Space Propulsion Technology for Small Spacecraft

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David Krejci and Paulo Lozano

***Abstract*—As small satellites become more popular and capable, strategies to provide in-space propulsion increase in importance. Applications range from orbital changes and maintenance, attitude control and desaturation of reaction wheels to drag compensation and de-orbit at spacecraft end-of-life. Space propulsion can be enabled by chemical or electric means, each having different performance and scalability properties. The purpose of this review is to describe the working principles of space propulsion technologies proposed so far for small spacecraft. 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-liter, 1 kg CubeSats and smaller.**

***Index Terms*—Space propulsion, Micropropulsion, Cubesat propulsion.**

I. Introduction

PROPULSION capability for satellites has been a priority for satellite developers since the early stages of spaceflight to increase spacecraft capabilities [1]–[5], with the first instances of electric propulsion (EP) occurring in 1964 onboard of the Russian Zond-2 carrying a Pulsed Plasma thruster [6] and the American SERT satellites with the first operation of an ion engine [7]. Significant other development highlights include the development of the Shell 405 catalyst allowing the utilization of high performance hydrazine-based chemical propulsion systems [8] and the SMART-1 mission showing the capability of an electric Hall thruster in an Earth-moon transfer [9], as well as the interplanetary Deep Space-1 mission, which was propelled by an NSTAR ion engine [10]. With the ABS-3A and Eutelsat 115 West B, we have seen the first instances of commercial platforms to use EP for orbit raising from a low earth orbit to the final geostationary orbit. However, while these missions have proven the high Δv capability of EP systems for traditional missions with high electrical power available, the recent trend towards miniaturization of spacecraft [11], [12] has resulted in tightened requirements for EP systems in terms of mass, volume and power consumption, while continuing to be competitive in terms of fuel efficiency, which is directly related to the system's capability to transform electric energy into kinetic energy of the exhaust.

This manuscript explores the different propulsion principles currently developed for small and miniaturized spacecraft. While designations for different satellite classes have been somehow ambiguous, a system mass based characterization approach will be used in this work, in which the term 'Small satellites' will refer to satellites with total masses below 500kg, with 'Nanosatellites' for systems ranging from 1-10kg, 'Picosatellites' with masses between 0.1-1kg and 'Femtosatellites' for spacecrafts below 0.1kg. In this category, the popular Cubesat standard [13] will therefore be characterized as Nanosatellite, whereas Chipsats, Wafersats and membrane-syle satellites [14], [15] will be categorized as Pico- or Femtosatellite, depending their mass and configuration.

Different satellite classes result in different power budgets available, as in a first approximation, available area for solar panels, as well as battery sizes decrease for decreasing satellite mass. Traditional mass based power estimation approximations assume specific power per subsystem mass of $25-200$W/kg [16]. Assuming a mass fraction of 10% for the power subsystem, the available power for Small satellites range from $1.25-10$kW to $25-200$W, whereas available power on Nanosatellites would range from $25-100$ down to $2.5-20$W and power levels for Pico- and Femtosatellites reduced by one and two orders of magnitude respectively compared to Nanosatellites. However, densification and fast integration of high efficiency components have shown that these assumptions can significantly underestimate the performance of current Nanosatellites, especiallcy Cubesats. For example, current state of the art 3U Cubesats can achieve $50-60$W of total BOL power when using deployable solar sails [17]. The available power level can have significant impact on the propulsive capabilities of a satellite platform in the case of EP, both on the choice of thruster principle as well as the resulting propulsive performance.

Propulsion is an enabling capability for spacecraft serving a variety of purposes to enable different applications, including:

- **Change of orbit altitude, orbit corrections:** The primary purpose of propulsive capabilities on spacecraft is to alter the orbit by changing orbital elements such as altitude or inclination. This can include changes in orbital altitude and corrections to maintain an orbit by counteracting perturbations. Low earth orbit raising maneuvers can range from 50m/s to 1.5km/s [18].
- **Life extension by drag compensation:** Miniaturized satellites are often deployed in orbit altitudes that guarantee natural decay within a certain timespan to comply with orbital debris guidelines, or to leverage relatively cheap launch opportunities, such as resupply missions to the International Space Station. During the active phase of the mission, propulsion can be used to extend the lifetime of the spacecraft, while relying on natural orbit decay caused by the rarefied atmosphere at the end of

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mission. While Δv requirements for drag makeup depend strongly on orbit altitude and satellite cross section and mass, typical values are in the 10s to 100s m/s.

- **Deorbiting:** Propulsive maneuvers can be used to decrease the obit altitude in low earth orbits, after the end of a mission to facilitate induced or natural orbit decay. Depending on the initial orbit, typical controlled re-entry Δv values range from $120 - 150$m/s [18]
- **Formation flight:** Much of the attractiveness of miniaturized space systems comes from the potential to facilitate the deployment of constellations, for a variety of purposes. Constellations bring benefits, such as an increase of revisit times, and the introduction of redundant or distributed architectures. Spacecraft in accurately controlled formation can also enable advanced instruments, as in the LISA mission for gravitational wave detection [19] and distributed apertures to increase resolution such as the proposed Terrestrial Planet Finder [20]. In most cases, such applications necessitate to maintain the relative spacecraft separation, requiring the capability for propulsive orbit correction, with required Δv ranging from 1 m/s to 100s of m/s depending on orbital mission parameters.
- **Constellation deployment:** Ridesharing and cheap development of larger numbers of miniaturized satellites make constellations of such systems very attractive. Constellation deployment from a shared launch allows to minimize launch cost [21], obviating dedicated launches in different orbits. Even small Δv delivered by the satellite after delivery from a shared launch vehicle can allow to evenly spread orbits and therefore revisit from a shared launch. Depending on mission architecture and time to deploy the constellation, the required Δv can range from 1 to 100s m/s.
- **Interplanetary missions:** Propulsion may be used for primary orbit transfer and control and adjustment as well as attitude control, and as means to desaturate reaction wheels in the absence of a planetary magnetic field. For primary orbit transfer, required Δv depends heavily on the initial launch orbit. For autonomous propulsion to an interplanetary body from a GEO orbit, the necessary Δv to reach earth escape velocity is ~ 1.3km/s, whereas attitude control is in the order of $1 - 10$m/s per year.

Small satellites could be orders of magnitude less costly than their larger counterparts, with development times in the 1-2 year range, which are very short compared to traditional project cycles, in some instances reaching a decade or more. This represents a dramatic change in the way satellites can be used, from utility platforms providing traditional services to systems spurring innovation that eventually will lead to vigorous scientific and economic development. Before that happens, however, the challenge of high capable and compact propulsion systems needs to be surmounted. For example, having access to high Δv propulsion in small satellites would allow them to perform rapid, frequent and affordable exploration of a myriad of objects of interest that are relatively close to the earth. This exploration would increase our understanding of our planetary neighborhood, while also could bring new areas of industrial development. Besides exploration, other areas will benefit from the inclusion of propulsion in small satellites, from imaging and communications, to astronomy and fundamental physics.

The different nature of these applications result in significantly differing requirements imposed onto the propulsion systems, with a general tradeoff between thrust and specific impulse as detailed in the following section.

II. Propulsion principles

The ideal change in velocity of a spacecraft Δv is described by the Tsiolkovsky rocket equation [22] as a function of the spacecraft mass before m_i and after m_f a propulsive maneuver.

$$\Delta v = I_{sp} g_0 ln\left(\frac{m_i}{m_f}\right) \tag{1}$$

Where $I_{sp} = u_e/g_0$ is the specific impulse, which is defined as the ratio of the propellant exhaust velocity relative to the propulsion system u_e and the standard acceleration due to gravity at Earth's sea level $g_0 \sim 9.81\text{m/s}^2$. The thrust F generated by a propulsion system is given by:

$$F = \dot{m}u_e + A_e p_e \approx \dot{m}u_e \tag{2}$$

where $\dot{m}$ is the exhaust mass flow, A_e the nozzle exit area and p_e the exhaust flow pressure at the nozzle exit plane, assuming plume expansion to vacuum. 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:

$$\eta \approx \frac{\frac{1}{2}\dot{m}u_e^2}{P} \tag{3}$$

where 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 EP.

