Design of a CubeSat Propulsion System using a Cylindrical Hall Thruster

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Abstract: Electric Propulsion (EP) systems for CubeSats are one of the most challenging technological issues in the small satellite community and reliable solutions are still scarce. Many companies and universities have been devoting considerable effort to developing units capable of high performance active thrusting and enabling, with this, orbital maneuvering and interplanetary missions with very low power satellites. Considering the proven high performance of the Cylindrical Hall Thruster (CHT) at low power levels and its potentially longer lifetime, the present work has the goal to present the design of a CubeSat propulsion unit using the CHT, which could turn out to be a commercially viable product.

Nomenclature

f_s	=	safety factor
r	=	tank radius
Lc	=	length of cylindrical section
P	=	tank pressure
M_t	=	mass of propellant tank
σ_Y	=	yield strength of propellant tank material

I. Introduction

The prospects of using the CubeSat paradigm to perform complex missions, like deep space exploration or formation flying, are encouraging. Unlike traditional programs, missions adopting this standard usually present inherent low costs, fast project life cycles and have vast launch opportunities.

In order to execute any of these non-usual missions, however, it is necessary to employ propulsion subsystems capable of providing the necessary ΔV to the spacecraft, which in general can range from less than

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100 m/s, for phasing maneuvers, to more than 3 km/s for interplanetary flight [1], using very low power levels.
Currently, there exist several commercial systems to perform low-ΔV maneuvers with CubeSats, such as miniaturized Pulsed Plasma Thrusters (PPTs) [2], chemical and cold gas thrusters [3]. These modules, that presents low specific impulse levels, are usually employed to perform drag compensation, de-orbiting maneuvers and even orbital phasing.
In order to accomplish high-ΔV missions, however, it is necessary to use alternative methods. The two most developed solutions are high specific propulsion systems and solar sails [1]. Despite the sails offering an interesting option, their long maneuver time (up to 5 years) restricts their application.
Universities and companies have been devoting considerable effort to developing several types of miniaturized high specific impulse propulsion systems. The most developed are the miniaturized FEEP (~800 sec) [4] and ion thrusters (~2500 sec) [5]. Despite these good characteristics, these systems present low thrust levels, which can turn out to generate high gravitational losses. A thruster type that presents an interesting balance of characteristics is the Hall-effect thruster, and still very little effort has been spent developing these thrusters for application to CubeSats.
Between the many types of Hall thrusters, one of the most appropriate for miniaturization is the Cylindrical Hall Thruster (CHT) [6] developed originally at the Princeton Plasma Physics Laboratory (PPPL) in the early 2000s. Considering the proven high performance of the CHT at low power levels and its potentially longer lifetime, the present work has the goal to present the design of a CubeSat propulsion unit using the CHT, which could turn out to be a commercially viable product. This subsystem aims to allow 6U CubeSats to execute 3.5 km/s-class maneuvers, such as the GEO-Moon orbital transfer and escape trajectories. The 6U CubeSat is assumed to have an initial wet mass of 8 kg.

II. Propulsion System Architecture

This work aims to address a feasible design of a CubeSat propulsion system that employs a miniaturized Cylindrical Hall Thruster to perform low-thrust orbital maneuvers. This propulsion system consists of a module with volume of approximately 3U [7] that is designed to be used on 6U CubeSat missions.
The propulsion system assembly consists of the propellant (Xenon) tank, the Flow Controller Unit (FCU), the Power Processing Unit (PPU), the Hollow Cathode (HC) and the Cylindrical Hall Thruster (CHT). The architecture of the subsystem is depicted in Figure 1.

