

SURVEY OF PROPULSION OPTIONS FOR CUBESATS

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ABSTRACT

At present, no CubeSat except one (CanX-2) has flown in space featuring propulsion. This was acceptable as long as CubeSats were used mostly as university experiments. As CubeSats become of interest to other users in the government and industry communities, a larger range of capabilities may be required than exhibited so far while maintaining the uniqueness of the Cubesat platform. Propulsion capability is crucial in increasing mission capabilities of future CubeSats, including orbit change and raising, formation flying, proximity operations, fine attitude control, or drag-make-up and de-orbit. While some of these tasks may be accomplished with propellantless devices, their applications are limited mostly to a single task, and bear their own risks. In this study, a survey of propulsion technologies applicable to CubeSats was conducted. Only few off-the-shelf design solutions exist today. The survey was thus expanded to include propulsion devices under development and approaching the required design envelope for CubeSats with respect to mass, volume, and power. In some cases, such as for electric propulsion devices, CubeSat architectures themselves may need to be adapted, and be required to feature deployable solar arrays to increase power capabilities. Given the vast amount of ongoing research into micropropulsion technologies, only thruster technologies could be included in this publication. Valves and other feed system components, as well as their integration, are equally important, however, have to be left to a future survey given the extent of material to be covered. Three major propulsion technology areas applicable to CubeSats emerged when conducting this review: (1) Existing technologies, such as butane systems, pulsed plasma thrusters, and vacuum arc thrusters are applicable with no or only minor changes, (2) New thruster technologies under significant development, such as hydrazine monopropellant systems, ion engines, or colloid thrusters could be adapted with some further development, especially in other subsystem areas such as feed systems, power processing units, and power generation, and (3) emerging technologies, such as micro electrospray arrays and micro cavity discharge arrays that offer even higher flexibility due to scalability for CubeSats, and enable compact integration.

INTRODUCTION

CubeSat Background and the Case for CubeSat Propulsion

CubeSats are gaining increased attention in government and industry. Primarily due to their low cost, mass, and size, new mission paradigms may be explored using this spacecraft architecture, such as low cost technology demonstrators, formation flying missions, or inspector satellites. CubeSats to date come in several standardized sets. The original CubeSat configuration is a 10 x 10 x 10 cm cube approximately 1 kg in mass, termed "1U" (Fig. 1). "2U" and "3U" CubeSats are being offered today as well, consisting of 2 or 3 integrated "1U" units up to 10 x 10 x 30 cm in size and approximately 3 kg in mass [1], [2]. Nugent et al [1] and Venturini et al [2] list typical state-of-the-art CubeSat performances for 1- and 3-U sats. Some of the key CubeSat design parameters are repeated here in Table 1 [2]. As can be seen, besides being low in mass and volume, power resources are particularly limited. This assumes no deployable solar arrays, as is currently the case for CubeSats. Bus voltages are also significantly lower as the current spacecraft standard of 28V. Pointing accuracies at present also are fairly loose, limited by the availability of current actuators for attitude control, mainly magnetic torquers. In fact, to date university-led CubeSat activities often do not provide any attitude control at all.



Figure 1: 1U CubeSat [1]

Table 1: Current and Projected CubeSat Capabilities (excerpted from Venturini et al. [2])

Parameter	Capability
Mass (kg)	1.0 (1U); 3.0 (3U)
Size (cm)	10 x 10 x 10 (1U); 30 x 10 x 10 (3U)
Power (W)	1.6 (1U); 10 (3U)
Payload Power (W)	0.5 (1U)
Pointing Accuracy (degree)	1
Pointing Control (degree)	0.5

The current capability constraints of Cubesats are rooted in their history and original intent, namely to provide students with a cost-efficient means to cheaply launch and operate microspacecraft in space and gain experience in actual, hardware oriented spacecraft design and operations. Developed in the 1990s as a result of a collaboration between Stanford and Cal Poly San Luis Obispo universities [2], the first launch of four university Cubesats took place in 2003 onboard a Russian Rockot vehicle and placed Stanford's successful QuakeSat-1 into orbit, among others [1]. Resource limitations, as typical for such student-led activities, limits subsystem capabilities. Initial government led experiments using CubeSats, such as the Ames GeneSat-1, launched in 2006 aboard a Minotaur 1 [1], still exhibits this heritage in that it featured no attitude control system at all, nor was it required due to the mission focus of a biological bacteria growth experiment.

As government and industry develop an increasing interest in CubeSats, required CubeSat capabilities are bound to become more demanding. CubeSat formations, inspector sats, or operations requiring significant maneuverability around Earth or extending even beyond Earth orbit will all require substantial propulsive capability, in terms of delta-v, impulse bit, specific impulse and thrust. A substantial number of attempts to develop micropropulsion capabilities have taken place to address these issues, and will be reviewed in this paper. Micropropulsion efforts underway were not exclusively focused on CubeSats, but involved the AF university nanosat activities, as well as various technology funding sources, both in the US and internationally [3], [4], [5], [6] to address microspacecraft needs in general, generally understood as including spacecraft up to 100 kg in mass. CubeSat applications so far have been extremely limited in number and capability. One CubeSat (CanX-2) flew a sulfur hexafluoride (SF_6) liquefied cold gas system. A cold gas system was also flown on the near CubeSat-sized MEMS Pico Sat Inspector (MEPSI) by the Aerospace Corporation. Propulsion systems are generally complex, involving multiple components such as thrusters, tanks, valves, other flow component devices (regulators, pressure transducers, filters, etc) as well as power processing units for electric propulsion applications, valve drivers, etc, and involve the use of potentially toxic or explosive propellants, all making the integration of propulsion systems into CubeSat architectures a relatively costly proposition, typically beyond the budgetary capabilities of typical university led projects so far. Therefore to date, almost no Cubesat has flown with onboard propulsion, and several of the university nanosat propulsion activities did not materialize into flight opportunities either. Thus, significant technology development in the micropropulsion field remains to be done, in particular in regards to actual flight applications.

Trends in CubeSat Propulsion Requirements

As for larger scale conventional spacecraft, propulsion requirements will vary greatly depending on specific mission requirements. It is therefore impossible to generate a single set of requirements addressing all future mission needs. Neither is possible for conventional spacecraft either. However, a few general trends may be observed.

As pointed out in Table 1, mass, volume and power constraints of CubeSats dictate stringent constraints on any propulsion subsystem, given that it can only use up a portion of these resources. Conventional interplanetary spacecraft may have propellant mass fractions in excess of 60%, simple attitude control function require propellant mass fraction of only a few percent. Requirements for delta-v may vary equally widely for CubeSats depending on the mission. Orbit raising or plane change require approx. 135 m/s per degree of plane change and 0.6 m/s per km of altitude change in low-Earth orbit (LEO) at 250 km altitude according to Wertz and Larson [7] (Table 2), slightly decreasing with altitude. Depending on the altitude and plane change requirements, total delta-vs may range from a few hundred m/s to perhaps well in excess of 1 km/s. Depending on how much time is available to conduct such a maneuver, this may require either high-Isp electric propulsion options, or high thrust-to-power (T/P) low Isp or chemical propulsion options.

Power resources available on CubeSats are likely the most constrained at the present time, not exceeding but a few Watts of total spacecraft power, however, also offer the largest growth potential. At present, CubeSats do not offer deployable arrays due to complexity, reliability, and cost. Where these are available, available power levels may be increased significantly, perhaps as high as 100 W for a 3U CubeSat [2]. Any survey of propulsion hardware applicable to CubeSats needs to account for the possibility of such “high-power” options, potentially available in the future.

Impulse bit (Ibit) requirements for attitude control have been estimated previously by Blandino, cited in Ref. [6]. Based on impulsive maneuvers, and a “deadband” time between firings of 20 and 100 seconds, respectively, required impulse bits for a 1 kg spacecraft may be as low as 29 μ Ns for 1 degree (17 mrad) pointing requirements for 100 sec deadband intervals, and decrease to as little as 0.034 μ Ns for smaller pointing requirements of 0.02 mrad [6]. Slew requirements, based on the arbitrary assumption to perform a 180 degree slew in one minute, may be as low as 0.06 mN [6]. These numbers may increase significantly, though, based on spacecraft mass and unique mission requirements. A 10-kg microinspector satellite under previous development by JPL, for example, required up to 10 mN thrust to allow for sufficient maneuverability during proximity operations for inspection of a larger spacecraft, while allowing for sufficient thrust margins for collision avoidance [8]. In that case, a chemical butane system was chosen to operate the spacecraft within its available power levels of under 14 W.

Formation flying thrust requirements will depend highly on the needed positioning control for the formation, which in turn depends on the mission focus. Optical observations by means of interferometry have very tight positioning control and thrust requirements, ranging into the milli and even micro-Newton range even for conventional spacecraft in the 1000 kg range, but may, given the physical constraints of CubeSats in accommodating the optics, not necessarily be an ideal candidate for CubeSat architectures. Looser requirements may result for distributed spacecraft architectures, whose sole purpose is to distribute subsystems over various small spacecraft to avoid detection, have redundancies in case of failures, or conduct tensor mapping of regions of space, such as measuring magnetic fields, solar winds,

Table 2: Delta-v Requirements for Orbit raising and Plane Change at Different orbital Altitudes (excerpted from Wertz and Larson [7])

Earth Orbital Altitude	Plane Change (m/s)/deg	1 km Altitude Change (m/s)
250	135.35	0.58
500	132.86	0.55
750	130.97	0.52

etc. Given this large range of potential mission applications, thrust requirements may range equally widely. Micro-Newton to milli-Newton levels at a continuous or near-continuous thrust may be required, but are subject to further detailed requirements studies.

Last, but by no means least, propulsion subsystem components other than thrusters have to be considered. Although thrusters are naturally the first focal point of propulsion subsystem miniaturization, other components, such as valves, regulators, filters, etc. will have to be miniaturized and light-weighted to match CubeSat mass, volume and power envelopes. As will be seen in the course of this paper, even the smallest conventional commercial-off-the-shelf valve hardware, based on solenoid architectures, require several Watts to open and hold open, often exceeding current CubeSat power budgets. New ultra-low power technologies will need to be explored, such as piezovalves or micro-isolation valves [8], [9]. Propulsion feed system architectures will also need to achieve higher degrees of integration to meet CubeSat volume constraints. Individual components plumbed and welded together will quickly use up valuable real estate even for the smallest available valve components. Integrated feed systems, based on MEMS [6] or ChEMS technologies (see Fig. 2) will be required, and will be reviewed below. Multi-functional structure approaches, where thruster and valve components directly flange onto the tank, are another option that were previously explored at JPL under the micro-inspector project [8] (Fig. 3).



Fig. 2: VACCO Integrated ChEMS Thruster/Feed System Module

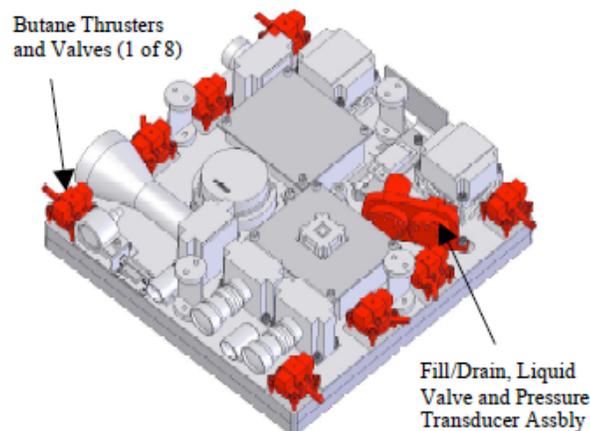


Fig. 3: Multi-functional Tank Structure with Integrated Propulsion Components for the JPL Micro-Inspector Satellite [8]

Reviewing these requirements trends, it becomes quickly apparent that they are as varied as for conventional spacecraft and thus require, not surprisingly, a suite of different propulsion options ranging from low-power, high thrust chemical to high specific impulse (Isp) electric propulsion systems. Even propellant-less systems, such as solar sails or tethers may find applications in limited, special cases.

Scope of this Survey

The large range in propulsion system requirements outlined in the previous sections will be reflected in the survey of propulsion options undertaken in this paper, including chemical, electric and propellantless propulsion systems. This survey builds upon a detailed micropropulsion surveys conducted previously [6], [9], [10], with a substantial amount of new material having been added. Compared to a detailed survey conducted by the author a decade ago [6] the increased level of international activities is particularly noteworthy, with US activities mostly focuses on DoD sponsored projects, possibly due to the shifting focus and resulting demise in NASA's technology funding in recent years.

This survey will not be limited to off-the-shelf hardware exactly matching CubeSat requirements today. There are only very few thruster technologies that fit into this category at present. Also, since CubeSat system requirements may change in the future, a survey of such existing options may be too limited given the potential of future applications. Therefore, thruster technologies, either existent or under significant development, are being reviewed that either fit or approach the CubeSat envelope, thus pointing to areas of possible development to match CubeSat requirements of the future.

The extent of this survey unfortunately does not allow for the inclusion of valve and other feed system components, although their miniaturization and integration are as important as similar efforts underway for thrusters. At this point the author may only refer to a previous survey of the large amount of MEMS activities in this field, mostly confined to research activities [9]. Off-the-shelf miniature valve technologies can also be found in the cold gas thruster section of this paper. A more detailed survey of this aspect of micropropulsion will need to be left to a future publication.

CHEMICAL PROPULSION OPTIONS

Chemical propulsion options are generally characterized by limited specific impulse (< 300 sec), but thrust levels that may exceed several Newton for small thruster technology. Chemical thruster options are this of interest in CubeSat applications, where rapid maneuvering is required, such as rapid plane change or orbit change, or proximity operations near other spacecraft, such as for inspection purposes, where sufficient agility is required for collision avoidance. Several thruster options exist, including monopropellant, bipropellant, and solid motors. Given the mass and volume constraints of CubeSats, less complex, lower dry mass monopropellant or solid motor options are of particular interest.

Monopropellant Thrusters – Hydrazine

Hydrazine thrusters are the workhorse for attitude control and small delta-v applications in conventional spacecraft applications. Hydrazine is decomposed catalytically in a catalyst bed consisting of iridium coated alumina pellets. Hydrazine thrusters emerged as early as 1949 through development work at JPL, and have been in flight applications since 1966 [6]. Commercially available thruster hardware, however, is too heavy and large, and thruster valves are too power consuming to be used on CubeSats even when used as a single main engine for delta v applications [6]. A recent development of a hydrazine milli-Newton thruster (HmNT) at JPL, however, may fit the mass and volume envelope of a CubeSat for such an application [11], [12] (Fig. 4). The HmNT was not developed, however, with CubeSat applications in mind, and rather targets precision pointing and formation flying missions of standard large scale spacecraft buses, allowing for the replacement of reaction wheels, eliminating wheel jitter and reducing mass if tied into an existing coarse hydrazine attitude control system.

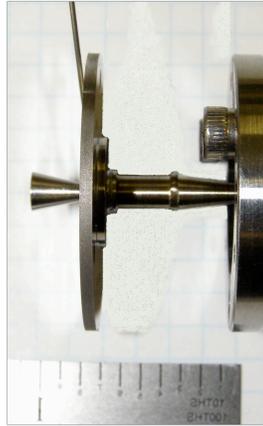


Figure 4: JPL Hydrazine Milli-Newton Thrusters (HmNT) by Parker et al [11], [12]

Table 3: Small Hydrazine Thruster Performance Comparisons

Parameter	SoA MR-103H	HmNT [11], [12]	MEMS/Yuan and Li [14]
Thrust (mN)	1,070	129	0.74 – 1.44
Isp (sec)	220	150 (est.)	162
Volume (cc)	94	8	-
Power – Valve (W)	5 (open)/1 (hold)	8 (open)/1 (hold)	-
Power – Catbed (W)	1.5	0.25 (est.)	-
Min. Impulse (μ Ns)	5000	50	-
Mass (g)	195	40	-

Typical HmNT characteristics are shown in Table 3. Noteworthy is the significant 100-fold reduction of the minimum impulse bit over state-of-the art COTS hardware to 50 μ Ns. Thrust levels have been measured on a JPL micro-Newton thrust stand to 129 mN. The thruster weighs 40 g in its current laboratory model stage, and is about an order of magnitude smaller in volume than SoA hardware.

While too large to be used in sets of multiple units for attitude control, one may contemplate the use of a single such thruster for small to intermediate delta-v applications. The use of hydrazine poses hazards in particular in the university environments due to its toxicity and flammability, however, future government or industry applications of CubeSats may not be as affected by such considerations due to the extensive experience in handling hydrazine. Valve power, however, remains too high for CubeSat applications with current power capacities (Table 1) at 8 W valve opening and 2 W valve holding power. However, small impulse bit performances, requiring fast acting valves, will not be needed in a delta-v applications, and outfitted with a slower acting valve will reduce power requirements, and may be compatible at least with higher power CubeSat version in the future. Already, the HmNT would be suitable for use on somewhat larger microspacecraft, in the 20-30 kg range.

Other developments in miniature hydrazine thrusters include a lower-power miniature hydrazine thruster version that was envisioned by Platt [13], outfitted with microvalves by Lee Inc., operating at power levels of 0.5 W (holding), however, limiting the flow to 10^{-4} gram/sec and currently constrained at 120 psi pressure. A MEMS based thruster versions was tested by Yuan and Li [14] in Taiwan. This thruster was fabricated out of silicon using MEMS micromachining techniques. It featured a catalyst bed consisting of micromachined iridium coated pillars (Fig. 5), and integrated a planar MEMS micro-nozzle with a throat width of 60 μ m and height of 77 μ m, half angle of 22.5° and an expansion ratio of 15. An integrated electrostatically actuated valve was also provided as part of the assembly. The valve

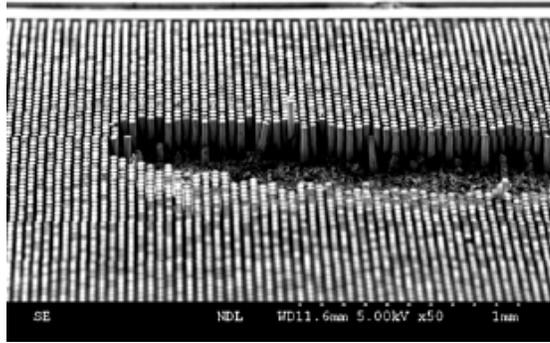


Figure 5: Micromachined Hydrazine Catalyst Bed featuring Ir-coated Si micromachined pillars [14]

diaphragm (poppet) was pre-stressed to increase contact forces with the seat, further aided by thermally stressing through heat soakback from the thruster – a feature that can obviously not be exploited when the thruster has not been in use for extended periods of time. Actuation forces provided by this type of valve, however, are small [9], and seating force through pre-stressing the membrane have to be traded against available force margins. Thrust stand tests were performed and indicated 0.74 – 1.44 mN of thrust at 0.48 – 0.96 mg/sec of hydrazine flow (Table 3). Vacuum specific impulse was estimated at 162 sec at 0.99 mN thrust [14]. Future development could render such a thruster as an option for CubeSat attitude control, ideally in connection with larger scale HmNT-derived devices for delta-v applications.