A. 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 converging-diverging 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:

1) Cold and warm gas propulsion: In these basic systems, a high pressure working gas is expanded through a converging-diverging nozzle to create thrust. This principle is limited by the storage pressure achievable in the tank system, typically limited by structural considerations, and often require a separate pressure regulator to avoid thrust and specific impulse decay as the storage pressure decays over the mission duration as indicated in Fig. 1(a). Typical propellants used are

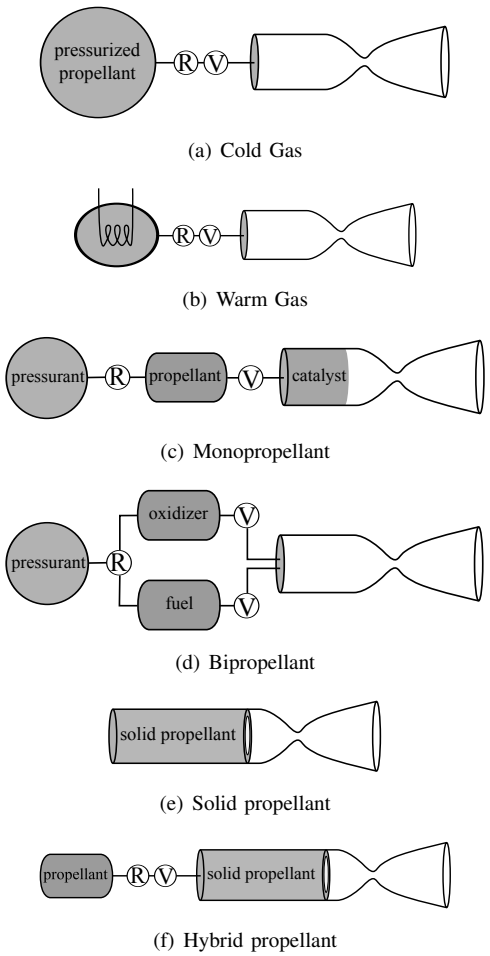


Fig. 1. Some chemical propulsion technologies

isobutane (C_4H_{10}), the refrigerants R236fa and R134a and Sulfur dioxide (SO_2) [23]. In warm gas propulsion systems, the working fluid is additionally heated before expansion (Fig. 1(b)), increasing the internal energy of the working medium using an external source, resulting in an increase in specific impulse of the system when compared to cold gas systems, see Sec. II-C1. Hybrid systems make use of phase changes, or storing a propellant, CO_2 or Nitrous oxide (N_2O), in supercritical condition [24]. Although such systems require active components separating the thruster from the high pressure propellant storage, the general simplicity of the system has allowed a high degree of miniaturization, especially with the emergence of MEMS based valve and other components. With generally low specific impulse of such systems, the total achievable Δv is typically limited for miniaturized satellite systems in which space to accommodate large propellant tanks is not available. However, the ability to achieve small impulse bits and large number of impulse cycles make these systems attractive for small and precise positioning and attitude control maneuvers.

Using a multiple staged valve architecture, typically with an intermittent smaller volume reservoir, between main valve and thruster selector valve, allows to decrease the risk of valve failure leading to failures that can prove critical to the satellite mission.

Typical specific impulse range from $\sim$ 10s to $\sim$ 80s, and thrust levels $\sim 100\mu$N to $\sim$ 100mN.

2) Monopropellant and advanced monopropellant thruster: In monopropellant thrusters (Fig. 1(c)), a high 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 for 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. In these designs, propellant selection is typically governing the design and nature of the decomposition chamber, with N_2H_4 requiring preheating of the catalyst bed. Decomposition temperatures of monopropellants are in general low enough to allow for radiation cooling without need for exotic materials for the chamber designs. Design tradeoffs are usually made between the toxicity and storability of the propellant, which can have a major impact on mission cost. N_2H_4, which is highly toxic, reactive and has high vapor pressure features provides high performance, reliability and long catalyst lifetime compared to available propellants. However, to comply with reducing mission cost typically strived for in small satellite missions, lower toxicity options such as hydrogen peroxide based propulsion systems have been recently developed [25]–[27]. As the density of the stored propellant determines the necessary tank volume, the density of the propellant in stored condition can be important in miniaturized designs, leading to increase volumetric specific impulse of high density propellants such as H_2O_2 when compared to liquids with lower density, such as N_2H_4. Another concern specifically for small satellites can arise by the fact that high energetic, and potentially unstable propellant can negatively impact launch opportunities for secondary payloads due to safety concerns.

Typical specific impulse range from $\sim$ 100s to $\sim$ 230s and thrust levels from $\sim$ 100mN up to 100s of Newtons.

Originally motivated by the search for less toxic propellant alternatives to hydrazine-based derivatives, monopropellant systems based on Ammonium dinitramide (ADN) or Hydroxylammonium nitrate (HAN) have been developed with considerable effort [28]. In such systems, the propellant is typically thermally and catalytically decomposed, with significantly higher performance than traditional monopropellants. Such systems are therefore similar to monopropellant systems in terms of necessary fluidic components, comprising of a single propellant tank and feed stream but provide energy release and therefore performance similar to traditional bipropellant systems. An ADN based propulsion system has already been successfully tested in space [29], and the Green propellant infusion mission (GPIM) [30], [31] is currently awaiting launch to increase the maturity of a HAN based propulsion system by in orbit validation.

Typical specific impulse for such systems are between $\sim$ 200s and $\sim$ 250s, and thrust levels range from hundreds of millinewton to tens of Newtons.

3) Bipropellant propulsion: In bipropellant systems as shown in Fig. 1(d), combustion of an oxidizer and a fuel are utilized to create a high temperature, high pressure gaseous mixture that can be expanded using a converging-diverging 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 used, storable, non-cryogenic propellant combinations are Monomethylhydrazine (MMH) or Unsymmetrical dimethylhydrazine (UDMH) 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 [32], [33]. Bipropellant systems feature the highest performance for chemical systems per stored propellant, but require generally complex propellant managements system with multiple active components. Due to the higher flame temperature compared to monopropellant thrusters, radiative cooling of the combustion chamber requires high temperature metals, often refractory metals such as Rhodium or Platinum alloys, and can lead to radiative energy transfer from the high temperature combustion chamber to the satellite structure which needs to be considered in design. Bipropellant systems are therefore typically used in missions with higher Δv requirements such as apogee orbit insertions or maneuvers involving significant orbit changes.

Typical specific impulse are $\sim$ 300s with thrust levels usually starting from 10N.

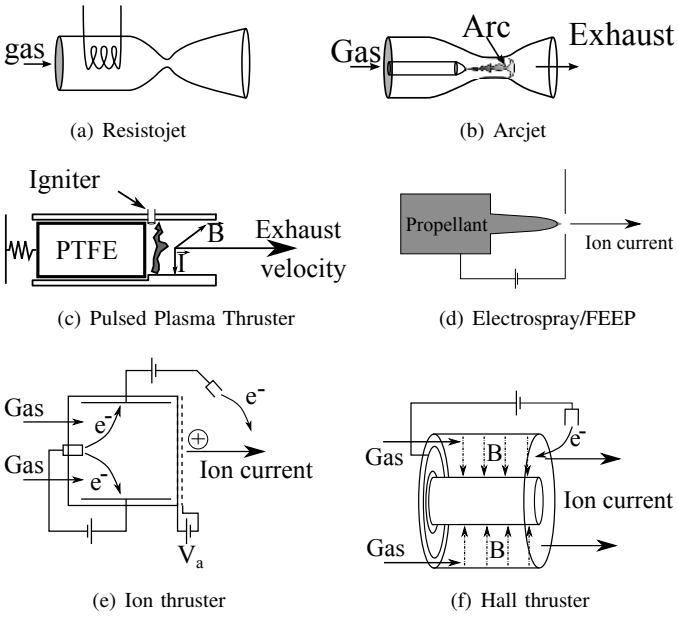
B. Solid and hybrid propulsion

By combusting a solid propellant, solid propulsion systems provide a hot working gas that is then expanded to produce thrust (Fig. 1(e)). 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 [34]. Restarting capability and increasing performance are investigated by introducing hybrid configurations where part of the reactants is fed into the solid combustion chamber using a fluidic system as depicted in Fig. 1(f). While increasing the utility of such propellant systems, this typically comes with added system complexity and is therefore currently considered of limited use for miniaturized space systems.

Typical specific impulses for miniaturized solid motors range from $\sim$ 150s to $\sim$ 280s, with thrust levels ranging from tens to hundreds of Newtons.

C. Electric propulsion

EP systems differ from chemical propulsion devices by the means of energy supply for exhaust acceleration. 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 that is used to accelerate the exhaust. While any source supplying electrical power such as nuclear reactors or Radioisotope thermoelectric generator (RTD) can be used, in most cases, especially in small satellites, electrical power is supplied by solar panels.