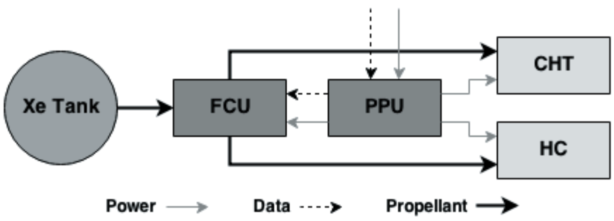

Figure 1. *Block diagram of the propulsion system.*

The propellant tank is designed to provide the necessary amount of gas for the specific mission to both the thruster and its cathode. The mechanical design of the tank suffers from a critical constraint in the case of this work due to the limited volume of CubeSat units.
The FCU controls the propellant flow rate to the CHT and HC and is usually composed by feed lines, valves and an electronic control system that receives the commands from the PPU to regulate the mass flow rate to the propulsion components.
The PPU is a critical component in all electric propulsion systems. Its main purpose is to treat the power provided by the spacecraft bus to the adequate voltage and current levels used by the thruster, cathode, and

all the other components of the propulsion system. The PPU also contains a digital interface to perform communication with the on-board data handing (OBDH) system of the spacecraft to, with this, control the operation of the propulsion system. There are two main possibilities in the architecture of the system, regarding the PPU: to use a single PPU for all systems or to employ dedicated units for the thruster and for the cathode [8]. Although using multiple PPUs greatly adds to the reliability of the system, it is chosen to employ the single configuration due to the harsh volume and mass constraints of the CubeSat standard.
In the next sections, a detailed description of each component of the propulsion system is given, showing the methodology and aspects considered in their design.

A. Cylindrical Hall Thruster

The Cylindrical Hall Thruster (CHT) was originally conceived with the goal of enabling the efficient downsizing of Hall thrusters without creating highly non-optimal magnetic fields [9]. Poor field topologies can strongly degrade the thruster efficiency, causing intensive ion losses, heating and erosion, thus turning the miniaturization of coaxial Hall Effect Thrusters (HET) impractical.
An End-HET, which is also known as a Kaufman ion source, is an alternative to the traditional HETs that tries to diminish the problems with miniaturization. It consists of a cylindrical discharge cavity with metallic walls. Tests of End-Hall thrusters showed major disadvantages and low efficiency mainly due to high plume divergence and hot electron losses, not cooled by secondary emission of traditional ceramic walls [9].
CHTs are a direct evolution of the End-HET, however with the main difference of the employment of ceramic walls, characteristic of traditional HETs, thus diminishing its major loss effects and increasing the radial magnetic field component, diminishing electron diffusion. Uniting the advantages of both the coaxial and End-HETs, the CHTs allows scaling down while maintaining acceptable performance levels even in power regimes below 0.1kW, as already demonstrated [10].
The design of a CHT, as shown in Figure 2, consists of a cylindrical ceramic channel with a ring anode that usually can also perform the task of distributing the propellant gas. The annular part in the center of the channel is usually either much shorter than the length of the channel or absent. Typically, the annular part is kept to enhance ionization, but for very low power applications its complete removing can be advantageous [6], as the so-called Fully Cylindrical Hall Thruster (FCHT) facilitates scaling without diminishing greatly the performance.

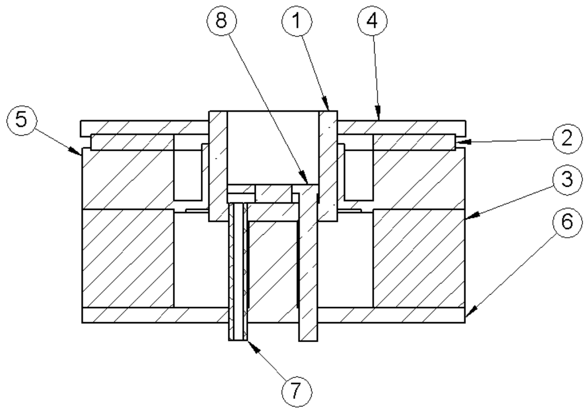

Figure 2. *Diagram of CHT, showing the ceramic channel (1), the permanent magnets (2, 3), the magnetic circuit (4, 5, 6) and the gas distributor (7).*

The magnetic field of the CHT has a cusp or mirror profile. Although the non-magnetized ion acceleration, in the CHT, occurs in a very similar manner to the traditional HETs, the electron trapping presents many different phenomena that enhance the effectiveness of the electromagnetic trapping and thus of the thruster.

