

State-of-the-Art for Small Satellite Propulsion Systems

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Abstract - *The NASA/Goddard Space Flight Center (NASA/GSFC) is in the business of performing world-class, space-based, scientific research on various spacecraft platforms, which now include small satellites (SmallSats). In order to perform world class science on a SmallSat, NASA/GSFC requires that their components be highly reliable, high performing, have low power consumption, at the lowest cost possible. The Propulsion Branch (Code 597) at NASA/GSFC has conducted a SmallSat propulsion system survey to determine their availability and level of development. Based on publicly available information and unique features, this paper discusses some of the existing SmallSat propulsion systems. The systems described in this paper do not indicate or imply any endorsement by NASA or NASA/GSFC over those not included.*

Keywords: small satellite, SmallSat, CubeSat, propulsion

1 Introduction

Small satellites (SmallSats) are miniature satellites that are categorized by their mass, which ranges from 10 g to <500 kg [1]. A CubeSat is a class of SmallSats of a standard size and volume (10 x 10 x 10 cm) known as a “U”, and mass (1.33 kg), originally developed in 1999, by California Polytechnic State and Stanford Universities [2]. These spacecraft have been used as a low cost method to conduct low-earth orbit research and technology demonstrations. As SmallSats become more capable there will be a growing need for them to have propulsion. To highlight an example, in January 2016, the International Space Station (ISS) deployed its first SmallSat mission that contained a propulsion system, called LONESTAR, using the refrigerant R-236fa as the propellant.

The current state-of-the-art (SoA) for SmallSat propulsion systems is rapidly evolving. However, the technology readiness level (TRL), i.e., their level of development, is still relatively low. The desired SmallSat propulsion system is high performing and highly reliable, with the simplest design feasible, in order to meet performance requirements. In order to keep the cost low, vendors and NASA must find the right balance between reliability, performance, and complexity. However current systems are either:

- Low-cost, unreliable, and low performing, or
- High-cost, reliable, and high performing

It will be to the advantage of current and future SmallSat missions to have simple, low-cost, highly reliable, and high performing propulsion systems. However, several obstacles must be overcome to achieve this balance.

1.1 Obstacles to SmallSat Propulsion

The main obstacles to any SmallSat technology are reliability and maturity. SmallSat propulsion technology, however, has the added obstacle of system safety. As it currently stands, primary payloads and NASA/Johnson Space Center (NASA/JSC) (for ISS) will not allow additional hazards to be flown, e.g., high pressure systems (>100 psia) or hazardous propellants. This restriction was primarily born out of the fact that most CubeSats are developed by hobbyist and university students that don't have the same rigorous quality standards as most government agencies or large private organizations that develop and build spacecraft. In addition to this, as discussed earlier, the reliability of current SmallSat components is questionable, at best. Add into the mix a system that uses a highly toxic, highly energetic, and/or highly complex pressurized propulsion system, designed to the same standards, then the restriction becomes understandable.

In addition to these safety challenges, there are cost drivers that must be addressed as well. The main cost driver for SmallSat *electric* propulsion systems is the development of a low-cost, highly reliable power processing unit (PPU). All self-contained propulsion systems have a PPU, or similar system, to distribute the power properly. Space-flight qualified systems (e.g., those that are radiation hardened) for large spacecraft buses are costly, in and of themselves. Designing PPUs for small spacecraft for the near-term low-Earth orbit (LEO) missions, and for more challenging lunar and interplanetary missions, will only increase this price to obtain reasonable levels of reliability (driven by radiation hardness). Innovation in this area is greatly needed in order to keep SmallSats cost effective.

The main cost driver for SmallSat *chemical* propulsion systems is ensuring system safety. This is particularly true for those that use high pressure, toxic, and highly energetic propellants. Any propulsion system (electric or chemical) will have to meet U.S. Range Safety requirements. However, for chemical systems, it is thought that SmallSat propulsion systems would have to significantly exceed current requirements, or at least well documented, in order to convince primary payloads or NASA/JSC that these systems will not create undesirable risk. While, many of these systems are very early in their development, private industry has been working with Range Safety representatives to ensure system compliance.

