State-of-the-Art
Small Spacecraft Technology

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January 2025

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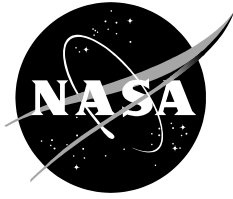
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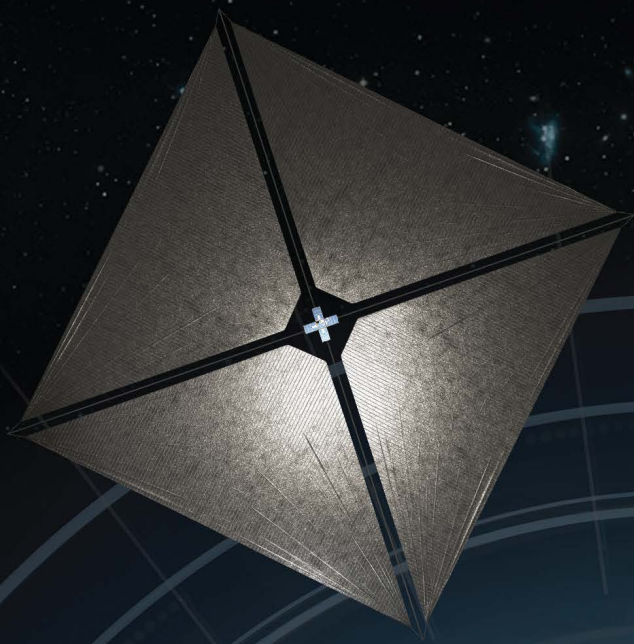This report is also available in electronic form at

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SMALL SPACECRAFT SYSTEMS VIRTUAL INSTITUTE (S3VI)

Small Spacecraft Technology
State-of-the-Art Report
2025 Edition

2025 Cover Page Image Credits

Left Vehicle

Advanced Composite Solar Sail System (ACS3) spacecraft artistic image.
Credit: NASA.

Right Vehicle

Burstcube spacecraft.
Credit: NASA.

NASA Ames Research Center, Small Spacecraft Systems Virtual Institute

January 2025

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Contents

Change Summary

Published Date	Edition	Chapter	Description of Changes
January 2025	2025	Complete Spacecraft Platforms	Updates to Hosted Payloads and Spacecraft Bus sections, and all technology tables.
		Power	All technology tables updated.
		In-space Propulsion	Edits pending next edition.
		Guidance Navigation & Control	Chapter updates; Formation Flying and Rendezvous and Proximity Operations section included.
		Structures, Materials, and Mechanisms	The Additive Manufacturing section updated.
		Thermal Control	Edits pending next edition.
		SmallSat Avionics	On-board Computing Systems table updated.
		Communications	Free Space Optical Communications section updated.
		Launch, Integration, Deployment, and Orbital Transport	Minor edits throughout the chapter; Orbital Maneuvering Vehicle section updated.
		Ground Data Systems and Mission Operations	Updated content in Ground Segment Services, Ground Station Components, Ground Data and Supporting Systems sections.
		ID and Tracking	Minor edits throughout chapter.
		Deorbit Systems	Passive and Active Systems sections updated.

Preface

NASA's *Small Spacecraft Technology State-of-the-art* report is updated annually to capture new information on publicly available small spacecraft systems from NASA and other sources. Each chapter captures the development status of current state-of-the-art SmallSat technologies, along with design considerations for the reader to consider when identifying components for their mission. The organizational approach for each chapter includes an introduction of the technology, current development status of the technology's procurable systems, and summary tables of the technologies surveyed. In this way, each chapter presents a stand-alone report on the subject spacecraft subsystem, with updated information on new and maturating technologies and reference missions as applicable.

When the first edition of this report was published in 2013, 247 CubeSats and 105 other non-CubeSat small spacecraft under 50 kilograms (kg) had been launched worldwide that represented less than 2% of launched mass into orbit over multiple years. Small satellite flight heritage has greatly increased since then as they have become the primary way for commercial, government, private, and academic institutions to access space. Since 2023 there has been an influx of constellations of mini-class small spacecraft with a mass of 201 – 600 kg, as well as a new generation of larger small spacecraft constellations weighing 600 – 1,200 kg (1). While updates in all chapters reflect this growth in the small spacecraft market, a focused effort was made to update areas with recent technology developments that may ultimately bridge existing technology gaps.

The current edition of the report features updates for hosted payload services (see Platforms chapter); Orbital Maneuvering/Transport Vehicle (OMV/OTV) services (see Integration, Launch, Deployment, and Orbital Transport chapter); Formation Flying and Rendezvous and Proximity Operations (see Guidance, Navigation, and Control chapter); and passive and active deorbit systems (see Deorbit Systems chapter) as these are rapidly growing fields which are now extensively represented at small spacecraft conferences. Although still in their infancy, commercial launch service providers are beginning to expand their offerings to also include In-Space Servicing, Assembly, and Manufacturing (ISAM), as well as mission modeling and simulation services. As these services evolve and reliable conventions and standards emerge, they will likely be integrated with additional details in future editions of this report.

This report should not be considered as a comprehensive overview of all the technologies but a rather general overview for the current state-of-the-art SmallSat technologies and their development status. It should be noted that technology maturity designations may vary with changes to payload, mission requirements, reliability considerations, and/or the environment in which performance was demonstrated. Readers are highly encouraged to contact companies for further information regarding the performance and maturity of the described technology. Any companies mentioned in this report are for informational purposes only and do not constitute an endorsement by NASA.

References

(1)   Bryce and Space Technology. "SmallSat by the Numbers, 2024." Accessed September 28, 2024. [Online] Available at: https://brycetech.com/reports/report-documents/Bryce_Smallsats_2024.pdf

Chapter Glossary

(EELV)	Evolved Expendable Launch Vehicle
(ESPA)	EELV Secondary Payload Adapter
(FASTSAT)	Fast, Affordable, Science and Technology Satellite
(LADEE)	Lunar Atmosphere and Dust Environment Explorer
(LCROSS)	Lunar Crater Observation and Sensing Satellite
(NODIS)	NASA Online Directives Information System
(SST)	Small Spacecraft Technology
(STMD)	Space Technology Mission Directorate
(TMA)	Technology Maturity Assessment
(TRL)	Technology Readiness Level
(U)	Unit

1.0   Introduction

1.1   Objective

The objective of this report is to assess and provide an overview of the state of the art in small spacecraft technologies for use by mission designers, project managers, technologists, and students, connecting current small spacecraft missions to available technologies. This report focuses on the spacecraft system in its entirety, provides current best practices for integration and key considerations for the reader where possible, and presents devices from publicly available sources for each specific spacecraft subsystem.

This report is a survey of small spacecraft technologies sourced from open literature; it is not an original source. In addition, this report only considers literature in the public domain. Information presented in this report is limited to SmallSat technology that is publicly available as of September 30, 2024. It does not include information on instrumentation, science payloads, or advances or developments that have not been publicly disclosed. Commonly used sources for data include manufacturer datasheets, press releases, conference papers, journal papers, public filings with government agencies, news articles, presentations, the compendium of databases accessed via NASA's Small Spacecraft Systems Virtual Institute (S3VI) Information Search, and engagement with companies. Data not appropriate for public dissemination, such as proprietary, export controlled, or otherwise restricted data, are not considered. We encourage the SmallSat community to publish mission outcomes and technology development milestones in publicly available conference papers, press releases, or company websites so they can be reflected in future editions of this report.

This report is funded by NASA's Space Technology Mission Directorate (STMD). It was first commissioned by the Small Spacecraft Technology (SST) program within NASA's STMD in mid-2012 in response to the rapid growth in interest in using small spacecraft for low-Earth orbit, low-cost missions. The report was subsequently updated in 2015, 2018, 2020, 2021, 2022, and 2023 to capture SmallSat technology growth and maturation. In addition to reporting currently available state-of-the-art technologies that have achieved Technology Readiness Level (TRL) 5 or above, a prognosis is provided describing technologies as "on the horizon" if they are being considered for future application.

1.2   Scope

The NASA-SmallSat era began at NASA Ames Research Center with the launch of Pioneer 10 and 11 in March 1972 and April 1973, respectively, where both spacecraft weighed < 600 kg. The NASA SmallSat mission gained momentum when NASA initiated the Small Explorer (SMEX) program in 1988 to encourage the development of small spacecraft with masses in the range of ~60–350 kg to reduce cost. In 1998 NASA Ames focused its SmallSat program on lunar exploration and launched Lunar Prospector (< 700 kg), followed by the Lunar Crater Observation and Sensing Satellite (LCROSS), (< 630 kg) in 2009, and the Lunar Atmosphere and Dust Environment Explorer (LADEE), (~380 kg) which was launched in September 2013. In late 2010, NASA launched its first minisatellite called Fast, Affordable, Science and Technology Satellite (FASTSAT), which had a launch mass ~180 kg. This decrease in spacecraft mass, reduced overall cost, and increase in science capabilities ignited interest in miniaturization and maturity of aerospace technologies which have proven to be capable of producing more complex missions for less cost.

In 2012, the Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (ESPA) provided a modular packaging approach for six payload slots of up to 180 kg mass allocation. As this report is focused on smaller platforms, the "180 kg mass limit" served as a convenient metric

to further classify the maximum "SmallSat" mass. SmallSats are generally grouped according to their mass, and this report adopts the following five small spacecraft mass categories (1):

- minisatellites are spacecraft with a total mass of 100 – 180 kg;

- microsatellites have a total spacecraft mass of 10-100 kg;

- nanosatellites have a total mass of 1 – 10 kg;

- picosatellites have a mass of 1 – 0.01 kg; and

- femtosatellites have a total spacecraft mass 0.01 – 0.09 kg.

Figure 1.1 offers examples of spacecraft categorized by mass. On the lower mass end, there are projects such as KickSat-2, which deployed 100-centimeter (cm) scale "ChipSat" spacecraft, or Sprites, from a 2U femtosatellite deployer in March 2019. These femtosatellite ChipSats are the size of a large postage stamp and have a mass below 10 grams.
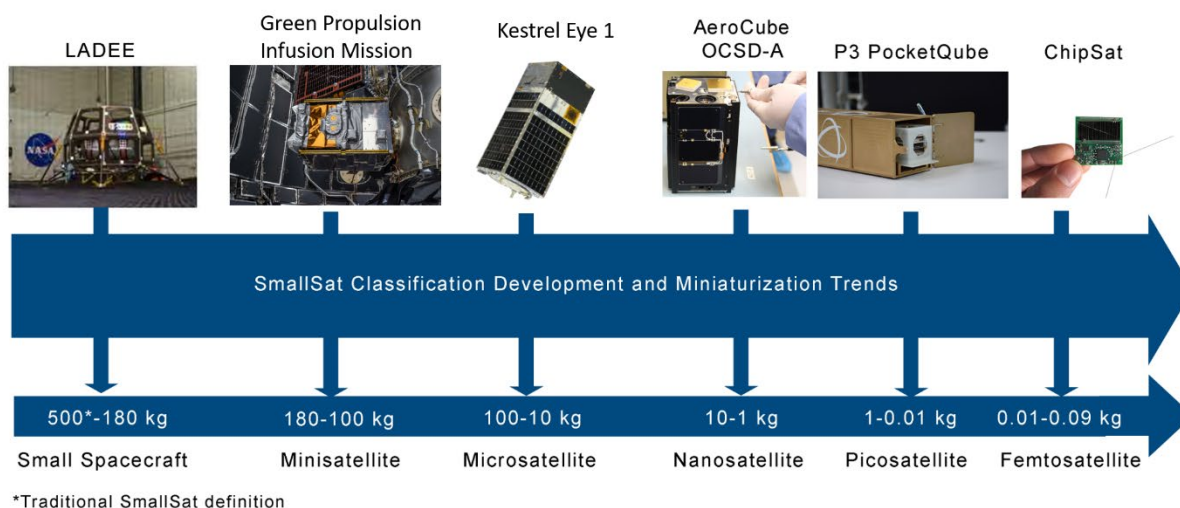
Figure 1.1: Overview of small spacecraft categories. Credit: NASA, SpaceX, Redwire Space, and Alba Orbital.

In 1999, a collaboration between California Polytechnic State University (Cal Poly) in San Luis Obispo and Stanford University in Stanford, California, developed a small educational platform called a "CubeSat" which was designed for academic space exploration and research. CubeSats are now a common form of small spacecraft that can weigh only a few kilograms and are based on a form factor of a 10 cm square cube, or unit (U) (1). While the original CubeSat was composed of a single 1U cube, in 2014 the CubeSat form factor expanded to 6U, and it is now common to combine multiple cubes to form larger units, as shown in Figure 1.2. These larger CubeSat sizes have become more standardized and popular in the past few years as much more science can be achieved at less cost with the additional volume, power, and overall increase in capability.
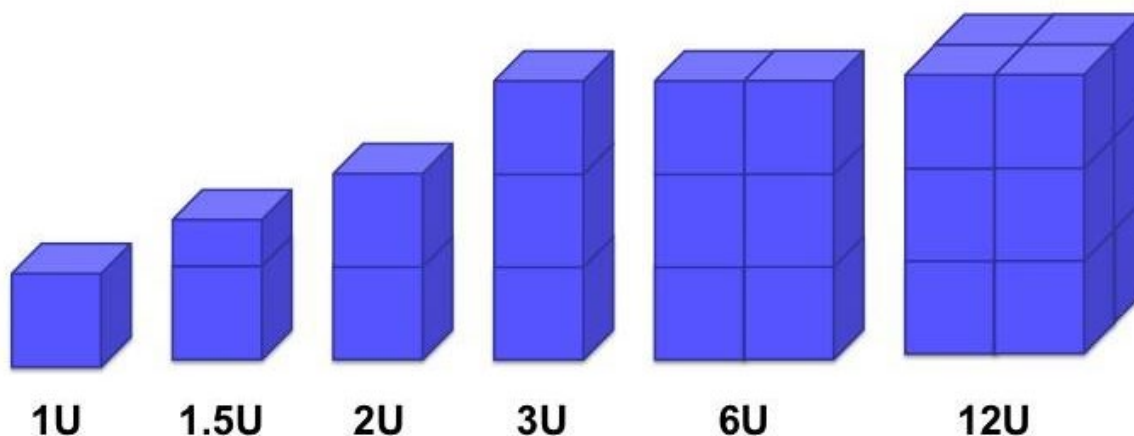
Figure 1.2: CubeSats are a class of nano- and microsatellites that use a standard size and form factor. Credit: NASA.

It is common to interchange the terms "CubeSat" and "NanoSat" (short for nanosatellite) as the original 1-3U CubeSat platforms fell under the nanosatellite category. Since the physical expansion of CubeSats now goes beyond the 6U form factor, CubeSats fall into both nanosatellite and microsatellite categories. Certain chapters of this report have a particular emphasis on CubeSat platforms as nanosatellite applications have expanded in recent years. Figure 1.3 illustrates the three smaller SmallSat categories: microsatellites, nanosatellites, and
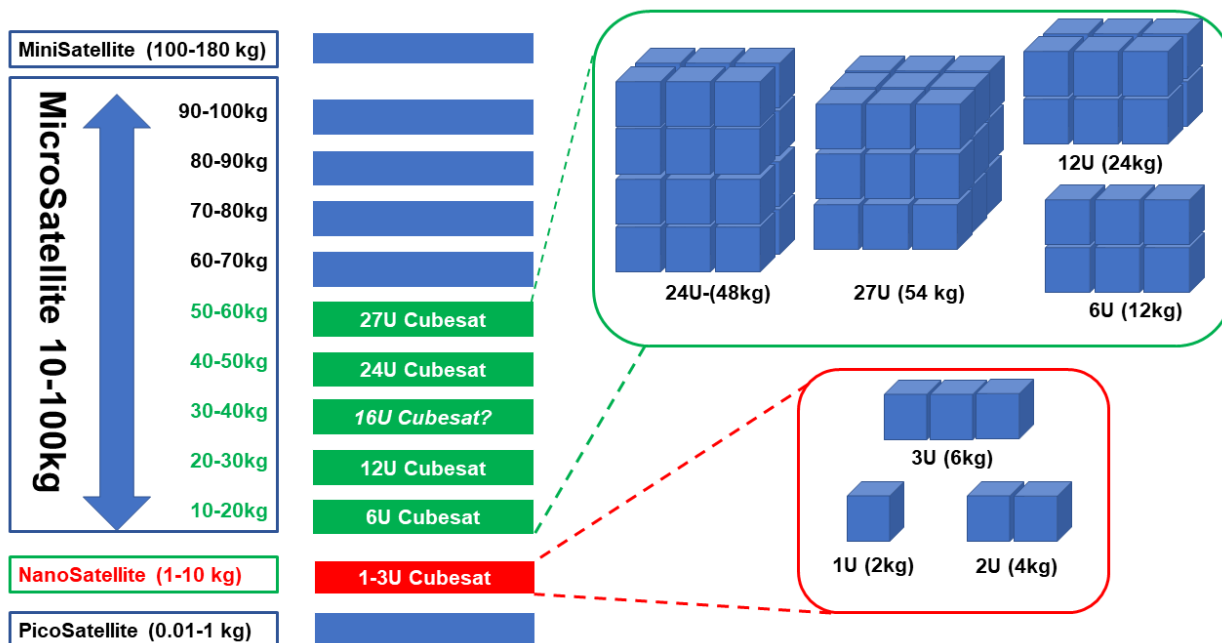
Figure 1.3: Nanosatellite sizes compared to CubeSat containerized sizes. Credit: NASA.

picosatellites.

1.3   Assessment

A central element of this report is to list state-of-the-art technologies by NASA standard Technology Readiness Level (TRL) as defined by the 2020 NASA Engineering Handbook, found

in NASA NPR 7123.1C *NASA Systems Engineering Processes and Requirements*. The authors have endeavored to independently verify the TRL value of each technology by reviewing and citing published test results or publicly available data to the best of their ability. Where test results and data disagree with vendors' own advertised TRL, the authors have attempted to engage the vendors to discuss the discrepancy. Readers are strongly encouraged to follow the references cited in the literature describing the full performance range and capabilities of each technology. Readers of this report should contact individual companies to further clarify information. It is important to note that this report takes a broad system-level view. To attain a high TRL, the subsystem must be in a flight-ready configuration with all supporting infrastructure—such as mounting points, power conversion, and control algorithms—in an integrated unit.

While "state-of-the-art" may be defined as the most recent development stage of technology, this report defines "state-of-the-art" in the context of NASA's TRL scale (Figure 1.4) when assessing SmallSat technology. A technology may be deemed state-of-the-art whenever its TRL is larger than or equal to 5. A TRL of 5 indicates that the component and/or brassboard with realistic support elements was built and operated for validation in a relevant environment so as to demonstrate overall performance in critical areas. Success criteria include documented test performance demonstrating agreement with analytical predictions and documented definition of scaling requirements. Performance predictions are made for subsequent development phases (2).

Figure 1.4: NASA's standard Technology Readiness Level scale. Credit: NASA.

An accurate TRL assessment requires a high degree of technical knowledge on a subject device, and an in-depth understanding of the mission (including interfaces and environment) on which the device was flown. TRL values vary depending on design factors for a specific technology. For example, differences in TRL assessment based on the operating environment may result from mechanical loads, mission duration, the thermal environment, or radiation exposure. The authors believe TRLs are most accurately determined when assessed within the context of a program's unique requirements. If a technology has flown on a mission without success, or without providing valid confirmation to the operator, such claimed "flight heritage" is discounted. Some older technologies may still be well suited to certain mission needs and still be regarded as "state-of-the-art." For a technology to be considered obsolete, "retired", or no longer "state-of-the-art", it's performance must have been surpassed by newer technology such that it is no longer used.

While a technology with a TRL value lower than or equal to 4 may not be state of the art, in some cases these technologies may considered "on the horizon." A TRL of 4 is defined as a component and/or breadboard validated in a laboratory environment with documented test performance demonstrating agreement with analytical predictions and a documented definition of the relevant environment. These promising technologies may soon be considered state-of-the-art for small spacecraft.

NASA standard TRL requirements for this report edition are stated in the NPR 7123.1C, Appendix E, which is effective through February 14, 2025. The criteria for selection of appropriate TRL are
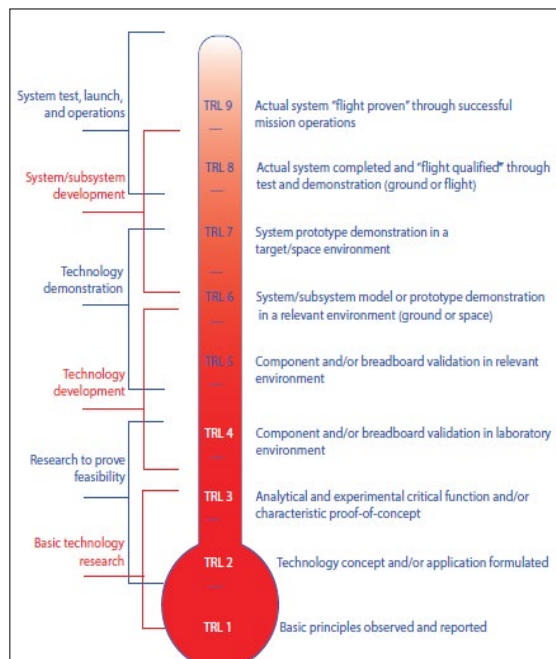
described in the NASA Systems Engineering Handbook 6105 Rev 2 Appendix G: Technology Assessment/Insertion. Please refer to the NASA Online Directives Information System (NODIS) website https://nodis3.gsfc.nasa.gov/ for NPR documentation. The following paragraphs in sections 1.3.1 and 1.3.2 of this Introduction are excerpts from the NASA Engineering Handbook 6105 Rev 2 (pp. 252 – 254). They highlight important aspects of NASA TRL guidelines in hopes of eliminating confusion on terminology and heritage systems.

1.3.1 Terminology

"At first glance, the TRL descriptions in Figure 1.4 appear to be straightforward. It is in the process of trying to assign levels that problems arise. A primary cause of difficulty is in terminology, e.g., everyone knows what a breadboard is, but not everyone has the same definition. Also, what is a "relevant environment?" What is relevant to one application may or may not be relevant to another. Many of these terms originated in various branches of engineering and had, at the time, very specific meanings to that particular field. They have since become commonly used throughout the engineering field and often acquire differences in meaning from discipline to discipline, some differences subtle, some not so subtle. "Breadboard," for example, comes from electrical engineering where the original use referred to checking out the functional design of an electrical circuit by populating a "breadboard" with components to verify that the design operated as anticipated. Other terms come from mechanical engineering, referring primarily to units that are subjected to different levels of stress under testing, e.g., qualification, protoflight, and flight units. The first step in developing a uniform TRL assessment (see Figure 1.5) is to define the terms used. It is extremely important to develop and use a consistent set of definitions over the course of the program/project."

1.3.2 Heritage Systems

"Note the second box particularly refers to heritage systems (Figure 1.5). If the architecture and the environment have changed, then the TRL decreases to TRL
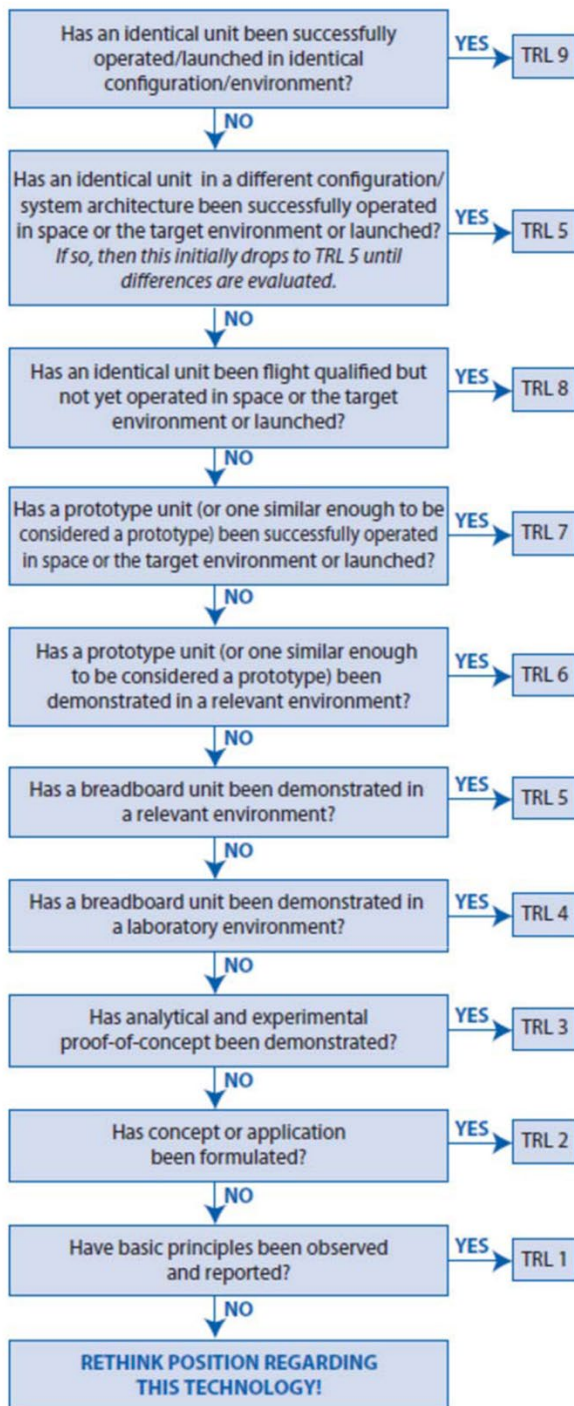
Figure 1.5: Technology Maturity Assessment thought process. Credit: NASA.

5—at least initially. Additional testing may need to be done for heritage systems for the new use or new environment. If in subsequent analysis the new environment is sufficiently close to the old environment or the new architecture is sufficiently close to the old architecture, then the resulting evaluation could be TRL 6 or 7, but the most important thing to realize is that it is no longer at TRL 9. Applying this process at the system level and then proceeding to lower levels of subsystems and components identifies those elements that require development and sets the stage for the subsequent phase, determining the new TRL."

References

(1)   NASA. What are SmallSats and CubeSats? February 26, 2015. Revised August 6, 2017. https://www.nasa.gov/content/what-are-smallsats-and-cubesats

(2)   NASA Systems Engineering Handbook. NASA/SP-2016 6105 Rev. 2. https://www.nasa.gov/feature/release-of-revision-to-the-nasa-systems-engineering-handbook-sp-2016-6105-rev-2

Chapter Contents

Chapter Glossary

(COTS)	Commercial-off-the-Shelf
(EELV)	Evolved Expendable Launch Vehicle
(ESPA)	EELV Secondary Payload Adapter
(GEO)	Geostationary Equatorial Orbit
(I&T)	Integration and Test
(kg)	Kilogram
(LEO)	Low Earth Orbit
(MEO)	Medium Earth Orbit
(MTBF)	Mean Time Between Failures
(NASA)	National Aeronautics and Space Administration
(SHF)	Super High Frequency
(SmallSat)	Small Satellite
(SPA)	Secondary Payload Adapter
(STMD)	Space Technology Mission Directorate
(TRL)	Technology Readiness Level
(UHF)	Ultra High Frequency
(UK)	United Kingdom
(Unk)	Unknown
(USA)	United States of America
(VLEO)	Very Low Earth Orbit
(VHF)	Very High Frequency
(W)	Watts
(xGEO)	Beyond Geostationary Equatorial Orbit

2   Complete Spacecraft Platforms

2.1   Introduction

The Small Spacecraft market allows a wide range of market utilization for mission implementations, from ground-up developments of a spacecraft to entire hosted orbital solutions. This chapter addresses the state of the art for complete spacecraft platforms in two distinct SmallSat market options.

1) **Hosted orbital services:** an emerging business model in the space industry that offers customers access to satellite capabilities to host their payloads (also known as *hosted orbital payloads*) without the need to build, launch, or operate their own spacecraft. With hosted orbital missions, a payload is typically delivered to the provider to be integrated onto an existing spacecraft.
2) **Spacecraft bus:** also known as a *satellite bus* or *spacecraft platform*, is the section of the flight segment that provides essential services to the payload and enables the mission objectives such as thermal management, power, communication, guidance, navigation and control, data processing, and propulsion. This option allows the user to purchase a spacecraft bus with optional system-level integration and test (I&T) services, but vendors may not offer hosted orbital services (spacecraft bus + system-level I&T + operations).

One option is not superior and selection may depend on the needs and constraints of each individual mission. The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. The information is for awareness and guidance only. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. Table 2-9 includes a list of providers with contact information and the source used to complete the tables. It is recommended to contact the organizations/companies directly for further clarification and applications for your specific mission needs.

This chapter organizes the state-of-the-art small spacecraft platforms into these two main categories. The dedicated small spacecraft bus section is further divided by PocketQube, CubeSat, and ESPA-Class offerings. Each subsection contains a summary table with a non-exhaustive list of commercially available small spacecraft platforms.

2.2.1   Hosted Orbital Services
2.2.2   Spacecraft Bus
        PocketQubes
        CubeSats
        ESPA-Class

Section 2.3 provides a brief explanation on systems engineering considerations that introduces newcomers to the design selection process and highlights specific resources for mission development.

2.2   State-of-the-Art – Spacecraft Platforms

2.2.1   Hosted Orbital Services

Hosted orbital service providers accept your science instruments and/or technology payloads and performs system-level integration and test, and operations. The actual implementation on the vendor side varies widely across the industry, depending on providers' vehicle type and business model. Examples of vendor implementations may include the following.

- Host your payload on its own dedicated spacecraft where the entirety of the spacecraft bus is at the disposal of a single customer and/or mission
- Host your payload on vendor's unused capability aboard their own, internally developed mission
- Host your payload along with other payloads from different customers on a single spacecraft
- Host your 'software-only' payload, also known as a 'virtual' payload on an existing spacecraft that supports virtual payloads

It should be noted that not all hosted orbital service providers sell their spacecraft bus to customers. For spacecraft bus providers, see Section 2.2.2. It is incumbent upon the user to conduct their own research and determine which solution best meets their mission requirements and constraints.

Common benefits of hosted orbital solutions include cost effectiveness, reliability, flexibility, faster access to space, and the ability for users to focus on the spacecraft payload. Customers can access space capabilities without the high upfront costs of building and launching their own spacecraft and can concentrate their efforts and resources on their specific instruments or technologies, leaving the spacecraft bus build, system-level integration and test, and operations to an experienced developer. Hosting of multiple payloads offers efficiencies through resource sharing, which can make this option more affordable than manifesting a payload on its own dedicated spacecraft. Common interfaces allow for rapid integration and test, decreasing the overall time to flight. Mission success can be improved by leveraging commercial provider's proven infrastructure, processes and expertise. Services can be scaled up or down based on mission needs, allowing for adjustments in bandwidth, coverage, or mission duration. Additional space-based hosting of small spacecraft payloads is also available on several vehicles that are not covered in this report, including payloads that are hosted in or on the ISS, attached to rocket bodies, or hosted in or on capsules (crewed or uncrewed) that return to earth.

Procurement of hosted orbital services is relatively new and could be challenging if the user does not have experience procuring such services. In 2023, NASA's Flight Opportunities program released a solicitation for Suborbital/Hosted Orbital Flight and Payload Integration Services that, for the first time, included commercial orbital platforms capable of hosting payloads. Fifteen companies were selected to provide flight and payload integration services for technology payloads to be demonstrated in high-altitude, reduced gravity, and other relevant testing environments on suborbital rocket-powered vehicles, high-altitude balloons, and hosted orbital platforms via Indefinite Delivery/Indefinite Quantity (IDIQ) contracts. These IDIQ contracts can be used by NASA centers and other government agencies to procure hosted orbital services. The Flight Opportunities competitions and solicitations (e.g., TechLeap, TechFlights) offer options for hosted orbital platforms, aiming to advance the maturity of a technology via flight tests. Additional information on NASA's Flight Opportunities program and the IDIQ contract for hosted orbital services can be found here: https://www.nasa.gov/stmd-flight-opportunities/nasa-contracted-flight-providers/.

Table 2-1: Hosted Orbital Service Providers						
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the Hosted Payloads category)						
Organization Headquarters	Max Volume	Max Mass (kg)	Peak Power (W)	3-σ Pointing Control/ Knowledge	Destination	US Office
Aerospacelab Belgium	0.125 m³	50	300	<0.005°/ <0.005°	LEO, GEO	Yes
Artemis Space Technologies UK	0.58 m³	500	1,500	0.01°/0.01°	LEO, MEO, GEO, Lunar and Deep Space	No
Astranis Space Technologies Corp. USA	0.02 m³	10	300	<0.1°/ <0.09°	GEO	Yes
Astro Digital USA	Unk	Unk	Unk	Unk	Unk	Yes
Axelspace Japan	0.2 m³	30	184	<0.05°/ <0.04°	LEO	No
Berlin Space Technologies Germany	1 m³	250	2,500	<0.017°/< 0.017°	LEO	Yes
C3S Electronics Development Hungary	16.5U	18.5	155	0.2°/ 0.2°	LEO, MEO	No
CREOTECH Poland	>50U	100	175	<0.015°/< 0.015°	LEO, MEO, GEO, Lunar	No
EnduroSat Bulgaria	150U	70	300	0.1°/ 0.008°	LEO	Yes
General Atomics EMS USA	0.46 m³	200	450	0.03°/ 0.02°	LEO	Yes
German Orbital Systems Germany	4U	8	50	<1°/< 1°	LEO	No
Gran Systems Taiwan	6U	12	50	2°/ 2°	LEO, Lunar	Yes
Harpy Aerospace India	6U	28	72	<0.1°/< 0.01°	LEO, GEO, Lunar, ISS	Yes
Hemeria France	0.1 m³	35	250	<0.03°/<0.01°	LEO, GTO, GEO	No
Innova Space Argentina	0.5U	0.5	4	<15°/< 15°	LEO	Yes
Loft Orbital USA	0.44 m³	85	>1,000	<0.035°/<0.03°	LEO	Yes
Momentus Space USA	1 m³	800	3,000	0.008°/ 0.008°	LEO, MEO, GEO, Lunar, Deep Space	Yes
NanoAvionics Lithuania	0.7 m³	150	378	0.15°/ 0.03°	LEO	Yes
Nara Space South Korea	16U	19	120	0.05°/ 0.03°	LEO	No
NearSpace Launch USA	8U	16	160	0.5°/ 0.2°	LEO	Yes
Northrop Grumman USA	0.37 m³	50	420	<4°/<1°	LEO	Yes
NovaWurks USA	1 m³	300	2000	0.002°/0.0004°	LEO, GEO, xGEO	Yes
NPC SPACEMIND Italy	12U	24	100	<0.1°/<0.1°	LEO, MEO	No

Table 2-1: Hosted Orbital Service Providers
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the Hosted Payloads category)

Organization [Headquarters]	Max Volume	Max Mass (kg)	Peak Power (W)	3-σ Pointing Control/ Knowledge	Destination	US Office
OHB LuxSpace [Luxembourg]	0.3 m3	90	600	<0.022°/ 0.01°	LEO	No
OHB Sweden [Sweden]	1 m^3	300	1,500	0.008°/ 0.008°	LEO, MEO	No
Open Cosmos [UK]	12U	18	160	0.03°/0.02°	LEO	No
Orbital Astronautics [UK]	0.163 m^3	100	5,000	<0.05°/<0.01°	LEO, MEO, GEO, Deep Space	No
Orion Space Solutions [USA]	14U	45	400	<1°/<1°	LEO, GEO, Lunar	Yes
Quantum Space [USA]	0.5 m^3	100	400	0.006°/0.006°	LEO, GEO, Cislunar, Lunar, Deep Space	Yes
Redwire Space [USA]	0.94 m^3	104	500	0.005°/0.0017°	LEO, MEO, GEO and Deep Space	Yes
Rocket Lab [USA]	Unk	Unk	Unk	Unk	LEO	Yes
SatRev [Poland]	3U	3	25	1°/0.6°	LEO	No
Sierra Space [USA]	0.48 m^3	250	500	0.001°/ <0.001°	LEO, MEO, GEO	Yes
SITAEL [Italy]	0.90 m^3	120	2,000	0.0017°/ 0.001°	LEO	No
SpaceX [USA]	Unk	Unk	Unk	Unk	LEO	Yes
Space Inventor [Denmark]	24U	50	400	<0.008°/ <0.008°	LEO, GEO, MEO	No
Spacemanic [Czech Republic]	12U	18	500	0.1°/ 0.05°	LEO, MEO, GEO, Lunar	No
Spire Global [USA]	12U	32	300	0.1°/ 0.05°	LEO	Yes
Surrey Satellite Technology Ltd. [UK]	0.4m^3	200	2,000	<0.01°/ <0.01°	LEO, MEO, GEO, Lunar	No
Terran Orbital [USA]	12U	100	500	0.014°/ 0.002°	LEO, GEO, Deep Space	Yes
Varda Space Industries [USA]	Unk	Unk	Unk	Unk	Unk	Yes
Xplore [USA]	0.125 m^3	55	210	0.17°/ 0.018°	VLEO, LEO, Cislunar	Yes
York Space Systems [USA]	-	300	1,500	0.008°/ 0.004°	LEO, GEO, Lunar	Yes

2.2.2 Spacecraft Bus

The SmallSat market offers complete spacecraft bus solutions including system-level I&T and operations options. Spacecraft bus providers in this section may only offer the spacecraft bus as a product or may also provide I&T and operations services. Contact the provider using the Table 2-8 to better understand the services offered if a provider seem to meet your basic requirements advertised in the tables.

PocketQubes

PocketQubes refer to small satellites that conform to a form factor of 5 cm cubes. PocketQubes use a standard deployer and follow a unit nomenclature of P. In this case 1P refers to a single 5 cm cube (see Figure 2.1). Consequently, 2P refers to two of these single units. A typical PocketQube deployer can deploy up to a 3P satellite but larger deployers may allow additional capability. PocketQube providers have developed spacecraft buses to simplify mission implementation; a list of providers is included in this section. Table 2-2 provides available commercial PocketQube products. Figure 2.2 is an example of a PocketQube deployer at Alba Orbital.
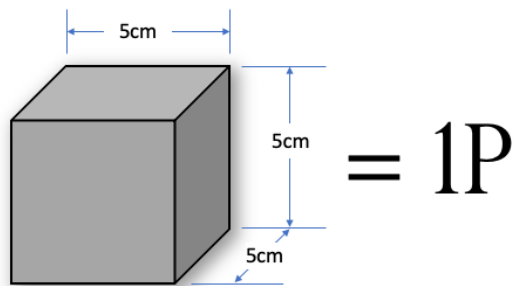
Figure 2.1: PocketQube dimensions.

Figure 2.2: Alba Orbital Integration of PocketQubes into the deployers. Credit: Alba Orbital.

Table 2-2: PocketQubes Market Solutions						
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the PocketQube category)						
Organization Headquarters	**Peak Power (W)**	**3-σ Pointing Control/ Knowledge**	**Comm Options**	**Intended Destination**	**Maturity**	**US Office**
Alba Orbital UK	15	5°/2°	UHF, S	LEO	Flown LEO	Yes
DIYSATELLITE Argentina	9	<5°/<5°	VHF, UHF, SHF	LEO, GEO, Lunar	Flown LEO	No
FOSSA Systems Spain	10	<5°/<5°	UHF, S	LEO	Flown LEO	No
Hydra Space Systems Spain	8	5°/5°	VHF, UHF	LEO	Flown LEO	No
Innova Space Argentina	3.9	N/A -Magnetic Passive	UHF	LEO	Flown LEO	Yes
Quub, Inc. USA	26	5°/2°	UHF, S	LEO, Lunar	Flown LEO	Yes

CubeSats

CubeSats refer to small satellites that conform to a form factor of 10 cm cubes. The CubeSat standard was created by California Polytechnic State University, San Luis Obispo, and Stanford University's Space Systems Development Lab in 1999 to facilitate access to space for university students. Low-cost access to space increased due to the adoption of the standard by launch providers. Many organizations are currently using the standard including academia, private industry, and government. More information on the history of CubeSats can be found in the introduction of this report. CubeSat providers have developed spacecraft buses to accommodate missions from <1U to 27U satellites. This section provides a list of providers separated by satellite size: 0.25U-3U (Table 2-3), 6U (Table 2-4), 12U (Table 2-5) and 16U+ (Table 2-6).

Multiple companies have developed deployers for CubeSats with different dimensions and external volume allocations. Contact your sponsoring organization and/or launch provider for specifics on which deployer to use in your mission. The primary two interfaces follow the classic corner rails or the tabs (clamped and unclamped). Most spacecraft bus providers in this chapter can adapt to different interfaces. Refer to the Launch, Integration, and Deployment chapter for further information on SmallSat deployers. Figure 2.3 includes images of CubeSat missions that have been successfully flown in space, Figure 2.4 provides examples of CubeSat deployers' locations on a rocket, and Figure 2.5 provides examples for 6U and 16U satellites from Spire Global.
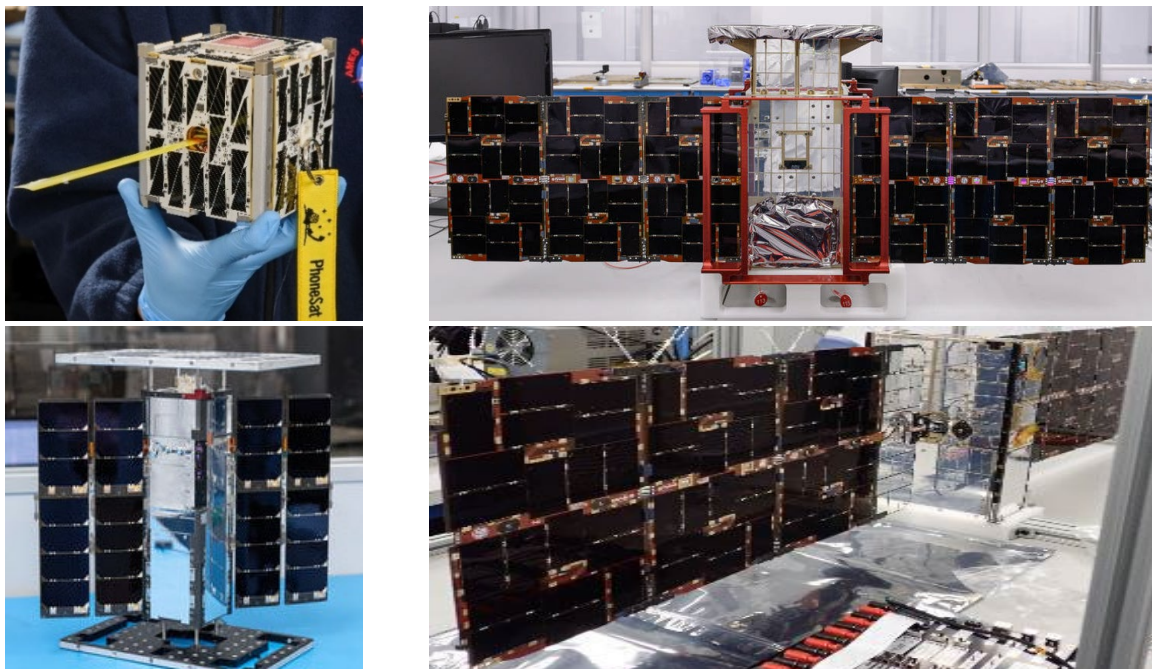
Figure 2.3: Examples of flown CubeSats. (Top left) 1U PhoneSat spacecraft, (top right) 12U CAPSTONE spacecraft, (lower left) 3U CLICK spacecraft, (lower right) 6U PTD-3 spacecraft. Credits: NASA and Terrain Orbital.
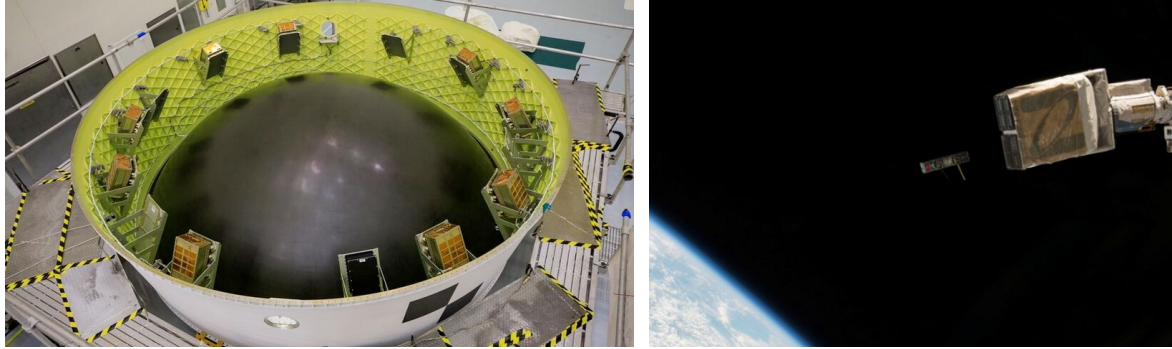
Figure 2.4: (left) Location of Artemis CubeSat deployers in between the Orion Crew Vehicle and the Interim Cryogenic Propulsion Stage (ICPS); (right) NASA Nodes mission deployment from ISS. Credit: NASA.
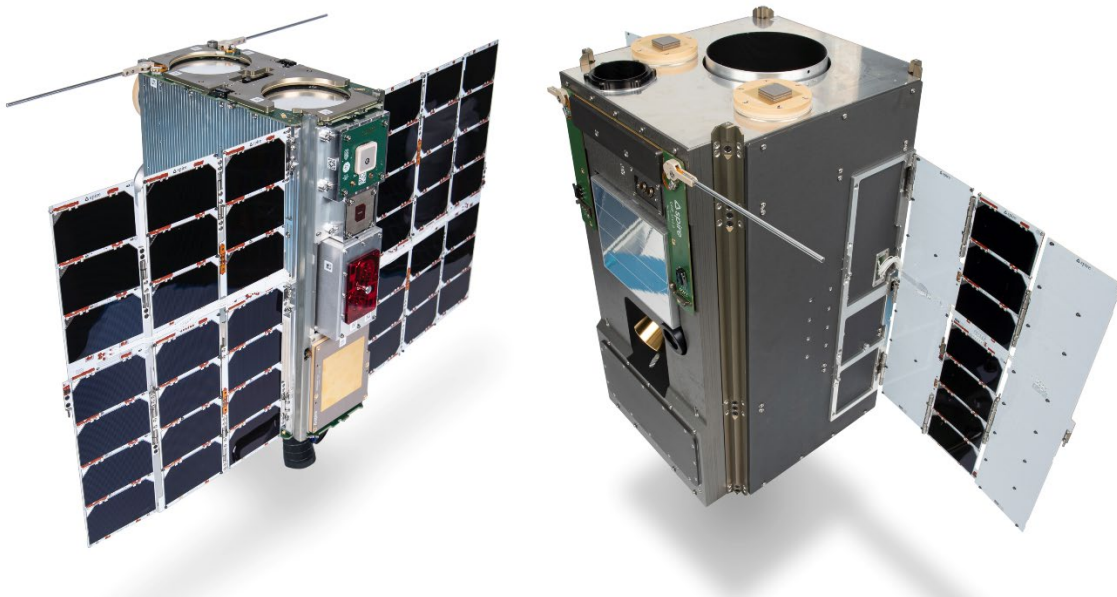
Figure 2.5: Examples of a 6U and 16U CubeSat. Credit: Spire Global.

Table 2-3: 0.25U-3U Market Solutions						
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 0.25U-3U category)						
Organization Headquarters	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
AAC Clyde Space Sweden	90	<0.1°/<0.01°	VHF, UHF, S, X	LEO	Flown LEO	Yes
Alén Space Spain	180	0.2°/0.1°	VHF, UHF, S	LEO	Flown LEO	No
Artemis Space Technologies UK	50	0.01°/0.01°	UHF, S, X, Ka, Ku	LEO	Flown LEO	No
Blue Canyon Technologies USA	27	0.003°/0.003°	L, S, X	LEO, GEO, Deep Space	Flown LEO Qualified GEO and Deep Space	Yes
C3S Electronics Development Hungary	35	0.2°/0.2°	UHF, S	LEO, MEO	Flown LEO	No
Deimos Space Spain	35	0.2°/0.2°	UHF, X	LEO	Flown LEO	No
EnduroSat Bulgaria	84	<1°/<0.6°	UHF, S, X	LEO	Flown LEO	Yes
FOSSA Systems Spain	30	1°/1°	UHF, S	LEO	Flown LEO	No
General Atomics EMS USA	10	0.28°/0.08°	UHF	LEO, MEO, GEO, xGEO	Under Development	Yes
German Orbital Systems Germany	24	<1°/<1°	UHF, S	LEO	Flown LEO	No
GomSpace Denmark	35	2.5°/2°	S	LEO	Flown LEO	Yes
Gran Systems Taiwan	50	2°/ 2°	VHF, UHF	LEO	Flown LEO	Yes
GUMUSH AeroSpace Turkey	80	<1°/ <0.1°	VHF, UHF, S, X	LEO	Flown LEO	No
Harpy Aerospace India	72	<0.1°/<0.01°	VHF, UHF, S, X	LEO, GEO, Lunar	Qualified LEO and Lunar	Yes
Hex20 Australia	25	0.003°/ 0.003°	UHF, S	LEO, MEO, GEO, Lunar	Flown LEO	No
IMT Italy	>5	10°/5°	VHF, UHF	LEO	Under Development	No
Innova Space Argentina	7.5	<15°/<15°	UHF	LEO	Flown LEO	Yes
ISISPACE The Netherlands	50	<15°/<15°	VHF, UHF, S	LEO	Flown LEO	No
NanoAvionics Lithuania	175	13.20°/12.93°	UHF, S, X	LEO	Flown LEO	Yes
NearSpace Launch USA	100	0.5°/0.2°	L, UHF, S, X	VLEO, LEO	Flown LEO	Yes
NPC SPACEMIND Italy	51.6	<0.1°/<0.1°	UHF, S, X, Ka	LEO, MEO, GEO, Lunar	Flown LEO and MEO	No
Open Cosmos UK	160	2.4°/0.67°	UHF, S	LEO	Flown LEO	No
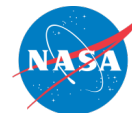
Table 2-3: 0.25U-3U Market Solutions						
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 0.25U-3U category)						
Organization Headquarters	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Orbital Astronautics UK	400	0.1°/ 0.01°	S, X, K, Ka, Optical	LEO, MEO	Flown LEO	No
Orion Space Solutions USA	8	1°/1°	L, S, X	LEO	Qualified LEO	Yes
Pumpkin Space Systems USA	200	0.05°/<0.05°	UHF, S, X, Ka	LEO	Flown LEO	Yes
Quub, Inc. USA	44	5°/2°	UHF, S	LEO, Lunar	Flown LEO	Yes
SatRev Poland	36	1°/0.6°	UHF, S	LEO	Flown LEO	No
SkyLabs Slovenia	100	0.3°/0.06°	VHF, UHF, S	LEO, MEO	Flown LEO and MEO	No
Space Flight Laboratory Canada	93	0.009°/0.004°	UHF, S, X, Ka	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Space Inventor Denmark	100	0.01° / 0.01°	VHF, UHF, S, X, L	LEO	Flown LEO	No
Spacemanic Czech Republic	30	0.1°/0.05°	VHF, UHF, S	LEO, GEO, Lunar	Flown LEO Qualified GEO	No
Spire Global USA	35	0.1°/0.05°	UHF, L, S, X, Ka, Ku	LEO	Flown LEO	Yes

Table 2-4: 6U Market Solutions						
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 6U category)						
Organization Headquarters	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
AAC Clyde Space Sweden	150	<0.1°/<0.01°	VHF, UHF, S, X	LEO	Flown LEO	Yes
Alén Space Spain	180	0.2°/0.1°	VHF, UHF, S	LEO	Flown LEO	No
Argotec Italy	80	<0.02°/<0.01°	UHF, S, X	LEO, GEO, Deep Space	Flown Deep Space Flown Lunar	Yes
Artemis Space Technologies UK	100	0.01°/0.01°	UHF, S, X, Ka, Ku, Optical	LEO, MEO, GEO, Lunar, Deep Space	Flown LEO Qualified MEO, GEO, Lunar, and Deep Space	No
Astro Digital USA	240	<0.1°/<0.05°	UHF, S, X, Ka	LEO	Flown LEO	Yes
Blue Canyon Technologies USA	108	0.003°/0.003°	L, S, X	LEO, GEO, Deep Space	Flown LEO and Lunar Qualified GEO and Deep Space	Yes
C3S Electronics Development Hungary	165	<0.2°/<0.2°	UHF, S	LEO, MEO	Under Development	No
Deimos Space Spain	40	0.05°/0.05°	UHF, S, X	LEO	Under Development	No
EnduroSat Bulgaria	112	0.08°/0.04°	UHF, S, X	LEO	Flown LEO	Yes
General Atomics EMS USA	10	0.28°/0.08°	UHF, S	LEO	Flown LEO	Yes
German Orbital Systems Germany	72	<1°/<1°	UHF, S, X	LEO	Flown LEO	No
GomSpace Denmark	102	0.07°/0.056°	S, X	LEO, Deep Space	Flown LEO Qualified Deep Space	Yes
GUMUSH AeroSpace Turkey	160	<0.1°/<0.05°	VHF, UHF, S, X	LEO	Under Development	No
Harpy Aerospace India	160	<0.1°/<0.01°	VHF, UHF, S, X	LEO	Qualified LEO	Yes
Hex20 Australia	45	0.003°/0.003°	UHF, S, X	LEO, MEO, GEO, Lunar	Flown LEO	No
IMT Italy	115	0.1°/0.1°	VHF, UHF, S, C, X	LEO	Under Development	No
ISISPACE The Netherlands	100	<0.3°/<0.3°	UHF, S, X	LEO, Lunar	Flown LEO Qualified for Lunar	No
NanoAvionics Lithuania	175	0.18°/0.12°	UHF, S, X	LEO	Flown LEO	Yes

| Table 2-4: 6U Market Solutions |||||||
| (The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 6U category) |||||||
Organization Headquarters	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
NearSpace Launch USA	160	0.5°/0.2°	L, UHF, S, X	LEO	Flown LEO	Yes
NPC SPACEMIND Italy	85.2	<0.1°/<0.1°	UHF, S, X, Ka	LEO, MEO, GEO, Lunar	Flown LEO	No
Open Cosmos UK	160	0.02°/0.01°	UHF, S, X	LEO	Flown LEO	No
Orbital Astronautics UK	1,000	0.1°/0.01°	S, X, K, Ka, Optical	LEO, MEO	Flown LEO	No
Orion Space Solutions USA	15	1°/1°	L, S, X	LEO	Flown LEO	Yes
Pumpkin Space USA	200	0.05°/<0.05°	UHF, S, X, Ka	LEO, Lunar	Flown LEO Qualified Lunar	Yes
Quub, Inc. USA	50	5°/2°	UHF, S, Ku	LEO, Lunar	Under Development	Yes
SatRev Poland	36	1°/0.6°	UHF, S	LEO	Qualified LEO	No
SkyLabs Slovenia	200	0.3°/0.06°	VHF, UHF, S	LEO, MEO	Flown LEO and MEO	No
Space Dynamics Laboratory USA	400	0.021°/0.021°	UHF, S, X, Ka	LEO, GEO, GTO, Cislunar, Deep Space	Flown LEO Qualified GEO, GTO, Lunar and Deep Space	Yes
Space Flight Laboratory Canada	240	0.009°/0.004°	UHF, S, X, Ka	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Space Inventor Denmark	200	<0.008°/<0.008°	VHF, UHF, S, X	LEO	Flown LEO	No
Spacemanic Czech Republic	500	0.1°/0.05°	VHF, UHF, S	LEO, GEO, Lunar	Flown LEO Qualified GEO	No
Spire Global USA	200	0.1°/0.05°	UHF, L, S, X. Ka, Ku	LEO	Flown LEO	Yes
Terran Orbital USA	180	0.021°/0.007°	UHF, S, X	LEO, GEO, Lunar	Flown LEO and Lunar	Yes

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Table 2-5: 12U Market Solutions (The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 12U category)						
AAC Clyde Space [Sweden]	400	<0.01°/<0.0075°	VHF, UHF, S, X, K, Ka, Ku, Optical	LEO	Qualified LEO	Yes
Alén Space [Spain]	180	0.2°/0.1°	UHF, S	LEO	Under Development	No
Argotec [Italy]	100	<0.02°/<0.01°	UHF, S, X	LEO	Under Development	Yes
Artemis Space Technologies [UK]	150	0.01°/0.01°	UHF, S, X, Ka, Ku, Optical	LEO, MEO, GEO, Lunar, Deep Space	Flown LEO Qualified GEO, MEO, Lunar, and Deep Space	No
Blue Canyon Technologies [USA]	108	0.0025°/0.0025°	L, S, X	LEO, GEO, Deep Space	Flown LEO and GEO Qualified Deep Space	Yes
C3S Electronics Development [Hungary]	165	<0.2°/<0.2°	UHF, S	LEO, MEO	Under Development	No
EnduroSat [Bulgaria]	250	0.08°/0.04°	UHF, S, X, K/Ka	LEO	Flown LEO	Yes
General Atomics EMS [USA]	115	0.3°/0.024°	UHF, S	LEO	Qualified LEO	Yes
GomSpace [Denmark]	102	0.07°/0.056°	S, X	LEO	Qualified LEO	Yes
GUMUSH AeroSpace [Turkey]	240	<0.05°/<0.05°	VHF, UHF, S, X	LEO	Under Development	No
Harpy Aerospace [India]	420	<0.1°/<0.01°	VHF, UHF, S, X, K, Ka, Ku, Optical	LEO	Qualified LEO	Yes
Hex20 [Australia]	110	0.003°/0.003°	UHF, X	LEO, MEO, GEO, Lunar	Flown LEO	No
ISISPACE [The Netherlands]	190	<0.03°/<0.03°	UHF, S, X, Ka	LEO	Under Development	No
NanoAvionics [Lithuania]	175	0.18°/0.09°	UHF, S, X	LEO	Flown LEO	Yes
Nara Space [South Korea]	120	0.05°/0.03°	S, X	LEO	Qualified LEO	No
NearSpace Launch [USA]	500	0.5°/0.2°	L, UHF, S, X	LEO, MEO	Under Development	Yes
NPC SPACEMIND [Italy]	96	<0.1°/<0.1°	UHF, S, X, Ka	LEO, MEO, GEO, Lunar	Flown LEO	No
Open Cosmos [UK]	160	0.031°/0.027°	UHF, S, X	LEO	Flown LEO	No
Orbital Astronautics [UK]	1,000	0.05°/0.01°	S, X, K, Ka, Optical	LEO, MEO, GEO	Flown LEO	No
Orion Space Solutions [USA]	40	<1°/<1°	L, S, X, Ka	LEO, GEO	Qualified LEO and GEO	Yes
Pumpkin Space [USA]	400	0.05°/<0.05°	UHF, S, X, Ka	LEO, Lunar	Qualified LEO	Yes

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
SkyLabs [Slovenia]	500	0.3°/0.06°	VHF, UHF, S	LEO, MEO	Flown LEO and MEO	No
Space Dynamics Laboratory [USA]	400	0.021°/0.021°	UHF, S, X, Ka	LEO, GEO, GTO, Cislunar, Deep Space	Flown LEO Qualified GEO, GTO, Lunar and Deep Space	Yes
Space Flight Laboratory [Canada]	322	0.009°/0.004°	UHF, S, X, Ka	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Space Information Laboratories [USA]	180	0.008°/0.008°	S, X, Ka	LEO, GEO, Lunar	Under Development	Yes
Space Inventor [Denmark]	200	<0.008°/<0.008°	VHF, UHF, S, X	LEO	Flown LEO	No
Spacemanic [Czech Republic]	500	0.1°/0.05°	VHF, UHF, S, X	LEO, GEO, Lunar	Flown LEO Qualified GEO	No
Spire Global [USA]	300	0.1°/0.05°	UHF, L, S, X, Ka, Ku	LEO	Qualified LEO	Yes
Terran Orbital [USA]	100	0.021°/0.007°	UHF, S, X	LEO, GEO, Lunar	Flown LEO and Lunar	Yes

Table 2-5: 12U Market Solutions
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 12U category)

Table 2-6: 16U+ Market Solutions
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 16U+ category)

Organization Headquarters	Format	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
AAC Clyde Space Sweden	16U	400	<0.01°/<0.0075°	VHF, UHF, S, X, K, Ka, Ku, Optical	LEO	Qualified LEO	Yes
Argotec Italy	16U+	220	<0.02°/<0.01°	UHF, S, X, K, Ka	GEO, Deep Space	Under Development	Yes
Artemis Space Technologies UK	16U	200	0.01°/0.01°	UHF, S, X, Ka, Ku, Optical	LEO, MEO, GEO, Lunar, Deep Space	Flown LEO Qualified GEO, MEO, Lunar, and Deep Space	No
Astro Digital USA	16U+	500	<0.05°/<0.01°	UHF, S, X, Ku, Ka, V, W, Optical	LEO	Flown LEO	Yes
Blue Canyon Technologies USA	16U	108	0.0025°/0.0025°	L, S, X	LEO, GEO, Deep Space	Qualified LEO, GEO and Deep Space	Yes
C3S Electronics Development Hungary	16U+	165	<0.2°/<0.2°	UHF, S	LEO, MEO	Under Development	No
Deimos Space Spain	16U+	100	0.025°/0.025°	UHF, S, X	LEO, Lunar, Deep Space	Under Development	No
EnduroSat Bulgaria	16U	250	0.08°/0.04°	UHF, S, X, K/Ka	LEO	Qualified LEO	Yes
German Orbital Systems Germany	16U	164	<1°/<1°	UHF, S, X	LEO	Qualified LEO	No
GomSpace Denmark	16U	150	0.07°/0.056°	S, X	LEO	Qualified LEO	Yes
GUMUSH AeroSpace Turkey	16U	240	<0.05°/<0.05°	VHF, UHF, S, X	LEO	Under Development	No
Harpy Aerospace India	16U, 27U	420	<0.1°/<0.01°	VHF, UHF, S, X, K, Ka, Ku, Optical	LEO	Qualified LEO	Yes
Hex20 Australia	27U	150	0.003°/0.003°	UHF, S, X	LEO, MEO, GEO, Lunar	Flown LEO	No
ISISPACE The Netherlands	16U	190	<0.03°/<0.03°	UHF, S, X, Ka	LEO	Under Development	No
NanoAvionics Lithuania	16U	175	0.18°/0.09°	UHF, S, X	LEO	Flown LEO	Yes
Nara Space South Korea	16U	120	0.05°/0.03°	S, X	LEO	Flown LEO	No
NPC SPACEMIND Italy	16U	120	<0.1°/<0.1°	UHF, S, X, Ka	LEO, MEO, GEO, Lunar	Under Development	No
Open Cosmos UK	16U	160	0.031°/0.027°	UHF, S, X	LEO	Flown LEO	No
Orbital Astronautics UK	16U, 27U	1,000	0.05°/0.01°	S, X, K, Ka, Optical	LEO, GEO, Lunar	Qualified LEO	No
Orion Space Solutions USA	16U+	400	<1°/<1°	L, S, X, Ka, Optical	LEO, GEO, Lunar	Qualified LEO, GEO and Lunar	Yes

Organization [Headquarters]	Format	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Pumpkin Space [USA]	16U, 27U	400	0.05°/<0.05°	UHF, S, X, Ka	LEO, Lunar	Qualified LEO	Yes
SkyLabs [Slovenia]	20U+	500	<0.005°/<0.003°	VHF, UHF, S	LEO, MEO	Flown LEO and MEO	No
Space Dynamics Laboratory [USA]	16U+	1,600	0.021°/0.021°	UHF, S, X, Ka, Optical	LEO, GEO, GTO, Cislunar, Deep Space	Flown LEO Qualified GEO, GTO, Lunar and Deep Space	Yes
Space Flight Laboratory [Canada]	16U	322	0.009°/0.004°	UHF, S, X, Ka	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Space Information Laboratories [USA]	27U	180	0.008°/0.008°	S, X, Ka	LEO, GEO, Lunar	Under Development	Yes
Space Inventor [Denmark]	16U	200	<0.008°/<0.008°	VHF, UHF, S, X, L, Ka, Ku, QV	LEO, GEO, MEO	Flown LEO and GEO	No
Spacemanic [Czech Republic]	16U, 27U	1,000	0.1°/0.05°	VHF, UHF, S, X	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Spire Global [USA]	16U	300	0.1°/0.05°	UHF, L, S, X, Ka, Ku	LEO	Flown LEO	Yes
Terran Orbital [USA]	16U	100	0.021°/0.007°	UHF, S, X	LEO, GEO, Lunar	Flown LEO and Lunar	Yes

Table 2-6: 16U+ Market Solutions
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the 16U+ category)

ESPA-Class

The term ESPA-class refers to the Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (SPA) or similar configurations. The ESPA ring typically separates the primary payload from the upper stage of the launch vehicle and permits additional mounting locations for secondary payloads. Multiple rings can be stacked to increase the number of secondary payloads.

Table 2-7 includes ESPA-class options that may not be designed for the ESPA ring but have the mass and volume to permit adaptability to rideshare opportunities. The information in this chapter is limited to offerings with mass under 500 kg even though some variants of the ESPA ring can support higher mass. Variants of the ESPA ring include, but are not limited to, ESPA-Heavy and ESPA-Grande. Examples of ESPA rideshares are provided in Figures 2.5, 2.6, and 2.7, while Figure 2.9 shows an example of an ESPA satellite from Muon Space.

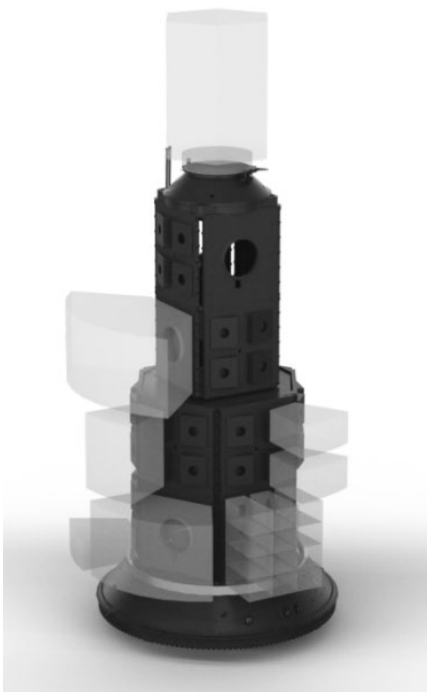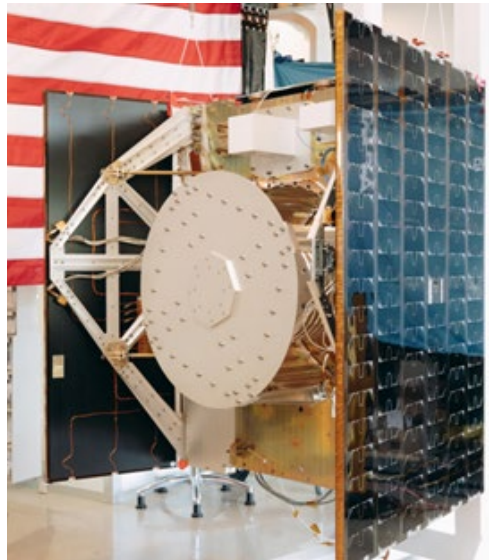*Figure 2.6: Example mission configuration using rideshare plates. Credit: SpaceX.*

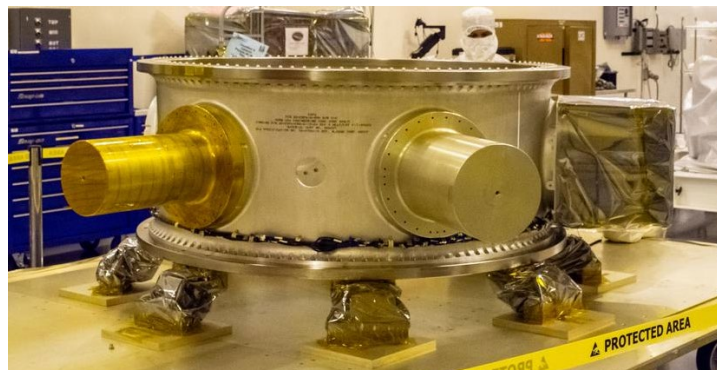*Figure 2.7: Apex Aries bus with payloads. Credit: Apex Space.*

Figure 2.8: LandSat-9 ESPA ring populated with payloads and mass ballasts. Credit: NASA/Randy Beaudoin.
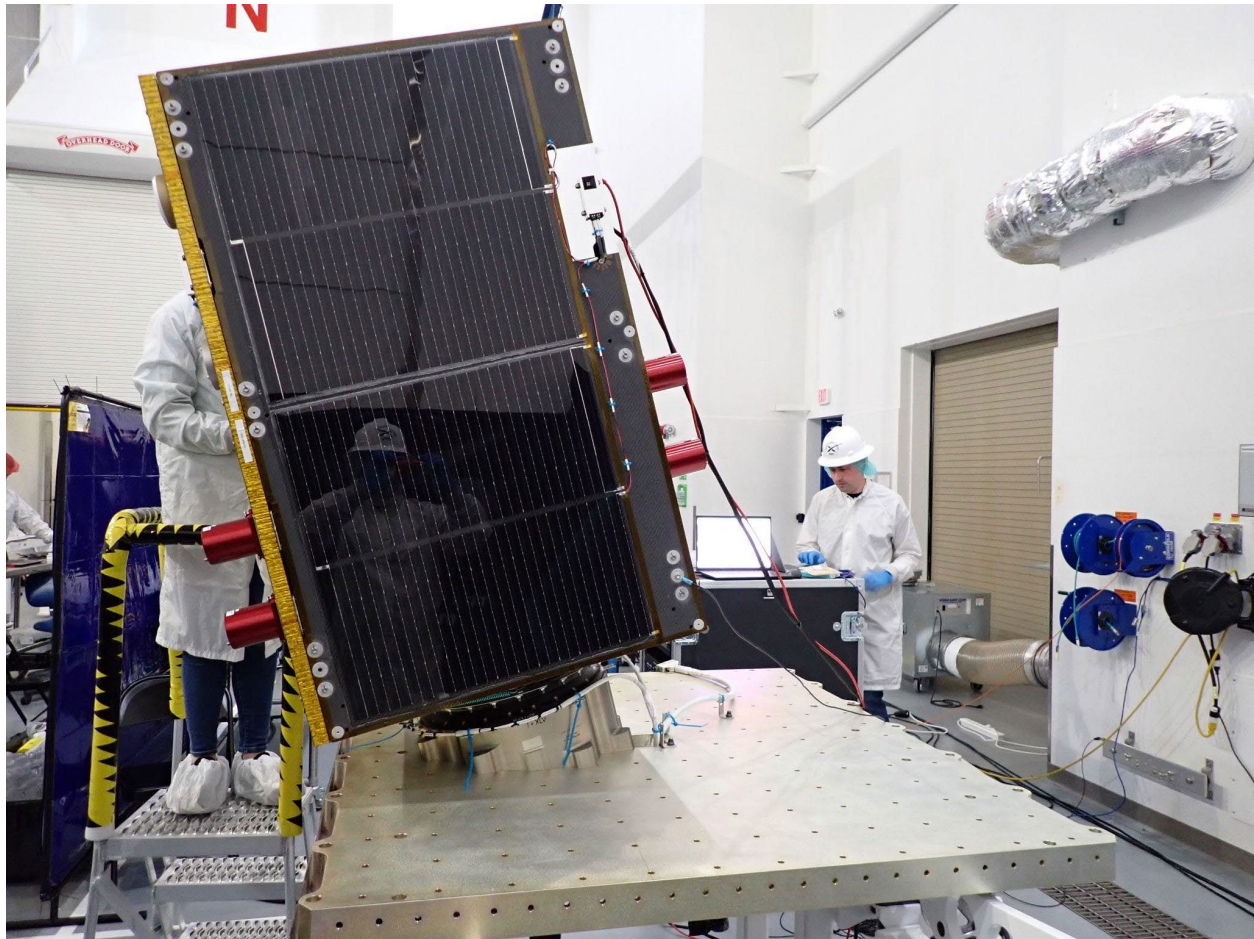
Figure 2.9: ESPA-Class satellite bus from Muon Space during integration at SpaceX facility for Transporter 8 rideshare mission. Credit: Muon Space via ExoLaunch and SpaceX.

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office

Table 2-7: ESPA-Class Market Solutions
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the ESPA-Class category)

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Aerospacelab [Belgium]	1,000	<0.003°/<0.003°	S, X, Ka, Optical	LEO, GEO	Flown LEO Under Development GEO	Yes
Airbus US Space & Defense [USA]	2,200	0.3°/0.3°	S, Ka, Optical	LEO	Flown LEO	Yes
Apex Space [USA]	4,000	0.005°/0.0025°	UHF, S, X	LEO, GEO	Flown LEO	Yes
Argotec [Italy]	400	<0.01°/<0.008°	UHF, S, X, K, Ka	LEO, MEO, GEO, Deep Space	Qualified LEO Under Development MEO, GEO, Deep Space	Yes
Artemis Space Technologies [UK]	1,250	0.01°/0.01°	UHF, S, X, Ka, Ku, Optical	LEO, MEO, GEO, Lunar, Deep Space	Qualified LEO, MEO, GEO, Lunar and Deep Space	No
Astranis Space Technologies Corp. [USA]	2,500	<0.1°/<0.01°	MIL-Ka, Ka, Ku, Q, V, X	MEO, GEO, Cislunar, Deep Space, Polar, High Inclination	Flown GEO	Yes
Astro Digital [USA]	2,000	<0.05°/<0.01°	UHF, S, X, Ku, Ka, V, W, Optical	LEO, GEO, Deep Space	Flown LEO	Yes
Astroscale [USA]	3,530	0.05°/0.025°	S, X, C	LEO, GEO	Under Development	Yes
BAES [USA]	2,500	<0.007°/<0.006°	L, S, X, Ka	LEO, MEO, GEO, Deep Space	Flown LEO, Qualified Deep Space	Yes
Berlin Space Technologies [Germany]	2,500	<0.017°/<0.017°	S, X	LEO	Flown LEO	Yes
Blue Canyon Technologies [USA]	1,082	0.0025°/0.0025°	L, S, X	LEO, GEO, Deep Space	Flown LEO and GEO Qualified Deep Space	Yes
CesiumAstro [USA]	4,500	<0.1°/<0.01°	S, L, Ka, Optical	LEO	Under Development	Yes
CREOTECH [Poland]	150	<0.015°/<0.015°	S, X, Optical	LEO, MEO, GEO, Lunar, Deep Space	Flown LEO	No
Deimos Space [Spain]	300	<0.005°/<0.005°	S, X, iDRS	LEO	Under Development	No
EnduroSat [Bulgaria]	300/3500	0.1°/<0.008°	UHF, S, X, K/Ka	LEO	Qualified LEO (300W) Under Development (3500W)	Yes
General Atomics EMS [USA]	450	0.03°/0.02°	S, X	LEO	Qualified LEO	Yes
Harpy Aerospace [India]	2,100	0.35°/0.35°	S, Ka, Optical	LEO	Qualified LEO	Yes

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Hemeria [France]	>1,000	<0.03°/<0.01°	S, X	LEO, GEO, GTO	Flown LEO Qualified GEO and GTO	No
LeoStella [USA]	2,000	0.013°/0.009°	UHF, S, X	LEO	Flown LEO	Yes
Lockheed Martin [USA]	500+	<0.1°/<0.1°	S, X, Ka	LEO, GEO, Lunar, Deep Space	Flown LEO Qualified GEO, Lunar and Deep Space	Yes
Loft Orbital [USA]	>1,000	<0.035°/<0.03°	S, X, L	LEO	Flown LEO	Yes
Magellan Aerospace [Canada]	200	0.01°/0.01°	S, X	LEO	Flown LEO	No
Malin Science Space Systems [USA]	918	<0.015°/<0.015°	UHF, X, Ka	Mars	Under Development	Yes
Moog [USA]	2,000	<0.050°/<0.033°	S, X, Ka, Optical	LEO, GEO	Flown LEO Under development GEO	Yes
Momentus Space [USA]	3,000	0.008°/0.008°	S, X, Ka, Optical	LEO, MEO, GEO, Lunar, Deep Space	Flown LEO	Yes
Muon Space [USA]	3,000	0.012°/0.03°	S, X, Ka	LEO	Flown LEO	Yes
NanoAvionics [Lithuania]	660	0.24°/0.09°	UHF, S, X	LEO	Flown LEO	Yes
Northrop Grumman [USA]	400	<0.01°/<0.008°	S, X, Ka	LEO, GEO, HEO	Flown LEO, GEO, and HEO	Yes
NovaWurks [USA]	>5,000	0.002°/0.0004°	UHF, S, L, X, Ka, Ku and Optical	LEO, MEO, GEO, GTO, HEO, Lunar and Deep Space	Flown LEO and GTO	Yes
OHB LuxSpace [Luxembourg]	834	<0.022°/ 0.01°	S, X	LEO	Qualified LEO	No
OHB Sweden [Sweden]	1,500	0.008°/0.008°	S, X, L	LEO, MEO	Flown LEO	No
Open Cosmos [UK]	2,200	<0.033°/0.03°	S, X	LEO	Under Development	No
Orbital Astronautics [UK]	5,000	0.05°/0.01°	S, X, K, Ka, Optical	LEO, MEO, GEO, Deep Space	Qualified LEO	No
Quantum Space [USA]	1,000	0.006°/0.006°	S, X, Ka	LEO, GEO, Cislunar, Lunar, Deep Space	Under Development	Yes
Reflex Aerospace [Germany]	>300	<0.01°/<0.01°	S, X, Ka, Ku, Optical	LEO	Under Development	No
Redwire Space [USA]	600	0.005°/0.0017°	X	LEO	Qualified LEO	Yes

Table 2-7: ESPA-Class Market Solutions
(The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the ESPA-Class category)

Organization [Headquarters]	Peak Power (W)	3-σ Pointing Control/ Knowledge	Comm Options	Intended Destination	Maturity	US Office
Table 2-7: ESPA-Class Market Solutions (The fields indicate maximum capability; organizations may offer multiple options including smaller capabilities within the ESPA-Class category)						
Sierra Space [USA]	500	0.001°/ <0.001°	UHF, S, X	LEO, MEO, GEO	Flown LEO	Yes
SITAEL [Italy]	2,000	0.0017°/0.001°	S, X	LEO	Qualified LEO	No
Southwest Research Institute [USA]	1,550	0.015°/0.002°	S, X, Ka	LEO, GEO	Flown LEO Under Development GEO	Yes
Space Dynamics Laboratory [USA]	1,600	0.021°/0.021°	UHF, S, X, Ka, Optical	LEO, GEO, GTO, Cislunar, Deep Space	Flown LEO	Yes
Space Flight Laboratory [Canada]	1,200	0.009°/0.004°	UHF, S, X, Ka	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	No
Space Inventor [Denmark]	400	<0.008°/<0.008°	VHF, UHF, S, X, L, Ka, Ku, QV	LEO, GEO, MEO	Qualified LEO	No
Surrey Satellite Technology Ltd. [UK]	2,000	<0.01°/<0.01°	S, X, Ka, Ku, ISL	LEO, Lunar	Flown LEO Under Development Lunar	No
Terran Orbital [USA]	1,500	0.014°/0.002°	UHF, S, X	LEO, GEO, Lunar	Flown LEO	Yes
XPLORE [USA]	950	0.17°/ 0.018°	S, X	VLEO, LEO, Cislunar	Under Development	Yes
York Space Systems [USA]	1,500	0.008°/0.004°	UHF, S, X, Ka, Ku, Optical	LEO, GEO, Lunar	Flown LEO Qualified GEO and Lunar	Yes

2.3   Programmatic and Systems Engineering Considerations

When determining the optimal mission design approach, small satellite mission developers should carefully evaluate the programmatic and systems engineering considerations that align most with their specific objectives and constraints. This assessment is crucial for making informed decisions that best serve the mission's goals and requirements. Example of these considerations include:

- Environments the system will endure during development and in flight
- Concept of operations, including desired orbit
- Functional and performance requirements
- Key performance parameters with appropriate margins (e.g., mass, volume, power, data link, data budget, pointing)
- Software considerations such as development environment and re-use
- Technology development considerations such as flight heritage, Technology Readiness Level (TRL), and reliability
- Risk posture for development and performance
- Trades between performance, cost, and schedule
- Procurement considerations such as production/lead time and contractual mechanisms
- Licensing requirements, as applicable (e.g., RF licensing, remote sensing, export control, re-entry)

In addition to the considerations listed above, hosted orbital service missions should also consider:

- Payload priority/mission lifetime for multi-customer/multi-manifest missions
- Balancing costs vs payload use of platform resources (e.g., mass, volume, power, data link, data budget, pointing)

Before finalizing any mission design decisions, it is essential to thoroughly analyze and consider these factors, along with numerous other considerations, for each potential option within the trade space. Given mission system performance requirements for key performance parameters like mass, volume, power, data link, data budget, and pointing, a functional importance rating and risk-based trade study should be used to screen the many options available. In addition to functional performance, relevant flight heritage or TRL, production lead time, and any available reliability data should be included in the trades. These, as well as cost, could drive the design to be done via COTS or commercial support.

Mission developers may want to take into consideration the following guides to help them in their selection and design process:

- NASA CubeSat 101 Book https://www.nasa.gov/content/cubesat-launch-initiative-resources
- NASA Systems Engineering Handbook https://www.nasa.gov/connect/ebooks/nasa-systems-engineering-handbook
- NASA Small Spacecraft Technology program Guidebook for Technology Development Projects https://www.nasa.gov/sites/default/files/atoms/files/smallsattechdevguidebook_rev-508d1.pdf

2.4   Summary

Several vendors have pre-designed fully integrated small spacecraft buses that are space-rated and available for purchase. The market ranges from companies that are willing to heavily modify their systems to fit the customer's needs to companies attempting to standardize their system

with little to no customization in favor of a better cost proposition. This chapter consolidated a long list of providers with key characteristics to facilitate the research and down-selection process for SmallSat practitioners.

For feedback about this chapter, email: arc-sst-soa@mail.nasa.gov. Please include a business email in case of follow up questions.

2.5   References

The references in this section are provided to facilitate the process in which practitioners can obtain information from the providers. The source indicates how the information provided in this chapter was obtained.

Source definition:

2025 = organization provided the information through direct communication with the State-of-the-Art team for the current edition of the document.

2024 = organization provided the information through direct communication with the State-of-the-Art team on the 2024 edition of the document, and the team was unable to communicate with the organization to update the current edition of the document.

Web = information obtained from a provider website or NASA website

Table 2-8: List of Contact Information for Organizations in this Chapter			
Organization	**Source**	**Contact Email**	**Website**
AAC Clyde Space	2025	enquiries@aac-clydespace.com	www.aac-clyde.space
Aerospacelab	2025	gerry.jansson@aerospacelab.com	www.aerospacelab.com
Airbus US Space & Defense	2024	deborah.horn@airbusus.com	www.airbusus.com
Alba Orbital	2025	contact@albaorbital.com	www.albaorbital.com
Alén Space	2025	sales@alen.space	www.alen.space
Apex Space	2025	General Inquiries Page	www.apexspace.com
Argotec	2025	info@argotecgroup.com	www.argotecgroup.com
Artemis Space Technologies	2024	info@spaceartemis.com	www.spaceartemis.com
Astranis Space Technologies Corp.	2025	scott@astranis.com	www.astranis.com
Astro Digital	2025	brian@astrodigital.com	www.astrodigital.com
Astroscale-us	2025	k.shahady@astroscale-us.com	www.astroscale-us.com
Axelspace	2024	Contact Page	www.axelspace.com
BAE Systems Inc., Space and Mission Systems (BAES)	2025	Space & Mission Systems Team Contact	www.baesystems.com
Berlin Space Technologies	2025	info@berlin-space-tech.com	www.berlin-space-tech.com
Blue Canyon Technologies	2024	info@bluecanyontech.com	www.bluecanyontech.com
C3S Electronics Development	2025	info@c3s.hu	www.c3s.hu
CesiumAstro	2025	info@cesiumastro.com	www.cesiumastro.com
CREOTECH	2025	kontakt@creotech.pl	www.creotech.pl/space
Deimos Space	2025	cmentrena@deimos-space.com	www.elecnor-deimos.com
DIYSATELLITE	2025	gus@diysatellite.com	www.diysatellite.com
EnduroSat	2025	Contact Page	www.endurosat.com
FOSSA Systems	2025	contact@fossa.systems	www.fossa.systems

Table 2-8: List of Contact Information for Organizations in this Chapter			
Organization	**Source**	**Contact Email**	**Website**
General Atomics EMS	2024	Chris.white@ga.com	www.ga.com/EMS
German Orbital Systems	2024	info@orbitalsystems.de	www.orbitalsystems.de
GomSpace	2024	info@gomspace.com	gomspace.com
Gran Systems	2025	info@gransystems.com	www.gransystems.com
GUMUSH AeroSpace	2025	gumush@gumush.com.tr	www.gumush.com.tr
Harpy Aerospace	2025	jayakumar@harpyaerospace.in	www.harpyaerospace.in
Hemeria	2025	Contact Form	www.hemeria-group.com/en
Hex20	2025	lloyd@hex20.com.au	www.hex20.com.au
Hydra Space Systems	2025	contacto@hydra-space.com	www.hydra-space.com
IMT	2024	giovanni.cucinella@imtsrl.it	imtsrl.it
Innova Space	2025	info@innova-space.com	innova-space.com/en
ISISPACE	2025	sales@isispace.nl	www.isispace.nl
LeoStella	2025	mike.kaplan@leostella.com	leostella.com
Lockheed Martin	2025	timothy.m.linn@lmco.com	-
Loft Orbital	2025	gautier@loftorbital.com	www.loftorbital.com
Magellan Aerospace	2025	rushi.ghadawala@magellan.aero	www.magellan.aero
Malin Space Science Systems	2025	yee@msss.com	www.msss.com
Moog	2025	Hfreeman2@moog.com	www.moog.com/markets/space
Momentus Space	2025	sales@momentus.space	www.momentus.space
Muon Space	2025	info@muonspace.com	www.muonspace.com
NanoAvionics	2025	info@nanoavionics.com	www.nanoavionics.com
Nara Space	2025	sales@naraspace.com	www.naraspace.com
NearSpace Launch	2025	nsl@nearspacelaunch.com	www.nearspacelaunch.com
Northrop Grumman	2025	John.Dyster@ngc.com	-
NovaWurks	2025	info@NovaWurks.com	www.novawurks.com
NPC SPACEMIND	2025	info@npcspacemind.com	www.npcspacemind.com
OHB LuxSpace	2025	info@luxspace.lu	www.luxspace.lu
OHB Sweden	2025	spacesales@ohb-sweden.se	www.ohb-sweden.se
Open Cosmos	2024	partnerships@open-cosmos.com	open-cosmos.com
Orbital Astronautics	2024	hello@orbastro.com	orbastro.com
Orion Space Solutions	2024	contact@orionspace.com	orionspace.com
Pumpkin Space Systems	2025	sales@pumpkininc.com	www.pumpkinspace.com
Quantum Space	2025	sales@quantumspace.us	www.quantumspace.us
Quub, Inc.	2025	info@quub.space	quub.space
Rocket Lab	Web	spacesystems@rocketlabusa.com	www.rocketlabusa.com
Redwire Space	2024	sales@redwirespace.com	www.redwirespace.com
Reflex Aerospace	2025	sales@reflexaerospace.com	www.reflexaerospace.com
SatRev	2025	engage@satrev.space	www.satrev.space
Sierra Space	2025	spaceapps@sierraspace.com	www.sierraspace.com
SITAEL	2025	sales.space@sitael.com	www.sitael.com
SkyLabs	2025	sales@skylabs.si	www.skylabs.si
Southwest Research Institute	2024	spacecraft-info@swri.org	-
Space Dynamics Laboratory	2025	info@sdl.usu.edu	www.sdl.usu.edu

Table 2-8: List of Contact Information for Organizations in this Chapter			
Organization	**Source**	**Contact Email**	**Website**
Space Flight Laboratory	2025	info@utias-sfl.net	www.utias-sfl.net
Space Information Laboratories	2024	sales@spaceinformationlabs.com	www.spaceinformationlabs.com
SpaceX	Web	Julianna.Scheiman@spacex.com	www.spacex.com
Space Inventor	2024	sales@space-inventor.com	space-inventor.com
Spacemanic	2025	sales@spacemanic.com	www.spacemanic.com
Spire Global	2024	joe.carroll@spire.com	www.spire.com
Surrey Satellite Technology Ltd.	2025	info@sstl.co.uk	www.sstl.co.uk
Terran Orbital	2025	info@terranorbital.com	terranorbital.com
Varda Space Industries	Web	info@varda.com	www.varda.com
Xplore	2024	inquire@xplore.com	www.xplore.com
York Space Systems	2025	BD@yorkspacesystems.com	www.yorkspacesystems.com

Chapter Contents

Chapter Glossary

(AFRL)	Air Force Research Laboratory
(BMS)	Battery Management System
(BOL)	Beginning-of-Life
(CFRPs)	Composite Fiber Reinforced Panels
(CIGS)	Cu(In,Ga)Se2
(COTS)	Commercial-off-the-Shelf
(EOL)	End-of-Life
(EPS)	Electrical Power System
(ESA)	European Space Agency
(GaN)	Galium Nitride
(GRC)	NASA Glenn Research Center
(KSC)	Kennedy Space Center
(Li-ion)	Lithium-ion
(LiCF$_x$)	Lithium carbon monofluoride
(LiPo)	Lithium polymer
(LiSO$_2$)	Lithium sulfur dioxide
(LiSOCl$_2$)	Lithium thionyl chloride
(MIL)	Military
(QML)	Qualified Manufacturers List
(NiCd)	Nickel-cadmium
(NiH$_2$)	Nickel-hydrogen
(OPV)	Organic Photovoltaic
(OSCAR)	Optical Sensors based on carbon materials
(PCB)	Printed Circuit Board
(PEASSS)	Piezoelectric Assisted Smart Satellite Structure
(PET)	polyethylene terephthalate
(PMAD)	Power management and distribution
(RHUs)	Radioisotopic Heater Units
(RTGs)	Radioisotope Thermoelectric Generators
(SABER)	Solid-state Architecture Batteries for Enhanced Rechargeability and Safety
(SP)	Specific Power
(SWaP)	Size, Weight, and Power
(TPV)	Thermophotovoltaic
(TR)	Thermoradiative
(TRL)	Technology Readiness Level
(Wh kg^{-1})	Watt hours per kilogram

3.0 Power

3.1 Introduction

The electrical power system (EPS) is a major, fundamental subsystem that encompasses electrical power generation, storage, and distribution, and commonly comprises a large portion of volume and mass in any given spacecraft. Power generation technologies include photovoltaic cells, panels and arrays, and radioisotope or other thermonuclear power generators. Power storage is typically applied through batteries; either single-use primary batteries or rechargeable secondary batteries. Power management and distribution (PMAD) systems facilitate power control to spacecraft electrical loads and take a variety of forms that are often custom designed to meet specific mission requirements. EPS engineers frequently target a high specific power or power-to-mass ratio (Wh kg^{-1}) when selecting power generation and storage technologies to minimize system mass. The EPS volume is most likely to be the constraining factor for nanosatellites.

CubeSats and SmallSats typically operate in a mild radiative environment for short periods in low-Earth orbits, so stringent qualification standards and high Technology Readiness Level (TRL) tend to not be as relevant as on deep space missions. Therefore, EPS engineers should note some fundamental differences between commercial-off-the-shelf (COTS) parts and space-qualified parts while weighing those differences against spacecraft requirements. Typically, Military or Space (MIL/QML) qualified parts go through a series of specific tests, while COTS go through less stringent ones. For example, Military or Space parts are typically tested and qualified to survive -55°C to 125°C, while the alternative COTS requirement is -40°C to 85°C. The same trend is true for other factors that are a part of the MIL/QML qualification process like radiation, reliability, etc. COTS parts are typically known to have higher performance, while space qualified parts typically have relatively higher reliability. Another key limitation in QML parts is their lack of availability and slow revision timeline. All in all, we find that COTS parts are in many cases more suitable for use in SmallSat designs.

In this chapter, the terms SmallSat and CubeSat are often used in the same context, however, the reader needs to be aware of distinctions between the two types of spacecraft. Please refer to the introduction of this report for more information on the categories of SmallSats. CubeSats fall under the category of both microsatellites and nanosatellites, and CubeSat missions commonly use COTS parts for space applications. Due to their nearly exclusive use in low-Earth orbit applications, CubeSats are more likely to incorporate COTS parts as they typically feature shorter mission lengths, more favorable environmental conditions, and as a result, need less stringent standards when qualifying parts. Knowing the distinction between a CubeSat and a SmallSat is necessary for determining the potential for incorporating COTS parts in a SmallSat design.

The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

In this chapter we will review the following categories:

- Sections 3.2 & 3.3: Power Generation, including solar cells, panels and arrays,
- Sections 3.4 & 3.5: Energy Storage, including Li-ion, Lipo, supercapacitors and solid-state batteries, and
- Sections 3.6 & 3.7: Power Management, including modular architectures and wireless power transfer and telemetry.

3.2   State-of-the-Art – Power Generation

Power generation on SmallSats is a necessity typically governed by a common solar power architecture (solar cells + solar panels + solar arrays). As the SmallSat industry drives the need for lower cost and increased production rates of space solar arrays, the photovoltaics industry is shifting to meet these demands. The standardization of solar array and panel designs,
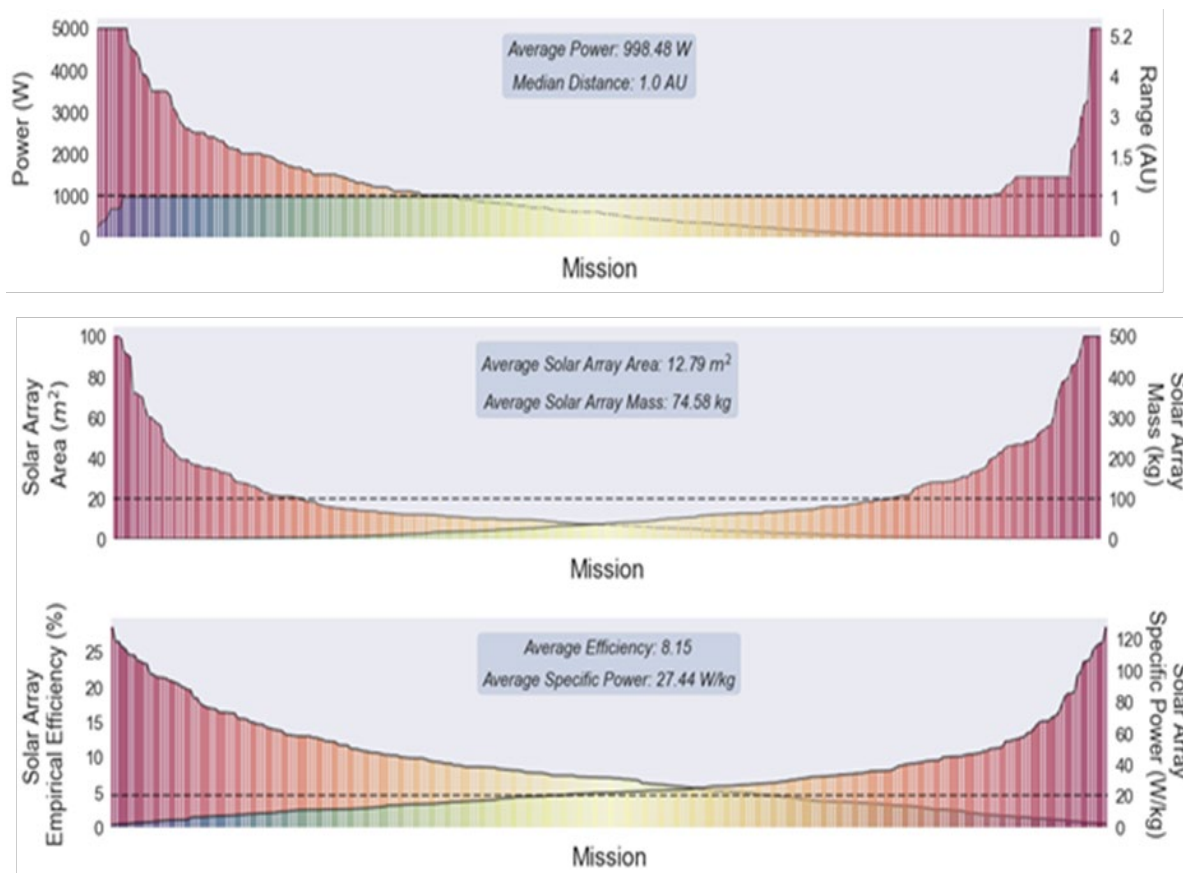
Figure 3.1: (Top) Distribution of mission range, or the furthest point from the sun that the spacecraft reaches, and mission power levels [power capped at 5 KW]. (Middle) Distribution of solar array surface area and solar array mass [mass capped at 500 Kg]. (Bottom) Distribution of solar array empirical efficiency (calculated at Earth) and specific power (for the entire array measured at the destination of the mission), Peretz et al. 2024 (92). X axis represents 389 missions and color scheme represents highest to lowest values (red to blue). Credit: NASA.

deployment mechanisms, and power integration will be critical to meet the desire for large, proliferated constellations.

EPS engineers should note beginning-of-life (BOL) vs end-of-life (EOL) performance of the systems as well as their planned testing hours for such systems while on the ground prior to operations. Typically, EPS for SmallSats is over-engineered to handle a dynamic thermal environment, eclipse durations while in LEO, or any other operational scenarios or mission needs while in eclipse at varying sun-angles. Figure 3.1 captures actual space system performances given such wide varying operational conditions for 389 space missions (not only SmallSats) launched since 1989 to 2021 reviewed in the publication (1). The dotted horizontal line indicated where more of the 389 missions surveyed fall under.

When possible, choosing a pre-designed and qualified panel is preferred over designing unique solar panels to reduce the cost and schedule as well as unforeseen design and manufacturing issues. Companies that have the capacity for mass production and automation are rare because space solar arrays, cells, and panels have always been a 'boutique' business; however, standardized designs like the OneWeb and StarLink constellations have been appearing more often to meet the demands of highly proliferated constellations.

The following subsections aim to capture the current state of the art and assist EPS engineers, mission designers, system engineers, etc., in designing, reviewing and ultimately constructing and operating power flight systems.

3.2.1   Solar Cells

Solar power generation is the predominant method of power generation on small spacecraft. As of 2021, over 90% of all nanosatellite/SmallSat form factor spacecraft were equipped with solar panels and rechargeable batteries (1). Limitations to solar cell use include diminished efficacy in deep-space applications, no generation during eclipse periods, degradation over mission lifetime (due to aging and radiation), high surface area, mass, and cost. To pack more solar cells into the limited volume of SmallSats and NanoSats, mechanical deployment mechanisms can be added, which may increase spacecraft design complexity and reliability, as well as risk. Photovoltaic cells, or solar cells, are made from thin semiconductor wafers that produce an electric current when exposed to light. The light available to a spacecraft solar array, also called solar intensity, varies as the inverse square of the distance from the Sun. The projected surface area of the panels exposed to the Sun also affects power generation and varies as a cosine of the angle between the panel and the Sun.

While single-junction cells are cheap to manufacture, they carry a relatively low efficiency, usually around 20%, and are not included in this report. Modern spacecraft designers favor multi-junction solar cells made from multiple layers of light-absorbing materials that efficiently convert specific wavelength regions of the solar spectrum into energy, thereby using a wider spectrum of solar radiation (2). The theoretical efficiency limit for an infinite-junction cell is 86.6% in concentrated sunlight (2). However, in the aerospace industry, triple-junction cells are commonly used due to their high efficiency-to-cost ratio compared to other cells.

The current state of the art for space solar cells are multi-junction cells ranging from 3 to 5 junctions based on Group III-V semiconductor elements (like GaAs). SmallSats and CubeSats typically use some of the highest performing cells that provide efficiencies over 32%, even though they have a substantially higher cost than terrestrial silicon solar cells (~20% efficient). Ultimately the size, weight, and volume of smaller satellites may be the determining factor in choosing solar cell technology, rather than solar cell efficiency. Being a life-limiting component on most spacecraft, the EOL performance at operating temperature is critical in evaluating their

performance. Common factors that degrade the functionality of solar cells include radiation exposure, coverglass/adhesive darkening, contamination, and mechanical or electrical failure.

This section individually covers small spacecraft targeted cells, fully integrated panels, and arrays. Table 3-1 itemizes small spacecraft solar cell efficiency per the available manufacturers. Note the efficiency may vary depending on the solar cells chosen.

Company	Cell Name	BOL Efficiency	Voc (V)	Vmp (V)	Jsc (mA/cm²)	Jmp (mA/cm²)	Pmp (W/m²)	Ref
AZUR Space Germany	Silicon S 32	16.8	0.628	0.528	45.8	43.4	229.2	(4)
	3G30-Adv	29.5	2.7	2.411	17.2	16.71	403	
	4G32-Adv	31.5	3.426	2.999	15.2	14.37	431	
	TJ 3G28C	28	2.667	2.37	16.77	16.14	1367	
Rocket Lab USA	ZTJ	29.5	2.726	2.41	17.4	16.5	397.7	(5)
	ZTJ+	29.4	2.69	2.39	17.1	16.65	397.9	
	ZTJ Omega	30.2	2.73	2.43	17.4	16.8	408.2	
	Z4J	30.0	3.95	3.54	12	11.5	407.1	
	IMMα	32.0	4.78	4.28	10.7	10.12	433.1	
	ZTJM	29.5	2.72	2.38	17.1	16.5	392	
SpectroLab USA	XTJ	29.5	2.633	2.348	17.76	17.02	399.6	
	XTJ-Prime	30.7	2.715	2.39	18.1	17.4	415.9	(7)
	XTE-SF	32.2	2.75	2.435	18.6	17.8	433.4	(6)
	XTE-HF	32.1	2.782	2.49	18	17.4	427.9	
	XTE-LILT	31.6	2.755	2.459	18.1	17.4	427.9	
	UTJ	28.4	2.66	2.35	17.14	16.38	384.93	(8)
	TASC	27	2.52	2.19	32	28	270	(9)
	ITJ	26.8	2.565	2.27	16.9	16	1353	(10)
Emcore Corporation USA	BTJ	28.5	2.7	2.37	17.1	16.3	386	(11)
	ZTJ	29.5	2.726	2.41	17.4	16.5	397	

Table 3-1: Solar Cells Product Table

3.2.2 Solar Panels & Arrays

Solar panels & arrays are constructed from individual solar cells connected in series to form strings and in parallel to form circuits mounted on a substrate backing (e.g., Figure 3.2). While very low-power CubeSats and SmallSats may only need body-mounted solar panels, most will require more power from deployed solar arrays. The deployed solar arrays for CubeSats and SmallSats are mostly on rigid substrates made of either a printed circuit board (PCB), composite fiber reinforced panels (CFRPs), or an aluminum honeycomb panel.
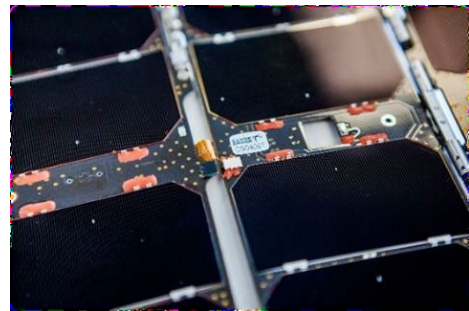
Figure 3.2: AAC Clyde Space solar arrays. Credit: AAC Clyde Space.

Deployed solar arrays are often the largest structure on a satellite; the ratio between the size of the deployed solar array and the size of the SmallSat may be much higher compared to other conventionally large spacecraft. The size and fundamental frequency of the solar arrays impact spacecraft pointing, propulsion, and delta-V needed for station keeping. Important considerations for SmallSat solar
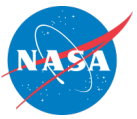
arrays are deployment mechanisms, deployed frequency, panel specific power, and power density, as well as stowed volume. Most of these metrics are not listed on manufacturer datasheets.

Solar array comparison can be challenging because SmallSat/CubeSat manufacturers who make solar arrays specific to their bus and payload designs often do not report solar array power using the same metrics. Their reported "power" can mean multiple things: power available to the payload, peak power provided by a combination of solar array and battery, or an orbital-specific average power. Reported solar array power (Peak BOL) mainly refers to the peak power of the solar array at the beginning of life, 28°C which is mission independent. Panel stiffness and moment of inertia usually need to be calculated for a specific spacecraft because they are dependent on multiple factors such as the size and mass of the panel, as well as spacecraft size and weight distribution. Examples of commercial solar array and panel products are shown in Table 3-2.

Table 3-2: Solar Array/Panel Products					
Company [Headquarters]	Product	Panel Type	Specific Power (W/kg)	Peak BOL Solar Array Power (W)	Ref
AAC Clyde Space [Sweden]	Photon	*	*	9.25 per 3U-12 Face	(12)
Agencia Espacial Civil Ecuatoriana [Italy]	Deployable Multifunction Solar Array	Deployed Rigid	-	-	
Airbus Netherlands [The Netherlands]	Sparkwing Solar Panel 600x570mm 36V	Body Mounted, 1-3 deployed panels, Rigid	18	88 per panel	(13)
	Sparkwing Solar Panel 600x700mm 36V	Body Mounted, 1-3 deployed panels, Rigid	21	111 per panel	
	Sparkwing Solar Panel 600x820mm 36V	Body Mounted, 1-3 deployed panels, Rigid	24	133 per panel	
	Sparkwing Solar Panel 600x965mm 36V	Body Mounted, 1-3 deployed panels, Rigid	25	155 per panel	
	Sparkwing Solar Panel 750x820mm 36V	Body Mounted, 1-3 deployed panels, Rigid	29	177 per panel	
	Sparkwing Solar Panel 750x965mm 36V	Body Mounted, 1-3 deployed panels, Rigid	30	199 per panel	
	Sparkwing Solar Panel 750x1100mm 36V	Body Mounted, 1-3 deployed panels, Rigid	34	243 per panel	
	Sparkwing Solar Panel 910x820mm 36V	Body Mounted, 1-3 deployed panels, Rigid	33	221 per panel	
	Sparkwing Solar Panel 910x965mm 36V	Body Mounted, 1-3 deployed panels, Rigid	37	265 per panel	
	Sparkwing Solar Panel 1000x570mm 36V	Body Mounted, 1-3 deployed panels, Rigid	29	176 per panel	
	Sparkwing Solar Panel 1000x1100mm 36V	Body Mounted, 1-3 deployed panels, Rigid	39	309 per panel	
	Sparkwing Solar Panel 1070x820mm 36V	Body Mounted, 1-3 deployed panels, Rigid	37	265 per panel	
	Sparkwing Solar Panel 600x570mm 50V	Body Mounted, 1-3 deployed panels, Rigid	19	92 per panel	
	Sparkwing Solar Panel 600x800mm 50V	Body Mounted, 1-3 deployed panels, Rigid	22	122 per panel	
	Sparkwing Solar Panel 600x965mm 50V	Body Mounted, 1-3 deployed panels, Rigid	25	153 per panel	
	Sparkwing Solar Panel 600x1100mm 50V	Body Mounted, 1-3 deployed panels, Rigid	28	183 per panel	

Table 3-2: Solar Array/Panel Products					
Company [Headquarters]	Product	Panel Type	Specific Power (W/kg)	Peak BOL Solar Array Power (W)	Ref
Airbus Netherlands [The Netherlands]	Sparkwing Solar Panel 750x570mm 50V	Body Mounted, 1-3 deployed panels, Rigid	23	122 per panel	
	Sparkwing Solar Panel 750x800mm 50V	Body Mounted, 1-3 deployed panels, Rigid	25	153 per panel	
	Sparkwing Solar Panel 750x965mm 50V	Body Mounted, 1-3 deployed panels, Rigid	36	241 per panel	
	Sparkwing Solar Panel 750x1100mm 50V	Body Mounted, 1-3 deployed panels, Rigid	34	244 per panel	
	Sparkwing Solar Panel 910x570mm 50V	Body Mounted, 1-3 deployed panels, Rigid	26	153 per panel	
	Sparkwing Solar Panel 910x800mm 50V	Body Mounted, 1-3 deployed panels, Rigid	32	214 per panel	
	Sparkwing Solar Panel 910x965mm 50V	Body Mounted, 1-3 deployed panels, Rigid	34	244 per panel	
	Sparkwing Solar Panel 910x1100mm 50V	Body Mounted, 1-3 deployed panels, Rigid	40	305 per panel	
	Sparkwing Solar Panel 1070x570mm 50V	Body Mounted, 1-3 deployed panels, Rigid	29	183 per panel	
	Sparkwing Solar Panel 1070x800mm 50V	Body Mounted, 1-3 deployed panels, Rigid	34	244 per panel	
	Sparkwing Solar Panel 1070x965mm 50V	Body Mounted, 1-3 deployed panels, Rigid	39	305 per panel	
	Sparkwing Solar Panel 1070x1100mm 50V	Body Mounted, 1-3 deployed panels, Rigid	45	366 per panel	
	Sparkwing Solar Panel 1230x570mm 50V	Body Mounted, 1-3 deployed panels, Rigid	31	214 per panel	
	Sparkwing Solar Panel 1230x800mm 50V	Body Mounted, 1-3 deployed panels, Rigid	36	275 per panel	
Blue Canyon Technologies [USA]	BCT Solar Array	Body Mount + Deployed Rigid	*	28 – 42 for 3U / 48-118 for 6U-12U	(14)
DHV Technologies [Spain]	CubeSats Solar Panels	Body Mounted (Polyimide)	50	2 for 1U Face	(15)
	CubeSats Solar Panels	Body Mounted (Polyimide)	49	4 for 2U Face	
	CubeSats Solar Panels	Body Mounted (Polyimide)	75	8 for 3U Face	
	CubeSats Solar Panels	Body Mounted (Polyimide)	68	18 for 6U Face	

Table 3-2: Solar Array/Panel Products					
Company [Headquarters]	Product	Panel Type	Specific Power (W/kg)	Peak BOL Solar Array Power (W)	Ref
DHV Technologies [Spain]	CubeSats Solar Panels	Deployed Rigid (Polyimide)	42	12 for 3U Double Deployable and Body Mounted	
	CubeSats Solar Panels	Deployed Rigid (Polyimide)	69	57 for 6/12U Double Deployable and Body Mounted	
	CubeSats Solar Panels	Deployed Rigid (Polyimide)	108	34 for 3U Quadruple Deployable	
	CubeSats Solar Panels	Deployed Rigid (CFRP)	69	68 for 6U Quadruple Deployable	
	CubeSats Solar Panels	Body Mounted (Polyimide)	50	2 for 1U Face	
	CubeSats Solar Panels	Body Mounted (Polyimide)	49	4 for 2U Face	
	Body mounted solar array panel	Sandwich CFRP substrate	84	179	
	Body mounted solar array panel	Low thickness monolithic CFRP substrate	140	96	
	Multiple deployable solar array wing	Sandwich CFRP substrate	57	697	
EnduroSat [Bulgaria]	1U Solar Panel	Deployed Rigid	63	2.4	(16)
	1.5U Solar Panel	Deployed Rigid	55	3.6	
	3U Solar Panel/Array	Deployed Rigid	66	14.4	
	6U Solar Panel/Array	Deployed Rigid	64	19.2	
	8U Solar Panel/Array	Deployed Rigid	65	24.0	
ExoTerra Resource [USA]	Fold Out Solar Arrays (FOSA)	Deployed Flexible	140	150	(17)
GomSpace [Denmark]	Nanopower DSP	Deployed Rigid	*	1.2	(18)
	NanoPower P110	Fixed	36.9 – 92	1.2 per cell 2.4	(19)
	NanoPower MSP 16 cell	Fixed	42	1.2 per cell 19.2	(20)
	NanoPower TSP Per wing	Deployed Rigid	60	45	
ISISPACE [The Netherlands]	Smallsat Solar Panels	Body Mount + Deployed Rigid	46	2.3 per U	(21)
Lockheed Martin [USA]	SmallSat Solar Array	Deployed Rigid with Additively Manufactured Substrate	53.6	170 per panel BOL at 28C	
MMA Design [USA]	Hawk	Deployed Rigid (PCB)	121	36-112	(22)

Table 3-2: Solar Array/Panel Products					
Company [Headquarters]	Product	Panel Type	Specific Power (W/kg)	Peak BOL Solar Array Power (W)	Ref
MMA Design [USA]	zHawk	Deployed Rigid (PCB)	95	36	
	Next-Gen HaWK	Deployed Rigid (PCB)	100	45 - 310	
NPC Spacemind [Italy]	SP1X	Body Mounted 1U	68	2.4	
	SP1Z	Body Mounted 1U Top	68	2.4	
	SP2X	Body Mounted 2U	64	4.8	
	SP2Z	Body Mounted 2U Top	64	4.8	
	SP3X	Body Mounted 3U	76	8.4	
	SP4X	Body Mounted 4U	75	10.8	
	SP4Z	Body Mounted 4U Top	58	9.6	
	SP6X	Body Mounted 6U	74	19.2	
	SP8X	Body Mounted 8U	68	24	
	SP3X Deployable	Double wing Deployable 3U	75	21.6	
	SP4X Deployable	Double wing Deployable 4U	58	28.8	
	SP6X Deployable	Double wing Deployable 6U	74	57.6	
	SP8X Deployable	Double wing Deployable 8U	68	72	
Pumpkin Space Systems [USA]	Dual Articulated Deployable Solar Array	Deployed Rigid	31	135	(24)
	Dual-Quad Articulated Deployable Solar Array	Deployed Rigid	30	240	
	Deployable Clamshell Solar Array (DCSA)	Deployed Rigid	44	220	(25)
	Deployable Clamshell Solar Array (DCSA)	Deployed Rigid	38	350	
Redwire Space [USA]	ROSA	Flexible PV blanket	100	1000	(26)
	Aladdin SmallSat Array	Hybrid Array: Flex Rigid	80	300	
Sierra Space Corp	Surface Mount Technology Solar Panel	Composite		2500	
Spacemanic [Czech Republic]	RA_Solar_Panels	Triple Junction GaAs	46	2.4	
Space Dynamics Laboratory [USA]	SDL Modular Solar Panels	Deployed Rigid	84.5	180 per panel	

* Available with inquiry to manufacturer

** For SmallSat use

While solar arrays efficiency has been the prevailing way to characterize solar array performance, discrepancies between theoretical and empirical data indicate that specific power (SP) of the solar array fundamentally governs space mission feasibility and flexibility. Figure 3.3 visualizes the landscape of mission architectures, parameterized by power (adjusted by a factor of squared distance) and mass; the relation between these axes is the SP.
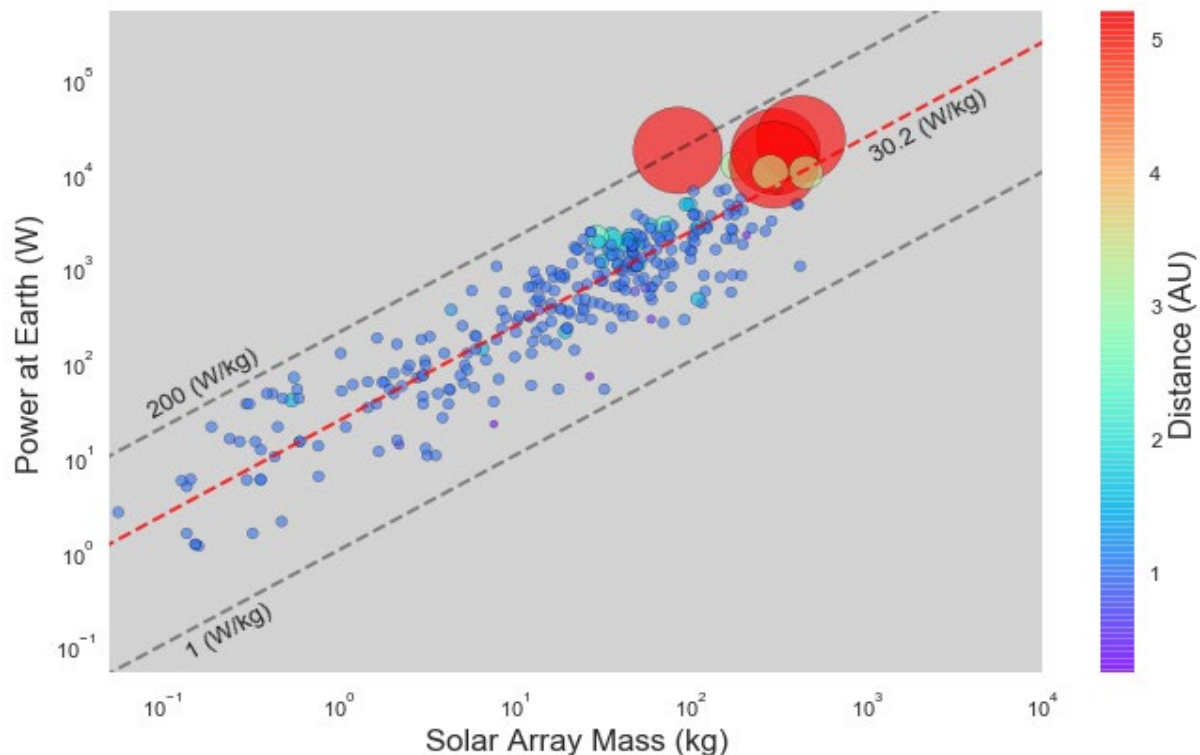
Figure 3.3: Specific power of a solar array at Earth. A minimum of 1 W/kg is plotted as a dashed black line, showing that no missions fly with any smaller specific power. The larger red circles represent the missions farther from Earth corresponding to the red color on the distance scale. Peretz et al (2024). Credit: NASA.

