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Conceptual study of a low power electric propulsion system for LEO and VLEO microsatellite missions

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ΔΙΠΛΩΜΑΤΙΚΗ ΕΡΓΑΣΙΑ

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ABSTRACT

The new space market trends and their influence in the whole space mission analysis, especially for small satellite (<200kg) applications, increase the research interest in several fields. In this frame, the primary goal of this study is to support this interest and pinpoint the advantages of lowering the altitude to LEO and VLEO regimes, identify the related mission challenges and focus on the importance of the satellite propulsion system to achieve the specific mission objectives. The advantageous utilization of electric propulsion systems in (V)LEO applications and the related growing perspectives are addressed and a conceptual study of a low power (<200W) electric propulsion system for LEO and VLEO microsatellite missions is provided.

A representative system (satellite platform) is specified and determined along with the related system application (earth observation SAR). Based on those criteria all mission parameters are extracted to drive the design of the related electric propulsion system. The related orbits are selected (one VLEO (400km) and two LEO (500km and 600km)), the use of the propulsion system (orbit maintenance, orbit correction after release, End-of-Life de-orbiting, collision avoidance) is defined, the atmospheric characteristics are evaluated, the related ΔV calculations are performed and mass, power and thrust requirements are set.

A thorough trade-off analysis is carried out to select the electric propulsion system that maximizes the payload mass and power fraction for the three selected orbits. Based on the trade-off for the defined application, the electric propulsion system selected is conceptually described, meaning the system architecture is defined including all required subsystems. The subsystems are analysed, containing the integrated thruster unit and its components, the propellant calculations and the related tank dimensioning and the specification of the power and control unit features.

In this way, the conceptual study approach fully specifies one optimized electric propulsion system that can efficiently cover the needs of three different orbit scenarios of a quite representative mission application that fits most of the current space market needs. This approach offers a useful tool to any satellite user to adopt this system and easily adjust it to any (V)LEO mission. It constitutes a possible plug and play option for applications in this specified operating envelope, easy to integrate and configure, leading to a faster, cheaper, efficient and more reliable electric propulsion system integration to any small satellite. This becomes the most important and novel characteristic of this study, with this concept-like focus on the electric propulsion system being the core of the whole analysis.

SUBJECT AREA: Low power electric propulsion for microsatellites

KEYWORDS: Electric propulsion, LEO, VLEO, microsatellites, earth observation, space applications, new space, mission analysis, ΔV calculation

ΠΕΡΙΛΗΨΗ

Οι νέες τάσεις της διαστημικής αγοράς και η επίδρασή τους στην ανάλυση των διαστημικών αποστολών, ειδικά σε εφαρμογές μικρών δορυφόρων (<200kg), αυξάνουν το ερευνητικό ενδιαφέρον σε πολλά πεδία. Σε αυτό το πλαίσιο, ο πρωταρχικός στόχος της παρούσας μελέτης είναι να υποστηρίξει αυτό το ενδιαφέρον και να εστιάσει στα πλεονεκτήματα της μείωσης της τροχιάς (από 300 έως 600km), να εντοπίσει τις σχετικές προκλήσεις των νέων αποστολών και να επικεντρωθεί στη σημασία του συστήματος προώθησης για την επίτευξη των συγκεκριμένων στόχων της αποστολής. Ερευνώνται τα πλεονεκτήματα της χρήσης συστημάτων ηλεκτρικής προώθησης σε εφαρμογές χαμηλής και πολύ χαμηλής γήινης τροχιάς καθώς και οι σχετικές προοπτικές ανάπτυξης και παρέχεται μια εννοιολογική μελέτη ενός συστήματος ηλεκτρικής προώθησης χαμηλής ισχύος (<200W) για αποστολές μικροδορυφόρων σε αυτές τις τροχιές.

Αρχικά, προσδιορίζεται ένα αντιπροσωπευτικό σύστημα (δορυφόρος) μαζί με τη σχετική εφαρμογή του (SAR για παρατήρηση της γης). Με βάση αυτά τα κριτήρια εξάγονται όλες οι παράμετροι της αποστολής που οδηγούν τον σχεδιασμό του συστήματος ηλεκτρικής προώθησης. Επιλέγονται οι σχετικές τροχιές (400km, 500km και 600km), ορίζεται η χρήση του συστήματος προώθησης, αξιολογούνται τα ατμοσφαιρικά χαρακτηριστικά, εκτελούνται οι σχετικοί υπολογισμοί ΔV και ορίζονται οι απαιτήσεις μάζας, ισχύος και ώσης.

Πραγματοποιείται λεπτομερής συγκριτική ανάλυση για την επιλογή του συστήματος ηλεκτρικής προώθησης που μεγιστοποιεί το διαθέσιμο ποσοστό μάζας και ισχύος για τις τρεις επιλεγμένες τροχιές. Με βάση την συγκριτική μελέτη για την καθορισμένη εφαρμογή, το επιλεγμένο σύστημα ηλεκτρικής προώθησης περιγράφεται εννοιολογικά, δηλαδή ορίζεται η αρχιτεκτονική του συστήματος και όλων των απαιτούμενων υποσυστημάτων. Αναλύονται τα υποσυστήματα, συμπεριλαμβανομένων της ενσωματωμένης μονάδας του προωθητήρα και των επιμέρους στοιχείων της, οι υπολογισμοί του καυσίμου και οι σχετικές διαστάσεις της δεξαμενής και οι προδιαγραφές των χαρακτηριστικών της μονάδας ισχύος και ελέγχου.

Με αυτό τον τρόπο, η εννοιολογική σχεδιαστική προσέγγιση προσδιορίζει πλήρως ένα βελτιστοποιημένο σύστημα ηλεκτρικής προώθησης που μπορεί να καλύψει τις ανάγκες τριών διαφορετικών σεναρίων τροχιάς μιας αντιπροσωπευτικής εφαρμογής αποστολής που ταιριάζει στις περισσότερες από τις τρέχουσες ανάγκες της διαστημικής αγοράς. Αυτή η προσέγγιση προσφέρει ένα χρήσιμο εργαλείο σε κάθε χρήστη του δορυφόρου για να υιοθετήσει αυτό το σύστημα και να το προσαρμόσει εύκολα σε οποιαδήποτε αποστολή σε αυτές τις τροχιές. Αποτελεί μια πιθανή επιλογή «plug and play» για εφαρμογές σε αυτό το καθορισμένο εύρος αποστολών, εύκολη στην ενσωμάτωση και τη διαμόρφωση, που οδηγεί σε ταχύτερη, φθηνότερη, αποτελεσματικότερη και πιο αξιόπιστη διαχείριση του συστήματος ηλεκτρικής προώθησης σε οποιονδήποτε μικρό δορυφόρο. Αυτό γίνεται το πιο σημαντικό και πρωτότυπο στοιχείο αυτής της μελέτης, με αυτή την εννοιολογική εστίαση στο σύστημα ηλεκτρικής προώθησης να αποτελεί τον πυρήνα της όλης ανάλυσης.

ΘΕΜΑΤΙΚΗ ΠΕΡΙΟΧΗ: Ηλεκτρική προώθηση χαμηλής ισχύος για μικροδορυφόρους

ΛΕΞΕΙΣ ΚΛΕΙΔΙΑ: Ηλεκτρική προώθηση, χαμηλή γήινη τροχιά, μικροδορυφόροι, παρατήρηση γης, διαστημικές εφαρμογές, ανάλυση αποστολής, υπολογισμός ΔV

Αφιερωμένο σε εκείνη, την μία.

Statement of Originality

This thesis and the work to which it refers, are the results of my own effort. Any ideas, data, figures or text resulting from the work of others (whether published or unpublished) are fully identified and highlighted as such within the thesis and attributed to their originators in the text and the relevant bibliography (references).

Alexandros Manoudis

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PREFACE

This dissertation thesis was written in the frames of the Inter-institutional Master Studies Program in Space Technologies, Applications and seRvices (STAR) provided by the Department of Informatics and Telecommunication of the School of Science of National and Kapodistrian University of Athens.

A Master that gave to me several opportunities and important information to better understand the Space technical and interdisciplinary domains and occupy with specific aspects of this field. This inspiration for space adventures and further research is the driving factor for my career from now on.

In this direction, this thesis is a result of an obtained personal interest in the field of electric propulsion that along with the proper and focused support and guidance of my supervisor Prof. Vaios Lappas became a reality. For this and the general cooperation during my study I would like to express my gratitude to him.

1. INTRODUCTION

1.1 New space

Under the term of “New Space”, there is an ongoing “revolution” in the space sector with new players/commercial entrepreneurs/businesses entering a domain traditionally occupied by institutional players (“Old Space” e.g., space agencies working with large companies) to exploit the new opportunities opening in front of them [1].

In this direction, several parameters play an important role to this revolutionary transition especially in near Earth applications. The great development of related space technologies over the years (payload size reduction, better efficiency and performance, lower costs, use of commercial of the shelf (COTS) parts), the reduction of cost per launch (reusable and multi-satellite launchers), the commercialization of satellite production lines and the high profitability of downstream value-added services are the most important factors that drive this “New Space” revolution.

The potential of using small satellites for commercial applications has successfully appraised the investment community into enabling “New Space” enterprises to launch several constellations like never before [2] searching for quick performance demonstration. More and more commercial players are involved in the Low Earth Orbit (LEO) applications that range from Earth Observation (EO), remote sensing and navigation to Telecommunications (e.g. 5G high speed internet, Internet of Things (IoT)) and other more exotic applications (e.g. space tourism). Those applications are recently planned to cover the majority of Earth territories in almost real time and in this frame, big constellations of small satellites combined together are required. The so called “megaconstellations” are already launched (Swarm Technologies, SpaceX, OneWeb, Spacety) exploiting the opportunities arisen in this field.

Most of these newly developed constellations are utilizing small satellites with a mass up to around 250kg to effectively exploit the applicable payloads, flying in several orbital planes in the lower limit of LEO range (between 300km and 600km) to achieve improved communications link budget, better optical resolution and full coverage.

Recent technology evolution, mainly the related subsystems miniaturization, has led to significant cost-reduction and enabled more flexible satellite development cycles even at very low altitudes, opening a new era in the space market sector. One of the major technical challenges that significantly affect the performance of the whole satellite constellation is the propulsion system. The research in this field offers great opportunities for improvement, in particular on electric propulsion systems, which are also hybrid systems that would utilize different modes of operation [1].

1.2 Small Satellites

Following on this “New Space” trend, small satellites (up to 500kg) have become increasingly popular in the last 10 years as they can offer cost reduction, greater reliability, are more affordable for a large variety of commercial applications, and can be easily integrated to this new megaconstellation philosophy.

Although the benefit of flying smaller and, therefore, lighter spacecraft from the overall launch cost perspective may be apparent, additional advantages make small satellites a viable alternative for a number of potential missions [3]. The functionality of a single satellite can be distributed among several smaller satellites, reducing the complexity of each individual satellite, decreasing the overall design and manufacturing costs per satellite, improving the reliability of the system. The transition to lower orbits that require smaller payloads (for EO or communications) and the growth in the payloads efficiency (smaller payloads can offer the same quality of application) make small satellites a great fit for constellation concepts in LEO and VLEO especially in altitudes around 300km to 600km. Moreover, the reduction of the complexity of the small satellites facilitates rapid prototyping, lowers the development life-cycle, and stimulate more efficient and economical production and manufacturing practices [3].

According to NASA [4], small satellites are defined as satellites with mass lower than 180kg and related sub-categories based on their mass. Recent studies though, expand the upper limit up to 500kg (for minisatellites category), creating an accepted small satellite classification (per kg) in the space community as follows [5]:

- Minisatellite: 150-500kg
- Microsatellite: 10-150kg
- Nanosatellite: 1-10kg
- Picosatellite: 0.01-1kg
- Femtosatellite: 0.001-0.01kg

There is also the broad category of small satellites called CubeSats, that is defined by its modular size measured in Units (U). A CubeSat is a type of miniaturized satellite usually for space research that is made up of multiple cubic units of 10x10x10cm size (1U). They have a mass of no more than 1.33kg per unit and to date 1U, 1.5U, 2U, 3U and 6U satellites have been launched, while missions planned reach up to 12U or even more.

Many of the satellites utilized in the newly designed megaconstellations are inside the Micro- and Mini- satellite regime, mainly focusing on a range of 100-250kg. Several types and configurations of those satellites are present in many different missions, but the most commonly used scheme incorporates a box-like main part that includes all required subsystems, antennas and payloads and deployable solar arrays for the power production, as shown in Figure 1.

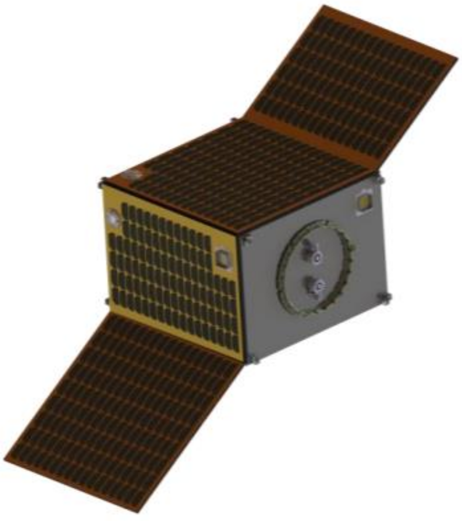

Figure 1: Small satellite reference configuration, Sitael S-200 platform, [6]

Lower cost electronics and "rideshare" launch opportunities have enabled the growth of small satellites. These smaller satellites can also be developed faster, allowing academic institutions, smaller companies, and developing nations to access space. Small satellites have since matured towards a mainstream market segment, providing opportunities for cost-effective in-orbit technology demonstration. When launched in greater numbers, they are also able to provide a new class of observation data with frequent updates and global coverage.

The contemporary emphasis on cost reduction and actual down-sizing of the satellites the recent years, has forced a re-evaluation of technologies which may have critical impact on satellite mass. The propulsion system has been a dominant contributor to the overall mass of the spacecrafts for many commercial, scientific, and military missions over the years and offers a great room for research and development.

1.3 Moving to Very Low Earth Orbits (VLEO)

Spacecraft orbits have been historically classified according to their altitude and the type of mission they are typically utilized in, as follows:

- Geostationary orbits (GEO, 35786km): They are mainly used for standard telecommunications (e.g. broadcasting) and Earth atmosphere observations (e.g. weather) due to their synchronized rotation with the Earth and their ability to accomplish almost global coverage with only a small number of satellites (usually three).
- Medium Earth Orbits (MEO, 2000km to 35786km): They are frequent used for navigation services through constellations, such as GPS, GLONASS and Galileo, that require balancing between global coverage and diversity (the number of satellites visible from the ground at the same time).
- Low Earth Orbits (LEO, <2000km): They are the primary selection for EO missions as they are closer to the surface and can in this way obtain higher resolution images. Recently, LEO has also become popular for telecommunications constellations due to the increased bandwidth and the reduced latency and power requirements. Typical LEO satellites operate above 500km to avoid the requirement for drag compensation and eliminate the effects of aerodynamic disturbances.
- Very Low Earth Orbits (VLEO, 250km to 450km, a subset of LEO): They have seen relatively little use since the early Cold War reconnaissance satellites. However, research is underway to explore the benefits of returning to these orbits and to address the challenges of operating sustainably at lower altitudes [7]. Generally, at 450km, the aerodynamic drag is strong enough to make a spacecraft decay in less than 5 years, requiring significant changes in traditional spacecraft designs (which usually come with a 5-year operational lifetime target) [8].

In LEO and VLEO several inclination orbits can be utilized based on the mission requirements, but the most common orbits include:

- Polar Orbit: An orbit that passes above or nearly above both poles of Earth on each revolution. Therefore, it has an inclination of (or very close to) either 90 degrees or −90 degrees. Usually within 30 degrees of the Earth’s poles, the polar orbit is used for satellites providing reconnaissance, weather tracking, measuring atmospheric conditions, and long-term Earth observation.
- Sun-Synchronous Orbit (SSO): A type of polar orbit, SSO satellites are synchronous with the sun, such that they pass the equator at the same local solar time on every pass. Useful for image-taking satellites because shadows will be the same on every pass.

1.3.1 Benefits of lowering the altitude

Reducing the orbital altitude of satellites in LEO at around (or below) 450km can offer several benefits that could drive the growth of a new generation of satellites that can succeed higher performance at a lower cost. Operating telecommunications and remote sensing satellites at lower altitudes offers significant technical and cost advantages. The benefits can be broadly categorized as those that improve payload performance, platform benefits, and the opportunity to exploit new technological approaches, which are summarized in Table 1 [9].

Moreover, as presented in [10], in the LEO range, several models can be used to globally represent the radiation environment. Radiation environment seems to be less aggressive moving from 600km and 300km. This is in part due to the increasing atmospheric density and demonstrates a further benefit in operating at lower orbital altitudes. With the increasing interest and use of COTS components without radiation-hardening, a reduction in radiation exposure at lower altitudes may enable longer duration missions as the lifetime dosage reduces correspondingly. Alternatively, even cheaper commercial components may be able to be successfully utilized in VLEO, further decreasing mission costs, system development time and need for redundancy.

Table 1: Benefits of operating satellites at lower altitudes, [9]

Category	**Benefits**
Payload performance	For optical payloads: • Increased resolution or reduced aperture size • Improved radiometric performance • Smaller and less expensive
	For radar and communications payloads: • Significantly improved link budgets • Reduced antenna size and transmission power • Reduced latency and improved frequency reuse
Platform	• More benign radiation environment • Improved launch vehicle payload mass (up-mass) • End-of-life disposal is enabled due to drag • Reduced space debris collision risk • Improved geospatial accuracy and relaxed pointing requirements
New technologies	• Aerodynamic attitude and orbit control • Atmosphere breathing electric propulsion for drag compensation

1.3.2 Challenges

Despite the extensive benefits, there are still some important challenges while operating in lower altitude orbits, thus specific attention and further research are required so as to accomplish efficient commercial exploitation.

The most significant issue is the increase of aerodynamic drag experienced by satellites operating at these altitudes. As orbital altitude is reduced, atmospheric density increases, consequently increasing the aerodynamic forces experienced by orbiting satellite and limiting its useful lifetime before the final decay and burn up in the atmosphere. This drag gradually enforces the satellite to reduce its velocity, lose altitude and change its orbit.

Therefore, in order to provide extended lifetime operations, current satellites have to be equipped with efficient propulsion systems and carry enough propellant increasing in this way satellite costs considerably. Variations in the atmospheric density and any presence of thermospheric winds can also have a disturbing effect on the satellite stability and pointing capability. These effects may have a detrimental effect on image quality or communications networks if not properly compensated.

The atmosphere at VLEO is principally composed of atomic Oxygen, the most abundant element from 180km to 650km. When the Sun’s ultraviolet radiation strikes the residual diatomic oxygen, it photo-dissociates into atomic oxygen. Their recombination with atoms of oxygen or nitrogen occurs at lower rates because the density of the atmosphere at this altitude is too low. The atomic oxygen is highly reactive and can deteriorate optical and thermal coatings [11]. In combination with the high orbital velocity and thermospheric temperatures, this atomic oxygen can damage the external surfaces of spacecraft through erosion. Sensitive optics, solar arrays, and antennas can also be adversely affected, reducing the mission performance and lifetime of the satellite. In order to bypass these degradation issues, special care shall be taken into account over mission lifetime or appropriate selection of materials that are resistant to atomic oxygen (high density materials and graphite).

The satellite communications window to downlink the produced data to a ground station gets reduced as the altitude decreases. The pass duration is reduced due to an orbital velocity increase and an elevation angle constraint. In this frame, and especially in EO satellites where the data generated are highly increasing (better resolution of images), a VLEO satellite may require a downloading data rate of a higher-orbit satellite, driving to the utilization of innovative solutions. The main trends are the usage of higher bandwidth communications subsystems, and the data transmission through communication relay satellites at GEO increasing the communication time (required bandwidth is reduced) and eliminating the need for ground stations.

1.4 Applications

Following the short description presented in the previous sections regarding the small satellites features and the lower altitude orbits characteristics there is a set of feasible applications mainly in the EO, telecommunications and scientific/development fields. More specifically, satellites at those altitudes provide value across a wide range of application areas including:

- Infrastructure monitoring and asset tracking
- Environmental monitoring
- Precision agriculture, and food security
- Defence, intelligence, and security
- Maritime surveillance
- Disaster monitoring, management, and response
- Energy and natural resources
- High-bandwidth, low-latency, global communications
- Navigation systems from LEO
- Space surveillance and tracking

According to [12], the Union of Concerned Scientists satellite database (last updated in 1/1/2021) there are 3372 satellites orbiting Earth in total. 2612 of them are flying in LEO altitudes (77.5%) but only 173 below 450km reflecting the small VLEO utilization up to now. This study is focusing on altitudes at the lower limit of LEO (below 600km) and in Table 2, a generic application classification of the (1912) orbiting satellites is presented.

From this classification it can be extracted that the majority of the satellites at those altitudes is used for communications services (56.54%), EO applications (28.61%) and technology development/demonstration (11.35%). This is reasonable due to the already existing presence of OneWeb and SpaceX (Starlink) megaconstellations in the communications domain. EO applications at those altitudes are gradually increasing over the years exploiting the “New space” opportunities, while technology development and demonstration is always ongoing to support the growth boost in these sectors.

Table 2: Applications classification of orbiting satellites below 600km

Application	Number	Percentage
Communications	1081	56.54%
Earth Observation	547	28.61%
Earth Science	12	0.63%
Technology development/demonstration	217	11.35%
Surveillance	3	0.16%
Educational	3	0.16%
Space science	44	2.30%
Unknown	5	0.26%

1.5 Payloads

In order to fulfil the wide range of applications and services provided by the satellites at those altitudes, a proportionally wide range of payloads is required. In the next table (Table 3), a short introduction of the main types of possible payloads and their example applications is provided. In most of those cases, constellations of satellites are required to achieve the goals of each application usually for big Earth coverage, low latency and short revisit times.

Table 3: Payloads and applications

<table>
<tr><th>Payload</th><th>Application</th><th>Remarks</th></tr>
<tr><td>SAR (Active)</td><td>Earth Observation:
Maritime and land surveillance,
Agriculture monitoring,
Land monitoring,
Interferometry.</td><td>SAR capability to acquire images with every lighting and weather condition ensure a great advantage</td></tr>
<tr><td>Optical Panchromatic</td><td>Earth Observation:</td><td>A large amount of imaging data is expected to be</td></tr>
</table>

High resolution multi-spectral	• Disaster monitoring, • Surveillance.	generated so the satellite shall be equipped with adequate storage device and downlink bandwidth
Thermal Infrared (TIR)	Earth Observation: • Land use, • Management of natural environmental and technological hazards.	4 TIR bands
Weather scanning radar	Earth Observation: • Monitoring of extreme weather events such as storms and tornados.	Passive microwave spectrometer with frequency close to the Oxygen absorbing one (~118 GHz)
GNSS Reflectometry	Earth Observation: • Information about the sea wind and waves or about the ground soil moisture for agriculture, • Tree vegetative biomass and ice data.	Receives the signals transmitted by GNSS satellites, such as GPS, Galileo or GLONASS, and process it to retrieve the required information
VHF Data Exchange System	Telecommunications: • Terrestrial and satellite radio communication links in the VHF maritime band to facilitate globally interoperable digital data exchange between ships, between ships and shore, between shore and ships and between ship and satellite	
Tactical UHF	Telecommunications: • Support voice and data transfer services among mobile/fixed user terminals for military, institutional and civilian applications in UHF band	
5G, High speed internet, IoT	Telecommunications: • Laser communications between satellite and ground stations and between satellites of the same constellation	A specific constellation architecture shall be utilized to ensure very low latency and global coverage communications
GNSS from LEO	Navigation	A specific number (>50) of satellites is required to

		achieve at least 4 navigational signals contemporary in view from a defined user receiver based on satellites orbital data
Space situation awareness Space surveillance and tracking	A space-based radar and/or optical system to acquire an accurate overview of the space debris population.	

