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ELECTRIC PROPULSION OPTIONS FOR CUBESATS

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The CubeSat platform is presently well established as a standard building block for nano-satellites and presents a large potential for a number of applications, particularly in its 3U (or larger) version. The aim of this study is to investigate the possibilities offered by the implementation of Electric Propulsion (EP) on CubeSat platforms and to analyse the relative advantages of several current EP technologies with respect to specific constraints of such minimal spacecraft. We focus on devices requiring limited power and with small mass and size, yet performing at relatively high specific impulse, such as Hall Effect Thrusters, Field Emission Electric Propulsion, Pulsed Plasma Thrusters and resistojets. Performance analysis is performed first analytically, then by direct simulation of representative mission profiles on Alta's SATSLAB simulator, whereby the coupled effect of orbit, attitude and on-board energy management is accounted for. We show that performance of CubeSats is significantly enhanced by a properly chosen EP system, extending the operational envelope to include manoeuvres such as residual drag compensation, change of orbital altitude and plane inclination, and rephasing in constellations, resulting in additional mission profile flexibility. New mission classes, enabled by the unique combination of CubeSat's tiny mass and EP manoeuvring capability, are outlined and discussed.

I. INTRODUCTION

The current financial and institutional limitations on space activities worldwide are motivating both the miniaturization of space platforms and the standardization of space proven components. Possibly, the cheapest way to access space today is the CubeSat concept, introduced at the Stanford University[1] together with the California Polytechnic State University (Cal-Poly)[2]. Such a kind of platforms can be easily developed with off-the-shelf components and the availability of a standard interface (P-POD[3]) for the deploy system further reduces the overall costs of these objects as piggyback payloads on standard launches.

The CubeSat concept was born in the university environment and, during the past decade, it has been considered suitable only for educational purposes. However, the high standardization level reached and the increasing miniaturization of the various space components are moving CubeSats outside of universities into the real space business. Several works are addressing and discussing the use of these platforms[4,5]. Proposed applications span from Earth observation, to technological demonstrators, in-orbit testing of space components and in-situ atmospheric measurements[6].

CubeSat platforms are based on a standard building block (1U unit) of 100x100x100 mm and 1.3 kg of mass and two (2U) or three (3U) of these modular cubes can be combined reaching a maximum weight of 4 kg.

During the last decade, approximately a dozen of CubeSats were successfully launched and almost the same number failed during the launch or the early operative phases[7]. None of these CubeSats was actively propelled, although some of them mounting an electric thruster failed the launch (for instance ION[8]). Also attempts to fly tether equipped CubeSats (such as the DTUsat and the CUTE-1.7) must be mentioned in the history of launched CubeSats.

Such platforms can be equipped, for instance, with an optical payload for Earth observation and with some standard sensors for atmospheric measurement. Alternatively, these cheap satellites offer an easy way to test space components, thus also the technological demonstrator role of the CubeSat has to be considered. Moreover, during the development of a real mission, after the feasibility phase, the CubeSat platforms can easily fill the gap up to the development of the real satellite offering cheap and easy pathfinder opportunities to retrieve preliminary data[9].

Typically CubeSat operative orbits are not selectable for the specific mission purposes, but they can only be roughly defined according with the launch primary payload requirements. A significant improvement of performance and scientific return of these tiny satellites could derive from the possibility to actively acquire operative orbits and/or to maintain a specific attitude. Thus, also the possibility to include small thrusters on these platforms has become a viable study option[10].

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Nevertheless, due to the present limitations of size, mass and power, the inclusion of a propulsion subsystem is particularly challenging. These propulsive subsystems have to be light, requiring very limited power, with reduced size and enough flexible to be integrated with the other subsystems. Classical chemical thrusters have the obvious disadvantage of requiring a relatively large propellant mass and tank; moreover the usual propellants are quite hazardous materials, forbidden by CubeSat launch specifications[11], although also solid propellant thrusters have been developed in miniaturized versions for nano-satellite applications[12].

The concept developed in this study is to consider the advantages of a manoeuvrable CubeSat equipped with an electric thruster for a number of mission tasks. Section II focuses on the issues arising from the integration of an electric thruster with one of these small platforms for near Earth missions. To do so, a preliminary mission analysis is carried out, in whch natural orbit lifetime at different altitudes is compared with the orbital lifetime extension enabled by electric propulsion. In Sec. III a preliminary sizing of one of these platforms is addressed including a payload margin and an electric propulsive subsystem. Both the mass and the power required by each subsystem is determined. In Sec. IV an Earth observation mission of a CubeSat equipped with the Simplified FEEP thruster is considered as reference mission scenario to emphasize the importance of an accurate power management already in the early phases of the mission definition[13].

The analyses have been carried out using Alta's orbital propagator, SATSLab[14]. This software is designed to determine and estimate the tight connections between orbital dynamics and the on-board available energy stored in batteries, the power generated by solar arrays and the payload and subsystems energetic requirements[15].

Finally, in Sec. V of this work, orbit-to-orbit transfer capabilities and deorbiting time achievable by a CubeSat equipped with an electric thruster are assessed.

II. CUBESAT LIFE EXTENSION

Due to their typical low budget, CubeSats are usually launched by means of limited performance launchers and as piggyback payload without possibility to impose constraints on the final orbit characteristics. Accordingly, typical operative orbits of these small objects lie in Low Earth Orbit (LEO) or Sun-Synchronous Orbit (SSO), with an altitude range from few hundred kilometres up to one thousand. One of the main perturbing force acting at these altitudes is the atmospheric drag, whose dissipative effect tends to decrease the orbital energy causing reentry.