Space missions using solar arrays, regardless of spacecraft mass, are strongly clustered around ~30 W/kg (red dashed line) and are strongly bounded: no missions fly with SP lower than 1 W/kg (lower dashed line), and the maximum empirical SP of this dataset is 200 W/kg (upper dashed line). There are two clearly unoccupied regions: the empty region of high-mass and low-power is of little interest since this is not generally desirable; however, it is interesting that the high-power low-mass regime is empty, indicating that while this is a highly desirable region, it is technologically inaccessible.

With the average mission power consumption of 1000 W and a medium value of 600 W, Figure 3.4 shows what maximal ranges can be achieved with three hypothetical solar array technologies with specific power levels of 10, 100 and 1600 W/Kg (the lines show how the Empirical Specific power changes as a function of range until it reaches the lower limit of 1 W/Kg). The color-coded regions are where maximal surface area of 100 m^2 could be contained for power levels of 100,

300, and 1000 W. The areas under the line and within the colored region indicate where current (empirical evidence) and future (theoretical) missions could exist.

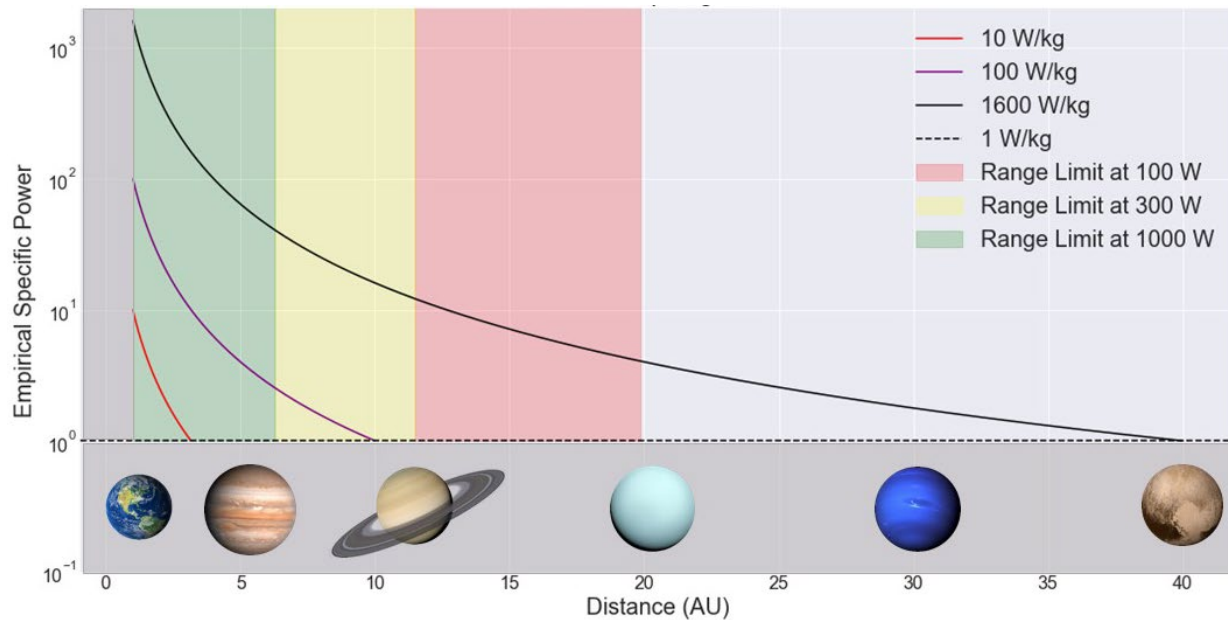*Figure 3.4: Practical power operation limit for spacecraft. Horizontal Axis is range in AU, Vertical is Specific Power in W/Kg. Lines represent a hypothetical solar array Specific Power. Colored regions show where a solar array with surface area smaller than 100 m^2 could exist (green is more spacecraft power and red is less spacecraft power). Peretz et al (2024). Credit: NASA.*

There are two possible approaches to improving solar array specific power: increase generated power or decrease array mass. The former has been the focus of the community for the past 40 years through improving efficiencies. However, even if triple-junction solar cell efficiency improves to the theoretical limit of 68%, the surface area, mass, and storage volume required to support median power requirements for exploration of deep space are beyond the point of feasibility. The mass required from such solar array structures would be measured in thousands of kilograms; clearly a reduction in solar array mass is needed for deep space exploration missions to be feasible.

Future work is needed to reduce solar array mass; deployment mechanism mass could be reduced with light-weight components or alternate configurations (for example, using a tension-based deployment instead of a motor-driven system) to eliminate the requirement for a motor and many spacecraft integration components. Mass reduction for the solar cells themselves can come by reducing cell thickness or increasing flexibility to increase launch vehicle stowage volume. Creating highly foldable arrays may be a desirable solution, with mass decreases possible in both deployment mechanisms and in the design of solar cells.

3.3   On the Horizon – Power Generation

New technologies continue to be developed for space-qualified power generation. Promising technologies applicable to small spacecraft include advanced multi-junction, flexible and organic solar cells, hydrogen fuel cells, and a variety of thermo-nuclear and atomic battery power sources.

3.3.1 Multi-junction Solar Cells

Fraunhofer Institute for Solar Energy Systems has developed different four-junction solar cell architectures that currently reach up to 38% efficiency under laboratory conditions, although some designs have only been analyzed in terrestrial applications and have not yet been optimized (Lackner). Fraunhofer ISE and EV have achieved 33.3% efficiency for a 0.002 mm thin silicon-based multi-junction solar cell, and future investigations are needed to solve current challenges of the complex inner structure of the sub-cells (28). Additionally, SpectroLab has been experimenting with 5- and 6-junction cells with a theoretical efficiency as high as 70% (29).

A collaboration between the Air Force Research Laboratory (AFRL) and SolAero has developed Metamorphic Multi-Junction (IMM-α) solar cells that are less costly with increased power efficiency for military space applications (2). The process for developing IMM-α cells involves growing them upside down, where reversing the growth substrate and the semiconductor materials allow the materials to bond to the mechanical handle, resulting in the more effective use of the solar spectrum (2). A single cell can leverage up to 32% of captured sunlight into available energy. This also results in a lighter, more flexible product. These cells had their first successful orbit in low-Earth orbit in 2018, and since then they have operated in low-Earth orbit on other CubeSat missions.

3.3.2 Flexible Solar Cells

Flexible and thin-film solar cells have an extremely thin layer of photovoltaic material placed on a substrate of glass or plastic. Traditional photovoltaic layers are around 350 microns thick, while thin-film solar cells use layers just one micron thick. This allows the cells to be flexible, lightweight, and cheaper to manufacture because they use less raw material. The performance of commercial flexible CIGS was investigated and reported with the potential for deep space applications at the University of Oklahoma. The authors found promising thin-film solar material using $Cu(In, Ga)Se_2$ (CIGS) solar cells with recorded power conversion efficiencies up to 22.7% (30).

3.3.3 Organic Solar Cells

Another on the horizon photovoltaic technology uses organic or "plastic" solar cells. These use organic electronics or organic polymers and molecules that absorb light and create a corresponding charge. A small quantity of these materials can absorb a large amount of light making them cheap, flexible, and lightweight.

Toyobo Co., Ltd. and the French government research institute CEA have succeeded in making trial organic photovoltaic (OPV) small cells on a glass substrate. Trial OPV modules on a lightweight and thin polyethylene terephthalate (PET) film substrate were demonstrated during their joint research project. Toyobo and CEA succeeded in making the OPV small cells on a glass substrate with the world's top-level conversion efficiency by optimizing the solvents and coating technique. In a verification experiment under neon lighting with 220 lux, equivalent to the brightness of a dark room, the trial product was confirmed to have attained a conversion efficiency of about 25%, or 60% higher than that of amorphous silicon solar cells commonly used for desktop calculators (31).

In October 2016, the Optical Sensors based on carbon materials (OSCAR) stratospheric-balloon flight test demonstrated organic-based solar cells for the first time in a stratospheric environment. While more analysis is needed for terrestrial or space applications, it was concluded that organic solar energy has the potential to disrupt "conventional" photovoltaic technology (32). Since then, a joint collaborative agreement between the German Aerospace Center and the Swedish National
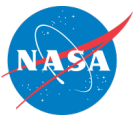
Space Board REXUS/BEXUS has made the balloon payload available for European university student experiments collaborating with the European Space Agency (ESA) (33).

No standardized stability tests are yet available for organic-based solar cell technology, and challenges remain in creating simultaneous environmental influences that would permit an in-depth understanding of organic photovoltaic behavior, but these achievements are enabling progress in organic-based solar cell use. In 2018, Chinese researchers in organic photovoltaics were able to reach 17% power conversion energy using a tandem cell strategy. This method uses different layers of material that can absorb different wavelengths of sunlight, which enables the cells to use more of the sunlight spectrum, which has limited the performance of organic cells (34).

3.3.4   Fuel Cells

Hydrogen fuel cells are appealing due to their small, light, and reliable qualities, and high energy conversion efficiency. They also allow missions to launch with a safe, storable, low-pressure, and non-toxic fuel source. An experimental fuel cell from the University of Illinois that is based on hydrogen peroxide rather than water has demonstrated an energy density of over 1000 Wh kg^{-1} with a theoretical limit of over 2580 Wh kg^{-1} (35). This makes them more appealing for interplanetary missions and during eclipse periods, however unlike chemical cells, they cannot be recharged on orbit. Carrying a large fuel tank is not feasible for small or nanosatellite missions. Regenerative fuel cells are currently being researched for spacecraft applications. Today, fuel cells are primarily being proposed for small spacecraft propulsion systems rather than for power sub-systems (36).

3.4   State-of-the-Art – Energy Storage

Solar energy is not always available during spacecraft operations; the orbit, mission duration, distance from the Sun, or peak loads may necessitate stored, onboard energy. Primary and secondary batteries are used for power storage and are classified according to their different electrochemistry. As primary-type batteries are not rechargeable, they are typically used for short mission durations. Silver-zinc is typically used as they are easier to handle and discharge at a higher rate, however, there are also a variety of lithium-based primary batteries that have a higher energy density, including lithium Sulfur dioxide (LiSO$_2$), lithium carbon monofluoride (LiCF$_x$) and lithium thionyl chloride (LiSOCl$_2$) (37).

Secondary-type batteries include nickel-cadmium (NiCd), nickel-hydrogen (NiH$_2$), lithium polymer (LiPo) and lithium-ion (Li-ion), which have been used extensively in the past on small spacecraft. Lithium-based secondary batteries are commonly used in portable electronic devices because of their rechargeability, low weight, and high energy, and have become ubiquitous on spacecraft missions. They are generally connected to a primary energy source (e.g., a solar array) and can provide rechargeable power-on-demand. Each battery type is associated with certain applications that depend on performance parameters, including energy density, cycle life, and reliability (38). Figure 3.5 shows some popular 18650 Lithium-Ion cells and their specific energy densities. While legacy cells had a specific energy of less 200 Wh/kg, latest cells have all exceeded 240 Wh/kg. Traditionally, vendors pack these 18650 cells in various configurations to meet customer needs. Table 3-3 shows a list of battery pack assemblers with their products and TRLs.

18650 Battery Cell Specific Energy Density
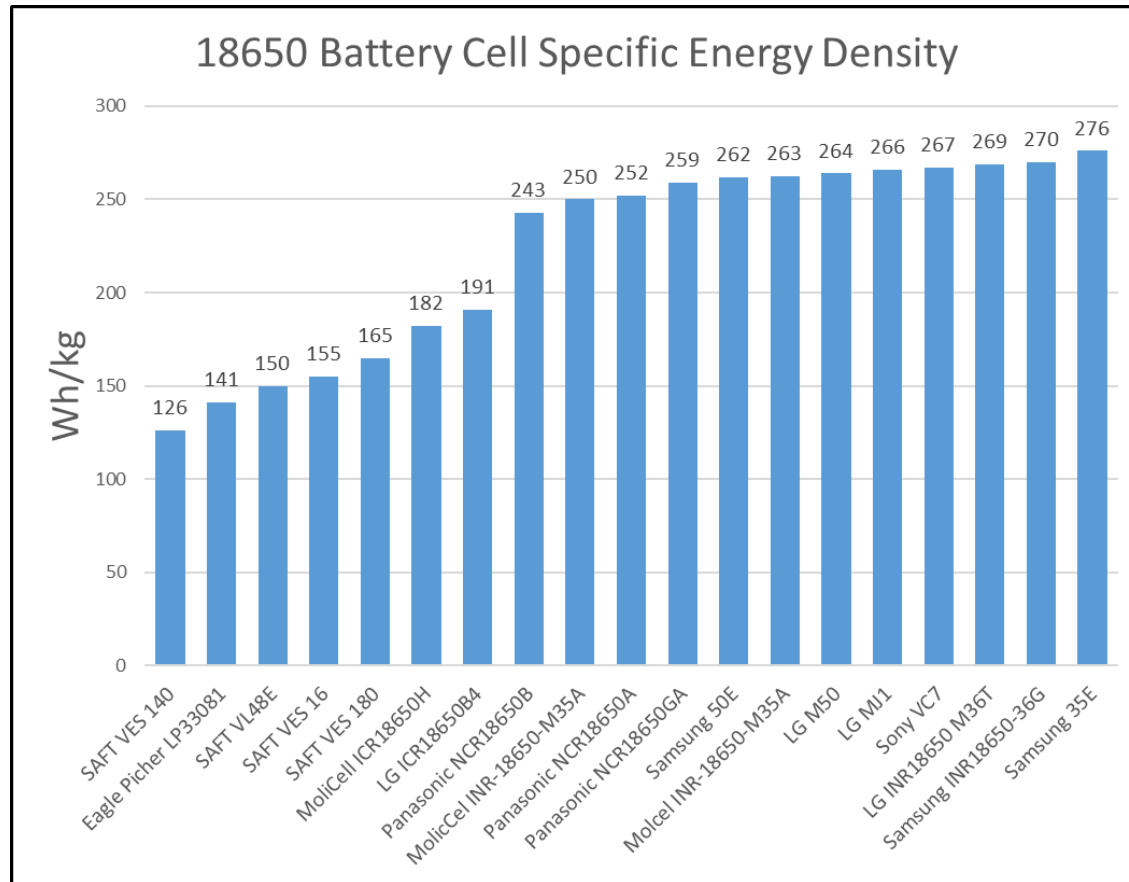
Figure 3.5: Battery cell energy density. Credit: NASA.

This section will discuss the individual chemical cells as well as pre-assembled batteries of multiple connected cells offered from multiple manufacturers. Due to small spacecraft mass and volume requirements, the batteries and cells in this section will be arranged according to specific energy, or energy per unit mass. However, several other factors are worth considering, some of which will be discussed below (37).

Table 3-3: Battery (Pack) Product Table							
Company [Headquarters]	Product	Volumetric Energy Density [Wh L^{-1}]	Specific Energy [Wh kg^{-1}]	Typical Capacity [Ah]	Max Discharge Rate [A]	Cells Used	Ref
AAC Clyde Space [Sweden]	Optimus	169.5	119.4	4.84	2.6	Clyde Space Li-Polymer	(39)
Aerospacelab [Belgium]	BM	380	-	14	12 (in / out)	4 independent battery packs of 8 lithium-ion cells each	
Astro Digital	Direct Energy Pack	187	144	10	50	21700 Li-Ion	
Argotech [Italy]	ELEKTRA	243.5	152.5	3.4	4	Li-Ion	(40)
Berlin Space Technologies [Germany]	BAT-110 Modular Battery (Nominal 3 strings)	69.73	57.75	7.5	3	Li-Fe	
EaglePicher Technologies [USA]	NPD-002271	271	153.5	14.5	15	EaglePicher Li-ion	
EnduroSat [Bulgaria]	EPS IV (BMS)	222.9	178.1	33.5	100	Li-Ion	
GomSpace [Denmark]	Nanopower BPX 3000mAh (e.g., 4S-2P)	262.2	172	6.0	4	GomSpace NanoPower Li-ion	(42)
	Nanopower BP4 3000mAh (e.g., 2S-2P)	239.8	168.8	6.0	4	GomSpace NanoPower Li-ion	(43)
	Nanopower BP8 (8S-1P)	227.36	177.3	3.0	4	GomSpace NanoPower Li-ion	
GUMUSH AeroSpace [Istanbul]	n-ART BAT	184.5	155.1	6.01	8	Li-Ion	
Ibeos [USA]	28V Modular Battery	151.1	109.8	8 - 30+	20	*	(44)
ISISPACE [The Netherlands]	PBP-2S1P	129.1	156.5	6.4	4	Li-Ion	
	PBP-2S2P	327.85	160	12.8	8	Li-Ion	
	PBP-4S1P	327.85	160	12.8	4	Li-Ion	
NPC SPACEMIND [Italy]	KANON	167	250	13.6	6	Li-Ion	
Saft [France]	VES16 4S1P	109.2	91	4.5	4.5 – Cont. 9 – Pulse	SAFT Li-ion	(46)
SkyLabs [Slovenia]	SKY-NANOeps-BMM	103	89	up to 18	up to 54	LiFePO4	(47)
Space Dynamics Laboratory [USA]	SDL 12V Battery Pack	144	75	12.0	96	EaglePicher LP32975	
Vectronic Aerospace GmbH [Germany]	VLB-X	101.96	74.6	12	10 – Cont. 20 – Pulse	SAFT Li-Ion	(48)

* Available with Inquiry to Manufacturer

The chemistry and cell design impacts the volumetric and specific energy densities. This limit represents the total amount of energy available per unit volume or weight, respectively. Current top-of-the-line Li-ion energy cells exhibit ~270 Wh kg^{-1}. Li-ion batteries exhibit lower energy densities due to the inclusion of a battery management system (BMS), interconnects, and sometimes thermal regulation.

There are generally two groups of cells – high energy or high power. High power cells use a low resistance design, such as increasing coating surface area, or multiple points of contact for the current collector to the cell which can allow for lower overall resistance values and a higher rate of discharge. High energy cells work to optimize gravimetric energy densities to obtain the most energy from the cell. Some common methods to increase gravimetric energy densities are via the addition of silicon to the anode, the use of high voltage cathodes, or using a metallic lithium anode. However, these methods can significantly reduce the cyclability of the battery system in exchange for increased energy density.

In general, for space applications, high energy density is important because a battery with high gravimetric energy density will be cheaper to launch into orbit (higher battery capacity per unit mass). However, for some high pulse applications, high-power cells would meet mission needs with less weight. However, energy density is not the only factor to investigate during cell selection. For non-space commercial applications, faster degradation (lower cyclability) of the battery can be beneficial as the electronic device often lasts as long as the battery, and faster turnover of a device may lead to increased revenue.

While space-designed cells typically underperform in energy density, they over-perform in cyclability with many space-designed cells used for longer (~5-15 year) missions. However, not all degradation modes for the lithium-ion trend in a linear fashion, and trends often take time to settle, thus the test results don't necessarily show the best performing cell until others are further along in testing.

Due to the extremely short mission durations with primary cells, the current state-of-the-art energy storage systems use lithium-ion (Li-ion) or lithium-polymer (LiPo) secondary cells, so this subsection will focus only on these electrochemical compositions, with some exceptions.

3.4.1 Secondary Li-ion and Lipo Batteries

Typically, Li-ion cells deliver an average voltage of 3.6 V, while the highest specific energy obtained is well over 150 Wh kg^{-1} (37). Unlike electronics, battery cells do not typically show significant damage or capacity losses due to radiation. However, in an experiment done by JPL, some capacity loss is seen among these latest lithium-ion battery cells under a high dosage of Cobalt-60. The results are shown below in Figure 3.6 (49).

In Lithium-ion batteries, repeated charging cycles of the battery eventually result in aging or degradation that affects the overall energy (Watt-hours) that the battery may provide. Many variables impact aging, such as temperature, charge/discharge rate, depth of discharge, storage conditions, etc. Due to the numerous variables that impact aging, lithium-ion batteries are typically put under life test in mission conditions before launch to ensure the battery will meet the specific mission life requirements.
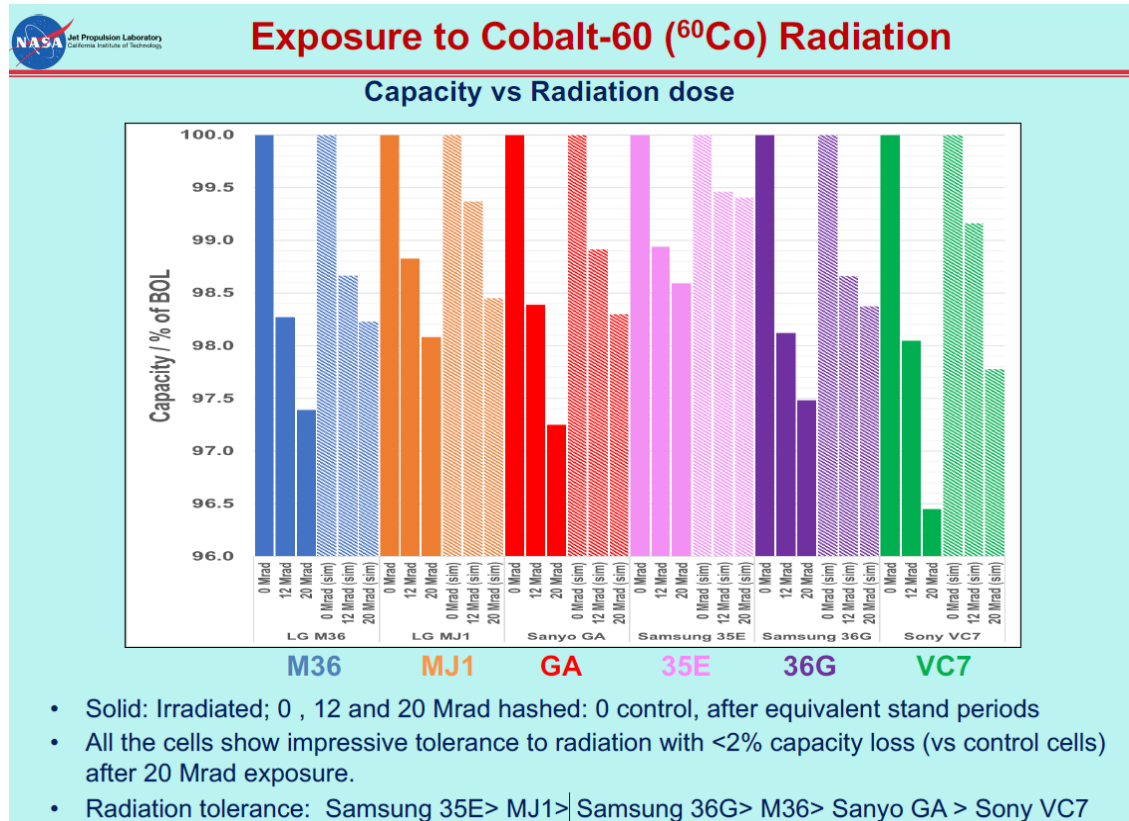
Figure 3.6: Capacity vs. radiation dose. Credit: JPL.

18650 Cells

18650 cylindrical cells (18 x 65 mm) have been an industry standard for lithium-ion battery cells. Many manufacturers have staple high-performance 18650 cells, some of which have flown on multiple spacecraft and are documented in Table 3-4 below.

Table 3-4: 18650 Cylindrical Cells		
Cell	**Specific Energy (Wh kg^{-1})**	**Flight Heritage**
LG ICR18650 B3 (2600 mAh)	191	NASA's PhoneSat, NoDES
Panasonic NCR18650B (3350 mAh)	243	MarCO, ADAPT (Sept 2022*: BioSentinel, Lunar Flashlight, NeaScout)
Molicel ICR18650H (2200 mAh)	182	NASA's EDSN mission
Canon BP-930s (3000 mAh)	112	NASA's TechEdSat missions
LG MJ1 (3500 mAh)	260	NASA's PACE mission

Cylindrical 18650s have become the most commonly used building blocks for many SmallSats, although prismatic and pouch formats are also available. The lithium-ion industry has seen incremental increases in energy density via the inclusion of silicon in the anode, high voltage cathodes, new electrolyte additives, and improved cell designs.

21700 Cells

21700 (21 x 70 mm) is another type of cylindrical cell that is getting more popular in the automotive industry. Samsung 50E and LG M50 both offer 5000 mAh of energy while the Samsung cells are slightly heavier. The specific energy densities are 262 Wh kg^{-1} and 264 Wh kg^{-1} respectively. Although 21700 cells are slightly larger than 18650 cells, they have some of the highest energy densities and could offer some mechanical packaging benefits with fewer cells for certain missions. Figure 3.7 shows various 21700 battery cell specific densities.
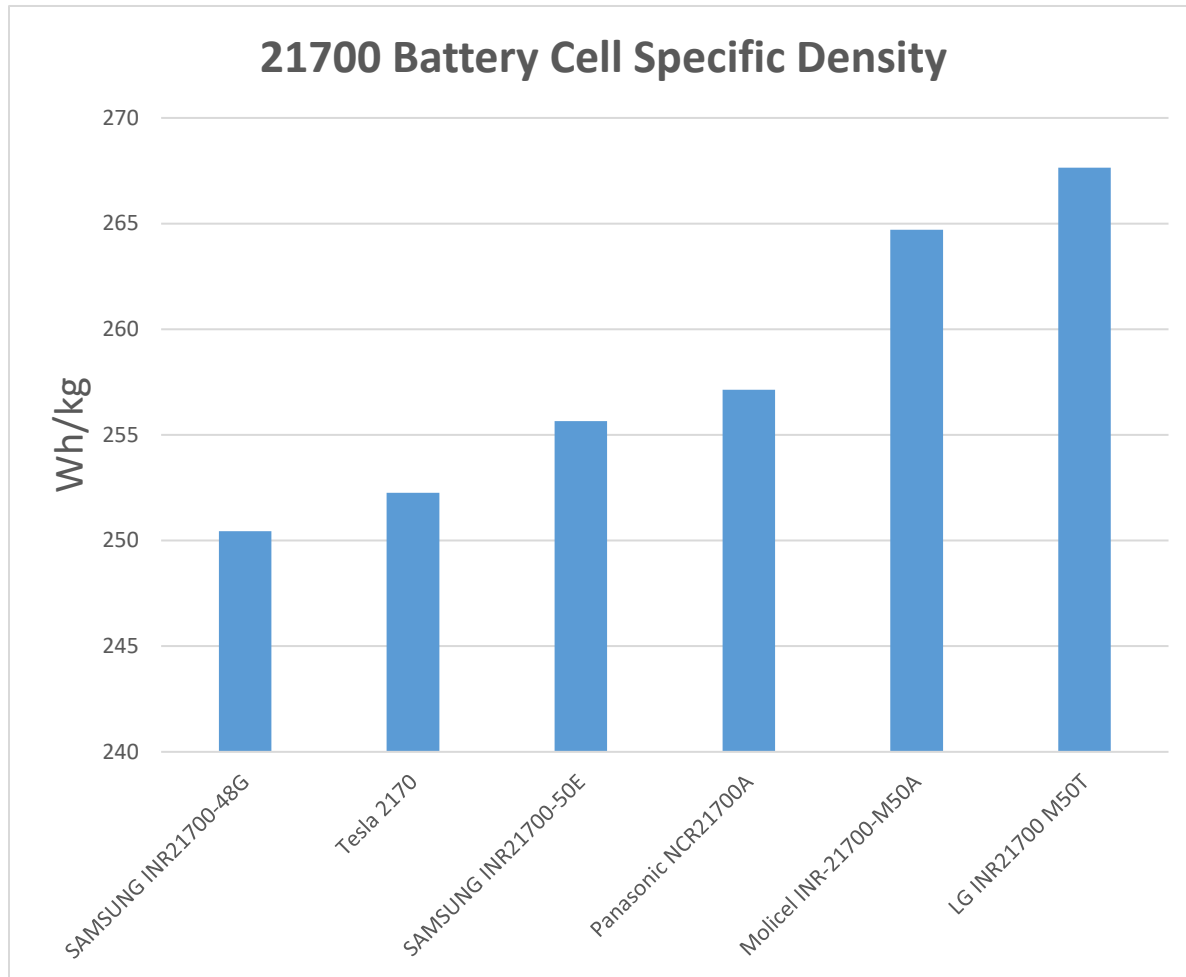
Figure 3.7: 21700 battery cell specific density. Credit: NASA.

4680 Cells

4680 (46 x 80 mm) cylindrical cells are a battery cell form factor that has been introduced to the energy storage scene by Tesla. The larger format cell potentially exacerbates several of the thermal management drawbacks (particularly internal temperature gradients and heterogeneity in current distribution) associated with other common smaller cells, however, to address these drawbacks, Tesla has a "tab-less current collection" method where the current collector foil is used in conjunction with an array of current collectors to reduce ohmic losses and the temperature increases that those losses can cause (50).

When it comes to the manufacturing of Li-ion batteries and battery cells, the companies listed in Table 3-5 are at the forefront of their respective sectors. Although China has some of the largest consumer electronics and EV battery manufacturers, their products are not known for high performance and high reliability and are rarely used in the space industry. Therefore, we do not currently include these products in this report.

Table 3-5: Commercial and Space Li-ion Manufacturers			
Commercial Li-ion Manufacturing		Space Li-ion Manufacturing	
Company	Headquarters	Company	Headquarters
Panasonic	Japan	EaglePicher Technologies	USA
LG Chem	South Korea	Enersys	USA
Samsung	South Korea	GS Yuasa	Japan
E-one Moli	Taiwan	Saft	France
Sony	Japan	Tesla	USA

3.5   On the Horizon – Energy Storage

In the area of power storage, there are several ongoing efforts to improve storage capability and relative power and energy densities; a Ragone Chart shown in Figure 3.8 illustrates different energy devices (51). For example, the Rochester Institute of Technology and NASA Glenn Research Center (GRC) developed a nanotechnology enabled power system on a CubeSat platform. The power system integrates carbon nanotubes into lithium-ion batteries that significantly increase available energy density. The energy density has exceeded 300 Wh kg^{-1} during testing, a roughly two-fold increase from the current state of the art. The results in this program were augmented from a separate high-altitude balloon launch in July 2018 organized through NASA GRC, which showed typical charge and discharge behavior on the ascent up to an altitude of 19 km (52). A collaborative project between the University of Miami and NASA Kennedy Space Center (KSC) is aiming to develop a multifunctional structural battery system that uses an electrolytic carbon fiber material that acts as both a load-bearing structure and a battery system. This novel battery system will extend mission life, support larger payloads, and significantly reduce mass. While several panel prototypes have shown successively increased electrochemical performance, further testing of the individual components can improve the accuracy of the computational models (53).
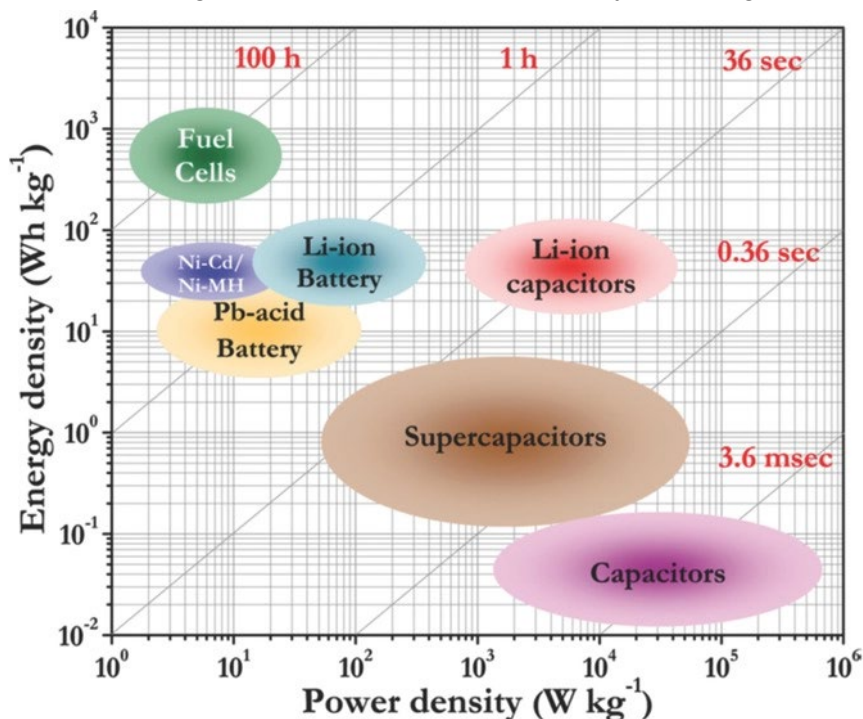

Figure 3.8 Relative power and energy densities of different energy devices. Ragone chart illustration reprinted with permission from Aravindan et al. Copyright (2014) American Chemical Society.

3.5.1 Supercapacitors

While the energy density for supercapacitors, also called ultracapacitors, is low (up to 7 Wh kg^{-1}), they offer a very high-power density (up to 100 kW kg^{-1}), which could be useful for space applications that require power transients. Their fast charge and discharge time, their ability to withstand millions of charge/discharge cycles and a wide range of operational temperatures (-40°C to +70°C), make them a perfect candidate for several space applications (launchers and satellites). This was demonstrated in an ESA Study entitled "High Power Battery Supercapacitor Study" completed in 2010 by Airbus D&S (54). The Nesscap 10F component and a bank of supercapacitors based on the Nesscap 10F component were space-qualified in 2020 after the completion of the ESA Study entitled "Generic Space Qualification of 10F Nesscap Supercapacitors." Although not likely to replace Li-ion batteries completely, supercapacitors could drastically minimize the need for a battery and help reduce weight while improving performance in some applications. Figure 3.9 shows a comparison chart (55), and Table 3-6 lists differences in Li-ion batteries and supercapacitors (56).
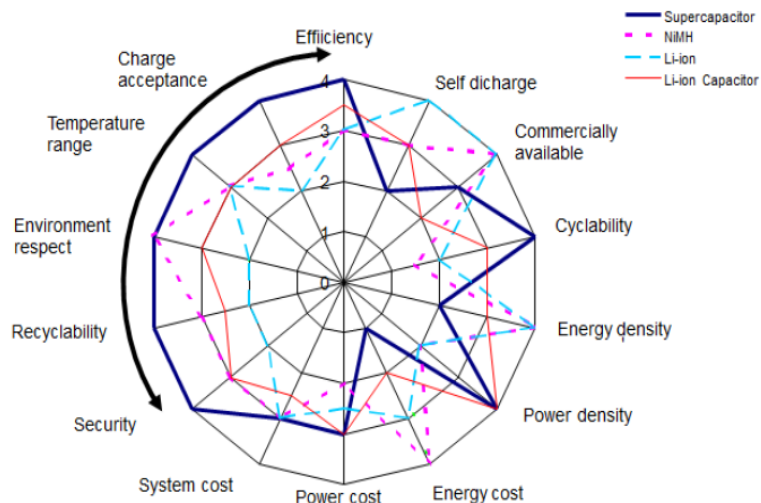
Figure 3.9: Supercapacitor comparison chart. Credit: Airbus Defense and Space and ESA (2016).

Table 3-6: Battery-vs-Supercapacitor Specifications		
Feature	Li-Ion Battery	Supercapacitor
Gravimetric energy (Wh kg^{-1})	100 – 265	4 – 10
Volumetric energy (Wh L^{-1})	220 – 400	4 – 14
Power density (W kg^{-1})	1,500	3,000 – 40,000
Voltage of a cell (V)	3.6	2.7 – 3
ESR (mΩ)	500	40 - 300
Efficiency (%)	75 – 90	98
Cyclability (nb charges)	500 – 1,000	500,000 – 20, 000,000
Life (years)	5 – 10	10 – 15
Self-discharge (% per month)	2	40 – 50 (descending)
Charge temperature	0 to 45°C	-40 to 65°C
Discharge temperature	-20C~60°C	-40 to 65°C
Deep discharge pb	yes	no
Overload pb	yes	no
Risk of explosion	yes	no
Charging 1 cell	complex	easy
Charging cells in series	complex	complex
Voltage on discharge	stable	decreasing
cost ($) per kW h	235 – 1,179	11,792

The lithium-ion capacitor is a promising recent development in the world of energy storage, combining the energy storage capabilities of both lithium-ion batteries as well as double-layered capacitors; they provide a middle ground between power density and energy density, but suffer from limited life-cycles. Some lithium-ion capacitors have minimum specific energy of 200 Wh kg^{-1} but are limited by a maximum specific power of <350 W kg^{-1} (58).

3.5.2   Solid-State Batteries

Most of the batteries used in contemporary space applications are lithium-ion batteries that use liquid electrolytes. However, these batteries carry an inherent risk of combustion from physical damage as well as thermal runaway due to overcharge. As a result, spacecraft often carry parasitic weight in the form of cooling systems and housing units. There is a long-standing need for battery designs that improve on energy and power density, as well as safety. NASA's Solid-state Architecture Batteries for Enhanced Rechargeability and Safety (SABER) project aims to create solid-state batteries that have significantly higher energy than the current state-of-the-art lithium-ion batteries and do not catch fire or lose capacity over time. Current strides in this project include examination and testing on unique battery chemistries including sulfur-selenium and "holey graphene" (57). See Table 3-7 for examples of solid-state batteries.

Table 3-7: Solid-State Batteries			
Manufacturer	**Product**	**Wh/kg**	**Wh/L**
Solid Power	Silicon EV Cell	390	930
Solid Power	Lithium Metal	440	930
Solid Power	Conversion Reaction Cell	560	785
QuantumScape	LFP (projected)	230	600
QuantumScape	NMC (projected)	300	1000

3.5.3   Batteries for Low-Temperature Applications

Typical Li-Ion batteries have an operating temperature range of -20°C to 60°C (3). This may not meet the requirements for missions that require lower operating temperatures. See Table 3-8 for batteries with low-temperature applications.

Table 3-8: Batteries for Low-Temperature Applications			
Company/Chemistry	**Package**	**Temperature**	**Specific Energy**
EEMB/Li-Ion (93)	Custom	-40°C ~ 60°C	193.5 (Wh kg^{-1})
Tadiran/LiSOCl$_2$	Custom	-80°C ~ 125°C -40°C ~ 85°C	1420 Wh/l 1420 Wh/l
GREPOW/LiPo	Custom/Pouch	-40°C ~ 60°C -55°C ~ 50°C	N/A
GREPOW/Li-Ion (LiFePO$_4$)	Custom/Pouch	-40°C ~ 50°C	N/A

3.6   State-of-the-Art – Power Management and Distribution

Power management and distribution (PMAD) systems control the flow of power to spacecraft subsystems and instruments and are often custom designed by mission engineers for specific spacecraft power requirements. However, several manufacturers have begun to provide a variety of PMAD devices for inclusion in small spacecraft missions. PMAD not only delivers power coming from energy sources (typically solar arrays in SmallSat applications) but also conditions energy as well, mitigating harmful transient disturbances and fault conditions from propagating downstream and hurting connected loads.

Several manufacturers supply EPS which typically have a main battery bus voltage of 8.2 V but can distribute a regulated 5.0 V and 3.3 V to various subsystems. The EPS also protects the electronics and batteries from off-nominal current and voltage conditions. As the community settles on standard bus voltages, PMAD standardization may follow. Well-known producers of PMAD systems that focus on the small spacecraft market include Pumpkin, GomSpace, Stras Space, and AAC Clyde Space. However, several new producers have begun to enter the PMAD market with a variety of products, some of which are listed below. Table 3-9 lists PMAD system manufacturers; it should be noted that this list is not exhaustive.

Key considerations in determining PMAD device selection often include conversion efficiency, input/output voltage range, output power capabilities, and size, weight, and power (SWaP). These metrics are critical to consider for good SmallSat PMAD designs, but it is important to note that PMAD devices are best chosen to suit the exact application of the SmallSat mission. SmallSat missions are often short and more flexible in terms of risk management than larger satellites, and therefore lend themselves to greater flexibility in design choices. One must leverage the benefits and risks to the mission at hand when choosing COTS PMAD systems, which may include the following:

- COTS PMAD may require less intensive integration and testing but have drawbacks to be addressed in a custom PMAD build
- Unnecessary features and peripherals (e.g., excess switching, fusing, current capability) can greatly increase SWaP metrics on a SmallSat
- Variability in designs of COTS PMAD devices means that important features and protections are not available in all devices (MPPT, Dead-bus protections, redundancy mechanisms, etc.)

Due to the variability of COTS PMAD options, many choice considerations, from internal power management topologies/materials to telemetry and protection options, are either included or omitted from products depending on the manufacturer. Internal power regulation topologies have traditionally been silicon-based, but relatively recent research into the performance improvements of Gallium Nitride (GaN) topologies has increased the number of GaN-based PMAD options in the consumer market with the following benefits over their silicon counterparts:

- Ability to achieve high switching rates and lower switching losses, allowing for the downsizing of inductors and capacitors, and improving SWaP metrics
- Lack of gate oxide layer in GaN-based field-effect transistors yields improvements in overall efficiency

It must also be noted that GaN-based PMAD options are not to be considered as drop-in replacements for silicon-based PMAD options. Despite the number of performance improvements, GaN architectures come with a variety of drawbacks including high complexity of control circuitry and lack of flight heritage.

In looking at the table below, one must note that there is no single COTS PMAD solution that can fit all needs of a mission at hand. In appealing to a broad range of applications, most COTS PMAD devices make sacrifices that can impact important metrics for SmallSats, including SWaP as well as the efficiency and quality of the power being managed. In choosing to use COTS PMAD devices, designers and system architects should be aware of, and try to minimize, unnecessary features not beneficial to the mission.

Table 3-9: Power Management and Distribution System Products								
Company	Product	Mass (kg)	Volume (cm³)	Peak Power Output (W)	Input Voltages (VDC)	Output Voltages (VDC)	Max Efficiency (%)	Ref
AAC Clyde Space [Sweden]	Starbuck Micro	2.45	3968	80.4	28	28 / 5	97	(59)
	Starbuck Mini	5.9	13133	1200	*	22-34 / 5 / 8/ 12 / 15	*	(60)
	Starbuck Nano	0.086	140	*	*	3.3 / 5/ 12	*	(61)
Aerospacelab [Belgium]	PCDU	1.635	1715.5	960	[0; 65] V	5V bus voltage [4.75, 5.25] V 12V bus voltage [11.4, 12.6]	>97	
Argotec [Italy]	PCDU VOLTA	0.97	600	100	18-22	1x 3.3 V, 1x 5 V, and 2x 12 V	75	(62)
	PCDU ZEUS	0.56	532	220 W (Solar + battery power)	8.5-24 V	3.3 V; 5 V; 12 V; 28 V (unregulated) (Config. at Manufacturing)	85	
	PCDU PHOENIX	1.5	1400	250+ W	31-55 V	3.3 V; 5 V; 12 V; 28 V (unregulated) (Config. at Manufacturing)	85	
Berlin Space Technologies [Germany]	PCU-110	1.08	1190	180	standard: 20-28 optional: 12, 15, 18	standard: 2x 12, 12x 5, 5x 3.8 optional: 1.8 – 28	*	(63)
Bradford Space [Luxembourg]	SuperNova modular PCDU	2.9	3045	1500	22 - 34	3.3 / 5 / 12 / unreg. batt	95	
C3S Electronics Development LLC [Budapest]	EPS1000	~0.860	~731	90	6…25V 6ch SA	3.3V, 5V, 9.9…12.3V	90	(64)
	EPS2000	6.8	7000	950	6-60V	23-33V	97%	
DHV Technologies [Spain]	MicroEPS	0.285-1.135 (+0.170✳)	392-1045	592 in eclipse/ 693 in sunlight	10-40 (X/Y) / 9-28 Z	3.3 / 5/ 12 / Batt	93	
	NanoEPS	0.155-0.402 (+0.109✳)	283-600	59 in eclipse/ 124 in sunlight	9-28 (X/Y) / 3-18 (Z)	3.3 / 5/ 12 / Batt	93	
	PicoEPS	0.110-0.190 (+0.1✳)	140-197	29 in eclipse/ 74 in sunlight	3-18	3.3 / 5/ 12 / Batt	93	
Ecarver GmBH [Germany]	PCU-SB7	1.5	1800	250	0-24	0-24	85	(65)
EnduroSat [Bulgaria]	EPS I	0.208	183	20-10	0-5.5	3.3 / 5 / Batt	86	(66)
	EPS I Plus	0.292	259	30	0-5.5	3.3 / 5 / Batt	86	(67)

Table 3-9: Power Management and Distribution System Products								
Company	Product	Mass (kg)	Volume (cm³)	Peak Power Output (W)	Input Voltages (VDC)	Output Voltages (VDC)	Max Efficiency (%)	Ref
	EPS II	1.28	742	250	10-36	3.3 / 5 / 6-12 / Batt	89	(68)
	EPS III	1-6 kg	860-4000	500	0-45	3.3 / 5 / 12 / Batt	87	(69)
	EPS IV	12	12784	7200	0 -70	12 / 28 / Batt	TBD	
GomSpace Denmark	P31U	0.1	127	30	0-8	3.3 / 5	96	(70)
	P60	**	**	100	16/32 V	3.3/5/8/12/18/ 24	-	
	P80	360-610g ¥	350–586	300	0-25	3.3/5/12/18V & Vbat	-	
GUMUSH AeroSpace Istanbul	n-ART EPS	0.098	160	100	4.5-30	3.3 / 5 / 8-36 / Batt	94	
Ibeos USA	150 W CubeSat EPS	0.14	124	150	18-42	3.3 / 5 / 12 / Unreg Batt	95	(71)
Ibeos USA	200 W CubeSat EPS	0.14	124	200	12-34	3.3 / 5 / 12/ Unreg Batt	96	
	Modular EPS	Starting at <1	Starting at 1150	500 – 2,000	12-26	5 / 12 / Unreg Batt	98	
ISISPACE The Netherlands	ICEPS2 Type A	0.184¥	223	39.2	--	3.3 / 5 / Unreg	96	(72)
	ICEPS2 Type B	0.31	275	78.4	--	3.3 / 5 / Unreg	95	
	ICEPS2 Type C	0.36	317	78.4	--	3.3 / 5 / 28.2 / Unreg	95	
Nanoavionics Lithuania	CubeSat EPS (EPS 1.0)	*	*	175	2.6-12	3.3 / 5 / 3-12	96	(73)
NPC SPACEMIND Italy	GEMINI	0.3	215	200	10 to 24	1 or 2 X 3.3 1 or 2X 5.0 2X 12V Regulated up to 24V	88	
Pumpkin Space Systems USA	EPSM 1	0.3	180	300	4-32	3.3-28	99	(74)
	AMPS	1.3	360	1200	5-32	--	99	
SkyLabs Slovenia	SKY-NANOeps-PCDU-23c-5d	0.2	216.125	72	10V Unreg	3.3, 5, 12, 10 Unreg	99	(75)(76)
Spacemanic Czech Republic	NANOeps-BMM-172wh	1.933 g	1668.485	540	Up to 12V	3.3, 5V, 12V, 10V Unreg	99%	

* Available with inquiry to manufacturer
** Configuration dependent
† Standard Configuration
‡ Optional radiation shielding case
¥ Flexible stacking options Standard options

3.7   On the Horizon – Power Management and Distribution

Power management and distribution have been steadily improving each year due to changes in technology, as well as from different approaches to maximizing the use of these systems, including modular architectures, wireless telemetry, and power transmission options.

3.7.1   Modular Architecture

For small spacecraft, traditional EPS architecture is centralized (each subsystem is connected to a single circuit board). This approach provides simplicity, volume efficiency, and inexpensive component cost. However, a centralized EPS is rarely reused for a new mission, as most of the subsystems need to be altered based on new mission requirements. A modular, scalable EPS for small spacecraft was detailed by Timothy Lim and colleagues, where the distributed power system is separated into three modules: solar, battery, and payload. This allows scalability and reusability from the distributed bus, which provides the required energy to the (interfaced) subsystem (77).

ISISPACE has a modular EPS for CubeSat missions (3U+) that includes a large amount of flexibility in output bus options with adjustable redundancy for certain parts of the device. The modular EPS consists of a power conditioning unit for solar panel input, secondary power storage, a battery holder with an integrated fuse, and a power regulation and distribution unit for subsystem loads. Each unit is designed to be independent, allowing for daisy-chaining and flexibility in redundancy and subsystem upgrades. This device is based on heritage from the Piezoelectric Assisted Smart Satellite Structure (PEASSS) CubeSat flown in 2017, with the device itself successfully flown in 2018 (78).

3.7.2   Wireless Power Transfer and Telemetry

In the commercial world, the technology already exists for wireless sensing and power transmission from the order of microwatts, all the way up to kilowatts. In the realm of SmallSats, wireless power transfer/detection would be useful as redundant options in dusty environments where physical connectors can be contaminated, or in situations where hardware needs to be swapped around and powered (battery swaps). While wireless power transfer/detection is highly inefficient when compared to conventional means, research and development in this technology for use in space applications has a lot of potential to increase the reliability and robustness of SmallSat power management and distribution.

3.8   Summary

Driven by weight and mostly size limitations, small spacecraft are using advanced power generation and storage technology such as >32% efficient solar cells and lithium-ion batteries. The higher risk tolerance of the small spacecraft community has allowed both the early adoption of technologies like flat lithium-polymer cells, as well as COTS products not specifically designed for spaceflight. This can dramatically reduce cost and increase mission-design flexibility. In this way, power subsystems are benefiting from the current trend of miniaturization in the commercial electronics market, as well as from improvements in photovoltaic and battery technology.

Despite these developments, the small spacecraft community has been unable to use other, more complex technologies. This is largely because the small spacecraft market is not yet large enough to encourage the research and development of technologies like miniaturized nuclear energy sources. Small spacecraft power subsystems would also benefit from greater availability of flexible, standardized power management and distribution systems so that every mission need not be designed from scratch. In short, today's power systems engineers are eagerly adopting certain innovative Earth-based technology (like lithium polymer batteries) while, at the same time,

patiently waiting for important heritage space technology (like fuel cells and RTGs) to be adapted and miniaturized. Despite the physical limitations and technical challenges these power generation technologies have, most small nanosatellites in the foreseeable future will still likely carry batteries to support transient loads.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email so someone may contact you further.

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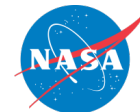(15)   DHV Technology. "Products - Solar Panels for CubeSats." [Online] 2020. Available at: https://dhvtechnology.com/products

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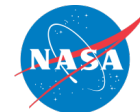
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Chapter Contents

Chapter Glossary

(ABS)	Acrylonitrile Butadiene Styrene
(AC)	Alternating Current
(ACE)	Apollo Constellation Engine
(ACO)	Announcement for Collaborative Opportunity
(ACS)	Attitude Control System
(ADN)	Ammonium Dinitramide
(AFRL)	Air Force Research Laboratory
(AIS)	Applied Ion Systems
(AOCS)	Attitude and Orbit Control System
(AR)	Aerojet Rocketdyne
(ARC)	Ames Research Center
(BOL)	Beginning of Life
(CHIPS)	CubeSat High Impulse Propulsion System
(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
(EOL)	End of Life
(EMC)	Electromagnetic Compatibility
(EMI)	Electromagnetic Interference
(EP)	Electric Propulsion
(EPL)	Electric Propulsion Laboratory, Inc.
(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
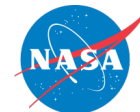
(GEO)	Geostationary Equatorial Orbit
(GIT)	Gridded-ion Thrusters
(GOCE)	Gravity Field and Steady-State Ocean Circulation Explorer
(GOX)	Gaseous Oxygen
(GPIM)	Green Propellant Infusion Mission
(GPS)	Global Positioning System
(GRC)	Glenn Research Center
(GSFC)	Goddard Space Flight Center
(HAN)	Hydroxylammonium Nitrate
(HET)	Hall-effect Thruster
(HTP)	High Test Peroxide
(HTPB)	Hydroxyl-terminated Polybutadiene
(IPS)	Integrated Propulsion System
(ISS)	International Space Station
(JANNAF)	Joint Army Navy NASA Air Force
(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

(TBD)   To Be Determined

(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, although 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 using 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, combined 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 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 tabulates procurable devices of each technology. Some sections also include an incomplete list of notable advancements. While the key integration and operational considerations are not comprehensive, they provide initial insights that may influence propulsion system selection. In the cases where a device has a long history of spaceflight, 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 or less 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 presented 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 for 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). The guideline was recently updated in 2022 to reflect the latest community input (4). This guideline suggests an interpretation of TRL for micro-propulsion and reflects both NASA and DOD definitions for TRL. The JANNAF panel consists 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 receives feedback from the non-Government propulsion community. While this JANNAF guideline focuses on micro-propulsion (e.g., 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 may continue 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 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 yet feasible to accurately and consistently apply the TRL scale. This novel classification system is discussed in detail below.

Readers 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 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 concept development. Once a handful of devices are identified as possible solutions for a specific mission concept, a detailed TMA and rigorous TRL assessment should be conducted. The PMI classification system sort 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 final 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 consistent with the DRM. 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 (e.g., 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. Furthermore, propulsion systems are often highly throttleable, and the devices surveyed herein may in many cases be capable of offering performance beyond the ranges presented in table 4-1.

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 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, tethers, electric sails (and plasma brakes), 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 future applications.

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 wet mass 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 – 28 N	180 – 285
Alternative Mono- and Bipropellants	50 mN – 22 N	150 – 310
Hybrids	8 – 222 N	215 – 300
Cold Gas	10 µN – 3.6 N	40 – 110
Solid Motors	37 – 461 N	187 – 269
Propellant Management Devices	-	-
4.6.2 ELECTRIC PROPULSION TECHNOLOGIES		
Electrothermal	0.1 mN – 1 N	20 – 350
Electrosprays	20 µN – 20 mN	225 – 3,000
Gridded Ion	0.1 – 20 mN	500 – 3,000
Hall-Effect	0.25 – 55 mN	200 – 1,920
Pulsed Plasma and Vacuum Arc Thrusters	4 – 500 µN	87 – 3,200
Ambipolar	0.5 – 17 mN	400 – 1,100
4.6.3 PROPELLANTLESS PROPULSION TECHNOLOGIES		
Solar Sails	TBD	-
Tethers	TBD	-
Electric Sails	TBD	-
Aerodynamic Drag	TBD	-
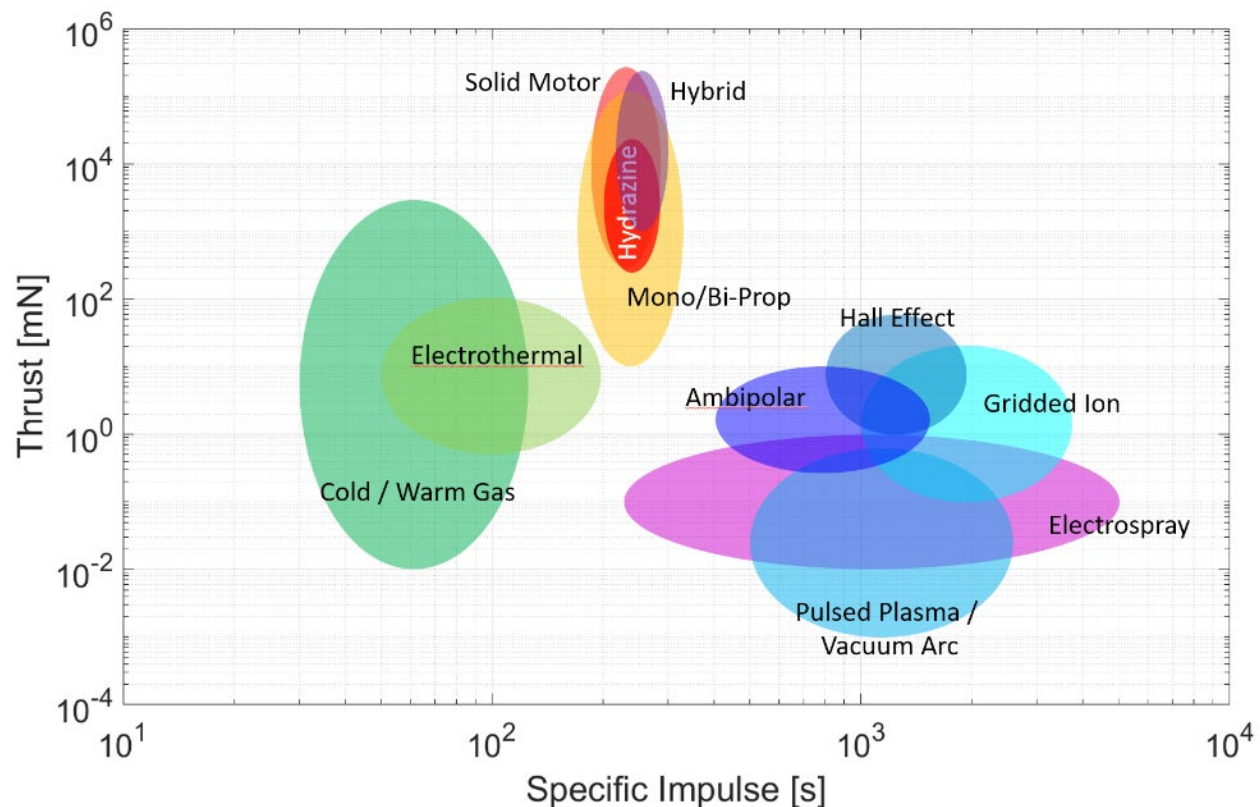
Figure 4.2: Typical small spacecraft in-space propulsion trade space (thrust vs. specific impulse). Credit: NASA.

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 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 heritage hydrazine propulsion systems makes them suitable for small spacecraft buses, and some hydrazine thrusters used on large spacecraft may be appropriate as the main propulsion system for small spacecraft. Hydrazine thrusters typically achieve a specific impulse between 200 – 235 seconds for 1-N class or larger thrusters.

b. Key Integration and Operational Considerations

- **Extensive Flight Heritage:** Since hydrazine has been used extensively in spaceflight applications, the technology's traits are well understood (5).
- **Extensive Component Ecosystem:** A robust ecosystem of components and experience exists because hydrazine systems are widely used. As such, hydrazine propulsion

systems are frequently customized for specific applications using the available components.

- **Qualified for Multiple Cold Restarts:** These systems have the advantage of typically being qualified for multiple cold starts.
- **Extensive Safety and Handling Requirements:** Hydrazine and its derivatives are corrosive, toxic, and potentially carcinogenic. Its vapor requires the use of Self Contained Atmospheric Protective Ensemble (SCAPE) suits. This overhead must be considered when planning ground processing workflow for spacecraft and may impose undesirable constraints on the spacecraft, the launch provider, or other spacecraft participating in the same launch opportunity. Hydrazine propulsion systems typically incorporate redundant serial valves to prevent spills or leaking vapor, which might harm ground personnel or hardware.

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 (6) (7). The company also offers a 20-N hydrazine thruster with extensive flight history, although not with small spacecraft.

Aerojet-Rocketdyne (L3 Harris company) has several small hydrazine thrusters with extensive flight histories. The 0.09-N MR-401, 1-N MR-103 series, 4-N MR-111G, and the 22-N MR-106L hydrazine thrusters are flight proven (8) (9) (10). In addition to four MR-107S 222-N thrusters, the OSIRIS-Rex spacecraft propulsion system has six MR-106L, sixteen MR-111G, and two MR-401 thrusters (11).

Moog has the 1-N MONARC-1, 5-N MONARC-5, and 22-N MONARC-22 series hydrazine thrusters (12). These thrusters have decades-long flight histories, although primarily as attitude control thrusters in larger satellites. NASA JPL's Soil Moisture Active/Passive (SMAP) spacecraft (a small satellite) used eight MONARC-5 thrusters (13).

Northrop Gruman has a line of hydrazine thrusters with long flight histories as attitude control thrusters for larger satellites. The thrust lines include the 1-N MRE-0.1, 5-N MRE-1.0, and 18-N MRE 4.0 (14).

Rafael offers 1-N, 5-N, and 25-N flight proven hydrazine thrusters (15).

IHI Aerospace offers the 1-N MT-9, 4-N MT-8A, and 20-N MT-2 flight proven hydrazine thrusters (17).

Stellar Exploration supplied their Monopropellant CubeSat System for an EchoStar spacecraft and the CAPSTONE spacecraft. The EchoStar nanosatellite built by Tyvak has been successfully commissioned and placed in the altitude prescribed in EchoStar's license for its S-band frequency (18). The CAPSTONE spacecraft was similarly built by Tyvak and launched on a Rocket Lab Electron from New Zealand on 28 June 2022. The CAPSTONE spacecraft is a 12U, 25 kg CubeSat that will help reduce risk for future lunar spacecraft by validating navigation technologies and verifying the dynamics of a halo-shaped orbit as planned for Gateway, the Moon-orbiting outpost that is part of NASA's Artemis program. Although CAPSTONE experienced communication and propulsion challenges along the way, the spacecraft achieved the expected lunar orbit (19) (20) (21).

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, in work sponsored by NASA GSFC, is developing a hybridization of ionic liquid and conventional hydrazine constituents to form the green hydrazine propellant blend (GHPB). "Green hydrazine" would provide the low vapor toxicity and high-density specific impulse of ionic liquids while retaining the low combustion and preheat temperatures of conventional hydrazine. This approach would provide a direct drop-in replacement for hydrazine propulsion systems. 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 (22).

Alternative Monopropellants and Bipropellants

a. Technology Description

For the past several decades, Earth storable propellants (i.e., storable at room temperature) have dominated in-space chemical propulsion systems, hydrazine for monopropellants and nitrogen tetroxide with either monomethylhydrazine or hydrazine for bipropellants. These propellants have the advantages of being ambient-temperature liquids (thus not requiring extensive thermal management) and easily ignited (catalyst for hydrazine, hypergolic for bipropellants). However, due to the extensive handling and toxicity concerns of the conventional chemical propellants, more propulsion systems using alternate propellants have been developed and adopted. Often described as "green" propellants, these alternative propellants are not necessarily benign, but are not vapor hazards like the conventional Earth storables and thus do not require specialized suits and breathing apparatus for handling.

Alternative monopropellants based on ionic liquids have received extensive technology development in recent years and have been matured to flight status. Ionic liquid monopropellants are ionic salts dissolved in water and blended with a fuel. Catalysts react the aqueous ionic salt and fuel blend for combustion. Thus, these formulations are not true monopropellants, since they have fuel and oxidizer components, but functionally they are no different than monopropellants.

The two matured ionic liquid monopropellant blends are LMP-103S, which is based on ammonium dinitramide (ADN) or ASCENT (Advanced Spacecraft Energetic Non-Toxic), formally referred to as AF-M315E, which is based on Hydroxylammonium Nitrate (HAN). These monopropellants do not present a vapor hazard and can be handled with conventional personal protection equipment (gloves, face shield). Depending on the formulations, they also can offer higher specific impulse and higher density-specific impulse than monopropellant hydrazine. The challenges with ionic liquid monopropellants are that their catalysts require more preheating than hydrazine and their higher combustion temperatures require higher-temperature, with more expensive catalyst and chamber materials. The ionic liquid monopropellants are not drop-in replacements for hydrazine, although their propulsion system components are similar.

Hydrogen peroxide is a high-density liquid that can be catalytically decomposed exothermically like hydrazine. Hydrogen peroxide does not present a vapor hazard, although high concentrations (> 90%) can rapidly react with impurities in storage containers. Hydrogen peroxide was used extensively in the early days of space propulsion before losing favor to the higher performing hydrazine. It is now being examined again as a nontoxic alternative to hydrazine. Development efforts have focused on high-test peroxide (HTP) which have concentrations from 85 to 98%.

Green bipropellant combinations have also been developed, particularly for small satellite applications. Hydrogen peroxide and nitrous oxide have been explored as the oxidizer options with an alcohol as the fuel. A water electrolysis system breaks water down to gaseous hydrogen

and gaseous oxygen, which serve as propellants for the engine. An electrolysis system, using water as the source of propellant, could enable the use of in-situ resources.

b. Key Integration and Operational Considerations

- **Improved Hazard Safety Classifications:** Air Force Range Safety AFSPCMAN91-710 (23) 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) (24). 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 (24). 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.

- **Simplified Safety and Handling Requirements:** Fueling spacecraft with green propellants, generally permitted as a parallel operation, may require a smaller exclusionary zone, allowing for accelerated launch readiness operations (25). 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.

- **Immature Component Ecosystem:** While there are thrusters that are relatively mature (PMI E/F), incorporating them into integrated 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 focused 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 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 Considerations for Green Propellants:** 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

ECAPS offers a range of High Performance Green Propulsion (HPGP) thrusters, including the LMP-103S (figure 4.3), at 100-mN, 1-N, 5-N, and 22-N thrust levels. The 1-N HPGP thrusters were first demonstrated on orbit in the Prototype Research Instruments and Space Mission technology Advancement (PRISMA) mission completed in June 2011. PRISMA consisted of two small satellites, each carrying hydrazine and HPGP systems to show off a side-by-side comparison of their performance (26).

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

ECAPS developed an LMP-103S based propulsion system for a constellation of Earth observing satellites, called SkySat. SkySat has a propulsion system that uses four 1-N HPGP thrusters. Thirteen SkySat satellites with the ECAPS propulsion system have been launched and are fully operational (27).

ECAPS LMP-103S propulsion system is also used on the Autoscale demonstration of rendezvous technologies called ELSA-d, which launched in March 2021. ELSA-d has eight 1-N ECAPS HPGP thrusters to provide re-orbiting and de-orbiting capability. A system issue impacted three of eight ECAPS thrusters and an unresolved root cause resulted in the loss of a fourth thruster. Nevertheless, many mission goals were successfully accomplished, improving the providers readiness for offering a commercial deorbit service (28).

VACCO provided a hybrid Micro Propulsion System (MiPS) for the ArgoMoon 6U CubeSat built for the Italian Space Agency. The ArgoMoon MiPS, figure 4.4, has an ECAPS 100mN LMP-103S thruster with four VACCO 25 mN R134a cold gas thrusters. ArgoMoon was successfully deployed in the Artemis I mission in November 2022 (29) (30) (31).
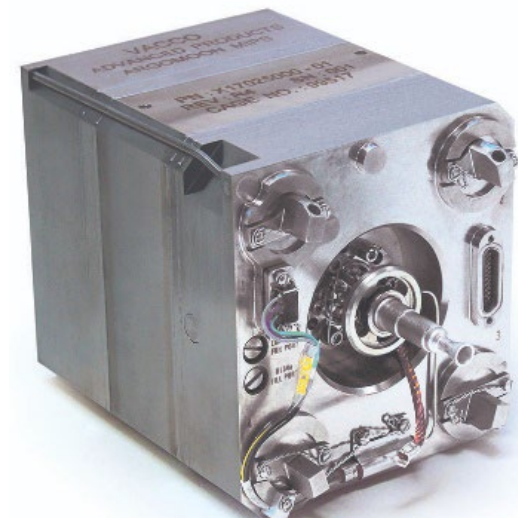
Figure 4.4: ArgoMoon hybrid Micro Propulsion System. Credit: VACCO.

Aerojet-Rocketdyne built the ASCENT-based propulsion system for the NASA Green Propellant Infusion Mission (GPIM). GPIM was an on-orbit demonstration of the ASCENT (then called AF-M315E) propulsion system, using five 1-N thrusters (figure 4.5) for small attitude control maneuvers (32). GPIM was integrated on a small spacecraft bus (Ball Aerospace's BCP-100) and launched in June 2019 as a secondary payload on a Falcon Heavy. The five thrusters were successfully fired in space, across a range of operating modes, testing their ability to control the satellite's attitude, and demonstrating their effectiveness at changing its orbital inclination (33).

Figure 4.5: GR1 thruster. Credit: Aerojet.

An ASCENT propulsion system, the Lunar Flashlight Propulsion System (LFPS) was developed for the JPL Lunar Flashlight mission (34). Lunar Flashlight (figure 4.6) was launched as a secondary payload in a December 2022 Falcon 9 launch, to map the lunar south pole for volatiles. LFPS was a pump-fed monopropellant system, using four 0.1-N ASCENT thrusters (figure 4.7) built by Rubicon Space Systems (a division of Plasma Processes) and a micro-pump built by Flight Works Inc. The propellant management system was fabricated using additive manufacturing. During the first few days of flight, it was found that 3 of the 4 thrusters were underperforming. Based on ground testing it was thought the underperformance might have been caused by obstructions in the fuel lines that limited propellant flow to the thrusters. Improvements were seen by increasing fuel pump pressure to clear the suspected obstructions. However, the effort was not sufficient to keep the spacecraft in the vicinity of the Moon and the mission was terminated in May 2023 (35).
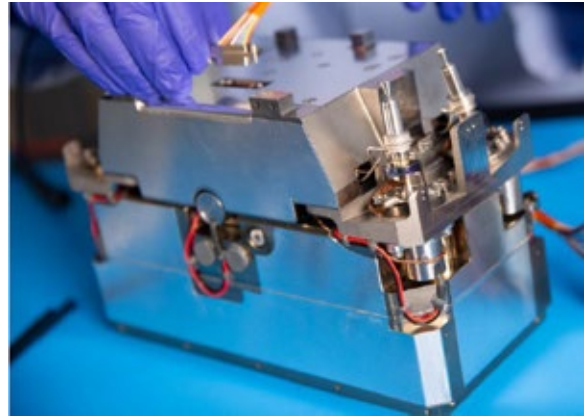
Figure 4.6: Lunar Flashlight Propulsion System. Credit: NASA.

Figure 4.7: Rubicon 0.1N ASCENT thruster. Credit: NASA MSFC.

Benchmark Space Systems offers thrusters that can be used with High Test Peroxide (HTP) as a monopropellant or as an oxidizer in a bipropellant combination with a fuel. Benchmark provided a bipropellant Halcyon Avant propulsion system that uses four 22-N thrusters using HTP and isopropyl alcohol for the Sherpa-LTC2 orbital transfer vehicle (OTV) (36). The Sherpa-LTC2 was launched on a Falcon 9 in September 2022. The OTV was used to elevate the orbit of the Varuna Technology Demonstration Mission satellite (37). In June 2021, three monopropellant HTP Halcyon systems launched aboard the SpaceX Transporter 2 rideshare mission. One of the systems debuted Orbit Fab's RAFTI refueling kit as part of their Tenzing mission. The other two missions supported an undisclosed mission partner (38).

CisLunar Explorer, part of a NASA Centennial Challenges program will use a water electrolysis propulsion system developed by Cornell University's Space Systems Design Studio on a pair of 3U CubeSat. In this system, the water is
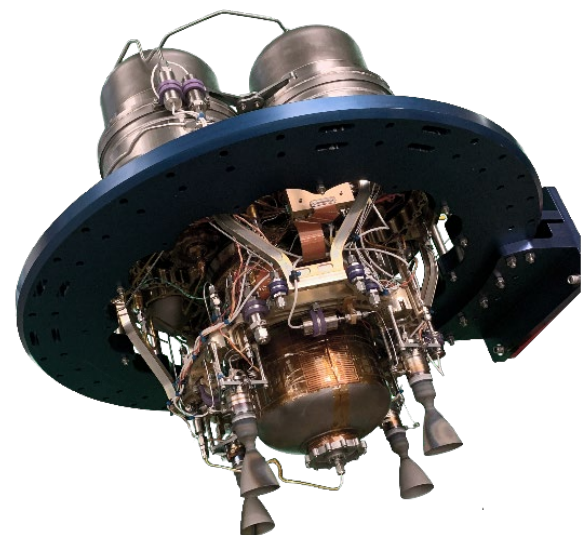
Figure 4.8: Halcyon Avant (Sherpa-LTC2 Configuration). Credit Benchmark Space Systems.

electrolyzed in the propellant tank. The gaseous hydrogen/gaseous oxygen mixture is flowed through a flame arrestor into the combustion chamber where it is ignited by a glow plug (39). The

CubeSat was originally intended to be launched on Artemis I but difficulties during integration bumped it from the mission (40) (41).

NASA's Small Spacecraft Technology (SST) program at Ames Research Center (ARC) launched the first Pathfinder Technology Demonstration (PTD) mission in January 2021 (42)(43)(44). PTD-1 (figure 4.9) 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 has been evaluated and made public (45). 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. Tethers Unlimited became a subsidiary of ARKA Group LP in 2020.
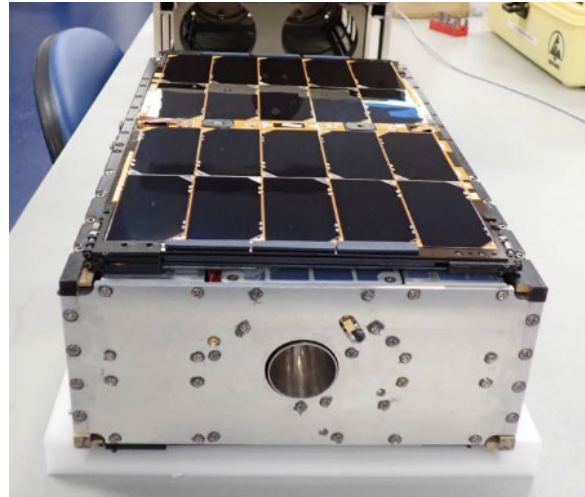
Figure 4.9: PTD-1 HYDROS-C. Credit: NASA.

NanoAvionics developed an ADN-based monopropellant propulsion system under the Enabling Propulsion System for Small Satellites (EPSS) program. The EPSS monopropellant system was demonstrated on LituanicaSAT-2, a 3U CubeSat, to correct orientation and attitude, avoid collisions, and extend orbital lifetime. 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 (46) (47).

   d. Summary Table of Devices

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

   e. Notable Advances

Aerojet Rocketdyne continues to develop 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 (48). 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 ASCENT propellant containing 10% added water. The heat transfer to surrounding spacecraft structure both during heat up and operation are comparable to conventional hydrazine thrusters.

Rubicon Space Systems (a division of Plasma Processes) is maturing 1N and 5N ASCENT thrusters (49), intended for SmallSat applications. Both offerings are built using the same materials and processes as those used on the 0.1-N thrusters delivered for the Lunar Flashlight Mission. Additionally, Rubicon Space Systems intends to engineer a short-life, lower cost version of the 5N thruster. The prototype thruster accumulated > 1-kg throughput and over 500 seconds before the end of the NASA Phase I SBIR. The Phase II effort will continue to develop the 5N thruster.

CU Aerospace LLC (CUA) has developed the Monopropellant Propulsion Unit for CubeSats (MPUC) system, figure 4.10 (50). The monopropellant is an H_2O_2-ethanol blend denoted as CMP-X. Tests on a thrust stand with a bladder-fed propellant tank have demonstrated continuous constant thrust for >50 minutes at a thrust level of 250 mN at I_{SP} of 179 s with an average input power of ~6 W during catalyst warmup. 1.5U and 2U system designs have an estimated 1400 N-s and 2120 N-s total impulse, respectively. A ~900°C flame temperature allows the thrust chamber to use non-refractory construction materials. CMP-X has low toxicity and was subjected to UN Series 1, 2, 3, and 6 testing; CMP-X demonstrated no detonation propagation when confined under a charge of high explosive, it exhibited thermal stability with no explosion or detonation during bonfire testing, and was not sensitive to drop impact or friction. CMP-X passed the criteria for either a 1.4S or a "Not Class 1" determination and may be excluded from the explosive class. Long-term storage testing shows no degradation over > 1200 days with testing ongoing. This flight-like, additively-manufactured, thruster passed environmental (vibration and thermal vacuum) testing and post-environmental thrust data were within experimental error of the pre-environmental data (51).

Dawn Aerospace has developed the CubeDrive bipropellant (nitrous oxide and propylene) system that consists of a single B1 thruster, propellant tank, valves, and electronics. The thruster can also be operated in cold gas mode (for smaller impulse bits) by not engaging the spark igniter. The 0.8U (figure 4.11) to 4U configurations provide 425 to 3,500 N-s of total impulse, respectively (52).

VACCO has integrated four ECAPS 1-N LMP-103S thrusters into their Integrated Propulsion System (IPS) (figure 4.12), a bolt-on propulsion module for delta-V and attitude control applications (53) (54).

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

Figure 4.10: MPUC System. Credit: CU Aerospace.

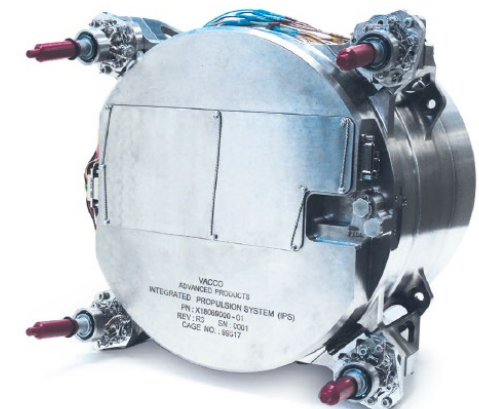Figure 4.11: CubeDrive. Credit: Dawn Aerospace.

Figure 4.12: VACCO Industries IPS. Credit: VACCO Industries.

the oxidizer is flowing, these systems can readily be started or shut down by controlling the oxidizer flow.

b. Key Integration and Operational Considerations

- **Improved Safety and Handling:** Hybrid systems are inherently safer to handle than solid motor systems because there is no oxidizer pre-mixed into the solid motor, which reduces the risk of pre-mature ignition.
- **Integrates Attributes of Solids and Liquids:** Hybrids achieve many positive attributes of both solid motors (storability & handling) and liquid engines (restart & throttling).
- **Combustion Efficiency:** Combustion efficiency tends to be lower than either solid motors or liquid engines.
- **Other Drawbacks:** Regression rate control and fuel residuals tend to be more problematic in hybrid designs.

c. Current & Planned Missions

An arc-ignition 'green' CubeSat hybrid thruster system prototype was developed at Utah State University and demonstrated in flight under the Undergraduate Student Instrument Project (USIP). The hybrid rocket design used a 3D printed acrylonitrile butadiene styrene (ABS) plastic as the fuel and high-pressure gaseous oxygen (GOX) as the oxidizer. For safety considerations the oxidizer was diluted to 60% nitrogen and 40% oxygen for the demonstration. 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 (55) (56). 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. Investigating different nozzle materials for low erosion in long duration burns is a key concern (57) (58).

JPL is developing a hybrid propulsion system for a 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 (59) (60) (61) (62).

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 (63) (64).

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 for enhanced performance. This design fits in a 1U space, for a 3 to 6U spacecraft (65).

Parabilis Space Technologies has developed 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 (66). 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 (67).

Cold Gas

a. Technology Description

Cold gas propulsion systems are simple, mature, and safe, although they provide relatively limited total impulse. There is no combustion in cold gas systems, with thrust produced by the expulsion of a gaseous propellant through a diverging nozzle. Propellants can be stored as compressed gases, saturated liquids, or solids. Gaseous nitrogen is a commonly used propellant for cold gas systems, although many other gases have been used in the long history of cold gas propulsion. For gases the tradeoff is between performance and storage; lower molecular weight gases offer higher specific impulse but require more voluminous storage. Saturated liquids are stored at low pressure and vaporized when flowed into a low-pressure chamber. The dense storage of saturated liquids and their property of self-pressurization has made them popular as a propellant for CubeSat missions. The liquid propellants may require a pressurant or are self-pressurized, if stored in a two-phase liquid-gas state. Solid propellants can be used in a cold gas system by subliming the solid propellant with adequate heat.

A derivative of cold gas systems is electrothermal or 'warm gas' systems, in which the propellant is somewhat heated without chemical reaction and accelerated through a nozzle. The additional heating results in a modest improvement in thrust and specific impulse compared to a pure cold gas system, but typically burdens the spacecraft with increased power consumption. Electrothermal systems are described in more detail in the Electric Propulsion section.

b. Key Integration and Operational Considerations

- **Low Cost and Complexity:** Cold gas thrusters are often attractive and suitable for small buses due to their relatively low cost and complexity.
- **Safe:** Most 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.
- **Small Impulse Bit:** Cold gas systems are often well suited to provide attitude control since they can provide very small minimum impulse bits for precise maneuvering.
- **Small Total Impulse:** The low specific impulse of these systems limits them from providing large orbital correction maneuvers.
- **Integrated Systems Optimized for CubeSats:** Designs optimized around a CubeSat's limited resources have improved the capability of these systems for nanosatellite buses.

c. Missions

The Micro-Electromechanical-based PICOSAT Satellite Inspector, or MEPSI, built by the Aerospace Corporation flew aboard STS-113 in 2002 and STS-116 in 2006. The spacecraft included both target and imaging/inspector vehicles connected via a tether. The two vehicles were each 4 x 4 x 5 in^3 in volume and had five cold-gas thrusters, producing approximately 20 mN. The

MEPSI propulsion system was produced using stereo-lithography. It was suited as a propulsion research unit for PicoSats (68).

Marotta developed a cold gas micro-thruster, CGMT-000-9, for fine attitude adjustment maneuvers that flew on the NASA ST-5 mission in 2006. The thruster operated in blowdown mode with gaseous nitrogen, starting at 2.4 N and ending at 0.05 N (69) (70).

In June 2014, Space Flight Laboratory at University of Toronto Institute for Aerospace Research (UTIAS) launched two 15 kg small spacecraft (CanX-4 and CanX-5) to demonstrate formation flying. The Canadian Nanosatellite Advanced Propulsion System (CNAPS), shown in figure 4.13, consisted of four thrusters fueled with liquid sulfur hexafluoride. This propulsion module is a novel version of the previous NanoPS that flew on the CanX-2 mission in 2008. The non-toxic sulfur hexafluoride propellant was selected because it has a high vapor pressure and density, which are important properties for making a compact self-pressurizing system. The propulsion system was successfully used for drift recovery and autonomously maintaining a formation from 1-km range down to 50-m separation (71) (72) (73) (74).
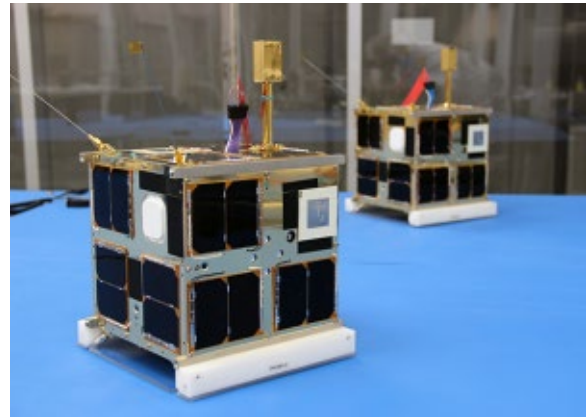
Figure 4.13: CanX-4 and CanX-5 formation flying nanosatellites with CNAPS propulsion systems. Credit: UTIAS SFL.

Microspace Rapid Pte Ltd of Singapore developed a cold gas propulsion system for the POPSAT-HIP1 CubeSat demonstration mission that launched June 2014. It consists of eight micro-nozzles that provide control for three rotational axes with a single thrust axis for translational applications. The total delta-v was estimated from laboratory data to between 2.25 and 3.05 ms^{-1}. Each thruster has 1 mN of nominal thrust using argon propellant. An electromagnetic microvalve with a very short opening time of 1 m-s operates each thruster (75).

GomSpace (acquired NanoSpace in 2016) has developed two related propulsion systems called the NanoProp CGP3 and NanoProp 6U. Both use proportional thrust control of four nozzles to control spacecraft attitude and provide delta-v. The CGP3 was flown on the TW-1 3U CubeSat launched in 2015. The 6U configuration was flown on GOMX-4B in 2018 as a formation flight demonstration (76) (77) (78) (79).

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 that launched on Artemis I in November 2022. The propulsion system enables detumbling and pointing for communication back to Earth. The propulsion system uses a 3D-printed propellant tank to reduce part count and make efficient use of the available volume. The system contains six RCS thrusters and one delta-v thruster. The delta-v thruster was included to allow for collision avoidance but was ultimately not needed. One of the RCS thruster valves failed closed during RCS checkout. Rather than further attempting to actuate the valve, and risk the valve failing open, a workaround was identified to perform momentum unloading with the remaining five RCS thrusters. As of May 2023, the propulsion system has accumulated 408 firings and continues to operate as expected. The initial phase of the mission was completed in April 2023. NASA has extended the mission by up to an additional 18 months, or as late as November 2024 (80) (81) (82) (83).

The ThrustMe I2T5 subliming iodine cold gas module, figure 4.14, was the first iodine propulsion system to be spaceflight tested, flown on the Xiaoxiang 1-08 satellite in 2019 (84) (85). Since then, the system was also launched on RTAF's NAPA-2 in June 2021 and on Spire's L3C in January 2023. Additional I2T5 are anticipated to launch on the Robusta-3A satellite, developed by CSUM, which will carry various scientific payloads related to meteorology and technology demonstration (86) (87).

The CubeSat Proximity Operations Demonstration (CPOD) is a mission led by Terra Orbital (88) to demonstrate autonomous on-orbit rendezvous and proximity operations using two identical 3U CubeSats. Each spacecraft incorporates a cold gas propulsion system built by VACCO Industries that provides up to 186 N-s of total impulse. This module uses the self-pressurizing refrigerant R236fa propellant, which is

Figure 4.14: I2T5 Iodine Cold Gas Module. Credit: ThrustMe.

exhausted through a total of eight thrusters distributed in pairs at the four corners of the module. CPOD launched in May 2022 as part of SpaceX's Transporter-5 mission. The CPOD mission demonstrated rendezvous of the CubeSats with intersatellite distances closing from 997 km to a minimum of 0.36 km. On-orbit limitations on experimentation were attributed to multiple factors as the launch date slipped due to many years of delay, including obsolete hardware, partial failure in the solar panels, and a propulsion system anomaly suspected to be a plenum leak (89) (90).

The Mars Cube One (MarCO) technology demonstration mission consisted of two identical CubeSats that followed the InSight spacecraft to Mars in loose formation in 2018. The MarCO spacecraft performed five trajectory correction maneuvers (TCMs) during the mission to Mars. The TCMs were conducted using an R236fa cold gas propulsion system developed by VACCO Industries, which contains four thrusters for attitude control and another four for the TCMs. MarCO-B developed a propulsion system leak which required constant maintenance. A tank to plenum leak was identified prior to launch, which the mission accepted. However, a second leak through a MarCO-B thruster valve developed during flight. The combination of both leaks resulted in a continuous moment on the MarCO-B spacecraft. The MarCO spacecraft succeeded in their mission to relay telemetry from the InSight lander during its descent to Mars (91) (92) (93) (94).

Near-Earth Asteroid Scout (NEA Scout) was a joint MSFC and JPL mission that had a VACCO cold gas MiPS (R236FA propellant), with six 25-mN thrusters, to assist the main propulsion system, a solar sail. However, after deployment from Artemis I in November 2022, the project team was not able to communicate with the spacecraft (95) (96) (97).

d. Summary Table of Devices

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

Solid Motors

a. Technology Description

Solid rocket motors have an oxidizer and fuel mechanical mixture stored in solid form (propellant grain). For small satellites, solid rocket motors may be used for impulsive maneuvers such as orbit insertion or quick de-orbiting. They achieve moderate specific impulses and high thrust magnitudes. There are some electrically controlled solid thrusters that operate in the milli-newton (mN) range that are restartable and have steering capabilities. Solid rocket arrays can be compact

and suitable for small buses. Composed of several miniature solid rockets, individual units can be fired, alone or together, as needed.

b. Key Integration and Operational Considerations

- **Thrust Vector Control:** Thrust vector control systems can be coupled with existing solid rocket motors to provide controllable high delta-v maneuvering.
- **Usually Single-Burn:** In general, solid motors are 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 considered safer than traditional solid energetic propellants.

c. Current & Planned Missions

Northrop Grumman offers the STAR 3 motor which has a maximum thrust of 2050 N. The STAR 3 motor was used as the Transverse Impulse Rocket System for the Mars Exploration Rover (MER) program, and was used in 2004 to reduce the lateral velocity of the MER Spirit Lander. The STAR 4G motor was developed and tested by NASA GSFC as an orbit adjust motor for deploying nanosatellite constellations but was not flown, although it is still offered by Northrop Grumman along wth other, higher total impulse motors in the STAR line (98).

The Pacific Scientific Energetic Materials Company (PacSci EMC) Modular Architecture Propulsion System (MAPS) array (figure 4.15) has a 10-plus year in-orbit lifespan. The MAPS system provides three axes capability to control 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 in 2017 aboard PacSciSat, which successfully completed all mission objectives (99) (100).
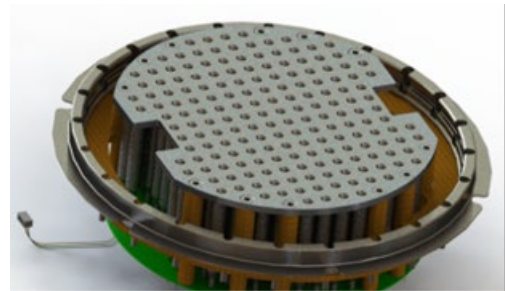
Figure 4.15: PacSci EMC MAPS sealed solid propellant rocket motor array. Credit: PacSci.

d. Summary Table of Devices

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

Propellant Management Devices

a. Technology Description

While not directly a thrust producing device, propellant management devices (PMDs) are frequently used in liquid propulsion systems to reliably deliver propellant to thruster units. PMDs are commonly a critical part of in-space liquid propulsion systems that do not use bellows or membrane type tanks. As small spacecraft look toward more complex propulsion system requirements, PMDs will undoubtedly play an integral role. 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 more conformal structures to be created and optimized for small spacecraft missions. As such, PMDs are briefly covered here for awareness. A more detailed treatment and explanation can be found in literature. A comprehensive, up-to-date list of the types of PMDs, as well as missions employing PMDs, is available in Hartwig (101).

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 storage 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.

c. Current & Planned Missions

The Lunar Flashlight Propulsion System used 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, were determined for the ASCENT propellant by Kent State University, funded and managed by NASA. The design and modelling was 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 the development of SmallSat and CubeSat scale diaphragm propellant tanks using materials that are compatible with hydrazine and some green monopropellant fuels (102), and demonstrated the utility of additive manufacturing in producing 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 such as nuclear reactors or beamed energy are conceivable. The energy conversion occurs by one of three mechanisms: electrothermal, electrostatic, or electromagnetic acceleration (133) (134). 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.

Instead of detailing the complete operating range for each propulsion device, the authors decided to provide only the metrics associated with the nominal operating condition to improve comprehension of the data and make initial device comparisons more straightforward. When a manufacturer does not specifically state a nominal operating condition in literature, the manufacturer may have been contacted to determine a preferred 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 the 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 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.

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 complex molecules may result 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 during spacecraft disposal. While most spacecraft materials burnup on re-entry, the re-entry behavior of devices using these high temperature materials will be scrutinized when assessing the danger of 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 direct current bus power to a high-frequency alternating current. 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 4.16) has been implemented by multiple customers operating in low-Earth orbit, including HawkEye 360, Capella Space, and BlackSky Global (135). All missions use the same Comet thruster head, while the BlackSky Global satellites use a larger tank to provide a greater total impulse capability. The HawkEye 360 Pathfinder mission is a constellation of three small microsatellites built by Space Flight Laboratory (SFL) based on its 15-kg NEMO platform; each spacecraft measures 20 x 20 x 44 cm^3 with a mass of 13.4 kg (136) (137). The Comet provides each HawkEye 360 pathfinder 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 (139).
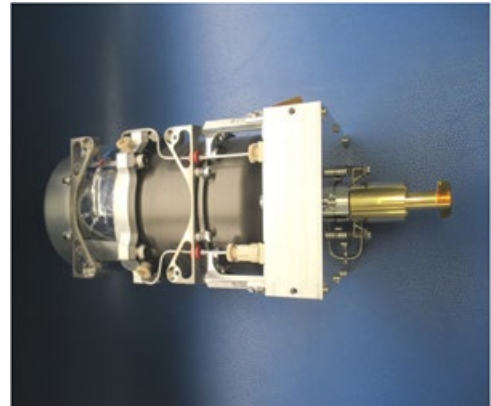
Figure 4.16: Comet-1000. Credit: Bradford Space.

The Propulsion Unit for CubeSats (PUC) system (140), figure 4.17, 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 (141). 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 (142). 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, although it remains unknown if any of the units have flown.

Figure 4.17: PUC module. Credit: CU Aerospace.

In 2019, CU Aerospace was selected for a NASA STMD Tipping Point award to design, fabricate, integrate, and perform mission operations for the Dual Propulsion Experiment (DUPLEX) 6U CubeSat. DUPLEX has two of CU Aerospace's micro-propulsion systems onboard, one Monofilament Vaporization Propulsion (MVP) system (143) (144) (145), figure 4.18, and one Fiber-Fed Pulsed Plasma Thruster (FPPT) system (149) (150) (151) (152) (153), figure 4.43. The MVP is an electrothermal device that vaporizes and heats an inert solid polymer propellant fiber to 725 K. The coiled solid filament approach for propellant storage and delivery addresses common

Figure 4.18: MVP module. Credit: CU Aerospace.

propellant safety concerns, which often limit the application of propulsion on low-cost CubeSats. In-orbit operations will demonstrate multiple mission capabilities including inclination change, orbit raising and lowering, drag makeup, and deorbit burns. Launch is manifested in early-2024 (155).

AuroraSat-1 is a technology demonstration 1.5U CubeSat that is demonstrating multiple propulsion devices by Aurora Propulsion Technologies. AuroraSat-1 carries Aurora's smallest version of Aurora Resistojet Module for Attitude control (ARM-A) (156), figure 4.19, and a demonstration unit of their Plasma Brake Module (PBM) (157). The ARM-A system integrated into AuroraSat-1 has six resistojet thrusters for full 3-axis attitude control and 70 grams of water propellant, providing a total impulse of 70

Figure 4.19: Aurora Resistojet Module for Attitude Control. Credit: Aurora Propulsion Technologies.

N-s. AuroraSat-1 is built by SatRevolution with Aurora providing the payloads. The satellite was launched by Rocket Lab in May 2022. (158) (159). See section 4.6.3 for discussion of the PBM module.

The OPTIMAL-1 technology demonstration 3U nanosatellite by ArkEdge Space Inc. launched in late 2022 and was deployed from the International Space Station in early 2023. Among other technology demonstration devices, OPTIMAL-1 contains a Pale Blue water resistojet thruster (160).

SPHERE-1 EYE, a 6U CubeSat developed by Sony Group Corporation, includes a Pale Blue Water Resistojet Thruster. The satellite was launched aboard a SpaceX Falcon 9 on January 3rd, 2023, and has been orbiting Earth with an altitude between 500 km to 600 km. The water propulsion system is expected to prolong the satellite's life by 2.5 years. The propulsion system operated for approximately 2 minutes on March 3rd, 2023, and the company confirmed the successful generation of thrust from the obtained flight telemetry (161).

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 4.20). This technology can be generally classified into the following types according to the propellant used:

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 colloid 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 gallium.

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.

Figure 4.20: Schematic of typical electrospray emitter and electrode configurations. Credit: NASA.

b. Key Integration and Operational Considerations

- **Charge Balance**: Ionic liquid propellants can support electrospray operations with just cation (i.e., positively charged) emission or bi-polar emission (i.e., both anions and cations). For cation-only emission, as with FEEP thrusters, a separate cathode neutralizer is needed to maintain overall charge balance; such neutralizers do not necessarily need to be tightly integrated with the thruster heads, with resultant mass, volume, and power impacts that must be understood for spacecraft integration. For electrospray technologies using bi-polar emission, reliable high-voltage switching in the power electronics becomes a critical consideration.

- **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. Consideration of only the primary beam plume may not be sufficient, as droplet emission and molecular fragmentation within the plume can generate off-axis species. Plume shields are frequently employed on spacecraft to protect sensitive surfaces in the absence of high-fidelity electrospray plume models or characterization data.

- **Propellant Handling and Thruster Contamination**: Ionic liquids and metallic propellants can be sensitive to humidity and oxidation, so care is needed if extended storage is required prior to flight. 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 operations. Due to environmental factors such as spacecraft outgassing and atomic oxygen levels, thruster operations following spacecraft deployment may need a conditioning or burn-in period before achieving full propulsive performance.
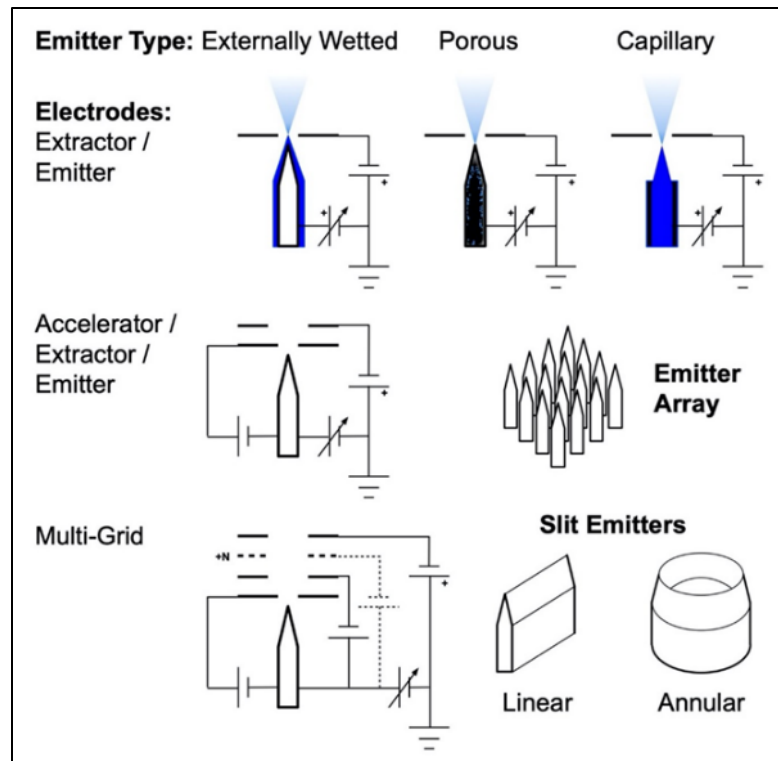
- **Performance Stability and Lifetime**: As an electrospray propulsion system operates over time, the propulsive performance can degrade as the plume impinges upon and deposits condensable propellant on thruster head surfaces; in time, sufficiently deposited propellant buildup can electrically short out the thruster electrodes and terminate thruster operation. 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 to provide very precise thrust 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 can 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 4.21) 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 mission-level performance requirements (163).
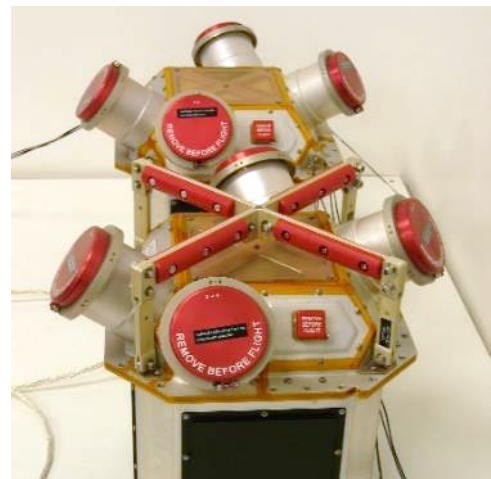
Figure 4.21: Flight CMNT modules for LISA Pathfinder. Credit: Busek.

The Enpulsion Nano FEEP (figure 4.22), formerly the IFM Nano, 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

Figure 4.22: Nano. Credit: Enpulsion.

agreement with the ~220 µN commanded thrust (164). Since this initial demonstration, the Enpulsion Nano has flown onboard other spacecraft, but limited on-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 (165) (166) and the DOD-funded Harbinger technology demonstrator (launched onboard Electron flight STP-27RD in May 2019) (167) (168). The Enpulsion 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 (169) (170). A summary of available on-orbit statistics, anomalies, and lessons learned for the Enpulsion Nano family (Nano, Nano R³, and Nano AR³) is available (171).

An Enpulsion Micro R³ (figure 4.23) was housed onboard the GMS-T mission, which launched in January 2021 onboard a Rocket Lab Electron. The telecommunications satellite uses an OHB Sweden Innosat platform with propulsive capability for collision avoidance. Inaugural on-orbit commissioning of the propulsion system was confirmed in March 2021 (173).

Figure 4.23: Micro R³. 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 (174) (175). Orbit lowering capability was demonstrated in 2020; of the four individual thrusters, three experienced anomalous behavior during the UWE-4 mission (176). A 3U-Cubsat implementation of the same NanoFEEP technology is shown in figure 4.24.

Figure 4.24: Eight NanoFEEP thrusters integrated on 3U-Cubesat bus. Credit: Morpheus Space.

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 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 4.25).

- **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 4.26).

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, krypton) 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 must be maintained during thruster assembly, transport, launch, and operations to minimize grid erosion. Random vibration tests at the protoflight level should be conducted to verify the survivability of the ion optics against launch loads. 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 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. For missions requiring extended thruster operations, a secondary propulsion system or reaction wheels may be needed to counter the torque buildup (177).
- **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.
- **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.
- **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
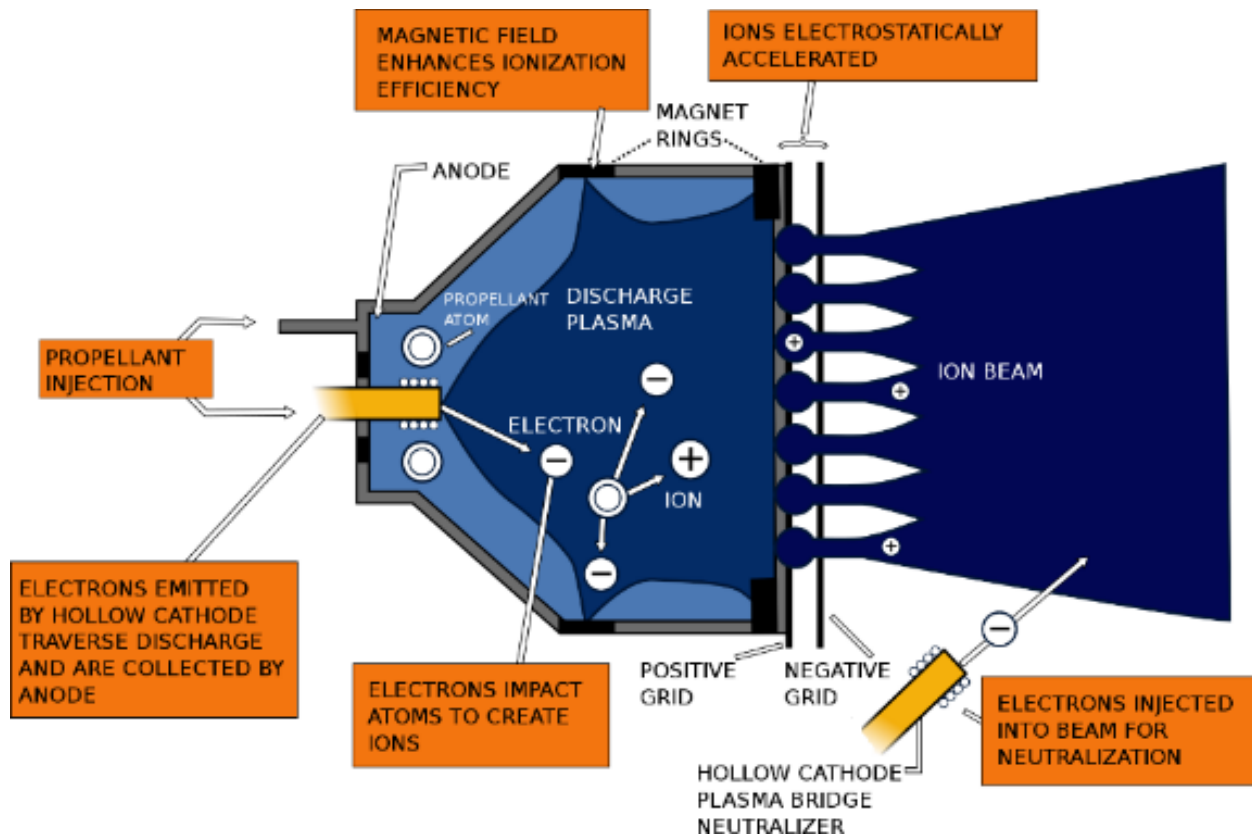
Figure 4.25: Schematic of typical DC-discharge gridded-ion thruster. Credit: NASA.
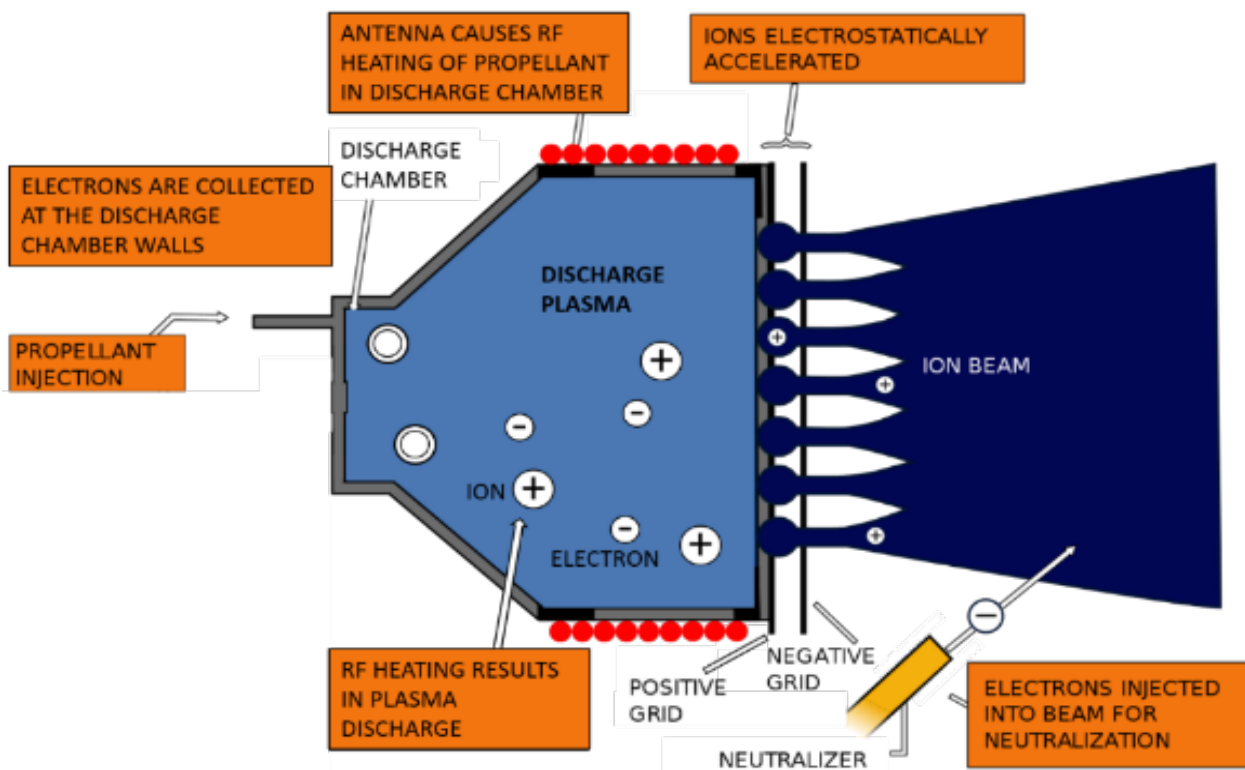
Figure 4.26: Schematic of typical RF-discharge gridded-ion thruster. Credit: NASA.

potential spacecraft interactions and the long-term reliability of feed system and thruster components exposed to iodine.

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 4.27), with one serving as a redundant backup, successfully provided drag-free control of the 1000-kg satellite until xenon propellant exhaustion in October 2013 (178) (179) (180) (181).

ThrustMe's NPT30-I2 1U or 1.5U (figure 4.28), an integrated, RF-discharge gridded-ion propulsion system, has at least two units in space as of July 2023. The Beihangkongshi-1 satellite was launched in November 2020 onboard a Long March 6 rocket. 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 (182) as described in a Nature publication (183). In April 2023, the 1.5U flew 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 (184), which included a demonstration of satellite collision avoidance maneuvers (185). With more than 100 systems ordered by clients (186) as of July 2023, upcoming launches include the NPT30-I2 onboard a GomSpace 12U CubeSat for the ESA GOMX-5 technology demonstration mission, NTU Singapore's INSPIRESAT-4, and Turion Space's DROID.002 (187).

Figure 4.27: T5 gridded-ion thruster for GOCE mission. Credit: QinetiQ.

Figure 4.28: NPT30-I2-1U. Credit: ThrustMe.

The Lunar IceCube and LunarH-Map missions both included a Busek BIT-3 propulsion system (figure 4.29) with solid iodine propellant. The BIT-3 system was the primary propulsion for each spacecraft's lunar transfer, orbit capture, orbit lowering, and spacecraft disposal. Each integrated BIT-3 system includes a low-pressure propellant tank with heated propellant-feed system components, a power processing unit to control the RF thruster and RF cathode, and a two-axis gimbal assembly.

The Lunar IceCube mission, led by Morehead State University, aimed to characterize the distribution of water and other volatiles on the Moon from a highly inclined lunar orbit with a perilune less than 100 km. The 6U spacecraft was launched as a secondary payload onboard Artemis I (188) (189). Communication issues with the CubeSat were reported shortly after launch (190).

Figure 4.29: BIT-3 thruster. Credit: Busek.

The Lunar Polar Hydrogen Mapper (LunaH-Map) mission, led by Arizona State University, aimed to map hydrogen distributions at the lunar south pole from a lunar orbit with a perilune less than 20 km. The 6U spacecraft was launched as a secondary payload onboard Artemis I (191). The mission ended in late-2023 as the BIT-3 was inoperable and unable to perform the required mission maneuvers. The mission team concluded that the propulsion system's tank valve was stuck closed (192) (193).

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 (194). The first flight of a U.S. manufactured HET, the Busek BHT-200, was onboard the TacSat-2 spacecraft (195), 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 (196), formerly BPT-4000. Launches of HETs greatly accelerated in 2019 with the launch of 120 SpaceX Starlink and 6 OneWeb spacecraft (197), each using an HET. As of November 2023, SpaceX has launched over 5,500 Starlink satellites, and OneWeb has launched over 600 satellites. 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 typically operate less efficiently with krypton propellant, and krypton has more challenging storage requirements, krypton gas is considerably lower cost than xenon gas. Lower cost is a compelling attribute when the potential number of spacecraft are projected in the thousands, as with constellations. While xenon is generally a superior propellant, krypton fed Hall thrusters will likely become more common in the future, especially for large constellations. SpaceX's 2nd Generation Starlink spacecraft made another first using argon as a Hall thruster propellant. While argon has been commonly tested with lab HETs due to its low cost and good availability, argon is not generally considered a good choice for spaceflight due to its challenging storage requirements. However, given SpaceX's need for such large volumes of propellant for the thousands of satellites planned, supply chain challenges may provide good justification for their use of this generally non-ideal propellant. Many other

propellants have been considered and ground tested for Hall-effect thrusters, but to date only Hall-effect thrusters using xenon, krypton, or argon have flown.

As schematically shown in figure 4.30, 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. Many recent thruster designs have begun centrally mounting the cathode in the HET body as shown in figure 4.30. 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.
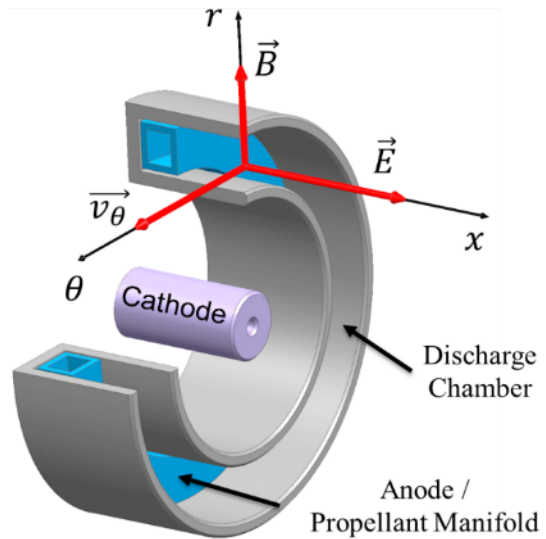
Figure 4.30: Hall-effect Thruster schematic. Credit: NASA.

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 depends 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 operating 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, may result 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 (or 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 (198) (199) (200).

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 (201) (202).

The Busek BHT-200 (figure 4.31) 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

Figure 4.31: BHT-200 thruster. Credit: Busek.

2018. A Busek PPU powered the 200 W HET for each of the FalconSat missions (203) (204) (205). Ground testing of the BHT-200 includes multiple propellants, although all spaceflights of the BHT-200 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 (206) (207).

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 4.32) 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 (208) (209) (210) (211) (212).

Figure 4.32: IHET-300 thruster. Credit: Rafael.

The European and Italian space agencies selected the SITAEL HT100 (figure 4.33) 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. The mission will use the thruster to raise and lower its orbit over a series of verification tests conducted during a planned two-year lifespan. The mission seeks to conduct at least one thousand ignition cycles. SITAEL recently completed 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 that is 75 kg with dimensions of 60 x 40 x 36 cm^3. The spacecraft launched December 1, 2023 (213) (214) (215) (216). The HT100 is also baselined for the Italian Space Agency Platino-1 and Platino-2 spacecraft (217).
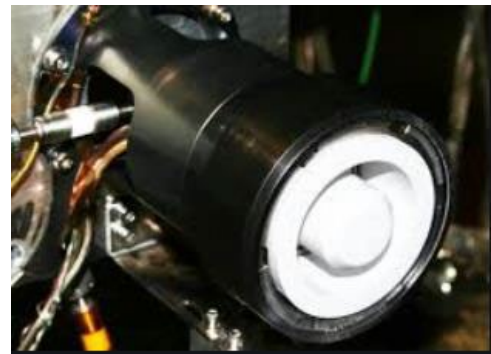
Figure 4.33: HT100 thruster. Credit: SITAEL.