Monopropellant Thrusters – Hydrogen Peroxide

Scharlemann et al. [15], [16] at Austrian Research Centres Seibersdorf (ARCS) have been developing hydrogen peroxide monopropellant micro-thrusters for microspacecraft applications (Fig.6). Hydrogen peroxide has been used since the dawn of the space age in many famous, and some infamous applications, such as propellant to power the fuel pumps of the V-2, and post-war US Redstone, Jupiter and Viking missiles, and as monopropellant on the early crewed US flights of Mercury, as well as the X-1 and X-15 rocket planes [6]. Its use in spacecraft applications decreased with the advent of higher performing hydrazine propellant.

However, the use of hydrogen peroxide has regained attention recently in the microspacecraft community, as its lower degree of toxicity may make it more amenable to use by university groups, which have led microspacecraft activities so far [6]. Measured thruster performances by Scharlemann et al [15],



Figure 6: Miniature Hydrogen Peroxide Thruster by Scharlemann et al. [15]

[16] indicate thrust capabilities in their laboratory prototype of 100 – 800 mN and a vacuum equivalent specific impulse of 153 sec (ambient specific impulse was determined to be around 100 sec). The thruster uses a monolithic catalyst bed impregnated with sodium permanganate [16] (Table 4). As for the HmNT described above, these thrusters are too large to be used for attitude control functions onboard a CubeSat, but maybe used as main engines for small to intermediate delta-v.

Hydrogen-peroxide thruster development was also underway by Sahara et al. in Japan [17] and by Kuan et al. [18], [19] in Taiwan, both similar in size as the Austrian device. No thrust and Isp performance are available in the literature for the Japanese device [17]. Kuan et al [18] paid particular attention to the thermal design of the catalyst chamber, and arrived at a device approx. 8 (dia.) x 13 mm in size with a throat diameter of 0.5 mm dia. and 5.8 g in weight, delivering 182 mN thrust at 101 sec specific impulse at ambient conditions, which was estimated to equal 221 mN of thrust and 125 sec of specific impulse under vacuum conditions [18] (Table 4). Silver flakes served as catalyst material and were packed into a catalyst bed retained by a stainless steel mesh. Typical operations requires preheating the bed with 10-15 Watts of power to attain steady state thrust values within 50 ms. Operations without pre-heating was conducted in laboratory tests and resulted in achieving steady state chamber pressures and thrust after 15 sec [18]. While this may be acceptable for delta-v applications, cat bed lifetimes may be compromised by cold starts, as for conventional hydrazine thrusters.

MEMS based devices had under development at NASA Goddard Space Flight Center (GSFC) by Hitt et al. [20]. The GSFC thruster consisted of an entirely MEMS fabricated planar catalyst bed and nozzle design machined into silicon. The catalyst consisted of micromachined silicon pillars, like those later reproduced by Yuan et al for the MEMS hydrazine thruster [14], coated in silver as the catalyst for peroxide decomposition (Fig. 7). The initial experiments were plagued by incomplete hydrogen peroxide decomposition, occurring only near the silver coated pillar surfaces, and not in the bulk of the propellant flow. This may have been due to incomplete mixing in the low Re number flow regime [20] or due to the large surface-to-volume ratio of a highly thermally conductive Si chamber, which may have led to excessive heat losses [18].

Table 4: Small Hydrogen Peroxide Thruster Performance Comparisons

Parameter	ARC Seibersdorf [15]	Kuan et al. [18]
Thrust (mN)	100-800	221
Isp (sec)	153	125
Size (mm)	-	8 x 13
Mass (g)	-	5.8
Heater Power (W)	-	10-15 (cat bed pre-heat)

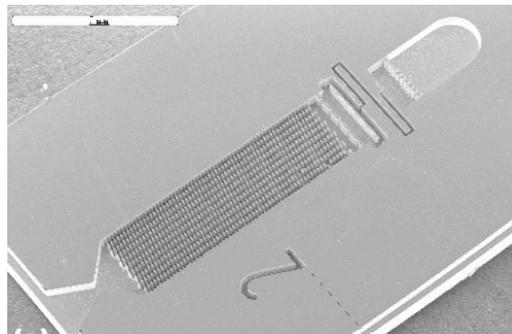


Figure 7: Micromachined Hydrogen Peroxide Thruster by Hitt et al [20]

Hydrogen peroxide thrusters may fill a similar function as HmNT derived hydrazine thrusters for CubeSats, and provide small to intermediate delta v with miniature conventional design, attitude control with MEMS based devices if they could be realized. The benefit of non-toxic hydrogen peroxide, however, must be weighed against storability concerns of this propellant. In a previous review [6] it was pointed out that hydrogen peroxide faces storability concerns. Hydrogen peroxide slowly decomposes when heated or exposed to a catalyst. Almost any organic residue can act as such a catalyst. Slow decomposition in propellant tanks can lead to tank overpressurization over time. However, Scharlemann et al. [16] point out that decomposition rates can be reduced by using high purity hydrogen peroxide, adding stabilizers, and employing surface passivations. Dedicated tests will be required to verify if hydrogen peroxide storage stability compatible with mission requirements can be achieved.

Cold Gas Thrusters

Cold gas systems offer the greatest degree of simplicity of all propulsion systems. Basically, gas from a high-pressure gas supply tank is vented through a valve and nozzle to produce thrust. Cold gas thrusters, besides their simplicity, offer low impulse bits and contamination-free and non-toxic propellant operation, and are space-proven technology since the 1960's [6]. They are particularly suitable for space applications where a combination of low impulse bit, exposure to contamination sensitive spacecraft surfaces, and low total impulse requirements is given. Given that they were designed for low-impulse bit attitude control for spacecraft, existing thruster hardware is small and light weight. Table 5 shows SoA cold gas thruster hardware. As can be seen, thrust values range from 4.5 mN to 100 N. The smallest thrusters, such as Moog's thrusters originally intended for attitude control on the Pluto Fast Flyby mission, weigh as little as 9 grams and are as small as 12 x 35 mm.

However, most of these thrusters were never designed with microspacecraft applications in mind, and the power levels required for valve actuation, although small by conventional standards, i.e. around 10 W (opening) down to < 1 W (holding) for the smallest units, still far exceed today's CubeSat capabilities (see Table 1), in particular if one further takes into account that at least two thrusters are typically fired in common for attitude control purposes. One exception is the thruster developed by Marotta, which was envisioned for microspacecraft use slightly larger than CubeSats on the NASA's ST-5 New Millennium mission, and has a pull-in power of less than 1 W, however, at the expense of a larger thruster mass of around 70 grams.

Other than the aforementioned miniature hydrazine and hydrogen peroxide thrusters, cold gas thrusters are also not suitable for delta-v applications since specific impulses are low, typically around 65 sec for nitrogen propellant, leading to large propellant mass fractions. This increased propellant mass fraction further increases propellant system mass since heavy tankage is required to contain the high-pressure gas supply. The large storage pressure also creates a risk of propellant leakage across valve seats, in particular after valves have seen extended use throughout the mission, increasing the likelihood of deposition of contaminants onto valve seats [6]. Finally, the cold gas thrusters listed in Table 5 require conventional integration using tube-welding approaches. This leads to relatively bulky systems in volume constrained CubeSat systems.

MEMS-based cold gas thruster systems have been developed in Sweden by Stenmark et al. [4], [6], [21] – [23]. These thrusters eliminate the integration concerns mentioned above by combining four thrusters with integrated valves, featuring stacked piezoactuators, filters and heaters (Fig.8). The use of piezovalves also reduces power requirements, and stacked piezoactuators provide sufficient valve stroke at merely 24 V [6], [22]. Despite these impressive miniaturizations and integration schemes, this system will still have to contend with high pressure gaseous propellant storage and the aforementioned leakage concerns and mass penalties resulting from it.

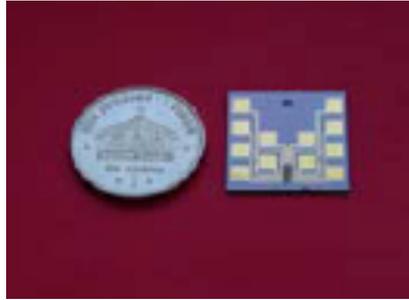
A novel gas generator concept has been under development in the Netherlands, addressing these high-pressure gaseous propellant storage and leakage concerns of conventional cold gas systems (Fig. 9) [24]. Rather than relying on high-pressure tanks, propellant is stored in solid gas generator cartridges

Table 5: Cold Gas Thruster Performance Comparisons

							
Manufacturer	Moog 58X125A	Moog 58E143 58E144 58E145 58E146	Moog 58E142	Moog 58E151	Marotta Cold Gas Micro- Thruster	Moog 58-118	Moog 50-820
Model							
Propellant	N2	N2	N2	N2	N2	N2	N2
Thrust (N)	0.0044	0.016 - 0.040	0.12	0.12	0.05 - 2.36	3.6	52-105
Mass (g)	9	40	16	70	<70	23	430
Size (mm)	11.9x34.7	13.97x57.2	14x20.3	19.05x 40.87		6.6x25.4	98.2x104.1
Valve Power (Open) (W)	10	10	<35	10.5	<1	30	47
Valve Power (Hold) (W)		1					
Isp (sec)	65	>60	>57	65	65	65	65
Operat. Pressure (psia)	0-50	0-36	50-300	0-400	100-2240	230	215-2515
Proof Pressure (psia)	300	290	600	1015	3360	1115	3765
Burst Pressure (psia)		508	1200	1615	5600		6265
Response (Open) (ms)	2.5	2.5	3.5	5	5	<4	<10
Response (Close) (ms)	2.5	2.5	3.5	3	5		<10
Minimum Ibit (mNs)					<44		
Life (No. of Cycles)	>15,000	500,000 - 2,000,000	20,000	1,000,000		>10,000	>6,000
Status	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual	Flight Qual
Comments	Brilliant Pebbles, SAFER, Pluto Fast Flyby	CHAMP, GRACE	SIRTF	SIRTF	Developped for GSFC Nanosats, ST- 5	SCIT, SAFER, Pluto Fast Fly- by	COMET, Pegasus Cluster of 3 thrusters: 2 @ 52N, 1 @ 105N



(a)



(b)

Figure 8 (a) MEMS –based Cold Gas Thruster Module, (b) Cold Gas Thruster Chip [4]

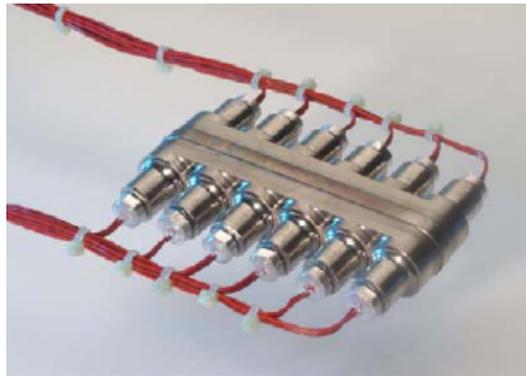


Figure 9: Gas Generator Cartridges Demonstrator [24]

and upon ignition of these released into a plenum from which nitrogen gas can be drawn for propulsive use. Since only a fraction of the gas is only stored in the plenum shortly before use at any time of the mission, while the bulk of the nitrogen gas is bound in the solid gas generator cartridges, tank volume requirements and leakage concerns are reduced. Originally developed by TNO and Bradford Engineering Corp. in the Netherlands, the cartridges release pure nitrogen from a proprietary substance upon pyrotechnic ignition (Fig.9). Since pyrotechnic igniters are banned on CubeSats, Rackemann et al. [24] developed a resistive igniter, estimated to require approximately 2.5 W for 30 seconds [24].

Further miniaturization of the demonstrator unit is required to be usable on CubeSat type spacecraft, requiring smaller cartridges to be used in sequence to allow for a more consistent gas supply from a smaller plenum as required for Cubesats. Rackemann et al. estimates that volume requirements of a gas generator system may be reduced by a factor of six over a high-pressure tank based cold gas system, however, mass reduction is significantly less, around 20%, due to cartridge mass, its solid base material, and required plumbing.

Liquefied Gas Thrusters

Liquefied gas thrusters use propellants, such as butane or ammonia, which can be stored in their liquid form, yet phase transfer into gas upon expansion. Within the propellant tank, due to self-pressurization due to ambient heating from the spacecraft, the gas remains in its liquid form for the largest part, the ullage being filled with gas-phase propellant at its vapor pressure for the given tank temperature. For butane, this pressure value ranges between 40 – 100 psia for typical spacecraft temperatures. This allows much lighter, lower pressure tank constructions, significantly reduces the leakage concerns, yet maintains the simplicity of cold gas thruster technology upon expansion of the liquid propellant from the tank into a plenum.

Butane thrusters were first built and flown onboard the SNAP-1A spacecraft by Surrey Space Centre in England in 2000 [25]. The Surrey system pursued a low-cost approach by constructing the tank from bend large diameter tubing, capped off at the ends with tube fittings. Given the low pressure propellant storage, this was a feasible approach. Unfortunately, the Surrey system did not use a plenum tank. Butane was expanded from the tank directly into the thrusters. Incomplete vaporization of butane may have led to the expulsion of butane in its liquid phase, which led to extremely low specific impulses and rapid propellant expenditure [25].

A JPL micro-inspector concept employed a separate gas plenum [8]. The JPL micro-inspector was based on a pre-cursor, the Low Cost Adjunct Microspacecraft (LCAM) pioneered by Collins at JPL [8]. Here, the central butane liquid propellant tank was surrounded by a plenum, into which the butane was allowed to expand and vaporize completely. This was aided by waste heat from the spacecraft transferred into the tank by mounting heat-generating avionics flush with the tank. This “multi-functional tank” (MFT) architecture invented by Collins is seen in a later iteration in Figs 3 and 10 for the Micro-Inspector spacecraft. As can be seen, the plenum surrounds the liquid tank, including its orbital weld and so acts as an additional safety feature containing the inner liquid tank. The low tank pressure (less than 100 psi even in worst temperature cases) allowed for a flat tank geometry except for slightly curved internal tank surfaces to facilitate the desired maximum tank pressure. The flat surfaces provided interfaces for avionics board mounting (providing heat of vaporization to the butane propellant and in turn allowing for cooling of the avionics), as well as flanged thruster and valve components (shown in red in Fig. 3). A latching and liquid control valve allows transfer from the liquid into the plenum tank based on pressure sensor feedback. From the plenum tank, eight thrusters are fed to produce attitude control and small delta-v. Total delta-v capability was 15 m/s for Micro-Inspector, sufficient for the intended use of spacecraft inspection of the host spacecraft.

As for CubeSats, Micro-Inspector was power limited to approximately 14 W, and lower power valve technology was required. VACCO Industries developed a cold gas thruster featuring piezovalue

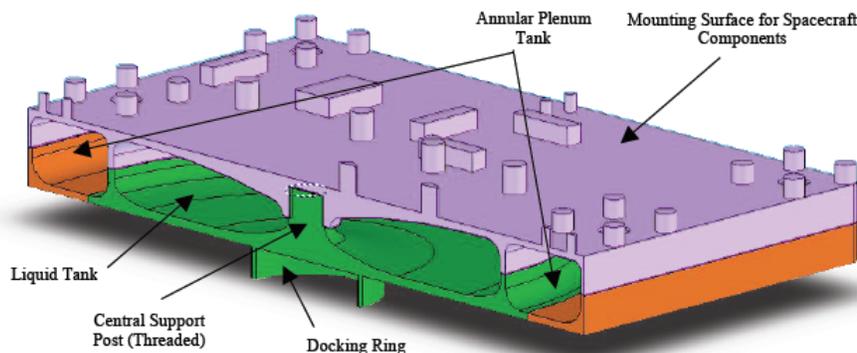


Fig. 10: JPL Micro-Inspector Multifunctional Tank Design featuring Liquid and Plenum Tank [8].

technology, as well as a piezovalue based liquid control valve, miniature latch valve and pressure transducer assembly, and the miniature fill & drain valve. The piezovalue allowed for opening operations of 1 W or less based on ramp up speeds, and micro-Watt levels of holding power, and is thus uniquely suited for microspacecraft applications. The cold gas thruster provide up to 25 mN of thrust and the specific impulse was estimated to be around 70 sec based on thrust and mass flow measurements conducted at JPL (Fig. 11). Thruster performance data are summarized in Table 6.

In parallel to the aforementioned activities, VACCO developed an integrated butane propulsion system based on its ChEMS™ (Chemical Etched Micro Systems) technology [26] – [28]. ChEMS™ is a VACCO developed metal-micromachining technique. While not allowing for the same level of precision micro-features as are achievable with MEMS in silicon, it far exceeds traditional metal-machining techniques and is sufficient for the types of miniaturizations needed for CubeSats. It relies on the chemical etching of metal disks that are then diffusion bonded into stacks to build up three-dimensional structures. Dropping in micro-coils allows for complex manifolds with integrated valves, thrusters, etc.[28].

