Chapter Contents

Chapter Glossary

(ABS)	Acrylonitrile Butadiene Styrene
(AC)	Alternating Current
(ACE)	Apollo Constellation Engine
(ACO)	Announcement for Collaborative Opportunity
(ADN)	Ammonium Dinitramide
(AFRL)	Air Force Research Laboratory
(AFRL)	U.S. Air Force Research Laboratory
(AOCS)	Attitude and Orbit Control System
(AR)	Aerojet Rocketdyne
(ARC)	Ames Research Center
(CMNT)	Colloid MicroNewton Thrusters
(CNAPS)	Canadian Nanosatellite Advanced Propulsion System
(CNES)	French National Center for Space Studies
(CPOD)	CubeSat Proximity Operations Demonstration
(CUA)	CU Aerospace LLC
(DFMR)	Design for Minimum Risk
(DRM)	Design Reference Mission
(DSSP)	Digital Solid State Propulsion LLC
(EMC)	Electromagnetic Compatibility
(EMI)	Electromagnetic Interference
(EP)	Electric Propulsion
(EPSS)	Enabling Propulsion System for Small Satellites
(ESA)	European Space Agency
(ESPs)	Electrically Controlled Solid Propellant
(FASTSAT)	Fast, Affordable, Science and Technology Satellite
(FEEP)	Field Emission Electric Propulsion
(FPPT)	Fiber-Fed Pulsed Plasma Thruster
(GEO)	Geostationary Equatorial Orbit
(GIT)	Gridded-ion Thrusters
(GOCE)	Gravity Field and Steady-State Ocean Circulation Explorer
(GOX)	Gaseous Oxygen
(GPIM)	Green Propellant Infusion Mission

(GPS)	Global Positioning System
(GRC)	Glenn Research Center
(GSFC)	Goddard Space Flight Center
(HAN)	Hydroxylammonium Nitrate
(HET)	Hall-effect Thruster
(HTP)	High Test Peroxide
(HTPB)	Hydroxyl-terminated Polybutadiene
(IPS)	Integrated Propulsion System
(ISS)	International Space Station
(JHU ERG)	Johns Hopkins University Energetics Research Group
(JPL)	Jet Propulsion Laboratory
(LFPS)	Lunar Flashlight Propulsion System
(LISA)	Laser Interferometer Space Antenna
(MAPS)	Modular Architecture Propulsion System
(MarCO)	Mars Cube One
(MCD)	Micro-cavity Discharge
(MEMS)	Microelectromechanical System
(MEO)	Medium Earth Orbit
(MMH)	Monomethyl Hydrazine
(MPUC)	Monopropellant Propulsion Unit for CubeSats
(MSFC)	Marshall Space Flight Center
(MVP)	Monofilament Vaporization Propulsion
(N_2O)	Nitrous Oxide
(NEA)	Near-Earth Asteroid
(NODIS)	NASA Online Directives Information System
(NSTT)	Nanosat Terminator Tape
(OTS)	Orbital Transfer System
(OTV)	Orbital Transfer Vehicle
(PacSci EMC)	Pacific Scientific Energetic Materials Company
(PBM)	Plasma Brake Module
(PMD)	Propellant Management Device
(PMDs)	Propellant Management Devices
(PMI)	Progress toward Mission Infusion
(PMMA)	Polymethyl Methacrylate

(PPT)	Pulsed Plasma Thrusters
(PPU)	Power Processing Unit
(PTD)	Pathfinder Technology Demonstration
(PTFE)	Polytetrafluoroethylene
(PUC)	Propulsion Unit for CubeSats
(ROMBUS)	Rapid Orbital Mobility Bus
(SAA)	Space Act Agreement
(SBIR)	Small Business Innovative Research
(SCAPE)	Self Contained Atmospheric Protective Ensemble
(SEP)	Solar Electric Propulsion
(SMAP)	Soil Moisture Active Passive
(SMART-1)	Small Missions for Advanced Research in Technology
(SME)	Subject Matter Experts
(SSTL)	Surrey Satellite Technology Ltd.
(SSTP)	Small Spacecraft Technologies Program
(TCMs)	Trajectory Correction Maneuvers
(TMA)	Technology Maturity Assessment
(TRL)	Technology Readiness Level
(UTIAS)	University of Toronto Institute for Aerospace Research
(VAT)	Vacuum arc thrusters
(VENuS)	Vegetation and Environment monitoring on a New Microsatellite
(WFF)	Wallops Flight Facility

4.0 In-Space Propulsion

4.1 Introduction

In-space propulsion devices for small spacecraft are rapidly increasing in number and variety. Although a mix of small spacecraft propulsion devices have established flight heritage, the market for new propulsion products continues to prove dynamic and evolving. In some instances, systems and components with past flight heritage are being reconsidered to meet the needs of smaller spacecraft. This approach minimizes new product development risk and time to market by creating devices similar to those with existing spaceflight heritage, albeit accounting for small spacecraft volume, mass, power, safety and cost considerations. Such incremental advancement benefits from existing spaceflight data, physics-based models, and customer acceptance of the heritage technologies, which eases mission infusion. In other instances, novel technologies are being conceived specifically for small spacecraft. These technologies often use innovative approaches to propulsion system design, manufacturing, and integration. While the development of novel technologies typically carries a higher risk and slower time to market, these new technologies strive to offer small spacecraft a level of propulsive capability not easily matched through the miniaturization of heritage technologies. Such novel devices are often highly integrated and optimized to minimize the use of a small spacecraft's limited resources, lower the product cost, and simplify integration. Regardless of the development approach, the extensive investments by commercial industry, academia, and government to develop new propulsion products for small spacecraft suggests long-term growth in the availability of propulsion devices with increasingly diverse capabilities.

In the near-term, the surge in public and private investments in small spacecraft propulsion technologies, in combination with the immaturity of the overall small spacecraft market, has resulted in an abundance of confusing, unverified, sometimes conflicting, and otherwise incomplete technical literature. Furthermore, the rush by many device developers to secure market share has resulted in some confusion surrounding the true readiness of these devices for mission infusion. As third parties independently verify device performance, and end-users demonstrate these new devices in their target environments, the true maturity, capability, and flight readiness of these devices will become evident. In the meantime, this report will attempt to reduce confusion by compiling a list of publicly described small spacecraft propulsion devices, identifying publicly available technical literature for further consideration, recognizing missions of potential significance, and organizing the data to improve comprehension for both neophytes and subject matter experts.

This chapter avoids a direct technology maturity assessment (TMA) based on the NASA Technology Readiness Level (TRL) scale, recognizing insufficient in-depth technical insight into current propulsion devices to perform such an assessment accurately and uniformly. An accurate TRL assessment requires a high degree of technical knowledge on a subject device as well as an understanding of the intended spacecraft bus and target environment. While the authors strongly encourage a TMA that is well-supported with technical data prior to infusing technologies into programs, the authors believe TRLs are most accurately assessed within the context of a program's unique requirements. Rather than attempting to assess TRL in the absence of sufficient data, this chapter introduces a novel classification system that simply recognizes Progress toward Mission Infusion (PMI) as an early indicator of the efficacy of the manufacturers' approach to system maturation and mission infusion. PMI should not be confused with TRL as PMI does not directly assess technology maturity. However, PMI may prove insightful in early trade studies. The PMI classification system used herein is described in detail in Section 4.4.2.

4.1.1 Document Organization

This chapter organizes the state-of-the-art in small spacecraft propulsion into the following categories:

1. In-Space Chemical Propulsion (4.6.1)
2. In-Space Electric Propulsion (4.6.2)
3. In-Space Propellant-less Propulsion (4.6.3)

Each of these categories is further subdivided by the prevailing technology types. The subsections organize data on each prevailing technology type as follows:

a. Technology Description
b. Key Integration and Operational Considerations
c. Current & Planned Missions
d. Summary Table of Devices
e. Notable Advancements

The organizational approach introduces newcomers to each technology, presents technology-specific integration and operation concerns for the reader's awareness, highlights recent or planned missions that may raise the TRL of specific devices, and finally tabulates procurable devices of each technology. Some sections further include an incomplete list of highlights of notable advancements. While the key integration and operational considerations are not all-inclusive, they provide initial insights that may influence propulsion system selection. In the cases where a device has significant flight heritage, this chapter reviews only select missions.

4.2 Public Data Sources and Disclaimers

This chapter is a survey of small spacecraft propulsion technologies as discussed in open literature and does not endeavor to be an original source. As such, this chapter only considers literature found in the public domain to identify and classify devices. Commonly used sources for public data include manufacturer datasheets, press releases, conference papers, journal papers, public filings with government agencies, and news articles.

This chapter summarizes device performance, capabilities, and flight history, as presented in publicly available literature. Data not appropriate for public dissemination, such as proprietary, export controlled, or otherwise restricted data, are not considered. As such, actual device maturity and flight history may be more extensive than what is documented herein. Device manufacturers should be consulted for the most up-to-date and relevant data before performing a TMA.

This chapter's primary data source is literature produced by device manufacturers. Unless otherwise published, do not assume independent verification of device performance and capabilities. Performance and capabilities described may be speculative or otherwise based on limited data.

The information described below is not intended to be exhaustive but to provide a general overview of current state-of-the-art technologies and their development status. It should be noted that technology maturity designations may vary with change to payload, mission requirements, reliability considerations, and/or the environment in which performance was demonstrated. Readers are highly encouraged to reach out to companies for further information regarding the performance and maturity of the described technology. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA.

Suggestions or corrections to this document should be submitted to the NASA Small Spacecraft Virtual Institute Agency-SmallSat-Institute@mail.nasa.gov for consideration prior to the publication of future issues. When submitting comments, please cite appropriate publicly accessible references. Private correspondence is not considered an adequate reference.

4.3 Definitions

- *Device* refers to a component, subsystem, or system, depending on the context.
- *Technology* refers to a broad category of devices or intangible materials, such as processes.

4.4 Technology Maturity

4.4.1 Application of the TRL Scale to Small Spacecraft Propulsion Systems

NASA has a well-established guideline for performing TMAs, described in detail in the NASA Systems Engineering Handbook (1). A TMA determines a device's technological maturity, which is usually communicated according to the NASA TRL scale. The TRL scale is defined in NASA Procedural Requirements (NPR) 7123 (2). The NASA Systems Engineering Handbook and NPR 7123 can be accessed through the NASA Online Directives Information System (NODIS) library. Assessment of TRLs for components, systems, or software allows for coherent communication between technologists, program managers, and other stakeholders regarding the maturity of a technology. Furthermore, TRL is a valuable tool to communicate the potential risk associated with the infusion of technologies into programs. For TRLs to be applied across all technology categories, the NASA TRL definitions are written broadly and rely on subject matter experts (SME) in each discipline to interpret appropriately.

Recently, U.S. Government propulsion SMEs suggested an interpretation of the TRL scale specifically for micro-propulsion. The Micro-Propulsion Panel of the JANNAF Spacecraft Propulsion Subcommittee in 2019 published the *JANNAF Guidelines for the Application of Technology Readiness Levels (TRLs) to Micro-Propulsion Systems* (3). This guideline suggests an interpretation of TRL for micro-propulsion and reflects both NASA and DOD definitions for TRL. The JANNAF panel consisted of participants from the Air Force Research Laboratory (AFRL), Glenn Research Center (GRC), Jet Propulsion Laboratory (JPL), and Goddard Space Flight Center (GSFC). The panel further received feedback from the non-Government propulsion community. While this JANNAF guideline focuses on micro-propulsion (e.g., for CubeSats), the guideline still has relevance to rigorously assessing TRLs for the more general category of small spacecraft in-space propulsion. By establishing a common interpretation of TRL for small spacecraft propulsion, a more coherent and consistent communication of technology maturity can occur between small spacecraft propulsion providers and stakeholders. The JANNAF guideline is open to unlimited distribution and may be requested from the Johns Hopkins University Energetics Research Group (JHU ERG). Ensure the use of the latest JANNAF guideline, as the guideline is anticipated to evolve with further community input.

A fundamental limitation of the JANNAF guideline for TRL assessment, and TMA in general, is an assumption of in-depth technical knowledge of the subject device. In the absence of detailed technical knowledge, especially in a broad technology survey as presented herein, a TMA may be conducted inaccurately or inconsistently. Furthermore, assessment of TRL assumes an understanding of the end-user application. The same device may be concluded to be at different TRLs for infusion into different missions. For example, a device may be assessed at a high TRL for application to low-cost small spacecraft in low-Earth orbits, while assessed at a lower TRL for application to geosynchronous communication satellites or NASA interplanetary missions due to different mission requirements. Differences in TRL assessment based on the operating environment may result from considerations such as thermal environment, mechanical loads, mission duration, or radiation exposure. Propulsion-specific variances between missions might include propellant type, total propellant throughput, throttle set-points, burn durations, and the total number of on/off cycles. As such, an accurate TRL assessment not only requires an in-depth technical understanding of a device's development history, including specifics on past flight-qualification activities, but also an understanding of mission-specific environments and interfaces. The challenge of assessing an accurate TRL in a broad technology survey poses a significant burden for data collection, organization, and presentation. Such activities are better suited for the programs seeking to infuse new technologies into their missions.

Given the rapid evolution of small spacecraft propulsion technologies and the variety of mission environments, as well as generally limited device technical details in open literature, the propulsion chapter implements a novel system to classify technical maturity according to Progress toward Mission Infusion (PMI). This novel classification system is not intended to replace TRL but is a complementary tool to provide initial insight into device maturity when it is not feasible to accurately and consistently apply the TRL scale. This novel classification system is discussed in detail below.

Readers using this survey are strongly encouraged to perform more in-depth technical research on candidate devices based on the most up-to-date information available, as well as to assess risk within the context of their specific mission(s). A thoughtful TMA based on the examination of detailed technical data through consultation with device manufactures can reduce program risk and in so doing increase the likelihood of program success. This survey is not intended to replace the readers' own due diligence. Rather, this survey and PMI seek to provide early insights that may assist in propulsion system down-select to a number of devices where an in-depth TMA becomes feasible.

4.4.2 Progress Toward Mission Infusion (PMI)

Rather than directly assessing a device's technical maturity via TRL, propulsion devices described herein are classified according to evidence of progress toward mission infusion. This is a novel classification system first introduced in this survey. Assessing the PMI of devices in a broad survey, where minimal technical insight is available, may assist with down-selecting propulsion devices early in mission development. Once a handful of devices are selected for further consideration, an in-depth technical examination of the selected devices may be more practical to conduct a TMA and rigorously assess TRL. The PMI classification system sorts devices into one of four broad technology development categories: Concept, In-Development, Engineering-to-Flight, and Flight-Demonstrated. The following sections describe the PMI classification system in-detail. Furthermore, figure 4.1 summarizes the PMI classifications.

Concept, 'C'

The *Concept* classification reflects devices in an early stage of development, characterized by feasibility studies and the demonstration of fundamental physics. Concept devices typically align with the NASA TRL range of 1 to 3. At a minimum, these devices are established as scientifically feasible, perhaps through a review of relevant literature and/or analytical analysis. These devices may even include experimental verification that supports the validity of the underlying physics. These devices may even include notional designs. While Concept devices are generally not reviewed herein, particularly promising Concept devices will be classified in tables with a 'C'.

In-Development, 'D'

The *In-Development* classification reflects the bulk of devices being actively matured and covered in this survey, where only a modest number of devices may progress to regular spaceflight. In-Development devices typically align with the NASA TRL range of 4 to 5. While In-Development devices may have specific applications attributed by their developers, no selection for a specific mission has been publicly announced. In the absence of a specific mission, device development activities typically lack rigorous system requirements and a process for independent requirement validation. Furthermore, qualification activities conducted in the absence of a specific mission typically require a delta-qualification to address mission-specific requirements. At a minimum, In-Development devices are low-fidelity devices that have been operated in an appropriate environment to demonstrate basic functionality and support prediction of the device's ultimate capabilities. They may even be medium- or high-fidelity devices operated in a simulated final environment, but lacking a specific mission pull to define requirements and a qualification program. They may even be medium- or high-fidelity devices operated in a spaceflight demonstration but lacking sufficient fidelity or demonstrated capability to reflect the anticipated final product. These devices are typically described as a technology push, rather than a mission pull. In-Development devices will be classified in tables with a 'D'.

Engineering-to-Flight, 'E'

The *Engineering-to-Flight* classification reflects devices with a publicly announced spaceflight opportunity. This classification does not necessarily imply greater technical maturity than the In-Development classification, but it does assume the propulsion device developer is receiving mission-specific requirements to guide development and qualification activities. Furthermore, the Engineering-to-Flight classification assumes a mission team performed due diligence in the selection of a propulsion device, and the mission team is performing regular activities to validate that the propulsion system requirements are met. Thus, while the PMI classification system does not directly assess technical maturity, there is an underlying assumption of independent validation of mission-specific requirements, where a mission team does directly consider technical maturity in the process of device selection and mission infusion. Engineering-to-Flight devices typically align with the NASA TRL range of 5 to 6. At a minimum, these are medium-fidelity devices that have been operated in a simulated final environment and demonstrate key capabilities relative to the requirements of a specific mission. These devices may even be actively undergoing or have completed a flight qualification program. These devices may even include a spaceflight, but in which key capabilities failed to be demonstrated or further engineering is required. These devices may even include a previously successful spaceflight, but the devices are now being applied in new environments or platforms that necessitate design modifications and/or delta-qualification. These devices must have a specific mission pull documented in open literature. A design reference mission (DRM) may be considered in place of a specific mission pull, given detailed documentation in open literature, which includes a description of the DRM, well-defined propulsion system requirements, maturation consistent with the DRM requirements, and evidence of future mission need. Engineering-to-Flight devices will be classified in tables with an 'E'.

Flight-Demonstrated, 'F'

The *Flight-Demonstrated* classification reflects devices where a successful technology demonstration or genuine mission has been conducted and described in open literature. Flight-Demonstrated devices typically align with the NASA TRL range of 7 to 9. These devices are high-fidelity components or systems (in fit, form, and function) that have been operated in the target in-space environment (i.e., low-Earth orbit, GEO, deep space) on an appropriate platform, where all key capabilities were successfully demonstrated. These devices may even be final products, which have completed genuine missions (not simply flight demonstrations). These devices may even be in repeat production and routine use for several missions. The devices must be described in open literature as successfully demonstrating key capabilities in the target environment to be considered Flight-Demonstrated. If a device has flown, but the outcome is not publicly known, the classification will remain Engineering-to-Flight. Flight-Demonstrated devices will be classified in tables with an ‘F’.

Concept, 'C'

- At minimum, an idea has been established as scientifically feasible.
- May even include experimental verification of the underlying physics.
- May even include notional device designs.
- Approximately aligns to NASA TRL 1-3

In-Development, 'D'

- At minimum, a low-fidelity device that has been operated in an appropriate environment to demonstrate the basic functionality and predict the ultimate capabilities.
- May even be a medium- or high-fidelity device operated in a simulated final environment, but the device lacks a specific mission pull to define requirements and a qualification program.
- May even be a medium- or high-fidelity device operated in a flight demonstration, but the device lacks sufficient fidelity or demonstrated capability to reflect the anticipated final product.
- Approximately aligns to NASA TRL 4-5

Engineering-to-Flight, 'E'

- At minimum, a medium-fidelity device that has been operated in a simulated final environment and demonstrates key capabilities relative to the requirements of a specific mission.
- May even include a qualification program in-progress or completed.
- May even include a spaceflight, but the device fails to demonstrate key capabilities.
- May even include a successful spaceflight, but the device is now being applied in a new environment or platform, necessitating a delta-qualification.
- A specific mission opportunity must be identified in open literature.
- Approximately aligns to NASA TRL 5-6

Flight-Demonstrated, 'F'

- At minimum, a high-fidelity component or system (fit, form, and function) that has been operated in the intended in-space environment (e.g., LEO, GEO, deep space) on an appropriate platform, where key capabilities have been successfully demonstrated.
- May even be a final product that has completed a mission (not strictly a technology demonstration).
- May even be a product in repeat production and routine use for a number of missions.
- A successful spaceflight must be identified and the outcome described in open literature.
- Approximately aligns to NASA TRL 7-9

Figure 4.1: Progress toward mission infusion (PMI) device classifications. Credit: NASA.

4.5 Overview of In-Space Propulsion Technology Types

In-space small spacecraft propulsion technologies are generally categorized as (i) chemical, (ii) electric, or (iii) propellant-less. This chapter surveys propulsion devices within each technology category. Additionally, liquid-propellant acquisition and management devices are reviewed as an important component of in-space propulsion systems. Although other key subsystems have not yet been reviewed, such as small spacecraft propulsion power processing units, they may be included in future updates of this publication. Table 4-1 lists the in-space propulsion technologies reviewed. Figure 4.2 graphically illustrates the range of thrust and specific impulse for these small spacecraft propulsion devices. The thrust and specific impulse ranges provided in table 4-1 and figure 4.1 only summarize the performance of small spacecraft devices covered in this survey and may not reflect the broader capability of the technologies beyond small spacecraft or the limits of what is physically possible with further technology advancement.