The Astra Spacecraft Engine (ASE), figure 4.34, successfully achieved orbital ignition onboard the Spaceflight Sherpa-LTE1 orbital transfer vehicle, which launched on SpaceX's Transporter-2 mission on June 30, 2021 (218). This single-string system is sized to achieve a controlled de-orbit of Sherpa-LTE1 (219). On-orbit performance was demonstrated by operating the system for 5-minute durations. The first 54 maneuvers have been reported (220). After outgassing, performance metrics were nominal within one standard deviation of ground test data. On-orbit thrust averaged 22.4 mN, and specific impulse for each 5-minute thrust maneuver averaged 1108 seconds. Total propulsion system power processing efficiency averaged 94%, including feed system power,

Figure 4.34: Astra Spacecraft Engine (ASE). Credit: Astra.

circuit efficiency, and housekeeping circuits (221). As of November 2023, the ASE aboard the Sherpa-LTE1 is continuing mission operations and has operated for more than 800 five-minute maneuvers (i.e., accumulated total duration of ~70 hours) (218). The ASE, (formerly the Apollo Constellation Engine) is a propulsion system that was acquired in Astra's purchase of Apollo Fusion in 2021. The ASE is designed for operation with xenon and krypton propellants and sized to fit ESPA-class missions. The propulsion system includes several key technologies, including permanent magnets, a heatless instant start cathode, and a radiation hardened PPU. Astra has also reportedly sold ASE units to OneWeb (222) and LeoStella (223), Apex (224), Astroscale (225), and Maxar (226). As of March 2023, Astra has nine ACE in orbit (224).

Exotrail's first in-orbit propulsion demonstration mission was in November 2020. NanoAvionics and Exotrail partnered to integrate Exotrail's spaceware - nano (figure 4.35) into a NanoAvionics M6P nanosatellite 6U bus, which was built, integrated, and qualified in 10 months. Following this, Exotrail signed a contract with AAC Clyde Space to provide propulsion for the Eutelsat ELO 3 (2022) and ELO4 (2023) 6U CubeSats (227) (228) (229) (230) (231), and a contract with Aerospace Lab to provide its spaceware - nano for Aerospace Lab's Risk Reduction Flight (RRF) mission. The Aerospace Lab spacecraft, known as "Arthur," was launched in 2021. The propulsion system will be used to demonstrate the spacecraft maneuver capabilities (232) (233). Blue Canyon Technologies (BCT) selected the Exotrail's spaceware

Figure 4.35: spaceware-nano thruster. Credit: Exotrail.

system for the company's Venus-class microsatellite platform, which will be used for the NASA's INCUS mission, which is expected to launch in 2026 (234) (235).

A spaceware - micro cluster[2] (figure 4.36) will be integrated onboard York Space Systems S-Class platform for a satellite mission scheduled for launch in late 2023, aiming to orbit the Moon and deliver Earth-to-Moon telecommunication services in support of Intuitive Machines' lunar south pole mission. With Exotrail's spaceware - micro cluster[2], York Space Systems will be able to execute maneuvers such as a lunar transfer orbit (236). Exotrail also has agreements to provide the spaceware – micro for missions by OHB Luxspace (237), Satrec Initiative (238), Astro Digital (239), and Muon Space (240).

Exotrail will also launch its spacevan In-Orbit Demonstration (IOD) mission in October 2023. The spacevan will use Exotrail's spaceware - micro cluster[2] to demonstrate its capabilities (e.g., plane change or altitude change) (241).
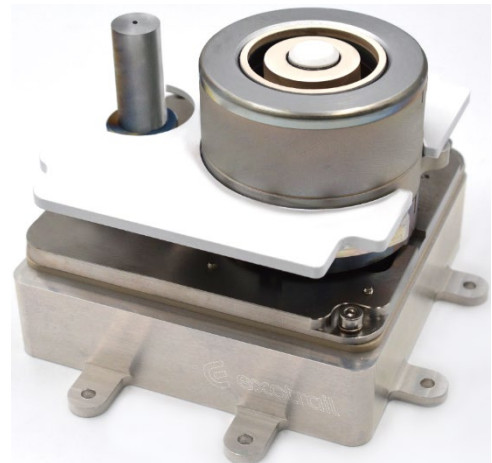
Figure 4.36: spaceware-micro thruster. Credit: Exotrail.

Airbus will integrate Exotrail's spaceware - mini thruster, Exotrail's 300W to 600W class electric propulsion system, onto Airbus' Earth Observation satellite platform portfolio. Exotrail anticipates completion of the thruster's qualification activities in 2024 (242).

Blue Canyon, a Raytheon subsidiary, is producing satellites for the DARPA Blackjack program. Blue Canyon selected Exoterra's Halo thruster (figure 4.37), for its Phase 2 and Phase 3 satellites (243). Exoterra has demonstrated Halo on three Blackjack satellites that launched in June 2023 on the SpaceX Transporter-8 rideshare. These are the first flights of ExoTerra's Halo electric propulsion system (244). Additionally, ExoTerra has received a NASA Tipping Point award to perform an in-orbit demonstration of their 12U Courier SEP spacecraft bus, tentatively planned for launch in 2024. The bus includes ExoTerra's Halo thruster, xenon flow control system (XFC), power processing unit (PPU), and deployable solar arrays. The Courier spacecraft provides up to 1 km/s of delta-v, while hosting a 2U, 4 kg payload. The Tipping Point mission objective is to demonstrate the

Figure 4.37: Halo thruster. Credit: ExoTerra Resource.

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. For the Tipping Point mission, the 0.85 kg, 1/3U thruster will nominally operate at 135 W discharge power and produce ~8 mN of thrust (245) (246) (247) (248) (249).

Exoterra Resource is commercializing as Halo12 a version of JPL's Magnetically Shielded Miniature (MaSMi) Hall thruster technology for multiple unnamed flight opportunities. JPL performed extensive development of the technology between 2011 and 2021, culminating in a 7,205-hour wear test of JPL's engineering model thruster. The life test processed over 100 kg of xenon propellant and produced a total impulse of approximately 1.55 MN-s. Exoterra's unnamed flight programs (250) require additional life time, beyond JPL's demonstration, so the same JPL engineering model thruster was recently installed in JPL's high-bay vacuum facility to extend the demonstration to 8187 hours and 1.73 MN-s. Other qualification tests conducted by JPL include a 25,000 ignition cycle heaterless cathode demonstration, a 3000-cycle coil TVAC test, and dynamic testing of the thruster (251). It is important to note that the thruster tested by JPL was not produced by Exoterra. However, given that the JPL produced thruster continues to be operated in support of Exoterra's commercialization activities, it is plausible that Exoterra has retained the key life limiting features of the JPL design (250) (252).

Orbion's Aurora Hall-effect thruster (figure 4.38) system was selected for a U.S. Space Force 400-kg prototype weather satellite, under contract with General Atomics Electromagnetic Systems (GA-EMS). The Aurora propulsion system consists of a thruster, cathode, power processing unit, propellant flow controller, and cable harness. The system will be used for orbit raising, orbit maintenance, and de-orbit over the 3-5 year mission (253) (254) (255).
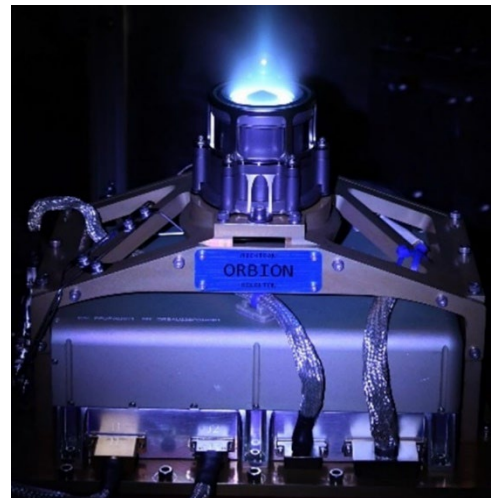
Figure 4.38: Aurora HET, PPU and XFC. Credit: Orbion Space.

Busek has supplied its BHT-350, figure 4.39, Hall-effect thruster to Airbus OneWeb Satellites (AOS) for a range of missions. Busek engineered and qualified the thrusters for orbit raising, orbit maintenance, and end-of-life de-orbit. Eighty OneWeb satellites with BHT-350s launched in December 2022 and January 2023. More than 100 units were operating on OneWeb satellites as of August 2023 with all thrusters reported as operational (256) (257) (258) (259).

Busek shipped its first flight BHT-600 Hall-effect thruster system to a U.S. Government customer in early 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 4.40. The thruster successfully demonstrated 70 kilograms of xenon propellant throughput before the test was terminated (260) (261).

Northrop Grumman's (NG) Space Systems Sector has developed the NGHT-1X (figure 4.41) thruster for its next generation satellite servicing vehicle known as the Mission Extension Pod (MEP). MEP carries power and propulsion for self-propelled orbit raising as well as station keeping and momentum management when docked to a client vehicle. MEP is designed for a 6-year mission life but can carry a propellant load that permits even longer lifetimes. The NGHT-1X is designed to operate nominally in the 600 to 1,000 W range, but has demonstrated stable operation from 200 to 1,100 W. Within the nominal power range, the thruster is designed to generate a minimum total impulse of 2.1 MN-s, not including margin, to enable the MEP mission. NG partnered with the NASA Glenn Research Center (GRC) to develop and commercialize the NGHT-1X, licensing NASA's technology for a high propellant throughput, sub-kilowatt hall-effect thruster. NG's SpaceLogistics has sold all three MEPs that are on the manifest for the first launch on a Falcon 9 rocket planned for 2025 (262) (263) (264) (265) (266) (267) (268).

The EOS SAT-1 developed by Dragonfly Aerospace for the EOS Data Analytics space mission successfully tested its SETS SPS-25 propulsion system. The EOS SAT-1 launched on a Falcon 9 rocket on January 3, 2023 (269) (270).
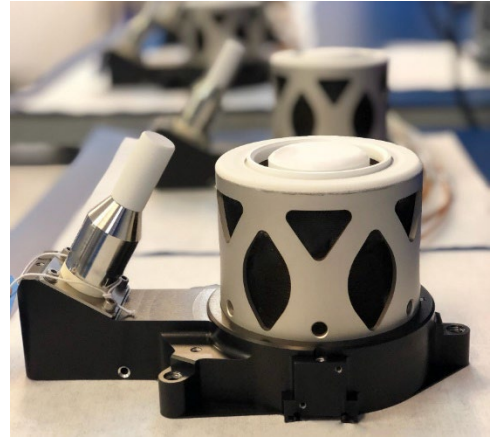
Figure 4.39: BHT-350 Flight Units. Credit: Busek Co.
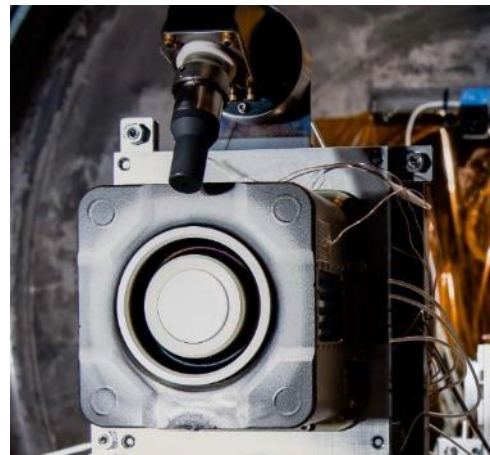
Figure 4.40: BHT-600 Installed in NASA GRC Vacuum Test Facility. Credit: Busek Co.

Figure 4.41: NGHT-1X Engineering Model Hall-Effect Thruster. Credit: Northrop Grumman.

The Aliena Pte Ltd MUSIC-SI Hall thruster (figure 4.42) was integrated on the NuX-1 3U nanosatellite in July 2021. In-orbit deployment was achieved in January 2022 during SpaceX Transport 3 Mission. Aliena's Hall thruster became the first to operate onboard such a small satellite, featuring the lowest power consumption for a commercial Hall thruster at 20W (271) (272) (273) (274) (275).

The MUSIC-HM Hall thruster from Aliena and four ARM-A resistojets from Aurora Propulsion Technologies form the Multi-modal Electric Propulsion Engine (MEPE). MEPE was deployed on an Orbital Astronautics 12U satellite (ORB-12 Strider) in July 2023 by a PSLV rocket (PSLV C-56 DS-SAR Mission). The mission will demonstrate the capability of these propulsion systems to support a small

Figure 4.42: NuX-1 satellite with MUSIC-SI Hall Thruster. Credit: Aliena Pte Ltd

satellite constellation. MEPE's dual electric propulsion technologies allows both the high thrust, low specific impulse of resistojets and low thrust, high specific impulse of Hall thrusters. Both systems share a common propellant, electronics, and fluidics (271) (275) (276).

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 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 (277).

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 constantly changes. 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 (278). 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 micronewton-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. The amount of material accelerated may be maximized through higher frequency pulsing.
- **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 drive 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, spacecraft contamination 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 onboard, one Monofilament Vaporization Propulsion (MVP) system (143) (144) (145), shown in figure 4.18, and one Fiber-Fed Pulsed Plasma Thruster (FPPT) system (149) (150) (151) (152) (153), shown in figure 4.43. 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

Figure 4.43: FPPT 1.7U module. Credit: CU Aerospace.

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. Thrust-vectoring capability of ±10° in the yaw and pitch axes (also with the potential for roll control authority) has been incorporated into the system allowing for limited ACS capability and wheel desaturation for deep space missions. In-orbit operations will demonstrate multiple mission capabilities including inclination change, orbit raising and lowering, drag makeup, and deorbit burns. Launch is manifested in early-2024 (155).

The Benchmark Space Systems' Xantus thruster (figure 4.44) is a millinewton-class non-toxic metal plasma thruster that is compatible with multiple solid metal propellants. The baseline molybdenum propellant has a measured specific impulse of 1774 seconds. The Xantus thruster first launched in January 2023 onboard the USSF Rapid Revisit Optical Cloud Imager (RROCI) 12U CubeSat demonstration mission. However, the RROCI spacecraft failed to deploy after launch. Xantus is also manifested in 2024 onboard Orion Space Solutions' EWS (Electro Optical Weather System) mission (279) (280).

Comat's Plasma Jet Pack (PJP) vacuum arc thruster can be configured with up to four nozzles operating on a solid/inert propellant to provide vectorized thrust (281).
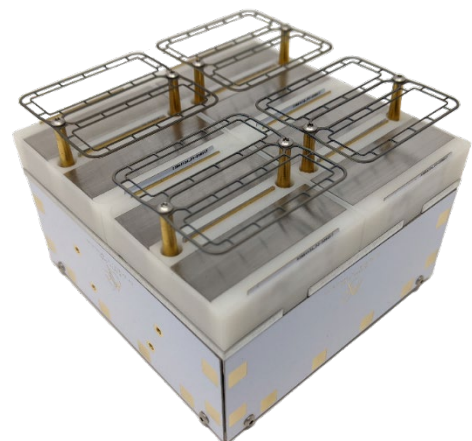
Figure 4.44: Xantus Thruster. Credit: Benchmark Space Systems.

The PJP has launched as part of the Vega VV23 mission on the ∑yndeo-2 6U CubeSat. At the time of this publication, no details have yet been reported on the commissioning of the propulsion system (282).

The Neumann Drive is a center-triggered pulsed cathodic arc propulsion technology developed by Neumann Space using molybdenum propellant. The Neumann Drive is offered in multiple configurations offering a range of propulsive performance depending on available spacecraft power (283). The ND-15 thruster, designed for nanosatellites, is currently in orbit undergoing testing on Australia's Skykraft satellite (284) (285) (286), and another ND-15 will be flown in late 2023 on the University of Melbourne's SpIRIT nanosatellite (287) (290). The ND-50, a more compact system, is planned for missions in 2024 on the 6U EDISON satellite (291) and 150 kg CarbSAR satellite (292).

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.

c. Missions

The SpaceX Falcon 9 Transporter-1 launch in January 2021 included two small satellites 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 (293). Phase Four has further reported in 2023

Figure 4.45: Maxwell Block 1. Credit Phase Four.

that its Maxwell Block 2, with improved modularity and performance compared to Block 1, has been demonstrated on orbit (294).

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 (50-I2-Small) module (figure 4.46), an integrated propulsion system that includes thruster, power processing unit, and heated propellant-feed components. The propulsion demonstration is expected to include orbit raising and lowering around the 600-km orbit (296) (297) (298).

Figure 4.46: REGULUS propulsion module. Credit: T4i.

A 6U CubeSat from Team Miles was awarded a rideshare slot onboard Artemis I, as one of the winning teams in NASA's Cube Quest Challenge. The objective of the mission was 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' current M1.4 system, were integrated onboard the CubeSat to provide primary propulsion as well as 3-axis control (299) (300). While the spacecraft was deployed in November 2022, communications contact was not established (405).

d. Summary Table of Devices

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

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

4.6.3 In-Space Propellant-less Propulsion

Propellant-less propulsion systems generate thrust via interaction with the surrounding environment (e.g., solar photon pressure, planetary magnetic fields, solar wind and ionospheric plasma pressures, and planetary atmospheres). By contrast, chemical and electric propulsion systems generate thrust by expulsion of reaction mass (i.e., propellant). Four propellant-less propulsion technologies have undergone in-space demonstrations to date, including solar sails, tethers, electric sails (and plasma brakes), and aerodynamic drag devices.

Solar Sails

a. Technology Description

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.

b. Missions

NASA's NanoSail-D2, figure 4.48, 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 (301).

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 September 2022 (302).

Figure 4.48: Deployment test of the NanoSail solar sail. Credit: NASA.

NASA's Near-Earth Asteroid (NEA) Scout (figure 4.49) mission launched as a secondary payload onboard Artemis I in November 2022. Unfortunately, contact was never made with the spacecraft, and the sail was not able to be tested. The 6U CubeSat was to deploy an 85-m^2 solar sail and conduct a flyby of an asteroid within two years of launch (303) (304).

NASA's Advanced Composite Solar Sail System (ACS3) technology demonstration uses composite materials for solar sail propulsion. The unfurled

Figure 4.49: Deployment test of the Near-Earth Asteroid Scout solar sail. Credit: NASA.

sail, approximately 9-m^2, will be deployed from a 12U CubeSat that is planned for launch in 2024 (305).

Tethers

a. Technology Description

NASA has developed tether technology for space applications since the 1960's. A tether is nothing more than a long wire, conducting or nonconducting, deployed from a spacecraft in orbit. Two types of tethers and systems can be used for space transportation, with both having had their fundamental physical principles demonstrated in space:

- **Electrodynamic Tethers (EDT)**: A propulsive force of $F = IL \times B$ (Lorentz force) is generated on a spacecraft-tether system when a current I from an onboard power supply is fed into a tether of length L against the electromotive force (emf) induced in it by the geomagnetic field B. This concept will work near any planet with a magnetosphere (Earth, Jupiter, etc.). The resulting force may be used to accelerate the deorbit of a spacecraft, boost a spacecraft's orbital altitude (by reversing the direction of current flow using an onboard power supply), or change its orbital inclination.
- **Momentum Exchange Tethers**: Using completely different physical principles, long nonconducting tethers can exchange momentum between two masses in orbit to place one body into a higher orbit or a transfer orbit for lunar and planetary missions. Recently

completed system studies of this concept indicate that it would be a relatively low-cost in-space asset with long-term, multi-mission capability.

b. Missions

Important milestones include retrieval of a tether in space on the Tethered Satellite System (TSS) mission (TSS-1, 1992), the accidental momentum-exchange boost of the TSS-1R satellite when the tether broke during flight (TSS-1R, 1996), successful deployment of a 20-km-long tether in space (SEDS-1, 1993), closed loop control of tether deployment (SEDS-2, 1994), and operation of an electrodynamic tether with tether current driven in both directions (i.e., power and thrust modes) (PMG, 1993).

More recently, 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 4.50, 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 caused 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 (306) (307) (308).

Figure 4.50: Nanosat Terminator Tape (NSTT). Credit: Tethers Unlimited.

The Naval Postgraduate School's NPSat-1 was launched as a secondary payload on STP-2 and deployed its NSTT in late 2020 (308). 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 used a 250-m NSTT (308) (309). A comparison of flight data for operation of the NSTT from each of these three missions has been publicly released (310).

Electric Sails

a. Technology Description

An electric sail (also known as an electric solar wind sail or an E-sail) uses the dynamic pressure of the solar wind as a source of thrust. By using positively charged wires deployed from a spacecraft to create electric fields around each wire, a large 'virtual' sail is created that deflects solar wind protons and extracts their momentum. Unlike EDTs, they do not derive thrust using the Lorentz force. Electric sails must be used outside the Earth's magnetosphere.

A closely related cousin to the electric sail is the plasma brake, an electrostatic deorbit propulsion system that, like the electric sail, uses Coulomb collisions between the charged wire and the ions in the surrounding environment (e.g., the Earth's ionosphere) to generate an electrostatic force orthogonal to the tether direction, which lowers the spacecraft's orbital altitude.

These technologies are relatively immature, but researchers in Europe and the USA have been making incremental progress on their development.

b. Missions

The AuroraSat-1 satellite was launched on an Electron rocket on May 5, 2022 (158) (159). The spacecraft was built by SatRevolution with Aurora Propulsion Technologies providing the payloads. The mission serves as a technology demonstration for a Plasma Brake module (157) (figure 4.51) and an Aurora Resistojet Module for Attitude control (ARM-A) (156) (figure 4.19), both produced by Aurora. The Plasma Brake module on AuroraSat-1 is a dual-redundant system for demonstration purposes. A 50-m tether will be deployed to demonstrate its deorbiting capability.

Figure 4.51: Plasma Brake Module (PBM) demo unit. Credit: Aurora Propulsion Technologies.

Aerodynamic Drag

a. Technology Description

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 the associated 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 is no longer sufficient, as the Federal Communications Commission in late 2022 adopted a draft rule that requires <2000-km altitude spacecraft to deorbit within 5 years of mission completion (312). 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**	**PMI Status**	**Missions**	**References**
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerojet Rocketdyne [USA]	MPS-120	Hydrazine	0.25 – 1.0 (4)	-	2 (2U) 0.8 (1U)	2.5† (2U) 1.6† (1U)	2U 1U	-	Y	D	-	(104)
Aerojet Rocketdyne [USA]	MPS-125	Hydrazine	0.25 – 1.0 (4)	-	14 (8U) 9.9 (6U) 5.2 (4U)	12.1† (8U) 9.3† (6U) 6.2† (4U)	8U 6U 4U	-	Y	D	-	(104)
Stellar Exploration [USA]	Monopropellant CubeSat System	Hydrazine	-	200s	-	-	-	-	Y	F	Echostar Global 3, NASA Capstone	(18) (19) (20) (21) (105)
Stellar Exploration [USA]	Bipropellant CubeSat system	Hydrazine/ NTO	-	285	-	-	-	-	Y	D	-	(105)
Thruster												
Aerojet Rocketdyne [USA]	MR-103	Hydrazine	1	202 - 224	183	0.33 - 0.37	-	< 16	-	F	numerous	(8)
Aerojet Rocketdyne [USA]	MR-106	Hydrazine	22	228 - 235	561	0.59	-	< 36	-	F	numerous	(8)
Aerojet Rocketdyne [USA]	MR-111	Hydrazine	4	219 - 229	262	0.37	-	< 16	-	F	numerous	(8)
Aerojet Rocketdyne [USA]	MR-401	Hydrazine	0.08	182	200	0.60	-	-	-	F	Numerous	(8)
ArianeGroup [France]	1 N	Hydrazine	1	200 - 223	135	0.29	-	-	-	F	ALSAT-2, numerous	(6) (7)
ArianeGroup [France]	20 N	Hydrazine	20	222 - 230	517	0.65	-	-	-	F	Numerous	(7)
IHI Aerospace [Japan]	MT-9	Hydrazine	1	208 - 215	(100 kg)*	-	-	-	-	F	Numerous	(17)
IHI Aerospace [Japan]	MT-8A	Hydrazine	5	212 - 225	(190 kg)*	-	-	-	-	F	Numerous	(17)
IHI Aerospace [Japan]	MT-2	Hydrazine	20	210 - 226	(115 kg)*	-	-	-	-	F	Numerous	(17)
Moog [USA]	MONARC-1	Hydrazine	1	227	111	0.38	113x50 mm	18 (valve)	-	F	numerous	(12)
Moog [USA]	MONARC-5	Hydrazine	4.5	226	613	0.49	203x380 mm	18 (valve)	-	F	NASA SMAP, numerous	(12) (13)
Moog [USA]	MONARC-22-6	Hydrazine	22	228	533	0.72	203x380 mm	30 (valve)	-	F	numerous	(12)
Moog [USA]	MONARC-22-12	Hydrazine	22	228	1,173	0.69	229x530 mm	30 (valve)	-	F	numerous	(12)
Northrop Grumman [USA]	MRE-0.1	Hydrazine	1	216	(34 kg)*	0.5	114x175 mm	15	-	F	numerous	(14)
Northrop Grumman [USA]	MRE-1.0	Hydrazine	5	218	(544 kg)*	0.5	114x188 mm	15	-	F	numerous	(14)
Northrop Grumman [USA]	MRE-4.0	Hydrazine	18	217	(249 kg)*	0.5	61x206 mm	30	-	F	numerous	(14)
Rafael [Israel]	1N	Hydrazine	1	205 - 214	100	0.31	-	9 (valve)		F	numerous	(15) (16)
Rafael [Israel]	5N	Hydrazine	6	210 - 220	74	0.31	-	9 (valve)		F	numerous	(15) (16)
Rafael [Israel]	25N	Hydrazine	28	205 - 220	100	0.53	-	15 (valve)		F	numerous	(15) (16)

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, "-" = denotes data not available or applicable, * denotes mass of propellant throughput, rather than total impulse

Manufacturer	Product	Propellant	Thrust per Thruster (Quantity)	Specific Impulse	Total Impulse	Mass	Envelope	Power	ACS	PMI Status	Missions	References
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerojet Rocketdyne [USA]	MPS-130	ASCENT	0.25 – 1.0 (4)	-	2.7 (2U) 1.1 (1U)	2.8† (2U) 1.7† (1U)	2U 1U	-	Y	D	-	(103) (104)
Aerojet Rocketdyne [USA]	MPS-135	ASCENT	0.25 – 1.0 (4)	-	19.4 (8U) 13.7 (6U) 7.3 (4U)	14.7† (8U) 11.2† (6U) 7.2† (4U)	8U 6U 4U	-	Y	D	-	(104)
Aerospace Corp. [USA]	HyPer	H_2O_2	-	-	-	-	0.25U	-	-	D	-	(106)
Benchmark Space Systems [USA]	Halcyon	HTP	0.25 1 5 10	150 - 170	(1.56 kg)* (6.6 kg)* (32.5 kg)* (65 kg)*	-	-	< 4	Y	E	Transporter 2 (Undisclosed missions)	(38) (107) (108) (109)
Benchmark Space Systems [USA]	Halcyon Avant	HTP + Fuel	2 10 22	290 - 310	(6.8 kg)* (34 kg)* (74.8 kg)*	-	-	< 15	Y	F	Sherpa-TLC2	(36) (37) (109)
Benchmark Space Systems [USA]	Peregrine	HTP & Alcohol	0.1 - 22 (1-8)	270 - 300	5 - 150	-	-	-	-	D	-	(109)
ECAPS [Sweden]	SkySat 1N HPGP Propulsion System	LMP-103S	1.0 (4)	> 200	21	22†	55 x 55 x 15	10	Y	F	Skysat, PRISMA, Astroscale ELSA-d	(26) (27) (28) (29) (110) (111) (112) (113)
Busek [USA]	BGT-X5 System	ASCENT	0.5	220 - 225	0.57	1.5†	1U	20	N	D	-	(114) (115) (116)
Cornell University [USA]	Cislunar Explorer	Water (Electrolysis)	-	-	-	-	6U total (2-units)	-	-	E	CubeQuest Challenge (removed from Artemis 1)	(39) (40) (41)
CU Aerospace [USA]	MPUC	(CMP-X) Peroxide/ Ethanol blend	0.23	178	1.4 (1.5U) 2.1 (2U)	2.5† (1.5U) 3.3† (2U)	1.5U 2U	3	N	D	-	(50) (51)
Dawn Aerospace [New Zealand]	CubeDrive	Nitrous Oxide & Propylene	0.5	-	0.425 1.450	1.17† 2.70†	0.8U 2U	-	N	D	-	(52)
Dawn Aerospace [New Zealand]	SatDrive	Nitrous Oxide & Propylene	0.5 – 16.7	250 - 280	5 – 500	-	-	-	N	D	-	(52)
Moog [USA]	Monopropellant Propulsion Module	Green or 'Traditional'	0.5	224	0.5	1.0†	1U	2 x 22.5 W/Thruster	N	D	-	(117)
Rubicon Space Systems [USA]	Sprite	ASCENT	0.05 - 0.15	215	> 1.2	< 2†	1.5U	2 - 16	N	D	-	(118)
Rubicon Space Systems [USA]	Phantom	ASCENT	0.3 - 0.5	-	> 9	< 10†	8U	15 - 50	Y	D	-	(119)
NASA MSFC [USA]	LFPS	ASCENT	0.1 (4)	> 200s	> 3.5	< 5.5†	~2.4U	15 - 47	Y	E	Lunar Flashlight	(34) (35)
NanoAvionics [Lithuania]	EPSS C1K	IADN-blend	1.0 BOL 0.22 EOL	213	> 0.4	1.2†	1.3U	9.6 (preheat) 1.7 (firing)	N	F	Lituanica-2	(46) (47)
NanoAvionics [Lithuania]	EPSS C2	IADN-blend	1.0 BOL 0.25 EOL	220	> 1.7	2.6†	2U	-	N	D	-	(46)

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, "-" = denotes data not available or applicable, * denotes mass of propellant throughput, rather than total impulse, BOL = Beginning of Life, EOL = End of Life

Table 4-3 (cont.): Other Monopropellant and Bipropellant Propulsion												
Manufacturer	Product	Propellant	Thrust per Thruster (Quantity)	Specific Impulse	Total Impulse	Mass	Envelope	Power	ACS	PMI Status	Missions	References
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems (cont.)												
Tethers Unlimited USA (Subsidiary of ARKA Group LP)	HYDROS-C	Water (Electrolysis)	> 1.2	> 310	> 2.1	2.7†	19 x 13 x 9.2	5 - 25	N	F	Pathfinder Technology Demonstration	(44) (45) (120) (121)
Tethers Unlimited USA (Subsidiary of ARKA Group LP)	HYDROS-M	Water (Electrolysis)	> 1.2 (1)	> 310	> 18	13.7†	38.1 dia. x 19.1	7 - 40	N	D	-	(122)
VACCO USA	ArgoMoon Hybrid MiPS	LMP-103S/ R134a	0.1 (1)	190	0.78	2.07†	1.3U	4.3 20 (max)	Y	F	ArgoMoon (Artemis I)	(29) (30) (31)
VACCO USA	Green Propulsion System (MiPS)	LMP-103S	0.1 (4)	190	3.3	5†	3U	15 (max)	Y	D	-	(123)
VACCO USA	Integrated Propulsion System	LMP-103S	1 (4)	200	12	14.7†	11U	50 (max)	Y	D	-	(53) (54)
Thruster Heads												
Aerojet Rocketdyne USA	GR-M1	ASCENT	0.25 - 0.5	206	(3.4 kg)*	--	--	14 (max)	-	D	-	(48)
Aerojet Rocketdyne USA	GR-1	ASCENT	0.3 - 1.4	231	(12 kg)*	-	-	18 (max)	-	F	GPIM	(8) (32) (33) (48)
Aerojet Rocketdyne USA	GR-22	ASCENT	8 - 25	250	74	-	-	28	-	F	GPIM	(8) (32) (33)
Benchmark Space Systems USA	Felicette	HTP	0.4 - 1.2	161	> 10	-	-	-	-	E	Transporter 2 (Undisclosed missions)	(38) (109)
Benchmark Space Systems USA	Lynx	HTP	2 - 2.5	295	> 20	-	-	-	-	D	-	(109)
Benchmark Space Systems USA	Ocelot	HTP	18 - 22	302	> 240	-	-	-	-	F	Sherpa-TLC2	(36) (37) (109)
ECAPS Sweden	0.1 N HPGP	LMP-103S	0.03 - 0.1	196 - 209	(1 kg)*	0.04 excl. FCV	-	6.3 - 8	-	F	ArgoMoon	(26) (29) (112)
ECAPS Sweden	1 N HPGP	LMP-103S	0.25 - 1	204 - 235	(24 kg)*	0.38	-	8 - 10	-	F	PRISMA, SkySat, ELSA-d	(26) (27) (28) (29) (112) (113)
ECAPS Sweden	1 N GP	LMP-103S/LT	0.25 - 1	194 - 227	(5 - 8 kg)*	0.38	-	8 - 10	-	D	-	(113)
ECAPS Sweden	5 N HPGP	LMP-103S	1.5 - 5.5	239 - 253	-	0.48	-	15 - 25	-	D	-	(112)
ECAPS Sweden	22 N HPGP	LMP-103S	5.5 - 22	243 - 255	(150 kg)*	1.1	-	25 - 50	-	D	-	(112)
Dawn Aerospace New Zealand	B20 Thruster	Nitrous Oxide & Propylene	6.1 - 16.7	-	-	0.6	17.6 x 8.0 x 7.9	-	-	F	numerous	(52)
Dawn Aerospace New Zealand	B1 Thruster	Nitrous Oxide & Propylene	0.46 - 1.28	-	-	0.26	10.8 x 7.9 x 4.0	-	-	F	numerous	(52)
Rubicon Space Systems USA	0.1N	ASCENT	0.03 - 0.28	215 – 235	(3.1 kg)*	0.06	-	7 - 9	-	E	Lunar Flashlight	(34) (35) (49)
Rubicon Space Systems USA	1N	ASCENT	0.2 - 1.1	236 - 250	(> 7 kg)*	0.18	-	< 15	-	D	-	(49)
Rubicon Space Systems USA	5N LT	ASCENT	1 - 6	221 - 261	(3 kg)*	0.25	-	< 25	-	D	-	(49)
Rubicon Space Systems USA	5N HT	ASCENT	1 - 6	221 - 261	(110 kg)*	0.30	-	< 25	-	D	-	(49)

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, "-" = denotes data not available or applicable, * denotes mass of propellant throughput, rather than total impulse

					Table 4-4: Hybrid Chemical Propulsion						
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**PMI Status**	**Missions**	**References**
---	---	---	[N]	[s]	[kN-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Aerospace Corp. [USA]	Propulsion Unit for CubeSats	Paraffin/Nitrous Oxide	-	-	-	-	1U	-	-	D		(65)
JPL [USA]	Hybrid Rocket	PMMA/GOX	-	> 300	-	-	-	-	-	D	-	(59) (60) (61) (62)
NASA Ames [USA]	Hybrid Rocket	PMMA/ Nitrous Oxide	25	247	-	-	-	-	-	D		(63) (64)
Parabilis Space Technologies [USA]	ROMBUS	Various/N2O	222	260	44.6	49	55 dia. x 46	-	Y	D		(66)
Parabilis Space Technologies [USA]	NanoSat Obrital Transfer System	HTPB/N2O	9.4	245	-	-	-	-	Y	C		(67)
Utah State Univ. [USA]	Green Hybrid Rocket	ABS/Nytrox	25 - 50	220 - 300	-	-	-	< 30 for 1-2 seconds	-	D		(57) (58)
Utah State Univ. [USA]	Green Hybrid Rocket	ABS/GOX	8	215	-	-	-	-	-	D	-	(55) (56)

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, "-" = denotes data not available or applicable

Table 4-5: Cold Gas Propulsion											
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**PMI Status**	**Missions**	**References**
---	---	---	[mN]	[s]	[N-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
Aerospace Corp. USA	MEPSI	R236fa	20	-	-	0.188†	-	-	Y	E	STS-113 and STS-116	(68)
Benchmark Space Systems USA	Starling	Nitrogen	10 - 1000	70	-	0.75†	0.5U	< 4	Y	D	-	(109)
GomSpace Sweden	NanoProp CGP3	Butane	0.01 – 1 (4)	60 - 110	40	0.35†	5 x 10 x 10	< 2	Y	E	TW-1	(76) (78) (79) (126)
GomSpace Sweden	NanoProp 6U	Butane	0.1 – 10 (4)	60 - 110	80	0.900†	2 x 10 x 20	< 2	Y	F	GOMX-4B	(76) (77) (127)
Lightsey Space Research USA	BioSentinel Propulsion System	R236fa	20	47	79.8	1.28†	4 x 10 x 20	< 1 idle, < 4 operating	Y	E	BioSentinel	(80) (81) (82) (83)
Microspace Rapid Pte Ltd Singapore	POPSAT-HIP1	Argon	0.083 – 1.1 (8)	43	-	-	-	-	-	E	POPSAT-HIP1	(75)
ThrustMe France	I2T5	Iodine	0.35	-	75	0.9†	0.5U	5	N	F	Xiaoxiang 1-08, NAPA-2, Spire L3C Demo, Robusta-3A (2024**)	(84) (85) (86) (87)
UTIAS/SFL Canada	CNAPS	Sulfur Hexafluoride	12.5 - 50	40	100	-	-	-	N	F	CanX-4/CanX-5	(71) (72) (73) (74)
VACCO USA	NEA Scout	R236fa	25 (6)	-	500	2.54†	2U	< 55 warmup < 9 operating	Y	E	NEA Scout	(95) (96) (97)
VACCO USA	MiPS Standard	R236fa	25 (4)	-	82 - 515	0.85 – 2.46†	0.4 – 1.5U	< 12	Y	D	-	(128)
VACCO USA	MarCO-A and -B MiPS	R236fa	25 (8)	-	755	3.49†	8 x 15 x 20	-	Y	E	MarCO-A & -B	(91) (92) (93) (94)
VACCO USA	C-POD	R236fa	10 (8)	40	186	1.25†	1U	5	Y	E	CPOD	(88) (89) (90)
Thruster Heads												
Marotta USA	CGMT	Nitrogen	105 - 2360	-	-	< 0.06	-	< 7	-	F	NMP ST5	(69) (70)
Moog USA	058E143-146	Nitrogen	10 - 40	60	-	0.04	1.4 x 5.7	< 10	-	F	CHAMP, GRACE	(129)
Moog USA	058E142A	Nitrogen	120	57	-	0.016	1.4 x 2.0	< 35	-	F	Spitzer Space Telescope	(129)
Moog USA	058E151	Nitrogen	120	65	-	0.07	1.9 x 4.1	< 10.5	-	F	Spitzer Space Telescope	(129)
Moog USA	058-118	Nitrogen	3600	57	-	0.023	0.7 x 2.5	< 30	-	F	SAFER, Pluto Fast Flyby	(129)
Moog USA	58E163A	Nitrogen, Xenon, Argon	1300	70 N2, 21 Xe, 54 Ar	-	0.115	2.4 x 5.3	< 10.5	-	F	GEO applications	(129)

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, "-" = denotes data not available or applicable, ** anticipated launch date

Table 4-6: Solid Motor Chemical Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust (Quantity)**	**Specific Impulse**	**Total Impulse**	**Mass**	**Envelope**	**Power**	**ACS**	**PMI Status**	**Missions**	**References**
---	---	---	[N]	[s]	[N-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
PacSci EMC USA	MAPS	-	-	-	-	-	-	-	-	E	PacSciSat	(99) (100)
Thruster Heads												
Industrial Solid Propulsion USA	ISP 30 sec. Motor	80% Solids HTPB/AP	37	187	996	0.95†	5.7	-	-	D	Optical target at Kirtland AFB	(131)
Northrop Grumman USA	STAR 3	TP-H-3498	461	266	1,250	1.16†	8 dia. x 29	-	-	E	Mars Exploration Rover Spirit lander	(132)
Northrop Grumman USA	STAR 4G	TP-H-3399	258	269	2,650	1.5†	11.3 dia. x 13.8	-	-	D	-	(132)

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, "-" = denotes data not available or applicable

Table 4-7: Electrothermal Electric Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**ACS**	**PMI Status**	**Missions**	**References**
---	---	---	[mN]	[s]	[N-s]	[kg]	[cm^3 or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems												
AIS [USA]	AIS-SWAG1-PQ	Adamantane	0.12	20	3	0.224†	4.2 x 4.2 x 5.4	5	N	D	-	(311)
AIS [USA]	AIS-SWAG1-DUO	Adamantane	0.24	20	6	0.449†	8.6 x 4.2 x 5.4	10	N	D	-	(311)
AIS [USA]	AIS-SWAG1-QUAD	Adamantane	0.48	20	12	0.898†	8.6 x 8.6 x 5.4	20	N	D	-	(311)
Aurora [Finland]	ARM-A	Water	0.5	100	70	0.280†	0.3U	10£	Y	E	AuroraSat-1 (2022), ORB-12 STRIDER	(156) (158) (159) (276)
Aurora [Finland]	ARM-C	Water	1	-	-	0.050†	45	12 (max)	N	D	-	(162)
Benchmark Space Systems [USA]	Starling Ardent	Nitrogen	10 - 1000	70 – 140	-	1.15	1U	10 – 100	Y	D	-	(109)
Bradford Space [Netherlands]	Comet-1000	H$_2$O	17	175	1,150	1.55†	2,300	50 (max)	N	F	HawkEye 360, Capella Space, Transporter 7	(135) (136) (137) (138)
Bradford Space [Netherlands]	Comet-8000	H$_2$O	17	175	8,348	6.68†	23,760	55 (max)	N	F	BlackSky	(135) (138) (139)
CU Aerospace [USA]	CHIPS-180	R134a	15	67	176	1.03†	540	20	Y	D	-	(313) (314) (315) (316)
CU Aerospace [USA]	CHIPS-500	R134a	25	69	505	1.84†	1300	25	Y	D	-	(313) (314) (315) (316)
CU Aerospace [USA]	CHIPS-1000	R134a	31	70	1,030	3.13†	2500	30	Y	D	-	(313) (314) (315) (316)
CU Aerospace and VACCO [USA]	PUC	SO$_2$	4.5	70	184	0.72†	0.35U	15	N	E	8 flight units delivered to AFRL	(140) (141) (142)
CU Aerospace [USA]	MVP	Delrin Fiber	4.5	66	280	1.06†	0.93U	45	N	E	DUPLEX (launch 2024**)	(143) (144) (145) (146)
EPL [USA]	APS 100	Ammonia	28	280	2,200	2.63†	2.5U	100	N	D	-	(317)
EPL [USA]	APS 500	Ammonia	150	350	25,200	13.4†	27,300	450	N	E	Atomos Space (2024**)	(317) (318)
Pale Blue [Japan]	PBR-9	Water	1.0	45	35	0.6†	0.5U	9	N	D	-	(319)
Pale Blue [Japan]	PBR-20	Water	1.0	70	200	1.5†	1U	20	N	E	ArkEdge OPTIMAL-1	(160) (319)
Pale Blue [Japan]	Water Resistojet Propulsion System	Water	2.7	60	170	1.4†	1.25U	< 30	N	E	SPHERE-1 EYE	(161)
SteamJet Space Systems [UK]	Steam TunaCan Thruster	Water	6	172	219	0.54†	402	< 19.9	N	D	-	(147)
SteamJet Space Systems [UK]	Steam Thruster One	Water	6	172	-	-	-	19.9	N	D	-	(148)
Thruster Heads												
SITAEL [Italy]	XR-50	Ar, Xe, N$_2$	100	55-85	≤ 72,000	0.22‡	45.8	≤ 50	-	D	-	(320) (321)
SITAEL [Italy]	XR-100	Ar, Xe, N$_2$	125	63-105	≤ 90,000	0.22‡	45.8	≤ 80	-	D	-	(320) (321)
SITAEL [Italy]	XR-150	Ar, Xe, N$_2$	250 (Ar) 100 (Xe)	58-110	≤ 180,000	0.22‡	45.8	≤ 95	-	D	-	(320) (321)

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, "-" = denotes data not available or applicable

Table 4-8: Electrospray Electric Propulsion												
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**Neutralizer**	**PMI Status**	**Missions**	**References**
---	---	---	[μN]	[s]	[N-s]	[kg]	[cm^3 or U]	[W]	---	C,D,E,F	---	---
Integrated Propulsion Systems												
Busek [USA]	CMNT (4x heads)	EMI-Im (ionic)	4 x 20	225	980	14.8†	29U	16.5	Carbon Nanotube	F	LISA Pathfinder	(163)
Busek [USA]	BET-MAX (Config. A)	EMI-Im (ionic)	4 x 55	850	4 x 92§	0.8†	1250	12	Carbon Nanotube	E	US Government	(322) (323) (324) (325) (326) (327) (328)
Busek [USA]	BET-MAX (Config. B)	EMI-Im (ionic)	4 x 40	2300	4 x 250	0.8†	1250	14	Carbon Nanotube	D	-	(328)
Enpulsion [Austria]	Nano	Indium (FEEP)	330	1,500 (alpha); 3,000 (gamma)	> 3,000 (alpha); > 5,000 (gamma)	0.90†	10 x 10 x 8.3	40	Thermionic	F	Flock 3p', ICEYE X2, Harbinger, NetSat, and others	(164) (165) (166) (167) (168) (169) (170) (171) (329) (330) (331)
Enpulsion [Austria]	Nano R^3	Indium (FEEP)	350	1,500 (alpha); 2,700 (gamma)	> 3,000 (alpha); > 5,000 (gamma)	1.4†	9.8 x 9.9 x 9.5	45	Thermionic	E	(Evolution of Nano design)	(171) (172) (332)
Enpulsion [Austria]	Micro R^3	Indium (FEEP)	1,000	2,100 (alpha); 2,750 (gamma)	> 25,000 (alpha); > 35,000 (gamma)	3.9†	14 x 12 x 13.3	105	Thermionic	F	GMS-T	(172) (173) (333) (334)
Enpulsion [Austria]	Neo	Indium (FEEP)	20,000	1,500	> 550,000	30†	34 x 34 x 15	800	Thermionic	D	-	(335)
Morpheus Space [Germany]	NanoFEEP (2x heads)	Gallium (FEEP)	< 40	-	-	0.16‡	9 x 2.5 x 4.3	< 3	Propellant-less	E	UWE-4	(174) (175) (336) (337)
Morpheus Space [Germany]	MultiFEEP (2x heads)	Gallium (FEEP)	< 140	-	-	0.28‡	9 x 4.5 x 4.5	< 19	Propellant-less	D	-	(336)

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, § demonstrated, "-" = denotes data not available or applicable

Table 4-9: Gridded-Ion Electric Propulsion												
Manufacturer	**Product**	**Type**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Power**	**Cathode Type**	**PMI Status**	**Missions**	**References**
---	---		---	[mN]	[s]	[kN-s]	[kg]	[cm³ or U]	[W]	---	C,D,E,F	---	---
Integrated Propulsion Systems													
Avant Space [Russia]	GT-50	RF Ion	Xenon	< 7	-	-	< 8†	< 4U	< 240	Hollow	D	-	(338) (339)
Busek [USA]	BIT-3	RF Ion	Iodine	1.0	1,960	31.7	2.9†	18 x 8.8 x 10.2	70	RF	E	Lunar IceCube; LunaH-Map	(188) (189) (190) (191) (192) (193) (340) (341) (342) (343) (344) (345)
Pale Blue [Japan]	PBI-40 Hybrid	RF Ion (Resistojet)	Water (Water)	0.15 (0.9)	> 500 (40)	-	< 2.5†	9 x 12 x 12	28 23	RF	E	RAISE-3 (DDL); RAISE-4 (2024**)	(346) (347) (348) (349)
ThrustMe [France]	NPT30-I2	RF Ion	Iodine	< 1.3	2,400	5.5 (1U) 14 (1.5U)	1.2† (1U) 1.85† (1.5U)	1U 1.5U	< 81	Thermionic	F	Beihangkongshi-1; NORSAT-TD; INSPIRESAT-4 (2023**); GOMX-5 (2024**); DROID.002 (2024**)	(182) (183) (184) (185) (186) (187) (350) (351) (352)
Thruster Heads													
Ariane Group [Germany]	RIT µX	RF Ion	Xenon	< 0.5	-	-	0.44‡	7.8 x 7.8 x 7.6	< 50	RF	D	-	(353) (354) (355) (356) (357) (358)
Ariane Group [Germany]	RIT 10 EVO	RF Ion	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)	(353) (355) (359)
QinetiQ [UK]	T5	DC Ion	Xenon	< 20	< 3,000	-	2‡	19 x 19 x 24.2	< 600	Hollow	F	GOCE	(178) (179) (180) (181)

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, "-" = denotes data not available or applicable, RF = Radio Frequency, DC = Direct Current, DDL = Destroyed During Launch

Table 4-10: Hall-Effect Electric Propulsion Thrusters											
Manufacturer	**Product**	**Propellant**	**Thrust***	**Specific Impulse***	**Total Impulse***	**Mass**	**Envelope**	**Thruster Power***	**Cathode Type**	**PMI Status**	**Missions**	**References**
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm³ or U]	[W]	Notes	C,D,E,F	---	---
Aliena Pte Ltd [Singapore]	MUSIC-SI	Xenon	< 0.25	< 200	1	2[†]	1.5U	20	None	E	NuX-1	(271) (272) (273) (274) (275)
Aliena Pte Ltd [Singapore]	MUSIC-HM	Xenon	3	1,000	15	5[†]	4U	100	EM-HL	E	ORB-12 STRIDER	(271) (275) (276)
Astra [USA]	ASE	Xenon	25	1,400	300	1.0	-	400[‡]	CM-HL	F	Sherpa-LTE, Aries (2024**), ELSA-M (2024**)	(218) (220) (219) (220) (221) (222) (223) (224)
Astra [USA]	ASE	Krypton	18	1,300	300	1.0	-	400[‡]	CM-HL	D	-	(218)
Busek [USA]	BHT-100	Xenon	6.3	1,086	150	1.2	275 wo cath.	105	EM-SH	D	-	(203) (360)
Busek [USA]	BHT-200	Xenon	13	1,390	84[§]	1.2	675 wo cath.	250[‡]	EM-SH	F	TacSat-2, FalconSat-5, -6	(203) (204) (205) (206)
Busek [USA]	BHT-200-I	Iodine	14	1390	-	1.2	675 wo cath.	250	EM-SH	E	NASA iSat (Cancelled)	(204) (206) (207)
Busek [USA]	BHT-350	Xenon	17	1,244	212[§]	1.9	-	350	EM-SH	F	OneWeb Satellites	(256) (257) (258) (259)
Busek [USA]	BHT-600	Xenon	39	1,500	1000[§]	2.8	1,470 wo cath.	680[‡]	EM-SH	E	US Government	(203) (260) (261) (361) (362)
Busek [USA]	BHT-600-I	Iodine	39	-	-	2.8	1,470 wo cath.	600	EM-SH	D	-	(206) (361) (362) (363)
EDB Fakel [Russia]	SPT-50	Xenon	14	860	126[§]	1.2	1,092	220	EM-SH	F	Canopus-V	(198) (199) (200) (201) (364)
EDB Fakel [Russia]	SPT-50M	Xenon	14.8	930	266	1.3	---	220	EM-SH	D	-	(364)
EDB Fakel [Russia]	SPT-70BR	Xenon	39	1,470	435[§]	2.0	1,453	660	EM-SH	F	KazSat-1, KazSat-2	(201) (202)
EDB Fakel [Russia]	SPT-70M	Xenon	41.3	1,580	-	-	-	660	EM-SH	D	-	(202)
EDB Fakel [Russia]	SPT-70M	Krypton	31.3	1,460	-	-	-	660	EM-SH	D	-	(202)
ExoTerra [USA]	Halo	Xenon	19.6	1,294	>375	0.83	375	400[‡]	CM-HL	F	Tipping Point (2024**), Blackjack	(243) (244) (245) (246) (247) (248) (249)
ExoTerra [USA]	Halo	Krypton	12	900	>175	0.83	375	300[‡]	CM-HL	D	-	(365)
ExoTerra [USA]	Halo12	Xenon	50	1,900	> 2,000	3.4	1,700	1,000[‡]	CM-HL	E[1]	Unnamed Flights	(250) (251) (252) (366)
ExoTerra [USA]	Halo12	Krypton	55	1,575	> 2,000	3.4	1,700	1,000[‡]	CM-HL	D[1]	-	(367)
Exotrail [France]	spaceware nano	Xenon	2.5	800	6	-	2.5U	60	EM-SH	F	M6P Demo, Arthur, ELO3 and ELO4, Otter Pup, INCUS (2026)	(227) (228) (229) (230) (231) (232) (233) (234) (235)
Exotrail [France]	spaceware micro	Xenon	7	1,000	60	-	960	150	EM-SH	E	York Space (2023**), SpaceVan (2023**), Astro Digital (2024), Satrec Initiative (2025), Muon Space (2026)	(227) (230) (236) (237) (238) (239) (240) (241)
Exotrail [France]	spaceware mini	Xenon	23	1,300	300	-	-	400	EM-SH	E	Airbus' Earth Observation satellite platform portfolio	(227) (242)

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, [1]ExoTerra is commercializing the JPL developed MaSMi thruster, "-" = denotes data not available or applicable

Manufacturer	Product	Propellant	Thrust*	Specific Impulse*	Total Impulse*	Mass	Envelope	Thruster Power*	Cathode Type	PMI Status	Missions	References
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm³]	[W]	Notes	C,D,E,F	---	---
JPL [USA]	MaSMi	Xenon	55	1,920	> 1,550§	3.4	1,700	1,000	CM-HL	D	-	(250) (251) (368) (369) (370) (371) (372) (373) (374) (375) (376) (377) (378) (379)
Northrop Grumman [USA]	NGHT-1X	Xenon	55	1,700	> 2,000	3.1	-	900	CM-SH	E	MEP (2025**)	(262) (263) (264) (265) (266) (267) (268)
Orbion [USA]	Aurora	Xenon	15	1,320	200	1.5	1,500	250	EM-SH	E	GA-EMS (**)	(253) (254) (255)
Rafael [Israel]	R-200	Xenon	13	1,160	200	-	-	250	EM-HL	D	-	(208) (380) (381)
Rafael [Israel]	IHET-300	Xenon	> 14.3	> 1,210	> 135	1.5	1,836	300	EM-SH	F	VENuS	(208) (209) (210) (211) (212)
Rafael [Israel]	R-800	Xenon	-	-	> 600	-	-	800	EM-HL	D	-	(208) (382)
Safran [France]	PPS-X00	Xenon	43	1,530	1,000	< 3.2	-	650	EM-SH	D	-	(383) (384) (385)
SITAEL [Italy]	HT100	Xenon	9	1,300	73	-	407 wo cath.	175	EM-SH	E	uHETSat (2023**)	(213) (214) (215) (216)
SITAEL [Italy]	HT400	Xenon	27.5	1230	1,000	2.77	1,330	615	EM-SH	D		(386) (387) (388)
SETS [Ukraine]	ST25	Xenon	7.6	1,000	82	0.95 (with 2 cathodes)	1,003	140	EM-SH	E	EOS SAT-1	(269) (270) (389) (390)
SETS [Ukraine]	ST40	Xenon	25	1,450	400	1.2 (with 2 cathodes)	1,170	450	EM-HL	D	-	(391)

Table 4-10 (cont.): Hall-Effect Electric Propulsion Thrusters

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, [1]ExoTerra is commercializing the JPL developed MaSMi thruster, "-" = denotes data not available or applicable

Manufacturer	Product	Propellant	Thrust*	Impulse Bit	Specific Impulse*	Total Impulse*	Mass	Envelope	Power*	ACS	PMI Status	Missions	References
---	---	---	[μN]	[μNs]	[s]	[N-s]	[kg]	[cm³ or U]	[W]	Y/N	C,D,E,F	---	---
Integrated Propulsion Systems													
AIS [USA]	AIS-VAT1-PQ	Bismuth	26	-	87	0.13	0.056†	37	5	N	D	---	(311)
AIS [USA]	AIS-VAT1-DUO	Bismuth	52	-	87	0.26	0.084†	74	10	N	D	---	(311)
AIS [USA]	AIS-VAT1-QUAD	Bismuth	104	-	87	0.52	0.177†	148	20	N	D	---	(311)
Benchmark Space Systems [USA]	Xantus Metal Plasma Thruster	Molybdenum	400	1	1,774	5,000	1.2†	0.53U	40	N	E	USSF RROCI (not deployed), USSF EWS (2024**)	(392) (279) (280)
Comat [France]	Plasma Jet Pack	Metal	150	20	2,000	120£	1.2†£	0.66U£	30	N	E	∑yndeo-2	(281) (282) (393)
CU Aerospace [USA]	FPPT-1.7	PTFE Fiber	500	165	3,200	24,000	3.0†	1.7U	96	Y	E	DUPLEX (2024**)	(149) (150) (151) (152) (153) (154)
Hypernova Space Technologies [South Africa]	NanoThruster A, Size XS	Solid-state fuel	80	-	≥500	-	0.8†	0.5	10	N	D	---	(394)
Mars Space Ltd [UK] Clyde Space [Sweden]	PPTCUP	PTFE	40	40	600	44	0.28†	0.33U	2	N	D	---	(395) (396)
Neumann Space [Australia]	ND-15	Molybdenum	3.75	45	2,000	880	1.9†	1.5U	15	N	E	Skyride, Apogee	(283) (284) (285) (286) (287) (288) (289) (290)
Neumann Space [Australia]	ND-50	Molybdenum	100	150	2,000	1,800	1.3†	1U	50	N	E	Edison (2024**), CarbSAR (2024**)	(283) (291) (292)

Table 4-11: Pulsed Plasma and Vacuum Arc Electric Propulsion

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, "-" = denotes data not available or applicable, ESV = Ejector Spring Volume, £ assumes power unit and 2 nozzles

Table 4-12: Ambipolar Electric Propulsion												
Manufacturer	Product	Propellant	Thrust*	Specific Impulse*	Total Impulse*	Mass	Envelope	Power	ACS	PMI 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	7	400	5	5.9‡	19 x 13.5 x 19	450	N	F	Capella	(293) (295) (397) (398) (399) (400)
Phase Four USA	Maxwell (Block 2)RF	Xenon	13	700	-	5.0 (without tank)	22 x 12 x 24 (without tank)	450	N	E	(Mission Not Specified)	(400) (401) (294)
Phase Four USA	Maxwell (Kr)RF	Krypton	13.6	1,110	-	-	-	550	N	D	-	(294) (402)
T4i Italy	REGULUS-50-I2RF	Iodine	0.55	550	3 (Small); 7 (Medium); 11 (Large)	2.5† (Small); 4† (Medium); 5.1† (Large)	9.4 x 9.5 x {15, 18, 20} {Small, Medium, Large}	50	N	E	UniSat-7	(296) (297) (403) (298) (404)
T4i Italy	REGULUS-150-I2RF	Iodine	2	1000	11	6.1†	12 x 12 x 27	150	N	D	-	(404)
Miles Space USA	M1.4	Water	17.2	-	-	1.0†	9.5 x 9.5 x 9.5	6	N	E	Team Miles	(299) (300) (405) (405)

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, "-" = denotes data not available or applicable, RF = Radio Frequency

Table 4-13: Propellant-less Propulsion												
Manufacturer	Product	Propellant	Thrust*	Specific Impulse*	Total Impulse*	Mass	Envelope	Power	ACS	PMI Status	Missions	References
---	---	---	[mN]	[s]	[kN-s]	[kg]	[cm³]	[W]	Y/N	C,D,E,F	---	---
Aurora Propulsion Technologies Finland	Plasma Brake	-	< 100 mN/m	-	-	< 2	1U	< 4	N	E	AuroraSat-1	(157) (158) (159)
Tethers Unlimited USA	NSTT	-	-	-	-	0.81	18 x 18 x 1.8	-	N	F	Prox-1, NPSat-1, DragRacer	(306) (307) (308) (309) (310) (407)

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, "-" = denotes data not available or applicable

See Chapter on Passive Deorbit Systems for review of aerodynamic drag devices.

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Chapter Glossary

(ADCS)   Attitude Determination and Control System

(AutoNGC)   Autonomous Navigation, Guidance, and Control

(CoCom)   Coordinating Committee for Multilateral Export Controls

(COTS)   Commercial-off-the-Shelf

(DOF)   Degrees of Freedom

(DSAC)   Deep Space Atomic Clock

(DSN)   Deep Space Network

(EAR)   Export Administration Regulations

(FOGs)   Fiber Optic Gyros

(GNC)   Guidance, Navigation & Control

(GSO)   Geo-stationary Orbit

(USAF)   U.S. Air Force

(HCI)   Horizon Crossing Indicators

(IMUs)   Inertial Measurement Units

(JPL)   Jet Propulsion Laboratory

(LiDAR)   Light Detection and Ranging

(LMRST)   Low Mass Radio Science Transponder

(MarCO)   Mars Cube One

(PFF)   Precision Formation Flying

(PMSM)   Permanent-magnet Synchronous Motor

(PNT)   Position, Navigation, and Timing

(RPO)   Rendezvous and Proximity Operations

(SDST)   Small Deep Space Transponder

(SGP4)   Simplified General Perturbations 4

(SWaP)   Size, weight, and power

(TLE)   Two-Line Element

(TRL)   Technology Readiness Level

(USTP)   University SmallSat Technology Partnerships

5.0   Guidance, Navigation & Control

5.1   Introduction

The Guidance, Navigation & Control (GNC) subsystem includes the components used for position determination and the components used by the Attitude Determination and Control System (ADCS). In Earth orbit, onboard position determination can be provided by a Global Positioning System (GPS) receiver. Alternatively, ground-based radar tracking systems can also be used. If onboard knowledge is required, then these radar observations can be uploaded and paired with a suitable onboard orbit propagator. Commonly, the U.S. Air Force (USAF) publishes Two-Line Element sets (TLE) (1) which are paired with a Simplified General Perturbations 4 (SGP4) propagator (2). In deep space, position determination is performed using the Deep Space Network (DSN) and an onboard radio transponder (3). There are also technologies being developed that use optical detection of celestial bodies such as planets and X-ray pulsars to calculate position data (4).

Using SmallSats in cislunar space and beyond requires a slightly different approach than the GNC subsystem approach in low-Earth orbit. Use of the Earth's magnetic field, for example, is not possible in these missions, and alternate ADCS designs and methods must be carefully considered. Two communication relay CubeSats (Mars Cube One, MarCO) successfully demonstrated such interplanetary capability during the 2018 Insight mission to Mars (5). This interplanetary mission demonstrated both the capability of this class of spacecraft and the GNC fine pointing design for communication in deep space.

ADCS includes sensors to determine attitude and spin rate, such as star trackers, sun sensors, horizon sensors, magnetometers, and gyros. In addition, the ADCS is often used to control the vehicle during trajectory correction maneuvers and, using accelerometers, to terminate maneuvers when the desired velocity change has been achieved. Actuators are designed to change a spacecraft's attitude and to impart velocity change during trajectory correction maneuvers. Common spacecraft actuators include magnetic torquers, reaction wheels, and thrusters. There are many attitude determination and control architectures and algorithms suitable for use in small spacecraft (6).

Miniaturization of existing technologies is a continuing trend in small spacecraft GNC. While three-axis stabilized, GPS-equipped, 100 kg class spacecraft have been flown for decades, it has only been in the past few years that such technologies have become available for micro- and nano-class spacecraft. Table 5-1 summarizes the current state-of-the-art of performance for GNC subsystems in small spacecraft. Performance greatly depends on spacecraft size with a range of values for nano- to micro-class spacecraft.

The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that Technology Readiness Level (TRL) designations may vary with changes specific to the 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 TRL of the described technology.

Table 5-1: State-of-the-Art GNC Subsystems		
Component	**Performance**	**TRL**
Reaction Wheels	0.00023 – 0.3 Nm peak torque, 0.0005 – 8 N m s storage	7-9
Magnetic Torquers	0.15 A m^2 – 15 A m^2	7-9
Star Trackers	8 arcsec pointing knowledge	7-9
Sun Sensors	0.1° accuracy	7-9
Earth Sensors	0.25° accuracy	7-9
Inertial Sensors	Gyros: 0.15° h^{-1} bias stability, 0.02° h$^{-1/2}$ ARW Accels: 3 µg bias stability, 0.02 (m s^{-1})/h$^{-1/2}$ VRW	7-9
GPS Receivers	1.5 m position accuracy	7-9
Integrated Units	0.002-5° pointing capability	7-9
Atomic Clocks	10 – 150 Frequency Range (MHz)	5-6
Deep Space Navigation	Bands: X, Ka, S, and UHF	7-9
Altimeters	~15 meters altitude, ~3 cm accuracy	7

5.2   State-of-the-Art – GNC Subsystems

5.2.1   Integrated Units

Integrated units combine multiple different attitude and navigation components to provide a simple, single-component solution to a spacecraft's GNC requirements. Typical components included are reaction wheels, magnetometer, magnetic torquers, fine and/or coarse Sun sensors, GPS, and star trackers. The systems often include processors and software with attitude determination and control capabilities that enable common mission profiles such as Sun tracking, inertial pointing, and Earth target tracking. In addition, providers such as Blue Canyon Technologies (BCT) and CubeSpace can provide overall GNC design and simulation with their product to ensure that

Figure 5.1: BCT XACT integrated ADCS unit. Credit: Blue Canyon Technologies.

desired mission objectives can be met within the desired orbit environment and hardware constraints of the integrated unit and spacecraft bus. Using these integrated units can increase overall mission success, as the included ADCS software is also at a high TRL. BCTs' XACT (figure 5.1) flew on the NASA-led MarCO and ASTERIA missions, both of which were 6U platforms, and has also flown on 3U missions (MinXSS was deployed from NanoRacks in February 2016). Table 5-2 describes some of the integrated systems currently available that are associated with a TRL value of 7-9.

Table 5-2: Currently Available Integrated Systems							
Manufacturer	Model	Mass (kg)	Actuators	Sensors	Processor	Pointing Accuracy	Ref
AAC Clyde Space	iADCS-200	0.470	3 reaction wheels 3 magnetic torquers	1 star tracker 1 IMU, Optionally high precision magnetometer and sun sensors	Yes	<1°	(7)
	iADCS-400	1.7	3 reaction wheels 3 magnetorquers	1 star tracker, 1 IMU, Optionally high precision magnetometer and sun sensors	Yes	<1°	(8)
Arcsec	Arcus ADC	0.715	3 reaction wheels 3 magnetic torquers	1 star tracker 3 gyros 6 photodiodes 3 magnetometers	Yes	0.1°	(9)
Berlin Space Technologies	IADCS-100	0.4	3 reaction wheels 3 magnetic torquers	1 star tracker 3 gyros, 1 magnetometer, 1 accelerometer	Yes	<<1 deg	(10)
Blue Canyon Technologies	XACT-15	0.885	3 reaction wheels 3 magnetorquers	1 star tracker 3-axis magnetometer	Yes	0.003/0.007°	(11)
	XACT-50	1.230	3 reaction wheels 3 magnetorquers	1 star tracker 3-axis magnetometer	Yes	0.003/0.007°	(11)
	XACT-100	1.813	3 reaction wheels 3 magnetorquers	1 star tracker 3-axis magnetometer	Yes	0.003/0.007°	(11)
	Flexcore	†	3 – 4 reaction wheels 3 magnetorquers	2 star trackers 3-axis magnetometer	Yes	0.002°	(11)
CubeSpace Satellite Systems	CubeADCS	0.26*	3/4(Pyramid) x reaction wheels, 3 x Magnetorquers distributed or integrated with ADCS core (3U/6U only)	Up to 4 x FSS, 2 x EHS, 2 x STR, 2 x MGTM. All distributed	5Hz	~70 arcsec (3 sigma) †	(12)

*Mass may vary if actuators are integrated inside CubeADCS core.
†Configuration dependent.

5.2.2   Reaction Wheels

Miniaturized reaction wheels can provide small spacecraft with three-axis precision pointing capability. Reaction wheels provide torque and momentum storage along the wheel spin axis, which results in the spacecraft counter-rotating around the spacecraft center of mass due to conservation of angular momentum from the wheel spin direction. For example, the spacecraft deployment process typically results is some tipoff momentum that can be seen in induced spacecraft angular rates. This tipoff momentum can be stored in the reaction wheels, reducing the angular rate of the spacecraft to zero. Reaction wheels must be carefully selected based on several factors including the moment of inertia of the spacecraft, required momentum storage capacity, and mission slew rate requirements. Once a wheel reaches its maximum speed, it becomes saturated and can no longer provide a torque or store momentum in that axis/direction. To prevent this, reaction wheels need to be periodically desaturated using an actuator that provides an external torque, such as thrusters or magnetic torquers (13). By providing an external torque in an opposing direction, the reaction wheel compensates and spins down to keep the spacecraft at its target (14).

While one or two reaction wheels can be used for a momentum-biased spacecraft, a full three-axis controlled spacecraft requires at least three wheels mounted orthogonally. As reaction wheel failure is a common mode of failure in many missions, it is common to include three additional wheels as a hot backup (14). On smaller spacecraft where SWAP requirements prohibit redundant reaction wheels, a four-wheel configuration is often used to provide fault tolerance (15). The multiple reaction wheels are often assembled in a "skewed" or angled configuration such that there exists a cross-coupling of torques with two or more reaction wheels. While this reduces the torque performance in any single axis, it allows a redundant, albeit reduced, torque capability in more than one axis. The result is that should any single reaction wheel fail, one or more reaction wheels are available as a reduced-capability backup option.

Though reaction wheels are typically the primary choice for precision three-axis pointing, other important considerations must be made. For example, they can introduce major attitude disturbances through static and dynamic imbalances, which can generate undesired forces, torques, and moments. While placing reaction wheels near the center-of-gravity of spacecraft can help minimize the force/torque effects, other disturbances are location independent and careful design considerations must be made (14). Reaction wheels are typically not magnetically clean, whether it is in the electro-magnetic controller for spinning the wheels or ferrous materials being used in the wheel rotor itself, so consideration must be made when placing the wheels within the spacecraft bus. Wheels typically should be placed with sufficient separation from magnetometers or other sensitive instrumentation. As magnetic torquers are the primary method of preventing reaction wheel saturation for small spacecraft in Earth orbit, they require an accurate measurement of the Earth's magnetic field to determine torquer commands, making wheel placement especially critical in the spacecraft bus design (17). Another consideration is how the reaction wheel is used in orbit and how precise pointing requirements are. Most reaction wheels have reduced performance around zero-crossings (transition between positive/negative speeds), so some missions choose to bias the momentum (i.e., wheels running at half maximum speed when at zero spacecraft angular rate) (18). This can provide finer control with less jitter, at the expense of reduced overall momentum capacity (effectively half).

Table 5-3 lists a selection of high-heritage miniature reaction wheels. Except for three units, all the reaction wheels listed have spaceflight heritage.

Table 5-3: High Heritage Miniature Reaction Wheels							
Manufacturer	Model	Mass (kg)	Peak Power (W)	Peak Torque (Nm)	Momentum Capacity (Nms)	# Wheels	Radiation Tolerance (krad)
AAC Clyde Space	RW210	0.48	0.8	0.0001	0.006	1	36
	RW400	0.375	15	0.008	0.050	1	36
	Trillian-1	1.5	24	47.1	1.2	1	Unk
Astrofein	RW1 Type A	≤ 0.025	< 0.375 + PWDE	23e-6	5.8e-4	1	/
	RW1 Type B	≤ 0.012	< 0.3 + PWDE	4e-6	1.0e-4	1	/
	RW25	≤ 0.2	< 2.8	0.002	0.03	1	/
	RW35	≤ 0.5	≤ 9	0.005	0.1	1	20
	RW90	≤ 0.9	≤ 16.5	0.015	0.35	1	20
	RW100	≤ 0.8	≤ 25	0.02	0.4	1	20
	RW150	≤ 1.3	≤ 42	0.03	1	1	20
	RWT150	≤ 1.5	≤ 120	0.1	1	1	20
	RW250	≤ 2.75	≤ 100	0.1	4	1	20
	RWT250	≤ 2.75	≤ 200	0.3	4	1	20
Berlin Space Technologies	RWA05	1.700	23.5	0.016	0.5	1	30
	RWP015	0.130	1	0.004	0.015	1	Unk
	RWP050	0.240	1	0.007	0.050	1	Unk
	RWP100	0.330	1	0.007	0.100	1	Unk
	RWP500	0.750	6	0.025	0.500	1	Unk
	RW1	0.950	10	0.07	1.000	1	Unk
	RW4	3.200	10	0.250	4.000	1	Unk
	RW8	4.400	10	0.250	8.000	1	Unk
Comat	RW20	0.180	1	0.002	0.02	1	Up to 20Krad*
	RW40	0.230	1	0.004	0.04	1	Up to 20Krad*
	RW60	0.275	1	0.006	0.06	1	Up to 20Krad*
CubeSpace Satellite Systems	CubeWheel CW0017	0.06	0.85	0.23	0.0017	1/3/4(Pyramid)	24
	CubeWheel CW0057	0.115	2.7	2	0.0057	1/3/4(Pyramid)	24
	CubeWheel CW0162	0.144	7.2	7	0.0162	1/3/4(Pyramid)	24
CubeSpace Satellite Systems	CubeWheel CW0500	0.310	15	16	0.05	1/3/4(Pyramid)	24

Table 5-3: High Heritage Miniature Reaction Wheels							
Manufacturer	**Model**	**Mass (kg)**	**Peak Power (W)**	**Peak Torque (Nm)**	**Momentum Capacity (Nms)**	**# Wheels**	**Radiation Tolerance (krad)**
	CubeWheel CW1200	0.450*	32	20	0.12	1/3/4(Pyramid)	24
	CubeWheel CW2500	0.750*	33	27	0.25	1/3/4(Pyramid)	24
	CubeWheel CW5000	1.084	48	37	0.5	1/3/4(Pyramid)	24
	CubeWheel CW10K	2.1*	50	37	1	1/3/4(Pyramid)	24
	CubeWheel CW40K	2.2*	85	37	4	1/3/4(Pyramid)	24
GomSpace	NanoTorque GSW-600	0.940	0.3	0.0015	0.019	1	Unk
NanoAvionics	RWO	0.137	3.25	0.0032	0.020	1	20
	4RWO	0.665	6	0.0059	0.037	4	20
NewSpace Systems	NRWA-T6	<5	136	0.3	0.00783	1	20
	NRWA-T065	1.55	1.7	0.02	0.00094	1	10
	NRWA-T2	2.8	0.4	0.09	0.00163	1	10
Rocket Lab	RW-0.03	0.185	1.8	0.002	0.040	1	20
	RW-0.003	0.048	Unk	0.001	0.005	1	10
	RW-0.01	0.122	1.05	0.001	0.018	1	20
	RW3-0.06	0.235	23.4	0.020	0.180	1	20
	RW4-0.2	0.6	Unk	0.1	0.2	1	60
	RW4-0.4	0.77	Unk	0.1	0.4	1	60
	RW4-1.0	1.38	43	0.1	1	1	60
Vectronic Aerospace	VRW-A-1	1.90	110	0.090	6.000	1	20
	VRW-B-2	1.00	45	0.020	0.200	1	20
	VRW-C-1	2.3	45	0.020	1.20	1	20
	VRW-D-2	2	65	0.05	2.0	1	20
	VRW-D-6	3	110	0.09	6	1	20

*Printed Circuit Board (PCB) level

5.2.3   Magnetic Torquers

Magnetic torquers provide control torques perpendicular to the local external magnetic field. They are divided into two categories, usually labeled as air coils or torquer bars/rods. Both operate by applying an electric field through a coiled conductor, generating a magnetic dipole through the center of the coil that loops around itself (14). This generated magnetic dipole interacts with the local external magnetic field, generating a torque perpendicular to the dipole and field. The strength of the dipole, and thereby the torque, is governed by the area of the coil and the number of

Figure 5.3: Magnetorquers for micro satellites. Credit: ZARM Technik.

turns in the coil. For torquer bars/rods, a ferrous core is added, which greatly amplifies the strength of the magnetic field without an increase in required power consumption. While these bars/rods typically provide volume and power savings, the ferrous material can hold a residual field even when the device is powered off, which can affect other sensitive instrumentation such as magnetometers. With air coils, however, residual magnetic fields dissipate very rapidly when powered off. Air coils typically need a larger coil radius and/or more windings with a higher power consumption requirement to provide an equivalent magnetic dipole as compared to torquer bars/rods. Many are custom designed specifically for the mission's spacecraft bus, such as being integrated into a components PCB inner layer. As such, not as many providers provide them as COTS components. Table 5-4 lists a selection of high heritage magnetic torquers and figure 5.3 illustrates some of ZARM Technik's product offerings.

As control torques can only be provided in the plane perpendicular to the local magnetic field and generated magnetic dipole, magnetic torquers alone cannot typically provide three-axis stabilization. A spacecraft will typically have three magnetic torquers mounted on orthogonal axes, and usually there is no need for redundant torquers as the electro-mechanical design provides internal redundancy (14). Some research shows that coarse three-axis pointing can be achieved, but this requires multiple orbits and sufficient orbit inclination to provide external magnetic field variability. Typically, magnetic torquers are often used to remove excess momentum from reaction wheels. As the torque generated from magnetic torquers acts externally to the spacecraft, the reaction wheels apply an internal counter torque to compensate and reduce their stored momentum. Magnetic torquers are also used on spinner missions, where they can slowly spin up the spacecraft by applying an external torque over time.

Use of magnetic torquers beyond low-Earth orbit and in interplanetary applications need to be carefully investigated since their successful operation is relying on a significant local external magnetic field. This magnetic field may or may not be available in the location and environment for that mission and additional control methods may be required.

Table 5-4: High Heritage Magnetic Torquers						
Manufacturer	**Model**	**Mass (kg)**	**Power (W)**	**Peak Dipole (A m²)**	**# Axes**	**Radiation Tolerance (krad)**
AAC Clyde Space	MTQ800	0.395	3	15	1	Unk
CubeSpace Satellite Systems	CubeTorquer CR0002	0.0165	0.49	0.2	1	24
	CubeTorquer CR0003	0.023	0.38	0.3	1	24

Table 5-4: High Heritage Magnetic Torquers						
Manufacturer	**Model**	**Mass (kg)**	**Power (W)**	**Peak Dipole (A m²)**	**# Axes**	**Radiation Tolerance (krad)**
CubeSpace Satellite Systems	CubeTorquer CR0004	0.023	0.63	0.4	1	24
	CubeTorquer CR0006	0.031	0.56	0.6	1	24
	CubeTorquer CR0008	0.028	0.56	0.8	1	24
	CubeTorquer CR0010	0.037	0.67	1	1	24
	CubeTorquer CR0012	0.045	0.68	1.2	1	24
	CubeTorquer CR0020	0.054	0.77	2	1	24
GomSpace	Nano Torque GST-600	-	-	0.31–0.34	3	-
	NanoTorque Z-axis Internal	0.106	-	0.139	1	-
ISISPACE	Magnetorquer Board	0.196	1.2	0.20	3	-
MEISEI	Magnetic Torque Actuator for Spacecraft	0.5	1	12	1	-
NanoAvionics	MTQ3X	0.205	0.4	0.30	3	-
	MTQ MP42	0.215	2.5	5	1	
NewSpace Systems	NCTR-M003	0.030	0.25	0.29	1	-
	NCTR-M012	0.053	0.8	1.19	1	-
	NCTR-M016	0.053	1.2	1.6	1	-
Rocket Lab	TQ-40	0.825	-	48.00	1	-
	TQ-15	0.400	-	19.00	1	-
ZARM Technik**	MT0.2-1	0.012-0.014	0.135-0.25	0.2	1	NA*
	MT0.5-1	0.009	0.275	0.5	1	NA*
	MT0.7-1-01	0.035	0.5	0.7	1	NA*
	MT1-1-01	0.065	0.23	1	1	NA*
	MT1.5-1-01	0.097	0.4	1.5	1	NA*
	MT2-1-02	0.1	0.5	2	1	NA*
	MT3-1-D22042701	0.15	0.7	3	1	NA*
	MT4-1	0.15	0.6	4	1	NA*
	MT5-1	0.19-0.3	0.73-0.75	5	1	NA*
	MT5-2	0.31	0.77	5	1	NA*
	MT6-2	0.25-0.3	0.48-1.1	6	1	NA*
	MT7-2	0.4	0.9	7	1	NA*
	MT10-1	0.35-0.4	0.53-0.8	10	1	NA*
	MT10-2	0.37-0.48	0.7-1	10	1	NA*
	MT15-1	0.4-0.55	1.0-1.55	15	1	NA*

* Only EEE parts are connector and wires. Magnetorquer is not sensitive to ionizing radiation.
** ZARM Technik: Over 200 models available with design to mass/power optimization.

5.2.4 Thrusters

Thrusters can generate forces and torques, providing spacecraft with both attitude and translational control capabilities (14). As they are not dependent on external fields or dynamics, they can be used in any orbit. However, they require some type of expendable fuel source, so the lifetime of the thruster can be limited. While not typically employed on SmallSats for attitude control, recent advances in the miniaturization of propulsion systems have resulted in some COTS availability. Thrusters have the advantage of providing large external torques to perform rapid maneuvers, quick reaction wheel desaturation, and spin stabilization. When used for attitude control, these systems can become more complex to prevent induced translational dynamics.

When these propulsion systems are used to provide orbit and/or translational control, attitude control remains necessary to control the direction of thrust. Even with a full three degree-of-freedom thruster system, depending on the thruster configuration and the center-of-gravity of the spacecraft, the thrusters can impart undesired torques that must be mitigated using attitude control systems such as reaction wheels or ACS thrusters.

One critical application of state-of-the-art thrusters is on proximity operation missions, an increasing area of focus for advanced mission concepts. Proximity operations require both high precision translational and attitude control that typically cannot be achieved with other actuators. Further discussion on this is provided in section 5.3.1. Translation and pointing accuracy are determined by minimum impulse bit and control authority by thruster force, with significant improvements recently being made in this field. An in-depth discussion on thrusters for attitude and translational control are described in Chapter 4: *In-Space Propulsion*.

5.2.5 Star Trackers

A star tracker can provide an accurate estimate of the absolute three-axis attitude by comparing a digital image to an onboard star catalog (15). Star trackers identify and track multiple stars and provide three-axis attitude up to several times a second (14). While simplistic in concept, star trackers are among the most expensive small spacecraft components with a significant variance in capabilities between manufactures. To operate, the sensor requires a clear field-of-view of a starfield such that an initial lost in space solution can be found. This initial acquisition depends on the FOV size, magnitude of the stars, and the star trackers internal processing capabilities. Factors such as spacecraft angular rate and external light sources and glare can corrupt and invalidate a solution. Missions dependent on accurate attitude information typically include an IMU propagated Kalman filter to maintain an attitude solution when the star tracker may lose track of an attitude solution. Mission design and star tracker integration into the spacecraft bus also requires careful geometry analysis to ensure that the FOV remains clear for critical mission operations. Missions with propulsion must also worry about contaminating the optics from thruster/exhaust plumes.

While all COTS star tracker suppliers can provide a full state attitude solution, it is their accuracy and capabilities in difficult situations that distinguishes the components and drastically increases their cost. Most modern star trackers can provide knowledge of the sensor's boresight to within a few arcseconds, with reduced knowledge about the roll-axis (14) when the spacecraft is inertially fixed. The accuracy is typically dependent on the optic's FOV, the sensor's resolution, and the processing power dictating the size of the star catalog. However, typically this accuracy can quickly degrade if the spacecraft has some angular rate. For example, cheaper star trackers might be able to track a solution at up to a 0.3 deg/s spacecraft angular rates, whereas more expensive options may track up to 3.0 deg/s. Other factors like stray light resilience, initial acquisition time, and precision also drive-up costs. State-of-the-art in this field revolves around improving the speed and accuracy of the sensors, as well as decreasing their size and mass. These advancements have allowed for new manufactures to enter the field with cheaper options,

allowing missions to get full 3-axis attitude information with a sensor that would otherwise be cost prohibitive. Table 5-5 lists some models suitable for use on small spacecraft. For example, Arcsec's Sagitta Star Tracker was launched on the SIMBA cubesat in 2020.

					Table 5-5: Star Trackers Suitable for Small Spacecraft			
Manufacturer	**Model**	**Mass (kg)**	**Power (W)**	**FOV**	**Cross axis Accuracy (3s)**	**Twist Accuracy (3s)**	**Radiation Tolerance (krad)**
Arcsec	Sagitta	0.275	1.4	25.4°	6	30	20
	Twinkle	0.04	0.6	10.4°	30	180	-
BAE Systems Inc, Space and Mission Systems	CT-2020	3.000	8	-	1.5"	1"	-
Berlin Space Technologies / AAC Clyde Space	ST200	0.040, 0.106*	0.65	22°	30"	200"	11
	ST400	0.250, 0.270*	0.75	15°	15"	150"	11
Blue Canyon Technologies	Standard NST	0.350	1.5	10° x 12°	6"	40"	-
	Extended NST	1.300	1.5	10° x 12°	6"	40"	-
Creare	UST	0.840	-	-	7"	15"	-
CubeSpace Satellite Systems	CubeStar	0.047	0.271	59.4°	0.02°	0.06°	24
Danish Technical University	MicroASC	0.425	1.9	-	2"	-	-
Leonardo	Spacestar	1.600	6	20° x 20°	7.7"	10.6"	-
NanoAvionics	ST-1	0.108	1.2	21° full-cone	8"	50"	20
Redwire Space	Star Tracker	0.475	2.5	14x19	10/27"	51"	75
Rocket Lab	ST-16RT2	0.185	1	8° half-cone	5"	55"	-
Sodern	Auriga-CP	0.225	0.8	30°	10"	70"	35
	Hydra-M	1.4	0.7	22°	3"	30"	50
	Hydra-TC	1.6	9.5	23,6°	3"	30"	50
Solar MEMS Technologies	STNS	0.14	1	12°	40"	70"	20
Space Micro	MIST	0.520	3	14.5°	15"	105"	30
	µSTAR-100M	1.800	5	-	15"	105"	100
	µSTAR-200M	2.100	8-10	-	15"	105"	100
	µSTAR-200H	2.700	10	-	3"	21"	100
	µSTAR-400M	3.300	18	-	15"	105"	100
Terma	T1	0.637 OH	0.8 OH*, 2.5	20° circular	3.0"	21"	55†
	T3	0.45 (DPU) 0.330	2	20° circular	3.2"	22"	35
Vectronic Aerospace	VST-41MN	0.7 - 0.9	2.5	14° x 14°	27"	183"	20
	VST-68M	0.470	3	14° x 14°	7.5"	45"	20

*Mass includes Baffle.
†SEE Immune

5.2.6   Magnetometers

Magnetometers provide a measurement of the local magnetic field, which can be used to provide a real-time estimate of 2-axis attitude information (20). This requires that a well modeled magnetic field model is available to reference the measured magnetic field (14). With a time-history of magnetometer data, algorithms exist (such as Kalman filters) that can provide full state attitude and rate knowledge from magnetometer only measurements. Using

Figure 5.4: NSS Magnetometer. Credit: NewSpace Systems.

other sources of unit vector measurements (i.e., Sun sensor for Sun vector knowledge) can provide real-time 3-axis attitude information by using algorithms such as TRIAD and QUEST. In addition to attitude information, knowing the local magnetic field is required to determine the necessary dipole for magnetic torquers to produce a desired external torque on the spacecraft.

As a magnetometer cannot differentiate between different sources of magnetic fields, placement on the spacecraft bus must be carefully considered. Sources like reaction wheels and magnetic torquer rods can change the local magnetic field, as well as other sources of electromagnetic fields like power systems, cabling harnesses, and even solar cells (14). Spacecraft with critical magnetic field measurement requirements may choose to place the magnetometer on a boom extending from the spacecraft, as magnetic dipole falls off at the cubed reciprocal of distance. In addition to careful spacecraft placement, experience has shown that magnetometers may require on-orbit recalibration, which requires orbit position and preferably attitude data. Table 5-6 provides a summary of some three-axis magnetometers available for small spacecraft, one of which is illustrated in figure 5.4.

Table 5-6: Three-axis Magnetometers for Small Spacecraft						
Manufacturer	**Model**	**Mass (kg)**	**Power (W)**	**Resolution (nT)**	**Orth ogon ality**	**Rad Tolerance (krad)**
AAC Clyde Space	MM200	0.012	0.01	1.18	-	30
	MAG-3	0.100	Voltage Dependent	-	1°	10
CubeSpace Satellite Systems	CubeMag Deployable	0.016	0.23	13	0.6	24
	CubeMag Compact	0.006	0.23	25	0.6	24
GomSpace	NanoSense M315	0.008	-	-	-	-
NewSpace Systems	NMRM-Bn25o485	0.085	0.75	8	1°	10
MEISEI	3-Axis Magnetometer for Small Satellite	0.220	1.5	-	1°	-
ZARM Technik	Analogue High-Rel Fluxgate Magnetometer FGM-A-75	0.33	0.75 W	±75000	1°	50
	Digital AMR Magnetometer AMR-D-100-EFRS485	0.18	0.3 W	±100000	1°	-

5.2.7   Sun Sensors

Sun sensors are used to estimate the direction of the Sun in a spacecraft body frame. Sun direction estimates can be used for attitude estimation, though to obtain a three-axis attitude estimate at least one additional independent source of attitude information is required (e.g., the Earth nadir vector or the direction to a star). Because the Sun is easily identifiable and extremely bright, Sun sensors are often used for fault detection and recovery. However, care must be taken to ensure the Moon or Earth's albedo is not inadvertently perturbing the measurement. Even glint off nearby spacecraft components can corrupt measurements for some types of Sun sensors (14). One method to limit albedo effects is to exclude measurements below a certain brightness threshold (albedo is typically measured as some fraction of solar maximum), but care must be taken with this method as it may limit the effective field-of-view of the sensor (cutoff threshold corresponds to the solar angle yielding the same cosine loss). Commercial examples of small spacecraft Sun sensors are described in table 5-7.

Figure 5.5: Redwire coarse Sun sensor detector (cosine type). Credit: Redwire Space.

There are several types of Sun sensors which operate on different principles.

Cosine detectors are photocells. Their output is the current generated by the cell, which is (roughly) proportional to the cosine of the angle between the sensor boresight and the Sun. Typically several cosine detectors (pointing in different directions) are used on a spacecraft for full sky coverage. These sensors are the most susceptible to albedo effects. Cosine detectors (e.g., figure 5.5) are inexpensive, low-mass, simple and reliable devices, but their accuracy is typically limited to a few degrees, and they do require analog-to-digital converters.

Quadrant detectors. Quadrant sun sensors typically operate by shining sunlight through a square window onto a 2 x 2 array of photodiodes. The current generated by each photodiode is a function of the direction of the Sun relative to the sensor boresight. The measured currents from all four cells are then combined mathematically to produce the angles to the Sun. While more accurate than Cosine detectors, they have a similar sensitivity to albedo effects.

Digital Sun Sensor. The Sun illuminates a narrow slit with a number of photodiodes located behind a geometric coded bit mask. Depending on the angle to the Sun, the photodiodes will be illuminated as per the geometric pattern resulting in correspondingly different photocurrents which are then amplified and thresholded against an average value. Given the known slit geometries, this digital bit output can then be converted to a sun angle.

Sun Camera. Some Sun sensors are built as a small camera imaging the Sun. Since the Sun is so bright, the optics include elements to decrease the thoughput. A computer will identify the image of the Sun and calculate the centroid. Sun sensors can be made very accurate this way and typically have built in albedo rejection. Sometimes, multiple apertures are included to increase accuracy.

Manufacturer	Model	Sensor Type	Mass (kg)	Peak Power (W)	Analog or Digital	FOV	Accuracy (3s)	# Measurement Angles	Radiation Tolerance (krad)	TRL
AAC Clyde Space	SS200	-	0.003	0.04	Digital	110°	<1°	-	>36	7-9
Berlin Space Technologies	FSSA-110	Fine Sun Sensor	With 3x sensor: 0.102	0.185	Analog	114°	5°	?	>20	7-9
Bradford Space	CoSS	Cosine	0.024	0	Analog	160° full cone	3°	1	40000	7-9
	CoSS-R	Cosine	0.015	0	Analog	180° full cone	3°	1	120000	7-9
	CSS	Cosine	0.215	0	Analog	180°x180°	1.5°	2	70000	7-9
	FSS	Quadrant	0.375	0.25	Analog	128° x 128°	0.3°	2	100	7-9
	Mini-FSS	Quadrant	0.050	0	Analog	128° x 128°	0.2°	2	20000	7-9
CubeSpace Satellite Systems	CubeSense	Camera	0.015	0.174	Digital	0.2°	2	24	CubeSense	7-9
GomSpace	NanoSense FSS	Quadrant	0.002	-	Digital	45°, 60°	±0.5°, ±2°	2	-	-
Lens R&D	BiSon64-ET	Quadrant	0.023	0	Analog	±58° per axis	0.5°	2	9200	7-9
	BiSon64-ET-B	Quadrant	0.033	0	Analog	±58° per axis	0.5°	2	9200	7-9
	MAUS	Quadrant	0.014	0	Analog	±57° per axis	0.5°	2	9200	7-9
Needronix	Eagle	Fine Sun Sensor (±55°)	0.003	<0.007	Digital	110°	< ±0.15°	2 plus Irradiation and Gyro	>20	7-9
	Eagle Plus	Fine Sun Sensor (±55°)	0.005	<0.015	Digital	110°	< ±0.1°	2 plus Irradiation and Gyro	>20	5-6
	Eagle Point	Fine Sun Pointing (±5.5°)	0.006	<0.015	Digital	11°	< ±0.01°	2 plus Irradiation and Gyro	>20	5-6

Table 5-7: Small Spacecraft Sun Sensors

Table 5-7: Small Spacecraft Sun Sensors										
Manufacturer	**Model**	**Sensor Type**	**Mass (kg)**	**Peak Power (W)**	**Analog or Digital**	**FOV**	**Accuracy (3s)**	**# Measurement Angles**	**Radiation Tolerance (krad)**	**TRL**
	Swan	Coarse Sun Sensor Pyramid	0.070	0	Analog	170° (~ 2π Steradian)	< ±1° (Region I.)	2	>1000	5-6
NewSpace Systems	NFSS-411	-	0.035	0.150	Digital	140°	0.1°	--	20	7-9
	NCSS-SA05	-	0.005	0.05	Analog	114°	0.5°	--	-	7-9
Redwire Space	Coarse Analog Sun Sensor	Coarse Analog Sun Sensor	0.045	0	Analog	±40°	±1°	1	>100	7-9
	Coarse Sun Sensor (Cosine Type)	Coarse Sun Sensor (Cosine Type)	0.010	0	Analog	~ Cosine, Conical Symmetry	±2° to ±5°	Configuration dependent	>100	7-9
	Coarse Sun Sensor Pyramid	Coarse Sun Sensor Pyramid	0.13	0	Analog	2π steradian+	±1° to ±3°	2	>100	7-9
	Digital Sun Sensor (±32°)	Digital Sun Sensor (±32°)	Sensor 0.3 kg Electronics ~1	1	Digital	±32° x ±32° (each sensor)	±0.125°	2	100	7-9
	Digital Sun Sensor (±64°)	Digital Sun Sensor (±64°)	Sensor0.25 Electronics 0.29 - 1.1	0.5	Digital	128° X 128° (each sensor) Note: 4π steradians achieved with 5 sensors	±0.25°	2	100	7-9

Table 5-7: Small Spacecraft Sun Sensors										
Manufacturer	**Model**	**Sensor Type**	**Mass (kg)**	**Peak Power (W)**	**Analog or Digital**	**FOV**	**Accuracy (3s)**	**# Measurem ent Angles**	**Radiation Tolerance (krad)**	**T R L**
	Fine Pointing Sun Sensor	Fine Pointing Sun Sensor	Sensor .95 Electronics 1.08	< 3	Digital	±4.25° x ±4.25° (Typical)	Better than ±0.01°	2	100	7-9
	Fine Spinning Sun Sensor (±64°)	Fine Spinning Sun Sensor (±64°)	Sensor 0.109 Electronics 0.475 – 0.725	0.5	Analog and Digital	±64° Fan Shaped (each sensor)	±0.1°	1 plus Sun Pulse	100	7-9
	Micro Sun Sensor	Micro Sun Sensor	< 0.002	< 0.02	Analog	± 85° Minimum	±5°	2	Approx. 10	5-6
	Miniature Spinning Sun Sensor (±87.5°)	Miniature Spinning Sun Sensor (±87.5°)	< 0.25	0.5	Digital	±87.5° (From normal to spin axis)	±0.1°	1 plus Sun Pulse	100	7-9
	Fine Sun Sensor (±50°)	Fine Sun Sensor (±50°)	-	-	Digital	100 X 100 Each Sensor	±0.01° TO ±0.05°	2	100, 150, or 300	7-9
Solar MEMS Technologies	nanoSSOC-A60	Orthogonal	0.004	0.007	Analog	±60° per axis	0.5°	2	100	7-9
	nanoSSOC-D60	Orthogonal	0.007	0.076	Digital	±60° per axis	0.5°	2	30	7-9
	SSOC-A60	Orthogonal	0.025	0.01	Analog	±60° per axis	0.5°	2	100	7-9
	SSOC-D60	Orthogonal	0.035	0.315	Digital	±60° per axis	0.5°	2	30	7-9
	ACSS	Quadrant & Redundant	0.035	0.072	Analog	±60° per axis	0.5°	2	200	7-9
Space Micro	CSS-01, CSS-02	Cosine	0.010	0	Analog	120° full cone	5°	1	100	7-9
	MSS-01	Quadrant	0.036	0	Analog	48° full cone	1°	2	100	7-9

5.2.8   Horizon Sensors

Horizon sensors can be simple infrared horizon crossing indicators (HCI), or more advanced thermopile sensors that can detect temperature differences between the poles and equator. The sensors are typically either static or scanning, but by characterizing horizon crossings over a series of measurements, the sensors can provide an accurate nadir vector, which can then be used for attitude determination and/or attitude control guidance (14). For terrestrial applications, these sensors are referred to as Earth Sensors, but can be used for other planets. Examples of such technologies are described in table 5-8 and illustrated in figure 5.6.

Figure 5.6: MAI-SES. Credit: Redwire Space.

In addition to the commercially-available sensors listed in table 5-8, there has been some recent academic interest in horizon sensors for CubeSats with promising results (4)(21)(22).

Table 5-8: Commercially Available Horizon Sensors								
Manufacturer	Model	Sensor Type	Mass (kg)	Peak Power (W)	Analog or Digital	Accuracy	# Measurement Angles	Rad Tolerance (krad)
CubeSpace Satellite Systems	CubeSense Earth	Infrared camera	0.018	0.28	Digital	1°	-	24
Servo	Mini Digital HCI	Pyroelectric	0.050	Voltage Dependent	Digital	0.75°	Unk	Unk
	RH 310 HCI	Pyroelectric	1.5	1	Unk	0.015°	Unk	20
Solar MEMS Technologies	HSNS	Infrared	0.120	0.150	Digital	1°	2	30

5.2.9   Inertial Sensing

Inertial sensors include gyroscopes for measuring angular change and accelerometers for measuring velocity change. They are packaged in different ways that range from single-axis devices (i.e., a single gyroscope or accelerometer), to packages which include 3 orthogonal axes of gyroscopes (Inertial Reference Unit (IRU)) to units containing 3 orthogonal gyros and 3 orthogonal accelerometers (Inertial Measurement Unit (IMU)). These sensors are frequently used to propagate the vehicle state between measurement updates of a non-inertial sensor. For example, star trackers typically provide attitude updates at a few Hertz. If the control system requires accurate knowledge between star tracker updates, then an IMU may be used for attitude propagation between star tracker updates.

Gyroscope technologies typically used in modern small spacecraft are fiber optic gyros (FOGs) and MEMS gyros, with FOGs usually offering superior performance at a mass and cost penalty (24). While MEMs are smaller and can provide sufficient performance, they are more susceptible to radiation and single event upsets, with radiation hardened models only recently becoming

available (14). Other gyroscope types exist (e.g., resonator gyros, ring laser gyros), but these are not common in the SmallSat/CubeSat world due to size, weight, and power (SWaP) and cost considerations.

Gyro behavior is a complex topic (25) and gyro performance is typically characterized by a multitude of parameters. Table 5-9 only includes bias stability and angle random walk for gyros, and bias stability and velocity random walk for accelerometers, as these are often the driving performance parameters. That said, when selecting inertial sensors, it is important to consider other factors such as dynamic range, output resolution, bias, sample rate, etc. Factors like turn-on bias and bias drift require onboard estimation of the bias so that the sensor can be used in attitude determination and control systems.

Table 5-9: Gyros Available for Small Spacecraft													
						Gyros				**Accelerometers**			
						Bias Stability		**ARW**			**Bias Stability**		**VRW**
Manufacturer	**Model**	**Sensor Type**	**Technology**	**Mass (kg)**	**Power (W)**	**# Axes**	**(°/hr)**	**stat**	**(°/rt(hr))**	**# Axes**	**(µg)**	**stat**	**(m/sec)/rt(hr)**
Emcore	QRS11	Gyro	MEMS	≤0.06	0.8	1	6	Typical	-	-	-	-	-
	QRS28	Gyro	MEMS	≤0.025	0.5	2	-	-	-	-	-	-	-
Honeywell	MIMU	IMU	RLG	4	34	3	0.05	-	0.01	-	100	-	-
	HG1700	IMU	RLG	0.9	5.000	3	1.000	1σ	0.125	3	1000	1σ	0.65
	HG4934SRS (PN: 68904934-BA60)	IRU	MEMS	0.145	<5.500 peak	3	< 3.0	3s	<0.20 max	None, IRU only	None, IRU only	None, IRU only	None, IRU only
L3	CIRUS	Gyros	FOG	15.400	40.000	3	0.000	1σ	0.100	0	-	-	-
NewSpace Systems	NSGY-001	IRU	Image-based rotation estimate	0.055	0.200	3	-		-	0	-	-	-
Northrop Grumman	LN-200S	IMU	FOG, SiAc	0.748	12	3	1.000	1σ	0.070	3	300	1σ	-
NovAtel	OEM-IMU-STIM300	IMU	MEMS	0.055	1.50	3	0.500	--	0.150	3	50	-	0.060
Safran	STIM202	IRU	MEMS	0.055	1.500	3	0.400	--	0.170	0	-	-	-
	STIM210	IRU	MEMS	0.052	1.500	3	0.300	--	0.150	0	-	-	-
	STIM300	IMU	MEMS	0.055	2.000	3	0.300	--	0.150	3	50	--	0.07

Table 5-9: Gyros Available for Small Spacecraft													
						Gyros				Accelerometers			
							Bias Stability		ARW		Bias Stability		VRW
Manufacturer	Model	Sensor Type	Technology	Mass (kg)	Power (W)	# Axes	(°/hr)	stat	(°/rt(hr))	# Axes	(µg)	stat	(m/sec)/rt(hr)
	STIM318	IMU	MEMS	0.057	2.500	3	0.300	--	0.150	3	3	--	0.015
	STIM320	IMU	MEMS	0.057	2.500	3	0.300	--	0.100	3	3	--	0.015
	STIM277H	IRU	MEMS	0.052	1.500	3	0.300	--	0.150	0	-	--	-
	STIM377H	IMU	MEMS	0.055	2.000	3	0.300	--	0.150	3	50	--	0.07
Silicon Sensing Systems	CRH03	Gyro	MEMS	0.42	0.2W	1	CRH03-010 – 0.03 CRH03-025 – 0.04 CRH03-100 – 0.04 CRH03-200 – 0.05 CRH03-400 – 0.1		CRH03-010 – 0.005 CRH03-025 – 0.006 CRH03-100 – 0.006 CRH03-200 – 0.008 CRH03-400 – 0.010	0	-	-	-
	CRH03 (OEM)	Gyro	MEMS	0.18	0.2W	1	CRH03-010 – 0.03 CRH03-025 – 0.04 CRH03-100 – 0.04 CRH03-200 – 0.05		CRH03-010 – 0.005 CRH03-025 – 0.006 CRH03-100 – 0.006	0	-	-	-

Manufacturer	Model	Sensor Type	Technology	Mass (kg)	Power (W)	Gyros				Accelerometers			
							Bias Stability		ARW		Bias Stability		VRW
						# Axes	(°/hr)	stat	(°/rt(hr))	# Axes	(µg)	stat	(m/sec)/rt(hr)
							CRH03-400 – 0.1		CRH03-200 – 0.008 CRH03-400 – 0.010				
	RPU30	Gyro	MEMS	1.35	<0.8W	3	0.06		0.006	0	-	-	-
	DMU41	9 DoF IMU	MEMS	<2	<1.5W	3	0.1		0.015	3	15	-	0.05
	CAS	Acc	MEMS	0.004	-	0	-		-	2	CAS2X1S - 7.5 CAS2X2S - 7.5 CAS2X3S - 7.5 CAS2X4S - 25 CAS2X5S - 75		CAS2X1S - TBC CAS2X2S - TBC CAS2X3S - TBC CAS2X4S - TBC CAS2X5S - TBC
VectorNav	VN-100*	IMU + magnetometers +barometer	MEMS	0.015	0.220	3	10.000	max	0.210	3	40	max	0.082
	VN-110*	IMU + magnetometers	MEMS	0.125	2.500	3	1.000	max	0.0833	3	10	max	0.024

*Small form-factor versions of these products available.
-- Represents unknown data

5.2.10 GPS Receivers

For low-Earth orbit spacecraft, GPS receivers are now the primary method for performing orbit determination, replacing ground-based tracking methods. Onboard GPS receivers are now considered a mature technology for small spacecraft, and some examples are described in table 5-10. There are also next-generation chip-size COTS GPS solutions, for example the NovaTel OEM 719 board has replaced the ubiquitous OEMV1.

By measuring the time between receiving signals between multiple GPS spacecraft, the GPS receiver can determine its position to a high degree of accuracy (14). A minimum of four spacecraft must be visible to the GPS receiver, where three provide position data and a fourth provides timing to correct the GPS receivers clock bias. With multiple GPS antennas, a spacecrafts GPS receiver can even be used for attitude determination by using phase differences in GPS signals, though the accuracy of this method is limited and prone to error sources. GPS accuracy is limited by propagation variance through the exosphere and the underlying precision of the civilian use C/A code (26). GPS units are controlled under the Export Administration Regulations (EAR) and must be licensed to remove Coordinating Committee for Multilateral Export Control (COCOM) limits (27).

Although the usability of GPS is limited to LEO missions, past experiments have demonstrated the ability of using a weak GPS signal at GSO, and potentially soon to cislunar distances (29)(28). Development and testing in this fast-growing area of research and development may soon make onboard GPS receivers more commonly available.

Table 5-10: GPS Receivers for Small Spacecraft					
Manufacturer	**Model**	**Mass (kg)**	**Power (W)**	**Accuracy (m)**	**Radiation Tolerance (krad)**
AAC Clyde Space	GNSS-701	0.16	Unk	<5	10
APL	Frontier Radio Lite	0.4	1.4	15	20
Aerospacelab	GNSS-VSP	0.394	2.4	1.5 (RMS)	*
General Dynamics	Explorer	1.2	8	15	-
	Viceroy-4	1.1	8	15	-
GomSpace		0.285	3	< 10	<20	0.285
NovaTel	OEM 719	0.031	0.9	-	*
SkyFox Labs	piNAV-NG	0.024	0.124	10	30
Spacemanic	Celeste_gnss_rx	0.025	~0.1	1.5	40
Surrey Satellite Technology	SGR-Ligo	0.09	0.5	5	5

*Heavy ions (61 MeV) No destructive event Except OEM719 receiver which is protected by smart current limit

5.2.11 Deep Space Navigation

In deep space, navigation is performed using radio transponders in conjunction with the Deep Space Network (DSN). As of 2020, the only deep space transponder with flight heritage suitable for small spacecraft was the JPL-designed and General Dynamics-manufactured Small Deep Space Transponder (SDST). JPL has also designed IRIS V2, which is a deep space transponder that is more suitable for the CubeSat form factor. Table 5-11 details these two radios, and the SDST is illustrated in figure 5.7. IRIS V2, derived from the Low Mass Radio Science Transponder (LMRST), has flown on the MarCO CubeSats in 2018, LICIACube that performed an asteroid flyby in September 2022, 12U lunar CAPSTONE spacecraft that entered a lunar orbit November 13, 2022, and was on six Artemis 1 secondary CubeSat payloads (Lunar Flashlight, LunaH-Map, ArgoMoon, CubeSat for Solar Particles, Biosentinel, and NEA Scout). It is also scheduled to fly on INSPIRE (30).

Figure 5.7: General Dynamics SDST. Credit: General Dynamics.

Table 5-11: Deep Space Transponders for Small Spacecraft					
Manufacturer	**Model**	**Mass (kg)**	**Rx Power (W)**	**Bands**	**Radiation Tolerance (krad)**
General Dynamics	SDST	3.2	12.5	X, Ka	50
Space Dynamics Laboratory	IRIS V2.1	1.1	10.3	X, Ka, S, or UHF	25
	Iris Radio V3	0.8	10	Simultaneous Multi-band: X, Ka, S	25

5.2.12 Atomic Clocks

Atomic clocks have been used on larger spacecraft in low-Earth orbit for several years now, however integrating them on small spacecraft is relatively new. Table 5-12 provides examples of commercially available atomic clocks and oscillators for SmallSats. The conventional method for spacecraft navigation is a two-way tracking system of ground-based antennas and atomic clocks. The time difference from a ground station sending a signal and the spacecraft receiving the response can be used to determine the spacecraft's location, velocity, and (using multiple signals) the flight path. This is not a very efficient process, as the spacecraft must wait for navigation commands from the ground station instead of making real-time decisions, and the ground station can only track one spacecraft at a time, as it must wait for the spacecraft to return a signal (31). In deep space navigation, the distances are much greater from the ground station to spacecraft, and the accuracy of the radio signals needs to be measured within a few nanoseconds.

More small spacecraft designers are developing their own version of atomic clocks and oscillators that are stable and properly synchronized for use in space. They are designed to fit small spacecraft, for missions that are power- and volume-limited or require multiple radios.

Table 5-12: Atomic Clocks and Oscillators for Small Spacecraft

Manufacturer	Model	Dimensions (mm)	Mass (kg)	Power (W)	Frequency Range (MHz)	Rad Tolerance (krad)	TRL
AccuBeat	Ultra Stable Oscillator	131 x 120 x 105	2	6.5	57.51852	50	7-9
Bliley Technologies	Iris Series 1"x1" OCXO for LEO	19 x 11 x 19	0.016	1.5	10 -100	39	7-9
	Aether Series TCVCXO for LEO	21 x 14 x 8	-	0.056	10 - 150	37	-
Microsemi	Space Chip Scale Atomic Clock (CSAC)	41 x 36 x 12	0.035	0.12	10	20	5-6
Safran Timing Technologies SA	MO	44 x 54 x 57	0.22	3.5 Nom 5.5 Max	10	100	-
	Space Qualified mRO-50	51 x 51 x 20	0.080	0.4 Nom	10	25 Min	-
	miniRAFS	108 x 53 x 68	0.45	< 12 Max	60 and 10	--	-
	LNMO	50 x 50 x 30	0.1	1.5 Nom 2.5 Max	5 – 40	100	-

- Represents unknown data

5.2.13 LiDAR

Light Detection and Ranging (LiDAR) is new type of sensor that is emerging. The technology has matured in terrestrial applications (such as automotive applications) over the last decade and is used in larger spacecraft that are capable of proximity operations, like Orion. This sensor type has applications for small spacecraft altimetry and relative navigation (e.g., a Mars helicopter, rendezvous and docking, and formation flying). Table 5-13 lists examples of flown LiDARs.

Table 5-13: Lidar for Small Spacecraft

Manufacturer	Model	Mass (kg)	Power (W)	Max Range (m)	Radiation Tolerance (krad)
ASC	GSFL-4K (3D)	3	30	>1 km in altimeter mode	-
Garmin	Lidar Lite V3	0.022	0.7	40	-

- Represents unknown data

5.3   On the Horizon

In general, technological progress in guidance, navigation, and control is advancing quickly in automotive research areas but is lagging slightly in the aerospace industry. Given the high maturity of existing GNC components, future developments in GNC are mostly focused on incremental or evolutionary improvements, such as decreases in mass and power, and increases in longevity and/or accuracy. This is especially true for GNC components designed for deep space missions that have only very recently been considered for small spacecraft. However, in a collaborative effort between the Swiss Federal Institute of Technology and Celeroton, there is progress being made on a high-speed magnetically levitated reaction wheel for small satellites (figure 5.8). The idea is to eliminate mechanical wear and stiction by using magnetic bearings rather than ball bearings. The reaction wheel implements a dual hetero/homopolar, slotless, self-bearing, permanent-magnet synchronous motor (PMSM). The fully active, Lorentz-type magnetic

Figure 5.8: High-speed magnetically levitated reaction wheel. Credit: Celeroton AG.

bearing consists of a heteropolar self-bearing motor that applies motor torque and radial forces on one side of the rotor's axis, and a homopolar machine that exerts axial and radial forces to allow active control of all six degrees of freedom. It can store 0.01 Nm of momentum at a maximum of 30,000 rpm, applying a maximum torque of 0.01 Nm (32).

Several projects funded via NASA's Small Spacecraft Technology (SST) program through the University SmallSat Technology Partnerships (USTP) initiative have begun advancing GNC systems. Listed below in table 5-14 are projects that focused on GNC advancement, and further information can be found at the USTP website:

https://www.nasa.gov/smallspacecraft/university-smallsat-technology-partnership-initiative/

Each presentation is from the USTP Technology Exposition that was held in May 2021 and June 2022.

Table 5-14: USTP Initiative GNC Projects			
Project	**University**	**Current Status**	**Reference**
On-Orbit Demonstration of Surface Feature-Based Navigation and Timing	University of Texas, Austin	Still in development	USTP Technology Expo presentation
Autonomous Nanosatellite Swarming (ANS) using Radio Frequency and Optical Navigation	Stanford University	Onboard Starling mission (Launched in 2023)	USTP Technology Expo presentation
Distributed multi-GNSS Timing and Localization (DiGiTaL)	Stanford University	Leveraged technology used in Starling mission	USTP Technology Expo presentation
Mems Reaction Control and Maneuvering for Picosat beyond LEO	Purdue University	Awarded a suborbital flight test through NASA's Flight Opportunities program	(29)
A Small Satellite Lunar Communications and Navigation System	University of Boulder, Colorado	Still in development	USTP Technology Expo presentation

A high-precision continuous-time PNT compact module for the LunaNet small spacecraft	University of California, Los Angeles	Still in development	USTP Technology Expo presentation

5.3.1 Formation Flying and Rendezvous and Proximity Operations

Other GNC advances involve research on SmallSats performing precision formation flying (PFF) and on-orbit rendezvous and proximity operations (RPO). Many research papers have discussed ways to accomplish this, and previous extravehicular free flyers have demonstrated this innovative capability in the past few decades. To enable PFF and RPO missions, spacecraft require a suite of sensors to determine relative positions and attitudes to operate the navigation filters, complex guidance laws to maintain formations and prevent on orbit conjunctions, as well as advanced methods and/or actuators to control the relative states of the spacecraft. While some COTS sensors and actuators exist specifically for enabling PFF and RPO missions, current research and missions are using novel hardware and methods to achieve these objectives using existing components.

To enable relative guidance and control, it is important to first measure the relative positions and velocity, as well as attitude, of the spacecraft. Relative navigation can be accomplished by hardware specifically designed for the task, such as LiDAR, or repurposed hardware. For example, the PY4 mission has been able to demonstrate on-orbit range measurements between two spacecraft using their S-band radio modules with the addition of a GPS on the anchor spacecraft, and using a batch Kalman filter to determine the relative position vector of a target spacecraft to an accuracy of approximately 4 meters (34). The MR/MRS SAT mission seeks to use two vision-based cameras to perform stereo imaging to determine the relative position and velocity of an uncooperative space object without a priori knowledge to conduct its proximity operations (35). Another mission using vision-based systems was Seeker, a 3U CubeSat that was deployed in September 2019 and designed to demonstrate safe operations around a target spacecraft with core inspection capabilities. While Seeker was unable to perform its underlining goal, there were still several benefits for improving future missions (36). The CubeSat Proximity Operations Demonstration (CPOD) mission used an optics module with four imagers for relative ranging and target attitude determination but required priori knowledge and markers on the tracked spacecraft (37). More recently, the NASA Starling mission, launched in 2023, used an experimental onboard vision-based sensor payload called Starling Formation-flying Optical eXperiment (StarFOX) to provide angles-only relative navigation of an object without a priori knowledge, demonstrating on orbit relative position knowledge with only 0.5% error relative to range using one or multiple observers (38). These missions illustrate the complexity of the navigation problem, especially as much of it must be done autonomously without ground-based commanding to achieve the objectives of PFF and RPO missions.