The ideal SmallSat propulsion system is one that balances reliability, high performance (i.e., relatively high specific impulse [I_{sp}] and thrust), has no chemical or electromagnetic contamination issues, is low pressure (or pressurizes post deployment), safely contains propellant (hazardous or non-hazardous), low cost, and has the simplest design feasible in order to meet performance requirements. There is work on-going in academia and industry to balance a subset of these desirements. A few of those systems are described in this paper.

2 SmallSat Propulsion System and Thruster Options

There are many SmallSat propulsion system and thruster options at various stages of development currently available in the commercial market. The technologies consist of a wide range of propulsion system types: chemical (e.g., cold gas and green), electric, solids, and non-propellant (e.g., solar sails and tethers). This section provides a brief description of some of the available chemical and electric propulsion technologies. The systems described in this paper do not indicate or imply any endorsement by NASA or NASA/GSFC over those not included.

In addition to the standard parameters used to quantify the performance of these propulsion systems (mass, thrust, I_{sp} , power), the parameter of volumetric impulse will be also used. This efficiency parameter that describes the amount of total impulse (Ns) a system imparts to a body per unit volume (U). The units for volumetric impulse are (Ns)/U.

2.1 SmallSat Cold Gas Propulsion Systems

Thus far, the systems that are the lowest cost, simplest, and the most developed are the cold gas systems. These systems use non-traditional in-space propulsion system propellants, such as saturated liquids. Examples of

these are refrigerants, such as R-134 and R-236fa, sulfur dioxide, and butane. The advantage of using these two-phase liquids as propellants is their high vapor pressure at ambient temperatures, essentially making them self-pressurizing. For example, R-236fa has a vapor pressure of 33 psia at 20°C. Combining these liquids with simple cold gas propulsion system architecture allows for a low-cost, highly reliability system at a high TRL.

2.1.1 CubeSat MicroElectroMechanical Systems (MEMS) Propulsion Module

The *CubeSat MEMS Propulsion Module* is a cold gas system (see Figure 1) that was developed by NanoSpace AB, based in Uppsala, Sweden. It is currently flying on a Chinese mission called Tianwang-1 [3] that was launched in late-September 2015. This formation flying mission employs two un-propelled 2U CubeSats and one 3U CubeSat that is maneuvered by the MEMS propulsion module. The propellant for this system is butane, a two-phase liquid that has a high vapor pressure at nominal temperatures. This system has a wet mass of 300 g with a propellant mass of 50 g. It consists of four MEMS thruster chips, a propellant tank (with integrated heater and temperature sensor), fill/drain valve, filter, normally closed isolation valve (one per thruster), proportional flow control valves (one per thruster), and front-end control electronics (one per pair of thrusters) to monitor the thruster's temperature and pressure sensors [[4],[5],[6]]. The tank heaters are used to maintain a desired system pressure. For an operating temperature range of 0 to +40°C, the butane vapor pressurizes the system from 0 to 58 psia.



Figure 1. NanoSpace CubeSat MEMS Propulsion Module [7]

This system was initially designed for a 3U spacecraft application and measures about 9.5x9.5x4.9 cm. The propellant tank, as shown in Figure 1 in the center of the unit, is a cylindrical “tuna can” that can be lengthened to provide more propellant to meet delta-v requirements. The size of the tank is limited by the available volume within the spacecraft bus. The performance of the baseline 3U, four (4) thruster, MEMS Propulsion Module was found to be <4 mN/110 sec (thrust/ I_{sp}). These parameters are based on a propellant mass of 50 g and an average operating power of <2.5W, which was verified between ground and

on-orbit testing. This system has a volumetric impulse of 133.3 (Ns)/U. NanoSpace is currently developing a 6U version.