Chemical systems have enabled in-space maneuvering since the onset of the space age, proving highly capable and reliable. These include hydrazine-based systems, other mono- or bipropellant systems, hybrids, cold/warm gas systems, and solid propellants. Typically, these systems are sought when high thrust or rapid maneuvers are required. As such, chemical systems continue to be the in-space propulsion technology of choice when their total impulse capability is sufficient to meet mission requirements.

On the other hand, the application of electric propulsion devices has been historically far more limited. While electric propulsion can provide an order of magnitude greater total impulse than chemical systems, research and development costs have typically eclipsed that of comparable chemical systems. Furthermore, electric propulsion generally provides thrust-to-power levels below 75 mN/kW. Thus, a small spacecraft capable of delivering 500 W to an electric propulsion system may generate no more than 38 mN of thrust. Therefore, while the total impulse capability of electric propulsion is generally considerable, these systems may need to operate for hundreds or thousands of hours, compared to the seconds or minutes that chemical systems necessitate for a similar impulse. That said, the high total impulse and low thrust requirements of specific applications, such as station keeping, have maintained steady investment in electric propulsion over the decades. Only in recent years has the mission pull for electric propulsion reached a tipping point where electric propulsion may overtake chemical for specific in-space applications. Electric propulsion system types considered herein include electrothermal, electrospray, gridded ion, Hall-effect, pulsed plasma and vacuum arc, and ambipolar.

Propellant-less propulsion technologies such as solar sails, electrodynamic tethers, and aerodynamic drag devices have long been investigated, but they have yet to move beyond small-scale demonstrations. However, growing needs such as orbital debris removal may offer compelling applications in the near future.

Some notable categories are not covered in this survey, such as nuclear in-space propulsion technologies. While substantial investment continues in such areas for deep space science and human exploration, such technologies are generally at lower TRL and typically aim to propel spacecraft substantially larger than the 180 kg limit covered by this report.

Whenever possible, this survey considers complete propulsion systems, which are composed of thrusters, feed systems, pressurization systems, propellant management and storage, and power processing units, but not the electrical power supply. However, for some categories, components (e.g., thruster heads) are mentioned without consideration of the remaining subsystems necessary for their implementation. Depending on the device's intended platform (i.e., NanoSat, MicroSat, SmallSat), the propulsion system may be either highly integrated or distributed within the spacecraft. As such, it is logical to describe highly integrated propulsion units at the system level, whereas components of distributed propulsion systems may be logically treated at the sub-system level, where components from a multitude of manufacturers may be mixed-and-matched to create a unique mission-appropriate propulsion solution.

Table 4-1: Summary of Propulsion Technologies Surveyed

Technology	Thrust Range	Specific Impulse Range [sec]
4.6.1 CHEMICAL PROPULSION TECHNOLOGIES		
Hydrazine Monopropellant	0.25 – 25 N	200 – 285
Alternative Mono- and Bipropellants	10 mN – 120 N	160 – 310
Hybrids	1 – 230 N	215 – 300
Cold / Warm Gas	10 μN – 3 N	30 – 110
Solid Motors	0.3 – 260 N	180 – 280
Propellant Management Devices	N/A	N/A
4.6.2 ELECTRIC PROPULSION TECHNOLOGIES		
Electrothermal	0.5 – 100 mN	50 – 185
Electrosprays	10 μN – 1 mN	225 – 5,000
Gridded Ion	0.1 – 20 mN	1,000 – 3,500
Hall-Effect	1 – 60 mN	800 – 1,950
Pulsed Plasma and Vacuum Arc Thrusters	1 – 600 μN	500 – 2,400
Ambipolar	0.25 – 10 mN	400 – 1,400
4.6.3 PROPELLANTLESS PROPULSION TECHNOLOGIES		
Solar Sails	TBD	N/A
Electrodynamic Tethers	TBD	N/A
Aerodynamic Drag	TBD	N/A

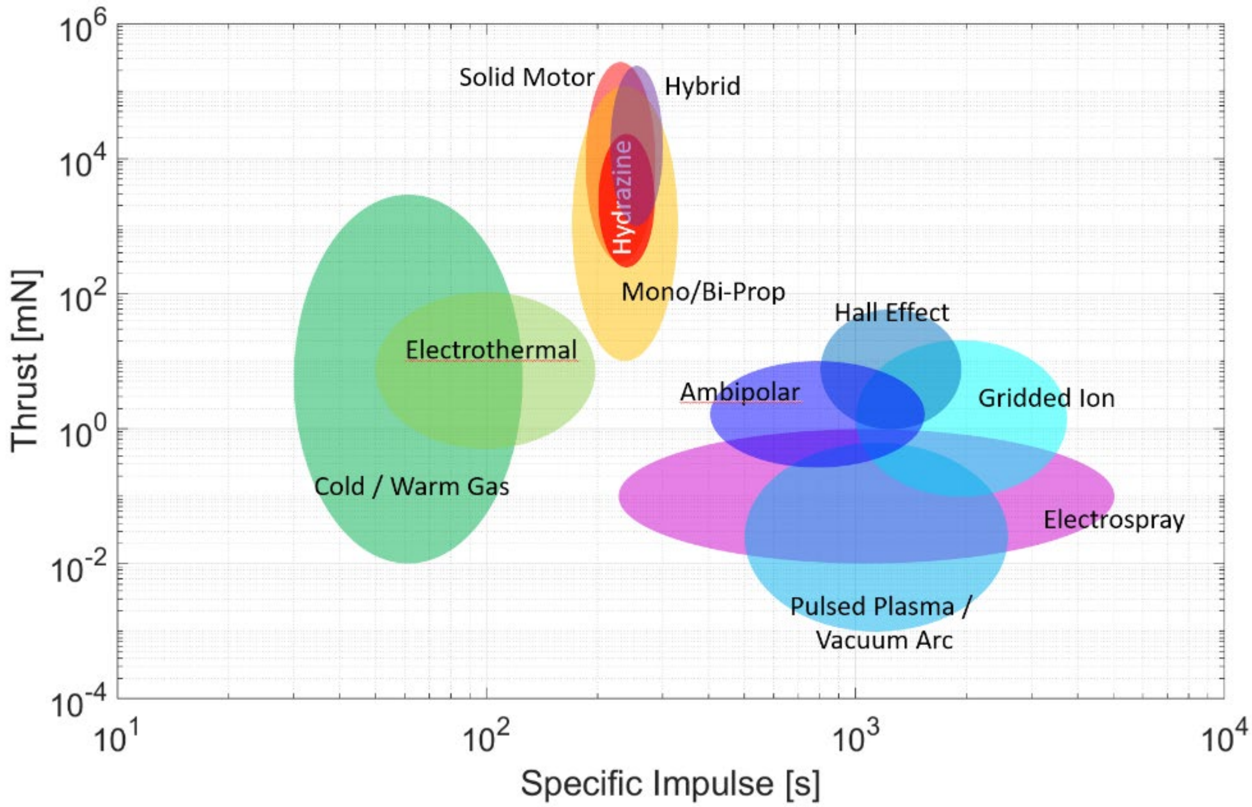


Figure 4.2: Typical small spacecraft in-space propulsion trade space (thrust vs. specific impulse). Credit: NASA.

4.6 State-of-the-Art in Small Spacecraft Propulsion

4.6.1 In-Space Chemical Propulsion

Chemical propulsion systems are designed to satisfy high-thrust impulsive maneuvers. They offer lower specific impulse compared to their electric propulsion counterparts but have significantly higher thrust to power ratios.

Hydrazine Monopropellant

a. Technology Description

Hydrazine monopropellant systems use catalyst structures (such as S-405 granular catalyst) to decompose hydrazine or a derivative such as monomethyl hydrazine (MMH) to produce hot gases. Hydrazine thrusters and systems have been in extensive use since the 1960's. The low mass and volume of a significant number of larger spacecraft hydrazine propulsion systems also allows those systems to be suitable for some small spacecraft buses. Thrusters that perform small corrective maneuvers and attitude control in large spacecraft may be large enough to perform high-thrust primary maneuvers for small spacecraft and can act as the main propulsion system. Hydrazine specific impulses are achievable in the 200 – 235 second range for 1-N class or larger thrusters.

b. Key Integration and Operational Considerations

Since hydrazine has been in use for some time, its traits are well defined (4). However, hydrazine (and its derivatives) is corrosive, toxic, and potentially carcinogenic. Its vapor pressure requires the use of Self Contained Atmospheric Protective Ensemble (SCAPE) suits. This overhead increases the ground processing flow of spacecraft and may impose undesirable constraints on secondary spacecraft. Hydrazine propulsion systems typically incorporate redundant serial valves to prevent spills or leaks.

Because hydrazine systems are so widely used for larger spacecraft, a robust ecosystem of components and experience exists, and hydrazine propulsion systems are frequently custom designed for specific applications using available components. Typically, they also have the advantage of being qualified for multiple cold starts, which may be beneficial for power-limited buses if the lifespan of the mission is short.

c. Current & Planned Missions

ArianeGroup has developed a 1-N class hydrazine thruster that has extensive flight heritage, including use on the ALSAT-2 small spacecraft (5) (6).

Aerojet Rocketdyne has leveraged existing designs with flight heritage from large spacecraft that may be applicable to small buses, such as the MR-103 thruster used on New Horizons for attitude control (7). Other Aerojet Rocketdyne thrusters potentially applicable to small spacecraft include the MR-111 and the MR-106 (8). These thrusters have successfully flown on several missions.

Moog-ISP has extensive experience in the design and testing of propulsion systems and components for large spacecraft. These may also apply to smaller platforms, as some of their flight-proven thrusters are lightweight and have moderate power requirements. The MONARC-5 thrusters flew on NASA JPL's Soil Moisture Active Passive (SMAP) spacecraft in 2015 and provided 4.5 N of steady state thrust. Other thrusters potentially applicable to small spacecraft buses include the MONARC-1 and the MONARC-22 series (9).

d. Summary Table of Devices

See table 4-2 for current state-of-the-art hydrazine monopropellant devices applicable to small spacecraft.

e. Notable Advances

Aerojet Rocketdyne (AR) has developed a new class of green hydrazine propellant blends providing the low vapor-toxicity and high density- I_{SP} of ionic liquids while retaining the low reaction and preheat temperatures of traditional hydrazine. This makes it possible to increase both safety and performance while still using conventional nickel-alloy catalytic thrusters. In testing completed to date, green hydrazine blends have demonstrated long-term thermal stability/storability, low shock/impact sensitivity, and good operational stability. Furthermore, they have demonstrated a 100-fold reduction in vapor pressure/toxicity and a similar low-temperature start capability as compared to pure hydrazine (10). Ongoing development efforts at Aerojet Rocketdyne, NASA GSFC, and the Aerospace Corporation are on track to advance the technical maturity of green hydrazine blends to flight-ready status by the end of 2022.

Alternative Monopropellants and Bipropellants

a. Technology Description

Alternative propellant technologies are increasingly being developed and adopted as a replacement for hydrazine, due to hydrazine's handling and toxicity concerns. These include replacements such as the emerging 'green' ionic liquids, and more conventional propellants like hydrogen peroxide or electrolyzed water (bi-propellant hydrogen/oxygen).

The primary ionic liquid propellants with flight heritage or upcoming spaceflight plans are LMP-103S, which is a blend of Ammonium Dinitramide (ADN), and AF-M315E (now: referred to as "ASCENT"), a blend of Hydroxylammonium Nitrate (HAN). Other alternative propellants, such as hydrogen peroxide, are also available and have been in use for many years. Some of these may be lower performing than hydrazine but offer more benign operating environments and require more readily available and lower-cost materials.

This group of ionic liquid propellants, commonly referred as 'green propellants', have reduced toxicity due in large part to the lower danger of component chemicals and significantly reduced vapor pressures as compared to hydrazine. The 'green' affiliation also results in potentially removing Self-Contained Atmospheric Protective Ensemble (SCAPE) suit requirements. The elimination of the SCAPE suit requirement reduces operational oversight by safety and emergency personnel, and potentially reduces secondary payload requirements. The 'green propellants' LMP-103S and ASCENT are ideally used as direct replacements for hydrazine. Usually, these green propellants are decomposed and combusted over a catalytic structure akin to hydrazine systems, which often requires pre-heating to decompose the propellant. However, they both require high catalyst pre-heating and have higher combustion temperatures. Therefore, these blends are not 'drop-in' replacements.

Green propellants also provide higher specific impulse performance than the current state-of-the-art hydrazine monopropellant thrusters for similar thrust classes and have a higher density-specific impulse achieving improved mass fractions. Additionally, these propellants have lower minimum storage temperatures which may be beneficial in power-limited spacecraft, as tank and line heater requirements are lower.

While other alternative propellant choices (such as electrolyzed water or hydrogen peroxide) are not 'green' propellants like the ionic liquids, they may also be considered within the 'green' category. They exhibit more benign characteristics relative to hydrazine and are therefore an alternative option to hydrazine. These alternative propellants are seen as particularly useful for small satellite applications, where the comparatively low mission cost can provide a mutual benefit in technology advancement and development while providing needed mission capabilities (11).

b. Key Integration and Operational Considerations

Air Force Range Safety AFSPCMAN91-710 (12) requirements state that if a propellant is less prone to external leakage, which is often seen with the ionic liquid 'green' propellant systems due to higher viscosity of the propellant, then the hazardous classification is reduced. External hydrazine leakage is considered "catastrophic," whereas using ionic liquid green propellants reduces the hazard severity classification to "critical" and possibly "marginal" per MIL-STD-882E (Standard Practice for System Safety) (13). A classification of "critical" or less only requires two-seals to inhibit external leakage, meaning no additional latch valves or other isolation devices are required in the feed system (13). While these propellants are not safe for consumption, they have been shown to be less toxic compared to hydrazine. This is primarily due to green propellants having lower vapor pressures, being less flammable, and producing more benign constituent product gases (such as water vapor, hydrogen, and carbon dioxide) when combusted.

Fueling spacecraft with green propellants, generally permitted as a parallel operation, may require a smaller exclusionary zone, allowing for accelerated launch readiness operations (14). These green propellants are also generally less likely to exothermically decompose at room temperature due to higher ignition thresholds. Therefore, they require fewer inhibit requirements, fewer valve seats for power, and less stringent temperature storage requirements. The reduced hazard associated with some of these propellants may enable projects to take a Design for Minimum Risk (DFMR) approach to address some propulsion system safety concerns, but only with the support of associated range and payload safety entities.

While there are thrusters that are relatively mature (PMI E/F), incorporating them into propulsion systems is challenging, and the maturity of stand-alone propulsion systems has lagged the pace of component development. Historically, research and development efforts, like Small Business Innovative Research (SBIR) efforts, have focus on component development, and not the entire system. Efforts are now being made to focus on the development of system solutions. Most of these non-toxic propellants are still in some phase of development. Additionally, data on the propellants is also widely restricted. Therefore, a comprehensive, public, peer-reviewed databased of compatible materials does not currently exist, and would-be system developers using these propellants may have difficultly accessing such data to guide their efforts.

Other ‘green propellants’ such as Hydrogen Peroxide, High Test Peroxide (HTP), and HTP/Alcohol bipropellants also have their own unique handling considerations. For instance, HTP is a strong oxidizer and can exothermically decompose rapidly if improperly stored or handled. Hydrogen Peroxide, however, has been used as a rocket propellant for many decades, and there is a lot of information on safe handling, materials selection, and best practices. Electrolyzed water is another propellant option, wherein water is decomposed into hydrogen and oxygen and combusted as a traditional bi-propellant thruster. However, generating and managing the power required to electrolyze the water in a compact spacecraft presents its own unique challenges. Yet it does provide a safe-to-launch system with very benign constituents.

c. Current & Planned Missions

Planet Labs launched a constellation of Earth observing satellites, called SkySat. These satellites are approximately 120 kg, and incorporate the Bradford-ECAPS HPGP system, a LMP-103S based system shown in figure 4.3. SkySats 3 – 21 include a propulsion system using four 1-N thrusters. As of August 2020, 13 SkySat satellites with the Bradford ECAPS propulsion system have been launched and are fully operational (15).

Figure 4.3: ECAPS HPGP thruster. Credit: Bradford ECAPS.

Astroscale has built and launched a highly maneuverable ‘chaser’ SmallSat called ELSA-d. ELSA-d is a twin SmallSat mission which will demonstrate key rendezvous and docking technologies, and proximity operational concepts in readiness for providing a commercial deorbit service (16). ELSA-d has an LMP-103S using eight 1-N Bradford ECAPS thrusters to provide both re-orbiting and de-orbiting capability. ELSA-d launched in March 2021.

The JPL-led Lunar Flashlight mission manifested for Artemis I will map the lunar south pole for volatiles. The mission will demonstrate several technological firsts, including being one of the first CubeSats to reach the Moon, the first planetary CubeSat mission to use green propulsion, and the first mission to use lasers to look for water ice (17).

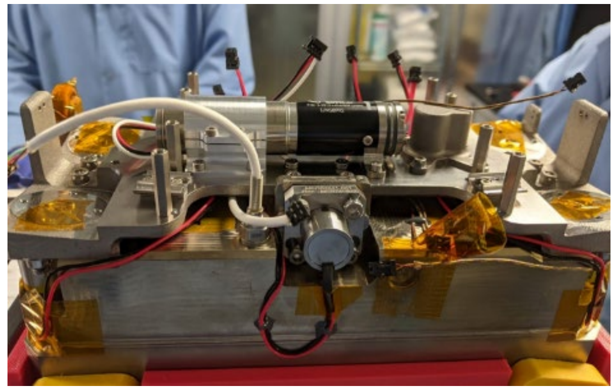


Figure 4.4: Lunar Flashlight Propulsion System. Credit NASA.

NASA Marshall Space Flight Center (MSFC) led the development of the Lunar Flashlight Propulsion System (LFPS), a self-contained unit that can deliver over 3000 N-s of total impulse for this mission (figure 4.4). The LFPS is a pump-fed system that has four 100-mN ASCENT thrusters (figure 4.5), built by Plasma Processes LLC., and a micro-pump built by Flight Works Inc. The LFPS employs a propellant management device (PMD) and newly developed isolation and thruster micro-solenoid valves and a micro-fill/drain valve. The LFPS system was delivered to JPL in May 2021 and is now being integrated into the LF Spacecraft. The LFPS structural design and electronics controller development was performed by the Georgia Institute of Technology (Atlanta, GA).

Figure 4.5: Plasma Processes LLC 100mN thruster. Credit: NASA MSFC.

Another ASCENT-based propulsion system flew as a technology demonstration on the NASA Green Propellant Infusion Mission (GPIM) launched in July 2019 (18). This small spacecraft was designed to test the performance of this propulsion technology in space by using five 1-N class thrusters (figure 4.6) for small attitude control maneuvers (19). Aerojet completed a hot-fire test of the GR-1 version in 2014 and further tests in 2015. Initial plans to incorporate the GR-22 thruster (22-N class) on the GPIM mission were deferred in mid-2015 to allow for more development and testing of the GR-22. As a result, the GPIM mission only carried and demonstrated five GR-1 units when launched (20).

Figure 4.6: GR1 thruster. Credit: Aerojet.

CisLunar Explorer is part of a NASA Centennial Challenge mission planned for Artemis I. The CisLunar Explorer's concept consists of a pair of spacecrafts on a mission to orbit the Moon. The two spacecraft are mated together as a "6U"-sized box, and after deployment from the launch vehicle, they will split apart, and each give their initial rotation in the process of decoupling. The spacecraft will then enter and attempt to maintain lunar orbit. The propulsion system for this mission is a water electrolysis system developed by Cornell University (21).