Once the navigation solution is complete, guidance laws are used to determine path planning to achieve PFF and RPO objectives, as well as avoiding conjunction of objects when they are in near proximity. While this field is primarily research based and dependent on algorithms instead of COTS hardware, there is some development in packaged software suites to enable PFF and RPO type missions. The Autonomous Navigation, Guidance, and Control Software (autoNGC) suite is being developed by NASA Goddard Space Flight Center to enable autonomous operations when ground communications are limited or unavailable, a critical need for cis-lunar and deep space missions (39). Lockheed Martin, in collaboration with government and industry partners, is developing the 12U LINUSS platform that will eventually offer spacecraft upgrade and servicing capabilities, a service heavily dependent on PFF and RPO technologies (40). The

mission tested Lockheed Martin's Horizon Command & Control and Compass Mission Planning Software, which seeks to provide a software solution for various mission operations including PFF (41). It is expected that this field will grow with the continued increased interest in PFF and RPO missions and that demand for software suites with proven flight heritage will become more common.

While there have been multiple successful demonstrations of on-orbit relative navigation for missions conducting PFF and RPO objectives, many of these missions experienced difficulties with the control systems and their actuators. The Seeker mission experienced multiple failures with the thruster system likely related to a FPGA controller failure, resulting in the inability to thrust in multiple axes, causing the mission to not meet its minimum objective (36). CPOD is one of the more recent CubeSat missions to attempt the characterization of low-power proximity operations technologies, however it was only able to demonstrate limited RPO and was unable to complete the docking maneuvers as planned before its mission ended June 2023. The CPOD mission experienced partial solar panel failures resulting in lower power margins that limited mission operations, and experienced a propulsion system anomaly that was most likely related to a plenum leak. Even with these failures, the mission was able to demonstrate both far- and near-field rendezvous and ingress maneuvers, achieving most of the mission objectives. The spacecraft ran out of fuel completing some of these objectives, making it not possible to exercise the docking procedure which sought to use a magnetic actuator to dock the two spacecraft. The Lockheed Martin LINUSS mission also experienced a plenum leak and required continuous ground interventions by the development team to overcome the continual challenges experienced (40). The Starling swarm, which ended its initial mission in May 2024, experienced a leak in the propellant system and initially a sticky refill valve, though both anomalies were able to be resolved through ground operations (43). The mission was ultimately able to resume its in-train formation using thrusters to lower swarm member's altitude and let differential drag establish required separations, allowing the mission to continue conducting various experiments. Other missions had success using non-propulsive control techniques like drag separation for formation maintenance. The PY4 mission successfully used drag separation for both in-track and cross-track maneuvers by controlling individual member's attitude to create the requisite drag differential (44). While such missions are less prone to failure by not relying on a propulsion system, they are typically limited to far-field rendezvous and cannot achieve PFF or near-field RPO objectives like docking.

The complexity of PFF and RPO type missions for SmallSats continues to challenge the hardware, software, and operation design for these missions, however these challenges will continue to be overcome as more industry, government, and university entities become involved. With this field expanding, the benefits of PFF and RPO type missions will soon be realized for the SmallSat platform.

5.4 Summary

Conventional small spacecraft GNC technology is a mature area, with many high TRL components previously flown around Earth offered by several different vendors. These GNC techniques are generally semi/non-autonomous as on-board observations are collected with the assistance of ground-based intervention. As the interest for deep space exploration with small spacecraft grows, semi-to-fully autonomous navigation methods must advance. It is likely that future deep space navigation will rely solely on fully autonomous GNC methods that require zero ground-based intervention to collect/provide navigation data. This is a desirable capability as the spacecraft's dependence on Earth-based tracking resources (such as DSN) is reduced and the demand for navigation accuracy increases at large distances from Earth. However, current methods advancing deep space navigation involve both ground- and space-based tracking in

conjunction with optical navigation techniques. To support this maturity, the small spacecraft industry has seen a spike in position, navigation, and timing (PNT) technology progression in inertial sensors and atomic clocks, and magnetic navigation for near-Earth environments.

The rising popularity of SmallSats in general, and CubeSats in particular, means there is a high demand for components, and engineers are often faced with prohibitive prices. The Space Systems Design Studio at Cornell University is tackling this issue for GNC with their PAN nanosatellites. A paper by Choueiri et al. outlines an inexpensive and easy-to-assemble solution for keeping the ADCS system below $2,500 (45). Lowering the cost of components holds exciting implications for the future and will likely lead to a burgeoning of the SmallSat industry.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email so someone may contact you further.

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Glossary

(ABS)	Acrylonitrile Butadiene Styrene
(ACS3)	Advanced Composite Solar Sail System
(AE)	Aerospace Corporation Electron
(AM)	Additive manufacturing
(AMODS)	Autonomous On-orbit Diagnostic System
(AP)	Aerospace Corporation Proton
(CAM)	Computer Aided Manufacturing
(CFRP)	Carbon Fiber Reinforced Polymers
(CNC)	Computerized Numerical Control
(COBRA)	Compact On-Board Robotic Articulator
(COTS)	Commercial-off-the-shelf
(CSLI)	CubeSat Launch Initiative
(CTD)	Composite Technology Deployment
(CTE)	Coefficient of Thermal Expansion
(DCB)	Deployable Composite Boom
(DDD)	Displacement Damage Dose
(DLP)	Digital Light Projection
(DOF)	Degrees of Freedom
(EEE)	Electrical, Electronic and Electro-mechanical
(EELV)	Evolved Expendable Launch Vehicle
(ESD)	Electrostatic Discharge
(ESPA)	EELV Secondary Payload Adapter
(FDM)	Fused Deposition Modeling
(FFF)	Fused Filament Fabrication
(FPGAs)	Field Programmable Gate Arrays
(FST)	Flame, Smoke, and Toxicity
(GCD)	Game Changing Development
(GEVS)	General Environmental Verification Standard
(HDT)	Heat Deflection Temperature
(ISS)	International Space Station
(MOSFETs)	Metal Oxide Semiconductor Field Effect Transistors
(PAEK)	Polyaryletherketone
(PC)	Polycarbonate

(PCB)   Printed Circuit Board

(PEEK)   Polyetheretherketone

(PEI)   Polyetherimide

(PEKK)   Polyetherketoneketone

(PLA)   Polylactic Acid

(PLEO)   Polar Low-Earth Orbit

(PSC)   Planetary Systems Corporation

(RECS)   Robotic Experimental Construction Satellite

(ROC)   Roll Out Composite

(SADA)   Solar Array Drive Actuator

(SEUs)   Single Event Upsets

(SLA)   Stereolithography

(SLS)   Selective Laser Sintering

(SPEs)   Solar Particle Events

(STELOC)   Stable Tubular Extendable Lock-Out Composite

(TID)   Total Ionizing Dose

(TRAC)   Triangle Rollable and Collapsible

(TRL)   Technology Readiness Level

(ULA)   United Launch Alliance

6.0 Structure, Mechanisms, and Materials

6.1 Introduction

Material selection is of primary importance when considering small spacecraft structures. Requirements for both physical properties (density, thermal expansion, and radiation resistance) and mechanical properties (modulus, strength, and toughness) must be satisfied. The manufacture of a typical structure involves both metallic and non-metallic materials, each offering advantages and disadvantages. Metals tend to be more homogeneous and isotropic, meaning properties are similar at every point and in every direction. Non-metals, such as composites, are inhomogeneous and anisotropic by design, meaning properties can be tailored to directional loads. Recently, resin or photopolymer-based AM has advanced sufficiently to create isotropic parts. In general, the choice of structural materials is governed by the operating environment of the spacecraft, while ensuring adequate margin for launch and operational loading. Deliberations must include more specific issues, such as thermal balance and thermal stress management. Payload or instrument sensitivity to outgassing and thermal displacements must also be considered.

Additive manufacturing (AM) has increased custom structural solutions for SmallSats and demonstrated high throughput of complex structures. Materials that were once out of reach of AM are now readily available in higher end systems. Once only for secondary structures, AM has seen an expansion in primary structures – especially in small CubeSat or PocketQube buses.

However, for larger CubeSats and Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (ESPA) SmallSats, conventionally machined assemblies constructed from aluminum alloys still have their place for primary structures. Secondary structures, such as solar panels, thermal blankets, and subsystems, are attached to primary structures. They stand on their own and transmit little to no critical structural loads. When a primary structure fails, catastrophic failure of the mission occurs, and while failure of a secondary structure typically does not affect the integrity of the spacecraft, it can have a significant impact on the overall mission. These structural categories serve as a good reference but can be hard to distinguish for small spacecraft that are particularly constrained by volume. This is especially true for SmallSats, as the capabilities of these spacecraft may be similar to full size buses, but the volume afforded by dispensers or deployment rings becomes the constraining factor. Therefore, it is imperative that structural components are as volume efficient as possible. The primary structural components need to serve multiple functions to maximize volume efficiency. Such functions may include thermal management, radiation shielding, pressure containment, and even strain actuation. These are often assigned to secondary structural components in larger spacecraft.

Structural design is not only affected by different subsystems and launch environments, but also the spacecraft application and intended environment. There are different configurations for spin-stabilized and 3-axis stabilized systems, and the instrumentation used places requirements on the structure. Some instruments require mechanisms, such as deployable booms, to create enough distance between a magnetometer and the spacecraft to minimize structural effects on the measurement. The spacecraft exterior and interior material and electronic subsystems need to be understood in the specific mission environment (e.g., in-space charging effects). Mitigation for charge build-up is provided in section 6.3.2 Thermoplastics and Photopolymers.

Highly configurable or modular systems may be desirable in quick-turn products, as prototyping and firmware and software development can be extended further into the spacecraft design cycle with flight hardware in the loop. Card slot systems not only provide those benefits, but when paired

with certain standards, they can still fulfill the same structural, mechanical, and thermal requirements as the current CubeSat method of "stacking" electronics and payloads.

Small satellite mechanisms have advanced with deployable structures, actuators, and switches. Deployable structures enable large structural applications with minimal volume requirements. Actuator and switch mechanisms expand the capabilities of small satellites with motion and deployment applications. These mechanisms enable increased small satellite capabilities beyond original structural volume constraints.

An overview of radiation effects and some mitigation strategies is included in this chapter because radiation exposure can impact the structural design of small spacecraft. For SmallSats operating out of low-Earth orbit with increased radiation exposure, mission planners may also want to consider risk mitigation strategies associated with specific radiation environments. This includes both interplanetary missions, where solar radiation dominates, and polar low-Earth orbit (PLEO) missions, where solar radiation risk increases over the poles. In addition, as solar maximum approaches in 2025 (1) with an increased number of solar particle events (SPEs), mission planners will need to consider many orbital environments.

The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

6.2   State-of-the-Art – Primary Structures

6.2.1 CubeSat Standard

Two general approaches are common for primary structures, often called frames or chassis, in the small spacecraft market: commercial-off-the-shelf (COTS) structures and custom machined or printed components. It is not surprising that most COTS offerings are for the CubeSat market. Often COTS structures can simplify development, but only when the complexity of the mission, subsystems, and payload requirements fall within the design intent of

Table 6-1: CubeSat Standard Structure Dimensions		
Type	Dimension (mm)	Average Structure Weight (kg)
1U	100 x 100 x 113.5	0.118
1.5U	100 x 100 x 170.2	0.142-0.25
2U	100 x 100 x 227	0.220
3U	100 x 100 x 340.5	0.352
6U	100 x 226.3 x 366	0.916-1.94
12U	226.3 x 226.3 x 366	1.84

a particular COTS structure. Custom machined structures enable greater flexibility in mission specific system and payload design. The typical commercially available structure has been designed for low-Earth orbit applications and limited mission durations, where shielding requirements are confined to limited radiation protection from the Van Allen Belts.

The CubeSat standard structure has evolved with increasing use over many years. The CubeSat standard structures, also referred to as canisterized satellites, include 1U, 1.5U, 2U, 3U, 6U, and 12U. Larger sizes exists but are less common. Table 6-1 shows the nominal weight limits and

dimensions of each CubeSat structure from the CubeSat Design Specification document. There is an extra volume (XL) option available for 3U, 6U, and 12U CubeSats; this additional volume, commonly referred to as the "tuna can" volume, is associated with an individual dispenser type. This cylindrical XL additional space allows for structural extensions of the CubeSat that can be used for various components. Steamjet Space has developed Steam Thruster, a tuna can-sized electrochemical thruster specifically designed for CubeSats. The 3U CubeSat Elfin mission used this tuna can space for antenna deployment. Shields mission also fit a radiator within its tuna can volume. Figure 6.1 shows this optional volume and location on the CubeSat.

There are several companies that provide CubeSat primary structures. Most are machined from aluminum alloy 6061 or 7075 and are designed with several mounting locations for components to allow flexibility in spacecraft configuration. The SmallSat community has witnessed an increase in CubeSat standard configuration over the last 10 years from 1U to 3U, to include 6U and 12U. This was due to a higher demand for more science on a smaller platform, and by the need for more volume to design more complex CubeSats that can handle greater responsibility. Table 6-2 lists several commercial primary CubeSat structures. Of the offerings included here, 1U, 3U and 6U frames are most prevalent, however 12U frames are becoming more widely available as there are now more dispensers for the 12U CubeSat structure. Figure 6.2 shows some commercial examples of 3U, 6U and 12U CubeSat structures.

8U and 16U CubeSat Structure

Following the trend of larger CubeSat structures that is driven by the needs of the SmallSat market, several companies are now offering CubeSat structures not officially recognized by the CubeSat

Figure 6.1: Optional extra volume shown on 3U and 12U –Z face (also known as a "Tuna Can"). Credit: Cal Poly CubeSat Laboratory.
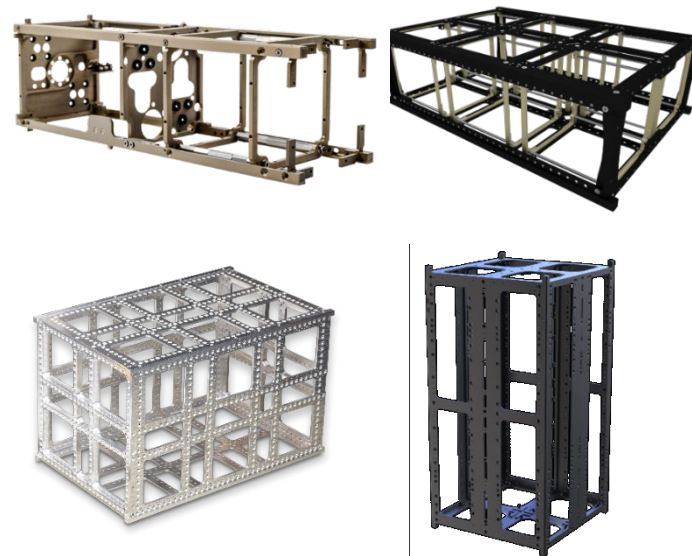
Figure 6.2: Various commercial CubeSat structures. Top Left: NanoAvionics 3U Structure. Credit: NanoAvionics. Top Right: 6U nanosatellite structure. Credit: GomSpace. Lower Left: 12U Structure. Credit: C3S Electronics Development, LLC. Lower Right: 16U structure. Credit: EnduroSat.

standard such as the 8U and 16U. Customized dispensers are available that will host these larger volumes.

Table 6-2: Commercial Primary CubeSat Structures	
Manufacturer [Headquarters]	Structure (U)
AAC Clyde Space [Sweden]	ZAPHOD 1U, 2U, 3U, 6U, 12U
C3S Electronics Development LLC [Hungary]	3U/3U Plus, 6U, 12U, 16U
Cervos Space [Turkey]	1U, 2U, 3U, 6U
EnduroSat [Bulgaria]	1U, 1.5U, 3U, 6UXL, 8U 12UXL, 16U
German Orbital Systems [Germany]	1U, 2U, 3U, 6U, 12U
GomSpace [Denmark]	3U, 6U, 8U, 12U, 16U
Gran Systems [USA]	1U, 1.5U, 2U, 3U, 6U, 6UXL
Gumush [Istanbul]	n-ART 1U, 2U, 3U
ISISPACE [The Netherlands]	1U, 2U, 2UXL, 3U, 6U, 8U, 12U, 16U
Ishitoshi Machining [Japan]	MBF-1U, 3U
NanoAvionics [Lithuania]	1U, 2U, 3U, 6U, 12U, 16U
Pumpkin Space Systems [USA]	Supernova 1U, 3U, 6U, 12U
NPC Spacemind [Italy]	SM 1U, 1.5U, 2U, 3U, 6U, 6U XL, 8U, 12U, 12U XL, 16U
Nara Space [South Korea]	12U, 16U

6.2.2 Custom CubeSat Primary Structures

A growing development in building custom small satellites is the use of detailed interface requirement guidelines. These focus on payload designs with the understanding of rideshare safety considerations for mission readiness and deployment methods. Safety considerations include safety switches, such as the "remove before flight" pins and foot switch, and requirements that the spacecraft remain powered-off while stowed in the deployment dispensers. Other safety requirements often entail anodized aluminum rails and specific weight, center of gravity, and external dimensions for a successful canister or dispenser deployment.

DiskSat Structure

The Aerospace Corporation is developing a DiskSat (figure 6.3) demonstration flight with support from NASA's Space Technology Mission Directorate (STMD). The DiskSat is a 1-m circular disk, 2.5 cm thick, graphite-epoxy composite sandwich, with a structural mass less than 3 kg/m^2. The volume is close to 20 liters, which is equivalent to a hypothetical '20U' spacecraft. While the entire volume will not be filled, the increased surface area is useful for power, aperture, thermal management, and manufacturing simplification. First launch for the demonstration mission is planned for 2026 (2).
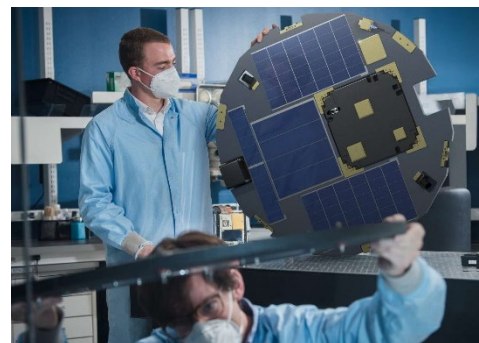

Figure 6.3: DiskSat structure. Credit: The Aerospace Corporation.

6.2.3 Primary Structure Standard Dispenser

The box that houses the CubeSats in the launch vehicle is called a dispenser (or deployer), and they dispense (or deploy) the CubeSat into the desired orbit. The CubeSat uses the entire volume of the dispenser to make use of its full capacity. Since the CubeSat adopts a standard size and form factor, CubeSat dispensers have also been standardized with two constraint systems: rail- or tab-type. This allows spacecraft designers and launch service providers to minimize launch integration cost, increase access to space, and sustain frequent launches (3). The CubeSat Design Specification document by the CubeSat Program at Cal Poly was created to provide CubeSat developers baseline requirements that are compatible with as many CubeSat dispensers and launch opportunities as possible to eliminate launch interface failures (4). To view the most updated versions of the CubeSat Design Specification, please visit: http://www.cubesat.org. The CubeSat Design Specification document includes rail systems. The Canisterized Satellite Dispensers (CSD) tab system created by Planetary Systems Corporation (now Rocket Lab) is the most widely available tab dispenser that offers design flexibility for structures that do not require the use of rails. See CSD datasheet for detailed information on tab dispenser (5).

A tab-style canister deployment system uses tabs that are loaded to hold the CubeSat to a wall of the canister which are released upon deployment. The vibrational load during launch passes from the launch vehicle to the canister structure with the pre-loaded CubeSat. A CubeSat using a rail dispenser is lightly loaded on the z-axis. On the x and y axis a thin gap exists between the rail of the dispenser and rails on the CubeSat which can cause vibrational chatter. The vibrational chatter adds to the mechanical load of the CubeSat during testing and launch. For more CubeSat rail vs tab dispensers, please refer to Chapter 10: Launch, Deployment, Integration, and Orbital Services.

The required interface documents originate with the rideshare integrator for the specific dispenser being used with the launch vehicle. The launch vehicle provider typically provides the launch vibrational conditions. The NASA CubeSat Launch Initiative (CSLI) requires CubeSat or SmallSat systems be able to withstand the General Environmental Verification Standard (GEVS) vibration environment of approximately 10 G_{rms} over a 2-minute period (6). The NASA CSLI rideshare provides electrical safety recommendations for spacecraft power-off requirements during launch and initial deployment. The detailed dispenser or canister dimensional requirements provide enough information, including CAD drawings in many cases, to enable a custom structural application.

Table 6-3 lists some dispenser and canister companies that provide spacecraft physical and material requirements for integration. In response to the demand for larger CubeSats, dispensers for 12U CubeSats are now available through several launch service providers like NanoRacks and United Launch Alliance (ULA) through the Atlas series. There are several European companies providing deployment for 16U platforms that expand the limits of the CubeSat Design Specification. The DSOD, EXOpod, and the Quad Pack are all dispensers that can fit a single 16-unit CubeSat platform or several smaller CubeSats.

Table 6-3: Spacecraft Physical Dimension and Weight Requirements from Deployers			
Deployer	U Accessible	Requirements	Available Documents
P-POD by Cal Poly	1U, 3U	Dimensions, Weight, Rail	Follows CubeSat Standard (4)
CSD by Planetary Systems Corp.	1U, 3U, 6U, 12U	Dimensions, Weight, Tab	
Railpod III, 6U NLAS, 12U Deployer by Tyvak	3U, 6U, 12U	Dimensions, Weight, Rail	Interface Control Documentation (8)
PSC by Rocket Lab	3U, 6U, 12U	Dimensions, Weight, Tabs	Interface Guide, CAD Drawings (5)
DuoPack, QuadPack by ISISPACE	1U, 2U, 3U, 4U, 6U, 8U, 12U, 6UXL, 12UXL, 16U	Dimensions, Weight, Rail	Follows CubeSat Standard (7)
MyPOD and Test PODs by Gran Systems	3U, 6U	Dimensions, Weight, Rail	Website (9)
DSOD by Dhruva Space	1U, 3U, 6U, 12U, 16U	Dimensions, Weight, Rail	Website (10)
EXOpod by Exolaunch	1U, 2U, 3U, 6U, 8U, 12U, 16U	Dimensions, Weight, Rail	User Manual (11)
SMPOD by NPC Spacemind	1U,2U,3U,4U,6U,6U XL, 8U, 12U, 12U XL, 16U	Dimensions, Weight, Rail	ICD (on request on website (12))

6.2.4 CubeSat Structures Construction Methods

Monocoque Construction

Monocoque structures are load-bearing skins that have significant heritage on aircraft. On small spacecraft, the intent of this design is several-fold – it maximizes internal volume, it provides more thermal mass for heat sinks or sources, it allows for more mounting points, and it has more surface area to potentially reduce total ionizing dose (TID). Monocoque construction is common, and "extruded" designs are relatively easy to fabricate through computerized numerical control (CNC) machining, waterjet, or laser cutting.

Modular Frame Designs

Modular frames allow for a flexible internal design for quick-turn missions, while still ensuring strict adherence to external dimensions of the CubeSat standard, especially when deployment from a standardized, reusable dispenser is required. Open frames are suitable for low-Earth orbit, as radiation shielding is not provided by the structure. Care must also be taken to design for thermal mass requirements, as modular frames are inherently light.

Additively Manufactured Designs

The use of additive manufacturing allows for designs of structures that cannot be made with traditional methods or would be excessively costly to manufacture using traditional machining. Additive manufacturing process allows for much more design flexibility to customize structures to specific mission needs. Topology optimization methods, which generally create structures that are only feasible for additive manufacturing, can be used to minimize the amount of material and thus mass needed for the spacecraft structure. Some CubeSat missions have already flown using polymeric printed structures as opposed to metal which can greatly reduce both mass and cost. Considerations need to be made regarding ESD, grounding, and other possible electrical considerations to a polymeric frame. Additionally, fasteners for polymeric structures need to be

evaluated differently than for metal, as plastic doesn't create as strong of threads and heat-set inserts may not hold well to the structure in the temperature extremes of space.

6.3 State-of-the-Art – Mechanisms

Spacecraft commonly contain onboard devices whose function are based on mechanical movement (i.e.: slide, roll, rotate, separate, unfold, or spin) to either modify part of the spacecraft's geometry or to ensure operational function of a component or instrument. These devices are known as mechanisms, and as spacecraft become more sophisticated with the advances in miniaturization of electronics and systems, their reliance of mechanisms greatly increases.

The domain of spacecraft mechanisms is quite broad as there are many different types in the design and life of a spacecraft that include the moving parts associated in each phase:

- Deployment: dispensing spacecraft into orbit
- Beginning of mission life: deployments of solar arrays, booms, antennas, instrumentation, etc.
- Mission maintenance: sun tracking, pointing antennas and instruments, active doors or shields, gyroscopes and reaction wheels, thrusters, etc.
- End-of-life: deorbiting methods

The technology within the mechanism to perform the movement is accomplished with an actuator. Depending on the actuation method, spacecraft mechanisms are either passively or actively driven. Passive mechanisms do not consume electric energy and provide driving power via spring load, and active mechanisms are motorized to produce driving power for mechanism operation. Most mechanisms can use both passive and active capabilities depending on the application. Table 6-4 provides an overview of common spacecraft mechanisms and examples of technologies used.

The state-of-the-art for small spacecraft mechanisms is quantified on their high reliability, low power, and light weight characteristics, and the common mechanisms listed below are considered state-of-the-art for small spacecraft use. For the purposes of this chapter, the mechanisms focus on deployable extensions, robotic manipulations, release actuation, component pointing, and gimbal mechanisms. Reliability considerations are provided for optimal operational capabilities, as well as a brief explanation of the factors that affect spacecraft mechanisms.

Table 6-4: Type of Spacecraft Mechanisms		
Type of Mechanism	Description	Technology Examples
Separation and Release	Reliable stowage and release of spacecraft and deployable components upon an external command (active) or spring-loaded (passive).	Clamp band systems, Frangibolts, release nuts, pin pullers, bolts, burn wire, hinges, and passive spring-loaded switches
Motorized	Allows for rotary motion of spacecraft components.	Solar Array Drive Assembly, directional antennas, combination of dampeners and absorbers
Attitude Control	Provides pointing accuracy and stability for spacecraft and components.	Reaction (momentum) wheel assembly, gimbals, component pointing, passive methods

6.3.1 Actuators

By classical definition, actuators are devices that convert electrical, thermal, hydraulic, and/or pneumatic energy into mechanical motion when said energy is allowed to flow. Active, or commanded, actuators use onboard data links and electrical transistors to determine the transfer of energy; whereas passive, or reactive, actuators allow the spacecraft environment (including external launch systems) to dictate actuator energy transfer. Table 6-5 provides some commercial actuators for SmallSats.

Specifically, spacecraft actuators are used for a variety of purposes, including:

- Attitude control and gimbaling: to control the orientation of either part (gimbaling), or all (attitude control), of a spacecraft in space. This is important for pointing sensors, instruments, and/or communications antennas in a direction required for their use.
  - o Attitude control general types: reaction control thrusters, momentum wheels, control moment gyros, magnetic torquers, aerodynamic control surfaces, solar sails, and gravity gradient stabilizers.
  - o Gimbal general types: single-axis, dual-axis, and triple-axis system.
- Propulsion: supporting attitude control system operations, maneuvering to a new orbit, or reducing orbital velocity to begin atmospheric reentry.
  - o General types: chemical rocket engines (which can be the same as the upper stage launch vehicle engines), reaction control thrusters, and electric propulsion systems. These systems typically require actuated valves to operate.
- Deployment, docking and separation: extend and unfold solar panels, antennas, and other spacecraft components requiring unpacking to function.
  - o Deployment general types: hinge-&-spring based, linear-actuator-&-scissor-frame based, roll-out systems, and inflatable structures.
  - o Docking general types: probe-and-drogue, peripheral, and soft-capture systems.
  - o Separation general types: spring-powered or gas-powered systems.
- Thermal control: manage all or part of the spacecraft's temperature. This is important for protecting internal components from extreme temperatures.
  - o General types: louvers, heat pipes, thermoelectric/Peltier devices, and pumped thermal fluid systems.

Mechanical actuation methods/techniques that are found in many of the above systems include:

- Electric & electromagnetic: AC/DC motor, piezoelectric ceramics, and push/pull & rotary solenoids (including solenoid valves), and microelectromechanical systems (MEMS).
- Thermal & thermoelectric: Shape memory alloys (SMA), phase-change liquids/solids (paraffin wax, liquid metals), thermofluidic gas systems, thermal bimorph structures, harmonic drive micro actuators (HMAs), thermal knife cutters, and magnesium alloy band systems.

Figure 6.4: (top left) SADM 1500. Credit: Comat. (right) TiNi Aerospace Frangibolt Actuator and (right) ML50 microlatch. Credit: Ensign-Bickford Aerospace & Defense.

Table 6-5: Commercial Actuators						
Manufacturer [Headquarters]	Product	Mass (kg)	Size (mm)	Power Consumption	Actuation method	Ref
Beyond Gravity [Switzerland]	Separation Nut PSM 3/8B	0.23	58.5x36x56	-	-	(23)
Comat Space [France]	Solar Array Drive Mechanism - 400	0.465	83x62x46	4 W	Geared motor	(18)
	Solar Array Drive Mechanism - 1500	3.5	201x132	13 W	Geared motor	(19)
DCUBED [Germany]	Micro Pin Puller (uD3PP)	0.08	25.5x25.5x25.5	-	SMA	(21)
	Nano Pin Puller (nD3PP)	0.025	17x17x17	-	SMA	(21)
	Micro Release Nut (uD3RN)	0.078	25x25x25	-	SMA	(22)
DHV Technology [Spain]	MicroSADA-10	<0.25	100x100x100	-	Stepper motor	(20)
	MicroSADA-18	<0.95	226x80x18	-	Stepper motor	(20)
Ensign-Bickford Aerospace & Defense Company [USA]	TiNi™ Flat Pack					(24)
	TiNi™ FD04 Frangibolt	0.007	13.72x10.16	15 W @ 9 VD	SMA	(14)
	TiNi™ ML50	0.010	11.45 diamx12.7	9 W @ 7 VD	SMA	(15)
	TiNi™P5 Nano Pin Puller	0.015	25.4 diamx16.5	1.5A to 3.75A	SMA	(25)
	TiNi™ P5 Pin Puller	0.018	28 diamx26	0.5A to 2A	SMA	
	NEA 9040 Mini	0.030	28 diamx32	0.4A to 1.5A	SMA	
	NEA 1120 Pin Puller	0.014	21x22	Min 3A	Split Spool	(26)
Honeybee and MMA Design [USA]	Solar Array Drive Actuator (SADA)	3.1	127x210	-	Stepper Motor	(17)
Nimesis Technology [France]	Triggy	0.004-0.271	*	*	SMA	(28)
	Gripper	<0.032	39x35x7.6	7.4 W (average)	SMA	
Moog [USA]	Type 2 Side-Drive Solar Array Drive Mechanism (SADM)	5	234x278.6	15 W	-	(16)
Revolv Space [Netherlands]	Solar Array Rotary Actuator (SARA)	<0.35	97x97x23	1 W (average)	-	(25)
Sierra Space [USA]	High-Output Paraffin Actuator (HOPA)	0.35	44x14x14	5W	Paraffin	
	C14E-TC Solar Array Drive Assembly (SADA)	2.9	210x140x140	-	Stepper Motor	(29)
	C14C-TC Solar Array Drive Assembly (SADA)	2.2	172x140x140	-	Stepper Motor	(29)
	C14 Bi-Axis Gimbal	1.23	155x72x72	-	Stepper Motor	(29)
	C14E Bi-Axis Gimbal	1.72	221x112x112	-	Stepper Motor	(29)

Data unknown is represented by -

* See reference

6.3.2 Deployable Structures

Space deployable mechanisms are structures folded into a compact configuration and deployed into a larger predetermined shape. The development of deployable structures on spacecraft is appealing to enable greater mission performance. Once deployed, the structures reconfigure, changing shape and size from folding and unfolding. Common spacecraft deployables include antennas, radiators, solar panels, gravity assists, and other science instruments. Small spacecraft are great candidates for using deployable structures to raise the functionality of a smaller platform. However, there are limited designs for compact, lightweight, low power deployable structures that can be folded or rolled up for launch and then self-deployed in space to support these kinds of systems on small satellites.

There are different types of deployment mechanisms to ensure the deployed structure effectively expands to the desired configuration in-orbit: folding, sleeve, truss, and inflatable. Deployable solar arrays are a common folded-type of passive deployment mechanism achieved by connecting the spring and hinge to increase solar energy for the spacecraft. Please refer to the *Power* chapter for deployable solar panels and arrays. The sleeve-type deployment mechanism is implemented using a rolling or sliding screw conveyor and is commonly seen on SmallSats for various antennas (30). Inflatable deployment structures are light-weight film material typically used for larger deployed structures, like solar sails. Please refer to the *Deorbit Systems* chapter for deployable mechanisms used for deorbit devices.

For SmallSat applications, it is common that deployable components are on a boom – a cantilever arm ejected from the spacecraft – that can perform various tasks once deployed. See figure 6.5 for NASA's GPX-2 CubeSat mission with a Redwire Space deployable boom to create gravity gradient stabilization as an example. SmallSat deployable structures are common and are associated with high reliability. Engineers have started developing deployables with different materials to decrease the stowage area, mass, and power. Table 6-6 lists a selection of commercially available deployable booms.

Figure 6.5: GPX-2 CAD image with gravity gradient boom deployed. Credit: NASA.

NASA Langley Research Center (LaRC) has developed Deployable Composite Booms (DCB) through the Space Technology Mission Directorate (STMD) Game Changing Development (GCD) program and a joint effort with the German Aerospace Center, see figure 6.6. DCBs have high bending and torsional stiffness, packaging efficiency, thermal stability, and 25% less weight than metallic booms (31). The Advanced Composite Solar Sail System (ACS3) project will demonstrate DCB
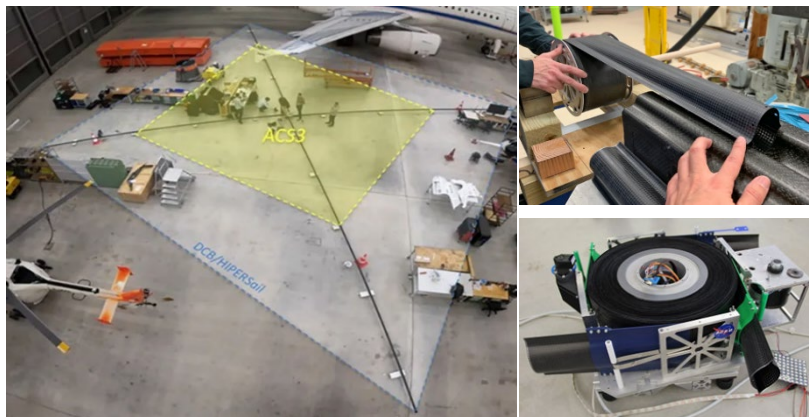

Figure 6.6: NASA Deployable Composite Boom (DCB) technology. Credit: NASA.

technology for solar sailing applications after being launched in April 2024. The DCB/ACS3 7-m boom technology is extensible to 16.5 m deployable boom lengths (32).

Engineers have started using origami – the art of paper folding – as a strategy of deployable structure design. Origami structures are flexible in their deployment direction so that they can be easily collapsed along the same path they are deployed. One advantage of origami-inspired mechanisms is potentially faster and cheaper prototyping. Instead of relying on laser cutting or 3D-printing, prototyping of origami-inspired mechanisms can be accomplished using inexpensive materials like paper before moving to other more expensive materials. Many resources and patterns already exist that detail how designs can be created and modified or adapted for engineering purposes (33). Solar panels and arrays, solar sails, and sunshades are examples of ongoing origami engineered SmallSat components.

Table 6-6: Commercial Deployable Booms		
Manufacturer	**Product**	**Reference**
Composite Technology Development	Stable Tubular Extendable Lock-Out Composite (STELOC)	(34)
Oxford Space Systems	AstroTube deployable boom	(35)
Redwire Space	Roll Out Composite (ROC) booms	(36)
Redwire Space	CubeSat ROC Boom Deployer	(37)
Redwire Space	ROC-FALL system	(37)
Magellan Aerospace	Deployable Boom	(38)
Rolatube Technology	Deployable Composite Booms	(39)
Northrup Grumman	Coil Booms	(40)
Northrup Grumman	Telescoping Tube Masts	(41)

6.3.3 Robotic Manipulator

The need for in-space servicing is receiving more attention from the SmallSat community with the increasing demand of more complex SmallSat with greater capability and longer mission life. These types of challenges are being solved with robotic manipulations that can perform intricate actions in space. Tasks such as repairing defunct satellites, in-orbit assembly, satellite servicing, debris capture, spacecraft system up-keep, construction, and repair are important advances for future space operations; these challenges are currently expensive and risky to perform. Current robotic solutions for in-space construction and repair involve humans and use very large, expensive, custom-built robotic arms with limited capabilities, such as the Canadian Arm. As NASA's Artemis program prepares for astronaut presence in lunar and deep space missions on the Lunar Gateway, there is a greater need for more advanced and maneuverable space robotic systems. The use of these sophisticated robotic systems on a SmallSat are more alluring than traditional larger platforms as SmallSats present a more cost-effective and agile solution. A more agile robotic system can be stowed in small space and deployed to perform several tasks automatically or semi-automatically.

This section provides an overview of the continuous work occurring to further develop robotic systems on SmallSats. Table 6-7 lists a non-exhaustive list of the ongoing work. This type of SmallSat mechanism is maturing with research and development at government, academia, and commercial entities (42). For example, the Naval Academy Satellite Team for Autonomous Robotics (NSTAR) has developed an autonomous 3U CubeSat robotic arm system called the Robotic
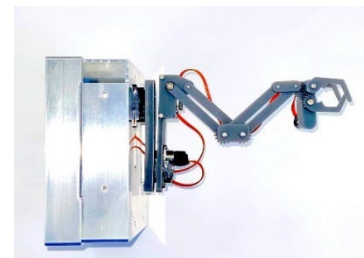

Figure 6.7: RECS robotic arm. Credits: US Naval Academy.

Experimental Construction Satellite (RECS) to be tested on the ISS. RECS was launched November 2022 (43).

Manufacturer	Product	Mass (kg)	Extendable length (mm)	Stowed Envelope (mm)	DOF	Power Consumption (W)	Actuator method	Form Factor
U.S. Naval Academy	3U CubeSat with two robotic arms	4	600	300x100x100	6 plus "claw" end-effector actuation	-	Stepper motor	RECS 3U (Nov 2022)
Redwire Space	-	-	1 to 4 m reach	-	5 to 7	8 to 65	-	ESPA class satellites
Sierra Lobo	Sierra Lobo Arm (SLAC 1)	-	100x100x100	30x50x65	3	1.5	-	-

Table 6-7: Robotic Arms for Small Spacecraft

- Represents unknown data

6.3.4 Reliability Considerations

Mechanisms add capabilities and complexities to small satellite design. Additional integration and testing are required. NASA Reliability and Maintainability Standard (44) describes maintainability, and "test as you fly," in addition to multiple other mitigation strategies and considerations. For mechanisms, it is important to test the full sub-system and system integration for power consumption and sub-system dependencies. Mechanisms have lifetimes, so it is important to have a maintainable mechanism and to understand the lifetime of the mechanism from test to flight. Because mechanisms add complexity and a single point failure risk in some instances, such as attitude control or solar panel pointing, directional antenna control, or one-time sub-system deployment switches, it is important to focus on reliability strategies. Mechanisms have contributed to over 10% of reported small satellite failures (45). Adding a mechanism to enable a mission increases risk.

The space environment adds to reliability considerations for operational considerations in vacuum, plasma, and/or thermal environments. For the reliability of mechanisms, there are multiple steps that contribute to risk mitigation. Sarafin et al. describes a multi-step approach for a reliable mechanism from design simplicity, margin, supplier selection, to test (46). The steps include guidance for torque margin for rotating parts, such as solar panel and antenna pointing motors (46)(47)(48). During ground testing of mechanisms, it is important to understand the mechanism lifetime, so that the component performs throughout the planned mission duration. Materials considerations contribute to mechanism reliability in the space environment, such as lubricant use and material coatings to avoid corrosion and welding of dissimilar materials (47)(48)(49). Because mechanisms are critical for advanced spacecraft capabilities for power, communications, and science/research instruments, it is important to add mechanism margin and tests to the spacecraft development and/or sub-system and system integration.

6.4   State-of-the-Art – Polymeric Additive Manufacturing

Additive manufacturing (AM) processes for primary spacecraft structures have long been proposed but only recently have such methodologies been adopted for flight. AM has been common for SmallSat secondary structural elements for many years. Typically, the advantage of AM is to free the designer from constraints imposed by standard manufacturing processes and allow for monolithic structural elements with complex geometry. In practice, additive manufacturing has a separate design space and design process, which has seen tighter integration into computer-aided design, computer-aided manufacturing, and modal and structural analysis packages in the past few years. Such tools can enable quicker turnaround times for SmallSat development, and have been instrumental in mass optimization, using AM materials in radiation shielding, and enabling high-throughput, high-quality manufacturing. As the AM field is rapidly evolving, this section makes a best attempt to cover as many materials and printers as possible that are potentially applicable to SmallSat development.

6.4.1   Applicability of TRL to Polymer AM

While AM systems and platforms might be considered mature and of high TRL, the TRL of AM parts configured for spaceflight depends on the material, the configuration of the actual part, the manufacturing process of the material, the postprocessing of the manufactured part, the testing and qualification process, and many other factors. For example, nylon fabricated with a fused filament fabrication (FFF) system will have different bulk structural properties from nylon fabricated with a selective laser sintering (SLS) system. Furthermore printer and material settings (e.g., temperature) also play a role in final part properties, meaning the same part from the same material on the same machine can have different properties if the parameters are adjusted.

In other words, a TRL might be assignable to a component created through a particular manufacturing process with a specific material. If a particular component manufactured with nylon on an FFF system was flown to LEO successfully, the TRL for this component would be 7. If this component was subsequently flown on another mission manufactured with Antero 840 PEEK also on an FFF system, the TRL would still be 7. Documentation of the manufacturing process is important to properly account for TRL. This section focuses on polymer AM and does not address metal AM for SmallSats.

6.4.2   Inspection and Testing

When new materials and/or processes are used, testing must be performed to minimize risk and bridge the gap between TRL levels. In particular, the only way to validate a tailored structure, component, or material is through testing, especially if more freedom is allocated to research and development. For new material types, if there is latitude afforded in upfront research and development, mechanical, modal, and thermal tests should be performed to compare against a known, proven structural design.

6.4.3   Thermoplastics and Photopolymers

With the expansion of available open-source AM platforms in the last decade, thermoplastics and photopolymer materials have rapidly gained traction and acceptance in many applications ranging from mechanical validation and fit-checking to engineering-grade, low-rate production products. Photopolymer or "thermoset" resins and associated manufacturing processes have improved to the point where microfluidics experiments may be additively manufactured, with the microfluidics channels and growth chambers directly manufactured as one piece, as opposed to the more traditional microfluidics approach of machining plastic or glass blocks.

As of publication, there are three primary methods of conducting AM for plastics: FFF (also referred to as fused deposition modeling (FDM)), which uses thermoplastics in either a spool or pellet form; stereolithography (SLA), which uses photopolymer resin; and SLS, which uses a fine powder or photo-set resin. Within SLA, there are two methods of curing resin: digital light projection (DLP), which uses a very high-resolution LED matrix – a monochrome display – to cure the entire layer nearly instantly; and polyjet (also known as resin jet) which deposit resin from a line array of jets, much like an inkjet printer with a large print head.

Certain thermoplastics are quickly gaining acceptance for high-reliability parts and applications on Earth, although, as of this writing, they have yet to gain widespread acceptance for space applications. One reason for this is AM methods cannot yet produce surfaces as smooth as machined metals, which is often a requirement for parts with tight tolerances. However, some thermoplastics are machinable, such as Nylon or polyetherimide (PEI). Like the manufacture of cast iron parts, machining to a final, high tolerance specification may allow these thermoplastics to gain further acceptance.

Except for some large-format AM centers, almost all thermoplastics are manufactured in spools, and may or may not be packaged for proprietary solutions. For SLA, almost all resins are used specifically for commercial solutions and AM centers. Additionally, some manufacturers may mix in additives to enhance material properties or ease the printing process. Because of this, the following sections on each material include a table of materials for both open-source and commercial solutions, and selected properties of interest. Availability of recommended nozzle and bed temperature is indicative of the ability to be printed on an open-source machine, except otherwise noted in the material description. Materials are not picked according to preference but through availability of technical specifications and potential applicability. For various types of AM solutions, readers are encouraged to use these sections as a rough guide for currently available commercial filaments. Additionally, the material tables will be expanded as more data is obtained on the following materials.

Surface discharge, or electrostatic discharge (ESD), is a result of in-space charging effects and is caused by interactions between the in-flight plasma environment and spacecraft materials and electronic subsystems (50). The field buildup and ESD can negatively affect the spacecraft and there are design precautions which must be considered depending on the spacecraft's operational environment. Per ESD guidelines from NASA Spacecraft Charging Handbook 4002A, dielectric materials above 10^{12} Ohm (Ω) cm should be avoided because charge accumulation occurs regardless. Please refer to the NASA Handbook 4002A, 5.2.1.5 Material Selection for more information. Historically, ESD due to faulty grounding has been a leading cause of spacecraft or subsystem failures (50). Most ESD rated thermoplastics have a surface resistivity between 10^6 and 10^9 ohms. Increasing the extrusion temperature decreases the surface resistance.

Polylactic Acid (PLA)

PLA is the most common filament used in AM and table 6-7 lists several PLA filaments. It exhibits very low shrinkage and is extremely easy to print because it does not require a high temperature bed nor a heated build chamber and requires a relatively low extruder (nozzle) temperature. It also has low off gassing during printing, important in open-frame AM systems in rapid prototyping environments such as lab settings. Unless the application has a very short-term exposure to harsh conditions, and if the conditions are well characterized and controlled, it is not recommended to use PLA for an application beyond TRL 3-4. For laboratory settings in controlled environments not subject to excessive mechanical forces, ESD-compatible filaments are available.

Table 6-7: Polylactic Acid Filaments								
Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk* (Ω-cm)
Prusament PLA	55	12 kJ/m^2	57	N/A	215	50-60	1.24	No
Verbatim PLA	50	16 kJ/m^2	63	N/A	210	50-60	1.24	No
ColorFabb PLA-PHA (51)	N/A	30 kJ/m^2	61	89	210	50-60	1.24	No
Stratasys PLA (52)	51	27 kJ/m^2	26	84	N/A	N/A	1.264	No, 10^{15}
3DXSTAT™ ESD-PLA	55	N/A	55	95	210	23-60	1.26	Yes, 10^6-10^9

Acrylonitrile Butadiene Styrene (ABS)

ABS has traditionally been the choice for higher strength, lightweight prints from the FDM process in the open-source community. In the early days of hobbyist 3D printing, ABS was a preferred material as there was an existing supply chain making ABS filaments for weed-whackers. It is generally temperature resistant and UV resistant but turns yellow and eventually becomes more brittle over time when exposed to sunlight. It is a marginally difficult filament to print, especially in open-frame systems. High temperature gradients during printing may cause warping as parts get larger. Enclosed AM systems with heated chambers print ABS well. For many systems passive chamber heating is sufficient, although active heating can provide better results. Extrusion and bed temperatures for ABS tend to be slightly higher than those for PLA. Additionally, ABS shrinks 1 to 3 percent of its printed size upon cooling – the shrinkage varies from manufacturer to manufacturer and to a lesser extent can also be affected by the rate of cooling. ABS has flown as the complete structure for KickSat-2, a FemtoSat deployer for chip-scale satellites (53). The single-use, short mission duration, and intricate dispenser frame made a conventionally machined deployer mass- and cost-prohibitive. Table 6-8 lists some examples of ABS filaments.

Table 6-8: ABS Filaments								
Filament Name	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk (Ω-cm)
Stratasys ABS-CF10	100	20-51 J/m	21	29-69	N/A	N/A	1.0972	Marginal 10^4-10^9
Stratasys ABS-ESD7	105	36.2 J/m	35	44	N/A	N/A	1.07	Marginal 10^4-10^9
3DXSTAT™ ESD-ABS	97	N/A	58	80	230	110	1.09	Yes, 10^6-10^9
Verbatim ABS	106 (ISO 306)	21 J/m	47	78	240-260	90	1.05	No
Kimya ABS Carbon	N/A	N/A	36.7	N/A	250-260	90-100	1.05	N/A

Nylon

Versatile and tough, there are multiple formulations for nylon that allow for a very wide range of applications and material properties. In general, nylon is more difficult to manufacture than ABS on open-source FFF systems because it requires an enclosure for thermal stability and additional bed preparation for higher adhesion. It is also extremely hygroscopic; if possible, filament should be baked out before printing and must be kept in a dedicated dry box while printing or stored. Nylon and composite blends are very common as they can achieve high part strength at a lower price than many of the higher-end thermoplastics. Secondary structural pieces have been flown through the TechEdSat program using Markforged Onyx carbon fiber filaments. Table 6-9 lists some examples of nylon filaments.

Table 6-9: Nylon Filaments								
Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk (Ω-cm)
Taulman3D Alloy 910 (54)	82	N/A	56	N/A	250-255	30-65	N/A	Unk
Taulman3D Alloy 910 HDT (54)	112	N/A	56	N/A	285-300	55	N/A	Unk
Taulman3D Nylon 680 Food Grade (55)	N/A	N/A	47	N/A	250-255	30-65	N/A	No
Markforged Onyx ESD (56)	138	44 J/m	52	83	285	non-heated	1.2	Yes, 10^5-10^7
3DXTECH CARBONX™ HTN+CF (57)	240	N/A	87	95	295	130	1.24	Marginal 10^9
Stratasys Nylon 12 (58)	92-95	71-138 J/m	33-42	55-57	N/A	N/A	1.01	No, 10^{13}
Novamid 1030 PA-CF10	153-184	N/A	38	70	275	60	1.17	--

Polycarbonate (PC)

Also known as Lexan™, this thermoplastic has some of the highest impact resistance, tensile strength, and temperature resistance available for most open source-based AM systems. After manufacturing, it is dimensionally stable and very stiff. However, it is difficult to print on open-frame, open-source AM systems due to very high warping especially when printing large components. Very high bed, chamber, and nozzle temperatures are required, and poor adhesion to the bed is a typical issue. Interface adhesives are recommended to improve bed adhesion. It is also highly hygroscopic; if possible, the filament should be baked out before printing and kept in a dedicated dry box while printing. Certain filaments, like the Prusament PC Blend, have additives to mitigate some of the difficulties of printing PC, particularly those related to higher temperature requirements. Table 6-10 lists some example polycarbonate filaments.

Table 6-10: Polycarbonate Filaments								
Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk (Ω-cm)
Prusament PC Blend (59)	113	No break for ISO 179	63	88-94	275	110	1.22	No
Prusament PC Blend Carbon Fiber (59)	114	35 kJ/m^2	55-65	85-106	285	110	1.16	No
Stratasys PC (60)	143	27-77 J/m	60	75	N/A	N/A	1.20	No
Polymax PC	99-114	21.5 kJ/m^2	59.7	94.1	280	100	1.19	--
3DXtech ESD-PC	135	N/A	68	95	295	130	1.24	10^7 - 10^9
3DXtech CF-PC	133	N/A	70	90	300	140	1.36	--
Polymaker PC-ABS	106	25.8 kJ/m^2	39.9	66.3	250-270	90-105	1.1	--

Windform

Manufactured by CRP Technology, these proprietary materials are classified as a carbon fiber reinforced polymer originally designed for the automotive racing industry. They are unique in that these composites are manufactured through SLS (61). This results in higher dimensional stability and more isotropic properties than FFF, as well as also being able to produce small features slightly better than the FFF process. Windform XT 1.0 and 2.0 have been used on CubeSat and PocketQube platforms and have flight heritage through KySat-2 (launched on ELaNa IV), and TANCREDO-1 (launched through the ISS via JEM in 2017) (62). The PACE-1 CubeSat also used Windform 3D printed components to mount optical devices. Table 6-11 lists CRP Windform filaments. The NASA GPX-2 Windform XT 2.0 structure launched in July 2022 and is operational. Some of the printed parts in the crew dragon spacecraft are made from Windform SP.

Table 6-11: CRP Windform								
Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Manufacturing process	Bed Temp (°C)	Density (g/cc)	ESD Risk (Ω-cm)
Windform XT 2.0	173	4.72 kJ/m^2	84	133	N/A, SLS	N/A, SLS	1.097	Yes, 10^8
Windform RS (64)	181	10.8 kJ/m^2	48-85	139	SLS	SLS	1.10	Yes, 10^8
Windform SP	187	5.82 kJ/m^2	76.1	120.1	SLS	N/A	1.11	Yes, $<10^8$

Polyetherimide

Polyetherimide (PEI), also known by the Saudia SABIC trade name Ultem™, is a very tough thermoplastic resin with high thermal and chemical stability. It is inherently flame-resistant and can be machined. Some formulations of PEI are FAA-approved for flame, smoke, and toxicity (FST), and may also have ESD formulations. PEI is also known for extremely low off gassing, crucial for optical components and sensitive scientific packages. PEI requires high nozzle and bed temperatures. For all but the smallest of parts a heated chamber is required, with temperatures in excess of 120°C producing the best results. Due to these factors, PEI is only practically printable on commercial FFF systems. It is also highly hygroscopic; filament should be baked out before printing and kept in a dedicated or heated dry box while both printing and in storage. PEI has similar characteristics to polyetheretherketone (PEEK), typically with slightly lower mechanical strength. PEI is a common bed material for higher end open-source FFF systems due to its adhesive properties with other thermoplastics at higher temperatures. Ultem 1010 tends to have better mechanical properties than Ultem 9085, however is generally more challenging to print due to lower bed adhesion. Table 6-12 lists some PEI filaments.

Table 6-12: PEI Filaments								
Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m^2) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk
THERMAX™ Ultem™ 9085	158	N/A	63	90	275	115	1.34	No
3DXSTAT™ Ultem™ 1010 CF-ESD (65)	205	N/A	62	115	395	150	1.34	Yes, 10^7-10^9
Stratasys Ultem™ 1010 CG(66)	212	22-27 J/m	81	82-128	N/A	N/A	1.29	No, 10^{14}
Stratasys Ultem™ 9085 (67)	153	39-88 J/m	69	80-98	N/A	N/A	1.27	No, 10^{15}
Zortrax Z-PEI 9085 (68)	186	N/A	54	90	N/A	N/A	1.34	No
Sabic Ultem™ AM9085F	175	33-104 J/m	80	90	345	180	1.28	10^{15}
CarbonX CF-Ultem™ 9085	165	N/A	93	120	390	140	1.39	--
CarbonX CF-Ultem™ 1010	205	N/A	145	120	385	140	1.31	--

PAEK

Polyetheretherketone (PEEK) and polyetherketoneketone (PEKK) – in the polyaryletherketone (PAEK) family – are the highest performing thermoplastics developed as of this writing. With certain additives and matrix materials, they can rival the strength of stainless steel and withstand over 200°C continuously in some formulations, after annealing. Annealing PEEK/PEKK can cause part shrinkage up to 3%; this should be accounted for in either the original design or in the part slicing software. PEEK/PEKK are naturally flame-retardant; they are accepted for use in aviation ducting. They also achieve extremely low off gassing in operation, which makes these thermoplastics good candidates for compatibility with optical components in space. Due to the extreme conditions required for manufacturing and the very high filament cost, these materials are only practically available for printing in extremely robust commercial FFF systems with sealed and heated chambers. It is also highly hygroscopic; filament should be baked out before printing and kept in a dedicated or heated dry box while both printing and in storage. PEEK has heritage on long-term, external ISS experiments, and structural elements on the Juno spacecraft, making it suitable for extreme radiation environments (69). Table 6-13 lists some PAEK-based filaments.

Filament Name (Citation)	ISO 75/ASTM D648 Deflection Temp (°C)	ISO 179-1 Hardness (kJ/m²) or Izod D256-10A (J/m)	ISO 527-1/ASTM D638 ZX Tensile strength (MPa)	ASTM D790/ISO 178 Flexural strength (MPa)	Nozzle Temp (°C)	Bed Temp (°C)	Density (g/cc)	ESD Risk (Ω-cm)
3DXSTAT™ ESD-PEEK (70)	140	N/A	105	141	380-400	150	1.32	Yes, 10^7-10^9
3DXSTAT™ ESD-PEKK	185	N/A	109	135	375	140	1.34	Yes, 10^7-10^9
CarbonX™ CF PEKK-Aerospace	285	N/A	126	178	390	140	1.33	Yes, 10^7
Stratasys Antero 840 (71)	150	28-43 J/m	95	87-139	N/A	N/A	1.27	Yes, 10^4-10^9
Zortrax Z-PEEK (72)	160	N/A	100	130	N/A	N/A	1.30	N/A
Thermax PEKK	150 (182 post anneal)	N/A	105	95 (134 post anneal)	350-370	140	1.27	--
Kimya CF-PEKK	150	N/A	39.1	85.9	370	150	1.27	--
3DGence PEEK	N/A	5 kj/m²	105	130	420	100	N/A	--
CarbonX PEEK-CF10	265	N/A	105	136	400	140	1.39	--

Table 6-13: PAEK-based Filaments

CarbonX PEEK-CF20	305	N/A	126	145	420	140	1.39	$<10^6$

Photopolymers

Otherwise known as "thermosets," these materials are liquid polymers cured by an optical and thermal process. Compared to other AM processes, photopolymers and their manufacturing processes allow for superior isotropic material properties, very high resolution, and the ability to manufacture optical quality parts. In general photopolymers produce parts that are more brittle than their thermoplastic counterparts; special selection is required for more ductile properties. Some formulations, especially from 3D Systems and Stratasys, are designed for extreme temperature resistance and strength, desirable in aerospace applications. In some cases, the listed heat deflection temperature (HDT) may be superior to those of PAEK. As previously discussed, there are three major methods of curing photopolymers, one of which is proprietary. Many photopolymers are specifically paired for commercial systems. As a result, the table 6-14 includes the commercial system associated with the photopolymer.

Some of the photopolymers listed below have several additional characteristics not listable in this table, including, but not limited to, elasticity, tear strength, optical clarity, water absorption, and medical grade certifications. Such characteristics may be useful for biological experiments in future SmallSats. Please consult the products' specific websites and datasheets for additional information. Additionally, photopolymers printed via a polyjet process have the advantage of being able to be mixed, in-situ, as the object is being manufactured. This allows for continuously varying material properties throughout the object. All photopolymer printing processes require at least a two-step post-processing consisting of a print wash and curing step. Table 6-14 lists some photopolymers.

Table 6-14: Photopolymers							
Photopolymer Name (Citation)	ISO 75/ASTM D648 HDT (°C)	ISO 179-1/ASTM D256-10A (J/m)	ISO 527-1/ASTM D638 Tensile (MPa)	ASTM D790 Flexural (MPa)	Density (g/cc) at 25°C	ESD Risk (Ω-cm)	Manufacturing and/or Machine Type
Accura Bluestone (73)	267-284	13-17	66-68	124-154	1.78	ND	3D Systems ProX 800
VisiJet M2S-HT250 (74)	250	10	51	83	1.15	ND	3DS MJP 2500 Plus
DSM Somos® Watershed XC	50	25	50	69	1.12	ND	Stratasys V650 Flex SL
Henkel LOCTITE® IND402 A70 Flex (75)	N/A	N/A	5.5	N/A	1.068	ND	Several

Henkel LOCTITE® 3D 3843 (76)	80	54	60	81	N/A	ND	DLP SLA types only
Somos® Perform	132-268	17-20	68-80	120-146	1.61	N/A	SLA types only

6.4.4 AM Design Optimization

Design optimization is an integral part of manufacturing validation and testing. As previously discussed for AM, validation, testing, and optimization encompass all materials and manufacturing processes. Software platforms help speed up this process, especially those that integrate toolpathing generation, computer aided manufacturing (CAM), load analysis, and fill generation. The inherent advantage of AM to allow monolithic structural elements implies a much-expanded design space compared to subtractive manufacturing. Furthermore, AM processes can create interior features and volumes that cannot be made using subtractive processes as subtractive methods need to have access from the part exterior to any feature. Software has kept up with the pace of manufacturing advances and incorporates tools to assist with AM designs.

The manufacturing ecosystem includes software ranging from simple CAM solutions generating toolpaths (G-code) to complete, structural analysis and high-fidelity manufacturing simulations. As of this writing, AM has gained significant traction and value in low-TRL demonstrations and physical validation, partly due to the ease of fabrication in typical AM ecosystems. It is beginning to displace traditional machining – "subtractive" manufacturing – as AM systems have matured enough to print advanced thermoplastics, resins, and metals.

Infill Patterns

Due to the flexibility that AM offers, new methods of lightweighting are now possible. "Lightweighting" refers to the reduction of mass of structural elements, without compromising structural integrity. The best examples of well-proven heritage methods of lightweighting are "honeycomb" sandwiched aluminum panels, subtractive machining, and truss structures. However, such methods have certain limitations. Honeycomb panels for example, do not have uniform, or isotropic, properties – they do not exhibit the same stiffness in all directions.

Lightweighting in AM encompasses what is called "infill," or the internal structure of a hollow body or panel. With a minimal increase in mass, an internal structure manufactured with AM can vastly increase the strength of a body. Very recently, the AM community has renewed interest in the use of the gyroid pattern, discovered by NASA researcher Alan Schoen in 1970, due to the ease of generation in AM toolpath programs, as well as its rotational nature minimizing any empty columns in a part. Aside from honeycomb and gyroids, several options for infill exist. Different options are offered with different AM-focused software packages. Newer software allows for variable density infills and higher pattern density (and thus strength where needed), and lower pattern density in other regions to reduce overall mass.

Multi Material Parts

Many AM platforms will use more than one type of material in their part fabrication process. The use of dissimilar materials creates volumes within a part of different strength, flexibility, chemical resistance, color, and others. Depending on the size of the volumes and their compatibility with each other, a geometric feature may be required to lock them together. FDM platforms primarily use multiple extruders with their own material feeds for multi-material printing. Tool changing printers that swap out different extruder assemblies during printing are also starting to enter the consumer market. At the time of writing there is a small but growing number of open-source FFF

printers that feed multiple different filaments though the same nozzle by automatically swapping filaments, having multi-in one out nozzles, splicing stands of different filament together, or using filaments made from multiple polymers. Methods such as these tend to be more limited than multi-tool methods as the extrusion temperature ranges of the materials used need to be similar to prevent printing issues. Multi-material printing is achieved in SLS platforms by depositing different powered materials or binding agents on a given layer. For SLA printing, polyjet systems can use multiple materials in a single print.

The most common use of multi material printing is to use a separate polymer for the support structure. A second polymer that is chemically soluble, in a solution the model material is not, can allow for the creation of more complex part geometries that could potentially be damaged, or would be inaccessible, from mechanical removal. Low temperature FFF applications primarily use polyvinyl alcohol (PVA) or butene-diol vinyl alcohol (BVOH) as a water-soluble support structure. Higher temperature FFF support materials tend to be proprietary formulations based on acrylate polymers that dissolve in basic solutions. Alternatively, for chemically soluble supports, some FFF platforms will use a support material that has poor adhesion to the model material, so it can be mechanically removed more easily.

Digital Materials

Both honeycomb panels and AM parts with infill have a common repetitive unit cell. By repeating this unit cell throughout the interior of a part, or as a structure on its own, a larger structure can be made. Further, by defining properties into this unit cell, information can effectively be encoded into the design, allowing for differing behavior of different parts of the structure. Digital materials can dramatically expand the design space of a structure, allowing for targeted optimization of various properties such as mass to strength ratios, flexibility structural lightweighting, and others. As previously discussed, with certain resin polyjet AM centers, resins can be mixed in real time to form an object that has continuously varying properties. For example, a part could be made to have a gradient from a rigid to a flexible material that would felicitate bending in one axis over another.

6.5   Radiation Effects and Mitigation Strategies

6.5.1   Shielding from the Space Environment

Radiation Shielding has been described as a cost-effective way of mitigating the risk of mission failure due to total ionizing dose (TID) and internal charging effects on electronic devices. In space mission analysis and design, the average historical cost for adding shielding to a mission is below 10% of the total cost of the spacecraft (77). The benefits include reducing the risk of early total ionizing dose electronics failures (78). Some of the key CubeSat and SmallSat commercial electronic semiconductor parts include processors, voltage regulators, and memory devices, which are key components in delivering science and technology demonstration data (79).

Shielding the spacecraft is often the simplest method to reduce both a spacecraft's ratio of total ionizing dose to displacement damage dose (TID/DDD) accumulation, and the rate at which single event upsets (SEUs) occur if used appropriately. Shielding involves two basic methods: shielding with the spacecraft's pre-existing mass (including the external skin or chassis, which exists in every case whether desired or not), and spot/sector shielding. This type of shielding, known as passive shielding, is only very effective against lower energy radiation, and is best used against high particle flux environments, including the densest portions of the Van Allen belts, the Jovian magnetosphere, and short-lived solar particle events. In some cases, increased shielding is more detrimental than if none was used, owing to the secondary particles generated by highly penetrating energetic particles. Therefore, it is important to analyze both the thickness and type

of materials used to shield all critical parts of the spacecraft. Due to the strong omni-directionality of most forms of particle radiation, spacecraft need to be shielded from the full 4π steradian celestial sphere. This brings the notion of "shielding-per-unit-solid-angle" into the design space, where small holes or gaps in shielding are often only detrimental proportionally to the hole's solid angle as viewed by the concerned electrical, electronic and electro-mechanical (EEE) components. Essentially, completely enclosing critical components should not be considered a firm design constraint when other structural considerations exist.

6.5.2 Inherent Mass Shielding

Inherent mass shielding consists of using the entirety of the pre-existing spacecraft's mass to shield sensitive electronic components that are not heavily dependent on location within the spacecraft. This often includes the main spacecraft bus processors, power switches, etc. Again, the notion of "shielding-per-unit-solid-angle" is invoked here, where a component could be well shielded from its "backside" (2π steradian hemisphere) and weakly shielded from the "front" due to its location near the spacecraft surface. It would only then require additional shielding from its front to meet operational requirements. The classic method employed here is to increase the spacecraft's structural skin thickness to account for the additional shielding required. This is the classic method largely due to its simplicity, where merely a thicker extrusion of material is used for construction. The disadvantage to this method is the material used, very often aluminum, is mass optimized for structural and surface charging concerns and not for shielding either protons/ions or electrons. Recent research has gone into optimizing structural materials for both structural and shielding concerns; currently an active area of NASA's Small Business Innovation Research (SBIR) program research and development.

The process to determine exactly how much inherent shielding exists involves using a reverse ray tracing program on the spacecraft solid model from the specific point(s) of interest. After generating the "shielding-per-unit-solid-angle" map of the critical area(s) of the spacecraft, a trade study can be performed on what and where best to involve further additional shielding.

Numerous CubeSat and SmallSat systems use commercial processors, radios, regulators, memory, and SD cards. Many of these products rely on silicon diodes and $metal\ oxide$ $semiconductor\ field\ effect\ transistors$ (MOSFETs) in these missions. A comprehensive NASA guidance document on the use of commercial electronic parts was published for the ISS orbit, which is a low-Earth orbit where the predominant radiation source is the South Atlantic anomaly. The hardness of commercial parts was noted as having a range from 2 – 10 kRad (80). For typical thin CubeSat shielding of 0.20 cm (0.080 in) aluminum, yearly trapped dose is 1383 Rad; with an additional estimated 750 Rad from solar particle events, the total dose increases to 2133 Rad for the ELaNaXIX Mission environment at 85 degrees inclination and 500 km circular orbit (table 6-16) (81). Adding a two-fold increase for the trapped belt radiation uncertainty brings the total radiation near the TID lifetime of many commercial parts (80), even before estimating a SPE TID contribution. The uncertainty of radiation model results of low-Earth orbit below 840 km has been estimated as at least two-fold; Van Allen Belt models are empirical and rely on data in the orbital environment (82). The NASA Preferred Reliability Series "Radiation Design Margin Requirements" also recommends a radiation design margin of 2 for reliability (83). Currently, The Aerospace Corporation proton (AP) (63) and The Aerospace Corporation electron (AE) (85)

Models do not have radiation data below 840 km, and radiation estimates are extrapolated for the lower orbits (82). For spacecraft interplanetary trajectories near the Sun or Earth, the radiation contributions from SPEs will be higher than low-Earth orbit, where there is some limited SPE radiation protection by the magnetosphere. By reducing the total ionizing dose on commercial parts, the mission lifetimes can be increased by reducing the risk of electronic failures on sensitive semiconductor parts. 6.5.3 Shields-1 Mission, Radiation Shielding for CubeSat Structural Design

Shields-1 has operated in polar low-Earth orbit and was launched through the ELaNaXIX Mission in December 2018. The Shields-1 mission increased the development level of atomic number (Z) Grade Radiation Shielding with an electronic enclosure (vault) and Z-grade radiation shielding slabs with aluminum baselines experiments (figure 6.14) (86). Preliminary results in table 6-15 show a significant reduction in total ionizing dose in comparison to typical modeled 0.20 cm (0.080 in) aluminum structures sold by commercial CubeSat providers. The 3.02 g cm^{-2} Z-shielding vault has over 18 times reduction in total ionizing dose compared to modeled 0.20 cm aluminum shielding (81).
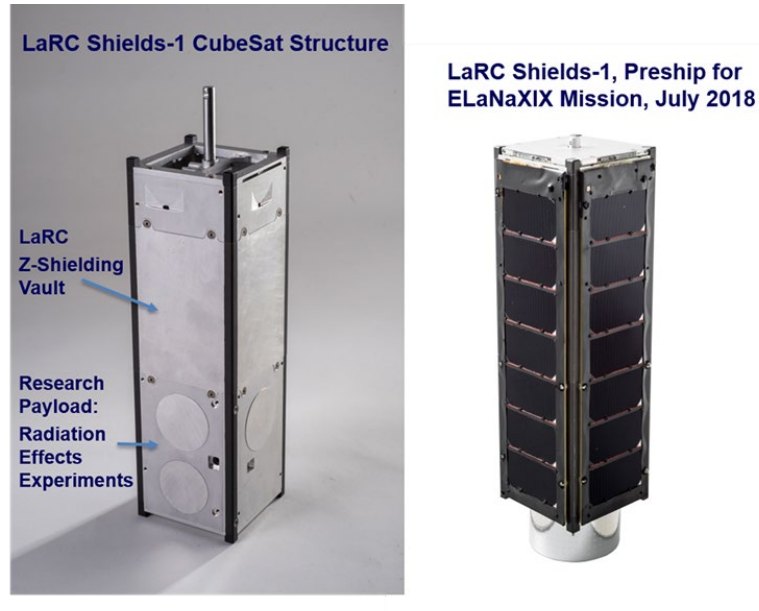
Shields-1 structure and Final Preship Picture with LaRC Z-Shielding Vault and Experiment, Solar Panels and Thermal Radiator

Figure 6.14: Shields-1 Z-shielding structure and final preship picture, ELaNaXIX Mission. Credit: NASA.
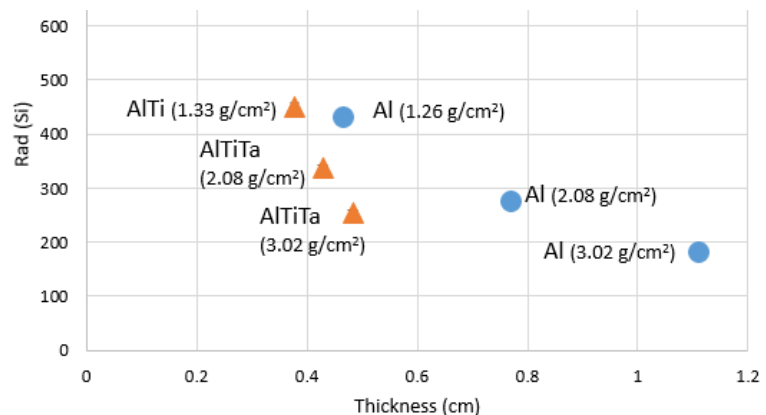
Figure 6.15: SPE Contribution to TID in PLEO, King Sphere Model, ELaNaXIX Shields-1 orbit. Credit: NASA.

Z-shielding enables a low volume shielding solution for CubeSat and SmallSat applications where reduced volume is important. AlTiTa, Z–shielding, at 2.08 g cm^{-2} reduces the dose from a SPE by half when compared to a standard 0.2 cm aluminum structure (figure 6.15). NASA has innovated "Methods of Making Z-Shielding" with patents in preparing different structural shieldings (87)(88)(89)(90), from metals to hybrid metal laminates and thin structural radiation shielding, to enable low-volume integrated solutions with CubeSats and SmallSats (91).

Table 6-15: Shields-1 Experimental Total Ionizing Dose Measurements in PLEO				
Shielding	Areal Density (g/cm^2)	Thickness (cm)	Trapped Belts TID Total (Rad (Si)/Year)	SPE King Sphere Model, (Rad (Si))
Al	0.535	0.198	1383+/-47 #	750+/-5
Al	1.26	0.465	**90.9 +/-2.7 (SL)**	432 +/- 7
Al	1.69	0.624	**84.3 +/-2.5 (SL)**	345 +/- 9
Al	3.02	1.11	**73.6 +/-3.2 (SL)**	183 +/- 11
AlTi	1.33	0.378	**89.7 +/-2.7 (SL)**	451 +/- 6
AlTiTa20	2.08	0.429	**84.3 +/-2.5 (SL)**	338 +/- 6
AlTiTa40	3.02	0.483	**81.9 +/-3.4 (SL) 75.6+/- 3.2 (Vault)**	253 +/- 6

*Shields-1 Experimental total ionizing dose measurements in PLEO in comparison to typical 0.20 cm aluminum shielding commercially available for CubeSats and SPE additional contributions to dose. **Bold values** Shields-1 experimental results. SL = Slab, Vault = Z-Shielding electronics enclosure. # sphere Space Environment Information System (SPENVIS) Multi-layered Shielding Simulation Software (MULASSIS) AP8 Min AE8 Max modeled results. SPE King Sphere Model SPENVIS MULASSIS modeled results.*

6.5.4 Ad Hoc Shielding

There are two types of ad hoc shielding used on spacecraft: spot shielding, where a single board or component is covered in shield material (often conformally), and sector shielding, where only critical areas of the spacecraft have shielding enhancement. These two methods are often used in concert as necessary to further insulate particularly sensitive components without unnecessarily increasing the overall shield mass and/or volume. Ad hoc shielding is more efficient per unit mass than inherent mass shielding because it can be optimized for the spacecraft's intended radiation environment while loosening the structural constraints. The most recent methods include multiple layer shields with layer-unique elemental atomic numbers which are layered advantageously (often in a low-high-low Z scheme), known as "graded-Z" shielding, and advanced low-Z polymer or composite mixtures doped with high-Z, metallic micro-particles. Low-Z elements are particularly capable at shielding protons and ions while generating little secondary radiation, where high Z elements scatter electrons and photons much more efficiently. Neutron shielding is a unique problem, where optimal shield materials often depend on the particle energies involved. Commercial options include most notably Tethers Unlimited's VSRS system for small spacecraft, which was specifically designed to be manufactured under a 3D printed fused

filament fabrication process for conformal coating applications (a method which optimizes volume and minimizes shield gaps).

6.5.5 Charge Dissipation Coating

The addition of conformal coatings over finished electronic boards is another method to mitigate electrostatic discharge on sensitive electronic environments. Arathane, polyurethane coating materials (92), and HumiSeal acrylic coatings (93) have been used to mitigate discharge and provide limited moisture protection for electronic boards. This simple protective coating over sensitive electronic boards supports mission assurance and safety efforts. Charge dissipation films have decreased electrical resistances in comparison to standard electronics and have been described by NASA as a coating that has volume resistivities between $10^8 - 10^{12}$ ohm-cm. In comparison, typical conformal coatings have volume resistivities from $10^{12} - 10^{15}$ ohm-cm (50).

6.5.6 LUNA Innovations, Inc. XP Charge Dissipation Coating

The XP Charge Dissipation Coating has volume resistivities in the range of $10^8 - 10^{12}$ ohm-cm (table 6-16) and is currently developing space heritage through the NASA MISSE 9 mission and Shields-1 (94). The XP Charge Dissipation Coatings were developed through the NASA SBIR program from 2010 to present for extreme electron radiation environments, such as outer planets, medium-Earth, and geostationary orbits, to mitigate charging effects on electronic boards.

Table 6-16: XP Charge Dissipation Coating and Commercial Conformal Coating Resistivity Comparisons	
Material	**Volume Resistivity (Ohm-cm)**
XP Charge Dissipation Coating	$10^8 - 10^{12}$, 4.7 x 10^9 at 25°C
Arathane 5750 A/B	9.3 X 10^{15} at 25°C, 2.0 X 10^{13} at 95°C
Humiseal 1B73	5.5 x 10^{14} Ohms (Insulation Resistance per MIL-I-46058C)
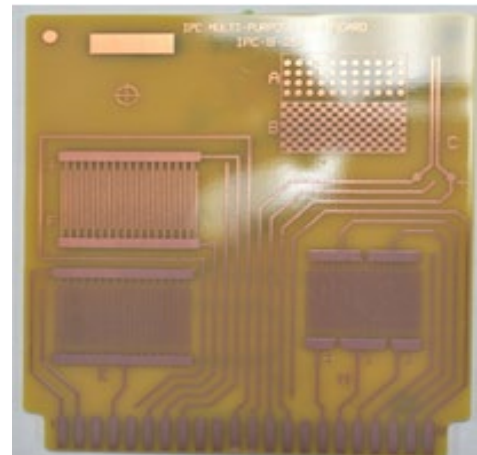
Figure 6.16: Transparent LUNA XP Charge Dissipation Coating on an electronic board. Credit: LUNA Innovations, Inc.

The LUNA XP Charge Dissipation Coating has reduced resistance compared to typical commercial conformal coatings, which reduces surface charging risk on electronic boards. LUNA XP Coating (figure 6.16) on an electronic board has transparency for visual parts inspection. For extreme radiation environments, a combination of radiation shielding, and charge dissipation coating reduces the ionizing radiation that contributes to charging and provides a surface pathway for removing charge to ground (50).

6.6 Summary

This chapter has been updated with the current status of structures, materials, and mechanisms for small satellite missions. Additions include custom structure references with the dimensional and material requirements of integrating deployment systems, new mechanisms technology to reflect the ongoing growth in SmallSat mechanical devices, and more commercially procured deployable booms and larger CubeSat primary structures (12U and 16U), as well as the upcoming DiskSat structure. The radiation environment section, state-of-the-art radiation shielding, and charge dissipation materials have been updated. Reflecting the fast pace of development in

additive manufacturing, a selection of available thermoplastics and resin-based materials suitable for different TRL levels have been detailed.

There has been high focus on deployment mechanisms with respect to light weighting and reliability. Small spacecraft subsystems related to antenna booms, gravity gradients, stabilization, sensors, sails, and solar panels are some examples. These technologies are gaining space heritage through operations and more often are being included in mission planning. The growth of these deployment mechanisms increases the capabilities of SmallSat technology and will be a continued focus in the next edition of this report.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email so someone may contact you further.

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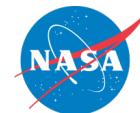
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Chapter Contents

Chapter Glossary

(APG)	Annealed Pyrolytic Graphite
(ARC)	Ames Research Center
(ATA)	Active Thermal Architecture
(BIRD)	Bi-Spectral Infrared Detection
(ESPA)	EELV Secondary Payload Adapter
(FEP)	Fluorinated Ethylene Propylene
(FETS)	Folding Elastic Thermal Surface
(FOX)	Flat-Plate Heat Pipe On-Orbit Experiment
(GFTS)	Graphite Fiber Thermal Straps
(GSFC)	Goddard Space Flight Center
(HEC)	High Efficiency Cooler
(IR)	Infrared
(ISS)	International Space Station
(KGS)	Kaneka Graphite Sheets
(LPT)	Linear Pulse Tube
(MLI)	Multi-Layer Insulation
(MPFL)	Mechanically Pumped Fluid Loop
(MWIR)	Midwave Infrared
(NLAS)	Nanosatellite Launch Adapter System
(OHP)	Oscillating Heat Pipe
(PCM)	Phase Change Materials
(P-POD)	Poly-Picosatellite Orbital Deployer
(PFL)	Pumped Fluid Loop
(PGF)	Pyrovo Pyrolytic Graphite Film
(PGS)	Pyrolytic Graphite Sheets
(PRISM)	Portable Remote Imaging Spectrometer
(q_{albedo})	Solar heating reflected by the planet
(Q_{gen})	Heat generated by the spacecraft
($Q_{out,rad}$)	Heat emitted via radiation
($q_{planetshine}$)	IR heating from the planet
(q_{solar})	Solar heating
(Q_{stored})	Heat stored by the spacecraft
(SDL)	Space Dynamics Laboratory

(SI)	International System of Units
(SPOT)	Standard Passive Orbital Thermal-Control
(SST)	Small Satellite Technology
(TAFTS)	Two Arm Flexible Thermal Strap
(TEC)	Thermoelectric Coolers
(TMT)	Thermal Management Technologies
(TRL)	Technology Readiness Level
(TSU)	Thermal Storage Unit
(UAM)	Ultrasonic additive manufacturing
(ULP)	Ultra-Low Power
(VDA)	Vacuum Deposited Aluminum

7.0 Thermal Control

7.1 Introduction

All spacecraft components have a range of allowable temperatures that must be maintained to meet survival and operational requirements during all mission phases. Spacecraft temperatures are determined by how much heat is absorbed, stored, generated, and dissipated by the spacecraft. Figure 7.1 shows a simplified overview of heat exchange from a satellite orbiting Earth, but the heating principles apply to any planet or body a spacecraft orbits.

Q_{gen}= Heat generated by spacecraft
q_{solar}= Solar heating
q_{albedo}= Solar heating reflected by planet
$q_{planetshine}$= Infrared heating from planet
$Q_{out,rad}$= Heat emitted via radiation
Q_{stored}= Heat stored by the spacecraft
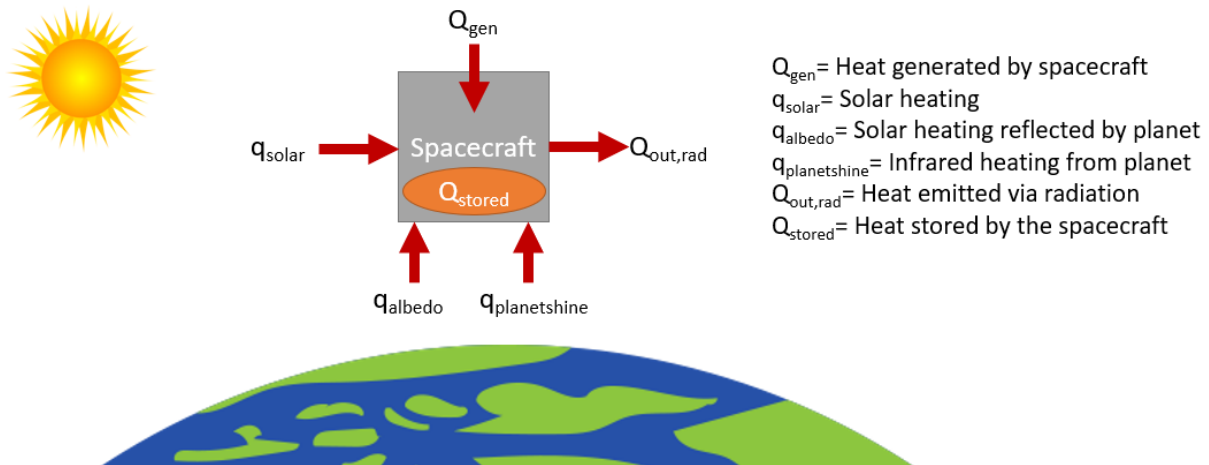
Figure 7.1: Orbiting spacecraft heating simplified overview. Q_{gen}, $Q_{out,rad}$, and Q_{stored} are represented as heat values, Watts per square meter in International System of Units (SI), whereas q_{solar}, q_{albedo}, and $q_{planetshine}$ are represented as heat fluxes. Credit: NASA.

The heat exchange depends on several factors listed below. Solar absorptivity and infrared (IR) emissivity are surface optical properties referenced below and described further in section 7.2.1: Paints, Coatings, and Tapes. Thermal control of a spacecraft is achieved by balancing the energy as shown in Equation 1:

$$q_{solar} + q_{albedo} + q_{planetshine} + Q_{gen} = Q_{stored} + Q_{out,rad} \qquad (1)$$

- Q_{gen} (heat generated by the spacecraft) depends on the power dissipation of spacecraft components.
- The amount of q_{solar} (solar heating) absorbed by the spacecraft depends on the solar flux, which is determined by distance to the sun, the surface area viewing the sun (view factor), and the solar absorptivity of that surface.
- The amount of q_{albedo} (solar heating reflected by the planet) absorbed by the spacecraft depends on the planet, the surface area viewing the planet (view factor), and the solar absorptivity of that surface.
- The amount of $q_{planetshine}$ (IR heating from the planet) absorbed by the spacecraft depends on the planet, the surface area viewing the planet (view factor), and the IR emissivity of that surface.
- $Q_{out,rad}$ (heat emitted via radiation) includes the surface area designated as radiator space, the IR emissivity of the surface, and the difference in temperature between the spacecraft radiator and the heat sink to which it is dissipating, typically and most effectively deep space. $Q_{out,rad}$ also include heat lost through insulation or other surfaces not specifically intended to function as radiators.

- Q_{stored} (heat stored by the spacecraft), is based on the thermal capacitance of the spacecraft.

Temperatures are regulated with passive and/or active thermal management technology and design methods. Many of the same thermal management methods used on larger spacecraft are also applicable to SmallSats and given the increased interest in small spacecraft over the last decade, some spacecraft thermal control technologies have been miniaturized or otherwise adapted to apply to SmallSats. Thermal control methods and technologies as applied to large spacecraft are considered state-of-the-art for the purposes of this review but may have a Technology Readiness Level (TRL) value less than 9 for small spacecraft applications.

Challenges of designing thermal control systems for SmallSats stem from several intrinsic properties, summarized in table 7-1. Due to the small size and volume limitations inside the deployer or around deployables, there is often no room for multi-layer insulation (MLI) for CubeSats. The thermal solution must be worked out as a coatings problem, exposing the CubeSat to more transient thermal behaviors.

Table 7-1: SmallSat Thermal Control Challenges	
SmallSat Property	**Challenge**
Low thermal mass	The spacecraft is more reactive to changing thermal environments.
Limited external surface area	There is less real estate to be allocated to solar cells, designated radiator area, and/or viewports required for science instruments.
Limited volume	There is less space for electronic components, science instruments, and thermal control hardware. Components can be more thermally coupled, and it can be harder to isolate different thermal zones.
Limited power	There is less power available for powered thermal control technology.
Power Density	There is a big challenge to dissipate power as electronics are stacked close to each other, sometimes with no direct path to radiator.
MLI Edge Effects	MLI can "short" along the edges resulting in degraded performance, not specific to SmallSats but more of a general spacecraft issue.

The information described in this section is not exhaustive but provides an overview of current state-of-the-art thermal technologies and their development. The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

7.2 State-of-the-Art – Passive Systems

Passive thermal control maintains component temperatures without using powered equipment. Passive systems are typically associated with low cost, volume, weight, and risk, and are advantageous to spacecraft with limited, mass, volume, and power, like SmallSats and especially

CubeSats. MLI, coatings/surface finishes, interface conductance, heat pipes, sunshades, thermal straps, interface materials, and louvers are some examples of passive thermal control technology.

In addition to passive thermal control technology, structural and electrical design methods also contribute to managing the thermal environment, passively. These design methods include:

- Material selection
  - o Structural component materials chosen based on needed heat transfer through the structure. A high or low thermal conductivity may be more advantageous based on the application.
- Spacecraft orientation
  - o If orientation is not dictated by science objectives, changing the orientation of the spacecraft can help maintain temperatures.
  - o Changing orientation may only be needed during certain mission phases, such as science operation, if larger amounts of heat are dissipated.
  - o This method is often used in conjunction with other thermal control methods, such as orienting the spacecraft so that the radiator area can face deep space.
- Thermal interfaces:
  - o Definition of the thermal contact between components through specific mounting methods can thermally isolate components or allow more heat to be transferred to a structural element (or radiator area) when each is needed. For example:
    - ▪ Heat transfer can be reduced by mounting a component through multiple stacked washers with low thermal conductivity.
    - ▪ Heat transfer can be increased by mounting components with more fasteners (if applicable) and can be further increased by using thermal interface materials between a component and mounting surface.
- Circuit board design considerations, include:
  - o Copper layers within each board can be increased, in number or thickness, to conduct heat away from electrical components through the boards to their structural connection points.
  - o Circuit boards can be mounted to increase heat transfer away from the boards to the structure, such as by mounting with wedge locks.

Table 7-2: SmallSat Passive Thermal Technologies	
Passive Technology	**Description**
Sprayable Thermal Control Coatings	Specialized liquid coatings applied to the spacecraft surfaces to manage and regulate the ratio of absorptivity and emissivity for optimal energy balance and thermal performance.
Films, Tapes, and MLI	Materials used to modify spacecraft surface properties to manage and regulate temperature.
Thermal Straps	Provide a conductive link between a heat source and thermal sink to conductively transfer heat.
Thermal Contact Conductance and Bolted Joint Conductance	Heat transfer by "contact" conductance between two surfaces pressed together by uniform pressure can be varied by using interface filler materials and conductive gaskets.
Sunshields	Deployed material that reduces the amount of incident solar flux on a spacecraft.
Thermal Louvers	Controls heat transfer between spacecraft surfaces.

Deployable Radiators	Dedicated surface for dissipating excess heat via radiative heat transfer.
Phase Change Materials	Energy is absorbed as material within a metal compartment changes phase because of exposure to a heat source.
Thermal Switches	An electromechanical device which controls the flow of electrical current in response to temperature change.

Table 7-2 provides a description of current passive thermal control technology as applied to SmallSats. One key factor to consider when choosing thermal control technology, both passive and active, is the temperature limit of the technology itself. The goal is to use the appropriate technology to maintain the temperatures of spacecraft components within their limits, but the technology used to achieve this also has limits. It is recommended to verify that the technology used is applicable to the given design not only with respect to needed function, but to the environment (temperature limits) as well.

7.2.1 Sprayable Thermal Control Coatings, Tapes, and MLI

In a vacuum, heat is transferred only by radiation and conduction, with no convection. The internal environment of a fully enclosed small satellite is usually dominated by conductive heat transfer, while heat transfer to/from the outside environment is driven via thermal radiation. Many missions with electrical surface resistivity requirements use coatings with these properties to handle these surface charging concerns (this also applies to MLI). For SmallSat missions where extensive use of MLI is not practical (see MLI paragraph on page 207), a mixed use of several different coatings is needed to achieve optimal energy balance and thermal performance. There are also coatings that better approximate the use of MLI by being relatively low emissivity (such as 0.25) with a lower alpha (0.1) so they don't overheat in the sun. These are colloquially known as tailorable emittance coatings that involve some oxide depositions starting with a vacuum deposited aluminum (VDA) base to drive up the emissivity while keeping the alpha low.

The thermal radiation band of the electromagnetic spectrum is between 0.1 and 100 μm in wavelength, as shown in Figure 7.2. Outside of the thermal radiation waveband, electromagnetic energy generally passes through objects or has very little heat energy under practical conditions. Thermal analyses are typically conducted using a two waveband absorptance model which subdivides the thermal energy spectrum into solar (< 3 μm) and IR (> 3 μm) wavelengths. Thermal radiation heat transfer is controlled by using materials that have specific optical surface properties, namely: solar absorptivity and IR emissivity. Solar absorptivity governs how much incident heating from solar radiation a spacecraft absorbs, while IR emissivity determines how much heat a spacecraft emits to space, relative to a perfect blackbody emitter, and what fraction of thermal radiation from IR sources (e.g., the Earth, Moon, any particularly hot spacecraft components) are absorbed by that spacecraft surface.

The surface properties of a spacecraft can be modified by adding specialized paints, coatings, surface finishes, or adhesive tapes, depending on the needs of the spacecraft. For example,
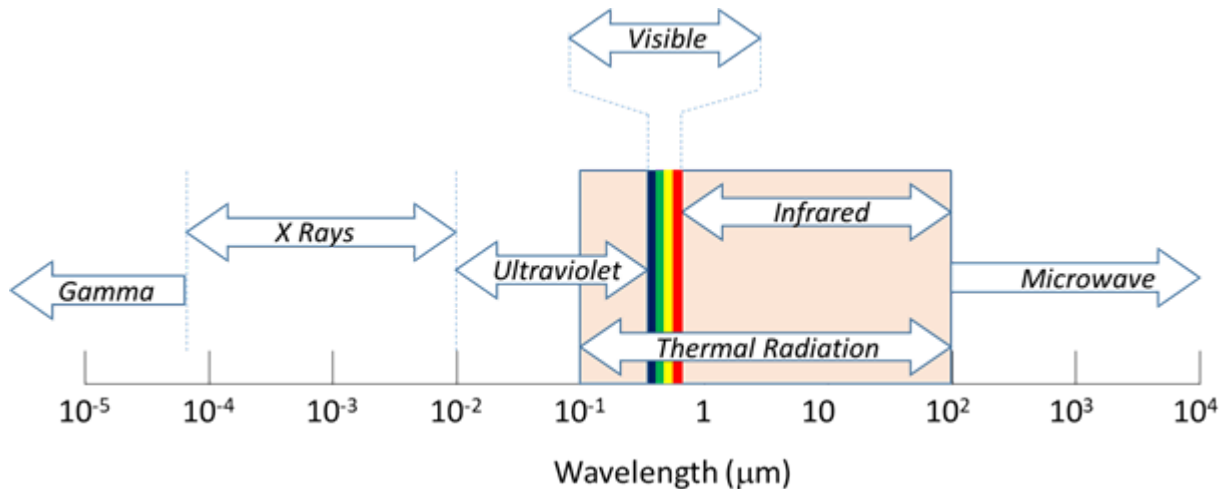

Figure 7.2: Electromagnetic spectrum showing the range of thermal radiation. Credit: NASA.

matte black paint has a high solar absorptivity and high IR emissivity for surfaces required to absorb a high percentage of solar heating and emit a high percentage of spacecraft heat. Alternatively, matte white paint has a low solar absorptivity and high IR emissivity (1) for surfaces required to absorb a low percentage of solar heating and emit a high percentage of spacecraft heat (e.g., radiator). Second-surface silver Fluorinated Ethylene Propylene (FEP) tapes offer excellent performance as radiator coatings, reflecting incident solar energy (low solar absorptivity) while simultaneously emitting spacecraft thermal energy efficiently (high IR emissivity). The selection between paints, coatings, and tapes depends on the application. Tape is typically easy to apply and remove, is comparatively inexpensive, and has a longer usable lifetime than paint. Tape can also be added later in the assembly process if changes to thermal control need to be made after the spacecraft has already begun assembly. Some tapes, however, must be handled carefully to maintain optical properties and can be difficult to bond properly to curved surfaces.

While optical films/tapes may have solar absorptivities and IR emissivities very close to the datasheet value as received from the manufacturer, these two values will change throughout the mission. Coatings will darken due to atomic oxygen bombardment (low earth orbit), UV light (high earth orbit, GEO, and beyond), and other cosmic rays. This will increase the solar absorptance values, some quite substantially. The longer the mission duration, the more the optical coating's properties will change. It's important the thermal engineer take this into consideration; thermal performance at the beginning-of-life (BOL) of a tape/film may not be the same at its end-of-life (EOL). Thermal analysis should take this into consideration and bias the optical property values in their thermal models to account for this degradation. BOL measurements from the datasheets (used for cold biased conditions) should not be used for EOL modeling, which are hot biased conditions.

Coatings and paints must often be applied earlier in the assembly process but can cover non-flat surfaces more easily. However, some paints, like Parker-Lord's Aeroglaze 306/307, are expensive and require extensive and highly specialized processes to apply. Different options may also have different temperature limits. All these factors must be considered with regard to the needed application when selecting the final solution. Table 7-3 has a list of non-exhaustive sprayable thermal control coatings that include application difficulty and specific notes on some of the products.

One example, BioSentinel, a 6U spacecraft that launched as a secondary payload on the Artemis I mission in 2022, made extensive use of Sheldahl metallized tape coatings and second-surface silvered FEP tapes to control its external thermal radiative properties and overall energy balance (2). Another example, Picard, a 150 kg SmallSat, used white paint on the Sun pointing face to reduce the amount of solar flux absorbed and lower temperatures. For most small spacecraft projects to date, adhesive tapes, such as silver FEP, or other standard surface finishes (e.g., polishing, anodize, alodine) have been the preferred choices.

Table 7-3: Sprayable Thermal Control Coatings					
Manufacturer	**Product**	**Color**	**Cost**	**Difficulty to Apply***	**Notes**
Socomore	Z306 Polyurethane	Black	$	1	
	Z307 Polyurethane	Black	$$	1	
	A276 Polyurethane	White	$	1	Not UV stable
Huntington Ingalls Industries (formerly Alion and IITRI)	Z93P Silicate	White	$$	3	**
	Z93-C55 Silicate	White	$$$	4	Thickness control paramount **
	S13GP:6N/LO-1 Silicone	White	$$$	1	Dedicated equipment/cross contamination
AZ Technology	AZ-93 Silicate	White	$$	3	**
	AZ2000 Silicate	Off-white	$$$	4	May require specialty, epoxy primer **
	AZW/LA-II Silicate	White	$$$$	5	Thickness control paramount **

*1 not difficult, 5 extremely
**All silicates require temperature and RH control

A MLI blanket is typically comprised of multiple inner layers of a thin material with low IR emissivity (usually 10 to 20 layers) and a durable outer layer. The amount of radiative heat transfer allowed is limited by the many layers of reflectors. The low IR emissivity layers are either embossed or alternated with thin netting to limit conduction through the layers. Perforations may be added to allow the MLI to vent trapped gas once arriving on-orbit, although this can also be achieved via edge venting. MLI is used as a thermal radiation barrier to both protect spacecraft from incoming solar and IR flux, and to prevent undesired radiative heat dissipation to space. It is commonly used to maintain temperature ranges for components in-orbit.

MLI is delicate and performance drops drastically if compressed (causing a thermally conductive "short circuit"), so it should be used with caution or avoided altogether on the exterior of small satellites that fit into a deployer (e.g., P-POD, NLAS). MLI blankets can also pose a potential snagging hazard in these tight-fitting, pusher-spring style deployers. Additionally, MLI blankets tend to drop efficiency as size decreases because heat transfer through the blanket increases closer to the blanket edges, and the specific attachment method has a large impact on performance because attachment to the spacecraft creates a heat path. Due to these challenges, MLI generally does not perform as well on small spacecraft (more specifically CubeSat form factors) as on larger spacecraft. Surface coatings are typically less delicate and more appropriate

for the exterior of a small spacecraft that will be deployed from a dispenser. Internal MLI blankets that do not receive direct solar thermal radiation can often be replaced by a variety of low emissivity tapes or coatings that perform equally well in that context, using less volume and at a potentially lower cost.

Table 7-4: Films, Tapes and MLI		
Manufacturer	**Film Product**	**Typical Solar Absorptivity BOL**
Astral Technologies Unlimited	StaMet/100XC Kapton	0.5
	StaMet/275XC Kapton	0.56
Astral Technologies Unlimited Multek/Sheldahl	0.010" FEP/Ag/Inconel	0.08
	0.005" FEP/Ag/Inconel	0.07
	ITO/0.010" FEP/Ag/Inconel, perforated	0.09
	ITO/0.005" FEP/Ag/Inconel, perforated	0.08
	Germanium/100XC Kapton	0.55
Dunmore Multek/Sheldahl	VDA/200HN/PSA	0.08
	200HN/VDA/PSA	0.41
	100HN/VDA/PSA	0.38
	VDG/200HN/PSA	0.2
NASA GSFC	GSFC Silver Composite Coating/ 200HN Kapton	0.1
	GSFC Aluminum Composite Coating/200HN Kapton	0.14
	GSFC Aluminum Composite Coating/100XC Kapton	0.2
	GSFC Aluminum Composite Coating/275XC Kapton	0.28
	GSFC Tailorable Emittance Coating/200HN Kapton	0.14
Thales	Optical Solar Reflector CMX 0.005"	0.08

7.2.2 Thermal Straps

A thermal strap is a flexible, thermally conductive link added between a heat source and sink to conductively transfer heat. They are often used between high heat dissipating chips or components and a chassis wall or other radiator surface. Their flexibility prevents the addition of structural loads. Thermal straps can be made of metal, traditionally copper or aluminum, or high conductivity carbon materials, such as graphite. They can be formed of multiple foil sheets or wound cables (also referred to as

Figure 7.3: Flexible thermal straps. Credit: Thermal Management Technologies. Q-Strap. Credit: Redwire Space.

ropes and braids), with end blocks at each end to hold the sheets/cables in place and to mount or otherwise attach to the needed surfaces. Straps with more than two end blocks and multiple material combinations can also be produced and have been used on large spacecraft. Advances in thermal straps are being developed to further increase heat transfer capability and custom

Figure 7.4: Thermal strap design with aluminum foils, copper foils, and PGS in aluminum end blocks (above), and their respective measured thermal conductance (left). The dashed lines connecting data points are based on material thermal conductivity curves. Credit: SDL.

thermal straps are now commonly fabricated and tested using graphite material due to improved thermal conductivity.

Space Dynamics Laboratory (SDL) developed solderless, flexible thermal straps without solder, epoxy, or other filler materials. Figure 7.4 shows a comparison of the as-tested conductance for the same strap geometry fabricated with three different foil materials of aluminum, copper, and pyrolytic graphite sheets (PGS), showing the PGS increases the overall thermal conductivity. SDL supplied Utah State University with a PGS strap for the Active Thermal Architecture (ATA) project sponsored by the Small Spacecraft Technology (SST) program. A follow-on to this ATA project is referenced in the cryocooler section. Boyd Corporation has designed thermal straps using their patented k-core technology that has an annealed pyrolytic graphite (APG) core within an encapsulating structure (4). Technology Applications, Inc. has specialized in testing and developing Graphite Fiber Thermal Straps (GFTS), with flight heritage only on larger spacecraft missions (Orion and Spice) (5). The Pyrovo Pyrolytic Graphite Film (PGF) thermal straps developed by Thermotive have flown in optical cooling applications for high altitude cameras and avionics on larger spacecraft, on JPL's ASTERIA CubeSat in 2017, and on the Mars 2020 rover mission (6).

Table 7-5: Thermal Straps		
Manufacture	**Product**	**Material**
Technology Applications, Inc.	Copper Cable	Copper
	Copper Foil	Copper
	Aluminum Foil	Aluminum 1100
	PGL	Pyro Graphite
	GFTS	Graphite Fiber
Thermotive	Copper Foil	Copper
	Aluminum Foil	Aluminum 1100
	Pyrovo	Pyro Graphite Film
Space Dynamics Laboratory	PGS	Pyrolytic Graphite Sheet
	Aluminum Foil	Aluminum 99.999% and 1235
	Copper Foil	Copper 101 and 110
	Copper Braid	Copper Braid
Redwire Space	Q-Strap	High-k graphite material

Table 7-5: Thermal Straps		
Manufacture	**Product**	**Material**
Thermal Management Technologies	TMT010-200 Series	Copper Foil
	TMT010-300 Series	Copper Braid (small)
	TMT010-400 Series	Copper Braid (large)
	Customized	Copper braid or foil or using aluminum foil
Boyd Corporation	k-Core Graphite-Encapsulated	Pyrolytic graphite

7.2.3 Thermal Contact Conductance and Bolted Joint Conductance

Two surfaces pressed together by uniform pressure will transfer heat via "contact" conductance. This conductance value is a product of the heat transfer coefficient and the contact surface area. The heat transfer between such interfaces can be varied by using interface filler materials and conductive gaskets (7).

Bolted joints experience non-uniform pressure, creating a more complex heat transfer scenario where the conductance depends on screw size, torque, surface properties, and other values. The conductance can be varied by changing torque, surface properties and finishes, and materials. Table 7-6 provides the conductance of various screws (7).

Table 7-6: Bolted Joint Thermal Conductance Design Guideline		
	Conductance [W/K]	
Screw Size	**Small Stiff Surface**	**Large Thin Surfaces**
2-56	0.21	0.105
4-40	0.26	0.132
6-32	0.42	0.176
8-32	0.80	0.264
10-32	1.32	0.527
1/4-28	3.51	1.054

7.2.4 Thermal Interface Materials and Conductive Gaskets

Thermal interface materials can be inserted between two components to increase the conductive heat transer between them. They are often made as a sheet or pad of material sandwiched between surfaces, but there are many different types that vary in material, thickness, thermal conductivity, temperature limits, and vacuum-compatibility. Thermal interface materials can also be a grease or paste.

Thinner sheets of materials are commonly used between heat dissipating electronics boxes and mounting surfaces to thermally sink the hot components to a colder surface and reduce the temperature of the electronics. The performance of these types of materials depends on reaching a certain contact pressure between components to ensure the needed heat transfer. Laird Performance Materials has developed many different types of thermal interface materials for a variety of applications. For example, their Tflex series, shown in figure 7.5, is about 1 to 5 mm thick with a thermal conductivity of 6 W mK^{-1} (8), whereas their Tgon series of materials are about 0.13 to 0.5 mm thick with a thermal conductivity of 5 W mK^{-1} (9).
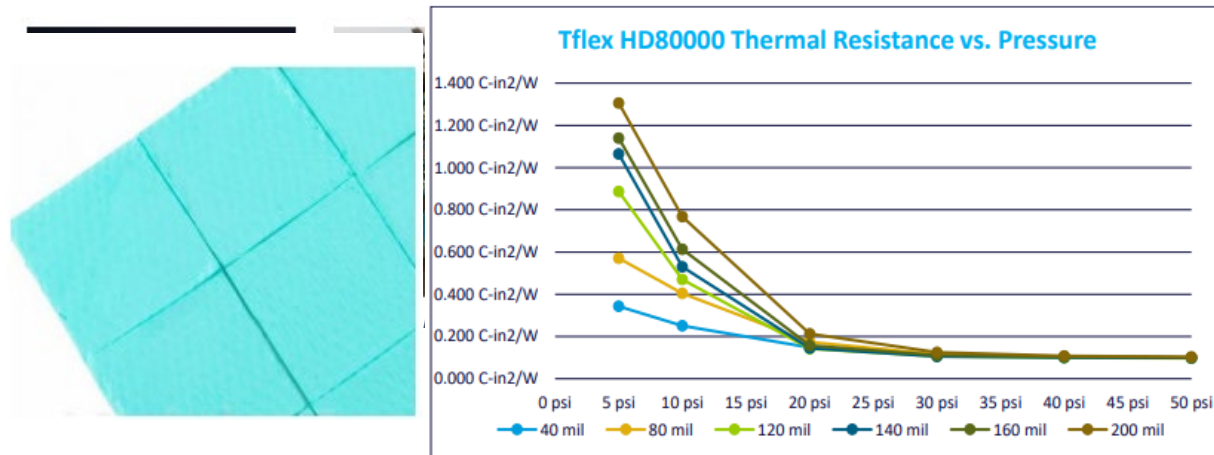
Figure 7.5: Laird Tflex HD80000 series sheets (left) and thermal resistance vs. pressure (right). Credit: Laird Performance Materials.

Thicker pad-like materials, such as Henkel brand GAP PADs®, are often used between high heat dissipating chips on an electronics boards and the electronics enclosure. These are also made to fit a variety of applications, with varying material, thickness, conformability, tear-resistance, electrical isolation, thermal conductivity, and more (10). Several additional thermal interface materials developed by Henkel Corporation are shown in figure 7.6. For conductive gaskets and interface materials, see table 7-7.

Table 7-7: Thermal Interface Materials and Conductive Gaskets		
Manufacturer	**Product**	**Recommended Conductance (W/in2-C)**
Parker Chomerics	Nusil CV-2946	3.5
NeoGraf	Egraf HT-1210	2
NeoGraf	Egraf HT-C3200	2.5
Parker Chomerics	Chotherm 1671	0.75
Parker Chomerics	Therm a gap 579	0.9
Laird Performance Materials	T-PLI 220	1.5
Henkel Corporation	Gap Pad 300	1.25
Indium Corporation	Indium foil 0.005"	0.9

7.2.5 Sunshields

A sunshield, or sunshade, is an often-deployed device made up of a material with low solar absorptivity that reduces the amount of incident solar flux impinging a spacecraft, by blocking the view to the sun. Sunshields are commonly used for spacecraft thermal control, although only recently designed for small spacecraft.

7.2.6 Thermal Louvers

Thermal louvers are thermally activated shutters that regulate how much heat the louvered surface can dissipate. As the louvers open, the average IR emissivity of the surface changes, changing how much heat the surface dissipates. Full-sized louvers on larger spacecraft have high efficacy for thermal control, however, integration on small spacecraft is challenging. Typical spacecraft louvers are associated with a larger mass and input power, which are both limited on small spacecraft. Although commonly defined as active thermal control, here we consider louvers as a passive thermal control component because the CubeSat-adapted design considered does not require a power input from the spacecraft.

7.2.7 Deployable Radiators

A radiator is a dedicated surface for dissipating excess heat via radiative heat transfer, and has a high IR emissivity and low solar absorptivity—an optical property combination typically referred to as "radiator properties." A deployable radiator is stowed during transit or when the radiator is not needed and deployed when excess heat dissipation is required. Deployable radiators on small spacecraft can be challenging due to volumetric constraints. While paint has been widely used to create efficient radiator surfaces on larger spacecraft, the relatively limited available external surface area on SmallSats that already have body-mounted solar cells reduces the potential for creating dedicated radiative surfaces. For a system that requires a large amount of heat dissipation, a passive deployable radiator would greatly enhance thermal performance by increasing the available radiative surface area. Since deployable radiators may be needed because of a lack of radiator surfaces on the spacecraft body due to body-mounted solar cells, an alternate approach (perhaps more common for CubeSats) is to use the chassis body as the radiator area and have a deployable solar array. Also, deployed solar arrays would be able to radiate off a high emissivity/low solar absorptance backside for improved thermal management of the array. Figure 7.7 shows an example of a deployable thermal dissipation technology from Redwire Space.
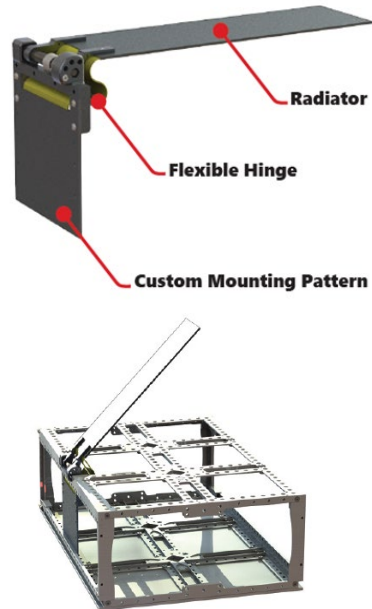
Figure 7.7: Q-Rad deployable radiator technology. Credit: Redwire Space.

A novel deployable radiator is being developed by JPL, California Polytechnic San Luis Obispo, and California State Los Angles. At the core of this technology is an Additively Manufactured Deployable Radiator with embedded Oscillating Heat Pipes (AMDROHP) that enables heat to be efficiently transported across moving interfaces. The current AMDROHP radiator design is shown in figure 7.8 and consists of an evaporator and a condenser plate, and a series of flexible joints connecting the two plates. AMDROHP can be stowed within a 3U CubeSat and can be passively deployed without use of an actuator. This AMDROHP technology is currently in the testing

Figure 7.8: Rendering of an AMDROHP radiator design. Credits: California State Los Angles, Jet Propulsion Laboratory, and California Polytechnic San Luis Obispo.

phase and further design optimization is ongoing. This project s funded by NASA's SST program in the University SmallSat Technology Partnerships initiative within the 2020 cohort.

7.2.8   Heat Pipes

A traditional heat pipe is a passive device comprised of a metal container (pipe) that holds a liquid under pressure and has a porous wick-like structure within the container. When heat is applied to one end of the tube, the liquid inside the tube near the hot end vaporizes into a gas that moves through the tube to the cooler end, where it condenses back into a liquid. The wick transports the condensed liquid back to the hot end via capillary action. There are also more complicated and non-passive types of heat pipes such as variable conductance, diode, and loop heat pipes, which are not further explained in this document.

Heat pipes are an efficient passive thermal transfer technology, where a closed-loop system transports excess heat via temperature gradients, typically from electrical devices to a colder surface, which is often either a radiator itself, or a heat sink that is thermally coupled to a radiator. Traditional constant conductance heat pipes are cylindrical in shape with a grooved inner wick, like those used on Bi-Spectral Infrared Detection (BIRD), a 92 kg satellite launched in 2001, to join satellite segments (13). Heat pipes can also be configured as flat plates with tubing sandwiched between two plates and charged with a working fluid inside. SDS-4, a 50 kg small spacecraft launched in 2012, incorporated the Flat-Plate Heat Pipe On-Orbit Experiment (FOX), developed at JAXA (14).

The FlexCool heat pipe by Redwire Space is a bent, flat heat pipe developed as a cross between a heat pipe and a thermal strap that can be customized for higher heat fluxes by increasing the thickness. This heat pipe flew on TechEdSat-10, a 6U CubeSat deployed from the ISS in 2020, to thermally manage the radio. An image of this technology in a 1U CubeSat model is shown in figure 7.9.
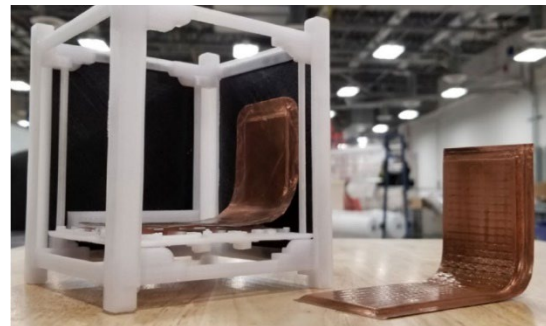
Figure 7.9: FlexCool conformable micro heat pipe before integrating with TechEdSat-10 DVB-S2 radio. Credit: Redwire Space.

7.2.9   Phase Change Materials and Thermal Storage Units

A phase change material used as a thermal storage unit is made up of a material (e.g., wax) within a metal housing with a heat source attached so that, as the source conducts heat to the enclosure, the phase change material within absorbs the energy as it changes phase (usually from solid to liquid). Then, as the heat source energy output reduces, the phase change material releases the energy as it changes back to its initial phase (usually from liquid to solid). Owing to the low thermal conductivity of the phase change material, the metal housing must conduct heat into the phase change medium for efficient solidification or melting. See table 7-8 for common phase change materials.

Table 7-8: Phase Change Materials		
Material	Melting Point (°C)	Heat of Fusion (kJ/kg)
Saliclic Acid	159	199
Bee Wax	61.8	177
Paraffin	20-60	140-280
Polyethylene glycol 600	20-25	146

Table 7-8: Phase Change Materials		
Material	**Melting Point (°C)**	**Heat of Fusion (kJ/kg)**
Glycerol	18	199
Acetic acid	17	187
Water	0	333
Isopropyl alcohol	-89	88
Butane	-135	76
Ethane	-172	93
Methane	-183	59

Thermal storage units are typically used with components that will experience repeated temperature cycling, or to slow down the temperature transient caused by a high heat dissipation event, or a temporary change in the environment such an eclipse. They can be challenging to apply to CubeSats and other small satellites because of the extra mass of the housing needed.