1.6 Propulsion system

As briefly introduced in §1.3.2, the major challenge when lowering the orbit is the increase of aerodynamic drag experienced by the satellites. Thus, in order to provide extended lifetime operations, the satellites shall be equipped with efficient propulsion systems and carry enough propellant. The main application of the on-board propulsion system is to offer the required drag compensation over the lifetime of the mission. Other applications of the propulsion system based on the mission requirements may include support of attitude control, orbital manoeuvres (between LEO or VLEO planes and/or collision avoidance), phasing manoeuvres and de-orbiting at the end of mission lifetime (not required for VLEO satellites). The efficiency of a propulsion system is a quite complex parameter to quantify since it incorporates several design and mission aspects. In general, the minimization of the propulsion system mass (including the whole propulsion system and the required propellant) to reliably fulfil the mission objectives can be a fair point to begin.

Equation 1 (Eq. 1) is known as the ideal rocket equation, or the Tsiolkovsky rocket equation after the Russian physicist who first derived it in 1903. It relates the change in velocity, or velocity increment ΔV of a spacecraft to the propellant ejection speed (V_e), the propellant mass (m_p) and the final mass (m_f, the satellite dry mass). The necessary ΔV to perform a manoeuvre or to accomplish a propulsion mission can be computed from orbital mechanics based on the type of operation required.

$$\Delta \mathrm{V} = \mathrm{V_e}\ \ln\left(1 + \frac{\mathrm{m_p}}{\mathrm{m_f}}\right) \qquad \textbf{Eq. 1}$$

An important parameter that defines the performance of a propulsion system is the specific impulse. The specific impulse I_{sp} is the impulse delivered per unit of propellant consumed or equivalent to the thrust produced per unit of propellant flow rate. It is conventionally measured as time (seconds). The specific impulse is a relevant figure of merit in the field of space propulsion as it is a measure of the efficiency of a thruster in terms of fuel consumption. As it is clearly presented in equation 2 (Eq. 2), the higher the specific impulse, the less propellant is needed to produce a given ΔV operation.

$$\mathrm{m_p} = \mathrm{m_f}\left(\mathrm{e}^{\left(\Delta \mathrm{V}/\mathrm{I_{sp}\,g}\right)} - 1\right) \qquad \textbf{Eq. 2}$$

Specific impulse offers the ability to compare among types of propulsion systems, but it does not give any straight indication of the time or the total impulse required to achieve that manoeuvre. Propulsion systems with high I_{sp} usually have low mass flow rates and consequently require a longer period to fulfil the same total impulse with a chemical propulsion system. Thus, satellite mission lifetime has an essential impact on the comparison of the propulsion systems.

Total impulse can give an overview of the amount of momentum change that a propulsion system can provide in total, but this is also dependent on the amount of propellant that the satellite can carry. So, when comparing propulsion systems in this way, the assumption that similar amount of propellant is carried shall be applied.

Impulse density is a measure of the total impulse per unit volume of the propellant. This allows for consideration of the volume limitations of small satellites. While no single performance parameter is ideal for analysing the propulsion systems, a combination of these performance parameters provides a good idea of the benefits and drawbacks for the various types of propulsion systems, [13].

Other parameters that are essential for the selection of the space propulsion system are the produced thrust (measure in mN) that gives an overview of the robustness of the system (how fast it can perform a specified manoeuvre) and the required power (only for electric propulsion systems, measured in W).

A good figure of the efficiency of a propulsion system can be defined as the fraction of the total source power that is transformed into kinetic power of the exhaust, often called jet power, as given by equation 3 (Eq. 3). P is the total input power, either released from energy stored in the chemical bonds of the propellant or supplied by an external power source in electric propulsion (EP), while $\dot{m}$ is the exhaust mass flow [14].

$$n \approx \frac{\frac{1}{2}\dot{m}\,V_e^2}{P} \qquad \textbf{Eq. 3}$$

A propulsion system usually includes the thruster or nozzle part, the propellant tank and the propellant feed system (valves, pipes, etc.). There are several types of propulsion systems that are utilized in small satellites, but the two major categories are the chemical propulsion and the electric propulsion as depicted in the following paragraphs. There are also some propellantless propulsion systems like solar sails, electrodynamic tethers, electric sails and magnetic sails that offer quite small quantities of thrust and are not so commonly utilized, or they are still in research phase.

In general, propulsion systems for small satellites must address the following challenges, [17]:

- <u>Volume and Mass</u>: Small satellites are volume-limited and subject to a stringent propellant mass requirement, and for higher ΔV missions this increases the incentive for high specific impulse propulsion systems.
- <u>ΔV</u>: The ΔV requirement is determined by the nature of the mission. Typical values range widely because small satellite mass and mission requirements can both vary significantly. Typical values range from 2m/s for satellite de-tumble and attitude adjustments to several km/s for orbit changes and deep space missions.
- <u>Low Electromagnetic Interference</u>: Due to the compactness of small satellites, all their subsystems are within close proximity to each other. Any electrostatic or electromagnetic interference caused by the electric thrusters may harm on-board electronic components, hindering the mission.
- <u>Thermal control</u>: Another issue associated with the compactness of small satellites is thermal control. Many electric thrusters are sources of large thermal loads which, if not properly dissipated, could cause satellite components to overheat.
- <u>Low power</u>: One of the major limitations for electric thrusters onboard small satellites is the low available electrical power due to the limited solar panel surface area of the satellite unless it has deployable solar panels, which can significantly

increase available surface area for solar cells. This of course comes at the cost of increased mass, cost, complexity and drag.

- Cost-Effectiveness: The general need for lowering cost of the small satellites puts a limit on the total cost of the propulsion system, which cannot be larger than the overall cost of the satellite. In this direction, propulsion systems should have a simple design and be assembled by low-cost components.

1.6.1 Chemical propulsion

In chemical propulsion systems, thrust is generated by acceleration of a compressed working fluid by expansion to a low-density exhaust stream with increased kinetic energy, typically using a specific nozzle geometry. Increasing the pressure and temperature of the working medium before expansion increases the resulting kinetic energy of the exhaust, and therefore the achieved specific impulse. Available systems are typically classified according to the principle of energy release in the working medium before acceleration, as briefly depicted below, [14].

- Cold Gas Propulsion: A high pressure working gas is expanded through a nozzle to create thrust. Typical propellants used are isobutane (C_4H_{10}), the refrigerants R236fa and R134a, and sulfur dioxide (SO_2). Typical specific impulse of those systems ranges from ~20 to ~80s and thrust levels from ~1mN to ~100mN.

- Monopropellant propulsion: A highly energetic propellant is typically decomposed catalytically or thermally into a high temperature working gas, before it is expanded through a nozzle to a low temperature and density exhaust stream with elevated exhaust velocity. This concept requires storable, and decomposable propellants, and commonly used fluids include hydrazine (N_2H_4) and derivatives, highly concentrated hydrogen peroxide (H_2O_2), and N_2O. Typical specific impulse ranges from ~100 to ~260s and thrust levels range from ~100mN up to more than 100N.

- Bipropellant propulsion: Combustion of an oxidizer and a fuel is applied to create a high-temperature, high-pressure gaseous mixture that can be expanded using a nozzle to create a high velocity exhaust stream. Such systems typically show highest performance in terms of specific impulse, but also come with most complexity due to typically two independent fluidic feed systems including two separate tanks and valve sets. Typically utilized, storable, noncryogenic propellant combinations, such as monomethylhydrazine or unsymmetrical dimethylhydrazine with oxidizers such as dinitrogen tetroxide (N_2O_4), MON-1 or MON-3, or less toxic combinations such as H_2O_2/kerosine, or H_2O_2/methane. Typical specific impulse reaches ~300s with thrust levels usually starting from ~10N.

- Solid Propulsion: By combusting a solid propellant, solid propulsion systems provide a hot working gas that is then expanded to produce thrust. Solid propulsion systems can be designed without complexity of moving actuators, but generally lack restarting capability and precise controllability, and have been considered as end-of-life deorbiting devices. Typical specific impulses for miniaturized solid motors range from ~150 to ~280s, with thrust levels ranging from tens to hundreds of newtons.

1.6.2 Electric propulsion

Electric propulsion is a technology that usually offers thrust generation with high exhaust velocities. Due to this main characteristic, compared to chemical propulsion methods, the result is a significant reduction of the propellant mass necessary for a given space mission. Reduced propellant mass can consequently decrease the launch mass of the spacecraft, leading to lower costs and smaller platforms. EP can also enable new missions that could not be feasible with the use of less efficient propulsion systems.

In this direction, the overall reduction in cost and mass budgets of an EP mission, has raised interest of commercial and institutional players in the last 10 years, so that today almost all LEO mega-constellations have an EP system onboard, and most of the GEO platforms rely on electric thrusters. Many space exploration missions also strongly benefit from the adoption of EP systems, as they are typically very demanding in terms of total ΔV and the propellant mass fraction would be too high if a conventional chemical propulsion system was used. LEO constellations, like the ones planned by OneWeb and SpaceX, are all equipped with EP systems. As of spring 2020, Starlink already has more than 400 satellites orbiting with krypton-fed HETs, while OneWeb satellites operate with low power xenon-fed HETs.

While in chemical propulsion systems the energy is generally stored within the molecular bonds of the propellants and is released by combustion, decomposition, or expansion, EP systems use an external energy source supplying electrical power (any source of electrical power, such as nuclear reactors or radioisotope thermoelectric generator can be used, but in most cases, especially in small satellites, electrical power is supplied by solar panels) that is used to generate thrust. Due to their working principle, they generally produce less thrust (generally up to tens of mN) and higher specific impulse (usually up to 3000s) compared to chemical propulsion systems. The future of electric propulsion is actively directed in two objectives: increasing the specific impulse and lifetime of high-power technologies and ameliorating the efficiency and reliability of low-power technologies.

There are three main types of EP systems based on the way the acceleration of the propellant is achieved:

- <u>Electrothermal (or thermal expansion)</u>: They use electrical power to heat a gas, which is then accelerated through a supersonic nozzle, transferring propellant enthalpy into kinetic energy. These types of propulsion systems include mainly resistojets and arcjets, while other not so common configurations include radio frequency (RF) heating, microcavity discharges and microwave heating, [13].
- <u>Electromagnetic:</u> They use a combination of electric and magnetic fields to accelerate plasma. These systems mainly include pulsed plasma thrusters (PPTs) and vacuum arc thrusters (VATs), while other options are thrusters utilizing a magnetic nozzle and magnetic plasma dynamic (MPD) thrusters, [13].
- <u>Electrostatic:</u> They accelerate charged particles, mostly ions, by electrical forces when falling through a potential drop across two electrodes. They often require a magnetic field to ionize propellant, while thrust is produced through electrostatic acceleration of the plasma. Electrostatic devices mainly include ion engines, Hall effect thrusters (HETs), electrospray thrusters and field emission electrostatic propulsion (FEEP), [14].

A nice overview of the main operating characteristics (thrust per power vs. specific impulse) of the three EP types is depicted in the following figure (Figure 2, [15]) showing the ability of each system to perform according to the required mission specifications.

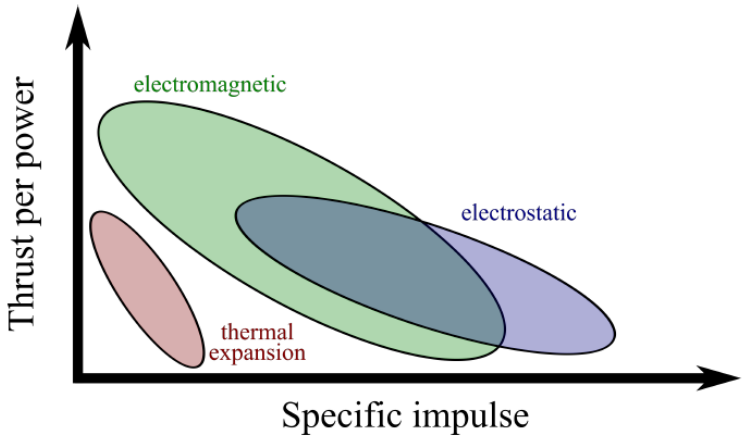

Figure 2: EP types overview in terms of thrust per power and specific impulse, [15]

A solid understanding of the overall performance of the propulsion systems is given in the following figure (Figure 3, [18]), where both chemical and electric propulsion systems are included showing the different operating characteristics of each subcategory. It is evident in this figure that EP systems present increased specific impulse and comparable thrust with chemical propulsion options, capturing the great potential of this technology for low power applications.

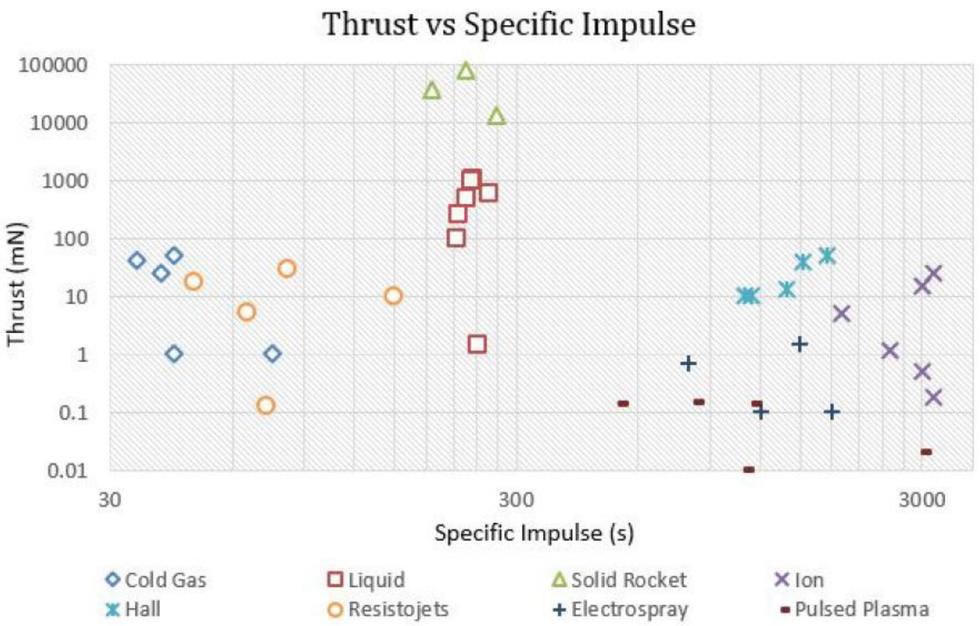

Figure 3: Propulsion systems Thrust vs. Specific Impulse overview, [18]

Another specific and innovative type of EP that is only applicable in VLEO is Atmosphere-Breathing Electric Propulsion (ABEP). ABEP is composed of two main components: the intake and the electric thruster. The ABEP system collects the residual atmospheric particles encountered by the satellite through the intake and uses them as propellant for the electric thruster. The system is theoretically applicable to any planetary body with an atmosphere and can drastically reduce the on-board propellant storage requirement while extending the mission's lifetime, [16]. ABEP can be combined with almost any type of EP system based on intake characteristics and it is mainly applicable in altitudes around 250km. Although it is a quite promising technology with a lot of research focusing on it recently, many challenges are still to transcend, mainly associated with the performance of the air intake, the thruster corrosion and the utilization efficiency of the propellant.

To sum up, several factors may change today's picture and could increase the use of EP on LEO satellites, [17]:

- Growing commercial incentives: In the past three years there has been a fast-growing commercial market for using LEO telecommunication satellite constellations to enable low-latency internet access to large parts of the world. These satellites usually require performing in quite demanding orbit raising injection manoeuvre. The associated ΔV required to perform the manoeuvre makes high specific impulse electric thrusters an attractive option for these satellite platforms, which is why companies such as SpaceX and OneWeb are developing their own EP-based platforms.
- Increased use of secondary launch opportunities: For many cases, even though launching as a secondary payload reduces the cost of launch, the satellite is usually not placed in the desired operational orbit, requiring the use of a higher performance on-board propulsion system for final orbit placement.
- Longer mission lifetime: The average LEO satellite lifetime is growing due to technological advances. Longer LEO missions will require additional propellant, making high specific impulse and long-life electric thrusters an attractive option.
- Deorbit requirement: To reduce the amount of space debris, many countries require that each LEO satellite be equipped with propulsion to enable the spacecraft to manoeuvre into a disposal orbit. The deorbit requirement increases the overall required ΔV for the satellite platform, again making electric thrusters more attractive for LEO missions.
- New manoeuvres: Low thrust high specific impulse capability allows for a variety of manoeuvres that have not been performed in the past. Altitude change, plane change, and phase change (to enable satellite servicing or re-positioning), drag compensation, and full attitude control (replacing the reaction wheels) are just a few of these manoeuvres, associated with the use of EP on future LEO missions.

1.7 Electric propulsion for small satellites

In the following table (Table 4), a summary overview of the main performance characteristics of the utilization of electric propulsion in small satellites (small size and compactness make them a special platform case as depicted in §1.2) is presented according to the major determinant factors identified.

Table 4: Electric propulsion for small satellites

Factor	EP performance
Volume and mass	Due to EP principles of operation and high specific impulse produced, it offers significant reduction in propellant requirement and overall volume and mass of the system.
Specific impulse	High, main advantage of EP.
Manoeuvrability	High accuracy with the ability of very small manoeuvre increments but quite longer manoeuvre times especially for high ΔV operations. New manoeuvres can be utilized, and new missions can be identified based on those advanced features.
Efficiency	Low in general at thruster level, typically decreases more at power levels below 100W. Due to the high specific impulse produced though this is overcome.

Mission flexibility	The ability of EP to operate at different thrust increments and with adjustable periods of time offer a flexible mission frame that can support on-mission deviations and extended lifetime.
Reliability	EP offers a reliable solution demonstrating several successful missions all over the years. The recent increased interest in this field creates an even more well defined and reliable environment for those systems.
Maturity/Qualification	With more than 50 years of research and development, EP can be considered a mature technology in general. Low power EP systems recent development though requires high life qualification time and extended testing to demonstrate their suitability for specific missions.
Thrust	Low, based on mission characteristics this may be a restricting factor.
Power required	For small satellites, the available power is generally more difficult to manage. Since EP systems require constant electric power to operate this may be a restricting factor.
Thermal control	Some EP systems may be sources of thermal loads that need special techniques to manage.
Cost	The general need for lowering cost sets a limit on the total cost of the propulsion system. In this frame, EP aims to have a simple design and be integrated by low-cost components. The general trend of “new space” for commercialization of the satellite production and the use of COTS components leads to a major cost reduction for EP systems that overcomes the need for extended qualification campaigns.
Electromagnetic Interference	Due to the compactness of small satellites, electrostatic or electromagnetic interference provoked by the EP systems may affect satellite electronics.
Propellant management	Based on the EP type, the need for pressurized propellants and specialized tanks and valves systems makes this part a critical challenge for the performance and safety of the EP systems.
Operability	With some of the EP technologies requiring high voltages and/or complex electronics to operate, this factor needs special attention and further investigation for the selection of the suitable EP system.
Lifetime	Lifetime of the EP systems may be a restricting factor due to the decay of thruster materials and mission requirements.

1.8 Thesis overview

This thesis aims to provide some fundamental information about the big outburst of the new space and the influence of this in the whole space mission analysis, especially for small satellite applications. In this direction, the primary concern of this study is to highlight the benefits of lowering the altitude to LEO and VLEO regimes and investigate the related mission challenges, identify the end-user applications of such missions and their related payloads and concentrate on the importance of the satellite propulsion system to achieve the mission objectives in this approach. An introductory presentation of the two main satellite propulsion categories, chemical and electric, is provided, focusing on the advantageous utilization of electric propulsion systems in (V)LEO applications and the growing perspectives they offer to the space market.

In this frame, the literature review (§2) aims to deeply examine the low power (up to ~200W) EP systems including their main operating principles and design characteristics. Moreover, a comprehensive presentation of the most common low power EP options is depicted along with multimode EP configurations. An overview of LEO (up to 600km) and VLEO micro and mini satellites missions, focusing mainly on EP systems application cases and the related mission analysis, is following, concluding with a critical analysis based on the literature review outcome to evaluate the lessons learnt through this review and highlight any possible research gaps that this study will contribute to fill.

The study continues with the mission analysis part (§3) where the system (satellite platform) requirements and specifications are determined along with the related system application (earth observation SAR). Based on those criteria all mission parameters are extracted and estimated to drive the design of the EP system. More specifically, the related orbits are selected (one VLEO and two LEO), the use (injection inaccuracies correction, End-of-Life de-orbiting and drag compensation) of the EP system is defined, the atmospheric characteristics (atmospheric density model and drag coefficient) are evaluated, the related ΔV calculations are carried out for the specified applications of the EP system and mass, power and thrust requirements are set.

Having set in the previous chapter the required EP system specifications envelope that fulfil the mission objectives for the three selected orbits, all available and suitable EP systems are then presented, evaluated and compared in §4. The main objective of this thorough trade-off analysis is to select the EP system that maximizes the payload mass and power fraction for the three orbits. Since, mass is not only affected by the mass of the thruster, but also from the mass of the required propellant, the specific impulse is also included in this analysis.

Based on the previous evaluation trade-off of the EP systems for the defined application, the EP system selected is conceptually described in §5. The EP system architecture is defined including all required subsystems and depicted in a functional block diagram. The subsystems are analysed, starting from the integrated thruster unit and its components, the propellant calculations and the related tank dimensioning and the specification of the power and control unit features. The EP system conceptual study is concluded with the summarized system, mission and operation characteristics.

Finally, this thesis is concluded in §6 highlighting the major results of this study and specifying future steps and considerations to expand the core findings of this conceptual approach of the electric propulsion system.

1.9 Novelty

It will be highlighted in the following literature review that there is a gap in the EP system analysis for satellites in the size envelope of 100-250kg and for specific missions' range in VLEO and lower LEO (up to 600km) regime, especially for the electric propulsion system dimensioning. In this direction, the main goal of this study is to try to cover this gap introducing a representative and already available electric propulsion system as a baseline scenario that will efficiently fulfil the main mission requirements.

In this direction, this is a conceptual study approach that fully specifies an optimized EP system that can cover the needs of three different orbit scenarios of a quite representative mission application that fits most of the current space market needs. The system is optimized in terms of payload mass and power fraction offering a useful tool to any satellite user to adopt this system and easily adjust it to any (V)LEO mission needs. It actually constitutes a possible plug and play option for applications in this specified operating envelope, easy to integrate and configure, leading to a faster, cheaper, efficient and more reliable EP system integration to any small satellite. This becomes the most important and novel characteristic of this study, with this concept-like focus on the EP system being the core of the whole analysis.