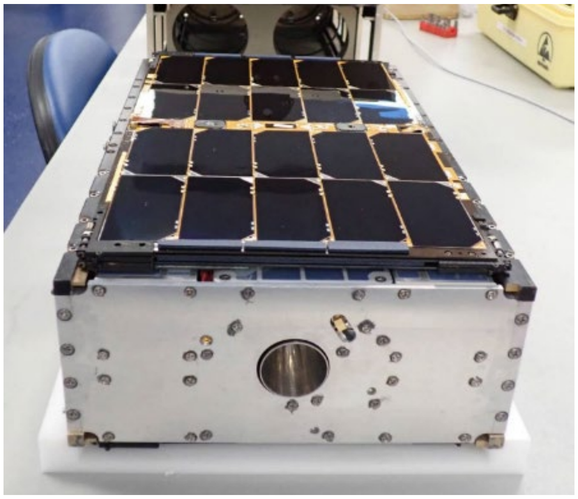
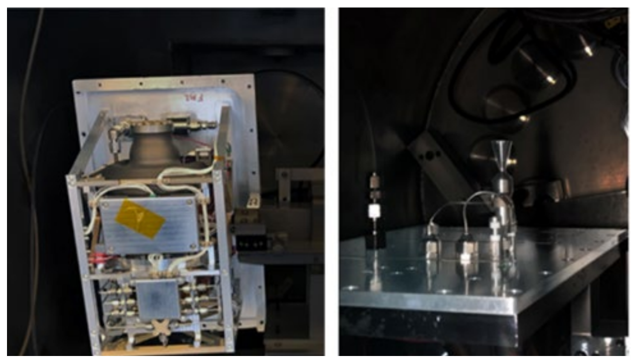


Figure 4.7: PTD-1 HYDROS-C Credit: NASA.

NASA's Small Spacecraft Technology (SST) program at Ames Research Center (ARC) have launched the first Pathfinder Technology Demonstration (PTD) mission in January 2021 (22) (23) (24). PTD-1 (figure 4.7) tested the HYDROS-C water electrolysis propulsion system, developed by Tethers Unlimited Inc. With a volume less than 2.4U, the HYDROS-C uses water as propellant. In-orbit, water was electrolyzed into oxygen and hydrogen, then combusted like a traditional bi-propellant thruster. Limited performance data is being evaluated, but is expected to provide an average thrust of 1.2 N with an I_{SP} of 310 s. The system requires 10 – 15 minutes of recharge time between pulses. A variant of the HYDROS-C system is the HYDROS-M system, which is intended to be sized for MicroSats.

Benchmark Space Systems has delivered its first three Halcyon propulsion systems (figure 4.8), which launched on June 30, 2021 on SpaceX's Transporter-2 rideshare mission. The Halcyon system combines an HTP thruster developed by legacy Tesseract with Benchmark's fluid handling and flight controller subsystems to provide a thrust of 1-N with an I_{SP} between 155-175s. It also uses proprietary on-demand pressurization technology, permitting it to be launch at low pressure (25).



Credit: Benchmark Space Systems.

VACCO Industries has built and delivered the first of its Integrated Propulsion System (IPS), which was designed to deliver 12,000 N-sec total impulse. The IPS (figure 4.9) features four 1-N LMP-103S Bradford ECAPS thrusters, using the LFP-103S propellant.

NanoAvionics has developed a non-toxic mono-propellant propulsion system called Enabling Propulsion System for Small Satellites (EPSS), which was demonstrated on LituanicaSAT-2, a 3U CubeSat, to correct orientation and attitude, avoid collisions, and extend orbital lifetime. It uses an ADN-blend as propellant, achieves 213 s of specific impulse, and provide 400 N-s of total impulse. LituanicaSAT-2 was launched in June 2017 and successfully separated from the primary payload (Cartosat-2) as part of the European QB50 initiative. According to product literature, multiple missions have since launched, with the latest being in April 2019 (26).

Figure 4.9: VACCO Industries IPS Credit: VACCO Industries.

Dawn Aerospace (formerly: Hyperion) has developed a 0.5 N bi-propellant system that consists of a single thruster with a gimbal to provide thrust in two axes. The 1U configuration (figure 4.10) provides 850 N-s of total impulse with a minimum impulse bit of 35 mN-s (27).

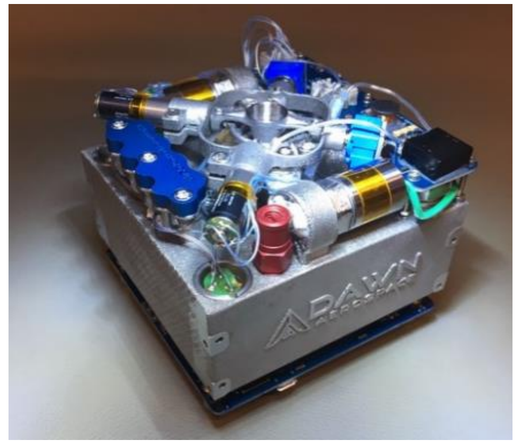


Figure 4.10: PM200.
Credit: Dawn Aerospace.

The Busek BGT-X5 CubeSat propulsion system (figure 4.11) occupies 1U+ volume and weighs ~1.5kg wet, including ~300g of ASCENT (AF-M315E) propellant. The module can deliver a maximum of 660 N-s of total impulse at I_{SP} of 225 s. Key components include a 0.5-N micro thruster, a bellows-based propellant tank, a custom micro solenoid valve, a secondary flow inhibit for range safety, integrated controller, and a Post-Launch Pressurization System (PLPS), enabling the system to be completely unpressurized at launch (28). The BGT-X5 system is currently undergoing qualification testing, with a projected launch date of Q2 CY2022.

Figure 4.11: BGT-X5 System
Credit: Busek Space Systems.

Rocket Lab's Electron rocket has a liquid propellant kick-stage that uses a cold-gas RCS. The Rocket Lab Kick Stage, powered by the Curie engine, was designed to deliver small satellites to precise orbits before deorbiting itself to leave no part of the rocket in space. The kick stage was flown and tested onboard the "Still Testing" flight that was successfully launched on January 21, 2018. With the new kick stage Rocket Lab can execute multiple burns to place numerous payloads into different orbits. The kick stage is designed for use on the Electron launch vehicle with a payload capacity of up to 150 kg, and will be used to disperse CubeSat constellations fast and accurately, enabling satellite data to be received and used soon after launch (29) (30).

d. Summary Table of Devices

See table 4-3 for current state-of-the-art other mono- and bipropellant devices applicable to small spacecraft.

e. Notable Advances

Aerojet Rocketdyne continues development of its GR-M1 Advanced Green Monopropellant CubeSat Thruster. It employs the same advanced techniques, ultra-high-temperature catalyst, and refractory metal manufacture as the GPIM GR-1 thruster, but on a nanosat scale (31). To partially mitigate thermal management challenges exacerbated at the miniature scale, the GR-M1 is designed to operate on a reduced-flame-temperature variant of the AF M315E propellant containing 10% added water. The heat transfer to surrounding spacecraft structure both during heat up and operation are comparable to conventional hydrazine thrusters.

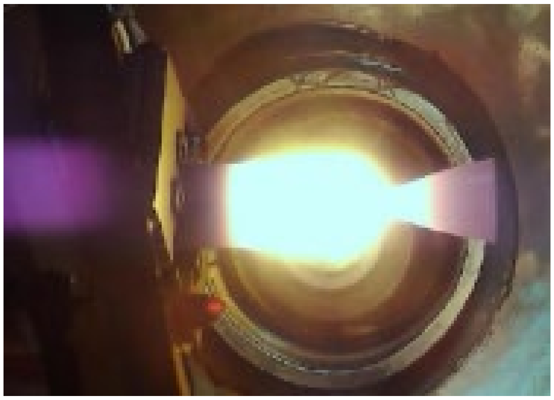


Figure 4.12: PP3616-A 5N ASENT Thruster. Credit: Plasma Processes.

Plasma Processes LLC is maturing a 1N and 5N ASCENT thruster (figure 4.12), intended for SmallSat application (32). Both offerings are built using the same materials and processes as those used on the 100mN thrusters delivered for the Lunar Flashlight Mission. Additionally, Plasma Processes intends to engineer a short-life, lower cost version of the 5N thruster. The prototype thruster accumulated > 1kg throughput and over 500 seconds before the end of the NASA Phase I SBIR. The Phase II effort will continue the development of the 5N thruster.

CU Aerospace LLC (CUA) is developing the Monopropellant Propulsion Unit for CubeSats (MPUC) system. The monopropellant is an H_2O_2-ethanol blend denoted as CMP-X. Tests on a thrust stand achieved a thrust level of >100 mN at I_{SP} >180 s with an average input power of ~3 W, for hot fire runs typically spanning >10 minutes. 1.5U and 2U systems are in development with an estimated 1550 N-s and 2450 N-s total impulse, respectively. The ~950°C flame temperature allows the thrust chamber to use non-refractory construction materials. CMP-X has low toxicity and was subjected to a scaled UN Series 1 detonation test series and demonstrated no detonation propagation when confined under a charge of high explosive. No special measures are anticipated for its long-term storage and permission for transport is expected. A NASA Phase II SBIR effort is currently underway.

Hybrids

a. Technology Description

Hybrid propulsion is a mix of both solid and liquid/gas forms of propulsion. In a hybrid rocket, the fuel is typically a solid grain and the oxidizer (often gaseous oxygen) is stored separately. The rocket is then ignited by injecting the oxidizer into the solid motor and igniting it with a spark or torch system. Since combustion can only occur while the oxidizer is flowing, these systems can readily be started or shutdown by controlling the flow of oxidizer.

b. Key Integration and Operational Considerations

Because there is no oxidizer pre-mixed with the solid motor, these systems are inherently safer from a handling standpoint than solid motor systems, as the risk of pre-mature ignition is greatly reduced. They offer the best of both worlds of solids (storability & handling) and liquids (restart & throttling). Yet they do have drawbacks, as combustion efficiency tends to not be as high as in either system, and regression rate control and fuel residuals tend to be more problematic in these designs.

c. Current & Planned Missions

An arc-ignition 'green' CubeSat hybrid thruster system prototype was developed at Utah State University. This system is fueled by 3-D printed acrylonitrile butadiene styrene (ABS) plastic known for its electrical breakdown properties. Initially, high-pressure gaseous oxygen (GOX) was to be used as the oxidizer. However, for the sake of the technology demonstration and after safety considerations by NASA Wallops High Pressure Safety Management Team, it was concluded the oxidizer needed to contain 60% nitrogen and only 40% oxygen. On March 25, 2018, the system was successfully tested aboard a sounding rocket launched from NASA Wallops Flight Facility

(WFF) into space and the motor was successfully re-fired 5 times. During the tests, 8 N of thrust and a specific impulse of 215 s were achieved as predicted (34) (35). The Space Dynamics Lab has miniaturized this technology to be better suited for CubeSat applications (0.25 - 0.5 N). A qualification unit is currently in development for the miniaturized system.

d. Summary Table of Devices

See table 4-4 for current state-of-the-art hybrid devices applicable to small spacecraft.

e. Notable Advances

Utah State University has an ongoing test series with Nytrox, a blend of nitrous oxide and oxygen, and ABS. This testing is focused on a 25-50 N system for a 12U sized vehicle and will cumulate in vacuum testing of the motor at MSFC in 2021. Investigation into different nozzle materials for low erosion in long duration burns is a key concern (36) (37).

JPL has pursed development of a hybrid propulsion system for 12U CubeSat and a 100 kg SmallSat. Testing included regression rate characterization of clear and black Poly (Methyl MethAcrylate) fuels with GOX to be included in propulsion system sizing. Later vacuum testing included an improvement of the ignition system to a laser operated system that eliminates the need for a separate ignition fuel gas to be carried (38).

NASA ARC developed a polymethyl methacrylate (PMMA) and nitrous oxide hybrid system that had ethylene and nitrous oxide thrusters. The ethylene and nitrous oxide also function as the hybrid ignition source. The hybrid system had a demonstrated efficiency of 91% and calculated I_{SP} of 247 sec, making it competitive with current small satellite propulsion systems (39) (40).

Aerospace Corporation and Penn State University developed an “Advanced Hybrid Rocket Motor Propulsion Unit for CubeSats (PUC)”. The design used additive manufacturing techniques for the carbon filled polyamide structure including the nitrous oxide tank and a paraffin grain within an acrylic shell, with acrylic diaphragms 3-D printed in-situ in the grain to aid in the performance of the grain. This design fits in a 1U space, for a 3 to 6U spacecraft (41).

Parabilis Space Technologies has done development work on two small satellite propulsion systems. Rapid Orbital Mobility Bus (ROMBUS) is a hybrid rocket-based system with nitrous oxide as the oxidizer and the attitude control system/reaction control system thruster propellant. It provides high-impulse thrust for satellite translational maneuvers which can be used for initial orbit insertion, rapid orbit rephasing, threat/collision avoidance, and targeted re-entry at the satellite’s mission end of life (42). Nano Orbital Transfer System (OTS) is a Hydroxyl-terminated polybutadiene (HTPB) and nitrous oxide (N_2O) hybrid system, with N_2O based ACS thrusters. Nano OTS leverages Parabilis’ proven hybrid engine and small satellite technologies for low-cost, high performance maneuvers using non-toxic green propellants. The OTS has a modular design, enabling rapid and low-cost configuration of stages to accommodate 3U size NanoSats up to >50 kg MicroSat size vehicles.

Cold Gas / Warm Gas

a. Technology Description

Cold gas systems are relatively simple systems that provide limited spacecraft propulsion and are one of the most mature technologies for small spacecraft. Thrust is produced by the expulsion of a propellant which can be stored as a pressurized gas or a saturated liquid. Warm gas systems, in which the propellant is heated but there is still no chemical reaction, have been used to increase thrust and specific impulse. Warm gas systems use the same basic principle as cold gas systems and have higher performance at the cost of added power requirements to heat the propellant.

Electrothermal systems, a type of warm-gas system where the gas is electrically heated in the thruster body or nozzle, are described in more detail in the Electric Propulsion section.

b. Key Integration and Operational Considerations

Cold gas thrusters are often attractive and suitable for small buses due to their relatively low cost and complexity. Many cold gas thrusters use inert, non-toxic propellants, which are an advantage for secondary payloads that must adopt "do no harm" approaches to primary payloads. Such systems are well suited to provide attitude control since they technically provide very low minimum impulse bits for precise maneuvering. However, the low specific impulse of these systems limits them from providing large orbit correction maneuvers. Recently, new designs have improved the capability of these systems for nanosatellite buses such as 3U CubeSats.

c. Missions

A cold gas thruster developed by Marotta flew on the NASA ST-5 mission (launch mass 55 kg) for fine attitude adjustment maneuvers. It incorporates electronic drivers that can operate the thruster at a power of less than 1 W. It has less than 5 ms of response time and it uses gaseous nitrogen as propellant (43).

The Micro-Electromechanical-based PICOSAT Satellite Inspector, or MEPSI, built by the Aerospace Corporation flew aboard STS-113 and STS-116. The spacecraft included both target and imaging/inspector vehicles connected via tether. The two vehicles were 4 x 4 x 5 in^3 in volume, each, and had five cold-gas thrusters, producing ~20 mN. The MEPSI propulsion system was produced using stereo-lithography. It was suited as a propulsion research unit for PicoSats (44).

Figure 4.13: SSTL butane propulsion system. Credit: Surrey Satellite Technology, Ltd.

Surrey Satellite Technology Ltd. (SSTL) has included a butane propulsion system in several small spacecraft missions for a wide range of applications in low-Earth orbit and Medium Earth Orbit (MEO). In this system, propellant tanks are combined with a resistojet thruster and operation is controlled by a series of solenoid valves (figure 4.13). It requires power to heat the thruster and improve the specific impulse performance with respect to the cold gas mode. (45) (46).

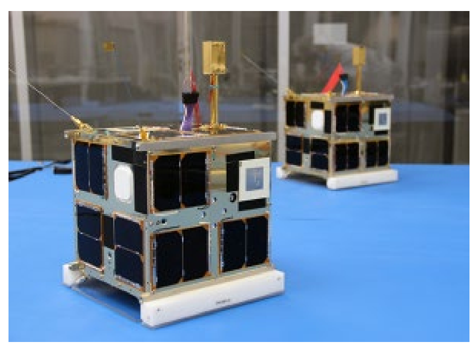


Figure 4.14: CanX-4 and CanX-5 formation flying nanosatellites with CNAPS propulsion systems. Credit: UTIAS SFL.

In June 2014, Space Flight Laboratory at University of Toronto Institute for Aerospace Research (UTIAS) launched two 15 kg small spacecraft to demonstrate formation flying. The Canadian Nanosatellite Advanced Propulsion System (CNAPS), shown in figure 4.14, consisted of four thrusters fueled with liquid sulfur hexafluoride. This non-toxic propellant was selected because it has high vapor pressure and density, which is important for making a self-pressurizing system (47). This propulsion module is a novel version of the previous NanoPS that flew on the CanX-2 mission in 2008 (48).

Another flight-demonstrated propulsion system was flown on the POPSAT-HIP1 CubeSat mission (launched June 2014), which was developed by Microspace Rapid Pte Ltd in Singapore. It consisted of a total of eight micro-nozzles that provided control for three rotation axes with a single axis thrust for translational applications. The total delta-v has been estimated from laboratory data to be between 2.25 and 3.05 ms^{-1}. Each thruster has 1 mN of nominal thrust by using argon propellant. An electromagnetic microvalve with a very short opening time of 1 m-s operates each thruster (49).

Two related butane propulsion systems have been developed by GomSpace: the NanoProp 3U and NanoProp 6U. Both use proportional thrust control of four nozzles to control spacecraft attitude while providing delta-v. The 6U configuration was flown on GOMX-4B in 2018 as a formation flight demonstration (50) (51).

An ACS cold gas propulsion system using R-236fa was produced and tested by Lightsey Space Research for the NASA ARC BioSentinel mission, a 6U CubeSat scheduled to launch on Artemis I. This propulsion system uses a 3D-printed propellant tank in order to reduce part count and use the available volume more efficiently (52) (53).

A complete cold gas propulsion system has been developed for CubeSats with a microelectromechanical system (MEMS) (figure 4.15) that provides accurate thrust control with four butane propellant thrusters. While thrust is controlled in a closed loop system with magnitude readings, each thruster can provide a thrust magnitude from zero to full capacity (1 mN) with 5-μN resolution. The dry mass of the system is 0.220 kg and average power consumption is 2 W during operation (54). This system is based on flight-proven technology flown on larger spacecraft (PRISMA mission, launched in 2010). The MEMS cold gas system was included on the bus of the TW-1 CubeSat, launched in September 2015 (55).

Figure 4.15: NanoSpace MEMS cold gas system. Credit: GomSpace.

The CubeSat Proximity Operations Demonstration (CPOD) is a mission led by Tyvak Nano-Satellite Systems (56). It incorporates a cold gas propulsion system built by VACCO Industries that provides up to 186 N-s of total impulse. This module operates at a steady state power of 5 W and delivers 40-s of specific impulse while the nominal thrust is 10 mN (57). It uses self-pressurizing refrigerant R236fa propellant to fire a total of eight thrusters distributed in pairs at the four corners of the module. It has gone through extensive testing at the US Air Force Research Lab. Endurance tests consisted of more than 70,000 firings.

JPL is supporting the InSight mission, launched in March 2018, which incorporated two identical CubeSats as part of the Mars Cube One (MarCO) technology demonstration. These spacecrafts performed five trajectory correction maneuvers (TCMs) during the mission to Mars. The CubeSats included an integrated propulsion system developed by VACCO Industries, which contained four thrusters for attitude control and another four for TCMs. The module uses cold gas refrigerant R-236FA as propellant, produces 75 N-s of total impulse, and weighs 3.49 kg (58) (59).

NEA Scout is a NASA MSFC mission that is going to be launched as part of Artemis I. For its main propulsion system, NEA Scout will deploy a sail of 80 m^2 with 0.0601-mm s^{-2} of characteristic acceleration and will be steered by active mass translation via a VACCO cold gas MiPS (R236FA propellant). This module is approximately 2U in volume and will use six 23-mN thrusters to provide 30 m s^{-1} of delta-v (60) (61).

The ThrustMe I2T5 iodine cold gas module, figure 4.16, is the first iodine propulsion system to be spaceflight tested, on-board of the Xiaoxiang 1-08 satellite. The demonstration was the result of a joint collaboration of ThrustMe and Spacety (62) (63). An I2T5 module is anticipated to launch in 2021 on the Robusta-3A satellite, developed by CSUM. The Robusta-3A will carry various scientific payloads related to meteorology and technology demonstration (64).

Figure 4.16: I2T5 Iodine Cold Gas Module. Credit: ThrustMe.

d. Summary Table of Devices

See table 4-5 for current state-of-the-art cold gas / warm gas devices applicable to small spacecraft.

Solid Motors

a. Technology Description

Solid rocket technology is typically used for impulsive maneuvers such as orbit insertion or quick de-orbiting. Due to the solid propellant, they achieve moderate specific impulses and high thrust magnitudes that are compact and suitable for small buses.

