

Development of a Microthruster Module for Nanosatellite Applications

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Abstract: There is a growing interest for the use of nano-satellites within the aerospace community. However, the applicability of useful missions for nano-satellite is limited by the lack of propulsion capability. This study have identified the mission needs, and the requirements that must be met by the propulsion system in terms of mass, dimensions, power and performance in order to enable nano-satellite as an emerging and disruptive technology. Trade-off studies led to the selection of the Pulsed Plasma Thruster as the most-promising technology which can best meet the full range of requirements or at least best meet them over the widest range for the class of spacecraft and mission of interest to this study. This paper describes the conceptual PPT microthruster module design which has been developed to cover the mission scenario selected.

Nomenclature

$\Delta L/L_0$	=	inductance variation during the discharge to initial discharge circuit inductance ratio
$\Delta m/E$	=	mass ablated per unit of energy
ΔV	=	change in velocity
d_{el}	=	electrode length
d_{PROP}	=	propellant bar width
E/A	=	discharge energy over the propellant surface exposed to the discharge
η	=	efficiency
h	=	electrode spacing
$Ibit_{em}$	=	electromagnetic acceleration component
$Ibit/E$	=	impulse bit to discharge energy ratio
L'	=	inductance variation per unit length
L_{PROP}	=	propellant bar length
Ψ	=	current parameter
t	=	electrode thickness

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w = electrode width

I. Introduction

ONE current trend in space technology is towards miniaturisation of spacecraft, from mini to micro pushing towards nano-satellites. However, the extent to which nano-satellites will be disruptive will depend heavily on the availability of suitable miniaturised propulsion systems. The challenge is to develop a new generation of micro-propulsion system capable of satisfying the stringent requirements imposed by resource limitations on nano-satellites such as mass, volume and power available. Nano-satellites are defined in this study as spacecrafts with masses between 1 and 20 kg.

In this frame, the University of Southampton, Mars Space Ltd, Swedish Space Corporation, NanoSpace, and with the consultancy of Clyde Space are currently performing a study funded by ESA to identify and develop the most promising propulsion technology for nano-satellite applications.

In this paper the range of potential mission classes which could benefit from the use of a micro-thruster module and the trade-off leading to the selection of the most promising propulsion technology is reported and then the preliminary design of the micro-thruster module is presented.

II. Mission Requirements

In this section the mission analysis task which provides the input to the requirements definition for the parameters related to the mission design and the propulsion system to cover the identified mission needs will be presented. Then, the requirements that must be met by the propulsion system in terms of mass, dimensions, power and performance will be given.

The first phase of the study addressed the mission types that could benefit from the miniaturisation of satellites and the use of a micro-thruster module. Propulsion systems perform a variety of tasks for both earth and interplanetary missions which include orbit insertion manoeuvring, orbit maintenance or station keeping, and attitude control.

A review of the propulsive requirements for several missions¹⁻¹¹ revealed that a micro-propulsion concept for nano-satellites needs to address the following features.

1. The ability to deliver a total ΔV range from 1 to 100 m/s, although some missions might require a total ΔV greater than the upper limit.

2. The ability to achieve several thrust levels which are associated to different manoeuvres. A thrust range from 1 mN to 1000 mN is considered in order to satisfy all the mission requirements (general requirement for orbit insertion, orbit maintenance, and attitude control). A smallest range will be the subject of specific mission requirements.

3. A maximum micro-thruster module power consumption of 10 W.

4. A minimum impulse bit range from 0.1 mNs to 100 mNs.

5. A maximum micro-thruster module wet mass of 3 kg.

6. A micro-thruster module volume from 0.0008 m³ to 0.009734 m³, which represents a 0.093 m to 0.214 m side cube.

7. A micro-thruster subsystem lifetime targeted at 2 to 5 years.

8. A thruster configuration from 1 up to 12 thrusters depending on the application envisaged.

9. Modular and configurable design to suit different missions. Low recurring and re-configuration cost (module for multi mission approach is envisaged).