Compared to the End-HET, the CHT has a stronger radial magnetic field component that greatly diminishes the electron diffusivity toward the anode. Moreover, because the field also has a strong gradient toward the anode, the so-called hybrid magnetoelectrostatic trap is formed [11], *e. g.*, electrons are trapped axially by the mirror-like effect, in the direction of the anode, and by the plume potential drop in the direction of the cathode.
Two methods to create the magnetic field are usually employed in this kind of thruster: electromagnetic coils and permanent magnets. The first one permits more controllability of the magnetic field intensity and topology but requires power and increases system complexity. The second option requires no power, decreases the thruster complexity and size, but removes the capability of controlling the magnetic field. Considering both options, usually permanent magnets are employed in the case of very low power and size applications [6], and electromagnetic coils are used in the case of larger ones.
Another important aspect is that CHT's lower surface-to-volume ratio, compared to traditional HETs, makes them less prone to channel erosion, due to ion sputtering, potentially increasing mission lifetime.
So far, CHTs have been tested at power levels from about 50 to 600W, which is still high a consumption range for CubeSat subsystems, thus making their application in low-cost microsatellite missions impractical. Nevertheless, the main characteristics of the CHT make it well suited for scaling down, and the next section will explore the possibility of scaling the CHT down to levels acceptable for CubeSat missions, 10-50 W, and propose a new miniaturized design that could be developed for this application.

1. Scaling

When designing new thrusters usually there are two basic approaches that can be adopted: the introduction of a new concept or the scaling of a known thruster based on existing experimental data [12]. The second approach usually is employed when it is necessary to preserve simplicity in the project lifecycle and to avoid extensive development efforts. In this direction, the present work aims to use the data available from the CHT to scale it to be more suitable to CubeSat mission applications. The method is mainly based on the physical phenomena occurring in the thruster and on empirical performance trends.

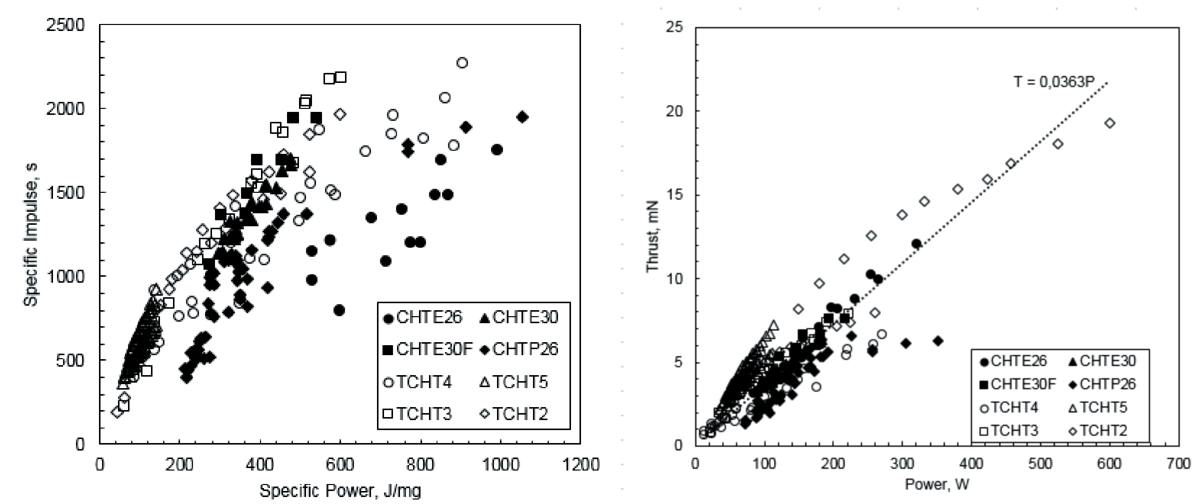

Figure 3. *Experimental data from several different CHTs. The graphic on the right shows the thrust-to-power relation and the graphic on the left shows the specific impulse to specific power (power per unit mass flow rate) relation.*