2. LITERATURE REVIEW

This chapter contains a summary of the literature relevant to the Master thesis subject and objectives. Firstly (in §2.1), a general overview of low power (up to ~200W) EP systems is provided including their main operating principles and design characteristics. Next, in the same section, a more comprehensive presentation of the most common low power EP options is depicted along with multimode (shared propellant or hardware) EP configurations. The next section (§2.2) provides an overview of LEO (up to 600km) and VLEO micro and mini satellites missions, focusing mainly on EP systems application cases and the related mission analysis. Finally, in section 2.3, a critical analysis based on the literature review outcome is presented, aiming to evaluate the lessons learnt through this review and highlight any possible research gaps that this study will contribute to fill.

2.1 Low power EP systems overview

The use of EP systems on satellites for commercial, defence and space science missions has been increasing in recent decades, from the first successful operation back in ‘60s to the present day. The literature includes a variety of research papers on the use, development and enhancement of EP technologies over the years. Recent research interest focuses more on the utilization of EP for specific applications such as LEO satellites or small spacecrafts that require special design characteristics especially in terms of size and power. For this thesis, the main focus is found on low power (up to ~200W) EP systems for LEO and VLEO micro/mini satellite (up to ~200kg) missions.

In this frame, [17] provides an overview of the technological and commercial development of EP systems that have been deployed up to date. A key constraint imposed on LEO satellite by EP systems is limited power. Unlike GEO communication satellites, LEO satellites are small, lower power platforms that typically have commensurately small, lower power payloads. Since these smaller LEO platforms are only capable of generating hundreds of watts, due to the limited size of their solar panels and only partial exposure to the sun arising from longer fractions of each orbit spent in the Earth's shadow, they can usually only accommodate low power electric propulsion systems requiring no more than a few hundred watts. Given this propulsion system framework, seven technology subclasses of electric thrusters have been used in LEO since 1981: (1) Electrospray thrusters, (2) resistojets, (3) arcjets, (4) hollow cathode thrusters, (5) PPTs, (6) ion thrusters and (7) Hall thrusters. These seven electric thruster technologies have been incorporated onboard a total of 167 LEO satellites weighing over 50kg. According to the breakdown of EP presented in this study, it is evident that resistojet technology is the most prolific and powers 74% of all EP based LEO missions, followed by Hall thrusters (15%). It is also evident that in the past decade Hall thrusters have become more prominent powering over 40% of all EP based LEO satellites for this time period. Especially in missions where high manoeuvrability is required, Hall thrusters are utilized because of their relatively high performance.

Several studies over the last 15 years are focusing on small satellites EP systems either surveying specific options or presenting their design characteristics and working principles.

In this direction, [18] provides a thorough overview of low power propulsion technologies that have been developed or are currently being developed. The operating principles and key design considerations for each class of propulsion system are outlined. Especially for EP, the following systems are analytically presented and constitute the main low power EP options:

- Resistojets: In this type of EP, the propellant is passed through a heat exchanger (or heating element) where it is super-heated and ejected through an expansion nozzle. Because of the propellant's high energy (gained by heating), an exhaust velocity much greater than that of a cold gas propulsion system is achieved in a resistojet. A major drawback of resistojets is that their performance (thrust, I_{sp}) is limited by the melting temperature of the heating element used. In addition, power and thermal losses during heating of the element contribute to the inefficiency of resistojets. Resistojets with a variety of propellants (R134a, R236fa, SO_2, Xe, even water) have been used on larger satellites, and like any other systems with liquid propellants, they have experienced issues due to sloshing within the tanks. They generally provide low thrust and low specific impulse (hundreds of s) compared to other EP systems, but they require low power and present very low system complexity.

- Radio-Frequency Ion Thrusters (RITs): RITs belong to a subset of gridded ion thrusters that generate thrust by accelerating the ionized propellant (plasma) through an electrostatic grid. The stored propellant (usually Xe, Kr or Iodine) is let into the discharge chamber where it is ionized (and becomes plasma) by means of RF power (from RF coils). The plasma is then extracted (from the discharge chamber) and accelerated by a series of grids (ion optics) called screen and accelerator grids. The screen grid extracts propellant cations from the ionized plasma and directs them downstream towards the accelerating grid. A neutralizer cathode, present on the exterior of the thruster in all ion engines, provides electrons to neutralize the ionized propellant that is emitted from the thruster. The specific impulse of a gridded thruster can be varied by changing the voltage that is applied to the accelerating grids. Ion thrusters are characterized by high thruster efficiency (60% to >80%) resulting in high specific impulse (from 2000s to over 10000s); however, they have been plagued with issues that are caused by cathode wear and contamination over prolonged usage. Contamination effects incurred can be mitigated by the use of inert propellants like xenon and krypton, however, it still leaves out the issues due to the plasma interactions.

- Hall Effect Thrusters: They are electrostatic devices that generate thrust by first ionizing and then accelerating the propellant in mutually perpendicular electric and magnetic fields. These thrusters work on the principle of the well-known Hall Effect: when electric current is applied to a conductive material (propellant) placed in mutually perpendicular electric and magnetic fields, a potential difference is developed that is perpendicular to the applied electric and magnetic fields. The applied magnetic field (either produced by permanent magnets or by electrically stimulated magnetic coils) is radial, while the accelerating electric field (acting from anode towards cathode) is axial. Note that, unlike gridded ion thrusters, HETs do not have the grid system, instead the grids are replaced with a strong magnetic field perpendicular to the flow of ions. This magnetic field reduces the mobility of electrons coming from the external cathode, thereby restraining their flow towards anode in the accelerating electric field. HETs have many advantageous features like high specific impulse (up to 2000s, higher than most systems except ion engines), higher thrust density and simplicity in design (when compared to gridded ion engines due to lack of accelerator grids). However, they also face some challenges with erosion of magnetic circuitry due to discharge plasma and lower efficiency (6-30% at 0.1-0.2kW and 50% at 1kW). Similar to ion engines, the Hall thrusters make use of heavy elements as propellants, for instance, Xe, Kr, iodine and Ar. Of these, xenon has been favoured for its lower ionization energy, higher atomic mass and easy storage. A neutralizer cathode in thruster output is also necessary for HETs similarly to RITs. Lifetime of a Hall thruster is mainly limited by the erosion of the components protecting its magnetic circuitry from discharged plasma (ionized propellant). Wall erosion is mainly caused when the ions are driven towards the wall material because of elevated parallel component of electric field and the high electron temperature. Recently, a new technique called Magnetic Shielding was proposed that could potentially eliminate wall erosion in Hall thrusters. It is to be noted that, the magnetic and electric fields that are supposed to be mutually perpendicular are not so when under electron pressure. When the walls are magnetically shielded, the electric field component parallel to the wall is nearly eliminated, resulting in the decrease of ion bombardment on the walls.

- <u>Electrospray Thrusters</u>: They are plasma-free electric propulsion systems that work on the principle of electrostatic extraction and acceleration of charged particles (ions) from a liquid (propellant) surface to produce thrust. Their fundamental working mechanism is based on a process by which the conductive liquid surface of the propellant is deformed into a sharp cone-shaped meniscus called Taylor Cone; when a certain threshold of the electric potential is surpassed, ions are extracted from the cone’s apex. Electrospray thrusters accelerate positive or negative ions, respectively generating either a positive or negative ion beams thereby eliminating the need for an external cathode to neutralize the ejected ions unlike in plasma propulsion devices (ion and Hall thrusters) where an external cathode is essential. The propellants used for electrospray thrusters are usually ionic liquids, and their negligible vapor pressure serves as an advantage by resolving the need for propellant pressurization and helps with system miniaturization. In an electrospray thruster, typically, the extraction of charged particles is done through two regimes: the cone-jet regime, in which the meniscus (of the propellant) breaks up into droplets; and the ionic regime where pure ions are extracted. The specific impulse observed in ionic regime is greater than in cone-jet regime. A thruster is typically designed to operate in only one of the two regimes, and the regime defines the specific thruster: either a colloid thruster or a field emission thruster. Electrospray thrusters use ionic liquids as propellants as they do not require heating, have low operating voltage, high conductivity in the pure state and negligible vapor pressure. An individual electrospray emitter operates in the milli-watt (power) and generates thrust in the order of micro-newtons; therefore, an array of emitters is required to form the thruster that can yield the desired thrust.

- <u>Pulsed Plasma Thrusters</u>: They operate by creating a pulsed, high-current discharge across the exposed surface of a solid insulator (for instance, Teflon) that serves as a propellant. The arc discharge ablates (sublimates/vaporizes) the propellant material from its surface, thereby ionizing and accelerating the propellant to high speeds. A current pulse lasting few micro-seconds is generally driven by a capacitor that is charged and discharged approximately once every second. The advantages of a PPT are its ability to provide small impulse bits for precision manoeuvring, robustness by programming impulse bits to cater to mission needs, design simplicity owing to the ability of using wide variety of propellants (solid/liquid), and its ability to maintain constant specific impulse and efficiency over a wide range of input power levels. However, these advantages come at the cost of issues that result due to electrode erosion, presence of macro-particles in the plume due to non-uniform ablation, very low thruster efficiency (10-20%) and low thrust levels.

For the aforementioned EP systems, two major comparisons are summarized: first, power and specific impulse; second, thrust-to-power ratio and specific impulse. Important conclusions are recorded through this analysis:

- Despite of their average power consumption being slightly higher than that of electrospray and pulsed plasma systems, resistojets provide a specific impulse an order of magnitude lesser. Resistojets also have the highest average thrust-to-power ratio amongst all the surveyed electric propulsion systems.
- It can be observed that electrospray, radio frequency ion, Hall and pulsed plasma thrusters provide similar performance.
- RF ion thrusters feature the highest specific impulse due to their high operating efficiency (approximately 70%) but offer a lower average thrust-to-power ratio.
- Hall thrusters on the other hand have a lower I_{sp} than RF ion thrusters because of their lower thruster efficiency, however they possess higher average thrust-to-power ratio.
- Though RF ion and Hall thrusters have the highest specific impulse amongst electric engines, they consume a considerably larger amount of power mainly owing to complex design systems that include the series of grids, external cathode and RF ion power for RF ion thrusters; external cathode, induced magnetic and accelerating electric fields for Hall thrusters.
- Electrospray thrusters have a specific impulse higher than PPTs and lower than both RF ion and Hall thrusters. They have a relatively lower specific impulse than Hall and RF ion thrusters because of the formation of larger clusters of droplets during the extraction of ions.
- The average thrust-to-power of the surveyed electrospray thrusters is found to be similar to that of RF ion thrusters.
- PPTs have a relatively lower thrust and specific impulse amongst the electric propulsion systems due to their very low thruster efficiency (10-20%). PPTs like electrospray systems require power of an order of magnitude lower than Hall and RF ion thrusters because of the relatively simpler design that involves the generation of an arc to ablate the propellant. They also have the lowest average thrust-to-power ratio amongst all surveyed electric engines.

Other studies in the same frame include:

A Review of High Thrust, High Delta-V Options for Microsatellite Missions presented in [19]. This paper provides a brief overview of propulsion technologies (chemical and electric) currently (back in 2009) available for microsatellites, and an evaluation of each technology for potential use in a demanding mission. The sample mission is that of a microsatellite inspector which, starting in a 200km parking orbit, must be diverted to rendezvous with another satellite in orbit at a different altitude and inclination. It is found that existing bipropellant microrocket designs provide a high thrust value, combined with a 300 s specific impulse, allowing for response times of only a few hours for such an inspector mission with ΔV requirements over 1km/s. Miniaturized electrostatic thrusters provide the largest ultimate ΔV capability, approaching 10km/s, but with a very low thrust level and therefore a response time capability of several months.

Comparisons between different EP concepts depicted in [20]. In this paper, an overview of the main EP concepts (arcjets, HETs, gridded ion engines and MPD thruster) is presented as recorded back in 2011, for a wide range of power, thrust and specific impulse.

Electric propulsion for small satellites analytically presented in [21]. This paper mainly focuses on very low thrust (in μN level) and power EP systems. Several modified configurations (miniaturizations) of the known EP systems are presented including ablative micro-PPT, micro-VAT, micro-laser plasma thruster and micro-cathode arc thruster.

Propulsion options for CubeSats analytically presented in [13]. This survey (carried out in 2017) includes propulsion systems that have been designed specifically for CubeSat platforms, and systems that fit within CubeSat constraints but were developed for other platforms. Throughout the survey, discussion of flight heritage and results of the mission are included where publicly released information and data have been made available. Major categories of propulsion systems that are in this survey are solar sails, cold gas propulsion, electric propulsion and chemical propulsion systems.

The most updated review of the EP systems available for small satellites is provided in [22]. This study reviewed electrostatic, electrothermal and electromagnetic propulsion methods based on state-of-the-art research and the current knowledge base. Performance metrics by which these space propulsion systems can be evaluated are presented. The article outlines some of the existing limitations and shortcomings of current electric propulsion thruster systems and technologies. Moreover, the discussion contributes to the discourse by identifying potential research avenues to improve and advance electric propulsion systems for small satellites. Except the already presented low power options of EP systems, some more details are provided for:

- <u>Electrospray Thrusters</u>: Electrospray thrusters operate in either droplet or ion emission mode. They generally come in three different variations which are characterised by the propellant being used. This largely determines if a neutralizer is needed to keep the spacecraft charge neutral. The neutralizer emits electrons into the exhaust to neutralise the particles. Colloid based thrusters typically emit larger charged droplets (droplet mode) using a propellant that is charge neutral. Some colloid thrusters may also require a neutralizer if the propellant is not doped with a salt which increases electrical conductivity. FEEP thrusters typically emit individual ions (ion emission mode) and require a neutralizer as they operate with a liquid metal. FFEP thrusters offer high thrust precision but low thrust forces (<1mN) and a wide range of specific impulses. The final type of electrospray thruster is Ionic Liquid Ion Source (ILIS) which does not need a neutraliser as it only uses molten salts as propellant.

- <u>Gridded Ion Thrusters (GITs)</u>: A GIT produces ions by bombarding a propellant with a high energy electron beam created either by a direct current (DC) discharge, an RF discharge or a microwave discharge. The ions are then ejected through a series of electrically charged grids. A potential difference is established between these grids, one a screening grid and another an accelerating grid. This potential difference is what determines the acceleration of the propellant. The negatively charged ions created are accelerated by the cathode grid (accelerator). The most common type of propellant is Xe though earlier versions of this thruster used metallic propellants such as mercury or cesium which have high atomic masses, ionize easily but have very high boiling points and are toxic chemicals. Xe, in comparison to cesium and mercury, ionises more easily, has a high atomic mass and critically, and it has a low boiling point making it more favourable. Ion thrusters have the highest efficiency in comparison to other propulsion methods and very high specific impulses.

- <u>Arcjets</u>: A constant current is passed through two electrodes of opposite polarity at either end of a constricting tube to induce a sustained electric arc, this heats the propellant to exit through a diverging nozzle at high velocity. Typically, the first part of the nozzle and the nozzle constrictor is the anode, mounted to a co-axial tube at the end of which is a cathode rod, the electrodes are separated by a high temperature insulator such as boron nitride or aluminium oxide. The arc is ignited by a high voltage, usually 1000-4000V and then dips to a low operating mode. Arcjets typically have four power levels ranging from very low, 100-300W to high power, up to 200 kW. The most suitable power range for smallsats is at the lower end of the power scale, within the 100W-1kW range arcjets. Since the arc can generate significantly higher temperatures in comparison to a heating coil, the specific impulse is usually greater than that of a resistojet and while similar to chemical thrusters, the arcjets also usually have 2-3 times higher specific impulses than chemical rockets. On the other hand, arcjets have low efficiency and a lot of heat loss, while they also typically require complex power processing units (PPUs) due to the presence of high voltage.

One of the major challenges identified for electric propulsion in this study in comparison to chemical propulsion is the duration of time that it takes for the system to reach high thrust, often taking a much longer time in comparison to minutes or seconds for chemical systems. This limits electric propulsion systems to very specific in-space applications such as station keeping, collision avoidance etc.

Moreover, some interesting statistics are provided, showing that traditionally, 62% of operational missions used resistojets with 34% choosing hall thrusters. Currently, no single thruster can achieve a wide array of manoeuvrability, specific impulse, continuous acceleration, lifetime and adequate thrust efficiency and sensitivity for a range of small satellite requirements. For example, EP systems proposed for station keeping typically include PPTs, resistojets, arcjets and electrosprays. Station keeping will be an important parameter for future small satellites as more enter orbit but even more so will be collision avoidance considering space debris. Collision avoidance would typically require high delta velocities depending on the notice period before a collision occurs. This would favour those EP systems that can therefore provide higher thrust forces, although if sufficient notice is given, most EP systems could avoid the collision. Orbit raising will require high thrust and long lifetime EP systems. Considering the mass and power budgets of small satellites most EP systems are not currently suited to orbit raising although they are gradually becoming more capable in this direction.

Some variants of the established technologies of EP systems along with some novel approaches and propellant selection considerations are studied in [22]. The main EP technologies are initially presented with their principles of operation, possible applications and basic mission calculations. Special focus is concentrated on HETs, where the main design challenges are identified including discharge oscillations, electron transport and plasma-wall interactions. In this frame, variants of HET are presented such as, thruster with anode layer, cylindrical Hall thruster, diverging cusped-field Hall thruster, magnetic shielding and wall-less Hall thruster. Other advanced EP concepts include negative ion thrusters, electrodeless plasma thrusters, RF and microwave plasma thrusters.

Finally, a review of the available propellant options is carried out highlighting that it determines to a large extent the thrust efficiency and the level of specific impulse. In addition, nature, storage feasibility, flow control and injection of the propellant have a major impact on the complexity and the overall cost of a spacecraft propulsion assembly as a whole. Selection of the right propellant is therefore a critical step in the development of an EP system. Xenon is currently the propellant of choice for various electric thrusters, including HETs and ion engines, whatever the spacecraft type. Although xenon has several advantages, it also has many disadvantages that might force the community to consider alternative propellant options as both the number of available technologies and the diversity of vehicles, missions and manoeuvres are growing.

A suitable propellant combines a low ionization threshold with a high ionization cross-section to minimize the amount of energy necessary to create a high-density plasma. Molecular propellants must generally be avoided as part of the available input power is lost into dissociation in smaller fragments and into excitation of internal vibration and rotation modes. The iodine molecule (I_2) might be an exception as it is attractive in many other ways. Some technical and practical aspects must also be accounted for when analysing propellant options. The propellant must be easy to store in order to reduce the complexity of the tank and associated sub-systems without changes in the power processing unit. Liquid and solid propellants bring here an obvious benefit as gaseous atomic propellants like Xe and Kr are stored in the supercritical fluid state, which necessitates large high-pressure (200-300 bars) tanks. Condensable propellants, on the other hand, introduce specific requirements since the mass flow system and the gas feeding system are more complicated to operate. A liquid/solid compound with a high vapor pressure and with low melting and boiling temperatures is preferred, as any power that is used for evaporation and to maintain the temperature of the transfer line and injection system reduces the overall efficiency of the thruster assembly. Another relevant point is that the propellant must be non-radioactive (e.g. radon), non-toxic and easy to handle. Contamination may also be a critical issue for thruster elements as well as for spacecraft elements like solar arrays and radiators. According to the aforementioned list of criteria for propellant selection in the field of EP, there are three attractive candidates for an alternative to xenon. Krypton generates a high specific impulse and no significant modifications of the thruster assembly are required. It is also a non-toxic propellant. However, the thrust efficiency is lower due to a poor ionization degree and its storage density is below that of xenon. Bismuth is a low-cost propellant that offers a large thrust-to-power ratio due to its high mass and high ionization efficiency. In addition, its storage density is six times that of xenon. The main drawback arises from the fact that a high temperature is required to prevent condensation. The thrust efficiency achieved with iodine is similar to that obtained with xenon. Although the ionization energy of I_2 is low and the cross-section large, energy is lost into dissociation and excitation and the average mass is close to the I atom mass. But an iodine-vapor feeding system can be operated at relatively low temperature. Moreover, the storage density of I_2 is three times higher compared to xenon and a high purity grade is relatively inexpensive. The disadvantage lies in the fact that iodine is a reactive compound and compatibility with the thruster and space vehicle must be guaranteed. In conclusion, one must keep in mind that there is no ideal propellant. The selection results from a trade-off between various criteria, evaluating pros and cons of each option. Furthermore, the final choice also depends upon the spacecraft design and the mission objective and duration.

A thorough review of possible propulsion options for very low Earth orbit microsatellites is carried out in [24]. In this study, VLEO mission characteristics are presented and calculated mainly focusing on station keeping application for the propulsion system and identifying four main mission scenarios in the 250-500km altitude range: C1 (100kg, 100W) and high solar and geomagnetic activity (HA), C1 and low solar and geomagnetic activity (LA), C2 and HA, C2 and LA. In this frame, several chemical (cold gas, monopropellant) and electric (resistojets, HET, RF ion) propulsion systems are presented that can fulfil the required mission based on several design assumptions. The main parameters that are evaluated in this analysis are: the mass fraction, the volume fraction and the power fraction showing mixed results based on the propulsion system, the altitude and the mission scenario. Finally, two earth observation case studies (C1, 275km, 4.5 years, LA and C2, 370km, 2 years, HA) are assessed calculating the required propulsion system characteristics. Based on the analysis the most suitable option for the first case study is HET or RF ion thruster, while for the second one liquid monopropellant or FEEP thruster.

Space Propulsion Technology for Small Spacecraft studied in [14]. The purpose of this review is to describe the working principles of space propulsion technologies proposed so far (2018) for small spacecraft including chemical and electric propulsion options, while also some propellantless concepts (solar sails, magnetic sails, electromagnetic tethers) are presented. Given the size, mass, power, and operational constraints of small satellites, not all types of propulsion can be used and very few have seen actual implementation in space. Emphasis is given in those strategies that have the potential of miniaturization to be used in all classes of vehicles, down to the popular 1-L, 1-kg CubeSats and smaller.

Except already presented types of EP, VATs are added in this study. VATs are similar to PPTs in terms of mechanical design but initiate a lower power discharge that ablates anode material and have been specifically developed for low power nanosatellites.

Moreover, interesting metrics of commercial propulsion systems for small satellites are provided confirming the already expected better performance of EP systems compared to chemical propulsion systems and showing a mixed result among the EP systems. It becomes evident that the selection of the appropriate EP system combines a set of parameters and it is strongly connected with the mission characteristics and the related EP applications. One important aspect presented in this study is correlated with the flight heritage those systems carry up to now that may be a critical one in the final selection of the system.

Finally, [25] surveys and provides a performance comparison for some promising microsatellite propulsion technologies. Two recently developed propulsion technologies, green monopropellants and ion electrospray, show great promise for increasing the manoeuvrability of severely volume and power constrained microsatellites.

Following Figure 3 performance overview of the available EP systems, the main research studies for small satellites EP systems for LEO missions (medium thrust usually required in the range of about 1-20mN, the higher possible specific impulse and low power in the range 100-300W) are recently concentrating on:

HET

Several studies over the years are focusing on HETs and their variants, and due to their improved performance characteristics, they show a boosted research interest recently.