This effect can be compensated by operating the propulsion system, acting with a small force for extended times to counteract the orbital decay. Consequently, we study in this section the orbital life of a CubeSat in the LEO region and assess the cost of performing drag compensation maneuvers, in terms of mass and power required.

II.1 CubeSat Orbit Lifetime

The determination of orbital lifetime of objects in the near-Earth region could represent a difficult task to perform with standard direct method as these may be very expensive in terms of computation time. A valid alternative method to model the perturbation effects is represented by the Gauss form of the Lagrange Planetary Equations[16] (LPEs), modelling the time evolution of the classical orbital parameters (semi-major axis a, eccentricity e, inclination i, right ascension of the ascending node Ω, argument of pericenter ω and mean anomaly M) under the influence of non-conservative perturbations. The perturbation accounted for in this section is atmospheric drag, which mostly affects semi-major axis and eccentricity. Thus, the estimation of deorbiting time is carried out focusing on the time evolution of these elements. In particular, it is possible to obtain an expression for their instantaneous rate of change due to the atmospheric drag combining LPEs with the expression of spacecraft relative velocity and the rate of change of eccentric anomaly. A specific atmospheric density model is then required to carry out the integrations of the resulting equations. The Harris-Priester model[17], which relies on a number of tables listing density values obtained from observational data within a complete solar cycle, has been used to assess the orbital lifetime in three different scenarios: constant minimum, medium and maximum atmospheric density.

The three different CubeSat sizes are assumed to be affected in the same way by the atmospheric drag, as the area-to-mass ratio of the three platforms can be considered almost constant assuming that the larger face is aligned with the velocity direction.

The evaluation of expected orbit lifetime is given in Fig. 1, assuming a drag coefficient of 2.2, for the constant medium density model in a range of altitudes up to 500 km.

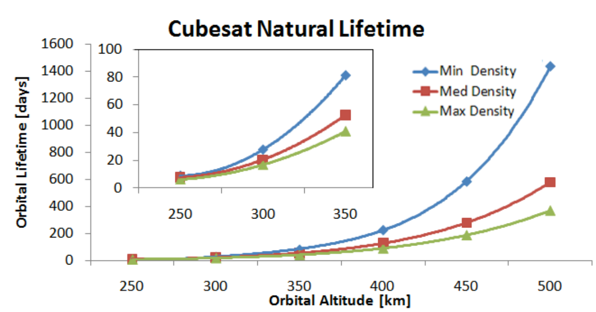

Fig. 1: CubeSat lifetime for a range of altitudes and three different atmospheric densities.

The upper altitude limit has been chosen to focus only on the more interesting LEO region as the exponential decrease of atmospheric density with altitude leads to a minimum lifetime of 4 years at 600 km and more than 100 years at 900 km. These durations are quite long, thus it is not required any additional effort to further increase the lifetime of an object as such altitudes. Thus, from Fig. 1, it is possible to identify a sort of maximum altitude range below which low thrust drag compensation maneuvers are beneficial.

II.2 Overview of suitable electric propulsion schemes

Several electric propulsion technologies are available as valuable options to provide CubeSat-like platforms with active maneuver capabilities. Electric thrusters usually provide low thrust levels, as the power required per unit thrust is relatively high. Nevertheless, the use of low thrust propulsion is very appealing for very small platforms, as the resulting thrust-to-mass ratio is high enough. Thus, a preliminary selection of suitable propulsion technologies focuses on small and light devices requiring limited power to work. Among the technologies coping with these requirements we have selected the resistojet[18], where a propellant stream is resistively heated and expanded through a nozzle, Field Emission Electric Propulsion (FEEP)[18], based on room temperature field ionization, and the Pulsed Plasma Thrusters (PPT)[18], where a solid propellant (typically Teflon) is ablated and accelerated by a pulsed electric discharge.

The above selection was done for the sake of comparison among very different concepts and is by no means exhaustive. Small Hall effect thrusters[18] (e.g. HT-100[19]) and downscaled ion thrusters[18] (e.g. MiXI[20]) can also be considered suitable for the sought application. However, such devices, given the present state-of-the-art, do not appear to fit into standard CubeSat platforms in terms of power and mass. In particular, the power electronics required for such thrusters are quite heavy and limits the real application of these devices to small platforms, like micro/mini satellites with slightly larger masses and powers. Furthermore, these devices have relatively low efficiency, thus a relevant part of the incoming power (of the order of tens of watts) must be dissipated as waste heat, potentially causing local thermal problems.

Considering the three technologies identified as appropriate for CubeSat applications, three specific thrusters are chosen as examples of each category, with suitable characteristics for the considered application: Alta's Simplified FEEP[21], a field emission thruster working with ionic liquids, the Micro-PPT[22], specifically developed in "micro versions" for CubeSats applications (currently funded by the ESA ITI program with the aim of doubling the lifetime of a 3U CubeSat at 600 km or altitude), and the Alta XR-100 resistojet[23]. The main characteristics of these devices are summarized in Table 1. For the Micro-PPT, the impulse-bit and the total impulse deliverable are given as representative figures.