(a) Resistojet (b) Arcjet

(c) Pulsed Plasma Thruster (d) Electrospray/FEEP

(e) Ion thruster (f) Hall thruster

Fig. 2. Electric propulsion configurations

1) Electrothermal acceleration: Historically, **resistojets** were the first instance in which an external energy source was used to electrothermally augment traditional chemical rockets and was originally applied to N_2H_4. In such devices, shown in Fig. 2(a), the propellant gas is typically heated by an electrically heated surface in various configurations [35] to increase the propellant gas beyond the stagnation temperature of the purely chemical propulsion system, and therefore augment the resulting exhaust velocity after expansion. Resistojets have been employed over a wide range of power levels and propellants, with gas-heater heat transfer usually being the premier design challenge. While traditional resistojet designs have not been found in small satellites so far, warm gas thrusters in which the entire propellant, or a small subset of the propellant is heated in a separate reservoir before injection into the acceleration chamber, can be seen as a somehow related thruster type that has been developed with special focus on small satellite applications.

Arcjets (Fig. 2(b)) on the other hand augment the propellant gas temperature by creating a steady arc discharge, typically through the nozzle throat from a cathode inside the combustion chamber that is annually surrounded by propellant gas flow. Heating the gas by means of a discharge allows to surpass the maximum working temperature of conventional heating elements, that typically limit the performance of resistojets. While providing higher performance than traditional monopropellant and resistojet propulsion systems, the added complexity of arcjets necessitating a fluidic and a power-intensive electrical subsystem have prevented usage of such propulsion systems in small satellite applications to this point.

2) Electrostatic acceleration: Electrostatic space propulsion devices accelerate charged particles, mostly ions, by electrical forces when falling through a potential drop across two electrodes and are the most evolved EP concepts originating back to the 1950s.

Ion Engines, also know as Kaufman Ion Engines or Electrostatic Thrusters, shown in Fig. 2(e), accelerate ions that are produced in an ionization chamber using a potential drop between an extractor and an acceleration grid. Ion production can be accomplished by collision with injected electrons, radio-frequency-, microwave- or contact ionization. The expelled ion beam is then typically neutralized by an external electron emitter. The first gridded ion thruster has been tested on a suborbital flight on the SERT I spacecraft in 1964 [7]. Ionization efficiency in such devices is a function of propellant and electron current as well as residence time of the injected gaseous propellant in the ionization chamber, limiting the miniaturization in terms of volume. Miniaturization of a durable cathode for electron supply is presenting a challenge with respect to miniaturization, in addition to the ability to manufacture flat grids out of materials that show high resistivity to ion erosion while space charge between the accelerating grids limits the achievable emission current density and therefor thrust. Recent developments in miniaturized ion engines make use of RF ioniziation, obviating the need for an internal electron emitter [36]–[38]. Ion engines have been employed in notable mission such as Deep Space-1 [10] , Dawn [39] and GOCE [40] , typically achieve specific impulse between $I_{sp} = 2000 - 3000$s.

Hall thrusters present a second form of highly developed electrostatic accelerator systems that was flown amongst others on the SMART-1 moon transfer mission [9], and has become attractive due to the absence of any acceleration grids which are typically life limiting components due to erosion and electrical breakdowns. Hall thrusters on the other hand feature an annular ionization and acceleration chamber in which a neutral gas is injected near an upstream anode through a manifold, and is subsequently ionized by electrons that are injected downstream as shown in Fig. 2(f). These electrons are attracted to a positive upstream anode, while strong magnetic fields perpendicular to the gas flow force the electrons one a precessing path along the annular chamber, increasing the residence time and therefore probability of collision with the injected neutral gas. The ions, due to their higher molecular mass, feel a proportionally weaker acceleration by the magnetic force, and are primarily accelerated electrostatically by the potential drop between the anode and a cathode situated at the exit of the otherwise insulating annular chamber. While specific impulse achieve by Hall thrusters are generally lower than in ion engines around $I_{sp} \sim 1500$s, they feature higher thrust densities (up to an order of magnitude higher than in ion engines) and can be adapted over a wide range of power levels.

Colloid- electrospray thrusters and Field Emission Electrostatic Propulsion (FEEP) (Fig. 2(d)) are propulsion systems with similar acceleration principles, but differ in terms of production of charged particles. In FEEP thrusters, a liquified metal propellant such as Indium, Gallium or Cesium, is suspended over a sharp emitter structure to increase the localized electric field at the apex. Balancing the electrostatic pull and the surface tension, the conductive liquid metal deforms into a Taylor cone [41] which further increases the local field strength at the apex, where the ionization threshold can be surpassed, leading to the ejection of ions. These ions are then accelerated by the same electric field used for extraction. In electrospray thrusters, ionic liquids or electrolytes are used as propellants, with the latter tending to produce significant ratios of slow moving droplets, whereas ionic liquid electrosprays are able to operate in pure ionic mode [42]. While electric conductivities, and therefore ultimately emission current per emission site, of such propellants are generally lower compare to liquid metals ion sources, they do not require energy to liquefy the propellant by heating and therefore have the potential for higher system efficiency, assuming similar beam properties. In addition, no ionization is required due to the nature of the ionic liquid propellant since molecular ions are readily extracted from the propellant bulk [43]. In addition, ionic liquid electrospray thrusters are capable of producing charged particles of both polarities, obviating the need for an external neutralization device as required in other EP systems [44]. Due to the absence (in case of electrosprays) or locally confined (FEEP) ionization process, both FEEP and electrospray systems feature uniquely small ionization/acceleration chambers and lend themselves favorably towards miniaturization.

3) Electromagentic acceleration: While in electrostatic propulsion systems, the force that is acting on the accelerated charged particle is reciprocally felt by the accelerating electrode, the definition of electrostatic pressure $T/A = 1/2\epsilon_0 E^2$ with T/A the thrust density, ϵ_0 the vacuum permittivity and E the accelerating electrical field strength, imposes a limit to the achievable thrust per area in real devices where E can typically not be increased indefinitely. In electromagnetic devices however, the force transfer from the accelerated particle beam to the structure occurs via magentic fields in such devices, allowing for higher thrust densities.

Magnetic Plasma Dynamic (MPD) thrusters are high power propulsion systems with self-induced magnetic fields and operate at power levels incompatible with small satellite technology, but would be interesting for larger spacecraft due to their ability to reach multi newton thrust levels.

Pulsed plasma thrusters (PPT) and Vacuum Arc Thrusters (VAT) on the other hand are a type of propulsion system that are operated in a unsteady regime, in which a solid propellant, typically Teflon, is ablated by an induced discharge, and acceleration of ablated material occurs by the Lorentz force. As the magnetic fields in the acceleration electrodes interact with the magnetic field induced in the perpendicular discharge, the ablated material is accelerated perpendicular to the discharge surface as indicated in Fig. 2(c). The pulsed operation, in which a capacitor is slowly charged and then rapidly discharged, suits itself well to small power budgets, and operation frequency can easily be adjusted to available power. While adaptability to available power budget, generally small impulse bits and mechanical simplicity due to the solid

propellant trade favorable for PPTs, they generally feature very low efficiency $< 10\%$ [6].

Vacuum arc thrusters 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 [45], [46].

D. Propellant-less propulsion

In addition to traditional propulsion system that operate by accelerating neutral or charged particles to produce thrust, a variety of propulsion concepts utilizing different physical mechanisms have been proposed and tested:

1) Solar sail propulsion: uses low density solar radiation pressure to generate thrust, by typically employing lightweight, expandable structures to obtain very high surface to mass ratio, contracting the low forces due to low radiation pressure. This concept has been notably tested by the Venus bound IKAROS spacecraft [47] and the Cubesat LightSail-1 [48]. In addition to radiative pressure propulsion, these expandable structures have been used as end-of-life deorbit devices to lower a low earth orbiting spacecraft by significantly increasing the atmospheric drag, as proven in the NanoSail-D mission [49].

Electrodynamic tethers are long, conductive wires or structures deployed in orbit in the presence of a planetary magnetic field and ionosphere, producing a force by the interaction of the external magnetic field and the charges moving along the tether which, for propulsion purposes, is held at a different potential with respect to the spacecraft using a power supply, thus adding kinetic energy to the orbit. To remove kinetic energy, eg. to lower the orbit altitude, the ends of the tether can be shorted. Motion of the spacecraft with respect to the magnetic field will induce a voltage difference, drawing a current across the tether, which interacts with the external magnetic field, effectively reducing the orbital velocity.
An extreme form of high specific impulse propulsion is **Photonic propulsion**, which utilizes the photonic pressure to produce thrust. While thrust levels are generally too low for power levels achievable in small satellites with respect to traditional solar system bound missions, this concept has attracted interest either by using external high power laser beams to propel Nanosatellites [50], or in the context of small propulsion maneuvers in very long mission durations, such as attitude corrections in interplanetary Femtosatellites [50].