Figure 3 shows the data collected from several different CHT experimental campaigns [13] [14] [15] [16] [17] [18] [19] [20]. It is possible to notice that the plots reveal some underlying trends of these parameters, possibly aiding in the scaling process and allowing one to compare the parameters obtained from traditional approaches with real data, permitting with this the evaluation of the scaling of a virtual thruster to determine

whether it was an optimistic or pessimistic estimation.
The need for scaling an electric thruster is closely related to the will of maintaining performance levels when operational characteristics, such as discharge power or mass flow rate, are changed. Thus, it is necessary to assess how the main thruster characteristics can be scaled in order to maintain important parameters virtually constant, at different power levels. A broad review of the existing Hall thruster scaling methods is presented in [21].
Due to the lack of available theoretical models in the literature about the operation of the CHT we chose to adopt a simplified approach for its scaling, in this case, the so-called “photographic” method [22] and compare it to the experimental data collected and shown in the plots in Figure 3. As commonly chosen, this analysis will be undertaken considering the specific impulse and thrust efficiency as the parameters that should remain theoretically constant during scaling.
The scaling occurs by first selecting a factor k that when applied to the original parameters gives the characteristics of the new thruster. The application of the scaling factor depends on the parameter to which it is applied. Table 1 shows the method of application to each major design parameter as widely demonstrated in the literature [12]. By definition, the “scaling relation” is defined as the ratio between the new and the original parameter values.
The selection of k depends on the goal of the specific scaling process. In the case of this work the objective is to determinate a design that operates with a specific low power level, for its usage in micro-satellite missions, so k is defined as the ratio of the target and the original power.

Parameter	Symbol	Scaling relation
Characteristic length	L	k
Density	n_0, n_n, n_e	k
Propellant mass flow rate	$\dot{m}$	k
Pressure	p	$1/k$
Current density	J	$1/k$
Electron current	I_e	k
Ion current density	J_i	$1/k$
Ion current	I_i	k
Power	P	k
Magnetic field	B	$1/k$

Table 1. *Scaling relation for each major parameter of the thruster design. “Photographic” method.*

The thruster chosen to be used as configuration for the scaling is the 2.6 cm diameter FCHT, with permanent magnets, designed at PPPL and tested at NASA’s Marshall Space Flight Center [6]. This thruster was designed to operate in the power range of 100-200 W, where it achieved efficiency levels up to 19% and thrust levels up to 7 mN. The thruster has 3.5 cm diameter and 5.5 cm length, weighting approximately 350 g.
As Conversano and Wirz [5] showed, an ion thruster working below a 30 W power level fits the application in a CubeSat mission with special permissions. The scaling will target a CHT that could operate in the 10-50 W (30 W nominal) power range. Table 2 shows the scaling from the 2.6 cm thruster to the target one. If the obtained parameters are compared to the plots in Figure 3, it is possible to observe that the scaling was reasonably conservative and showed a good correlation with the empirical observations. In particular, the thrust could even be higher (1.8 mN at 50 W), while the specific impulse could actually be lower, as suggested by the large data scatter in the left-hand plot in Figure 3.

Parameter	2.6 cm FCHT	CubeSat CHT
Power, W	200	50
Thrust, mN	6	1.5
Specific Impulse, sec	1500	1500
Thrust Efficiency	0.19	0.19
Channel Diameter, cm	2.6	0.65
Mass Flow Rate, mg/s	0.5	0.125
Maximum Magnetic Field, T	0.1	0.4

Table 2. *2.6 cm FCHT and the new CHT characteristics.*

2. Design considerations

The general design of the thruster being considered is shown in Figure 2: the basic components of a CHT are the cylindrical discharge chamber, which in this case does not contain any annular part, a ring anode, which can serve also as gas distributor, the permanent magnets and the magnetic circuit, usually made from low-carbon steel.

The discharge chamber of general CHTs consists of a ceramic channel, typically made of Boron Nitride in order to diminish material sputtering and secondary emissions [23]. This ceramic insulation also serves to decrease heat transfer to the peripheral components, as the magnets, magnetic circuit and close electronic systems. Some models of CHT contain a short annular ceramic part to increase propellant ionization but, in this case, it was chosen to use the so-called Fully Cylindrical configuration, without annular part, to help increasing simplicity in the miniaturization process without major performance losses [24].

The choice to use permanent magnets instead of coils, to generate the needed magnetic field, is due to the possibility of increasing compactness, simplicity and decreasing consumed power and weight [6]. These advantages fit well in the design of miniaturized thrusters. The chosen magnet type is the Samarium-Cobalt (SmCo), due to its good levels of magnetic remanence, when compared to Ferrite and Alnico, and at the same time high Curie temperature when compared to the Neodymium magnets.