There are some electrically controlled solid thrusters that operate in the milli-newton (mN) range. These are restartable, have steering capabilities, and are suitable for small spacecraft applications, unlike larger spacecraft systems that provide too much acceleration.

b. Key Integration and Operational Considerations

Thrust vector control systems can be coupled with existing solid rocket motors to provide controllable high delta-v in relatively short time. While some solid motors are restartable, in general solid motors are often considered a single-burn event system. To achieve multiple burns, the system must be either electrically restartable (aka electric solid propellants), or several small units must be matrixed into an array configuration. Because electrically controlled solid propellant (ESPs) are electrically ignited, they are safer than traditional solid energetic propellants.

c. Current & Planned Missions

A flight campaign tested the ability of thrust vector control systems coupled with solid motors to effectively control the attitude of small rocket vehicles. Some of these tests were performed by using state-of-the-art solid rocket motors such as the ISP 30 developed by Industrial Solid Propulsion and the STAR 4G by ATK (now: Northrop Grumman) (66).

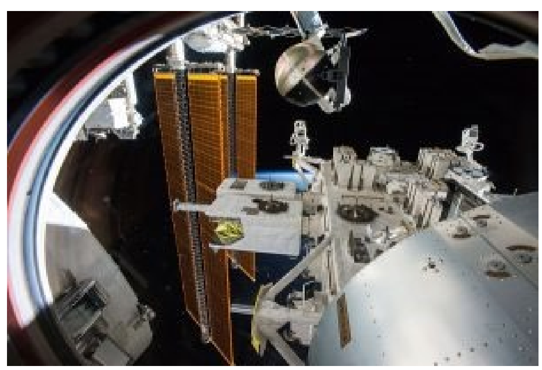


Figure 4.17: SpinSat at the ISS. Credit: NASA.

SpinSat, a 57 kg spacecraft, was deployed from the International Space Station (ISS) in 2014 and incorporated a set of first-generation solid motors, the CubeSat Agile Propulsion System (figure 4.17), which was part of the attitude control system developed by Digital Solid State Propulsion LLC (DSSP). The system was based on a set of ESP thrusters that consist of two coaxial electrodes separated by a thin layer of electric solid propellant. This material is highly energetic but non-pyrotechnic and is only ignited if an electric current is applied. The thrust duration can be better controlled and allows for better burn control, and the lack of moving parts make the system suitable for small spacecraft (67).

The Modular Architecture Propulsion System (MAPS) by Pacific Scientific Energetic Materials Company (PacSci EMC) Propulsion array (figure 4.18) has a 10-plus year in-orbit lifespan. The MAPS system provides three axes capability to control such areas as attitude control, deorbit, drag makeup, and plane and attitude changes with a delta-v greater than 50 m s^{-1}. The capability of MAPS "plug-and-play" bolt-on design and clean-burning propellant array is scalable and can be custom fit for a range of interfaces. MAPS was flown aboard the PACSCISAT (68) (69).

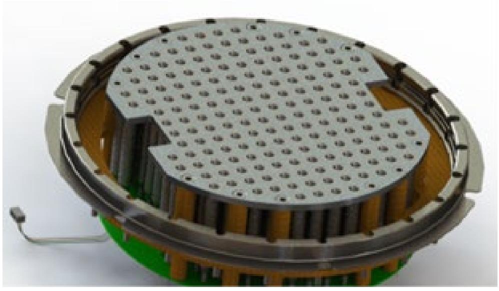


Figure 4.18: PacSci EMC MAPS sealed solid propellant rocket motor array. Credit: PacSci.

d. Summary Table of Devices

See table 4-6 for current state-of-the-art solid motor devices applicable to small spacecraft.

Propellant Management Devices

a. Technology Description

While not specifically a propulsion type, propellant management devices (PMDs) are frequently used in larger liquid propulsion systems to deliver propellant to the thruster units. As small spacecraft start looking toward more complex propulsion systems, PMDs will undoubtedly play an integral part. Historically, small spacecraft have used bellows or membrane tanks to ensure propellant delivery and expulsion. However, there is the potential to incorporate PMD structures into additively manufactured tanks and propulsion systems, permitting much more conformal structures to be created for small spacecraft missions while still meeting mission performance targets. Hence, PMDs are a critical part of any in-space propulsion system that doesn't use bellows or membrane type tanks, and they are briefly covered here for awareness. A more detailed treatment and explanation can be found in the literature, and a good overview is provided by Hartwig (70).

b. Key Integration and Operational Considerations

The purpose of PMDs is to separate liquid and vapor phases within the propellant storage tank upstream of the thruster, and to transfer vapor-free propellant in any gravitational or thermal environment. PMDs have flight heritage with all classical storable systems, have been flown once with LMP-103S, have no flight heritage with cryogenic propellants, and have been implemented in electric propulsion systems. Multiple PMDs are often required to meet the demands of a particular mission, whether using storable or cryogenic propellants. A comprehensive, up-to-date list of the types of PMDs, as well as missions employing PMDs, is available in Hartwig (70).

c. Current & Planned Missions

The Lunar Flashlight Propulsion System will employ a PMD sponge and ribbon vane. The sponge was additively manufactured, while the ribbon vane was cut from sheet metal and bent to conform to the required dimensions. Surface tension properties, a necessary parameter for PMD sizing, have been determined for the ASCENT propellant by Kent State University, funded and managed by NASA. The design and modelling effort were a joint effort between MSFC and GRC.

d. Summary Table of Devices

No summary table is included for propellant management devices in this report edition.

e. Notable Advances

Northrop Grumman has made advances in development of SmallSat and CubeSat scale diaphragm propellant tanks using materials with known compatibility with hydrazine and some green monopropellant fuels (71). Some effort has been made to demonstrate the application of additive manufacturing to produce tank shells.

4.6.2 In-Space Electric Propulsion

In-space electric propulsion (EP) is any in-space propulsion technology wherein a propellant is accelerated through the conversion of electrical energy into kinetic energy. The electrical energy source powering in-space EP is historically solar, therefore these technologies are often referred to as solar electric propulsion (SEP), although other energy sources are conceivable such as nuclear reactors or beamed energy. The energy conversion occurs by one of three mechanisms: electrothermal, electrostatic, or electromagnetic acceleration (112) (113). Each of these technologies are covered herein.

This survey of the state-of-the-art in EP does not attempt to review all known devices but focuses on those devices that can be commercially procured or devices that appear on a path toward commercial availability. The intent is to aid mission design groups and other in-space propulsion end-users by improving their awareness of the full breadth of potentially procurable EP devices that may meet their mission requirements.

Metrics associated with the nominal operating condition for each propulsion device are published herein, rather than metrics for the complete operating range. A focus on the nominal operating condition was decided to improve comprehension of the data and make initial device comparisons more straightforward. When a manufacturer has not specifically stated a nominal operating condition in literature, the manufacturer may have been contacted to determine a recommended nominal operating condition, otherwise a nominal operating condition was assumed based on similarity to other devices. For those metrics not specifically found in published literature, approximations have been made when calculable from available data. Readers are strongly encouraged to follow the references cited to the literature describing each device's full performance range and capabilities.

Electrothermal

a. Technology Description

Electrothermal technologies use electrical energy to increase the enthalpy of a propellant, whereas chemical technologies rely on exothermal chemical reactions. Once heated, the propellant is accelerated and expelled through a conventional converging-diverging nozzle to convert the acquired energy into kinetic energy, like chemical propulsion systems. The specific impulse achieved with electrothermal devices is typically of similar magnitude as chemical devices given that both electrothermal and chemical devices are fundamentally limited by the working temperature limits of materials. However, electrothermal technologies can achieve somewhat higher specific impulses than chemical systems since they are not subject to the limits of chemical energy storage.

Electrothermal devices are typically subclassified within one of the following three categories.

1. *Resistojet devices* employ an electrical heater to raise the temperature of a surface that in turn increases the bulk temperature of a gaseous propellant.
2. *Arcjet devices* sustain an electrical arc through an ionized gaseous propellant, resulting in ohmic heating.

3. *Electrodeless* thrusters heat a gaseous propellant through an inductively or capacitively coupled discharge or by radiation.

Systems where the propellant enthalpy is increased by electrical heating within the propellant tank, rather than heating in the thruster head, are covered in the chemical propulsion section under cold/warm gas systems.

b. Key Integration and Operational Considerations

- **Propellant Selection:** Electrothermal technologies offer some of the most lenient restrictions on propellant selection for in-space propulsion. Whereas chemical systems require propellants with both the right chemical and physical properties to achieve the desired performance, electrothermal systems primarily depend on acceptable physical properties. For example, electrothermal devices can often employ inert gases or even waste products such as water and carbon dioxide. They also allow use of novel propellants such as high storage density refrigerants or in-situ resources. That said, not all propellants can be electrothermally heated without negative consequences. Thermal decomposition of many complex molecules results in the formation of polymers and other inconvenient byproducts. These byproducts may result in clogging of the propulsion system and/or spacecraft contamination.
- **Propellant Storage:** Electrothermal devices may require that propellants be maintained at a high plenum pressure to operate efficiently. This may require a high-pressure propellant storage and delivery system.
- **High Temperature Materials:** The working temperature limit of propellant wetted surfaces in the thruster head is a key limitation on the performance of electrothermal devices. As such, very high temperature materials, such as tungsten and molybdenum alloys, are often employed to maximize performance. The total mass and shape of these high temperature materials are a safety consideration for spacecraft disposal. While most spacecraft materials burnup on re-entry, the behavior of these high temperature materials will be considered when assessing the danger of re-entry debris to life and property.
- **Power Processing:** While some simple resistojet devices may operate directly from spacecraft bus power, other electrothermal devices may require a relatively complex power processing unit (PPU). For example, a radio-frequency electrodeless thruster requires circuitry to convert the DC bus power to a high-frequency alternating current (AC). In some cases, the cost and integration challenges of the PPU can greatly exceed those of the thruster.
- **Thermal Soak-back:** Given the high operating temperatures of electrothermal devices, any reliance on the spacecraft for thermal management of the thruster head should be assessed. While the ideal propulsion system would apply no thermal load on the spacecraft, some thermal soak-back to the spacecraft is inevitable, whether through the mounting structure, propellant lines, cable harness, or radiation.

c. Missions

The Bradford (formerly Deep Space Industries) Comet water-based electrothermal propulsion system (figure 4.19) has been implemented by three customers operating in low-Earth orbit: HawkEye 360, Capella Space, and BlackSky Global (114). All three missions employ the same Comet thruster head, while the BlackSky Global satellites use a larger tank to provide a greater total impulse capability. The three HawkEye 360 pathfinder spacecraft employ the Space Flight Laboratory NEMO platform with each spacecraft measuring 20 x 20 x 44 cm^3 with a mass of 13.4 kg (115) (116). The Comet provides each HawkEye 360 a total delta-v capability of 96 ms^{-1}. The approximate dimensions of the BlackSky Global spacecraft are 55 x 67 x 86 cm^3 with a mass of 56 kg (117).

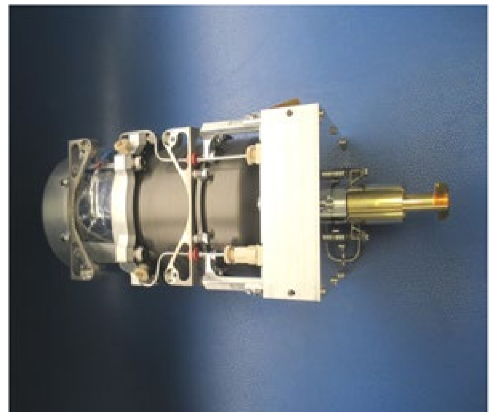


Figure 4.19: Comet-1000. Credit: Bradford Space.

The Propulsion Unit for CubeSats (PUC) system (118), figure 4.20, was designed and fabricated by CU Aerospace LLC (Champaign, IL) and VACCO Industries under contract with the U.S. Air Force to supply two government missions (119). The system was acquired for drag makeup capability to extend asset lifetime in low-Earth orbit. The system uses SO_2 as a self-pressurizing liquid propellant. The propulsion system electrothermally heats the propellant using a micro-cavity discharge (MCD) and expels the propellant through a single nozzle (120). It can alternatively use R134a or R236fa propellants, but only in a cold-gas mode with reduced performance. Eight (8) flight units were delivered to the Air Force in 2014, though it is unknown if any of the units have flown.

Figure 4.20: PUC module. Credit: CU Aerospace LLC.

In 2019, CU Aerospace was selected for a NASA STMD Tipping Point award to design, fabricate, integrate, and perform mission operations for the DUPLEX 6U CubeSat having two of CU Aerospace's micro-propulsion systems on board, one Monofilament Vaporization Propulsion (MVP) system (121) (122), figure 4.21, and one Fiber-Fed Pulsed Plasma Thruster (FPPT) system (123) (124) (125) (126), figure 4.44. The MVP is an electrothermal device that vaporizes and heats an inert solid polymer propellant fiber to 1100 K. The coiled solid filament approach for propellant storage and delivery addresses common propellant safety concerns, which often limit the application of propulsion on low-cost CubeSats. In-orbit operations will include inclination change, orbit raising and lowering, drag makeup, and deorbit burns demonstrating multiple mission capabilities with approximately 20 hours of operation for MVP and >1,000 hours for FPPT. Launch is anticipated in mid-2022 (127).

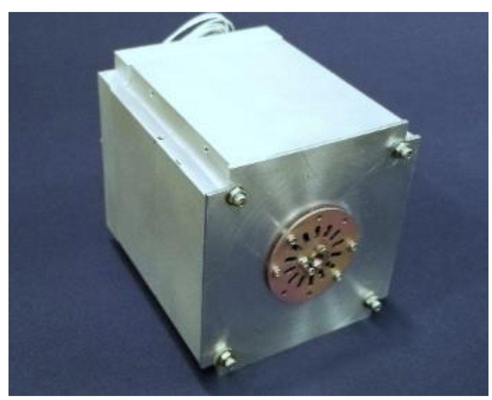


Figure 4.21: MVP module. Credit: CU Aerospace.

AuroraSat-1 is a technology demonstration 1.5U CubeSat that will demonstrate multiple propulsion devices by Aurora Propulsion Technologies. AuroraSat-1 will carry Aurora's smallest version of their Attitude and Orbit Control System (AOCS) (128), figure 4.22, and a demonstration unit of their Plasma Brake Module (PBM). The AOCS integrated in AuroraSat-1 has six resistojet thrusters for full 3-axis attitude control and 70 grams of water propellant, providing a total impulse of 70 N-s. AuroraSat-1 is built by SatRevolution with Aurora providing the payloads. The satellite is anticipated to be launched on an Electron rocket in Q4 2021 (129) (130) (131). See section 4.6.3 for discussion of the PBM module.

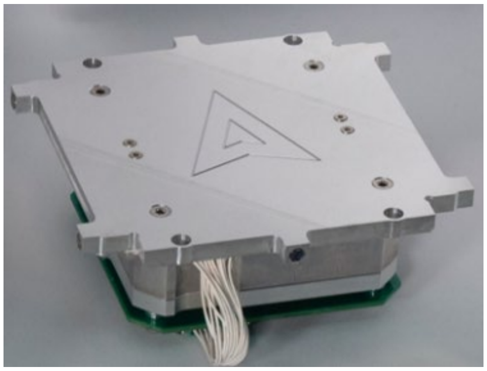


Figure 4.22: ARM-A AOCS module. Credit: Aurora Propulsion Technologies.

d. Summary Table of Devices

See table 4-7 for current state-of-the-art electrothermal devices applicable to small spacecraft.

Electrosprays

a. Technology Description

Electrospray propulsion systems generate thrust by electrostatically extracting and accelerating ions or droplets from a low-vapor-pressure, electrically-conductive, liquid propellant (figure 4.23). This technology can be generally classified into the following types according to the propellant used:

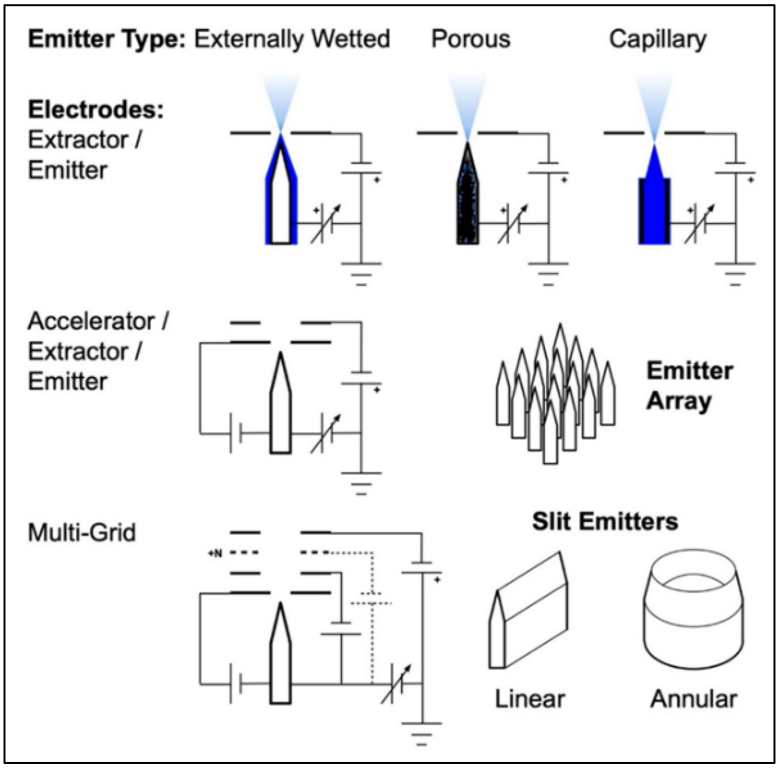


Figure 4.23: Schematic of typical electrospray emitter and electrode configurations. Credit: NASA.

Ionic-Liquid Electrosprays: These technologies use ionic liquids (i.e., salts in a liquid phase at room conditions) as the propellant. The propellant is stored as a liquid, and onboard heaters may be present to maintain propellant properties within the desired operational temperature range. Commonly used propellants include 1-ethyl-3-methylimidazolium tetrafluoroborate (EMI-BF4) and bis(trifluoromethylsulfonyl)imide (EMI-Im). Thrusters that principally emit droplets are also referred to as colloidal thrusters.

Field Emission Electric Propulsion (FEEP): These technologies use low-melting-point metals as the propellant. The propellant is typically stored as a solid, and onboard heaters are used to liquefy the propellant prior to thruster operations. Common propellants include indium and cesium.

Feed systems for electrospray technologies can be actively fed via pressurant gas or passively fed via capillary forces. The ion (high-I_{SP}) or droplet (moderate-I_{SP}) emission can be controlled by modulation of the high-voltage (i.e., >1 kV) input in a closed-loop feedback system with current measurements. Stable operations in either emission mode can provide very precise impulse bits. Propellants that result in both anion and cation emission may not require the presence of a cathode neutralizer to maintain overall charge balance; such neutralizers are included as part of the electrospray propulsion system for propellants that only emit positively charged species.

b. Key Integration and Operational Considerations

- **Plume Contamination**: Because propellants for electrospray propulsion systems are electrically conductive and condensable as liquids or solids, impingement of the thruster plume on spacecraft surfaces may lead to electrical shorting and surface contamination of solar panels and sensitive spacecraft components.
- **Propellant Handling and Thruster Contamination**: Ionic liquids and metallic propellants can be sensitive to humidity and oxidation, so care is needed if extended storage prior to flight is required. Electrospray technologies can also be sensitive to contamination of the thruster head during propellant loading, ground testing (e.g., backsputter or outgassed materials from the test facility), and handling (i.e., foreign object debris). Precautions should be taken to minimize contamination risks from manufacturing, through test, and to launch. Post-launch, ionic liquids can outgas (e.g., water vapor) when exposed to the space environment, and such behavior should be accounted for in the mission ConOps.
- **Performance Stability and Lifetime**: As an electrospray propulsion system operates over time, the propulsive performance can degrade as the plume impinges upon and deposits condensable propellant on thruster head surfaces; in time, sufficiently deposited propellant buildup can electrically short out the thruster electrodes and terminate thruster operations. Especially for missions with large total impulse requirements, lifetime testing or validated life models of the electrospray propulsion system in a relevant environment is important for understanding end-of-life behavior.
- **Specific Impulse**: Even for electrosprays that principally emit ions, operational thruster modes and instabilities can result in droplet emission that degrade the specific impulse and thrust efficiency. Caution is advised when considering claimed specific impulse or other propulsive properties (e.g., thrust vector and beam divergence) derived from plume characteristics; verification test data in a relevant environment is important for properly assessing these claims.
- **Precision Thrust**: Electrospray devices have the potential of providing very fine thrust precision during continuous operations. For devices that can operate in pulsed mode via pulsed modulation of the high-voltage input, fine impulse bits (i.e., <10 μN-s) may be achievable. Such operations permit precise control over spacecraft attitude and maneuvering. Verification test data in a relevant environment should be used to properly assess the degree of thrust precision.

c. Missions

The ESA Laser Interferometer Space Antenna (LISA) Pathfinder spacecraft was launched in December 2015, on Vega flight VV06. Onboard were two integrated propulsion modules associated with the NASA Space Technology 7 Disturbance Reduction System (ST7 DRS). Each propulsion module contained four independent Busek Colloid MicroNewton Thrusters (CMNT), propellant-less cathode neutralizers, power processing units, digital control electronics, and low-pressure propellant tanks. The propulsion system was successfully commissioned in-orbit in January 2016, after having been fully fueled and stored for almost eight years. The electrospray modules (figure 4.24), were operated at the Earth-Sun Lagrange Point 1 for 90 days to counteract solar disturbance forces on the spacecraft; seven of the eight thrusters demonstrated performance consistent with ground test results, and the full propulsion system met the mission-level performance requirements (132).