Developed originally for the MEMS PicoSat Inspector (MEPSI) by Aerospace Corporation, this system was the first butane propulsion system designed specifically for CubeSats. Termed the Micro Propulsion System (MiPS), it consists of a self-contained unit featuring five thrusters, integrated liquid and plenum tank with internal control/isolation valve, pressure and temperature sensors, and filters (Fig. 12). The self-contained unit thus contains all propulsion components and the butane fuel, allowing for a modular design that can be cleanly interfaced with a CubeSat bus by merely attaching it on to its side (Fig. 13). In fact, in addition to the micro-pulsed plasma thruster (PPT) by Busek and the Vacuum Arc Thruster (VAT) by Almeda Applied Science Corp (AASC Inc.) to be reviewed in the Electric Propulsion Section further below, it is one of only a few propulsion systems that are available off-the-shelf today, ready for CubeSat use, and are able to operate within its current spacecraft mass, volume, power, and bus voltage constraints. As the PPT and VAT are very low thrust and inherently impulsive thrusters, the

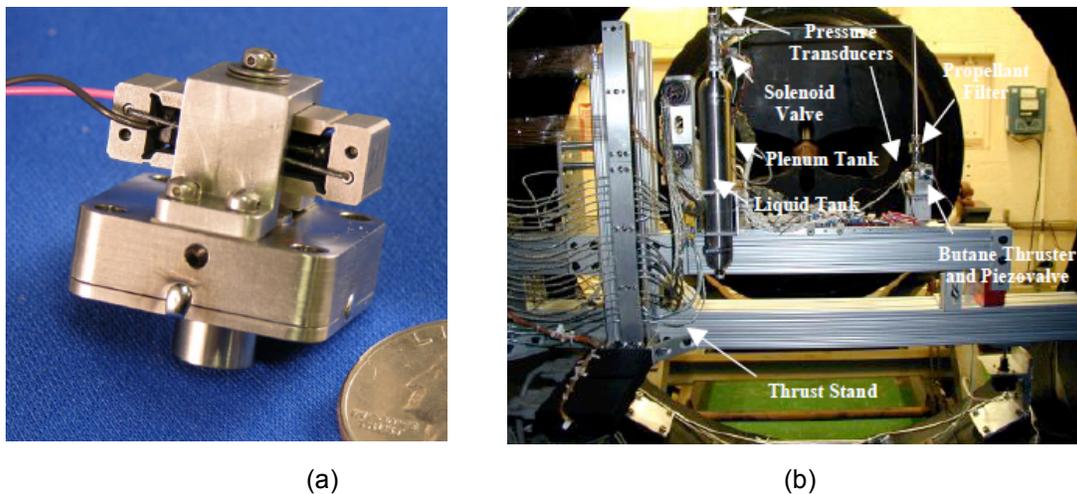


Fig. 11: (a) VACCO Piezoactuated Butane Thruster [27] (b) on JPL Micro-Newton Thrust Stand [8]

Table 6: Micro Butane Thruster Performances

Vendor	Propellant	Thrust (mN)	Isp (sec)	Mass (g)	Power (W)
VACCO	Butane	10-25	70	30	1 (open)/ 10 μ W (hold)



Fig. 12: VACCO MiPS Module [27]

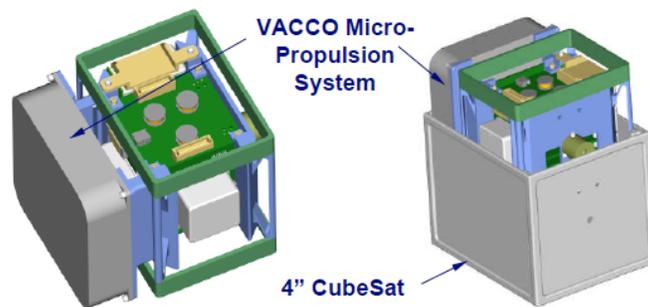


Fig. 13: VACCO MiPS CubeSat Integration [28]

VACCO ChEMS™ butane module is the only readily available chemical thruster system, able to operate with continuous thrust force.

Typical performance data for the VACCO MiPS is listed in Table 7. Total system dry mass (including the tank) is 456 grams and propellant capacity is 53 g of isobutene. The thrusters deliver 55 mN thrust, and the total impulse capability of the units as shown in Fig. 12 is 34 Ns at a total delta-v capability of 34 m/s. The valves are of the solenoid type and have demonstrated over 90,000 cycles.

VACCO extended the MiPS design in collaboration with Boeing [27] to the “Palomar” design (Fig. 14). This system has increased capability, and in addition to butane may be operated on nitrous oxide (N₂O). Nitrous oxide was found to decompose exothermally when heated. The Palomar system features two decomposing thrusters for delta-v applications, and 16 cold gas N₂O thrusters for attitude control, plus tankage, control valves, filters, pressure and temperature sensors. It weighs 1.39 kg dry, has a propellant capacity of 96 grams at 264 – 1500 psia storage pressures for nitrous oxide, requiring spheroid tank shapes compared to lower pressure butane applications, which can employ flat tank designs. Performances estimated based on nitrous oxide propellant are 141 Ns (decomposing) and 56 Ns (cold gas). The Palomar system to date has been demonstrated with isobutane [28].

These demonstrated performances and high levels of integration aided by low pressure butane storage make butane systems ideal candidates for low delta-v applications on CubeSats and attitude control, where rapid reaction and slews are required and power levels are limited.



Fig. 14: VACCO/Boeing Palomar MiPS Module [27]

Table 7: VACCO MiPS and Palomar Module Performances [27] – [28]

Parameter	VACCO MiPS	VACCO Palomar
Propellant	Butane	N ₂ O
Thrust (mN)	55 (per thruster)	-
Mass (g)	456	1,393
Propellant Mass (g)	53	96
Total Impulse (N-s)	34	56 (cold gas); 141 (decomposing)
Delta -v (m/s)	34 (for 1 kg MEPSI CubeSat)	-

Solid Rocket Motors (SRMs)

Solid rocket motors have been used since the dawn of the space age, mostly in the function of kick-stages or orbit insertion [6]. It is for these functions that they may become of interest for CubeSat based missions as well, i.e. orbit raising, trajectory insertions, rapid de-orbiting etc. As for conventional spacecraft, SRMs would not be integrated into the spacecraft bus, but the motor will be attached to the spacecraft, in this case to the CubeSat. Thus, the SRMs do not necessarily have to meet CubeSat mass or volume constraints, although there will be some short term power demands for motor ignition. However, for spacecraft as small as CubeSats, available thrust levels and resulting vehicle accelerations are a major selection criteria.

Most SRMs today use a propellant consisting of an HTPB (Hydroxyl-terminated Polybutane) matrix with aluminum particulates as fuel additive and ammonium perchlorate (NH₄ClO₄) as oxidizer. Since the propellant is solid and no valves are required, SRMs allow for tight packaging and easy integration while meeting sizable delta-v requirements given that specific impulse values even for small motors reach above 250 sec, higher than those of mono-propellant systems and approaching those for small bipropellant thrusters. Disadvantages of solid motor are that they are generally not restartable and require orbit trimming due to thrust and total impulse uncertainties, thus requiring additional propulsive capability unless inaccuracies of the solid burn can be accepted. In addition, as mentioned above, the selection of suitable small motors is limited. Most smaller motors have been used as retro rockets, or for stage separation on launchers and were required to deliver large thrust values over very short burn durations, often under 1 sec [6]. Resulting vehicle accelerations for a CubeSat would be excessive, requiring costly requalification of spacecraft subsystems for high accelerations.

A few exceptions to this trend are listed in Table 8. The STAR 5A motor by ATK is a so called end-burner, burning slower than most motors, producing 169 N of thrust over a burn duration of 32 sec at an Isp of 250 sec (Fig. 15). Thrust forces and accelerations would still be high, but significantly less than for most space qualified off-the-shelf SRM design [6]. The STAR 5A could provide around 1.3 km/s delta-v to

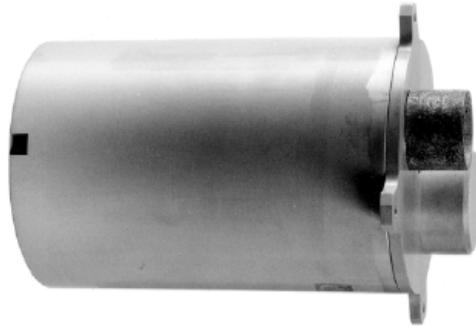


Fig. 15: ATK STAR 5A Solid Rocket Motor (5 in. dia.) (from ATK STAR Motor Catalogue)

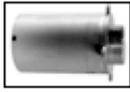


Fig. 16: ATK STAR 4G Solid Rocket Motor (from ATK STAR Motor Catalogue)

a 1U CubeSat at an average vehicle acceleration of around 4 g. The ATK STAR 4G listed in Table 8 was designed for the New Millennium ST-5 mission (Fig. 16). At 257 N thrust (average) and a burn time of just over 10 sec, accelerations would be even larger. A 1U CubeSat could attain a delta-v of approximately 1.35 km/s at an average vehicle acceleration of around 13 g.

An even smaller motor was designed by Teasdale et al. [29] for the Smart Dust mission concept, where distributed microspacecraft based sensor arrays were to be dispersed in a region of space. The motor was designed to facilitate the initial dispersion of these sensor arrays, not formation control subsequent to that. The motor consisted of an Alumina ceramic thrust chamber, and silicon micromachined nozzles with integrated polysilicon heaters for igniters. The igniter requires approx. 0.2 – 0.3 W for less than 1.5 sec. A HTPB fuel and ammonium perchlorate oxidizer propellant combination was used. Interestingly, no Al particles were added to the propellant mix as Al slag formation during burning clogged the 0.5 mm dia. micro-nozzle throat. The motor weighed 0.75 gram fueled with a propellant mass of 0.12 gram. Combustion chamber length is 12.7 mm, OD is 4.8 mm and ID is 3.2 mm. Peak thrust values of 15 mN were measured over the course of the first 1 sec of the burn, settling into a steady state thrust value of 2 mN over a burn time of approx. 7 sec. The specific impulse was given as only 14 sec [29]. From that a total impulse value of approx. 16.5 Ns can be estimated, using the propellant mass above.

Table 8: ATK Small STAR Solid Rocket Motor Performances

		
Manufacturer	ATK	ATK
Model	STAR 4G	STAR 5A
Propellant	TP-H-3399	TP-H-3399
Thrust (avg) (lbf)	58	38
Thrust (max) (lbf)	69	38
Mass (Loaded) (lbm)	3.3	10.24
Propellant Mass (lbm)	2.16	5.05
Burnout Mass (lbm)	1.07	5.08
Case Material	Graphite-Epoxy	Al
Size (in)	4.45x5.43	5.13x8.84
Burn/Action Time (sec)	10.3/10.8	32.0/35.6
Specific Impulse (sec)	269.4	250.8
Total Impulse (lbf-sec)	595	1289
Status	Developm.	Flight-Proven
Comments	Developed for GSFC for nanosats. High chamber pressure, low-eroding nozzle insert. Rel. long burn time	Long burn time motor for slow acceleration. Well suited for small spacecraft use.

A team of the Universities of Singapore and Arkansas took SRM design to an even greater extreme by building entirely MEMS fabricate SRMs out of silicon [30] – [32]. The planar thrust chamber and nozzle can be seen in Fig. 17. The silicon die is capped by a Pyrex 7740 cover through anodic bonding. An Au/Ti heater provides ignition. Propellant again is a composite of HTPB binder (14%), ammonium perchlorate oxidizer (74%) and added Al particles to increase performance [30]. Maximum performances obtained after several design iterations were 4.6 mN average thrust over a burn time of 169 ms, yielding a total impulse of 0.79 mNs [32].

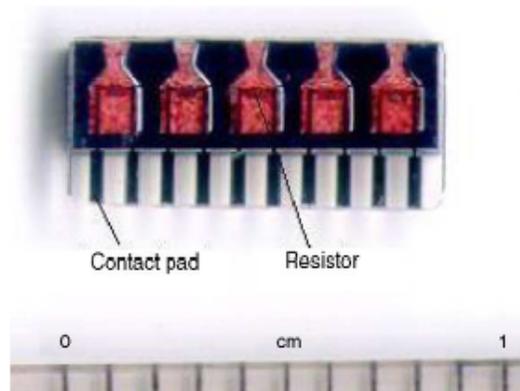


Fig. 17: MEMS Micromachined Silicon Solid Rocket Motors by Zhang et al. [31]

Digital Microthruster Arrays

Digital microthruster arrays extend the concept of micromachined solid rocket motors into arrays of potentially 100,000s of individual, single-pulse units. Such concepts have been pioneered by teams at TRW (now NGST), Aerospace Corp. and Caltech [6], [33], [34], LAAS at CNES in France [6], [35] – [37], Princeton, Honeywell, Atlantic Research, and the University of Minnesota [6], [38], [39], KAIST in South Korea [40], and at NASA Glenn [6], [41].

All concepts basically consist of three micro-machined wafers bonded into a stack, including a nozzle and burst disk, a cavity, and an igniter wafer. In the case of the LAAS group, the igniter is integrated with the burst disk. The fact that the thrusters are etched vertically into the wafer, as opposed to the planar design by Zhang et al [31] results in the typical square nozzle geometries through KOH etch with 35.3° half angles.

The NGST/Aerospace/Caltech design uses silicon nozzle/burst disk and igniter wafers and a FORTURAN glass wafer featuring cavities loaded with lead styphenate propellant, bonded using cyanoacrylate [34]. The LAAS/CNES design uses a silicon nozzle wafer, a second silicon wafer featuring the igniters and burst disks, and a third glass or silicon wafer holding the propellant cavities, loaded with glycide azide polymer mixed with ammonium perchlorate and doped with zirconium particles (termed GAP/AP/Zr) or zirconium perchlorate potassium (termed ZPP) [37]. An intermediate space wafer is bonded between the burst/burst disk wafer and the wafer holding the propellant cavities to increase poor ignition rates by increasing the ullage volume per cavity, decreasing pressure rise time such that complete combustion could be achieved before the burst disk ruptures, increasing likelihood of ignition, completeness of combustion and thus array performance [37]. The Honeywell/Princeton concept uses a two stage silicon cavity wafer design, the first one featuring cavities filled with lead styphenae, triggering a combustion in a second wafer adjacent to the first, featuring cavities filled with a nitrocellulose mixture [39]. This mixture burns very rapidly, allowing for complete combustion before the burst disk ruptures, increasing impulse bit performance (Fig. 18). In the NASA Glenn concept, solid gas generator pellets were placed into cavities. The propellant was LAX 112 ($C_2H_4N_6O_2$) [6], [41]. Upon ignition, the product decomposes into nitrogen, hydrogen and oxygen. This concept uses no burst disk, reducing debris issues around the spacecraft.

The NGST/Aerospace/Caltech design produced 0.1 mNs per cavity over a 1 ms pulse duration from arrays featuring 16 thruster cavities. Repeatability showed variations of up to 19% from cavity to cavity. The LAAS/CNES design produced up to 7.3 mNs using ZPP using a $250\ \mu\text{m}$ nozzle throat over a duration of approx. 500 ms. Ignition energy (resistive heater) for this propellant ranged between 17 to 24 mJ.

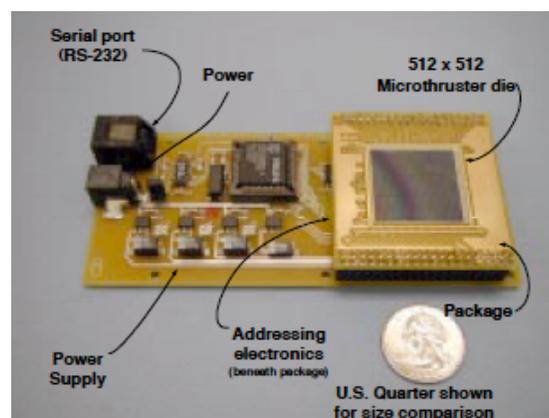


Fig. 18: Honeywell/Princeton Digital Micro-Thruster Array [39]

There is an obvious fascination in the microfabrication communities with digital thruster arrays given the large number of groups around the world that have worked these designs. However, upon closer inspection, this thruster design faces multiple implementation challenges.

As indicated above, care has to be taken to ensure proper timing of burst disk rupture. Too early a rupture leads to incomplete combustion, poor performance, failure to ignite, or poor repeatability from cavity to cavity. Once the disk does rupture, it generates debris around the spacecraft. While these generally will drift away from the spacecraft, they may pose a contamination risk in formation flying applications. It has also been observed that igniting one cavity can set off multiple cavities in the immediate neighborhood of the first through heat losses through the wafer structure, leading to uncontrolled firings of multiple cavities at once (Fig. 19). The designers have sought to address this issue through a variety of means. Using glass cavity wafers instead of the highly thermally conductive silicon reduces heat transfer. Youngner et al [39] placed silicon dioxide “firewalls” at intervals across the wafer, surrounding subsets of cavities to limit progression of uncontrolled firings across large sections of the wafer. Rossi et al [37] placed grooves surrounding each cavity to decrease thermal conduction. Such steps, however, decrease the packing density of cavities per wafer, limiting the number of firings available and the total impulse.

Even if operating properly, note that each thruster fires from a different position with respect to the spacecraft center line, each creating a different torque, placing challenges onto the attitude control system, in addition to similar challenges incurred by impulse repeatability. Delta-v applications would require at least two cavities to be fired at opposite and at equal distances from the spacecraft centerline. Any repeatability uncertainties from cavity to cavity will induce torques, requiring additional attitude control. The total impulse per wafer, in particular when taking into account reduced packing densities to avoid uncontrolled firings, may also be low for delta-v applications.

Finally, the thruster wafers would have to be placed onto external wall surfaces of the spacecraft and thus occupy potentially valuable wall space needed for solar arrays, instruments, cameras, louvers, patch antennas and other equipment, especially at reduced cavity densities per wafer.

Digital micro-thrusters thus appear to pose several design and integration challenges. However, they may have an application niche for very small spacecraft, even smaller than CubeSats, where propulsion options are even more limited, and where only crude propulsive capability may be needed for simple tasks not requiring any precision, such as initially dispersing a sensor net without the need for much control, for example.



Fig. 19: Unintended multiple cavity firing in a digital micro-thruster array due to heat conduction across the wafer [37]

Bipropellant Thrusters

Bipropellant thrusters, as opposed to monopropellant thrusters, offer higher Isp performances approaching 300 sec for small, storable propellant engines, however, at the cost of added complexity and dry mass [6]. The propellant mass savings realized through the use of higher-Isp propellant combinations thus have to be balanced against the added dry mass of (at least) dual tanks for oxidizer and fuel, and of the mass of the associated feed system branches and components (valves, filters, etc.). In order to maintain required mixture ratios throughout the mission, bipropellant engines are also pressure regulated, requiring a pressurization system consisting of a gas supply tank and associated components (regulators, valves, filters, etc.). For this reason, a trade with monopropellant systems, which require only one (mono)propellant feed system branch and can be operated in blow-down modes, not requiring a pressurization system, outperform bipropellant engines in terms of wet propulsion mass below delta-v's of approximately 500 – 1000 m/s, depending on spacecraft mass. Cost of bipropellant systems are also typically higher than for monopropellant systems due to higher engine complexity [6], part count of the feed system, and integration. For microsattellites, given the higher dry mass fractions and more severe cost constraints, this trade-off between mono and bipropellant engines will likely be pushed towards the higher end of this delta-v range, although it will always be a mission specific trade. At the higher delta-v end of this range it should be noted that much simpler and cheaper solid motors, available for microsatellite applications (see above) also become competitors to bipropellant engines for simple, one time delta-v maneuvers, such as orbit insertion. Bipropellants, however, do have an advantage over solids when greater flexibility, multi-mode operation, and restartability is required.