3. Cathode

The electron source is one of the most critical components in typical electric propulsion systems. One of the most employed technologies is the Hollow Cathode (HC) for its high reliability, technology maturity, and current generation capability [23].

Permanent magnet CHTs present usual anode voltage profiles in the range of 200-600 V [6]. It is expected therefore that the present thruster will operate with current levels in the order of 100-300 mA.

Operational low-current hollow cathodes are very scarce nowadays and most of the experimental campaigns of miniaturized thrusters employ over-sized cathodes, which, in general, greatly degrades the overall efficiency of the system [23]. The most successful miniaturized cathodes are presented by Conversano and Wirz [5], with a low-current hollow cathode, developed by JPL, which is used in the MiXI thruster assembly, and by Busek, that designed the BHC-50 that is used as a neutralizer in the BIT-1 assembly. The present work considers the average characteristics of these cathodes just for the sake of this system study.

B. Power Subsystem

Each component of the propulsion system needs a distinct type of supply to work properly: the PPU main purpose is to convert the power provided by the spacecraft bus to the specific needs of each component. As commonly designed [25] the PPU specified for this system is composed by four regulated power supplies, discharge supply, cathode heater supply, cathode keeper supply, FCU supply, and a digital system to perform the interface with the spacecraft OBDH system and perform control of all power supplies and the FCU, as shown in Figure 4.

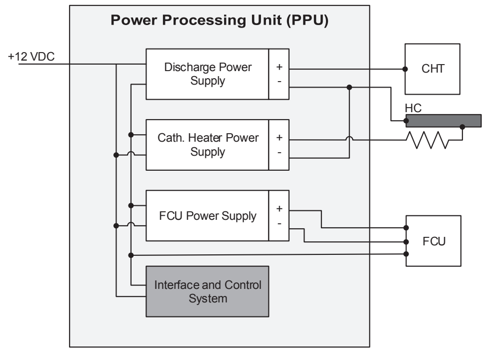

Figure 4. *Block diagram of the Power Processing Unit (PPU).*

All power supplies consist of commercial DC-DC converters [26] working under a regulated bus voltage input of +12 V, as normally provided by CubeSat electrical power systems (EPS). The discharge supply consists of a 60 W converter with variable output voltage in the range of 0-600 V, working mainly with current levels in the order of 100-300 mA. The cathode heater supply is a 10 W converter that is able to provide 20-40 V at higher currents, 400-500 mA.

By analyzing a theoretical CubeSat mission, which employs the designed propulsion system, it is possible to verify the necessary power generation and storage capabilities of the complete spacecraft. The preliminary power budget of the system is shown in Table 3.

In the estimation, the considered Onboard Data Handing (OBDH) system was the ISIS OBC [27], which presents an average power consumption of 400 mW. The telemetry, telecommunication & command (TT&C) system was assumed the Clyde Space UTRX Half Duplex UHF Transceiver [28], which presents an average receiving power of 250 mW and transmitting power in the range of 4-10 W. The attitude determination and control system (ADCS) was assumed the MAI-400 [29], produced by the company Maryland Aerospace Inc., presenting an average consumption of 3 W when using the reaction wheels actuators and the rest of components, 1 W when using just sensors and the ADCS computer and 200 mW in idle mode. Lastly, the electrical power system (EPS) of the spacecraft was assumed the 3G Flex EPS [30], provided again by Clyde Space together with stand-alone batteries [31] by the same company. It was assumed that the spacecraft has onboard a generic payload that consumes 3 W when activated.

