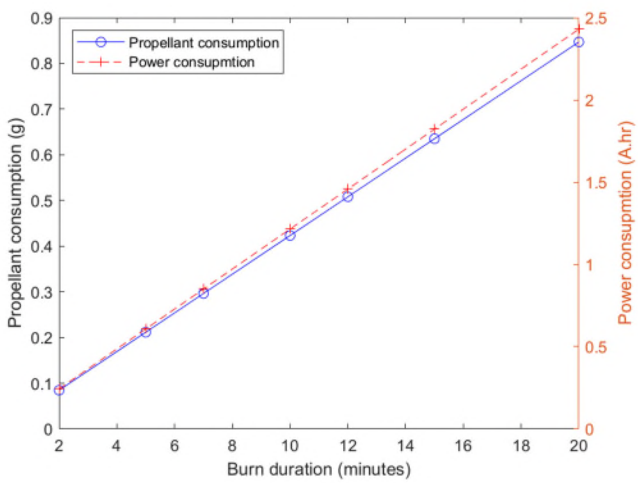

Figure 3: Propellant and power consumptions versus burn time for the HT100 Hall thruster.

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Figures 4 and 5 show the trade space results for the same CDM when thruster type MiXi is selected. The plots in the figure clearly show that in this case, a 50-minute burn starting 2.5 hours before TCA, requiring just 0.15 grams of Xe propellant and 1.2Ahr of battery power, is sufficient to bring the miss distance and PoC parameters to an acceptable level.

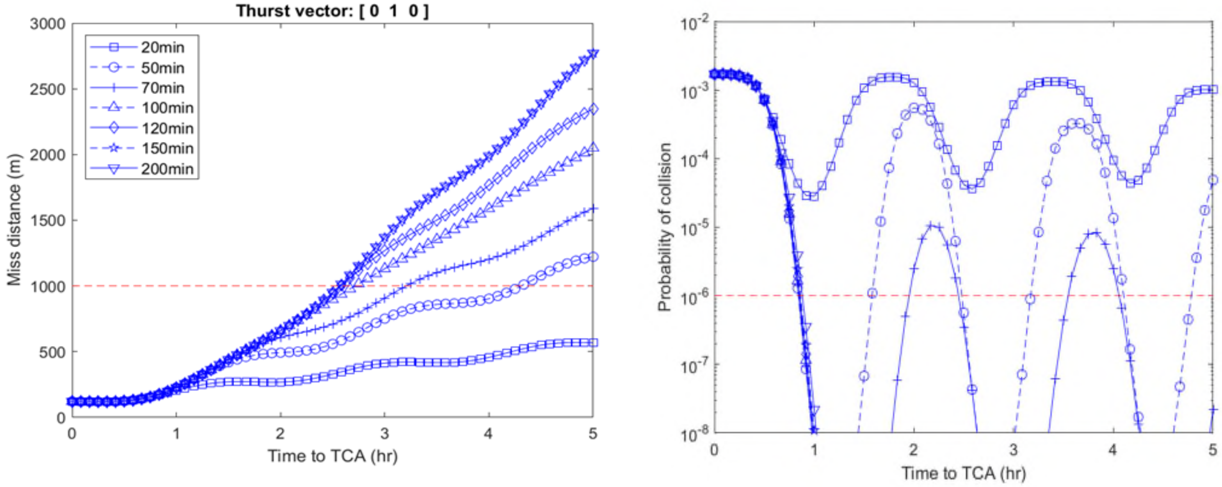

Figure 4: The miss-distance curves (left) and POC curves (right) versus time to TCA and burn time for the MiXi thruster.

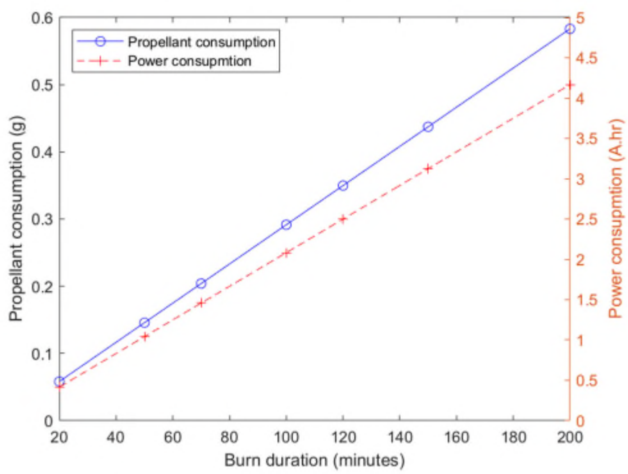

Figure 5: Propellant and power consumptions versus burn time for the MiXi thruster.

4. Conclusions

This study examined the application of electric thrusters for collision avoidance maneuvers, leveraging their high specific impulse to enhance fuel efficiency and extend satellite mission lifetimes. A continuous-thrust COLA manuever strategy was developed based on the impulse response characterization of the orbital dynamics. This capability allowed optimization of key control parameters, including time to closest approach (TCA), burn duration, thrust vector orientation, and thrust magnitude, to maximize miss distance and minimize collision probability while adhering to operational constraints. Moreover, the method quantified power consumption, battery discharge, and propellant usage, providing a comprehensive assessment of the maneuver's feasibility. Case study simulations demonstrated the effectiveness of the proposed approach, showcasing the potential of electric propulsion-based CDMs for enhancing space situational awareness and collision risk mitigation in operational satellite missions.

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