Feasibility of Utilizing Current Propulsion Technologies in Support of Very Low Earth Observation Space Platforms

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1. Abstract

An emerging low earth orbit spacecraft platform is being sought after due to many advantages of operation at low earth orbits of altitude less than 450Km. These encompasses better imagery and enhanced communications capabilities in addition of spacecraft debris self-clearing. However, operation at low earth altitudes requires a propulsion system that can counter react the atmospheric drag which is the dominant force acting on the spacecraft. The paper, therefore is focused on measures and methodology to select off the shelf ion thrusters in order to enhance the life time of low earth orbiting platforms. The study is focused on three types of ion thrusters manufactured by Busek. Models investigated are the BIT-1, BIT3 and BIT-7. The analysis, revealed, BIT-1 thruster is suitable for larger CubeSats at elevations beyond 300 km. Smaller CubeSats provide too little power and cannot provide the required thruster power. BIT-3 thruster is only suited to large CubeSats (24U and upwards) as they can gain the solar power required to run the thruster. BIT-7 thruster is unsuitable for any CubeSat arrangement as its required thruster power level that is beyond the means of the CubeSats in question. BIT-7 would be more suited to larger satellites. The analysis is based on solar energy consumption of non-deployable solar panel. The analysis and methodology implemented will accelerate the development of spacecraft platforms by relaxing the design requirements through selecting off the shelf suitable propulsion system. This will also reduce the time requirement for design and development and will lower the cost associated with development.

2. Introduction

There is an ongoing trend in operations of space platforms that are more sustainable and can provide better earth observation and communications capabilities. These platforms operate at relatively very low earth orbits of altitudes less than 500km. Operating at these altitudes subject the space platforms to high magnitude of atmospheric drag. To increase the operation life times and extend earth observation and communication critical missions, a force that counter react the drag is necessary. From the sustainability point of view, the cost of fabricating of such platforms is less expensive due less stringent requirements on the payload. The platform proximity to Earth implies better imagery operational capacities with less advanced optics, cameras and antennas. Simplifying the payloads will lower the cost and weight which means less demanding financial burdens on companies to enter the market. Furthermore, at End of Life due to higher gravitational pull of Earth, a self- decay is encountered and no space debris will remain in space orbits.

To counter react the atmospheric drag a propulsion system should be developed. There are a variety of space propulsion systems that is available on the market. These can be termed as cold gas thrusters, electrostatic and electromagnetic thrusters. Cold gas thrusters can provide the highest magnitude of thrust, but very low specific impulse. On the other hand, both electrostatic and electromagnetic thrusters have better capabilities, and are currently used on space platforms extensively. Electrostatic thrusters due to its high specific impulse is used for satellite de-orbiting and orbit transfer and final orbit insertion, while electromagnetic pulsed plasma thrusters are more utilized for attitude control. In general, electromagnetic thrusters are less energy hungry systems as compared with electrostatic thrusters. It is worth noting that operation at those low altitudes creates some other challenges, the ionized gas environment can impose shorten life of instruments and sensors. Atomic oxygen reacting with materials can cause performance degradation due surface corrosion.

3. Electric Propulsion General Scaling:

There are wide variety of electric propulsion technologies that are available in the market dedicated to various applications. This makes developing unified scaling approach more challenging than other types of propulsion systems for example such as chemical propulsion systems. Therefore, the approach adopted in this paper is by looking at the system as a whole, based on this methodology the thrust can be written as the multiplication of the mass flow rate by the specific impulse.

$$T = \dot{m} I_{sp} = 2 P_R n / Isp \qquad (1)$$

Where, $\dot{m}$ is the mass flow rate, I_{sp} the specific impulse, P_R is the required power, and n is the thruster efficiency. Assuming the specific impulse and the thrust efficiency are constant, then the thrust is proportional to the mass flow rate and the required power.

The observation based on equation 1 is the thrust is proportional to the mass flow rate, however the mass flow rate scales to the cube of the characteristic linear dimension of the spacecraft. On the other hand, the power available is based on the energy consumption by the solar panels. This implies, the power available can be scaled based on square of linear dimension. The power required to power available ratio is proportional to characteristic linear dimension of the spacecraft.

$$\frac{P_{Req}}{P_{available}} \; \alpha \; \frac{mass}{area} \; \alpha \; \frac{L^3}{L^2} \; \alpha \; L \qquad (2)$$

By examining at the above relation, a large propulsion power is required as the spacecraft is getting smaller in size, given that the thrusters scaled to be smaller in size accordingly. For drag force that must be counter reacted, the drag force will depend on the frontal area and the ratio of power available to power required is independent of the spacecraft dimensions based on relation 2.