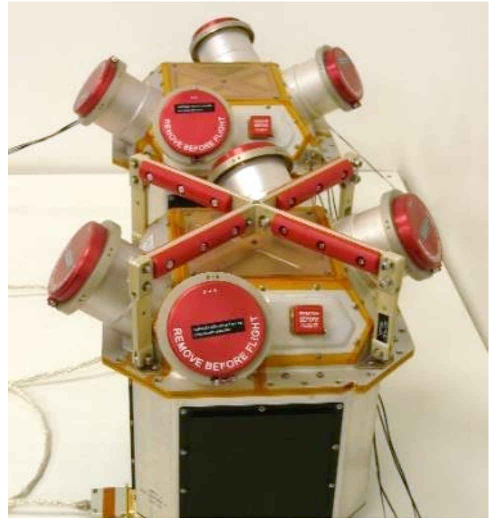


Figure 4.24: Flight CMNT modules for LISA Pathfinder. Credit: Busek.

Enpulsion's IFM Nano FEEP (figure 4.25), was first integrated onboard a 3U Planet Labs Flock 3P' CubeSat and launched via PSLV-C40 in January 2018. The indium-propellant propulsion system (with integrated thruster head, propellant storage, and power processing unit) was demonstrated in a 491 km by 510 km orbit. Two thruster firing sequences were reported, with the first a 15-minute firing in non-eclipse and the second a 30-minute firing in eclipse. Global Positioning System (GPS) telemetry data onboard the spacecraft indicated good agreement with the ~220 μN commanded thrust (133). Since this initial demonstration, the IFM Nano has flown onboard other spacecraft, but limited in-orbit data is publicly available. These missions include the ICEYE-X2 (launched onboard Falcon-9 flight F9-64 in December 2018) to provide low-Earth orbit interferometric synthetic aperture radar observations (134) (135) and the DOD-funded Harbinger technology demonstrator (launched onboard Electron flight STP-27RD in May 2019) (136) (137). The IFM Nano was also integrated onboard the Zentrum für Telematik (Würzburg) NetSat formation-flying demonstrator mission, which launched as a Soyuz-2 rideshare in September 2020 (138) (139).

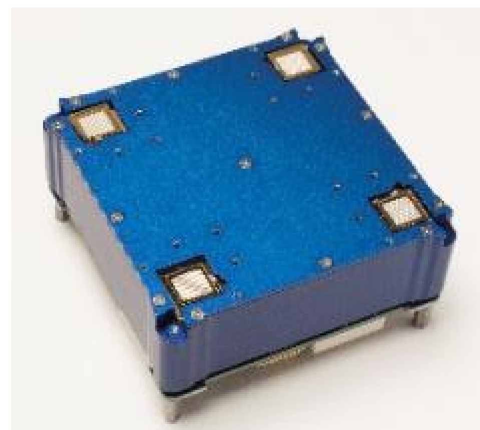
Figure 4.25: IFM Nano. Credit: Enpulsion.

The GMS-T mission was launched in January 2021 onboard a Rocket Lab Electron. The telecommunications satellite uses an OHB Sweden Innosat platform and houses an Enpulsion Micro R^3 (figure 4.26). Inaugural on-orbit commissioning of the propulsion system was confirmed in March 2021 (140).

Figure 4.26: IFM Micro R^3. Credit: Enpulsion.

The University Würzburg Experimental Satellite 4 (UWE-4) was launched as a secondary payload onboard the Soyuz Kanopus-V 5 and 6 mission in December 2018. This 1U spacecraft housed two Morpheus Space NanoFEEP systems, with each system consisting of two gallium-propellant thrusters, a power processing unit board for the UNISEC Europe bus, and a propellant-less cathode neutralizer. An experiment using one thruster as an attitude control actuator was reported, with the increased spacecraft rotation rate corresponding to a derived thrust magnitude of ~5 μN; anomalous torque was attributed to unexpected impingement of the thruster plume upon the spacecraft antenna (141) (142). Orbit lowering capability was demonstrated in 2020; of the four individual thrusters, three experienced anomalous behavior during the UWE-4 mission (143). A 3U-Cubsat implementation of the same NanoFEEP technology is shown in figure 4.27.

Astro Digital's Tenzing satellite, which was integrated with a Sherpa-LTE Orbital Transfer Vehicle onboard the SpaceX Falcon 9 Transporter-2 launch in June 2021, houses two Accion Systems' TILE-2 units (figure 4.28) to demonstrate on-orbit rendezvous and proximity operations maneuvers (144). Another TILE-2 system is integrated onboard the Massachusetts Institute of Technology's BeaverCube, an educational mission that is expected to launch as a secondary payload onboard the SpaceX CRS-23 mission in August 2021 (145) (146) (147).



Systems.

Accion's TILE-3 technology (consisting of an integrated unit with thruster heads, propellant storage, and power processing unit) is expected to be demonstrated onboard the D2/AtlaCom-1 mission. The spacecraft, a NanoAvionics M6P bus, was deployed in low Earth orbit following a SpaceX Falcon 9 Transporter-2 launch in June 2021 (148). Under a NASA Tipping Point Partnership, this mission seeks to demonstrate comparable propulsive capability as the MarCO CubeSats, but instead using electrospray technology (149). A TILE-3 unit is shown in figure 4.29.

Figure 4.29: TILE-3. Credit: Accion Systems.

d. Summary Table of Devices

See table 4-8 for current state-of-the-art electrospray devices applicable to small spacecraft.

Gridded-Ion

a. Technology Description

Gridded-ion propulsion systems ionize gaseous propellant via a plasma discharge, and the resultant ions are subsequently accelerated via electrostatic grids (i.e., ion optics). This technology can be generally classified into the following types according to the type of plasma discharge employed:

- **Direct-Current (DC) Discharge**: The propellant is ionized via electron bombardment from an internal discharge cathode (figure 4.30).
- **Radio-Frequency (RF) Discharge**: No internal discharge cathode is present. Instead, the propellant is ionized via RF or microwave excitation from an RF generator (figure 4.31).

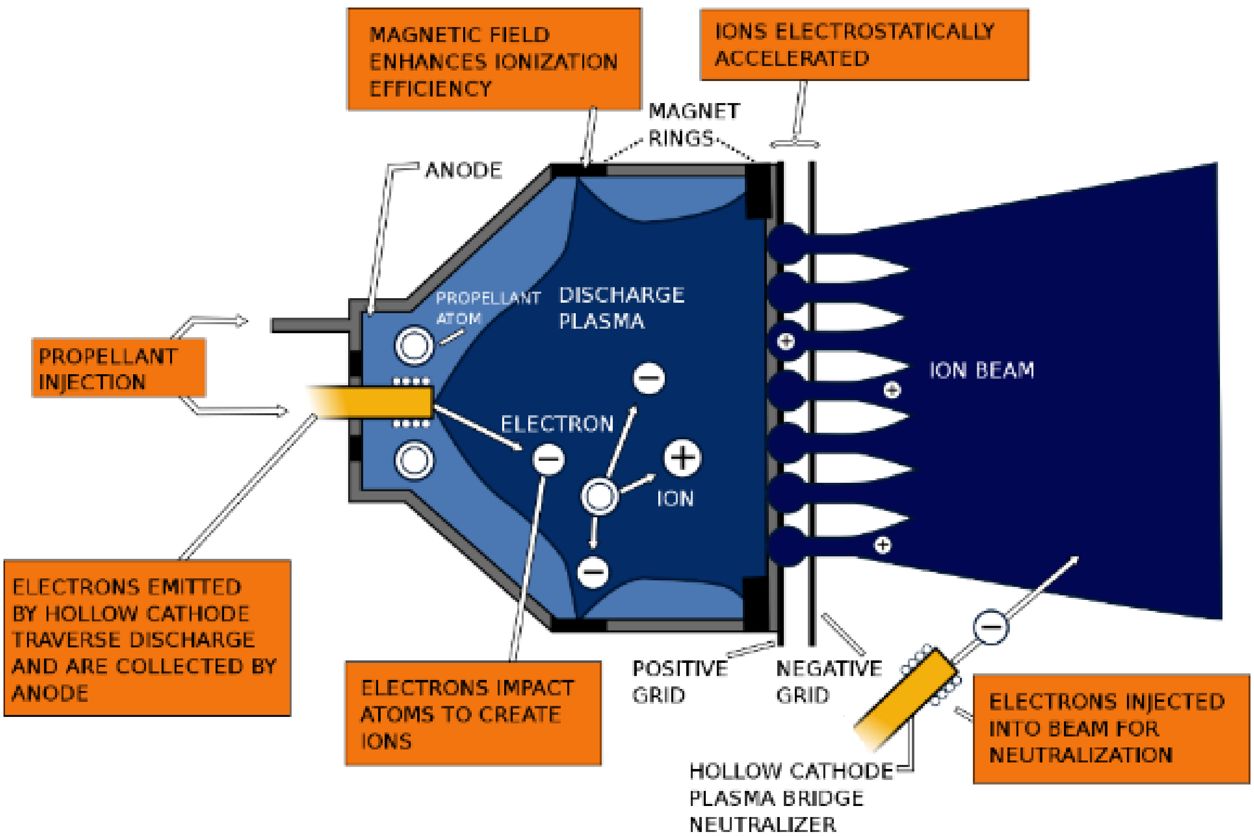


Figure 4.30: Schematic of typical DC-discharge gridded-ion thruster. Credit: NASA.

Gridded-ion thrusters typically operate at high voltages and include an external neutralizer cathode to maintain plume charge neutrality. High specific impulses can be achieved, but the thrust density is fundamentally limited by space-charge effects. While the earliest thruster technologies used metallic propellants (i.e., mercury and cesium), modern gridded-ion thrusters use noble gases (e.g., xenon) or iodine.

b. Key Integration and Operational Considerations

- **Performance Prediction:** Due to the enclosed region of ion generation and acceleration, gridded ion thrusters tend to be less sensitive to test-facility backpressure effects than other devices such as Hall thrusters. This allows for more reliable prediction of in-flight performance based on ground measurements. Furthermore, the separation between ion generation and acceleration mechanisms within the device tend to make calculations of thrust and ion velocity (or I_{SP}) more straightforward.

- **Grid Erosion**: Charge-exchange ions formed in between and downstream of the ion optics can impinge upon and erode the grids. Over time, this erosion can lead to a variety of failure modes, including grid structural failure, an inability to prevent electrons from back streaming into the discharge chamber, or the generation of an inter-grid electrical short due to the deposition of electrically conductive grid material. Proper grid alignment is important to reducing grid erosion, and this alignment must be maintained during thruster assembly, transport, launch, and operations. Random vibration tests at the protoflight level should be conducted to verify the survivability of the ion optics against launch loads, and validated thermal modeling may be needed to assess the impact of grid thermal expansion during thruster operations.
- **Foreign Object Debris**: The grids are separated by a small gap, typically less than 1 mm, to maximize the electric field and thrust capability of the device. As a result, gridded-ion thrusters tend to be sensitive to foreign object debris, which can bridge the inter-grid gap and cause electrical shorting. Precautions should be taken to minimize such contamination risks from manufacturing, through test, and to launch.
- **Cathode Lifetime**: Cathodes for plasma discharge or plume neutralization may be sensitive to propellant purity and pre-launch environmental exposure. Feed system cleanliness, bake-out, and use of a high-purity propellant are key factors in maximizing cathode lifetime. The technology provider may recommend a maximum cumulative atmospheric exposure and humidity to reduce risk.
- **Roll Torque**: Misalignments in the ion optics can lead to disturbances in the thrust vector, resulting in a torque around the roll axis that cannot be addressed by the mounting gimbal.

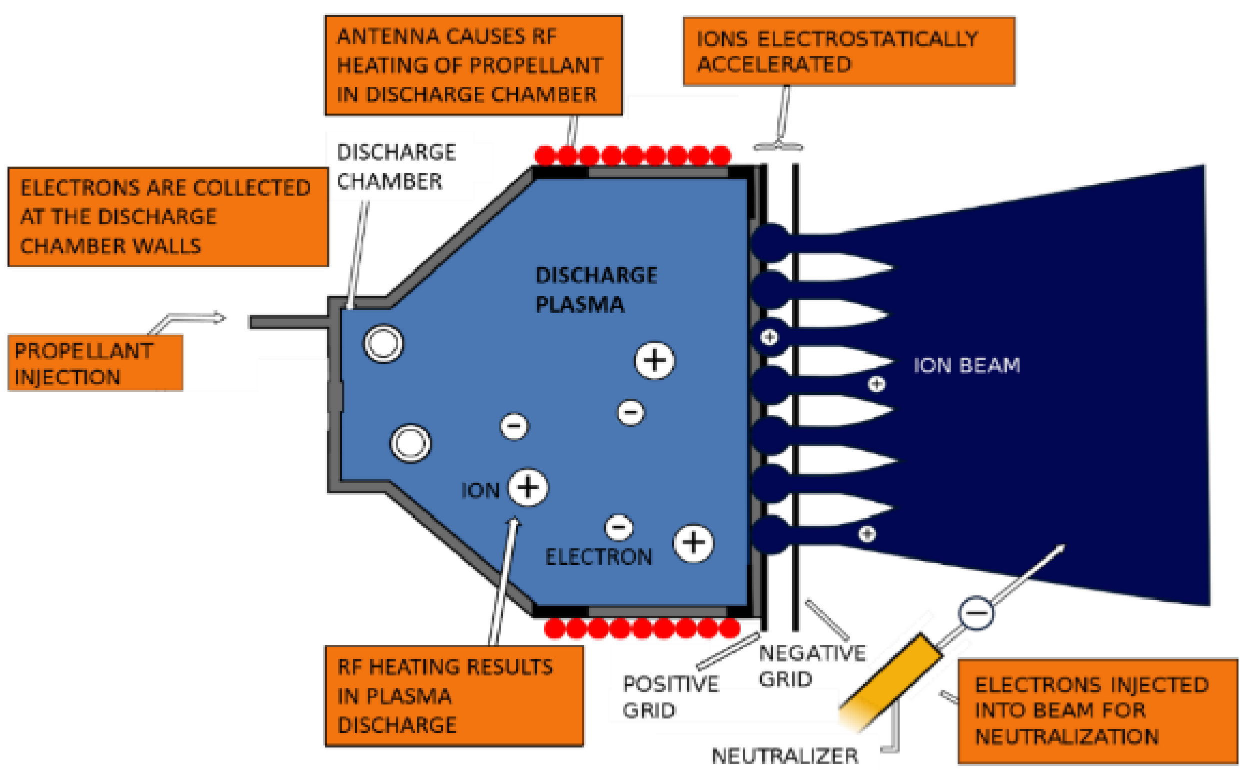


Figure 4.31: Schematic of typical RF-discharge gridded-ion thruster. Credit: NASA.

For missions requiring extended thruster operations, a secondary propulsion system or reaction wheels may be needed to counter the torque buildup (150).

- **Electromagnetic Interactions**: For RF-discharge thrusters, electromagnetic interference and compatibility (EMI/EMC) testing may be critical to assess the impact of thruster operations on spacecraft communications and payload functionality.
- **Iodine Propellant:** To address the volume constraints of small spacecraft, iodine is an attractive propellant. Compared to xenon, iodine's storage density is three times greater. Furthermore, iodine stores as a solid with a low vapor pressure, which addresses spacecraft integration concerns associated with high-pressure propellant storage. However, iodine is a strong oxidizer and long-duration impact on the thruster and spacecraft remain largely unknown. Upcoming flights will provide insight into potential spacecraft interactions and long-term reliability of feed system and thruster components.
- **Power Electronics:** Operation of gridded ion thrusters requires multiple high-voltage power supplies for discharge operation (ion generation), ion acceleration, and neutralization, leading to potentially complex and expensive power electronics.

c. Missions

The ESA Gravity Field and Steady-State Ocean Circulation Explorer (GOCE) was launched in March 2009 onboard a Rokot / Briz-KM to provide detailed mapping of Earth's gravitational field and ocean dynamics from an altitude of ~220-260 km. Two QinetiQ T5 DC-discharge gridded-ion thrusters (figure 4.32), with one serving as a redundant backup, successfully provided drag-free control of the 1000-kg satellite until xenon propellant exhaustion in October 2013 (151) (152).

Figure 4.32: T5 gridded-ion thruster for GOCE mission. Credit: QinetiQ.

The Beihangkongshi-1 satellite was launched in November 2020 onboard a Long March 6 rocket. The 12U Spacety CubeSat housed a ThrustMe NPT30-I2-1U (figure 4.33), a 1U-integrated, RF-discharge gridded-ion propulsion system. As part of the first on-orbit demonstration of iodine-propellant electric propulsion, two 90-minute burns provided an orbit altitude change of 700 m (153). A 1.5U version of the NPT30-I2 is expected to fly onboard a Space Flight Laboratory of the University of Toronto, Institute for Aerospace Studies (UTIAS) 35-kg DEFIANT bus for the Norwegian Space Agency's NorSat-TD mission; expected to launch in 2022, this mission includes a demonstration of satellite collision avoidance maneuvers (154). NPT30-I2-1.5U is also expected to fly onboard a GomSpace 12U CubeSat for the 2022 ESA GOMX-5 technology demonstration mission (155).

Figure 4.33: NPT30-I2-1U. Credit ThrustMe.

Lunar IceCube is an upcoming NASA-funded CubeSat mission to characterize the distribution of water and other volatiles on the Moon from a highly-inclined lunar orbit with a perilune < 100 km. Led by Morehead State University, the mission will be conducted via a 6U spacecraft that is manifested as a secondary payload onboard Artemis I (156) (157).

Lunar Polar Hydrogen Mapper (LunaH-Map) is an upcoming NASA-funded CubeSat mission to map hydrogen distributions at the lunar south pole from a lunar orbit with a perilune < 20 km. Led by Arizona State University, the mission will be conducted via a 6U spacecraft that is manifested as a secondary payload onboard Artemis I (158).

Figure 4.34: BIT-3 thruster. Credit: Busek.

Both Lunar IceCube and LunarH-Map missions use an onboard Busek BIT-3 propulsion system (figure 4.34) with solid iodine propellant. The BIT-3 system will be used as primary propulsion during the lunar transfer trajectory, followed by lunar orbit capture, orbit lowering, and spacecraft disposal. Each integrated BIT-3 system includes a low-pressure propellant tank with heated propellant-feed components, a power processing unit to control the RF thruster and RF cathode, and a two-axis gimbal assembly.

d. Summary Table of Devices

See table 4-9 for current state-of-the-art gridded-ion devices applicable to small spacecraft.

Hall-Effect

a. Technology Description

The Hall-effect thruster (HET) is arguably the most successful in-space EP technology by quantity of units flown. The Soviet Union first flew a pair of EDB Fakel SPT-60 HETs on the Meteor-1-10 spacecraft in 1971. Between 1971 and 2018, over 300 additional HETs flew internationally, although EDB Fakel produced the vast majority. The first flight of a non-Russian HET was on board the European Space Agency (ESA) Small Missions for Advanced Research in Technology (SMART-1) spacecraft in 2003. SMART-1 employed the French PPS-1350 HET, produced by Safran (159). The first flight of a U.S. manufactured HET, the Busek BHT-200, was onboard the TacSat-2 spacecraft (160), a U.S. Air Force Research Laboratory (AFRL) experimental satellite in 2006. In 2010, Aerojet, another U.S. entity, began commercially delivering their 4.5 kW XR5 HET (161), formerly BPT-4000. Launches of HETs greatly accelerated in 2019 with the launch of 120 SpaceX Starlink and 6 OneWeb spacecraft (162), each including an HET. By late-June 2021, an additional 1,617 SpaceX and 212 OneWeb satellites launched into low-Earth orbit with HETs. Suffice to say that HETs have become a mainstream in-space propulsion technology.