	Simplified FEEP	**XR-100**	**Micro-PPT**
Power [W]	5	70	0.3
Thrust [mN]	0.1	125	
Isp [s]	2000	60	600
Mass [g]	400	115	180
Lifetime [hours]	>5000	1000	
Impulse-bit [μNs]			34
Total Impulse [Ns]			44

Table 1: Simplified FEEP, Micro-PPT and XR-100 characteristics (see Ref. 21, 22 and 23).

The low specific impulse provided by the resistojet thruster may result in a high propellant mass consumption. For this reason, the analysis carried out with this thruster is also indicative of the typical performance of chemical thrusters, such as cold gas thrusters. This thruster, indeed, is characterized by a high thrust value and, even in case of malfunctioning, it can still work (with reduced performance) as a cold gas thruster.

II.3 Low-Thrust Drag Compensation

In order to investigate the role of an active electric propulsion system to compensate the drag force, the thruster accelerative term has to be included in the perturbation term of the equations of motion.

This term takes into account the effect of EP; furthermore also the mass variation of the spacecraft due to thruster operation is modelled. The analysis is performed by means of an averaging of the Lagrange planetary equations. Differently from the natural orbital lifetime estimation, however, in this case also the increase of the semi-major axis due to the thruster is considered. This average variation is given by[17]:

$$\Delta a_{rev} = \frac{8\pi\left|\vec{a}_{th}\right|a^3}{2\mu} \quad [1]$$

where $\vec{a}_{th}$ is the acceleration exerted by the thruster considering the instantaneous spacecraft mass and μ is the gravitational parameter of the central body

A fixed propellant mass of 50 g (regardless of the CubeSat size) is considered for the lifetime estimation. As a consequence, the resulting mission durations can be intended as a sort of "worst case" performance for the 2U and 3U case where larger tanks can be foreseen. Simulations, summarized in Fig. 2, are stopped either at the propellant exhaustion or when the maximum operative life of the thrusters has been reached.

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It is clear that, as the CubeStas have very limited mass and volume, the high specific impulse of FEEP offers the significant advantage to have small tanks and small propellant mass fractions, as well a potential system life of several years. From a technological point of view, moreover, the Simplified FEEP, the Micro-PPT and the XR-100 have also simple feeding systems and they can operate with non-hazardous materials, in compliance with the CubeSat specifications[11]. In this case performance is not the same for the three CubeSat sizes, contrary to the natural deorbiting case. This is a consequence of the different masses of the three platforms that result in different acceleration magnitudes provided by the same electric thruster. This explains the presence of the three curves in each plot of Fig. 2.

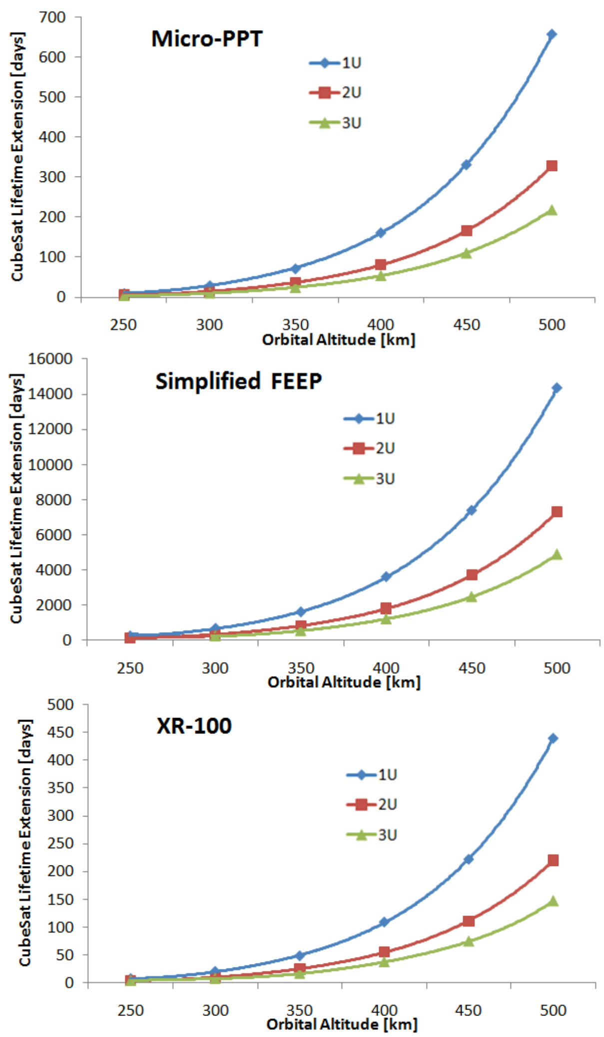

Fig. 2: Lifetimes for Micro-PPT, Simplified FEEP and XR-100 equipped CubeSat.

In light of the above, the 1U CubeSat performs better than the others, due to its smaller mass. Drag compensation can be performed for approximately three times the duration obtainable with the same thruster on a 3U platform. The lower the orbital altitude, the more substantial is the orbital lifetime increase brought by the use of an electric thruster.

For example, the natural orbital lifetime of a 3U CubeSat at 300 km of altitude in case of medium atmospheric density is approximately 60 days. Considering a Simplified FEEP thruster, lifetime increases by 221 days, with 2734 hours of thruster operation and 50 g of propellant mass consumption. On a 3U CubeSat at an altitude of 250 km, the same thruster does not provide enough acceleration to compensate for atmospheric drag. Moreover, at this orbital altitude (250 km) the lifetime of a 1U CubeSat is only 19 days and can be extended by 11 days if the 1U would be equipped with a Micro-PPT thruster and by 229 days in the case of the Simplified FEEP.