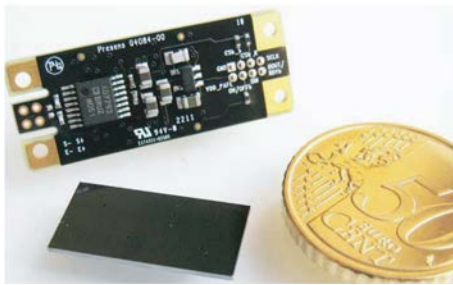


Figure 2. MEMS Thruster chip (bottom) and Front-end Electronics (top) (€0.50 and \$0.25 about the same diameter) [5].

The unique features of this system are the MEMS thruster chips, which contain the majority of the flow components, and the closed-loop control electronics, which allow the system to vary the thrust level within its range of 0.01 to <1 mN. Both are shown in Figure 2. Each MEMS thruster chip contains a proportional flow control valve, the thrust chamber with a gas heater (for warm gas performance), a mass flow and temperature sensor, and the thruster nozzle. The proportional flow control valve is controlled by the mass flow sensor and front-end electronics. This tandem allows the thruster to be operated continuously within its designed range, and is able to achieve a thrust resolution knowledge of 0.01 mN.

2.1.2 3D Printed Cold Gas Propulsion Module

Dr. E. Glenn Lightsey, while at the University of Texas-Austin (UT-Austin) [8], led a group to develop a 3-dimensional (3D) printed cold gas propulsion system that uses the refrigerant R-236fa as propellant (see Figure 3). The crux of this technology is the development of architecturally complex cold gas systems using the stereolithography (SL) process. These systems include thruster, tanks (main and plenums), internal piping, and ports to attach valves and sensors all in one unit made of a resin block. The advantage of the SL process is that the system's form factor is very flexible, in that, it could be made to fit "around" other components, if necessary. Through testing, UT-Austin found that the Accura Bluestone, a nanocomposite resin, offered better material strength and temperature tolerance than others tested.

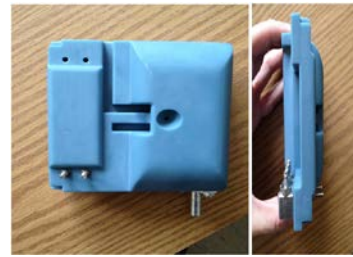


Figure 3. Bevo-2 3D Printed Cold Gas System [8]

To operate the system, pressure and temperature sensors, miniaturized solenoid valves, and the control electronics were mounted to the system's exterior. The control electronics board is used to control valve actuation and read the signals received from the pressure and temperature sensors. During operation, propellant is transferred, via a solenoid valve, from the main tank into a smaller plenum as vapor due to the pressure differential between the two chambers. The size of the tank and plenum are scaled to meet performance requirements. When the spacecraft is ready to maneuver, another valve is actuated to allow the vapor to move from the plenum out through the thruster.

The system shown in Figure 3 is currently on-orbit. A joint mission between UT-Austin and Texas A&M University (TAMU), called LONESTAR, was deployed from the ISS in January 2016. This is a proximity operations mission where a 50+ kg SmallSat, developed by TAMU, deployed a 3U spacecraft, developed by UT-Austin. The smaller spacecraft, called Bevo-2, contains the propulsion system. During the mission, Bevo-2 will fly around the larger spacecraft, performing proximity detection maneuvers. The Bevo-2 propulsion system had a wet mass of 0.38 kg and was 10.0 x 9.0 x 4.4 cm in size. Based on mission requirements, the Bevo-2 propulsion system was designed to provide (thrust/ I_{sp}) 40 mN/35 sec while maintaining a propellant temperature of 24°C. Bevo-2 has a volumetric impulse of 146.5 (Ns)/U. Two more SL cold gas systems, also developed by UT-Austin, will fly by the end of 2017: JPL's INSPIRE mission [9] and Georgia Tech's Prox-1 mission [10].