More specifically, in [26] a design of a low power HET is analytically presented. A low-power HET and a corresponding low-current hollow cathode were designed. The ceramic-walled low-power thruster is designed for a power range of 50-150W, uses a coil for generating the magnetic field, and has an outer discharge chamber diameter of 23mm. The thruster has been manufactured and is ready for the operation with the low current hollow cathode. The hollow cathode has a classical structure and is designed for currents of about 1A.

In [27], VENμS, a novel technological mission using EP is analysed. VENμS is a recently launched satellite, for super-spectral earth imaging and EP system demonstration. The system includes a novel design, developed, qualified, manufactured and integrated by Rafael and during this mission, system enhancement capabilities and its space performance will be characterized. The system and the satellite were carefully designed to comply with the mission goals and constraints. VENμS mission duration is approximately four years, during which the satellite will operate in two major orbits. The first one with a 30-month duration is at an altitude of 720km in SSO, while the second one at 410km SSO for 12 months. The EP system will be used to control the orbit of the satellite in the first orbit, perform the orbit transfer required and control the orbit in a high drag environment present in 410km altitude. HET-300 thruster is used to fulfil mission objectives utilizing Xenon as propellant and operating nominally at 300W power. However, its useful range of operation is between 250-600W. Its nominal thrust is around 14.3mN and the specific impulse 1210s.

In [28], the scaling relations for a sub-kilowatt HET were derived in the form of linear equations from the physical relationships and the approaches found in previous studies. In this direction, a collection of the available sub-kilowatt HET (50-660W, 3.9-40mN, 839-1593s) is presented including their main design, performance and physical characteristics. Because the anode power and voltage are major constraints in thruster design, the equations were given as a function of those parameters. A low-power HET that consumes 360W and applies 300V at the anode was designed based on the proposed relationship.

In [29] and [30], Sitael's HT100 HET development process is analytically presented. HT100 in-orbit validation through μHETSat mission is one of the main steps of the development of this thruster. For a significant validation process, the thruster is expected to fire over 1000 hours, bringing the spacecraft from an initial altitude of 525km to a target altitude of 425km and back. After cumulating about nearly 1000 hours of operation through nine up and down manoeuvres, the satellite altitude will be lowered down to 350km and there it is going to stay for several months by means of the HT100 (fired for 30 minutes every 'n' days, where 'n' is strictly dependent of the atmospheric density at that altitude, which is in turn dependent on the solar activity). The whole mission can be completed with about 3kg of Xe, although 4.3kg will be loaded in the tank to keep an adequate margin. The thruster unit consists of one thruster (HT100) and two hollow cathodes. It can be operated in a power range between 120 and 300W, with a peak efficiency exceeding 35%, a maximum specific impulse of about 1300s and a maximum thrust of 9mN. With a design based on permanent magnets and a total mass lower than 450g, HT100 is also the most lightweight Hall thruster of this class. In general, the thruster unit consists of the HET, the hollow cathode for the electron generation, the propellant management assembly and the PPU aimed to provide the required power at the subunits. Moreover, a full ground qualification process is ongoing to demonstrate the potential operation and design capabilities of this thruster unit. The thruster, tested both with xenon and krypton and operating at power levels that can be as low as 100W, is a cornerstone for future small satellites missions that will require a significant total ΔV and have tight constraints on maximum propellant mass. HT100 has been developed in parallel with a dedicated feeding system and power processing and control units, which have also been extensively tested both in standalone mode and coupled with the thruster unit. The qualification test campaigns can be broadly divided in three main segments: thruster performance verification, coupling tests with the other relevant subsystems and environmental tests.

In [31], Orbion Space Technology's Aurora HET thrust (19mN at 300W, 5.7mN at 100W), specific-impulse (1400s at 300W, 950s at 100W) and total efficiency (41% at 300W, 24% at 100W) are reported based on ground measurements using xenon propellants. The Aurora system is a 100-300W, magnetically-shielded HET. Magnetically-shielded HETs show vastly superior lifetimes compared to traditional HETs, but typically suffer by reduced performance as compared to non-shielded thrusters, especially in the lower power classes. The data presented here show that the Aurora thruster has performance on-par with best-in-class, non-magnetically-shielded, low-power HETs.

In [32], a review of low-power EP Research at the Space Propulsion Centre Singapore is reported. The research is focusing on a variety of EP subunits, including low power HETs (25-200W), cathode development, vacuum testing and plasma diagnostics. The first class of HET developed at the research centre is a 200W-class Hall thruster. The thruster has a classical design with a ceramic discharge channel and is operated with an external thermionic electron emitter. The final iteration of the thruster employed electromagnetic coils for both the internal and external magnetic circuit. The upgraded thruster with the new generation cathode and a picture of the thruster plume is presented in Figure 4. The thruster can stably fire at discharge potentials from 110 to 270V, leading to an anode power from 50 to 220W for Xe. When operating with Xe and at 200W of discharge power, the thruster can yield a thrust of 9.34mN and a specific impulse of 1729s, while the thrust efficiency is 39.3%. At 100W, when operating with Xe, the thruster produces a thrust of 5mN with a specific impulse of 940 s, at a thrust efficiency of 23.4%.

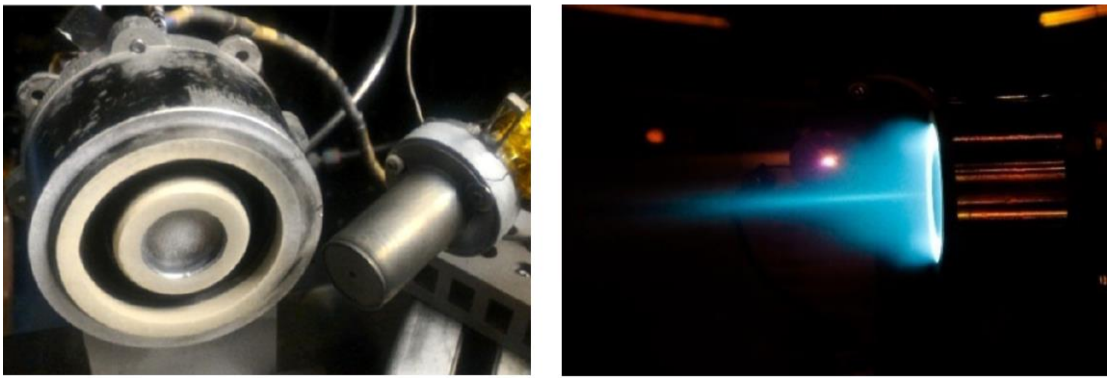

Figure 4: HET design and plasma plume operating with Xe at 200 W, [32]

In [33], the development of advanced low-power Hall thrusters by EDB Fakel, in co-engineering with Airbus Defence and Space Toulouse is depicted with the flight-proven SPT-50 thruster. To guarantee the requested operational lifetime, innovative magnetic field topology and discharge chamber material have been implemented. The main parameters of this system are a power level of 220W with a thrust of 14mN, specific impulse of 860s and total efficiency of 26%.

In [34] and [35], the development of low power HET SPT-25 and SPT-40 by Space Electric Thruster Systems (SETS) is presented. SPT-25 thruster shows a range of power 150-200W, thrust 7-11mN, specific impulse <1200s and a total efficiency of 30%. SPT-40 shows a range of power 200-400W, thrust 10-20mN, specific impulse 1200s and a total efficiency of 38%.

In [36], an overview of the EP systems development of Busek is presented. Regarding the HET, several options have been developed with different power and thrust levels. The ones that lie inside the required thrust envelope are BHT-100 (100-150W, 6.3mN, 1200s) and BHT-200 (200W, 13mN, 1390s) both operating with Xe, but Iodine compatible versions have also been developed.

Finally, in [37], CAM200 low power HET recent development is presented. Being part of several validation projects, this thruster performs in the 100-300W regime, with a thrust of 6-14mN and specific impulse between 900s and 1500s, while the efficiency reaches 43% in the highest power operating point.

RIT

Many research studies are concentrated on gridded Ion thrusters and their variants, especially in RITs, a quite mature and promising technology.

In [38], Ariane Group Radio Frequency Ion Thruster "RIT" family is presented. It consists of three members: RIT-μX a miniaturized thruster system for the Micro- and Milli-Newton thrust regime, RIT 10 EVO (5-25mN) a thruster derived from the flight proven RIT 10, and the new RIT 2X Series systems capable to deliver more than 200mN thrust per engine. The thrusters are embedded in a system. Regarding the RIT 10 EVO the main operating points include 15mN/3000s at 435W and 5mN/1900s at 145W with Xe as propellant. The main parts of a RIT based EP System are: the Neutraliser aimed to neutralise the expelled positive charged ions, the RF Generator which converts the DC current into the required AC current for the RF coil inside the thruster, the management of the propellant flow and the PPU since the thruster needs one positive and one negative high voltage for the grid system and an alternating current for the thruster's ionization coil. In fact, the PPU has to provide all voltages required by the electric propulsion sub-system.

In [39], the activities carried out by Research Institute of Applied Mechanics and Electrodynamics of the Moscow Aviation Institute in the field of low power ion thrusters in 2010-2020 are presented. The performance of RIT-8 laboratory model at different power levels is presented showing 3.4mN of thrust, 1930s specific impulse and 28% efficiency at 116W, 5.6mN, 2590s and 36% at 195W and 8.9mN, 3790s and 53% at 310W respectively.

In [40], L-3 Electron Technologies Inc. 8-cm Xenon Ion Propulsion System (XIPS©, Figure 5) is presented. Based on the engineering experience gained from the development, testing and flight heritage of its product line, L-3 ETI has designed this thruster intended for small satellite or low power applications. The main design characteristics include a power range 100-350W, a thrust range 2-14mN, a specific impulse between 2000-3000s and total efficiency reaching 55%.

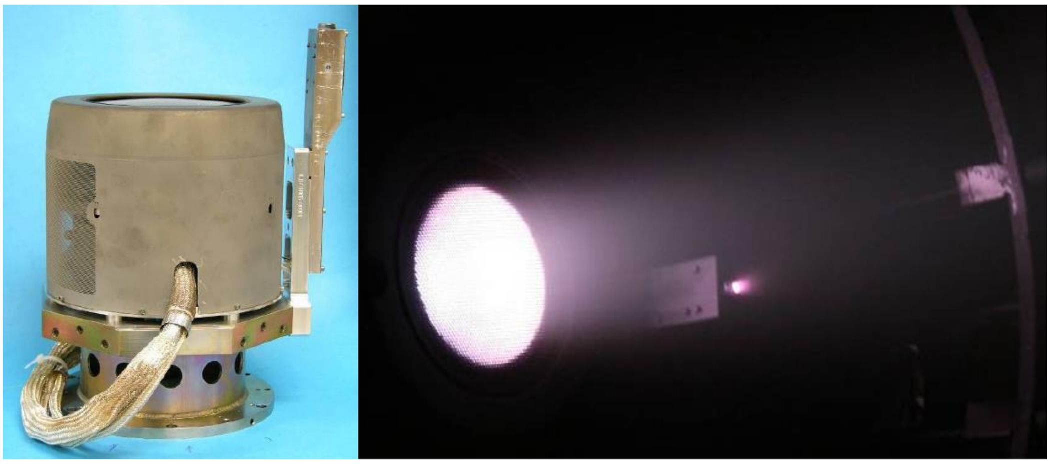

Figure 5: RIT design and plasma plume operating with Xe, [40]

In [41], REGULUS propulsion system is presented. This system integrates a Magnetically Enhanced Thruster and its subsystems (i.e., fluidic line, electronics, and thermo-structural components) in a 2U envelope of weight lower than 3kg. This type of thruster is an RF cathode-less thruster capable of providing thrust in the range 300μN to 900μN, with maximum specific impulse of 900s and maximum operation power of 60W. Despite its low thrust capability that better fits in CubeSats, the easy scalability of the system makes it a possible option also for bigger satellites. Moreover, this thruster can operate with different types of propellants, such as Iodine, Xe and Ar.

Finally, in [36], the overview of the EP systems development of Busek also includes a RIT option. The fully integrated BIT-3 propulsion system represents a significant technological achievement as it is one of the first solid iodine-fuelled, flight-ready EP systems with maximum power of 80W, maximum thrust of 1.27mN and specific impulse of 2290s.

Electrospray

Electrospray (and FEEP) technology thrusters belong at the low end of the required thrust level, but the good performance characteristics and the scalability of such systems make them an interesting and attractive research area over the years.

In [42], some iconic liquid electrospray options are studied showing their operating principles and concentrating on their scalability applying a modular configuration based on the needs of the application. The basis can be S-iEPS Propulsion module performing with 3840 emitter tips and providing 100μN of thrust and 1150s of specific impulse.

In [43], the TILE electrospray emitters are presented consisting are the basis of Accion TILE systems that offer high modularity and configurability. The TILE-3 module presents 0.45mN of thrust at 20W and 650s of specific impulse. The modular design can be flexibly configured to meet any mission needs. In this direction, aggregated units that multiply thrust, or distributed components across the spacecraft can be utilised.

In [44] and [45], the ENPULSION NANO and the ENPULSION MICRO thrusters' development is presented. They are both based on liquid metal FEEP principle, producing thrust by electrostatically accelerating previously extracted and ionized propellant to high exhaust velocity. The FEEP technology contains no moving parts and uses non-toxic indium as propellant (Figure 6). More specifically, the MICRO module provides 1.35mN of thrust, up to 6000s of specific impulse at 120W power and it is easily scalable.

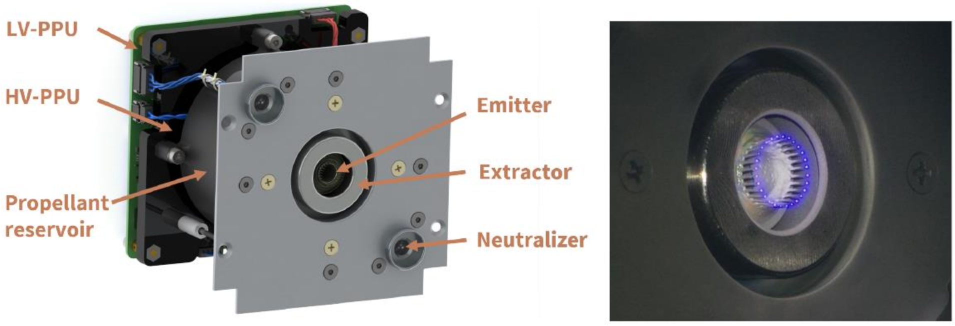

Figure 6: FEEP design and operation, [45]

Finally, in [36], the overview of the EP systems development of Busek also includes some active and passive electrospray options. More specifically, Busek's BET-300-P passively fed electrospray thruster is being actively developed as part of a high precision reaction control system. Up to four BET-300-P thrusters per centralized PPU can be integrated as desired to provide attitude or orbital control within a wide range of spacecraft platforms. Each thruster can provide throttled continuous thrust from <1μN up to 150μN at <2.5W of thruster power with sub-μN resolution over the full range and maximum specific impulse of 2300s.

Multimode systems

Another propulsion research field that recently attracts attention especially in the small satellites range, regards multimode propulsion systems. Multimode propulsion refers to the combination of two or more propulsion systems or operating modes of them with shared propellant and/or hardware into a single propulsion system. It is an emerging technology that offers enhanced capabilities (flexibility and adaptability) for any space mission and can therefore play a crucial role in the future of space propulsion, while mass savings can be also achieved. Although it is usually applied to combinations of chemical and electric propulsion systems/modes, this technique can be also integrated to EP systems only, providing some extra performance characteristics to the required applications. One of the core design goals for multimode EP systems, in addition to using a common propellant, is to make use of common hardware to reduce the mass of the propulsion system and increase the deliverable payload.

In this direction, recent development programs include multimode attitude and orbit control systems sharing a common back-end architecture for the propellant (Xe), power electronic control and fluidics, based on a HET and a resistojet. Such a system can support missions with the most demanding requirements benefitting from the low-thrust-high-impulse performance of the HET, as well as the high-thrust-low-impulse mode of the resistojet.

In [46], two all-electric multimode propulsion systems are presented. These systems are adjustable between a high specific impulse mode and what is often referred to as a high thrust-to-power (and correspondingly lower specific impulse) mode. The first system is a Xe HET system, and the second system is an electrospray propulsion system. Additionally, dual mode ion thrusters and hybrid Hall-ion thrusters have also been investigated.

All-electric multimode systems do not necessarily share the same thruster between modes. [47] study focuses on a multimode propulsion system integrating two electric thrusters, a resistojet and an ion thruster both fed with water propellant. This system is driven by the requirement for small satellite propulsion that can provide high ΔV, multi-axis thrust with a safe propellant. At only 50W of power level, the analyses suggest that the gridded ion thruster can provide 800s of specific impulse and 0.3mN of thrust, while the resistojet can provide 72s and 3.9mN thrust, respectively. Moreover, an interesting efficiency benefit to this approach is the use of the waste heat from the ion thruster power supply to support vaporization of the water propellant. In this configuration the ion thruster and four resistojets could be operated at the same time with only 50W power.

Resistojet

As reported in the previous sections, although resistojets generally provide adequate thrust but very low specific impulse (up to hundreds of s) compared to other EP systems, they require low power and present very low system complexity, thus they can be good candidates for EP multimode systems or simplified solutions where needed. Research interest over the years is focusing on innovative resistojet concepts to cover specific application needs and expand their applicability.

In [48], a next generation of high-performance xenon resistojet delivering specific impulse above 50s is studied. This would be of significant benefit to both small and newer all-electric spacecraft. This paper thus presents a validated model of the conventional SSTL-T50 thruster that operates with Xe, Butane or N2, requires 50W of power, with 57s of specific impulse and more than 30mN of thrust.

In [49], an overview of the commercially available resistojet options is presented along with a conceptual design of a 50mN coiled heater resistojet. In general, several propellant options are compatible with technology, including R236fa, R134a, Xe, Kr and Ammonia. The power range is usually low (up to 50W), the specific impulse goes up to 100s, while the thrust reaches 30mN.

2.2 LEO and VLEO micro/mini satellite missions

As briefly analysed in §1.6, along the lifetime of a satellite's mission, several operational manoeuvres to change some parameters of its orbit may be required. For a typical LEO mission, the main manoeuvres of a satellite usually include:

- Orbit correction after release from the launch vehicle
- Altitude/orbit/inclination changes based on mission requirements

- Continuous drag compensation
- Collision avoidance
- Phasing station keeping
- End-of-Life De-orbiting

In this frame, several studies are focusing on the efficient utilization of EP systems to cover the mission lifetime needs of micro/mini satellites orbiting in LEO and VLEO. It is of great importance to fully analyse and specify in detail the mission characteristics and the propulsion system application requirements so as to properly design the system. Thus, many studies depict mission analysis parameters for LEO and VLEO missions and specify all required information to solidly approach the EP system design.

EP system application cases

In [50], three main orbital manoeuvres are calculated and studied since they are considered important in the characterization of EP use for satellites in near-Earth orbits. The first manoeuvre changes an orbit from low-Earth orbit (LEO) at 800km to medium-Earth orbit (MEO) at 20000km. The second manoeuvre changes the inclination of an orbit at LEO by 90°, while the third manoeuvre rephases a satellite in LEO orbit by 180°. Each manoeuvre considers thruster specific impulse, I_{sp}, from 1000 to 3000s and thruster power from 100W to 1.5kW for a 500kg satellite to obtain propellant mass and transfer time. Thruster efficiency is set to 0.5, which is approximately the efficiency of current commercial, low-power EP systems at the power levels under investigation.

Study [51] presents the drag-free and attitude control (DFAC) of the European Gravity field and steady-state Ocean Circulation Explorer satellite (GOCE), during the science phase. DFAC aims to enable the gravity gradiometer to operate so as to determine the Earth's gravity field especially in the so-called measurement bandwidth (5-100mHz), making use of ion and micro-thruster actuators. More specifically, the DFAC technology is based on the following EP actuators:

- A pair of ion thrusters (1.5-20mN), in cold redundancy, for along-track drag compensation (single-axis control).
- Eight micro-thrusters (electrical, 0.002-1.2mN), for attitude tracking and compensation of lateral non-gravitational forces and torque disturbances (five-axis control).

Paper [52] studies the application of an electric propulsion system for autonomous station-keeping of a remote sensing spacecraft (100kg) flying at low altitude. The considered propulsion system exploits a xenon propellant bus, which operates both a low-power Hall-effect thruster and a resistojet (multimode operation). The former is used for continuous in-track control, while the latter provides the impulsive thrusts necessary for cross-track manoeuvres. More specifically, the HET (100W) features a high specific impulse (1000s) and provides continuous low thrust (2.5-6mN) in the tangential direction, while the resistojet (30W) generates a higher thrust level (10-50mN) to perform out-of-plane impulsive manoeuvres. This results in a hybrid continuous/impulsive control scheme. The orbit is nearly circular, with an altitude of around 228km (VLEO), which corresponds to a 5-day ground track repeat period.

In [24], the feasibility of use of EP options for a VLEO (250-500km) microsatellite aimed to support EO missions, is analytically calculated and studied. More specifically, two configuration scenarios are studied, one utilizing a 100kg satellite with 100W available for propulsion system, and the other utilizing a 10kg satellite and 15W for the propulsion system. All propulsion system parameters are analytically calculated taking into account the required orbital and operational specifications and two solar and geomagnetic activity extremes (high and low).

In [25], two main manoeuvrability (one for LEO and one for GEO) use cases for microsatellites are presented. The one related to LEO, is a station-keeping use case that requires performing 14m/s of station keeping manoeuvres per year divided evenly into a correction every two days to maintain a 10km station-keeping box at 600km orbit. The figures of merit for the station keeping are to minimize the percentage of time devoted to station-keeping manoeuvres and to minimize the average power needed for station-keeping.

Study [53] focuses on using various EP and ABEP systems to increase the lifetime and usefulness of the satellites up to 3U by providing drag compensation. The scope of the study is limited to VLEO ranging from 100 to 300km and all required design parameters are analytically estimated.

In [14], some flight experience with small satellite EP systems is presented. More specifically, one small satellite (200kg) case utilizing a 100-300W Xenon HET (Busek BHT-200) and two nanosatellite cases (1.5U) utilizing VAT, PPT and Electrospray systems, are depicted.

Paper [41] provides analytical calculations for a 6U satellite required orbital manoeuvres performed by EP in the 400-800km altitude envelope. The analysis includes Manoeuvre duration (Δt), velocity variation (ΔV), fuel consumption (Δm) for basic orbital transfers causing altitude variation, inclination change and node shift as functions of initial altitude (circular orbit).

In [54], INSPIRESat-4 Atmospheric Coupling and Dynamics Explorer mission is presented. It concerns a ring-deployed 27U spacecraft that aims to achieve a sustained flight in the VLEO region to make in-situ Ionospheric plasma measurements. The mission consists of two main phases, an initial 535x450km orbit for 6 months of science observations and another 6 months on a 300km altitude. The orbit lowering can be done passively (relying on drag) or actively (through retrograde thruster firing). Once at 300km, prograde thruster operation must be used to prevent rapid decay of the orbit. The NPT-30 ion thruster from ThrustMe is operated during the VLEO part of the mission providing 0.3-1.1mN of thrust, with a specific impulse up to 2400s and 65W of required power.