4. Ion electric propulsion-Scaling

The systems that are most used to this day are the Gridded Ion Engine and the Hall Effect Thruster [1]. The methods of propulsion for both systems are similar but differ in some ways. The basis of ion propulsion is ionization of the fuel to produce ions. This is often done using electron bombardment where electrons collide with propellant atoms, causing the atoms to expel electrons, leaving behind positively charged ions i.e. a plasma. Electrons are produced via thermionic emission of a hollow discharge cathode [2]. From here, the systems differ. For the Gridded Ion Engine, the electrons are then attracted to the discharge chamber walls that contain the propellant [2]. Strong magnets are used to direct the electrons into the discharge chamber [2]. After bombardment, the ions travel to the first of two grids containing thousands of apertures [2]. The voltage drop between the two grids causes the ions to accelerate to very high speeds out of the second grid [2]. The ions exit as an ion beam, leading to thrust. To prevent the ions from coming back to the thruster, a neutralizer is used to expel an equal number of electrons, effectively leaving the thruster neutral [2].

The net screen voltage required across the accelerator to achieve a certain exhaust velocity [1] is given by

$$V_S = \frac{m_i v_i^2}{2q} \qquad (3)$$

The most commonly used propellant is Xenon and $m_i$the atomic mass of Xenon. and q is the charge of the ion particle. v_i is the velocity of the ion particle. For a given grid spacing and voltage, a certain current of ions can pass through a unit area of the grid. This is termed, the space charge limited current and is provided by the Child-Langmuir Law [1]:

$$I_B = \frac{4}{9} A \varepsilon_0 \frac{V_S^{\frac{3}{2}}}{x_i^2} \sqrt{\frac{2q}{m_i}} \qquad (4)$$

In equation 4, I_B is the beam current, A is the thruster exit grid area, x_i is the grid spacing and v_s is the applied accelerating voltage and ε_0 is the permittivity of vacuum. Based on equation (4), the thrust can be written as

$$T = \frac{8}{9} A \varepsilon_0 \frac{V_S^2}{x_i^2} \qquad (5)$$

For scaling purposes associated with ion thrusters there are two processes that are relevant for scaling discussion, the first is the acceleration which determines the thrust and the specific impulse and the second is the ion creation process which determines the efficiency of the device. In order for the efficiency to be constant as the thruster chamber gets smaller in size, this requires that the physics will remain the same compared with the unscaled propulsion system. In order for the efficiency to remain constant, the temperature of the electrons as well as ions and neutrals should remain constant, this assumption is essential to ensure that the scaling of the various parameters is viable and successful. To maintain the same frequency of collisions, in the scaled thruster the variation of mean free path λ should scale with the characteristic thruster length L, which implies

$$\lambda \sim \frac{1}{nQ}\ \alpha\ L \qquad (6)$$

Where n is the number density of given species and Q the relevant collisional cross section. Since the assumption is Q remains constant, the cross section is only dependent on temperature, then Q remain constant. The conclusion is the number density must scale inversely with characteristic length. Another aspect related to scaling, as the thruster chamber gets smaller, the electron containment characteristics should be similar to the original thrusters. This implies that the electron gyro should also scale with the characteristic thruster length.

$$R_L = \frac{C_e m_e}{eB}\ \alpha\ L \qquad (7)$$

Where R_L is the gyro radius, $C_e$is the mean electron speed, m_e is the mass of the electron, e is the electron charge and B the magnitude of the local magnetic field. However, the mean electron speed is a function of temperature which is assumed constant, then B must be scaled inversely as compared with the characteristic length. This implies higher magnetic field for electron containment as the thruster length gets smaller. This means that a lower limit of the length size will be reached at a certain point since it will be impossible to come up with large magnetic field to confine or contain the electrons. The feasibility to maintain the ion engines efficiency by increasing the magnetic field is possible to some extent, but constraint by the magnitude of the magnetic field.