Thermal Management Technologies developed a phase-changing thermal storage unit (TSU) that considers desired phase-change temperatures, interfaces, temperature stability, stored energy, and heat removal methodologies. This device will allow the user to control temperature peaks, stable temperatures and/or energy storage (15).

Redwire Space developed multiple phase change materials (PCM)-based thermal energy storage panels for the CubeSat form factor that can be easily stacked in between critical components (16). Shown in figure 7-10 are two examples of thermal storage technology solutions, Q-Store and Q-Cache.
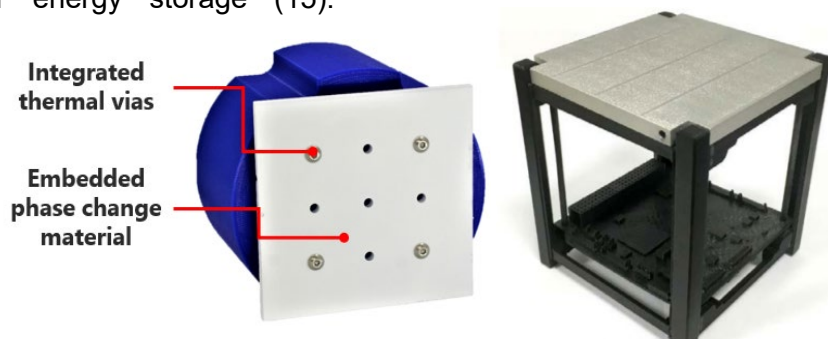
Figure 7.10: Redwire Space's thermal energy storage technologies: (left) Q-Store and (right) Q-Cache. Credit: Redwire Space.

7.2.10 Thermal Switches

A thermal switch is a device that switches a heat conduction path between either a strong thermal coupling or weak thermal coupling (thermal isolation) as needed to control the temperature of heat producing components. A switch typically connects a heat producing component and a low temperature sink, such as a radiator. Heat switches differ from thermostats in that they passively modulate a thermal coupling while thermostats modulate heater circuits (17). Part of the challenge with integration of a thermal switch in SmallSats is that they take up additional space between a component and heat sink. Typical, heat switches may provide a conduction ratio of 10:1 with a technology goal of 100:1 (18). See table 7-9 for example commerical thermal switches.

Table 7-9: Thermal Switches							
Manufacture	**Product**	**Type**	**Max Conductance**	**Min Conductance**	**Mass**	**Max Q**	**Notes:**
ACT	(Study)	VCHP	20 W/K	0.01 to 0.04 W/K	0.3 to 0.5 kg		VCHP Variable Cond. Heat Pipe)
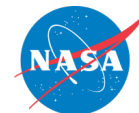
Table 7-9: Thermal Switches							
Manufacture	Product	Type	Max Conductance	Min Conductance	Mass	Max Q	Notes:
Sierra Nevada	Passive Thermal Control Heat Switch	Mechanical	1 W/K	0.015 W/K	~0.110 kg	6 W	Originally designed for Martian surface ops.
Sierra Nevada	HP Thin Plate	Mechanical	607 W/m^2/°C	7.8 W/m^2/°C	2.72 g/cm^3	12 W	Based on single "cell" design (25.4 mm x 25.4 mm)
Sierra Nevada	Diaphragm Thin Plate	Mechanical	607 W/m^2/°C	7.1 W/m^2/°C	2.72 g/cm^3	12 W	Based on single "cell" design (41.3 mm x 41.3 mm)

7.2.11 Multifunctional Thermal Structures

A newer development in passive thermal control for small spacecraft are multi-functional thermal structures. These integrate thermal control capabilities directly into the structure. This is particularly advantageous for small spacecraft due to strict mass and volume constraints. Currently, Thermal Management Technologies has adapted its multifunction heat spreading structure technology, scaled it to smaller satellite configurations, and called it Standard Passive Orbital Thermal-control (SPOT) Structures. SPOT Structures come in four standard configurations: 6U, 12U, Launch U, and ESPA (19). Each incorporates heat-spreading technology that improves the ability to radiate waste heat. They incorporate features such as low mass, high stiffness/strength, and integrated heat pipes. This new technology is at TRL 4.

7.3 State-of-the-Art – Active Systems

Active thermal control methods rely on input power for operation and have been shown to be more effective in maintaining tighter temperature control for components with stricter temperature requirements or higher heat loads (20). Typical active thermal devices used on large-scale spacecraft include electrical resistance heaters, cryocoolers, thermoelectric coolers, and fluid loops. Electrical heaters are usually easily integrated into SmallSat architectures as they do not typically use much mass or volume. Heaters are frequently used in all space applications, including small and large satellites, so they are often included as passive thermal control technology. Other active systems are challenging to integrate into CubeSats and other small satellites because of the power, mass, and volume needs associated with each given technology. Until spacecraft designers can miniaturize existing actively controlled thermal techniques and reduce power requirements or increase available spacecraft power, most of active thermal systems in small spacecraft are limited and have a TRL range of 3-6. Descriptions of SmallSat active thermal technologies are shown in table 7-10.

Table 7-10: SmallSat Active Thermal Technologies	
Active Technology	Description
Electrical Heaters	Space heaters use an electrical-resistance element in between two sheets of flexible electrically insulating material.
Cryocoolers	Miniature refrigeration system designed to cool components to extremely low temperatures.
Thermoelectric Coolers (TEC)	Miniature solid-state heat pumps which provide localized cooling via the Peltier effect, which is cooling resulting from passing electric current through a junction formed by two dissimilar metals.

Fluid Loops	System that circulates a working fluid through the spacecraft to a heat sink.

7.3.1 Heaters

Electrical resistance heaters used on small spacecraft are most often Kapton heaters, which consist of a polyimide film with etched foil circuits that produce heat when a current is applied. Kapton heaters also often have a pressure sensitive adhesive on one side for easy application. Heaters are usually controlled by a thermostat or temperature sensor and used in cold environments to maintain battery temperature, typically the component with the narrowest temperature limits. The low mass of SmallSats requires little additional heater power to maintain temperature limits, and so heaters do not typically need to be very high power to effectively manage temperatures. The TRL values for electrical heaters on SmallSats are 7-9 in LEO environments.

The 1U CubeSats Compass-1, MASAT-1, and OUTFI-1 each required an electrical heater attached to the battery in addition to passive control for the entire spacecraft system to maintain thermal regulation in eclipses (21). Additionally, as biological payloads become more common on small spacecraft, their temperature limits must be considered and maintained as well. NASA ARC biological nanosats (GeneSat, PharmaSat, O/OREOS, SporeSat, EcAMSat, and BioSentinel) all used actively controlled heaters for precise temperature maintenance for their biological payloads, with closed-loop temperature feedback to maintain temperatures. See table 7-11 for examples of electrical heaters for active thermal systems.

Table 7-11: Active Thermal Systems - Electrical Heaters					
Manufacturer	**Product**	**Power (W/cm²)**	**Temp Limits**	**Material**	**Notes**
Omega	KHLVA, PLM-Series	0.4 to 1.56	-40°C to 149°C (w/PSA) -57°C to 232°C (w/o PSA)	Kapton (polyimide)	Various sizes, rectangular/square/round, w & w/o pressure sensitive adhesive (PSA), Low-Outgassing
Omega	Polyimide Heater Kit	0.4 to 1.56	-200°C to 200°C (w/PSA)	Kapton (polyimide)	Various shapes. Temp limit max for short excursions. 149°C for continuous ops due to PSA limits.
Tempco	SHK Series	0.8 to 6.2	-35°C to 150°C (w/ Al Foil)	Kapton (polyimide) Some w/ Al Foil	Rectangular/square/round, w/ (PSA). Some model available w/ AL foil for extended temp limits.
Minco	Polyimide Thermofoil HK Series	0.8 to 5.1	-200°C to 200°C -200°C to 150°C (w/ Al Foil)	Kapton (polyimide) Some w/ Al Foil	Mountings: #12 & arcrylic PSA, epoxy, shrink band, stretch tape, clamping. NASA rated for materials & vacuum.

Table 7-11: Active Thermal Systems - Electrical Heaters					
Manufacturer	Product	Power (W/cm^2)	Temp Limits	Material	Notes
Birk Manufacturing	Polyimide-Traditional	0.02 to 3.1	See Notes	Kapton (polyimide)	Max Temps: FEP-Bonded & All Polyimide(260°C); Acrylic Bonded(120°C).
Birk Manufacturing	Polyimide-High Temp	0.02 to 7.75	See Notes	Kapton (polyimide)	Max Temp: up to 300°C
Chromalox	KPH, KPM Series	0.4 to 1.56	-200°C to 200°C* -200°C to 121°C**	Kapton (polyimide)	* FEP Construction, Acrylic construction
All Flex Solutions	Custom Designs				See website for design guide, options.

7.3.2 Cryocoolers

Cryocoolers are refrigeration devices designed to cool around 100K and below. A summary of cryocooler systems is given in figure 7.11, and a detailed review of the basic types of cryocoolers and their applications is given by Radebaugh (22). The first two systems (a) and (b) are recuperative cycles, and (c), (d), and (e) are regenerative cycles. Cryocoolers are used on instruments or subsystems requiring cryogenic cooling, such as high precision IR sensors. Instruments such as imaging spectrometers, interferometers and midwave infrared (MWIR) sensors require cryocoolers to function at extremely low temperatures. The low temperature improves the dynamic range and extends the wavelength coverage. The use of cryocoolers is also associated with longer instrument lifetimes, low vibration, high thermodynamic efficiency, low mass, and supply cooling temperatures less than 50K (23). Cryocoolers on small spacecraft are still a new concept, however there have been two CubeSats with cryocooling on board. Lunar IceCube, a 6U secondary payload launched from Artemis I in 2022 and developed by Morehead State University, used a 600 mW cryocooler for its BIRCHES point spectrometer (24). For more examples of cryocooler solutions, see table 7-12.
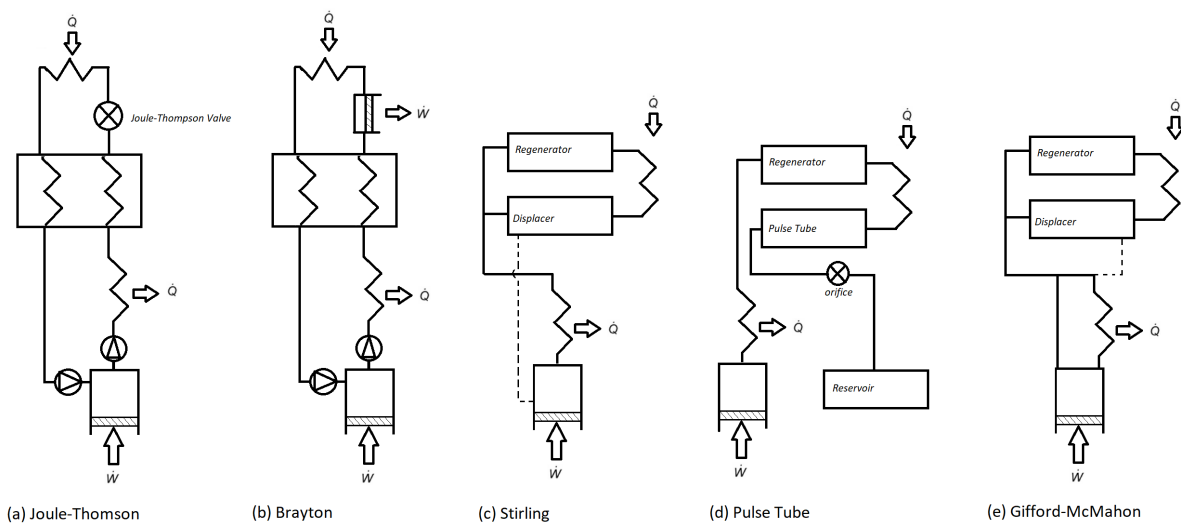
Figure 7.11: A comparison of cryocooler types. Credit: NASA.

Table 7-12: Active Thermal Systems - Cryocoolers							
Manufacturer	**Product**	**Form factor**	**Life (yrs)**	**Mass (kg)**	**Power (W)**	**TRL**	**Notes:**
Lockheed Martin Space	MICRO 1-1	~1U	10	0.350	15	7 - 9	1 W cooling @150K cold tip. Integrated on LunIR CubeSat (launcher: Artemis I).
	MICRO 1-2	~1U	10	0.475	25	6	2 W cooling @105K cold tip. Tested for MISE. Tentatively slated for integration into Europa Clipper.
	MICRO 1-3	~1U	10		40	5	Original design point: 1.4 W cooling @105K (40 W). Max Cooling: 2.9 W @200K (30 W).
AIM Infrarot-Module GmbH	SF070	~1U	>3.5 (MTTF)	0.85	24	6 - 9	0.6 W @80K cold tip. Slated from HyTi: Hyperspectral Thermal Imager (6U CubeSat).
SunPower - Ametek	CryoTel DS Mini	<2U	>14	1.20	45	6	1.8 W @77K (23°C reject). Op to 40K.
	CryoTel MT	<3U	>23	2.10	80	6	5 W @77K (23°C reject). Op to 40K.
Ricor	K508N	1U+	15	0.475	5.5	4 - 6	0.2 W @80K (23°C reject). Demo for 'Active CryoCubeSat (ACCS)' project.

TRL values provided are specific to low-Earth environments for SmallSats <180 kg.

7.3.3 Thermoelectric Coolers (TECs)

TECs are miniature solid-state heat pumps which provide localized cooling via the Peltier effect, which is cooling resulting from passing electric current through a junction formed by two dissimilar metals. TECs have been used to cool star trackers, IR sensors and low noise amplifiers on large spacecraft. Advantages of TECs are that they have no moving parts, are reliable, noiseless, lightweight, and compact. Their use is limited below temperatures of 130K by low efficiency and low performance with large temperature differences. Furthermore, the TECs are fragile to mount and highly sensitive to thermal expansion stresses. External stresses can be mitigated by adding a conductive strap on the cold side (25). Table 7-13 provides an overview of some commercial thermoelectric coolers.

Table 7-13: Thermoelectric Coolers					
Manufacturer	**Product**	**Cooling Capacity**	**ΔT @ 27°C**	**Operating Temp**	**Notes:**
Laird Technologies	*Examples:* CP Series PolarTec PT Series UltraTEC UTX Series HighTemp ETX Series Multistage MS Series	1.8 - 118 W 18 - 71 W 69 - 299 W 7.7 - 322 W 0.3 - 38 W	68 - 74°C 68 - 74°C 68 - 74°C 68 - 74°C 81 - 129°C	< 80°C < 80°C < 80°C < 150°C < 80°C	
Custom Thermoelectric	04801-9A30-18RB	6.1 W	70°C (Max)	< 200°C	
FerroTec	9502/031/018 M	4.1 W	70°C (Max)		Values @ 50°C

Table 7-13: Thermoelectric Coolers					
Manufacturer	**Product**	**Cooling Capacity**	**ΔT @ 27°C**	**Operating Temp**	**Notes:**
II-IV Coherent	MT30-0.9A-21AN	2.2 W	70°C (Max)		Values @27°C, Designed for zero/high G
II-IV Coherent	CM23-1.9-08AC	3.4 W	71°C (Max)		Values @27°C, Designed for zero/high G

7.3.4   Fluid Loops

A pumped fluid loop (PFL) consists of a circulating pump that moves a liquid through tubing connected to a heat exchanger and heat sink. A heat source is mounted to the heat exchanger and the pumped fluid carries the heat from the source to a heat sink, typically a radiator, and then the cooled fluid is returned to the heat source to continue providing cooling. A PFL is capable of cooling multiple locations via forced fluid convective cooling. Mechanically pumped fluid loops (MPFL) are not typically used on SmallSats because they are associated with high power consumption and mass.

The Active Thermal Architecture (ATA) system is an advanced, active thermal control technology for small satellites in support of advanced missions in deep space, helio-physics, earth science, and communications. The ATA technology is capable of high-power thermal rejection and zonal temperature control of satellite busses, payloads, and high-energy density components. The ATA system includes integrated heaters and a PID-based control algorithm to dynamically tune the satellite's thermal rejection and zonal temperature control as a function of payload and mission parameters. The ATA project was developed by the Center for Space Engineering at Utah State University and funded by the NASA SST program in partnership with JPL. The ATA is a sub 1U two-stage active thermal control system targeted at 6U CubeSat form factors and larger. The first stage consists of a MPFL with a micro-pump that circulates a single-phase heat transfer fluid between an internal heat exchanger and a deployed tracking radiator. The second stage is composed of a miniature tactical cryocooler, which directly provides cryogenic cooling to payload instrumentation. The conceptual operation of the ATA system is shown in figure 7.13. The ATA system is scheduled to fly on the upcoming ACMES mission, a technology demonstration flight funded by NASA ESTO through an InVEST grant.
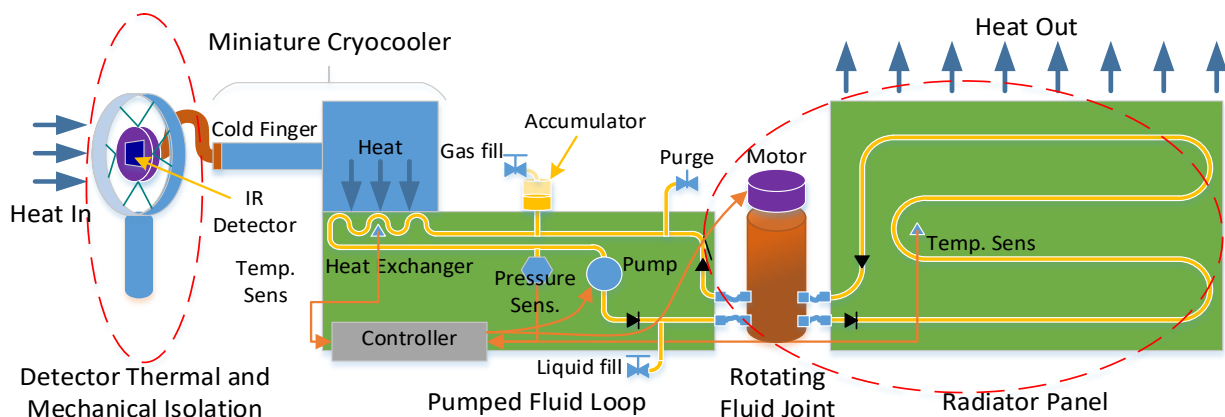
Figure 7.13: Conceptual operation of the ATA thermal control system. Credits: CSE/USU/ NASA/JPL.

Ultrasonic additive manufacturing (Fabrisonics UAM) techniques were used to simplify and miniaturize the ATA system by embedding the MPFL fluid channels directly into the integrated

HX, CubeSat chassis, and the external radiator, to create integrated multi-function structures. By leveraging UAM 3D printing, the ATA system can improve baseline thermal performance by >200%, a mass savings of more than >30%, and a savings on cost and schedule of better than >50%. The ATA system also features flexible, multi-axis rotary fluid unions and an integrated geared micro-motor, which allows for the two-stage deployment and solar tracking of the ATA radiator. The ATA features passive vibration isolation and jitter cancellation technologies such as a floating wire-rope isolator design, particle damping, flexible PGS thermal links and a custom Kevlar isolated cryogenic electro-optical detector mount. Figure 7.14 shows some of the technologies developed for ATA, as well as the ground-based prototype CubeSat.
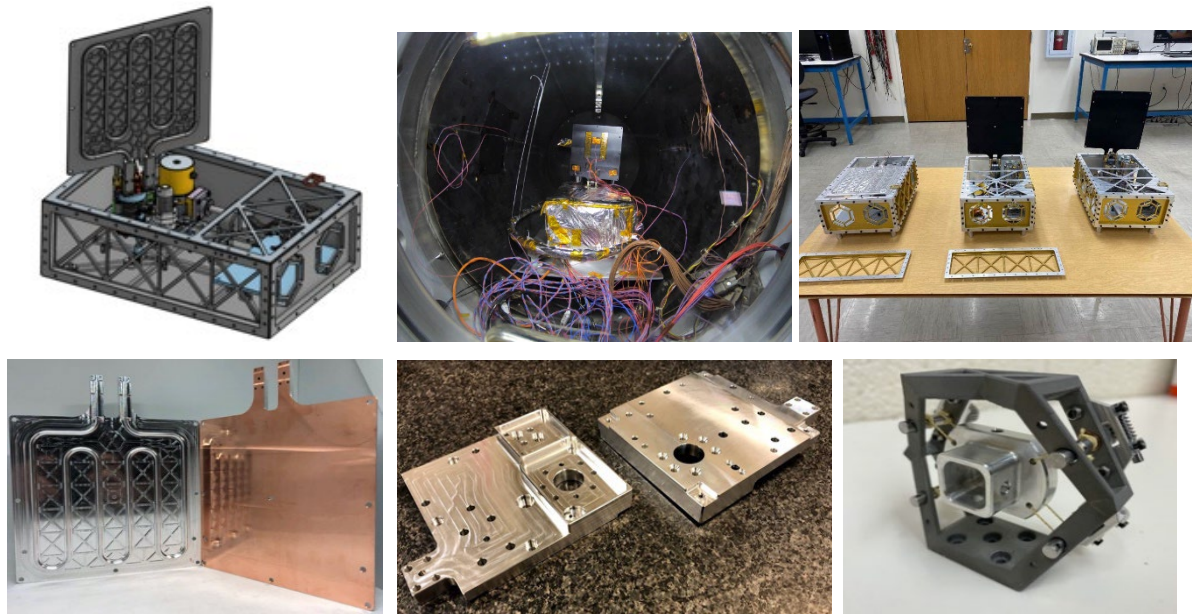
Figure 7.14: From top left: ATA CubeSat prototype, ATA subsystem testing, ATA prototypes, UAM radiator with copper backing, UAM heat exchanger, Kevlar isolated cryogenic electro-optical prototype mount. Credits: CSE/USU/NASA/JPL.

Table 7-14: Active Thermal Systems - Fluid Loops						
Manufacturer	**Product**	**Cooling Capacity**	**Mass**	**Volume**	**TRL**	**Notes**
Netherlands Aerospace Centre	Mini-MPL	>20 W	-	< 1U	4 - 6	Scalable to ≤ 200W
Lockheed Martin	Joule-Thomson Microcryocooler	-	0.2 kg	< 1U (Est.)	4 - 5	

TRL values provided are specific to low-Earth environments for SmallSats <180 kg.

7.4 Summary

As thermal management on small spacecraft is limited by mass, surface area, volume, and power constraints, traditional passive technologies such as paints, coatings, tapes, MLI, and thermal straps dominate thermal design. Active technologies, such as thin flexible resistance heaters have also seen significant use in small spacecraft, including some with advanced closed-loop control. Many technologies that have to date only been integrated on larger spacecraft are being designed, evaluated, and tested for small spacecraft to meet the growing needs of SmallSat developers as small satellites become more advanced. Deployable solar panels that have been

used by many other SmallSats are paving the way for thermal deployable components, while advanced deployable radiators and thermal storage units are still undergoing testing for small spacecraft.

More active thermal control system technologies are being developed to accommodate volume and power restrictions of a smaller spacecraft; cryocoolers are being designed to fit within 0.5U volume that will allow small spacecraft to use optical sensors and imaging spectrometers.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email.

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Glossary

(AI/ML)	Artificial Intelligence/Machine Learning
(ASICs)	Application-specific Integrated Circuits
(CDH)	Command and Data Handling
(ConOps)	Concept of Operations
(COTS)	Commercial-off-the-shelf
(CRAM)	Chalcogenide RAM
(DDD)	Displacement Damage Dose
(DRAM)	Dynamic RAM
(EPS)	Electrical Power System
(FERAM)	Ferro-Electric RAM
(FPGAs)	Field Programmable Gate Arrays
(FSW)	Flight Software
(I/O)	Input & Output
(LEO)	Low-Earth Orbit
(MRAM)	Magnetoresistive RAM
(OBC)	Onboard Computer
(PCM)	Phase Change Memory
(PSA)	Payload and Subsystems Avionics
(Rad-hard)	Radiation-hardened
(SDRs)	Software-defined Radios
(SEEs)	Single-event Effects
(SEL)	Single-event Latch-up
(SEUs)	Single-event Upsets
(SoCs)	System-on-chip
(SRAM)	Static Random-Access Memory
(SSA)	Small Spacecraft Avionics
(SWaP)	Size, Weight, and Power
(SWaP-C)	Size, Weight, Power, and Cost
(TID)	Total Ionizing Dose
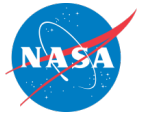
8.0 Small Spacecraft Avionics

8.1 Introduction

Small Spacecraft Avionics (SSA) consist of all the electronic subsystems, components, instruments, and functional elements of the spacecraft platform, including the primary flight sub-elements Command and Data Handling (CDH) and Flight Software (FSW), as well as other critical flight subsystems such as Payload and Subsystems Avionics (PSA). These subsystems are configured for specific mission platforms, architectures, and protocols, and are governed by appropriate operations concepts, development environments, standards, and tools. The CDH and FSW are the brain and nervous system of the integrated avionics system, and generally provide command, control, communication, and data management interfaces with all other subsystems in some manner, whether in a direct point-to-point, distributed, integrated, or hybrid computing mode. The avionics system is essentially the foundation for all components and their functions integrated on the spacecraft. As the nature of the mission influences the avionics architecture design, there is a large degree of variability in avionics systems.

There are two major factors to consider for SmallSat avionics:

1. Spacecraft scale: a traditional spacecraft is a high-size, weight, power, and cost (SWaP-C), flagship system and typically has a high-SWaP-C avionics system to reduce risk and address higher reliability requirements. A SmallSat is a low-SWaP-C miniature system, with a low-SWaP-C avionics system. Typically, more risk is tolerable with low cost, but nonetheless enhancements can be applied to increase reliability. Individually, the avionics system scales with the spacecraft, however constellations of SmallSats can "match" the capabilities of a traditional spacecraft by distributing the avionics and increasing reliability through functional redundancy of the spacecraft.

2. Architecture design: the architecture design is not necessarily dependent on the scale of the spacecraft. In both traditional spacecraft and SmallSats, the avionics system can be either centralized or decentralized, single-string or redundant, and modular or monolithic. Traditional spacecraft reduce risk by employing redundancy such that if one element fails the entire architecture is able to continue, but SmallSat avionics designs are usually single-string, whereby if one element fails, the entire system fails. Figure 8.1 illustrates an architectural block diagram of a centralized small spacecraft system. In anticipation of extended durations in low-Earth orbit (LEO) and deep space missions, designers are now incorporating radiation-hardened (rad-hard) or radiation-tolerant architecture designs in their avionics packages to further increase their overall reliability.

While a significant focus of this chapter is on commercial products and developments, vendors are not the only ones developing avionics platforms; there are also numerous government and academic efforts worth considering, with a few examples below:

- SpaceCube and MUSTANG, by NASA GSFC (government)
- Sabertooth by JPL
- CHREC/SHREC Space Processor, by NSF SHREC (academic)
- RadPC by Montana State University (academic)

Given the distributed and integrated nature of modern avionics, this chapter organizes the state-of-the-art in SmallSat Avionics into CDH (8.3) and FSW (8.4). On-the-Horizon activities (TRL <5) for CDH and FSW (8.5 and 8.6, respectively) highlight recent developments in next-generation SSA systems. Avionics Systems Platform and Mission Development Considerations (8.2) discusses how these considerations are being addressed and/or mitigated by state-of-the-art advances in CDH, FSW, and PSA products. A summary of future SSA systems is provided in (8.7).
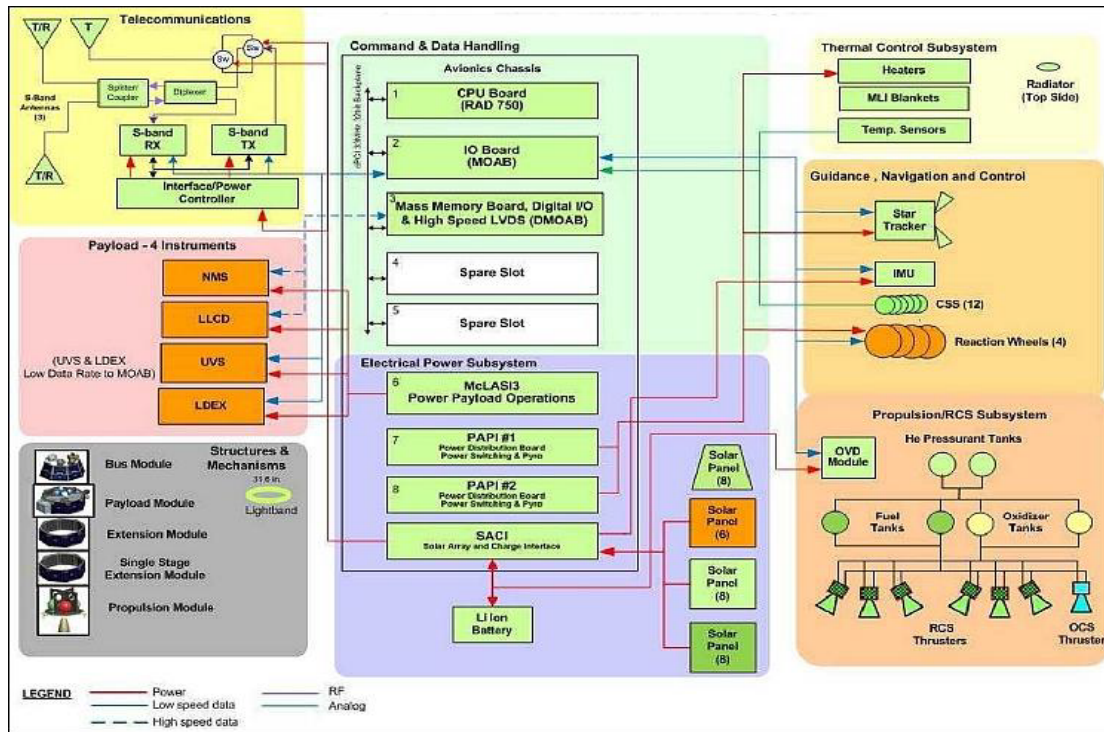
Figure 8.1: Functional block diagram of the LADEE spacecraft. Credit: NASA.

The information described below is not intended to be exhaustive but provides an overview of current state-of-the-art technologies and their development status for a particular small satellite subsystem. The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The information is for awareness and guidance only. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

8.2 Avionics Systems Platform and Mission Development Considerations

There are many factors considered in selecting the configuration and implementation of avionics subsystems, components, and elements for small spacecraft missions. Overall spacecraft concerns of Size, Weight, and Power (SWaP) always need to be considered. Some of the more pertinent issues and concerns that all small spacecraft missions must address include:

- Mission applicability and tailoring
- Element, module, and component modularity and interoperability
- Manufacturing and production efficiency, complexity, and scaling
- Mission environment, especially radiation and long-duration space exposure
- Standards and regulatory concerns
- SWaP-C constraints

In addition to CDH and FSW, state-of-the-art SSA systems should consider the following subsystem/payload specific electronic systems:

- Small spacecraft platform size ranges and configurations
- Integrated avionics platform architectures
- Mission avionics configurations
- Spacecraft and mission autonomy

Flight payload and subsystems avionics elements include:

- Subsystem integrated onboard computer (OBC) controllers
- Integrated systems health avionics
- Onboard payload processors
- Cloud-based processors

Modular avionics architectures for small spacecraft can be characterized as either federated or integrated. In a federated avionics architecture, each subsystem of the spacecraft is considered an independent, dedicated autonomous element, with the avionic components performing all functions independently and exchanging data over standardized communications protocols and interfaces. An integrated avionics architecture is a shared, distributed functionality, that can be configured with distributed, heterogeneous and/or mixed criticality elements. In either case, modular avionics architectures can be configured with smart subsystem capabilities, redundancy, fault tolerance, radiation mitigation, and anomaly mitigation procedures.

Constellation networks and swarms, synchronized formations, and other multi-satellite cluster formations are creating new opportunities for SSA. The increased need for synchronization, intersatellite communications, controlled positioning for integrated CDH functionality, coordination and conduct, Concept of Operations (ConOps), and autonomous operations impose new constraints on the avionics system. This is true not only for single satellites, but now also for multi-satellite configurations, whereby overall mission performance is dependent on all the platform elements acting in a co-dependent fashion.

8.3 State-of-the-Art (TRL 5-9): Command and Data Handling

Current trends in small spacecraft CDH generally appear to be following those of previous, larger scale CDH subsystems. The current generation of microprocessors can easily handle the processing requirements of most CDH subsystems and will likely be sufficient for use in spacecraft bus designs for the foreseeable future. Cost and availability are likely primary factors for selecting a CDH subsystem design from a given manufacturer, but many groups develop their own custom platforms. The ability to spread nonrecurring engineering costs over multiple missions and reduce software development through reuse are both desirable factors in a competitive market. Heritage designs work well for customers looking to select components with proven reliability for their mission as long as the key components are still in production. SmallSat CDH should consider the following:

1. Avionics and onboard computing form factors
2. Highly integrated onboard computing products
3. Radiation-tolerant processors and FPGAs
4. Memory, electronic function blocks, and components
5. Bus electrical CDH interfaces
6. Radiation mitigation and tolerance schemes

As small satellites move from the early CubeSat designs with short-term mission lifetimes to potentially longer missions, radiation tolerance becomes a significant factor when selecting parts. These distinguishing features, spaceflight heritage and radiation tolerance, are the primary differentiators in the parts selection process for long-term missions, verses those which rely heavily on commercial-off-the-shelf (COTS) parts. Experimental missions typically focus on low-cost, easy-to-develop systems that take advantage of open-source software and hardware to provide an easy entry into space systems development, especially for hobbyists or those who lack specific spacecraft expertise.

Small spacecraft CDH technologies and capabilities have been continuously evolving, enabling new opportunities for developing and deploying next-generation SSA. When small spacecraft were first introduced, a primary purpose was to observe and send information back to Earth. As awareness and utility have expanded, there is a need to improve the overall capability of data collection for specific mission environments beyond LEO. Small spacecraft, including nanosatellites and CubeSats, currently perform a wide variety of science in LEO, and these smaller platforms are emerging as candidates for more formidable missions beyond LEO.

The adoption of CubeSat and SmallSat technology is enabled by the miniaturization of electronics, sensors, and instruments. As spacecraft manufacturers begin to use more space-qualified parts, they find that those devices can often lag their COTS counterparts by several generations in performance but may be the only means to meet the radiation requirements placed on the system. Presently, there are several commercial vendors who offer highly integrated systems that contain the onboard computer, memory, electrical power system (EPS), and the ability to support a variety of Input & Output (I/O) for the CubeSat class of small spacecraft. Several CDH developments for CubeSats have resulted from in-house development, the rise of new companies that specialize in CubeSat avionics, and the use of parts from established companies who provide spacecraft avionics for the space industry in general. While parallel developments are impacting the growth of CubeSats, vendors with ties to the more traditional spacecraft bus market are increasing CDH processing capabilities within their product lines.

In-house designs for CDH units are being developed by some spacecraft bus vendors to better accommodate small vehicle concepts. While these items generally exceed CubeSat form factors in size, they can achieve similar environmental performance and may be useful in small satellite systems that replicate more traditional spacecraft subsystem distribution.

8.3.1 Avionics and Onboard Computing Form Factors

The CompactPCI and PC/104 form factors continue generally to be the industry standard for CubeSat CDH bus systems, with multiple vendors offering components that can be readily integrated into space-rated systems. Overall, form factors should fit within the standard CubeSat dimension of less than $10 \times 10 \text{ cm}^2$. Spacecraft avionics components are performance-driven and not necessarily dependent on spacecraft platform sizes, but some noncontainerized spacecraft platforms may need to consider the availability and utility of higher TRL avionics products. The PC/104 form factor was the original inspiration to define standard architecture and interface configurations for CubeSat processors, but with space at a premium, many vendors have been using all available space exceeding the formal PC/104 board size. Although the PC/104 board dimension continues to inspire CubeSat configurations, some vendors have made modifications to stackable interface connectors to address reliability and throughput concerns. Many vendors have adopted the use of stackable "daughter" or "mezzanine" boards to simplify connections between subsystem elements and payloads, and to accommodate advances in technologies that maintain compatibility with existing designs. A few vendors provide a modular package which allows users to select from a variety of computational processors.

8.3.2 Highly Integrated Onboard Computing Products

A variety of vendors are producing highly integrated, modular, onboard computing systems for small spacecraft. These CDH packages combine processors and/or Field Programmable Gate Arrays (FPGAs) with various memory banks, and with a variety of standard interfaces for other subsystems onboard. FPGAs and software-defined architectures also give designers a level of flexibility to upload software modifications to adapt to new requirements and interfaces. Table 8-1 summarizes the current state-of-the-art for some of these components. It should be noted that while some products have achieved TRL 9 by virtue of a space-based demonstration, what is relevant in one application may not be relevant to another, and different space environments and/or reliability considerations may result in lower TRL assessments. Some larger, more sophisticated computing systems have significantly more processing capability than what is traditionally used in SmallSat CDH systems, however the increase in processing power may be a useful tradeoff if payload processing and CDH functions can be combined (note that overall throughput should be analyzed to assure proper functionality under the most stressful operating conditions).

System developers are gravitating towards ready-to-use hardware and software development platforms that can provide seamless migration to higher performance architectures. As with non-space applications, there is a reluctance to change controller architectures due to the cost of retraining and code migration. Following the lead of microprocessor and FPGA vendors, CubeSat avionics vendors are now providing simplified tool sets and basic, cost-effective evaluation boards.

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	Product	Processor/ SoC/ FPGA	Radiation Hardness Assurance (RHA)	Board Dimension (cm)	Power Consumption (W)	Orbits Flown	Ref
AAC Clyde Space Sweden	Kryten-M3	Smart Fusion 2 SoC including an ARM Cortex-M3 processor delivering 62.5 DMIPS	TID 20 krad (system level)	9.589x9.017x 0.551	0.4	LEO	(1)
	Sirius OBC & TCM	32-bit LEON3FT (IEEE-1754 SPARC v8) fault-tolerant processor	TID 20 krad (system level)	9.589x9.017x 1.720	1.3	LEO, lunar lander	(2)
Alén Space Spain	DARA OBC	ARM 32-bit Cortex-M7 with FPU	N/A	8.93x9.33x1. 2	Max: 0.280	LEO, 2024	(4)
	TRISKEL OBC+TTC	ARM 32-bit Cortex-M7 with FPU	N/A	8.93x9.33x1. 2	OBC_max:0.280 TTC_max: 5.865	LEO, 2025	(3)
Aitech Systems Inc.	SP0-S	Power PC 1020, Alcatel RTAX	100 krad TID, SEL/LET 40 Mev-cm2/mg (without enclosure)	3U cPCI: 10.06x16x20. 3	12	LEO, MEO, GEO, Lunar, Deep Space*	(5)
	SP1-S	64-bit Arm® Cortex®-A72 cores @ 1.8 GHz	100 krad TID (Target)	3U Space VPX: 10.06x16x20. 3	<25	TBD	
	S-A1760	Pascal™ Architecture GPU w/256 CUDA® cores NVIDIA Denver 2 Dual-Core ARM® CPU + Cortex® A57 Quad-Core ARM® CPU	1.05 krad TID, < 1 type 2 SEFI update in 158 days (without enclosure)	12.7x12.7x5. 2	Configurable ≤ 5 Idle, 8-10 under typical CUDA load,<20 when System on Module is fully utilized	LEO*	(6) (7)
	S-C8780	Intel® Pentium-D or Xeon-D, discrete 2D GPU, Xilinx UltraScale+ FPGA w/Dual-core ARM® CPU	TBD	3U Open VPX: 10.06x16x20. 3	Configurable 25-55	TBD	(8)
	S-C8500	Intel® Tiger Lake UP3 SoC, 4 cores/8 threads, integrated GPGPU with 96 Execution Units, Xilinx UltraScale+	1 krad TID, SEL/LET TBD (without enclosure)	3U Open VPX: 10.06x16x20. 3	CPU Configurable from 12-28, Total system 22-	TBD	(9)

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
		FPGA w/Dual-Core ARM® CPU is Optional			42 configurations.		
Aerospacelab Belgium	OBC	Xilinx Zynq-7045 ARM dual-core Cortex A9 running at 667MHz Kintex-7 FPGA with 350k Programmable Logic Cells, 218,600 Look-up tables (LUT) and 437,200 flip-flops (FF) Linux and FreeRTOS compatible	Radiation hardened supervisor MCU Radiation hardened watchdog Heavy ions (62.5 MeV) No destructive event	187x84.1x116.6 mm³ L x W x H Mass 1.325 kg	<4	LEO	
Argotec Italy	OBC FERMI	Dual-Core LEON3FT SPARC V8 Processor with fault-tolerant memory controller +Microchip RTG4	Rad-hard	10.24x10x4.49	5 (depending on load)	Deep Space, Lunar Orbit, LEO	(10) (11)
Berlin Space Technologies GmbH Germany	CDH-110	STM32F103	TID 30kRad (PCB level)	17.6x9.4x6.9	< 2	LEO	
Bradford Space Netherlands	Stellar OBC + PDHS	Dual Polarfire SoC	30 krad (Si) + functional SEE resilience through SW EDAC and internal cold redundancy	13.3x11.5	< 25W Full Load / < 3 Typical / < 1 Idle	TBD	(12)
C3S Electronics Development LLC Hungary	OBC	32-bit ARM Cortex-M7	Continuous integrity check on the program memory, multi-level watchdog system, MRAM storage	0.92x0.895x0.123 without daughterboard, 0.92x0.895x0.136 with daughterboard	0.46 (measured in test mode, using eMMC as mass storage)	LEO	(13)
CesiumAstro USA	SBC-1461	1.4 GHz ARM Cortex	Rad Tolerant	5x8.4x1.3	8 typ.	LEO	(14)
	RDP-23FV	Xilinx Versal	Rad Tolerant to Rad Hard	16x10x2.54	50	LEO (2025)	

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
EnduroSat	OBC	ARM Cortex M7	Tested at 40 krad TID	8.9x9.4x2.3 (with integrated GNSS)	1.5 Peak 0.2 Idle	400-600 km SSO, equatorial	(15)
	Payload Controller	Xilinx UltraScale+ SoC	Tested at 20 krad TID	9.8x10x6.4	<50 Peak 15Idle	Designed for LEO	
GomSpace	NanoMind HP MK3	Xilinx Zynq 7030/7045	>20 krad	9.5x9.5x3.15	Dependent on customer application	LEO	(16)
Ibeos	EDGE-1100 (3U SpaceVPX)	AMD Ryzen SOC	TID: 30 krad SEE: >37 MeV	3U SpaceVPX; 16(L) x 10(W) x 2.5(Pitch)	6-35	Designed for LEO and GEO	
	EDGE-1100 Genie	AMD Ryzen SOC	TID: 30 krad SEE: >37 MeV	14.8(L) x 10.8 (W) x 3.4(H) – Stand-alone box	8-35	Designed for LEO and GEO	
	Core-250	High Performance Space Computer Processor (HPSC), 8 x RISC-V cores, 3U SpaceVPX	--	16(L) x 10(W) x 2.5(Pitch)	<25	Designed for LEO, GEO, and Beyond	
Innoflight	CFC-400XS	Defense Grade Xilinx UltraScale+ MPSoC	TID: 30 krad	8.2x8.2x2.7	6-25	LEO (2021) & GEO (2022)	(17)
	CFC-400XP	Defense Grade Xilinx UltraScale+ MPSoC	TID: 30 krad	17.2x10x2.5	4-30	LEO in 2024	(18)
	CFC-510P	AMD Ryzen GPGPU	TID: 30 krad	17.2x10x2.5	12-40	LEO in 2024	

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
KP Labs	Antelope OBC and DPU	OBC – RM57 Herkules microcontroller (Dual 300 MHz ARM Cortex-R5F with FPU in lock-step) DPU – AMD Xilinx Zynq UltraScale+ MPSoC (ZU2EG, ZU3EG, ZU4EG, ZU5EG), Quad ARM Cortex-A53 CPU, Dual ARM Cortex-R5 in lock-step	COTS with SEE mitigation	Motherboard – 1x1x1 Daughterboard - 7x4x2	From less than 0.5 (DPU is off) to 6 (DPU processes the data)	LEO	(19)
	Leopard D PU	AMD Xilinx Zynq UltraScale+ MPSoC (ZU6EG, ZU9EG, ZU15EG); Quad ARM Cortex-A53 CPU; Dual ARM Cortex-R5 in lock-step	COTS with SEE mitigation	Non-redundant: 9x9.5x5 Redundant: 9x9.5x7.8	7 static power consumption; up to 10 for deep learning inference	LEO	(20)
	Lion DPU	AMD Xilinx Kintex Ultrascale FPGA (KU035, KU060, KU095)	COTS with SEE mitigation	16x10x5	<15 for KU035 version	TBD	(21)
Laboratory for Atmospheric and Space Physics	LASP Generic Small-sat FPGA Board	Kintex-7	25 krad	8.763x8.763	1	LEO	(24)
Micro Aerospace Solutions	MAS-SBC-107	Arm® Cortex®-M7	Total Ionizing Dose of 30 krad (Si)	9×9.6	<30	LEO	
Nara Space Technology	NSTOBC	Atmel ARM 9 based Microprocessor Unit	<24 krad	9x9x1.6	0.429 (Idle)	LEO	(22)
NOVI	Gen 2 OBC	AMD/Xilinx Versal (AI Edge Series): Dual-core ARM® Cortex®-A72, Dual-core ARM® Cortex®-R5F, Programmable Logic (FPGA), AI Engines. Vorago ARM MCU: Low-power ARM® Cortex®-M4, Embedded rad-hard FRAM.	Radiation-hardened Vorago ARM MCU, SEL immune Xilinx processor, Optional radiation-hardened memory, Built in SEU/SEFI mitigation,	9.3x9.3x2.5	<0.75W (Low Power Mode) 5-50W (High Performance Modes, Application Dependent)	LEO	(23)

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	Product	Processor/ SoC/ FPGA	Radiation Hardness Assurance (RHA)	Board Dimension (cm)	Power Consumption (W)	Orbits Flown	Ref
			ECC protected memory interfaces, Redundant storage for boot images.				
Novo Space	SBC002AV	Zynq Ultrascale+	TID: 30 krad; Automotive parts with Rad-tolerant in high criticality parts; Rich telemetry with local event response; Board sectorization and power control; ECC in Volatile memories; CRC / Reed-Solomon / Interleaving in Non-Volatile memories; FPGA and fabric use selective TRM on critical functionality; Scrubbing on soft FPGAs.	16x10	application dependent active mode max: 15	TBD	(25)
	SBC003AV	SmartFusion2		16x10	application dependent active mode max: 8	TBD	(26)
	SBC004AF	Versal ACAP		16x10	Under development	TBD	(27)
	SBC005AF	Polarfire		16x10	application dependent active mode max: 8	TBD	(28)
	GPU001AF	Jetson TX2i		8.7x5	active mode min:9 active mode max:15	TBD	(29)
	GPU002AF	Jetson Orin Nano	TID: 30 krad	8.7x5	active mode min:6 active mode max:11	TBD	(30)
Pumpkin Space Systems	PPM A1	MCU+DSP	N/A	90x96	0.05	LEO	

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
Pumpkin Space Systems	PPM A2	MCU+DSP	N/A	90x96	0.05	LEO	
	PPM A3	MCU+DSP	N/A	90x96	0.05	LEO	
	PPM B1	MCU+DSP	N/A	90x96	0.05	LEO	
	PPM D1	MCU	N/A	90x96	0.05	LEO	
	PPM D2	MCU+DSP	N/A	90x96	0.05	LEO	
	PPM E1	MCU	N/A	90x96	0.05	LEO	
	PPM B1	MCU+DSP	N/A	90x96	0.05	LEO	
	PPM D1	MCU	N/A	90x96	0.05	LEO	
	MBM2 w/BBB	MCU	~5 krad	90x96	2	LEO	
Ramon Space	NuPod	Space Grade Xilinx Versal ACAP, Cortex A72 4x / 16x Core ARM	TID: 30 krad SEL: 43 MeV·cm2/mg	28.7x21.5x7.5 Mounted enclosure	20-90	TBD	(31)
Resilient Computing	RadPC-SBC-001	RISC-V 32-Bit (AMD Artix 7)	COTS with SEE mitigation	10x10	1.5	LEO, Lunar (2024)	(32)
Sierra Space	UMCD	Microchip RTSX72	100kRad TID (Si)	11.4x11.4x4.4	7	LEO, GEO	(33)
SkyLabs	NANOhpc-obc	4x RISC-V 64-bit / PolarFire / SoC	COTS w/ SEE mitigation	9.5x9.1x1.3	~10 Peak	TBD	(34) (36)
	NANOhpm-obc	32-bit RISC-V core / PolarFire	COTS w/ SEE mitigation	9.5x9.1x1.3	<5	LEO, MEO	(35) (38)
	NANOobc-2	PicoSkyFT / IGLOO 2	COTS w/ SEE mitigation	9.5x9.1x1.1	<1 Peak, <0.9 Idle	LEO, MEO	(37)
Space Dynamics Laboratory	SDL-25	UT700 LEON3-FT + RT3P FPGA	10 krad	13x9	2-6	LEO, GEO	
	LEON 4	GR740 LEON4 + RTG4	50 krad	SpaceVPX 6U	12	LEO, GEO	
SPiN USA	MA61C CubeSat	GR712RC dual-core 32-bit LEON3 fault-tolerant, SPARC V8 processor	Processor is 300 krad	9.599x9.027	1-1.2	TBD	(39)
	MA61C smallsat	GR712RC dual-core 32-bit LEON3 fault-tolerant SPARC V8 processor	Processor is 300 krad	10.5x10.5	1-1.2	TBD	(40)

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
	MA61C cPCI serial space	GR712RC dual-core 32-bit LEON3 fault-tolerant SPARC V8 processor	Processor is 300 krad	16x10	1-1.2	TBD	(41)
Spiral Blue	Space Edge One	Nvidia Jetson Xavier NX	N/A	10x10	7	LEO	(42)
Southwest Research Institute® (SwRI®)	Centaur SBC	GR712RC dual core LEON3-FT CPU, Xilinx Ultrascale or Microchip RT-ProAsic or RTG4 FPGA	TID: Up to 60 krad (Si), more with shielding SEL: Immune to >67eV/mg/cm2 SEU: < 1 error per 24-hour period	Available in 3U/6U cPCI form factors	Low power < 4 Operation	LEO	
	HP-SBC	PowerPC based MPC8548e CPU, Microsemi RT-ProASIC FPGA, Optional: Microsemi RTG4 FPGA, Optional: Xilinx Ultrascale FPGA	TID: Up to 60 krad (Si), more with shielding SEL: Immune to >67eV/mg/cm2 SEU: < 1 error per 24-hour period	Available in 3U/6U cPCI form factors	12-20 depending on clock frequency and FPGA pairing	Leo and GEO	(43)
Spacemanic	DeepThought_OBC	SAMV71	-	6.7x4.2x0.7	0.1	GRBAlpha BDSat1 & 2 Planetum Veronika CroCube	(44)
	Eddie_OBC	MSP430	-	6.7x4.2x0.7	0.1	LEO (500-550 SSO), Planetum	(45)

Table 8-1: Sample of Highly Integrated Onboard Computing Systems							
Manufacturer Headquarters	**Product**	**Processor/ SoC/ FPGA**	**Radiation Hardness Assurance (RHA)**	**Board Dimension (cm)**	**Power Consumption (W)**	**Orbits Flown**	**Ref**
						Veronika CroCube	
Trident Space Electronic Systems	VDRT	Versal VC1902	30 krad / 50MeV LU	10x14.6x2.54	<60	LEO/MEO capable	(46)
	UDRT	MPSoC ZU19	30 krad / 40MeV LU	10x14.6x2.54	<50	Multiple LEO 400-1200km missions	(47)
Unibap	iX10-100A	AMD Ryzen V1000 (CPU and GPU), Intel Myriad-X (VPU)	Radiation Tolerant COTS with SEE mitigation	10x10x6	25-40	LEO	(48)
	iX5-106	AMD Steppe Eagle (CPU and GPU), Intel Myriad-X (VPU) Microchip SmartFusion2 (FPGA)	Radiation Tolerant COTS with SEE mitigation	10x10x5	15-25		
Xiphos	Q7S	AMD-Xilinx Zynq-7020 Dual-core ARM Cortex-A9	25 krad	7.8x4.3x0.9	2	LEO	
	Q8S	AMD-Xilinx Zynq Ultrascale+ MPSOC Quad-core ARM Cortex-A53	30 krad	8x8x1.12	>5	LEO	
	Q8Js	AMD-Xilinx Zynq Ultrascale+ MPSOC Quad-core ARM Cortex-A53	30 krad	8x8x1.12	>5	LEO	

*Orbits flown are on larger spacecraft

8.3.3 Radiation-Hardened Processors

Several radiation-hardened embedded processors have recently become available. These are being used as the core processors for a variety of purposes including CDH. Some of these are the Vorago VA10820 (ARM M0) and the VA41620 and VA41630 (ARM M4); Cobham GR740 (quad core LEON4 SPARC V8); BAE 5545 quad core processor; and LS1043 quad processor. These have all been radiation tested to at least 50 kRad total ionizing dose and a high onset for single-event effects.

8.3.4 Memory, Electronic Function Blocks, and Components

The range of onboard memory for small spacecraft is wide, typically starting around 32 kB and increasing with available technology. For CDH functions, onboard memory requires high reliability. A variety of different memory technologies have been developed for specific traits, including volatile memory, such as Static Random-Access Memory (SRAM) and Dynamic RAM (DRAM), Magnetoresistive RAM (MRAM), Ferro-Electric RAM (FERAM), Chalcogenide RAM (CRAM) and Phase Change Memory (PCM). SRAM is typically used due to price and availability, with numerous SRAM choices (up to 4M x 39 [20 MB]). There are many manufacturers that provide a variety of electronic components that are space-rated with high reliability. A chart comparing the various memory types and their performance is shown in table 8-2.

Table 8-2: Comparison of Memory Types						
Feature	**SRAM**	**DRAM**	**Flash**	**MRAM**	**FERAM**	**CRAM/ PCM**
Non-volatile	No	No	Yes	Yes	Yes	Yes
Operating Vo Itage, ±10%	2.5 – 5 V	1.35-3.3 V	3.3 & 5 V	3.3 V	3.3 V	3.3 V
Organization (bits/die)	512 k × 8	128 M × 8 1Gb × 8	16 M × 8; 4G × 8	2M × 8	16 k × 8	Unk
Data Retention (70°C)	N/A	N/A	10 years	10 years	10 years	10 years
Endurance (Erase/Write cycles)	Unlimited	Unlimited	1E5	1E13	1E13	1E13
Access Time	10-25 ns	25 ns	50 ns after page ready; 200 μs write; 2 ms erase	300 ns	300 ns	100 ns
Radiation (TID)	50K - 1 Mrad	50 krad	30 krad	1 Mrad	1 Mrad	1 Mrad
Temperature Range	MIL-STD	Industrial	Commercial	MIL-STD	MIL-STD	MIL-STD
Power	500 mW	300 mW	30 mW	900 mW	270 mW	Unk
Package	4 MB-20 MB	128 MB 1GB	128MB – 4 GB	2 MB	1.5 MB (12 chip package)	Unk

8.3.5 Bus Electrical Interfaces

CubeSat class spacecraft continue to use interfaces that are common in the microcontroller or embedded systems world. Highly integrated systems, especially systems-on-chip (SoCs), FPGAs, and application-specific integrated circuits (ASICs), will typically provide several interfaces to accommodate a wide range of users and to ease the task of interfacing with peripheral devices and other controllers. FPGAs are commonly used for these interfaces because of their flexibility and ability to change interfaces as needed. Some of the most common bus electrical interfaces are listed below with applicable interface standards:

- Serial Communication Interfaces (SCI): RS-232, RS-422, RS-485 etc.
- Synchronous Serial Communication Interface: I2C, SPI, SSC and Enhanced Synchronous Serial Interface (ESSI)
- Networks: Ethernet, LonWorks, etc.
- Fieldbuses: CAN Bus, LIN-Bus, PROFIBUS, etc.
- Timers: PLL(s), Capture/Compare and Time Processing Units
- Discrete IO: General Purpose Input/Output (GPIO)
- Analog to Digital/Digital to Analog (ADC/DAC)
- Debugging: JTAG, ISP, ICSP, BDM Port, BITP, and DB9 ports
- SpaceWire: a standard for high-speed serial links and networks
- High-speed data: RapidIO, XAUI, SerDes and MGT protocols are common in routing large quantities of mission data in the gigabit per second speeds

8.3.6 Radiation Mitigation and Tolerance Schemes

Deep space and long-duration LEO missions compel developers to consider reliability requirements and possibly incorporate radiation-mitigation strategies into their respective spacecraft designs. CubeSats are often either composed of only COTS components or a hybrid combination of COTS and rad-hard and radiation-tolerant components. COTS components typically offer superior performance, energy efficiency, and affordability compared to their rad-hard alternatives; however, COTS devices tend to be highly susceptible to radiation. The advantages of COTS components have enabled low-cost CDH development, while also allowing developers to leverage start-of-the-art technologies in their designs. A hybrid design combines COTS and rad-hard components, such as COTS processor and memory with rad-hardened supporting electronics (e.g., EPS, watchdog, etc.), to maximize the benefits of both technologies. These designs may also incorporate radiation-mitigation techniques to further enhance overall system reliability.

For space applications, the effects of radiation on electronic devices can vary broadly (44). Radiation effects are often categorized into long-term cumulative effects and transient single-event effects (SEEs). Long-term effects include total ionizing dose (TID) and displacement damage dose (DDD). TID, measured in krad, is the ionizing radiation absorbed by the device material over time causing parametric or functional degradation of the device. DDD is the nonionizing damage caused by particle collisions with the device structure over time. SEEs occur when a single radiation particle strike deposits enough charge to cause an effect. SEEs can be destructive or nondestructive. Single-event upsets (SEUs) are nondestructive SEEs that can affect the logic state of a memory cell. Single-event latch-up (SEL) are destructive SEEs that manifest as parasitic structures in CMOS logic or bipolar transistor structures, potentially causing a high-current state.

Other areas of consideration for CDH elements include memory, imaging, protection circuits (watchdog timers, communications watchdog timers, overcurrent protection, and power control),

memory protection (error-correction code memory and software error detection and correction), communication protection (several components), and parallel processing and voting.

8.4 State-of-the-Art (TRL 5-9): Flight Software

The FSW, at a fundamental level, communicates the instructions for the spacecraft to perform all operations necessary for the mission. These include all the science objectives as well as regular tasks (commands) to keep the spacecraft functioning and ensure the storage and communication of data (telemetry). The FSW is usually thought of as all the programs that run on the CDH avionics but should also include all software running on the various subsystems and payload(s).

There are many factors in selecting a development environment and/or operating system for a space mission. A major factor is the amount of memory and computational resources. There are always financial and schedule concerns. Another factor is what past software an organization may have used and their experiences with that software. The maturity of the software and its availability for the target subsystem or payload are additional factors to be considered in the final selection.

FSW complexity can refer to the architecture design (e.g., the interactions between subsystems, especially for spacecraft autonomy) as well as the number of operations to be performed. The more software is required to do, the bigger the task and cost. This complexity (and the associated verification effort) is what primarily drives the cost and schedule for a program or mission. Required reliability and fault management can also increase complexity and cost, regardless of the size of the spacecraft. Changing requirements is also a huge factor, which may be mitigated by involving the software team early in the planning process.

With the increase in processing capability with CDH and other processors, more capable FSW has been enabled. Traditionally, larger spacecraft require rad-hard processors which have poor performance, while CubeSats and SmallSats can take more risks with COTS processors that offer substantially more performance. Several advances have increased the processing capabilities available for CubeSats. Low-power ARM-based processors and embedded COTS SoCs, as well as advances in radiation hardened processors, have brought similar processing capabilities down to the small size of CubeSats. All of this has resulted in increased demands and requirements for FSW.

Generally, CDH and other subsystems need to be able to supervise several inputs and outputs as well as process and store data within a fixed time-period. These all need to be performed in a reliable and predictable fashion throughout the lifetime of the mission. The needs of each mission can vary greatly, but basic deterministic and reliable processing is a fundamental requirement. The following are important when considering FSW design:

- Implication of CDH processors on FSW
- Frameworks
- Operating systems
- Software languages
- Mission operations and ground support suites
- Development environment, standards, and tools

8.4.1 Implication of CDH Processors on FSW

The processor and memory available on the CDH can put significant limitations on the FSW. For some of the smaller jobs, or to reduce electronic complexity, smaller processors are used (distributed processing). These have typically been thought of as embedded processors, with many of them containing dedicated memory. Modern integrated space avionics, including heterogeneous and mixed criticality architectures, also impact operational constructs and can

contribute to advanced configurations (such as multiple modular redundant systems architectures) which can allow advanced paradigms for radiation tolerance and system redundancies in critical small spacecraft missions.

8.4.2 Frameworks

In the context of SSA, a FSW framework can be described as a hierarchal architecture, sometimes referred to as a set of lego-like building block constructs, partitions, and functions. This emerging system-of-systems concept describes the large-scale integration of many independent, self-contained systems that work together to satisfy a global need. Examples of commonly used frameworks include:

- cFS (https://cfs.gsfc.nasa.gov)
- F' (https://github.com/nasa/fprime)
- NanoSat Mission Operations Framework (https://nanosat-mo-framework.github.io/)
- Spacecloud (https://space-cloud.io/)
- ROS (https://www.ros.org/)

8.4.3 Operating Systems

Operating systems manage computer hardware, software resources, and provide common services for computer programs. Examples of commonly used operating systems include:

- VxWorks
- RTEMS
- FreeRTOS
- Linux

8.4.4 Software Languages

System programming involves designing and writing computer programs with software languages that allow the computer hardware to interface with the programmer and the user, leading to the effective execution of application software on the computer system. State-of-the-art small spacecraft have used C, C++, Python, Arduino and other software languages.

8.4.5 Mission Operations and Ground Support Suites

Although not directly used on the spacecraft, mission operations and ground support suites must also use software and systems for testing, and to monitor, command, control, and communicate with the spacecraft, as well as display status and disseminate data across all aspects of a space mission (including spacecraft performance and procedures, systems health, science and technology data handling and management, and telemetry tracking and control). For smaller spacecraft and missions, it is usually best to use the same ground support software for mission operations, integration and testing, and development and testing. There are numerous open-source and proprietary tools and programs available for these activities. A small set of tools that have been used at NASA are described below. For more information, please refer to the *Ground Data System and Mission Operations* chapter.

8.4.6 Development Environment, Standards, and Tools

Development environment, standards, and tools are used to design, develop, validate, and operate small spacecraft missions, with adherence to accepted software and space mission standards. Examples of commonly used development tools include:

- Version control tools
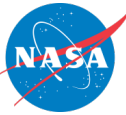
- Auto-generation of software
- Simulations and simulators
- Software best practices and NPR7150

8.5 On the Horizon (TRL 1-4): Command and Data Handling

Many CDH systems will continue to follow trends set for embedded systems. Short-duration missions in LEO will continue to take advantage of advances made by industry leaders who provide embedded systems, technologies, and components. In keeping with the low-cost, rapid development theme of CubeSat-based missions, many COTS solutions are available for spacecraft developers.

While traditional CDH processing needs are relatively stagnant, as small satellites are being targeted for flying increasingly data-heavy payloads (i.e., imaging systems) there is new interest in advanced onboard processing for mission data. Typically, these higher performance functions would be added as a separate payload processing element outside of the CDH function.

Next-generation SSA/PSA distributed avionics applications are integrating FPGA-based software-defined radios (SDRs) on small spacecraft (45). A SDR can transmit and receive in widely different radio protocols based on a modifiable, reconfigurable architecture, and is a flexible technology that can enable the design of an adaptive communications system. This can increase data throughput and enable software updates on-orbit, also known as re-programmability. Additional FPGA-based functional elements include imagers, Artificial Intelligence and Machine Learning (AI/ML) processors, and subsystem-integrated edge and cloud processors. The ability to reprogram sensors or instruments while on-orbit have benefited several CubeSat missions when instruments do not perform as anticipated, or when entering an extended mission phase that requires subsystems or instruments to be reprogrammed.

In keeping with trends seen in other disciplines and industries, the Industry 4.0 and "digitally managed everything" is absolutely of critical importance for technological and programmatic efficiencies in SSA systems development. Following are some modern tools, technologies, and approaches that should be considered when developing and deploying next-generation small spacecraft avionic systems:

- Artificial intelligence, machine learning, and machine vision
- Robotics and automation
- Model-based systems engineering
- Embedded systems / edge computing
- Internet-of-space-things
- Cloud computing
- Augmented reality/ virtual reality / mixed reality
- Software-defined-everything
- Advanced manufacturing
- Digital twin

8.6 On the Horizon (TRL 1-4): Flight Software

FSW is key to mission success. The field of software is a very dynamic environment that is continuously evolving. The challenges with flight software usually remain the same regardless of the size of the spacecraft (CubeSat to SmallSat) and are related to the size and complexity of the endeavor. Overall, FSW can be known to cause scheduling and implementation issues, especially
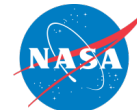
during integration and test. There is usually a temptation to add additional features, and all these factors can drive up overall complexity of the FSW and increase risk to the mission as a whole.

It is essential that FSW be as simple as possible. It is critical to survey options and plan early in any FSW effort. Wherever possible, early development and testing should be performed. Efforts to add additional features should be looked at very critically with a strong effort to stick to the existing plan. With good planning and careful execution, a favorable outcome can be achieved. It is becoming more common to update software after the hardware is delivered (or even launched), and there are now software frameworks such as cFS that have features to enable software updates after deployment.

On the horizon FSW will soon include multicore processor operating systems and programming, as learning how to harness multicore processors differently than Microsoft Windows does will enable true real-time multiprocessing. On the horizon FSW will also include artificial intelligence (e.g., Nvidia); FSW for multicore, multiprocessor, and heterogenous platforms (e.g., AMD-Xilinx Versal); and FSW (middleware) for constellations of SmallSats with resource management, scheduling and task assignment, and fault tolerance.

Spacecraft autonomy is an emerging capability and SmallSat designers have particular interest in the following characteristics for autonomous systems:

- Situational and self-awareness
- Reasoning and acting
- Collaboration and interaction
- Engineering and integrity

Spacecraft autonomy can be considered part of management, direction, and control for all subsystems and functions in a spacecraft. CDH takes input from, and provides direction to, all subsystems (ADCS, Power, Propulsion, Comm, vehicle health, etc.). Those subsystems may also have a degree of autonomy depending on the complexity of its local "smart subsystems" processor. The NASA 2020 Technology Roadmap defines autonomous systems as a cross-domain capability that enables the system to operate in a dynamic environment independent of external control (46).

Some autonomous systems now implement a heterogeneous architecture, meaning they contain multiple processors with varying levels of performance and capabilities. For instance, higher performance modules and components can be used for sophisticated data processing, AI and onboard computing for both spacecraft and mission performance optimization—as well as real-time adaptive analysis of science data—while lower performance onboard processors and FPGAs conduct the routine spacecraft operations functions and interact with the subsystems which also may include distributed performance cascades.

Cybersecurity is also becoming a driving factor in FSW. This is mostly handled through the encryption of commands, but as space becomes more crowded, more sophisticated cybersecurity schemes may be required of SmallSats. SmallSats can also be a testbed for demonstrating new hardware and software for cybersecurity.

8.7 Summary

Space applications now require considerable autonomy, precision, and robustness, and are refining technologies for such operations as on-orbit servicing, relative and absolute navigation, inter-satellite communication, and formation flying. An exciting trend is that small spacecraft missions are becoming more complex as these platforms are now being used for lunar and deep space science and exploration missions. Small spacecraft technology is expanding to meet the needs of increasing small spacecraft mission complexity. This has accelerated over the past few

years to achieve the next gen goals of using small spacecraft to collect important science in deep space, and mitigate risk for larger, more complex mission-critical situations. In parallel, spacecraft electronics have matured with higher performance and reliability, and with miniaturized components that meet the growing needs of these now very capable spacecraft.

The 2022 Small Spacecraft Avionics chapter has been updated with a broader, interrelated framework, where CDH, FSW, and smart payloads are not just independent space platform subsystems but are part of an integrated avionics ecosystem which includes all electronic elements of a space platform, now primarily digitally based and or managed. Also, SSA should not be considered as an isolated spaceflight technology component, but rather as a core digital engineering technology emphasis area, capable of taking advantage of and integrating products, processes, and technologies from other disciplines. To continue to be relevant and efficient, the SSA communities must remain cognizant and receptive of the continuously evolving nature of the digital based Industry 4.0 technology revolution now being evidenced in other related and/or associated vertical disciplines and solutions.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email for further contact.

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Chapter Contents

Chapter Glossary

(ADCS)	Attitude Determination and Control System
(BPF)	BandPass Filters
(CDH)	Command and Data Handling
(COTS)	Commercial-off-the-Shelf
(DORA)	Deployable Optical Receiver Aperture
(DLR)	German Aerospace Center
(DSN)	Deep Space Network
(DSP)	Digital Signal Processing
(DVB-S2)	Digital Video Broadcast Satellite Second Generation
(FCC)	Federal Communications Commission
(FIPS)	Federal Information Processing Standard
(FPGAs)	Field Programmable Gate Arrays
(FSM)	Fine-steering Mirror
(FSO)	Free Space Optical
(IARU)	International Amateur Radio Union
(IEEE)	Institute of Electrical and Electronics Engineers
(ISARA)	Integrated Solar Array and Reflectarray Antenna
(ISM)	Industrial, Scientific, and Medical
(ISOC)	Inter-spacecraft Optical Communicator
(ISS)	International Space Station
(JAXA)	Japanese Aerospace Exploration Agency
(JPL)	Jet Propulsion Laboratory
(LADEE)	Lunar Atmosphere and Dust Environment Explorer
(Lasercom)	Laser Communications
(LCH)	Laser Clearing House
(LCT)	Laser Communication Terminals
(LDPC)	Low-Density Parity-check Code
(LLCD)	Lunar Laser Communications Demonstration
(LNA)	Low Noise Amplifier
(LSRB)	Laser Safety Review Board
(MA)	Multiple Access
(MarCO)	Mars Cube One
(MEMS)	Micro-Electro-Mechanical Systems

(MRR)	Modulating Retro-Reflector
(NEN)	Near Earth Network
(NICT)	National Institute of Information and Communications Technology
(NOAA)	National Oceanic and Atmospheric Administration
(NPR)	NASA Procedural Requirements
(NTIA)	National Telecommunications and Information Administration
(OCSD)	Optical Communication and Sensor Demonstration
(OCTL)	Optical Communication Telescope Laboratory
(OSIRIS)	Optical Space Infrared Downlink System
(PAT)	Pointing, Acquisition, and Tracking
(RF)	Radio Frequency
(SBIR)	Small Business Innovative Research
(SCaN)	Space Communications and Navigation
(SDR)	Software Defined Radios
(SME)	Subject Matter Expert
(SNR)	Signal-to-Noise Ratio
(SOTA)	Small Optical Transponder
(SWaP)	Size, Weight, and Power
(SST)	NASA Small Spacecraft Technology Program
(TDRS)	Tracking and Data Relay Satellite
(TMA)	Technology Maturity Assessments
(TRL)	Technology Readiness Levels
(TT&C)	Tracking, Telemetry & Command
(USTP)	University Smallsat Technology Partnerships
(VSOTA)	Very Small Optical Transponder
(WFF)	Wallops Flight Facility

9.0 Communications

9.1 Introduction

For most missions the communication system enables the spacecraft to transmit data and telemetry to Earth, receive commands from Earth, and relay information from one spacecraft to another. A communications system consists of the ground segment: one or more ground stations located on Earth, and the space segment: one or more spacecraft and their respective communication payloads. The three functions of a communications system are receiving commands from Earth (uplink), transmitting data down to Earth (downlink) and transmitting or receiving information from another satellite (crosslink or inter-satellite link) (Figure 9.1). There are two types of communication systems: radio frequency (RF) and free space optical (FSO), FSO is also referred to as laser communications (lasercom).
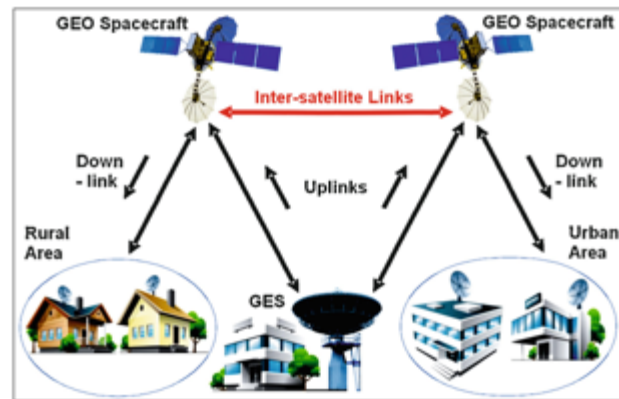
Figure 9.1: Satellite uplink, downlink, and crosslink. Credit: D. Stojce (2019).

Most spacecraft communications systems are radio frequency based. They typically operate within the designated Institute of Electrical and Electronics Engineers (IEEE) radio bands of 300 MHz to 40 GHz. A RF system communicates by sending data using electromagnetic waves to and from antennas. Information is modulated onto radio frequency electromagnetic waves and sent over a channel, through the atmosphere or space, to the receiving system where it is demodulated (Figure 9.2).

Although RF systems are typically used for low-rate space communication, recent developments in FSO communications have made it a compelling alternative to RF systems, particularly for high-rate communication. FSO systems consist of a transmitting terminal and receiving terminal. Like an RF system, information is modulated onto electromagnetic waves (at optical frequencies) and sent over a channel to the receiving system. FSO links operate at a much higher frequency than RF links, generally at near-infrared bands (e.g., 1064 nm or 1550 nm). Visible light is often not
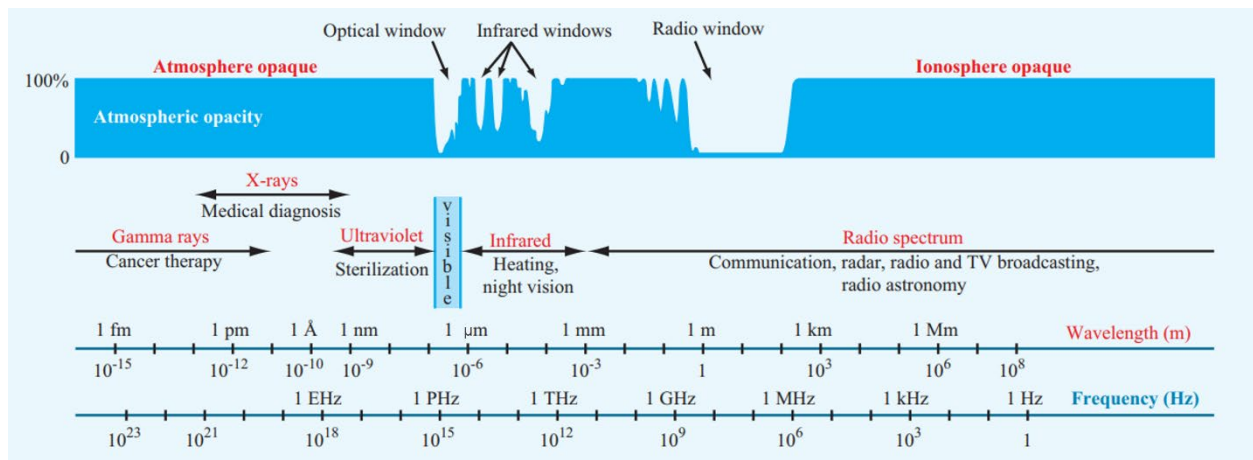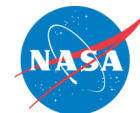
Figure 9.2: Atmospheric opacity of the electromagnetic wave spectrum with the infrared and radio windows used by spacecraft for communication. Credit: Microwave Radar and Radiometric Remote Sensing by Ulaby and Long.

used due to eye safety concerns for technicians at the terminals. The use of higher frequencies and wider bandwidths can support higher data rates, but the shorter wavelengths also result in narrower beamwidths that require more accurate pointing towards the communication terminal both more accurately and precisely.

This chapter organizes the state-of-the-art in small spacecraft communications technologies into two main categories: RF and FSO. Tables at the end of each section list hardware options for RF and developing FSO technologies for mission designers to consider.

The information described below is not intended to be exhaustive but provides an overview of current state-of-the-art technologies and their development status for a particular small satellite subsystem. The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The information is for awareness and guidance only. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

9.2 Radio Frequency Communications

A radio communication system includes a radio transmitter, a free space communication channel, and a radio receiver. At the top level, a radio transmitter system consists of a data interface, modulator, power amplifier, and an antenna. The transmitter system uses the modulator to encode digital data onto a high frequency electromagnetic wave. The power amplifier then increases the output RF power of the transmitted signal to be sent through free space to the receiver using the transmit antenna.

The radio receiver system uses a receiving antenna, low noise amplifier, and demodulator to produce digital data output from the received signal. The receiving antenna collects the electromagnetic waves and routes the signal to the receiver, which then demodulates the wave and converts the electrical signals back into the original digital message. Low noise amplifiers are sometimes employed to minimize thermal noise in certain frequency bands and/or increase the received signal strength. In many cases, the functions of the modulator and demodulator are combined into a radio transceiver that can both send and receive RF signals.

Radio frequency communications for spacecraft are conducted between 30 MHz and 60 GHz. The lower frequency bands (up to S-band) are typically more mature for SmallSat use, however extensive use of these bands has led to crowding and challenges acquiring licensing. Higher frequencies offer a better ratio of gain-to-aperture-size, but this is offset by the increased atmospheric attenuation at those frequencies and the higher free space loss that is directly proportional to the square of the frequency.

9.2.1 Frequency Bands

Satellite communications are conducted over a wide range of frequency bands. The typical bands considered for small satellites are UHF, S, X, and Ka. The most mature bands used for CubeSat communication are VHF and UHF frequencies. There has been a shift in recent years towards S and X, with Ka-band also being used for recent & future small satellite communications. The move to higher frequency bands has been driven by a need for higher data rates. At the higher frequencies, there is generally greater atmospheric and rain attenuation adding to increased free space loss. This needs to be compensated for with higher power transmission and/or high gain

antennas narrower beamwidths. Moving to higher-gain antennas increases the pointing accuracy required for closing the link. See Table 9-1 for a list of RF bands.

NASA spacecraft, which use the government bands of S-band, X-band and Ka-band, may use the NASA Near Space Network (NSN). The primary frequency bands of S, X, and Ka are more advantageous than using the UHF band, which has a higher probability of local interference. Satellite Tracking, Telemetry & Command (TT&C) is typically conducted over S-band. Non-NASA spacecraft have access to a wide variety of ground system options ranging from do-it-yourself to pay-per-pass services.

In L-band, CubeSats can take advantage of legacy communications networks such as Globalstar and Iridium by using network-specific transponders to relay information to and from Earth. These networks remove dependence on dedicated ground station equipment. However, they can only be used at orbital altitudes below the communication constellation and require experimental frequency authorization.

Table 9-1: Radio Frequency Bands	
Band	**Frequency**
VHF	30 to 300 MHz
UHF	300 to 1000 MHz
L	1 to 2 GHz
S	2 to 4 GHz
C	4 to 8 GHz
X	8 to 12 GHz
Ku	12 to 18 GHz
K	18 to 27 GHz
Ka	27 to 40 GHz
V	40 to 75 GHz

Ku-, K-, and Ka-band communication systems are the state-of-the-art for large spacecraft, especially in spacecraft-to-spacecraft communications, but they are still young technologies in the CubeSat world. They are becoming more attractive to SmallSat designers as the lower frequencies become more congested. At the higher frequencies, rain fade becomes a significant problem for communications between a spacecraft and Earth (1). Nonetheless, the benefits of operating at higher frequencies have justified further research by both industry and government alike. At JPL, the Integrated Solar Array and Reflectarray Antenna (ISARA) mission demonstrated high bandwidth Ka-band CubeSat communications with over 100 Mbps downlink rate (2). The back of the 3U CubeSat was fitted with a high gain reflectarray antenna integrated into an existing solar array. The successful demonstration of the reflectarray on ISARA became the basis for the Mars Cube One (MarCO) mission to Mars. The MarCO mission uses two twin CubeSats for a communications relay between the InSight lander and Earth. Using a X-band reflectarray they were able to successfully complete their mission (3). Another mission to use Ka-band for DTE communications was the Kepler telescope, launched in 2009. With future missions being increasingly data hungry, we are likely to see a shift towards Ka-band and, possibly, even higher frequencies.

CubeSats have also used the unlicensed Industrial, Scientific, and Medical (ISM) bands for communications. The Ames TechEdSat team has successfully demonstrated WiFi to downlink data at 1 Mbps. Notably, a group at Singapore's Nanyang Technological University used a 2.4 GHz ZigBee radio on its VELOX-I mission to demonstrate commercial-off-the-shelf (COTS) land-based wireless systems for inter-satellite communication (4). Similarly, current investigations are looking at using wireless COTS products, such as Bluetooth-compatible hardware, for inter-satellite communications (5).

9.2.2 System Architecture

A small satellite RF communications system consists of a transceiver comprised of a radio, an amplifier, and an antenna. Radios receive a message from the Command and Data Handling (CDH) subsystem, then produce and modulate an electromagnetic wave to create a signal. They are responsible for generating the signal and modulating or demodulating it. The radio is also where coding may be added to the signal. Channel coding is added to provide data error detection

and correction capabilities, which ensures reliable communication under the conditions imposed by the satellite transmission path. From Shannon's Equation (6), it is known that the information capacity of a channel is related to its bandwidth and signal-to-noise ratio (SNR). The channel capacity (information flow) can be increased by increasing the SNR or the bandwidth, and many modulation and coding schemes make effective use of this tradeoff.

Radios offer some power amplification, but often the signals from small satellites require a greater boost. The power amplifier will take the signal from the radio and increase the RF output power before sending it to the transmit antenna. On the receive side, a low noise amplifier will take the weak signal from the receive antenna and amplify it while minimizing thermal noise. A bandpass filter might be used before the LNA to reject undesired frequencies. The radio will then be able to process the stronger signal with higher accuracy. In RF communications the role of the antenna is to increase and focus the strength of the signal in a specific direction. The digital message encoded on the RF carrier signal will be sent to and from the antennas of each system. See Figure 9.3 for an example transmit and receive block diagram.
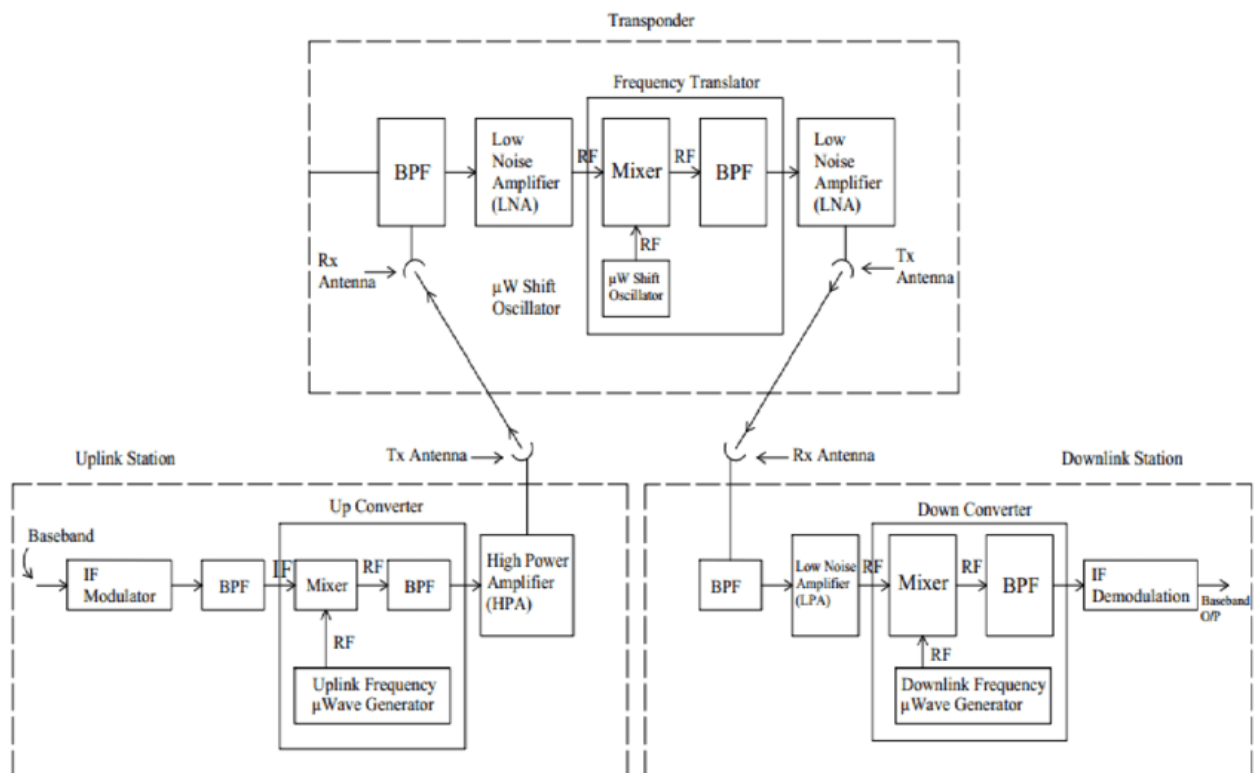
Figure 9.3: Transmit and receive block diagram. Credit: Karim et al. (2018). http://creativecommons.org/licenses/by/4.0/

9.2.3 Major Components in SmallSat Communication Systems

- Radio or Modulator/Demodulator: on the transmit side it produces, modulates, codes, and amplifies an electromagnetic wave to create a signal. Adds modulation and coding as needed. As a receiver it decodes and demodulates received signals.
- Mixers: RF mixers are used in communications systems to change the frequency of the signal. If the frequency generated by the radio is not the desired transmit frequency, then an upconverter will convert the signal to a higher frequency for transmit. Similarly, the downconverter will down convert a receive frequency to a lower one for processing.

- Filters: bandpass filters are used to reject undesired frequencies, typically before the LNA or downconverter.
- Amplifier: a power or gain amplifier is required for a transmit system. A low noise amplifier (LNA) is required for a receive system. LNAs, in addition to amplifying the (low power) received signal, serve to minimize the system noise temperature.
- Antenna: increases the strength of a signal in a specific direction, relative to the same signal strength without directionality. Transmits signals fed to it by a transmitter and receives signals propagated across free space. Antennas can be low-gain & omni-directional with a broad beam, or high-gain & directional with a narrow beam.
- Encryption: a cryptographic unit is an integrated encryptor/decryptor device that provides secure uplink, downlink, or crosslink for satellite communication links. Most small satellite designers will not require a cryptographic payload unit based on their threat level and may be able to use the communications radio for simple encryption schemes.
- Spread-spectrum communication applies a known frequency spreading function to the signal, which helps reduce interference from other transmitters, and provides more secure communications; as such, it is often used for multi-way communication networks. For example, the NASA Tracking and Data Relay Satellite (TDRS) multiple-access mode requires spread spectrum signals to support multiple simultaneous communication links.

9.2.4 Design Considerations

As with all spacecraft subsystems, there are power and mass constraints placed on the comm system. Based on these restrictions several trade studies need to be performed to choose the optimal design. When designing an RF communications system, the first trades performed are for data rate, power consumption, and total mass. For example, a mission with high data rate needs would select a high frequency such as X-band for downlink and a directional high-gain antenna. Based on the ground station locations available, engineers would perform link budget analyses to determine the minimum power needed for a specific ground station antenna. This analysis would factor in rain and atmospheric attenuation, as well as modulation and coding. A few different link budget trades will be run, varying antenna size, RF output power and data rate. Each link will return a different margin of decibels, representing the reliability of the system. The engineers will proceed to calculate the final mass and power for each configuration. The mission designer will have a limit on mass and power constraints for the communications subsystem. Each configuration traded will compare data rate, power, and mass. A high data rate downlink may cost a high amount of mass for the antenna and power for the amplifier and radio. Conversely, a low-power, low-mass system may have a lower data rate.

Another factor that is considered in the design phase is pointing. Depending on the orbit of the satellite and whether the link is Uplink/Downlink, or Crosslink, the system may have a specific pointing requirement. Large satellites frequently use gimbals--platforms that can pivot to point their antennas. The addition of a gimbal will increase the overall mass and power draws of the system. CubeSats frequently trade high-gain antennas for low-gain, omni-directional ones to maintain the link regardless of directionality. CubeSats may also change their attitude to point a body-mounted antenna, rather than use a gimbal.

9.2.5 Policies and Licensing

Any non-Federal US spacecraft with a transmitter must be licensed by the Federal Communications Commission (FCC). The types of RF licenses used by small satellites are: Amateur (FCC Part 97) and Experimental (FCC Part 5) (7). An amateur license type of authorization is limited to hobbyists and non-profit use and comes with many FCC restrictions. Experimental Part 5 licenses are commonly used for university CubeSats and can be granted for

a CubeSat operating in the amateur band (A SmallSat or SmallSat constellation can also apply under provisions of Part 25). A spacecraft with any sort of remote sensing capability must contact the National Oceanic and Atmospheric Administration (NOAA) to find out if a NOAA license is required. A NOAA license is not an RF license and conveys no authority for the radiation of RF energy for communication. For government missions the National Telecommunications and Information Administration (NTIA) is the licensing authority.

For amateur licensing, there must be an FCC licensed amateur radio control operator. Downlink telemetry and communications cannot be obscured (encrypted). Use of science gathered via amateur radio downlink for profit ("pecuniary interest") is prohibited. Frequency "assignment" in the amateur-satellite allocations requires coordination, a process administered by the International Amateur Radio Union (IARU) (8).

In 2018, the FCC adopted a Notice of Proposed Rulemaking to develop a new authorization process tailored specifically to small satellite operations, keeping in mind efficient use of spectrum and mitigation of orbital debris. Small satellites that would qualify for the new rules include those with 10 or lesser number of satellites under a single license. All individual satellites will have to be 10 cm or larger in the smallest dimension and weigh less than 180 kg. The maximum in-orbit lifetime of each individual satellite will be six years, including de-orbiting time, and they would have to be deployed under 600 km altitude. Each satellite will have a unique telemetry marker for tracking and will not release any debris (9).

9.2.6 Encryption

Encryption is the process of encoding information to conceal it from outside actors. Small satellites can use a cryptographic unit to encrypt or decrypt data prior to transmission. When data is being prepared for transmission, it is broken up into packets. These packets are then scrambled according to the encryption scheme being used. An encryption scheme uses an encryption key generated by an algorithm to encode the data. The authorized receiver of the encrypted data will be able to decrypt the message using the appropriate key. Without the authorized key, decrypting the data will be extremely difficult.

With the increased proliferation of small satellites in low-Earth orbit comes an increase in vulnerabilities. Many SmallSats are comprised of COTS hardware and/or open-source software. While this strategy allows for a more flexible design approach, adversaries can gain insight into the design. Encryption of data in transit prevents other actors from commanding satellites or intercepting transmissions.

NASA requires any of its propulsive spacecraft within 2 million kilometers of Earth to protect their command uplink with encryption that is compliant with Level 1 of the Federal Information Processing Standard (FIPS) 140-3 (10). The FCC has also considered requiring encryption on the telemetry, tracking, and command communications as well as mission data for propulsive spacecraft, but decided not to incorporate a specific requirement at this time. A satellite with an amateur license cannot encrypt transmissions in any way and must consist of open information. The eligibility rules are listed in 47 CFR Part 97 (11).

9.2.7 Antennas

Antennas are used for propagating data through free space using electromagnetic waves. RF antennas are typically sized for their respective frequencies. This means that antennas are often chosen or designed specifically for their mission. COTS antennas are available for SmallSats and can be built to order. For missions that don't have high data rate requirements, a simple patch or monopole antenna with low gain and efficiency will suffice. Due to their low directionality, these antennas can generally maintain a communication link even when the spacecraft is tumbling, which is advantageous for CubeSats lacking good attitude and accurate pointing control. New

developments in antenna design have put technologies like the deployable reflector antenna, reflectarray, and passive or active array antennas on the horizon for small satellites. Please see Table 9-3 for information on commercially available antennas for SmallSats/CubeSats.
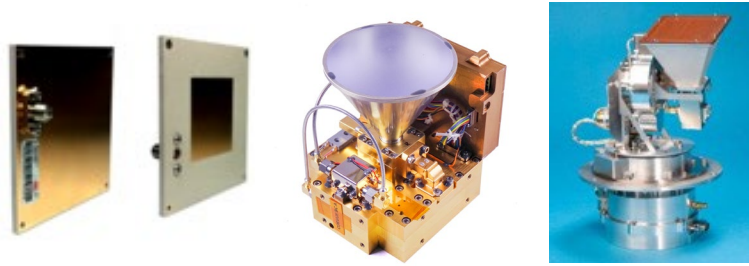
There are two primary classifications of antenna: fixed or deployable. Fixed antennas do not require any power or triggering mechanisms.

Figure 9.4: (from left to right) CubeSat-compatible S-band patch antenna (IQ Wireless), X-band high-gain antenna and pointing mechanism (Surrey Satellite Technology, Ltd.), and Ka-band transmitter with a horn antenna (Astro Digital).

They remain stationary in the position that they are attached to the spacecraft. This includes patch antennas, array antennas, monopole antennas, omni-directional antennas, and horn antennas (see Figure 9.4). Deployable antennas require power to deploy and use mechanisms to configure into their final position. This includes whip antennas, parabolic reflectors, reflectarrays, helical and turnstile antennas (see Figure 9.5).

A communications link is often characterized by the frequency and data rate. The antenna is a key design decision for meeting data rate objectives by increasing link margin. Increasing the aperture or diameter of an antenna inceases the link margin, which can allow designers to increase the data rate of the system or reduce the necessary transmit power.
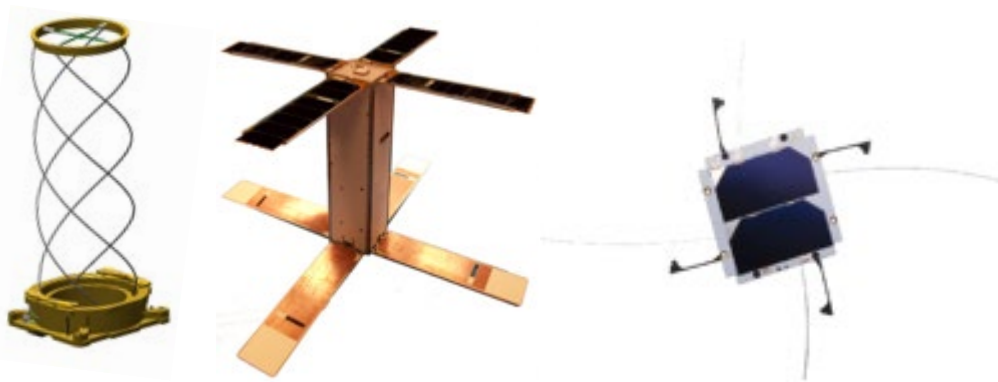
Figure 9.5: (from left to right) Example of deployable quadrifilar helical antenna (Helical Communication Technologies), SNaP spacecraft with Haigh-Farr's deployable UHF Crossed Dipole antenna (Space Missile and Defense Command), and EnduroSat UHF antenna with EnduroSat solar panels (EnduroSat).

9.2.8 Radios

Radios for SmallSat downlink are transceivers (transmitter and receiver in one). Transceivers convert digital information into an analog RF signal using a variety of modulation and coding schemes. Radios for TT&C are designed for low data rates, with high reliability and only need to transmit health data and receive commands. Traditional radios may be locked to a single frequency band and modulation/coding scheme based on their design and build. Software defined radios (SDR) have some or all of the radio's functions implemented in Digital Signal Processing

(DSP) software rather than hardware, Figure 9.6. Furthermore, spacecraft teams can change such characteristics in-flight by uploading new settings from the ground. By using Field Programmable Gate Arrays (FPGAs), SDRs have great flexibility that allows them to be used with multiple bands, filtering, adaptive modulation, and coding schemes, without much (if any) change to hardware (12). SDRs are especially attractive for use on CubeSats, as they are becoming increasingly small and efficient as electronics become smaller and require less power. NASA has been operating the Space Communications and Navigation (SCaN) Testbed on the International Space Station since 2012 for the purpose of SDR TRL advancement, among other things (13). Many radios can provide RF output power to the antenna directly. For higher power applications, an external RF amplifier or high gain antenna may be

Figure 9.6: Example of software defined radio, tunable in the range 70 MHz to 6 GHz. Credit: GomSpace.

used. The reader is encouraged to refer to the SmallSat Avionics chapter for further information on FPGAs and SDRs. See Table 9-4 for information on commercially available radios for SmallSat/CubeSats.

This report recommends efficient modulation and coding schemes for spacecraft power and bandwidth to increase the data rate and meet bandwidth constraints with the limited power and mass for CubeSat spacecraft. Advanced coding, such as the CCSDS low-density parity-check code (LDPC) family, with various code rates is a powerful technique to provide bandwidth and power with high-order modulation to achieve high data rate requirements for CubeSat missions. Digital Video Broadcast Satellite Second Generation (DVB-S2), a significant satellite communications standard, is a family of modulations and codes for maximizing data rates and minimizing bandwidth use. DVB-S2 uses power and bandwidth efficient modulation and coding techniques to deliver performance approaching theoretical limits of RF channels. NASA's NSN has conducted testing at NASA Wallops Flight Facility (WFF) to successfully demonstrate DVB-S2 over a S-band 5 MHz channel achieving 15 Mbps with 16 APSK LDPC 9/10 code (14).

9.2.9 On the Horizon RF Communications

A CubeSat constellation may involve numerous CubeSats in the constellation (e.g., tens or hundreds). Each CubeSat is typically identical from a communication perspective. One or more CubeSats may be mother ship-capable while the others may be subordinate (e.g., daughterships). CubeSat constellations optimize coverage over specific areas or improve global revisit times to fulfill mission objectives. There is growing interest among the NASA science community in using constellations of CubeSats to enhance observations for Earth and space science. NASA GSFC has conducted research on future CubeSat constellations, including CubeSat swarms, daughter ship/mother ship constellations, NEN S- and X-band direct-to-ground links, TDRS Multiple Access (MA) arrays, and Single Access modes. The MA array requires the use of spread-spectrum to support multiple simultaneous communications links to increase coverage and link availability

As CubeSat missions employ more automation, constellations could exchange information to maintain precise positions without input from the ground. Radiometric ranging is a function recently incorporated into CubeSat transceivers. A timing signal is embedded into the radio signal and is used to determine the range to the spacecraft. Using this method along with directional vectors obtained from ground antennas allows for trajectory determination of satellites beyond low-Earth orbit. Spacecraft may relay data to increase the coverage from limited ground stations. Inter-CubeSat transponders may very well become a vital element of eventual deep space missions, since CubeSats are typically limited in broadcasting power due to their small size, and may be better suited to relay information to Earth via a larger, more powerful mothership.

Spacecraft routinely use transponders, however, networked swarms of CubeSats that pass information to each other and then eventually to ground, have not flown. Developing networked swarms is less of a hardware engineering problem than a systems and software engineering problem in that one must manage multiple dynamic communication links. As of 2024, the two MarCO SmallSats, CAPSTONE, BioSentinel, ArgoMoon, LunaH-Map, have operated beyond low-Earth orbit and both use the IRIS CubeSat Deep Space transponder. Both the MarCO satellites used a deployable reflectarray panel at X-band and were equipped with a full-duplex radio providing both UHF and X-band coverage (15). This allowed for near real-time updates of the InSight rover's landing. After this success, more SmallSats may be deployed beyond low-Earth orbit. The ability to provide crosslink relay hops for large spacecraft will prove to be critical for deep space missions.

Table 9-3: Antennas									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Frequency Band**	**Gain**	**Polarization**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
MMA Design	LAMBDA	Deployable Sinuous Antenna	100	UHF	>-1	Dual CP	4000	20x15x10 (stowed) 100x100 (deployed)	N
Oxford Space Systems	Yagi antenna	Deployable	156.5-162.5	VHF	>6.5	Dual Linear	<1000	92.5 x 50 (deployed)	Y
Spacemanic	SAM_Antenna_Module	Dipole Cross Dipole	144 399 420	VHF, UHF	2.1	Linear/RHCP	55	9.8X9.8X0.56	Y
Haigh-Farr, Inc.	Part Number: 17100	Crossed Dipole	307	VHF, UHF	--	RHCP	267	32x8x1	Y
GomSpace	NanoCom ANT430	Omni Canted Turnstile	400-435	VHF, UHF	1.5	Circular	30	10x10	Y
Helical Communications Technologies	Helios Deployable Antenna	Helical	400-3000	VHF, S	3	Circular	180	10x10x3.5	Y
NanoAvionics	CubeSat UHF Antenna System 1x1U	Turnstile	400-500	UHF	1.37	--	33	10x10x0.7	Y
NanoAvionics	CubeSat UHF Antenna System 1x2U	Turnstile	400-500	UHF	2.31	--	50	20x10x0.7	Y
NanoAvionics	CubeSat UHF Antenna System 2x2U	Turnstile	400-500	UHF	3.4	--	65	20x20x0.7	Y
EnduroSat	UHF Antenna III	Whip/Burn-wire	435-438 or 400-403	UHF (VHF option)	> 0	RHCP	85	10x10	Y
ISISPACE	CubeSat Antenna System for 1U/3U	Tape	--	VHF, UHF	0	Circular, Linear	89	10x10x0.7	Y
Flexitech Aerospace	600MHz - 10GHz Spiral Antenna	Spiral	600-10000	UHF, L, S, C, X	3	Circular	1283	17x17x8.5	N

National Aeronautics and Space Administration

Table 9-3: Antennas									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Frequency Band**	**Gain**	**Polarization**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
Oxford Space Systems	Helical antenna	Deployable	862–928	UHF	>6.5-7.5	RHCP	<235	30 (deployed helical length)	Y
CesiumAstro	Vireo L-Band Phased Array	APA	960	L	22	Circular	49100	114x100x13.5	N
SkyFox Labs	piPATCH-L1E1	Patch	1575.42	GPS-L1 GALILEO E1	--	--	50	9.8x9.8x1.3	Y
NAL Research Corporation	Antenna SYN7391-A/B/C (Iridium)	Flat Mount	1610-1626.5	L	4.9	RHCP	31	4.6x.4.3x1.0	Y
Oxford Space Systems	Helical antenna (high RF power handling)	Deployable	1980-2200	S	>4.0-4.5	RHCP	<1200	60 (deployed helical length)	Y
IQ Spacecom	S-Band Single Patch Antenna	Patch	1980-2500	S	6	Circular	49	7x7x0.34	Y
IQ Spacecom	S-Band Dual Patch Antenna	Patch	1980-2500	S	6	Circular	62	8x10x0.34	Y
IQ Spacecom	S-Band High Gain Patch Antenna	Patch	1980-2500	S	11.5	Circular	179	16x16x0.34	Y
Flexitech Aerospace	2-2.5GHz Turnstile Antenna	Turnstile	2000-2500	S	5	Circular	173	--	N
SkyLabs	S-band Patch Antenna	Patch	2025-2110	S	6	LHPC/RHPC	70	8.2x8.2x1.1	Y
Vulcan Wireless	ANT-S/S Unified S-Band Antenna	Patch	2025-2300	S	6.5	Circular	76	8x8x1	Y
EnduroSat	S-band Commercial Patch Antenna	Patch	2025-2110	S	7	Selectable Circular	81	9.8x9.8x0.6	Y
EnduroSat	S-band Wideband Patch Antenna	Patch	2025-2110 2200-2290	S	5	RHCP	115	9.8x9.8x0.7	Y

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Table 9-3: Antennas									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Frequency Band**	**Gain**	**Polarization**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
ANYWAVES	S-Band TT&C Antenna	Patch	2025-2290	S	6.5	RHCP/LHCP	132	8x8x1.2	Y
Haigh-Farr, Inc.	P/N 21060	Waveguide	2020-2120	S	25	LHCP	667	10x10x4.1	N
ISISPACE	S-Band Patch Antenna	Patch	2200-2290	S	6.5	RHCP	50	8x8x1	N
CesiumAstro	Nightingale AAA-2SF1	Patch	2025 2200	S	5	RHCP LHCP	480	12x9.5x5	Y
AeroSpace Lab	SBA-TX	Patch	2200–2290	S	6.1 dBic	RHCP	118	9.5x9.5x2.4	Y
AeroSpace Lab	SBA - RX	Patch	2025–2110	S	6.1 dBic	CP	117	9.5x9.5x2.4	Y
Haigh-Farr, Inc.	S-band Patch Antenna	Patch	2245-2245	S	--	RHCP	48	4.8x6.5x6.5	Y
EnduroSat	S-band ISM Patch Antenna	Patch	2400-2450	S	8.3	LHCP	64	9.8x9.8x0.6	Y
Oxford Space Systems	Offset Reflector Antenna (muti feed)	Deployable	5030-5500	C	> 42	Linear	<2720	350 (deployed reflector diameter, scalable to 600)	N
IQ Spacecom	X-Band Single Patch Antenna	Patch	7145-7250 8025-8400	X	6	Circular	10	3.5x3.5x0.18	Y
IQ Spacecom	X-Band High Gain Antenna	Patch	7145-7250 8025-8400	X	10	Circular	12	4x6x0.18	Y
Oxford Space Systems	(Single) Hinged Rib Metal Mesh	Deployable	7200-8500	X	30	RHCP/LHCP	5200	90	N
Oxford Space Systems	"Hinged Rib" Cassegrain Antenna	Deployable	7200-8500	X	>30	RHCP/LHCP	< 5200	85 (deployed)	N
EnduroSat	X-band Patch Antenna	Patch	8025-8400	X	6	RHCP	2.2	2.4x2.4x0.2	Y

Table 9-3: Antennas									
Manufacturer	Product	Type	Min Frequency	Frequency Band	Gain	Polarization	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
EnduroSat	X-band 2x2 Patch Antenna	Patch	8025-8400	X	12	RHCP	23.2	6.0x6.0x0.3	Y
EnduroSat	X-band 4x4 Patch Antenna	Patch	8025-8400	X	16	RHCP	53	9.8x8.3x0.3	Y
EnduroSat	X-Band 8x12 Patch Antenna	Patch	8025 - 8400	X	26	RHCP	490	25.4 x 17.2 x 0.7	N
ANYWAVES	X-band Payload Telemetry Antenna	Patch	8025-8400	X	11.5	Circular	65	7.3x7.3x11	Y
MMA Design	T-DaHGR	Deployable Reflectarray	8400 to 10000	X	29 – 42.5	Configurable	1300 to 11000	10 x 10 x 10 - 20 x 20 x 20 (stowed) Ø70-Ø200 (deployed)	N
Oxford Space Systems	"Wrapped Rib" Cassegrain Antenna	Deployable	9200 - 10400	X	> 45.6	Dual Linear	<4500	300 (deployed reflector diameter)	N
Oxford Space Systems	"Wrapped Rib" Cassegrain Antenna	Deployable	9200 - 10400	X	> 50	Dual Linear	<6000	500 (deployed reflector diameter)	N
MMA Design	NeuSAR	Deployable Reflectarray	10000	X	>45.5	V+H Linear	16550	52 x 52 x 25 (stowed) Ø300 deployed	Y
EnduroSat	K-band Patch Antenna	Patch	17700-20200	K	18	RHCP	76	6.5x6.5x1.0	N
Oxford Space Systems	"Hinged Rib" Cassegrain Antenna	Deployable	17500-20200 27500-30000	K/Ka	>38/41	Dual Circular (2 channels for Tx and Rx respectively)	< 2500	60 (deployed reflector diameter, scalable to 150)	N

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Table 9-3: Antennas									
Manufacturer	Product	Type	Min Frequency	Frequency Band	Gain	Polarization	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
CesiumAstro	APA-1AT1	APA	24500	Ka	26.5	RHCP or LHCP, selectable at manufacturing	500	18x18x2	Y
CesiumAstro	Vireo Ka 288 Tx DRA	APA	20200	Ka	28	Circular, switchable on orbit	14500	27 x 23 x 16	N
CesiumAstro	Vireo Ka 576 Tx DRA	APA	17000	Ka	31	RHCP or LHCP, selectable at manufacturing, switchable on orbit	26000 to 34000	40x 40x26.5	N
CesiumAstro	Vireo Ka 288 Rx DRA	APA	30000	Ka	28	Circular, switchable on orbit	7000	18x15.5x15.5	N
CesiumAstro	Vireo Ka 576 Rx DRA	APA	27000	Ka	31	RHCP or LHCP, selectable at manufacturing, switchable on orbit	23000 to 29500	40x40x26.5	N
EnduroSat	W-band Patch Antenna	Patch	71000-75000	W	23-29	RHCP	37	8.7x8.1x2.0	N
PICOSATS	BEAMSAT K/Ka band horn antenna	Horn antenna	TX 17300 - 21200 RX 27000 - 31000	K/Ka	TX>19,6 RX>23,75	TX: LHCP RX: RHCP	570	88.0 x 88.8 x 218	Y
PICOSATS	BEAMSAT Ku band horn antenna	Horn antenna	TX 10700 - 12750 RX	Ku	TX>15 RX>16	Linear	633	109.1x109.1x 205.3	N

Table 9-3: Antennas									
Manufacturer	Product	Type	Min Frequency	Frequency Band	Gain	Polarization	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[dBi]	--	[g]	[cm]	---
			12750 - 14800						
PICOSATS	BEAMSAT V band ADE antenna	ADE antenna	59000 - 71000	V	>30	RHCP & LHCP	83	79.5x79.5x43.4	N
PICOSATS	BEAMSAT Ka band 8 x 8 patch antenna	Patch antenna	27000 - 31000	Ka	>20	RHCP	76	78x78x13.5	N
PICOSATS	BEAMSAT K band 8 x 8 patch antenna	Patch antenna	17300 - 21200	K	>20	LHCP	135	94.5x94.5x14	N
PICOSATS	BEAMSAT X band 4 x 4 patch antenna	Patch antenna	8000 - 8500	X	>16	RHCP	125	98x98x10.8	N
PICOSATS	BEAMSAT Q band horn antenna	Horn antenna	37500 - 43500	Q	>20	RHCP & LHCP	150	49x49x110	N
PICOSATS	BEAMSAT QV band horn antenna	Horn antenna	TX 37500 - 43500 RX 47500 - 52500	QV	>30	RHCP & LHCP	250	119x119x65	N

Table 9-4: Radios									
Manufacturer	Product	Type	Min Frequency	Freque ncy Band	Data Rate	Tx Power	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
Aerospace Lab	SSDR	SDR	6	S	1840	29 dBm RMS	575	12.6x11.1x5.3	Y
Vulcan Wireless	NSR-SDR-K/Ka	Radio	20.100-21200, 30000-31000	K	200,000	5W	615	18.3x9.2x3.6	N
Aerospacelab	XSDR	SDR	58	X	206,000	29 dBm RMS	555	12.6x11.1x5.3	Y
Space Micro	MicroSDR-C	SDR	70-3000	VHF, UHF, L, S, C	42,000	0	750	10x10x8	Y
Rincon Research	ASTROSDR	SDR	70-6000	VHF, UHF, L, S, C	--	5 dBm	95	9.0x9.0x1.613	Y
GomSpace	NanoCom SDR	SDR	70-6000	VHF, UHF, L, S, X	--	--	271	9x9x6.6	Y
NI Ettus Research	B205mini	SDR	70-6000	VHF, UHF, L, S, X	--	10 dBm	24	8.3x5.1x8	Y
Alén Space	TOTEM	SDR	70 - 6000	VHF, UHF, L, S	–	7 dBm	130	9.33x8.93x1.36	Y
Alén Space	TREVO	SDR	70 - 6000	VHF, UHF, L, S	–	7 dBm	<714	9.33x8.93x27.88	Y
Vulcan Wireless	NSR-SDR-MUOS	Transceiver	300-320, 360-380	UHF	64	8W	450	10x10x5.6	N
Alén Space	TRISKEL	TTC	395 - 410, 430 - 440	UHF	2.2 - 1.9.2	30 dBm	200	9.33x8.93x126	N
AAC Clyde Space	PULSAR-VUTRX	SDR	--	VHF, UHF	9.6	1.5 W	100	9.6x9x1.6	Y
SkyLabs	NANOcomm-2	Transceiver	130-470	VHF, UHF	Up to 25	31 dBm	110	9.5x9.1x1.2	Y
AstroDev	Helium-100	Transceiver	120-150, 400-450	VHF, UHF	38.4	3 W	78	9.6x9x1.6	Y

Table 9-4: Radios									
Manufacturer	Product	Type	Min Frequency	Frequency Band	Data Rate	Tx Power	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
AstroDev	Lithium-1	Transceiver	130-450	VHF, UHF	9.6	0.25-4 W	48	1.0x3.3x6.5	Y
AstroDev	Beryllium-2	Transceiver	130-450	VHF, UHF	9.6	0.25-4 W	52	1x3.3x6.5	Y
GomSpace	NanoCom AX100	Transceiver	143-150, 430-440	VHF, UHF	0.1-38.4	30 dBm	24.5	6.5x4x7	Y
Spacemanic	Murgas_trx_VHF	Transceiver	144	VHF, UHF	9.6	+30dBm	25	6.7x4.2x0.7	Y
LY3H	SatCOM TP0	FM Repeater	144-146, 430-440	VHF, UHF	--	217 mW	59	--	Y
ISISPACE	TRXVU	Transceiver	145.8-150.05, 400.15-440	VHF, UHF	9.6	27 dBm	75	9x9.5x1.5	Y
CesiumAstro	SDR-1001	SDR	300	UHF, L, S, C		-5 to 6	110	5x8.4x1.35	
CesiumAstro	SDR-2104	SDR	600	L, S, C, X		-5 to 6	900	16x10x2.54	
CesiumAstro	Nightingale RFPU	SDR	300, 24500	UHF, L, S, C, Ka		-5 to 6	3500	20x14.85x13.4	
CeisumAstro	SDR-1001	SDR	300 – 6000	UHF, L, S, C	up to 62,500	-	100	5x8.4x1.3	N
AAC Clyde Space	TRX-U	Transceiver	390-450	UHF	19.2	2	140	8.3x5.7x1.6	Y
NanoAvionics	SatCOM UHF	Transceiver	395-440	VHF, UHF	2.4-38.4	3 W	7.5	5.6x3.3x6.6	Y
Spacemanic	Murgas_trx_UHFlow	Transceiver	399	UHF	9.6	+30dBm	25	6.7x4.2x0.7	N
EnduroSat	UHF Transceiver Type II	Transceiver	400-403, 430-440	UHF	≤ 9.6	30 - 33 dBm	94	9.6x9.6x1.1	Y
Spacemanic	Murgas_trx_UHF	Transceiver	420 MHz	UHF	9.6	+30dBm	25	6.7x4.2x0.7	Y
L3 Communications Inc. /SDL	Cadet	SDR	450	VHF, UHF	3,000	--	200	6.9x7.4x1.34	Y

Table 9-4: Radios									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Freque ncy Band**	**Data Rate**	**Tx Power**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
CesiumAstro	Vireo RPU	SDR	600	L, S, C, X		-5 to 6	5000	25 x 15 x 20.3	N
sci_Zone, Inc.	LinkStar-STX3	Transmitter	1610-1625	L	0.009	--	48	8.6x5.3x2.9	Y
Qualcomm	GSP-1720	Transmitter	1610-1626.5, 2483.5-2500	L, S	9.6	31 dBm	60	11.9x6.5x1.5	Y
NAL Research Corporation	NAL Iridium 9602-LP	Iridium Satellite Tracker	1616-1626.5	L	--	1 W	136	6.9x5.5x2.4	Y
NearSpace Launch	EyeStar-S4 Iridium	Transceiver	1618.75	L	0.1	31.96 dBm	29	6x3x2.1	Y
L3Harris	CXS-1000	Transponder	1700-2100	L, S	20,000	1-5 W	1360	10x10x11	Y
Tethers Unlimited	SWIFT-SLX	SDR	1700-2500	S	6,000	33 dBm	300	9x9.8x3.6	Y
Tethers Unlimited	SWIFT-XTS S Transceiver X Transmitter	SDR	1700-2500, 7000-8500	S, X	6,000-25,000	34 dBm	800	9x9.8x6	Y
AAC Clyde Space	TX-2400	Transmitter	2000-2300	S	6,000	2.5	70	6.8x3.5x1.5	Y
Syrlinks	EWC27 + OPT27-SRX S/X Transceiver	Transceiver	2025-2110	S	100,000	27-33 dBm	400	9x9.6x3.9	Y
Vulcan Wireless	NSR-SDR-S/S	Transceiver/T ransponder	2025-2115, 2200-2300	S	4,000	4W	375	9.2x9.5x3.4	Y
Vulcan Wireless	NSR-SDR-X/S	Transceiver/T ransponder	2025-2115, 8000-8500	S, X	10,000	5W	375	9.2x9.5x3.4	N
Vulcan Wireless	NSR-SDR-X/S HP	Transceiver/T ransponder	2025-2115, 8000-8500	S, X	10,000	10 W	615	18.3x9.2x3.6	N

Table 9-4: Radios									
Manufacturer	Product	Type	Min Frequency	Frequency Band	Data Rate	Tx Power	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
CesiumAstro	Vireo RPU	SDR	17000 27000	Ka		-5 to 6	7400	25x22.6x20.3	N
IQ Technologies for Earth and Space GmbH	HISPICO	Transmitter	2100-2500	S	1,000	27 dBm	100	9.5x4.6x1.5	Y
Innoflight, Inc.	SCR-104	SDR with AES-256 Encryption	Tx: 2200-2290 Rx: 1760-1840 Rx: 2025-2110	Tx: S Rx: L/S	Tx: >4,500 Rx: 1000	--	250 - 290	8.2x8.2x2.5 9.8x8.2x3.3	Y
CUBECOM	STXG2	Transmitter	2200-2290 2400-2480	S	25,000	--	137	9.6x9.0x13	N
Emhiser Research, Inc.	ETT-01EBA102-00	Transmitter	2200-2400	S	--	1 W	57	3x8.6x0.8	Y
Quasonix	NanoTX	Transmitter	2200.5-2394.5	S	50	1-10 W	Request	3.3x8.6x0.8	Y
IQ Technologies for Earth and Space GmbH	SLINK-PHY	Transceiver	2200-2290, 2025-2110	S	64-4000	30 dBm	275	6.5x6.5x13.7	Y
ISISPACE	TXS	Transceiver	2200-2290	S	4.3	27-33 dBm	132	9.8x9.3x1.4	Y
Syrlinks	S-band Transponder	Transponder	2200–2290	S	8-2000	27-33 dBm	--	--	Y
EnduroSat	S-band Transmitter	Transmitter	2200-2290, 2400-2450	S	≤ 20000	27 - 33 dBm	250	9.6 x 9.6 x 1.5	Y
General Dynamics	S-Band TDRSS/DSN	Transponder	Tx: 2200-2300 Rx: 2025-2220	S	12,000	0.03 W	4900	19x23x15	Y
Microhard	Nano N2420	Modem	2400-2483.5	S	230	0.1-1 W	210	5x3x0.6	Y
Laboratory for Atmospheric and Space Physics (LASP)/Blue Canyon	X-band Radio	SDR	Tx: 2200-2500, 8000-8500, 21000-33000 Rx: 1760-1840, 2000-2110, 21000-23000	Downlink: S, X, Ka Uplink: L, S, Ka	100000	30 dBm	--	4.5x4.35x1.25	Y

Table 9-4: Radios									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Frequency Band**	**Data Rate**	**Tx Power**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
Technologies (BCT)									
SkyLabs	NANOlink-base Gen2	SDR	2200-2300	S	Up to 4000	30 dBm	110	9.5x9.1x1.2	Y
SkyLabs	NANOlink-boost Gen2	SDR	2200-2300	S	Up to 4000	37 dBm	250	9.5x9.1x2.2	Y
SkyLabs	NANOlink-boost-dp Gen2	SDR	2200-2300	S	Up to 4000	31.5 dBm	385	9.5x9.1x3.2	Y
Tethers Unlimited	SWIFT-XTX X Transmitter	SDR	7000-8500	X	25,000	33 dBm	300	9x9.8x6	N
General Dynamics	X-Band Small Deep Space	Transponder	7145-7230, 8400-8500	X	100,000	0.06	3200	18x17x11	Y
Space Dynamics Laboratory	IRIS V2.2	SDR Transponder	RX: 7145–7235 GHz TX: 8400–8500 GHz	X, Ka, S, or UHF	0.1 - 10000	36	875	10.1x10.1x5.6	Y
Innoflight, Inc.	SCR-106	SDR with AES-256 Encryption	Tx: 7800-8500 Rx: 1760-1840 Rx: 2025-2110	Tx: X Rx: L/S	Tx: 20,000 Rx: 1,000	0.02-2.5 W	250 - 290	8.2x8.2x2.5 9.8x8.2x2.8	Y
Innoflight, Inc.	SCR-108	SDR with AES-256 Encryption	Tx: 19200-21200 Rx: 29000-31000	Tx: Ka Rx: Ka	Tx: 100,000 Rx: 20,000	0.02-3 W	404	9.8x8.7x3.9	Y
EnduroSat	X-band Transmitter	Transmitter	7900-8400	X	≤ 150000	27-33 dBm	270	9.6x9.6x2.6	Y
EnduroSat	S-band Transceiver II	Transceiver	2200-2290 (downlink)	S	< 125	27-33 dBm	195	9.6x9.0x1.9	Y
EnduroSat	S-band Transceiver ISL	Transceiver	2025 - 2110 (TM & TC)	S	< 50	27-33 dBm	205	9.6 x 9.6 x 1.9	N

Table 9-4: Radios									
Manufacturer	Product	Type	Min Frequency	Freque ncy Band	Data Rate	Tx Power	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
EnduroSat	S-Band Transceiver III	Transceiver	2200 - 2290 (TM), 2025 - 2110 (TC)	S	≤ 3000	27 - 33 dBm	250	9.6x9.6x2.5	N
CUBECOM	XTXG2	Transmitter	8025-8400	X	250,000	--	137	9.6x9.0x1.3	N
CUBECOM	µHDTRX-X	Transmitter	8025-8400	X	1,500,000	--	250	9.6x9.0x2.0	Y
CUBECOM	HDRTX	Transmitter	8025-8400	X	1,000,000	--	250	9.6x9.0x2.0	Y
IQ Technologies for Earth and Space GmbH	XLINK	Transceiver	8025-8500, 7145-7250	X	64-25,000	30 dBm	--	<1 U	Y
Syrlinks	EWC27	Transmitter	8025-8500	X	140,000	27-33 dBm	235	9x9.6x2.6	Y
Syrlinks	EWC27 + OPT27-SRX	Transceiver	Rx: 2025-2110 Tx: 8025-8500	S, X	RX: 256 TX:100000	33 dBm	320	9.6x9x3.9	Y
Syrlinks	EWC31	Transceiver	Rx: 2025-2110 Tx: 2200-2290	S	RX: 256 TX: 2,000	33 dBm	405	9.5x9.5x5.3	Y
Syrlinks	EWC31-NG	Transceiver	Rx: 2025-2110 Tx : 2200-2290	S	RX: 512 TX: 2,000	33 dBm	360	9.5x9x3.2	N
Syrlinks	N-XONOS	Transmitter	Rx: 2025-2110 Tx: 8025-8400	S, X	RX: 256 TX: 400,000	33 dBm	385	9.5x9x3.1	Y
Syrlinks	EWC15-NG	Transceiver	Rx: 2025-2110 Tx: 2200-2290	S	RX: 512 TX: 2,000	33 dBm	1280	17.2x12x6.7	N

Table 9-4: Radios									
Manufacturer	Product	Type	Min Frequency	Freque ncy Band	Data Rate	Tx Power	Mass	Dimensions	Flight Heritage
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
Syrlinks	XONOS	Transmitter	Rx: 2025-2110 Tx: 8025-8500	X/S	RX: 256 TX: 628,000	40 dBm	2400	20,6x15,2x6,9	N
Argotec	UST-Lite	Transponder	2025 – 2120 2200 – 2300 7145 – 7235 8400 – 8500 22550 – 23550 25500 – 27500	S, X, K, Ka	Up to 100,000 (higher possible with NRE)	8 dBm (input to SSPA)	<4000 (quad-band)	19.9 x 14.1 x 12.1 (quad-band)	N
Tethers Unlimited	SWIFT-KTX Ka Transmitter	SDR	20200-21200 24000-27000	Ka	25,000	33 dBm	300	9x9.8x4	N
Tethers Unlimited	SWIFT-KTRX Ka Transmitter	SDR	24000-27000	Ka	---	35 dBm	1,000	16x9.6x6	N
SpaceMicro	microKaTx-300	Transmitter	25250-27250	K	1,000,000	2	1000	10x10x8	Y
AAC Clyde Space	PULSAR-DATA XTX X-Band Transmitter	SDR	--	X	50,000	2 W	130	9.6x9x1.1	Y
EnduroSat	X-band Transmitter II	Transmitter	8000 - 8400	X	≤ 350,000	27 - 33 dBm	< 2500	9.2 x10x8.5	N
EnduroSat	K-band Transmitter	Transmitter	25500 - 27000	K	≤ 1,000,000	27 - 33 dBm	< 3000	9.2x10x10	N
AAC Clyde Space	PULSAR-DATA STX S-Band Transmitter	SDR	--	S	7,500	1 W	100	9.6x9x1.7	Y
Honeywell	STC-MS03	Transceiver	--	S	6,250	3.16 W	1000	16x11x4.4	Y
Innoflight, Inc.	SCR-106HDR	SDR with AES-256 Encryption	Tx: 7800-8500 Rx: 1760-1840 Rx: 2025-2110	Tx: X Rx: L/S	Tx: 100,000 Rx: 20,000	-	250 - 290	8.2 x 8.2 x 2.5 9.8 x 8.2 x 2.8	Y
EnduroSat	Versatile Wideband SDR (VW-SDR)	Transceiver	75 - 6000	VHF, UHF, L, S, C	982,000	-10 dBm	<1500	9.8x9.8x7.5	N

Table 9-4: Radios									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Freque ncy Band**	**Data Rate**	**Tx Power**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
Trident	RDRT	SDR – RF System on Chip (RFSoC)	100	L/S	8-channels 250MHz – 6554MHz TX, 8-channels 250MHz – 4096MHz RX	-1 dBm Full-scale	571	10 x 14.6 x 2.54	N
Trident	ADCM	SDR – MPSoC Basecard and -SP converters - RX only	100	L/S/C	6.4GSPS single channel, 3.2 GSPS dual-channel	2.8dBm Full scale input	690	10 x 14.6 x 2.54	N
Space Dynamics Laboratory	Iris Radio V3	SDR Transponder	Various	Simulta neous Multiban d: X, Ka, and S	0.1 - 10000	34	720	10.1x10.1x3. 8	N
PICOSATS	RADIOSAT Ka band transponder	Transponder	TX 17300 - 21200 RX 27000 - 31000	K/Ka	-	5	690	93.5 x 114.15 x 45.6	Y
PICOSATS	RADIOSAT Ku band transponder	Transponder	TX 10700 - 12750 RX 12750 -13250 RX 13750 - 14800	Ku	-	5	< 1000	93.48 x 114.15 x 62.11	N
PICOSATS	RADIOSAT Ka band transceiver	Transceiver	TX 17300 - 21200 RX 27000 - 31000	K/Ka	TX 4000000 RX 1000000	5	900	83.50 x 114.15 x 53	N
PICOSATS	BEAMSAT Ku band transceiver	Transceiver	TX 10700 - 12750 RX 12750 -13250	Ku	TX 4000000 RX 1000000	5	1000	93.48 x 114.15 x 70.6	N

Table 9-4: Radios									
Manufacturer	**Product**	**Type**	**Min Frequency**	**Freque ncy Band**	**Data Rate**	**Tx Power**	**Mass**	**Dimensions**	**Flight Heritage**
---	---	---	[MHz]	--	[kbps]	--	[g]	[cm]	---
			RX 13750 - 14800						
PICOSATS	RADIOSAT SDR	Modem/Tran sceiver (SDR)	300 - 7000	UHF, L, S, C	TX 4000000 RX 1000000	0,001	< 500	83.50 x 96 x 25	N
PICOSATS	BEAMSAT Ka band transmitter	Transmitter	25500 - 27000	Ka	4000000	5	900	83.50 x 114.15 x 53	N
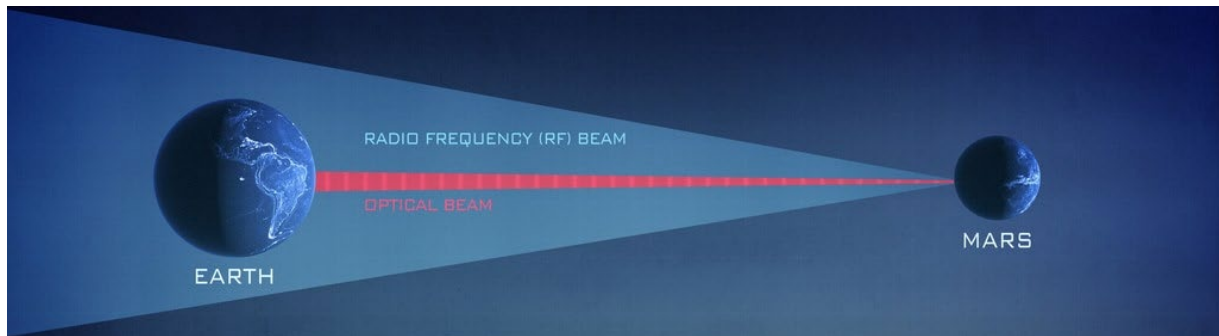
Figure 9.7: Laser vs RF link and data downlink. Credit: NASA

9.3   Free Space Optical Communications

Free space optical communications, or lasercom, uses optical wavelengths of electromagnetic radiation to transmit messages wirelessly between user terminals. While few small satellite optical communications terminals have flown, availability is improving, and optical communication is becoming a more common wireless communication technology for small satellites.