Further concepts of propellant-less propulsion include **Magnetic sail propulsion**, in which a loop structure induces a magnetic field that interacts with solar wind, leading to an exchange of momentum to the magnetic sail structure [51], and **Electric sail propulsion**, in which a large conductive mesh is kept at a positive potential relative to a solar wind plasma, blocking positive particles, therefore leading to a theoretical exchange of momentum towards the spacecraft [52]. Due to the small order of magnitude of such interactions which necessitates large scale structures, such concepts have not yet evolved beyond conceptual studies.

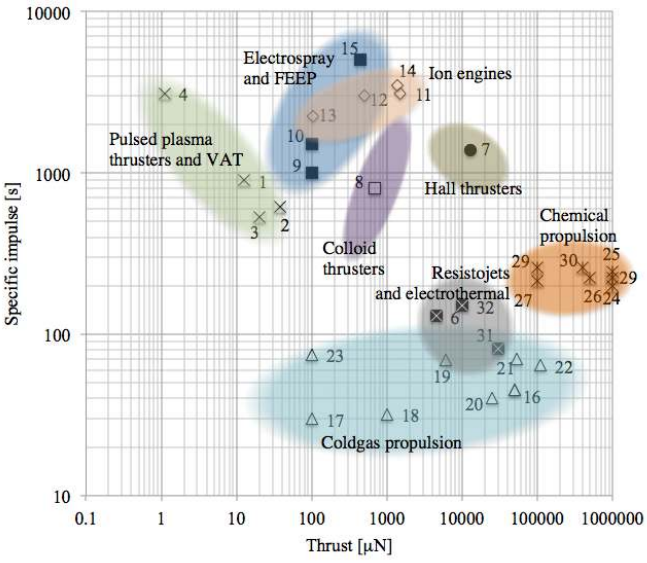


Fig. 3. Commercial micro-propulsion systems: Thrust versus specific impulse at nominal operation point. **1** PPT, Austrian Institute of Technology [53], **2** PPTCUP, Mars Space [54], **3** BmP-220 Busek (Falcon-Sat 3), **4** μCAT, George Washington University (BricSat-P) [55], **5** VAT, University of Illinois, **6** RFT, Phase Four [56], **7** BHT-200, Busek, **8** BET-1mN, Busek, **9** BET-100 Busek, **10** iEPS, Massachusetts Institute of Technology (Aerocube-8) [57], **11** MiXi, Jet Propulsion Laboratory, **12** RIT-μX, Airbus, **13** BIT-1, Busek, **14**, BIT-3, Busek, **15** IFM Nano, Enpulsion/Fotec [58], **16** NANOPS, University of Toronto (CanX-2) [59], **17** MEPSI, The Aerospace Corporation, (MEPSI-3) **18**, Cold gas micropropulsion system, Micro Space (POPSAT-HIP1) [60], **19** T^3μPS, TNO (Delfi-n3Xt) [61], [62], **20** CPOD MiPS, VACCO, **21** PUC, VACCO, **22** Bevo-2 cold gas, University of Texas [63], **23**, CubeProp, NanoSpace (STU-2B), [64], **24** MPS-120, Aerojet Rocketdyne, **25** MPS-130, Aerojet Rocketdyne, **26** BGT-5X, Busek, **27** BGT-1X, Busek, **28** ADN Micro Propulsion System, VACCO/ECAPS, **29** LMP-103S, ECPAS (PRISMA) [29], [65], **30**, HYDROS, Tethers Unlimited [66], **31** CHIPS, CU Aerospace, **32** Resistojet, Busek. Unless otherwise noted, data is taken from vendor data sheets or Ref. [23]

III. Performance Metrics of Small Satellite Propulsion Technology

A plethora of propulsion systems suited for small satellites and beyond has been developed and researched in the recent years, with multiple different propulsion principles matured through inspace testing or ready for commercialization, from both chemical propulsion systems and EP systems. While chemical propulsion systems used as primary propulsion means on small satellites showed some overlap with existing technology used for station keeping on larger space platforms, miniaturization of propulsion devices to smaller platforms such as Nanosatellites and beyond required significant effort to achieve miniaturization without significant decrease in performance, and led to a variety of new developments altogether.
Fig. 3 shows a classification of the different propulsion systems for small satellites discussed in Sec. II, supplemented with data from commercially existing systems in terms of thrust and specific impulse. For space qualified systems, the satellite name is included in brackets. While this plot is intended to provide a general classification of the different propulsion solutions, care should be taken with the individual

performance metrics of the propulsion systems, as these rely on data provided by suppliers, and where determined with varying degree of accuracy and detail. In this plot, the data shown for each propulsion system corresponds to the nominal operational point provided by the suppliers. It should however be noted that especially EP systems are capable of throttling over significant ranges, as discussed later. Propellantless systems are not included in this plot as specific impulse becomes a meaningless metrics, and bipropellant engines are omitted due to their limited utility for miniaturized satellites. A notable exception is the **30** HYDROS propulsion system [66], which is based on the combustion of H_2 and O_2 provided by electrolysis of stored water, and does not require two separate propellant streams as do traditional bipropellant systems.
While there is significant overlapping for electrical propulsion systems based on different acceleration principles, several trends can be clearly identified:

- Cold and warm gas propulsion systems occupy the lower specific impulse range, but are available over a large range of thrust. As expected, cold gas propulsion systems are found on the lower bound in terms of specific impulse, while heated systems are capable of increasing the specific impulse to $\sim< 100$s. Ability to generate large numbers of small impulse bits make these systems attractive for small orbit and attitude correction applications.
- Chemical propulsion systems based on decomposition of an energetic compound, such as HAN-, ADN-, or hydrazine-based systems are found on the high thrust region of the plot, with specific impulse significantly higher than cold and warm gas systems, but lower than the bulk of EP systems.
- EP systems feature increased specific impulse, and range from thrust levels comparable to cold gas systems to μN levels. Inside this group, the highest performing systems in terms of specific impulse are FEEP thrusters, which generally show a wide range of throttleability. A variety of commercially available ion engines are available, with thrust levels comparable or higher than electrospray thrusters, with lower specific impulse than FEEP thrusters.
- Pulsed plasma thrusters are found on the lower thrust and lower specific impulse region of the studied EP systems.
- One miniature Hall thruster was considered, to be on a mature level compared to the other systems investigated. Hall thrusters are capable of providing the highest thrust of all EP systems considered, but feature lower specific impulse compared to the available ion engines.

In order to compare propulsion systems, it is important to consider, besides propellant mass, the physical parameters of the propellant during storage, most notably the propellant density. It is therefore useful to introduce the concept of volumetric specific impulse, defined as $I_{sp}^{vol} = \rho_{prop} I_{sp}$ given in [kg s / m^3], with ρ_{prop} the propellant density at storage conditions, allowing to draw conclusions of the specific impulse achievable per required storage volume. Fig. 4 gives a comparison of the volumetric specific impulses for commercial chemical propulsion systems as a function

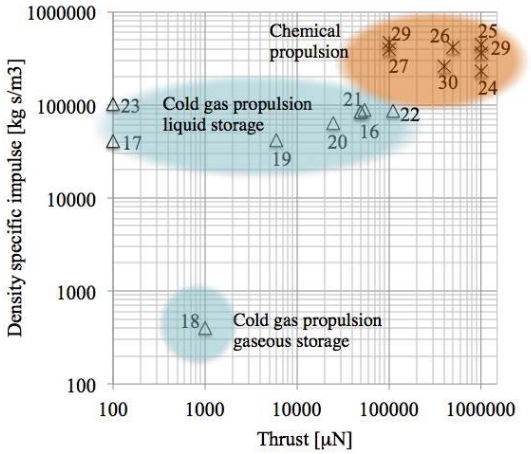


Fig. 4. Commercial chemical micro-propulsion systems: Thrust versus volumetric specific impulse at nominal operation point. **16** NANOPS, University of Toronto (CanX-2) [59], **17** MEPSI, The Aerospace Corporation, (MEPSI-3) **18**, Cold gas micropropulsion system, Micro Space (POPSAT-HIP1) [60], **19** T$^3\mu$PS, TNO (Delfi-n3Xt) [61], [62], **20** CPOD MiPS, VACCO, **21** PUC, VACCO, **22** Bevo-2 cold gas, University of Texas [63], **23**, CubeProp, NanoSpace (STU-2B), [64], **24** MPS-120, Aerojet Rocketdyne, **25** MPS-130, Aerojet Rocketdyne, **26** BGT-5X, Busek, **27** BGT-1X, Busek, **28** ADN Micro Propulsion System, VACCO/ECAPS, **29** LMP-103S, ECPAS (PRISMA) [29], [65], **30**, HYDROS, Tethers Unlimited [66].

of nominal thrust. If no information on storage pressure was available, storage under liquid conditions was assumed, unless information indicating otherwise. For the TNO T3μPS it is noted, that due to the gas generation principle from a solid material, that the volumetric specific impulse has only limited significance, and therefore a generic propellant density was calculated based on gas generator volume and propellant mass stored, conservatively attributing 12.5% of the volume to the propellant (assuming same density for propellant and gas generator). It should be noted that volumetric specific impulse can be an important metric for EP systems as well, however in most cases auxiliary system mass such as the mass of the power processing unit can significantly alter the total system mass, and therefore inhibits a systems comparison solely on volumetric specific impulse. As the PPU masses vary considerably for the EP systems considered, a comparison of EP systems based on volumetric specific impulse is not considered here. Nevertheless, the importance of propellant storage density as a fundamental parameter is highlighted for EP systems as well.