Subsystem	Component	Thrusting mode, W	Umbra mode, W	Science mode, W	Safe mode, W
Propulsion		11.60-51.40	8.60	0.00-8.60	0.00
	Main discharge	10-50 (1.0)	10-50 (0.0)	10-50 (0.0)	10-50 (0.0)
	Cathode heater	8.00 (0.0)	8.00 (1.0)	8.00 (0-1.0)	8.00 (0.0)
	FCU	1.00 (1.0)	1.00 (0.0)	1.00 (0.0)	1.00 (0.0)
	ICS	0.60 (1.0)	0.60 (1.0)	0.60 (1.0)	0.60 (0.0)
OBDH		0.60	0.60	0.60	0.60
	Onboard computer	0.60 (1.0)	0.60 (1.0)	0.60 (1.0)	0.60 (1.0)
TT&C		0.95	2.35	2.35	0.25
	UHF TX	7.00 (0.1)	7.00 (0.3)	7.00 (0.3)	7.00 (0.0)
	UHF RX	0.25 (1.0)	0.25 (1.0)	0.25 (1.0)	0.25 (1.0)
ADCS		4.00	0.50	0.50	0.50
	Sensors & computer	1.00 (1.0)	1.00 (1.0)	1.00 (1.0)	1.00 (0.1)
	Reaction wheels	3.00 (1.0)	3.00 (0.0)	3.00 (0.8)	3.00 (0.0)
Payload		0.00	0.00	3.00	0.00
	Sensor	3.00 (0.0)	3.00 (0.0)	3.00 (1.0)	3.00 (0.0)
Total		17.15-57.15	12.05	6.45-15.05	1.35

Table 3. *Power budget of the spacecraft, showing duty cycles between parentheses.*

If it is considered that while in daylight, the spacecraft operates mainly in “thrusting mode”, and while in shadow it operates mainly in “umbra mode”, it is possible to estimate the maximum needed power input from the solar panels. In order to run the needed subsystems and recharge the batteries in daylight phase the solar panels need to generate about 69.2 W. Depending on the orbit profile the power input level for the batteries can be decreased or even eliminated, for example in the case of a GEO to Moon transfer in a high-inclination orbit.
Considering for the analysis GaAs solar cells, which have an area of 30.2 cm² and present an end of life (EOL) efficiency of approximately 27% [32], also considering the sun constant as approximately 1350 W/m², it is estimated that it is necessary to use at least 63 solar cells. The installation of this number of solar cells is only achievable with the use of deployable systems as shown in [1] and [5]. For this theoretical mission approximately 4 double folded panels with pointing capability would be needed.

C. Propellant Storage and Supply System

The xenon tank will have a cylindrical configuration with hemispherical ends. This component is the one that suffers the most with the volume restrictions of the CubeSat standard. It is important to note that in the 6U standard the maximum width is 10 cm, limiting the tank diameter to values slightly less than this. In order to ensure the symmetry in mass balance the tank will be positioned transversally, as shown in Figure 4. This position limits its length to 20 cm and consequently its volume to a maximum of 1300 cm³.
Considering the 3.5 km/s mission profile, that the spacecraft have initial wet mass of 8 kg and that the thruster presents the performance characteristics shown in Table 1, it is estimated that the needed initial propellant for the mission is 1.7 kg. Assuming that the tank volume is 1250 cm³, in order to leave room for connections, and that the tank temperature will not exceed 50 ºC during the mission, the tank pressure is estimated to be approximately 280 bar.
The CubeSat standard, however, requires that the spacecraft should not have pressure vessels with more than one atmosphere [7]. Thus, for the application of the proposed tank, the launch provider should grant the mission a special exception to board the high-pressure vessel. An interesting and straightforward solution to this question could be the operation of the CHT with the solid-storable propellant iodine [33], nevertheless assessment of its performance with this type of propellant should still be made before it is considered for application.
The mass M_t of the tank can be simply estimated using the pressure vessel equation applied for a cylindrical tank with hemispherical ends,

$$M_t = 2\pi \frac{\rho f_s}{\sigma_Y} P(r + L_c)\, r^2 \tag{1}$$

Where ρ and σ_Y are the density and the yield strength of the material respectively, P the internal pressure of the tank, r and L_c are its radius and length of the cylindrical part respectively and f_s is the safety factor. The CubeSat standard requires safety factors, for pressure vessels, of no less than 4 while AISI/AIAA S-080 and S-81 standards require safety factors of no less than 1.5. If it is considered that the tank is made of titanium-lined carbon fiber overwrapped composite ($\rho \approx 2500$ kg/m³, $\sigma_Y \approx 650$ MPa), it would weight nearly 1 kg for the current CubeSat standard and 360 g for the AISI/AIAA standard. For the present discussion, the lighter tank version will be considered, once that exceptions have to be made anyway because of the needed pressure level.
The regulating valves, feed lines, electronic interface and control system compose the Flow Control Unit (FCU), responsible for propellant distribution. The μFCU developed by the company AST GmbH [34] presents an interesting solution that approximately matches the requirements of the present propulsion system. This module presents a 54 x 46 x 25 mm geometric envelope, has a total weight of about 62 g, and is capable of providing between 0.01 and 100 sccm of Xenon. Because this system is designed to operate at

relative low pressure, before the connection with the µFCU the propellant gas must suffer a previous pressure reduction from the storage pressure to approximately 2 bar. Once that the FCU design is out of the scope of this work, it will be assumed that the discussed propulsion system employs the µFCU.