Due to the higher frequencies of electromagnetic energy used in lasercom, the amount of bandwidth available for communicating is much larger compared to RF. This increase in bandwidth over RF enables much higher data rates. The beam width of a lasercom link is also typically much narrower than a RF link (Figure 9.7). The amount that a transmitted beam spreads as a function of its propagation distance is called its divergence. The divergence of a beam is proportional to the wavelength of the electromagnetic wave transmitted divided by the transmitted beam diameter. The high frequencies used in lasercom mean the wavelength of the transmitted energy is orders of magnitude smaller than RF systems. These small wavelengths mean the transmitter diameters and beam divergence of lasercom systems can also be much smaller, which enables the size, weight, and power (SWaP) of lasercom systems to be lower than similar performing RF systems. Laser communications have a low probability of intercept, are difficult to jam, and encounter very little interference because of the narrow beamwidth. At present, optical frequencies are unregulated, unlike RF systems which require a licensing process to be able to communicate with a spacecraft. Lasercom is not without its disadvantages, which include the required pointing of the beam and the impact weather has on the signal. The small beam divergence of lasercom transmit beams means that the acceptable pointing error of the narrow beam is much smaller than that of typical RF systems. The frequencies used in lasercom systems are also susceptible to large amounts of attenuation due to moisture in clouds. This attenuation prohibits communication while there is cloud cover and incentivizes operators to build their optical ground stations in areas that have infrequent cloud cover. While larger missions such as the Geosynchronous Lightweight Technology Experiment (GeoLITE), Near Field Infrared Experiment (NFIRE), and Lunar Laser Communication Demonstration (LLCD) demonstrated laser communications downlinks and crosslinks nearly two decades ago, small satellites and CubeSats have also now successfully demonstrated laser communication downlinks from space. For example, the Aerospace Corporation, in cooperation with NASA ARC, launched three
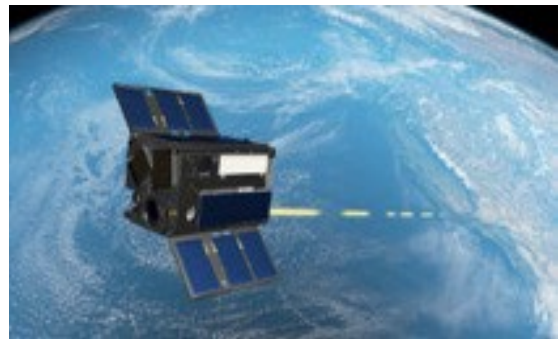
Figure 9.8: An artist rendering of laser communications for the OCSD. Credit: NASA.

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CubeSats in its AeroCube Optical Communication and Sensor Demonstration (OCSD) series (Figure 9.8). OCSD-B & C demonstrated a 200 Mbps downlink from a 1.5U CubeSat to a 40 cm ground station (16). The Aerospace Corporation transmitter has also successfully flown on follow-on missions that were able to use lasercom systems to downlink science data (17).

9.3.1 System Architecture

An optical modem, optical amplifier, and optical head typically comprise a lasercom terminal (LCT) (see figure 9.9 for an example system diagram of an optical head). As with radio terminals, component locations in optical terminals can vary; for example, the modulator may not be located proximal to the optical front end. Also, the pointing mechanism might differ from the one shown in Figure 9.9.

The key parameters of an optical communication system are frequency, modulation, aperture size, and range. Successful optical communications links require high pointing accuracy. The optical communication terminal on a spacecraft typically has a two-stage pointing system, with a coarse-pointing stage and a fine-pointing stage. The optical communication system often relies heavily on the

Figure 9.9: Optical head architecture diagram. Credit: Wang et al. 2023).

spacecraft attitude determination and control system (ADCS) for coarse-pointing, and may use a second pointing mechanism such as a gimbal as additional support for coarse pointing. Fine pointing is often implemented with additional mirrors in the payload. However, pointing that is solely dependent on spacecraft attitude control has also been demonstrated. On transmit, energy passing through the optical aperture forms a very narrow beam. The larger the aperture, the narrower the beam; this creates higher power density at a receiver for a given range , but comes at the expense of more demanding pointing requirements of the transmit beam.

In order for two communication terminals to locate each other, they may shine higher power and broader-beam "beacon" lasers to find each other before engaging the narrower and higher data rate link. Other strategies are possible, e.g., scanning the direction of the data beam until acquisition is achieved. The beacon itself may also be modulated. Optical modems may be software defined and can support multiple modulation and coding schemes, similar to RF.

9.3.2 Optical Ground Stations

The ground stations for optical communications understandably differ significantly from RF ground stations due to the need to have the receiving aperture (typically a mirrored telescope) maintain an optical-quality surface to focus the collected optical energy onto a receiver. Optical ground stations are often located at or near astronomical telescope sites, as they are located in favorable "seeing" environments, i.e., locations with low chance of cloud cover and calm, non-turbulent air. Optical ground stations are typically mounted inside protected domes or other structures to cover them during bad weather. These structures typically need to be opened for clear access to the sky. Since optical ground stations often have beacons, it is important to consider laser safety and proximity to airports. Typical ground-to-space beacons are tens of watts of optical power for low-Earth orbit missions. Most optical ground stations are experimental facilities used for campaigns with specific research missions, although there have been recent developments in commercial optical ground stations. For a more detailed outline of existing optical ground stations, refer to the *Ground Data Systems and Mission Operations* chapter.
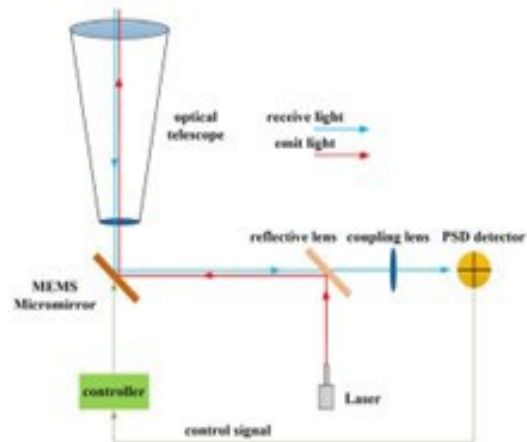
9.3.3   Design Considerations

Lasercom terminals can offer a smaller footprint and lower power draw compared to an RF terminal. However, lasercom pointing requirements are significantly more demanding. One of the largest challenges to widespread implementation is the required pointing for the LCT. To manage the pointing challenges, there are different types of system implementation and components that can be used, depending on resource constraints, performance requirements, and mission lifetime. The LCTs that have been designed, built, and operated on small satellite and CubeSat platforms have some significant differences from LCTs designed for larger spacecraft. Given the size, weight, and power constraints for SmallSat LCTs mechanical gimbals are typically not used. SmallSat lasercom systems may rely entirely on the body pointing of the satellite to point the LCT at the ground station, or they may use an internal fine pointing mechanism to achieve the required fine-pointing performance.

On SmallSat platforms, the limited volume and tight packaging is often a major challenge in the design of low-SWaP LCTs. There are thermal management challenges during operation, as it is difficult to radiate enough heat with limited surface area for radiators. There are also power constraints, due to limited surface area for solar arrays and secondary battery systems. In addition, not all SmallSat platforms can achieve the pointing requirements necessary for laser communications. Typically, both precise three-axis reaction wheels for actuation and attitude determination sensing from at least one star tracker are necessary.

While RF bands with high frequency and bandwidth are also affected by clouds and rain, cloud cover can prove difficult or even insurmountable for optical communications systems due to the high levels of attenuation. If the cloud coverage is too great at a specific ground station, the transmission may be held for a later time or passed off to a different ground station. With advances in intersatellite networking and the development of extensive networks of optical communication ground stations, routing data around weather may become more feasible.

The atmosphere is also a source of aberrations for optical communication systems, particularly for terminals with larger apertures and more complex receivers to support higher data rates. For example, some high-rate optical downlink terminals that require coupling the received light into fiber receivers must use adaptive optics to correct atmospheric effects on the incident wavefront. Without correction of the wavefront, there would be a lack of received signal power to couple into the receive optical fiber due to the perturbed wavefront of the received light. Adaptive optics systems take a sample of the incident wavefront before it reaches the receiver and measure the aberrations of the wavefront to feed into the control of the adaptive optics system that acts on the received light and counters the aberrations.

Lasercom crosslinks can provide a high bandwidth connection between two satellites, as well as perform ranging between the satellites, potentially with high ranging precision. Connecting two satellites across different orbit planes can help with data routing and can reduce how long it takes to route data to the end user. Lasercom crosslink systems are now in use for both commercial and government missions. Lasercom crosslink demonstrations have been performed from GEO-LEO, LEO-GEO, LEO-LEO, and are operational as part of the European Data Relay Service (43)(44), but these LCTs were developed for much larger spacecraft (20)(21). Crosslinks also have the challenge of both terminals being resource-constrained onboard a spacecraft. Space-to-ground links have an advantage in that the ground station apertures can be large with essentially unconstrained resources relative to what is available on a satellite. The challenges facing inter-satellite optical communications also centers on pointing, acquisition, and tracking (PAT) requirements. Satellites in different orbital planes can have high relative velocities and performing pointing, acquisition, and tracking of the terminal can be a challenge. An advanced

opto-mechanical system may be needed to surmount this challenge, and special consideration may be needed in the design of the receive optics to manage high Doppler shift.

9.3.4   Policies and Licensing

Given the early stages of development for satellite optical communication systems, both policy and regulatory approaches are still evolving. In the policy realm, there is an initial draft CCSDS Pink Book in process (CCSDS 141.0-P-1.1) with a goal to facilitate interoperability and cross-support between different communication systems. There is also an optical communication working group with NASA and ESA participation. The U.S. government Space Development Agency has also recently released their Optical Communications Terminal (OCT) Standard Version 4.0.0 (28th June 2024), which is a widely used design reference for space lasercom terminals, although not yet targeted specifically to small satellite terminals.

Regarding licensing and regulation, the situation is very different from the radio frequency domain. Currently there are no licensing requirements for laser communications. In the radio frequency spectrum, the main goal for licensing is to prevent interference between transmitters.

Lasercom interference is not currently coordinated by a regulatory body (like the ITU or NTIA in RF) although there have been historical ITU World Radiocommunication Conference resolutions that have considered implementing regulation. There is currently no regulation for two major reasons:

1) Laser communications is highly directional, which makes interference unlikely, due to the narrow divergence of the transmitting beam and corresponding small beam footprint at the receiver.
2) The small number of organizations that deployed laser communication systems doesn't warrant a complex coordination body like the ITU.

However, in the US there are three regulatory entities that are concerned with aspects of outdoor laser operations: The FAA, DoD Laser Clearing House (for DoD missions) and the NASA Laser Safety Review Board (for NASA missions).

FAA coordination is required if potentially harmful laser irradiance is transmitted through navigable airspace. This includes prevention of injury as well as potential distraction of pilots by visible lasers. The FAA will most likely only be concerned about transmitters at ground stations because transmitters on spacecraft are hundreds of miles above the highest-flying aircraft and beam dispersion is large enough that there are usually no safety implications. Missions should coordinate with their local FAA service center to get approval, documented with a "letter of non-objection."

The DoD Laser Clearinghouse (LCH) works to ensure that DoD and DoD-sponsored outdoor laser use does not impact orbiting spacecraft or their sensors. That includes both US DoD and foreign assets. LCH and mission operators might enter close cooperation where LCH permits specific laser engagements. The process of coordinating with LCH to get to that point can take many months and should be started as early as possible. However, currently LCH will only engage DoD and DOD-sponsored missions.

NASA's Laser Safety Review Board (LSRB) is focused on personnel safety for all outdoor laser operations. NASA missions prepare safety documentation and submit to LSRB for review before launch. LSRB will also verify FAA concurrence. Further information on regulations can be found in ANSI Z136.6 "American National Standard for Safe Use of Lasers Outdoors" and in (45).

9.3.5 Mission Examples

Missions demonstrating lasercom terminals on small satellite and CubeSat platforms have shown viable pathways for overcoming the challenges associated with lasercom to enable high bandwidth communications. Please refer to Table 9-5 for more information on lasercom missions.

The Small Optical Transponder (SOTA) was developed by the National Institute of Information and Communications Technology (NICT) in Japan and launched in 2014. This LCT is capable of up to 10 Mbps and has successfully demonstrated a laser space-ground link from a 50 kg microsatellite (22). The Very Small Optical Transponder (VSOTA) LCT, also developed by NICT, is capable of 1 Mbps. VSOTA was integrated into the Rapid International Scientific Experiment Satellite (RISESAT) from Tohoku University and launched in 2019 (23).

The German Aerospace Center (DLR) has been developing LCTs as part of its Optical Space Infrared Downlink System (OSIRIS) program to support lasercom from small satellites. The first, OSIRISv1, is capable of 200 Mbps downlinks and is integrated into the University of Stuttgart's Flying Laptop satellite. This LCT uses a body pointing-only approach. The OSIRISv1 LCT, launched in 2017, has completed commissioning and is being used by DLR to test their optical ground stations. The OSIRISv2 LCT, launched in 2016, is capable of 1 Gbps and is integrated into the BiROS satellite from DLR Berlin. This LCT uses closed-loop body pointing with a beacon reference. The OSIRISv2 LCT has been undergoing commissioning with parts of the terminal having been commissioned (24)(25)(26).

The Aerospace Corporation completed the first demonstration of optical communication from a CubeSat platform with the NASA-sponsored Optical Communication and Sensor Demonstration (OCSD) mission. These terminals were integrated into a 1.5U CubeSat and rely only on body pointing. The use of body pointing-only to achieve the necessary pointing required is due to the beam divergence being tuned, the pointing performance capability of their spacecraft, and a high optical power amplifier to ensure sufficient optical power is received at the optical ground station. The terminals achieved a 200 Mbps downlink data rate to a 40 cm ground station and do not use a beacon for a pointing reference (16). This transmitter has been flown since on multiple missions such as R3 (17) and the Rogue Alpha and Beta CubeSats (18). An updated version of this terminal has also been flown on the Slingshot mission, which has achieved 250 Mbps downlinks (19).

As part of NASA's CLICK mission, MIT developed the 1.2U CLICK-A terminal. The first phase of the mission flew the CLICK-A downlink terminal on a 3U CubeSat to demonstrate an optical design that uses a secondary fine pointing micro-electromechanical systems (MEMS) fine-steering mirror (FSM) to achieve the necessary pointing requirements for optical communication without imposing those requirements on the spacecraft pointing or needing large gimbals. This LCT uses closed-loop fine pointing with a beacon reference and is designed to close its link with a 28 cm ground station. The terminal was integrated into a Blue Canyon Technology's XB1 spacecraft bus and was launched to and deployed from the ISS in 2022. The mission was able to successfully demonstrate improved pointing with the MEMS FSM compared to body-only pointing. CLICK-A ultimately served as a risk-reduction phase for the CLICK-B/C phases of the mission described later in this section (27).

DLR has also been developing their OSIRIS4 CubeSat transmitter. This optical communication terminal is designed to demonstrate an optical downlink in a 0.3U package. This transmitter also uses a MEMs FSM fine pointing mirror and was launched on the PIXL-1 mission in 2021. A beacon is used for fine pointing reference with this terminal. This terminal is designed to be used with a 60 cm optical receiver and has been commercialized through TESAT with the product name CubeLCT (28).

Sony Computer Science Lab and the Japanese Aerospace Exploration Agency (JAXA) jointly developed a LCT called Small Optical Link for ISS (SOLISS). This LCT is capable of bidirectional 100 Mbps links and was launched to and mounted on the ISS in 2019. This LCT has been successfully demonstrated with NICT's ground station (29)(30).

MIT Lincoln Laboratory developed the TBIRD terminal, which achieved 200 Gbps downlinks. The transmitter uses commercial fiber telecommunication transceivers and amplifiers to support very high data rate downlinks. This project used NASA JPL's Optical Communication Telescope Laboratory (OCTL), which hosts a 1 m telescope with the adaptive optics necessary to couple the received light back into a fiber transceiver for demodulation. This terminal development was sponsored by NASA and was launched on the PDT-3 6U CubeSat mission in June of 2022 (33). It has thus far achieved the transmission of 4.8 Tb in a single pass and has demonstrated the value of implementing an automatic repeat request protocol in the use of lasercom downlinks (31)(32).

The CubeCat LCT is a commercial product by AAC Clyde Space that offers a bidirectional space-to-ground communication link between a CubeSat and an optical ground station. This LCT offers downlink speeds of up to 1 Gbps and an uplink data rate of 200 Kbps (34). It was recently launched for a demonstration mission called NorSat-TD and has successfully communicated with the ground (34).

Future mission launches include demonstrations of optical crosslinks on CubeSats. The CLICK-B/C phase of the CLICK mission is developing a 1.5U crosslink LCT. The CLICK-B/C crosslink LCT is designed to establish a 20 Mbps link at separations from 25 to 580 km. CLICK-B & C will each be integrated into its own 3U Blue Canyon Technologies XB1 spacecraft. The LCTs are designed to be capable of precision ranging up to a precision of 50 cm relative to each other. The spacecraft will be launched to and deployed from the ISS in 2025 and fly in the same orbital plane (27).

DLR is also developing an optical crosslink terminal called CubeISL. This terminal is a modification of their OSIRIS4CubeSat terminal, with the addition of a receive optical path and fiber amplifier. This terminal is designed for 100 Mbps crosslinks out to 1000 km and 1 Gbps downlinks. The design goals for this terminal are a mass less than 1 kg, a volume of 1U, and an operating power of 30 W (35). A demonstration mission of these terminals is planned with two 6U CubeSats used to host the terminals in 2025 (36).

Table 9-5: LCT Technologies									
Vendor/Developer	Terminal	Platform	Data Rate	Mass	Power	Wavelength	Modulation	Launch Date	Reference
---	---	---	[Mbps]	[kg]	[W]	[nm]	---	---	---
NICT	SOTA	SOCRATES	10	5.9	16	976/800/1549	OOK	5.2014	21
DLR	OSIRISv2	BiROS	1000	1.65	37	1550	OOK	6.2016	24
DLR	OSIRISv1	Flying Laptop	200	1.3	26	1550	OOK	7.2017	23, 24, 25
Aerospace Corporation	OCSD-B&C	AeroCube-7	200	<2.3	20	1064	OOK	12.2017	16
NICT	VSOTA	RISESAT	1	<1	4.33	980/1550	OOK/PPM	1.2019	22
Sony/JAXA	SOLISS	ISS	100	9.8	36	1550	OOK	7.2019	28, 29
DLR	OSIRIS4CubeSat	PIXL-1	100	0.4	10	1550	OOK	1.2021	27
MIT Lincoln Labs	TBIRD	PDT-3	200,000	<3	100	1550	QPSK	5.2022	30
MIT	CLICK-A	CLICK	10	1.2	15	1550	PPM	7.2022	26
AAC Clyde Space	CubeCat	NorSat-TD	1000	<1.33	15	1550	OOK	4.2023	31
MIT	CLICK-B/C	CLICK	20	1.5	30	1537/1563	PPM	Est. 2025	26
DLR	CubeISL	CubeISL	100	1	30	1537/1553	OOK	Est. 2025	LC5, LC6

9.4 Future Technologies

While free space optical communication technology development has been making strides towards fielding operational systems, other avenues of research have also been explored. Quantum key distribution is a protocol that shares a secret cryptographic key through entangled photons. Sources and optical front ends have been in development for transmitting these keys from small satellite spaceborne platforms (38)(39). Another approach to expanding the communication windows for small satellites in low-Earth orbit is to form an intersatellite link to geosynchronous orbit. Major programs, such as the previously mentioned European Data Relay System, use this type of link. NICT is looking to establish this type of link with a CubeSat through the CubeSOTA program (42). In addition to CubeSat terminals, larger terminals for larger SmallSats have been developed by Tesat, Mynaric (27), SpaceMicro (28), and SA Photonics. Deployment of these terminals has been driven by the development of the Space Development Agency's Proliferated Warfighter Space Architecture constellation and the use of LCT in their architecture. Beyond this, DARPA has funded the Space-BACN program that seeks to develop a reconfigurable and multi-protocol inter-satellite LCT that can be supported on small satellites.

Several projects funded via NASA's Small Spacecraft Technology (SST) program through the University Smallsat Technology Partnerships (USTP) initiative have began advancing RF and optical communication systems. The Deployable Optical Receiver Aperture (DORA) project, in partnership with Arizona State University and JPL, successfully launched from the ISS in October 2024 to demonstrate a novel approach to deploying large apertures (40)(41)⌷. Listed below in Table 9-6 are USTP projects focused on SmallSat communications technology advancement. Further information can be found at the USTP website:

https://www.nasa.gov/smallspacecraft/university-smallsat-technology-partnership-initiative/

Table 9-6: STP Initiative Communication Projects			
Project	**University**	**Current Status**	**Reference**
*FIGARO, 5G arrays for lunar relay operations	San Diego State	Still in development	USTP Technology Expo presentation
*A Small Satellite Lunar Communications and Navigation System	University of Colorado, Boulder	Still in development	USTP Technology Expo presentation
Deployable Optical Receiver Aperture DORA	Arizona State University	Launched October 2024	

*Presentation is from the USTP Technology Exposition held June 2022.

9.5   Summary

There is already strong flight heritage for many UHF/VHF and S-band communication systems for CubeSats. Less common, but with growing flight heritage, are X-band systems. Higher RF frequencies already have CubeSat flight heritage, but with limited (or yet to be demonstrated) performance. Although there are limited Ka-band systems for CubeSats today, high-rate transmitters such as the Astro Digital AS-10075 demonstrated 320 Mbps in the Landmapper-BC 3 v2 mission. While laser communication has been demonstrated on multiple CubeSat platforms, it is still not yet considered to be established technology for SmallSats. More demonstrations are in development, with some already launched and operating, to show higher data rates and increased pointing performance. Since optical communications uplink and downlink can be

blocked by clouds, RF is considered complementary to maintain contact under all conditions. There is growing interest among the NASA science community in using constellations of CubeSats to enhance observations for Earth and space science.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email for further contact.

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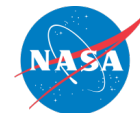
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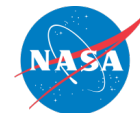
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Chapter Contents

Chapter Glossary

(CBOD)	Clamp Band Opening Device
(CDS)	CubeSat Design Specification
(CSLI)	CubeSat Launch Initiative
(CSOs)	Client Space Objects
(DPAF)	Dual Payload Attach Fittings
(EAGLE)	ESPA Augmented Geostationary Laboratory Experiment
(EELV)	Evolved Expendable Launch Vehicle
(ENRCSD)	Nanoracks External CubeSat Deployer
(ESA)	European Space Agency
(ESPA)	EELV Secondary Payload Adapter
(GEO)	Geostationary Equatorial Orbit
(HEO)	Highly Elliptical Orbit
(ISS)	International Space Station
(J-SSOD)	JEM Small Satellite Orbital Deployer
(JAXA)	Japan Aerospace Exploration Agency
(JEM)	Japanese Experimental Module
(JEMRMS)	Japanese Experimental Module Remote Manipulator System
(LEO)	Low-Earth Orbit
(LTAN)	Local Time at Ascending Node
(M-OMV)	Minotaur Orbital Maneuvering Vehicle
(MEO)	Medium Earth Orbit
(MET)	Microwave Electrothermal Thrusters
(MLB)	Motorized Light Bands
(MPAF)	Multi Payload Attach Fittings
(MPEP)	Multi-Purpose Experiment Platform
(NICL)	Nanoracks Interchangeable CubeSat Launcher
(NOAA)	National Oceanic and Atmospheric Administration
(NRCSD)	Nanoracks ISS CubeSat Deployer
(OMV)	Orbital Maneuvering Vehicle
(OTVs)	Orbital Transport Vehicles
(PCBM)	Cygnus Passive Common Berthing Mechanism
(RUG)	Rideshare User Guide
(SL-OMV)	Small Launch Orbital Maneuvering Vehicle
(SSMS)	Small Spacecraft Mission Service
(SSOD)	Small Satellite Orbital Deployer
(TRL)	Technology Readiness Level

10.0 Integration, Launch, Deployment, and Orbital Transport

10.1 Introduction

Of the 2,938 total spacecraft launched in 2023, 68% had a mass less than 600 kg and 27% were under 200 kg mass. This represents a 17% increase in total spacecraft launches since 2022 and is partially due to the emergence of next-generation SmallSat constellations. Small spacecraft weighing 400−600+ kg, including Iridium Next and Starlink v2-mini, have gained popularity in the aerospace community for their overall price reduction and increased capabilities. While fewer traditional small spacecraft (mass below 200 kg) have been launched, they have consisted of roughly 40% of the total launched spacecraft in the past ten years (1). The recent expansion in SmallSat and CubeSat constellations by operators like SpaceX, OneWeb, and Planet have increased the overall presence of spacecraft in low-Earth orbit (LEO). With more SmallSat missions currently being planned, the demand for SmallSat launches is expected to rise exponentially (1).

To address the growing demand for SmallSat launches, the 'rideshare' launch market has emerged as a popular means for SmallSats to access space. This trend has been fueled by an increase in launch providers and improved launch capabilities, as well as the development of adapters and dispensers specifically designed for SmallSat and CubeSat missions. Rideshare launches are typically multi-manifest launches that either consist of: 1) a large primary spacecraft that determines all the mission requirements (e.g., orbit, schedule, concept of operations, etc.), where secondary payloads are manifested to take advantage of the launch vehicle's surplus mass, volume, and other performance margins, or 2) a single launch vehicle consisting entirely of SmallSats, also known as a 'dedicated rideshare'. It should be noted that the term 'payload(s)' in this chapter refers to the entire free-flying spacecraft deployed from a launch vehicle and is defined differently in chapter 2 *Complete Spacecraft Platforms* as instruments/technologies on spacecraft.

Additionally, nanosatellite form factors are increasing in dimensions and mass, requiring larger dispensers to accommodate these larger CubeSat sizes. Orbital transport vehicles (OTVs), along with generally more capable orbital maneuvering vehicles (OMVs) can offer "last mile" delivery services to intended orbits for small satellites. These vehicles are becoming a more common paradigm for SmallSat and CubeSat deployment and operational logistics. While historically referred to as "Space Tugs," this term is in the process of being phased out by many in the commercial industry. The terms OTV and OMV seem to also be falling out of favor, with many commercial offerings simply referring to the available systems as vehicles capable of in-space transportation services. For the purpose of this report, the terms OTV/OMV describe systems with on-board propulsion that can be launched to an approximate orbit and then propel themselves to one or more target orbits. There they can either deploy the small spacecraft or serve as an integral part of the hosted payload. This chapter focuses on the use of the vehicles as deployers, while more information about hosted payload services is found in chapter 2 *Complete Spacecraft Platforms*, section 2.2.1. Some OTVs/OMVs are based on a traditional rocket kick stage and are intended to work with specific launch vehicles. Several commercial companies have successfully flown OTVs/OMVs and are booking future launch manifests.

Expanding future capabilities of small satellites will demand dedicated launchers. Flying a spacecraft as a dedicated payload may be the best method of ascent for missions that need a very specific orbit, near complete capability of available launcher performance, interplanetary trajectories, precisely timed rendezvous, or special environmental considerations. Technology developers and hard sciences can take advantage of the quick iteration time and low capital cost of small spacecraft to yield new and exciting advances in space capabilities and scientific

understanding. The emergence of very small launch vehicles has altered the landscape by providing dedicated rides for small spacecraft to specific destinations on more flexible timelines.

NASA's Launch Services program developed a new Indefinite Delivery/Indefinite Quantity (IDIQ) mechanism in Q1 2022: the Venture Class Acquisition of Dedicated and Rideshare (VADR) launch services. The VADR IDIQ contract mechanism was created to accommodate a more risk tolerant approach for NASA's new class of launches using selected commercial providers. VADR-procured launch services enable unique launch capabilities for Class D or higher risk tolerant payloads and provide FAA licensed launch services capable of delivering payloads to a variety of orbits. This contract mechanism offers a broad range of commercial launch services for traditional and dedicated rideshare options. The commercial approach uses a lower level of mission assurance for payloads with higher risk tolerance and contributes to NASA's science and research development efforts as an ideal platform for technical development. The 2022 Heliophysics Small Explorers Announcement of Opportunity and Mission of Opportunity were the first NASA AO's to use this contract structure. The VADR IDIQ contract provides a new mechanism for traditional and dedicated rideshare launches for risk-tolerant payloads. While 13 companies were initially selected, an on-ramp provision allows new launch services and capabilities to be proposed (2).

For educational small spacecraft, several initiatives have helped provide these opportunities. NASA's CubeSat launch initiative (CSLI) for example, has provided rides to a significant number of schools, non-profit organizations, and NASA centers. As of September 2024, the initiative launched 165 successful CubeSat missions, and continues to select CubeSats for launch (10). The European Space Agency (ESA) "Fly Your Satellite" program is a similar program which provides launch opportunities to university CubeSat teams from ESA Member States, Canada, and Slovenia (11).

The list of organizations/companies in this chapter is not all-encompassing and does not constitute an endorsement from NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. The information is for awareness and guidance only. The performance advertised may differ from actual performance since the information has not been independently verified by NASA subject matter experts and relies on information provided directly from the manufacturers or available public information. It should be noted that TRL designations may vary with changes specific to the 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 TRL of the described technology.

10.2   State-of-the-Art – Launch Integration Role

Launch options for SmallSats include dedicated launches or rideshare launches. Regardless of the approach, integration with the launch vehicle is a complex and critical portion of the mission. The launch integration effort for a primary spacecraft typically includes the launch service provider, the spacecraft manufacturer, the spacecraft customer, the launch range, and sometimes a launch broker or integrator (3). When launching on a rideshare launch, the launch integration may become complex due to its multi-customer, multi-mission nature.

There are several options for identifying and booking a launch for a SmallSat. For rideshares, the spacecraft customer may choose to use a launch broker or launch integrator to facilitate the launch manifest or work directly with the launch service provider. A launch broker matches a spacecraft mission with a launch opportunity, whereas an integrator provides additional services related to multi-mission manifesting and/or integration.

Whether a spacecraft customer chooses to use a launch integrator or not, it is the responsibility of the spacecraft owner/operator to obtain the appropriate licensing for their spacecraft. These

licenses may include: radio frequency licensing, remote sensing licensing, and laser usage approval (4) (5). The launch integrator or the launch service provider will require proof of licensure before integrating and/or launching the satellite. They will also require additional safety-related analyses and supporting data/documentation prior to launch, which typically includes orbital debris information, materials and venting data, and spacecraft specific models (6).

For rideshare launches, many satellites are subject to a "do no harm" approach to protect the other spacecraft on the launch vehicle. A list of "do no harm" requirements are imposed on the rideshare satellites by the launch provider, launch integrator, or primary mission owner. These requirements vary by launch provider and launch integrator, but usually include restrictions on transmitters, post separation mechanical deployments, and hazardous materials. A comprehensive list of typical "do no harm" requirements is provided in the NASA Rideshare User Guide (RUG) (7).

10.2.1 Launch Brokers and Launch Integrators

A launch broker for small satellites is an individual or organization which matches a spacecraft with a launch opportunity, usually as a rideshare spacecraft. Typically, a launch broker does not provide any additional launch integration services beyond coordinating the relationship between the spacecraft manufacturer or customer and the launch service provider. Their purpose is to fill excess capacity on a launch, and they can also bolster negotiations between the launch provider and payload for scheduling, integration, safety testing, and cost (8).

Launch integrators work with the satellite customer and the launch vehicle provider to ensure that the customer's spacecraft is compatible with the launch vehicle by performing analyses and physical integration services. The launch integrator may provide the CubeSat dispenser, separation system, or other hardware required for integration, or these may be provided by the spacecraft customer.

10.3 Launch Paradigms

The SmallSat market has grown considerably over the past decade experiencing a 23% compound annual growth rate from 2009 to 2018 (9), and this growth continues unabated. From 2013 to 2017 there was an average of ~140 SmallSats (less than 200 kg) launched per year. From 2017 to 2021 this number increased to an average 332 SmallSats per year, with more than 550 SmallSats launched in 2022 thanks to Starlink, Plan (1). Of the total spacecraft launched in 2023, 200-600 kg type spacecraft accounted for 40%, while micro, nano, pico, and femto spacecraft were the next most launched spacecraft (1). The 200-600 kg class spacecraft has seen a considerable launch growth since 2020, primarily due to mega constellation Starlink by SpaceX.

This increase in small satellite demand has caused a shift in the launch vehicle market, with many companies creating or advertising launch platforms centered around small satellites. While other chapters in this report cite specific companies providing "state-of-the-art" technologies, this section will provide an overview of two types of launch methods for SmallSats and the current state of these markets: dedicated and rideshare.

10.3.1 Dedicated Launches

In the context of this report, dedicated launches for SmallSats are those generally used to launch satellites with a mass less than 180 kg. However, this does not mean that the maximum mass to orbit is 180 kg or less, as some dedicated launchers have a payload maximum of 1000 kg, and many launch vehicles marketed for SmallSats can deliver masses to orbit that are higher than 180 kg. The primary orbit for this type of launch is low-Earth orbit (LEO), with very few companies

currently targeting highly elliptical orbit (HEO), medium-Earth orbit (MEO), or geostationary equatorial orbit (GEO).

Dedicated launches for SmallSats have many advantages. A SmallSat on a dedicated launch controls the mission requirements in whole — what they need, when they want to launch, and where they want to go. They generally have a readiness "go/no-go" call on launch day in case something goes wrong with their satellite pre-launch. They can also request special launch accommodations, such as a nitrogen purge or late battery charge, that are generally not available to a rideshare launch (this may be as a standard service or with an additional cost as mission-unique). The downside to a dedicated launch is that they are generally more expensive than a rideshare launch.

10.3.2 Rideshare Launches

Until recently, there were only a few launchers that allowed small spacecraft to ride as primary spacecraft. The majority of small spacecraft are launched as rideshares. Rideshare arrangements provide more launch options with demonstrated launch vehicles. Since almost any large launcher can fit a small payload within its excess mass and volume margins, there is no shortage of options for spacecraft that want to fly as a secondary spacecraft. On the other hand, there are downsides of ridesharing. For rideshares as a secondary payload to a larger primary spacecraft, the launch date and trajectory are determined by the primary customer, and the secondary payloads must take what is available. In some cases, they need to be delivered to the launch provider and be integrated on the adapter months before those spacecraft that are manifested in dedicated rideshares. Generally, the secondary spacecraft riding along with a larger primary spacecraft are only allowed to be deployed once the primary spacecraft successfully separates from the launch vehicle, but there are instances where the rideshare spacecraft separate prior to the primary spacecraft (12).

Dedicated rideshare launches are those that deploy multiple small spacecraft into orbit using a single launch. The orbits in which the spacecraft are deployed are similar but may vary in altitude. These types of launches are growing in popularity with many launch vehicle providers offering regular dedicated rideshare launches to the same altitude at regular intervals throughout the calendar year. While challenging, the logistics of these missions are managed by various launch integrators throughout the market, many of which are new to industry but are forging a new path in ridesharing.

10.4   Deployment Methods

The method by which SmallSats are deployed into orbit is a critical part of the launch process. The choice of deployment method depends on the form factor of the satellite. This section will discuss the deployment of CubeSats, which generally use CubeSat dispensers, and the deployment of free-flying SmallSats.

10.4.1 CubeSat Dispensers

The CubeSat form factor is a very common standard for spacecraft up to approximately 24 kg (12U CubeSat) but can also be extended to approximately 54 kg in a 27U configuration (13). The most updated CubeSat Design Specification document is found at http://www.cubesat.org, a website maintained and operated by California Polytechnical State University, San Luis Obispo, the creators of the CubeSat form factor.

The CubeSat form lends itself to container-based integration systems, or dispensers, which serve as an interface between the CubeSat and the launch vehicle. It's a rectangular box with a hinged door and spring mechanism. Once the door is commanded to open, the spring deploys the CubeSat. Many companies currently manufacture dispensers for the CubeSat form factor which

follow one of two constraint systems: the rail-type dispenser, and the tab-type dispenser. Due to the large number of dispenser manufacturers, the different companies are not listed here. Instead, a brief overview of the two types of dispensers is provided.

A rail-type dispenser (figure 10.1) supports CubeSats that have rails which extend the length of the CubeSat on four parallel edges. The rails on the CubeSat prevent it from rotating inside the dispenser. After the dispenser door opens, the rails slide along guides inside the dispenser and the CubeSat is deployed. As such, it is important that any rail-based CubeSat follow the current development specifications to ensure compliance. This type of dispenser is the most widely manufactured configuration, with more than fifteen manufacturers worldwide. Some rail-type dispensers use a clamping mechanism to securely hold the CubeSats in place during launch.
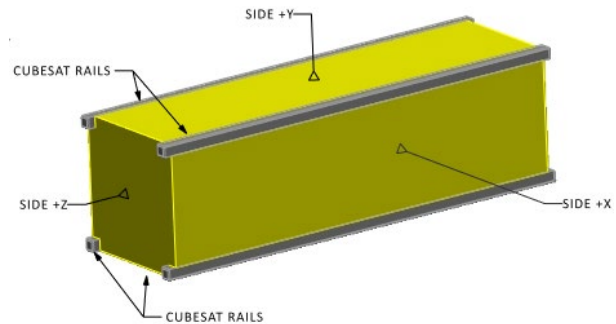
Figure 10.1: The Rail-type CubeSat. Credit: CalPoly's CubeSat Program.

A tab-type dispenser (figure 10.2) supports CubeSats with tabs which run the length of the CubeSat on two parallel edges. Typically, the dispenser grips the tabs to hold the CubeSat in place, only releasing it after the door has been commanded to open. More developers are beginning to develop their own tab-based designs for CubeSat dispensers. Many are based on the original Planetary Systems Corporation standard, however some offer features such as built-in isolation to accommodate for the launch vehicle environment. In addition, there are some tab-based dispensers that do not grip the tabs. Rather, they provide a slot to accommodate the tab, which slides freely within the slot. While use of tab-type dispensers is growing, they remain a minority among dispensers purchased and used by developers.
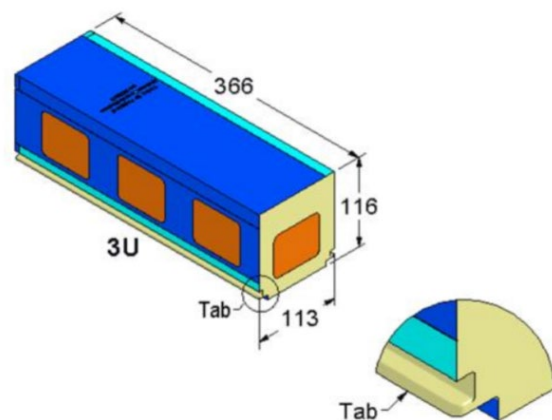
Figure 10.2: The Tab-type CubeSat. Credit: Planetary Systems Corporation.

While CubeSats can generally pick their dispenser type (rail vs. tab), the choice of the actual dispenser is not always a decision made by the CubeSat. In many cases, the launch vehicle provider or launch broker/integrator has already determined which dispensers will be installed on the launch vehicle. As each dispenser manufacturer has slightly different volumes and requirements, it is beneficial for the CubeSat to design for as wide a range of dispensers as possible to maximize launch opportunities.

Additionally, some dispenser manufacturers have features that may violate the "do no harm" requirements set forth by the launch vehicle or launch integrator. Therefore, it is beneficial for the CubeSat to evaluate "do no harm" recommendations from a variety of organizations, as these requirements can vary from flight to flight on the same launch vehicle based on the risk posture of the primary payload and/or the mission "owner" (7).

10.4.2 SmallSat Separation Systems

Small satellites which do not meet the CubeSat form factor, or will not be using a CubeSat dispenser for integration to the launch vehicle, require a different separation mechanism. Separation systems for SmallSats generally follow either a circular pattern or a multi-point (3 or 4 point) pattern. Depending on the launch vehicle or integrator, separation systems may already be in place and available to secondary spacecraft. It should be noted that separation systems are often some of the most complicated pieces of hardware involved with launching spacecraft. If a spacecraft is given the option to bring its own separation system to launch, great care should be taken in selection, including the development maturity and flight heritage for any separation system.

Circular separation systems use two rings held together by a clamping mechanism. One ring is attached to the launch vehicle and the other ring is attached to the spacecraft. Once the clamping mechanism is released, the two rings separate and are pushed apart by springs. Each ring then remains with the spacecraft or the launch vehicle. There are two primary types of clamping configurations, motorized light bands (MLB) and Marman clamps.

The MLB (figure 10.3) is a motorized separation system ranging from 8 inches to 38 inches in diameter. Smaller MLB systems are used to deploy spacecraft less than 180 kg, while larger variations may be used to separate larger spacecraft or other hosted payload hardware (see section 2.2.1), which are discussed below. The MLB's separation system eliminates the need for pyrotechnic separation, resulting in lower shock deployments with no post-separation debris.
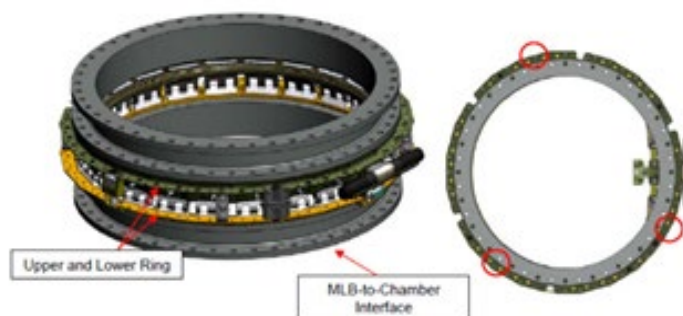
Figure 10.3: MkII Motorized Lightband. Credit: Planetary Systems Corporation.

Marman band separation systems use energy stored in a clamp band, often along with springs, to achieve separation. The Marman band is tensioned to hold the spacecraft in place. Some Marman bands use pyrotechnic devices to cut the clamping bolt, however many companies offer a low shock release mechanism which is potentially better for spacecraft sensitive to shock. Sierra Space produces a Marman band separation system known as Qwksep, which uses a series of separation springs to help deploy the spacecraft after clamp band release. RUAG Space provides several circular separation systems which use their Clamp Band Opening Device (CBOD) release mechanism to reduce shock impact on the spacecraft (14).

Several companies are now providing multi-point separation systems instead of the circular band. Using a multi-point separation system may result in mass savings over a circular separation system. However, some systems require additional simultaneous signals from the launch vehicle provider to ensure proper release. The RUAG PSM 3/8B is a low-shock separation nut developed to fit OneWeb satellites (15). It requires additional firing commands from the launch vehicle or a dedicated sequencing system. ISISPACE has also developed the M3S Micro Satellite Separation System (see figure 10.4) which is designed for satellites up to 100 kg but can be configured for higher masses (16).

The Advanced Satellite Dispenser (ASD) by Rocket Lab is a deployment mechanism for small payloads that complies with the Planetary Systems Corp Rocket Lab (PSC-RL) document. The

walls and door are optional meaning the ASD can fly with or without canisterization. This flexibility allows the freedom to significantly increase the payload's volume if they can use tab rails.

Given the stiffness and fundamental frequency requirements of traditional rideshare missions, many companies are also shifting to 4-point separation systems for MicroSats and SmallSats as a viable alternative to traditional MLB or clamp band systems. These systems function in a similar way to the systems above and are typically rated for microsatellites (≤100 kg), however boast less complexity than a traditional MLB or Clampband. The rapid acceptance of this launch solution is driven

Figure 10.4: ISISPACE M3S Micro Satellite System. Credit: ISISPACE.

by the fundamental frequency requirements of traditional rideshare launches, with the hope that reduced stiffness at the interface will increase the compatibility of SmallSats and MicroSats for those types of launches. In addition to reduced complexity, many of these result in cost savings as well, which can be passed on to both the integrator and the SmallSat manufacturer. Many integrators are exploring the addition of such systems into their portfolio to accommodate launches in the near future.

Cake Topper and Plate System for Rideshares

SpaceX has developed a system that differs from the SPA system for rideshare missions to SSO (Transporter Missions) and mid-inclination orbits (Bandwagon Missions). This system of plates rather than a ring is intended to allow more payloads to be included in the circumferential space for flight on their commercial rideshare missions. In addition, for larger spacecraft or spacecraft that cannot be horizontally mounted during flight, they also offer a cake topper option. Figure 10.5 shows these two options. The blue box shows the plate option which has a specific set of rideshare loads for the missions' part of transporter or bandwagon. The green box denotes the cake topper option which also has separate environments. User guides for both configurations are

Figure 10.5: SpaceX cake topper (green) and Plate System (blue) configurations. Credit: SpaceX.

available on the launch provider's website. For missions that will fly under the VADR contract, please contact NASA's Launch Services Program (LSP) for additional guidance and enveloping environments.

10.4.3 Integration Hardware

A main driver for CubeSat utility is their adhesion to a standard that can be integrated into several different launch configurations. The physical hardware that attaches both a containerized and non-containerized small spacecraft and keeps it insulated from a rocket body include deployers, adapters, dispensers, and launchers. The purpose of this hardware is to eject the spacecraft

safely into orbit, and most services offer different features, interfaces, connections, and designs for small spacecraft specifications. The exact configuration and standards vary by launch vehicle, and the determination of an appropriate and reliable launch option is part of the launch qualification process (8). With this rise in CubeSat constellations, integration hardware capable of launching multiple SmallSats simultaneously and consecutively is now a standard. This section will highlight some existing examples of integration flight support hardware applicable to both SmallSats and CubeSats, but the reader is highly encouraged to identify other integration services.

Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (ESPA)

The ESPA ring (Figure 10.6, top) is a multi-payload adapter for large primary spacecraft originally developed by Moog Space and Defense Group. Six 38 cm (15") circular ports can support six auxiliary payloads up to 257 kg each. It was used for the first time on the Atlas V STP-1 mission in 2007. The ESPA Grande (Figure 10.6, lower) uses four 61 cm (24") circular ports which can carry spacecraft up to 450 kg (991 lb) (17). Although developed by Moog, several other companies now offer similar designs in different configurations.

Small Spacecraft Mission Service (SSMS) Dispenser

ESA has developed the Small Spacecraft Mission Service dispenser for the Vega launch vehicle (Figure 10.7). This dispenser comes in a variety of different modular parts which can be configured based on the satellite launch manifest. The modularity of the dispenser provides greater flexibility for accommodating different customers (18).

Figure 10.6: [top] ESPA Ring [lower] ESPA Grande Ring. Credit: Moog, Inc.

Dual / Multi Payload Attach Fittings (DPAF / MPAF)

Many launch vehicle providers have existing accommodations for two or more payloads which are sometimes referred to as Dual Payload Attach Fittings (DPAF) or Multi Payload Attach Fittings (MPAF). As these are generally launch vehicle specific, and occasionally mission specific, they are not discussed here.
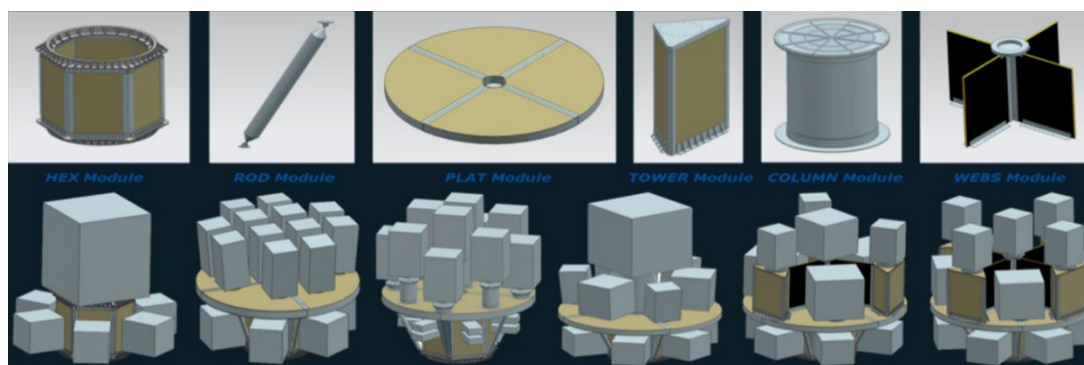
Figure 10.7: The European Space Agency Small Spacecraft Mission Service Dispenser for the Vega Launch Vehicle (19). Credit: European Space Agency.

10.5   Orbital Transfer / Maneuvering Vehicles (OTV/OMV)

One of the main disadvantages of riding as a rideshare is the inability to launch into the desired orbit. For rideshares with a primary spacecraft, the primary mission determines the orbital destination, so the secondary spacecraft orbit usually does not perfectly match their specific needs. However, by using an OTV or OMV, secondary spacecraft can maneuver much closer to their desired orbits. OTVs are generally more coarsely propulsion-capable, while OMVs may offer hosted systems more in terms of power, pointing, and communications. The OTV / OMV market is nascent, with many planned systems but few with existing flight heritage. This emerging technology is an area of interest in the near term for both SmallSats and CubeSats, as it offers a significant capability to reach destinations not previously achievable with systems of this scale. The ability for small spacecraft to reach new orbits could enable a much wider range of mission designs for destinations both near and far.

This section primarily focuses on systems that offer delivery or deployment satellite services, while systems that primarily offer instrument/technology payload hosting can be found in Chapter 2 *Complete Spacecraft Platforms*. The distinction between OTV / OMVs and hosted payload providers are not always clear, with many OTV / OMVs offering both the deployment of free-flying spacecraft at secondary orbits from the vehicle, and the hosting of instruments/technologies that stay attached to the vehicle. Similar OTV / OMV systems can be used to service other spacecraft, including deorbiting or relocation to a graveyard orbit. For information regarding those capabilities, refer to Chapter 13 *Deorbit Systems*.

While commercial delivery and deployment system launches have increased in cadence since the early 2020s, their launches have slowed a bit in 2024. Below is a launch timeline of OTV/OMVs with commercial payload services as part of the vehicle's documented purpose. This cadence is expected to continue an upward trend as more vehicles become space qualified and as new companies emerge. The data is derived from publicly accessible data and only contains launches by the providers described in this report. There may be more launches than indicated herein, and the success of the launch/mission is not factored.
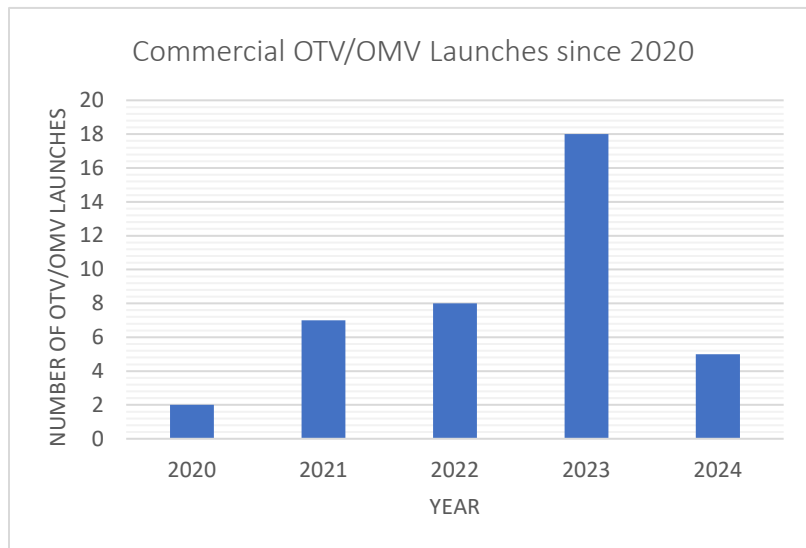
Figure 10.8: Number of commercial OTV/OMVs launched per year since 2020. Credit: NASA

10.5.1 Commercial Development/Services

As discussed above, the ESPA ring provides structure to which SmallSats or CubeSat dispensers are mounted. The idea of adding propulsion to an ESPA ring led to many of the early commercial OMV/OTV vehicle designs. Systems that were originally derived from the ESPA ring include the Spaceflight Inc. (now Firefly) SHERPA (19), the Orbital Sciences Corporation (now Northrup Grumman Space Systems) ESPAStar Product Line (20), and the Moog METEOR (21). The ESPAStar product line has gone through multiple programs with multiple names. The ESPAStar

line has significant flight heritage from the Long Duration Propulsive ESPA (LDPE) satellites, now known as Rapid On-Orbit Space Technology Evaluation Ring (ROOSTER) United States Space Force (USSF) program. While the SHERPA system is no longer commercially available, FireFly is now offering rides on its Elytra OTV/OMV product line, which appears to build upon the ESPA ring heritage for specific compatibility with FireFly's small- and medium-lift launch vehicles. Rocket Lab is another launch provider offering an OMV/OTV platform. The Photon system is an evolution of the Electron launch vehicle Kick Stage (22).

Other systems are being developed from the ground-up by commercial entities to provide orbital transport, hosting, and deployment services. These include the Momentus Vigoride vehicles, the Epic Aerospace Chimera vehicles, the Exolaunch Reliant, the Moog SL-OMV, the TransAstra WorkerBee, the Atomos Quark, the Impulse Space Mira and Helios vehicles, and the D-Orbit ION vehicles. Northrup Grumman, in addition to its ESPAStar series, has a proven Mission Extension Vehicle (MEV) that can host payloads, but is primarily developed to dock and extend the life of large GEO satellites. To date, the MEV has successfully extended the life of two IntelSat systems.

The current state-of-the-art OTV vehicle offerings tend to advertise 1000's of km/s of delta-V, which enables the following services:

- Altitude change / planetary transfer – changing the altitude from where the vehicle was deployed by the launch vehicle. This could be an altitude raising or lowering operation, depending on the needs of the integrated payloads. In some cases, if escape velocity is reached, this maneuver can be used to transfer to lunar orbits or beyond. A payload delivery and deployment vehicle could perform multiple altitude raises and lowers during a single mission.
- Inclination change – changing the angle of the orbit with respect to the angle at which the vehicle was initially deployed by the launch vehicle. This can be done during the same burn as an altitude change to increase efficiency, or as a standalone operation. This operation dramatically changes what is viewable on the ground swath by the orbiting spacecraft. It can also dramatically change the solar incidence of the orbit.
- Phasing – this refers to changing the position of a spacecraft within a given orbit. Assuming the same orbit of two spacecraft, the phasing of each of them would dictate at what time each of them are over a specific ground swath. This is typically achieved with a "slow-down" burn to achieve a smaller orbital period, and then a "recircularization" burn to return the spacecraft to its original orbit.
- Constellation deployment – this is the ability for a single payload delivery vehicle to drop multiple spacecraft into a constellation formation. This dramatically reduces the propulsive need of the individual small spacecraft and can enable constellations of small spacecraft to achieve much more than they could if they needed the propulsion to deploy and phase themselves.

Many OTVs and OMVs are designed with a large excess propellant volume so that after the initial contracted services, the hosting vehicle can remain in orbit and be contracted for additional future services such as deorbiting via "pushing" another spacecraft, or contracted inspection services. Some providers are developing systems specifically for these in-space services, including Astroscale, Starfish Space, and Turion Space. Technology gaps to developing all these highly capable space vehicles include RF licensing challenges, ACDS component availability, reliable propulsion systems, and lack of standardization for payload interfacing.

The following sections contain an overview of commercial OTV/OMV vehicles and their development status, and Table 10-2 provides corresponding vehicle parameters.

Firefly

The Elytra vehicle line is a series of orbital vehicles offered by Firefly Aerospace. The smallest vehicle is the Elytra Dawn, optimized for intra-LEO mobility. The largest vehicle is the Elytra Dark, which offers payloads transportation to lunar orbit and beyond. The Elytra line builds off Firefly's previous Space Utility Vehicle design and experience, as well as the Spaceflight SHERPA (Firefly acquired Spaceflight in 2023). The first Elytra mission is planned to fly aboard a Firefly Alpha rocket at the end of 2024 in support of a National Reconnaissance Office (NRO) mission.

Epic Aerospace

Epic Aerospace has a line of Chimera systems ranging from smaller LEO vehicles up to larger GEO-targeting vehicles. The LEO vehicles are intended to enable altitude changes from 450 km from the deployment location to larger, intensive and fast phasing maneuvers (3 hours of Local Time at Ascending Node (LTAN) change in less than 90 days is advertised (23)). The GEO systems are advertising capabilities beyond Earth, with trans-lunar injection from GTO/LEO as an option. The first LEO Chimera system was launched in January of 2023 aboard a SpaceX Transporter and is currently operational.

D-Orbit

The D-Orbit ION Satellite Carrier system was first used in 2020 to deploy Planet Labs constellation satellites and has been used 14 times as of August 2024 to deploy small satellites. Readers are encouraged to reach out to the company for more information.

Rocket Lab

The Kick-Stage-derived Photon system is a flight-proven vehicle that deploys payloads to target orbits not otherwise achievable by the Electron launch vehicle. In addition, the system can be mounted on an ESPA port of other launch vehicles as a secondary payload. Rocket Lab advertises other vehicles alongside Photon that can be used for hosted payload services and orbital insertion services; Explorer, Lightning, and Pioneer,

The Photon system has been flown four times as of September 2024. The third Photon mission successfully delivered the NASA CAPSTONE spacecraft to a near-rectilinear halo orbit around the Moon via a trans-lunar injection maneuver. The system used for CAPSTONE is now referred to as the Rocket Lab Explorer. The Explorer platform is now being used as the basis for the spacecraft envisioned for the ESCAPADE mission to Mars.

Atomos

Atomos has plans to offer many in-space services to small spacecraft such as deployment and orbit changes (raising, phasing, and inclination), rendezvous, docking, and satellite life extension. The first version of their OTV/OMV, the *Quark-LITE* launched in spring 2024 and despite communication and tumbling issues, it successfully demonstrated the integrated hardware and software developments for high delta-V orbital transfer capability and life extension in GEO (24). Upcoming Atomos GEO-1 and -2 missions will use the *Quark-Alpha* with payload delivery and hosting services to GEO.

Momentus Vigoride

Momentus Space has developed and flown the Vigoride orbital service vehicle. The maximum payload mass on Vigoride is 800 kg to LEO, and can be launched from an ESPA Grande ring, SpaceX XL rideshare plate, or a dedicated launch vehicle. Vigoride uses microwave electrothermal thrusters (MET), water plasma engines, to change the orbit prior to releasing payloads at their final orbit (25). Like all OMV/OMTs, the Vigoride is capable of changing inclination, altitude, and orbital planes. The first flight for Vigoride occurred in May of 2022. While

the inaugural flight had issues due to failed solar array deployment, two additional flights of the system occurred in 2023 and were both successful. The company has additional Vigoride systems ready for customer missions planned to fly in 2026.

UARX Space

UARX Space is developing the Orbit Solutions to Simplify Injection and Exploration (OSSIE) orbital transfer and hosted payload vehicle. This spacecraft is designed to be modular and scalable to satisfy customer requirements by using either electric or chemical propulsion. OSSIE has been developed together with other partners such as DLR to transport up to 400 kg. The vehicle uses green propellant thrusters made by Dawn Aerospace to maneuver. In September 2023, UARX space signed a cooperation agreement with Sener to manage the guidance, navigation, and control of the transfer missions. The first OSSIE qualification mission, scheduled to launch in June 2025, will hold 12 customers spanning from PocketQubes, CubeSats, and hosted payloads (28).

Impulse Space

The Mira Orbital Transfer Vehicle by Impulse Space provides several specific services such as orbital transport, constellation deployment, payload hosting, deorbiting maneuvers, and others. Mira's first launch was November 2023 on the SpaceX Transporter-9 mission (LEO Express 1) and has since then met all mission objectives over the course of nine months (29). Mira is capable of 600 m/s delta-v for a 300 kg payload via eight (8x) Saiph (27N) bi-propellant thrusters and sixteen (16x) cold-gas thrusters, in four clusters of four, enabling 6DOF control. There are plans to launch a second LEO-Express mission late 2024 for further technology development and validation.

While the Mira vehicle was designed for payload hosting, deployment, and orbital adjustments, the Helios vehicle is a high-powered kick stage advertised to provide transportation services out to Cislunar environments. Helios uses a combination of liquid oxygen and liquid methane to transport more than 4,000 kg of payload from LEO to GEO in less than 24 hours. The first flight is planned for no earlier than 2026 (30).

Space Machines Company

The Optimus vehicle is developed to carry hosted payloads and perform close inspections of Client Space Objects (CSOs) in LEO. The vehicle was first flown in 2024 with 8 hosted payloads from international customers and was powered by a bi-propellant chemical propulsion system (31). Optimus will be updated before next planned launches in 2026 and 2027 to LEO mid-inclination and SSO orbits. The updated 'Block-2' platform will be a maneuverable, rapid-response vehicle carrying improved sensors for detailed inspection images to be taken of satellites from 10km. Updates also include an RCS system that will allow for close approach maneuvers to within 1km of a CSO.

Blue Origin

The Blue Ring space mobility vehicle by Blue Origin is advertised to provide in-space computing capability, hosting services, and delivery services for more than 3000 kg of commercial and government payloads. Blue Ring aims to support missions in medium Earth orbit out to the cislunar region as a host spacecraft platform. It uses a combination of chemical and solar electric propulsion with rollout solar arrays. Blue Origin plans to test Blue Ring's mission operation capabilities and core flight systems in space for the first time on their New Glenn rocket in 2025 (32). The New Glenn rocket completed its first second-stage hot fire test in September 2024 and is on track for its first mission, DarkSky-1 (33).

Table 10-1: Commercial Orbital Transfer / Maneuvering Vehicles						
Company Headquarters	Vehicle	Heritage	Operational Altitudes	Maximum Payload Capacity (kg)	Delta-V	Ref.
Atomos Space USA	Quark	Flown	LEO	400	-	(40)
Blue Origin USA	Blue Ring	Upcoming	Not disclosed	-	-	(32)
D-Orbit Italy	ION	Flown Successfully	Not disclosed	-	-	(38)
Epic Aerospace Argentina	Chimera LEO Block 0	Flown Successfully	+/- 450 km from LEO deployment	300	200 m/s at max payload	(23)
	Chimera LEO Block 1	Upcoming	+/- 450 km from LEO deployment	300	950 m/s at max payload	(23)
	Chimera GEO Block 0	Upcoming	GTO to TLI	300	1800 m/s at max payload	(23)
	Chimera GEO Block 0 / Bus	Upcoming	GTO to TLI	290	1800 m/s at max payload	(23)
Exolaunch Germany	Reliant Standard	Upcoming	+/- 275 km from LEO(?) deployment	200	-	(39)
	Reliant Pro	Upcoming	+/- 275 km from LEO(?) deployment	260	-	(39)
Exotrail France	Spacevan LEO	Flown Successfully	LEO (~400-2000 km)	~200	~500 m/s	(46)
	Spacevan GEO	Upcoming	Above LEO (e.g., MEO, GEO, Cislunar >2000 km)	~150	~2200 m/s	(47)
Firefly USA	Elytra Dawn	Upcoming	LEO	-	-	(41)
	Elytra Dusk	Upcoming	LEO to GEO	-	-	(41)
	Elytra Dark	Upcoming	LEO to Lunar/Planetary	-	-	(41)
Impulse Space USA	Mira	Flown Successfully	LEO	300	500 m/s at max payload	(29) **Error! Reference source not found.**

Table 10-1: Commercial Orbital Transfer / Maneuvering Vehicles						
Company Headquarters	Vehicle	Heritage	Operational Altitudes	Maximum Payload Capacity (kg)	Delta-V	Ref.
	Helios	Upcoming	GEO	-	-	(30)
Moog USA	SL-OMV	Upcoming	400 - 700 km	6 x 12 (+ additional, if containerized)	Up to 200 m/s	(31)
	METEORITE	Upcoming	500 – 1200 km	100 (or more, with caveats)	>175 m/s	(35)
	METEOR	Upcoming	500 – 1200 km	750	>400 m/s	(36)
Momentus USA	Vigoride	Flown Successfully	250 - 2000 km (GEO and LLO also available)	800	up to 2000 m/s	(37)
Rocket Lab USA	Photon	Flown Successfully	LEO, MEO, GEO, and beyond	-	-	(22)
Space Machines Company Australia	Optimus	Flown	300 – 700 km	50	500 m/s	(44) (45)
TransAstra USA	WorkerBee (multiple configurations)	Upcoming	LEO, MEO, GEO, HEO, Lunar/Planetary	200 – 2000	-	(42)
UARX Space Spain	OSSIE	Upcoming	350 – 1000 km	200	690 m/s	(26)
Quantum Space USA	RANGER	Upcoming	LEO, MEO, GEO, cislunar	6000	Up to 10 km/s	(43)

Information not disclosed represented as -

10.6   International Space Station Options

The International Space Station (ISS) provides several methods for deploying CubeSats and SmallSats. The sections below discuss SmallSat deployment from the ISS as well as deployment above the ISS. The ISS also accommodates hosted payload instruments/experiments, but those accommodations are outside the scope of this chapter as they are for individual instruments/experiments themselves and are not satellites.

10.6.1 Deployment from ISS

The ISS provides several options for deploying satellites. Generally, satellites are launched below the ISS to avoid potential contact with the ISS. Below are several options available for launching from the ISS.

Nanoracks ISS CubeSat Deployer (NRCSD)

Nanoracks CubeSat Deployer (NRCSD) (figure 10.9) is a self-contained CubeSat dispenser system that mechanically and electrically isolates CubeSats from the ISS, cargo resupply vehicles, and ISS crew. The NRCSD is a rectangular tube that consists of anodized aluminum plates, base plate assembly, access panels, and deployer doors. The inside walls of the NRCSD are a smooth bore design to minimize and/or preclude hang-up or jamming of CubeSat appendages during deployment, should they become released prematurely.

For deployment, the platform is moved outside via the Kibo Module's Airlock and slide table, which allows the Japanese Experimental Module Remote Manipulator System (JEMRMS) to move the dispensers to the correct orientation and provides command and control to the dispensers. Each NRCSD can hold six CubeSat units as large as a 6U (1 x 6U). The NRCSD DoubleWide can accommodate CubeSats up to 12U (2 x 6U) with Nanoracks being able to launch up to 48U per cycle. The CubeSats deploy at a 51.6° inclination, 400 – 420 km orbit 1 to 3 months after berthing at the station.

Figure 10.9: Nanoracks CubeSat Deployer. Credit: Nanoracks.

Nanoracks ISS MicroSatellite Deployment – Kaber Deployer Program

Nanoracks Kaber Microsat Deployer is a reusable system that provides command and control for satellite deployments into orbit from the Japanese Experimental Module Airlock Slide Table of the ISS. The Kaber supports satellites with a form factor of up to 24U and mass of 82 kg and uses a Nanoracks separation system with circular interface similar to the separation systems discussed above. Satellites are launched to the ISS on a pressurized launch vehicle, mounted to the Kaber deployer, and deployed outside the ISS (48).

JEM Small Satellite Orbital Deployer (J-SSOD)

The Japanese Experimental Module (JEM) Small Satellite Orbital Deployer (J-SSOD) is a Japanese Aerospace Exploration Agency (JAXA) developed CubeSat deployer used to launch CubeSats from the ISS. The J-SSOD can launch CubeSats up to the 6U form factor (2x3

configuration). The satellites, with their dispensers, are installed on the Multi-Purpose Experiment Platform prior to Kibo's robotic arm Japanese Experiment Module Remote Manipulator System (JEMRMS) transferring the Multi-Purpose Experiment Platform (MPEP) to the release location. At that point, the CubeSats are deployed (49).

Bishop Nanoracks Airlock Module

A new airlock module, Bishop, was developed for the ISS by Nanoracks, Thales Alenia Space, and Boeing, and is the first commercialized, private module for the space station (50). Bishop provides more than five times the volume of the current Japanese Experimental Module (JEM) airlock, allowing for larger satellites and payload experiments. Bishop can host satellites and payloads, as well as deploy them, based on the needs of the mission. It has been attached to the exterior of the ISS since December 21, 2020 and has been instrumental in deploying CubeSats from the ISS (51).

10.6.2 Deployment Above ISS

Regular access to the ISS is very attractive for many satellite providers. However, the lower altitude of the ISS means the in-orbit lifetime for the satellite is generally shorter. This section discusses the options that have been developed to deploy CubeSats above the ISS using a cargo resupply module.

Nanoracks Interchangeable CubeSat Launcher (Previously E-NRCSD)

The Nanoracks Interchangeable CubeSat Launcher (NICL) is a system to deploy CubeSats into orbit above the ISS by using the Northrop Grumman Cygnus ISS Cargo Resupply vehicle. The first mission to use the ENRCSD was on the OA-6 mission in March 2016; the updated E-NRSD design (NICL) was scheduled to have its first flight in March 2023, however the geopolitical situation between Ukraine and Russia has impacted the conops that would have enabled this demonstration. Specifically, the ISS program currently will not allow the Cygnus mission to boost above station to deploy CubeSats, so the NICL will be delayed until that changes.

In the past, up to 36U of CubeSats in any form factor up to 16U could be deployed above the ISS with each Cygnus mission. CubeSats are installed in the Nanoracks deployer and mounted externally to the Cygnus vehicle before launch. They remain external to the ISS for the duration of time that Cygnus is attached to the station, so the associated space environment should be taken into consideration when designing spacecraft (e.g., survival of spacecraft/components in extreme temperatures, and unexpected behaviors or annealing of materials that undergo long durations of thermal cycling). The deployment altitude is dependent upon the propellant margins remaining in the Cygnus but is typically 465-500 km, meeting a minimum of 45 km above the ISS altitude (52). It is hoped that this capability will return soon, allowing additional deployment options for CubeSats from the ISS.

SEOPS SlingShot

SEOPS SlingShot is a system to deploy CubeSats into orbit above the ISS using the Northrop Grumman Cygnus ISS Cargo Resupply vehicle. The first mission to use the SlingShot was in 2019. SlingShot can fly up to 72U of CubeSats per Cygnus mission; the largest CubeSat form factor it can fly is 12U. This deployment method differs from the ENRCSD in that the satellites and their dispensers are flown to the ISS as pressurized cargo on a resupply mission. Astronauts remove the satellites and install the dispensers onto the Cygnus Passive Common Berthing Mechanism (PCBM) just prior to Cygnus' departure from the station. Once Cygnus departs the ISS, it raises to an altitude of approximately 500 km and deploys the CubeSats (53). As these CubeSats are hosted in a different location and manner than the ENRCSD CubeSats, it is possible

for Cygnus to carry CubeSats in both locations on a single mission. Due to the geopolitical situation in Europe, this capability is also on hold with hopes that it will return in the future.

10.7 On the Horizon

10.7.1 Launch Brokering and Integrators

From a launch broker perspective, several companies have developed online booking systems for launches similar to web-based airline ticket platforms. Some companies, including SpaceX, even accept credit card payment options for launch services (50). The premise is that you click on your preferred destination and timeline and the website provides you with launch options. As the supply of launches increases, there will most likely be an increase in demand for both brokers and integrators.

10.7.2 Launch

As discussed in the launch section above, there are always several new launch vehicles in development. The number continues to grow every year, and how many become realized remains to be seen.

10.7.3 Deployment

There are several emerging capabilities for SmallSat deployment, including CubeSat dispensers, SmallSat separation systems, and orbital maneuvering and transfer vehicles. Many companies, such as Rocket Lab and Exolaunch are commercializing deployment systems intended to function primarily or exclusively on their space vehicles (54).

10.8 Summary

A wide variety of integration and deployment systems exist to provide access to space for small spacecraft. While leveraging excess launch vehicle performance will continue to be profitable into the future, dedicated launch vehicles and new integration systems for small spacecraft are becoming popular. Dedicated launch vehicles take advantage of rapid integration and mission design flexibility, enabling small spacecraft to dictate mission parameters and schedule. New brokering and integration services will greatly increase the mission envelope of small spacecraft riding as secondary spacecraft. Advanced systems used to rideshare to dedicated orbits can increase mission lifetime, expand mission capabilities, and enable orbit maneuvering for smaller spacecraft. In the future, these technologies may yield exciting advances in space capabilities. The expanding popularity of OMVs are bolstering these new opportunities.

These emerging launch capabilities associated with SmallSat structure cost efficiencies will continue to flourish in the SmallSat market, allowing more countries and new operators to procure, manufacture, and launch their own SmallSat systems. These new launch providers will further increase the launch cadence of SmallSats. The previous few years have shown an increase in the number of available launch vehicles dedicated to small spacecraft. Additionally, the CubeSat Design Specification (CDS) has been revised to include the nanosatellite classification to 12U (6), which has led to the design of dispensers that can be accommodated on a variety of launch vehicles. Regardless of the evolution of the CDS, the dispenser and bus market are symbiotic and seems to be expanding.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email.

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Chapter Glossary

(API)	Application Programming Interface
(ASGS)	ASRF SmallSat Ground Station
(ASRF)	Atmospheric Sciences Research Facility
(AWS)	Amazon Web Services
(C-STS)	Celestia Satellite Test & Simulation BV
(C2)	Command & Control
(CCSDS)	Consultative Committee for Space Data Systems
(CFDP)	CCSDS File Delivery Protocol
(CDMA)	Code Division Multiple Access
(CPAW)	Collection Planning and Analysis Workstation
(CRC)	Cooperative Centre
(CS)	Commercial Services
(CSP)	CubeSat Space Protocol
(DIU)	Department of Defense's Defense Innovation Unit
(DLR)	German Aerospace Center
(DoD)	Department of Defense
(DSN)	Deep Space Network
(DSOC)	Deep Space Optical Communications
(DSS-17)	Deep Space Station-17
(DTE)	Direct-to-Earth
(DVB-S2)	Digital Video Broadcast Satellite Second Generation
(EDUs)	Engineering Development Units
(EGSE)	Electrical Ground Support Equipment
(EIRP)	Effective Isotropic Radiated Power
(ESA)	European Space Agency
(ESOC)	European Space Operations Centre
(FCC)	Federal Communications Commission
(FDMA)	Frequency Division Multiple Access
(FEP)	Front End Processors
(FFRDCs)	Federally Funded Research and Development Centers
(GBPA)	Ground-Based Phase Array
(GEO)	Geosynchronous Equatorial Orbit
(GNSS)	Global Navigation Satellite System
(GPS)	Global Positioning System
(GSOC)	Global Security Operations Center
(GSFC)	Goddard Space Flight Center
(GUI)	Graphic User Interface
(HEO)	Highly Elliptical Orbit
(HF)	High Frequency

(HPAs)	High-Power Amplifiers
(I&T)	Integration and Test
(IARU)	International Amateur Radio Union
(IC)	Intelligence Community
(IDL)	Interactive Data Language
(IMEI)	International Mobile Equipment Identity
(INCOSE)	International Council on Systems Engineering
(INNOVA)	IN-orbit and Networked Optical Ground Stations Experimental Verification Advanced Testbed
(ISAN)	Integrated Space Access Network
(ISO)	International Organization for Standardization
(ISS)	International Space Station
(ITOS)	Integrated Test and Operations System
(ITU)	International Telecommunications Union
(JPL)	Jet Propulsion Laboratory
(KSAT)	Kongsberg Satellite Services AS
(KSC)	Kennedy Space Center
(LADEE)	Lunar Atmosphere and Dust Experiment Explorer
(LCRD)	Laser Communications Relay Demonstration
(LDT)	Lowell Discovery Telescope
(LEOP)	Launch and Early Orbit Phase
(LLCD)	Lunar Laser Communications Demonstration
(LNA)	Low-Noise Amplifier
(MCS)	Mission Control Software
(MEO)	Medium Earth Orbits
(MOC)	Mission Operations Center
(MSFC)	Marshall Space Flight Center
(MSPA)	Multiple Spacecraft Per Aperture
(NEN)	Near Earth Network
(NSN)	Near Space Network
(NICT)	National Institute of Information and Communications Technology
(NIMO)	Networks Integration Management Office
(NIST)	National Institute of Standards and Technology
(NORAD)	North American Aerospace Defense Command
(NTIA)	National Telecommunications and Information Administration
(OCTL)	Optical Communications Telescope Laboratory
(OGS)	Optical Ground Stations
(PNT)	Position Navigation and Timing
(PPM)	Pulse Position Modulation
(PPP)	Public-Private Partnership
(PPS)	Precise Positioning System

(RBAC)	Role Based Access Control
(R&D)	Research and Development
(RF)	Radio Frequency
(RTE)	Real-Time Earth
(RTSR)	Real Time Space Relay
(SA)	Single Access
(SDR)	Software Defined Radio
(SETH)	Science Enabling Technology for Heliophysics
(SFCG)	Space Frequency Coordination Group
(SMA)	S-band multiple access
(SN)	Space Network
(SNSPD)	Superconducting Nanowire Single Photon Detector
(SOC)	Science Operations Center
(SPD-5)	Space Policy Directive 5
(SSBV)	Satellite Services B.V.
(SSC)	Swedish Space Corporation
(SWaP)	Size, Weight, and Power
(TDMA)	Time-Division Multiple Access
(TDRS)	Tracking and Data Relay Satellites
(TDRSS)	Tracking and Data Relay Satellite System
(TLE)	Two-Line Element set
(TNC)	Terminal Node Controller
(TNO)	The Netherlands Organization
(TOGS)	Transportable Optical Ground Station
(TT&C)	Telemetry, Tracking and Control
(UHF)	Ultra-High Frequency
(USRP)	Universal Software Radio Peripheral
(VHF)	Very high frequency
(VICTS)	Variable Inclination Continuous Transverse Stub
(VMs)	Virtual Machines
(WFF)	Wallops Flight Facility
(XTCE)	XML Telemetric and Command Exchange

11.0 Ground Data Systems & Mission Operations

11.1 Introduction

The ground segment is a critical part of the end-to-end science data return, and it includes all the ground-based elements that are used to collect and disseminate information from the satellite to the user (Figure 11.1). The primary elements of a ground system are summarized in Table 11-1.

There are exciting changes in the government and commercial sector ground stations and services, and the shifting synergy between these. From its inception in 1958, whenever NASA needed to receive data from one of its Earth observing satellites or talk to its astronauts in orbit, it used equipment and services it had needed to develop and build itself. Over time, commercial enterprise acquired the proficiencies necessary to reliably and securely communicate with objects in low-Earth orbit (LEO), services NASA is now pursuing to purchase as any other near-Earth space customer.

Figure 11.1: Functional relationship between the space segment, ground segment and final user for a small satellite mission. Credit: NASA.

The agency combined NASA's Near Earth Network (NEN) and NASA's Space Network (SN) into NASA's Near Space Network (NSN) in October of 2020. To support the commercialization initiative, NASA plans to have increased reliance on industry-provided communications services for missions close to Earth by 2030 (59). As of 2024, commercial providers do not service the Sun-Earth Lagrange Points or Deep Space, thus the Deep Space Network (DSN), and large NSN assets (18 m) continue to play a critical and needful role in returning science data from these regions for Heliophysics, Astrophysics and Planetary Science directorates.

Table 11-1: Primary Elements of a Ground System	
Element	**Function**
Ground Stations	Telemetry, tracking, and command interface with the spacecraft
Ground Networks	Connection between multiple ground elements
Control Centers	Management of the spacecraft operations
Remote Terminals	User interface to retrieve transmitted information for additional processing

The NSN provides Direct-to-Earth (DTE) services via a global system of commercial and NASA-owned ground stations that provide line of sight communications and tracking services to missions ranging from low-Earth orbit and extending to Sun-Earth Lagrange Points 1 & 2. These services

are augmented by Space Relay services via relay satellites in geosynchronous orbit.

The ground segment design can depend on several factors which may include, but are not limited, to the following:

- Data volume to satisfy mission requirements
- Location of the ground assets relative to mission orbit parameters
- Budget limitations
- Distribution of the team
- Affiliation of who controls the spacecraft (federal vs. non-federal users)
- Regulatory requirements
- Latency requirements

The ground system is responsible for collecting and distributing the most valuable asset of the mission: the data. Using the proper ground system is key to mission success.

All small satellites use some form of a ground segment to communicate with the spacecraft, whether it be hand-held radios using an amateur frequency, or a large dish pulling down data on a non-federal or federal frequency. The commercial marketplace for Telemetry, Tracking and Commanding (TT&C) services continues to expand and has reached a maturity to enable commercialization of Direct-to-Earth (DTE) radio frequency communications services. NASA is encouraging a growing commercial market by leveraging commercial capabilities to increase efficiency and robustness of ground networks. In addition, NASA plans to enhance its communications capabilities to provide near-continuous communications support to the Artemis lunar missions through Communication Relay and Navigation services in Lunar space.

11.2 Ground Systems Architecture

A typical small satellite mission has the following elements within the ground system architecture:

- Ground Station Terminal: Transmitter and receiver or transceiver at the ground station to transmit and receive information, including related hardware such as antennas. These may be in a Radio Frequency (RF) or in an optical wavelength.
- Mission Operations Center (MOC):
  - o Commands the spacecraft
  - o Monitors spacecraft performance
  - o Requests and retrieves data as necessary
- Science Operations Center (SOC):
  - o Generates and disseminates science data products
  - o Determines science operations to be relayed to the MOC
- Ground Station Data Storage and Network:
  - o Provides live connectivity to a MOC for commands and telemetry
  - o Temporarily stores data to be retrieved by the MOC and/or SOC

Figure 11.2 shows a generic small satellite ground architecture that uses NASA's Near Space Network (NSN) for nominal ground passes and the NASA Space Network (SN) for low-latency messaging.

In this architecture, the MOC is responsible for all communication to and from the spacecraft,

while the SOC and engineering teams can work both directly through the MOC to process commands. This is especially helpful during commissioning and troubleshooting instances where the engineering team needs direct access to the flight system. This architecture also provides a
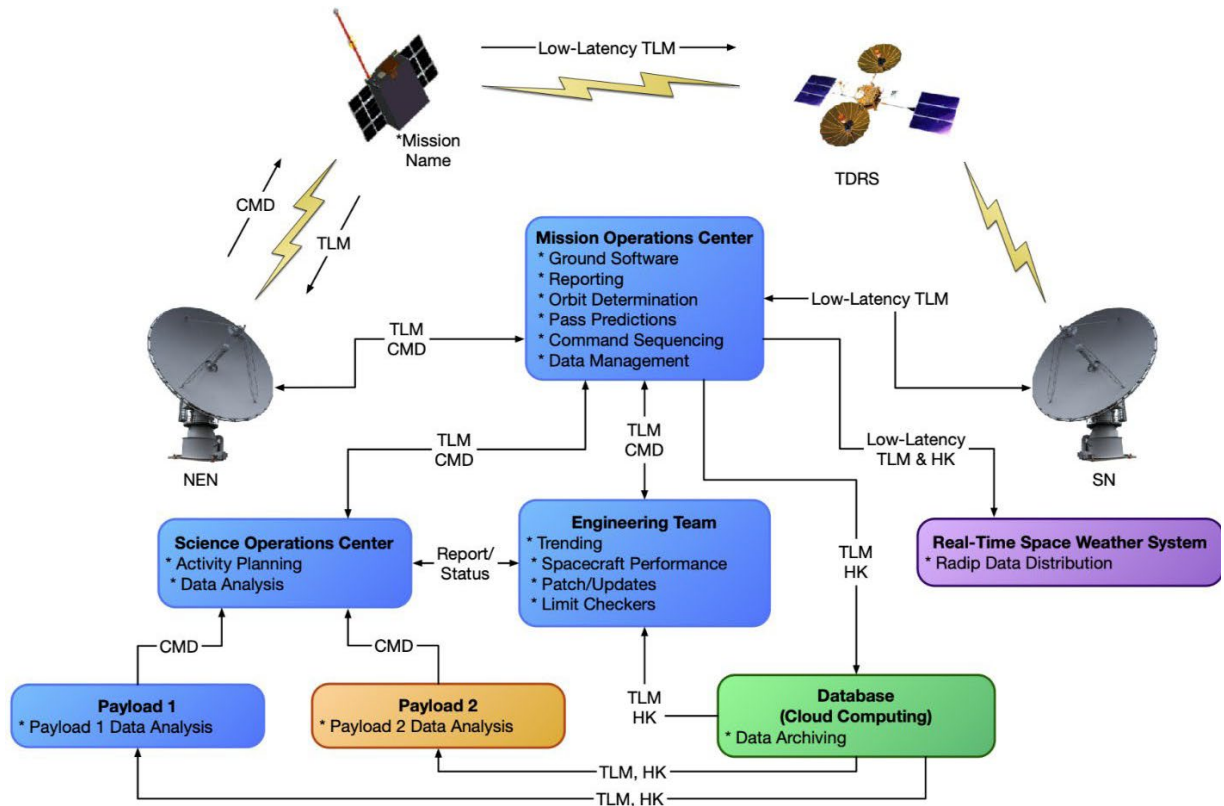
Figure 11.2: Example of a ground system architecture for a small satellite using NASA's Near Space Network. Credit: NASA.

separate database generated from the MOC of telemetry and housekeeping data that is accessible to stakeholders.

11.2.1 Types of Communication Infrastructures

Communications services may be either Direct-to-Earth (DTE) or augmented by space relay. DTE ground stations provide direct point-to-point access with antennas at ground stations which are strategically located and equipped with telemetry, command, and tracking services. DTE antennas for NASA small satellites are high gain parabolic dish antennas used to support S, X, and Ka bands, while some universities still use parabolic or UHF Yagi antennas. DTE ground stations could also incorporate phased array antenna systems or equipment for optical communications. The DTE services are especially effective for missions needing frequent, short-duration contacts with high data throughput. They are also capable of handling longer latency durations due to orbital dynamics and station visibility.

Space relay services involve an intermediate satellite that communicates with a ground station on the Earth's surface. Relay communication satellites for low-Earth orbit spacecraft can be in Geosynchronous Equatorial Orbit (GEO), about 36,000 km from Earth, or in low-Earth orbit. Relays are essential for providing communication and tracking when direct-to-ground communications are not feasible due to physical asset visibility constraints. It is common for a low-Earth orbit spacecraft to only be in a DTE ground station's line of sight for a portion of the orbit. The addition of space-based relay assets can provide missions with full-time coverage and continuous access to

communication and tracking services. They are most useful for missions that need continuous coverage, low latencies, and coverage of launch, critical events, or emergencies.

Communication with DTE ground stations can achieve much higher data rates than what is possible for space-based relays. When considering a GEO relay satellite, it can be ten times the distance from the low-Earth orbit spacecraft than the DTE ground station. With communication propagation losses being a function of the reciprocal of the distance squared, the same communications system can achieve orders of magnitude higher data rates with the DTE ground station. Achieving comparative data rates for a relay system would require a significant increase in power. The current low-Earth orbit relays have hardware limitations that permit data rates of 9.6 kbps or less, which is low relative to SmallSats being able to achieve 3 Mbps or more with DTE ground stations.

11.3 Frequency Considerations

The spacecraft transceiver and ground station need to be on a coordinated frequency to communicate. Selecting transmit and receive frequencies are a critical part of the spacecraft communications system design process. Frequencies are divided into different bands as shown in Table 11-2. See a list of supported frequencies per ground station in their specific sections.

Typical bands considered for small satellites and therefore ground stations are Ultra High Frequency (UHF), S, X, and Ka. UHF was the band of choice for early small satellites, but in recent years, there has been a shift to S and X and Ka. A ground station needs to maintain antennas and receivers such that the ground receive matches the space segment's transmit frequency and vice versa. Since Transmit (Tx) and Receive (Rx) have different key drivers and requirements, many ground stations are dual or tri-band.

Table 11-2: Frequency Bands	
Band	**Frequency**
HF	3 to 30 MHz
VHF	30 to 300 MHz
UHF	300 to 1000 MHz
L	1 to 2 GHz
S	2 to 4 GHz
C	4 to 8 GHz
X	8 to 12 GHz
Ku	12 to 18 GHz
Ka	27 to 40 GHz
V	40 to 75 GHz
W	75 to 110 GHz
mm	110 to 300 GHz

Ground Station Receive (Spacecraft Return, Telemetry)

Ground station receive frequencies are mostly S and X band from a LEO/ GEO orbit, and X and Ka band from deep space. Ka band has been implemented for transmit and is NASA's desired band for future small satellite missions. This shift has been driven by higher data return demands and frequency control. The higher frequencies permit more data to be transmitted over a given period but require more stringent pointing. UHF is appealing to some universities, due to the lower cost of hardware for both the spacecraft and ground station, good link margins, and more omni-directional pattern capability with the spacecraft but yields lower data rates and has a higher probability for interference. Higher frequencies provide wider bandwidths, and the matching antennas have narrower beamwidths or are arrayed for a higher gain, thus more stringent pointing is required.

Ground Station Forward (Spacecraft Commanding)

The key driver to successfully command a satellite from the ground, is the ability to reach it. The most critical period is after a satellite's release from the launch vehicle, at which point the satellite does not yet have full control over its attitude, thus the most important thing is a wide beamwidth for the spacecraft receiving antenna(s) in the selected frequency. For this reason, ground stations are designed with more power and Low Noise Amplifiers (LNA) to counter the low gain, ideally omni-directional single patch receive antennas in the lower frequency bands.

11.3.1 Frequency Selection: Link Budget

Calculating the RF link budget is the first step when designing a telecommunications solution. It is a calculation of the end-to-end performance of the communications link with the constraint of maintaining a required link margin. Maintaining a 3 dB link margin is adequate for data return from a satellite in low-Earth orbit at a slant range of 1,500 km. Usually commanding to a Near Earth orbit has plenty of margin because of the high power and aperture size of the ground station, and the lower required data rate on the account of the commands' low volume. When considering deep space communication, a 3 dB link margin is desired, but for distant spacecraft, such as New Horizons at 7 billion kilometers from Earth, 1 dB or less margin may be all that is practically possible. The budget calculation adds and subtracts all the power gains and losses that a communication signal will experience within the system. Factors such as uplink amplifier gain and noise, transmit antenna gain, slant angles and corresponding free space loss, satellite transceiver noise levels and power gains, receive antenna and amplifier gains and noise, cable losses, and atmospheric attenuation are considered. There is a duality to frequency effects: free space loss over the same range is less for lower frequencies; however, the wavelength is much smaller for higher frequencies, thus a same size ground aperture provides a much higher Gain over temperature (G/T). On the spacecraft end, a multi-element high-gain Ka-band antenna array for example fits in the palm of a hand. For high volume data return, which is where communications bottlenecks occur, higher frequencies are desirable – all the way up to optical wavelengths at 1550 nm (see section 11.10.1, *Free Space Optical Communications*).

11.3.2 Frequency Licensing

RF communication frequencies are intentionally protected. Within each frequency band there are government and non-government designations amongst the frequencies. Some frequencies are government use only, others are non-government only, and some are shared. Government bodies that regulate the frequency usage in the United States are the Federal Communications Commission (FCC) and the National Telecommunications and Information Administration (NTIA). Other countries may have their own national governing bodies, and all national bodies around the world must coordinate with the International Telecommunications Union (ITU), which is the governing body at the international level. The FCC is responsible for issuing communications licenses to non-government users and the NTIA handles government users. Licenses are required for both the satellite and ground station to transmit on a designated frequency or frequencies. It is becoming more common for small satellites to use multiple bands. For example, some missions have used UHF for uplink and S- band for downlink, while others have used S-band for uplink and X-band for downlink. Some of the non-government frequencies are dedicated for amateur usage.

Early university small satellites relied heavily on the use of amateur frequency bands. In recent years, there has been movement by the International Amateur Radio Union (IARU) and the FCC to significantly limit the use of amateur frequencies for small satellites. Those interested in using these frequencies are expected to first communicate their intention with the IARU and obtain a coordination letter prior to submitting an application with the FCC. It is recommended that missions with a new communication system design apply with the FCC or NTIA once an operations concept and a spacecraft design are defined, in order to verify a proper communications approach and associated hardware has been selected. Missions using a legacy communications approach can typically wait until they have been given a launch manifest. The licensing process can take several months and needs to be completed prior to launch. Some of the processing time is associated with the FCC and NTIA having to also coordinate with the ITU. Both the FCC and ITU are working to implement more streamlined small satellite licensing options. Such improvements will be necessary as constellations of small satellites become more prevalent.

11.4 Ground Segment Services

Ground segment services may include the below four categories. The NSN is a full-service ground station network and offers all four major service categories. Not all commercial services offer all services.

1) Mission Integration – this includes development of service agreements, interfaces, documentation, support of reviews, etc.

2) Mission Planning and Scheduling – this includes performing link and loading analyses, supporting service requests, and generating and implementing operational schedules.

3) User Mission Data Transfer – this includes primarily spacecraft forward command and return telemetry data.

4) Position, Navigation and Timing (PNT) – this includes navigation.

Position information (4) is critical for commanding the spacecraft (3). Commanding may be scripted by the mission and is actuated through ground services. Challenges are usually associated with the initial satellite-to-ground station link closure. Typically, two-line elements (TLE) or state vectors are established and shared by the launch provider after deployment. This information can be used to create an initial orbit solution for ground station antenna pointing. Low-Earth orbit missions can use North American Aerospace Defense Command (NORAD) TLE data (see https://www.space-track.org) for satellite location. However, it could take up to a week or more for NORAD to add the new object to their tracking list. This process could be delayed further if multiple spacecraft are ejected in close proximity, and it may not be clear which NORAD element set corresponds to which spacecraft. It is not uncommon to spend weeks attempting contact with different NORAD-tracked objects until the correct one is found. The position prediction accuracy based on the NORAD TLE also diverges over time and a new TLE will be needed to maintain data link. This is typically not an issue since the TLE is updated regularly, but on-board Global Positioning System (GPS) data (if equipped) can help determine the orbital parameters for the ground station to define latest orbital parameters.

Another method is to locate the satellite as it rises from the horizon. Ground station operators can point a directional antenna 5-10 degrees above the horizon to detect the satellite and synchronize with the radio. Most antenna tracking software will commence automatic tracking after the initial acquisition is successful. A half-duplex or full-duplex system could make a difference as well. Program track instead of auto-track is used for half-duplex. With a full-duplex system, the ground antenna attempts to acquire the downlink first. Predicts (NORAD or state vectors) are still used to initially acquire the spacecraft. If the predicts are off, the antenna can initiate a mechanical scan to increase the search area. Once the downlink is acquired, the ground antenna can auto-track and automatically point at the satellite for the duration of the pass. Additional passes are scheduled during spacecraft and payload commissioning. Table 11-3 describes NSN's transport and tracking capabilities.

Table 11-3: NSN Interfaces and Capabilities		
Interface/ Capability[1]	Direct to Earth	Space Relay
Terrestrial Link Data Transport Capabilities		
Data Storage [1]	Station Storage: 5-30 days Cloud-based: Mission-driven	7 days
Network Data Rate [1]	Mission-driven (up to 1.2 Gbps)	
SLE Protocols	F-CLTU, EF-CLTU (Forward) RAF, RCF, ROCF (Return)	
SLE Versions Supported [2]	CCSDS 910.4, CCSDS 911.1, CCSDS 911.2, CCSDS 911.5, CCSDS 912.1, CCSDS 912.11, CCSDS 912.3, CCSDS 913.1	
Offline-Data Transfer	CFDP, SFTP	
Security	Trusted Networks (Access Controls, Firewalls, Authentications, etc.)	
Spacecraft Navigation Tracking Capabilities		
Radiometric Tracking Services [1]	Tone Ranging 1-way or 2-way Doppler Antenna Angle Data	Spread Spectrum Ranging 1-way or 2-way Doppler Antenna Angle Data
Radiometric Measurement Accuracy [1]	Range: S-band: < 5 meters, 1σ Doppler (Range-Rate): S-band 1-way: ≤ 30 mm/s, 1σ S-band 2-way: ≤ 15 mm/s, 1σ X-band 1-way: ≤ 7 mm/s, 1σ Ka-band 1-way: ≤ 2 mm/s, 1σ Antenna Angles: S: 0.03°, X: 0.05° Ka: 0.01° (auto), 0.05° (program)	Range: ≤ 2.73 meters, 1σ Doppler (Range-Rate): 1-way ≤ 1.55 mm/s, 1σ 2-way ≤ 3.1 mm/s, 1σ Antenna Angles: ≤ 0.1°
Radar Tracking Service Bands	C-band (5.4-5.9 GHz) Single Object X-Band (10.499 GHz) Multi Object	N/A
Radar Tracking Loop Gain (dB)	C-Band: 212-245 (227 Typical) X-Band: 246 (nominal)	
Other [1]	Ground Antenna Slew Rate: Azimuth and Elevation: ≥ 10°/sec (10°/sec^2) * Train: ≥ 5°/sec (5°/sec^2) * WS1 18-m system ≥ 2°/sec (1°/sec^2)	Time Transfer Measurement: User Spacecraft Clock Calibration System: ≤ ±5 µs Return Channel Time Delay: ±25% of a bit period

[1] Services and performance (Data Rates, EIRP, G/T, etc.) are not uniform across assets.
[2] Additional capabilities above those listed could be supported as well.
[3] NASA may consider adding technologies not currently on its roadmap.
[4] 2nd and 3rd Generation TDRS only.

Another critical time in the life of a spacecraft is commissioning; either commissioning of the spacecraft bus, or commissioning of science instruments, including in-space calibration. During

commissioning phases, additional time and support personnel are typically scheduled (1).

11.4.1 Ground Networks – NASA and Partners

The ground stations, MOC, SOC, and the supporting infrastructure connecting them together, make up a ground network. Ground station antenna dish diameters, LNAs, frequency feeds, station gain over temperature (G/T) requirements are carefully selected for each network and are optimized for targeted ranges. NASA's NSN ground network provides services to satellites up to 2 million km range from Earth; NASA owns and JPL maintains the DSN for missions beyond two million km, including planetary.

At NASA's Goddard Space Flight Center, the Exploration and Space Communications (ESC) projects division oversees the operations, maintenance and advancement of the Space Communications and Navigation (SCaN) program office's NSN. Operating at a high-level of reliability and proficiency, the NSN provides communications and navigation services for missions within 2 million kilometers of our planet, bringing down an average of almost 30 Terabytes of critical data daily. Through space relays and ground-based assets, The NSN provides data delivery and satellite tracking services, empowering new discoveries about the universe and our home planet. JPL is responsible for managing and maintaining the DSN.

NASA Near Space Network

"The newly established NSN is more than just an aggregation of the NEN's and SN's space-based technologies, ground stations and antennas; it's the network through which NASA and other space users will now arrange for support services for their near-Earth missions. Critically, those support services may be provisioned through government or commercial network assets in a way that is seamless to users—a cornerstone in SCaN's effort to incorporate increasing levels of commercial service while ensuring mission needs are met." (2)

The NSN provides direct-to-earth telemetry, commanding, ground-based tracking, and data and communications services to a wide range of customers. The network consists of NASA, commercial, and partner S-band, X-band, and Ka-band ground stations supporting spacecraft in low-Earth orbit, GEO, Highly Elliptical Orbit (HEO), Lunar orbit, and Lagrange point L1/L2 orbit up to one million miles from Earth. The NSN supports multiple robotic and launch vehicle missions with NASA-owned stations and through cooperative agreements with interagency, international, and commercial services. Table 11-4 shows the radio frequencies that the NSN supports via the NTIA.

Table 11-4: NSN Supported Radio Frequencies and Bandwidths		
Band	**Function**	**Frequency Band (MHz)**
S Uplink	Earth to Space	2,025 – 2,110
X Uplink	Earth to Space	7,190 – 7,235 (Two NEN sites to 7,200)
S Downlink	Space to Earth	2,200 – 2,300
X Downlink	Space to Earth, Earth Exploration	8,025 – 8,400
X Downlink	Space to Earth, Space Research	8,450 – 8,500
Ka Downlink	Space to Earth	25,500 – 27,000

A comprehensive list of Forward and Return capabilities per frequency are in Table 11-5. Systems are compliant with most CCSDS recommendations. The NSN consists of geographically-

dispersed ground stations operated by NASA and its commercial partners (Figure 11.3).