10. Reduced AIT effort by reconfigurable module.

A single propulsion system for all the tasks is highly desired based on mass and cost, but with such a wide range of propulsion needs is not considered an achievable goal by the authors. In this sense, a study case scenario has been selected and will be used to demonstrate the potential application that the micro-propulsion module could provide to a nano-satellite. The authors have proposed the use of a 6 thruster propulsion system that provides moderate ΔV range from 1 to 40 m/s for station keeping manoeuvres and for minimum impulse bit attitude control for a generic nano-satellite. An overall total impulse from 10 Ns to 800 Ns will be considered and it is assumed that each thruster can deliver the same total impulse to the satellite.

III. Determination of the Most Promising Propulsion Technology

This task aimed to evaluate the existing thruster technologies either available as “off-the-shelf” items or under significant development which might be able to meet the propulsion requirements previously identified. Thruster

technologies such as pulsed plasma thrusters, resistojets, cold gas thrusters, Hall Effect thrusters, gridded ion engines, colloids thrusters, field electric propulsion thrusters, hollow cathode thrusters, and chemical propulsion systems were critically reviewed and evaluated in view of potential nano-satellite applications¹². The literature review included the functional principles, performances, current state-of-the-art, scaling-down and miniaturisation constraints, along with mass, volume and power budgets.

After the review, a first trade-off using a traffic light approach was performed in order to identify the most promising technologies. The evaluation criteria were listed as propulsion and general parameters. The propulsion system parameters of significance to the overall design of the mission included the following: thrust range and minimum impulse bit; system specific impulse and ΔV capability; design life, reliability and Technology Readiness Level (TRL). The general parameters are system constraints related to the propulsion system parameters such as, dimensions, size and accommodation; complexity and spacecraft interface, modularity, AIT aspects; power available for the propulsion module; and cost.

The results of the trade-off can be summarised as follows:

1. In terms of meeting the requirements for the reference mission scenario Hall Effect thrusters, gridded ion engines, colloid thrusters, and field effect electric propulsion thrusters do not satisfy the mass budget requirement. These technologies require additional effort in subsystem miniaturization, such as feed system and power processing.

2. The lowest power level that a Hall Effect thruster, gridded ion engine, and hollow cathode thruster has been operated is somewhere above the maximum power available requirement. The power processing of Hall Effect thrusters, gridded ion engines, and hollow cathode thrusters is relatively complex since separate power supplies are needed for the various components.

3. Use of liquid metal propellant in field effect electric propulsion thrusters poses certain concerns with regards to spacecraft interaction.

4. The most promising propulsion technologies that were carried out to the second trade-off are pulsed plasma thrusters, resistojets, and cold gas thrusters. Pulsed plasma thrusters have their strongest point in the high specific impulse, low power requirement and in the capability of using solid propellant hence reducing the propellant volume and making it easier to extend the range of ΔV that can be achieved with the same thruster given a volume constraint. Resistojets have on their own the easiest electronics design and the fact that they have already being produced on MEMS scale hence allowing the possibility of staking more thrusters together. Cold gas thrusters are considered the simplest propulsion technology and are small devices with low total system mass, nonetheless the propellant and tank mass needs to be added to the total system.

A second trade-off was carried out through the use of weighting factors. Each criterion was given a weighting factor that is input into the technology evaluation approach. The following criteria were selected to evaluate the different micro-propulsion technologies: performance; modularity, scalability, and miniaturisation; complexity; cost; power; lifetime; and TRL.

The performance; modularity, scalability, and miniaturisation; and complexity were the most important evaluation criteria and were given the highest weighting factor. The cost, power and lifetime had the next highest weighting factor. Because the scope of this study is to design, manufacture and test a new micro-thruster module the TRL was the least significant evaluation criterion and thus had the lowest weighting factor.

Modularity of design is essential for an easily reconfiguration; the propulsion subsystem shall be built as a module that could be integrated and customized to accommodate different mission requirements. The technology selected shall be extremely scalable, and able to provide a wide range of performances for nano-satellite applications. Design flexibility without reducing capability is essential in the thruster design. The propulsion system shall be built as a stand-alone module that could be integrated and tested independently from the satellite. The system complexity envisaged shall allow a short development and manufacturing process.