2.2 SmallSat Green Propulsion Systems

The emergence of green propellants (AF-M315E and LMP-103S) is driving the development of user technologies at all scales, from small to large spacecraft buses. These systems will have the inherent risks associated with using high pressure, highly energetic propellants. However, due to their low toxicity, they offer similar performance and complexity to their hypergolic counterparts without the handling risks. This is promising for SmallSat use, allowing for high performance and capability without the cost and risk of handling hazardous propellants. There are several of these SmallSat green propulsion systems currently under development that are maturing and working

towards flight qualification status, one of which is discussed here.

2.2.1 Advanced Monoprop Application for CubeSats (AMAC)

Busek Co. Inc. is developing a SmallSat green propulsion system known as the Advanced Monoprop Application for CubeSats (AMAC) [13], shown in Figure 4. The prototype system has a 1U form factor and uses the Air Force developed green propellant, AF-M315E, for a 3U spacecraft bus. The highlights for the AMAC are the titanium (Ti) bellows propellant tank, the 500 mN AF-M315E thruster, and their patented post-launch pressurization system (PLPS) used to pressurize the pressurant side of the bellows-driven propellant tank. The PLPS can re-pressurize the tank on demand.

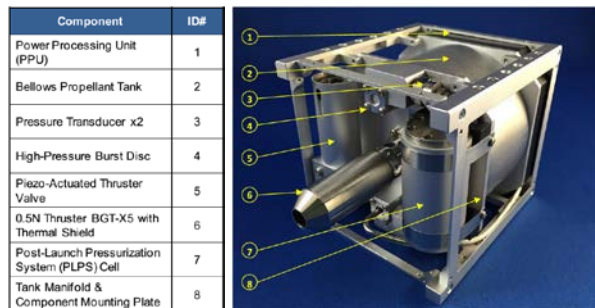


Figure 4. View of Busek's AMAC system [13]

In order to meet U.S. Range Safety requirements, AMAC includes two inhibitors: a piezo microvalve, located just upstream of the thruster, and a 50 psi Ti burst disc, located in between the valve and the thruster. To prevent tank over-pressurization, an 800 psi burst disc has been placed in the pressurant manifold. If this disc is ruptured it will not only depressurize the tank, but also disable the AMAC, rendering it unusable. The propellant manifold, tank, valves, and propellant-wetted gaskets are all titanium, which is fully compatible with the propellant. Finally, the PPU, which is currently under development, will have a peak consumption of 20W for catalyst bed heating and operation of the PLPS.

Busek has developed a 100 mN (BGT-X1) and a 500 mN (BGT-X5) AF-M315E thruster for SmallSat applications. The 500 mN thruster is shown in Figure 5 and will be discussed here. The BGT-X5 is able to provide thrust from 100 to 500 mN depending on feed pressure, which ranges from about 100 to 400 psia. During hot fire testing, the protoflight BGT-X5 has demonstrated 425 mN of thrust, 220 sec. of I_{sp} . The AMAC has a volumetric impulse of 565.0 (Ns)/U. Busek has also developed their own robust, non-granular catalyst for the AF-M315E that must be pre-heated to a minimum temperature of 430°C, which would take about four minutes given 20W of power.

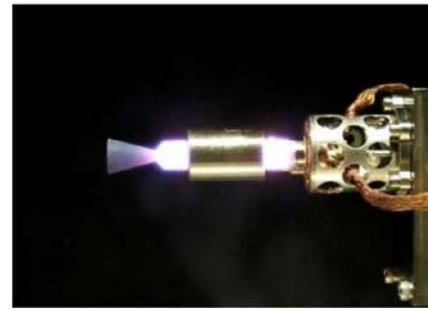


Figure 5. Busek BGT-X5 Thruster during testing [13]