The rapid growth in demand for HETs can be attributed to their simple design, historically well-demonstrated reliability, good efficiency, high specific impulse, and high thrust-to-power ratio. Although, the higher voltage gridded-ion thrusters (GIT) can achieve even higher specific impulse, HETs can achieve higher thrust-to-power ratios because the HET's higher density quasi-neutral plasma is not subject to space-charge limitations. The HET's higher thrust-to-power ratio will typically shorten spacecraft transit time. On the other end of the spectrum, arcjets provide significantly higher thrust than HETs, however material limitations prevent arcjets from matching the HET's electrical efficiency and specific impulse. For many missions, HETs provide a good balance of specific impulse, thrust, cost, and reliability.

HETs are a form of ion propulsion, ionizing and electrostatically accelerating the propellant. Historically, all HETs flown in space have relied on xenon propellant, given its high molecular weight, low ionization energy, and ease of handling. The recent exception is the SpaceX Starlink spacecraft using krypton propellant. While HETs operate less efficiently with krypton propellant and krypton has more challenging storage requirements, krypton gas is considerably lower cost than xenon gas, which is a compelling attribute when the potential number of spacecraft are projected in the thousands, as with constellations. Many other propellants have been considered and ground tested for Hall-effect thrusters, but to date only Hall-effect thrusters using xenon or krypton have flown.

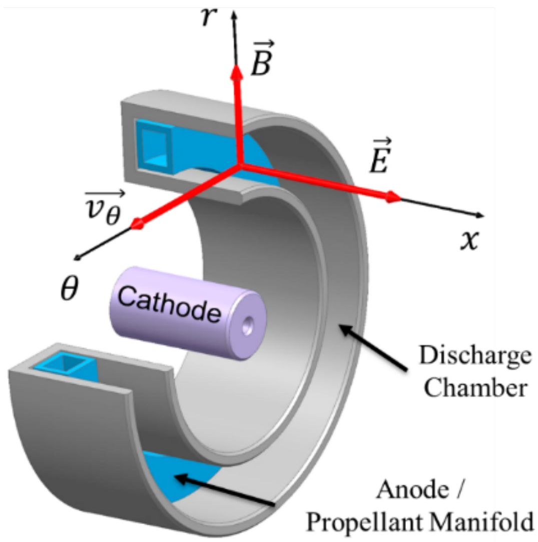


Figure 4.35: Hall-effect Thruster schematic. Credit: NASA.

As schematically shown in figure 4.35, HETs apply a strong axial electric field and radial magnetic field near the discharge chamber exit plane. The **E x B** force greatly slows the mean axial velocity of electrons and results in an azimuthal electron current many times greater than the beam current. This azimuthal current provides the means by which the incoming neutral propellant is collisionally ionized. These ions are electrostatically accelerated and only weakly affected by the magnetic field. The electron source is a low work function material typically housed in a refractory metal structure (i.e., hollow cathode), historically located external to the HET body, although many recent thruster designs have begun centrally mounting the cathode in the HET body as shown in figure 4.35. The cathode feeds electrons to the HET plasma and neutralizes the plasma plume ejected from the thruster. The high voltage annular anode sits at the rear of the discharge chamber and typically functions as the propellant distribution manifold.

b. Key Integration and Operational Considerations

- **Ground Facility Effects:** Ground facility effects may result in inconsistencies between ground and flight performance. The significance of the inconsistencies depend on factors such as test facility scale, test facility pumping speed, intrusiveness of diagnostics, and thruster electrical configuration.
- **Contamination:** Plume ions of an HET can affect spacecraft surfaces by erosion or contamination, even at large plume angles. Ground facility measurement of ion density at large angles may under predict flight conditions.
- **Thermal Soak-Back:** HET core temperature may exceed 400°C with the cathode exceeding 1000°C. Most HET waste heat radiates directly from the HET surfaces. However, some thermal soak-back to the spacecraft will occur through the mounting structure, propellant feed lines, electrical harness, and radiation.
- **Survival Heaters:** Given the thermal isolation between the HET and spacecraft, the HET may require a survival heater depending on the qualification temperature and flight environments.
- **Performance:** HET performance may vary over the life of the device due to erosion and contamination of the plasma wetted HET surfaces. Magnetically shielded thrusters demonstrate less time dependency to their performance than classical HETs.

- **Thruster Lifetime:** Classical HETs are primarily life-limited by erosion of the discharge chamber wall. Magnetically shielded HETs are primarily life-limited by erosion of the front pole covers.
- **Cathode Lifetime:** Cathode lifetime may be sensitive to propellant purity and pre-launch environmental exposure. Feed system cleanliness, bake-out, and use of a high purity propellant are key factors in maximizing cathode lifetime. The HET manufacturer may recommend a maximum cumulative atmospheric exposure and humidity. Some cathode emitter formulations are less sensitive to propellant impurities and atmospheric exposure, but these formulations may require other trades such as a higher ignition temperature.
- **Roll Torque**: The **E x B** force results in a slight swirl torque. For missions requiring extended thruster operations, a secondary propulsion system or reaction wheels may be needed to counter the torque buildup. The roll torque may largely be countered by periodically reversing the direction of the magnetic field. Field reversal requires switching the polarity of current to the magnet coils. Field reversal is only possible with HETs using electromagnets.
- **Thrust Vector**: Non-uniformity of the azimuthal plasma, magnetic field, or propellant flow may result in slight variations of the thrust vector relative to the HET physical centerline. Temperature variation of the HET, such as during startup, also results in a slight walking of the thrust vector.
- **Heaterless Cathodes:** Heaterless cathode technologies continue to mature. The benefit of a heaterless cathode is elimination of the cathode heater, typically an expensive component due to rigorous manufacturing and acceptance processes. However, the physics of heaterless cathode life-limiting processes require further understanding. Nevertheless, heaterless cathode demonstrations have empirically shown significant promise. Heaterless cathode requirements on the EP system differ from an HET with a cathode heater. Impacts on the power processing unit and feed system should be well understood when trading a heaterless versus heated cathode.
- **Throttling Range:** HETs typically throttle stably over a wide range of power and discharge voltage. This makes an HET attractive for missions requiring multiple throttle set-points. However, an HET operates most efficiently at specific throttle conditions. Operating at off-nominal conditions may result in decreased specific impulse and/or electrical efficiency.

c. Missions

Canopus-V (alternative spelling Kanopus-V) is a Russian Space Agency spacecraft for Earth observation with a design life of 5 years. The 450 kg spacecraft launched in 2012 employed a pair of EDB Fakel SPT-50 thrusters. Similarly, the Canopus-V-IK (Kanopus-V-IK) launched in 2017 with a pair of SPT-50. The SPT-50 thrusters have a long history of spaceflight dating back to the late 1970s. Although the Canopus bus exceeds 450 kg, the power class and physical scale of the SPT-50 are appropriate for smaller spacecraft. The SPT-50 is nominally a 220 W thruster operated on xenon propellant (163) (164) (165).

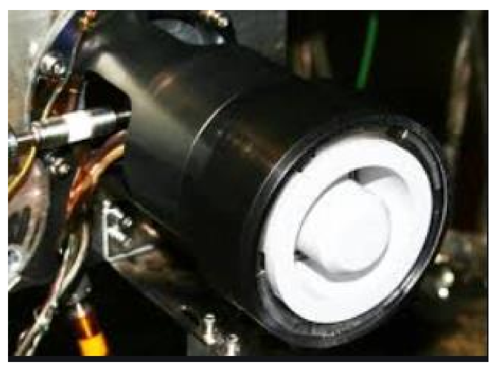
The KazSat-1 and KazSat-2 spacecraft produced by Khrunichev Space Center in cooperation with Thales Alenia Space launched in 2006 and 2011, respectively. The KazSat spacecraft are geosynchronous communication satellites. These spacecrafts employ the EDB Fakel SPT-70BR thruster. The SPT-70BR is Fakel's latest version of the SPT-70 product line. EDB Fakel optimized the SPT-70 for operation between 600 and 700 W, but no more than 900 W. Experiments demonstrate a lifetime of 3,100 hours, equating to about 450 kNs. The SPT-70 thrusters have a long history of spaceflight dating back to the early 1980s. Control of KazSat-1 was lost in 2008 (166) (167).

The Busek BHT-200 (figure 4.37) has the distinction of being the first U.S.-made HET to operate in space. The BHT-200 has flight heritage from demonstrations on the TacSat-2 mission launched in 2006, FalconSat-5 mission launched in 2010, and FalconSat-6 mission launched in 2018. A Busek PPU powered the 200 W HET for each of the FalconSat missions (168). Ground testing of the BHT-200 includes multiple propellants, although all spaceflights have used xenon. Busek developed an iodine compatible derivative of the BHT-200 for the NASA iSat mission. It was determined during the course of the iSat project that additional development related to iodine compatible cathodes was required before conducting an in-space demonstration of the technology at this scale of thruster (169) (170).

The Israel Space Agency and the French National Center for Space Studies (CNES) jointly developed the Vegetation and Environment monitoring on a New Microsatellite (VENuS) spacecraft launched in 2017. The 268 kg VENuS spacecraft includes a pair of Rafael IHET-300 thrusters (figure 4.36) and 16 kg of xenon propellant. Inflight operations have demonstrated operation between 250 and 600 W. Rafael developed the IHET-300, nominally operating at 300 W, specifically for small spacecraft (171) (172) (173) (174) (175).

Figure 4.36: IHET-300 thruster. Credit: Rafael.

The European and Italian space agencies selected the SITAEL HT100 (figure 4.38) for an in-orbit validation program to evaluate the device's capabilities for orbital maintenance and accelerated reentry of a small spacecraft. The uHETSat mission will be the first in-orbit demonstration of the HT100. SITAEL is currently performing ground qualification of the complete propulsion system. The HT100 is nominally a 175 W device operating on xenon propellant. The uHETSat will use the SITAEL S-75 microsatellite platform. The S-75 is 75 kg with dimensions of 60 x 40 x 36 cm^3. The anticipated launch date targets 2021 (176) (177) (178).



The Astro Digital Ignis satellite is a technology demonstration spacecraft built to the 6U CubeSat standard. The spacecraft bus is the Astro Digital Corvus-6 design, which is 32 x 21 x 11 cm^3 with a mass no more than 12 kg. The Ignis includes the Apollo Fusion Apollo Constellation Engine (ACE), shown in figure 4.39. Apollo Fusion offers the ACE compatible with xenon, krypton, and a proprietary high-density propellant. This first flight of the ACE HET was anticipated to employ 1.1 kg of the proprietary propellant, providing approximately 12,000 Ns

Figure 4.39: ACE thruster. Credit: Apollo Fusion.

of total impulse. The anticipated lifetime of the spacecraft is less than 3 years in low-Earth orbit with an altitude of 500 km. Ignis was anticipated to launch in 2021 (179) (180). The ACE was further selected as the electric propulsion sytem for Spaceflight Inc.'s Sherpa-LTE using xenon propellant. The Sherpa-LTE is an orbital transfer vehicle (OTV) claimed to be capable of delivering customers to GEO, Cislunar, and Earth-escape orbits.The first Sherpa-LTE launched on SpaceX's Transporter-2 mission on June 30, 2021 (181) (182).

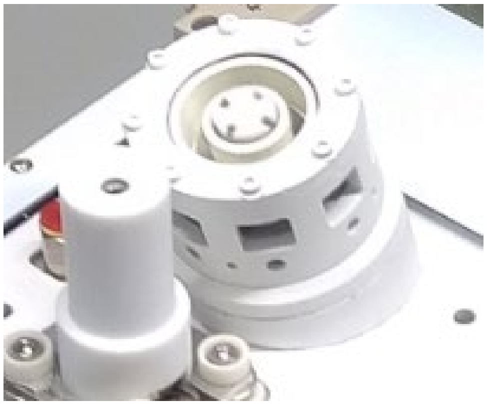
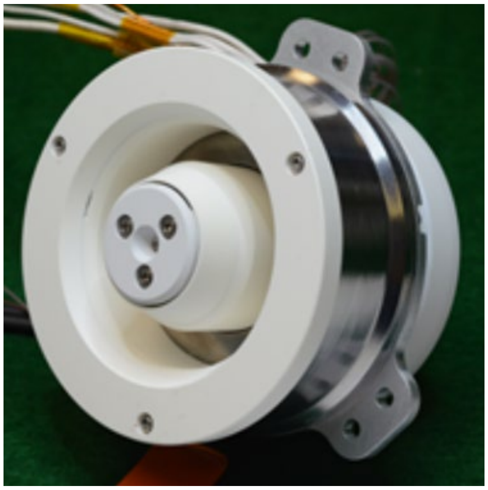


Figure 4.40: ExoMG-nano thruster. Credit: Exotrail.

Exotrail launched its first in-orbit demonstration mission including the 50 Watt ExoMG-nano (figure 4.40) thruster in November 2020. NanoAvionics and Exotrail partnered to integrate the ExoMG-nano into NanoAvionics' M6P nanosatellite 6U bus. Exotrail and its partners designed, built, integrated, and qualified the ExoMG-nano demonstrator in 10 months. Exotrail further signed a contract with AAC Clyde Space to provide propulsion for the Eutelsat ELO 3 and ELO4 6U CubeSats anticipated to launch in 2021 (183) (184) (185) (186) (187).



ExoTerra has received a NASA Tipping Point award to perform an in-orbit demonstration of their 12U Courier SEP spacecraft bus with a target launch date of December 2021. The bus includes ExoTerra's Halo thruster (figure 4.41), propellant distribution, power processing unit and deployable solar arrays. The Courier spacecraft provides up to 1 km s^{-1} of delta-v, while hosting a 2U, 4 kg payload. The Tipping Point mission objective is to demonstrate the SEP system by spiraling to 800 km from a drop-off orbit of 400 km and then deorbiting. Primary mission objectives include demonstration of the solar array deployment and power generation, PPU efficiency, and 2 kg of thruster propellant throughput. The 0.67 kg, 1/4U thruster will nominally operate at 135 W. During the mission operations, a variation in thruster power and discharge voltage will demonstrate a performance range of 135 to 185 W and 150 to 400 V, respectively (188) (189) (190).

Figure 4.42: Two flight Aurora HETs undergoing qualification testing. Credit: Orbion Space Technology.

AST & Science (AST) of Midland, Texas, selected the Aurora Hall-Effect Propulsion System (figure 4.42) manufactured by Orbion Space Technology for its SpaceMobile network. AST anticipates SpaceMobile to be a low-Earth orbit constellation of hundreds of satellites providing cellular coverage for 4G and 5G smartphones. Orbion's Aurora thrusters will provide propulsion for orbital maintenance, collision avoidance, and de-orbiting at end-of-life. Orbion's Aurora propulsion system consists of a thruster, cathode, power processing unit, propellant flow controller, and cable harness. The anticipated launch date for the first satellite of the SpaceMobile constellation is March 2022 (191) (192) (193).

Blue Canyon has also selected the Orbion Aurora thruster for DARPA Blackjack satellites. Blue Canyon is producing four satellites for the DARPA program as one of multiple satellite bus suppliers. Blackjack satellites are about 150 kilograms (194).

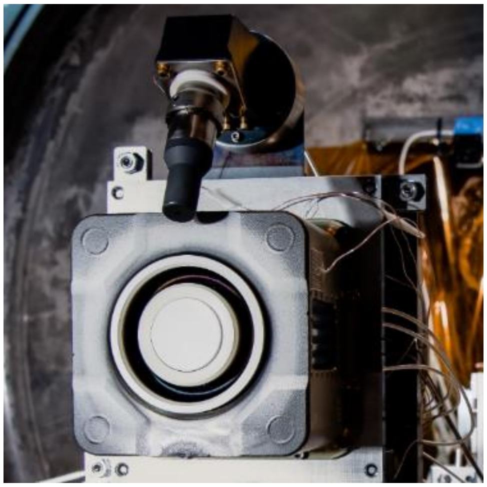


Figure 4.43: BHT-600 Installed in NASA GRC Vacuum Test Facility. Credit: Busek Co.

Busek shipped its first flight BHT-600 Hall-effect thruster system to a U.S. Government customer in early 2021 for an anticipated flight in 2021. The BHT-600 previously demonstrated a 7,000-hour ground test performed at NASA GRC as part of a NASA Announcement for Collaborative Opportunity (ACO) Space Act Agreement (SAA), figure 4.43. The thruster successfully demonstrated 70 kilograms of xenon propellant throughput before the test was terminated. The BHT-600 is designed for operation from 400 W to 1 kW (195) (196).

d. Summary Table of Devices

See table 4-10 for current state-of-the-art HET devices applicable to small spacecraft.

Pulsed Plasma and Vacuum Arc Thrusters

a. Technology Description

Pulsed Plasma Thrusters (PPT) produce thrust by first triggering an electric arc between a pair of electrodes that typically ablates a solid-state propellant like polytetrafluoroethylene (PTFE) or ionizes a gaseous propellant. The plasma may be accelerated by either electrothermal or electromagnetic forces. Whether the mechanism of acceleration is electrothermal, electromagnetic, or often some combination thereof is determined by the device topology (197).

Electrothermal PPTs characteristically include a chamber formed by a pair of electrodes and solid propellant, wherein propellant ablation and heating occurs. During and immediately following each electric discharge, pressure accumulates and accelerates the propellant through a single opening. Electromagnetic PPTs characteristically do not highly confine the propellant as plasma forms. The current pulse, which may exceed tens of thousands of amps, highly ionizes the ablated material or gas. The current pulse further establishes a magnetic field, where the **j x B** force accelerates the plasma. PPT devices that are predominantly electrothermal typically offer higher thrust, while devices that are predominantly electromagnetic offer higher specific impulse.

The simplest PPTs have no moving parts, which may provide a high degree of reliability. However, as the solid propellant is consumed, the profile of the propellant surfaces is constantly changing. Thus, PPTs with static solid propellant demonstrate a change in performance over their life and inherently have a relatively limited lifetime. More complex solid propellant PPTs include a propellant feed mechanism. Typically, the propellant surface profile changes during an initial burn-in period, but then settles into a steady-state behavior where the propellant advancement is balanced by the propellant ablation.

PPT devices are suitable for attitude control and precision pointing applications. PPTs offer small and repeatable impulse bits, which allow for very high precision maneuvering. The complete propulsion system consists of a thruster, an ignitor, and a power processing unit (PPU). Energy to form the pulsed discharge is stored in a high voltage capacitor bank, which often accounts for a significant portion of the system mass. Once the capacitors are charged, resulting in a large differential voltage between the electrodes, the ignitor provides seed material that allows the discharge between the electrodes to form. Various materials and gases (including water vapor) have been tested with PPTs, however PTFE remains most common.

Vacuum arc thrusters (VAT) are another type of pulsed plasma propulsion (198). This technology consists of two metallic electrodes separated by a dielectric insulator. Unlike PPTs, one VAT electrode is sacrificial, providing the propellant source. The mechanism for propellant acceleration is predominantly electromagnetic, resulting in a characteristically high specific impulse and low thrust. One variant of the VAT is predominantly electrostatic, by the inclusion of a downstream electrostatic grid.

b. Key Integration and Operational Considerations

- **Safety**: PPT capacitor banks often store tens of joules of energy at potentially a couple thousand volts. Follow good electrical safety practices when operating and storing PPTs in a laboratory environment.
- **Input Power Range**: PPTs and VATs are pulsed devices, which operate by discharging energy stored in capacitors with each pulse. Thus, the propulsion system's average power draw from the spacecraft bus can be quite low or high depending on the capacitor energy storage and pulse frequency. This flexibility allows PPTs to be applied to spacecraft with limited power budgets of just a few watts, or ample power budgets of hundreds of watts.
- **Minimum Impulse Bit:** A compelling capability of pulsed devices is the ability to generate small, precise, and well-timed impulse bits for precise spacecraft maneuvering. By controlling the discharge voltage, very small impulse bits on the order of micronewtons-seconds are easily achieved.
- **Compact and Simple Designs:** PPTs and VATs are typically very simple and compact devices. While the total impulse capability is small compared to other forms of EP, these devices offer a particularly attractive solution for CubeSats, where low cost may be a more significant consideration than total impulse. The systems are also attractive for learning environments where propulsion expertise such as high-pressure feed systems and propellant management may be lacking.
- **Late-Time Ablation**: Although pulsed devices allow for operation over a wide range of pulse frequency, thruster efficiency typically improves with higher pulse rate. Late time ablation is a key inefficiency of solid propellant pulsed devices, where material continues to ablate from the propellant surface well after the discharge pulse. Through higher frequency pulsing, the amount of material accelerated may be maximized.
- **Thrust-to-Power:** Pulsed devices suffer from several inefficiencies including late time ablation, frozen flow, and wall heating. Propulsion system efficiency is typically below 20% and may be as low as a few percent. Thus, although pulsed devices may have high specific impulse, the thrust-to-power is low. Small spacecraft with limited power for propulsion may find that large propellant loads provide little benefit as there is inherently a limitation to the number of pulses achievable over the life of the power-limited spacecraft.
- **Thermal Soak-back:** The low thruster efficiencies may result in large thermal loads on the spacecraft due to thermal soak-back, especially at high rates of pulsing. The spacecraft's ability to radiate this energy to limit heating may set an upper bound on pulse frequency.
- **Ignitor:** Pulsed devices usually require some form of ignitor to provide seed material to lower the impedance between the electrodes and initiate the discharge pulse. As such, the lifetime of the ignitor may dictate the lifetime of the thruster. Ignitors may fail due to

erosion or fouling that prevents sparking. Some devices may include multiple redundant ignitors to increase system lifetime.