Nonetheless, micro bipropellant rocket engine developments have been undertaken over the course of the last decade. Scharlemann et al [42], [43] of the Austrian Research Centres Seibersdorf have developed a conventionally, yet highly miniaturized engine and tested it on ethanol and gaseous oxygen [42]. Gaseous oxygen, while acceptable for initial experimentation, obviously presents system integration and reliability challenges in practical applications as this would lead to the same system issues (high-pressure storage, heavy, large tankage, leakage concerns) as for cold gas systems, although much higher specific impulse would obviously be achieved. Hydrogen peroxide was therefore envisioned as oxidizer. The advantage of hydrogen peroxide is that it could also be used in a separate catalytic converter to power the engine turbines. Operation with hydrogen peroxide as oxidizer, however, was not achieved due to failure to ignite the engine, attributed to too short a combustion chamber length to facilitate sufficient mixing. The commercially procured turbopump used also was unable to provide sufficient chamber pressures [42]. It was concluded that a redesign of the engine is required [42].

Impressive gains in bipropellant thruster miniaturization have been achieved by London and Epstein et al. at MIT, who have developed a MEMS-microfabricated bipropellant rocket engine assembly machined entirely from silicon or silicon carbide [44 – 47]. The thruster chip consists of an integrated thrust chamber, nozzle, turbine pumps, and inlet valves. The entire assembly is about 20 mm square by 5 mm in height (Fig. 20). MEMS construction of this engine is complex. The thrust chamber features integrated cooling passage allowing for regenerative cooling of the thrust chamber [47]. The turbopumps, entirely machined out of silicon, run on fluid bearings. The inlet valves use electrostatically actuated pilot valves, which allow for fast actuation, yet offer only small stroke and actuation force [9]. These valves therefore actuate main valves with 10 times the flow rate capability of the pilot valve. Ten of these valves are required to provide sufficient flow rate to the engine.

At this point, the entire engine assembly has not yet operated in the integrated configuration, rather each component was operated and tested by itself. Although operation on LOX and hydrocarbon propellants, as well as storable propellant combinations are being envisioned, at this stage thruster testing was only performed with gaseous oxygen and ethanol. At a chamber pressure of 30 atm, thrust of 2.7 N (0.6 lbf) at a vacuum equivalent Isp value of 300 sec were obtained.

The high level of integration, once demonstrated at sufficient lifetimes and levels of reliability, alleviates some of the dry mass concerns of bipropellant systems outlined above. However, using gaseous oxygen will lead to similar concerns as for cold gas thrusters – the success of this engine thus rests on using liquid, and preferably storable propellants for in-space use on microspacecraft.

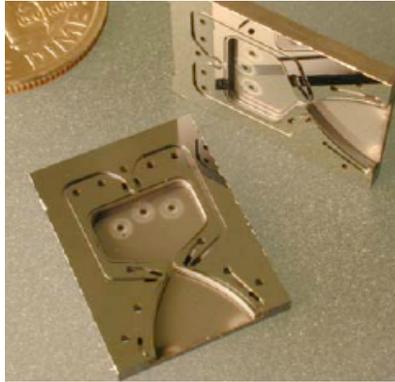


Fig. 20: MIT Micro-Bipropellant Engine Concept [47]

ELECTRIC PROPULSION OPTIONS

Electric propulsion options offer the potential of both ultrafine impulse bits for precision attitude and formation flying control, as well as large delta-v applicability due to high specific impulse available for many of these thruster concepts. They do require electric power, however, and as a result of this and the high specific impulse are generally low thrust devices. Some concepts, such as colloid systems, operate at higher thrust to power levels due to lower specific impulses and may fill performance gaps between chemical and higher-Isp electric propulsion systems. The need of power will place additional demands on the CubeSat bus, requiring additional power resources above what is available today, and likely deployable solar arrays in many cases. Power processing units will add additional mass, and their miniaturization along with the thruster and feed system is crucial to the successful integration of EP options on CubeSats.

Pulsed Plasma Thrusters (PPTs)

Pulsed plasma thrusters were the first application of electric propulsion in space onboard the Russian Zond 2 spacecraft on its way to Mars in 1964/65 [48]. The spacecraft was ultimately lost due to a faulty radio transmitter. PPTs have been used since, in the West on the Navy's NOVA navigation satellites [6], [48], and finally on the New Millennium Earth Observing (EO) – 1 spacecraft [6]. In a PPT, a capacitor, connected to two thruster electrodes, is charged and a discharge is triggered between these two electrodes. The ensuing arc ablates Teflon material off a solid fuel rod. This ablated material, due to Lorentz forces as a result of the current flowing through it and its interaction with its surrounding, self-generated magnetic field accelerates the material and expels it from the thruster to produce thrust [6], [48].

PPTs offer very small impulse bits, solid propellant storage in the form of Teflon fuel rods, modularity, and proven operation. As such, they inevitably became a technology of interest for microspacecraft applications [6], [49], [50]. Conventional technology, including the EO-1 thruster, are too massive for microspacecraft use, in particular for Cubesats [6]. Mass drivers, accounting for 40 – 45% of the thruster system mass, are the power electronics and energy storage capacitors [49]. Focus on miniaturization of PPT technology thus has to focus on mass reduction not only of the thruster head, but the supporting power electronics as well, and in particular on lighter capacitor designs. The thruster itself faces challenges during miniaturization, namely in the tailoring of discharge energy to the decreased fuel rod cross sectional area. If the discharge energy, as a result of overall system miniaturization, is too small, carbon neutrals in the plasma arc can return to the fuel rod surface and result in charring, which ultimately can lead to shorting of the thruster electrodes [51].

Recent advances in PPT miniaturization include the Aerojet “Dawgstar” thruster [49], [50], [52], [53], the AF Micro-Pulsed Plasma Thruster (μ PPT) by Spanjers et al. [51], a micro PPT developed by Austrian Research Centres Seibersdorf by Pottinger et al. [54], and a MEMS-based design by the Applied Physics Laboratory (APL) at Johns Hopkins University by Simon et al. [55] – [57]. Micro-PPTs have also been built and flown by Busek Inc. The “Dawgstar” thruster is the largest of these, and was developed for an AF funded university nanosat formation flying project, relying on spacecraft in the 15 kg class [52], [53]. As a result, its design is still recognizable as relatively conventional, although significant strides were made in system components miniaturization, light-weighting, and system simplification. Given its intended target of somewhat larger microspacecraft, it may also be just too big for a CubeSat applications. Its design, nonetheless reveals impressive gains in the miniaturization of this technology.

Shown in Fig. 21, the “Dawgstar” thruster system consists of four thruster modules with two thruster heads, both fed by a single capacitor per module. A single PPU charges the capacitors and triggers its discharges. Up to 4 capacitors can be charged simultaneously [52]. The “Dawgstar” thruster has delivered average $60 \pm 4 \mu\text{Ns}$ impulse bits at thrust levels ranging from $61 - 119 \mu\text{N}$, operating at 2 and 1 Hz. The thruster was operated at a frequency as high as 3 Hz. It has accumulated 1.1 million pulses over 153 hrs of continuous run time at 2 Hz (Table 9). Additional tests brought the total number of pulses to 1.7 million. The total system mass is 4.2 kg for 4 thruster modules and the PPU [52]. The single capacitor per module is a Mica capacitor rather than previously used oil filled capacitors, affording higher energy densities per mass [52]. Peak PPU input power ranges from 15.6 – 36 W depending on charging rates and thruster frequencies. The thruster efficiency is 1.8%, which is low, but typical for PPTs.

The AFRL μ PPT is a novel concept, differing substantially from previous PPT designs, including the “Dawgstar” thruster. Rather than using spring-loaded fuel rods, propellant housing structure, separate electrodes and igniters, it uses a coaxial fuel rod that combines all of the above into a compact package [51]. Two types are being distinguished – the 2-electrode and the 3-electrode configuration. In the former, a conductive center rod is surrounded by an annulus of Teflon, contained in turn by an outer conductive rod, forming a coaxial thruster design. The center and outer conductive rods serve as electrodes, charged by a capacitor until surface breakdown self-triggers a discharge, followed by acceleration of the ablative material similar to the processes in conventional designs [51]. However, the Teflon surface recesses into the fuel rod, rather than being pushed forward by spring-loaded action as in conventional PPTs [6], [48], in turn consuming the inner conductive rod as well. An issue with this design is that self-triggered discharges feature large shot-to-shot variations. If discharge energies are too low, neutrals from the plasma can redeposit onto the Teflon surface, causing charring and eventually shortening of the two electrodes [51].

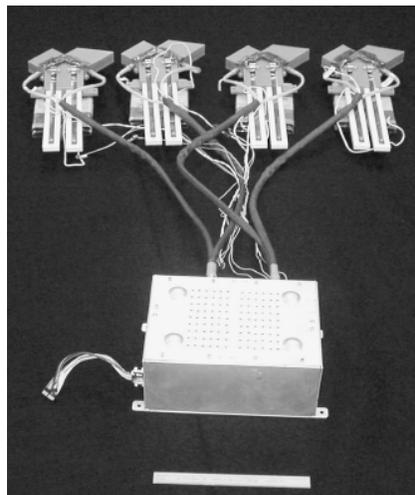
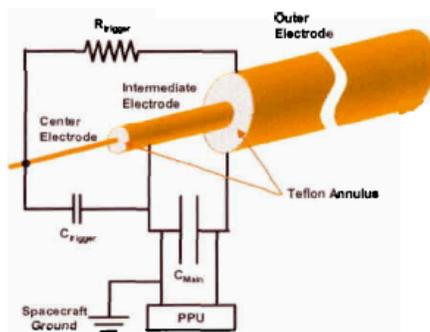


Fig. 21: The “Dawgstar” PPT Thruster System [52]

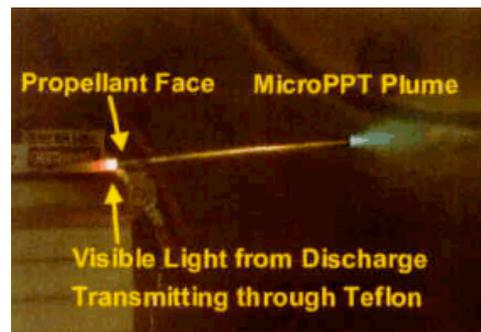
Table 9: Micro-PPT Performances

Parameter	Dawgstar	AFRL μ PPT	ARCS μ PPT	MILIPULT
Ibit (μ Ns)	60	2	2-25	0.4 – 0.6
Thrust (μ N)	61-264	2-30		
Isp (sec)	266			
Mass (g)	4200 (system)	500 (system)	30 (5 g propellant)	13.5
Total Impulse (Ns)	1000			
Power (W)	15.6 - 36	1-20	0.5 – 4 (est.)	
Discharge Energy per Shot (J)	4.9	2.25		0.018 – 0.22
Capacitance (μ F)	1.3 (per module)	0.42	2-6	0.937
Voltage (kV)	2.8	2.45 – 5.36		0.2 – 0.7
Frequency (Hz)	1-2			
Efficiency (%)	1.8			
Life (pulses)	1.7			
Propellant	Teflon	Teflon	Teflon	Water
Reference	[52]	[6], [51]	[54]	[55]

A three-electrode coaxial μ PPT was thus devised. In this design, the outer conductive rod is surrounded by a second annulus of Teflon, in turn surrounded by yet another conductive rod containing the Teflon fuel (Fig.22). This design emulates the triggered conventional PPT designs featuring external igniters. In fact, the discharge between the inner and intermediate conductive rods are self-triggered as in the two electrode design. The voltage between the intermediate and outer conductive rod is kept below surface breakdown voltages, and are instead triggered by seed ionization from the inner discharge, which now in effect acts as an igniter for the outer, or “main” discharge, allowing it to operate in a more predictable “triggered” mode. As with the 2-electrode system, the inner electrodes are consumed along with the Teflon during firing, and the Teflon surface recedes into the coaxial housing. The AFRL μ PPT can be seen operating in Fig. 22. The system weighs approx. 0.5 kg, delivered 2 μ Ns per shot at less than 1-2 J per pulse, and average thrust of 2 – 30 μ N and 1 – 20 W of power [6], [48] (Table 9).



(a)



(b)

Fig. 22: (a) AFRL μ PPT Concept [51], (b) AFRL μ PPT Firing in the Laboratory [51] – Plume to the right, discharge afterglow through the Teflon on the left.

The Austrian micro PPT has demonstrated impulse bits as high as 90 μNs , with most data obtained in the 10-20 μNs range [54]. Careful examinations of fuel rod aspect ratios were conducted to optimize performances. Predicted performance ranges for this thruster are 0.5 – 4 W power, with a total impulse capability of 49 μNs if a specific impulse of 1000 sec is assumed. Estimated mass for a flight like thruster is 30 g with 5 g propellant mass.

PPT miniaturization has been taken to an even greater extreme by Simon et al [55] – [57] using a planar, 2-D microfabricated architecture. This design emulates the AF μPPT in that it uses a 2-D equivalent of the 3-electrode system, with the difference that electrodes are no longer conductive rods as in the AF design, but conductive strips of metal deposited onto a thruster chip. Three thin strips of copper are deposited onto G10 fiberglass or Alumina substrates, called the common, trigger and sustain electrodes. Triggering a discharge between the trigger and common electrode via a surface discharge triggers a discharge between the sustaining and common electrode, which is held at voltage differences just below surface breakdown. As in the case of the 3-D AF coaxial thruster, this triggered electrode design allows for more predictable impulse bit performances. Teflon tape deposited onto the substrate between the electrodes served as propellant.

However, frequent thruster failure was observed with this ablative Teflon thruster design, likely due to leakage passages along the substrate material as a result of carbon deposition as a result of Teflon ablation [56]. The longest lasting thruster provided 10,000 pulses, most failed after shorter durations [56]. For this reason, the design was adapted for liquid propellant use. Termed the Micro Liquid Pulsed Plasma Thruster (MILIPULT), it featured a small liquid reservoir that allows flow to creep between the electrodes (Fig. 23). These experimental devices, however, featured no means to control propellant flow, especially between discharges [57]. For operational use this would be unacceptable, as it would mean that propellant would be lost through continuous leakage, reducing the effective Isp of the system and potentially posing contamination concerns to the spacecraft. Microvalves would be required to control the flow, posing significant challenges on their own. These valves would have to be chip based to match the envelope of this thruster concept, be fast acting to allow for shot frequencies of one to several Hz typical for PPTs, and have lifetimes of potentially millions of cycles to match the PPT lifetimes. As was shown in a previous survey by the author [9], obtaining valve designs offering these features in a MEMS package poses significant challenges. Electrostatic or piezovalves are the only options allowing for such fast actuations and may be offered in micromachined packages, but many details, such as leak tight seat designs, lifetimes etc remain to be worked out [9].

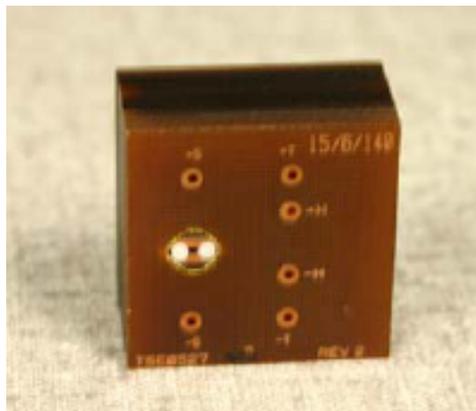


Fig. 23: APL MILIPULT Liquid Micro-PPT [55]

The MILIPULT thruster was tested without this feature for performance demonstration purposes. It produced 0.4 – 0.6 μNs with a thruster capacitance of 0.937 μF charged to 200 – 700 V, using water as a propellant [55] (Table 9).

Vacuum Arc Thruster (VAT)

The Vacuum Arc Thruster (VAT) was developed by Alameda Applied Sciences Corp. (AASC), and is another ablative pulse plasma type, yet using metal electrodes [58] – [61] (Fig. 24). Applying a high voltage between two electrodes leads to electric field intensification near surface roughness spikes of a metal electrode. The high power density in this spike leads to vaporization and ultimately electric breakdown and plasma generation in an arc discharge. Different cathode materials can be used, including Carbon (C), Aluminum (Al), Tungsten (W), Bismuth (Bi), or Chromium (Cr), among others. The plasma plume is then allowed to expand into vacuum and produces a thrust pulse. Impulse bits vary with electrode material chosen and power deposited into the arc. Demonstrated performances ranged from 1 μNs for Cr [59], 30 μNs for W [58] to 40 μNs for Bi [58]. Larger impulse bits can be built up by rapid firing of discrete impulse bits, ablating multiple roughness spikes. Up to 200 Hz firing sequences have been demonstrated. Test on a JPL micro-Newton thrust stand demonstrated up to 100 μNs for 100 Hz pulse trains using Cr [59].

A CubeSat VAT module was developed, making the VAT together with the PPT and the chemical VACCO MiPS system, one of the few near term thruster systems available for CubeSat propulsion today. In a past design study, which delivered thruster hardware intended for use on the ION-F CubeSat, a 150 g PPU was built, able to supply 4 VAT thruster heads with power using a 12-24 V bus voltage [59]. Thruster heads are estimated to weigh 100 grams and average power, depending on thrust values needed, may range between 1 – 100 W [59]. Thrust to power values of up to 10 $\mu\text{N/W}$ are expected to be obtained from such a thruster.

The advantages of the VAT are its very small impulse bits for microspacecraft attitude control, its solid propellant storage, with the propellant being integrated into the thruster head, allowing for inherently compact and modular designs. Given the small impulse bits per ablative pulse, however, even high cycling frequencies limit thrust and total impulse values, and large delta-v applications may not be possible, despite high specific impulse that can be achieved with this thruster, ranging from 1000 – 3000 sec [58],[59] depending on propellant. Rapid maneuvering, as in the case of an inspector sat, may also not be feasible with the VAT due to its low thrust values. Chemical systems, such as the VACCO MiPS may be more suitable for that application. The VAT may, however, be well suited for precision pointing and attitude control.