Government

- NASA's Alaska Satellite Facility, Fairbanks — Supports: S/X Band — Assets: 11.3m, 11m, 9.1m

- NASA's Kennedy Uplink Station — Supports: S-band - Assets: 6.1m

- NASA's Ponce de Leon Station — Supports: S-band - Assets: 6.1m

- NASA's Wallops Ground Station (GS), Virginia — VHF, S/X Band — Assets: 11m/5m

- NASA's White Sands GS, New Mexico — Supports: VHF, S/Ka Band — Assets: 18.3m

- NASA's White Sand Complex, New Mexico — Supports VHF, S/Ka Band — Assets: 11m

- NASA's McMurdo Ground Station, Antarctica — Supports: S/X Band — Assets: 10m

- Fairbanks Command and Data Acquisition Station (NOAA partnership), Gilmore

Figure 11.3: NSN Global Ground Station Locations. Credit: NASA

Creek, Alaska

Commercial

- KSAT Singapore — Supports: S/X Band — Assets: 9.1m

- KSAT Svalbard, Norway — Supports: S/X Band — Assets: 11.3m/11.3m/13m

- KSAT TrollSat, Antarctica — Supports: S/X Band — Assets: 7.3m/7.3m

- KSAT Punta Arenas – Supports: S/X Band — Assets: 11.5m

- KSAT Inuvik -- Supports: S/X Band — Assets: 13m

- SANSA Hartebeesthoek, South Africa — Supports: S/X Band — Assets: 12m/10m

- SSC Kiruna, Sweden — Supports: S/X Band — Assets: 13m/13m

- SSC Santiago, Chile — Supports: S Band — Assets: 9m/12m/13m

- SSC Space US North Pole, Alaska — Supports: S/X Band — Assets: 5m/7.3m/11m/13m

- SSC Space US Dongara, Australia — Supports: S/X Band — Assets: 13m

- SSC Space US South Point, Hawaii — Supports: S/X Band — Assets: 13m/13m

Table 11-5: NSN Direct to Earth Command and Telemetry Capabilities per Frequency		
Interface/ Capability[1]	Direct to Earth	Space Relay
Forward (Command) Communications		
Frequency Bands (Near-Earth Use)	S-band: 2025-2110 MHz X-band: 7190-7235 MHz	S-band: 2025-2110 MHz Ku-band: 13.775 GHz Ka-band: 22.55-23.55 GHz [4]
Maximum Bandwidth	S-band: 5 MHz X-band: 10 MHz	S-band: 6 MHz Ku-band: 50 MHz Ka-band: 50 MHz [4]
Forward Max Data Rate [1,2] (prior to encoding)	S-band: 5 Mbps X-band: 5 Mbps	S-band MA: 300 Kbps S-band SA: 4.2 Mbps Ku-band: 50 Mbps Ka-band SA: 50 Mbps [4]
Antenna System EIRP (dBW) [1]	S-band: 51-81 (56 Typical) X-band: 85-86	S-band MA: 42 [4] S-band SA: 48.5 [4] Ku-band SA: 48.5 [4] Ka-band SA: 63 [4]
Modulation [2,3]	PM, FM, PCM, PCM/PM, PCM/PSK/PM, BPSK, QPSK, OQPSK, UQPSK	Spread spectrum: BPSK or UQPSK Non-spread: BPSK, QPSK, OQPSK, PCM/PM, or PCM/PSK/PM
Encoding [2,3]	Uncoded, or LDPC ½ or 7/8	Uncoded, Rate ½ Conv., Reed-Solomon, Concatenated (½ Conv. + RS), LDPC ½ or 7/8
Polarization	Circular (LHC, RHC)	Circular (LHC, RHC) (LHC only for MA services)
Return (Telemetry) Communications		
Frequency Bands (Near-Earth Use)	S-band: 2200-2290 MHz X-band: 8025-8400 MHz X-band (SRS): 8450-8500 MHz Ka-band: 25.5 – 27 GHz [4]	S-band: 2200-2290 MHz Ku-band: 15.0034 GHz Ka-band: 25.25 – 27.5 GHz [4]
Maximum Bandwidth	S-band: 5 MHz X-band: 375 MHz X-band (SRS): 10 MHz Ka-band: 1500 MHz	S-band (MAR & SAR): 6 MHz Ku/Ka-band: 225 MHz [4] Ka-band (Wide): 650 MHz [4]

Return Max Data Rate [1,2] (prior to encoding)	<u>Rates will vary – examples:</u> S-band: 2.2 Mbps (PACE) X-band: 220 Mbps (ICESat-2) X-band (SRS): 13.1 Mbps (IRIS) Ka-band: 3.5 Gbps (NISAR)	S-band MA: 1 Mbps S-band SA: 14.1 Mbps Ku/Ka-band: 600 Mbps [4] Ka-band (Wide): 1200 Mbps [4]
Antenna System G/T (dBW) [1]	S-band: 19.1-29.6 (21 Typical) X-band: 30.5-37.8 (32 Typical) Ka-band: 38-45 (41.3 Typical)	S-band MA: 3.2 (for LEO) S-band SA: 9.5 (for LEO) Ku-band: 24.4 (for LEO) Ka-band: 26.5 (for LEO) [4]
Demodulation [2,3]	PM, FM, PCM, PCM/PM, PCM/PSK/PM, BPSK, QPSK, OQPSK, AQPSK, SQPN, 8PSK	Spread spectrum: BPSK or UQPSK Non-spread: BPSK, QPSK, OQPSK, PCM/PM, or PCM/PSK/PM
Decoding [2,3]	Uncoded, Rate ½ Conv. and/or Reed-Solomon, LDPC ½ or 7/8, or Turbo Rate ½	Uncoded, Rate ½ Conv., Reed-Solomon, Concatenated (½ Conv. + RS), LDPC ½ or 7/8, Rate 7/8 TPC
Polarization	Circular (LHC, RHC)	Circular (LHC, RHC) (LHC only for MA services)

[1] Services and performance (Data Rates, EIRP, G/T, etc.) are not uniform across assets.
[2] Additional capabilities above those listed could be supported as well.
[3] NASA may consider adding technologies not currently on its roadmap.
[4] 2nd and 3rd Generation TDRS only.

While NASA's NSN is often reserved for NASA-funded missions, other ground network options exist for non-government-funded satellite operators. One common option, especially amongst amateur operators, is to take advantage of the UHF and VHF amateur network around the world.

The NSN is exploring how to provide higher data rates for CubeSat missions with techniques such as Digital Video Broadcast Satellite Second Generation (DVB-S2). Higher data rates either increase science return or reduce the number of minutes per day of required ground station contacts. Higher data rates also enable mother-daughter small satellite constellations, where the mother spacecraft handles the communication with Earth for multiple daughter spacecraft. Functions such as Multiple Satellite per Aperture (MSPA) are planned to be implemented on the Lunar Exploration Ground Sites (LEGS) mission (see the State-of-the-Art section).

The NSN facilitates Commercial Services (CS) and negotiated a bulk-buy discount for all NASA missions. This allows for contacts on the NSN Contractor/University Operated and CS apertures to be at no-cost for NASA missions. The NSN does schedule CS in accordance with NASA mission-defined priority. The Networks Integration Management Office (NIMO) at NASA GSFC is the liaison for customers that wish to use NSN services. NIMO has a variety of services and capabilities available and can coordinate support from providers throughout NASA, other US agencies, US commercial entities, and foreign governments. Some of the services that NIMO can provide include:

- Requirements Development
- Communications Design Support & Guidance
- Optical Communications Analysis
- Network Feasibility Analysis
- Spectrum Management

- RF Compatibility Testing
- Launch Support

Network Feasibility Analysis includes determining NSN station loading as a function of the mission's priority and determining the availability of planned stations for the contacts requested. Prior to the mission deployment, the NSN commits to providing the requested stations and contact time as outlined in the network feasibility analysis.

For new customer mission service requests please fill out the NSN Service Inquiry Form at: http://go.nasa.gov/NSNServiceInquiry.

If interested in more information on using the Near Space Network (NSN), please also refer to https://esc.gsfc.nasa.gov/projects/NSN.

NASA Deep Space Network

The DSN is optimized to conduct telecommunication and tracking operations with space missions in GEO. This includes missions at lunar distances, the Sun-Earth LaGrange points, and in highly elliptical Earth orbits, as well as missions to other planets and beyond. The DSN has supported, or is currently supporting, missions to the Sun as well as every planet in the Solar System (including dwarf planet Pluto). Two missions (Voyager I and Voyager II) have reached interstellar space and still communicate with the DSN. The DSN offers services to a wide variety of mission customers, as shown in Table 11-6.

Table 11-6: DSN Customers, Mission Characteristics, Frequencies, and Services	
Customers	**Mission Phases**
- NASA - Other Government Agencies - International Partners	- Launch and Early Orbit Phase (LEOP) - Cruise - Orbital - In-Situ
Mission Trajectories	**Frequency Bands –** Includes Near-Earth and Deep Space Bands, Uplink and Downlink, Command, Telemetry, and Tracking Services
- Geostationary or GEO - HEO - Lunar - LaGrange - Heliocentric - Planetary	- S-Band (2 GHz) - X-Band (7, 8 GHz) - Ka-Band (26, 32 GHz)

DSN services include:

- Command Services
- Telemetry Services
- Tracking Services
- Calibration and Modeling Services
- Standard Interfaces
- Radio Science, Radio Astronomy and Very Long Baseline Interferometry Services
- Radar Science Services
- Service Management

Custom and tailored DSN services can also be arranged for missions and customers. DSN-provided data services are accessed via well-defined, standard data and control interfaces:

- The CCSDS
- The Space Frequency Coordination Group (SFCG)
- The ITU
- The International Organization for Standardization (ISO)
- De facto standards widely applied within industry
- Common interfaces specified by the DSN

The use of data service interface standards enable interoperability with similar services from other providers.

Figure 11.4 shows the DSN antennas and their locations. Each DSN ground station in California (United States), Madrid (Spain), and Canberra (Australia) currently as of 2021 was

Figure 11.4: DSN antennas and their locations. Credit: NASA.

operating four 34 m Beam Wave Guide antennas and one 70 m antenna. By the late 2020s, this is planned to increase to include one 70 m plus four 34 m antennas at each DSN site.

The DSN supports RF testing using the following facilities:

- Development and Test Facility (DTF-21), located near NASA Jet Propulsion Laboratory (JPL)

- Compatibility Test Trailer (CTT-22), able to come to the spacecraft site

For more information on DSN, please see:
https://www.nasa.gov/directorates/space-operations/space-communications-and-navigation-scan-program/scan-services-and-scheduling/
https://deepspace.jpl.nasa.gov/about/commitments-office/
https://deepspace.jpl.nasa.gov

Swedish Space Corporation

Swedish Space Corporation (SSC) is a global provider of ground station services, including support to launch and early operations, on-orbit Telemetry, Tracking and Control (TT&C) and data downlink, and even lunar services (see https://sscspace.com/). The SSC Infinity Network is specifically designed for constellations of small satellites in low-Earth orbits. The global network provides TT&C and data download and delivery services to SmallSat operators, and customer interfaces consist of web- based portals for pass scheduling on 5-meter and smaller antennas. SSC Infinity also uses standard configurations and standardized ground system hardware, limiting the number of mission configurations to help keep costs lower for satellite operators.

Using ground services will generally require some degree of pre-coordination (or "onboarding") between the operator and provider, which is usually done before launch. This will vary between providers but may include contracting mechanisms; frequency licensing and coordination between the operator and the provider; compatibility testing; and the sharing of mission and vehicle specific information to ensure the ground stations are properly configured for the operator to use. Once the onboarding process is complete, satellite operators can schedule passes between their satellite(s) and desired ground station(s) in advance (the time window varies for each provider). The schedules for each ground station are deconflicted based on scheduling priority, and all frequency and modulation adjustments for the satellite are completed in advance of the pass by the service provider.

KSAT^{LITE}

The baseline of Kongsberg Satellite Services AS (KSAT)'s 3.7-meter KSAT^{LITE} antennas provide X-band and S-band for downlink and S-band for uplink. KSAT operates over 100 KSAT^{LITE} antennas at 15+ ground station sites across the globe, supporting over 1.5 million passes in 2023 alone. In addition, KSAT^{LITE} offers a global Ka-band network capable of supporting missions with higher data rates. KSAT^{LITE} is an extension of the existing KSAT ground station antenna network with lower costs, increased flexibility, and improved availability and pass selection. The KSAT

Figure 11.5: 2024 KSAT^{LITE} ground network map. Credit: KSAT.

network has uniquely located polar stations in the Arctic and Antarctic regions, providing 100% availability on passes for spacecraft in polar orbit. The network also includes mid-latitude ground stations, providing access for diverse orbits and mission profiles.

Additionally, KSAT has a network of large aperture antennas that are purpose built for supporting missions in cis-lunar space and beyond. In 2023 KSAT invested in a network of three 20m LEGS-class antennas that will be placed around the globe to provide full coverage of the Moon, which will be fully operational by Q1 of 2027 (5). See Figure 11.5 for 2024 KSAT^{LITE} ground network map.

11.4.2   Ground Segment as a Service (GSaaS)

Ground Station as a Service (GSaaS) is a managed service which enables customers to communicate, downlink, & process data from their satellites/spacecrafts on as a pay-as-you go basis without needing them to build their own satellite ground stations. These services are usually scalable and use edge cloud services as an intermediate for customers data (5) (6).

AWS Ground Station

AWS Ground Station is a managed service (Figure 11.6.) that lets customers build ground segment architectures in the cloud to control their satellites, process satellite data, and scale satellite operations without having to worry about building or managing their own antenna infrastructure. Customers can stream satellite data from any of the AWS antennas to the Amazon Elastic Compute Cloud (EC2) for real-time processing or to directly store data in the Amazon Simple Storage Service (S3). Additionally, customers can easily integrate their space workloads with other AWS services in near real-time using Amazon's low-latency, high-bandwidth global network. For example, customers who downlink terabytes of data daily can easily access AWS services such as Amazon SageMaker to quickly derive useful information. Other AWS services include Amazon VPC, Amazon Rekognition, and Amazon Kinesis Data Streams. These services

Figure 11.7. AWS Ground Station Locations (2024). Credit: Amazon Web Services

Figure 11.6. GSaaS Flow Chart. Credits: NASA/Amazon Web Services.

allow operators to reduce data processing and analysis times for use cases like weather prediction or natural disaster imagery from hours to minutes or seconds. This also enables operators to quickly create business rules and workflows to organize, structure, and route the satellite data before it can be analyzed and incorporated into key applications such as imaging analysis and weather forecasting (8). A map of the AWS Ground Station antenna regions is shown in Figure 11.7.

AWS Ground Station offers 5.4-meter apertures at each of the antenna regions. AWS Ground Station provides satellite antennas direct access to AWS services for faster, simpler, and more cost-effective storage and processing of downloaded data. Frequencies and link parameters are as follows:

- S-band uplink: 2025-2120 MHz

- S-band downlink: 2200-2300 MHz with G/T of 16 dB/K

- X-band downlink: 7750-8400 MHz with G/T of 30.5 dB/K

AWS Ground Station antennas are interconnected via Amazon's low-latency, highly reliable, scalable, and secure global network backbone. As of September 2024, the AWS Cloud spans 108 Availability Zones within 34 geographic regions around the world, with announced plans for 18 more Availability Zones and 6 more AWS Regions in Mexico, New Zealand, the Kingdom of Saudi Arabia, Thailand, Taiwan, and the AWS European Sovereign Cloud.

Customers have access to multiple billing structures including (1) pay-as-you-go (PAYG) option, where customers pay only for what they use on a per minute basis; (2) Dedicated Mission Support (DMS) option, where customers would receive schedule prioritization on one antenna for their constellation; (3) or Dedicated Antenna Solution (DAS) option, where AWS collaborates with customers to design a custom solution tailored to specific mission requirements, including support for specific spectrum bands, antenna apertures, and/or locations. The PAYG, DMS, and DAS solutions can also be combined for additional capacity needs.

Scheduling: AWS Ground Station provides an easy-to-use graphical console that allows operators to reserve contacts and antenna time for their satellite communications. They can review, cancel, and reschedule contact reservations up to 15 minutes prior to scheduled antenna times. Access can be scheduled to AWS Ground Station antennas on a per-minute basis, so operators only pay for the scheduled time. They can access any antenna in the ground station network, and there are no long-term commitments.

Viasat

Viasat has delivered hundreds of ground antennas to NASA, other civilian space agencies, governments, and satellite industry partners throughout the world since the 1960s and has recently become a ground service provider. This move leverages decades of hardware engineering, with the global Real-Time Earth (RTE) network of ground terminals, complemented by a satellite relay network philosophy in GEO for low latency (near-real time) applications. A map of ground assets is shown in Figure 11.8. Viasat operates 7.3 m and 5.4 m antennas operating in 3 frequency bands (Table 11-7) (9) and plans to include 13m class antennas in the near future. The RTE ground segment service enables communications for next-generation and legacy LEO satellites using S, X, and Ka-bands. RTE offers downlinks from low megabits per second to multiple gigabits per second empowered by cutting edge software-defined radios at each site. The service includes high-speed connectivity for backhaul, real-time data streaming, and real-time monitoring of overhead passes.

Figure 11.8: Viasat RTE global ground stations. Credit: Viasat.

Viasat's fully automated system allows users their choice of machine-to-machine or manual GUI-based scheduling over a highly resilient cloud-based platform. Instantaneous pass confirmation without human intervention lowers cost and risk of human error. Data is delivered to your choice of cloud provider, operations center, or government cloud destination. Viasat's Real Time Space Relay (RTSR) service is designed to allow LEO satellite operators to send data back to Earth in real time through the ViaSat-3 network of high-capacity Ka band GEO satellites. The RTSR service provides on-demand data delivery up to 50 GB/orbit without the need for a traditional ground station, enabling continuous custody of the spacecraft. It is designed to be available over the poles, oceans, and remote areas with persistent bi-directional connectivity to simultaneously serve/support hundreds, scalable to thousands, of LEO and non-LEO users and sensor platforms without pre-coordination or complicated scheduling/de-confliction algorithms. In partnership with NASA CSP, Viasat is developing a capability suite to address the full range of space missions called the Integrated Space Access Network (ISAN). ISAN is a one stop shop for multi-band, multi-path, secure connectivity solutions for on-orbit communications with unprecedented flexibility & capacity. Viasat is integrating the RTE ground solution with its relays for on-orbit demonstrations in 2026, allowing users to select between the low latency space relays and the high throughput global ground network (9). Viasat's L-band relay solutions are also designed to provide real-time, low-data latency communications. An additional part of Viasat's NASA CSP work includes the InCommand system, which will use Viasat's global L-band network to provide real-time telemetry, tracking and command (TT&C) operations at any point in the spacecraft's orbit. InCommand works in tandem with the RTSR system, offering a robust solution for on-demand satellite command and control. This service ensures continuous, real-time communication with GEO satellite infrastructure, facilitating rapid tasking and data transmission for LEO satellites.

Table 11-7: Viasat Ground Stations						
Location	**Antenna**	**L-band Uplink**	**S-band Uplink**	**S-band Downlink**	**X-band Downlink**	**Ka-band Downlink**
Pendergrass, GA, USA (PGG)	5.4 m X-Y Full motion 5 °/sec max speed	N/A	2025-2110 MHz EIRP: 53.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 17 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 30 dB/K Simultaneous RHCP and LHCP	N/A
Guildford, UK (GDD)	5.4 m X-Y Full motion 5 °/sec max speed	N/A	2025-2110 MHz EIRP: 53.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 17 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 30 dB/K Simultaneous RHCP and LHCP	N/A
Alice Springs, AU (ASP01, ASP02)	7.3 m X-Y Full motion 6 °/sec max speed quantity 2	1755-1850 MHz EIRP: 63.4 dBW	2025-2110 MHz EIRP: 65.0 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 18 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.5 dB/K Simultaneous RHPC and LHCP
Accra, Ghana (ACC)	7.3 m X-Y Full motion 6 °/sec max speed	1755-1850 MHz EIRP: 63.4 dBW	2025-2110 MHz EIRP: 65.0 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 18 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.5 dB/K Simultaneous RHCP and LHCP
Cordoba, AR (COR)	5.4 m X-Y Full motion 5 °/sec max speed	N/A	2025-2110 MHz EIRP: 53.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 17 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 30 dB/K Simultaneous RHCP and LHCP	N/A
Öjebyn, Sweden (OJY)	7.3 m X-Y Full motion 6 °/sec max speed	N/A	2025-2110 MHz EIRP: 55.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 18 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.5 dB/K Simultaneous RHCP and LHCP
Pretoria, South Africa (PRY)	7.3 m X-Y Full motion 6 °/sec max speed	N/A	2025-2110 MHz EIRP: 55.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 18 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.5 dB/K Simultaneous RHCP and LHCP

Table 11-7: Viasat Ground Stations						
Location	Antenna	L-band Uplink	S-band Uplink	S-band Downlink	X-band Downlink	Ka-band Downlink
Obihiro, Japan (OBO)	7.3 m X-Y Full motion 6 °/sec max speed	N/A	2025-2110 MHz EIRP: 55.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 18 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.5 dB/K Simultaneous RHCP and LHCP
North Pole Alaska USA 7.3 m (NTP01)	7.3 m El-Az-Train El & Az: 15 °/s max speed Train 6 °/s max	N/A	2025 – 2120 MHz EIRP: 53 dBW Selectable RHCP or LHCP	2200–2300 MHz G/T: 19 dB/K Simultaneous RHCP and LHCP	8000–8500 MHz G/T: 32 dB/K Simultaneous RHCP or LHCP	N/A
North Pole Alaska USA (9.1 m) (NTP02)	9.1 m El-Az-Train	N/A	2042–2052 MHz EIRP 38 dBW Selectable RHCP or LHCP	N/A	8000–8500 MHz G/T: 34 dB/K RHCP	N/A
Tierra del Fuego, Argentina (USH)	7.3 m El-Az-Train El & Az: 15 °/s max speed Train 6 °/s max	N/A	2025-2110 MHz EIRP: 55.2 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 17.9 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 32 dB/K Simultaneous RHCP and LHCP	25500-27000 MHz G/T: 34.7 dB/K Simultaneous RHCP and LHCP (Upgrade required)
Dubai, UAE	9.1 m El-Az-Train		2025-2110 MHz EIRP 38 dBW Selectable RHCP or LHCP	2200-2290 MHz G/T: 20 dB/K Simultaneous RHCP and LHCP	8025-8400 MHz G/T: 34 dB/K Simultaneous RHCP and LHCP	N/A

*Denotes antenna redundancy

Leaf Space

Leaf Space is a ground-segment-as-a-service provider, operating a globally distributed ground station network to support satellites of any size in low-Earth (LEO), Medium-earth (MEO) and Geostationary (GEO) orbits. Leaf Space's Leaf Line network (see Figure 11.9) enables TT&C and payload data transmissions to and from the satellite operators' mission control software and cloud platform of choice. Leaf does so via a simple API interface, a proprietary autonomous scheduling software, and global coverage of ground stations, supporting satellites in every LEO orbit – mid, high, equatorial-inclination. Leaf Line antennas are fully owned or fully managed, ensuring maximum availability, flexibility, and independence of operations. Leaf Line is powered and

orchestrated by a secure cloud architecture designed to support multiple satellite missions and operators at the same time. The network's secure architecture allows seamless use of any antenna without requiring any further resource overheads or worrying about varying performance over different ground stations. Leaf Space works closely with its customers to execute both routine operations and complex tasks such as LEOP or custom CONOPS.

Leaf Line consists of 3.7m and 3.9m diameter antennas and supports operations in S-band uplink (2025-2110 MHz, EIRP: 50 dBW), S-band downlink (2200-2290 MHz, G/T: 12.8 dB/K), X-band downlink (8025-8500 MHz, G/T: 25.4 dB/K), and Ka-band downlink (25.5-27 GHz, G/T: 30.0 dB/K). The network includes a limited UHF (400.15-403 MHz) support capability as well. Leaf Space is expanding its ground station network in terms of sites, capacity, and antenna performances to cater to a wide variety of customers. With an eye towards the future of the sector, Leaf Space is working to integrate larger-aperture antennas for upcoming GEO missions, optical ground systems as well as cislunar-oriented solutions to its existing network.

In addition to providing access to the Leaf Line network, Leaf Space also procures, deploys and operates dedicated and/or custom ground stations for its customers (service offering called Leaf Key), enabling operators with non-standard frequencies, access or data requirements to leverage Leaf Space's cloud architecture and ground segment experience without compromising on antenna time. For more information on the company or on how Leaf Space can support current and future missions, please contact sales@leaf.space or visit https://leaf.space.
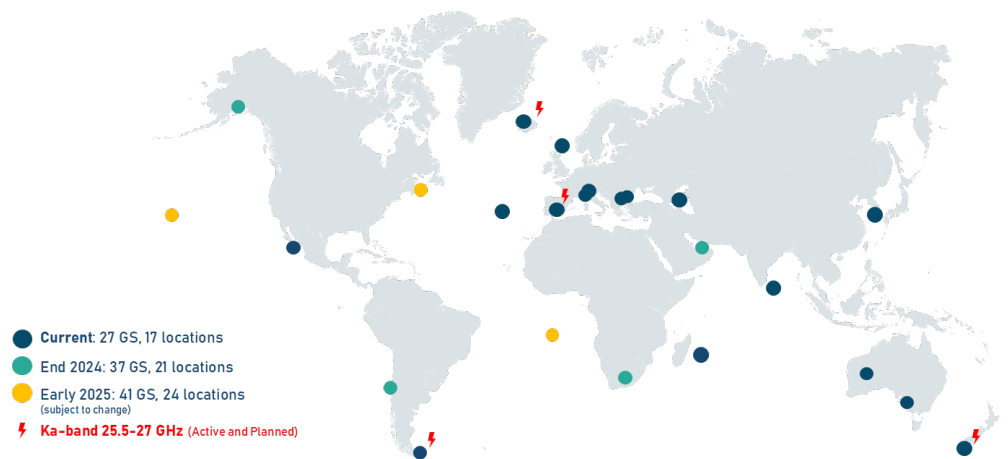
Figure 11.9: The Leaf Line network, September 2024. Credit: Leaf Space.

Laser Light

Laser Light Communications operates a Global Network platform, delivering a first-of-a-kind 21st century data service that will transform the way high volume communications traffic is carried. Using a hybrid infrastructure spanning terrestrial, subsea, and space domains; an end-to-end software defined architecture offering all-optical: up to 400 Gbps connectivity and provisioning within minutes (see https://www.laserlightcomms.com/).

Network On Demand

- Pay-as-you-go: only pay for the duration you use the service with no upfront or fixed costs

- Cost Effective: automation and end-to-end control yields significant operating cost savings

- Secure: highly targeted, dynamic, laser links are extremely difficult to intercept and can be encrypted

- High Capacity/ Performance: data delivery at up to 400 Gbps in the most direct route from origination to destination, automatically bypassing points of congestion

Data Transport as a Service

- Pay-as-you-go: only pay for volume you use when you use it with no upfront or fixed costs

- Cost Effective: automation and end-to-end control yield significant operating costs savings

- Secure: highly targeted, dynamic, laser links are extremely difficult to intercept and can be encrypted

- High Capacity/Performance: data delivery at optical speeds -- up to 400 Gbps directly from the point of origination to the point of destination, automatically bypassing points of congestion along the way

Atlas Global Network

ATLAS Space Operations, Inc. provides satellite RF communication services to the government and commercial sectors through geographical dispersion and cloud services, offering GSaaS on a global network of 34 ground stations and 51 antennas (https://atlasspace.com/).
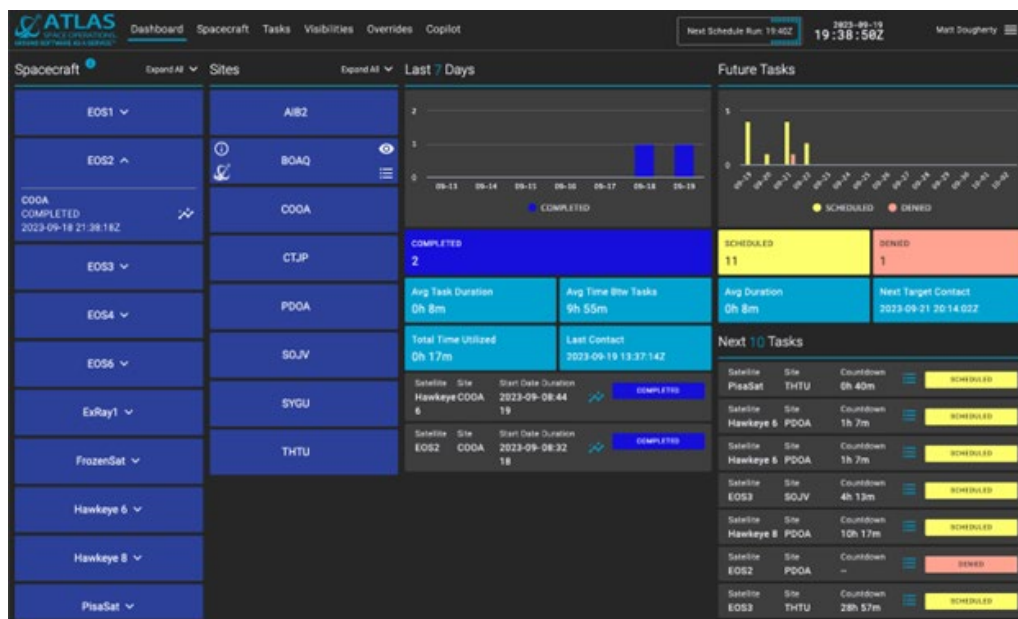
Figure 11.10. The Freedom™ Software Platform is a simple and scalable ground network management system. Credit: ATLAS Space.

All ATLAS ground stations are built upon the Freedom™ Software Platform (Figure 11.10) within a cloud-based distributed operations center which facilitates dynamic demand and scalable growth. Through geographical dispersion of its ground stations and cloud services, ATLAS provides a resilient capability that delivers dependable low latency data. ATLAS bases its mission success model on its global network of operational and traditional RF parabolic ground stations that are integrated with network management and scheduling software. These platforms work together as a mission architecture to meet customer requirements for routine satellite-to-ground communications.

ATLAS' Global Antenna Network is fully integrated with the Freedom Software, providing users with a robust, low latency, and secure communications solution, including: automated network operations, set-and-forget scheduling, mixed modem capability, real-time metrics, and single secure VPN access. Once integrated into the ATLAS Network, a single secure VPN enables access and load balancing of network resources. Freedom Core Services advance operations beyond legacy constructs and enable users the freedom and flexibility to reliably schedule satellite passes with minimal human interaction via machine-to-machine capabilities. Entire data processing and forwarding workflows can be automated within the cloud to ensure spacecraft data is ready for use as soon as it arrives at the Mission Operations Center.

ATLAS supports deploying clusters and serverless instances onto AWS. ATLAS hardware parameters are highlighted below (ATLAS Enterprise Sites and Digital Partner Sites). In addition to S-band and X-band capabilities, ATLAS can provide VHF, UHF services in Japan and Guam. The existing and planned ATLAS antenna systems support RF connectivity for LEO, MEO, GEO, and L1 orbits. ATLAS is actively pursuing technology development for deep space capabilities. ATLAS' global federated network is also displayed below. A summary of government and commercial ground stations and antennas in Table 11-8 and shown in Figure 11.11 is ATLAS' Global Federated Network of ground stations.

Freedom Space Technologies, Inc. dba Freedom Space was established in 2023 as a non-traditional small business and wholly-owned subsidiary of ATLAS Space Operations. Freedom Space, headquartered in Colorado Springs, CO, is purpose built to address and support the Department of Defense (DoD) and Intelligence Community (IC) mission requirements.

Figure 11.11: ATLAS Space Operations ground network map September 2024. Credit: ATLAS Space Operations.

Table 11-8: ATLAS Federated Antenna Network

Location	Lat	Long	Antenna Size (m)	Rx Freq (MHz)	G/T Long (dB/K)	Tx Freq (MHz)	EIRP (dBW)	Polarization
Utqiagvik (Barrow), AK, USA	71.27	-156.81	3.7	S: 2200-2300 X: 7900-8400	12.8 25.4	S: 2025-2120	60.0	R/LHCP
Dundee, Scotland	56.40	-3.18	3.7	S: 2200-2400	13.6 26.5	S: 2025-	48.0	R/LHCP

Table 11-8: ATLAS Federated Antenna Network								
Location	Lat	Long	Antenna Size (m)	Rx Freq (MHz)	G/T Long (dB/K)	Tx Freq (MHz)	EIRP (dBW)	Polarization
				R/LHCP X: 7800-8400		2120		
Chitose, Japan	42.77	141.62	3.4	S: 2200-2300 X: 7900-8500	11.5 26.4	S: 2025-2120	52.0	R/LHCP
Mojave, CA, USA	35.05	-118.15	3.0	S: 2200-2300 X: 7900-8500	11.3 25.9	S: 2025-2110	54.4	R/LHCP
Dubai, United Arab Emirates	24.94	55.35	3.7	S: 2200-2300 X: 7900-8400	12.8 25.4	S: 2025-2120	50.0	R/LHCP
Paumalu, Hawaii, USA	21.67	-158.03	7.3	S: 2200-2300 X: 7900-8500	20.7 31.0	S: 2025-2120	65.0	R/LHCP
Harmon, Guam	13.51	144.82	3.7	S: 2200-2300 X: 7900-8400	13.6 25.1	S: 2025-2110	52.4	R/LHCP
Sunyani, Ghana	7.34	-2.34	3.0	S: 2200-2300	12.4	S: 2025-2110	50.1	RHCP
Mwulire, Rwanda	-1.96	30.39	9.3	S: 2200-2300 X: 7900-8500	21.5 36.0	S: 2025-2120	60.0	R/LHCP
Tahiti, French Polynesia	-17.63	-149.60	3.7	S: 2200-2300 X: 7900-8400	13.9 27.4	S: 2025-2110	47.5	R/LHCP
Awarua, New Zealand	-46.52	168.38	3.7	S: 2200-2300 X: 8025-8500	13.7 27.0	S: 2025-2120	49.0	R/LHCP
Sodankylä, Finland*	67.36	26.63	7.3	S: 2200-2300 X: 7900-8500	19.8 32.1	S: 2025-2110	54.8	R/LHCP
Öjebyn, Sweden	65.33	21.42	7.3	S: 2200-2290 X: 8025-8400 Ka: 25500-27000	18.0 32.0 35.7	S: 2025-2110	55.2	R/LHCP
North Pole, Alaska, USA	64.79	-147.53	7.3	S: 2200-2300 X: 8000-	19.0 32.0 35.7	S: 2025-2120	53.0	R/LHCP

Table 11-8: ATLAS Federated Antenna Network							
Location	Lat	Long	Antenna Size (m)	Rx Freq (MHz)	G/T Long (dB/K)	Tx Freq (MHz)	EIRP (dBW)	Polarization
				8500 Ka: 25500-27000				
Stockholm, Sweden*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Dublin, Ireland*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Guildford, United Kingdom	51.24	-0.61	5.4	S: 2200-2290 X: 8025-8400	17.0 30.0	S: 2025-2110	53.2	R/LHCP
Portland, Oregon, USA*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Obihiro, Japan	42.59	143.45	7.3	S: 2200-2290 X: 8025-8400 Ka: 25500-27000	17.9 31.5 35.0	S: 2025-2110	55.2	R/LHCP
Columbus, Ohio, USA*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Seoul, South Korea*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Pendergrass, Georgia, USA	34.17	-83.67	5.4	S: 2200-2290 X: 8025-8400	17.0 30.0	S: 2025-2110	53.2	R/LHCP
Deadhorse, Alaska, USA*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Zallaq, Bahrain*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Kapolei, Hawaii, USA*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Accra, Ghana	5.74	-0.30	7.3	S: 2200-2290	18.0 32.0	S: 2025-	65.0	R/LHCP

Table 11-8: ATLAS Federated Antenna Network								
Location	Lat	Long	Antenna Size (m)	Rx Freq (MHz)	G/T Long (dB/K)	Tx Freq (MHz)	EIRP (dBW)	Polarization
				X: 8025-8400 Ka: 25500-27000	35.7	2110		
Singapore*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Alice Springs, Australia*	-23.75	133.88	7.3	S: 2200-2290 X: 8025-8400 Ka: 25500-27000	18.0 32.0 35.7	S: 2025-2110	65.0	R/LHCP
Pretoria, South Africa	-25.88	27.70	7.3	S: 2200-2290 X: 8025-8400 Ka: 25500-27000	18.0 32.0 35.7	S: 2025-2110	55.2	R/LHCP
Mingenew, Australia*	-29.01	115.34	5.0	S: 2200-2300 X: 8025-8500	14.0 29.5	S: 2025-2120	55.0	R/LHCP
Cordoba, Argentina	-31.52	-64.46	5.4	S: 2200-2290 X: 8025-8400	17.0 30.0	S: 2025-2110	53.2	R/LHCP
Cape Town, South Africa*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Dubbo, Australia*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Punta Arenas, Chile*	-	-	5.4	S: 2200-2300 X: 7750-8400	16.0 30.5	S: 2025-2120	50.0	R/LHCP
Ushuaia, Argentina	-54.51	-67.11	7.3	S: 2200-2290 X: 8025-8400 Ka: 25500-27000	17.9 32.0 33.0	S: 2025-2110	56.0	R/LHCP

11.4.3   Space Relay Network - NASA

Unlike a traditional ground network that goes direct from a "client" satellite to a ground station on the ground, space relay networks consist of communication satellites that relay data from the "client" satellite down to a ground station. While some no longer consider it state-of-the-art, NASA's Tracking and Data Relay Satellite System (TDRSS), shown in Figure 11.12, is one of the most well-known space relay networks. TDRSS relays data from the International Space Station (ISS) and the Hubble Space Telescope to NASA ground stations around the world.

Figure 11.12: NASA's Tracking and Data Relay Satellite System. Credit: NASA.

Space relay networks can be beneficial for small satellites in low-Earth orbit because those SmallSats are only in view of a ground station for a portion of their orbit. However, depending on the orbit of the relay satellites, a low-Earth orbit SmallSat could be in view of a relay satellite for most of its orbit. This makes a relay network beneficial for a SmallSat, especially right after SmallSat deployment when a ground station is still trying to locate the satellite. The space relay can transmit satellite telemetry, tracking, and control data to the ground, enabling faster satellite identification. This proves to be even more valuable when the satellite is deployed with several others for a given rideshare opportunity. This data can also contain satellite health information to give mission teams either peace of mind while awaiting acquisition by the ground station, or information for troubleshooting prior to the commissioning phase. Another benefit is the ability to obtain real-time event notifications without the need for prior scheduling. Scientists are interested in using this technology to alert the science community when certain phenomenon are observed. Space relay networks often require special hardware or software that must be added to a satellite before launch. In general, a satellite operator will purchase a modem compatible with the relay network and fly it on their satellite to access the network. It is common for the network providers to license their proprietary chipset to developers who build the modem hardware and serve as a service broker. Because of this added hardware component, the decision to leverage a space relay network must be made before the satellite has been launched.

11.4.4   Low Latency, Low Rate (Short Burst) Space Relay Providers

Space relay solutions are less common than traditional direct-to-Earth solutions, but there are a few options that exist for small satellites (see Table 11-9). To access the space relay, a satellite operator purchases a modem from the relay manufacturer and flies that on their satellite to access

the relay services. In general, space relays are ideal for obtaining satellite TT&C data (health and safety of the vehicle) rather than for large data downlinks.

Table 11-9: Service Providers for Space Relay Networks			
Product	Manufacturer	TRL	Specifications
Iridium Global Network	Iridium	9	LEO relay requiring 9600 series transceivers onboard the satellite
Fast Pixel Network	Analytical Space	6	Establish a data transport network in LEO

The Iridium network is one example satellite customers can rely on for delivering low latency messages. Iridium uses a combination of Frequency Division Multiple Access (FDMA) and Time-Division Multiple Access (TDMA) for its communication waveforms. L-band (1616 – 1626.5 Mhz) is used for uplink and downlink between the user spacecraft and the Iridium spacecraft. Inter-satellite communication links between Iridium satellites is accomplished through Ka-band (23.18 – 23.28 GHz). Operators install an Iridium transceiver (9600 series) onboard their spacecraft to communicate with the Iridium network. Messages are relayed through Iridium's Short Burst Data Service, which is hosted on Iridium's cloud platform for easy user operation. For each transceiver unit, a data plan must be chosen and purchased, much like cellular phone data plans, and the plan details are linked to the unit's ID, which is referred to as International Mobile Equipment Identity (IMEI). The special feature of this system is that it has as an option for "IMEI-to-IMEI" transmission. When an Iridium IMEI is activated, five output destinations may be specified. Most vendors allow for a combination of emails addresses, fixed IP address, or another device with an IMEI ID (6).

Iridium has announced the commercial availability of its Certus 100 'midband' service providing 88 kbps connectivity via small antennas and battery-powered devices, for basic data communications and IoT applications (10).

Analytical Space is another company to watch for future services. Their recent contract puts a LEO relay in space to aggregate data from GEO satellites. Through Fast Pixel Network, Analytical Space Inc. plans to establish a data transport network in low-Earth orbit to ingests data from geospatial intelligence satellites, send the data from node to node via high speed optical intersatellite links, and deliver the data to military, intelligence and commercial customers (11). High speed MEO and GEO commercial relays are not presently operating, but several are planned. These are listed under the State-of-the-Art Ground Data and Supporting Systems section (11.9).

11.5 Ground Stations Components

The hardware for ground stations consists of the tracking antenna, its feed, and the modem that converts the RF waveform into digital packets and vice versa.

11.5.1 Ground Station Operation

A DTE ground station is comprised of a system of hardware and software working together to convert the RF signal from a satellite signal into digital data. The first key element of the system is the antenna. It is chosen based on the frequency and gain required to talk with a satellite. NASA uses parabolic reflector antennas for RF ground satellite communications, while some universities use dish or Yagi antennas.

The dish antenna uses a parabolic reflector to collect signals from the spacecraft and focus them onto a feed antenna. The feed antenna is typically a horn antenna with a circular aperture. The size of a dish is at least several wavelengths in diameter at the frequency of operation and can be increased for higher gains. The distance between the feed antenna and parabolic reflector can

also be several wavelengths. For example, a Ka-band 34 m deep space antenna with a feed distance of 15 m would be approximately 3,000 wavelengths for the dish diameter and 1,500 wavelengths for the feed distance relative to a 1 cm Ka-band wavelength. The gain of a dish reflector (Figure 11.13) is frequency-dependent and it is directly related to the square of its diameter. Dish antennas are available in sizes from 1 meter to 70 meters in diameter.

The antenna collects RF waves and the antenna feed converts the electromagnetic waves into conducted RF electrical signals. The feed consists of a resonant pickup that is tuned to the transmit or receive frequency, a low gain low-noise amplifier, a sharp filter, and a second LNA with more gain than the first amplifier. These elements condition the signal. The signal then traverses through a coaxial cable to a nearby location where a radio demodulates the RF signal into digital data. In the uplink direction, the radio modulates the data bits onto an RF carrier which is amplified to 10 or more watts. The amplified RF resonates in the antenna feed, and the antenna amplifies the electromagnetic waves and focuses them towards the satellite.

Figure 11.13: Ground Antenna in Fairbanks, Alaska. Credit: NASA/ Clare Skelly.

It is desirable to have significant antenna gain, but as the gain increases, the beamwidth of the antenna decreases. There is a practical compromise where the beamwidth is so small that tracking is difficult and when the antenna gets so large that it is difficult to procure or manage. A typical antenna pattern is shown in Figure 11.14. There is a center lobe where most of the transmitted energy is contained. The remaining energy is stored in the sidelobes on either side of the main lobe. The diminished side lobes are intentional so that ground noise from other emitters on Earth are not collected when receiving and so that interference to terrestrial systems is not created when transmitting. The blue arrows in the Figure indicate the full-width-half-max gain point at about ±6°, which should result in an antenna pointing error of less than 6° and the full-width- half-max gain of 16 dBi to be used in a link budget. If more gain is needed, then the antenna will increase in size and the beamwidth will correspondingly decrease.
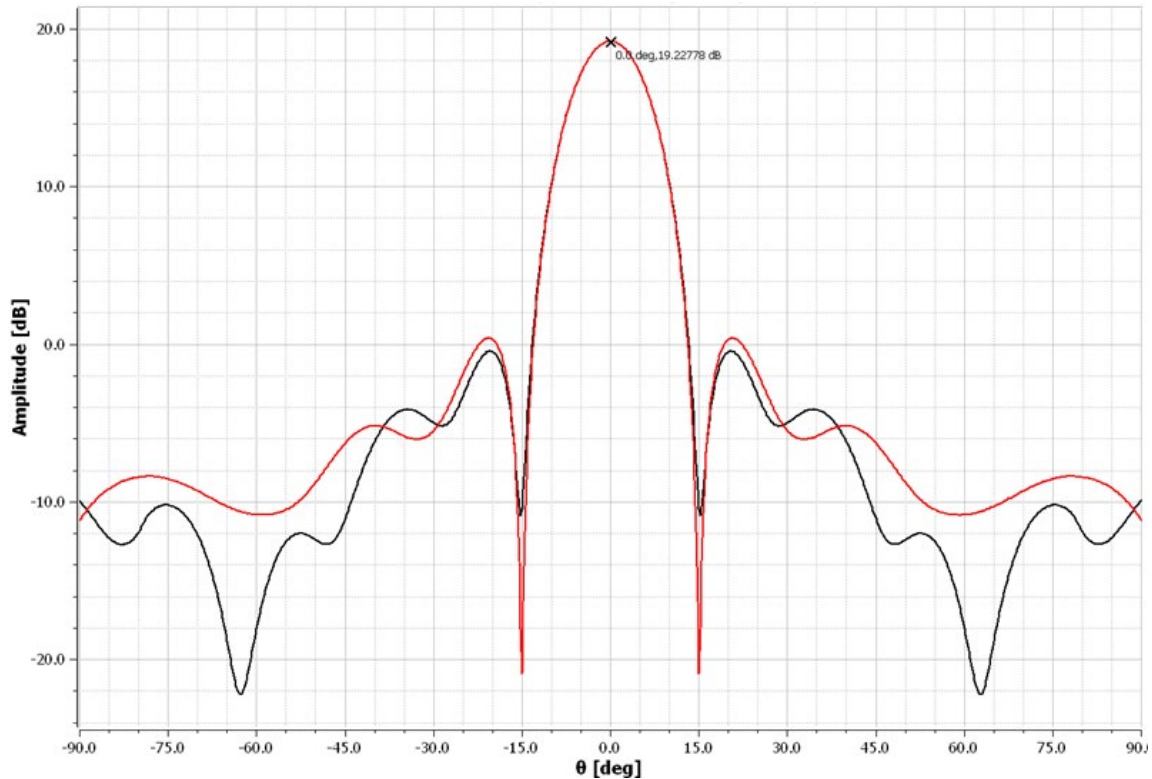
Figure 11.14: Antenna pattern from a 1.8-meter diameter parabolic dish operating at 915 Mhz with a high gain center lobe and diminished side lobes. Credit: NASA.

Directional antennas point towards the satellite as it moves over the ground station. Pointing adjustments are necessary in both the vertical (elevation) and horizontal (azimuth) directions. These movements are accomplished using motors and gears. Tracking software is used to predict the satellite's future location. The satellite position and time are processed through additional software that converts this information into commands for the motor controller. Time is an important factor and GPS time is used by the computer generating the satellite position estimate. A dedicated GPS receiver is connected to the computer for that purpose.

The cost of a DTE ground station is directly correlated with the size of the aperture, which drives the ground station foundation, pedestal, motors, and gears. The Yagi is less expensive. It sustains low wind loads and therefore can use a smaller foundation for support. In contrast, the dish antenna reflector sustains comparatively high wind loading and therefore needs a stronger concrete foundation and larger motors and gear box elements than the Yagi antenna.

11.5.2   Component Hardware for Ground Systems (GS)

This section lists examples of GS components and some supporting equipment. Table 11-10 lists example products in each category. The antenna feed consists of the RF pickup, LNA and mechanical filters located directly on the antenna. A radome is an RF transparent enclosure that protects the antenna from weather.

Table 11-10: Ground System Components		
Product	**Manufacturer**	**Type of Product**
Tracking Antenna	Viasat, Safran	Antennas for small satellites in and S, X and Ka-band frequencies
Antenna Feed	See End-to-End Hardware Section	RF pickup, mechanical filters, LNA

	11.7.2	
Radio, Software Defined	NI Ettus Research	USRP X410, up to 7.2 GHz with RFSOC advanced FPGA and meeting wide bandwidth requirements. USRP X310: DC-6 GHz with up to 160 MHz of baseband bandwidth, multiple high-speed interfaces
Data Receiver	Safran Data Systems	Cortex CRT (low data rate) and HDR (high data rate) (previously by Zodiac)
Modem, for TT&C and Payload Reception	Safran Data Systems	Satcore, plug-and-play modem for TT&C and Payload Reception
Digital Processing	Kratos	SpectralNet: Digital IF product that converts analog signals at RF frequencies up to S-band into digital IF packets.
Radome	Infinite Technologies	Antenna radomes
Ground Station Dongle	GAUSS	A USB low-power board to simulate ground station in laboratory conditions. The USB dongle integrates both a low-power UHF transceiver and a TNC, thus miniaturizing common ground station rack systems
Integrated Testing Systems (EGSE) & Ground Station TT&C Modems	Celestia Satellite Test & Simulation	Hardware and software elements all operating within a single reference platform and environment

Cortex HDR

Several NSN, SSC and NOAA stations use the Cortex HDR High Data Rate Receiver, which performs demodulation, decoding and frame synchronization on the (X-band) data stream. Each virtual channel in the AOS frame that is received by the station X-band receiving system will be written into separate files. Files are separated into small one-minute file sizes for a single VCID that allow for faster turn-around time on the data and smaller transmission cycles in case of transfer problems. File-based data is stored in a buffer (e.g., for 7 days) used for retransmissions and failure recovery when necessary. At the end of a pass, ground stations such as the NSN ones, perform an automatic secure file transfer protocol (SFTP)/secure copy protocol (SCP) push to the customer. If the customer wants to "replay" a dataset, they may use the self-service /SFTP/SCP interface on the system to pull their data to their site. Alternatively, the customer may choose to manually retrieve files and not select automatic file transfer (12).

USRP X310 and X410 Open-Source Software Defined Radio for SatCom Applications

The NI Ettus Research brand includes the Universal Software Radio Peripheral (USRP™) family of products. The USRP is one of the most popular open platforms for small satellite communications with options from high-performance to low-cost to highly deployable. One of the most popular hardware units for satellite communication applications is the USRP X310 with the UBX RF daughterboard. The USRP X310 is a high-performance software defined radio with the ability to transmit and receive modulated signals. With up to 160 MHz of instantaneous bandwidth and a frequency tuning range up to 6 GHz, the X310 with UBX has the raw hardware performance to cover many ground station satellite communication needs. The USRP family supports a wide range of software tool chains from LabVIEW to GNU Radio, with many existing IP modules for modulation and demodulation. The USRP X310 is intended for lab environments, however, it can

be built in rugged weatherproof configurations. Many small satellite researchers are using the USRP as their ground station equipment for its adaptability with open-source software and its embedded FPGA pre-processing capability. The USRP X310 offers 2 channels, 10 GIGE and PCIE bus, whereas the NI Ettus USRP X410 is equipped with two dual 100 GbE interfaces capable of moving much more data (13).

Kratos OpenSpace® SpectralNet®

OpenSpace SpectralNet eliminates the distance constraints of RF transport by digitizing RF signals into standard VITA-49 or DIFI IP packets for transport over IP networks in a way that preserves both frequency and timing characteristics, and then uniquely restores the RF signals at their destination. By eliminating the distance constraints between antennas and signal processing equipment, this technology enables operators to deploy new ground architectures with numerous advantages, such as the ability to mitigate the effects of rain fade for Ku/Ka satellites, reduce costs by centralizing operations, simplify disaster recovery and system maintenance, optimize antenna placement, and develop a migration path toward virtual ground systems. SpectralNet does all of this while protecting the operator's current investment in existing equipment (14). In addition to removing distance constraints, digitizing RF signals into DIFI packets also provides the on-ramp to virtualizing network functions such as modems, receivers, etc.

Integrated Testing Systems & Ground Station TT&C Modems

Celestia Satellite Test & Simulation BV (C-STS) provides ground-based solutions in the domains of satellite simulation, testing, communication, and data processing. Established in 1985, Satellite Services B.V. (SSBV) was acquired by Celestia Technologies Group in 2016 and re-branded to Celestia Satellite Test & Simulation B.V. to continue as a competence center for Electrical Ground Support Equipment (EGSE) and TT&C solutions. Celestia STS has more than 30 years of experience in the space industry. More than 300 EGSEs and TT&C modems were delivered to space agencies, large system integrators, and specialized flight-equipment manufacturers around the world.

On-board computers, mass memory units, and transponders are tested every day with C-STS equipment. Celestia EGSE solutions have been used in more than 80% of all European Space Agency (ESA) missions. Celestia STS testing equipment is available in standard functionality or configured to meet specific customer needs. System options include:

- Multi-Purpose Interface Platform
  - Test and simulation of spacecraft electrical interfaces
  - Discrete TM/TC, power and digital data interfaces
  - RS422/RS485/LVDS, SpaceWire, SERDES and parallel/serial LVDS
  - Up to two different bus voltage inputs for LCL and heater interfaces
  - Two gigabit ethernet ports for control/monitoring and data via TCP/IP (RJ45)
  - FMEA reports available for all interface modules
- Telemetry and Telecommand Processing System
  - TM acquisition and simulation
  - TC generation and acquisition
  - Bit error rate tester
  - TC authentication support
  - TM/TC deciphering (API/DLL/LAN)

- - o Includes control and monitoring software for data processing and visualization
- Wizardlink High-Rate Interface System
  - o Up to 4 Wizardlink bi-directional channels operating in parallel (2Gbps per channel)
  - o 10Gbit LAN for data streaming via TCP/IP (using SFP+)
  - o LVDS Flow Control TIA/EIA-644-A compliant
  - o Includes software for high speed ingest, data archiving and export
  - o Optional software for Level Zero data processing
- LVDS High-Rate Interface System
  - o Up to 4 parallel LVDS inputs and outputs
  - o 8-bit parallel up to 1Gbps per channel
  - o 10Gbit LAN for data streaming via TCP/IP (using SFP+)
  - o TIA/EIA-644-A compliant LVDS interfaces
  - o Teaming of 2 LVDS input and output channels to 16-bits
  - o 8-bit Parallel >1Gbps per channel, 16-bit parallel >2Gbps per channel
  - o Includes software for high speed ingest, data archiving and export
  - o Optional software for Level Zero data processing
- TT&C Integrated Modem and Baseband Unit
  - o Single or dual channel modulation and demodulation (70MHz I/F)
  - o Ranging measurement
  - o Doppler simulation
  - o Bit error rate tester
  - o Baseband TM and TC processing (optional)
- MIL-STD-1553 Interface System
  - o Up to 4 independent, dual redundant MIL-STD-1553 channels
  - o Bus controller, remote terminal and bus monitoring functionality
  - o External time/reference inputs, such as 10MHz, PPS and IRIG
  - o Fully compliant to MIL-STD-1553B Notice II/IV
  - o Specific protocol adaptations available
- CANopen Interface System
  - o Compliant with CAN specification 2.0A (11-bit ID) & 2.0B (29-bit ID)
  - o Master, slave node and bus monitoring functionality
  - o Transceivers support 1Mbps high-speed CAN bit rate (ISO-11898-2)
  - o Simultaneous real-time and non-intrusive monitoring of multiple CAN channels
- Level Zero Processor Software for High-Speed Data Processing
  - o Data directly from the local disk drive or shared network drive
  - o Processing of TM data from bitstream to frame and packet level
  - o Configurable frame and packet checking rules
  - o Configurable frame and packet output data storage and sorting

- o   Live frame and/or packet distribution via LAN
- o   Real-time statistical analysis, error checking, and reporting
- o   CFDP data processing support
- Optical Modem
  - o   Standard compatibility: SDA, CCSDS O3K RS Branch, custom
  - o   Multiple tunable optical Tx outputs with programmable skew (optical C-band)
  - o   Powerful receiver for very low optical input powers
  - o   FC/APC inputs and output mating ports
  - o   10Gbit LAN for data ingress/egress

Efforts are on-going to improve product capability with a focus on modular, flexible, scalable multichannel systems that take advantage of the latest technologies (15).

Infinite Technologies Radomes

A well-designed radome serves as a protective cover for an antenna while minimizing any adverse effects on its electrical performance. Figure 11.15 illustrates an example of a radome supplied by Infinite Technologies. Radomes create a controlled environment for the antenna system, shielding sensitive equipment from environmental stresses such as wind, snow, ice, and salt spray. By protecting the antenna from these elements, a radome can extend its operational life and reduce overall maintenance costs.

Figure 11.15: Infinite Technologies small radome. Credit: Infinite Technologies.

Incorporating a radome early in the system design phase is crucial. Doing so allows for the use of lighter and more cost-effective components, such as drive motors and foundations, since the radome eliminates wind loads on the antenna. Additionally, the controlled environment within the radome increases system availability, enabling the antenna to operate efficiently under harsh environmental conditions with minimal signal degradation. Maintenance personnel also benefit from the radome, as it offers protection from the weather during antenna maintenance activities.

For a radome to be truly effective, it must be tailored to the specific requirements of the system it

protects. A carefully selected and well-designed radome can enhance overall system performance and reliability by:

- Enabling operation in severe weather conditions, shielding the antenna from wind, rain, snow, hail, sand, salt spray, insects, animals, UV damage, windblown debris, and extreme temperature fluctuations.

- Creating a controlled environment that minimizes downtime, prolongs component and system life, and supports more economical choices in antenna pedestals, foundations, and drive systems.

- Providing security and protection for the antenna system from observation, vandalism, and other threats.

11.5.3   Ground Software

Ground station visualizes and calculates the satellite location in orbit and controlling the tracking antenna. Command and control software manages command scripts to be sent to the satellite and can display and analyze telemetry. Many software options are open source and free. Other software may be purchased from companies with a long history in ground segment solutions who had previously provided hardware products to do these tasks (Table 11-11).

Table 11-11: Software for Ground Systems			
Product	**Manufacturer**	**TRL**	**Type of Product**
softFEP	AMERGINT	9	Emulation ground systems software
OpenSpace quantumFEP	Kratos	9	Software that performs data formatting and interface conversion for commands and telemetry, with full support for NSA Type 1 and AES encryption/decryption devices
Gpredict	Alexandru Csete	9	Open-source software that tracks satellites and provides orbit prediction in real-time. Radio and antenna rotator control for autonomous tracking
GNU Radio	GNU Project	9	Free software development toolkit that provides signal processing blocks to implement software-defined radios and signal processing systems
HWCNTRL	DeWitt & Associates	9	Ground station control program with an automation software package

AMERGINT softFEP

AMERGINT softFEP applications are deployed virtually on cloud architectures or hosted on dedicated servers. The applications perform control center data formatting and interface conversion for commands and telemetry, with full support for NSA Type 1 and AES encryption/decryption devices. SoftFEP applications are built on a proven library of more than 1,000 software devices. This allows each softFEP application to be tailored to the requirements specific to the ground system. Processing chains configured via Python scripts move satellite downlink data from Earth receipt for processing and uplink data to the radiating site. Deploying softFEP on multiple virtual machines (VMs) or within the cloud is inherent in the product architecture. Virtualized softFEP deployments support a wide range of ground system architectures while taking advantage of cloud-computing benefits. When applications are deployed in VMs, they can be hosted locally or run remotely in a cloud and interoperate across network connections. Customers have deployed their softFEP applications as independent network gateways, black front-end processors, red front-end processors, and data recorders, flowing data between the VMs as a satellite contact is processed (17).

Kratos quantumFEP

OpenSpace quantumFEP (qFEP) is a virtual front-end processor for Telemetry, Tracking, and Command (TT&C). The system has been specifically designed to match the requirements, schedules, and budgets of quick turn programs. qFEP connects C2 systems to RF signal processing equipment, handling command and telemetry stream formatting, encryption/decryption devices, CCSDS processing, and network interfaces to either quantumRadio (qRADIO) or third-party ground antenna networks. Using generic compute servers, virtual machine environments, bare metal installations, or orchestration through Kratos OpenSpace Platform are supported operational types, and qFEP's small memory footprint allows for more efficient use of system resources. In addition, the product achieves independence by eliminating custom drivers, firmware, and hardware cards. Figure 11.16 provides an illustration of the qFEP system architecture (18).

Key features of quantumFEP are:

- Pure software implementation for signal processing functions
- Support for CCSDS TC, TM and AOS frame protocols, as well as Cadet, Ax.25 and Reed Solomon Encoding
- Suitable for all types of satellite programs
- Compatibility tested with widely used ground modems
- Built-in test functions reduce integration and test (I&T) effort and costs
- Configurable as mission requirements change or as new missions come online
- Commercial AES encryption/decryption standard feature with built in AES Key Manager
- Standard TCP/IP, GEMS, REST and VITA-49 interfaces make integration seamless
- Access and control from anywhere through the web. No client software to install or maintain
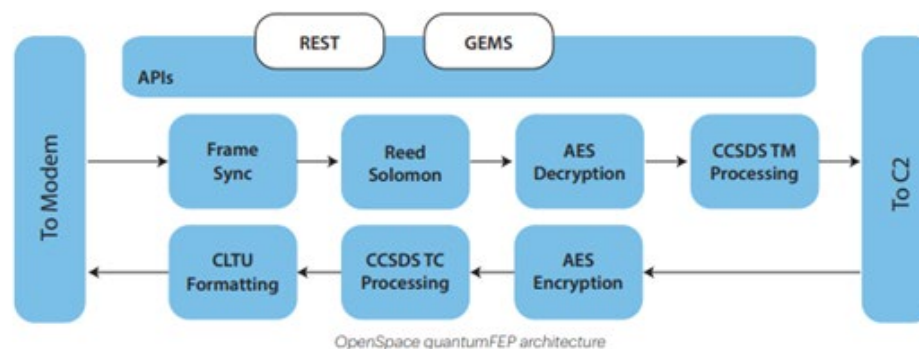
Figure 11.16: Kratos qFEP system architecture. Credit: Kratos.

Gpredict

Gpredict is a real-time satellite tracking and orbit prediction application. It can track a large number of satellites and display their position and other data in lists, tables, maps, and polar plots (radar view) as shown in Figure 11.17. It can also predict the time of future passes for a satellite and provide detailed information about each pass. Gpredict is different from other satellite tracking programs in that it allows the satellites to be grouped into visualization modules. Each of these modules can be configured independently from others, allowing unlimited

flexibility in the look and feel of the modules. It will also allow satellite tracking relative to different observer locations at the same time (19).

The following are key features of the software:

- Fast and accurate real-time satellite tracking using the NORAD SGP4/SDP4 algorithms
- No software limit on the number of satellites or ground stations
- Appealing visual presentation of the satellite data using maps, tables and polar plots (radar views)
- Allows satellites to be grouped into modules, each module having its own visual layout, and being customizable on its own. Of course, several modules can be used at the same time
- Radio and antenna rotator control for autonomous tracking
- Efficient and detailed predictions of future satellite passes. Prediction parameters and conditions can be fine-tuned by the user to allow both general and very specialized predictions
- Context sensitive pop-up menus allow future passes to be quickly predicted by clicking on any satellite
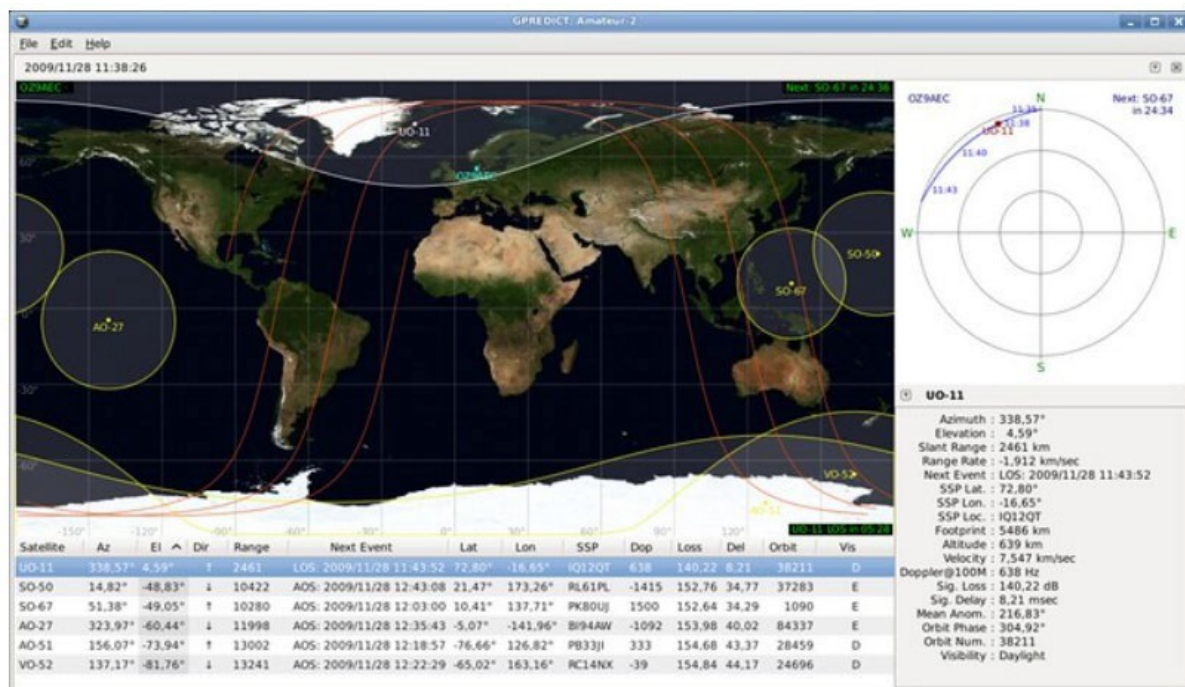
Figure 11.17: Gpredict graphical display with multiple satellites. Credit: Gpredict.

- Exhaustive configuration options allowing advanced users to customize both the functionality and look & feel of the program
- Automatic updates of the Keplerian Elements from the web via HTTP, FTP, or from local files
- With a robust design and multi-platform implementation, Gpredict can be integrated into modern computer desktop environments, including Linux, BSD, Windows, and

      Mac OS X

- As free software licensed under the terms and conditions of the GNU General Public License, it can be freely used, learned from, modified, and re-distributed

GNU Radio

GNU Radio is a free & open-source software development toolkit for developing radio systems in software as opposed to completely in hardware. It can be used with readily available low-cost external RF hardware and runs on most modern computers to create software-defined radios. It can also be used without hardware in a simulation-like environment.

GNU Radio performs all the signal processing and can be used to write applications to receive data out of digital streams or to push data into digital streams, which are then transmitted using hardware. GNU Radio has filters, channel codes, synchronization elements, equalizers, demodulators, vocoders, decoders, and many other elements (referred to as blocks) typically found in radio systems. More importantly, it includes a method of connecting these blocks and then manages how data is passed from one block to another. Blocks can be implemented in C++ or Python.

Since GNU Radio is software, it can only handle digital data. It operates with digitally sampled waveforms that can correspond to an RF signal with digital or analog modulation. Usually, complex baseband samples are the input data type for receivers and the output data type for transmitters. Analog hardware is then used to shift the signal to the desired center frequency. That requirement aside, any data type can be passed from one block to another–be it bits, bytes, vectors, bursts, or more complex data types. GNU Radio supports heterogeneous computing, where some of the blocks run on an FPGA or GPU. These acceleration techniques are particularly important for processing large bandwidths or data rates, and also in embedded platforms, where size and power consumption are usually constrained. Execution scheduling and data movement in a heterogeneous environment are active areas of development and research in the GNU Radio runtime. GNU Radio can be used on embedded systems, such as ARM SoCs running Linux, as well as on more powerful desktop or server PCs. GNU Radio is frequently used as part of the ground station, both for standard protocols such as CCSDS and for custom modems. Some commercial ground-station-as-a-service solutions that support GNU Radio modems are Azure Orbital Ground Station and AWS Ground Station. Another example is the open-source community-driven SatNOGS network. GNU Radio is also very useful for prototyping and lab testing. Additionally, some small satellites run GNU Radio on-board, usually as part of a highly flexible SDR payload. Figure 11.18 shows an example GNU Radio block diagram (20).
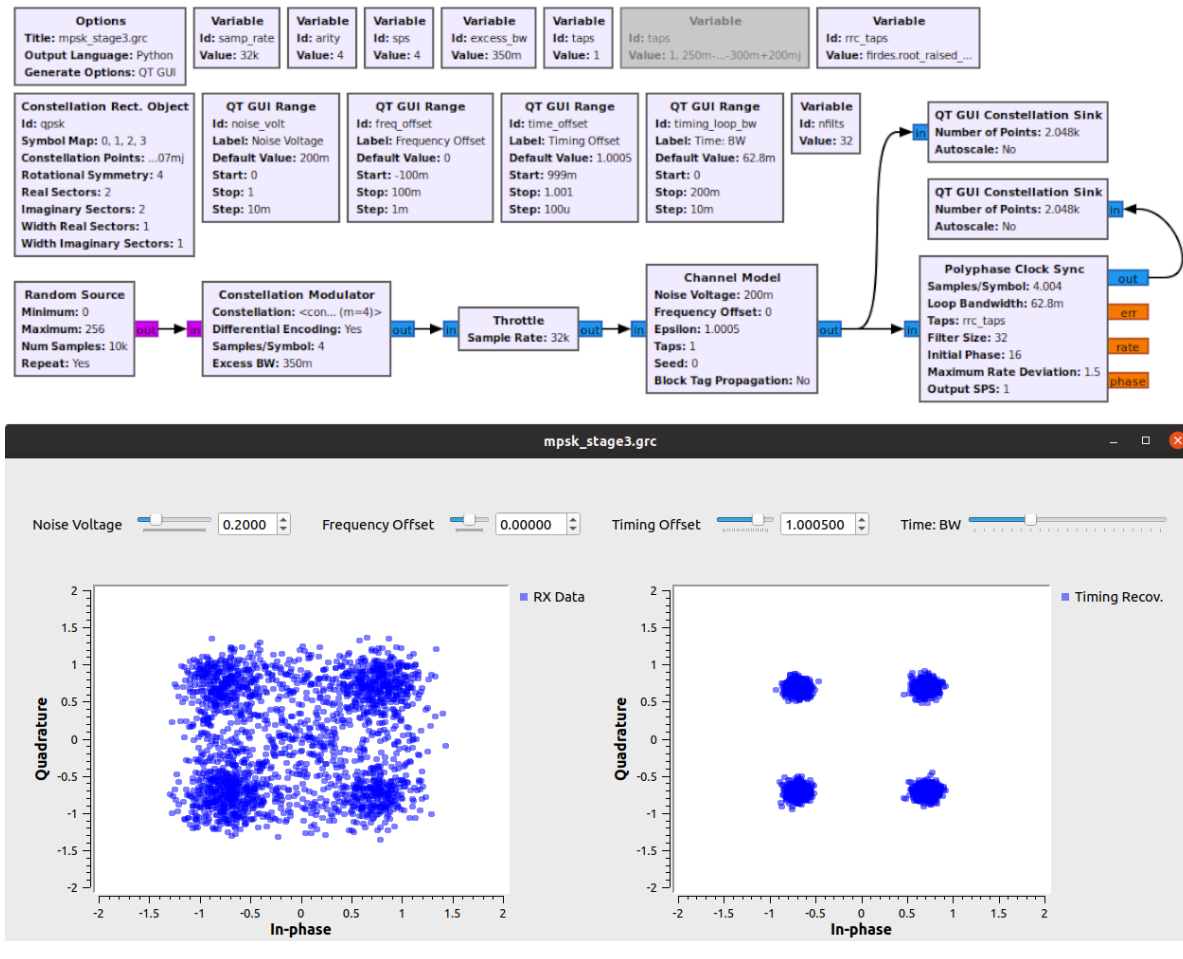
Figure 11.18: GNU Radio block diagram example for a 2-meter NBFM receiver. Credit: GNU Radio.

HWCNTRL

HWCNTRL is a satellite ground station control program that is installed in more than 30 sites throughout the world. This automation software package can support multiple antennas and instruments simultaneously. Satellite passes are generated by user request based on the ephemeris set, and users can select specific passes to be added to the schedule. Scheduled events can be single-use or reoccurring on a daily or weekly basis. A control/status screen is accessible for each instrument in the system, and the user can view and change the settings of any instrument through these screens (21).

11.6 Mission and Science Operations Centers

The Mission Operation Center (MOC) is where all satellite commanding is generated, ground station control is managed, and satellite telemetry is archived. It is typically a physical location where everything required to operate the satellite is located. It is often in a secure room with controlled access to protect the satellite operating equipment and prevent unauthorized satellite control. Inside the room are typically several terminals so that multiple subsystem experts can be reviewing telemetry or running their analysis programs concurrently. An example of a MOC with multiple terminals is shown in Figure 11.19.

Figure 11.19: Mission Operation Center at NASA Ames Research Center. Credit: NASA

The size of the MOC is determined by the complexity of the mission. There are more experts on during critical events or to resolve an anomaly. For a SmallSat mission, the complexity is usually lower and the MOC is a much smaller room. In addition to the terminals and telemetry analysis software are other resources for managing the satellite. These may include physical models of the satellite to study when contemplating anomalous telemetry. In the case of CubeSats, due to their small size, a functioning spacecraft engineering model may be useful to test commands and reproduce anomalies.

All tasking requests for future satellite operations are managed by the mission operations team. They will generate command plans, simulate satellite response to verify those plans, and if confidence in the simulations is not sufficient, they will run the commands on engineering model hardware prior to approving them for upload. The MOC team will also manage downloads. They will decide what ground resources are available when. If the MOC does not own its ground stations, a request for contact will be submitted to the ground station managing company. The MOC submits data necessary for commanding the satellite for upload which includes commands and parameter settings for the payloads, a schedule of events for the flight computer, and ephemeris and pointing tables for the attitude control system along with its own timeline of events. When the contact is complete, the data will be sent back to the MOC by the ground station.

Prior to a launch, there will be rehearsals with everyone at their stations, and simulated telemetry with anomalous readings inserted will be used to test the team. This ensures that they are ready with the proper analysis software or integration test data available to quickly diagnose the problem and propose a plan of action. At the time of launch, the MOC will be fully populated, as this is a critical event. Telemetry will have to be interpreted and acted upon in short order.

The SOC is the focal point for all mission science data. The science team will use it to store and analyze the data. From that analysis, the science team generates satellite tasking requests that are sent to the mission operations team. External requests for additional data collection come through the science team first to assess feasibility with the instrumentation before tasking requests are made to the operations team.

The SOC is typically separate from the MOC. The payload developer will have their own operations center located at their facility with easy access to supporting resources. Before cloud

data storage, the SOC was a physical place was where data servers resided to archive the mission science data. Prior to secure network solutions, dedicated computers were located inside the SOC that would run programs written specifically to analyze the science data. If the mission was secure and the data classified, then the physical SOC would be protected behind a locked door. Missions that do not produce classified data can take advantage of a virtual SOC instead of a physical location and the science data and special programs for analyzing data can reside in the cloud. The virtual SOC allows scientists to log on from anywhere and perform work without the need to come to a physical location and pass through secure doors. In the future, as cyber security techniques improve, it is likely that more and more secure missions will be comfortable with the virtual SOC solution and only the highest classification missions will maintain a secure physical SOC.

11.6.1 Software for Mission Operations

Mission operations rely on software across all the elements of the ground segment. Figure 11.20a outlines software functions for each of these elements. Satellite flight software not only manages state-of-health telemetry and payload data, but also software specific to the ground segment. Figure 11.20b provides an example of a command and telemetry data flow for a mission using DTE and relay services. Transmission can start autonomously by programming the satellite to know when it is over a ground station or within sight of a relay satellite. It can also be triggered by a command received from the ground station or relay satellite. When a communications link is established, the radio enters a higher power transmit mode and sends the data. The flight software manages the flow control of information

Figure 11.20a: Software functions for elements within the ground segment. Credit: NASA.

into and out of the radio, making sure that no buffers are overflowed. It also formats the housekeeping and science data to be transmitted into a packetized file format that can be accepted by the ground station. Ground networks have specific data protocol standards developed from experience. For example, the NASA's NSN incorporates standards proposed by the CCSDS. The flight software unpacks received packets, retrieving the uploaded commands and data.

Software supporting the ground segment exists in the satellite, at the physical ground stations and in the MOC (server infrastructure and end-user software). The ground station uses various software for controlling the antenna, commanding, signal formatting and encoding, scheduling passes, and interfacing with the MOC. One software computes the pointing direction by using a Two-Line Element set (TLE) to define the satellite motion, an accurate model of the pointing system mount, and GPS time. It generates motor commands as a function of time. The motor controller uses these commands to actively track the satellite during a pass. During the pass, another software suite is used to monitor the link, process and encode commands for transmission, handle any signal formatting or encryption, and demodulate and decode the received transmissions. This software also manages the network connection with the MOC over which the TLE is passed, as well as data for uploading and requests for data to be downloaded. When the contact is complete, the data received from the satellite is transferred back to the MOC. The ground station may also have its own telemetry for that contact. That data is used to trend its performance. Trending the performance of each contact provides insight and notice of degradation for both the satellite and the ground station. The ground station may also use scheduling software when handling multiple missions. This software uses orbit simulation and current TLE information

to determine when the contacts are expected. It will indicate when there are conflicts between contact opportunities and can assist with schedule optimization. A schedule is generated for a given period and then programmed into the ground station control system for execution. This process can be automated, but there is typically an operator on staff to monitor the system.
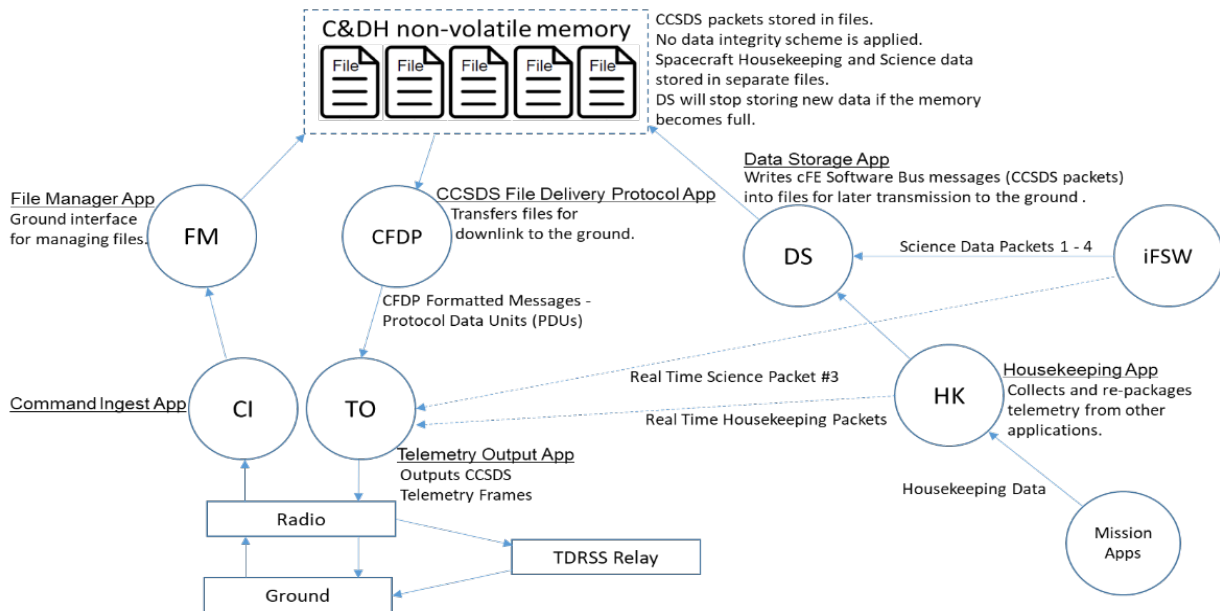

Figure 11.20b: Command and telemetry data flow for a SmallSat mission using DTE and relay services. Credit: NASA.

For the MOC, mission planning software is necessary for missions that require complex satellite behavior such as pointing at a target during science data collection. The software will include a model of the satellite dynamics and the capability of its components. The event is planned by listing a series of actions that must occur in a certain order and are spaced out by times that are approximated. The software will simulate the satellite response and then the times and actions are iteratively adjusted as needed to optimize the plan and not cause a satellite fault condition. The output of the plan is all the commands and databases that are required by the satellite. This output is submitted to the ground station ingest software for upload at a time prior to the planned event.

The SOC uses software to handle the receipt, unpacking, reconstruction and post processing of the mission science data. Using an ISS payload as an example, the science data is downlinked via TDRSS to NASA Marshall Space Flight Center (MSFC) where it is separated into different science streams and piped to the correct payload SOCs. At the SOC, but outside the company firewall, a computer is constantly running and ready to receive the data from MSFC. On that computer, the TReK software provided by NASA is running and it properly handshakes with the MSFC software assuring the data transfer. The science team periodically retrieves the data and safely brings it through the corporate firewall into the SOC. The science team writes parsing software to unpack the data which is stored in CCSDS format. They write another software to arrange the data back into the original image seen by the payload. Still more custom software will process the image to produce post-processed data products that are stored in the SOC archive and distributed to interested customers. The computer languages vary but Interactive Data Language (IDL) and Python are common choices for this type of software.

11.7   End-to-End Communications and Compatibility Testing

A SmallSat undergoes various tests through its development cycle to verify proper functionality. For the communication subsystem, end-to-end communication and compatibility testing with the selected ground network is its most critical test. Compatibility testing verifies that the ground station can properly communicate with the satellite on the uplink and downlink RF channels. Ideally, compatibility would be validated by testing the flight spacecraft with the actual ground station that will be supporting the mission. This may not be practical for larger or high-cost satellites, due to logistics associated with shipping and risk of damage. Two alternatives to shipping the satellites are typically used. One includes sending a replicate set of ground station hardware to the satellite facility for testing. A second option is to test with only the flight or an ETU radio (also common to include the flight computer) at the ground station or at a test lab configured with the ground station hardware. Drawbacks to the alternative options would include not testing the exact command path or determining whether ground sensitivity is sufficient.

For CubeSats, it is commonly feasible to bring the CubeSat to the ground station for testing. If that is not feasible, then at a minimum, the radio and flight processor (or Engineering Development Units [EDUs]) should be used. Testing at the ground station allows for the entire equipment chain to be part of the test, including the low-noise amplifier (LNA) and transmit/receive switch, if used. It is desirable to first test in a closed-loop configuration, where the satellite is connected to the ground system at the antenna port via a cable (with appropriate attenuators in line). If the satellite is fully integrated, disconnecting the flight antenna may not be feasible. In this case, a small monopole antenna located indoors near the CubeSat can be connected to the ground system. The monopole antenna connection to the ground system may vary depending on the ground antenna configuration but should include as much of the ground system electronics as practical.

Some missions elect to include an outdoor open-loop test with the CubeSat and ground antenna. This method allows for the entire ground system, including the ground antenna, to be included in the test. However, the ground antenna typically cannot point directly at the CubeSat due to mechanical limitations or to limit the received signal so the ground system RF components will not be overdriven. Off-pointing and reflections from the ground and local structures can also make it difficult to achieve a valid test.

End-to-end network testing primarily validates the ground station to MOC interface. This test verifies that the MOC can properly receive downlink data from the ground station and verifies that the ground station can receive and process uplink command data from it. Initial end-to-end testing will validate network connectivity, showing that network connections can be established and firewall rules at the ground station and MOC are in place. Once network connectivity is established, the MOC can transmit commands to the ground station for capture. The ground station can then transmit simulated or recorded data to the MOC for validation.

It is preferable to conduct initial end-to-end network testing prior to compatibility testing. In cases where the satellite can be brought to the ground station, a full end-to-end test can be conducted. Command transmissions from the MOC, through the network and ground system to the satellite can be validated. A complete end-to-end telemetry dataflow from the satellite to the control center can also be validated.

11.7.1   End-to-End Hardware for Ground Systems

A complete ground system can be provided as a kit with all the necessary components bundled together and setup to work seamlessly. These end-to-end solutions include the antenna, its controller, and the RF feed with all the necessary filtering and low noise amplification for the particular wavelength of interest. They use a software defined radio or a dedicated transceiver to convert between digital packets and RF waveforms. Software is included to process the satellite

position and direct the antenna to track it. Additional software is used to archive and display the information within the digital packets. Three vendors, GAUSS, Innovative Solutions In Space (ISISPACE) and GomSpace, listed in Table 11-12 provide solutions for the low-cost CubeSat and small satellite market. One vendor, Surrey Satellite Technology Limited, offers a higher end system, installation service, and personnel support. The final vendor listed, Kratos, offers a different end-to-end solution that begins with the digitized RF waveform. The Kratos Quantum software then demodulates, filters, unpacks, parses, displays, and archives the data (17).

Table 11-12: End-to-End Hardware for Ground Systems			
Product	**Manufacturer**	**TRL**	**Type of Product**
Complete Ground Solution	GAUSS	9	Small satellite provider offering a complete ground solution. UHV, VHF, and S-band
Complete Ground Solution	ISISPACE	9	Small satellite provider offering a complete ground solution. UHV, VHF, and S-band
Complete Ground Solution	GomSpace	9	Small satellite provider offering a complete ground solution. UHV, VHF, and S-band
Surrey Ground Segment	Surrey Satellite Technology Ltd.	9	Major contractor offering a complete ground solution. S-band uplink/downlink and X-Band downlink capability
Quantum	Kratos	9	Major contractor with a complete ground solution

GAUSS Ground Station Kit

The GAUSS ground station is a turnkey solution. It can be configured with UHF, VHF and S-band on the same pointing system. An example of the associated hardware is shown in Figure 11.21.

Figure 11.21: (left) GAUSS ground station hardware, transceiver and (right) tracking antenna. Credit: GAUSS Srl.

Hardware features of the systems offered include (22)(23)(24):

- High gain Yagi-Uda VHF and UHF antennas (>16 dBi for UHF)
- Low-noise amplifiers and band-pass filters for VHF and UHF bands
- Low-loss RF coaxial cables
- 1.5-meter parabolic dish for higher frequencies downlink (up to 6 GHz, default feed is for S-band)
- VHF: uplink and downlink up to 100 W using radio and Terminal Node Controller (TNC),

software defined radio (SDR) optional

- UHF: uplink and downlink up to 70 W, using radio and TNC, SDR optional

- TX using ICOM-9100 hardware, RX recording and decoding via SDR

- Several RF and electrical fuses for lightning protection

- S-Band: downlink using SDR for recording and post-processing of I/Q RF data

- Az/El rotor for high-torque maneuvering

- Hardware components power switch on/off to minimize power consumption

- Full HD camera for instant antenna monitoring and picture logging

The features of the software that accompanies the system include:

- Automatic TLE download from publicly available repositories

- SGP4 propagator as suggested by USAF NORAD's Space-Track

- Rotor control (compatibility with several rotor controllers, e.g. Yaesu, RF Hamdesign)

- Assisted rotor pointing calibration and verification using Sun position

- Fully compatible with ICOM-9100 satellite radio and GAUSS USB ground dongle

- Separated Doppler shift corrections for uplink and downlink frequencies

- DUPLEX TX/RX mode

- Instant weather check and logging to operate the ground station safely

- Lightning detection for safe antennas operation

- Instant logging of all subsystems operation

- Ground map with live Earth clouds

- Compatible with several TNCs (Kantronics, Symek, Paccomm, Kenwood)

- Email report to ground station operators

- Instant email alerts for non-nominal conditions of the satellite or GS hardware components

- Session programming for weeks of unattended ground station operations

- GUI command recording for easy session programming

- One button programming to include a whole set of commands in the session

- Manual override during pass for last-minute command addition

- Control and handling of multiple satellites using configurable priorities

- Satellite TLM decoding, graphing, and archiving into a database accessible by web

- Integrated satellite payload data handling and decoding (e.g., for image file processing)

- TCP/IP connections for remote ground station & TNC operations

Innovative Solutions In-Space Ground Station Kit

The ISISPACE small satellite ground station is a low-cost, turnkey solution that is designed to

communicate with satellites in low-Earth orbit that operate in either amateur frequency bands or commercial bands. The frequency bands covered are S-band, UHF, and VHF. The ground station consists of an antenna and a 19" rack which houses the transceiver, rotor control and computer which make the system very compact. Examples of these components are shown in Figure 11.22. The transceiver makes use of a SDR that provides flexibility to swiftly reconfigure modulation/coding/data-rate on the run. Most commonly used modulation schemes and coding methods are already implemented, and any customization requests can also be handled (25).
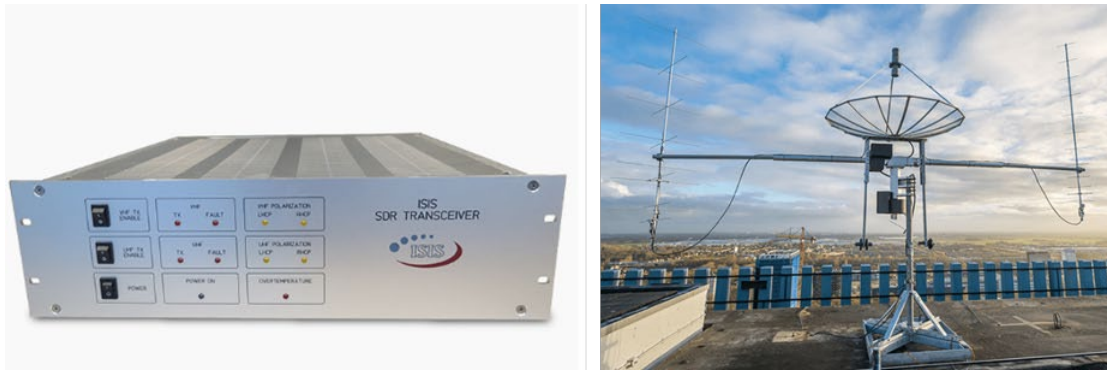
Figure 11.22: (left) ISIS ground station hardware, transceiver rack and tracking antenna (right). Credit: ISISPACE.

GomSpace Ground Station Solutions

GomSpace offers two products (A) NanoGround, and (B) Hands-off Operations Platform (HOOP) that can be used as ground segment building blocks for a wide variety of missions.

A.NanoGround provides a communication gateway for spacecraft using GomSpace AX2150 and NanoCom Link S/SX S-band and X-band radios. NanoGround provides TMTC link over S-band as well as IP networking over S and X-bands and is designed to be easily integrated with commercial ground station service providers. As of September 2024, NanoGround offers built-in integration for KSAT's ground station network. AWS Ground Station and LeafSpace ground station network support will be released in the near term. Other ground station networks are being considered and can be discussed based on client need.

B.HOOP is GomSpace's Mission Control software solution, providing autonomous "hands-off" satellite operations for single spacecraft and constellations. HOOP can either be operated by GomSpace's Mission Operations Team for client missions or it may be purchased by clients on a subscription basis to manage their own missions. HOOP comes pre-built with integration for GomSpace NanoGround, providing out-of-the-box solution to operate missions flying GomSpace satellites. HOOP's capabilities will be expanded in the near term with an up-coming release to support CCSDS communication protocols. This expanded capability will allow HOOP's application domain to include 3rd party satellites. HOOP has been in operation since 2020 and currently manages LEO missions. As of September 2024, five additional HOOP missions are in development and planned for near-term execution (26).

Surrey Satellite Technology Ltd. Ground Station Kit

Surrey can provide complete turnkey ground segment solutions for a range of space platforms, including all the hardware and software necessary to operate, maintain, process and archive data. Services provide by Surrey include:

- S- and X-band ground stations with full motion antenna systems from 2.4 meter to 7.3 meter in diameter, with radome options available for harsh climates

- SSTL Pilot Satellite Control Software
- Mission planning systems
- Radiometric and geometric image processing
- Data storage solutions
- Site surveys, ground segment installation and training
- Technical and maintenance support packages

In addition, Surrey can work with customers to integrate their ground segment solutions with existing ground infrastructure or with 3rd-party ground station networks (27).

Kratos Ground Station Solutions

Kratos virtual ground solutions begin with the OpenSpace SpectralNet digitizer converting analog signals at RF frequencies up to S-band into digital IF VITA-49 or DIFI IP packets. Kratos quantum virtualized network functions (VNFs) then process the digitized RF waveform. Kratos virtualized network functions run on general-purpose compute and can be dynamically instantiated for different mission directives at the click of a button. Figure 11.23 provides a visualization for the system concept.
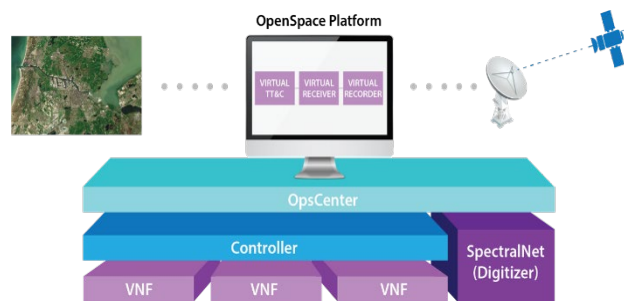
Figure 11.23: Visualization for the Kratos quantum system concept. Credit: Kratos.

All components are available separately to support an existing C2 solution or third-party ground network with existing signal processing and antenna resources. The quantum products include:

1) quantumCommand (qCMD), COTS software application for command and control (C2) of small spacecraft;

2) quantumFEP (qFEP), connects C2 systems to RF signal processing equipment: handling command and telemetry stream formatting, encryption/decryption devices, CCSDS processing, and network interfaces to either qRADIO or third-party ground antenna networks;

3) quantumRadio (qRadio), the software modem for RF signal processing on premise or in the cloud;

4) quantumDRA (qDRA), a data recording and archiving application supporting CCSDS/non-CCSDS header and channel data routing with IP-based interfaces;

5) quantumRX (qRX), a fully virtualized wideband software receiver, specifically tuned to streaming earth observations in near-real time with 500 MHz bandwidth using Digital IF digitizers;

6) quantumTX (qTX), a fully virtualized wideband software transmitter, specifically tuned for Earth observation and remote sensing satellites with over 1Gbps throughput for uplinks.

In 2021, Kratos introduced a virtual, software-defined architecture solution called the OpenSpace Platform. As an enterprise level, end-to-end system, it provides satellite operators and service providers the flexibility to scale on-demand as their operations grow in size and capability. By leveraging Digital IF over IP with time deterministic latency and software defined networks, the platform allows virtualized functions such as modems, channelizers, recorders and combiners to be orchestrated on generic servers in a ground station or public or private cloud environment. The

virtual architecture lends itself to upgrades and/or updates automatically, ensuring ongoing
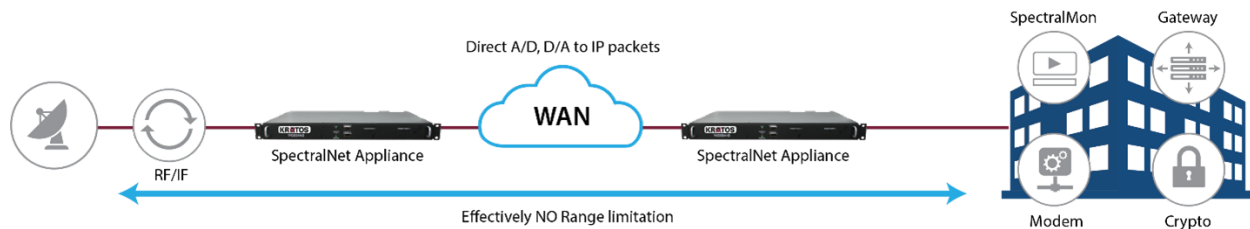
Figure 11.24: Kratos OpenSpace architecture concept keeps most of the RF ground equipment remote from the ground station. Credit: Kratos.

reliability and security. In addition, the ability to test software releases in real-time allows ground equipment functions to be included in continuous integration and continuous delivery cycles. Software defined architectures are more agile, programmable, and automated, enabling the ground system to work in tandem with dynamic satellite payloads. By shifting from RF signals and analog equipment to a virtualized, IP-based infrastructure, orchestration can occur on the fly. Figure 11.24 provides an illustration of the OpenSpace architecture concept (29).

Remos Space

Remos Space offers a versatile multi-mission software modem called Expedite that is designed specifically for the new space segment. Expedite can reconfigure modulation, coding, and data rates on-orbit, allowing it to adapt to different mission requirements dynamically. There are three configurations available: a plug-and-play system can be integrated directly into existing ground systems and includes a COTS 4U server with an integrated SDR and Remos software; the turnkey solution includes the software modem and necessary hardware such as antennas, sequencers, High-Power Amplifiers (HPAs), Low-Noise Amplifiers (LNAs), and other related components for a complete ground station setup; and the pure soft modem allows customers to deploy the modem software on their existing infrastructure, with licenses charged annually. Expedite is highly compatible with various spacecraft transceivers and does not require a specific transceiver for integration, making it a flexible choice for various mission profiles. Their antenna size ranges between 1.9 – 2.4 m with plans to supply larger apertures in the future. Remos Space is currently adding VITA49 and DIFI to Expedite that will enable seamless integration of the pure software modem with Amazon ground stations.

11.8 Cyber Security

Security of a space system needs to consider all aspects of the system, including the space platform, payloads, and all supporting functions. To a remote attacker, the most accessible portion of a spacecraft is the end-to-end command path as accessed through direct contact (RF link), subversion of any command path transports (space or ground networks), and subversion of the command authority (e.g., MOC). Another enticing option for an attacker is to cause a mission impact through manipulation of key space system dependencies, such as the Global Navigation Satellite System (GNSS), ground stations, and external service or data providers. Related concerns include degrading or denying the use of the command path through jamming or denial of service of the command path and supporting functions. Finally, vulnerabilities in any system, sub-system, or component, can be exploited by an attacker in creative ways to ultimately gain the ability to affect the overall system. Supply chain risk management is needed to address some portions of these challenges.

Effective space system security efforts begin early in the lifecycle and continue through to mission termination. Early architecture and design decisions will have the most significant impact on the

overall system's security outcomes and can help avoid rework in later phases. Integrating cybersecurity capabilities into a system fits well into standard system engineering practices. Many organizations are establishing a sub-discipline for systems security engineering, to serve as a focal point within the mission team to enable system security. This may overlap with or work closely with related roles such as a cybersecurity systems engineer. The International Council on Systems Engineering (INCOSE) has established a working group to support system security engineering, and NASA is actively implementing a system security engineering capability (30).

Operators need to maintain command authority over their spacecraft, preventing unauthorized access. Use of authenticated encryption, between the point where the command sequence is generated and the spacecraft, is the best method to ensure command authority and data integrity. Encryption provides confidentiality, and authentication ensures that a trusted source initiated the command. This approach generally addresses initial concerns about an attacker attempting to gain access through use or subversion of the command path to the spacecraft.

Key spacecraft external dependencies can be manipulated to cause a mission impact. Example dependencies might include the ground station and GNSS signal. Ground station impacts could disrupt data flows with the spacecraft. Jamming and measurement spoofing of GNSS signals has become common in various regions, with observed impacts in the maritime and airspace domains. Orbiting systems have detected similar effects. Space systems should be prepared to tolerate loss or interference with GNSS signals.

For the overall space system and its component parts to function properly and securely, each part must in turn be sufficiently secure. Traditional cybersecurity efforts apply well to most software and terrestrial systems. Attention to the security of each component, as well as ensuring the interplay between components is secure, will help protect the overall system. On-board security should also be considered, particularly for multi-customer and multi-payload spacecraft. Increasingly, spacecraft developers should consider that external defenses (e.g., command authentication or encryption) may be bypassed or subverted (similar to how network firewalls may be bypassed). Without further consideration, the on-board systems are likely vulnerable to further exploitation.

Supply chain risk management is an essential related discipline. Particularly for key components and software, understanding the vendor's sourcing, manufacturing approach, and cybersecurity and assurance management will help ensure any associated risk can be identified and managed. If a vendor relies on other vendors, additional scrutiny may be appropriate. Review of the sourcing may address whether the vendor is rebranding, assembling/integrating, or internally manufacturing components, and whether there is sufficient control to deliver a trustworthy product. Reviewing the manufacturing approach, whether hardware, software, or some combination, allows the customer to determine whether the vendor uses repeatable processes that yield deterministic and trustworthy results. And understanding how the vendor addresses cybersecurity, quality, and protection topics in their components provides insight into whether or not it is appropriate to integrate the component into the overall system.

U.S. National Guidance and Regulations

In September 2020, the U.S. National Space Council issued Space Policy Directive 5 (SPD-5), Cybersecurity Principles for Space Systems. Amongst other elements, the directive calls for use of "risk-based cybersecurity-informed engineering," anticipating and adapting to evolving malicious activities, and recommending capabilities to maintain positive control of space vehicles. Another element implies that Federal agencies may issue or update guidance, rules, or regulations to adopt the principles in this directive (31).

U.S. regulatory agencies, such as the Federal Communications Commission (FCC), have

considered and not yet issued rules that may require specific cybersecurity measures to be adopted. Prior proposed (and not issued) rules included requirements for encryption on the telemetry, tracking, and command communications for propulsive spacecraft.

U.S. Agency Guidance

Several U.S. government agencies and their support ecosystem have made various frameworks, standards, and other guidance available to address cybersecurity and protection concerns.

NASA Technical Standard

NASA requires its missions, including small satellites, to comply with NASA STD-1006 "Space System Protection Requirements." This standard covers protection of the "command stack," critical information, Position Navigation and Timing (PNT) sub-system resilience, and reporting detected and unexplained interference. The tailoring guidance within the standard allows for some flexibility in certain small satellite scenarios, such as non-maneuverable systems. Protecting the command stack involves use of encryption complying with FIPS 140 (level one). PNT resilience addresses the loss of or temporary interference with external PNT services, such as a GNSS (32).

National Institute of Standards and Technology (NIST)

NIST has several useful publications addressing cybersecurity and system security engineering. The NIST Cybersecurity Framework is a voluntary guidance framework that can be used by organizations to manage cybersecurity risk. The framework provides a structured and tailorable approach for organizational security capabilities and includes informative references to other NIST documents (e.g., SP 800-53), as well as other standards or guidance organizations (e.g., International Organization for Standardization, ISO). NIST's SP 800-160 Volumes I and II offer a thorough approach for system security engineering practices. In particular, SP800-160 Vol I's Appendix F, Design Principles for Security, can be used as an effective foundation for systems security (33)(34)(35)(36).

Additional Resources

Two U.S. Federally Funded Research and Development Centers (FFRDCs), the Aerospace Corporation (37) and MITRE (38), have published guidance for space system cybersecurity. These FFRDCs are expecting to make additional recommendations widely available. Aerospace's "Defending Spacecraft in the Cyber Domain" includes a brief survey of known cybersecurity initiatives and standards, challenges with legacy engineering approaches, emerging threats, and principles for "cyber-resilient spacecraft." The paper also includes a section specific to small satellites. MITRE has published a paper "Cyber Best Practices for Small Satellite" that briefly addresses cyber threats to space systems and includes a discussion on applying lessons learned from other industries to space systems.

Consultative Committee for Space Data Systems (CCSDS)

CCSDS provides a variety of guidance documents for implementing security measures in various aspects of the mission. For an introduction to the CCSDS approach and available guidance, see CCSDS 350.7-G-2 "Security Guide for Mission Planners" that provides a perspective on approaching security in space systems (39). The CCSDS 350.0-G-3 "The Application of Security to CCSDS Protocols" informative guidance document provides an introduction and discussion on various topics, including protecting the command path (40).

11.9   State-of-the-Art – Ground Data and Supporting Systems

11.9.1   Technologies

Multiple Spacecraft Per Aperture

The Annual Small Satellite Conference on the grounds of Utah State University is the premier event amongst small satellite stakeholders, and its themes reflect the trends of the times. In 2021 one of the key talks was on how to "Maximize Contact Availability of SmallSat Clusters through MSPA Technique on GSaaS," (41).

As scientists are increasingly interested in characterizing fields (going beyond single point measurements) requiring swarms of satellites, as well as the emergence of Distributed Satellite Missions (multiple satellites working in concert towards one common goal), MSPA is a critical enabler. While it is not a new concept, few ground stations have invested in such upgrade. The DSN has the capability to track multiple spacecraft per antenna (MSPA) (up to four) if they are all within the scheduled antenna's beam. The 34 m antennas at each complex can be combined into an array, with or without the co-located 70 m antenna. The combined G/T depends on several factors but is approximately increased by the sum of the antenna areas from the arrayed apertures minus approximately 0.3 dB combining loss. For instance, arraying four 34-meter antennas results in an increase of 5.72 dB.

Automation and Modeling

The MOC of the future will include a "lights out" or fully automated option. This requires software on the ground station side to run the antenna automatically. Automation software will receive a list of times that the antenna should track the satellite and it will manage that list. It will send TLEs and data to the antenna with no one present, receive downlinked telemetry, and archive it. Software automatically parses the telemetry, compares key watch items to defined limits, and alerts the team via email or phone text message. FreeFlyer by AI solutions combine astrodynamics/ spacecraft propagation, coverage and contact analysis (including swarms), attitude and maneuver modeling, and orbit determination (42).

Large Ground Antennas: to the Moon and Beyond

For years there had been a gap between NSN's largest 18m and DSN's 34/70m antennas, and such large antennas were not available from commercial ground providers. This gap has now been filled.

In 2022 Viasat introduced new 19/24m aperture antennas (Figure 11.25) at their Antenna Systems campus in Duluth, GA supporting several ongoing programs. The size and architecture of these larger apertures support current programs while offering the flexibility and scalability to support future and forward planning missions (43).

Figure 11.25: Viasat's new large-aperture space-to-ground communication antennas are ready to support lunar, cislunar, deep space and DoD missions. Credit: Viasat.

NASA Lunar Exploration Ground Sites

SCaN announces Lunar Exploration Ground Sites (LEGS) to provide direct-to-earth communication and navigation services for missions operating from 36,000 kilometers (km) in the

GEO to cis Lunar and other orbits out to 2 million km. To fully support distant orbits there will be three LEGS sites equally spaced around the Earth. The Ground sites use CCSDS Modulation and coding schemes for forward and return data. Specialized/unique Mod-Cods are optional. User Local Equipment on-site is optional. The 18m assets are listed as White Sands, USA: 32.544863, -106.612504 Matjiesfontein, South Africa: -33.231224, 20.58163 (TBD) and Pacific Region TBD. MSPA is planned for up to 4 simultaneous return services per aperture (Max 3 for Ka). Use of LEGS for other than Artemis support is TBD. See Table 11-13 for projected performance of LEGS assets.

Table 11-13: Projected Performance of LEGS Assets Pending Finalization			
RF Performance Criterion	Radio Frequency Performance (Return)		
	S-Band	X-Band	Ka-Band
G/T (minimum)	28 dB/K	39 dB/K	47.5 dB/K

11.9.2 Ground Aggregators

Table 11-14 lists those Ground Service providers who own and or operate their own brand of ground assets. Irrespective of the nature of ownership, satellite operators are reliant on the limited ground stations they have access to. Satellite operators have well defined windows for exchanging information with their satellites. To meet evolving demand from within the fast-growing segment of the space industry, multiple aggregator models have evolved from private market participants. Services from companies such as RBC Signals, Infostellar, Amazon Web Services, and Spaceit are offered through specialist ground station capacity aggregator platforms. These are digital solutions enabling ground station operators to provide their excess capacity to a global user base. Since this is very similar to the business model of Uber, these aggregator services represent the ongoing "Uber-ization" of ground station services within the space industry. The downstream service markets are observing new players with new products and services. With increasing competition, the differentiating factors are shrinking in number. When the upstream capabilities start resembling each other, the key differentiators will include the ability to communicate with the satellites on-demand (44).

RBC Signals

RBC Signals is provider of global satellite data communication products and solutions. It offers secure space communication solutions in every major frequency band using a worldwide network of company-owned and partner-owned systems. RBC Signals delivers dynamic solutions offering affordability, flexibility, and resiliency. Its diverse products and services offer complete end-to-end solutions for best-in-class multi-network solutions (see https://rbcsignals.com).

RBC has aggregated a growing network of over 80 antennas in nearly 60 locations worldwide offering unmatched capabilities. A map of these locations is shown in Figure 11.26. As of 2024, RBC owns about 20% of the ground stations, and the rest are partner stations. For customers needing turnkey access to existing antennas, RBC Signals offers ground station antenna-as-a-service, with the flexibility to secure unlimited satellite passes ("core") or 'pay-by-the-pass/minute/GB'. This is made possible through a combination of their own network of highly capable systems and the unique 'sharing economy' model, wherein they leverage the unused excess capacity of dozens of partner-owned antennas worldwide. RBC Signals also offers turnkey bring-your-own-antenna hosting solutions that pair customer-owned equipment with reliable, high-end ground infrastructure almost anywhere in the world. RBC employs a distributed compute architecture where most processing occurs at a data center/cloud, with some processing on the satellite or at

Figure 11.26: RBC Signals ground network map September 2024. Credit: RBC Signals.

the terrestrial edge at the ground station. RBC Signals can host AWS and Microsoft on premise cloud infrastructure, as well as virtual servers at the ground. Most recently, RBC has a new additional service for intersatellite links called Go.BIC, a partnership with IQ SpaceCom and Viasat (45).

Table 11-14: Service Providers for DTE Ground System Network			
Product	Dish Sizes	Services	MSPA
ATLAS Global Network	Various partners with other stations	S-band, X-band, UHF (Ka-band in 2017) Built on AWS cloud infrastructure	Partner dependent
KSAT and KSAT^{LITE} by Konsberg Sat. Services	> 10 m part of NSN or 3.7 m (KSAT^{LITE})	X-band and S-band D/L and S-band U/L. VHF, UHF, Ka-band D/L. KSAT^{LITE} designed specifically for SmallSats	No
SSC Infinity by Swedish Space Corporation	13 m, 7.3 m NSN partner	Designed specifically for SmallSats; Uses standardized HW	Not found
AWS Ground Station by Amazon	5.4 m	Built on AWS cloud infrastructure	Not found
Viasat	7.3 m, 5.4 m	Global network operating in S, X, and Ka-bands that can reach LEO, MEO, GEO, and HEO orbits	Mission Dependent
Leaf Space	3.7 m	Standardized global network accessible through UHF, S, X and Ka-band	Yes

NASA, Near Space Network	9 to 11 m, & 5 m	Global network operating in S, X, and Ka- bands that can reach LEO, GEO, HEO, and Lunar orbits; up to 2 mil km	Legacy: NoLEGS (18m): planned, Yes
NASA/ JPL, Deep Space Network	34 m, 70 m	Operating at S, X, K, Ka bands. Includes Morehead State 21m in X-band. 8 m optical receive aperture planned for 2030s	Yes
NASA UHF Ground Station	18 m	Operates in UHF (400 – 470 MHz)	No
RBC Signals Global Ground Station Network		VHF, UHF, S, C, X, Ku, and Ka-bands	Partner Dependent

11.9.3   Scheduling and Mission Operations Software

With the growing number of ground operators and aggregators, to take advantage of the plethora of assets, scheduling is emerging as the single most important enabler. As individual providers may have their own scheduling formats, for seamless operations, a common scheduler is critical, and this is true for the NASA commercialization efforts as well.

Scheduling: InfoStellar

InfoStellar offers communication services in the VHF, UHF, S, X, and Ka bands. Table 11-15 lists the frequency bands offered by InfoStellar which can be filtered by the tool on their website. Figure 11.27 shows all 34 antennas on the platform that are either live or for which integration is planned. With multiple commercial small-satellite operators in the market, the need for enhanced mission operations is much more than an industry-wide requirement (44).

Table 11-15: Select Frequency Bands by InfoStellar			
Downlink		Uplink	
X Band 8 – 12GHz	S Band 2 – 4GHz	S Band 2 – 4GHz	UHF Band 300MHz – 1GHz
Ka Band 27GHz – 40GHz	None No Uplink Channel	VHF Band 30 – 300MHz	None No Uplink Channel

The following section provides an overview of mission operations and scheduling software products that can be integrated into a MOC (see Table 11-16). While the specific aspects of each of these products is discussed below, they all have some common features. In general, these software applications cover functions related to mission scheduling and tasking, commanding and telemetry, and monitoring and control. Many of them also have automation features that enable "lights-out" operations or reduced manpower requirements.

All these products are highly customizable. They can not only adapt to multiple missions, satellites, and ground stations, but these products also allow for customized visualizations, analyses, and user interface views. Additionally, many of these products are cloud-based or have a web interface to enable easier access for an operator from almost anywhere.

Figure 11.27: All 34 antennas on the StellarStation platform that are live (25 - red) or for which integration is planned (9 - blue). Credit: InfoStellar.

Table 11-16: Mission Operations and Scheduling Software			
Product	**Manufacturer**	**TRL**	**Type of Product**
COSMOS	OpenC3	9	Open-source command and control system that can be used in all phases of testing and operations
Galaxy	The Hammers Company	9	Command and telemetry system that has been available since 2000
Orbit Logic Family of Products	Orbit Logic	9	Group of mission planning and scheduling products for both aerial and satellite imaging applications
Yamcs	Space Applications Services	9	Open-source command and control system
ACE Premier Family of Products	Braxton Technologies	8+	Group of hardware and software components for end-to-end Satellite Operations Center (SOC)
Mission Control Software	Bright Ascension	8+	Monitoring and control interface with "lights-out" automation features built-in
Major Tom	Xplore	8+	Cloud-based command and telemetry system that can interface with some COTS flight software

OpenC3 COSMOS

OpenC3 COSMOS is a free, open-source, open-architecture, command, control and communications system providing commanding, scripting, and data visualization capabilities for embedded systems and systems of systems. With the release of version 5, COSMOS is now a fully containerized, microservice-based architecture with a web frontend. COSMOS is intended for use during all phases of testing (board, box, and integrated system) and during operations. OpenC3 COSMOS is made up of a set of applications that can be grouped into four categories: real-time commanding and scripting; telemetry visualization; offline analysis; and utilities. Figure 11.28 shows how data flows through the microservices and is made available to users through an API and from a web-based interface. Any embedded system that provides a communication interface can be connected to COSMOS. All real-time communications of both commands and telemetry are logged in a cloud-native data store, which can use local hardware, or cloud-based buckets for potentially infinite storage.
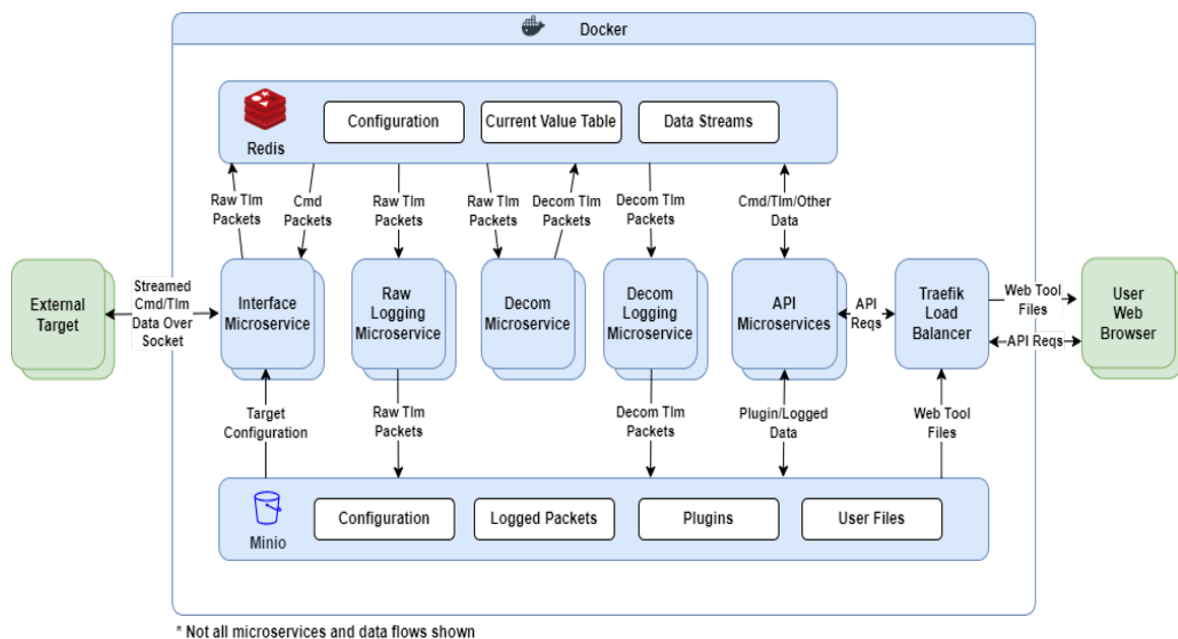
Figure 11.28: COSMOS5 architecture and context diagram. Credit: OpenC3, Inc. https://openc33.com/

Additionally, program specific tools can be written using the open-source OpenC3 COSMOS libraries, and these tools can interact with the commands and telemetry of all targets connected to the system. A paid enterprise edition of COSMOS is also available that adds on important scaling and security features including: user accounts, Role Based Access Control (RBAC), Kubernetes support with helm charts, calendar and automation capabilities, and a library of prebuilt plugins for common devices and protocols like SCPI devices and CFDP (45).