The other salient feature of the AMAC is its patented PLPS. One of the challenges of flying chemical propellant systems is the hazard of high pressures needed for operation. Busek's answer to this question was to develop a system that pressurizes the propellant tank well after the SmallSat has deployed from its launch vehicle. Prior to deployment, the propellant is at low blanket pressure. Once the PLPS is activated, an inert solid material is heated to $>130^{\circ}\text{C}$ to produce gaseous CO_2 , which is the pressurant used in the propellant tank. This novel system is repeatable, making it a "hybrid blowdown system." As the pressure reduces in the expanding bellows tank, the PLPS can be reactivated to increase the tank pressure in order to provide optimal thruster performance. The repeatability of this operation depends on available power and the amount of consumable solid material.

2.3 SmallSat Electric Propulsion Systems

Increased efficiency has always been the goal of propulsion systems. Electric Propulsion (EP) systems have, historically, provided high specific impulse and high delta-v. However, these systems have historically been more expensive as well. One of the many challenges for EP system technologists has been in the development of highly reliable electronic components for their PPU's for SmallSat applications, in a cost effective way. Fortunately, there are many academic and private organizations working to develop a wide variety of high performing systems that work to balance complexity, cost, performance and reliability.

2.3.1 PUC and CHIPS

CU Aerospace, in partnership with VACCO Industries, has produced two CubeSat propulsion solutions, known as Propulsion Unit for CubeSats (PUC) [14] (see Figure 6, left) and CubeSat High Impulse Propulsion System (CHIPS) [15] (see Figure 6, right). Both propulsion units can operate in a cold gas or warm gas mode. A warm gas system is, essentially, a cold gas system that uses an electrothermal approach to increase the temperature of the exiting gas to produce higher thrust and I_{sp} . PUC uses an electrothermal technology known as microcavity discharge (MCD), while CHIPS uses a micro-resistojet. Available

propellants are refrigerants R-134a, R-236fa, or sulfur dioxide, which produce various levels of performance in each system.



Figure 6. Propulsion Unit for CubeSats [14] (left), CubeSat High Impulse Propulsion System (CHIPS) [15] (right)

During operation of either PUC or CHIPS, system temperature and pressure are maintained within the set operational envelope by integrated control electronics located within the pressure-controlled vapor plenum which supplies the thrusters with propellant. With the exception of the CHIPS ACS capabilities, cold gas operation is similar between the two systems: a thrust (pressure) set point and burn time is selected and the system fires, exhausting gas through the selected thruster. The approach to warm gas operation is where they differ.

In warm-fire mode, PUC uses a microcavity discharge (MCD) plasma to increase propellant temperature ahead of an optimized single micro-nozzle, resulting in a significant performance boost. The MCD plasma is driven by the PUC PPU subsystem, a tightly-integrated AC power supply sharing the same space with the control electronics.

The TRL 8 PUC is 0.25U + “hockey puck” (8.9 x 8.9 x 6.7 cm), designed for a 1U - 3U spacecraft bus. Looking at Figure 6, the “hockey puck” is the protruding portion of the unit that contains the MCD thruster. PUC has demonstrated cold gas performance with 46 seconds of I_{sp} at 3.5 mN thrust and 6.0W input power; warm gas performance is 72 seconds I_{sp} at 5.4 mN thrust and 15.0W input power. PUC has a volumetric impulse of 514.5 (Ns)/U. To date, eight (8) units have been delivered to AFRL Edwards and are awaiting launch opportunities.

CHIPS warm-fire propulsion uses a micro-resistojet to superheat the propellant ahead of an optimized micro-nozzle, significantly improving performance over cold gas operation. The CHIPS micro-resistojet PPU is a DC power supply that can draw directly from the spacecraft bus or from an optional integrated battery pack.