In [55], the use of an ABEP system for drag compensation in the frames of DISCOVERER (developing technologies to enable commercially viable sustained operation of satellites in VLEO for communications and remote sensing applications) program is presented.

Finally, in [56], an economical remote sensing concept from a low altitude with continuous drag compensation is analytically presented. The planned mission is focusing on reducing the altitude very considerably, to around 300km or even below from the usual 600-800km. In this case, a significant aerodynamic drag force will be experienced by the satellite, but this can be balanced by a suitable propulsion system. For this, the use of gridded ion thrusters operating at very high exhaust velocity is suggested, thereby enabling the propellant required to be well below that needed if the chemical propulsion alternative was adopted. An analytical comparative influence of thruster type on spacecraft mass (around 300kg), for an altitude of 280km and mean solar activity is presented including several electric thruster types and configurations.

Mission analysis

Many studies are focusing on LEO and VLEO mission analysis of all required mission parameters including orbital specifications, lifetime considerations, environmental aspects (solar and geomagnetic indices, atmospheric modelling, drag coefficient (C_D)

estimations) and satellite configurations. In the frame of this study, only studies related with the required calculations (especially for drag compensation and orbital decay estimations) for the EP system are reviewed.

Low Earth orbiting satellites experience orbital decay and have physical lifetimes determined almost entirely by their interaction with the atmosphere. Prediction of such lifetimes or of a re-entry date is quite important for the mission analysis and the dimensioning of the EP system.

The prediction of satellite lifetimes depends upon a knowledge of the initial satellite orbital parameters, the satellite mass to cross-sectional area (in the direction of motion), and a knowledge of the upper atmospheric density and how this responds to space environmental parameters which must also be predicted. Even with a complete atmospheric model describing variations with time, season, latitude and altitude, complete specification of orbital decay is not possible because of uncertainties in the prediction of satellite attitude (which affects the relevant cross-sectional area), and solar and geomagnetic indices (which substantially modify the atmospheric model). Even when most of the quantities are known there appears to be an irreducible level below which it is not possible to predict. Thus, it is important to approach those mission analysis estimations with enough margin so as to fulfil the required mission objectives and properly design the EP system.

In [57], a simple model for atmospheric density as a function of space environmental parameters is presented. It shows how this may be applied to calculate decay rates and orbital lifetimes of satellites in essentially circular orbits below 500km altitude. Specific analysis is depicted regarding the atmospheric model approach, the drag coefficient estimation and the space environment parameters.

Study [58] concentrates on the modelling and simulation of VLEO. The characteristics of the VLEO environment and how it affects to the performance of a satellite are depicted while results comparing a satellite flying at LEO (700km) and at VLEO (350km) are shown, and the main differences are highlighted. In order to get realistic values of the perturbations affecting the satellite the following models were used:

- Atmospheric model: The Drag Temperature Model DTM2013
- Earth's magnetic field model: International Geomagnetic Reference Field IGRF12
- Atmospheric wind: Horizontal Wind Model HWM14.

Master thesis presented in [59], studies the feasibility of providing small satellites with a propulsion system that would enable them to perform orbit control manoeuvres all along the mission duration. The concept is to create a computer tool able to carry out a rapid analysis of the satellite mission, for the determination of the needed ΔV, and then a preliminary design of the main components of the required propulsion system. Different propulsion technologies are in this way considered, offering a trade-off option to select the best solution, in terms of mass and performance. Satellite models ranging from nano to mini-sat standard in LEO-VLEO missions of different durations (2, 5 and 7 years) have been used for feasibility simulations, and the results show that the use of some propulsion technology is possible to reach the fixed mission goals.

In [53], a thorough analysis of the feasibility of using EP systems for drag compensation of small satellites (up to 3U) in VLEO (up to 300km) is presented. In this frame, four types of thrusters are analysed (PPT, Ion, FEEP and ABEP) for different operational scenarios. For the calculation of the required drag compensation, five different atmospheric models are examined (Exponential, NRLMSISE, CIRA, ISA and MSIS) selecting NRLMSISE as the most accurate one. Moreover, drag variation with altitude, latitude, longitude and solar activity is evaluated. Finally, in order to estimate the orbital decay of satellites with and without electric propulsion systems, accurate orbital propagators have to be employed. The orbital propagators show the variation of some of the orbital parameters through time. Two main methods were employed, the first one estimates the loss of energy after each orbit, where the second one uses the Gauss's Planetary Equations (GPE). An extended GPE propagator can handle and monitor the changes in all 6 orbital elements (also verified with software STELA of CNES and GMAT).

In study [60], an evaluation of available methods for the computation of the aerodynamic drag coefficient of LEO satellites is depicted. This paper focuses on the thermospheric atmospheric region which is a high-altitude layer that exist above 85km. Generally, the solar flux and the geomagnetic activity are the main energy sources affecting its structure. The extremely low level of oxygen in the atmosphere of LEO demands a different aerodynamics methodology than that used in the continuum regime. In this direction, free molecular flow methodology is presented, along with numerical methods (panel, ray-tracing panel, test-particle Monte Carlo and direct simulation Monte Carlo) to compute the required parameters. Using the aforementioned analysis, a mathematical model is created that computes the drag coefficient. Verification and validation of this model is achieved by calculating the drag coefficient of flat plate, of a sphere and finally of a satellite (cubic with 1m side at 300km) with estimation at different orbital altitudes and velocities.

In [61], a deep analysis of drag coefficient computation is presented, focusing on the Cercignani–Lampis–Lord Gas–Surface Interaction Model. Drag coefficient calculations using the Cercignani–Lampis–Lord quasi-specular gas–surface interaction model have been used to derive modified closed-form solutions for several simple geometries (cube, cuboid, sphere, flat plate, parallel and vertical cylinder). The key component of the modified closed-form solutions is a relation between the normal energy and normal momentum accommodation coefficients, which is valid within ~0.5% over the global parameter space. The modified closed-form solutions are made self-consistent by relating the effective energy accommodation to the partial pressure of atomic oxygen through a Langmuir isotherm. The modified closed-form solutions are compared to fitted drag coefficients and drag coefficients computed using two other gas-surface interaction models: diffuse reflection with incomplete accommodation and Maxwell's model. Comparison during solar maximum conditions shows that both the diffuse reflection with incomplete accommodation and Cercignani–Lampis–Lord models agree with fitted drag coefficients within ~2% below ~500km altitude. Further comparison shows that solar minimum drag coefficients are up to ~24% higher than those at solar maximum based on global ionosphere–thermosphere model atmospheric properties. Drag coefficients computed with atmospheric properties from the Naval Research Laboratory mass spectrometer incoherent scatter extended model and the global ionosphere–thermosphere model agree within ~2% at solar maximum but disagree by up to ~11% at solar minimum.

Paper [62] reviews currently available and most common methods to calculate drag coefficients of spacecraft traveling in LEO. Aerodynamic analysis of satellites is necessary to predict the drag force perturbation to their orbital trajectory which for LEO is the second in magnitude after the gravitational disturbance due to the Earth's oblateness. Historically, accurate determination of the spacecraft drag coefficient was rarely required. This fact was justified by the low fidelity of upper atmospheric models together with the lack of experimental validation of the theory. However, advances on the field, such as new atmospheric models of improved precision, have allowed for a better characterization of the drag force. They have also addressed the importance of using physically consistent drag coefficients when performing aerodynamic calculations to improve analysis and validate theories. In this frame, a common approach to calculate the drag acceleration experienced by a body is:

$$a_{drag} = \frac{1}{2}\rho V^2 C_D \frac{S}{m} \qquad \textbf{Eq. 4}$$

where ρ represents the atmospheric density, V is the relative velocity of the satellite with respect to the atmosphere, C_D is the drag coefficient, S is the reference surface area in the direction of motion and m the mass of the body. Especially for the C_D, it has been a common practice to assume it constant and equal to 2.2 for LEO satellites. Due to the lack of precision of existing atmospheric density models, any modelling effort to refine the drag coefficient was normally considered of little advantage, since it does not compensate for the imprecise density model. Nowadays, it is widely accepted that the drag coefficient is not constant and can present very different values depending on the spacecraft shape and the atmospheric temperature and composition at the flying altitude. Moreover, two indices are generally used to measure the solar radiation and geomagnetic activity levels (usually mean, minimum and maximum values are estimated and applied in the analysis):

- F10.7 index for solar flux: It is a measure of the solar flux emitted at a wavelength of 10.7cm. Since the extreme ultraviolet (EUV) radiation is absorbed in the thermosphere it is difficult to obtain a measurement of the solar flux at these wavelengths using instruments at the Earth's surface. It has been found that the F10.7 index presents a good correlation with solar activity. Currently it is used as a proxy for EUV radiation in atmospheric models.
- A_P index for geomagnetic activity: This index is a measure of the general level of geomagnetic activity on the Earth for a given day. It is obtained from measurements of the magnetic field variations made at different locations. Geomagnetic storms are characterized by a sudden increase of this index.

Finally, two of the most modern models are briefly presented:

- NRLMSISE00: This model provides temperature and gas species number densities (for He, O, N_2, O_2, Ar, H and N) covering all the range from sea level up to the exosphere. Inputs to the model are altitude, latitude, longitude and the two indices F10.7 and A_P. A component named "anomalous oxygen" is introduced in the model for drag estimation. It accounts for the contribution of non thermospheric species to the drag at high altitudes, such as O^+ and hot oxygen (energetic oxygen atoms resulting from photochemical processes in the upper atmosphere).
- JB2006: This model provides neutral density and temperature from 120km to the exosphere. Input parameters are F10.7 and A_P, however, it also incorporates new solar indices (S_{10} and Mg_{10}) to obtain better density variation correlations with UV radiation together with a model of the semi-annual density variation. A further improvement in the modelling and results is the JB2008 version.

According to the European Cooperation for Space Standardization (ECSS) standard on Space environment, the NRLMSISE00 model shall be used for calculating neutral temperature, detailed composition and total density of the atmosphere, whereas the JB2006 (or JB2008) model may be used for calculating the total density above 120km.

In [63], the JB2006 empirical thermospheric density model is analytically presented. This new empirical atmospheric density model is developed using the CIRA72 (Jacchia 71) model as the basis for the diffusion equations. New solar indices based on orbit-based sensor data are used for the solar irradiances in the extreme and far ultraviolet wavelengths. New exospheric temperature and semi-annual density equations are employed to represent the major thermospheric density variations. Temperature correction equations are also developed for diurnal and latitudinal effects, and finally density correction factors are used for model corrections required at high altitude (1500-4000km). The new model, Jacchia-Bowman 2006, is validated through comparisons of accurate daily density drag data previously computed for numerous satellites. For 400km altitude the standard deviation of 16% for the standard Jacchia model is reduced to 10% for the new JB2006 model for periods of low geomagnetic storm activity.

Finally, in study [64], a critical assessment of satellite drag and atmospheric density modelling is depicted. This paper examines atmospheric drag models and data usage involved with propagating near-Earth satellites. The following important conclusion is extracted:

- No single atmospheric model is best for all applications. As a general outcome, JB2008 is the most accurate model below 300km, JB2008 and DTM2009 perform best in the 300-500km altitude range, whereas above 500km NRLMSISE00 and DTM2009 are most accurate. The precision of JB2008 decreases with altitude, which is due to its modelling of variations in local solar time and seasons, in particular of the exospheric temperature rather than modelling these variations for the individual constituents.

System design

Study [65] discusses two microsatellite system design approaches, namely Technical University of Berlin heritage and University of Surrey heritage. Both Universities provide approaches for system design and build of microsatellite systems. The design approaches are being compared along with lessons learned. Five sample satellites from each satellite design heritage are compared, including 15 bus parameters, payload profiles, and satellite weight and volume at launch. From the comparison, it is found that major differences in the satellite bus are in the choice of main computers and their associated link configuration and in the attitude control modes that also affect the design. Another major difference is in the satellites' structure design, which resulted in much higher density in the TU Berlin heritage satellites than the University Surrey heritage satellites. In the early design, there are differences in the choice of satellite's batteries. However, as soon as Li-ion batteries became available, both design heritages used such technology. In answering the increasing needs in payload data handling, both design heritage use of FPGA-based payload data handling and high downlink data rate in X-band. GPS is also the technology adopted by both design heritages for orbit determination and imager's ancillary data.

2.3 Critical analysis

This chapter contained a summary of the literature relevant to the Master thesis subject and objectives. A general overview of low power (up to ~200W) EP systems was provided including their main operating principles and design characteristics followed by a more comprehensive presentation of the most common low power EP options. Finally, an overview of LEO (up to 600km) and VLEO micro and mini satellites missions, focusing mainly on EP systems application cases and the related mission analysis was recorded.

From this thorough literature review, a critical gap in the EP system analysis and study for satellites in the size envelope of 100-250kg and for well-defined missions' in VLEO and lower LEO (up to 600km) regime is identified. More specifically, for the electric propulsion system study and dimensioning to fit a range of real use cases applications and to be directly applied in a range of missions, literature does not present so much analysis and related information. The literature concentrates mainly on low power EP systems and the related technologies as an independent research unit. Moreover, missions and applications of the EP systems tend to be more generic and either for smaller satellites (nanosats, CubeSats) or for higher orbits. Finally, most studies that analyse the mission and system design parameters do not focus on EP system design and analytical features and do not present conceptual EP system approaches.

Taking into account this critical analysis, the main contribution of this study is to combine all available information, concentrate on the EP system concept as a part of realistic applications and cover this literature gap by introducing a representative and already available electric propulsion system as a well-defined scenario that can efficiently fulfil the main mission requirements of the space market in this operating envelope.

In this direction, this conceptual study approach that fully specifies an EP system that can satisfy the requirements of three different (V)LEO mission scenarios of a representative mission application that covers most of the current space market needs. The EP system is defined in detail with a focus on the optimization of payload mass and power fraction offering a very effective tool to any satellite integrator to evaluate this system and easily adjust it to any (V)LEO mission requirement. In this way, it constitutes a feasible plug and play solution for applications in the defined operating envelope, easy to integrate and configure, leading to a faster, cheaper and more reliable EP system integration to any satellite need. This leads to the most important and novel characteristic of this study, with this concept-like focus on the EP system being the core of the whole study interest.

3. MISSION ANALYSIS

As analysed in the previous sections there is a gap in EP system analysis for satellites in the size envelope of 100-250kg and for specific missions' range in VLEO and lower LEO (up to 600km) regime. Scope of this study is to cover this gap introducing representative and already available systems as a baseline scenario that will efficiently fulfil the main mission requirements, especially in terms of propulsion. In this frame, a satellite platform in the above-mentioned range is selected and specified, along with one representative application that can be applied to three different orbits. Finally, based on those criteria all mission parameters are extracted and estimated to drive the design of the EP system.

3.1 System requirements and specifications

Based on mission characteristics investigated during the literature review, typical satellite configurations at micro/mini size level for LEO and VLEO applications usually incorporate a box-like main body (that includes the main satellite Bus and the payload), small deployable solar arrays in some directions of the body, and the rest external subsystems (antennas, AOCS, thrusters, etc.) placed accordingly in the open sides of the body. A quite representative satellite platform inside all the required specifications set for this analysis, is SITAEL S-200 ([6]).

SITAEL S-200 is a Minisatellite-class Platform, with up to 200kg max launch mass, characterized by high payload embarking capability, modular structure, designed for multi-purpose missions. It is an all-electric platform that can be equipped with a low power electric propulsion subsystem to enhance orbit control capabilities. It is characterized by high power availability on-board due to the large deployable solar arrays and high battery capacity and high performance AOCS (Attitude and Orbital Control Subsystem). In this way, S-200 Platform, with its flexibility and multi-purpose features, is suitable for a wide series of space missions, in particular for Earth Observation applications and for Telecom constellations thanks to the autonomous orbit deployment capability by electric propulsion. An expanded overview of the configuration of the platform is depicted in Figure 8, a sketch of the relative platform design in terms of motion coordinates is presented in Figure 7, while its main specifications are presented in S-200 related datasheet in Figure 9.

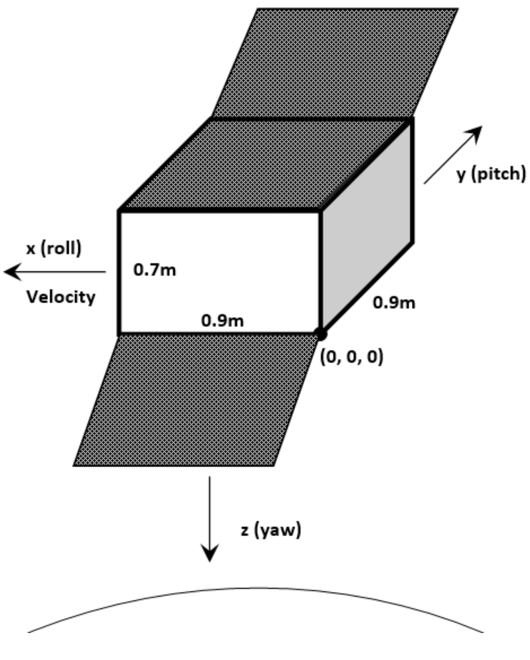

Figure 7: S-200 Platform relative motion sketch

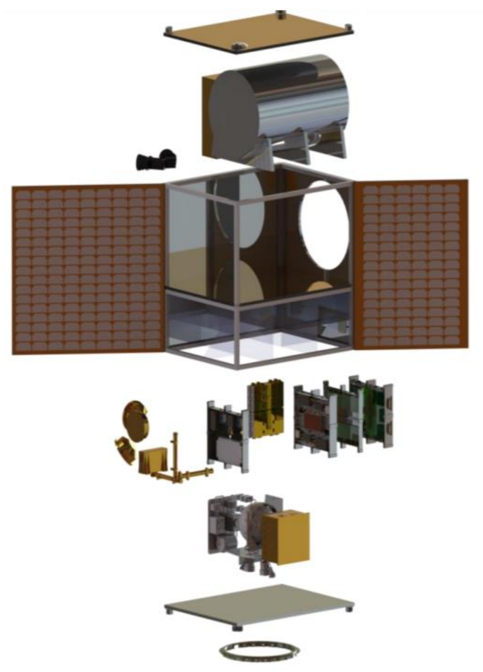

Figure 8: S-200 Platform expanded view, [6]

Targeted mission	EO SSO in very-LEO @350-800 km
P/L max mass	Up to 80 kg
P/L avg power cons.	Up to 120 W
P/L allowable volume	Up to 900x700x560 mm^3
S/C launch mass (kg)	<150 / <200 kg (*)
S/C envelope LxWxH	900 x 700 x 900 mm^3
S/C power gen.(W)	up to 200 W Avg, 510 W Peak
Solar array cells	GaAs TJ, 30% Eff. Cells
Battery capacity	Li-Ion, 880 Whr
Pointing accuracy	<0.05°, 3-axis stabilization
Pointing knowledge	0.006°
Slew rate	Up to 3 °/sec (0.5 °/s^2)
Delta-V	Up to 1 km/s (*)
TT&C	S-band, up to 1 Mbps (TM TX)
PDHT data rate	X-band, up to 400 Mbps
PDHT data storage	Up to 256 GB
S/C redundancies	Full-cold / partially hot P/F red.
Lifetime	Up to 5 years

Figure 9: S-200 Platform specifications, [6]

The platform is thermally and mechanically designed to withstand the environmental and lifetime requirements in the specified orbits and includes all required operating subsystems, as follows in brief description:

- Electrical Power Subsystem: The system responsible for the production (by deployable solar arrays), storage (with dedicated batteries) and distribution (through the power conditioning and distribution unit) of the required power in all satellite units (including the payload) and under all mission operating conditions (daylight and eclipse).

- AOCS: The system responsible for the attitude and orbital control of the satellite, aimed to provide the required compensation for any torque and accumulated angular momentum disturbance (solar torque, drag torque, gravity gradient and thruster firing and misalignment disturbance). It performs attitude measurement (through the dedicated sensors) and control (through the dedicated actuators) of the satellite in order to fulfil specific mission pointing (accuracy, stability and knowledge), orbital and agility requirements.
- EP subsystem: It is actually a subunit of the AOCS and it is usually responsible for the continuous drag compensation and orbit maintenance of the satellite. Based on the mission requirements this system can provide also orbit correction after release from the launch vehicle, altitude/orbit/inclination changes, collision avoidance, phasing station keeping and End-of-Life de-orbiting.
- Data Handling and Computing Subsystem: This system constitutes the heart of the satellite including a central processing unit and an associated mass memory. It is responsible for the whole operation of the satellite and its subsystems. It performs all local housekeeping tasks to make the dialogue run smoothly between the functional units and the CPU.
- Communication Subsystem: The system responsible for the communication of the satellite with the associated ground station. All telemetries transfer and telecommands receival for the control and operation of the satellite are passing through two dedicated satellite links (downlink and uplink accordingly) with their respective antennas.
- Payload: The core of the mission objective and the most important part of the satellite since it defines the mission application. All systems are actually working for the payload so as to provide the required means to achieve its mission, e.g. provide EO images, provide high speed communication etc.

3.2 System selected application and payload

As synoptically presented in §1.3.1, lowering the altitude offers significant advantages in the payload performance especially for EO applications. Almost 30% of currently orbiting satellites below 600km are providing EO services (as depicted in §1.4) either utilizing optical payloads or radars. As the radar payloads are capable of acquiring EO images with every lightning and weather condition (not affected by atmosphere elements), they offer a great advantage to such applications. For this reason, a SAR is applied as a representative payload for this study, targeting remote sensing applications such as maritime and land surveillance, agriculture monitoring, land monitoring and interferometry.

According to NASA [66], Synthetic Aperture Radar is a type of active data collection where a sensor produces its own energy and then records the amount of that energy reflected back after interacting with the Earth. While optical imagery is similar to interpreting a photograph, SAR data require a different way of thinking in that the signal is instead responsive to surface characteristics like structure and moisture. The spatial resolution of radar data is directly related to the ratio of the sensor wavelength to the length of the sensor's antenna. For a given wavelength, the longer the antenna, the higher the spatial resolution. From a satellite in space operating at a wavelength of about 5cm (C-band radar), in order to get a spatial resolution of 10m, you would need a radar antenna about 4250m long. An antenna of that size is not practical for a satellite sensor in space. Hence, scientists and engineers have come up with a clever workaround -the synthetic aperture. In this concept, a sequence of acquisitions from a shorter antenna are combined to simulate a much larger antenna, thus providing higher resolution data.

Radar sensors utilize long wavelengths at the centimetre to meter scale, which gives it special properties, such as the ability to see through clouds. The different wavelengths (frequencies) of SAR are often referred to as bands, with letter designations such as X (8-12GHz, high resolution SAR targeting urban monitoring, ice and snow, little penetration into vegetation cover, fast coherence decay in vegetated areas), C (4-8GHz, SAR main operation envelope targeting global mapping, monitoring of areas with low to moderate penetration and higher coherence, ice, ocean maritime navigation), L (1-2GHZ, medium resolution SAR aimed for geophysical monitoring, biomass and vegetation mapping, high penetration, interferometry), and P (0.3-1GHz, Experimental SAR targeting mainly biomass and vegetation mapping and assessment) being the most commonly used.