- **Shorting:** The electrodes of pulsed devices are separated by isolating elements. Shadow shielding or other physical features are typically necessary to avoid shorting between electrodes as conductive material ejected by the thruster accumulates. While PTFE is an insulator, the PTFE is reduced to carbon and fluorine when ablated, where carbon accumulation provides a potentially conductive path. VATs employ metal propellants that can similarly result in unintended shorting.
- **Spacecraft Contamination:** As with any conductive propellant, contamination of the spacecraft is a concern. Plume interaction with the spacecraft must be understood to assess the impact of the plume on the operation of critical surfaces such as solar panels, antennas, and radiators.

c. Missions

In 2019, CU Aerospace was selected for a NASA STMD Tipping Point award to design, fabricate, integrate, and perform mission operations for the DUPLEX 6U CubeSat having two of CU Aerospace's micro-propulsion systems on board, one Monofilament Vaporization Propulsion (MVP) system (121) (122), shown in figure 4.21, and one Fiber-Fed Pulsed Plasma Thruster (FPPT) system (123) (124) (125) (126), shown in figure 4.44. The FPPT can provide a large total impulse primary propulsion for micro-satellites through implementation of a novel PTFE fiber propellant storage and delivery mechanism. A major enhancement of the FPPT technology over classical PPTs is the ability to control both the propellant feed rate and pulse energy, thereby providing control of both the specific impulse and thrust. The FPPT can also provide precision control capability for small spacecraft requiring capabilities such as precision pointing or formation flying. In-orbit operations will include inclination change, orbit raising and lowering, drag makeup, and deorbit burns demonstrating multiple mission capabilities with approximately 20 hours of operation for MVP and >1,000 hours for FPPT. Launch is anticipated in mid-2022 (127).

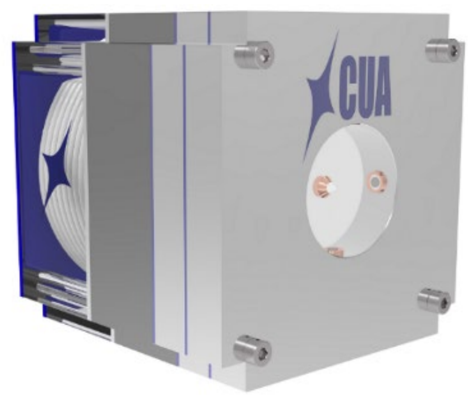


Figure 4.44: FPPT module. Credit: CU Aerospace.

d. Summary Table of Devices

See table 4-11 for current state-of-the-art pulsed plasma and vacuum arc devices applicable to small spacecraft.

Ambipolar

a. Technology Description

Ambipolar thrusters ionize gaseous propellant within a discharge cavity via various means, including DC breakdown or RF excitation. The escape of high-mobility electrons from the discharge cavity creates a charge imbalance in the plasma discharge, and the subsequent ambipolar diffusion accelerates ions out of the cavity to generate thrust.

Because the thruster plume is charge neutral, no neutralizer assembly is necessary. A variety of propellants are theoretically usable due to the absence of exposed electrodes (and their associated material compatibility concerns).

b. Key Integration and Operational Considerations

- **Propellant Agnostic**: While ambipolar thrusters may be operable on a variety of propellants thanks to the devices' lack of exposed electrodes, different propellants will have different ionization costs (i.e., impact on thruster efficiency), plume behavior, and propellant storage requirements that should be considered during propellant selection.
- **Electromagnetic Interactions**: For RF-discharge thrusters, electromagnetic interference and compatibility (EMI/EMC) testing may be critical to assess the impact of thruster operations on spacecraft communications and payload functionality.
- **Thermal Soakback**: Low thruster efficiencies may result in large thermal loads on the spacecraft due to thermal soakback. Validated thermal modeling should be considered to assess impacts to the host spacecraft.

Figure 4.45: Maxwell Block 1. Credit Phase Four.

c. Missions

The SpaceX Falcon 9 Transporter-1 launch in January 2021 included two SmallSats with the Phase Four Maxwell Block 1 onboard. This integrated propulsion system (figure 4.45) includes the RF thruster and power electronics along with a xenon propellant tank and feed system (199).

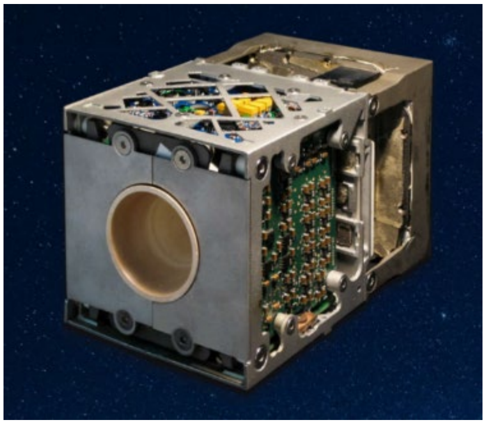


Figure 4.46: REGULUS propulsion module. Credit: T4i.

The UniSat-7 mission, led by GAUSS, is a 36-kg microsatellite that launched via Soyuz-2-1a Fregat in March 2021. This technology demonstration mission included a T4i iodine-propellant REGULUS module (figure 4.46); the integrated propulsion system includes thruster, power processing unit, and heated propellant-feed components. The propulsion demonstration is expected to include orbit raising and lowering between orbital altitudes of 300 and 400 km (201) (202).

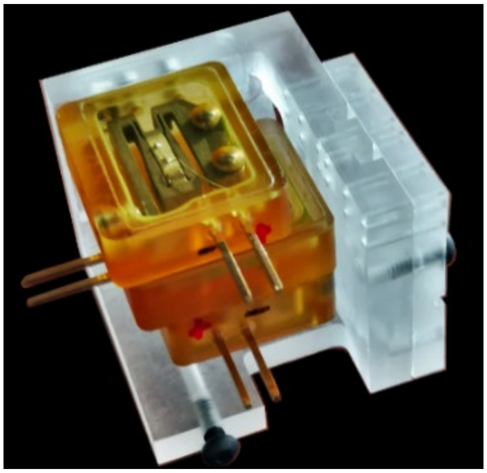


Figure 4.47: ConstantQ thruster head. Credit: Miles Space.

A 6U CubeSat from Team Miles has been awarded a rideshare slot onboard Artemis I, as one of the winning teams in NASA's Cube Quest Challenge. The objective of the mission is to demonstrate deep space communications from beyond a 2.5 million mile range. Twelve ConstantQ water-propellant thrusters (figure 4.47), an earlier version of Team Miles' M1.4 system, are integrated onboard the CubeSat to provide primary propulsion as well as 3-axis control (203) (204).

d. Summary Table of Devices

See table 4-12 for current state-of-the-art ambipolar devices applicable to small spacecraft.

4.6.3 In-Space Propellant-less Propulsion

Propellant-less propulsion systems generate thrust via interaction with the surrounding environment (e.g., solar pressure, planetary magnetic fields, and planetary atmosphere). By contrast, chemical and electric propulsion systems generate thrust by expulsion of reaction mass (i.e., propellant). Three propellant-less propulsion technologies that have undergone in-space demonstrations to date include solar sails, electrodynamic tethers, and aerodynamic drag devices.

Solar Sails

Solar sails use solar radiation pressure to generate thrust by reflecting photons via lightweight, highly-reflective membranes. While no commercial products are presently available, a handful of missions have sought to demonstrate the technology using small spacecraft. Recent missions include:

- NASA's NanoSail-D2 launched as a 3U CubeSat secondary payload onboard the Fast, Affordable, Science and Technology Satellite (FASTSAT) bus in November 2010. The 10 m^2 sail made of CP-1 deployed from a 650 km circular orbit and de-orbited the spacecraft after 240 days in orbit (205).
- The Planetary Society's LightSail 2 mission launched as a 3U CubeSat secondary payload on the Department of Defense's Space Test Program (STP-2) in June 2019. The 32 m^2 mylar solar sail was deployed at 720 km altitude and demonstrated apogee raising of ~10 km. Its mission was still ongoing as of August 2020 (206).
- The University of Illinois (Urbana, IL) and CU Aerospace LLC (Champaign, IL) teamed to develop CubeSail, which launched as one of ten CubeSats on the Educational Launch of Nanosatellites ELaNA-19 mission on a Rocket Lab Electron rocket in December 2018. CubeSail launched as a mated pair of 1.5U CubeSats. When separated, it intended to deploy a 250 m-long, 20 m^2 aluminized mylar film between them. The development team envisions the CubeSail mission as the first of many missions of progressively increasing scale and complexity (207). Satellite beacons at the correct frequency were observed post-launch once on 18 Dec. 2018, but not with sufficient signal to noise ratio to demodulate the call sign in the beacons. No further communications were received from CubeSail. After more than 2 years of continued efforts to establish full communication with CubeSail, it is believed that the satellites irrevocably failed. While it is uncertain the specific cause, the best assessment is that the radios failed in orbit. Due to the lack of communications, CubeSail was never able to attempt sail deployment or attempt to demonstrate sail control and deorbiting (208).
- NASA's Near-Earth Asteroid (NEA) Scout mission is expected to launch as a secondary payload onboard Artemis I. The 6U CubeSat will deploy an 85 m^2 solar sail and conduct a flyby of Asteroid 1991VG, approximately 1 AU from Earth (209).

Electrodynamic Tethers

Electrodynamic tethers employ an extended, electrically conductive wire with current flow. In addition to atmospheric drag on the wire, its interaction with the ambient magnetic field about a planetary body causes a Lorentz force that can be used for orbit raising or lowering. This technology currently provides a means for end-of-mission small spacecraft deorbit.

a. Missions

Georgia Institute of Technology's Prox-1 mission was launched as a secondary payload on the Department of Defense's Space Test Program (STP-2) in June 2019. The 70 kg spacecraft served as the host and deployer for the LightSail 2 mission. The Prox-1 spacecraft housed a Tethers Unlimited Nanosat Terminator Tape (NSTT), shown in figure 4.48, which deployed a 70 m tether in September 2019 to lower the orbit from 717 km. Data from the Space Surveillance Network indicate that the NSTT is causing Prox-1 to deorbit more than 24 times faster than otherwise expected. This rate of orbital decay will enable Prox-1 to meet its 25-year deorbit requirement (210) (211) (212).

Figure 4.48: Nanosat Terminator Tape (NSTT). Credit: Tethers Unlimited.

The Naval Postgraduate School's NPSat-1 was launched as a secondary payload on STP-2 and was expected to deploy its NSTT later in 2020 (212).

TriSept's DragRacer technology demonstration mission, launched as a rideshare onboard an Electron rocket in November 2020, sought to conduct a direct comparison of the deorbiting rates of two Millennium Space Systems satellites, one of which will use a 250 m NSTT (212) (213).

The AuroraSat-1 satellite is anticipated to be launched on an Electron rocket in Q4 2021 (131). The spacecraft is built by SatRevolution with Aurora Propulsion Technologies providing the payloads. The mission serves as a technology demonstration for a Plasma Brake module (figure 4.49), and an Attitude and Orbit Control System (AOCS) (128) (figure 4.22), both produced by Aurora. The Plasma Brake module on AuroraSat-1 is a dual redundant system for demonstration purposes. A 500-m tether will be deployed to demonstrate its deorbiting capability (129).

Figure 4.49: Plasma Brake Module (PBM) demo unit. Credit: Aurora Propulsion Technologies.

Aerodynamic Drag

Satellites have historically deorbited from low-Earth orbits with the aid of thrusters or passive atmospheric drag. Given the increasing rate of new spacecraft launched, and in-turn potential for new orbital debris following completion of missions, orbital debris management has gained increasing attention. Space debris poses a growing threat to active satellites and human activity in space. Allowing decades for defunct spacecraft to decay naturally from low-Earth orbit may soon be insufficient. Aerodynamic drag devices may provide one method to rapidly remove spacecraft from low-Earth orbits upon mission completion.

Below about 1,000 km altitude, the atmosphere exerts a measurable drag force opposite the relative motion of any spacecraft, which results in a slow orbital decay. The intensity of the drag force exerted on the spacecraft depends on numerous factors such as local atmospheric density, the spacecraft forward facing area, the spacecraft velocity, and a drag coefficient. The drag coefficient accounts for the drag force's dependency on an object's unique geometric profile. While the spacecraft velocity and local atmospheric density are largely mission dependent, a spacecraft's forward-facing area and drag coefficient can be altered by introducing aerodynamic drag devices such as exo-brakes and ballutes. These deployable or inflatable parachutes and balloons can greatly increase the drag force exerted on spacecraft by an order of magnitude or more and significantly increase the rate of orbital decay.

Furthermore, aerodynamic drag devices may be useful to reduce spacecraft propellant mass required for orbit capture and disposal at other planetary bodies, given sufficient atmospheric density exists.

For further details on these devices, see chapter on Deorbit Systems.

Table 4-2: Hydrazine Chemical Propulsion												
Manufacturer	Product	Propellant	Thrust per Thruster (Quantity)	Specific Impulse	Total Impulse	Mass	Envelope	Power	ACS	Status	Missions	References
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerojet Rocketdyne	MPS-120	Hydrazine	0.25 – 1.0 (4)	N/A	>2 (2U) >0.8 (1U)	1.6 – 2.5 † 1.2 – 1.5 ‡	1U – 2U	N/A	Y	D	-	(73)
Aerojet Rocketdyne	MPS-125	Hydrazine	0.25 – 1.0 (4)	N/A	>19 (8U) >13 (6U) >7 (4U)	6.2 – 12.1 † 3.6 – 5.1 ‡	4U – 8U	N/A	Y	D	-	(73)
Stellar Exploration	Biprop 12U CubeSat system	Hydrazine/ NTO	3 N	>285	N/A	N/A	N/A	N/A	Y	D	-	(74)
Thruster												
Aerojet Rocketdyne	MR-103	Hydrazine	1 N	202-224	183	0.33-0.37	-	16 max total	-	F	numerous	(8)
Aerojet Rocketdyne	MR-111	Hydrazine	4 N	219-229	262	0.37	-	16 max total	-	F	numerous	(8)
Aerojet Rocketdyne	MR-106	Hydrazine	22 N	228-235	561	0.59	-	36 max total	-	F	numerous	(8)
ArianeGroup	1 N	Hydrazine	1 N	200 – 223	135	0.29	-	N/A	-	F	numerous	(6)
Moog	MONARC-1	Hydrazine	1 N	227.5	111	0.38	-	18 (Valve)	-	F	numerous	(9)
Moog	MONARC-5	Hydrazine	4.5 N	226.1	613	0.49	-	18 (Valve)	-	F	numerous	(9)
Moog	MONARC-22	Hydrazine	22 N	228-229	533 – 1,173	0.69-0.72	-	30 (Valve)	-	F	numerous	(9)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available

Table 4-3: Alternative Monopropellant and Bipropellant Propulsion												
Manufacturer	Product	Propellant	Thrust per Thruster (Quantity)	Specific Impulse	Total Impulse	Mass	Envelope	Power	ACS	Status	Missions	References
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerojet Rocketdyne	MPS-130	AF-M315E	0.25 – 1.0 (4)	N/A	>2.7 (2U) >1.1 (1U)	1.7 – 2.8 † 1.1 – 1.4 ‡	1U – 2U	N/A	Y	D	-	(72) (73)
Aerojet Rocketdyne	MPS-135	AF-M315E	0.25 – 1.0 (4)	N/A	>19 (8U) >13.7 (6U) >7.3 (4U)	7.2 – 14.7 † 3.5 – 5.1 ‡	4U – 8U	N/A	Y	D	-	(73)
Aerospace Corp.	HyPer	Hydrogen Peroxide	N/A	N/A	N/A	N/A	~0.25U	N/A	N/A	D	-	(75)
Benchmark Space Systems	Halcyon	HTP & Alcohol	100 mN-22 N	270	1.7-10	2.5-7.5†	2000 – 7800 cm^3	up to 10 W	Y	F	Tenzing-01 (2021)	(25) (76) (77)
Bradford-ECAPS	Skysat 1N HPGP Propulsion System	LMP-103S	1.0 (4)	200	>17	17	27U	10	Y	F	Skysat, PRISMA	(15) (84)
Busek	BGT-X5 System	AF-M315E	0.5	220 – 225	N/A	1.5 (BOL)	1U	20	N	D	-	(85)
Cornell Univ.	Cislunar Explorer	Water (Electrolysis)	N/A	N/A	N/A	N/A	6U total (2-units)	N/A	N/A	E	CubeQuest Challenge (Artemis I)	(21)
CU Aerospace	MPUC	(CMP-8) Peroxide/ Ethanol blend	0.1 (1)	160 – 180	2.5	1.277 † 0.650 ‡	1U	3	N	D	-	(80) (87)
Dawn Aerospace	PM200	Nitrous Oxide & Propene	0.5 (1)	>285	>0.4 – 0.8	1.0 – 1.4	0.7 – 1U	12	Y	D	-	(27)
Moog	Monopropellant Propulsion Module	Green or 'Traditional'	0.5 (1)	224	0.5	1.01†	1U (baseline)	2 x 22.5 W/Thruster	N	D	-	(82)
MSFC	LFPS	AF-M315E	0.1 (4)	>200s	>3.5	<5.5kg	~2.4U	15 – 47W*	Y	E	Lunar Flashlight (Artemis I)	(17)
NanoAvionics	EPSS C1K	ADN-blend	1.0 (1) BOL 0.22 (1) EOL	213	>0.4	1.2 † 1.0 ‡	1.3U	0.19 (monitor) 9.6 (preheat) 1.7 (firing)	N	F	Lituanica-2	(26)
Rocket Lab	Kick Stage	Unk.	120	N/A	N/A	N/A	N/A	N/A	Y	F	Electron 'Still Testing'	(29) (30)
Tethers Unlimited	HYDROS-C	Water (Electrolysis)	1.1 (1)	>310	>2	2.61 † 1.87 ‡	190 mm x 130 mm x 92 mm	5-25	N	F	Pathfinder Technology Demonstration	(24) (81) (88)
Tethers Unlimited	HYDROS-M	Water (Electrolysis)	>1.2 (1)	>310	>18	12.6 † 6.4 ‡	381 mm dia. x 191 mm	7-40	N	D	-	(81)
VACCO	ArgoMoon Hybrid MiPS	LMP-103S/ cold-gas	0.1 (1)	190	1	14.7 † 9 ‡	~1.3U	13.6 20 (max)	Y	E	ArgoMoon (Artemis I)	(57) (91)
VACCO	Green Propulsion System (MiPS)	LMP-103S	0.1 (4)	190	4.5	5 † 3 ‡	~3U	15 (max)	Y	D	-	(57) (89)
VACCO	Integrated Propulsion System	LMP-103S	1.0 (4)	200	12.5	14.7 † 9 ‡	~1U – 19,000 cm^3	15 – 50 (max)	Y	E	-	(57) (90)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available