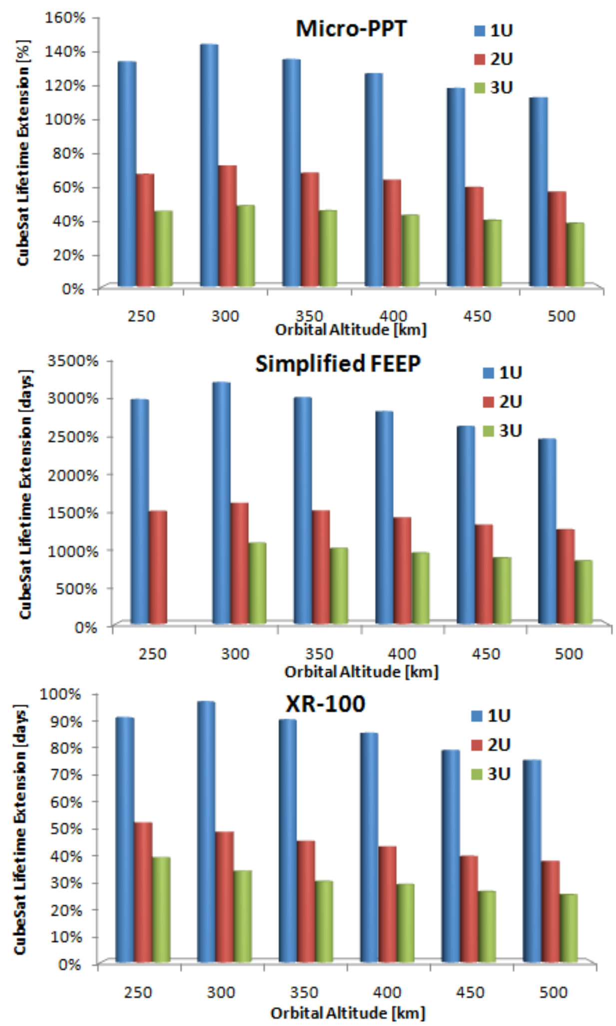

Fig. 3: Relative lifetime extension as a function of the orbital altitude in a constant medium atmospheric density scenario for each thruster.

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To complete the analysis, also the relative gain in orbital lifetime was computed. Figure 3 shows for each thruster the lifetime extension as a function of the orbital altitude considering the medium density model.

III. CUBESAT CONFIGURATION

After this overview on performance of CubeSats equipped with electric thrusters, it is interesting to approach the problem from a spacecraft system point of view. The aim is to define a preliminary sizing of the platform and a mass and power breakdown.

The standardization of the CubeSat platforms triggered the development of off-the-shelf components and pieces of equipments[24,25], making the in-house development of the basic CubeSat components an impractical proposition. Such commercially available components are considered in the following analyses.

III.1 Standard CubeSat Configuration

The goal of this section is to highlight the existence of a mass and power range allowing the introduction of an electric propulsion subsystem and, perhaps also of a small payload, considering as far as possible off-the-shelf components. With this aim, a preliminary selection of subsystems for the three CubeSat sizes is performed. This analysis is schematically summarized in Fig. 4. Values in the plots were obtained considering the state-of-the-art figures for mass and power of the main subsystems augmented with a safety margin (5-15%)[26].

In the following it is assumed that the incoming power generated by these platforms depends on the number of units. This is a natural assumption considering body mounted solar arrays. In particular, 5 W of incoming power are assumed for the 1U, 8 W for the 2U and 13 W for the 3U.

The spacecraft structure is the first off-the-shelf components; it is composed by milled aluminium and is already equipped with the standard separation springs[11]. The mass of this subsystem increases almost linearly with the number of CubeSat units.

The communications subsystem has to provide the link to relay data to ground station. It has to send commands to/from the CubeSat such as telemetry and command sequences. Furthermore, also the data collected by the payload and sensors on the satellite must be transferred to the ground base. It is assumed that the spacecraft uplinks and downlinks in UHF (400–450 MHz) or VHF (130–160 MHz, where frequencies close to 145 MHz are reserved for radio amateurs). The data rate is considered to be approximately around 300–1200 bps. Standard omnidirectional dipole antennas for these platforms (the one considered in the mass/power breakdown) are composed of two to four wires, deployable up to half a meter length, encapsulated in the top or bottom side of the CubeSat. The mass of this subsystem is not particularly critical, but it requires approximately 2.5 W. This subsystem, however, is supposed to operate only when the ground station is in the field of view, thus not continuously during the mission.