From the second trade-off, the PPT technology scored the highest, followed by the cold gas and resistojet technologies. We conclude that according to the trade-off criteria the PPT is the technology which can best meet the full range of requirements or at least best meet them over the widest range for the class of spacecraft and mission of interest to this study. The PPT technology is very scalable and suitable considering the significant limitations imposed on mass, power and volume budgets for the propulsion subsystem. An appealing feature of PPT technology is the use of solid non-toxic propellant feed system which can eliminates safety, handling, and leakage issues common to on-board fluids and their systems. Another advantage of the PPT is its small impulse bit combined with high specific impulse, and the flexibility to operate in a wide range of power or thrust by varying pulse energy or pulse rate. PPTs exhibit a simple spacecraft/PPT interface, which is limited to physical mounting hardware and electrical connections for power, commands and telemetry. Moreover, only two power supplies are allocated to the

propulsion system. Hence PPTs will be considered as the technology to be developed, built and tested during the study.

IV. Thruster Design

In this section the thruster design will be presented. The thruster configuration, mainly determined by the propellant bar arrangement and dimensions will be described and justified, together with the discharge energy level selection.

Semi-empirical correlations using data relative to many different thrusters have been reported¹³ in order to characterise the behaviour of a PPT. Fig. 1 shows the semi-empirical correlations comparison between the breech and side-fed thruster geometries which link the specific impulse and the ratio discharge energy over the propellant surface exposed to the discharge, E/A . In the side-fed configuration there is a considerable increase in the area of propellant exposed to the electric discharge with respect to the breech-fed one. A larger area of propellant exposed to the discharge ensures, a higher impulse bit for a given discharge energy. Nevertheless, the greater amount of mass ablated reduces the specific impulse accordingly.

As seen from the graph, an energy increase will augment the specific impulse. On the other hand, larger discharge energies could imply an increase of the power system mass that could easily exceed the propellant mass saving due to the specific impulse enhancement. Therefore, the improvement of specific impulse by increasing the discharge energy has to be carefully considered for the PPT integration to nano-satellites.

Taking into account the data reported in¹⁴, on average the discharge energy to area ratio is of the order of 2 J/cm² for PPTs with discharge energies in the range of 1.7 to 24J. Moreover, a good indicator of the E/A ratio to be considered for nano-satellite applications is the μ PPT developed for Cubesats applications by Clyde Space Ltd, Mars Space Ltd and the University of Southampton¹⁵, which is a side-fed thruster, designed to consume ~ 0.3 W of power, with a discharge energy of 2.3 J, and an E/A ratio of 2.13 J/cm².

To achieve a ΔV of 40m/s on a 20kg satellite as specified by the mission requirements the design of the μ PPT thruster¹⁵ developed by Clyde Space Ltd, Mars Space Ltd and the University of Southampton can be scaled up increasing the E/A ratio to increase the specific impulse, thus for further calculations, an E/A equal to 2.5 J/cm² will be assumed as baseline.

At the selected E/A ratio, and the above semi-empirical correlations, the specific impulse could be estimated for the breech and side-fed configuration. From the trends reported in Fig. 1 a specific impulse of 548 s and 727 s is obtained for the breech and side-fed configuration respectively at E/A equal to 2.5 J/cm².

Evaluating the specific impulse trend as a function of the propellant mass at the previous specific impulses of 548 s and 727 s for the breech and side-fed configuration respectively, a propellant mass of 24.8 g and 18.7 g is obtained per thruster. It is assumed that in a 6 thruster configuration, as stated in the mission requirements section, all the thrusters will deliver the same total impulse, thus about 133.3Ns.

A margin equal to 20% has been selected for the propellant mass, thus a propellant mass of 29.8g and 22.4g will be considered for the breech and side-fed configuration, respectively. Once the propellant mass needed to accomplish the mission is known, the determination of the propellant bar dimensions is performed.

As previous studies have shown¹⁹⁻²¹ electrode geometry plays a significant role in thruster performance, especially it has been shown that the electrode spacing to width ratio h/w is a critical design criterion. Additionally, the electromagnetic acceleration component $Ibit_{em}$ and efficiency η are found¹⁴ to be proportional to the inductance variation per unit length L' and to the ratio between the inductance variation during the discharge and the initial discharge circuit inductance $\Delta L/L_0$, respectively as shown by Eq. 1 and Eq. 2.