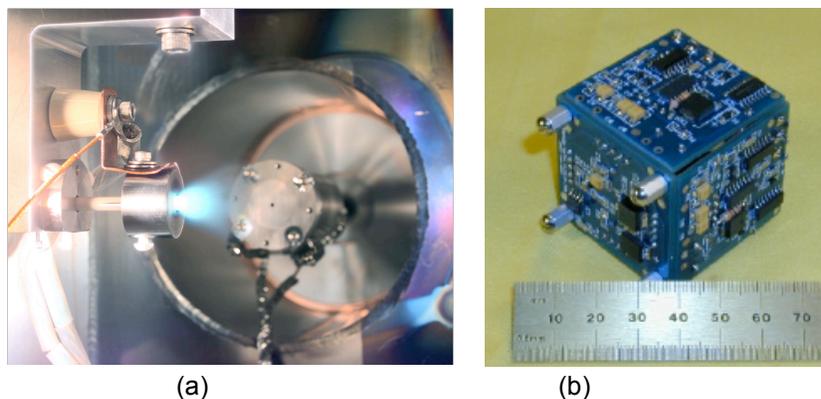


Fig. 24 (a): VAT Thruster firing on the JPL Micro-Newton Thrust Stand, (b) VAT PPU configured for CubeSat [59]

Miniature Ion Engines

In an ion engine, a plasma is generated in a discharge chamber, and ions are extracted from it via an electrostatic grid system. Typically xenon is being used as propellant gas, although historically mercury vapor has been employed, as well as other inert gases, such as argon and krypton. Two main types of ion engines are being distinguished, depending on how the plasma is generated [6], [62]. In a DC electron bombardment, or Kaufmann-type thruster, a cathode (typically a hollow cathode) is used to generate a plasma from which electrons are extracted and injected into the discharge chamber to ionize the propellant gas by accelerating the electrons towards an anode surface. Permanent magnets mounted near the anode reduce electron loss and increase the electron path through the discharge, thus increasing engine efficiency. The hollow cathode is fed the same propellant as the main discharge. In an radio-frequency (RF) ion engine, no cathode is required for plasma generation, rather, an electromagnetic coil is wrapped around a dielectric discharge chamber to generate an oscillating, predominantly axial magnetic field that according to Maxwell's laws generates an oscillating azimuthally electric field following the curvature of the cylindrical discharge vessel. Electrons present to small degrees within any gas are accelerated, causing ionization, and additional electrons are produced in these reactions to eventually cause plasma generation. Electron energy cost per thrust-producing ion generated by an RF thruster are typically higher than for the DC discharge, but no life limiting engine cathode is required other than the neutralizer. This neutralizer, in both cases (RF and DC), consists of another hollow-cathode mounted externally to the engine, providing electrons for beam neutralization of the positively charged ion beam, thus avoiding spacecraft charging.

Several miniature ion engines have recently been developed, although not specifically for CubeSats, but rather for formation flying applications of future space telescopes [63], [64]. The use of inert, non-contaminating xenon propellant near sensitive optical surfaces, as well as the ability to smoothly modulate the thrust amplitude rather than using chemically or PPT-based pulsed thruster approaches, or reaction wheels, which each may induce jitter, have made miniature ion engines a critical component of future development for these types of missions. Both types (DC and RF) have been miniaturized for this application.

Proposals to use micro-ion engine for microspacecraft were made initially by Janson at Aerospace Corp. [6], [65], and by Brophy et al. at JPL in the form of an ion thruster on a chip (ITOC) concept [66], although miniature RF engines existed decades before that (see below). Initial studies at JPL of this ITOC concept included critical subcomponent developments to study the feasibility of the overall approach, including microfabricated accelerator grid testing [67] – [69] and field emission based cathode development for discharge generation and neutralization [70], [71]. The ITOC thruster concept eventually morphed into a 3-cm diameter meso-scale ion engine, appearing more feasible and useful to future applications of this technology [72] – [78]. Termed the Micro Xenon Ion Thruster (MiXI) its primary application goal became the Terrestrial Planet Finder (TPF), and similar exoplanet missions [63], [64], [76]– [78]. A laboratory prototype of MiXI is shown in Fig. 25 (a) and (b). Table 10 summarizes recently demonstrated performances of this engine, compared to other devices reviewed in the following. MiXI has demonstrated up to 1.5 mN thrust at 50 W power and a specific impulse of 3200 sec on xenon propellant. Thruster mass is 200 g.

RF-ion engines are attractive candidates for miniaturization as no internal cathode is required. A 4-cm diameter RF ion thruster was originally developed at Giessen University in 1972, far ahead of its time [79], but this concept was abandoned for several decades. As in the US, recent interest in formation flying missions in Europe led to a reactivation of the 4 cm engine concept, as well as smaller RF ion engines since the middle of the last decade [79], [80]. Figure 26 shows a 2.5 cm dia RF ion engine named the μ NRIT-2.5. Its performance is summarized in Table 10. The 2.5-cm device weighs 210 gram, comparable to MiXI, and delivers 0.5 mN of thrust using 30 W of RF power with xenon.

RF-micro-ion engines have also been under development in the US in recent years. A Miniature Radio Frequency Ion Thruster (MRIT) of 1 cm diameter has been under development at The Pennsylvania State University (Figure 27) [81]. Initial laboratory testing was conducted on argon,



(a)



(b)

Fig. 25 (a) JPL Miniature Xenon Ion Thruster (MiXI), (b) MiXI in Operation

Table 10: Micro-Ion Engine Performances

Parameter	MiXI	μ NRIT-2.5	MRIT
Thrust (mN)	0.01 – 1.5	0.05 -0.6	0.001 – 0.06
Isp (sec)	2500 - 3200	2861	5480
Power (W)	13-50	13-34	
Electrical Efficiency (%)	> 40	4 - 47	15
Mass Utilization (%)	> 70	15 - 52	
Dia (cm)	3	2.5	2
Mass (g)	200	210	
Propellant	Xenon	Xenon	Argon



Fig. 26: The μ NRIT-2.5 Miniature Ion Thruster of the University of Giessen [79]

however, future operation is envisioned using xenon. Performances demonstrated with MRIT are listed in Table 10 along with MiXI and the 2.5-cm and 4-cm μ NRITs. MRIT has demonstrated 59 μ N of thrust and 5480 sec Isp at an RF power level of 15 W.

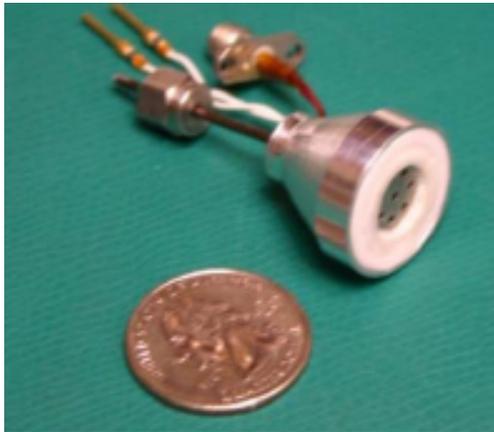


Fig. 27: The MRIT Micro-Ion Thruster of The Pennsylvania State University [81]

Miniature ion engine technology, although developed for formation flying missions, could potentially be adapted to CubeSats or similar microspacecraft and deliver large delta-v capability, providing significant orbit raising, multiple orbital plane changes, and extended drag-make up capabilities in low Earth orbits, thus allowing for significant and extended maneuverability and associated mission flexibility, even allowing CubeSats to eventually venture beyond Earth orbit. In the case of the latter it must be noted, however, that stand-alone microspacecraft use beyond Earth orbit, due to communication link issues from very small spacecraft platforms, may be limited. A more likely scenario for microspacecraft use beyond Earth orbit may be their deployment from larger spacecraft for certain aspects of an interplanetary mission, e.g. for particularly risky parts of a mission, such as the in-situ exploration of Saturn's rings, for example, or distributing widely distributed sensor nets requiring out of ecliptic maneuvers [6]. Even in these cases, delta-v requirements may still be significant, and may be benefitted by the development of high-Isp thruster options such as micro ion engines.

To realize micro-ion engine propulsion onboard CubeSats, in addition to reducing the size of the engines other required subsystems have to be equally miniaturized, including feed systems and power processing unit (PPU). VACCO's ChEMS module approach has already been adapted to xenon flow controllers, albeit for conventionally sized engines, but could be adapted for micro-ion engines for CubeSats given its already demonstrated feasibility on CubeSats under the MEPSI program (above). If miniaturized light weight PPUs could be developed as well, a "3U" CubeSat architecture may be envisioned, consisting of one "1U" module featuring the micro-ion engine and a micro-feed system, the second "1U" unit hosting the PPU, and the third "1U" unit remaining available for the payload and other spacecraft subsystems. The PPU unit would also require deployable arrays to provide power levels in the range of several 10s of Watt as required for micro-ion engine systems.

Miniature Hall Thrusters

In a Hall thruster electrons emitted by a cathode external to the thruster are accelerated towards an anode at the end of an annular discharge channel. On their way, they cross a radial magnetic field near the entrance of the channel. Due to Lorentz force action, they gyrate around the magnetic field lines and drift azimuthally through the annular channel, in the course of which they cause ionization. The ions are accelerated out of the thruster by the same field that accelerated the electrons. No grid systems as in ion engines are required. Due to the higher inertia of the ions, they are significantly less effected by the magnetic field, and can exit the channel to produce thrust. Electrons from the external cathode also neutralize the beam. The high electron density in the annular channel reduces space charge effects as in an ion engine, thus a more compact engine design can be realized for a given ion beam strength [6].

Hall thrusters typically deliver a specific impulse in the 1500 – 2000 sec range [6], [62]. This makes them generally more suitable for near-Earth missions. Optimization of electric propulsion devices requires balancing the propellant mass savings realized by the higher specific impulse of the device with the power requirements and associated masses (for power generation and conditioning) to generate this higher specific impulse. For the more moderate delta-v requirements in Earth orbit additional propellant mass savings by raising the specific impulse significantly above a level of approx. 2000 sec are no longer offset by the higher power system masses. Hence, a Hall thruster system operates more efficiently in Earth orbit than an ion engine. This optimization strictly only holds for state-of-the-art conventional electric propulsion technologies, as deliverable specific impulses, efficiencies, and specific power values used in this optimization are tied to existing technologies. Micro technologies may have different performance characteristics, which may shift this optimum. However, at this point, absent any data to the contrary, it seems reasonable to consider micro Hall thrusters as a potentially interesting electric propulsion option for near Earth operations of microspacecraft as well.

Hall thrusters, however, face particular challenges in their miniaturization given their plasma generation scheme described above [6]. Reducing the size of the thruster requires to reduce the Larmor radius (i.e. the radius of curvature of the electron trajectories in the magnetic field) to avoid excessive electron wall and efficiency losses. This in turn requires increased magnetic field strengths that need to be realized as the overall thruster size is reduced. Furthermore, increased heating of the magnets by the plasma is to be expected in smaller devices, which could lead to demagnetization, and loss of magnetic field strength, and hence electron containment.

Consequently, initial attempts at Hall thruster miniaturization were fraught with significant challenges. A miniature Hall thruster concept by MIT [82], 4 mm in dia., delivered 1.8 mN thrust and 826 sec specific impulse at 126 W of power as shown in Table 11 [6], [82]. Efficiencies were poor at only 6% due to the aforementioned demagnetization issues (Figure 28).



Fig. 28: MIT 50-W Hall Thruster [82]

Table 11: Small and Micro-Hall Thruster Performances

	BHT-200	SPT-30	MIT	PPPL CHT 2.6	PPPL CHT 3.0
Thrust (mN)	4-17	5.6 - 13	1.8	2.5-12	3 - 6
Isp (sec)	1200 - 1600	576 - 1370	865		1100 - 1650
Power (W)	100 - 300	99 - 258	126	50-300	90 - 185
Efficiency (%)	20 - 45	16 - 34	6	15 - 32	20 - 27
Mass (g)	<1	~1			
Dia (cm)	2.1	3	0.4	2.6	3
Propellant	Xenon	Xenon	Xenon	Xenon	Xenon
Reference	[87]	[6]	[82]	[83]	[84]

A novel type of miniature Hall thruster developed at the Princeton Plasma Physics Laboratory (PPPL) overcame several of the issues in Hall thruster miniaturization, specifically the low efficiencies observed in previous versions. The PPPL type uses a cylindrical Hall thruster (CHT) architecture, which distinguishes itself from the conventional annular Hall geometry by an open cylindrical, rather than an annual flow channel and a largely axial, rather than radial magnetic field [83] – [87]. Electrons are once again emitted by an external cathode, and accelerated towards an anode located at the end of the cylindrical channel. However, the axial magnetic field is concentrated at the anode end of the channel and therefore features a strong axial gradient at that location, causing electrons to be mirrored back, away from the anode, as they approach it. In a CHT, a region of high electron density is thus being generated due to the balance of attracting electrostatic forces due to the anode, and the mirror effect of the concentrated magnetic field (Figure 29) [84].

Cylindrical Hall thrusters of 9-, 3-, and 2.6-cm diameter have been built and tested. The 9-cm devices is not a miniature Hall thruster type, but sized for a power levels on the order of 1 kW. Performances of the 3- and 2.6 cm devices are listed in Table 11. Significant is the increase in efficiency ranging into the 30%. The 2.6-cm thruster delivers up to 12 mN at 300 W, whereas the 3 cm thruster, operated at lower power levels, delivers 3 – 6 mN at power levels from 90 – 185 W and specific impulses from 1,100 - 1,650 sec on xenon. Also shown in Table 11 are BHT-200 Hall thruster performance data by Busek for comparison. The BHT-200 is currently the state-of-the-art of existing, commercially available Hall thruster technology, and was the first US Hall thruster operated in space onboard Tac-Sat 2 in Dec. 2006.

Integration of miniature Hall thrusters systems into CubeSats faces similar challenges as for ion engines, i.e. miniaturized feed systems, power conditioning units and deployable solar arrays to provide the necessary power. Power processing unit miniaturization may be aided by the fact that typically only 300 V anode voltage are required, vs in excess of 1 kV for ion engines, and fewer power supplies are needed for Hall thrusters than for ion engines. However, due to the magnet mass, Hall thrusters will be heavier than miniature ion engines.

Electrospray Thrusters – Colloid

As becomes evident when surveying ion and Hall thruster miniaturization, significant challenges arise from the miniaturization of the plasma chamber, and efficiency losses associated with it. Electrostatic thruster devices do exist that do not require a plasma discharge, and emit ions or charged directly droplets directly from a liquid propellant column if a strong electrostatic field is applied [6]. In an

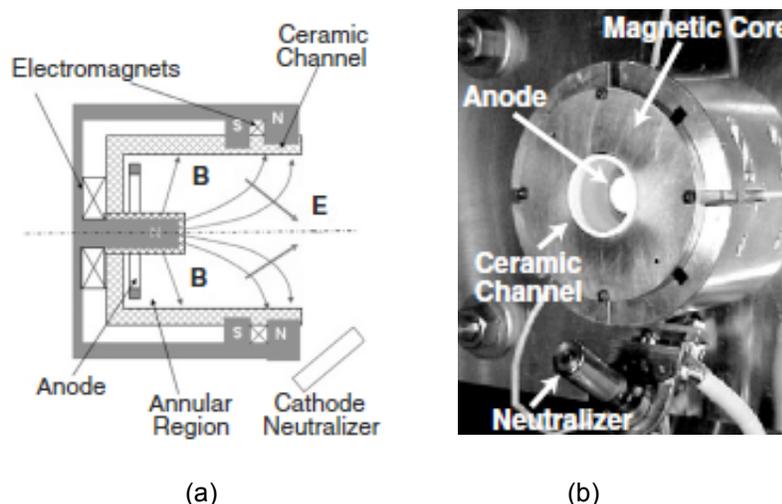


Fig. 29: Cylindrical Hall Thruster (CHT) by PPPL, (a) Concept, (b) 3-cm CHT device [84]

electrospray thruster, propellant is fed up an emitter, which can take the form of a sharp needle, capillary, or narrow slit [6]. An electric field is established between the emitter and an opposing electrode. The field is intensified near the emitter tip and the electrically conductive propellant is distorted into a sharp so called Taylor cone as a result of a balance of surface tension and electrostatic forces. The shape of this Taylor cones intensifies electric field strengths even further. Depending on the type of propellant, either ions are emitted from the propellant directly by field emission, as in the preferred case for liquid metal propellants, or charged liquid droplets are extracted in the case of doped glycerol or ionic liquid propellants. Liquid metal ion sources (LMIS) are typically also referred to as Field Emission Electric Propulsion (FEEP) and are reviewed in the next section. Thrusters relying primarily on droplet emissions are commonly referred to as colloid thrusters and are the subject of this section. Note that the difference between these two thruster types is less well defined in the case of newer, so called ionic liquid propellants, which are essentially molten salts at room temperature, and can emit both ions or droplets depending on operating conditions, such as flow rate, temperature and applied voltage [88]. Not requiring the generation of a plasma discharge, these thrusters are extremely compact and efficient, and given the sub-millimeter sizes of emitter tips, lend themselves extremely well to miniaturization [6].

Colloid thrusters were extensively studied in the US in the 1960s by Perel and Mahoney [6], [89] – [91], and at TRW (now NGST) [92], [93]. Research focused on glycerol propellants, doped with sodium iodine or sulfuric acid – the latter producing negative charged droplets allowing for a bipolar thruster operation without the need for a neutralizer. Demonstrated thruster performances ranged from 1 – 334 μN per emitter at specific impulses from 450 – 1,450 sec and specific power levels ranging from 4 – 10 W/mN [6]. The highest specific impulses were achieved at very large emitter voltages approaching 20 kV [6]. Emitter sizes ranged from 100 μ to over 2 mm I.D. for the higher thrust devices. Data obtained with these historic thruster devices are summarized in detail in Ref. [6].

Interest in colloid thrusters waned over the ensuing decades as spacecraft grew larger, and no application for colloid thrusters materialized. This changed dramatically in the late 1990s and early 2000s, as two spacecraft applications emerged in need of either precision thrust or micropropulsion devices, namely formation flying missions and microspacecraft. Initial developments were focused on microspacecraft applications. Perel et al. [94] re-engaged in colloid thruster development with a 1 μN I.D. capillary emitter and Pranajaya et al. [95] developed a 50 μN I.D. capillary device with up to 100 emitters for a university nanosat mission. The latter produced up to 1 μN per emitter at an estimated specific impulse of 500 sec and a specific power value of 10 W/mN [95]. The overall thruster module, including feed system and PPU under development had a mass target of 0.5 kg at a size of 10 x 10 x 20 cm^3 [95]. As such it would fit a 3U CubeSat bus and leave a 1U module for payload and other spacecraft subsystems.

Colloid thruster developments made significant further strides shortly after these renewed forays through NASA's New Millennium ST-7 and Laser Interferometry Space Antenna (LISA) flight programs. NASA selected a colloid thruster system developed by Busek, in collaboration with JPL for these missions [96] – [98]. LISA requires three spacecraft to fly in a triangular formation, spaced approximately 100,000 km apart, allowing gravity waves, should they exist, to be detected by moving the spacecraft with respect to each other, and detecting these movements by laser interferometry. In order to distinguish between potential gravity wave effects and other disturbances forces, such as solar wind, for example, the spacecraft are required to precisely follow a spacecraft internal proof-mass to within 10 nm. Micro-Newton levels of thrust to a precision of 0.1 μN are required to accomplish this task.