The 1U+ baseline CHIPS is designed for a spacecraft bus up to 6U in size. The differences with respect to PUC are that CHIPS offers higher performance, four ACS cold gas thrusters, and an optional battery pack. With a warm-fire input power of 30W, CHIPS has demonstrated warm gas performance of 82 sec I_{sp} , 30 mN thrust, and a volumetric impulse of 526.2 (Ns)/U. Cold gas performance is 47 seconds I_{sp} at 19 mN thrust with an input power of

about 8W. To reduce the spacecraft power draw, the unit can draw power from an optional battery pack.

Both PUC and CHIPS are engineered to be readily customized to fit form factor and performance requirements for a wide array of spacecraft bus configurations and mission profiles.

2.3.2 CubeSat Ambipolar Thruster (CAT)

The CubeSat Ambipolar Thruster (CAT) is a helicon plasma thruster (see Figure 7) that was originally developed by the Plasmadynamics & Electric Propulsion Laboratory at the University of Michigan, but is now being matured by Phase Four, Inc [[16], [17]]. This electric propulsion system uses radio-frequency (RF) power to create helicon waves, a low frequency electromagnetic wave, inside of a quartz liner to ionize a gas propellant into a plasma. The plasma is then expanded through a diverging magnetic nozzle by way of an 800 G magnetic field that is generated by annular neodymium magnets. The strength of the generated magnetic field also has the advantage of preventing the plasma from making contact with the walls of the quartz liner, reducing the chances of liner erosion during operation. Another advantage of this system is that there is no net current which needs to be neutralized, removing the need for a cathode. The CAT main components are all housed in a titanium Faraday shield that is used to protect the spacecraft bus from the generated RF energy. CAT is designed to operate with several choices of gas propellants, such as xenon, iodine, argon, water vapor, and krypton.

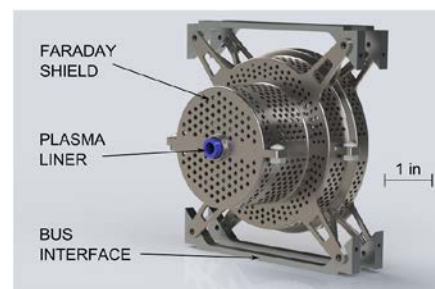


Figure 7. CAD Model of CAT [19]

The CAT prototype is currently 1.2U in size and originally designed for a 3U spacecraft bus. Phase Four is working to get this down to 1U. The goal of this design is to provide greater delta-v and thrust over other electric propulsion systems. Figure 8 shows the prototype CAT during ground testing. Based on the Phase Four specification sheet for CAT [17], xenon is projected to produce thrust/ I_{sp} /delta-v of 2.77 mN/498 sec/219 m/s with an input power of 50W and a propellant mass of 0.23 kg. CAT has a projected volumetric impulse of 936.7 (Ns)/U. This system has demonstrated performance with xenon ~1.0 mN/800 sec with an input power of ≤ 5 W and a propellant mass of 500 g. Phase Four is currently

conducting studies to better understand plume constituents and CAT system refinements, as well as optimizing the theoretical max thrust-to-power ratio of 120mN/kW.

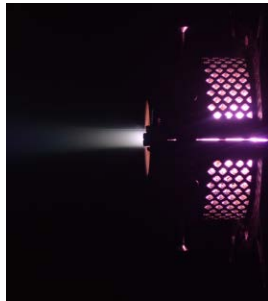
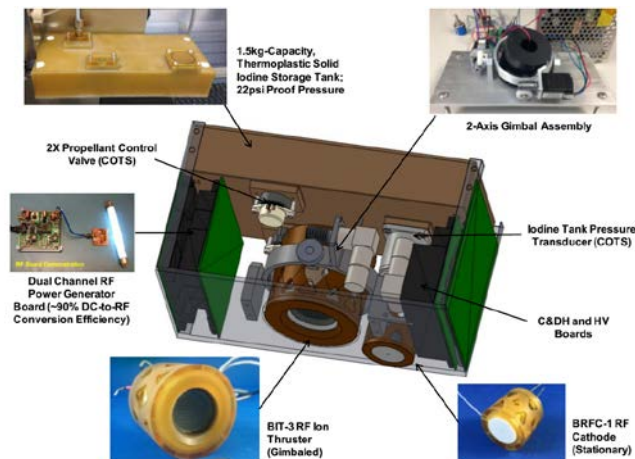