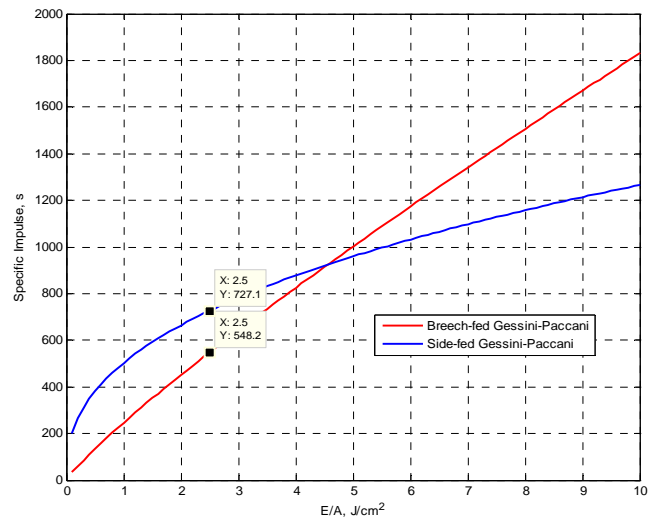


Figure 1. Semi-empirical correlations for breech and side-fed thruster geometries.

$$Ibit_{em} = \frac{1}{2}L\Psi \quad (1)$$

$$\eta \leq \frac{\Delta L}{L_0} \approx \frac{L'd_{el}}{L_0} \quad (2)$$

Where, Ψ is the current parameter and is calculated from Eq. 3.

$$\Psi = \int_0^{t_{fin}} i^2 dt \quad (3)$$

Hence to maximize the thruster performance, L_0 must be minimized and L' maximized. To minimize the initial inductance the thruster must be designed to allow the capacitors to be mounted as close as possible to the discharge chamber, whereas to maximize the value of L' the electrodes dimensions must be suitably selected.

The inductance (in μH) of a closed circuit of rectangular conductors of length d_{el} , cross section w by t , separated by h is defined¹⁴ by Eq. 4:

$$L = 0.4d_{el} \left(\ln \frac{h}{w+t} + \frac{3}{2} \frac{h}{d_{el}} + 0.22 \frac{w+t}{d_{el}} \right) \quad (4)$$

Hence, the inductance variation per unit length in $\mu\text{H}/\text{m}$ can be calculated as shown in Eq. 5:

$$L' = 0.6 + 0.4 \ln \frac{h}{w+t} \quad (5)$$

To maximize L' the electrode spacing must be increased and the electrode width and thickness decreased. However, if the value of h/w is too high, important non-uniformities may arise in the electromagnetic field, thus reducing the acceleration process efficiency²². Moreover, an increase of h will also increase the resistance of the plasma, hence reducing the current parameter Ψ and consequently reducing the electromagnetic impulse bit.

Past studies^{18, 19, 23-27} showed that the spacing between electrodes needs to be tuned in order to optimize the performance of the PPT. In work on a side fed PPTs¹⁸ it was shown that as the electrode spacing continues to increase beyond some optimal point, efficiency begins to decrease. Additionally, it was found that an excessive increase in electrode gap led to carbonisation of the Teflon surface.

Because further study of the aspect ratio is not within the scope of the project, it has been decided to choose an h/w ratio that is within the range of the data ($h/w = 0.5 - 3$) that was found in the literature^{14-16, 20, 21} for micro PPTs with similar characteristics to the intended design (especially taking into account the power consumption and the power capability limits of a nano-satellite), since the h/w ratio selected was 2.

For a breech-fed geometry, the electrode width is equal to the propellant bar width.