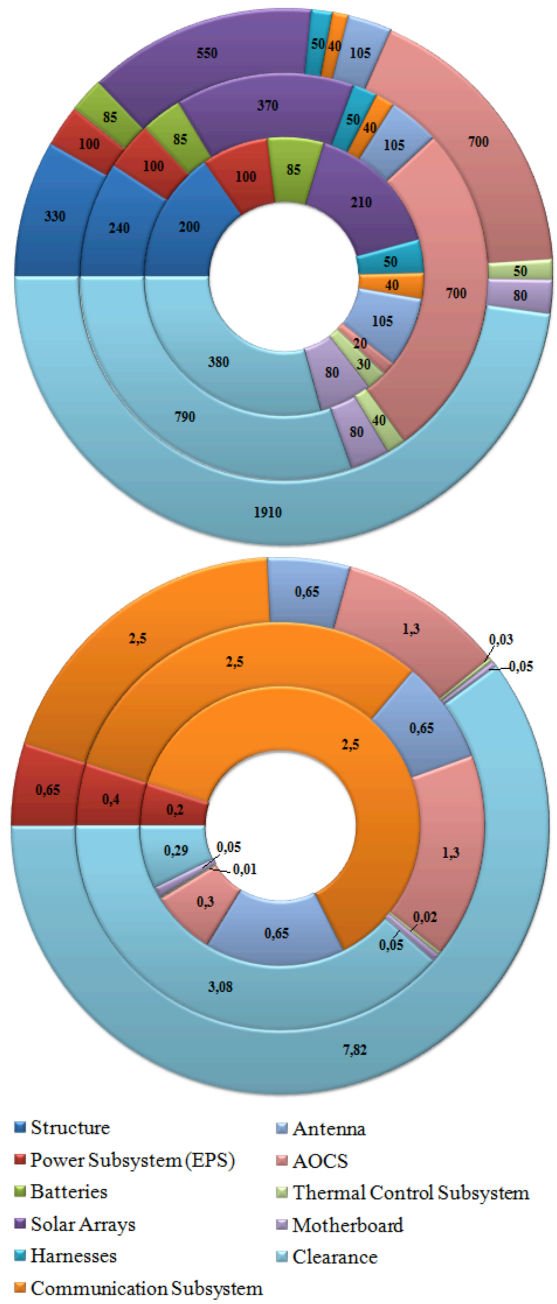

Fig. 4: Mass and power breakdown for 1U, 2U and 3U CubeSat (from the outermost to the innermost ring) equipped with standard subsystems. Masses are expressed in g and powers in W.

The attitude control of these small spacecraft is often performed by passive means. A set of magnetic coils, mounted along the structure, is frequently used to orient the spacecraft along the Earth magnetic field; this is the approach assumed in the present analysis for the 1U size, while active systems are considered for the 2U and 3U sizes. The use of miniaturized reaction wheels results in increase of pointing performance, but also in higher mass (about 700 g instead of 20 g) and power consumption (about 1.3 W instead of 0.3 W). On the single unit CubeSat, the reaction wheel option is not available due to the very limited mass, power and volume available.

The Power Subsystem (EPS) consists of the power control and distribution unit and solar arrays. The power control unit is in charge of the management and protection of batteries and solar arrays. The mass of this subsystem is almost constant on the different CubeSat sizes, while their power requirement increases with the number of units.

Several companies worldwide are now producing off-the-shelf solar arrays for CubeSat[27] applications. Such arrays are often placed over a printed circuit board substrate (PCB), covering almost all the CubeSat external surface. Their total mass depends on the size of external surface covered, thus it is larger for the 3U size.

Batteries usually consist of commercial Lithium Polymer Cells with high energy density. In Fig. 4, standard batteries were considered in all cases. These provide approximately 10 Wh of energy that would result in 5 hours of autonomy considering a basic absorption of 2 W (see Sec. IV for a more detailed analysis). This duration is much larger than 35 min of eclipse, a realistic value for LEO orbits. Considering also the thruster and/or an additional payload operating at the same time, the CubeSat autonomy would decrease. More refined computation are performed in Sec. IV.

The CubeSat thermal control is often achieved relying on passive methods such as radiating surfaces or thermal coatings. Protection of the energy storage subsystem sometimes might require the usage of active thermal control devices, such as heaters, to avoid the freezing of batteries; this scenario, however, is not addressed in the present study. For the mass of this subsystem, a value of 30–50 g has been assumed in Fig. 4.

The onboard computer provides the general data handling both for the payload and the spacecraft itself. Data storage is often included in this component. In the considered applications, 4–12 Mbytes are considered for data storage in standard space proven flash memories, independently on the platform size. This component is not crucial, either mass- or power-wise.

Finally, an additional 50 g of harness were included, taking into account all cabling and connection components required, in particular, by the on-board electronics.

The assumptions made so far for the system size do not take into account several factors that could increase the total mass, the power required or the cost of each subsystem. These factors have to be considered in a more detailed analysis of the platform, assessing the impact of the actual operative orbit and payload requirements. Nonetheless, data resulting from the mass budget of Fig. 4 show that there exist mass and power margins for all CubeSat sizes. For the 1U, up to 380 g of mass are still usable, becoming 790 g for the 2U CubeSat and 1910 g for the 3U. Even assuming all subsystem working at the same time, 0.3 W might be available on the 1U size, 3.1 W on the 2U and 7.8 W on the 3U. Within these margins, the goal of Sec. III is to consider the possibility to include an electric thruster, the associated propellant and possibly an additional payload.

For the sake of completeness, it should be remembered that the cost of the standard CubeSat configurations outlined is about 40 k€ for the 1U platform, 45 k€ for the 2U and 50 k€ for the 3U.

III.2 Electric Propulsion Subsystem

The recent interest for space component miniaturization and for the development of scaled down thrusters with reduced performance has opened the way for the introduction of EP in the CubeSat and nano-satellite domain, overcoming the perceived limitations of older devices. Besides the system mass, the most relevant CubeSat constraint is represented by its size and the external constraints imposed by the P-POD deployer. Accordingly, the described propulsion system has been designed to fit in a 1U CubeSat occupying one side of the satellite. Moreover, the overall dimensions of the thrusters and of the other components of the propulsion system must be compatible with the presence of other payloads in the CubeSat internal volume.