Figure 8. CAT prototype during firing using xenon gas [19]

2.3.3 BIT-3

The Busek Co, Inc, has developed an iodine-fueled RF ion propulsion system known as the BIT-3, which was designed to propel 6U spacecraft [20]. This system is due to fly on two missions that are due to launch in September 2018: Lunar IceCube (Morehead State University) and LunaH-Map (Arizona State University). This thruster produces ions using RF power from gaseous propellant to create a plasma. The positively charged plasma is then accelerated with an electrostatic grid. In order to prevent charging of surrounding surfaces, an RF cathode is used to generate electrons that neutralize the plume.



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Figure 9. Busek's BIT-3 RF Ion Propulsion System [20]

The BIT-3 system is about 2U (18.0 x 8.8 x 10.2 cm) in size, and has a wet mass of 3 kg, of which 1.5 kg is solid iodine. Figure 9 shows the individual components that comprise the BIT-3 propulsion system. The iodine is stored in a plastic storage tank. The advantage of using iodine, unlike other liquid or gaseous propellants, is the fact that it can be stored in the spacecraft as a solid block, with negligible vapor pressure. To use iodine as a propellant, the block is heated (between 60-100°C) within its storage tank to generate a gas (similar to dry ice) that powers the

system. The thermoplastic storage tank has been pressure proof tested to 22 psia and tested to its maximum expected operating pressure (MEOP) of 14.7 psia. This low pressure system answers one of the Range Safety concerns about pressurized systems. The BRFC-1 RF cathode is a modified version of Busek's 1 cm RF ion thruster, BIT-1, in that its polarity was reversed such that it would generate electrons instead of ions. This was done by adding an ion collector to the screen grid and other modifications in order for it to operate more like a conventional RF cathode. The PPU manages various power and control tasks for the propulsion system, which include, but not limited to, controlling the feed system, serving as the spacecraft avionics interface, converting spacecraft input power into RF power, and high voltage generation to accelerate ions. In order to optimize the limited volume, the PPU was split into 2 parts: the RF power generator (left of BIT-3 thruster in Figure 9), and the command & data handling (C&DH) and high voltage power generator (right of BIT-3 thruster in Figure 9).



Figure 10. BIT-3 RF Ion Thruster during ground testing: Xenon (left), Iodine (right) [20]

The BIT-3 is a 3 cm RF ion thruster. The current BIT-3 engineering model (EM) is designed to consume 65W, based on the available 6U spacecraft bus power. It is being developed using xenon gas because that is the baseline propellant for this type of thruster. However, the flight version will be fueled with iodine. During ground testing with xenon (shown in Figure 10, right side), the BIT-3 demonstrated $0.65-1.15 \pm 10\%$ mN of thrust with in input power that ranged between 28W to 45W. The corresponding I_{sp} was shown to be $1200-2100 \pm 10\%$ sec. The BIT-3 propulsion system has a volumetric impulse of 15,451 (Ns)/U.