An additional assumption is needed for the side-fed configuration. Considering that the values of h in PPTs with discharge energy in the joule level are normally of the order of half to some centimetres (0.5cm – 2.54cm) as found in the literature reviewed^{14-16, 20, 21} we will chose an electrode

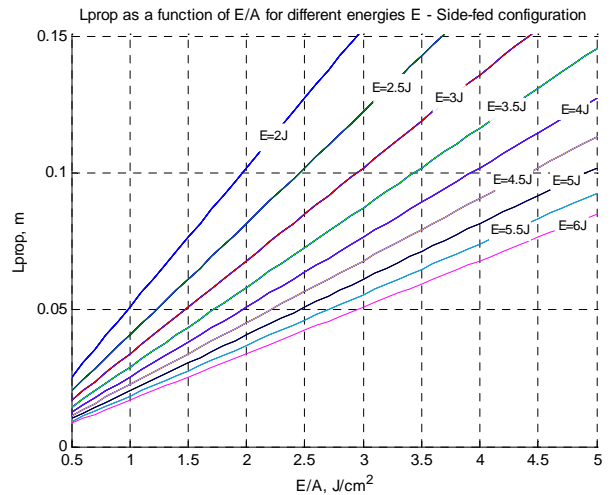


Figure 2. L_{PROP} trend for a side-fed configuration as a function of E/A for different discharge energies.

spacing h of 2cm, which past designs proved feasible, and consequently an electrode width of 1cm.

In Fig. 2 and Fig. 3 the trends of the L_{PROP} and d_{PROP} for different E/A ratios and discharge energies are reported for a side-fed configuration.

Regarding the power discharge, as previous studies^{20, 21, 28} have reported if the discharge energy is small propellant carbonisation would limit the operational lifetime of the PPT. Since, our baseline choice will be then to evaluate the propellant bar dimensions at 5 J discharge energy and energy over area ratio of 2.5 J/cm² for both thruster configurations. In the case of the breech-fed configuration, this will give us a length of the propellant bars of 6.8 cm and a propellant bar width of 1cm, hence the electrode spacing will be equal to 2 cm (an h/w ratio of 2 was assumed). In the side-fed configuration, the length of the propellant bars will be 5.1 cm on each side of the thruster, and a propellant width of 0.5 cm.

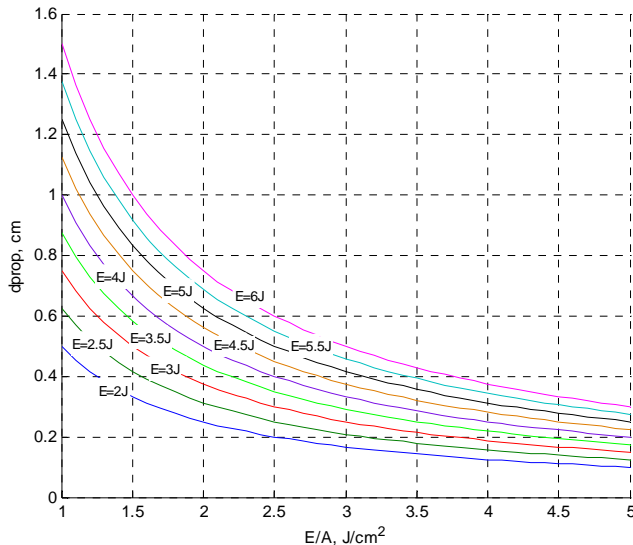


Figure 3. d_{PROP} trend for a side-fed configuration as a function of E/A for different discharge energies.

V. Theoretical Performance Analysis

The performance that may be expected from the thruster will be analysed assuming two different values of the impulse bit to discharge energy ratio (I_{bit}/E), an optimal and a non-optimal case.

For the side-fed configuration, the impulse bit to discharge energy ratio achieved by the μ PPT previously designed¹⁵ by Clyde Space Ltd, Mars Space Ltd and the University of Southampton will be considered for the non-optimal case, thus a value of 17 μ Ns/J. Since the μ PPT developed for Cubesats applications enable the consortium to gain expertise in the side-fed configuration, and the lessons learned could be implemented in the current design, an impulse bit to discharge energy ratio equals to 23 μ Ns/J will be considered for the optimal case. Additionally, a value of 23 μ Ns/J has been previously reported¹⁴ for side-fed PPTs.

Given that in a side-fed configuration there is a higher area of propellant exposed to the electric discharge with respect to the breech-fed one, and consequently a higher impulse bit for a given discharge energy, a value of 15 μ Ns/J will be considered for the optimal case and 10 μ Ns/J for the non-optimal one.