More in detail, the MicroPPT system takes approximately 90x90x25 mm [22], does not require any feeding line or tank due to the fixed quantity of propellant mass (Teflon) and, for this reason, a 1U CubeSat can be equipped with more than a single device. The Simplified FEEP system is designed to replace one side of a CubeSat unit and it is composed by the propellant tank filled with ionic liquid, the control electronics and the thruster itself. Given the Simplified FEEP system external volume (max. 100x100x50 mm), this propulsion system could not fit into a 1U CubeSat, while it could be more suitable for a 2U or even 3U CubeSat. Beside the thruster volume, indeed, also the power consumption is such that a 1U CubeSat might require storing a relatively large amount of energy to operate the thruster. Accordingly, a large part of CubeSat mass and volume should be kept to store the batteries.

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The resistojet thruster is a quite complex system with a pressurized propellant tank with a pressure control unit, some control valves and the thruster itself with an overall volume of almost 1U. Its high peak power consumption requires a heavy control unit but, considering its high thrust value, it can provide an high velocity increment with just few seconds of operation. Accordingly, this thruster does not require a relevant quantity of stored energy. Considering its low efficiency (~50%), the resistojet has a large thermal loss with respect to other thrusters, making it more suitable for a 3U CubeSat.

Considering the mass ranges identified for the three CubeSat sizes in Sec. III.1, beside the electric propulsion devices described, many additional instruments may still fit into CubeSat platforms as payload. A standard optical payload, for instance, able to reach a resolution of 80 m at 650 km of altitude is available[28]. It weighs approximately 200 g and requires up to 0.6 W for the image acquisition. As alternative payload, several universities developed instruments for charged particle detection, atmospheric observation, and radiation measurement. By way of example, a complete instrument package for the measurement of magnetic field and charged/neutral particles can be miniaturized up to <200 g (single sensors weigh few tens of g) requiring 500 mW of power.

The on-board communication subsystem may also be used to test network protocols without any further mass requirement (but with a slightly increased power required). Finally, it must be noticed that the development of CubeSat-related technology is constantly advancing and many applications, considered beyond the limits of such small spacecraft, are now becoming feasible. In this context, one can conceive the possibility that a Synthetic Aperture Radar (SAR) can be implemented on a CubeSat platform[29].

IV. MISSION ANALYSIS REFINEMENT

Among the electric thrusters considered, the one providing the best compromise between thrust and power required is the Simplified FEEP. Considering the results of the analysis of Sec. III.2, this section is devoted to a mission analysis refinement of a 2U CubeSat equipped with this device. The necessary orbital propagations and the related on-board power management analysis were carried out using the Alta mission simulator SATSLab[14].

The thrust strategy considered is implemented in order to maintain the spacecraft altitude. The orbit maintenance maneuvers consist in keeping the spacecraft within a pre-defined box. The electric thruster fires in the velocity direction to compensate the loss of orbital energy, drained by the atmospheric drag. The upper displacement limit for the orbital semi-major axis has been set to 100 m and the energy consumption has been also considered as a limiting factor for the thruster firing in case of batteries depth of discharge beyond a predefined threshold (95%). To model a real worst case scenario, such a maneuver is supposed to be performed regardless from the illumination/eclipse condition of the spacecraft.