To drive the discussion for completion a discussion on how scaling parameters will affect the thrust of the engine should be addressed. The thrust is the product of the mass flow rate and the exhaust velocity. Since the exhaust velocity remains constant, and the mas flow rate scales linearly with the characteristic length, the outcome is the thrust must scale linearly with characteristic length. In order for the thrust to scale linearly, the ratio of the thrust diameter and the gap thickness must change based on the following relations, here D is the thruster diameter and d is the gap witdth.

$$T\ \alpha\ \frac{D^2}{d^2}\ \ \alpha\ \ \frac{L^2}{d^2}\ \ \alpha\ L \qquad (8)$$

$$d\ \alpha\ \sqrt{L} \qquad (9)$$

In order to get the correct thrust after scaling, while maintaining the efficiency and specific impulse constants, the ratio of the gap width with the thrust diameter must scale inversely with the square root of the characteristic linear dimension. So, the gap width will get larger as the thruster is scaled down.

5. Mission Statement

A communication or Earth observation CubeSat is set to be launched into Low Earth Orbit and will be at an altitude between 300km to 600km. At these altitudes, the CubeSat will encounter drag and this would affect how long the CubeSat can stay in orbit. To counter this, the thrust provided by the propulsion system must be equal to or above the value of the drag. This will require a new set of calculations to determine the drag at these altitudes, the orbital velocity, the density of air at these altitudes and the power available to the thruster by the CubeSat.

The CubeSat derive its power though the use of solar panels. Currently, the solar panels used for CubeSats are quite advanced, and can track the Sun's position for optimal energy absorption. The power absorbed is determined by the area of the solar panel. The greater the area, the greater the power absorbed as there is more surface area exposed to the Sun, increasing the chance of photon collection. The equation for power collected is the following:

$$E = A_P r R P \qquad (10)$$

Where E is the energy collected in kW, A_P is the surface area of the panel in m^2, r is the solar panel efficiency, R is the average solar radiation in kW/m^2 and P is the performance ratio.

At LEO, the CubeSat will be going at a certain orbital velocity that is dependent on its distance from Earth. The equation for the orbital velocity is the following:

$$v_o = \sqrt{\frac{GM_E}{R_E + H}} \qquad (11)$$

Where v_o is the orbital velocity in m/s, G and is the gravitational constant in $m^3 kg^{-1} s^{-2}$, M_E is the mass of the Earth in kg, R_E is the radius of the Earth in m and H is the elevation above Earth in m.

The drag the CubeSat encounters is determine by the following equation:

$$D = \frac{\rho v_o^2 A_R C_D}{2} \qquad (12)$$

Where D is the drag in N, ρ is the density of air in kg/m^3, v_o is the orbital velocity in m/s, A_F is the reference area in $m^2$and C_D is the drag coefficient.

For the spacecraft to remain in orbit, the thrust must be greater than or equal to the atmospheric drag at a certain elevation. The power required by the thruster to counteract the atmospheric drag is calculated using a modification of (1), Here, the thrust is replaced by the atmospheric drag.

$$P_R = \frac{D I_{Sp} g}{2\eta_T} \qquad (13)$$

6. Results and Analysis

Busek creates many variations of miniature thrusters and they have a range of RF Ion Thrusters suitable for Satellites. These comprise of the BIT-1, BIT-3 and BIT-7 systems. Values for all three systems are listed in the following table, table 1. The propellant used by all three systems is Xenon. Based on the values listed in the table1, the thruster performance parameters can be evaluated, the thrust, ion velocity, specific impulse, effective exhaust velocity, gap size and maximum thrust will be calculated.

Table 1 Busek Thruster parameters

Parameters	Thruster		
	BIT-1	BIT-3	BIT-7
Total Thruster Power P_I (W)	10	60	360
Ion Beam Current I_B (mA)	1.5	21	157
Screen Grid Voltage V_S (V)	2000	2000	2000
Mass Flow Rate $\dot{m}_p$ (mg/sec)	0.0049	0.042	0.29
Gap Size x_i (mm)	28.8	45.7	-
Radius of Grid r (mm)	5	15	35

Figure 1a illustrates the exhaust velocity versus propellant mass flow rates, and Figure 1b illustrates the engine specific impulse as a function of the propellant mass flow rate. Based on ion engine performance theory both exhaust velocities and specific impulses has an inverse linear relation with propellant mass flow rate and this is demonstrated clearly in figure 1a and 1b, note that the horizontal axis is plotted in logarithmic scale. It is evident that the specific impulse and the exhaust velocity share similar values however those values are scaled for the same corresponding mass flow rate of the propellant.