The total impulse delivered is calculated for the breech and side-fed configurations, and the trend as a function of the mass ablated per unit of energy, $\Delta m/E$, is shown in the Fig. 4 and Fig. 5, together with the total impulse required per thruster in order to accomplish the mission requirements.

With respect to the breech-fed configuration, for the optimal case (I_{bit}/E ratio equal to 15 μ Ns/J) the mission will be accomplished given that the mass ablated per unit of energy is below 3.348 μ g/J. In the same way, for the non-optimal case (I_{bit}/E ratio of 10 μ Ns/J), the mission requirements can be met if the mass per unit of energy does not exceed 2.232 μ g/J. In the case of the side-fed configuration, for the optimal case (I_{bit}/E ratio of 23 μ Ns/J) the mission will be accomplished given that the mass ablated per unit of energy is below 3.87 μ g/J. For the non-optimal case (I_{bit}/E ratio equal to 17 μ Ns/J), the mission requirements can

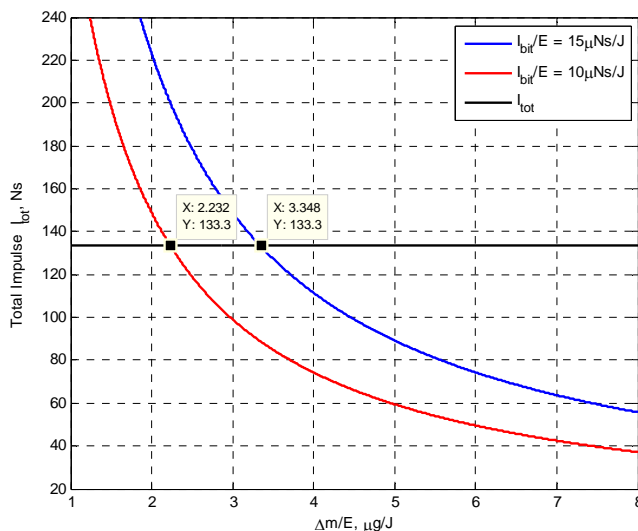


Figure 4. Total Impulse as a function of $\Delta m/E$ for an optimal and non-optimal I_{bit}/E ratio for a breech-fed thruster configuration.

be met if the mass per unit of energy does not exceed $2.861 \mu\text{g}/\text{J}$.

In terms of thruster efficiency the previous results translate for the breech-fed configuration into a thruster efficiency of 3.36% for the $15 \mu\text{Ns}/\text{J}$ optimal case, and an efficiency of 2.24% for the $10 \mu\text{Ns}/\text{J}$ non-optimal case. With respect to the side-fed configuration, the efficiency is 6.83% for the $23 \mu\text{Ns}/\text{J}$ optimal case, and 5.05% for the $17 \mu\text{Ns}/\text{J}$ non-optimal case.

For the breech-fed configuration, the required number of shots is 1,777,778 and 2,666,667 for the optimal and non-optimal impulse bit to discharge energy cases, respectively. With respect to the side-fed configuration, the required number of shots is 1,159,420 and 1,568,627 for the optimal and non-optimal impulse bit to discharge energy cases, respectively. It can be noticed that with the side-fed configuration the required number of shots in order to accomplish the mission requirements is about 600,000 and 1,000,000 shots lower than for the breech-fed optimal and non-optimal case, respectively.

As reported¹⁵ in the μPPT activity carried out by Clyde Space Ltd, Mars Space Ltd and the University of Southampton, the selection of the most appropriate capacitor technology for PPT applications is not a straightforward task. In particular the capacitors shall withstand the very short high current pulses typical of the PPTs, exhibit low equivalent series resistance and inductance, and to satisfy the stringent mass and volume requirements of nano-satellites. Thus, the difference in the number of shots between the side and breech-fed configuration could be of significance with respect to capacitor lifetime, and should be considered a key factor in the thruster configuration decision-making.