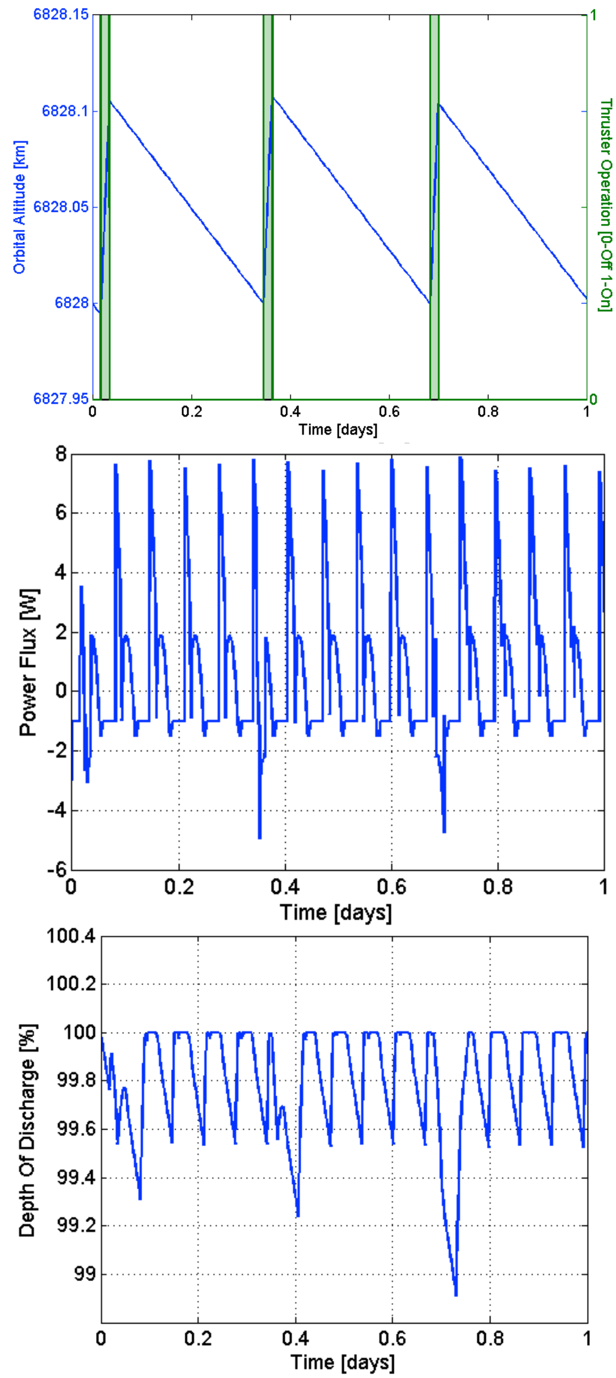

Fig. 5: Top: Orbital altitude and thruster operation. Centre: Incoming/outgoing power flux. Bottom: battery depth of discharge evolution over half day.

The scenario under consideration consists in a 300 km, non-sunsynchronous Earth orbit (under non-optimal illumination conditions), with the J_2 perturbation and the atmospheric drag duly taken into account. In particular, atmospheric drag has been modelled using the Harris-Priester model[17]. The power required to operate the CubeSat subsystems and the thruster is generated by the solar arrays considering the relative sun position. The power requirements of an hypothetic payload operating for half an orbit, a base consumption of the spacecraft itself and the power required by the thruster are considered during the whole simulation.

Two of the more relevant outputs of this refinement are the time evolution of the spacecraft energy and incoming/outgoing power. These are shown in Fig. 5 over one day of simulation. In Fig. 5 the orbital altitude evolution over one day is plotted together with the thruster operation condition (top). The plot in the middle shows the different power exchange conditions resulting in different power consumption levels and the bottom plot shows the depth of discharge of the 10 Wh batteries mounted in the CubeSat. From the centre plot of Fig. 5, the highest value (~8 W) corresponds to an illumination condition with no power required by any payload or by the thruster. The first lower value (2 W) represents the instantaneous power flux in the same illumination conditions with the payload switched on. The first negative value (-1 W) corresponds to eclipse conditions (zero incoming power) during which the base bus power requirement is the only power load. The lowest negative value (~5 W), instead, corresponds to eclipse situations where the batteries have to supply energy both to the thruster and to the payload.

Intermediate negative power flux values correspond to eclipse conditions where payload and spacecraft bus are the only active loads. This results in a peak absorption slightly larger than 7 W against an average power generation of 2.6 W.

From these simulations, altitude turns out to be maintained within the defined box for approximately 180 days with a total propellant mass consumption lower than 5 g. The stability in the battery depth of discharge reveals that, even in presence of the highlighted negative power peaks, the energy storage is capable of sustaining the CubeSat energetic requirements for the whole mission duration. In general, also considering the other platform sizes and longer simulation periods, it results that in all cases the considered incoming power is sufficient to sustain both the propulsion subsystem and the remaining spacecraft loads.

V. EP-EQUIPPED CUBESAT PERFORMANCE

In order to give a more detailed overview of the increased capabilities enabled by adding an electric propulsion system onto a nano-satellite, and in particular on a CubeSat, this section describes the obtainable performance in terms of transfer capabilities and deorbiting time with respect to propellant mass.

Considering the thruster options listed in Sec. II.2, Tsiolkovsky's equation[17] is used to estimate the obtainable velocity increment (ΔV). Moreover, also the time required to acquire target conditions is assessed for the FEEP and resistojet thrusters under the assumption of constant acceleration assuming an average spacecraft mass. The same quantity is computed for the PPT considering the impulse bit number per day and the velocity increment obtained with each impulse.

Figure 6 shows the velocity increment and the time required to gather it with respect to the propellant mass for the three thrusters considered. The areas highlighted in the plots of Fig. 6 show the range of achievable performance obtained considering the acceleration exerted by each thruster on 1U (best case) and 3U (worst case) CubeSats.

It should be noticed that the time required to acquire a given ΔV is almost the same for each CubeSat regardless of its mass. The decrease in the acceleration value, indeed, is counterbalanced by an almost equal reduction of the achievable velocity increment. The maximum considered propellant mass (150 g) represents a reference value and this mass might not be easily storable in the CubeSat.

By way of example, considering the plots in Fig. 6, a velocity increment of 750 m/s can be obtained in about 11 months considering a 3U CubeSat equipped with a Simplified FEEP thruster and 150 g of propellant while a 2U CubeSat with the same configuration may result in a ΔV larger than 1 km/s. The same propellant quantity on a 2U CubeSat results in a velocity increment of about 500 m/s if equipped with the Micro PPT thruster or 80 m/s with the XR-100. A 1U CubeSat equipped with a Micro PPT thruster and 150 g of propellant may achieve a ΔV of about 720 m/s, corresponding to an altitude increase of 1450 km and an inclination change of 3.4 deg.