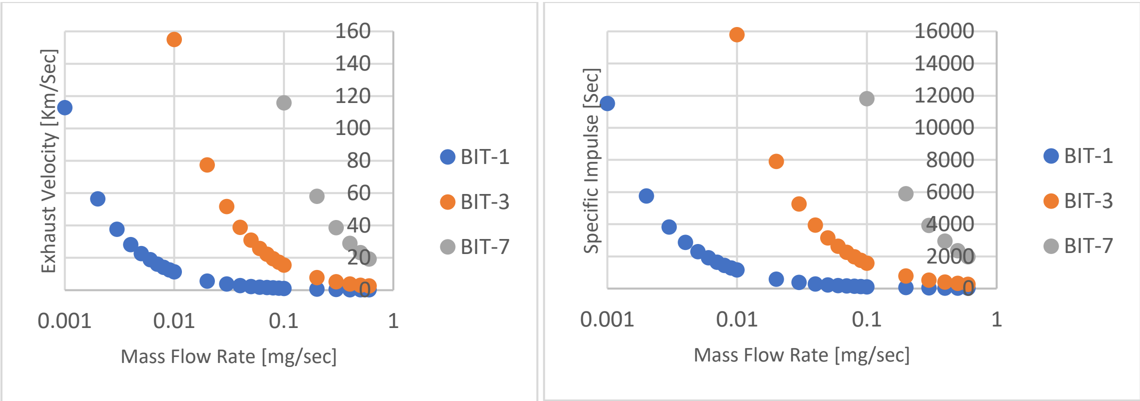

Figure 1: (a) Ion Engines exhaust velocities as a function of the mass flow rate (b) Ion Engines specific impulse as a function of the mass flow rate.

Figures 3a, 3b and 3c, indicates as the voltage is increased, the thrust, specific impulse and exhaust velocity also increase, therefore for a more effective thruster, a higher voltage is more desired. However, very high voltages can lead to the breakdown of the thruster as higher voltages require larger gap sizes, and this is considered impractical [3].