The first important conclusion evident from the data shown Fig. 4 is the importance of considering the thermodynamic state in which the propellant is stored under storage conditions, evident from the significantly lower volumetric specific impulse for the only system considered that features propellant stored in gaseous state, compared to the systems that guarantee liquid propellant storage conditions. Except for the gaseous Argon based system, most cold gas systems feature very similar propellant densities, therefore comparing similar than when compared on specific impulse level, except for the systems using SF_6, which has densities comparable to HAN- and ADN-based chemical propellants. The only higher

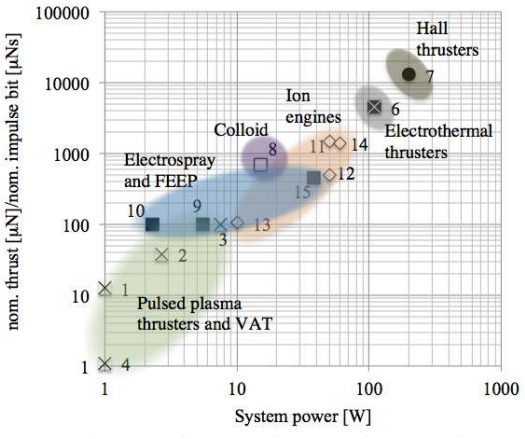


Fig. 5. Commercial EP systems: Nominal thrust as a function of system power, with most systems showing significant throttling capability not indicated. **1** PPT, Austrian Institute of Technology [53], **2** PPTCUP, Mars Space [54], **3** BmP-220 Busek (Falcon-Sat 3), **4** μCAT, George Washington University (BricSat-P) [55], **6** RFT, Phase Four [56], **7** BHT-200, Busek, **8** BET-1mN, Busek, **9** BET-100 Busek, **10** iEPS, Massachusetts Institute of Technology (Aerocube-8) [57], **11** MiXi, Jet Propulsion Laboratory, **12** RIT-μX, Airbus, **13** BIT-1, Busek, **14**, BIT-3, Busek, **15** IFM Nano, Enpulsion/Fotec [58]. Unless otherwise noted, data is taken from vendor data sheets.

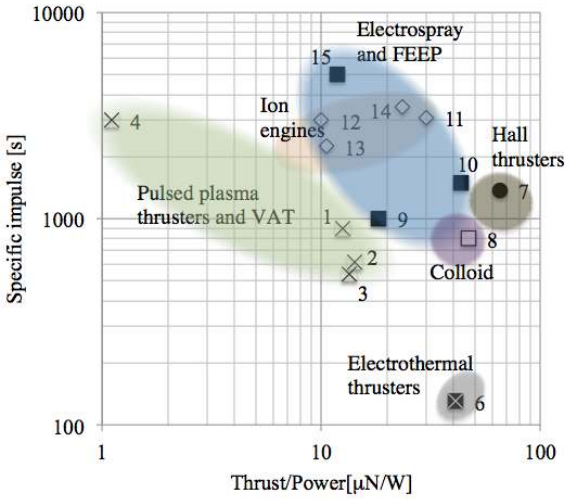


Fig. 6. Commercial EP systems: Thrust to power versus specific impulse. **1** PPT, Austrian Institute of Technology [53], **2** PPTCUP, Mars Space [54], **3** BmP-220 Busek (Falcon-Sat 3), **4** μCAT, George Washington University (BricSat-P) [55], **6** RFT, Phase Four [56], **7** BHT-200, Busek, **8** BET-1mN, Busek, **9** BET-100 Busek, **10** iEPS, Massachusetts Institute of Technology (Aerocube-8) [57], **11** MiXi, Jet Propulsion Laboratory, **12** RIT-μX, Airbus, **13** BIT-1, Busek, **14**, BIT-3, Busek, **15** IFM Nano, Enpulsion/Fotec [58]. Unless otherwise noted, data is taken from vendor data sheets.

performing chemical systems with lower density compared to other systems are the N_2H_4 based MPS-120 and the water based HYDROS thruster.

The discussion so far has neglected the power consumption of EP systems, which, with increased miniaturization of the spacecraft itself, can act as a main factor limiting usability. Fig. 5 plots the nominal thrust as a function of system input power for a variety of EP systems. For pulsed thrusters, the nominal impulse bit is plotted. The systems operational at lowest power levels are pulsed plasma thrusters and electrospray thruster systems, with the latter achieving higher thrust for the same power consumption compared to pulsed plasma thrusters, which corresponds well to the generally lower total efficiencies found for pulsed plasma thrusters. It should be noted that due the adjustability of the duty cycle, pulsed systems such as PPTs or VATs can be operated at even smaller average power consumptions, albeit at the cost of decreasing the achieved impulse bit further. Electrospray thrusters generally exhibit throttleability without significant impact on performance, and it is noted that they can be operated well below the indicated power and thrust levels. FEEPs using liquid metals show excellent throttleability in terms of thrust, but typically can not be steadily operated below a certain power threshold required for heating the propellant. Ion engines on the other hand occupy the higher power spectrum, showing comparable thrust levels than FEEPs or colloid thrusters, but can typically not be operated below 10W. Miniature Hall thrusters are found at the high power spectrum of this comparison, with the potential for increased thrust levels.

The investigated EP systems are available over a wide range of power levels and it is therefore useful to look at the relationship of specific impulse and the thrust to power ratio. As the relationship between system power, thrust and specific impulse is typically not trivial for the individual systems, a comparison of the stated performance at the nominal operational point is plotted in Fig. 6. In this comparison of specific thrust per power to specific impulse, it is again evident that Ion engines and FEEPs, providing highest specific impulse, trade in the middle in terms of thrust per input power, whereas electrospray, colloid and Hall thrusters feature the highest thrust to power ratio.

Comparison of Figs. 5 and 6 also shows that while thrusters with predominantly electrothermal acceleration, such as the Phase Four RFT thruster, can achieve similar thrust to power ratios, they feature significantly reduced specific impulse compared to other EP systems.

Fig. 3, Fig. 5 and Fig. 6 compare the specific impulse of different propulsion systems at the nominal operational point. It should however be noted that especially EP systems are often capable to operate over significant throttling ranges. Fig. 7 shows the throttling capability of selected EP systems, plotting the thrust and specific impulse as a function of input system power. The systems compared show the MIT iEPS system, which features constant specific impulse over the thrust and power range tested [57], the University of Washington VAT thruster that increases delivered impulse bit by increasing the pulsing frequency, the Busek BeP-220 Pulsed plasma thruster with similar operational properties compared to the VAT, the Busek BIT-3 ion thruster with linear specific impulse and thrust increase with increasing power [37], [67], and the Enpulsion IFM Nano thruster with variable thrust/power tradeoff for increasing specific impulse [58].

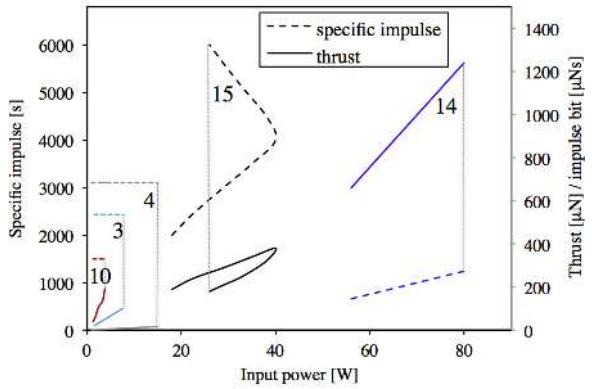


Fig. 7. Throttling range for sample EP thrusters showing dependency of specific impulse and thrust on input power. **3** BmP-220 Busek (FalconSat 3), **4** μCAT, George Washington University (BricSat-P) [55], **10** iEPS, Massachusetts Institute of Technology (Aerocube-8) [57], **14**, BIT-3 full system including neutralizer, Busek [67], **15** IFM Nano full system including neutralizer, Enpulsion/Fotec [58], [68].