SAR data can also enable an analysis method called interferometry, or InSAR. InSAR uses the phase information recorded by the sensor to measure the distance from the sensor to the target. When at least two observations of the same target are made, the distance, with additional geometric information from the sensor, can be used to measure changes in land surface topography. These measurements are very accurate (up to the centimetre level) and can be used to identify areas of deformation from events like volcanic eruptions and earthquakes.

A representative SAR payload for small satellite (well fitted inside the aforementioned satellite platform) remote sensing applications can be approached summarized with the following main specifications:

- Mass: ~70kg
- Power consumption during operation: 120W
- Power consumption in stand-by mode: 30W
- Dimensions (including electronics): 600x400x400mm^3
- Selected SAR band: X (8-12GHz)

This payload can be easily utilized and adjusted to the whole orbit envelope of interest (300-600km) providing accurate (high resolution) images and high spatial resolution (down to 5m) independently of the selected orbit. The lower orbit though, offers significantly improved link budgets, reduced SAR antenna size and transmission power, reduced latency and improved frequency reuse.

3.3 Mission requirements and specifications

Orbit selection

According to [12] satellite database, the vast majority (around 75%) of the EO satellites orbiting in the 350-600km regime, fly in a Sun-Synchronous polar orbit. Usually within 30 degrees of the Earth’s poles, the polar orbit is used for satellites providing reconnaissance, weather tracking, measuring atmospheric conditions, and long-term Earth observation. In the same way, SSO satellites are synchronous with the sun, such that they pass the equator at the same local solar time on every pass being very useful for image-taking applications because shadows are the same on every pass.

In order to adequately cover the specified orbit envelope for this study, three different altitudes are selected that can support the specified payload EO application and consist representative VLEO and LEO cases. Regarding the mission lifetime for each orbit, a typical lifetime assumption based on similar EO applications at those altitudes ([12]) is applied, driving to 2 years lifetime for 400km altitude, 5 years for 500km and 7 years for 600km respectively. The main orbit parameters are summarized in the following table (Table 5).

Table 5: Selected orbits main parameters

	Orbit 1 (VLEO)	**Orbit 2 (LEO)**	**Orbit 3 (LEO)**
Orbit type	SSO dawn-dusk, almost-circular orbit with drifting RAAN	SSO dawn-dusk, almost-circular orbit with drifting RAAN	SSO dawn-dusk, almost-circular orbit with drifting RAAN
Average altitude, [km]	405	509	614
Eccentricity, e	7.37×10^{-4}	1.31×10^{-3}	2.00×10^{-3}
Inclination, i	97.050º	97.438º	97.845º
Argument of perigee, ω	90º	90º	90º
Semimajor axis, α, [km]	6783	6887	6992
Period, T, [min]	92.660	94.799	96.975
Orbits per day	15.495	15.146	14.806
Orbiting Velocity, [km/s]	7.666	7.608	7.550
Mission lifetime, [yrs.]	2	5	7

EP system utilization

As synoptically presented in literature review (§2), along the lifetime of a satellite's mission, several operational manoeuvres to alter some parameters of its orbit slightly or significantly, are required. Especially, for a typical (V)LEO mission, the main manoeuvres of a satellite usually include orbit correction after release from the launch vehicle, altitude/orbit/inclination changes based on mission requirements, drag compensation for orbit maintenance, collision avoidance, phasing station keeping and End-of-Life de-orbiting.

In altitudes around 300-600km, the most critical manoeuvre that significantly affects and defines the propulsion system specifications is orbit maintenance (compensation of orbit decay due to the atmosphere) due to the increasing atmospheric drag with the altitude reduction. In order to achieve any EO mission at those altitudes, this orbit drag compensation shall be properly covered by the electric propulsion system of the satellite for the specified mission lifetime. From the rest possible manoeuvres:

- An amount of propellant is allocated for orbit correction after release from the launch vehicle based on launcher injection inaccuracy.
- No altitude/orbit/inclination changes are included in the frames of this study (no mission adjustment is foreseen).
- No phasing station keeping is foreseen (more useful for satellite constellations).
- End-of-Life de-orbiting is foreseen, targeting an altitude decrease at the end of the mission lifetime, so as to reach a controlled satellite re-entry in less than 10 years.
- Collision avoidance manoeuvres are assumed to be covered by the extra margin of propellant applied.

Orbit decay approach

As briefly analysed in literature review (§2), the common approach to calculate the drag acceleration (and correspondingly the required drag acceleration compensation) experienced by a body is given by the Eq. 4. The parameters that define this acceleration are the atmospheric density, the relative velocity of the satellite with respect to the atmosphere, the drag coefficient, the reference surface area in the direction of motion and the mass of the body. From those, the relative velocity is easily calculated depending on the altitude of the satellite for circular orbits $\left(V = \sqrt{\frac{\mu}{\alpha}}\right)$, the reference surface area in the direction of motion can be extracted based on Figure 7 ($0.7x0.9m^2$) and the mass of the satellite is known (initially 200kg, decreasing over the lifetime due to propellant consumption, but any estimation considering the initial mass lies in the safe regime). The rest two parameters, the atmospheric density and the drag coefficient concentrate the major inaccuracy impact and difficulty to approach in the orbit decay estimation.

Atmospheric density model

As explained in §2, there are several atmospheric models that approximate the atmosphere characteristics, including total density, in different altitudes, but no single atmospheric model is the best for all applications. The two most modern and usually applicable models on LEO satellite missions are the NRLMSISE00 and the JB2006 (or the updated JB2008). According to the ECSS standard on Space environment (ECSS-E-ST-10-04C Rev.1, [67]), the NRLMSISE00 model shall be used for calculating total density in all cases, whereas the JB2006 model may be used for calculating the total density above 120km. According to other studies (e.g., [64]), the use of NRLMSISE00 model is also recommended for altitudes above 500km while for lower altitudes JB2006 is suggested.

Moreover, the two indices referred in §2 (F10.7 index for solar flux and A_P index for geomagnetic activity) cause significant impact on the outcome of the atmospheric models based on their estimated intensity over the time of the mission. A proper approximation of those parameters usually contains the calculations for low (F10.7=65, A_P=0), moderate (F10.7=140, A_P=15) and high long term (F10.7=250, A_P=45) solar and geomagnetic activity.

As depicted in [58], another parameter that affects the atmospheric density is the longitude and the related magnetic field and atmospheric wind in different longitude inclinations. According to the data of [58] though, satellite inclinations of around 90^o (as in all selected orbit cases) lies in the lower density regime and the effect can be neglected since with the calculated mean values the dimensioning is surely in the safe region. Moreover, any latitude effect can be also neglected since the fast passing of all latitudes during each satellite orbit (around 15 times per day) eases the analysis by utilizing the average values calculated.

To conclude, in order to better review the atmospheric density for the altitudes of interest, both atmospheric models are applied based on ECSS [67] reference data for low, moderate and high solar and geomagnetic activity, and the results per orbit are presented in the following table (Table 6). For better accuracy, linear interpolation was applied to reach the required average altitude. Moreover, to evaluate in a realistic way the solar and geomagnetic activity during a satellite mission lifetime (usually around 5 years), study [68] is used. According to the data inside this study (from 1965 to 2016), there is a 10-15 years repetitive trend for solar and geomagnetic maximum lasting for 2-3 years followed by 2-3 years of drop and 2-3 years at minimum. Exploiting this trend, it can be assumed that for a regular LEO mission lasting about 5 years, about one third of the time is consumed in low activity, one third in moderate activity and one third in high activity. In this way a worst-case biased result can be extracted that safely drives the calculations of the required drag compensation propellant at any launching time. Thus, inside the following table (Table 6) this approach is also estimated for both atmospheric models and it is the value that is going to be applied to the rest of the drag compensation analysis.

Table 6: Atmospheric density calculations

Orbit 1 (VLEO)			
Average altitude, [km]	405		
F10.7	65	140	250
A_P	0	15	45
Density, ρ, [kg/m^3] NRLMSISE00	5.228×10^{-13}	3.678×10^{-12}	1.315×10^{-11}
	5.784×10^{-12}		
Density, ρ, [kg/m^3] JB2006	4.203×10^{-13}	3.920×10^{-12}	1.025×10^{-11}
	4.863×10^{-12}		
Orbit 2 (LEO)			
Average altitude, [km]	509		
F10.7	65	140	250
A_P	0	15	45
Density, ρ, [kg/m^3] NRLMSISE00	5.818×10^{-14}	6.919×10^{-13}	3.391×10^{-12}
	1.380×10^{-12}		
Density, ρ, [kg/m^3] JB2006	4.689×10^{-14}	7.606×10^{-13}	2.725×10^{-12}
	1.177×10^{-12}		
Orbit 3 (LEO)			
Average altitude, [km]	614		
F10.7	65	140	250
A_P	0	15	45
Density, ρ, [kg/m^3] NRLMSISE00	10.249×10^{-15}	1.492×10^{-13}	9.762×10^{-13}
	3.785×10^{-13}		
Density, ρ, [kg/m^3] JB2006	9.790×10^{-15}	1.717×10^{-13}	8.497×10^{-13}
	3.437×10^{-13}		

As depicted from the above analysis, in all orbit cases, NRLMSISE00 model gives worst density results (higher value of density that it is translated into higher required drag compensation). For this reason, those values are going to be used for the calculation of drag compensation also following the suggestion from ECSS standards.

Drag coefficient

The final parameter that has significant impact on the estimation of required drag compensation is the drag coefficient. As presented in §2, it has been a common practice to assume it constant and equal to a representative value of 2.2 for LEO satellites. Recently though, it is widely accepted that the drag coefficient is not constant and can present very different values depending on the spacecraft shape and the atmospheric temperature and composition at the flying altitude.

Nowadays, several studies are concentrating on estimating the drag coefficient of different satellite configurations in different orbits to predict the required propulsion specifications more accurately but due to the uncertainty of all involved modelling parameters (including the atmospheric models) the advantage acquired is usually minimal. One of the most thorough analysis of drag coefficient calculation is provided in [61], where many parameters in respect to the drag coefficient estimation are evaluated and deeply analysed. The shape of the satellite, the altitude, the temperature, the velocity, the density, the energy and the momentum are some of them. In order to include in the present analysis some better accuracy also regarding the drag coefficient estimation, the results of [61] study (Figure 6a) regarding the orbiting velocity versus the shape of the satellite is utilized. More specifically, the satellite can be approximated with a cube shape (cuboid shape inside the study has a quite big length in respect to the other dimensions and does not fit the selected platform, deployable solar arrays due to their thin layer in the direction of motion can be neglected) and taking into account the calculated orbiting velocity (~7500km/s) the extracted drag coefficient is $\mathbf{C_D \approx 2.45}$ in all cases.

3.3.1 ΔV calculations

Having up to now clarify and specify all required mission, satellite, orbiting and system sizing parameters, the next step is to calculate the required ΔV for each manoeuvre to be covered by the EP.

Injection inaccuracies correction

Usually, when a satellite is released from the launcher, its orbit is not exactly the specified one, having a slightly different altitude and inclination, due to regular injection inaccuracies of the launcher's last stage. Thus, the satellite is required to perform an initial small correction manoeuvre utilizing its EP system to reach the exact mission orbit. Modern launchers present very high injection accuracy especially for LEO, but a slight orbit correction is always required. This orbit correction normally consists of an altitude change and an inclination change.

For impulsive thrusters those manoeuvres can be carried out separately (in two impulsive steps), but low thrust EP thrusters are usually non impulsive ones as analysed in §2.1, working continuously to perform the required manoeuvre altering their plume direction. In this case the orbit correction is performed in one step including both required manoeuvres. In this case the required ΔV can be calculated with adequate accuracy by the Edelbaum's equation (Eq. 5):

$$\Delta V_{inj} = \sqrt{V_i^2 + V_f^2 - 2V_iV_f\cos\left(\frac{\pi}{2}\theta\right)} \quad \textbf{Eq. 5}$$

where V_i is the velocity in the injection orbit, V_f the velocity in the final desired orbit and θ is the required inclination angle change in rads.

One of the most modern, widely applicable and representative launchers for LEO earth observation missions is Vega from Arianespace, a program developed by the ESA. It is used as a reference for the calculations of this study since it serves the majority of European LEO launches. Vega has the ability to deliver multiple satellites directly into sun-synchronous orbits, polar circular orbits, or circular orbits of different inclinations with very high accuracy. According to Vega user's manual [69], it nominally provides a semimajor axis accuracy of ±15km and an inclination accuracy of ±0.15 degrees (0.002618 rads). The ΔV calculations for the injection inaccuracies correction for each selected orbit are presented in the following summary table (Table 7).

Table 7: ΔV calculations for injection inaccuracies correction

Orbit 1 (VLEO)			
	Inaccuracy (-)	**Nominal**	**Inaccuracy (+)**
Average altitude, [km]	390	405	420
Velocity, [km/s]	7.674	7.666	7.657
Inclination change, θ, [deg]	±0.15	-	±0.15
ΔV calculated, [m/s]	**32.664**	-	32.624
Orbit 2 (LEO)			
Average altitude, [km]	494	509	524
Velocity, [km/s]	7.616	7.608	7.599
Inclination change, θ, [deg]	±0.15	-	±0.15
ΔV calculated, [m/s]	**32.384**	-	32.344
Orbit 3 (LEO)			
Average altitude, [km]	599	614	629
Velocity, [km/s]	7.558	7.550	7.542
Inclination change, θ, [deg]	±0.15	-	±0.15
ΔV calculated, [m/s]	**32.108**	-	32.069

End-of-Life de-orbiting

According to the IADC (Inter-Agency Space Debris Coordination Committee) relevant debris mitigation guidelines, any spacecraft in LEO must have an orbital life not greater than 25 years. It is thus necessary to adopt a strategy to control the satellite's presence in orbit after the end of mission lifetime. In order to include a more challenging requirement in the near future, since debris mitigation becomes more and more important, especially for (V)LEO satellites, in the frames of this thesis a controlled de-orbit by lowering the orbit altitude at the end of the mission to achieve a satellite re-entry in about 10 years is foreseen.

Based on [57] calculations, an orbit of about 410km is required to fulfil a 10-year re-entry under worst case conditions (low solar and geomagnetic activity). It is evident that this requirement is already reached for Orbit 1 (VLEO) and no further propellant allocation is required. For the other two selected orbits, a controlled de-orbit at 410km is foreseen utilizing the EP system. The ΔV required for this manoeuvre can be approximated using the following equation (Eq. 6):

$$\Delta V_{deorbit} = V\left[1 - \sqrt{\frac{2(R_e + H_e)}{2R_e + H_e + H_i}}\right] \quad \textbf{Eq. 6}$$

where V is the satellite velocity at its nominal orbit, R_e is Earth's radius (6378km), H_i the nominal orbit and H_e the End-of-Life (EOL) orbit (410km). The ΔV calculations for End-of-Life de-orbiting for each selected orbit are presented in the following table (Table 8).

Table 8: ΔV calculations for End-of-Life de-orbiting

Orbit 1 (VLEO)	
Average altitude, H_i, [km]	405
Velocity, V, [km/s]	7.666
End-of-Life orbit, H_e, [km]	410
ΔV calculated, [m/s]	**0**
Orbit 2 (LEO)	
Average altitude, H_i, [km]	509
Velocity, V, [km/s]	7.608
End-of-Life orbit, H_e, [km]	410
ΔV calculated, [m/s]	**27.588**
Orbit 3 (LEO)	
Average altitude, H_i, [km]	614
Velocity, V, [km/s]	7.550
End-of-Life orbit, H_e, [km]	410
ΔV calculated, [m/s]	**56.096**

Drag compensation

For the calculation of the required ΔV for the drag compensation, Eq. 4 as depicted for the drag acceleration can be transformed to the total drag velocity reduction (that needs to be compensated) over the mission lifetime simply multiplying the drag acceleration with the total time of the satellite (expressed in seconds) in orbit, as follows:

$$\Delta V_{drag} = \frac{1}{2}\rho V^2 C_D \frac{S}{m} T_{mission} \quad \textbf{Eq. 7}$$

In the following table (Table 9) the related ΔV calculations for drag compensation for each selected orbit are presented taking into account the defined satellite and environmental parameters.

Table 9: ΔV calculations for drag compensation

Satellite parameters: $S=0.9x0.7=0.63m^2$, m=200kg, $C_D=2.45$	
Orbit 1 (VLEO)	
Average altitude, [km]	405
Velocity, V, [km/s]	7.666
Density, ρ, [kg/m^3]	$5.784x10^{-12}$
Lifetime, $T_{mission}$, [s]	63113851.949 (2 years)
ΔV calculated, [m/s]	**82.778**
Orbit 2 (LEO)	
Average altitude, [km]	509
Velocity, V, [km/s]	7.608
Density, ρ, [kg/m^3]	$1.380x10^{-12}$
Lifetime, $T_{mission}$, [s]	157784629.874 (5 years)
ΔV calculated, [m/s]	**48.629**
Orbit 3 (LEO)	
Average altitude, [km]	614
Velocity, V, [km/s]	7.550
Density, ρ, [kg/m^3]	$3.785x10^{-13}$
Lifetime, $T_{mission}$, [s]	220898481.823 (7 years)
ΔV calculated, [m/s]	**18.393**

Summary

In the following table (Table 10) the summary results of all applicable ΔV calculations per orbit, as depicted in this chapter, are presented along with the total ΔV requirement for each orbit.

Table 10: ΔV calculations summary

Orbit 1 (VLEO)	
Average altitude, [km]	405
Injection inaccuracy correction, $ΔV_{inj}$, [m/s]	32.664 (28.29%)
Drag compensation, $ΔV_{drag}$, [m/s]	82.778 (71.71%)
End-of-Life De-orbiting, $ΔV_{deorbit}$, [m/s]	0 (0.00%)

Total ΔV calculated, **ΔV_1**, [m/s]	**115.442**
Orbit 2 (LEO)	
Average altitude, [km]	509
Injection inaccuracy correction, ΔV_{inj}, [m/s]	32.384 (29.82%)
Drag compensation, ΔV_{drag}, [m/s]	48.629 (44.78%)
End-of-Life De-orbiting, $\Delta V_{deorbit}$, [m/s]	27.588 (25.40%)
Total ΔV calculated, **ΔV_2**, [m/s]	**108.601**
Orbit 3 (LEO)	
Average altitude, [km]	614
Injection inaccuracy correction, ΔV_{inj}, [m/s]	32.108 (30.12%)
Drag compensation, ΔV_{drag}, [m/s]	18.393 (17.26%)
End-of-Life De-orbiting, $\Delta V_{deorbit}$, [m/s]	56.096 (52.62%)
Total ΔV calculated, **ΔV_3**, [m/s]	**106.597**

From the ΔV calculations summary results, it can be concluded that despite the differences of the three selected orbits especially in terms of atmospheric density (drag compensation) and de-orbiting, the total required ΔVs present quite close values supporting the objective of this study to approach a unique EP system as a suitable option for a wide range (V)LEO missions adjusting only the mission lifetime. The different percentage allocation of each manoeuvre in the total ΔV requirement is a driving factor for the proper sizing of the EP system.

3.3.2 Mass, power and thrust requirements

In order to properly design and size the EP system of the satellite, the available mass and power envelope shall be defined along with the required thrust based on the drag compensation concept to be followed.

Mass and power

The selection of the most suitable EP system will be based on efficient cover of all propulsion needs as estimated in the previous chapter optimising in parallel the payload mass and power fraction (the percentage of payload mass and power in respect to the total satellite mass and power available, the higher the better) for the three orbits.

In this direction, a general simplified rule is set for the available mass, setting it to a representative 10% of the total satellite mass, meaning about **20kg** maximum for the whole EP system. This includes the required propellant, the propellant tank, the propellant management subsystem, the thruster unit and the required power processing unit. From this, a safe assumption regarding the propellant mass is around 30% of the total mass translating in 6.5kg max including margin.

Regarding the available power, according to the platform specifications (Figure 9), the peak power generated reaches 510W during sunlight period. Assuming a 10%

degradation of this value over the years (also a safety margin) around 450W are available during sunlight. The usual concept for the operation of the propulsion system is to utilize it only during sunlight period and when payload and communication subsystems are OFF. Moreover, assuming that data handing subsystem and electric power subsystem consuming a reasonable 60W each and the payload standby power is 30W as specified in §3.2, about **300W** maximum remain available for the operation of the EP system.

Thrust

In order to estimate an adequate thrust envelope to be provided by the EP system, a propulsion operating concept shall be defined. As already described, EP systems are systems with low thrust generation and non-impulsive operation, meaning that they should be operated for a certain period to succeed their propulsion objective. In general, several different control concepts can be applied, from continuous to once per some time firing.

As analysed before, the operation of the EP system is performed only during sunlight period to exploit the available peak power. In this way, continuous firing is not possible, so the general concept to be followed is to operate the EP system once per some completed satellite revolutions based on orbital characteristics and defined concept. In this direction, to create an adequate operation frame applicable for the three selected orbits, the following steps and rules are followed:

- The expected thrust for satellites of this size and power availability normally lies in the range of 1-15mN, as analysed in the related literature review (§2). Increments of 1mN are examined for simplification reasons.
- Calculations for Orbit 1 are assessed first, since the drag compensation ΔV is the maximum calculated.
- Altitude reduction due to atmospheric drag does not exceed 0.5% of the nominal altitude.
- The operating time of the EP system does not exceed the 50% of the total sunlight period TS per orbit leaving adequate time for communications and payload subsystems operation. A minimum of 10min operation per revolution is also applied for efficiency reasons. If the required time is less than 10min, then the operation of the thruster is performed after an integer number of revolutions.
- Based on relevant studies the total firing time shall not exceed 5000hr while total firing cycles (ON/OFF) of the thruster shall not exceed 10000 for reliability reasons. A 20% margin is applied to these maximum values (4000hr and 8000 cycles) for maintenance of the thruster and collision avoidance manoeuvres if required.
- The thrust envelope is verified for the three selected orbits for minimum and maximum thrust values. Propulsion operating concept, total impulse (thrust multiplied with the operating time), total firing time and firing cycles are defined accordingly. Thruster firing can be applied only in integer number of revolutions.

The following equations are used to define the aforementioned parameters:

$$\mathrm{TE} = \frac{2\varphi}{360^{o}}\mathrm{T} \qquad \textbf{Eq. 8}$$

where TE the maximum time of eclipse per orbit (min)

$\varphi = sin^{-1}\left(\frac{\mathrm{R_e}}{\mathrm{H_i}+\mathrm{R_e}}\right)$ Earth's angular radius (deg)

then TS=T-TE, the sunlight time per orbit (min)

$$\Delta\alpha_{rev} = -2\pi\left(C_D\frac{S}{m}\right)\rho\alpha^2$$ where $\Delta\alpha_{rev}$ the altitude reduction per revolution due to atmospheric drag (m)	**Eq. 9**
$$F_{drag} = \frac{1}{2}\rho V^2 C_D S$$ where F_{drag} the atmospheric drag force/thrust (N) then $I_{rev}=F_{drag}\cdot T$, the impulse required per revolution (N·s) and the minimum thrust required is the result of I_{rev} divided by TS/2 meaning that a firing is applied in every revolution for TS/2 maximum time	**Eq. 10**
$$F_{man} = m \cdot a = m\frac{\Delta V}{\Delta t}$$ where F_{man} force/thrust required for any other manoeuvre (N) with ΔV calculated then with known thrust, mass and ΔV, the required firing time is calculated assuming TS/2 firing time per revolution	**Eq. 11**
$$m_p = m\left(1 - e^{\left(-\Delta V/_{I_{sp}\,g}\right)}\right)$$ where m_p the required propellant mass for the defined ΔV and specific impulse based on EP system characteristics, m is the wet mass of the satellite (200kg) then setting the m_p at a maximum of 6.5kg as specified before, the minimum specific impulse to fulfil the defined ΔV of an EP system is estimated	**Eq. 12**

In the following table (Table 11) the preliminary work logic and results for the minimum required thrust, along with impulse and firing calculations are analytically presented to assess all involved parameters that affect the selection of the appropriate EP system suitable for the three selected orbits.