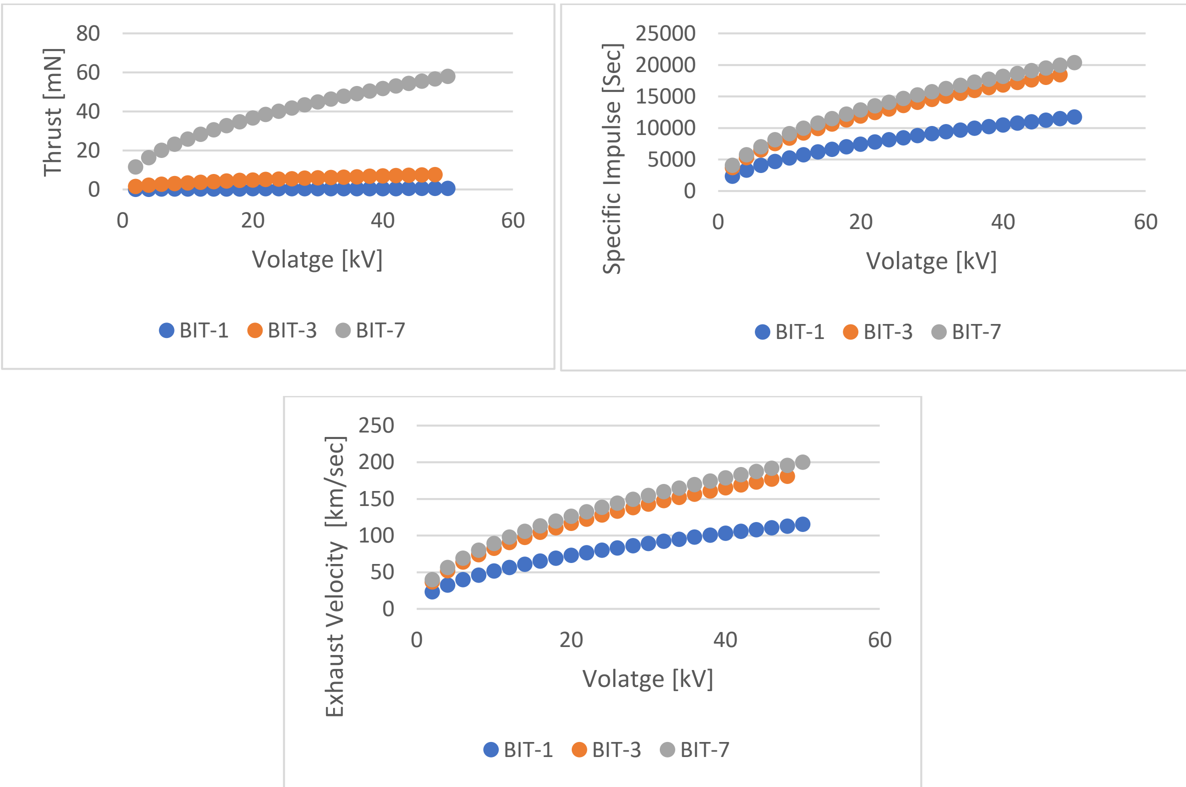

Figure 3 (a) Ion Engine thrust as a function of the screen voltage, (b) Ion Engine specific Impulse as a function of screen voltage, (c) Ion Engine Exhaust Velocity as a function of screen voltage

6.1 CubeSat Parameters

The average solar radiation in the upper regions of the atmosphere is approximately $1.368\ kW/m^2$ [4]. The region from which the CubeSat will be in orbit is within a range of 300-600 km. The efficiency of the solar cells in use will be considered at a 29.5%, taken from the EnduroSat website, a designer of CubeSat solar panels [5]. The performance ratio will be considered at 0.75 as the performance ratio is within a range of 0.5-0.9 [6]. CubeSats sizes can vary with the smallest being 1U, where the dimensions are 10cm x 10cm x 10cm. Combining multiple 1U systems create the larger systems i.e. 3U,6U,27U etc. Large CubeSat systems would require larger solar panels; therefore, the solar panel area will increase. For the first set of calculations, a 3U system will be used with dimension of 10cm x 30cm x 10cm, therefore the solar panel area will be 0.03m^2. From these values, the energy collected can be calculated. $E = A_p rRP = 0.03 \times 0.295 \times 1368 \times 0.75\ ,\ E = 9.0801\ W$, Table 2 illustrates the energy collected by the solar panel for CubeSats in the range 1U to 27Us.

Table 2 magnitude of energy collected by the solar panel for CubeSats in the range 1U to 27Us

$CubeSat$	$A\ (m^2)$	$E\ (W)$
1U	0.01	3.0267
2U	0.02	6.0534
3U	0.03	9.0801
6U	0.06	18.1602
12U	0.12	36.3204
18U	0.18	54.4806
24U	0.24	72.6408
27U	0.27	81.7209

a range of orbital velocities can be calculated based in equation 11, for range of different orbital altitudes of 200Km to 600km, Then the atmospheric drag is calculated based on equation 12. For calculation purposes, first use the corresponding values for a CubeSat at an elevation of 200 km from Earth with mean solar activity. The value of the drag coefficient of the 3U CubeSat, a rectangular rod, is taken to be 2.1 [7].

6.2 Atmospheric Drag Values

The magnitude of the drag value for mean solar activity is listed in table 3

Table 3 The magnitude of the drag value for mean solar activity

$H\ (km)$	$D\ (\mu N)$							
	1U	2U	3U	6U	12U	18U	24U	27U
200	244.95	489.90	734.85	1469.69	2939.39	4409.08	5878.78	6613.62
250	47.36	94.71	142.07	284.14	568.29	852.43	1136.58	1278.65
300	23.81	47.63	71.44	142.89	285.78	428.66	571.55	643.00
350	5.91	11.82	17.73	35.46	70.91	106.37	141.83	159.55
400	3.03	6.05	9.08	18.15	36.31	54.46	72.61	81.69
450	0.92	1.84	2.76	5.52	11.03	16.55	22.07	24.82
500	0.48	0.96	1.44	2.88	5.77	8.65	11.54	12.98
550	0.23	0.46	0.69	1.38	2.75	4.13	5.51	6.20
600	0.06	0.12	0.18	0.36	0.72	1.08	1.44	1.62