The following step in the design has been to address the carbonisation problems that both thruster configurations, breech and side-fed, have reported. Regarding breech-fed configurations, flight thrusters such as LES-6, LES-8/9 and SMS went through their flight qualification and acceptance programs, and carbonisation problems were not reported²⁹⁻³¹. The Tokyo Metropolitan Institute of Technology evaluated²⁰ the effect of the discharge energy with the breech-fed TMIT-5 thruster at 2.4 J operation, although the whole propellant surface was successfully ablated in the first 10,000 shots, non uniform ablation occurred after 100,000 shots and carbonisation was observed at the edge of the sublimation area. By increasing the discharge energy from 2.4 J to 3.6 J, the carbonisation did not occur after 100,000 shots firing. Another breech-fed configuration is the μPPT developed²¹ by the Austrian Research Centers (ARC) with a propellant area of 1cm^2 , the μPPT has been analysed and tested at discharge energies of 8J, 4.7J and 2J. During the study, the test at 2J was aborted after about 2,000 shots due to propellant carbonisation.

With respect to side-fed configurations, carbonisation of some components has been identified by different authors^{16, 23-26, 32}, as well as, during the thruster characterisation¹⁵ of the μPPT for Cubesats applications developed by Clyde Space Ltd, Mars Space Ltd and the University of Southampton. Nevertheless, as demonstrated by the Research Institute of Applied Mechanics and Electrodynamics of the Moscow Aviation Institute (RIAME MAI)²³⁻²⁶, the optimal correlation (found by testing different configurations) between discharge circuit parameters and discharge channel dimensions ensures the performances and that the energy coupling is efficient, thus preventing carbon deposition on the surfaces of the propellant bars.

Since, it has been decided to proceed with the side-fed thruster configuration due to the higher performance exhibited by this geometry, the lower number of pulses required to accomplish the mission requirements, and that some researchers claim to have solved the carbonisation problems. In a further step, during the testing phase, the test procedure will be defined in order to understand the operating limits of the thruster design.

VI. Conclusion

In this study, a conceptual PPT microthruster module design consisting of 6 thrusters has been developed to provide moderate ΔV range from 1 to 40 m/s for station keeping manoeuvres and for minimum impulse bit attitude

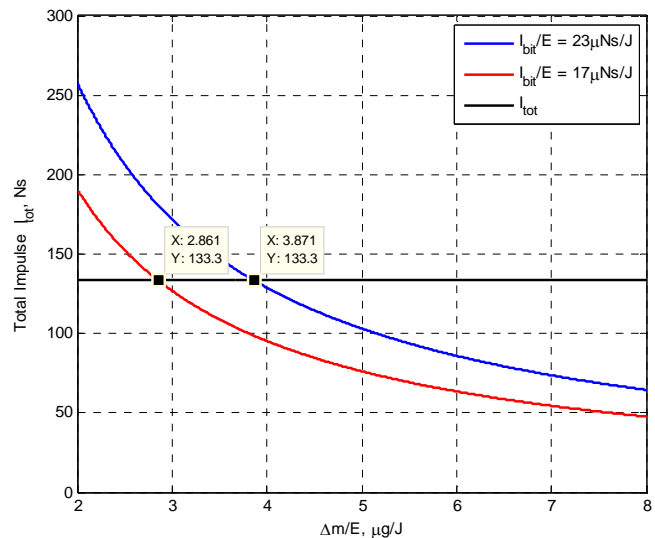


Figure 5. Total Impulse as a function of $\Delta m/E$ for an optimal and non-optimal I_{bit}/E ratio for a side-fed thruster configuration.

control for a generic nano-satellite. An overall total impulse from 10 Ns to 800 Ns has been considered and it is assumed that each thruster can deliver the same total impulse to the satellite.

A particular attention has been paid to the E/A ratio, the choice of this value based on the comparison with other low energy PPT designs and previous μ PPT developed for Cubesats applications by Clyde Space Ltd, Mars Space Ltd and the University of Southampton, shall ensure the specific impulse necessary to meet the mission requirements. An analysis of the expected thruster performances has been carried out, which suggests that the PPT would be able to provide the required total impulse even in a non-optimal case scenario.

Future work will consist of manufacturing and testing the PPT microthruster development model, and obtaining the actual PPT microthruster module characteristics (mass, power, and volume) will be obtained, as well as, the performance in terms of impulse bit and specific impulse.

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