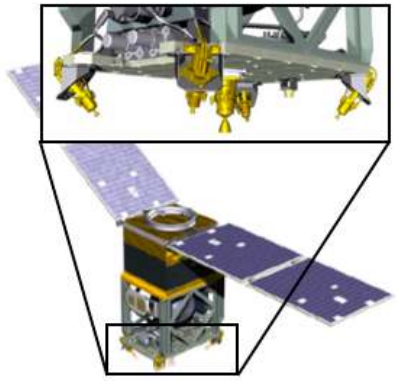


Fig. 8. GPIM small satellite featuring 1N advanced monopropellant thrusters using AF-M315E, from Ref. [69]

IV. Flight Experience with Small Satellite Propulsion Systems

A variety of propulsion systems have been flown on small satellites and smaller platforms, and many Nanosatellite missions are currently in preparation that will be using some type of propulsion. In this section, some notable flight tests of propulsion systems will be discussed.

A. Small satellite case: ADN- and HAN-based advanced monopropellants

While commercial propulsion systems for small satellites, both chemical and electrical, are readily available and have been used in space, two missions are highlighted in this section: A reduced toxicity, ADN-based propulsion system was tested onboard of the PRIMSA satellite, featuring 2 ECAPS LMP-103S thrusters, providing 1N at 252s specific impulse, providing a ΔV capacity of $\sim$ 60m/s to the satellite. The mission was able to prove successful operation of this thruster system in space, and the thrusters capability to be used in autonomous formation flying, as well as conducing propulsion system performance measurements [29].

The Green Propellant Infusion Mission (GPIM) [30], [31] is noted for testing another reduced toxicity, advanced monopropellant propulsion system on a Small Satellite platform with $<$ 180kg total mass shown in Fig. 8. The mission, which is scheduled for launch in spring 2018, will carry 4 Aerojet GR1 1N thrusters propelled by AF-M315E, a HAN based propellant, maturing this type of high performance propulsion system which is expected to provide a specific impulse of 235s. The total impulse of the propulsion system is stated as 23kNs, and while this mission is intended as a technology demonstration only, it will prove the ability to achieve considerable Δv using a system that can be implemented in smaller spacecraft as primary propulsion as well.

Various entities are currently maturing propulsion systems based on AF-M315E [30], [70], [71] or similar ADN [29] based propellants with applications ranging from small satellites to Cubesats, including the planned utilization of a VACCO MiPS system featuring 4 100mN ADN-based thrusters on the Lunar Flashlight Cubesat mission [72].

B. Small satellite case EP: Hall thruster

The Busek BHT-200 hall thruster, operated with Xenon propellant at 100-300W was flown on TacSat-2, a $\sim$380kg satellite, becoming the first US built Hall thruster to be operated in space. TacSat-2 was launched in 2006, followed by successful thruster operation [73]. The same thruster is scheduled to be flown on the 180kg FalconSat-6 microsatellite. A version of the BHT-200 using iodine as propellant is planned to be used in the iodine satellite (iSAT) mission, a 12U Cubesat intended as a rapid orbital demonstration of the iodine Hall thuster technology [74], [75]. The mission is intended to demonstrate small spacecraft maneuverability and mitigate concerns regarding iodine deposition on spacecraft structures. Featuring a 200W thruster, this mission will demonstrate relatively high power propulsion in a Cubesat envelope. This program includes testing of a higher power BHT-600 iodine thruster, and in space validation of the iodine Hall thruster technology is seen a precursor for Nano- and Small satellite missions with high Δv capabiliy, enabling future Near Earth Asteroid Orbiter and Lunar Orbiter missions [76].

C. Nanosatellite case 1: Cold gas- and warm gas thrusters

A variety of cold gas systems which are commercially available have been flown on Small satellites, and recently on Nanosatellites, including the 3U "PRopulsion Operation Proof SATellite -High Performance 1" (POPSAT-HIP1) mission, demonstrating attitude control (4 degrees) using microfabriacted nozzles. with an specific impulse estimated from orbital data, of 32s [60].

Delfi-n3Xt, a 3U Cubesat by Delft University of Technology verified a cold gas propulsion system in which the propellant is stored in a solid phase, operated in a blow down mode with continuously decreasing thrust level [62].

CubeProp, another cold gas system, developed by NanoSpace was tested onboard the STU-2B Cubesat (launched

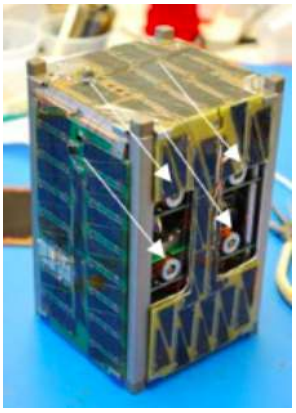


Fig. 9. 1.5U BRICSat-P featuring 4 VAT thruster heads developed by the University of Washington, indicated by white arrows, from Ref. [78]

in 2015), raising the orbital altitude by $\sim$ 600m, and performing attitude control tasks [64].

While still awaiting launch, the JPL Marco mission is a notable example of a Cubesat mission using cold gas propulsion for trajectory correction maneuvers on its flight path to a Mars flyby. The 6U Cubesats will use 4 R-236FA propelled thrusters for minor trajectory correction, and 4 canted thrusters for attitude control maneuvers, all fed by a single propellant tank [77].

D. Nanosatellte case 2: Vacuum Arc Thrusters and Pulsed Plasma Thrusters

A type of Vacuum arc thruster called Micro-Cathode Arc Thruster (μCAT) of the University of Washington [78], [79] has flown on the BRICSat-P mission, a 1.5 U Cubesat launched in spring 2015, featuring 4 μCAT thruster heads. Fig. 9 shows an image of the satellite. After satellite deployment, the thrusters were successfully used in the detumbling of the satellite, but a premature loss of communication to the satellite prevented further investigation of the propulsion system. The VAT propulsion system is planned to be further tested on an upcoming GWSat mission in 2018.

PPTs developed by Busek were flown on the 50kg Falconsat-3, featuring 3 MPACS units [80]. Each thruster had a discharge energy of $\sim$ 2J at an average specific impulse of $\sim$ 820s. While this system was tested on a Small Satellite, the size and suitability to small power budgets by reducing the discharge energy and therefore the impulse bit per discharge, or the duty cycle of the system, make PPTs relatively compatible to Nanosatellite.

E. Nanosatellite case 3: Electrospray thrusters

MEMS based electrospray thrusters developed at MIT SPL have been flown on several Aerocube-8 satellites, which are 1.5U Cubesats launched in 2015 and 2016. Each of the satellites carried a PPU capable of firing 8 individual thrusters, each consisting of a micromachined emitter array featuring 480 emitter tips in a 1 square centimeter area. Fig. 10 shows engineering units of a fully integrated electrospray propulsion

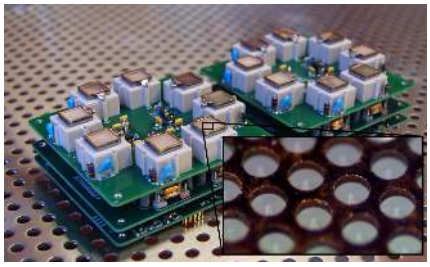


Fig. 10. Engineering units of the MIT electrospray propulsion system featuring 8 individually controllable micromachined emitter arrays.

Fig. 11. Busek BIT-3 iodine thruster, from Ref. [83]

system. In this system, the propellant is fed passively by capillary forces from a zero pressure propellant reservoir. The highly miniaturized electrospray emitter array allows for scaled up systems featuring 10s and 100s of emitter arrays, achieving unique controllability and redundancy.

V. Flight-ready High Performance Nanosatellite Propulsion Systems

A. Ion engines

Various entities have developed miniature radio frequency ion engines targeting 6-12U Cubesats, based on their power level and volume required to house thruster, neutralizer cathode, as well as propellant tank and feed system [36], [37]. Busek is currently maturing the BIT-3 propelled by iodine, which allows for an unpressurized tank system and therefore reduces the structural requirements of such a system, providing 1.4mN at 60W beam power, which reduces to $\sim$0.8mN at 60W PPU input power when including the neutralizer, and 1600s specific impulse [67]. Fig. 11 shows the thruster firing with iodine propellant together with a neutralizer cathode. Various missions are currently in the planning stage that intend to use the BIT-3, including the 6U Lunar Icecube [81] and the 6U LunaH-Map Cubesat mission [82].

B. FEEP

A Cubesat-compatible liquid indium FEEP propulsion module featuring 28 emission sites has been developed by FOTEC/Enpulsion [58], and is scheduled for flight by end of 2017. Fig. 12 shows the flight propulsion system during acceptance testing. A prototype of this propulsion system is currently undergoing a long duration firing tests, firing in a laboratory vacuum chamber continuously since over 2.5 years.