Table 11: Thrust calculations assessment for minimum thrust

Orbit 1 (VLEO)		
Average altitude, H_i, [km]	405	2 years mission
Total ΔV calculated, ΔV_1, [m/s]	115.442	
Orbital period, T, [min]	92.660	
Revolutions per day, η	15.495	
Sunlight time, TS and TS/2 [min]	56.574 || 28.287	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	12.904	157 revolutions required to reach 0.5% max altitude reduction (2025m)
Drag force, F_{drag}, [mN]	0.262	
Impulse required per revolution, I_{rev}, [N·s]	1.458	
Minimum thrust required, [mN]	0.859	<1mN
Minimum specific impulse required, [s]	356	
First assessment with minimum thrust 1mN		

Time required to operate, [min] Starting from revolution number 1 and for every single revolution	24.281	<TS/2 >10min
Firing time of the thruster, [hr]	5337	For drag compensation
Firing cycles (ON/OFF)	11320	
Firing time [hr] and firing cycles required for injection correction	1800 || 3818	
Firing time [hr] and firing cycles required for EOL de-orbit	0 || 0	
Total firing time, [hr]	7137	>4000hr max
Total firing cycles	15138	>8000 max
Reassessment with 2mN thrust (firing in every revolution is not possible due to firing cycles limitation exceeded)		
Time required to operate, [min] Starting from revolution number 1 and for every 2 completed revolution	24.385	<TS/2 >10min
Firing time of the thruster, [hr]	2300	For drag compensation
Firing cycles (ON/OFF)	5660	
Firing time [hr] and firing cycles required for injection correction	900 || 1909	
Firing time [hr] and firing cycles required for EOL de-orbit	0 || 0	
Total firing time, [hr]	3200	<4000hr max
Total firing cycles	7569	<8000 max
Total impulse required, I_{tot}, [kN·s]	23.040	
Orbit 2 (LEO)		
Average altitude, H_i, [km]	509	5 years mission
Total ΔV calculated, ΔV_2, [m/s]	108.601	
Orbital period, T, [min]	94.799	
Revolutions per day, η	15.146	
Sunlight time, TS and TS/2 [min]	59.074 || 29.537	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	3.174	802 revolutions required to reach

		0.5% max altitude reduction (2545m)
Drag force, F_{drag}, [mN]	0.062	
Impulse required per revolution, I_{rev}, [N·s]	0.351	
Minimum thrust required, [mN]	0.198	<1mN
Minimum specific impulse required, [s]	335	
First assessment with minimum thrust 2mN (from Orbit 1 calculations)		
Time required to operate, [min] Starting from revolution number 1 and for every 10 completed revolutions	29.100	<TS/2 >10min
Firing time of the thruster, [hr]	1342	For drag compensation
Firing cycles (ON/OFF)	2766	
Firing time [hr] and firing cycles required for injection correction	880 || 1788	
Firing time [hr] and firing cycles required for EOL de-orbit	780 || 1584	
Total firing time, [hr]	3002	<4000hr max
Total firing cycles	6138	<8000 max
Total impulse required, I_{tot}, [kN·s]	21.611	
Orbit 3 (LEO)		
Average altitude, [km]	614	7 years mission
Total ΔV calculated, ΔV_3, [m/s]	106.597	
Orbital period, T, [min]	96.975	
Revolutions per day, η	14.806	
Sunlight time, TS and TS/2 [min]	61.520 || 30.760	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	0.897	3421 revolutions required to reach 0.5% max altitude reduction (3070m)
Drag force, F_{drag}, [mN]	0.0017	
Impulse required per revolution, I_{rev}, [N·s]	0.097	
Minimum thrust required, [mN]	0.052	<1mN

Minimum specific impulse required, [s]	329	
First assessment with minimum thrust 2mN (from Orbit 1 calculations)		
Time required to operate, [min] Starting from revolution number 1 and for every 38 completed revolutions	30.607	<TS/2 >10min
Firing time of the thruster, [hr]	508	For drag compensation
Firing cycles (ON/OFF)	996	
Firing time [hr] and firing cycles required for injection correction	880 || 1716	
Firing time [hr] and firing cycles required for EOL de-orbit	1550 || 3023	
Total firing time, [hr]	2938	<4000hr max
Total firing cycles	5735	<8000 max
Total impulse required, I_{tot}, [kN·s]	21.155	

As depicted from the calculations inside Table 11, a minimum thrust of 2mN is required to fulfil the propulsion objectives and the propulsion operating concept steps set for the three selected orbits. In the following table (Table 12) the same approach is applied also for maximum thrust to verify if any constraint or limitation is appeared.

Table 12: Thrust calculations assessment for maximum thrust

Orbit 1 (VLEO)		
Average altitude, H_i, [km]	405	2 years mission
Total ΔV calculated, ΔV_1, [m/s]	115.442	
Orbital period, T, [min]	92.660	
Revolutions per day, η	15.495	
Sunlight time, TS and TS/2 [min]	56.574 || 28.287	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	12.904	157 revolutions required to reach 0.5% max altitude reduction (2025m)
Drag force, F_{drag}, [mN]	0.262	
Impulse required per revolution, I_{rev}, [N·s]	1.458	
Minimum thrust required, [mN]	0.859	<1mN
Minimum specific impulse required, [s]	356	

<table>
<tr><td colspan="3">First assessment with maximum thrust 15mN</td></tr>
<tr><td>Time required to operate, [min]
Starting from revolution number 1 and for every 17 completed revolutions</td><td>27.463</td><td><TS/2
>10min</td></tr>
<tr><td>Firing time of the thruster, [hr]</td><td>305</td><td rowspan="2">For drag compensation</td></tr>
<tr><td>Firing cycles (ON/OFF)</td><td>666</td></tr>
<tr><td>Firing time [hr] and firing cycles required for injection correction</td><td>121 || 257</td><td></td></tr>
<tr><td>Firing time [hr] and firing cycles required for EOL de-orbit</td><td>0 || 0</td><td></td></tr>
<tr><td>Total firing time, [hr]</td><td>426</td><td><4000hr max</td></tr>
<tr><td>Total firing cycles</td><td>923</td><td><8000 max</td></tr>
<tr><td>Total impulse required, I_{tot}, [kN·s]</td><td>22.992</td><td></td></tr>
<tr><td colspan="3">Orbit 2 (LEO)</td></tr>
<tr><td>Average altitude, H_i, [km]</td><td>509</td><td>5 years mission</td></tr>
<tr><td>Total ΔV calculated, ΔV_2, [m/s]</td><td>108.601</td><td></td></tr>
<tr><td>Orbital period, T, [min]</td><td>94.799</td><td></td></tr>
<tr><td>Revolutions per day, η</td><td>15.146</td><td></td></tr>
<tr><td>Sunlight time, TS and TS/2 [min]</td><td>59.074 || 29.537</td><td></td></tr>
<tr><td>Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]</td><td>3.174</td><td>802 revolutions required to reach 0.5% max altitude reduction (2545m)</td></tr>
<tr><td>Drag force, F_{drag}, [mN]</td><td>0.062</td><td></td></tr>
<tr><td>Impulse required per revolution, I_{rev}, [N·s]</td><td>0.351</td><td></td></tr>
<tr><td>Minimum thrust required, [mN]</td><td>0.198</td><td><1mN</td></tr>
<tr><td>Minimum specific impulse required, [s]</td><td>335</td><td></td></tr>
<tr><td colspan="3">First assessment with maximum thrust 15mN</td></tr>
<tr><td>Time required to operate, [min]
Starting from revolution number 1 and for every 75 completed revolutions</td><td>29.244</td><td><TS/2
>10min</td></tr>
<tr><td>Firing time of the thruster, [hr]</td><td>180</td><td></td></tr>
</table>

Firing cycles (ON/OFF)	369	For drag compensation
Firing time [hr] and firing cycles required for injection correction	120 || 244	
Firing time [hr] and firing cycles required for EOL de-orbit	102 || 207	
Total firing time, [hr]	402	<4000hr max
Total firing cycles	820	<8000 max
Total impulse required, I_{tot}, [kN·s]	21.695	
Orbit 3 (LEO)		
Average altitude, [km]	614	7 years mission
Total ΔV calculated, ΔV_3, [m/s]	106.597	
Orbital period, T, [min]	96.975	
Revolutions per day, η	14.806	
Sunlight time, TS and TS/2 [min]	61.520 || 30.760	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	0.897	3421 revolutions required to reach 0.5% max altitude reduction (3070m)
Drag force, F_{drag}, [mN]	0.0017	
Impulse required per revolution, I_{rev}, [N·s]	0.097	
Minimum thrust required, [mN]	0.052	<1mN
Minimum specific impulse required, [s]	329	
First assessment with maximum thrust 15mN (from Orbit 1 calculations)		
Time required to operate, [min] Starting from revolution number 1 and for every 284 completed revolutions	30.607	<TS/2 >10min
Firing time of the thruster, [hr]	68	For drag compensation
Firing cycles (ON/OFF)	133	
Firing time [hr] and firing cycles required for injection correction	119 || 232	
Firing time [hr] and firing cycles required for EOL de-orbit	208 || 406	

Total firing time, [hr]	395	<4000hr max
Total firing cycles	771	<8000 max
Total impulse required, I_{tot}, [kN·s]	21.330	

From this calculations table (Table 12) it becomes evident that no limitation or constraint appears due to maximum thrust level. On the contrary, using an EP system with higher thrust capability drives to quite lower thruster firing utilization (less time required to perform the manoeuvres) and lower firing cycles.

4. EP SYSTEM TRADE-OFF

4.1 EP suitable systems

Having set in the previous chapters the required EP system specifications envelope that fulfil the mission objectives for the three selected orbits, all available suitable EP systems are presented, evaluated and compared in this chapter. The main characteristics that are taken into consideration are:

- Power (up to 300W)
- Mass (up to 20kg, including propellant, propellant tank, propellant management system and PPU)
- Specific impulse (higher than 356s)
- Thrust (2-15mN)
- Total impulse (more than 25kN·s)
- Firing cycles (>10000)
- Lifetime (>5000hr)

The main objective of this trade-off analysis is to select the EP system that maximizes payload mass and power fraction for the three orbits, meaning that the required propulsion manoeuvres as estimated in §3.3, are fully covered with the minimum power and mass consumed by the EP system.

Thus, all available information for the suitable EP systems as found in the related literature and market research, is analytically depicted in the following table (Table 13). This table is the basis of this trade-off analysis creating a contemporary and completed database of the EP systems able to support microsatellite missions in the specified (V)LEO range.

The following rules are applied on the selection of the EP systems:

- Systems that their power or thrust range lies inside the specified ranges are included in the database with appropriate adjustments of their related characteristics (specific impulse etc.).
- Multiple systems of the same manufacturer (either of different type or similar type with different power or thrust specifications) are included in the database.
- A maximum quantity of five similar systems to cover the specified needs is foreseen.
- When a range of power, thrust and specific impulse is presented for a system the lower power is connected with the lower thrust and the lower specific impulse and vice versa.
- The mass and the dimensions of the system include the whole thruster unit (thruster, cathode, electrodes, frame etc.) unless otherwise noted.
- The references are either related papers or related datasheets of the systems. If no such data can be found, NASA state of the art document is used, [70].
- The mandatory available parameters to include a system are power, mass, specific impulse and thrust.
- A level of EP system market maturity and/or space proven compatibility shall be demonstrated, so research or under development systems are excluded.

Table 13: Suitable EP systems summary

#	Manufacturer	Model	Type	Quant.	Power [W]	Mass [g]	I_{sp} [s]	Thrust [mN]	Propellant	Dimensions [mm]	Remarks	Ref.
1	Rafael	R-200	HET	1	150-300	875*	800-1100	5-14	Xe/Kr	100x120x90*	*Based on HET300	[27], [37], [70]
2	Sitael	HT100	HET	1	100-300	800*	900-1300	6-15	Xe	65x65x100**	*With 2 cathodes **Cathodes excluded	[29], [30], [70]
3	Orbion	Aurora	HET	1	100-250	1500	950-1320	5.7-15	Xe/Kr	167x92x98		[31], [70]
4	ExoTerra	Halo	HET	1	125-300	750	730-1100	4-15	Xe/Kr	0.25U		[71], [70]
5	EDB Fakel	SPT-50M	HET	1	225	1320	930	14.8	Xe	169x120x88		[33], [70]
6	EDB Fakel	SPT-50	HET	1	225	1230	860	14	Xe	160x120x91		[33], [70]
7	SETS	ST-25	HET	1	150-200	750	1100-1300	5-11	Xe/Kr	79x79x79.5*	*Cathode excluded	[34], [72], [70]
8	SETS	ST-40	HET	1	250-300	1100*	1450	14-15	Xe/Kr/Ar	140x117x122**	*With 2 cathodes **Cathodes excluded	[35], [70]
9	Busek	BHT-100	HET	1	100	1160	1000	7	Xe/I	55x80x80		[36], [73], [70]
10	Busek	BHT-200	HET	1	200	1100	1390	13	Xe/I	-		[36], [74], [70]
11	Exotrail	ExoMG-micro	HET	1	150	850*	1000	7	Xe	75x95x95	*Estimated based on datasheet	[75], [76], [70]
12	Aliena	Music	HET	1	100	~750*	1000	3	Xe/Kr	-	*Estimated based on datasheet	[77]
13	LAJP	HEET-05	HET	1	100	470	950	5	Xe	-		[79]
14	LAJP	HEET-10	HET	1	200	520	1070	10.2	Xe	-		[79]
15	Ariane Group	RIT 10 EVO	RIT	1	145	1800	1900	5	Xe	186x186x134		[38], [78], [70]
16	T4-i	Regulus-50*	RIT	4	4x50	4x3000	550	4x0.65	I	-	*Integrated system (tank, propellant and PPU included) for 6kNs	[41], [70]
17	Busek	BIT-3*	RIT	2	2x75	2x2900	2150	2x1.1	I	2x180x88x102	*Integrated system (tank, 1.5kg of propellant and PPU included)	[36], [80], [70]

18	ThrustMe	NPT300*	RIT	1	200-300	10000	1200-1800	8-10	Xe/I	-	*Integrated system (tank, propellant and PPU included) for 50kNs	[81], [70]
19	ThrustMe	NPT30-I2*	RIT	2	2x65	2x1700	2400	2x1.1	I	2x93x93x155	*Integrated system (tank, propellant and PPU included) for 9.5kNs	[82], [70]
20	QinetiQ	T5	RIT	1	60-300	2000	500-2100	2-10	Xe	190x190x242		[70]
21	Miles Space	M1.4	Plasma	1	19	535	1340	2.8	H_2O based	90x90x95		[83], [70]
22	Accion	Tile 3*	Electrospray	5	5x20	5x1250	1650	5x0.45	Ionic	5x1U	*Integrated system (tank, propellant and PPU included) for 755Ns	[43], [84], [70]
23	Enpulsion	Micro R^3*	FEEP	2	2x120	2x3900	2000	2x1.35	Indium	2x140x120x98.6	*Integrated system (tank, propellant and PPU included) for 50kNs	[44], [85], [70]

4.2 EP system selection

In order to select the most suitable EP system for the required applications, the most important parameters are extracted from Table 13, so as to reflect the main target of this analysis, that is to optimise payload mass fraction and payload power fraction for the three orbits with one EP system. Since, mass is not only affected by the mass of the thruster, but also from the mass of the required propellant, the specific impulse is also included in the following analysis (the higher the specific impulse the lesser the required propellant). In this direction, in the following tables (Table 14, Table 15 and Table 16), the ten best ranked EP systems for each critical parameter (mass, power, specific impulse) are presented in descending order starting from the best one.

For easier reference, the tables include the # number of the initial table (Table 13) and the three critical parameters starting in each case from the one examined. In the cases of power and specific impulse, when a range is applicable, the lowest power is used (reflecting the lowest specific impulse) and the highest specific impulse is used (reflecting the highest power).

Table 14: EP systems ranking based on related system mass

Ranking	**# (Table 13)**	**System**	**Mass [g]**	Power [W]	I_{sp} [s]
1	13	LAJP HEET-05	470	100	950
2	14	LAJP HEET-10	520	200	1070
3	21	Miles Space M1.4	535	19	1340
4	4	ExoTerra Halo	750	125-300	730-1100
4	7	SETS ST-25	750	150-200	1100-1300
4	12	Aliena Music	750	100	1000
7	2	Sitael HT100	800	100-300	900-1300
8	11	Exotrail ExoMG-micro	850	150	1000
9	1	Rafael R-200	875	150-300	800-1100
10	8	SETS ST-40	1100	250-300	1450
10	10	Busek BHT-200	1100	200	1390

Table 15: EP systems ranking based on related system power

Ranking	**# (Table 13)**	**System**	**Power [W]**	I_{sp} [s]	Mass [g]
1	21	Miles Space M1.4	19	1340	535
2	20	QinetiQ T5	60	500	2000
3	2	Sitael HT100	100	900	800
3	3	Orbion Aurora	100	950	1500

3	9	Busek BHT-100	100	1000	1160
3	12	Aliena Music	100	1000	750
3	13	LAJP HEET-05	100	950	470
3	22	Accion Tile 3	100	1650	6250
9	4	ExoTerra Halo	125	730	750
10	19	ThrustMe NPT30-I2	130	2400	3400

Table 16: EP systems ranking based on related system specific impulse

Ranking	**# (Table 13)**	**System**	**I_{sp} [s]**	Mass [g]	Power [W]
1	19	ThrustMe NPT30-I2	2400	3400	130
2	17	Busek BIT-3	2150	5800	150
3	20	QinetiQ T5	2100	2000	300
4	23	Enpulsion Micro R3	2000	7800	240
5	15	Ariane Group RIT 10 EVO	1900	1800	145
6	18	ThrustMe NPT300	1800	10000	300
7	22	Accion Tile 3	1650	6250	100
8	8	SETS ST-40	1450	1100	300
9	10	Busek BHT-200	1390	1100	200
10	21	Miles Space M1.4	1340	535	19

For the selection of the most suitable EP system for the defined applications the following rules are applied:

- The system shall be present in all three ranking tables (Table 14, Table 15 and Table 16) and,
- No weighting factors are applied (the three critical parameters are considered equal), thus the system with the lowest ranking sum among all is the winner.

Following this evaluation procedure, the only system present in all three ranking tables is EP system **Miles Space M1.4** (#21) that is eventually the most suitable system for the required applications.

5. EP SYSTEM ARCHITECTURE

Based on the previous evaluation trade-off of the available and suitable EP systems for the thesis defined application, the following EP system (Table 17) was selected as the most appropriate according to the evaluation rules applied.

Table 17: Selected EP system main characteristics

Manufacturer	Model	Type	Quant.	Power [W]	Mass [g]	I_{sp} [s]	Thrust [mN]	Propellant	Dimensions [mm]
Miles Space	M1.4	Plasma	1	19	535	1340	2.8	H_2O based	90x90x95

According to the thruster interface and user manual [86], the M1.4 device uses ConstantQ propulsion technology, converting electricity and water vapor into thrust. Thrust is derived from electrostatic acceleration of separated ions and electrons using a combination of classic collisionless flow and electrohydrodynamic regimes, all working in a cycle determined by propellant temperature, input power and device geometry.

The operating cycle is self-stabilizing and does not require real time active control once initiated, though altering temperature and/or power will alter delivered thrust. The process is self-neutralizing and does not require a neutralizer device. Pressures throughout the system generate water vapor through sublimation, avoiding the need for water to boil and tolerating ice as the propellant.

5.1 EP system overview

The EP system consists of four thrust heads, a propellant tank with an integrated heater, four main valves for the control of the flow of the propellant, associated pressure and temperature sensors and a power processing and control unit. +5V are required for the logic part and +12V for the power (thrust) part. The maximum required system power is 14W (maximum flow rate with all thrusters active), while the required thermal power is 11.5W (Heater on, no thrusters active).

Each thrust head contains integrated high voltage generation electronics and a valve, to generate and accelerate plasma. Any combination of the four thrust heads (including all four operating) can be applied simultaneously, reaching a maximum thrust of 2.8mN.

The propellant tank shall hold pure liquid water (H_2O) and include enough volume and design features to tolerate freezing the water into solid ice and yet still generate thrust.

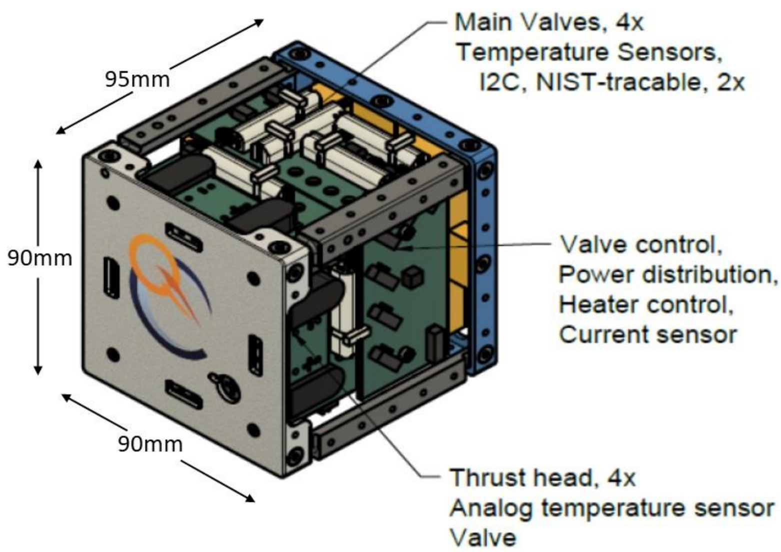

Figure 10: Viewpoint of the main thruster part, [86]

Temperature management is achieved with a high-density flat heater on the tank, a valve self-heating feature, and routing of electronics waste heat to minimize freezing of water vapor. Thrust is attained with a wide range of temperatures, including a tank full of frozen water ice. Water vapor, not liquid water, reaches the thrust heads and is converted into plasma and thrust. Should liquid water get near the thrust heads, it would rapidly sublimate into vapor due to vacuum exposure.

A functional block diagram of the main subsystems of the EP system is depicted below (Figure 11), while in Figure 10 a viewpoint of the main thruster part and its integrated components is presented.

The thruster unit already incorporates the logic circuit board that needs to be supplied with 5V and controlled accordingly by the external power unit and control module, the main valves module (contains 4 valves, arranged in 2 sets of 2 valves, denoted the "Horizontal" and "Vertical" valves, each set can be actuated simultaneously, two digital high precision temperature sensors placed so that flowing water vapor will change their temperature readings) and the thruster heads with their HV electronics that are supplied with 12V by the power unit. The heater module is integrated inside the propellant tank to ensure proper heating and temperature control of the propellant and it is supplied with 12V by the power unit. A dedicated external board is required to interface the satellite with the EP system and provide the required power and control.