The US is contributing a colloid propulsion system to ST-7/LISA. The thruster system consists of multiple clusters featuring four thruster heads each clocked at 90 degrees with respect to each other to provide precision control (Fig.30). The cluster module also houses the propellant supply and feed system, consisting of compressible bellows housing the propellant to pressure-feed each thruster head, and the PPU. Each thruster head has demonstrated a thrust range of 5 – 38.5 μN of thrust to a precision of less than 0.01 μN [96]. Specific impulses range from 190 – 240 sec depending on propellant temperature. Temperature control is achieved with individual thruster heaters, driving the power per cluster to

approximately 24 W during warm up, and 16 W during normal operation. The ST-7 colloid thruster performance characteristics are listed in Table 12.

Obviously, the cluster is too large for CubeSat applications, however, each thruster head could form a stand-alone thruster for a CubeSat for delta-v applications. It currently features 9 individually machined emitters (Fig. 31), the number of which could be tailored to the desired application. A dedicated feed system and PPU would be required, and miniaturization challenges of a colloid system are likely to focus more on these two subsystems than the thruster head itself, an experience already made by Pranajaya et al. [95]. Using ionic liquids, operating colloid thrusters in both droplet as well as ionic modes would allow

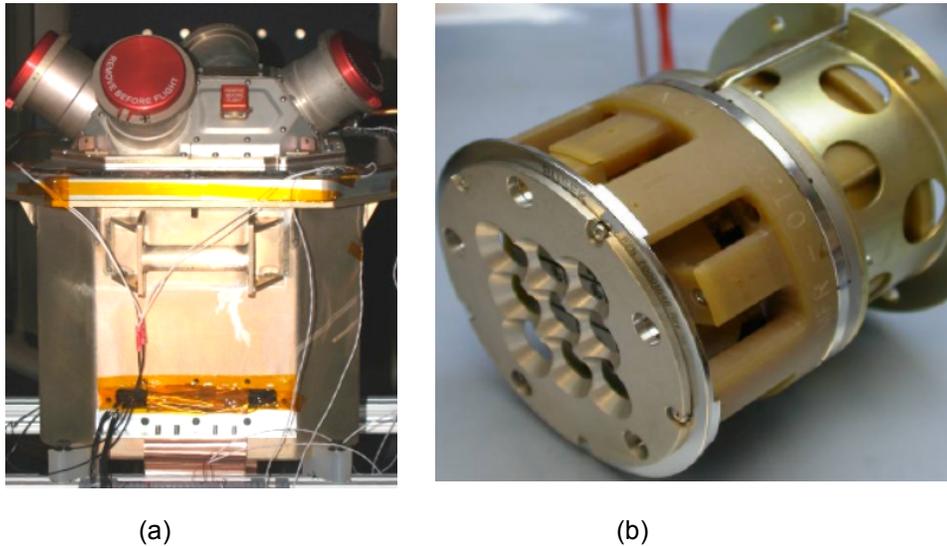


Fig. 30: (a) ST-7 Colloid Thruster Cluster [97]; (b) ST-7 Cluster head (4 per cluster) [98]

Table 12: ST-7 Colloid Thruster Performance Characteristics [96]

Parameter	Demonstrated Performance
Thrust (μN)	5 – 35.8
Thrust Noise (μN)	<0.01
Isp (sec)	240
Mass (kg)	15
Power (W) - Max	24.6 (all thruster heaters at max at 2 W each)
Power (W) - nominal	16
Beam Divergence (Half Angle) (degree)	<23



Fig. 31: Individual ST-7 Colloid Thruster Emitter [98]

such a system to cover a wide range of specific impulse from a couple hundred to a few thousand second specific impulse, allowing for substantial maneuverability of a CubeSat equipped with it, from high thrust-to-power, low-Isp modes for rapid maneuvering, to low thrust-to-power high-Isp modes for large delta-v maneuvers.

Electrospray Thrusters – FEEP

FEEP thrusters function similarly to colloid thrusters, as described above, however, emit predominantly ions. External wetted needle or internally wetted slit emitters are commonly used [6]. The former use indium, the latter cesium as a propellant. Unlike colloid thrusters, FEEP thrusters are not pressure fed, and propellant is supplied to the emitter tip by capillary forces, making for very compact feed systems featuring no valves or pressurized propellant tanks.

Field emission electric propulsion was initially studied in the US by Perel et al. in the 1960s [90], [91], focusing on capillary tubes to feed cesium propellant. As with colloid thrusters, research into FEEP thrusters in the US waned as spacecraft grew larger, and no unique applications requiring FEEPs materialized. Unlike colloid thrusters, however, FEEP development was continued in Europe, initially at ESA-ESTEC in The Netherlands, then at the University of Pisa and Centrosazio, later Alta in Italy under funding from ESA [6]. This work focused on cesium FEEPs and ultimately led to the development of slit emitter designs of varying slit widths, ranging from 2 mm to 70 mm producing thrust from 40 μ N to 1.4 mN. Two relatively recent FEEP slit emitter designs are listed in Table 13. Specific power values are 66 W/mN – considerably higher than for colloid thrusters due to high-Isp operation around 9,000 sec in the cases of these thrusters. The FEEP-5 thruster is shown in Fig. 32.



Fig. 32: FEEP-5 Thruster by Centrosazio/Alta

Table 13: FEEP Performance Characteristics

Supplier	Designation	Thrust (mN)	Mass (kg)	PPU Mass (kg)	Isp (sec)	Power (W)
ARCS	In-FEEP 100	0.001 – 0.1	0.3	Incl.	8000 - 12000	0.5 - 10
ARCS	GOCE MTA	0.002 – 0.65	3.5	Incl.	8000 - 12000	6-52
ARCS	In-FEEP 1000	0.001 - 1	1.5	Incl.	8000 - 12000	2-80
Centrosazio	FEEP-5	0.04	0.6	1	9000	2.7
Centrosazio	FEEP-50	1.4	1.2	1.2	9000	93

Cesium FEEP thrusters suffer from complications in the propellant handling, as cesium is very corrosive and readily reacts with even minute amounts water, oxygen, or CO₂, forming cesium hydroxides, oxides, and carbonates with high melting temperatures, which may clog emitter slits [6]. A lid mounted in front of emitter slits is needed to isolate the thruster from exposure to the environment, and may only be opened in space. Subsequent to the opening of the lid the thruster must immediately be fired again to avoid contamination of cesium located at the emitter exit slit due to spacecraft outgassing.

A significantly more benign propellant is indium. Solid at room temperature up to its melting point of 154 C, it requires heater power to liquefy it, however, can be handled under atmospheric conditions, thrust greatly simplifying propellant handling [15]. Its very low vapor pressure and lack of corrosiveness also reduces spacecraft contamination concerns, which is of particular importance for microspacecraft if used in formation flying missions with the potential of cross contamination between neighboring spacecraft intersecting each others thruster plumes.

The Indium FEEP or LMIS (Liquid Metal Ion Source) was developed at Austrian Research Centres Seibersdorf over 20 years ago, although originally not used as a propulsion devices but as an ion source [15]. In this function, it has seen flight applications onboard the Russian MIR space station in 1991 as part of a mass spectrometer experiment, and subsequently on multiple other space missions for spacecraft potential controls and in mass spectrometers [15].

Since then development has focused on developing Indium LMIS for space propulsion applications. Several devices currently under development at ARCS are listed in Table 13. Single emitter thrust performances typically range up to 15 μ N – higher thrust levels can be achieved by assembling arrays of multiple emitters [15]. A 100- μ N device was originally developed for the ST-7 mission, and featured 9 emitters (Fig. 33) [15]. An array for the European GOCE mission required 16 emitters (Fig.34).

CubeSat integration of In FEEP would require accommodation of the PPU, and additional deployable solar arrays as for other electrostatic thruster devices reviewed above. Feed system integration, however, is extremely simplified, as the propellant reservoir and capillary feed are part of the thruster head. FEEP thrusters allow for extremely high specific impulse values, however, at reduced thrust-to-power ratios, requiring either long transfer times or high power levels. At present, droplet emission with Indium thrusters has not yet been demonstrated in a controlled fashion. Were it feasible, a dual mode operation as for colloids operating on ionic liquids (above) could greatly extend mission flexibility.



Fig. 33: ARCS In-FEEP-100



Fig. 34: ARCS GOCE FEEP Array [15]

Microfabricated Electro spray Arrays

Inspecting Fig. 34, it becomes obvious that integrating emitters by conventional means is not a promising approach to arrive at large arrays. Large emitter arrays, however, may be desirable to scale the obtainable thrust of electro spray devices for a variety of applications – extending beyond the current focus on precision pointing of formation flying spacecraft to delta-v applications as well, thus fulfilling multiple functions with a single type of thruster, simplifying propulsion subsystem architecture. The large conventional array sizes are in particular disappointing as the critical thruster features – i.e. the emitters – are already sub-millimeter, even micron-sized in many applications. While cesium slit emitter thrusters can easily be scaled up by lengthening the slit, and in fact CS FEEP thrusters capable of milli-Newton thrust levels have been built [6], use of cesium is not desirable for reasons stated above.

A more efficient way to integrate emitters into arrays is through the use of microfabrication techniques [6]. In fact, the first ever proposed MEMS thruster concept was conceived by Mitterauer in 1991 in the form of a microfabricated FEEP concept [6], [99]. No devices were ever built, however. The idea resurfaced again in 1998 through a proposal by Marcuccio et al [100]. Initial forays into the experimental study of microfabricated colloid and FEEP thruster concepts were made at JPL under small discretionary funds with initial focus on the wetting abilities of silicon substrates [75]. Work on microfabricated electro spray arrays gained momentum with significant microfabrication and testing efforts at MIT in collaboration with Yale [101] – [106], the University of London in the UK [107] – [113], after initial efforts at the University of Southampton there [107], and at Sandia National Laboratories in Albuquerque, NM, in collaboration with ARCS of Austria [114], [115]. Initial work was also conducted at NASA Goddard Space Flight Center (GSFC), but not continued [116]. Quite early this concept also gained attention in China, where a couple of unique microfabricated electro spray arrays were developed and tested in the early part of the previous decade [117], [118]. So far unpublished work was subsequently sponsored by DARPA at SEI Inc., and at Almeda Applied Sciences Corp (AASC) in a joint effort with JPL and the University of California at Irvine (UCI), and finally presently at JPL.

The MIT/Yale work made significant contributions both in the fabrication and testing of silicon micromachined electro spray arrays [101] – [106], as well as the development of novel electro spray propellants [88], namely ionic liquids, which demonstrated dual mode operation (ionic and droplet). Initial development efforts focused on linear arrays [102], [105], featuring a self-aligning cathode attaching to the emitter array via a unique silicon micromachined “clip” mechanism (Fig. 35). Limited packaging densities achievable with linear arrays led the team to explore 2-D array architectures [105], [106]. A unique contribution was the development of “black silicon”: employing a chlorine plasma etch, the surface of a silicon substrate is roughened, aiding wettability. It is called “black” because of its appearance as the result of the etch and ensuing surface roughening [105]. The MIT group also explored grooved emitter tips for the first time, called “truncated volcanos” (Fig. 36 (a)), as well as more traditional “pencil” shaped emitters (Fig. 36 (b)) [106]. The emitter grooves, however, were not primarily designed to aide wetting,

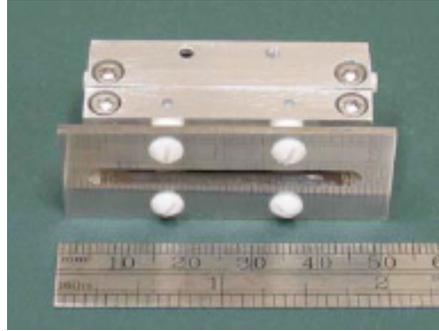


Fig. 35: Linear Microfabricated Colloid Thruster Array by MIT [102]

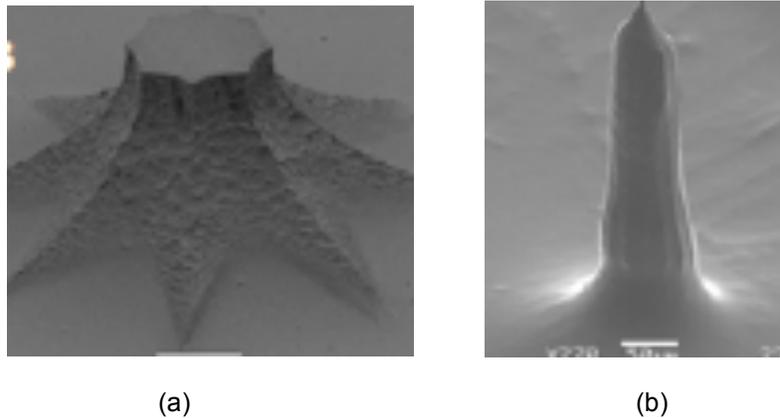


Fig. 36 (a) Micromachined “Truncated Volcano” Emitters and (b) Pencil Shaped Emitters by MIT [106]

but to produce sharp tips at the upper surface of the truncated cone to serve as emission sites [106]. Volcano emitters were $270\ \mu\text{m}$ tall and had an upper truncated surface diameter of $90\ \mu\text{m}$. The radius of curvature of the tips was $6\ \mu\text{m}$ [106]. Pencil shaped emitters were also $270\ \mu\text{m}$ tall, had a diameter of $50\ \mu\text{m}$ and a tip radius of curvature at the tip of $3\ \mu\text{m}$ [106]. Emitters were spaced $200\ \mu\text{m}$ apart. Pencil emitter arrays of up to 1025 emitters have been fabricated and were shown qualitatively to emit using EMI- BF_4 ionic liquid, as evidenced by markings left by impinging propellant on an opposing electrode [106]. Volcano emitter arrays, featuring 2×2 as well as 8×8 8-tip truncated volcano emitters have also been tested and emitted currents in the range of 0.2 to 200 nA per emitter using the same propellant.

In the British development up to 20,000 capillary emitters have been fabricated per 100-mm dia. silicon wafer [108]. Initial experiments using the traditional glycol propellant doped with sodium iodine, used $400\ \mu\text{m}$ O.D. and $100\ \mu\text{m}$ I.D capillary emitter tips and produced between 300 – 450 nA per emitter at voltages between 7.3 kV and 8.6 kV and various flow rates ranging from 54 nl/s to 151 nl/s [108]. More recently, emission using EMI- BF_4 propellant was demonstrated first using a single $30\ \mu\text{m}$ silica tip, then with silicon micromachined 19-emitter arrays (Fig. 37). Dual mode operation was observed, likely due to ionic and droplet emissions [111]. With the aide of the Ecole Polytechnique Federale de Lausanne (EPFL) in Switzerland, unique bead filled silicon micromachined capillary emitter arrays were fabricated (Fig. 38) [113]. By filling the capillaries with beads, hydraulic resistance can be tailored to provide the appropriate flow rates. Once again, 19 emitter arrays were built with capillaries $70\ \mu\text{m}$ in height, $34\ \mu\text{m}$ O.D and $24\ \mu\text{m}$ I.D. Silica beads of $4.74\ \mu\text{m}$ average diameter were filled into the capillaries via an alcohol based carrier fluid, which was subsequently evaporated. Heating the assembly fused the silica beads via a silanization process [113]. Both EMI- BF_4 as well as EMI-Im ionic liquids were tested. Using these arrays with EMI- BF_4 , specific impulse values as high as 3500 sec was measured at an emitter voltage of 1.2 kV and at thrust levels of 40 pN, averaging to about 2 pN per emitter [113].

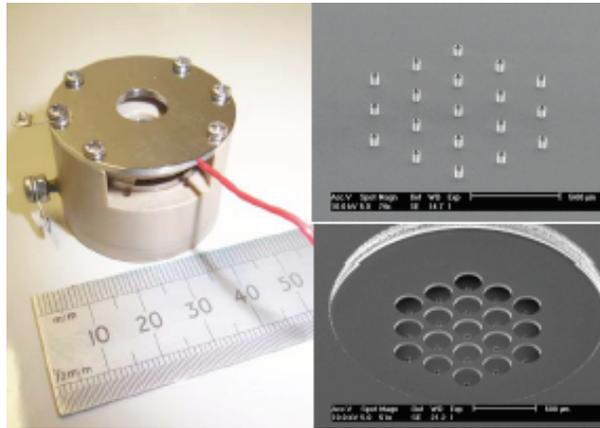


Fig. 37: Micromachined Colloid Emitter Array by the University of London [111].

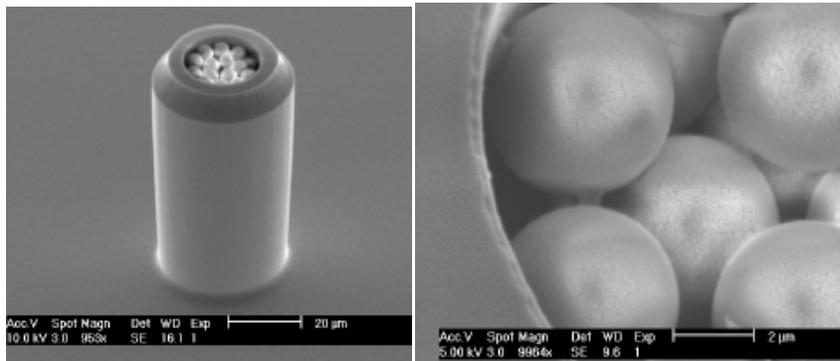


Fig. 38: Silica Bead Filled Silicon Micromachined Capillary Emitters by EFPL [113].

The Sandia/ARCS collaboration explored micro FEED arrays using Indium as a propellant [114], [115]. Several different techniques to micromachine emitter arrays, including laser etching, micro Electric Discharge Machining (EDM), photochemical etching, and silicon micromachining were explored [114], [115]. Although the most precise machining results were achieved with silicon micromachined arrays, no coatings had been developed yet to allow indium to wet silicon. Hence, laser micro-etched arrays were used for initial demonstration tests (Fig. 39). Arrays consisting of 21 x 21 capillary emitters, 90 µm in height with an I.D of 10 µm were machined and used in these tests [114], [115]. Currents of up to 4.5 µA were collected from the entire array. There was evidence that not all emitters fired at all times, which was attributed to the early development stage and related alignment issues, among others [115].

Microfabricated colloid arrays were also developed and tested quite early at Tsinghua University in Beijing, China [117], [118]. Tests with both silicon micromachined arrays [117] as well as a rather unique PCB board based array type [118] were conducted. In the case of the silicon based arrays, up to 192 emitters per array were fabricated (Fig. 40). Test units featured 4 emitters, and produced a measured thrust determined on a cantilever thrust stand of 6.8 µN. No information on the propellant was given [117]. An emitter voltage of 1480 V was required for onset, and voltage was driven to 1970 V to obtain the aforementioned thrust data [117].