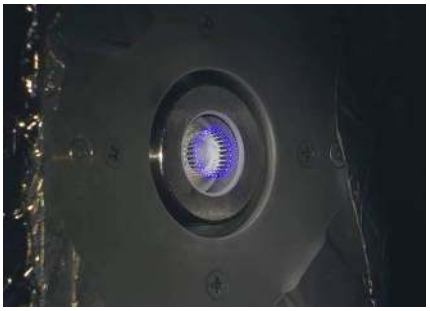


Fig. 12. Fotec/Enpulsion IFM Nano thruster during acceptance testing

The technology has been derived from single emitter ion sources with considerable flight heritage, including NASA's MMS and ESA's Rosetta mission [84]. The compact size and moderate power requirements together with very high specific impulse and the ability to throttle over a large range of thrust and specific impulse make this system additionally attractive for high Δv Small Satellite missions, with multiple FEEP units being operated in parallel to increase thrust levels.

VI. Scaling laws for Small Satellite Propulsion Technology

A. Gas dynamic acceleration

Direct scaling of existing propulsion systems to smaller envelopes in terms of volume and power is generally limited by a non-linear increase in the inefficiency beyond a certain point, as inefficiencies do not scale linearly with devices [85]. Perhaps the biggest challenge in maintaining high efficiency while miniaturizing gas dynamic systems is the scaling of the Reynolds number, according to [86]

$$Re \propto p_0 D_t / T_0^{1.2 \ldots 1.5} \tag{4}$$

where D_t is the throat diameter and the subscript 0 refers to stagnation conditions. In general, for a nozzle to scale without viscous losses, the Reynolds number would need to remain constant [86]. For this to be possible, the stagnation pressure of a system would need to increase proportional to the decrease in thrust, which is hardly ever possible for engineering reasons in actual devices. Experimental data of micromachined nozzles with throat diameters in the order of $100s$ of μm, Reynolds numbers ranging from 10^2 to 10^4 have been reported, with the extreme case of Re<1000 for nozzles $\sim 20\mu$m, where a significant portion of the flow is found in the subsonic boundary region, leading to significant viscous losses therefore significantly decrease in nozzle efficiency [86].

Another example of a physical loss mechanism not decreasing linearly with decreased thruster size are thermal losses in chemical propulsion systems. Considering exemplarily the widely accepted correlation from Bartz for the convective heat transfer coefficient at the throat of a nozzle [87]:

$$\dot{q} = h_g \left(T_g - T_w\right), \quad h_g \propto \frac{D_t^{1.6}}{D^{1.8}} \left(\frac{P_c}{c*}\right)^{0.8} \tag{5}$$

where , P_c the chamber pressure and $c*$ the characteristic velocity and D the diameter at which the equation is evaluated. This relation assumes constant Reynolds number scaling, which from the discussion above, is evidently an optimistic assumption. From Eq. 5 it is evident that the losses due to convective heat transfer scale according to $D_t^{1.6}$, whereas the thrust, which is linear to the mass flow, scales according to $T \propto D_t^2$. This indicates the non-linear increase of thermal losses, and therefore reduced efficiency, with increased miniaturization of propulsion systems which convert thermal energy into kinetic energy of the exhaust by expansion. In addition, this relationship indicates increased heat load on the throat region of such systems with increased miniaturization, which can become crucial for higher performing thrusters: As active cooling, as accomplished in larger chemical propulsion devices, is not expedient for miniature systems due to a number of reasons, including size constraints, complexity and cost, Eq. 5 can dictate the choice of an inefficient operational regime to lower the total heat load by decreasing the hot gas temperature T_g, for example by operating in a non-stoichiometric combustion regime in bipropellant engines.
The miniaturization of reaction chambers in propulsion systems utilizing decomposition or combustion of compounds, such as monopropellant and bi-propellant thrusters, are additionally limited by the residency time τ that the reactants take to complete the intended reaction, which can be formulated as

$$\tau = \frac{L}{v} = \frac{L\rho A}{\dot{m}} \tag{6}$$

where v is the mean propellant velocity, L and A the chamber length and cross section respectively, ρ the mixture density, and $\dot{m}$ the mass flow. This relationship establishes a minimum chamber volume necessary for a given propellant mass flow $\dot{m}$, and thus thrust, that is required in order to avoid inefficiencies due to incomplete reaction. While this provides a minimum volume to avoid losses from incomplete reaction, the increased thermal losses described in Eq. 5 will become increasingly decisive for miniature devices. This trend is amplified by the typical temperature dependency of chemical reactions, which amplifies the negative impact of thermal losses by decreasing the rate of reaction.

B. Electric acceleration

1) Gas phase ionization: With the exception of electrospray and FEEP thrusters, most EP systems rely on ionization of a neutral gas, by either electron collision, RF ionization or contact ionization. In these gas phase ionization systems, namely ion engines and Hall thrusters, the propellant utilization efficiency is therefore primarily governed by the ability to ionize the propellant within the acceleration chamber. In electron collision ionization, the probability for a collision to take place is dependent on the mean free path λ

$$\lambda = \frac{1}{n_e Q} \tag{7}$$

where Q is the collision cross section and n_e the number density of electrons in the plasma. As the mean free path relates to the probability of a neutral particle to experience a collision and ionization, miniaturization of the ionization

chamber needs to scale with the mean free path to retain thruster efficiency. This typically means operating in a higher density regime, which can have a detrimental impact by causing higher losses, such as higher ion fluxes to the chamber walls. In Hall thrusters, permanent magnetic fields are used to force the externally injected electrons on a precessing orbit along the cylindrical discharge chamber to increase residence time and therefore probability of collision with the injected neutral propellant. The radius of precession r_e is given by

$$r_e = \frac{m_e v_e}{eB} \tag{8}$$

where m_e, v_e and e are the electron mass, velocity and charge respectively, and B is the magnetic field strength. As the chamber dimensions need to be matched to the electron precessing radius, increased miniaturization in such devices without efficiency deficiency can only be achieved by increasing the magnetic field strength B, which is either a material parameter of the permanent magnets used, or determines the scaling of power for electromagnets, which increases with increasing miniaturization, reducing the overall efficiency.

Note that in addition, miniaturization of ionization chambers, especially in light of increasing the plasma density, comes with increasing particle flux towards the chamber walls, leading to increased erosion rates, which has a negative effect on lifetime.

2) Electromagnetic, pulsed thrusters: In PPTs and VATs ionization is accomplished in a small envelope along a usually solid propellant surface, whereas the energy transfer into the plasma discharge, and therefore the final kinetic energy of the exhaust, is proportional to the change in the circuit impedance and therefore proportional to the area the discharge circuit is encompassing during acceleration. Ref. [88] showed a quasi linear decrease in thruster efficiency with decreasing pulse energy, showing the increasing significance of plasma resistance for decreasing pulse energy.

3) Liquid phase and field emission ionization: Ion evaporation from a liquid phase or field emission ionization are processes occurring when a conductive liquid is electrically stressed above a threshold where particle extraction is achieved. While the field strength required for such processes is generally high, the electrostatic stressing forces the conductive liquid into a sharp structure, such as a Taylor cone (FEEP), [41] or similar electrified meniscii [89], which amplifies the local field strength at the tip of the structure, allowing for a region of high enough field strength for ionization or ion evaporation. While dependent on the geometry and liquid properties, the force balance at the apex can be expressed by

$$\frac{1}{2}\epsilon_0 E^2 \sim 2\frac{\gamma}{r^*} \tag{9}$$

where ϵ_0 is the vacuum permittivity, E is the local field strength, γ the liquid surface tension and $r*$ the radius of curvature of the emitting region that is formed by the mass flow balance of depleted and replenishing ion flow. The beam composition of FEEP are found to be largely dominated by singly charged ions, in the presence of slower microdroplets, reducing the average beam velocity and therefore specific impulse [90]. While similar in terms of the engineering implementation, electrospray emitters operate by extracting charged particles from an ionic liquid propellant, capable of operating in the pure ionic mode, in which only ions and ions solvated to the n-th degree are emitted [42]. In current versions of thrusters with larger total emission currents achieved by multiplexing the number of emission sites, operation is typically in an ion-droplet mixed regime [57], in which droplets with lower charge-to-mass ratio particles are accelerated to lower exhaust speed, lowering the specific impulse but increasing the thrust produced.