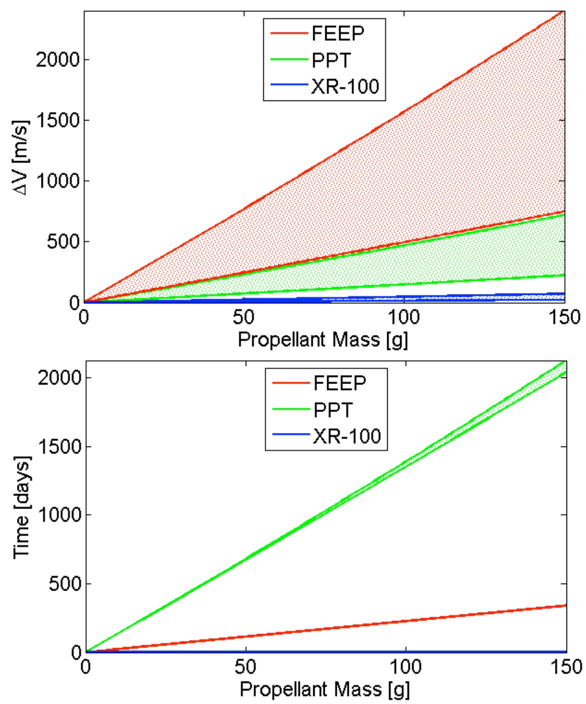

Fig. 6: Velocity increment and time required to achieve it with respect to ejected propellant mass for the three thrusters considered.

Considering the obtained ΔV values, a reference orbital altitude of 300 km is assumed to estimate the achievable semi-major axis and inclination change, by means of the Edelbaum analytical approximation[30]. Plots in Fig. 7 show the range of achievable altitude and inclination change as a function of the ejected propellant mass.

From the plots of Fig. 7, 150 g of propellant on a 3U CubeSat equipped with a Simplified FEEP thruster may enable an orbital altitude increase of 1500 km or an inclination change of 3.5 deg. A 2U CubeSat equipped with a PPT thruster and 150 g of propellant may increase its orbital altitude by 920 km or change its orbital inclination by 2.2 deg. The same propellant quantity on a 1U CubeSat equipped with a resistojets thruster results in an altitude increase of 126.5 km with about 700 s of thrusting time.

For the sake of completeness, also CubeSat deorbiting scenarios are here presented. Considering a 3U CubeSat on a 900 km altitude circular orbit with a drag exposed area of 0.01 m^2, its natural lifetime in a constant minimum atmospheric density scenario is of hundreds of years. Thus, it would not be compliant with the most recent international guidelines for spacecraft end-of-life disposal. In this very conservative scenario, a Simplified FEEP allows the CubeSat to deorbit from its initial altitude within 180 days with a propellant mass consumption of 78 g and a corresponding ΔV of 387 m/s. An XR-100 thruster mounted on the same 3U CubeSat would require more than 2.3 kg of propellant to perform the same task with a total ΔV of 504 m/s. The same CubeSat, equipped with a Micro-PPT thruster, requires about 200 g of propellant and more than 7 years to deorbit from the same altitude.

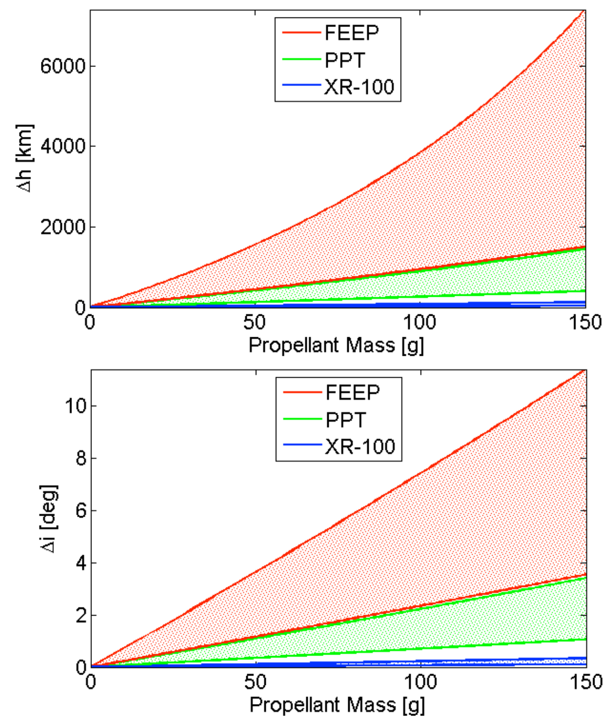

Fig. 7: Achievable altitude and inclination change range vs. ejected propellant mass for the three thrusters considered.

VI. CONCLUSIONS

From the preliminary mass and power breakdowns derived in our analyses, it results that an electric propulsion subsystem can be embedded in a standard CubeSat platform. This holds even for the smaller 1U version; however, in such case the mass and power margins left by the other subsystems do not seem to allow the inclusion of both a propulsion subsystem and an additional payload.

While CubeSat has already shown over the years its potential, even encouraging the development of miniaturized components and the birth of dedicated parts suppliers, the addition of an electric thruster opens a wide range of new possibilities for this platform, paving the way for the actual demonstration of CubeSat as viable platforms for simple, low cost access to space.

All in all, the Micro-PPT represents the most suitable choice for the 1U CubeSat both in terms of mass/volume and performance. 2U or 3U CubeSats too can be equipped with this thruster, but the achievable benefits and performance levels are definitely more modest and less interesting. Conversely, the Simplified FEEP thruster is not a good candidate for a 1U CubeSat but it may easily fit into larger CubeSats providing a high total impulse with a small quantity of propellant. Finally, the XR-100 resistojet thruster can be considered a good choice for a 3U CubeSat if a fast orbit transfer is required.

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