The main two requirement that must be met is the thrust must be greater or equal to the atmospheric drag to maintain orbit. Another requirement is for the CubeSat to provide enough power to counteract the atmospheric drag. For example, the power required for mean solar activity for BIT-1 thruster at low earth altitudes is illustrated in table 4

Table 4 Power required for BIT-1 thruster at Mean Solar Activity)

$H\ (km)$	$P_R\ (W)$							
	1U	2U	3U	6U	12U	18U	24U	27U
200	22.135	44.270	66.405	132.810	265.620	398.430	531.240	597.645
250	4.279	8.559	12.838	25.677	51.354	77.031	102.708	115.546
300	2.152	4.304	6.456	12.912	25.824	38.737	51.649	58.105
350	0.534	1.068	1.602	3.204	6.408	9.612	12.816	14.418
400	0.273	0.547	0.820	1.640	3.281	4.921	6.562	7.382
450	0.083	0.166	0.249	0.498	0.997	1.495	1.994	2.243
500	0.043	0.087	0.130	0.261	0.521	0.782	1.043	1.173
550	0.021	0.041	0.062	0.124	0.249	0.373	0.498	0.560
600	0.005	0.011	0.016	0.033	0.065	0.098	0.130	0.146

From Tables 2 to 4 and some of the relevant above equations, a table can be created to determine if the 3 systems meet the criteria for successful orbit. The three parameters for suitability are:

- The power required should be less than or equal to the thruster power,
- The power required should be less than or equal to the power gained from the solar panels
- The thruster power should be less than or equal to the power gained from the solar panels.

The following tables will detail whether the systems are suitable for a specific elevation. A check mark indicates that the thruster is suitable and a cross indicates that it is not suitable. From table 1, the total thruster powers (P_I) are 10 W, 60 W and 360 W for BIT-1, BIT-3 and BIT-7 respectively. The power gained from the solar panels are listed in table 6 and 7.

Table 5 Overall pass criteria (BIT-1 at Mean Solar Activity, 1U-6U)

$H\ (km)$	$P_R \leq P_I$				$P_R \leq E$				$P_I \leq E$				Overall			
	1U	2U	3U	6U	1U	2U	3U	6U	1U	2U	3U	6U	1U	2U	3U	6U
200	✗	✗	✗	✗	✗	✗	✗	✗	✗	✗	✗	✓	✗	✗	✗	✗
250	✓	✓	✗	✗	✗	✗	✗	✗	✗	✗	✗	✓	✗	✗	✗	✗
300	✓	✓	✓	✗	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✗
350	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓
400	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓
450	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓
500	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓
550	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓
600	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✓	✗	✗	✗	✓

Table 6 Overall pass criteria (BIT-1 at Mean Solar Activity, 12U-27U)

$H\ (km)$	$P_R \leq P_I$				$P_R \leq E$				$P_I \leq E$				Overall			
	12U	18U	24U	27U	12U	18U	24U	27U	12U	18U	24U	27U	12U	18U	24U	27U
200	✗	✗	✗	✗	✗	✗	✗	✗	✓	✓	✓	✓	✗	✗	✗	✗
250	✗	✗	✗	✗	✗	✗	✗	✗	✓	✓	✓	✓	✗	✗	✗	✗
300	✗	✗	✗	✗	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗	✗	✗
350	✓	✓	✗	✗	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✗	✗
400	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
450	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
500	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
550	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
600	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓	✓

7. Conclusions

The conclusion for this project is as follows:

- Performing trade-off studies to scale down the spacecraft and the propulsion system for application of atmospheric drag counter reacting based on pure spacecraft geometric scaling is not straight forward since the power available to power ratio is independent of spacecraft dimensions.
- The BIT-1 thruster is suitable for larger CubeSats at elevations beyond 300 km. Smaller CubeSats provide too little power and cannot provide the required thruster power.
- The BIT-3 thruster is only suited to large CubeSats (24U and upwards) as they can gain the solar power required to run the thruster. (based on similar analysis to BIT-1)
- The BIT-7 thruster is unsuitable for any CubeSat arrangement as its required thruster power level is beyond the means of the CubeSats in question. BIT-7 would be more suited to larger satellites. (based on similar analysis to BIT-1)
- The methodology implemented demonstrates the feasibility of selecting an off the shelf electric propulsion system, this has the benefits of reducing the time and cost of the design and development for communications and low earth orbit Satellites an CubeSats as well.

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