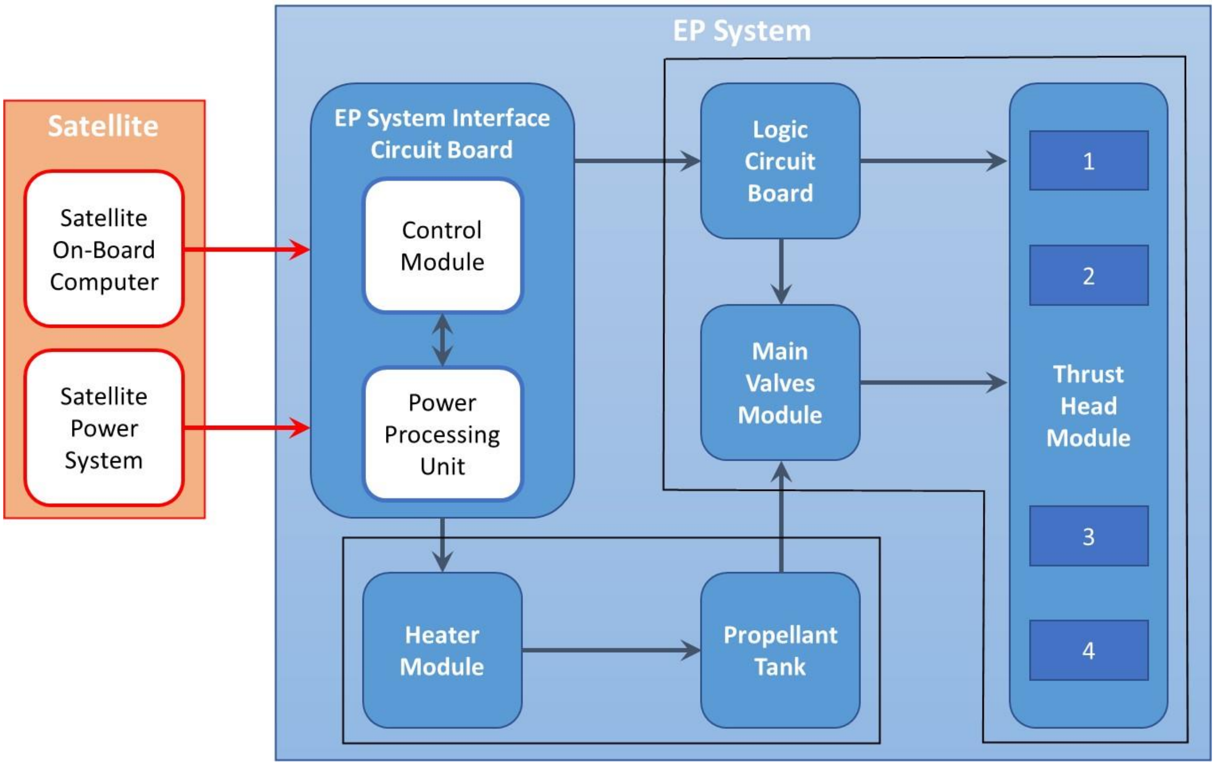

Figure 11: Functional block diagram of the EP system

5.2 Thruster operation logic

As denoted in [86], a ConstantQ thruster consists of a plasma formation region containing spark electrodes, two exhaust ports each one ringed by acceleration electrodes and a single power supply providing spark and acceleration power. Vapor enters the plasma formation region, expanding and changing pressure on its path towards the exhaust ports. Paschen's law ensures a spark occurs within the vapor at the point where the supply voltage meets the pressure on the Paschen curve.

Each exhaust port is surrounded by high voltage electrodes. One exhaust port voltage acts to extract positive ions from the plasma, while the other affects electrons. Electrons, being far less massive than ions, leave the plasma before ions, generating thrust from their interaction with the acceleration electrodes. Once outside the thruster, the electrons form a virtual cathode that pulls upon the ions remaining within the thruster.

As the ions leave, thrust is also obtained from the acceleration electrodes. However, the ions also derive kinetic energy from the virtual cathode, slowing the exhaust electrons and even causing electrons to flow back toward the thruster. This gives an increased acceleration voltage upon the ions, expanding the classic limits for space-charge flow rate and thrust density.

As the ions exit the thruster, they meet the returning electrons, neutralizing the plasma. With water vapor, the interface between exiting ions and returning electrons appears as a white-hot sphere 5-8mm outside the ion's exhaust port. This phenomenon is believed to be due to the presence of multiple ion species with different velocity profiles. A representative scheme of the main operations and the related regimes of the thruster are presented in the following figure (Figure 12).

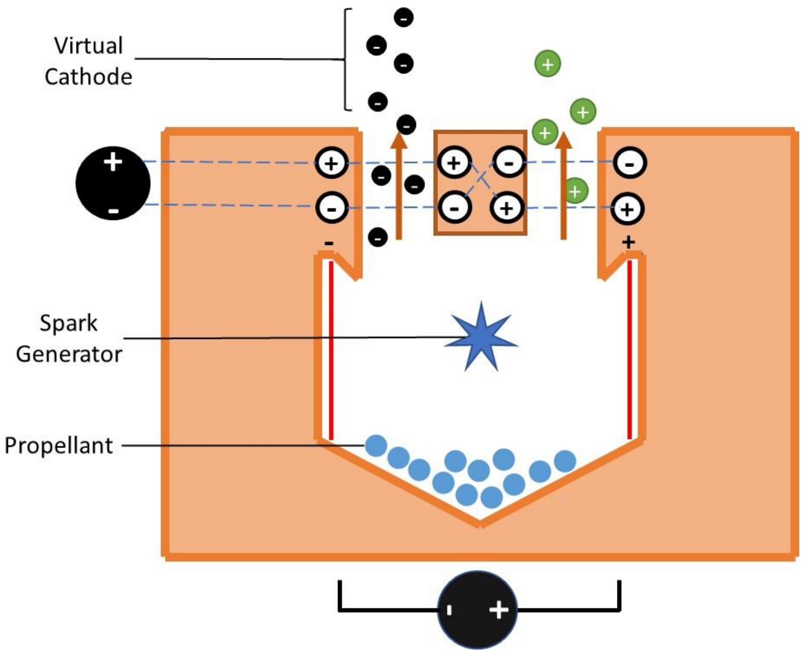

Figure 12: Thruster operation logic representation

A resonance occurs between the incoming gas pressure, spark push back and plasma drain rate through the exhaust ports (as driven by the supply's high voltage which can be varied to align with mission I_{sp}). The thruster utilizes a very specific geometry to drive this resonance, minimize wear, reduce power supply complexity, and reduce flight computing demands.

The main thruster operation procedure is then described by the following steps operated by the satellite's on-board computer according to the mission requirements:

- Heat the propellant tank
- Measure temperature with and without flowing propellant:

  Water vapor is passively generated via sublimation of liquid water or solid ice. Sublimation occurs when system pressure is less than the propellant vapor pressure, which varies with temperature.
- Set valve actuation direction:

Valves are controlled in a two-step process. First, the valve direction must be established.

- Actuate valves:
  Second, the desired valve must be actuated by briefly connecting it to the valve direction power lines. The valve will open or close depending upon its physical configuration and the valve direction setting. The main valves open with opposite polarity than the thrust head valves.
- Enable/disable high voltage:
  When the water vapor reaches a thrust head, it is made into plasma via a high voltage spark, then accelerated by high voltage fields. The amount of thrust generated depends upon the propellant temperature and technically upon the voltage actually provided on the 12V input line. Given sufficient mass flow, the spark and acceleration cycle occur on their own when high voltage is enabled.

5.3 Propellant mass calculation

Having selected the most suitable system for the application, the first step for the overall design of the EP system, is the calculation of the required propellant (H_2O) mass. The following equation (Eq. 12 as depicted in §3.3.1) is used for this purpose:

$$m_p = m\left(1 - e^{\left(-\Delta V / I_{sp}\, g\right)}\right) = 1.749\text{kg}$$

where m_p the required propellant mass for the defined ΔV (115.442m/s, the maximum calculated according to Table 10), I_{sp} the specific impulse (1340s according to the selected EP system) and m is the wet mass of the satellite (200kg). Thus, a total of 1.749kg of propellant is required to cover the needs of the defined application.

In order to cover any mission uncertainties (e.g. collision avoidance, mission adjustment after placing in orbit, mission lifetime extension, safety issues) an extra propellant margin of 25% is applied, while a 5% of residual (not usable for any possible reason) propellant is assumed. In this frame, the overall required propellant is calculated to be **2.274kg**.

5.4 Propellant tank dimensioning

Following the propellant mass calculation, the propellant tank dimensioning can be extracted. Since the propellant tank shall integrate the heater module and be directly connected with the thruster valve subsystem, it is efficient to be attached downstream the thruster unit. In this frame, the maximum width of the tank cannot exceed the dimension of the thruster which is a square of 90mm.

The usual shapes of the propellant tank are either spherical or cylindrical. In this case, since the attaching position of the tank has a quite restricted area (diameter of 90mm max), a cylindrical propellant tank is applied, leaving the elongated side of the cylinder to be adjusted according to the requirements of the mission.

The propellant is liquid water, with a density at room temperature of 997kg/m^3. Applying the well-known equation of volume, density and mass, the minimum volume of the tank is calculated 2.281L. Taking into account that the diameter of the cylinder is 90mm, the minimum length of the cylinder is calculated to be 358.5mm. Thus, applying a margin, the cylinder is dimensioned to have **a diameter of 90mm and length of 400mm**, reaching a total volume of **2.545L**.

According to the thruster datasheet ([86]), the main material used for the thruster is **Aluminum 6061-T6**. In order to avoid any material safety issues and corrosion due to vacuum in space, the same material is used for the wall of the propellant tank. This material presents the following properties: tensile strength σ_t=240MPa, density ρ_{al}=2700kg/m^3. Based on thruster related datasheet ([86]), the tank walls shall be able to withstand a burst (maximum) pressure of 65psi (4.5bar). Solving for the required thickness of the tank walls to withstand the defined propellant conditions, 0.084mm is calculated. Applying adequate margin for safety and applicability reasons, the thickness of the propellant tank walls is designed to be 1mm. This results in a calculated mass for the propellant tank of **340g**.

5.5 Power and control unit

The last part of the EP system that needs to be studied, is the power and control unit. It is actually an external board dedicated to interface the satellite On-Board Computer (OBC) that controls all operations of the satellite with the Control Module of the EP system and the Power Distribution Unit of the satellite with the PPU of the EP system.

This board includes:

- A microcontroller subunit that directly communicates with the OBC and receives commands and sends telemetry. This workflow is programmed based on the mission characteristics and requirements and according to predefined procedures. It is supplied by the PPU with 5V and 3.3V and drives the internal thruster logic circuit board to control the operation of the thruster. Specific timing of all commands and telemetry is applied to ensure that all valves operation, heating of the propellant, flow of the propellant and HV generation are controlled properly.
- A power subunit that receives unregulated voltage (usually in the range of 22-34V) form the power unit of the satellite and generates all required voltages for the operation of the EP system. Especially:
  - 5V, 1.5W for the operation of the microcontroller of the Control Module
  - 3.3V, 1.5W for the operation of the microcontroller of the Control Module
  - 5V, 0.75W for the logic circuit board
  - 5V, 1.25W for the activation of each valve (activated one by one)
  - 12V, 11.5W for the heater module (when Heater is on, no thrusters active)
  - 12V, 3.5W for the maximum thrust production per thrust head (14W in total for all operating)

  Based on the above analysis, the maximum required power by the PPU is then **19W**.

An estimation for the dimensions and the mass of a board of these specifications, including margin, is **150x150x40mm and 1kg** based on relevant systems available in the market and similar studies.

5.6 EP system summary characteristics

In the following table (Table 18) the EP system summary properties (mass and dimensions) are provided based on the previous paragraphs analysis. An extra margin of 5% is then applied for any uncertainty compensation, leading to the final mass and dimensions of the selected EP system. The total dimensions are extracted assuming a serial placement of the subsystems.

Table 18: EP system summary properties

EP Subsystem	Mass [g]	Dimensions [mm]
Thruster unit	535	90x90x95
Propellant tank (including propellant)	2274+340=2614	Cylindrical, diameter 90, length 400
Heater module	500	Integrated in the tank
PPU and Control Module	1000	150x150x40
Total	4649	~240x490x95
Margin 5%	232	12x25x5
EP System total	**4881**	**252x515x100**

From this table, it can be extracted that the approached concept of the EP system that fully covers the defined applications of the mission, will have a total mass of 4.881kg and rough dimensions of 252x515x100mm^3, constituting a quite efficient sized system.

5.7 EP system operation summary

As a final step for the illustration of the EP system operation, a summary table (Table 19) with the defined operating mission characteristics of the studied EP system for the three orbit scenarios is depicted based on the same calculation principles presented in §3.3.1. The EP system is assumed operating at the maximum thrust applicable of 2.8mN (all four thrust heads operating) with a specific impulse of 1340s.

The calculations presented inside this table are reflecting the baseline scenario of operations for the EP system and the typical mission characteristics as analytically described in §3.3.

Table 19: EP system operation summary

Orbit 1 (VLEO)		
Average altitude, H_i, [km]	405	2 years mission
Total ΔV calculated, ΔV_1, [m/s]	115.442	
Orbital period, T, [min]	92.660	
Revolutions per day, η	15.495	
Sunlight time, TS and TS/2 [min]	56.574 || 28.287	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	12.904	157 revolutions required to reach 0.5% max altitude reduction (2025m)
Drag force, F_{drag}, [mN]	0.262	

Impulse required per revolution, I_{rev}, [N·s]	1.458	
Thrust provided, [mN]	2.8	
Specific impulse, [s]	1340	
Time required to operate, [min] Starting from revolution number 1 and for every 2 completed revolution	17.407	<TS/2 >10min
Firing time of the thruster, [hr]	1642	For drag compensation
Firing cycles (ON/OFF)	5660	
Firing time [hr] and firing cycles required for injection correction	650 || 1379	
Firing time [hr] and firing cycles required for EOL de-orbit	0 || 0	
Total firing time, [hr]	2292	<4000hr max
Total firing cycles	7039	<8000 max
Total impulse provided, [N·s]	23103	
Propellant loaded, [kg]	2.274	
Propellant consumption typical scenario, [kg]	1.749	
Spare propellant, [kg]	0.525	
Orbit 2 (LEO)		
Average altitude, H_i, [km]	509	5 years mission
Total ΔV calculated, ΔV_2, [m/s]	108.601	
Orbital period, T, [min]	94.799	
Revolutions per day, η	15.146	
Sunlight time, TS and TS/2 [min]	59.074 || 29.537	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	3.174	802 revolutions required to reach 0.5% max altitude reduction (2545m)
Drag force, F_{drag}, [mN]	0.062	
Impulse required per revolution, I_{rev}, [N·s]	0.351	
Thrust provided, [mN]	2.8	
Specific impulse, [s]	1340	

Time required to operate, [min] Starting from revolution number 1 and for every 10 completed revolutions	20.874	<TS/2 >10min
Firing time of the thruster, [hr]	962	For drag compensation
Firing cycles (ON/OFF)	2766	
Firing time [hr] and firing cycles required for injection correction	650 || 1320	
Firing time [hr] and firing cycles required for EOL de-orbit	550 || 1117	
Total firing time, [hr]	2162	<4000hr max
Total firing cycles	5203	<8000 max
Total impulse provided, [N·s]	21796	
Propellant loaded, [kg]	2.274	
Propellant consumption typical scenario, [kg]	1.646	
Spare propellant, [kg]	0.628	
Orbit 3 (LEO)		
Average altitude, [km]	614	7 years mission
Total ΔV calculated, ΔV_3, [m/s]	106.597	
Orbital period, T, [min]	96.975	
Revolutions per day, η	14.806	
Sunlight time, TS and TS/2 [min]	61.520 || 30.760	
Altitude reduction per revolution, $\Delta\alpha_{rev}$, [m]	0.897	3421 revolutions required to reach 0.5% max altitude reduction (3070m)
Drag force, F_{drag}, [mN]	0.0017	
Impulse required per revolution, I_{rev}, [N·s]	0.097	
Thrust provided, [mN]	2.8	
Specific impulse, [s]	1340	
Time required to operate, [min] Starting from revolution number 1 and for every 38 completed revolutions	21.932	<TS/2 >10min

Firing time of the thruster, [hr]	364	For drag compensation
Firing cycles (ON/OFF)	996	
Firing time [hr] and firing cycles required for injection correction	640 || 1716	
Firing time [hr] and firing cycles required for EOL de-orbit	1100 || 2146	
Total firing time, [hr]	2104	<4000hr max
Total firing cycles	4858	<8000 max
Total impulse provided, [N·s]	21210	
Propellant loaded, [kg]	2.274	
Propellant consumption typical scenario, [kg]	1.616	
Spare propellant, [kg]	0.658	

6. CONCLUSIONS AND FUTURE CONSIDERATIONS

The possibility of applying a conceptual design approach for the utilization of the electric propulsion system for small satellite (V)LEO missions is examined and assessed inside this thesis. This final chapter highlights the most important outcome of this study and the related contributions that the thesis can provide to the state of the art and identifies future considerations that will further expand this work.

This conceptual study approach that is successfully presented inside this thesis, shows the potential use of such studies. One fully specified and optimized EP system that can cover the needs of three different orbit scenarios of a quite representative mission application that fits most of the current space market needs is the basis for concept driven studies in this field. The system is optimized in terms of payload mass and power fraction providing a useful tool to any satellite user to adopt it and easily configure it to any (V)LEO mission. This becomes the most important and novel characteristic of this study, with this conceptual focus on the EP system being the core of the whole study consideration.

6.1 Thesis summary

This study aims to provide some information about the outburst of the new space and the influence of this in the whole space mission management, especially for small satellite applications. In this direction, the first objective of this thesis is to pinpoint the advantages of lowering the altitude to LEO and VLEO range and identify the related mission challenges, investigate the possible applications and payloads of such missions and focus on the importance of the satellite propulsion system to satisfy the mission requirements. An initial rough presentation of the two main satellite propulsion technologies, chemical and electric, is provided, highlighting the advantageous utilization of electric propulsion systems in (V)LEO applications and the growing perspectives they offer to the space market.

In this frame, the literature and state of the art assessment presented in §2, tries to deeply investigate the low power EP systems (<200W) along with their main operating principles and design characteristics providing a detailed analysis of the most common low power EP solutions. Moreover, an overview of LEO (up to 600km) and VLEO micro and mini satellites (up to 200kg) mission cases is carried out, concentrating mainly on EP systems applications and the related mission analysis. A critical analysis based on the literature review outcome to assess any lessons learnt through this review and highlight any possible research gaps follows to conclude this analysis.

The thesis proceeds with the mission analysis part presented in §3 where the satellite requirements and specifications are analytically defined along with the related application (earth observation SAR). Based on those parameters, all mission characteristics are determined and calculated to support the design of the electric propulsion system. The related orbits are selected (one VLEO and two LEO), the utilization (injection inaccuracies correction, End-of-Life de-orbiting and drag compensation) of the EP system is defined, the atmospheric characteristics (atmospheric density model and drag coefficient) are assessed, the related ΔV calculations are performed for the required use of the EP system and mass, power and thrust requirements are determined.

With all required EP system specifications and operations envelope analytically set in the previous chapter to satisfy the mission requirements for the three selected orbits, all available and suitable EP systems are identified and assessed in §4. The major goal of this in-depth trade-off evaluation is to select the EP system that optimizes the payload mass and power fraction for the three selected orbits and provide the basis for the conceptual design of the whole EP system.

Following the evaluation trade-off of the EP systems, the EP system selected is conceptually defined in §5. The EP system architecture is determined including all required subsystems and presented in a functional block diagram. The subsystems characteristics are then estimated, beginning with the integrated thruster unit and its components, the propellant mass and tank dimensioning and the specification of the power and control unit features. The EP system conceptual study is concluded with the summarized system, mission and operation characteristics.

6.2 Contribution to the state-of-the-art

The main contribution points that this study offers to the state of the art are:

- Combination of all available low power electric propulsion system information in one concept
- Concentrate the analysis on the EP system concept as a part of realistic applications
- Study and analysis of a representative and already available electric propulsion system as a well-defined scenario that can efficiently fulfil the main mission requirements
- Conceptual study approach that fully specifies the EP system for three different (V)LEO mission scenarios of an earth observation application that covers most of the current space market needs
- A trade-off database with all the important low power EP systems data is provided
- The EP system is defined in detail with a focus on the maximization of the payload mass and power fraction
- The creation of a very effective tool for any satellite integrator to evaluate this system and easily adjust it to any (V)LEO mission requirement
- A feasible plug and play EP solution is studied, which is easy to integrate and configure leading to a faster, cheaper and more reliable EP system integration to any satellite requirement

6.3 Future considerations

The main points that need to be considered in the future in order to further develop the current study analysis and expand its findings, are:

- Safety issues related with the EP system and its integration inside the satellite are not considered and may affect the design approach
- The mechanical aspects of the EP system (structure, materials, placement) are not examined, and they are required to complete the EP system design
- An analytical propellant assessment shall be carried out to investigate the features and related pros and cons of the available propellants used for electric propulsion leading to further completion of the EP system design approach
- More manoeuvres can be evaluated for further utilization of the EP system like collision avoidance, altitude change, plane change and phase change (to enable satellite servicing or re-positioning)
- Perform specific analysis related with the mission lifetime and how it can be expanded focusing on specialized operating scenarios

- A more analytical design of the electric propulsion subsystems shall be carried out to fully specify all the components to be used
- Space environmental aspects (e.g. temperature, radiation, vacuum) can be studied that may affect the operation of the EP system

ABBREVIATIONS - ACRONYMS

ABEP	Atmosphere-Breathing Electric Propulsion
AOCS	Attitude and Orbital Control Subsystem
COTS	Commercial of the shelf
DC	Direct Current
EO	Earth Observation
EP	Electric Propulsion
EOL	End-of-Life
ECSS	European Cooperation for Space Standardization
EUV	Extreme Ultraviolet
FEEP	Field Emission Electrostatic Propulsion
GPE	Gauss's Planetary Equations
GEO	Geostationary Orbit
GNSS	Global Navigation Satellite System
GIT	Gridded Ion Thruster
HET	Hall-effect Thruster
HA	High solar and geomagnetic activity
IADC	Inter-Agency Space Debris Coordination Committee
IoT	Internet of Things
ILIS	Ionic Liquid Ion Source
LEO	Low Earth Orbit
LA	Low solar and geomagnetic activity
MPD	Magnetic Plasma Dynamic
MEO	Medium Earth Orbit
OBC	On-Board Computer
PPU	Power Processing Unit
PPT	Pulsed Plasma Thruster
RF	Radio Frequency
RIT	Radio-Frequency Ion Thruster
RAAN	Right Ascension of the Ascending Node
SSO	Sun-Synchronous Orbit
SAR	Synthetic-aperture Radar
TIR	Thermal Infrared

UHF	Ultra High Frequency
VAT	Vacuum Arc Thruster
VHF	Very High Frequency
VLEO	Very Low Earth Orbit

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