Charged particle extraction processes require local field strengths in the order of 10^9V/m for ion emission. In the case of an ionic liquid emission site, Eq. 9 allows to estimate the size of the region of ion extraction of $r^* \sim 15nm$ (for an ionic liquid with 1Si/m conductivity, $\gamma \sim 0.05$N/m). While the characteristic dimension of the Taylor cone itself could be orders of magnitudes larger than the region of actual charge emission described by r^*, it should be noted that theoretically, thrust densities in the order of MN/m^2 would be possible, for experimentally found emission currents of 100s of nA per emission site, which translates to $\sim 0.05 - 0.1$nN. However, due to the high local field strength required to surpass the threshold for ion evaporation, and the need to amplify the applied electric potential using field enhancing structures to support the Taylor cone, current technology does not allow for such dense packing of emission sites in engineering implementations. The current MIT electrospray thrusters feature 480 emitter structures per square centimeter, while test emitters with up to 4 times the emitter density have been successfully fabricated to date.

C. Systems consideration

Increasing miniaturization of the thruster heads without compromise in efficiency is considered advantageous regardless of acceleration principle, as it not only decreases the structural mass and volume of the system, but may also allow redundant systems. However, any miniaturization only achieves its full potential if its merit is not outweighed by the mass and volume required by auxiliary propulsion components in the propulsion system. While advances have been made in the miniaturization of auxiliary components such as the Busek Microvalve used for the colloid thruster system on LISA Pathfinder [91], the propellant stored itself constitutes a natural barrier towards miniaturization. Any meaningful miniaturization of a thruster technology therefore needs to consider the entire propulsion mass, highlighting the importance of maintaining, or increasing, the propulsion system efficiency with advancing miniaturization. This becomes most important for technologies for space missions with high Δv requirements, as in such cases, the stored propellant mass easily outweighs the mass of the thruster and auxiliary components and it may be more productive to increase the specific impulse of the thruster, than striving to decrease the structural mass of the thruster and components without considering the impact on the overall propulsion system. Based on the discussion in Sections III and VI, two promising approaches shall be highlighted:

Technologies that miniaturize without impact on efficiency: Liquid phase ionization technologies such as the ionic liquid electrospray emitters do not suffer from an inherent decrease of performance with advancing miniaturization of engineering devices beyond sub-millimeter scale. Such systems scale therefore linearly in terms of thrust, and are, in theory, not subject to an adverse impact on the specific impulse. Such systems thus allow miniaturization of the thrusters, without increasing the amount of stored propellant to accomplish a given mission requirement, therefore leading to an overall decrease in the mass and volume of the propulsion module.

Technologies with novel aspects that have positive systems impact: Novel technologies can outweigh inefficiencies that originate from miniaturization of a thruster by having a net positive impact when considering the overall propulsion system. The HYDROS system developed by Tethers Unlimited [66] is a bipropellant system which uses electrolysis of stored water to produce the reactants. Such a system could provide specific impulse similar to a bipropellant engine in the future, but may come with significantly lower structural mass for the propellant storage system compared to a traditional bipropellant system. Even for a modest decrease in efficiency of the thruster, caused by the small size of the bipropellant combustion chamber and considering the added electric power required, the overall system can favorably trade off in terms of the overall system mass and volume required.

An independent systems aspect, that can become significant with increased miniaturization, is potential electromagnetic interference (EMI), especially in the case of miniaturized EP systems. Such interference can become an issue for thrusters which require components with strong, unsteady magnetic fields, RF-thrusters or thrusters with unsteady operation principle, such as PPTs, which require a high current discharge. While shielding measures can to some extent decrease such unwanted interactions, miniaturization necessarily requires closer spacing and smaller margins on components, and care needs to be taken to avoid undesired interaction of high voltage components with sensitive subsystems such as onboard computers.

D. Pushing the envelope

1) Chipsats and membrane spacecrafts: Pushing the boundaries of miniaturization of spacecraft, Kicksat [14] was an early instance of radical miniaturization of semi-autonomous systems, with a 3U Cubesat designed to deploy 128 subsatellites called Sprites, intended to perform basic functionality including communication with the dispensing satellite. While limited in capability, and ultimately not successful due to a malfunction in the dispensing mechanism, this and followup missions in which individual chipsats were launched as independent piggyback payloads to other satellites, highlighted the theoretical capability of extremely miniaturized spacecraft, especially in large swarm constellation concepts. A related concept is pursued by MIT Lincoln Lab and MIT Space System Laboratory, called Wafersat, in which a satellite platform with full capabilities similar to a larger sized Cubesat will be reduced to a mass produceable Silicon wafer form-factor. While such missions pose challenges to a variety of subsystems, including thermal and communications, specific challenges are imposed on future propulsion systems that are deemed an enabling technology for such missions, both for orbit and attitude control. Using the baseline of Wafersat, requirements for propulsion systems regarding mass, volume, power and total impulse can be derived for a propulsion system useable for minor orbit correction and attitude control. Based on a standard 8 inch wafer form factor, the total impulse required for a year of drag compensation ranges from ~ 22.5Ns to ~ 3.35kNs in a 400km altitude orbit, depending on the angle of attack. For an angle of attack of zero, that is with minimal drag cross section, a state-of-the-art electrospray emitter with specific impulse of 1500s could perform a year of drag compensation in such an orbit using $< 0.2g$ of propellant. Another example are ultrathin spaceraft such as the membrane-based Brane Craft, studied by the Aerospace Corporation [15]. In this concept, the spacecraft itself consists of a flexible, thin membrane with large surface to volume ratio, imposing strict volume requirements on the propulsion system that is required to fit within the 30 μm envelope. Ref. [15] concluded that electrospray emission from sharp emitter structures in combination with a separate extractor and acceleration grid could be accomplished within the scale of 10 μm, fed passively with propellant that could be distributed around the emitters in a gap between the spacecraft membrane layers, leading to significant propellant mass per emitter, with 40 hours of accumulated firing time identified as a baseline. In addition, their analysis showed that acceleration of up to 0.1m/s^2 could be achievable, which is orders of magnitude higher than typical EP systems, partly enabled by the favorable large surface area, enabling large solar panel area and therefore a high power per spacecraft mass configuration.

Fig. 13. Deployed sail of the Lightsail 1 Cubesat, from [48]

As outlined in Sec. VI, the currently only available system that theoretically allows controlled propellant acceleration at sufficient efficiency within sub millimeter scale are liquid phase ionization systems, and both the MIT Wafersat as well as the Brane Craft studies use electrospray emitters as their current baseline technology [15].

2) Infinite specific impulse propulsion: Infinite specific impulse propulsion, that is propulsion without the need for propellant stored onboard of the spacecraft, is attractive for long duration, very high Δv missions, potentially enabling interplanetary, asteroid encounter (including the planned NASA

NEA Scout Cubesat mission [92]) and deep space exploration missions, such as the scientifically rewarding targets in the region of the outer planets and Oort cloud, positioning a spacecraft in the gravitational focus point of the sun to form a future powerful telescope [93] or even interstellar flight attempts as proposed by the project Starshot [94], which attempts to propel a miniaturized spacecraft to Proxima centauri using a man made high power light source. Besides mission studies of such long term goals, technology demonstration missions of solar radiation pressure based sail technology have been launched so far, including Jaxa's IKAROS sail [95] and the LightSail-1 Cubesat mission by the Planetary Society [48]. In addition, similar technology has been tested on NASA's Nanosail-D2 Cubesat, but for the purpose of increasing the spacecraft atmospheric drag area for deorbiting purposes [95]. The $14m^2$ solar sail was deployed in June 2010 onboard of the 300kg IKAROS satellite and was able to accumulate a change in orbital velocity of approximately 100m/s until December 2010 [95], [96]. Despite a variety of technical issues on Lightsail-1, it was able to successfully deploy the 32 m^2 solar sail, acting as a precursor for the followup Lightsail-2 misison [48].

VII. Conclusion

The success of small satellites has led to increased mission and scientific capabilities of miniaturized spacecraft, spurred by the decreased mission cost and faster development schedules. As mission complexities evolve, the need for high capability propulsion, previously mostly reserved for traditional large satellite platforms with high power budgets, has driven the development of a plethora of propulsion solutions for small satellites, Cubesats and beyond. This review discusses the different propulsion principles applicable to small satellites, and presents a classification of available propulsion solutions, including a variety of different chemical and EP systems of varying complexity and performance. A classification of these propulsion systems is given, based on the tradeoffs in performance parameters, including thrust, specific impulse and input power. A review of selected space qualified propulsion systems is presented, highlighting key technologies for specific satellite classes. A discussion of the predominant scaling laws for miniaturization of different propulsion systems is given, identifying the limiting factors in miniaturization on thruster, and system level is given. Based on this, a section discussing the propulsion related requirements and state-of-the art technologies for the extreme cases of Chip- and Membrane-Satellites, and Infinite Specific Impulse missions are discussed.

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