D. Structure and Mass Budget

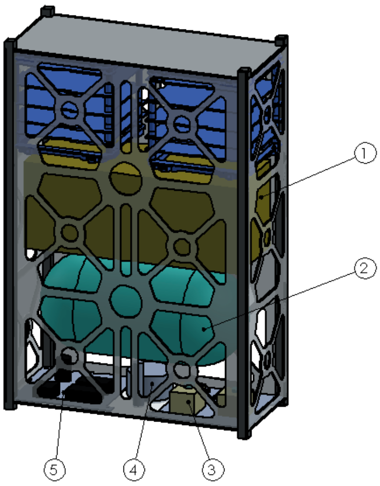

Figure 5. *Preliminary configuration of the 6U CubeSat with the discussed PS, showing general subsystems of the spacecraft (1), Xenon tank (2), FCU (3), CHT and HC (4), and PPU (5).*

The structure of the propulsion system, as the rest of the spacecraft, will follow the CubeSat standard [7], using Aluminum 7075 for both the internal structure and the interface rails. The rails' material, however, must be anodized to avoid spacecraft charging during its sliding in the deployment system. The construction is shown in Figure 5.

The mass of the complete spacecraft is designed to be no more than 8 kg, which is the general recommendation for a 6U. Table 4 shows the preliminary mass budget for the system.
For this preliminary estimation, a generic scientific payload was assumed that weights 500 g, which is an acceptable approximation when compared to real subsystems [1]. The CHT mass, approximated following the same approach presented previously, is assumed to be about 60 g. The PPU mass is estimated based on commercial power electronics characteristics, the DC-DC converter group is estimated to weight around 500 g [26] and the peripheral electronics, plus cables, to weight around 150 g [30].

Subsystem	Component	Mass, g
Propulsion		3072.00
	CHT	60.00
	Hollow cathode	100.00
	FCU	62.00
	PPU (with cables)	650.00
	Tank (with feed lines)	400.00
	Propellant (with 10% margin)	1800.00
OBDH		94.00
	Onboard computer	94.00
TT&C		190.00
	UHF Transceiver	90.00

	Antenna	100.00
EPS		426.00
	EPS Board	170.00
	Batteries	256.00
ADCS		694.00
	Module	694.00
Payload		500.00
	Sensor	500.00
Structure		2160.00
	Solar panels	1160.0
	Structure	1000.00
Total		7136.00

Table 4. *Mass budget of the spacecraft.*

The 864 g mass left over from the maximum design mass of 8 kg is considered as a margin to guarantee that the spacecraft would remain within the weight specifications even if margins were not considered or mass of components underestimated during this preliminary design approach.

III. Conclusions

In the first part of this work, a brief review was made of the propulsive technologies that are usually considered for CubeSat missions. Already in the second part, the description of the propulsion system design began, presenting first the scaling methodology for CHTs and sizing a thruster for application in a CubeSat mission. After, a brief discussion was made of the power components and requirements for the propulsion system and the spacecraft boarding it. Then, the necessary propellant storage system was calculated and the needed characteristics for the FCU were shown. Lastly, the main structural characteristics of the system were presented, together with the mass budget of the propulsion system and of a theoretical 6U CubeSat mission using it.

Despite being very preliminary, this work proved the feasibility of using propulsion systems employing CHTs in 6U CubeSat missions. The study considered a 3.5 km/s ΔV mission profile.

Future work will focus on detailing of design, simulation of subsystems, and addressing new solutions for critical issues. One of such first issues to be discussed is the elimination of high-pressure vessels with the employment of alternative solid-storable propellants, such as iodine.

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