A lower-cost approach was pursued by the Tsinghua team, replacing silicon micromachined arrays with PCB-board derived arrays [118]. Here, the copper deposited onto the PCB board is etched to form circular holes, connecting to the conducting liquid to apply the emitter voltage. A second PCB circuit

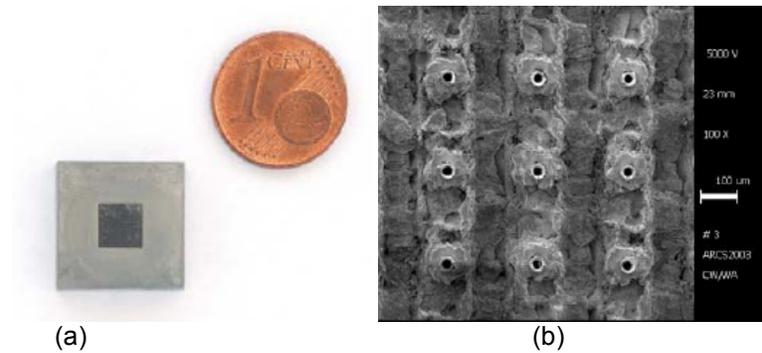


Fig. 39 (a) Laser Micromachined Emitter Array by Sandia, (b) Emitter Array Chip [115]

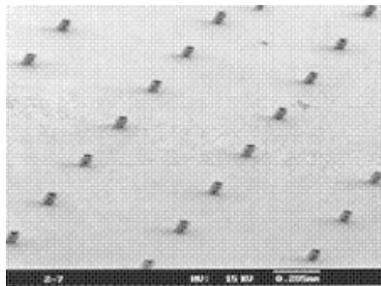


Fig. 40: Microfabricated Colloid Thruster Array of Tsinghua University, Beijing, China [117]

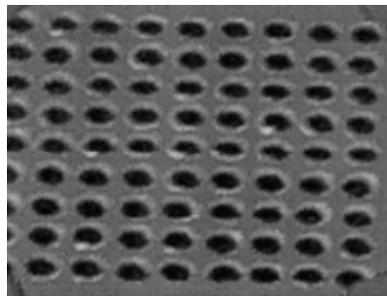


Fig. 41: PCB Board based Micro-Colloid Arrays by Tsinghua University, Beijing, China [118]

board with larger diameter holes served as an extractor (Fig. 41) [118]. The array featured 81 emitters of 0.3 mm I.D. Thrust values of $2.6 \mu\text{N}$ were achieved with the array, measured on a cantilever thrust stand. Once again no information reg. the propellant used was given. Tests were conducted in atmosphere, and voltage breakdown lead to limited thruster lifetimes [118]. A significant amount of effort was spend on the development of a silicon-based microfabricated pump to supply propellant to the chip, with a flow rate capability from 10 – 300 $\mu\text{l}/\text{min}$. Pump mass was 20 g, pump chip dimensions were 15 x 15 x 4 mm, and the pump required 200 mW to operate.

Microfabricated electrospray arrays offer significant potential for future microspacecraft applications - and beyond. They combine the inherent small physical dimensions of emitter tips in a compact, microfabricated package. Micro-colloid arrays offer high thrust-to-power ratios as a result of their lower specific impulse performance and offer dual mode operation (droplet and ion), significantly extending mission flexibility, yet typically have required pressure fed systems. Ionic propellants used are also highly water absorbent, requiring careful propellant handling and feed system design to avoid bubble formation due to water evaporation. Micro-FEEP arrays, in addition to offering the same level of miniaturization as

colloid arrays, offer the system benefit of capillary propellant feed, allowing for extremely compact, low-complexity, no moving parts feed systems, significantly easing integration. However, at the present time, no dual mode operation has been demonstrated with FEEP yet, and currently conventional FEEP systems are constrained to high specific impulse operation at lower thrust-to-power ratios than colloids. Despite the miniaturization achievable with colloid and FEEP arrays, both will require miniaturized PPUs to meet the CubeSat design envelope, which will be an area of required future development. Depending on array size and thrust levels desired, needed power levels may also require deployable solar arrays. If dual mode operation is feasible, micro-electrospray arrays could span a significant range of propulsive applications, from delta-v (both low-thrust/high-Isp and high-thrust/low-Isp), precision pointing, and slew in a single package.

Nanoparticle Field Extraction Thrusters (NanoFETs)

A novel thruster concept, the Nanoparticle Field Extraction Thruster (NanoFET) is an electrostatic thruster concepts that emulates electrospray thrusters, yet does not rely on droplet formation or ion emissions from a conductive liquid. Rather, it extracts solid particles of predetermined size from a liquid or powder reservoir [119]-[126]. Using particles of known size allows one to target specific impulses as needed for the mission, if multiple reservoirs with different particle sizes were available. the specific impulse could be tailored throughout the mission as needed, i.e. from high thrust-to-power and low-Isp for rapid maneuvers to low-thrust-to power and high-Isp for large delta-v applications.

Two different approaches are currently being explored. A liquid reservoir NanoFET relies on microparticles, either small conductive Al spheres or potentially nanotubes in later applications, suspended in a liquid with low vapor pressure, such as silicone oil [119], [122]. A micropump recirculates the liquid, transporting the particles past a charging electrode. An integrated extraction and accelerator grid system extracts the charged particles from the liquid reservoir and accelerates them to produce thrust. By emitting oppositely charged particles from different cells, no separate neutralizer would be required. The system was demonstrated with 1.5 mm long by 300 μm wide cylindrical aluminum particles, including in a reduced gravity flight experiment [125].

A dry powder particle extractor was designed to overcome several challenges identified with the liquid reservoir system, including maintaining a stable liquid/vacuum interface without fluid loss due to evaporation, fluid loss due to wetting of the particles, viscous drag in the liquid during extraction, and maintaining liquid purity to avoid electric breakdown [123]. In addition, fluid flow would need to be maintained to continually pass by the charging electrodes, requiring some sort of pumping mechanism, which could add complexity and would need to have adjustable pumping speeds unless accepting lower thrust levels with time as the reservoir is being increasingly depleted of particles.

Using a dry powder of ultrafine particles may lead to caking or clumping of particulates due to van der Waals forces. The dry NanoFET therefore relies on pushing a reservoir of particles through a sieve via piston action to separate the particles. Beyond the sieve, a grid system similar to the one in the liquid NanoFET is used to charge and accelerate the particles to produce thrust [123], [124]. The piston approach seems complex, and piezoelectric actuation of the sieve is envisioned to aid the extraction process [123]. Spherical particles ranging in size from 53 μm to 305 μm were used to demonstrate this concept [124].

No performance data have been obtained for either NanoFET type yet. Estimates range from under 200 sec Isp for solid 50-nm gold particles at up to 40 kV, up to 500 sec for solid ceramic particles of the same size, and up to 900 sec for hollow ceramic particles with a 5% (by diameter) shell thickness, which have yet to be demonstrated. Specific impulses are thus limited and accelerating voltages very high, which may make these concepts difficult to realize on Cubesats.

Other Nanoparticle Thruster Concepts

Two other nanoparticle thruster concepts have been proposed. The University of Kiel in Germany is exploring contact charging [127]. A needle emitter array is either coated or sprayed with nanoparticles, which acquire a charge upon contact and are emitted from the emitter needles and accelerated by a counter electrode. Obvious issues with this concept are propellant supply and feeding mechanisms.

At the University of Washington, a liquid colloid thruster concept is being studied that does not rely on capillary or needle emitters [128]. Instead, a conducting liquid propellant sheath deposited onto a diaphragm is excited into a standing wave pattern via a piezoelectric actuator attached to the diaphragm. This principle is used today in commercially available humidifiers. However, in this concept the wave amplitudes are adjusted such that they remain just below the conditions for atomization. Standing wavelengths are on the order of microns [128]. An electric field is then applied to the standing wave, exerting an electrostatic pull near the wave crest, overcoming the surface tension forces to eject charged liquid droplets. Using salt water, the basic feasibility of this concept was proven, emitting a current of approx. 65 nA at 3 kV applied voltage [128]. An obvious issue with this concept is the liquid/vacuum interface in zero-gravity conditions.

Resistojets

Resistojets are among the simplest electric propulsion devices in design. A liquid or gaseous propellant is passed through a heat exchanger, increasing its temperature, then expanded through a nozzle to produce thrust. Elevating the propellant temperature increases specific impulse and thrust. State-of-the-art thruster hardware, however, is far too heavy and power consuming for microspacecraft, and in particular Cubesats [6]. However, a few newer developments deserve mention.

Ammonia resistojets are being developed by Busek in collaboration with GSFC (Fig. 42) [129]. The thruster weighs 15 grams without valving, delivers 150-210 second specific impulse at 5-12 mN of thrust and a power level of 25 W [129].

A water resistojet has been under development at Surrey Space Centre in England, and was operated in space onboard an 88-kg microsatellite as part of the Disaster Monitoring Constellation (DMC) (Fig. 43) [130]. Heater requirements were limited to 3 W to be compatible with future Cubesat requirements [130]. A single tank was used, without a plenum, and heater wire was wrapped around downstream tubing to ensure complete vaporization. The propellant vapor would be further heated in the resistojet thruster itself. However, during testing ice formation was observed in the nozzle since the water vapor expanded and cooled. Additional heating or reducing the expansion ratio were considered two solutions. The latter was chosen. In-space demonstrations indicated that significantly more thrust was

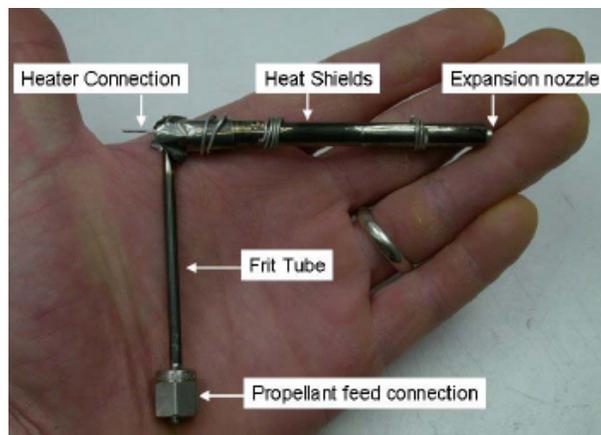


Fig. 42: Busek Ammonia Microresistojet [129]

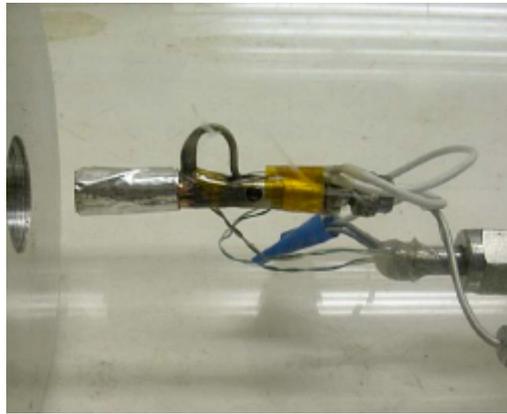


Fig. 43: Surrey Micro-Water Resistojet [130]

than anticipated produced (3.3 vs 0.2 mN), and in subsequent maneuvers it was found that the propellant tank was depleted. Apparently, liquid propellant has exited the thruster.

The icing issue using water propellant was previously observed in experiments with a MEMS-fabricated water resistojet at JPL, called the Vaporizing Liquid Micro-Thruster (VLM) (Fig. 44) [131]. The VLM was constrained to a maximum of 2 W of heater power in order to be compatible with future microspacecraft. At these low power levels, however, the thruster was not able to operate stably, as flow rates and hence feed pressures had to be set so low that vapor pressure building up inside the thruster chip induced chugging. The aforementioned icing problem was a result of the same low power constraints. As a result of these issues and power constraints, a butane thruster design by VACCO was adopted for the JPL Micro-Inspector spacecraft [8].

Ketsdever et al. has been exploring a novel type of Free Molecule Micro-Resistojet (FMMR) [132], [133]. In this concept, propellant is fed from a plenum as a very low pressurant gas such that the flow is in the free molecular regime, i.e. collisions over the characteristic thruster dimensions only take place with the hot wall surfaces, not among the molecules of the propellant flow. Due to the lack of momentum exchange between flow particles, all particles colliding with the heater wall surfaces will have the kinetic energy equivalent to the heater wall temperature. Thruster geometry is chosen such that all particles entering the thruster must intersect the wall heater surface[132]. Liquid propellant can be used if it is vaporized first when entering the plenum tank. Using water propellant, performances of 129 μN at 79 sec specific impulse were demonstrated at 3.2 W steady state power (5W maximum during heating transients) [133].

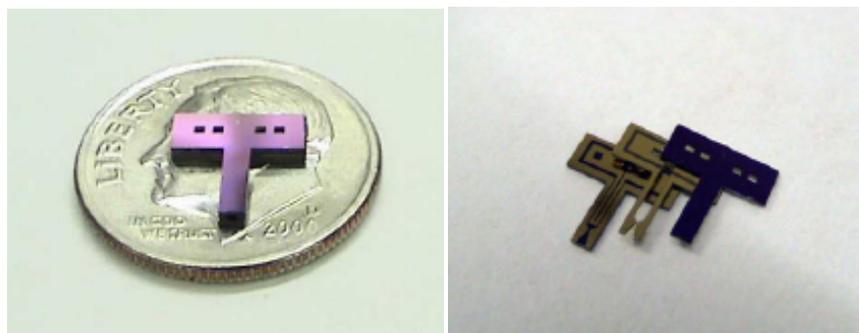


Fig. 44: JPL Vaporizing Liquid MicroThruster (VLM) [131]

Figure 45 shows the FMMR heater chip, as well as the technology demonstrator propulsion system lay-out, which was constrained by commercially available valve hardware. This latter image demonstrates well the need for integrated feed system approaches, as alluded to in the introduction, a fact that Lee et al. [133] are pointing out when referring to future, MEMS-based valve options that could be directly integrated with the FMMR chip, such as MEMS based piezovalves [134] or isolation valves, like the JPL Micro-Isolation Valve (MIV) [135].

Resistojets, despite their simplicity, offer only little performance improvements over cold gas systems at power levels, which, although small by conventional standards, may present a significant demand on a CubeSat power system. In the case of the aforementioned JPL Micro-Inspector satellite, the JPL VLM concept was therefore abandoned in favor of a newly developed butane thruster by VACCO [8]. Although lower in specific impulse performance than resistojet concepts, the power demands, other than during valve cycling, are significantly reduced over resistojet concepts.

Microcavity Discharge Thruster

The Microcavity Discharge Thruster (MCD) is a novel electrothermal thruster concept. It relies on flat panel microplasma heritage, realizing discharges in cavities as small as 10 μm in diameter at pressures up to well above 1 atm. Gas temperatures may reach 1500 K or higher at up to 2 W of power deposited per cavity, and if expanded through a nozzle, an electrothermal microthruster array concept can be realized [136]. Arrays with up to 1 million cavities have already been fabricated, allowing for significant scalability of this thruster concept, similar only to micro-electrospray arrays, potentially spanning power ranges from milli-Watt to kilo-Watt levels [136], yet operating in a higher thrust and lower specific impulse regime as the latter due to its electrothermal character.

The thruster concept consists of two perforated aluminum foil electrodes onto which an aluminum oxide layer is grown. The two electrode sheaths are then bonded, and the perforations form the discharge cavities. A chemical etch forms a nozzle for each cavity at one side of the electrode sheath. Applying a 50-150 kHz, 400-1200 VAC to the electrodes creates alternating electric fields inside the gas filled cavity, leading to low ($\ll 1\%$) degrees of ionization and subsequent heating of the partially ionized gas in the alternating electric field. Figure 46 shows the thruster concept and etched nozzle [136]. Computational modeling of such discharges [136], [137] and initial thrust stand experiments have been performed with thruster prototypes at up to 0.25 W per cavity using neon with added nitrogen and water vapor to improve power absorption [136]. Thrust values of 0.6 – 2.7 mN were obtained at 120 – 240 kPa at 0.99 – 5.22 mg/s propellant flow or a 4-cavity array with 120 μm throat and 210 μm exit plane diameter.

Microcavity discharge thruster arrays are an interesting concept in early stages of development, for both microspacecraft applications and beyond. It extends the thrust range that could be delivered from electro-spray arrays to higher thrust levels at higher thrust-to-power, yet lower specific impulse values.

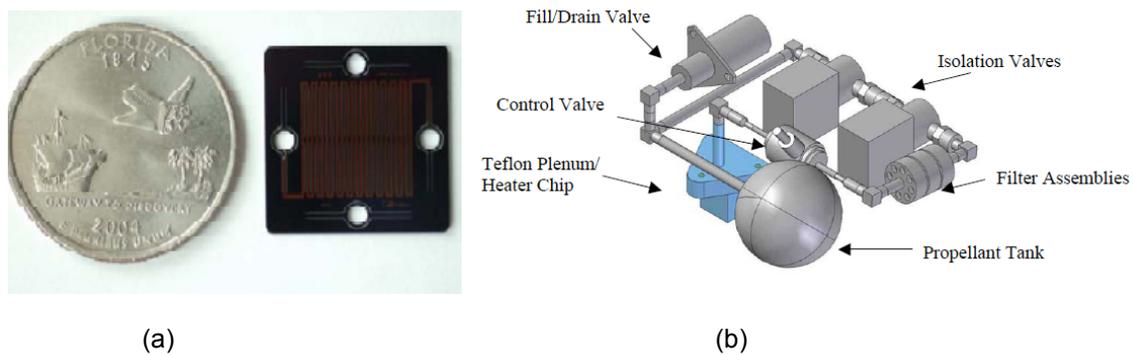


Fig. 45: FMMR (a) Heater Chip and (b) System Layout [133]

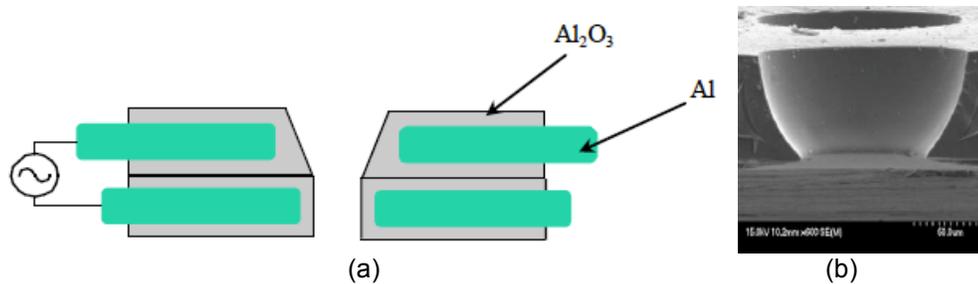


Fig. 46: (a) MCD Electrothermal Thruster Concept and (b) Micronozzle [136]

Yet it operates at higher gas temperatures than conventional resistojets, promising higher specific impulses than the latter, and is highly scalable by appropriately sizing the array. Microcavity arrays may therefore fill a gap between higher Isp and lower thrust-to-power electro spray arrays and higher thrust, lower Isp chemical systems, and be used for attitude control, but also for rapid slew maneuvers and proximity operations, were relatively responsive spacecraft are required, as well as for low delta-v applications.