2.3.4 Scalable ion Electro spray Propulsion System (S-iEPS)

The Scalable ion Electro spray Propulsion System (S-iEPS) is a MEMS electric propulsion system that was initially developed by the Massachusetts Institute of Technology [21], and is now being matured by Accion Systems, Inc. The S-iEPS is comprised of three main components: the thruster cells, the propellant tank, and the power electronics. Each system is made up of 2 to n thruster cells (an individual thruster cell is shown in Figure 11, left) depending on the thrust level required. The thruster

cells are comprised of porous glass emitter chips each supporting nearly 500 ion emission sites. These cells are mounted onto propellant tanks of custom volume. The S-iEPS uses an ionic liquid (a molten salt) as its propellant. The ionic liquid is composed of positive and negative ions with very low vapor pressure. The porous emitter chip is the heart of the S-iEPS. An electric potential is applied between the propellant and the emitter grid. The potential difference overcomes the propellant surface tension and electrostatically pulls the propellant through the emitter tips to form what is known as a Taylor cone (see Figure 12). Near the emitter grid, the potential is high enough to extract and accelerate positive and negative ions. Since both ion polarities are extracted, there is no need for ion neutralization.

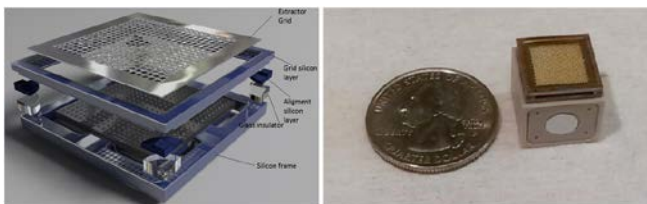


Figure 11. Exploded view of S-iEPS MEMS thruster cell (left); Single S-iEPS Thruster (right) [21]

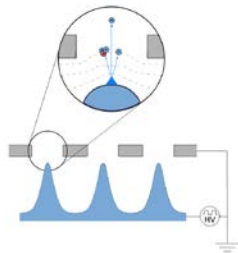


Figure 12. Taylor Cone formation and ion extraction [21]

A single cell mounted on its tank is 1.44 x 1.44 x 1.44 cm in size and has a wet mass of less than 3.5g (see Figure 11, right). In one product configuration, an array of eight S-iEPS thrusters is packaged on a PPU, as shown in Figure 13, and fired at alternating polarities to allow for charge neutralization. Ground testing of this system consisted of firing eight (8) of S-iEPS thrusters. The system demonstrated an average thrust of $74 \pm 3.7 \mu\text{N}$. Taking all losses into consideration, the I_{sp} is about 1000 sec. Its volumetric impulse is about 260.6 (Ns)/U.



Figure 13. S-iEPS 8 thruster array mounted on PPU [21]

3 Conclusions

SmallSats have been employed as a low cost method to conduct low-earth orbit research and technology demonstrations. As SmallSats become more capable, there is a growing need for them to have propulsion. NASA/GSFC has conducted a survey study of SmallSat propulsion systems to determine their availability and level of development. There are many systems that are under development that can fit a variety of needs based on mission requirements. The challenge that the SmallSat propulsion community faces is to develop systems that find the right balance between reliability, performance, and complexity in order to keep system cost down. Some of those systems working to strike that balance were discussed here. Table 2 summarizes the as-demonstrated performance of the discussed systems.

Table 1. SmallSat Propulsion System Performance Summary

	Thrust (mN)	I_{sp} (sec)	Power (W)	Mass (kg)		Vol. Imp. (Ns/U)
				Dry	Prop	
<i>CubeSat MEMS (NanoSpace)</i>	<4	110	<2.5	0.25	0.05	133.3
<i>3D Printed (UT-Austin)</i>	40	35	1.5	0.29	0.09	146.5
<i>AMAC (Busek)</i>	425	220	20	1.2	0.29	565.0
<i>PUC (CU Aerospace)†</i>	5.4	72	15	0.45	0.27	514.5
<i>CHIPS (CU Aerospace)†</i>	30	82	30	0.50	0.70	526.2
<i>CAT (Phase Four)‡</i>	1.00	800	5	1.00	0.50	936.7
<i>BIT-3 (Busek)‡</i>	1.15	2100	45	1.5	1.5	15,451
<i>S-iEPS (MIT)</i>	0.074	1000	1.5	0.018	0.010	260.6

† Warm gas performance

‡ Xenon Propellant

Acknowledgment

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