Table 4-3 (cont.): Other Monopropellant and Bipropellant Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust per Thruster (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**Status**	**Missions**	**References**
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems (cont.)												
VACCO	Green Propulsion System (MiPS)	LMP-103S	0.1 (4)	190	4.5	5 † 3 ‡	~3U	15 (max)	Y	D	-	(57) (89)
VACCO	Integrated Propulsion System	LMP-103S	1.0 (4)	200	12.5	14.7 † 9 ‡	~1U – 19,000 cm^3	15 – 50 (max)	Y	D	-	(57) (90)
Thruster Heads												
Aerojet Rocketdyne	GR-M1	AF-M315E	0.25	195	3.45	--	--	7	-	D	-	(31)
Aerojet Rocketdyne	GR-1	AF-M315E	0.4-1.1	231	23	N/A	-	12	-	F	GPIM	(8) (13)
Aerojet Rocketdyne	GR-22	AF-M315E	8.0-25	248	74	N/A	-	28	-	E	GPIM	(8) (13)
Aerospace Corp.	Hydrogen Peroxide Vapor Thruster (HyPer)	Hydrogen Peroxide	<10 mN	N/A	N/A	N/A	-	N/A	-	D	-	(75)
Bradford-ECAPS	0.1 N HPGP	LMP-103S	0.03 – 0.10	196 – 209	N/A	0.04 excl. FCV	-	6.3 – 8	-	E	ArgoMoon	(78)
Bradford-ECAPS	1 N HPGP	LMP-103S	0.25 – 1.0	204 – 235	N/A	0.38	-	8 – 10	-	F	SkySat	(15) (78)
Bradford-ECAPS	1 N GP	LMP-103S/LT	0.25 – 1.0	194 – 227	N/A	0.38	-	8 – 10	-	D	-	(79)
Bradford-ECAPS	5 N HPGP	LMP-103S	1.5 – 5.5	239 – 253	N/A	0.48	-	15 – 25	-	D	-	(78)
Bradford-ECAPS	22 N HPGP	LMP-103S	5.5 – 22	243 – 255	N/A	1.1	-	25 – 50	-	D	-	(78)
Busek	BGT-X1	AF-M315E	0.02 – 0.18	214	N/A	N/A	-	4.5	-	D	-	(86)
Busek	BGT-X5	AF-M315E	0.50	220 – 225	0.5	1.5†	1U	20	-	D	-	(85) (86)
Busek	BGT-5	AF-M315E	1.0 – 6.0	> 230	N/A	N/A	-	50	-	D	-	(86)
Dawn Aerospace	20N Thruster	N20/Propene	7.3 – 19.8N	>285		0.4	-	12W	-	F	numerous	(33)
NanoAvionics	EPSS-C1	ADN-blend	0.22 – 1.0	213	>0.4	N/A	-	9.6 (preheat) 1.7 (firing)	-	F	Lituanica-2	(26)
Plasma Processes	100mN Thruster PP3490-B	AF-M315E	0.1 – 0.17	195 - 208	N/A	.08	-	7.5 – 10	-	E	Lunar Flashlight	(17)
Rocket Lab	Curie Engine	unk.	120	N/A	N/A	N/A	-	N/A	-	F	Electron 'Still Testing'	(29) (30)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available

Table 4-4: Hybrid Chemical Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**Status**	**Missions**	**References**
---	---	---	[N]	[s]	[N-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Aerospace Co.	Propulsion Unit for CubeSats	Paraffin/Nitrous Oxide	N/A	N/A	N/A	N/A	1U	N/A	-	D		(41)
JPL	Hybrid Rocket	PMMA/GOX	N/A	>300	N/A	N/A	N/A	N/A	-	D	-	(38) (93) (94) (95)
NASA Ames	Hybrid Rocket	PMMA/ Nitrous Oxide	25	247	N/A	N/A	N/A	N/A	-	D		(39)(40) (94)
Parabilis	ROMBUS	Various/N2O	222	260s	Configurable	N/A	ESPA, ESPA Grande	N/A	Y	D		(42)
Parabilis	NanoSat Obrital Transfer System	HTPB/N2O	9.4	245s	N/A	3U OTS	Modular, 3U to 50kg sat	N/A	Y	C		(96)
Utah State Univ.	Green Hybrid Rocket	ABS/Nytrox	25-50	220-300	N/A	N/A	3-25U	<30W for 1-2 sec	Y	D		(36)(37)
Utah State Univ.	Green Hybrid Rocket	ABS/GOX	8	215	N/A	N/A	N/A	N/A	-	D	-	(34) (35) (92)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available

Table 4-5: Cold and Warm Gas Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**Status**	**Missions**	**References**
---	---	---	[mN]	[s]	[N-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerospace Corp.	MEPSI	R236fa	20	N/A	N/A	0.188	4 in. x 4 in. x 5in.	N/A	Y	E	STS-113 and STS-116	(44)
GomSpace / NanoSpace	Nanoprop CGP3	Butane	0.01 – 1 (x4)	60-110	40	0.3‡ 0.35†	0.5U	<2	Y	D	-	(50) (108)
GomSpace / NanoSpace	Nanoprop 6U	Butane	1 – 10 (x4)	60-110	80	0.770‡ 0.900†	200 mm x 100 mm x 50 mm	<2	Y	F	GomX-4	(50) (51) (109)
Lightsey Space Research	BioSentinel Propulsion System	R236fa	40 - 70	40.7	79.8	1.08 kg ‡ 1.28 kg †	220 mm x 100 mm x 40 mm	<1 W idle <4 W operating	Y	E	BioSentinel	(52) (53)
Marotta	MicroThruster	Nitrogen	0.05 – 2.36 N	70	N/A	N/A	N/A	<1	N/A	F	numerous	(43)
Micro Space	POPSAT-HIP1	Argon	0.083 – 1.1 (x8)	43	N/A	N/A	N/A	N/A	N/A	F	POPSAT-HIP1	(49)
SSTL	Butane Propulsion System	Butane	0.5 N							D	-	(45) (46)
ThrustMe	I2T5	Iodine	0.2		75	0.9†	0.5U	10	N	F	Xiaoxiang 1-08, Robusta-3A (2021**)	(62) (63) (64) (65)
UTIAS/SFL	CNAPS	Sulfur Hexafluoride	12.5 – 40	30	81	N/A	N/A	N/A	N	F	CanX-4/CanX-5	(110) (111)
VACCO	NEA Scout	R236fa	N/A	N/A	500	2.54†	2U	9	Y	E	NEA Scout (2021**)	(60) (61)
VACCO	MiPS Standard Cold Gas	R236fa	25 (x4)	40	98 – 489	553 – 957‡	0.4 – 1.38U	12 W (max)	Y	D	-	(57) (105)
VACCO	MarCO-A and -B MiPS	R236fa	25 (x8)	40	755	3.5	2U	15	Y	F	MarCO-A & -B	(57) (58) (59) (106)
VACCO	C-POD	R134A	25 (x8)	40	186	1.3	0.8U	5	Y	E	CPOD	(57) (107)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available, ** anticipated launch date

Table 4-6: Solid Motor Chemical Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**Status**	**Missions**	**References**
---	---	---	[N]	[s]	[N-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
D-Orbit	D-Raise	N/A	N/A	N/A	N/A	50 – 78	N/A	N/A	N	D	-	(103)
D-Orbit	D3	N/A	N/A	N/A	N/A	16 – 257	32 cm x 32 cm x 25 cm to 1100 cm x 500 cm x 1000 cm	N/A	N	D	-	(104)
DSSP	CAPS-3	HIPEP-501A	0.3 (3)	N/A	0.125	0.023	0.92 cm x 2.79 cm x 4.2 cm	< 2.3	N	F	SPINSAT	(67) (97)
DSSP	MPM-7	HIPEP-H15	N/A	200	1.5	<750 g (PPU)	< 0.75 U	200	N	D	-	(98)
PacSci EMC	MAPS	N/A	N/A (176 per lightband)	210	N/A	N/A	38 cm x 10.5 cm	N/A	N/A	F	PACSCISAT	(68) (69)
PacSci EMC	P-MAPS	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	D	-	(68)
Thruster Heads												
DSSP	CDM-1	AP/HTPB	186.8	235	226.4	0.046	0.64 dia x 0.47 length	< 5	-	D	Listed as "flight qualified"	(99) (100)
Industrial Solid Propulsion	ISP 30 sec. Motor	80% Solids HTPB/AP	37	187	996	0.95	5.7 cm	-	-	D	Optical target at Kirtland AFB	(66) (101)
Northrop Grumman (Former Orbital ATK)	STAR 4G	TP-H-3399	258	276	595	1.49	11.3 cm dia. x 13.8	-	-	D	-	(66) (102)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
† denotes a wet mass, ‡ denotes a dry mass, N/A = Not Available

Table 4-7: Electrothermal Electric Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**ACS**	**Status**	**Missions**	**References**
---	**---**	**---**	**[mN]**	**[s]**	**[N-s]**	**[g]**	**[cm^3 or U]**	**[W]**	**Y/N**	**C,D,E,F**	**---**	**---**
Integrated Propulsion Systems												
Aurora Propulsion Technologies Finland	AOCS	H_2O	0.5	100	70	280†	0.3U	10£	Y	E	AuroraSat-1 (2021**)	(128) (129) (131)
Busek USA	Micro Resistojet	Ammonia	10	150	404	1,250†	1U	15	Y	D	---	(214)
Bradford Space Netherlands	Comet-1000	H_2O	17	175	1,155	1,440†	2,600	55	N	F	HawkEye 360, Capella Space	(114) (115) (116)
Bradford Space Netherlands	Comet-8000	H_2O	17	175	8,348	6,675†	23,760	55	N	F	BlackSky Global	(114) (117)
CU Aerospace and VACCO USA	CHIPS	R134a	31	76	478	1,375†	1U	30	Y	D	---	(215) (216) (217) (218)
CU Aerospace and VACCO USA	CHIPS	R236fa	23	60	433	1,510†	1U	30	Y	D	---	(215) (216) (217) (218)
CU Aerospace and VACCO USA	PUC	SO_2	4.5	70	184	718†	0.35U	15	N	E	8 flight units delivered to AFRL	(118) (119) (120)
CU Aerospace USA	MVP	Delrin Fiber	4.5	66	334	1,140†	1.15U	45	N	E	DUPLEX (launch mid-2022**)	(121) (122)
Thruster Heads												
Sitael Italy	XR-150	Xe	65	57	NA	220‡	21.6	100	NA	D	---	(219) (220)
Sitael Italy	XR-150	Kr	67.2	70	NA	220‡	21.6	100	NA	D	---	(219) (220)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, £ per active thruster, NA = Not Applicable

Table 4-8: Electrospray Electric Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**Neutralizer**	**Status**	**Missions**	**References**
---	---	---	[μN]	[s]	[N-s]	[kg]	[cm^3 or U]	[W]	---	C,D,E,F	---	---
Integrated Propulsion Systems												
Accion Systems USA	TILE-2	EMI-BF4 (ionic)	50	1,650	35	0.45†	0.5U	4	NA	F	Astro Digital Tenzing, BeaverCube (2021**)	(145) (146) (147) (233)
Accion Systems USA	TILE-3	EMI-BF4 (ionic)	450	1,650	755	2.25†	1U	20	NA	F	D2/AtlaCom-1	(147) (148) (149) (234)
Busek USA	CMNT (4x heads)	EMI-Im (ionic)	4 x 20	225	980	14.8†	29U	16.5	Carbon Nanotube	F	LISA Pathfinder	(132)
Busek USA	BET-300-P (4x heads)	EMI-Im (ionic)	4 x 55	850	360	0.8†	1150	15	Carbon Nanotube	D	---	(221) (222) (223) (224) (225)
Enpulsion Austria	IFM Nano	Indium (FEEP)	350	3,500	---	0.90†	10 x 10 x 8.3	40	Thermionic	F	Flock 3p', ICEYE X2, Harbinger, NetSat	(133) (134) (135) (136) (137) (138) (139) (226) (227) (228)
Enpulsion Austria	IFM Micro R^3	Indium (FEEP)	1,000	3,000	---	3.9†	14 x 12 x 13.3	100	Thermionic	F	GMS-T	(140) (229) (230)
Morpheus Space Germany	NanoFEEP (2x heads)	Gallium (FEEP)	<40	---	---	0.16‡	9 x 2.5 x 4.3	<3	Propellant-less	E	UWE-4	(141) (142) (231) (232)
Morpheus Space Germany	MultiFEEP (2x heads)	Gallium (FEEP)	<140	---	---	0.28‡	9 x 4.5 x 4.5	<19	Propellant-less	D	---	(231)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, NA = Not Applicable

Table 4-9: Gridded-Ion Electric Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**Cathode Type**	**Status**	**Missions**	**References**
---	**---**	**---**	**[mN]**	**[s]**	**[kN-s]**	**[kg]**	**[cm^3 or U]**	**[W]**	**---**	**C,D,E,F**	**---**	**---**
Integrated Propulsion Systems												
Avant Space [Russia]	GT-50 [RF]	Xenon	<7	---	---	<8†	<4U	<240	Hollow	D	---	(235) (236)
Busek [USA]	BIT-3 [RF]	Iodine	1.15	2,100	32	2.9† (with gimbal)	18 x 8.8 x 10.2	75	RF	E	Lunar IceCube (2021**); LunaH-Map (2021**)	(156) (157) (158) (237) (238) (239)
ThrustMe [France]	NPT30 [RF]	Xenon	<1.1	---	---	<1.7†	<2U	<60	Thermionic	D	---	(240)
ThrustMe [France]	NPT30-I2 [RF]	Iodine	<1.1	---	---	1.2† (1U) or 1.7† (1.5U)	1U or 1.5U	<65	Thermionic	F	Beihangkongshi-1; NORSAT-TD (2022**); GOMX-5 (2022**)	(153) (154) (155) (241) (242) (243)
Thruster Heads												
Ariane Group [Germany]	RIT µX [RF]	Xenon	<0.5	---	---	0.44‡	7.8 x 7.8 x 7.6	<50	RF	D	---	(244) (245) (246) (247)
Ariane Group [Germany]	RIT 10 EVO [RF]	Xenon	<15	---	---	1.8‡	18.6 x 18.6 x 13.4	<435	Hollow	E	(Identical to flight-heritage RIT-10 with contemporary grid design)	(244) (246) (248)
QinetiQ [UK]	T5 [DC]	Xenon	<20	<3,000	---	2‡	19 x 19 x 24.2	<600	Hollow	F	GOCE	(151) (152) (249) (250)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, NA = Not Applicable, RF = Radio Frequency

Table 4-10: Hall-Effect Electric Propulsion Thrusters												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Thruster Power***	**Cathode Type**	**Status**	**Missions**	**References**
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm^3]	[W]	Notes	C,D,E,F	---	---
Apollo Fusion USA	ACE	Xenon	22	1,300	200	1.0	---	400‡	CM-HL	F	Sherpa-LTE (2021)	(181) (182) (251)
Apollo Fusion USA	ACE	Krypton	16	1,200	200	1.0	---	400‡	CM-HL	D	---	(251)
Apollo Fusion USA	ACE	Proprietary	24	1,250	---	1.0	---	400‡	CM-HL	E	Astro Digital Ignis (2020**)	(179) (180)
Busek USA	BHT-100	Xenon	6.3	1,086	150	1.2	275 wo cath.	105	EM-SH	D	---	(168) (252)
Busek USA	BHT-200	Xenon	13	1,390	84§	1.2	675 wo cath.	250‡	EM-SH	F	TacSat-2, FalconSat-5, -6	(168) (169) (253) (254)
Busek USA	BHT-200-I	Iodine	14	1390	---	1.2	675 wo cath.	250	EM-SH	E	NASA iSat (Cancelled)	(169) (170) (253)
Busek USA	BHT-600	Xenon	39	1,500	1000§	3.3	1,470 wo cath.	680‡	EM-SH	E	US Government (2021**)	(168) (195) (255) (256)
Busek USA	BHT-600-I	Iodine	39	---	---	3.3	1,470 wo cath.	600	EM-SH	D	---	(169) (255) (256) (257)
EDB Fakel Russia	SPT-50	Xenon	14	860	126§	1.2	1,092	220	EM-SH	F	Canopus-V	(163) (164) (165) (166) (258)
EDB Fakel Russia	SPT-50M	Xenon	14.8	930	266	1.3	---	220	EM-SH	D	---	(258)
EDB Fakel Russia	SPT-70BR	Xenon	39	1,470	435§	2.0	1,453	660	EM-SH	F	KazSat-1, KazSat-2	(166) (167)
EDB Fakel Russia	SPT-70M	Xenon	41.3	1,580	---	---	---	660	EM-SH	D	---	(167)
EDB Fakel Russia	SPT-70M	Krypton	31.3	1,460	---	---	---	660	EM-SH	D	---	(167)
ExoTerra USA	Halo	Xenon	7.1	1,110	100	0.67	220	185	CM-HL	E	Tipping Point (2021**)	(188) (189) (190)
Exotrail France	ExoMG nano	Xenon	2.0	800	5	---	---	53	EM-SH	F	M6P Demo (2020**), ELO3 and ELO4 (2021**)	(183) (184) (185) (186) (187)
Exotrail France	ExoMG micro	Xenon	5	1,000	19	---	---	100	EM-SH	D	---	(183) (186)
JPL USA	MaSMi	Xenon	55	1,920	3,000	3.4	1,700	1,000	CM-HL	D	---	(259) (260) (261) (262) (263) (264) (265) (266) (267)
Orbion USA	Aurora	Xenon	12	1,220	200	1.5	1,147	200	EM-SH	E	AST SpaceMobile (2022), DARPA Blackjack (**)	(191) (192) (193) (194) (268)
Rafael Israel	R-200HT	Xenon	---	---	---	---	---	200	EM-HL	D	---	(171)
Rafael Israel	IHET-300	Xenon	>14.3	>1,210	>135	1.5	1,836	300	EM-SH	F	VENuS	(171) (172) (173) (174)
Rafael Israel	R-800HT	Xenon	---	---	560	---	---	800	EM-HL	D	---	(171)
Safran France	PPS-X00	Xenon	43	1,530	1,000	---	---	650	EM-SH	D	---	(269)
SITAEL Italy	HT100	Xenon	9	1,300	73	---	407 wo cath.	175	EM-SH	E	uHETSat (2021**)	(176) (177) (178)
SITAEL Italy	HT400	Xenon	27.5	1230	1,000	2.77	1,330	615	EM-SH	D		(270) (271) (272)
SETS Ukraine	ST25	Xenon	7.6	1,000	82	0.75	1,003	140	EM-SH	D	---	(273) (274)
SETS Ukraine	ST40	Xenon	25	1,450	450	1.1	1,170	450	EM-HL	D	---	(275)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, ‡ PPU input power, § demonstrated, CM = Center Mounted, EM = Externally Mounted, SH = Swaged Heater, HL = Heater-less, JPL = Jet Propulsion Laboratory, SETS = Space Electric Thruster Systems, EDB = Experimental Design Bureau

Table 4-11: Pulsed Plasma and Vacuum Arc Electric Propulsion													
Manufacturer	Product	Propellant	Thrust*	Impulse Bit	Specific Impulse*	Total Impulse*	Mass	Envelope	Power*	ACS	Status	Missions	References
---	---	---	[µN]	[µNs]	[s]	[N-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems													
Applied Sciences Corp. USA	Metal Plasma Thruster	Molybdenum	600	150	1,756	4,000	0.85	0.7U	50	N	D	---	(276)
Busek USA	BmP-220	PTFE	20	20	---	175	0.5	375 + ESV	3	N	D	---	(277)
Comat France	Plasma Jet Pack	(metal)	288	29	---	4,000	1.0	1U	30	N	D	---	(278) (279)
CU Aerospace USA	FPPT-1.6	PTFE Fiber	270	180	2,400	20,700	2.8†	1.6U	48	N	E	DUPLEX (launch mid-2022**)	(123) (124) (125)
Mars Space Ltd UK Clyde Space Sweden	PPTCUP	PTFE	40	40	655	48	0.27	0.33U	2.7	N	D	---	(280)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, NA = Not Applicable, ESV = Ejector Spring Volume

Table 4-12: Ambipolar Electric Propulsion												
Manufacturer	Product	Propellant	Thrust*	Specific Impulse*	Total Impulse*	Mass	Envelope	Power	ACS	Status	Missions	References
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm³]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Phase Four USA	Maxwell (Block 1)RF	Xenon	6	400	5	5.9‡	19 x 13.5 x 19	450	N	F	Capella	(199) (200) (281) (282) (283) (284)
Phase Four USA	Maxwell (Block 3)RF	Xenon	5.5	800	---	---	22 x 12 x 24 (without tank)	450	N	D	---	(284)
T4i Italy	REGULUSRF	Iodine	0.55	550	3	2.5†	1.5U	50	N	F	UniSat-7	(201) (202) (285)
Miles Space USA	M1.4	Water	2.8	1340	3.3	0.8†	9 x 9 x 9.5	<11.5	N	E	Team Miles (2021**)	(203) (204) (286)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, NA = Not Applicable, RF = Radio Frequency

Table 4-13: Propellant-less Propulsion												
Manufacturer	Product	Propellant	Thrust*	Specific Impulse*	Total Impulse*	Mass	Envelope	Power	ACS	Status	Missions	References
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm³]	[W]	Y/N	C,D,E,F	---	---
Aurora Finland	Plasma Brake Module	NA	---	NA	NA	<1	1U	<4	N	E	AuroraSat-1 (2021**)	(129) (131) (287)
Tethers Unlimited USA	NSTT	NA	---	NA	NA	0.81	18 x 18 x 1.8	---	N	F	Prox-1, NPSat-1, DragRacer	(210) (211) (212) (288)

Note that all data is documented as provided in the references. Unless otherwise published, do not assume the data has been independently verified.
*nominal values (see references for full performance ranges), ** anticipated launch date, † denotes a wet mass, ‡ denotes a dry mass, NA = Not Applicable
See Chapter on Passive Deorbit Systems for review of aerodynamic drag devices.

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