PROPELLANTLESS PROPULSION

Propellantless propulsion tends to peak the interest of systems engineers in an attempt to avoid the complexities of propulsion systems. Several different propellantless propulsion options exist. As will be seen, these may be applicable to niche applications, yet bear with them their own set of complexities.

Electromagnetic Formation Flying

Electromagnetic formation flying has been develop for potential application to future interferometer missions for conventional spacecraft, and are currently under development by two research groups at MIT and in Japan [138], [139]. Since microspacecraft applications may involve swarms of these craft, in particular in Earth orbit where disturbance forces acting on a constellation may induce perturbations that need to be corrected against, electromagnetic formation flying may appear as a candidate for formation flying control as well. However, electromagnetic formation flying involves several limitations, which are amplified for microspacecraft.

In order to generate sufficient forces, high current electromagnetic coils must be used. For conventional spacecraft, the use of high-temperature superconducting coils is therefore considered, requiring to operate at temperatures below 77 K [138]. For conventional future interferometry missions, requiring cryocoolers for optical instruments, cryocooling of the electromagnetic coils is therefore not a significant added impediment. Microspacecraft, and in particular CubeSat platforms are unlikely candidates for sensitive optical instruments due to aperture limitations based on their size, and cryocooling would in all likelihood be a requirement difficult to realize in such a small spacecraft envelope. Obviously, forces required to control a microspacecraft constellation are less than those for conventional craft, but as was seen in the Introduction, power levels available on CubeSats are equally constrained, shifting the problem to smaller, lower power electromagnets, but not likely not eliminating the need for the same high-temperature super conductors.

In addition to either attractive or repelling forces, two interacting electromagnets may also induce torques when pointed orthogonally with respect to each other. These torques will cause the formation to rotate around its center of mass. Conservation of angular momentum will then induce spinning of the individual craft around their center of mass, with an angular momentum equal and opposite to that of the formation. This added complexity requires the use of momentum wheels to compensate. Theses wheels, however, may saturate over time, requiring propulsive means to desaturate them, magnetic torquers, or a scheme to sinusoidally vary the magnetic dipoles to result in a zero net gain of momentum of the wheels [139].

Electromagnetic formation flying, although avoiding complexities related to the integration of a propulsion system, thus has its own complexities to contend with, including the management of spacecraft torques, cryocooling, limited range of force action for a given power level, and the fact that only interacting forces between spacecraft can be induced. Electromagnetic formation flying offers no delta-v, drag-make-up, or deorbit capability, and maneuverability is thus rather constrained. Furthermore, reaction wheels are required.

Electrodynamic Tethers

While electromagnetic formation flying can only provide relative spacecraft motion and attitude control via reaction wheels, electrodynamic tethers have the opposite constraint. Tethers may provide de-orbit [140] or orbit raising capability [141] when used as a momentum exchange tether between two coupled objects. Tethers do not allow for attitude control or precise relative motion, as may be required for inspection satellites. Orbit raising also requires coupled spacecraft, placing further constraints on the spacecraft and mission design.

Probably the most interesting application of tethers are their use for de-orbit, as this is an issue of growing concern with an increasing number of space objects in near Earth orbit. An electrodynamic tether used for de-orbit relies on Lorentz forces action on the electrons in the conductive tether as it intersects Earth's magnetic field lines along its orbital path. A voltage difference is thus set up between the ends of the tether. If the circuit is closed by contacting the ends of the tether to the ionosphere by means of field emission cathodes, for example, a current flows through the tether that will induce a force by interaction with Earth's magnetic field, perpendicular to it and the tether, causing it to decelerate, and the spacecraft connected to it along with it [140].

Electrodynamic tethers may thus serve as a de-orbiting device if no propulsive means are provided on the spacecraft. In case they are, adding more propellant to an existing tank may be a more mass efficient option than adding an additional tether system for this purpose, pending a trade involving delta-v required for deorbit, specific impulse of the existing propulsion system, required tether length and its deployment system. CubeSats, however, have been used in the Multi-Application Survivable Tether (MAST) experiment to deploy and diagnose a tether experiment in low earth orbit for micro-meteoroid impact[142]. Thus, in this experiment CubeSats are serving the tether experiment as a diagnosis tool, rather than the tether serving the CubeSat.

Solar Sails

Solar sails have been proposed for space missions in the past [143], however, no missions have utilized them yet. Solar sails rely on reflecting sunlight, and propelling the spacecraft due to the transfer of momentum from the sun's photons to the sail. Solar pressure acting on sails is small, on the order of 4.7×10^{-6} N/m² in Earth's vicinity [144], thus requiring large sail areas to propel a conventional spacecraft. In order to not add a significant amount of mass to the spacecraft, sails have to be extremely light-weight, which needs to include the deployment mechanism. Microspacecraft may offer another application for this technology. Given the much lower mass of microspacecraft, sail sizes can be reduced, easing fabrication and deployment.

Solar sail applications would typically be for delta-v, in particular for interplanetary travel. However, as was noted in the Introduction, use of microspacecraft for interplanetary missions may not take the form of deploying individual microspacecraft as stand-alone spacecraft to their destination, for reasons of limitations with respect to instruments that could be carried along, especially with respect to optical instruments (realizable aperture sizes on microspacecraft) as well as communication constraints due to limited space for high-gain antenna equipment. Rather, microspacecraft may be used for these and other reasons as adjunct spacecraft to a larger mother ship that deploys microspacecraft for particular portions of the missions, such as those involving particular risks, for example. Therefore, this traditionally

envisioned applications of solar sails to provide large delta-v along interplanetary trajectories may not be a major driver for microspacecraft.

A few niche applications have been identified, however. These may involve missions to study earth's magnetotail [143], [144], or an early solar storm warning network [144]. In the former, a constellation of microsats deployed throughout the magnetotail region allows this constellation to move along with the magnetotail, i.e. the Earth-Sun line, as it moves around the sun. In the latter, solar sails allow a spacecraft to orbit the sun at a smaller radius than Earth's orbit, yet with an angular velocity equaling that of Earth, placing the spacecraft always between the Earth and the Sun. The difference in force between Sun's gravity and centrifugal forces due to orbiting at a lower angular velocity than required for this orbit is made up by forces acting on the solar sail. Placing a constellation of microspacecraft at a location between the Earth and the Sun allows for an early warning network against solar storms [144]. New technologies are being studied that may be applicable to microspacecraft, such as curved thin shells produced by Mevicon, Inc. [144]. These shells are stowed compactly, but deploy to their inherent stiffness without required actuation mechanisms or booms (Fig. 47) [144].

Except for these unique cases, near Earth applications of solar sails may take a rather different form. Rather than relying on the reflectivity of the sail to produce thrust, sails may be deployed from CubeSats as drag chutes for deorbit (Fig. 48) [144]. This approach would compete with simpler drag devices, such as spring-loaded rigid surfaces deployable from the spacecraft, or inflatable balloons. It may be used in such applications where a large area is required for relatively rapid deorbit, warranting the added complexity of sail deployment. Once again, if a propulsion system is already onboard the spacecraft, the added cost, complexity, mass, and risk of adding a separate de-orbit system would have to be traded against adding additional propellant to an existing propellant tank for this purpose.



Fig. 47: Self-deploying Thin Film Shells by Mevicon [144]

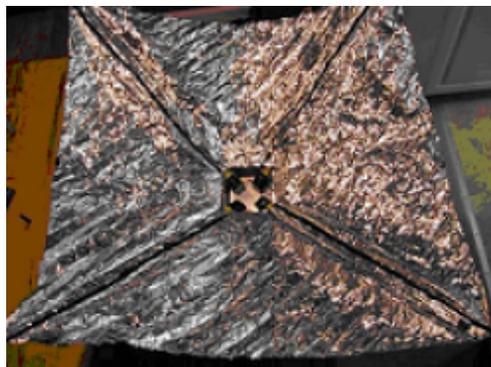


Fig. 48: Stanford Drag Sail for CubeSat Applications made from Mylar Survival Blanket Material [144]

Finally, as for tethers, CubeSats have been envisioned to be used as technology demonstrators for sub-scale solar sails, demonstrating deployment, solar sail dynamics or survival in the space environment. The CubeSat mission “NanoSail D” was to serve this purpose, but unfortunately did not make it to orbit due to a launch failure [145].

SUMMARY AND CONCLUSIONS

In order to increase the capability of future CubeSats, propulsion, notably absent on present day CubeSats, is required. Propulsion will allow for formation flying applications, orbit change (altitude and plane change), proximity operations, more precise attitude control, or de-orbit and therefore significantly increase mission flexibility over present day CubeSats. There are a vast amount of candidate propulsion systems available for future Cubesat applications, either under significant development or currently in research. In fact, since this author conducted his last review of the field about a decade ago, over a hundred new references could be added to this database.

Given the vast scope of this review, limits had to be self-imposed to keep it manageable. Most notably, no microvalve and other flow component technologies could be included in this review. This by no means should convey the impression of lessened importance of these components. In the contrary – miniaturization of thruster components only will not enable micropropulsion systems for CubeSats or other microsats, unless miniaturization of other components is addressed as well. At the same time, novel integration schemes need to be developed. Conventional integration by means of welded tube joints would give away a significant amount of the gains made in component miniaturization, and be labor intensive and costly. Novel approaches like MEMS-based, on-the-chip integration or ChEMS™ technologies currently being pioneered by VACCO Industries, will be required.

In the case of electric propulsion systems, the integration aspect has to involve the power and power conditioning systems. Current CubeSat designs offer too little power for many electric propulsion concepts, thus forfeiting enhanced mission flexibility, i.e. the possibility of significantly increased delta-v capability as may be required for orbit change. Deployable solar arrays are required and need to be developed in tandem to enable high-Isp electric propulsion systems onboard such spacecraft in order to fully capitalize on the benefits of electric propulsion systems. Power conditioning units, required to provide the electric propulsion systems with the required voltages currently are a major mass driver in already miniaturized systems, such as micro-PPTs, for example. Providing compact and lightweight units may likely dominate several already quite mature micro electric propulsion developments going forward.

Comparing this review with the one conducted a decade ago [6], several additional trends can be observed:

First, although a decade ago, micropropulsion was often, although by no means by everybody, viewed almost synonymously with MEMS, this trend underwent a significant reevaluation. While MEMS does offer the potential for significant miniaturization and very high precision, it also present the propulsion engineer with many new challenges. Among the former are the challenges to deal with new materials and their properties, not necessarily overlapping with the requirements of propulsion systems. The high degree of miniaturization enabled by MEMS may be more than what is required, or even feasible for some thruster concepts. Reducing thruster or plasma chambers to drastically increases wall losses, and may thus lead to efficiency losses. On the other hand, the chip-based integration of MEMS thrusters with valves and other components so far has only been realized in few cases, as the realization of reliable, high-force and leak-tight valves in MEMS poses even greater challenges than the thruster itself. Packaging MEMS thrusters individually and connecting them with conventional components, however, as pointed out above, does not realize the full miniaturization potential of this approach. At the same time, miniature metal machining techniques have matured and produced impressively small, and highly integrated systems.

There are a couple notable exceptions to this trend, however. Micro-electrospray arrays or future MCD arrays may greatly benefit, or even be enabled by microfabrication approaches. While neither of

these approaches necessarily has to rely on silicon microfabrication, it is clearly the basis for most of the electrospray array approaches currently being pursued. Assembling large numbers of emitters conventionally would be costly, likely not realizable at comparable packaging densities as obtainable with MEMS, and be less reliable due to fabrication tolerances. In these areas, MEMS microfabrication may have found a truly unique role in realizing these concepts. In fact, the scalability of these devices through MEMS to much larger sizes may allow them to be applied to much larger spacecraft and power levels than would be possible with conventional architectures.

A second trend that could be observed over the last decade is the increased relative level of international activities in the field. While these were present from the beginning, their relative weighting appears to have increased compared to US activities. This is neither an avoidable nor an undesirable trend, and signifies the excitement that this technology has generated worldwide. For many smaller nations microspacecraft are also the only affordable way to enter space. On the other hand, activities in the US, although still dominant, have lost breadth. While in the defense area several focused activities have been maintained and have achieved impressive results, such as in the area of electrospray arrays, overall the number of activities has been on the decline. This may have had one of its reasons in the demise of technology funds in recent years at important national institutions, such as NASA, a trend that only just now is in the process of reversal.

Where does all this leave the CubeSat designer? Concretely, three major areas of micropropulsion applicable to CubeSats and similar sized microspacecraft can be identified: (1) Existing or Nearly-off-the-shelf technologies, (2) New technologies under significant development that may enable significant increases in propulsive capability, and (3) Emerging research and development that allows an even greater level of integration and compactness, while affording far reaching propulsive capabilities for the future. These three areas will be briefly elaborated upon in the following.

(1) Existing or Nearly Off-the-Shelf Propulsion Technologies:

The following technologies are all available for near-term use on CubeSats today with only minor or no modifications required. However, they currently offer only limited propulsive capability. They include:

- **Butane Systems:** These systems combine liquid propellant storage and low-mass tankage with the simplicity of a cold gas system, at high levels of integration using VACCO ChEMS™ approaches, yet at limited specific impulse and delta-v capability. They are sufficient, however, for proximity operations. Using nitrous oxide, tank masses increase, but so will delta-v capability. Systems developed for MEPSI are ready for CubeSat usage.
- **Solid Rocket Motors (SRMs):** Solid rocket motors are a mature technology, and available even for very small spacecraft due to recent nanosat motor developments. However, spacecraft accelerations for a CubeSat would still be significant, and spacecraft integration requires additional considerations with respect to c.g. and attitude control, either requiring spinning the spacecraft, or providing three-axis control. Attitude control and orbit trimming propulsive capability may be required, which would add complexity to a CubeSat design so far not considered.
- **Micro-Pulsed Plasma Thrusters (PPTs):** These systems are based on the earlier AFRL μ PPT development, and integrated systems have been developed by Busek that have flown on 100-kg class microsats, which could also be integrated into CubeSats. Challenges remain in the area of PPU miniaturization, and the fact that total impulses of these systems are limited and that operation is inherently pulsed mode. They are suitable for precision pointing and small delta-v.
- **Vacuum Arc Thrusters (VATs):** These systems have been adapted to CubeSats. As for PPTs, however, operation is inherently pulsed and total impulse is limited. They are suitable for precision pointing.

(2) New Technologies under Past Significant Development:

These technologies have seen significant development, but require additional effort in subsystem miniaturization, such as feed system and power processing. If realized, they will significantly extend propulsive capabilities of future Cubesats. They include:

- **Monopropellant Systems:** Miniature hydrazine thrusters would offer significantly increased specific impulse and delta-v capability, yet provide thrust levels required for rapid orbit transfer. If feedsystem (valve) technologies can be optimized, lower power operation than presently available may be possible, which at this point prohibits their use on CubeSats. Hydrazine use onboard CubeSats has been viewed with skepticism in the past due to safety concerns, however, with the interest by government and industry into CubeSats this may no longer be a major impediment. Hydrogen peroxide use has been proposed, however, long term storability needs to be verified.
- **Ion Propulsion Systems:** Ion propulsion offers significant increases in Isp. Both DC as well as RF micro-ion thrusters have already been demonstrated, micro-feed systems and PPUs, however, remain to be developed. Also, to be applicable to CubeSat, their power capability needs to be increased significantly to greater than 50 W, likely through the use of deployable solar arrays. CubeSats of the “3U” architecture could be envisioned with a propulsion module featuring thruster, propellant tank and integrated feed system, a power module, featuring PPU and deployable arrays, and a payload/avionics/telecom module. Significant miniaturization has also been achieved recently for Hall thrusters at much improved efficiency levels, yet thruster masses are high due to magnetic circuitry.
- **Colloid Thrusters:** Leveraging significant development in colloid thruster system development for the New Millennium ST-7, single thruster heads of the ST-7 thruster could be adapted to CubeSats to provide delta-v capability. The high thrust-to-power ratios of colloid thrusters at a lower Isp than ion systems fills a gap in performances between ion and chemical systems that may be of interest in particular to Earth-observing missions.

(3) Emerging Technologies:

- **Microfabricated Electrospray Arrays:** Microfabricated Electrospray arrays were the first MEMS thruster concepts being proposed in the early 1990s, however, have emerged as working laboratory devices only over the course of the last decade. Electrospray arrays offer significant advantages with respect to compactness, in particular for liquid metal FEEP arrays not requiring pressure-fed feed systems, inherent modularity in those cases, and resulting ease of integration, as well as a large specific impulse range if dual mode operation (droplet and ion emission modes) can be implemented. The latter have already been demonstrated for colloid systems. The theoretically feasible scalability to very large arrays using microfabrication techniques may actually feed back into applications on conventional (non-micro) spacecraft. If the technology can be matured, electrospray arrays may perform a wide range of propulsive tasks, incl. attitude control, precision pointing, as well as delta-v applications in a single package.
- **Micro-Cavity Discharge Thrusters:** Using microfabricated cavity discharges arranged into arrays, they are the electrothermal equivalent of electrospray arrays, yet operating at lower specific impulses and higher thrust levels than the latter, and are thus applicable for attitude control and delta-v applications with faster transfer requirements. Since they use gaseous propellants, they do require feed system components that would need to be miniaturized.

Propellantless propulsion concepts, although repeatedly garnering attention, only have limited application to CubeSats and similar sized microsats. They either are unlikely to be used on microspacecraft, such as electromagnetic formation flying, due to high-temperature superconducting electromagnetic coils and associated cooling required, or their limited flexibility, either for specific niche applications, such as solar sails for magneto tail or solar storm warning networks, or de-orbit. In the case of the latter, such systems may make sense if no propulsion system is available on a Cubesat. Higher capable craft that feature propulsion systems may not benefit from adding a separate system for such specific applications when additional propellant can be loaded into already existing propellant tanks.

Adding propulsion systems to CubeSats will add complexity and cost in exchange for additional spacecraft capability. In particular the large non-recurring development costs are beyond the capabilities of the typical university CubeSat user. The recent interest by industry and government into CubeSats, however, offers an opportunity to contribute to the field with such developments and their subsequent flight application, thus reducing future costs of these systems and making them more widely available. Thus, CubeSats may not only have a role in future government and industry space activities, adding new capability due to their small size, but, vice versa, government and industry may yet have a role in CubeSat development by increasing the capabilities of these unique space assets.

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