Galaxy

GALAXY is a real-time command and telemetry system intended for use throughout the spacecraft, instrument, and/or component life cycle from development through integration and on-orbit operations. GALAXY has been used on missions ranging from stratospheric balloons and CubeSats through great observatories and interplanetary missions. GALAXY can accept telemetry from, and send commands to, multiple spacecraft and ground stations simultaneously. It can perform uplink authenticated encryption and downlink decryption and authentication. Users can customize GALAXY for a particular mission via a database in which they provide telemetry and command specifications. Users can design telemetry displays, plots, sequential prints, configuration monitors, and spacecraft commands and table loads in simple text files stored on the host computer's file system. GALAXY is CCSDS compliant, and can communicate over a wide variety of transports and protocols including TCP/IP networking (optionally with TLS), synchronous and asynchronous serial ports, SpaceWire, MIL-STD-1553, and the GMSEC message bus (47).
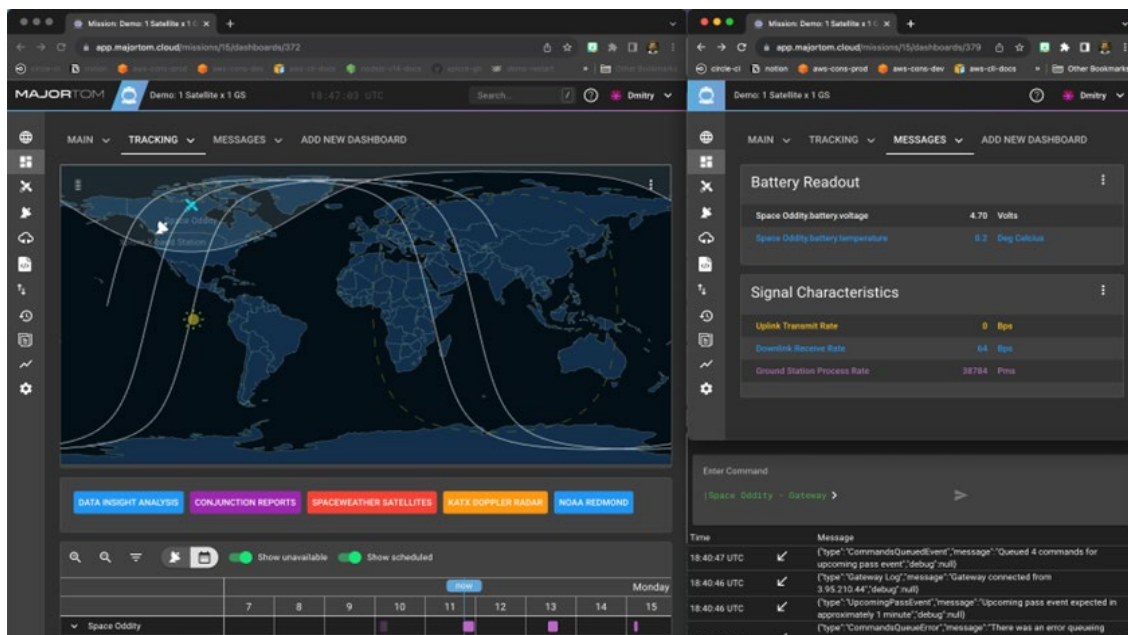
Figure 11.29: Major Tom's configurable dashboards allow operators to oversee and interact with the mission in the way they are most comfortable. Credit: Xplore.

Major Tom

Xplore's Major Tom® is a scalable, cloud-based mission control software that provides satellite mission operations and planning tools for ground station scheduling, satellite tasking and telemetry monitoring. It provides satellite operators the ability to integrate and control ground segment applications and services, and further de-risks mission operations with features that include out-of-the-box ground network integrations, data analytics, real-time dashboards, and a customizable commanding API. Major Tom's gateway API architecture, definitions, and protocols can integrate with custom ground station provider(s). This software is compatible with a heterogenous network of ground station providers and supports constellation and mission operations flexibility in satellites communication and commanding. New ground station network locations and providers can be added, and the same data model will unify accessing their functionality. Interface capabilities include control panel scheduling and pass management as well as data interface for downlink and uplink. Integrated ground station providers are ATLAS Space Operations, Inc., Leaf Space and Microsoft Azure Orbital. Figure 11.29 provides a screenshot of

the dashboard (48).

Orbit Logic Family of Products

Orbit Logic specializes in mission planning, scheduling, and space situational awareness software. The software suite consists of multiple applications that support analysis and operations for aerial and satellite imaging and space-to-ground networking. The mobile, web, desktop, and onboard scheduling applications have a variety of features, including: configurable systems, constraints, and goals; high performance algorithms; deconflicted scheduling plans; visualizations and animations on the user interface, and flexible process flows and automation. Figure 11.30 shows Orbit Logic's Collection Planning and Analysis Workstation (CPAW) (49).
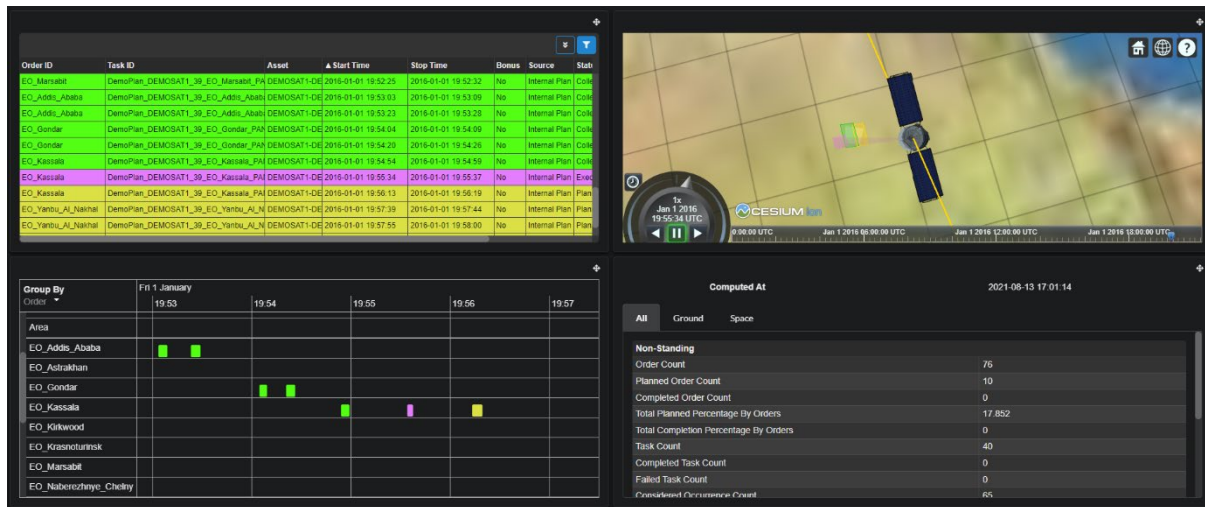
Figure 11.30: Orbit Logic CPAW dashboard. Credit: Orbit Logic.

ACE CrtlPoint

The ACE CrtlPoint from Parsons (acquired Braxton Technologies January 2021) is an automated space vehicle and ground station command and control (C2) application with a plug-in architecture that provides nearly lights-out TT&C operations. ACE CrtlPoint includes the hardware and software components necessary for a satellite MOC. Key applications include: integration with antenna scheduling; ground station control and status; data forwarding for analysis; command plan execution; anomaly detection; and a turnkey TT&C system. COTS capabilities plug into standardized environments, allowing the product to be ready immediately within a range of mission architectures (50).

Bright Ascension Mission Control Software

In addition to their tried and tested mission control software designed for easy-to-use monitoring and control during development and flight, Bright Ascension is currently developing an end-to-end suite of highly integrated software solutions for the entire mission from flight software development, testing and simulation, to ground operations and end user insights and analytics. As part of the suite, the company will offer two new products for mission operations – Groundkit, which provides reusable components and services for assembling complete and bespoke ground software systems, and Ops, a suite of applications for mission operations designed to reduce the complexity and cost of operations. The early versions of both products are expected to be released in the first months of 2025.

Yamcs

Yamcs is a mission control software developed by Space Applications Services that is publicly

available under the GNU Affero GPL license. It is designed for the command and control of the spacecraft, payload, ground station, and ground equipment, and supports telemetry reception, archiving, telecommand sending, alarm generation, and replay processing. Yamcs features a comprehensive web interface. The backend can be extended or customized using Java (though some plugins are only available to customers) and can be deployed in the cloud or on-premises. The platform also supports industry standards like Consultative Committee for Space Data Systems (CCSDS), CCSDS File Delivery Protocol (CFDP), CubeSat Space Protocol (CSP) and XML Telemetric and Command Exchange (XTCE). The extensive API includes both REST and WebSocket. A desktop tool is available for authoring operator displays. Plugins for popular open-source visualization frameworks, such as Grafana and OpenMCT, are also available. A companion tool, the Yamcs Gateway, enables the development of Electrical Ground Support Equipment (EGSE) for spacecraft and payload development, supporting low-level interfaces like CAN-bus, MIL-1553, SpaceWire and RS422. Figure 11.30 is a screenshot of the web GUI.
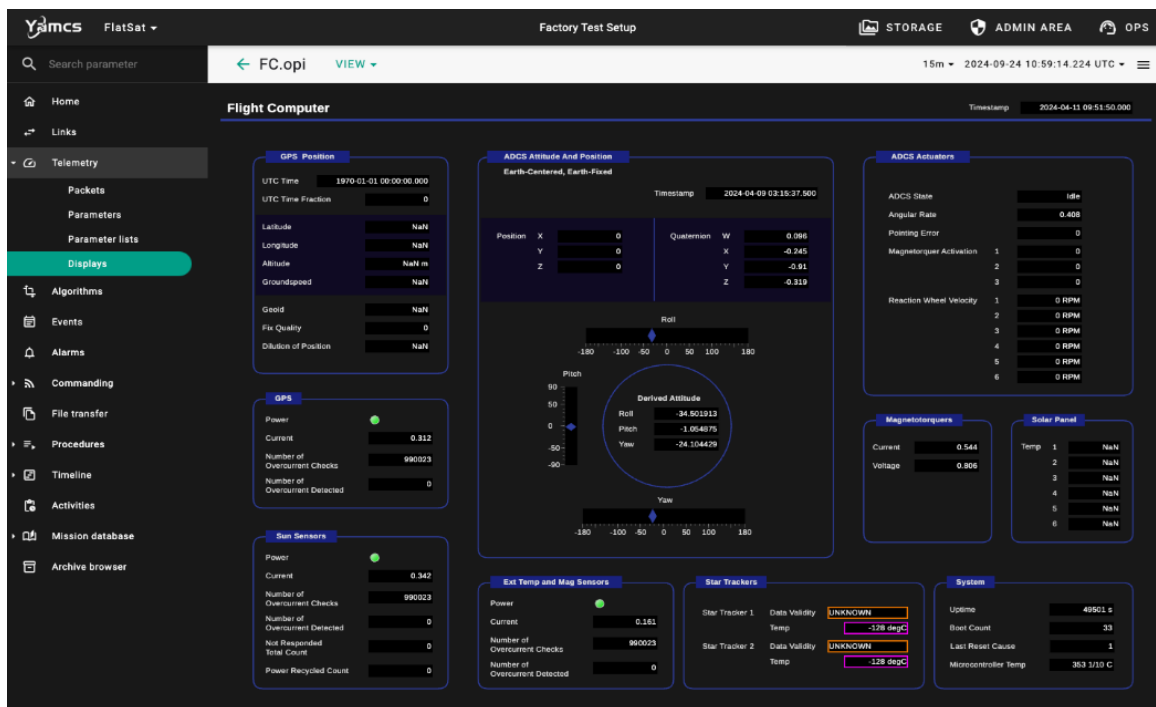
Figure 11.30: Screenshot of Yamcs web GUI. Credit: Yamcs.

11.10   On the Horizon

Ground data systems must continue to evolve to keep up with the furious pace of small satellite technology. Advancements in onboard processing and data storage will demand more capability in getting data to the ground. Mass production of small satellites is quickly becoming a reality and large constellations are now starting to find their way to orbit. This will require ground system technology that can communicate with multiple satellites simultaneously. Free Space Optical communications and phased array ground systems are emerging solutions to these needs. While both technologies have seen years of investment, they are now just starting to find their way into the ground networks. While it may still be years before becoming a staple for these networks, the following sections provide insight to the state of these technologies and where they are headed in the future.

11.10.1   Free Space Optical Communications

Increasing demand for data from NASA missions has led to a migration over the past few decades to increasingly higher RF bands (X, K, and Ka) and ultimately to the optical and near-infrared regime. Free Space Optical (FSO) communications are expected to increase data rates by two orders of magnitude over traditional RF links (see Communications chapter for more on FSO communications). The next generation systems will incorporate optical communications, and several early flight demonstrations and uses of optical communications in the coming decade are expected to be transformational for NASA and other space organizations. Whereas Ka-band frequencies go up to 40 GHz frequency, the optical signal reaches up to 200,000 GHz. Higher frequencies have the potential for huge increases in data rates, theoretically proportional to frequency-squared if all other factors are equal. At optical wavelengths, other factors, such as atmospheric losses, receiver sensitivity, aperture, and power, must also be considered, but nonetheless, optical communications offer the potential for orders of magnitude improvement in data throughput.
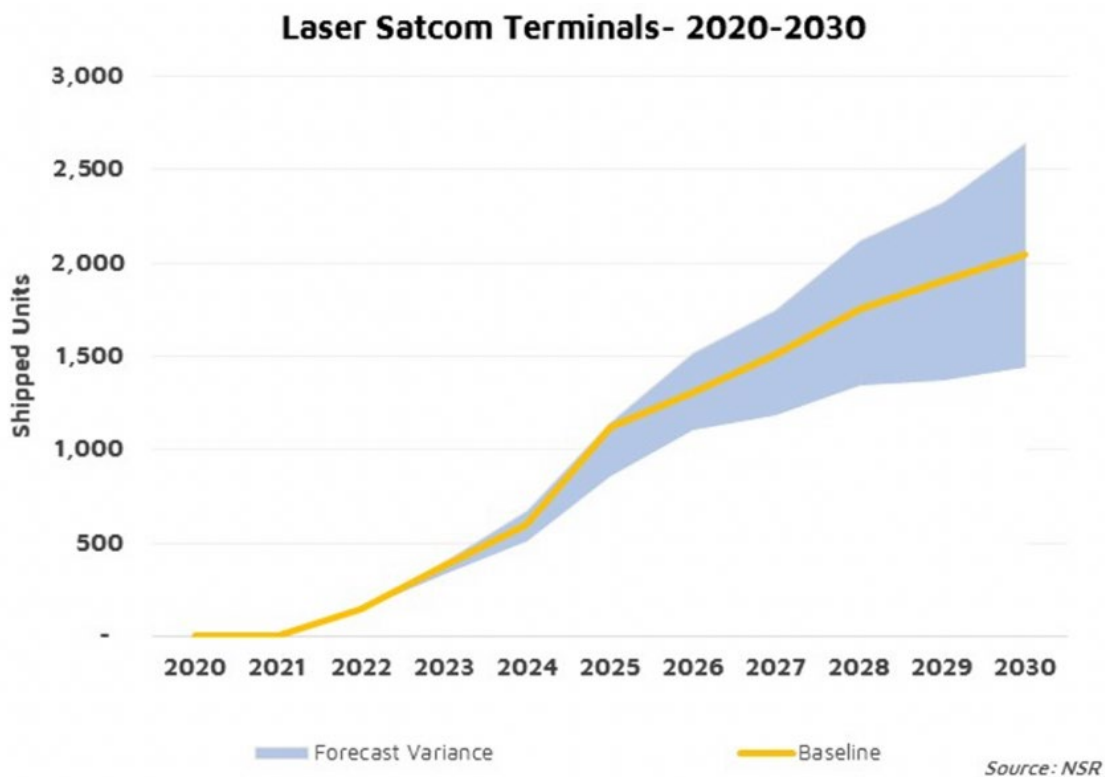
Figure 11.31: Laser terminals future forecast. Credit: Northern Sky Research.

For space applications, lasers are being used as the light source. Laser systems with dynamic systems such as fast-steering mirrors are used to accurately point the laser on the spacecraft to the ground terminal. Other methods using laser arrays for beam pointing are also being developed to reduce the need for complex dynamic systems. Data is transmitted in the form of hundreds of millions of short pulses of laser light every second. The light is made of photons and the optical ground terminals are setup to collect the light at the photon level. In fact, the ground terminals are designed for an environment where relatively few photons may be received from the transmitter spacecraft, especially from deep space. Direct photon detection with Pulse Position Modulation (PPM) is used instead of the common RF technique of direct carrier coherent modulation to convey information. PPM modulation uses a time interval that is divided into a number of possible pulse locations, but only a single pulse is placed in one of the possible positions, determined by the

information being transmitted. To detect extremely faint optical signals with relatively few photons through the atmosphere, optical ground stations can use a superconducting nanowire single photon detector (SNSPD), which, to increase the sensitivity of the nanowires, uses a 1-Kelvin cryocooler. A real-time signal processing receiver uses time-stamped photon arrivals to synchronize, demodulate, decode, and de-interleave signals to extract information code-words. Hence, while the specific technologies employed differ in some respects from those used in radio frequency ground terminals, the higher-level functions performed by the optical communication ground terminal are similar.

Optical communication is attractive for mission designers using small, resource-constrained spacecraft, because it offers a path to relatively high data rates with relatively small, low-power spacecraft equipment. The same volume and power savings can be experienced on the ground terminal side as well. This is driven by the size of the wavelengths. Because RF wavelengths are longer, the size of their transmission beam covers a wider area, therefore, the capture antennas for RF data transmissions must be very large. Laser wavelengths are 10,000 times shorter, allowing data to be transmitted across narrower, tighter beams. This results in the ability to deliver the same amount of signal power to much smaller collecting areas. The reduction in antenna size applies for ground and space receivers, which allows for size and mass reductions on the spacecraft side.

NASA made great strides with its optical communication demonstration on the Lunar Atmosphere and Dust Experiment Explorer (LADEE) mission. The pivotal NASA Lunar Laser Communications Demonstration (LLCD) was able to achieve 622 Mbps from a lunar distance. NASA has several exciting optical communications demonstrations, including O2O, Illuma-T, T-Bird and the Laser Communications Relay Demonstration (LCRD). LCRD is supported by Optical Ground Station (OGS)-1 at OCTL, and OGS2 in Hawaii (51).

Northern Sky Research (NSR) predicts growth for optical communications, which needs to be matched by OGS (Figure 11.31). "The demand now is not for just one gigabit per second, not 10 gigabits per second, but tens if not hundreds of gigabits per second. And it's growing exponentially. The only way to achieve that is by starting to use optical communications or laser communications to augment or to complement RF communications" (Barry Matsumori, CEO of BridgeComm, 2022) (52).

11.10.2   Optical Ground Stations and Future Infrastructure Requirements

OGS contain notably different equipment than RF stations, including an optics assembly, photon counter assembly (usually involving a photon counting nanowire detector and cryostat), and signal processing assembly with a time-to-digital converter. Since optical communications use a frequency higher than RF, (e.g., 1,550 nm downlink and 1,065 nm uplink wavelengths), the optical dishes can be smaller than RF antennas. To receive optical signals from a low-Earth orbit, 40 – 60 cm telescopes are sufficient. For successful deep space optical communications, calculations show that 3 m, 4 m, or even 8 m diameter ground apertures are required, depending on the distance from Earth. For these size apertures, when a 3 – 8 m OGS is not available, partnerships can be formed with large astronomy telescopes. For example, JPL-designed OGS equipment has been integrated at the Palomar Observatory (Hale 5-m telescope) for future use by the Deep Space Optical Communications (DSOC) demonstration. Note that OGS for LEO and deep space need different types of modems. It is also important for OGS to have spatial diversity. Weather,

atmospheric conditions, turbulence, and aerosols in the air can degrade laser propagation. Because certain types and depth of cloud covers can cause signal loss, probability of link success increases with multiple diverse locations.

For interoperability between SmallSats and public and private optical ground stations, a common communications standard is key. The Consultative Committee for Space Data Systems (CCSDS) and Space Development Agency (SDA) provide recommendations for communications standards, including optical communications. Adhering to these standards by both SmallSats and ground stations allows for multi-mission optical ground stations.

JPL is operating the Optical Communications Telescope Laboratory (OCTL) at Table Mountain, CA, with a 1 m telescope, as shown in Figure 11.32. This dish was used for the LADEE mission and offered great performance from a lunar distance (53).

JPL most notably operates the Deep Space Network (DSN), supporting 2-way RF communications and ranging services. Given the existing infrastructure, it is advantageous to augment a DSN RF antenna by installing optical segments at its center, making it a dual-purpose, RF-Optical hybrid antenna. The installation is being implemented in two phases.

In 2022 a small prototype RF-Optical system, including the mirror, cameras, and backend has been installed into DSS-13 (Figure 11.33) at the Goldstone Deep Space Communications Complex. DSS-13 is the R&D 34 m BWG antenna at Goldstone. The combination of seven small (0.5m) mirrors comprises a synthesized prototype optical aperture of about 1.3 m diameter. As of 2022, the seven-segment prototype mirror system has been undergoing alignment, test, and checkout. Control was verified to maintain segment position to <1 microradian, with first light successfully received from natural light sources. 1550 nm light was measured through the 100-meter fiber at the pedestal. JPL was able to track multiple sources across the sky from 20-80 degrees elevation.

A JPL designed communications detector and optical receiver been installed and tested over the air on DSS-13 as well. Other demo opportunities which could occur during the wait for Psyche (launch now delayed until next year) are being investigated.

The operational RF-Optical hybrid will ultimately

Figure 11.32: JPL's OCTL showing a 1-meter optical aperture. Credit: NASA JPL.
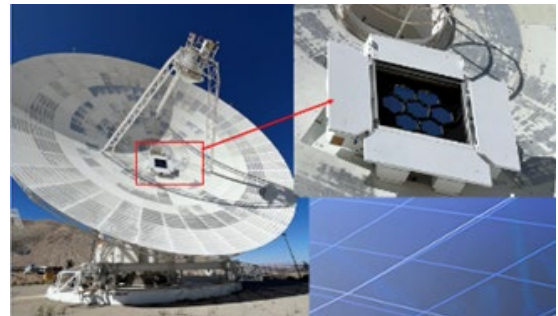
Figure 11.33: JPL's DSS-13, a 34m RF antenna, showing a 1.3-meter optical aperture in its center. Credit: NASA JPL.

Figure 11.34: Artist overlay of built DSN RF antenna and planned 8m optical segments at its center. Credit: NASA.

include 64 mirrors each of 1m diameter, installed as a segmented 8 m optical receive aperture/mirror physically inside one of the new DSN 34 m radio frequency ground terminals (DSS-23, in California). The final phase includes completion of the full 64- segment aperture on DSS-23, as illustrated in Figure 11.34, including a full year of field tests. This 8 m equivalent optical ground aperture will be operational in the early 2030s.

DSS-23 will be capable of a full set of RF services with the 34 m antenna in addition to high-rate optical communications with its 8 m optical assembly. Before the full operational readiness dates for optical communications, the 1.3 m partial optical systems will be usable at various times for best effort demonstration optical communications passes in the near-Earth or Lunar regimes, as well as for deep space missions.

The DSS-23 optical receiver is the same design that JPL is delivering to the Palomar Observatory for use with the DSOC optical communications technology demonstration on the NASA Psyche mission. This receiver is also being installed in ground terminals at White Sands and other locations for other near- and deep-space missions, as well as Artemis. One exciting implication of this 8m equivalent optical aperture is that it meets the 230 Mbps downlink data rate requirement for human exploration of Mars.

Looking at the broader optical communications landscape, over the past decade the community has been confronted with a chicken and egg problem: due to the lack of a ground segment, the FSO space segment has been slow to develop, and additional investments have not been made due to the low number of satellites flying an FSO ground segment. Further, due to the experimental nature of the first flying optical payloads, there has not been sufficient operational budget to pay for optical ground station services. Nevertheless, optical downlinks have the potential to become an essential part of data downlinks, especially in the new-space domain (54).

Europe

The European Optical Nucleus Network was formed between ESA ESOC, Germany Aerospace Centre (DLR), Global Security Operations Center (GSOC) and KSAT. Parties agreed to have an interoperable multi-mission approach based on the Consultative Committee for Space Data Systems (CCSDS) standards. Starting locations and characteristics are summarized in Table 11-17. The Nemea OGS (Figure 11.35) was operational as of 2022 after initial investments in the first commercial OGS, with the other stations soon to follow. The goal is to bring all building blocks together into an automated, cost-efficient, operational, multi-mission optical ground station service. This groundbreaking OGS uses modified COTS components to reduce cost, and could also be a blueprint for future stations. KSAT co-located its first OGS with KSAT RF antennas in the mid-latitudes at Nemea, due to the temperate weather and sharing the existing infrastructure. The OGS is compliant to the Optical On-Off Keying (CCSDS 141b1) draft standard and is designed to be cost competitive to the KSAT^{LITE} service which KSAT is currently offering in the RF domain. The

Figure 11.35: KSAT's low-complexity optical ground station in Nemea (2021).

telescope is mounted >3 m above the ground to avoid ground layer turbulence. The dome is a one-part, completely retractable structure with UV resistant plastic fabric. It is not connected to the telescope foundation to avoid coupling of vibrations caused by wind to the telescope system.

Table 11-17: European Optical Nucleus Network OGS Key Parameters			
	Nemea	**Almeria**	**Tenerife**
Downlink wavelength support range	1529-1569nm	1529-1569nm	1529.5-1568 nm
Telescope main aperture diameter	50 cm	60 cm	80 cm
Elevation angle support range	20° to 90°	10° to 85°	10° to 87° (LEO) 10° to 89° (GEO)
Max. operation wind speed	15 m/s (18 m/s gust)	15 m/s (18 m/s gust)	14m/s
Operational min. sun-distance	20°	20°	TBC
Tracking modes	Program-track Auto-track	Program-track Auto-track	Program-track Auto-track
Temperature operational range	-15°C to 40°C	-5°C to 35°C	TBC
Acquisition Beacon Source	1589.3nm, 5W (peak)	1589 – 1591nm, 5W (peak) TBC	1591.26 nm, 5 W 1590.4 nm, 8 W
Acquisition beacon divergence angle (1/e² full-angle)	632 μrad	1000 μrad TBC	500 μrad
FoV of auto-track system	8.5x6.8 arcmin	15x12 arcmin TBC	8.6 arcmin (diameter)
Absolute pointing error*	< 5 arcsec	< 10 arcsec	< +- 103 arcsec (99% confidence)
Auto-track accuracy	1 arcsec RMS max. error 3 arcsec	1 arcsec RMS	0.41 arcsec RMS
Site (mean) clear-sky probability	Summer: 80-95% Winter <60%	Summer: >75% Winter: >55%	81%
Site long term average seeing	Day: 2.7 arcsec Night: 3.6 arcsec	TBD	Day: 2.2 arcsec Night: 1.8 arcsec
Site WAN connectivity	10 Gbps	10 Gbps	10 Gbps
Location coordinates	Lat. 37°50'42.5"N Long. 22°37'24.0"E 278.7 m above sea level	Lat. 37°5'36.3''N Long. 2°21'31,1''W 498 m above sea level	Lat. 28°17'58.7"N Long 16°30'38.5"W 2382 m above sea level

- KSAT co-located its first OGS with KSAT RF antennas in the mid-latitudes at Nemea, due to the temperate weather and sharing the existing infrastructure,

- The OGS is compliant to the Optical On-Off Keying (CCSDS 141b1) draft standard and is designed to become cost competitive to the KSAT[LITE] service, which KSAT currently offers in the RF domain.

- The goal was to bring all building blocks together into an automated, cost-efficient, operational, multi-mission optical ground station service.

- For cost reduction modified COTS components have been selected.

- The telescope is mounted more than three meters above the ground to avoid ground layer

turbulence.

- The dome is a one-part completely retractable structure with UV resistant plastic fabric. It is not connected to the telescope foundation to avoid coupling of vibrations caused by wind to the telescope system.

The First Data Link Between Commercial Optical Terminals has been validated through a temporary Sony optical terminal placed on the ISS, with a channel data-rate of 150 Mbit/s with BER varying from 1e-3 and <1e-6.

Japan

The National Institute of Information and Communications Technology (NICT) optical ground station in Japan also received transmission from the SOLISS system by Sony CSL installed on the Kibo's exposed facility on the ISS (55). In 2021 NICT also reported that the 1 m optical ground station in Koganei, Tokyo received via optical communication images taken by the satellite's camera the using the SOTA optical communications device mounted on a 50 kg class microsatellite.

Germany

The DLR German Aerospace Center is another organization active in optical communications. About 25 km west of Munich, Germany is their Optical Ground Station Oberpfaffenhofen (OGS-OP) that houses a 40 cm Cassegrain telescope (56). The German Aerospace Center has also developed a transportable optical ground station (TOGS). It has a 60 cm deployable telescope in a Ritchey- Chretien-Cassegrain configuration with a focal ratio of f/2.5. The telescope is supported by an altazimuth mount on a structure with four adjustable legs for leveling the mount and compensating for rough terrain. It has been successfully used to track the OPALS instrument on the ISS and serves as the primary ground station for the OSIRIS payload on the BiROS satellite. The German Aerospace Center OGS-OP and TOGS are shown in Figure 11.36.

WORK

Figure 11.36: (left) OGS-OP and (right) TOGS. Credit: German Aerospace Center. https://creativecommons.org/licenses/by/3.0/de/legalcode.

Microwave is a system integrator and provider of satellite communication technologies and products. WORK Microwave supplies Optical Ground Stations (OGS) as a turn-key solution that can be customized upon mission request. WORK Microwave collaborated with KSAT and Astelco to establish the first commercial ground station for ESA at Nemea Greece, and will install a complete SDA standard compliant OGS for KSAT by the end of 2024. WORK Microwave manufactured the OGS with the Digital Optical Ground (DOG) station suite for system configuration, monitoring and control, and communication. Figure 11.37 shows the OGS DOG schematics, with telescope control from WORK Microwave's partners. The DOG Suite is a technology platform for further developments in optical communications across LEO, MEO and GEO orbits, as well as for upcoming lunar and deep space missions. More info on https://work-microwave.com/digital-optical-groundstation/.
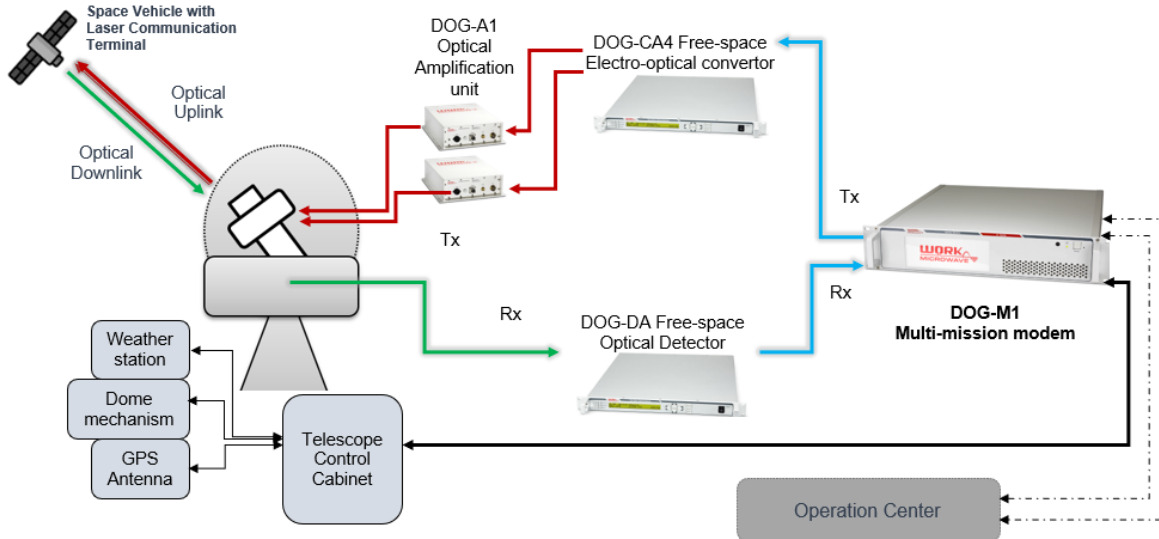
Figure 11.37: Schematics of the OGS with the Digital Optical Ground (DOG) station suite. Credit: WORK Microwave Inc.

Australia – New Zealand

The Australian Optical Ground Station Network (AOGSN) will eventually be made up of four ground stations in Western Australia, South Australia, the Australian National University (Australian Capital Territory), and New Zealand. The plan is to tie these stations together to produce a communication network that can support optical, RF, and future quantum communications. In spring 2021, Thales Australia signed a research extension with SmartSat Cooperative Centre (CRC) for the development of advanced optical communications technologies (57).

Sascha Schediwy, head of the research group responsible for designing and building the WA Optical Ground Station (Figure 11.38), believes lasers will play a crucial role in the next human missions to the Moon. "It's likely to be how we'll see high-definition footage of the first woman to walk on the Moon," Dr Schediwy said (abc.net.au).

Figure 11.38: The 70 cm Western Australian Optical Ground Station (WAOGS) is installed on a rooftop at the University of Western Australia. Credit: The International Centre for Radio Astronomy Research.

United States

In the U.S. the Aerospace Corporation is a player in the Optical Communication arena. Their manned OGS 40 cm telescope, located in El Segundo, CA, demonstrated 200 Mbps from 725 km. It is operating at 1064 nm wavelength, thus it is not compatible with other optical ground stations or most COTS optical space terminals.

11.10.3   Techniques to Improve Optical Comm Reliability

Laser communication is essential for future telecom networks to supplement RF communications and enable:

- Very high throughput links (> 10 Gb/s and up to Tb/s)
- Communication without frequency band limitation
- Highly secure, stealthy, non-interceptable links

It is essential for operational use cases:

- Multispectral observation of Earth from space (very bandwidth intensive)
- Securing sovereign communication
- Telecommunication constellations that rely on very broad bandwidth links

Cailabs

Cailabs designs and manufactures turnkey Optical Ground Stations for laser communication, offering unique turbulence mitigation based on its Multi-Plane Light Conversion (MPLC) technology, in both downlink (TILBA©-ATMO) and uplink (TILBA©-IBC). The company developed a standard design for 10 Gbps, LEO-to-ground, turbulence-proof laser comms, called TILBA©-OGS L10. The TILBA-ATMO is a product that takes a perturbed beam, corrects and couples it into a standard single-mode fiber. This compensates for atmospheric turbulence, improving free-space optical links. TILBA-ATMO makes it possible to use conventional telecom equipment and direct or coherent modulation formats to provide robust high-throughput links to optical ground stations. In an open-air trial with DLR, link stability with TILBA-ATMO was similar to that of the adaptive optics unit in low and medium turbulence, and more effective in strong scintillation
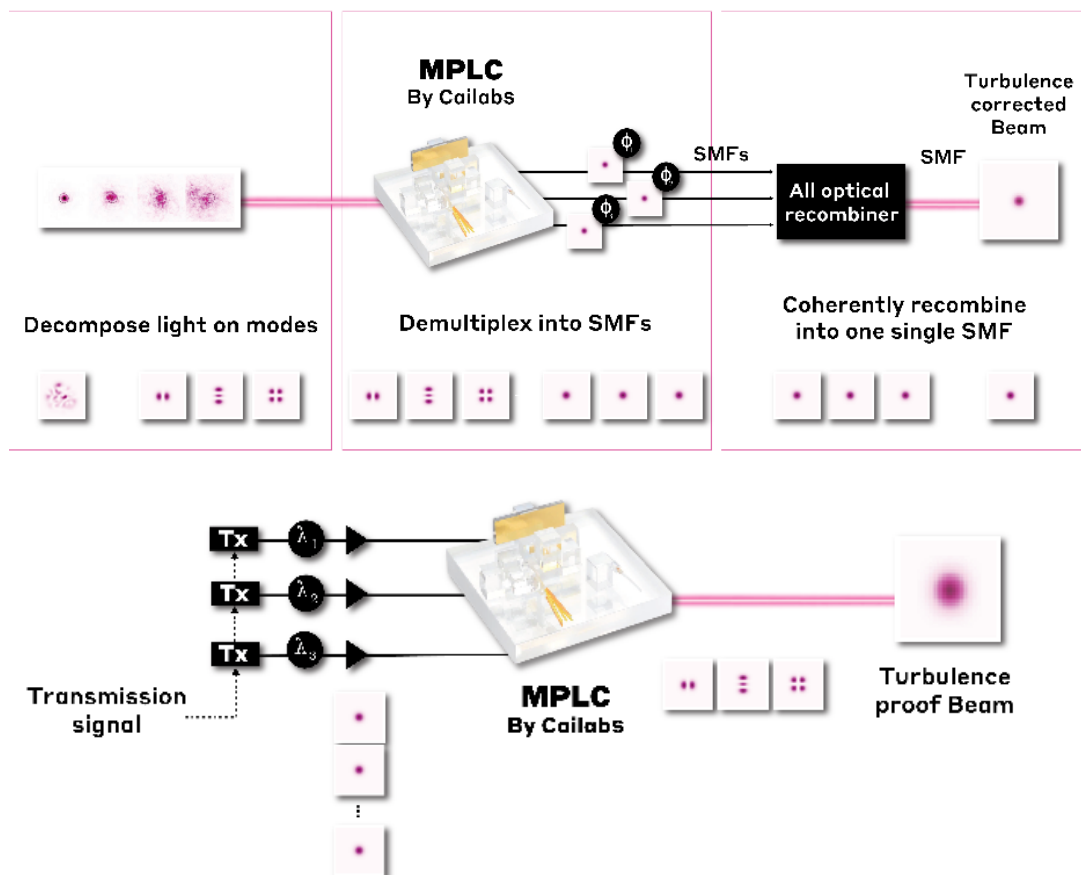
Figure 11.39: Cailabs Tilba Atmo equipment and schematic of TILBA-IBC. Credit: Cailabs.

conditions. They have also developed a second building block called TILBA-IBC, providing spatial diversity based on incoherent beam combination of spatial modes generated by the MPLC. This component can mitigate atmospheric turbulence in uplinks from ground to space, without the need to use multiple telescopes.

Cailabs is involved in multiple government and commercial OGS projects featuring 60 to 80 cm telescopes, compatible with both CCSDS and SDA standards, and is remotely operable (58). (https://www.cailabs.com/en/)

11.10.4   Role of Optical Relays

Optical inter-satellite links are "critical to the success of the Space Development Agency's low Earth orbit constellation" known as Transport Layer. SDA-Funded Laser Terminal Technology could connect to multiple satellites simultaneously. Each satellite in the Pentagon's "planned mesh network of communications satellites could have as many as many as four laser links so they can talk to other satellites, airplanes, ships and ground stations." BridgeComm, which recently received an SDA contract, "developed a so-called 'one-to-many' optical communications technology for point-to-multipoint transmissions" which could "help reduce the cost of building constellations by requiring fewer terminals," Michael Abad-Santos, senior vice president of business development and strategy at BridgeComm] said. BridgeComm first demonstrated "point to multipoint optical communications in 2019 in a project with Boeing, and has since continued to mature the technology," Abad-Santos said (59).

Warpspace is a private Japanese company developing an inter-satellite communication system based on laser communication (Figure 11.40). The WarpHub InterSat link relays will enable low latency data delivery from MEO, and in the future it will connect to the Lunar Gateway or planetary deep space via optical communications (see https://warpspace.jp/).
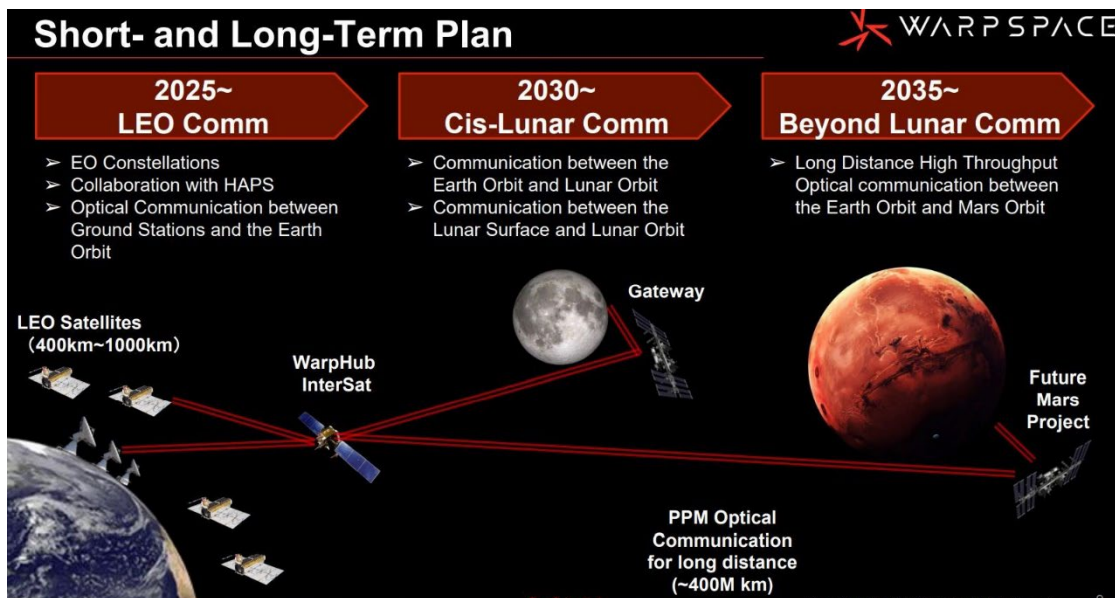
Figure 11.40: Warpspace plans a satellite network in MEO orbit by 2025 with optical communications hardware able to be reached by any LEO satellite by 2025. Credit: Warpspace.

11.11 Summary

The ground segment serves as the gateway to getting valuable data collected by the satellite into

the hands of the user. It is a critical component of the satellite system that requires attention at the earliest stages of mission planning. Understanding what ground solution best meets the needs of the mission has a direct impact on the spacecraft design, concept of operations, launch schedule, mission operations cost, and expected data volume for processing. Much effort also goes into preparing for the interaction between the satellite and ground network. Developing software and simulations, drafting operations manuals, conducting operations rehearsals, and performing compatibility tests are all par for the course. Post launch, the ground station also plays a key role in locating and commissioning the spacecraft.

In looking forward to the future of ground systems, the clear objective is how to bring the data down more efficiently. Great strides are being made with optical communications where it is possible to have increases in data per pass that are orders of magnitude above what can be achieved with RF communications. Optical communication technology is now being infused into ground system architectures, and flight hardware is becoming miniaturized enough to fit within small satellites. The ability of these systems to quickly change beam directions and acquire multiple targets will be critical for communicating with constellations of small satellites.

While the tried-and-true RF ground system solution remains the workhorse for small satellites, the innovative nature of the small satellite platform will soon challenge the community to adapt to systems capable of handling hundreds of satellites and high data volumes. Efforts are ongoing to keep pace, but only time will tell whether ground systems will advance or impede the small satellite revolution.

For feedback solicitation, please e-mail: arc-sst-soa@mail.nasa.gov. Please include a business e-mail so someone may contact you further.

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Chapter Contents

Chapter Glossary

(18 SDS)	U.S. Space Force 18th Space Defense Squadron
(19 SDS)	U.S. Space Force 19th Space Defense Squadron
(CA)	Conjunction Assessment
(CARA)	NASA's Conjunction Assessment Risk Analysis program
(CAESAR)	Conjunction Analysis and Evaluation Service, Alerts and Recommendations
(CCR)	Corner Cube Reflectors
(CNES)	Centre National d'Etudes Spatiales (French Space Agency)
(COTS)	Commercial-off-the-Shelf
(CUBIT)	CubeSat Identification Tag
(D/T/I)	Detection, Tracking and Identification
(EGTN)	ExoAnalytic Global Telescope Network
(ELROI)	Extremely Low Resource Optical Identifier
(EUSST)	European Union Space Surveillance and Tracking program
(FCC)	Federal Communications Commission
(GEO)	Geosynchronous Equatorial Orbit
(GNSS)	Global Navigation Satellite System
(GPS)	Global Positioning System
(GUI)	Graphical User Interface
(HEO)	Highly Elliptical Orbit
(HUSIR)	Haystack Ultrawideband Satellite Imaging Radar
(IDs)	Identification
(ILRS)	International Laser Ranging Service
(LEDs)	Light Emitting Diodes
(MEO)	Medium Earth Orbit
(NPR)	NASA Procedural Requirement
(NTE)	Nanosatellite Tracking Experiment
(NTIA)	National Telecommunications and Information Administration
(OCAP)	Orbital Conjunction Assessment Plan
(OEM)	Orbit Ephemeris Message
(O/Os)	Owner/Operators
(OSAs)	Orbital Safety Analysts
(PNT)	Position, Navigation, and Timing
(RF)	Radio Frequency
(RFID)	Radio Frequency Identification
(SRI)	Stanford Research Institute
(SSA)	Space Situational Awareness
(SSN)	Space Surveillance Network
(SWaP)	Size, Weight, and Power
(TLE)	Two-Line Element
(TraCSS)	Traffic Coordination System for Space
(USIR)	Ultrawideband Satellite Imaging Radar

12.0 Identification and Tracking Systems

12.1 Introduction

In the past, most launches involved a single, large satellite launching on a dedicated launch vehicle. Small satellites as secondary payloads were sometimes 'dropped off' along the way to the primary payload's orbit or rode along to the final orbit with the primary payload. In either case, it typically was not that difficult for tracking radars to distinguish between primary and secondary payloads via size and operational parameters.

Recently, however, multi-manifest or "rideshare" launches have become more common, and providers (1)(2)(3) are launching multiple CubeSats, or bundling CubeSats and other smaller payloads with larger payloads to fill up the excess capacity of almost any given launch vehicle. For technical and cost reasons, such launches generally deploy small satellites and CubeSats into very similar orbits over a short time window. "Batch" launches with a lack of separation between satellites can prevent effective tracking and create "CubeSat confusion" (4). When CubeSats are deployed close together in space and time they can be hard to distinguish from each other by tracking radars, making it difficult to determine which orbits correlate to which spacecraft, preventing a unique orbital state for each object from being added to the catalog of on-orbit objects (5)(6). At times it can take weeks to months to sort out which object is which, while some may never be uniquely identified at all. If the orbital states cannot be determined and the spacecraft cannot be added to the catalog of on-orbit objects, neighboring satellites may be unaware of close approaches, hindering their ability to mitigate risks of collision.

Due to their standardized shape and size, CubeSats look very similar to one another, especially when they are in orbit hundreds of kilometers away. If there are unidentified objects from a launch, then the possible number of associations of object identifications (IDs) to tracked objects scales as n! (n-factorial, where n is the number of unidentified space objects from the launch). For example, if there are just two objects, say a payload and an upper stage, there are two ways in which you can associate the IDs with the tracked objects, and even that can be a challenge (7). However, if there are ten unidentified objects, there are 3,628,800 possible combinations; with 20 this rises to 2.4×10^{18} combinations. The magnitude of the problem gets big quickly.

Small satellites can improve their chances of being identified and tracked through good coordination with tracking agencies pre-launch, through community sharing of position and covariance data in clearly defined, consistent, standard formats (such as Orbit Ephemeris Message (OEM), and through careful selection of deployment direction and timing (8). Good spacecraft design choices can also improve the chances of small satellites surviving launch and early orbit (9) and can even make use of in-space commercial radio networks as a "back-up" method of communicating should primary systems fail (10). However, despite improvements in both design and coordination, many small satellites still go unidentified. This has led to the introduction of tracking aids – independent systems that help owners and trackers identify small satellites and CubeSats, in some cases even if the satellite is malfunctioning.

Even if a small satellite is trackable (or rendered trackable through one of the enhancements discussed below), it is important that this position information be combined with satellite planned maneuver information and packaged into a predicted ephemeris that can be sent regularly (at least daily) to a conjunction assessment (CA) screening authority. The only way that CA screenings can be meaningful is if a satellite's future position is used by the screening authority in the screening process. Non-cooperative tracking, even if it is well and frequently performed, cannot anticipate satellite maneuvers; and this information is essential to a robust conjunction assessment system. Maneuverable satellites (which include satellites that employ non-propulsive trajectory modification methods, such as differential drag) must assemble and furnish durable predicted ephemerides to the screening authority, either through indigenous means (i.e., unaided

tracking and their own flight dynamics system's production of a predicted ephemeris) or via one of the position determination approaches to be described in order to ensure the operators are aware of close approaches with them. Non-maneuverable spacecraft need only ensure that they are trackable, although the CA best practices documents (41) all stress the importance of owner/operators (O/Os) taking responsibility for ensuring that screening authorities possess timely and accurate predicted position information for their spacecraft.

O/Os need not perform ephemeris generation themselves; there are several commercial companies that will perform this function on their behalf. O/Os can furnish a third-party provider with their on-board Global Navigation Satellite System (GNSS) data and their maneuver plans, and this provider can produce a predicted ephemeris with a realistic covariance regularly (i.e., at least daily) and furnish it to the appropriate screening authority. It is important that quality orbit determination be performed in such cases, which means both accurate predicted state estimates and, perhaps even more important for orbital safety, realistic covariance matrices accompanying each ephemeris point. Because satellite collision likelihood calculations require a statement of the state prediction uncertainty in order to compute the probability of collision, it is very important that realistic covariance estimates be produced and included with the ephemeris; this is not always a routine flight dynamics product, so it is necessary that ephemeris "vendors" be queried about their abilities to tune their organization development process to produce realistic covariances and their methodologies for verifying covariance realism. The NASA CARA github (33), search for "NASA CARA Analysis Tools") contains covariance realism evaluation code and associated theoretical and technical documentation. Small missions or university research CubeSat endeavors are offered special rates by some firms (e.g. Kayhan Space) for ephemeris generation, so such O/Os should ask about special pricing structures that may make ephemeris data quite affordable.

While the thrust of the above comments pertains to routine satellite identification and tracking once a satellite is firmly established in its final orbit and placed into the space catalogue, a particular problem occurs directly after satellite injection but before cataloguing—the collision avoidance "COLA Gap." During large launch deployments, it is often difficult to reliably distinguish multiple spacecraft, and it can take days or even weeks for enough spacecraft separation and regular tracking to occur to allow durable identification and therefore cataloguing. Because only catalogued objects are accommodated by the conjunction assessment enterprise, spacecraft caught in the COLA Gap are not included, thus creating a collision danger both to other satellites and to themselves. Producing an ephemeris from on-board GPS information or using one of the beacon or tracking aid devices enumerated below can allow a deployed satellite's current position, and therefore predicted position ephemeris, to be assembled rapidly. This ephemeris can be forwarded as a special submission to the CA screening process, thus closing the COLA Gap.

Finally, it should be recognized that regular predicted orbit information is required until spacecraft demise, not just during active operations. If a satellite is not regularly trackable by non-cooperative sources such as the United States Space Force Delta 2, this problem must be addressed through other means, such as processing of on-board GPS information to produce a predicted ephemeris. This mechanism needs to persist fully through spacecraft re-entry or placement in a disposal orbit. This requirement is one reason that independently powered GPS beacons are an attractive solution for satellites that are not easily trackable and will not pursue active deorbit at the end of their main mission.

The information described below is not intended to be exhaustive but provides an overview of current state-of-the-art technologies and their development status for a particular small spacecraft subsystem. It should be noted that Technology Readiness Level (TRL) designations may vary with changes specific 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 TRL of described technology. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA.

12.2   Identification and Tracking Ground Systems

The United States Space Force Delta 2 is responsible for performing space surveillance and providing foundational Space Situational Awareness (SSA) for the US Department of Defense as well as for other agencies and space entities. Delta 2's 18th Space Defense Squadron (18 SDS), located at Vandenburg Space Force Base in California, performs all catalog maintenance functions including detection, tracking and identification (D/T/I) of artificial objects in Earth orbit and maintaining the space catalog which is publicly available on Space-Track.org. As part of their activities, they provide launch support, re-entry assessment, and other SSA functions: Orbital safety activities, such as conjunction assessment (which identifies close approaches between launch and other catalogued in-orbit objects) are provided by Delta 2's 19th Space Defense Squadron (19 SDS) at the Naval Support Facility at Dahlgren, VA. Maintaining the catalog is achieved via the US Space Surveillance Network (SSN) that is formed by a suite of sensors around the world (28). 18 SDS is currently tracking more than 45,000 objects in Earth orbit and can provide data for pieces as small as 10 cm^3. They issue two-line elements (TLEs) that are updated on a regular basis and can be used to compute predicted orbit position for spacecraft communications acquisition and other purposes. They also produce precision vectors with covariance that can be used to perform conjunction analyses. TLEs are not accurate enough to be used for conjunction assessment, and do not include covariances that are needed to compute a probability of collision.

The US Space Force next generation SSA sensor, known as the Space Fence, was declared operational in March 2020 and can track objects below the previous 10 cm^3 limit. It is located on Kwajalein Atoll, in the Republic of the Marshall Islands and consists of a S-band radar system to track objects primarily in low-Earth orbit, although it can track objects in medium-Earth orbit (MEO) and geostationary equatorial orbit (GEO) as well. The 20th Space Control Squadron based in Huntsville, Alabama, manages the Space Fence and provides data to 18 SDS to augment the space catalogue (29). Another important sensor in the SSN is the Haystack Ultrawideband Satellite Imaging Radar (HUSIR), which is the highest-resolution, long-range sensor in the world. HUSIR simultaneously generates X- and W-band images that can provide valuable information about the size, shape an orientation of Earth orbiting objects (30). These are just two examples of sensors that make up the SSN, many having specific unique capabilities that support the SSN's various functions, including conjunction assessment.

The NASA Conjunction Assessment Risk Analysis (CARA) program acts as the intermediary between 18/19 SDS and NASA satellite missions. CARA gathers orbit ephemeris and covariance files from the NASA spacecraft operations teams and provides this data to NASA's Orbital Safety Analysts (OSAs) at Vandenberg for screening and close approach prediction. CARA provides risk assessment of these predicted close approaches to NASA missions beyond the 19 SDS support provided to non-NASA users, including operations concept development, specialized probability of collision computation, high interest event notification, and conjunction geometry analysis among other functions. In 2012, the French Space Agency (CNES) created a conjunction risk assessment team called Conjunction Analysis and Evaluation Service, Alerts and Recommendations (CAESAR) that provides risk assessment services to their missions (33)(34) as part of the European Union Space Surveillance and Tracking (EUSST) program.

In 2023, NASA released Revision 1 to their best practices handbook entitled "Spacecraft Conjunction Assessment and Collision Avoidance Best Practices Handbook," which is a great reference for satellite operators with respect to collision avoidance topics (31). NASA Procedural

Requirements (NPR) 8079.1 codifies those best practices as requirements for NASA space flight programs, projects, and vehicles to protect the space environment and reduce the risk of collision (32).

Besides USSF Delta 2, several commercial entities are providing tracking capabilities that can be purchased by stakeholders. ExoAnalytic and Slingshot Aerospace each operate global telescope networks consisting of over tens of observatories and hundreds of telescopes tracking orbiting objects in GEO, highly elliptical orbit (HEO), and MEO. The Exoanalytic Global Telescope Network (EGTN) can collect angles and brightness measurements. They each maintain a proprietary catalog of satellites and space debris that are regularly tracked and cataloged.

LeoLabs is another commercial entity providing spacecraft tracking and support services. They use a group of distributed Earth-based, phased-array radars to make a commercial-off-the-shelf (COTS) satellite tracking service targeted to the specific requirements of SmallSat operators in low-Earth orbit. They currently have radar locations in New Zealand, Costa Rica, the Azores, Poker Flat, AK, Midland, TX, Western Australia, and Argentina. There are currently seven functioning radar sites as of 2024, with plans for more radars strategically located around the world to track objects down to size levels approaching that of the DoD. The predicted performance also includes a revisit time of over 10 observations per day for specific objects. Through their LeoTrack platform, they can use their radar data to perform precision tracking and curate orbit information products for satellites as small as 1U. Their system includes an open-source graphical user interface capable of displaying all the catalog in real time, as well as fundamental orbit information about each individual object. They also offer a commercial launch and early orbit service, with SpaceX as one of their historic customers (38).

Catalogs provided by these commercial entities are different from the one maintained by 18 SDS in accuracy and objects included. Spacecraft owner/operators should be aware of the differences before choosing to use a particular service for a particular purpose. For conjunction assessment purposes, having multiple differing solutions can be confusing. The Department of Commerce was charged in Space Policy Directive-3 with creating a space traffic coordination system that enables commercial capabilities for conjunction assessment. In the future they may offer a conjunction assessment service that merges data form multiple sources in one solution as they work to transition the service currently provided by 19 SDS. Initial capability of the Traffic Coordination System for Space (TraCSS) is planned for September 2024. Lessons learned from NASA's recent Starling (launched July 2023) and Starlink constellations will be fed into the design of TraCSS.

NASA's Starling 1.5 mission extension, using the four-CubeSat Starling swarm of spacecraft, will build on the Starling primary mission demonstration of autonomous maneuver planning and execution. Partnering with SpaceX and their Starlink constellation, Starling 1.5 will demonstrate an advance space traffic management solution for co-located groups of autonomous spacecraft from different owner/operators. The onboard autonomous maneuvering software for the original Starling mission, consisting of Emergent Space Technologies (acquired by York Space Systems) Navigator and Autopilot products, can perform probability of collision calculations and propose risk mitigation maneuvers for conjunctions with co-located vehicles. A space traffic management hub on the ground will be able to receive maneuver responsibility claims from an operator, indicating that they intend to maneuver to reduce the conjunction risk, receive and screen proposed new trajectories in near-real-time, and provide operators conjunction data messages with screening results. This will enable highly automated operations, including conjunction risk assessment and mitigation between satellite owner/operators.

12.3 Tracking Aids

For spacecraft that cannot be routinely tracked by the SSN, it is important to ensure trackability by another means to enable other owner/operators to know where your spacecraft is to prevent debris-producing collisions. This is especially important for SmallSats that have orbital lifetimes that exceed operational lifetimes, as the risk to orbital neighbors remains after tracking activities have stopped. For NASA spacecraft, trackability until demise is required in NPR 8079.1 and assessed as part of the required Orbital Conjunction Assessment Plan (OCAP) written during design.

Tracking aids come in several categories, each with benefits and drawbacks (11). Table 12-1 discusses the broad categories available, with representative examples discussed below. Size, weight, and cost vary for each of the examples, but all can be considered compatible with a CubeSat mission; see the references for detailed information on size, weight, and power (SWaP) and cost, all of which become more favorable each year with technology improvements and economies of scale. Once the augmented tracking data is collected, the ephemeris data is made available to CARA/19 SDS for screening.

Several commercial companies offer services that will process the data produced by tracking aids and produce predicted orbit ephemeris data that can be used to perform conjunction assessment. These include SpaceNav, COMSPOC, and Kayhan Space, the latter of which has a capability that focuses on small spacecraft support.

Table 12-1: Types of Tracking Aids			
Technology Scheme	**Description and Reference Mission**	**TRL**	**Citation**
CubeSat position and ID via radio	A position, navigation, and timing (PNT) receiver is attached to a CubeSat, along with a radio to transmit the information via a LEO communications provider (or directly to the ground); example: BlackBox, Blinker.	7-9	(12) (13)
Coded light signals from light source on exterior of CubeSat	Exterior-mounted LEDs with large-aperture telescopes to receive the signal or diffused LED lasers with ground-based photon-counting cameras.	6-7	(14) (15)
Radio Frequency interrogation of an exterior Van Atta array	For example, exterior mounted radio frequency identification (RFID) tag & commensurate radar.	7-9	(16)
Laser interrogated corner cube reflectors (CCR)	One or several small CCRs can be attached to CubeSat exterior; ground-based laser and receiver telescope needed to distinguish number of CCRs.	7-9	(17)
Passive augmentations to visibility	Use of high-albedo paint or tape, improving overall conductance of the exterior of the satellite or other methods to increase visibility.	7-9	
Orbital Whereabout Locator (OWL)	A tuna can mounted plug and play device equipped with a battery that helps identify and keep track of a satellite with a GNSS based tracker, monitors key parameters of a satellite and downloads data using an omnidirectional antenna in any orientation.	7-9	

12.4 Devices that Communicate Position and ID via Radio

The most comprehensive (but also potentially the most complex and SWaP-intensive) option involves equipping a small satellite with an independent positioning, navigation, and timing (PNT) receiver and independent radio capable of transmitting data to an independent communications provider. An example technology is the Black Box system (Figure 12.1), described by NearSpace Launch, Inc., in a recent conference paper (18). This system comes in several form factors for mounting internally or externally to a small satellite or CubeSat. The patch antenna shown in the first image is approximately 10 cm by 8 cm and can weigh as little as 22 grams; larger systems such as the one shown in the third image of Figure 12.1 have flown and are considered TRL 9. Another example often used for CubeSats is the NovAtel OEM7700. JAXA offers a strap-on tracking device called mini-Mt. Fuji; they can provide sensor tracking, or operators can track it themselves. These systems combine a low-power GPS receiver with a low-power radio capable of communicating with a low-Earth orbit communication provider (in the case of Black Box, the Iridium network) and operate independently from the spacecraft's regular command and telemetry links. Externally mounted versions often include solar cells for independent power generation. A Black Box system is currently flying on Spaceflight Sherpa-FX orbital transfer vehicle, launched on January 24, 2021, and is returning GPS fixes to the developer. The GPS fixes were analyzed, and reports were presented at the October 2021 and October 2022 International Astronautical Congress (40)(41). As a further service, the vendor (NearSpace) can generate a low- or high-precision ephemeris for one's satellite and furnish this to the O/O, who can then submit it to a CA screening authority.

The advantages to such a system are that it provides complete data on a satellite's position and requires no specialized ground equipment (other than the equipment used by the communications provider). Some such systems are independently powered and can provide data even if the host satellite never powers up, though others are dependent on spacecraft power to function. These systems are the most complex of the tracking aids described, however, and despite their relatively small size, are still the most SWaP-intensive of the options examined. Systems that rely on power from the host vehicle are also useless if the host vehicle suffers a power anomaly or failure. Having an additional onboard radio that communicates with other space assets necessitates additional oversight by the Federal Communications Commission (FCC) (or National Telecommunications and Information Administration (NTIA) for US Government missions licensing and coordination).

In choosing such systems, it is important to choose one that is space rated, otherwise simulation is required to perform that testing pre-launch. The system should also provide fast convergence after launch to allow rapid computation of an orbit determination solution to provide to 18/19 SDS
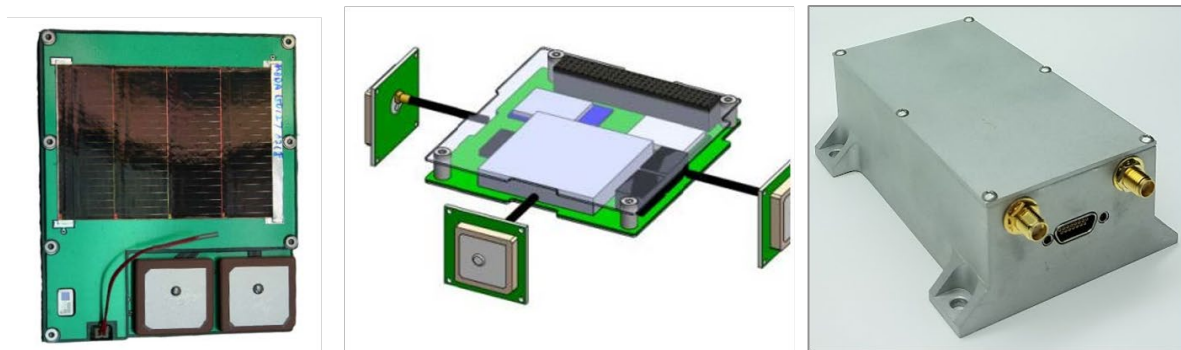

Figure 12.1: (left) Thin Patch or Stamp Black Box for side mounting. (Middle) PC104 Black Box for internal stack mounting. (Right) Standard Black Box for larger satellites. TRL 9: flown on spaceflight launch. Solar array and antennas not shown. Credit: NearSpace Launch, Inc.

for conjunction assessment screening as soon as possible after separation; this allows solution of the "COLA Gap" problem, namely providing orbital safety information and capabilities between payload injection and the time of the satellite's cataloging, which could be as long as several weeks. Other available features, like multi-frequency, multi-GNSS, and availability of pseudoranges are nice-to-have, but the benefit for conjunction assessment is marginal vs. just having a receiver. If flying a GPS system, it is important to ensure that the GPS data be available in the download stream and not just internally available to the spacecraft computer for onboard use.

Overall, when selecting such a system, the following considerations should be considered:

- Independent power of the device to ensure that it continues to function in the presence of internal spacecraft faults and be able to issue position information all the way until spacecraft demise;
- An ephemeris generation capability, either as a service from the device manufacturer, an indigenous capability of the O/O, or a service provided by a third-party provider. These ephemerides need to be submitted at least daily to the CA screening authority;
- The inclusion of planned satellite maneuvers in the predicted ephemeris to be furnished; and
- The existence of a ground network and communications path to allow the regular receipt of the broadcast data to achieve the requirements stated above.

12.5   Devices that Use Coded Light Signals

Identification systems and devices that make use of light emitting diodes (LEDs) and coded light signals have the advantage of being relatively simple and capable of identifying satellites uniquely. However, all systems flown to date have required power from the host
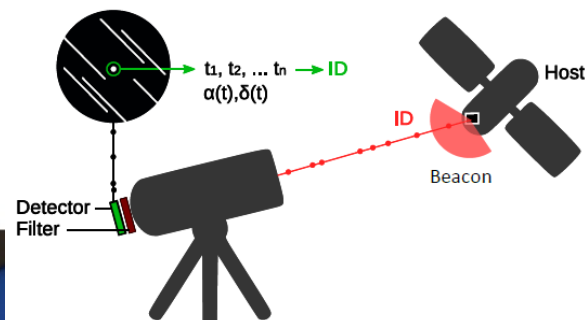
Figure 12.3: ELROI Optical Detection System. Credit: Los Alamos National Laboratory.
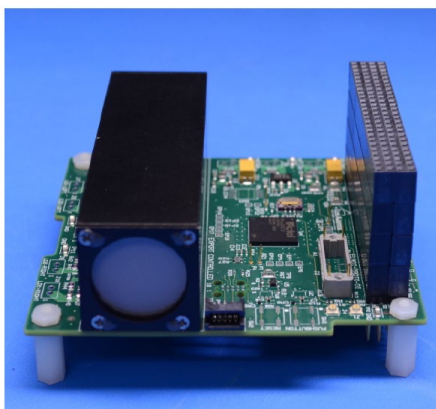
Figure 12.2: (left) ELROI PC104 beacon unit that was installed on NMTSat.d; (right) Two ELROI beacon units delivered for a launch in 2021. Credit: Los Alamos National Laboratory.

satellite, leading to issues with detection (19) if the host satellite does not power up. Current implementations are also relatively large, though future systems are expected to be much smaller and may include independent power. Devices such as the Extremely Low Resource Optical Identifier (ELROI) beacon (Figure 12.2), under development by Los Alamos National Lab (19), use exterior-mounted LEDs or diode lasers that blink in a prescribed sequence to uniquely identify

satellites. The ELROI system is designed to be independently powered by a small solar cell and battery and is packaged into a system as small as a Scrabble tile, though only larger systems – with power provided by the host satellite – have flown.

The emitters on such devices can be regular LEDs or diffused diode lasers but require specialized ground equipment – either a large-aperture telescope or a photon-counting camera –to track the object as it passes overhead. Figure 12.3 shows how the ELROI system works: a photon-counting camera attached to a telescope tracks the signal from a diode laser and decodes the ID of the host satellite from the on/off pattern of flashes.

Another similar system (35) proposes to use red, blue, and green LED lights on specific faces of the satellite, which blink in a unique pattern, and standard astronomical optical telescopes to track and identify the LED flash pattern (14). LEDSAT, a CubeSat to test this concept on-orbit, launched in August 2021 (36). A test of an exterior-mounted blue LED on a CubeSat was attempted in March 2021, but was indeterminate due to a lengthy period of bad weather at the single designated telescope site.

LED-based systems require relatively clear night-time skies for identification, and dedicated ground equipment (telescope and sensor). The light sources are too faint to allow blind searching of the sky for the satellite; orbital information from a SSA provider is also required to find and track the CubeSat, although the process of tracking the satellite via an optical telescope allows the orbital ephemeris to be updated. Therefore this tracking enhancement alone cannot be used to identify and catalogue the spacecraft. Issues with attitude control on the host satellite can also complicate the identification process. In addition, using LEDs or other light sources on a satellite while in Earth's shadow should be done carefully to minimize interference with astronomical observations. The SatCon1 report (37) on page 6 lists several recommendations to be followed: 1) assure the light source is fainter than apparent magnitude of V ~7 (and the fainter the better), and 2) advance notice of any illumination times, including accurate orbital elements.

12.5.1 Van Atta Arrays and RF Interrogation Receivers

Another method for increasing the ability to track and possible identification of small satellites involves devices that respond when interrogated by a radio frequency (RF) signal of appropriate wavelength. One such system, the CubeSat Identification Tag (CUBIT) shown in Figure 12.4, is similar to the RFID devices used in proximity badges (16). Built by SRI International and partnered with NASA Ames, CUBIT responds with a short burst of information when interrogated by a radio signal of the correct frequency. CUBIT is relatively small and designed to be independent of host vehicle power. The implementations that have flown contain a small battery suitable for 30 days of in-orbit life, which covers the most critical early orbit identification period. It could therefore be coupled with a coded light emitter to overcome the inability of that system to allow object identification. The device is separated into an internally mounted electronics unit attached to an exterior antenna to minimize the exterior footprint of the unit. Two units have flown and were successfully demonstrated in space onboard TechEdSat-6 in 2017 and TechEdSat-7 in 2020. A relatively large ground architecture (in CUBIT's case, a 30 m antenna and an array of antennas) are required to interrogate the system and successfully
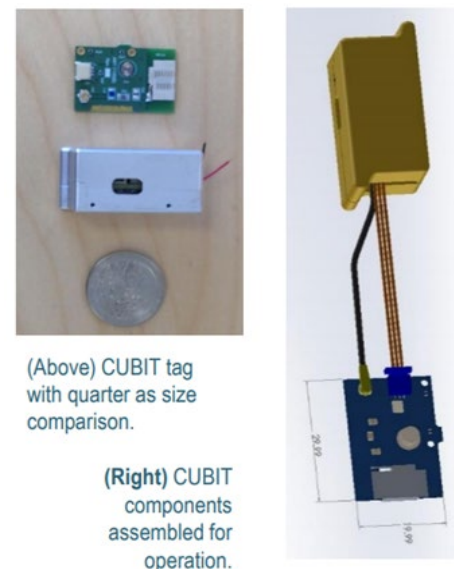
(Above) CUBIT tag with quarter as size comparison.

(Right) CUBIT components assembled for operation.

Figure 12.4: CUBIT. Credit: SRI International.

acquire the low-power response. CUBIT is patent-pending, and SRI has reached commercialization agreements with potential vendors. Future research will continue with a recently awarded AFWERX Phase 1 study.

Another example of an RF-interrogated device is a Van Atta array, a passive device which re-radiates RF energy back toward the source of that energy (20). One such device, the Nanosatellite Tracking Experiment (NTE) consists of a 64-element Van Atta array of tiny, paired antennas tuned to a Ku-band RF frequency, as shown in Figure 12.5 (21). When interrogated at the proper frequency range, the incident RF field received by each antenna is fed to a corresponding antenna via a passive transmission line, where it is re-radiated. This significantly increases the radar cross-section of the object, allowing it to be more easily tracked. Unique identification is difficult,

however, and requires specialized ground stations which tend to be expensive to operate. A satellite carrying a Van Atta array device will be distinguishable from one not carrying such a device, or from one carrying a device tuned to a different frequency band, but two satellites carrying the same Van Atta array will return the same signature. The RF interrogation also requires a ground source of the appropriate frequency. However, Van Atta array devices are entirely passive and extremely low SWaP, making them easy to include on small satellites and CubeSats. NTE devices have flown in space but results from those flight experiments have not been published to date.
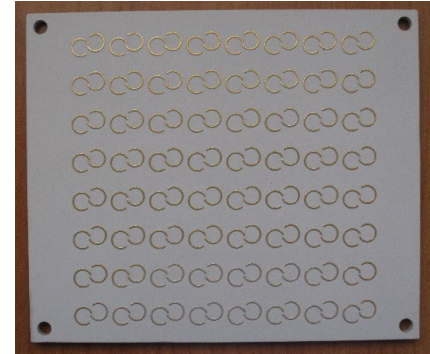
Figure 12.5: NTE Van Atta array retro-reflector in the Ku-band, fits standard 1U panel, tuned to HAX RADAR frequency. Credit: Naval Information Warfare Center.

No capability currently exists at the SSN to process Van Atta Array data, so it would be incumbent on the spacecraft operator to produce its own orbit determination solution from the data to provide an ephemeris to 18/19 SDS for CA screening.

12.5.2 Laser-Interrogated Corner Cube Reflectors

Corner cube reflectors (CCRs), long used in the space industry, are special mirrors designed to reflect laser light back in the direction from which it arrived. They require no internal energy source. When illuminated by a laser, they provide a return signal that can be detected on the ground by a fast camera, as seen in Figure 12.6. Putting a different number of CCRs on a set of CubeSats allows the ground station to differentiate between the CubeSats (i.e., a CubeSat with one CCR will produce a different return signal from another with two CCRs or three CCRs, etc.). One can use a laser and telescope system like those employed by the International Laser Ranging Service (ILRS) (22), which are high TRL and have been operating for decades. Precise orbital information is required to lase the CubeSat and receive a return signal, and the number of satellites that can be uniquely identified is limited by the number of corner cube reflectors that can be attached.
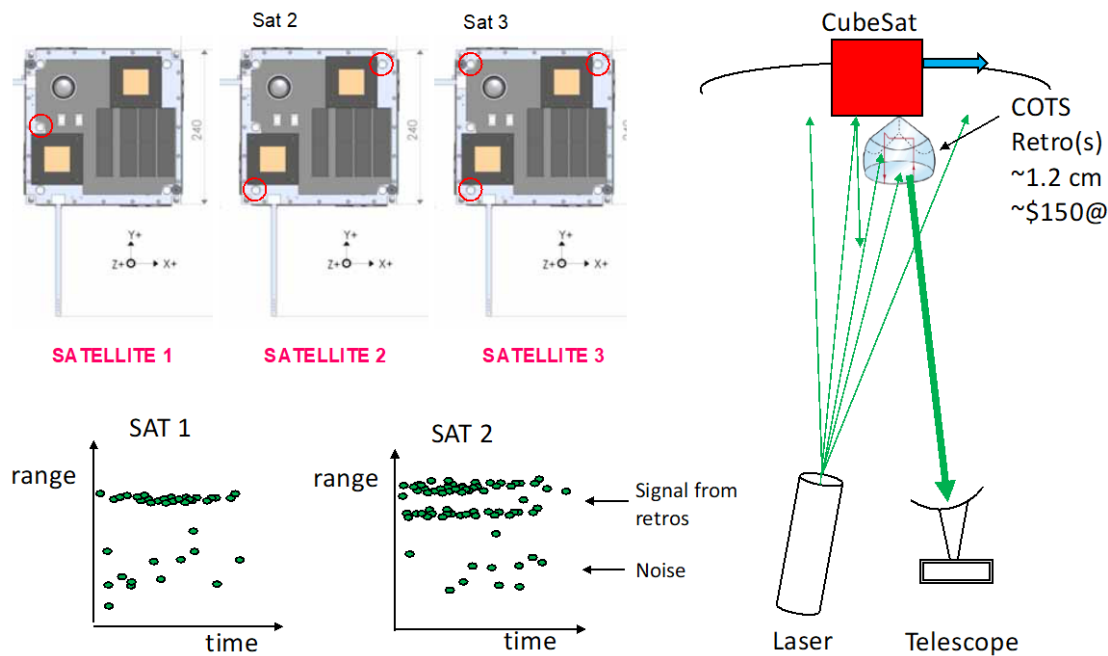
Figure 12.6: Corner Cube Reflectors. Credit: The Aerospace Corporation.

12.5.3 Passive Increase in Albedo

The simplest method of increasing trackability of satellites involves using high-albedo paint, special tape, or other simple methods to increase the optical or radar visibility of a small satellite, allowing it to be more easily detected by ground-based systems (23). White-colored thermal paint has been used for years to increase the ability of satellites to reject heat, which also helps make the satellites more visible and more trackable. Additionally, CubeSats often deploy a mission-specific configuration of wire antennas and/or cylindrical boom structures which can serve as unique identifiers using ground-based optical or radar characterization, as well as simply increasing the radar cross-section or photometric return to make small objects regularly trackable (24). Such approaches are simple, require little to no SWaP, and are readily available, but don't uniquely identify the satellite, and are limited in their effectiveness.

12.6 Future Efforts

Many in the community are aware of the "CubeSat confusion" issue, and there is a groundswell of desire to make progress in mitigating this problem. Regulators have recognized the issue (26), and one of the consolidators, SpaceFlight, Inc., has announced their Sherpa orbital transfer vehicle will take tracking and identification technologies into space as hosted payloads aboard some of their upcoming dispenser satellite flights to increase their TRLs (27).

On the horizon, High Earth Robotics plans to create the Argus constellation – twelve optical 6U HERO-1 nanosatellites with space telescope payloads in GEO that can identify objects, take high resolution images of damaged satellites, and help identify solutions to avoid further decomposition. The constellation is intended to be resilient to interference and communications link interruption (39).

12.7   Summary

Small satellites and CubeSats are likely to continue increasing in popularity, and multi-manifest launches provide a very cost-effective way to get large numbers of satellites to space. Improving the ability to identify and track small satellites in space – especially those deployed in batches from a single launch vehicle – can help both small satellite owners and the entire space enterprise avoid the pitfalls of "CubeSat confusion." Tracking SmallSats from launch through demise is critical to avoid collisions and preserve the space environment from orbital debris. It is important that the end-to-end cost and resulting capability are evaluated when choosing a tracking option to ensure that the needed functionality is available.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email so someone may contact you further.

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Chapter Glossary

(ADCS)   Attitude Determination and Control System

(AEOLDOS)   Aerodynamic End-of-Life Deorbit system for CubeSats

(AFRL)   Air Force Research Laboratory

(ARC)   Ames Research Center

(CRD2)   Commercial Removal of Debris Demonstration

(D3)   Drag Deorbit Device

(DOM)   De-orbit Mechanism

(EOL)   End-Of-Life

(ESEO)   European Student Earth Orbiter

(FURL)   Flexible Unfurlable and Refurlable Lightweight

(GCD)   Game Changing Development

(GTO)   Geosynchronous Transfer Orbit

(HSC)   High Strain Composite

(IADC)   Inter-Agency Space Debris Coordination Committee

(ISS)   International Space Station

(JAXA)   Japan Exploration Space Agency

(MSFC)   Marshall Space Flight Center

(RODEO)   Roll-Out DeOrbiting Device

(RPO)   Rendezvous and Proximity Operations

(SBIR)   Small Business Innovation Research

(SSO)   Sun-synchronous orbit

(STMD)   Space Technology Mission Directorate

(TRL)   Technology Readiness Levels

(TTP)   Thermosphere Test Probe

(UTIAS-SFL)   University of Toronto Institute for Aerospace Studies Space Flight Laboratory

(VESPA)   Vega Secondary Payload Adapter

13 Deorbit Systems

13.1 Introduction

Space debris, also known as orbital debris or space pollution, are derelict artificial objects left in space on purpose or accidentally that include larger nonfunctional spacecraft and rocket bodies, and smaller disintegrated mission-related objects such as lens caps, ejected bolts, or even paint flakes. Additionally, larger space debris are commonly broken up into even smaller fragments due to collisions, erosion, or expelled particles from the spacecraft or rocket bodies. This presents a major problem in the space environment as spacecraft can be damaged or destroyed by space debris collisions due to the very high velocities of the debris objects, which then produces even more space debris.

While space debris is present throughout space, there is a large accumulation around Earth particularly in low-Earth orbit (LEO) where most space operations take place. This is also attributed to the increased launch cadence of small spacecraft and the recent surge in constellations in the past decade. Improved access to space has made LEO accessible and less expensive for more countries, organizations, and institutions to launch small spacecraft, which adds to the associated risks and threats of space debris. Estimates of the accumulation of orbital debris suggest approximately 1,100,000 objects with a diameter 1 – 10 cm, and over 36,500 pieces with diameters >10 cm, are in orbit between geostationary, equatorial, and LEO

Figure 13.1: Orbital debris around Earth. Credit: NASA.

altitudes (1). Figure 13.1 shows a representation of the orbital debris around Earth. Additionally, the orbital lifetime of space debris can be extremely long since atmospheric drag is only effective at <250 km (2).
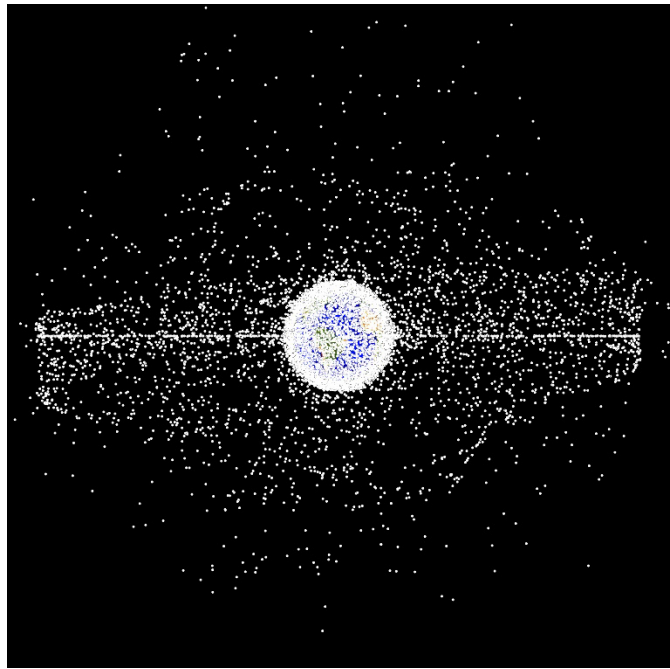
Due to the inherent problem of space debris, there are ongoing policy measures to establish the importance of mitigating and removing space debris. The general guideline is that spacecraft in LEO must deorbit, also known as decay, or be placed in graveyard orbit within a maximum of 25 years after the completion of their mission (3).This standard spacecraft lifetime regulation has been recently updated that directs NASA and other national agencies to reassess current mitigation policies, especially regarding the potential advantages and cost implications resulting from limits on the space debris orbit lifetime (4). These regulations have incorporated spacecraft decay capabilities into mission design.

The rate of spacecraft decay in LEO depends on several factors, including the initial orbit insertion, the ballistic coefficient of the spacecraft, and solar weather conditions, which all play a fundamental role in the ability to comply with decommissioning regulations. Small spacecraft designers have examined various strategies for complying with decommissioning regulations to accelerate spacecraft decay post mission: spacecraft are launched in a lower orbit for a natural decay within a few years or equipped with deorbit systems to encourage altitude decay and ultimately reenter and burn up in the Earth's atmosphere. Natural decay in <5 years can be

achieved for most smallsats at altitudes <400 km, however several smallsat missions must be in orbits beyond 400 km making them susceptible to use deorbit methods.

Spacecraft deorbit methods are either passive or active. Passive deorbit methods have gained maturity since the last iteration of this report, and there are more devices with high Technology Readiness Levels (TRL≥8) that are guaranteed to satisfy current lifetime requirements. Traditionally, passive systems were the main option for deorbiting due to their increased simplicity, however recently active methods are gaining traction. Common active deorbiting requires attitude control and, in some cases, surplus propellant post-mission, such as a steered drag sail that relies on a functioning attitude control system, or on actuators for pointing the sail. Propulsion devices for deorbiting techniques are considered risky due to potential failure or malfunction of either the spacecraft, up until its final stage of decommission, or the propulsive technology itself. Adequate attitude control and navigation capabilities after the mission for a controlled reentry are never a guarantee. Some of the new active deorbiting solutions include a separate spacecraft that can attach to the defunct satellite to bring it down to lower orbits where the satellites can complete the deorbit using their own drag decay.

The influx of small spacecraft in LEO has also developed space situational awareness and space traffic management data. For more information, please see the *Identification and Tracking Systems* chapter.

Chapter Organization

This chapter is organized as follows:

- Orbital Debris Regulations   (13.2)
- State-of-the-Art – Passive Deorbit Systems   (13.3)
- State-of-the-Art – Active Deorbit Systems   (13.4)

Orbital space debris has been a known issue, and there has been ongoing research in this field. It is now a hot topic item and the Orbital Debris Regulations section provides a comprehensive understanding of the current policy regulations for deorbit mitigations, when they were initiated, and the organizations that implement space orbiting debris regulations. The Passive and Active Deorbit System sections contain technology description, summary table of devices; and previous, current and planned missions. This chapter provides a comprehensive guide to existing commercial technologies and technology demonstrations for both methods, and the authors have attempted to highlight technology gaps within existing deorbiting capabilities and current development status on each deorbit method.

The information described below is not intended to be exhaustive but provides an overview of current state-of-the-art technologies and their development status as discussed in open literature. Companies mentioned in this chapter are presented as informational only and do not represent an endorsement by NASA. There is no intention of mentioning certain companies and omitting others based on their technologies or relationship with NASA. It should be noted that TRL designations may vary with changes specific 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 TRL of described technology.

Definitions

- *Disposal* refers to removal of spacecraft from orbital environment.
- *Deorbit* refers to lowering spacecraft's orbital altitude, also referred to as *Decay.*
- *Decay* refers to a gradual decrease of the distance between a spacecraft and the Earth.
- *Atmospheric Drag* refers to molecular collisions with the spacecraft body.

- *Drag Area* refers to the spacecraft surface area experiencing atmospheric resistance.
- *Orbital Lifetime* refers to total time spacecraft is in orbit.

13.2 Orbital Debris Regulations

Space debris has been a concern for several decades, but with visible sightings of reentry fragments of spacecraft and rocket bodies, the urgency to address space debris has grown. NASA's Orbital Debris Program Office[1] was created in 1979, the Air Force Space Debris Research Program was initiated in the 1980s. NASA was among the first organizations to implement plans for mitigation and remediation of space debris in the early 1990s, and in 1993 the Inter-Agency Space Debris Coordination Committee (IADC) was founded internationally.

NASA collaborated with the Department of Defense in 1997 to develop the U.S. Government Orbital Debris Mitigation Standard Practices (ODMSP) (5). The agency's most updated orbital debris guidelines can be found in NASA NPR 8715.6B "NASA Procedural Requirements for Limiting Orbital Debris and Evaluating the Meteoroid and Orbital Debris Environments" (6) and NASA Standard 8719.14C "Process for Limiting Orbital Debris" (3). These technical documents describe the processes and requirements to limit orbital debris for all NASA spacecraft missions. The guidelines, among other considerations, include a limit on the risk of potential human casualties caused by reentering debris, which shall not be greater than 1 in 10000 (5). Of the three spacecraft disposal methods identified – direct retrieval, atmospheric re-entry, and maneuvering into a storage orbit – atmospheric reentry was deemed as the most feasible for the majority of spacecraft missions. Therefore, a maximum 25-year post-mission orbital lifetime (no longer than 30 years after launch or a move into a graveyard orbit for safe storage) was established for all US spacecraft. The rationale for this specific orbital lifetime was based on the least amount of propellant required to maneuver to a lower orbit as predicted by various orbital debris models (7).

The IADC is an entity formed by national and multi-national space agencies, including NASA, ESA, JAXA and several others, and is widely recognized by the international community as the technical authority on space debris. In 2002, the IADC established the Space Debris Mitigation Guidelines to address orbital debris. Their findings and procedures are submitted to the United Nations (UN), as space debris has been one of the main interests of the UN Committee on the Peaceful Uses of Outer Space (COPUOS). In 2007, space debris mitigations guidelines based on the IADC procedures were accepted by the COPUOS and endorsed by the UN (5). The IADC adopted the 25-year orbital lifetime guideline for space objects in LEO.

The U.S. Federal Communications Commission (FCC) regulates all radio communication across the U.S. and all U.S. spacecraft must be licensed for space communications. Since the early 2000s, the FCC has deliberated over how best to mitigate orbital debris from FCC-authorized space activities, and formally adopted debris mitigation regulations (2) in 2004. These FCC regulations include orbital debris mitigation plans as part of license applications, and require applicants to disclose "the design and operational strategies that they will use, if any, to mitigate orbital debris," and to "identify particular methods by which a proposed satellite system will mitigate orbital debris" (2). The FCC adopted the ODMSP 25-year lifetime guideline as well, and commented that the 25-year "rule" should be tightened, as this no longer adequately addresses current orbital debris issues arising from the launch of large constellations and the expected increase in future LEO space activity. On September 29th, 2022, the FCC adopted a new rule for all FCC-licensed satellites within the LEO region (<2000 km) to reduce the lifetime requirement to 5 years after launch (8). There are ongoing discussions at the agency and federal level to determine the final policies.

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[1] https://www.orbitaldebris.jsc.nasa.gov/

Since this updated "5-year lifetime rule" by the FCC, there has been increased focus on space debris removal activities. In April 2023, the FCC created a new Space Bureau responsible for the regulation of satellites and space debris (9). The World Economic Forum (WEF) released the 'Space Industry Debris Mitigation Recommendations' document in June 2023 to standardize a series of recommended behaviors for satellite operations. One of these listed recommendations is to target five years or less after end-of-life for spacecraft removal. The document was signed by several companies including Airbus, The Aerospace Corporation, SES, and Planet. The WEF collaborated with ESA, the MIT Media Lab, and other stakeholders to establish a Space Sustainability Rating (SSR) system to provide a measurable score that can characterize spacecraft mission compliance with the international space debris remediation guidelines (10)(11).

The Federal Aviation Administration (FAA) announced on September 20th, 2023, a proposal to create a new rule to limit the growth of debris from commercial launch vehicles in order to reduce collision risk and limit space debris in populated LEO environments (12). The new regulation will give commercial launch operators specific options for orbital debris countermeasures, requiring disposal of their rocket upper stages by performing a controlled reentry within 30 days after mission completion, moving to a less congested or graveyard orbit (within 30 days), placing them in an Earth escape trajectory (within 30 days), retrieving them with active debris removal within five years after launch, or performing an uncontrolled atmospheric disposal within 25 years.

13.2.1 Considerations for Orbital Lifetime Requirements in LEO

Small spacecraft launched at or around the 400 km altitude naturally decay in under five years, however at orbital altitudes beyond 500 km, there is no guarantee the spacecraft will deorbit within that timeframe and some may have trouble deorbiting in under 25 years. This is due to potential low atmospheric density conditions and the effects on various ballistic coefficients, as seen in Figure 13.2. This graph displays various cases of SmallSats with distinct masses, drag areas, and initial orbits, under the atmospheric density conditions during the 11-year solar cycle maximum and minimum.
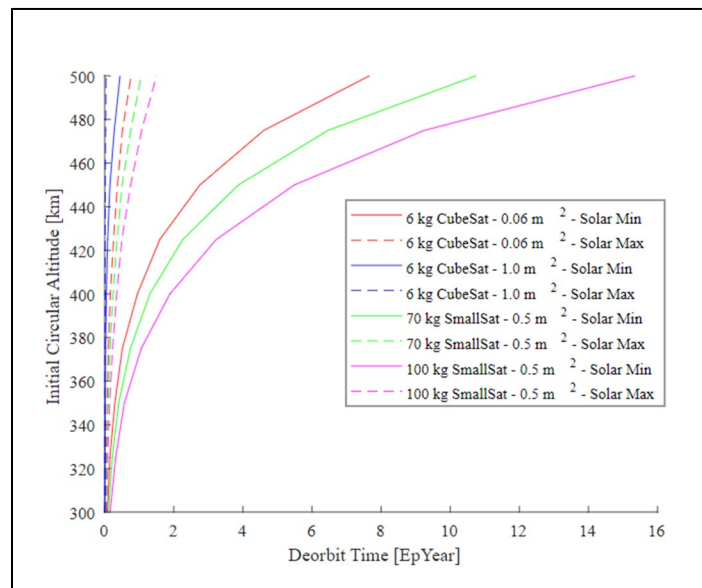

Figure 13.2: Initial orbit altitudes yield different lifetimes depending on the ballistic coefficient of the spacecraft. Three representative area-to-mass ratios are shown. Note that the propagation stops at 16 years, but the initial altitudes yield even longer times. Credit: NASA.

The varying solar weather conditions can affect the deorbit performance for a given altitude and can have a significant impact on orbital lifetimes. The atmospheric drag force that satellites experience is increased during solar maximum, resulting in a faster decay. In this situation, the Sun emits extra energy in the atmosphere and creates higher density layers in LEO altitudes that produce a stronger drag force on the satellites (13). It is common for some missions to plan their launch periods around the solar cycle, and if the stricter 5-year orbital lifetime requirement becomes widely accepted, more companies may want to consider this, as the deorbit time can be reduced by more than 10 years as seen in Figure 13.2.

Another important factor that affects orbit propagation in LEO is the spacecraft's Ballistic Coefficient (BC). The BC is defined in this chapter as the mass to area ratio multiplied by the inverse of the drag coefficient, that is assumed to equal 2.2. By this definition, a spacecraft with a lower ballistic coefficient will decay faster due to the smaller mass to area ratio. As shown in Figure 13.2, a 6U spacecraft with an area of 0.06 m^2 and an assumed mass of 6 kg has a ballistic coefficient of 45, which is significantly lower than a 100 kg spacecraft of 0.5 m^2 with BC of 90.

Since timing the launch for a particular solar weather scenario may not be feasible, another strategy for satellite operators to comply with orbital lifetime requirements is to decrease their spacecraft ballistic coefficient or mass to area ratio. Deorbit technologies such as drag devices can effectively increase the spacecraft's drag area and may become even more important for spacecraft operations in LEO.

13.3 State-of-the-Art – Passive Deorbit Systems

Passive deorbit methods require no further active control after deployment, relying only on natural perturbations and forces to deorbit. They are popular for their minimal mass, power, and cost constraints, and high reliability. Passive methods consist of a deployed structure, such as drag sail, boom, and electromagnetic tether, to reduce the spacecraft's ballistic coefficient. This raises the atmospheric drag on the spacecraft which accelerates the reentry process. These techniques have greatly matured in the last ten years, and recent developments have made passive deorbit technology well-known and space ready.

This section provides information on drag sails, deployable booms, and electromagnetic tethers. In the last decade, there have been several SmallSat missions that have demonstrated passive deorbit capabilities with a deployed sail, boom, or tether. Some notable deorbit missions led by academic programs, research and government organizations, and commercial entities are described under *Missions* for each deployed structure.

13.3.1 High TRL Drag Sails

Drag devices are the most common deorbit device for satellites orbiting in LEO. They are advantageous due to simplicity and small stowed volumes. For certain area-to-mass ratios in altitudes equal to or lower than 800 km, drag devices can be deployed to increase the drag area for faster deorbiting in compliance with the new 5-year requirement. This technology has been implemented in several small spacecraft missions, and several companies and institutions are developing prototypes that are increasingly more mature, providing solutions to the space debris problem for missions that do not have resources for an active system.

Many mission designers are discovering that drag sails do not need to be complicated in material or shape to be effective. It is important to note some missions described below have more information since they are publicly available. Found at the end of this section, Table 13-1 summarizes state-of-the-art passive deorbit systems. Solar sail technology development and mission information are found in the *In-Space Propulsion* chapter.

Missions

In January 2011, the NanoSail-D2 mission successfully demonstrated a drag sail to deorbit the spacecraft from a 650 km altitude and 72° inclined orbit. NanoSail-D2 was deployed from the minisatellite FASTSat–HSV to demonstrate the deorbit capability of a low mass, high surface area sail. The 3U spacecraft, developed at NASA Marshall Space Flight Center (MSFC) and Ames Research Center (ARC), reentered Earth's atmosphere in September 2011.

CanX-7 started at an initial ~700 km Sun-synchronous orbit (SSO), deployed a drag sail in May 2017 and re-entered April 2022. The sail was developed and tested at University of Toronto Institute for Aerospace Studies Space Flight Laboratory (UTIAS-SFL) (Figure 13.3). The CanX-7 deorbit technology consists of a thin film sail that is divided into four individual modules that each provide 1 m^2 of drag area. These sail sections are deployed mechanically with spring booms, which help to preserve the geometry. Each module also has electronics for individual telemetry and command that allowed different sections to be controlled separately to mitigate risk of a single failure, and for custom adaptability to various spacecraft geometries and ballistic coefficient requirements for other missions (22).
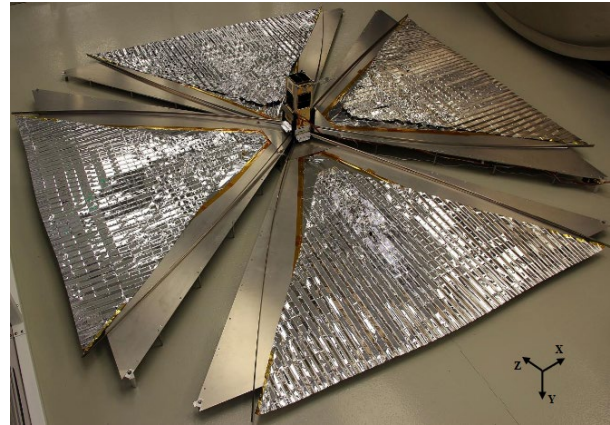

Figure 13.3: CanX-7 deployed drag sail during testing. Credit: Cotten et al. (2017).

The Surrey Space Centre based in the United Kingdom developed DragSail technology that was implemented in a family of missions. Funded by the European Commission QB50 program and the DEPLOYTECH partnership that included German Aerospace Centre (DLR) and NASA Marshall Space Flight Center, among others, the Inflatesail 3U CubeSat first demonstrated the DragSail in 2017 and included a mast/drag-sail technology that successfully deorbited the satellite in just 72 days. This achievement was the first time a spacecraft has deorbited using European inflatable and drag-sail methods (18).

The RemoveDebris mission, developed under the European Commission FP7 program by a consortium of several institutions such as Airbus and the Surrey Space Centre consisted of a 100 kg small spacecraft deployed from the ISS in 2018. One of the experiments it carried was a passive drag augmentation device consisting of a sail. The sail was deployed in March 2019, however, trajectory data showed it only partially deployed since no significant altitude change was measured. The lessons learned from this incident were implemented in another version for the Space Flight Industries' SSO-A mission that incorporated two of these sails. In that case, the assembly did not include an inflatable boom (21).

As part of the ESA CleanSat program, Cranfield Aerospace Solutions developed a variety of drag augmentation systems called Icarus which are similar to other drag devices where drag is increased by deploying a membrane sustained by rigid booms. The first demonstrated was the Icarus-1 sail that flew on the TechDemoSat-1 mission from SSTL launched in 2014 and re-entered May 2019. Another version also flew on the Carbonite-1 spacecraft also built by SSTL that launched in 2015. The Icarus sail is a thin aluminum structure located around the satellite side panel that

Figure 13.4: Icarus-3 drag sail implemented in the Carbonite-1 mission. Credit: Cranfield Aerospace Solutions.

contains four stowed Kapton trapezoidal sails and booms. The Icarus-1 sail system had a mass of 3.5 kg for about 5 m^2 of sail area, and the Icarus-3 sail had a mass of 2.3 kg for 2 m^2 sail area (Figure 13.4). Another technology developed by Cranfield Aerospace Solutions is a de-orbit mechanism (DOM) device similar to the Icarus drag sail but in a smaller cuboid outline. The mechanical system varies from Icarus since the sails are triangular and the booms work as tape springs themselves. This system flew on the European Student Earth Orbiter (ESEO) that was a
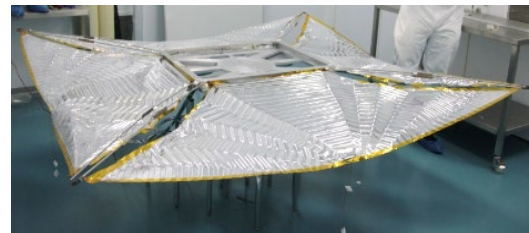
45 kg satellite with several student payloads launched in March 2018. The DOM module has a mass of 0.5 kg and will deploy a sail with an area of 0.5 m^2 to deorbit ESEO after decommissioning (16).

MMA Design LLC patented the dragNET de-orbit system, which is a 2.8 kg module (Figure 13.5) featuring four stowed thin membranes that deploy through a single heater-powered actuator. The sail has an area of 14 m^2 that can effectively deorbit a 180 kg spacecraft at an altitude of 850 km in less than 10 years. The dragNET was a part of the ORS-3 Minotaur Upper Stage and facilitated its deorbit in 2.1 years after launch in November 2013. In October 2022, the dragNET deorbit system was launched as part of the General Atomics GAzelle satellite, as seen in Figure 13.6, and is still operational (15).

Purdue University, CalPoly and Georgia Tech jointly developed a drag device with a pyramid geometry that aims to deorbit a satellite from a geosynchronous transfer orbit (GTO). The Aerodynamic Deorbit Experiment (ADE) is a 1U CubeSat technology demonstration that once deployed, will occupy an area of about 1 m^2 to decrease the spacecraft's ballistic coefficient to the perigee altitude in each pass, resulting in an expected lifetime of 50 – 250 days instead of the initially estimated seven years (42). The ADE mission is slated to launch in 2025.

The ADE technology was licensed to Vestigo Aerospace which is commercializing it with their Spinnaker series of drag sails and was awarded by NASA's Phase II Small Business Innovation Research (SBIR) Program. An initial flight test of an 18 m^2 sail was attempted in September 2021 aboard the first Firefly Aerospace Alpha rocket, however the launch was unsuccessful (45). Vestigo is developing two main products: a sail targeted for small satellites that has a surface area of 1.77 m^2, and a larger 18 m^2 sail for objects weighing up to 1000 kg (46). In 2023, Vestigo was awarded a NASA Phase II-S SBIR contract to contribute to the development of a technology demonstration mission to qualify the Spinnaker 2, an 8 m^2 sail for small satellites, and the Spinnaker 3, more targeted to orbital transfer vehicles and upper stages (47).

NPC Spacemind developed and launched a series of CubeSat missions that demonstrated their ARTICA deorbit system, which consists of a deployable 2.1 m^2 drag sail. The total size of the deorbiting system is 0.3 U which makes it suitable for CubeSats as small as 1U (28). In November 2022 and in January 2023, the DanteSat 3U CubeSat, and the Future-SM3 6U CubeSat, Futura-SM3, were respectively launched

Figure 13.5: [top] The dragNET module. [bottom] dragNET module attached to the GAzelle satellite prior to launch. Credits: MMA design.

and successfully operated with an ARTICA system onboard. These two new missions extend the ARTICA flight heritage after the earlier UrsaMaior, 1-Kuns, and Alpha missions, launched in 2017, 2018 and 2020 respectively (28).
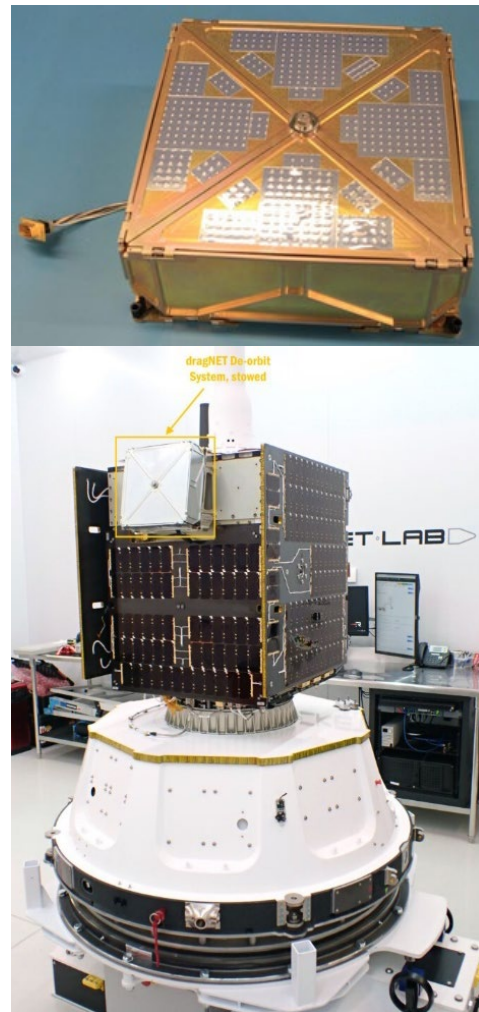
The Planetary Society's LightSail-2 was a 3U CubeSat mission with a solar sail launched in June 2019 and deployed from the Prox-1 satellite once in orbit. The mission demonstrated that solar sail technology can be used in LEO by modifying its orbit altitude along the course of the mission. The 32 m^2 sail was able to extend the mission lifetime by reducing orbital decay and on some occasions, it was also able to overcome drag entirely. In late November 2022, the mission successfully re-entered the atmosphere according to orbital predictions (30).
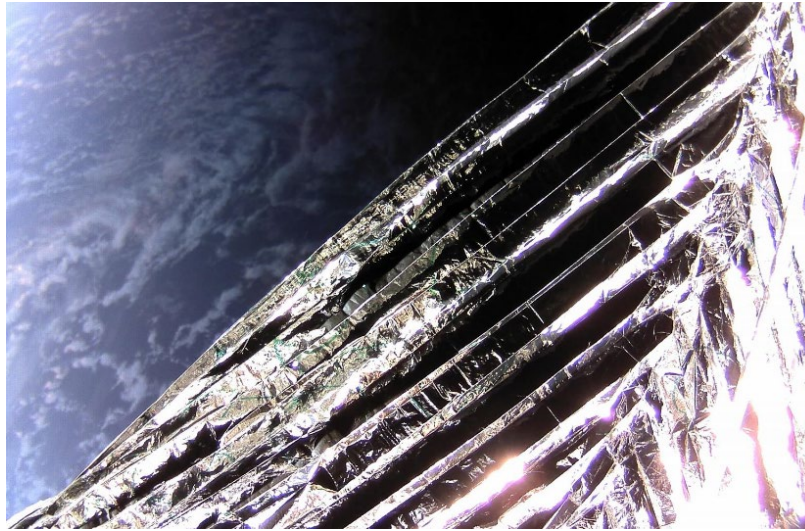

Figure 13.6: The ADEO-2 system deployed in LEO in December 2022, picture captured by the D-orbit's ION spacecraft carrier. Credits: ©HPS GmbH, Germany (www.hps-gmbh.com).

The Drag Augmentation Deorbiting System (ADEO) is a scalable drag sail developed by High Performance Space Structure Systems (HPS). There are various configurations of the ADEO sail: the ADEO-N series is tailored for small satellite missions of 20-250 kg, while the ADEO-M and ADEO-L target larger sizes, 100-700 kg and 500-1500 kg respectively. The ADEO-N series corresponds to sail sizes of 5±2 m^2, while ADEO-M covers areas within 15 ± 5 m^2. There are other smaller versions as well for picosatellites (ADEO-P) and CubeSats (ADEO-C) in particular, and the option to configure the sail size according to customer needs. Various configurations of the ADEO-N product family have been tested already. The NABEO-1, launched in 2018, attached to the center of a Rocket Lab Electron rocket Kick Stage and the sail was deployed 90 minutes after launch (24). In late December 2022, the ADEO-2 sail was deployed with a 10x10x10 cm package from the D-orbit ION-2 satellite carrier (Figure 13.6).

Gama launched its first spacecraft mission, a 6U CubeSat name Gama Alpha, in January 2023 into a ~550 km LEO and is currently still in orbit. This first technology demonstration mission aims to test the deployment and control of their 73.3 m^2 solar sail, and the final phase of the mission will use the sail to rapidly deorbit the satellite (32).

SBUDNIC, a CubeSat designed and built by Brown University students with support from D-Orbit shown in Figure 13.7, AMSAT-Italy, La Sapienza-University of Rome, and NASA Rhode Island Space Grant, demonstrated a practical, low-cost method to cut down on space debris. Rather than taking debris out of orbit after it becomes a problem, this $30 drag device made from Kapton polyimide can be added onto satellites to radically reduce how long they're in space. SBIDNIC was launched on a SpaceX rocket May 2022 as part of the Transporter 5 ridesharing mission. The plastic drag sail was deployed at about 520 kilometers and the satellite re-entered August 2023, about five years ahead of schedule (50).
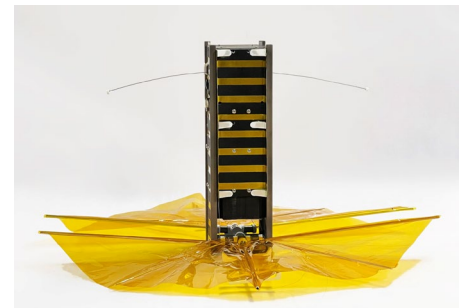

Figure 13.7: SBUDNIC CubeSat with drag sail made from Kapton polyimide film. Credit: Brown Univ.

The Advanced Composite Solar Sail System (ACS3) is a mission developed at NASA Langley and NASA Ames that consists of a spacecraft that will deploy an 81 m^2 solar sail in a 1000 km sun-synchronous orbit (see Figure 13.8). The main objective of the mission is to demonstrate that solar radiation pressure can propel the spacecraft to change the semimajor-axis and obtain a different orbit altitude. The sail is composed of a combination of composite materials with distinct properties, and it will be deployed with lightweight booms from a 12U CubeSat bus, developed by NanoAvionics. The spacecraft was launched aboard an Electron launch vehicle in April 2024. Although the main objective of the mission is to show the propulsive capabilities of the solar sail, the

Figure 13.8: The ACS3 sail fully deployed during its pre-integration fit test. Credits: NASA Langley.

device can be used for deorbiting purposes, and it may be used at the end of the ACS3 spacecraft lifetime for decommissioning (31).

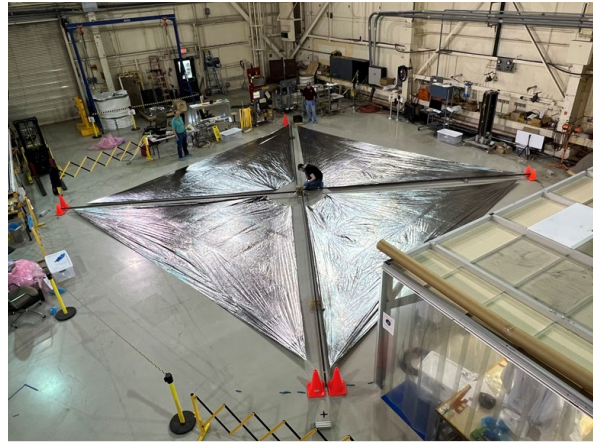**TechEdSat (Passive) Exo-Brake Series**

The Technology Educational Satellite program, also known as TechEdSat-n (TES-n), at NASA Ames Research Center (ARC) has contributed significantly to the development of drag devices. It consists of a series of nanosatellite technology demonstrations in collaboration with several universities including San Jose State University and the University of Idaho. One of the main goals of the program is to test and improve deorbiting techniques and develop a unique targeting capability with their own drag device design known as the Exo-Brake. Being used as both a passive and a controlled active deorbit system, the Exo-Brake is an atmospheric braking system that is distinguished from other drag devices with its parachute qualities instead of a solar sail due to its primary tension-based

Figure 13.9: TechEdSat-10 aft-imaging captures Exo-Brake deployment July 2020. Credit: NASA.

elements – see Figure 13.9. This is fundamental for accurate deorbit targeting since the device must retain its shape without collapsing during those critical reentry moments occurring at the atmosphere interface altitude of 100 km, known as the Von Karman line (35). Development of the Exo-Brake is funded by the Entry Systems Modeling project within the NASA Space Technology Mission Directorate's (STMD) Game Changing Development (GCD) program.
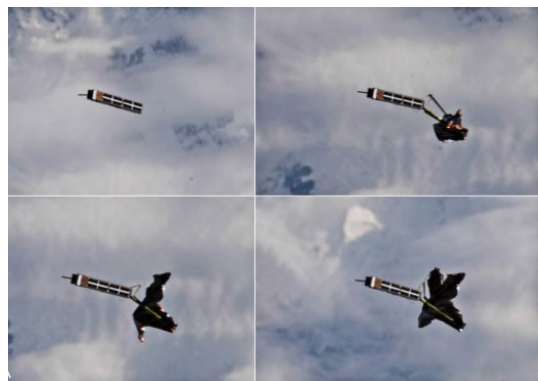
The Exo-Brake was first implemented as a passive deorbit device on TES-3 and TES-4, and then on TES-5 and TES-7. In comparison to other objects without a disposal device, the TES-7 flight is shown with other deployed objects from the same carrier, as seen in Figure 13.10. With a low ballistic coefficient induced by the deployed Exo-Brake, it is seen that the TES-7 re-enters in roughly 1.3 years from an initial 500 km orbit. This design incorporated a compact, low volume gas generator cartridge that inflated the Exo-Brake struts.

More recently, a simpler mechanical deployment scheme was developed (patent initiated) that improves the packaging performance in terms of increasing the ratio of a key performance parameter (deployed-drag area per stowage volume) applied to TES-11. This has been further miniaturized for TES-22 which is expected to launch in December 2024 as a test for a proposed future Thermosphere Test Probe (TTP) that could be part of a small cohort during solar coronal events. The passive Exo-brake (not drag-modulated for de-orbit targeting) has advanced recently for disposal activities to meet the new 5-year de-orbit lifetime requirement. Table 13-1 provides a comprehensive overview on the different TES-n missions with passive Exo-Braking capability.
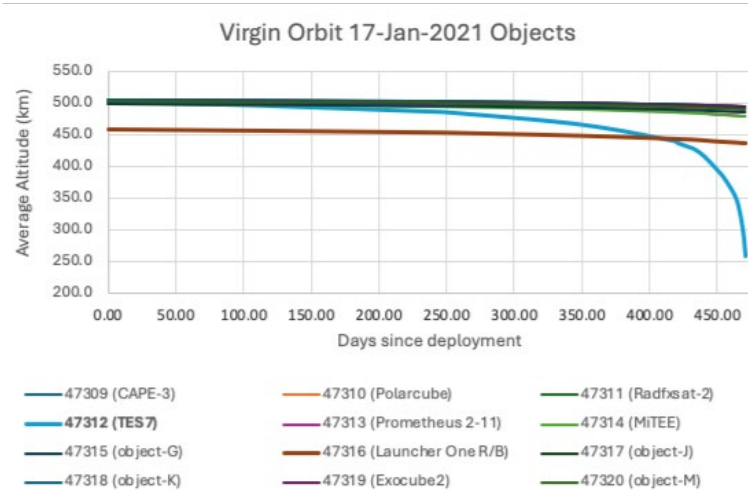

Figure 13.10: TES-7 deorbit compared to other deployed objects from the same carrier. Credit: NASA/Virgin Galactic

Table 13-1: Passive TechEdSat-N Spacecraft						
TES-n mission (Volume)	Deployment (month, year)	Deorbit Duration, Reentry Year	Nominal Drag Area (m²)	Deployed Orbit Alt. & Inc.	Technology Tested Notes	Ref
TES-1 (1U)	October 2012	May 2013	NA	412 km, 51.7 deg	-	(36)
TES-2 (1U)	April 2013	April 2013	NA	223 km, 51.6 deg	First test of Iridium	(37)
TES-3 (3U)	November 2013	January 2014	0.35	413 km, 51.6 deg	Original Passive Exo-Brake	
TES-4 (3U)	March 2015	April 2015	0.35	398 km, 51.6 deg	Upgraded Exo-Brake design	
TES-5 (3.5U)	March 2017	July 2017	0.35	403 km, 51.6 deg	Two-state Drag Modulated Exo-Brake for targeted re-entry	(38)
TES-6 (3.5U)	November 2017	May 2018	0.564	402 km, 51.7 deg	Variable Exo-Brake settings	(39)
TES-7 (2U)	January 2021	May 2022	1.2	499 km 60.7 deg	High Packing Density Exo-Brake	(40)
TES-9 (3U)	Planned†	-	-	-	Automatic deorbiting using the Exo-Brake system	
TES-11 (6U-W)	July 2024	Future*	0.790	508 km, 97.4 deg	Advanced disposal Exo-Brake mechanism	

*At the time of writing, spacecraft is still on orbit.

Table 13-2: Drag Sail Missions									
Product/Mission	Manufacturer	Mission host and launch mass	Device mass (kg)	Initial orbit (alt and inc.)	Launch Year	Deployment Year	Drag Area (m^2)	Ref.	TRL
NanoSail-D2	NASA MSFC/ARC	FASTSAT (4.2 kg)	N/A	650 km 72 deg	2010	2011	10	(3)	7-9
Drag-Net	MMA Design	ORS-3 Deployed a Minotaur Upper Stage (100 kg)	2.8	N/A	2016	2016	14	(14)	7-9
Drag-Net	MMA Design	General Atomics GAzelle Satellite	2.8	N/A	2022	TBC	14	(15)	7-9
Icarus-1	Cranfield Aerospace Solutions	SSTL TechDemoSat-1 (157 kg)	3.5	635 km	2014	2019	6.7	(16)	7-9
Icarus-3	Cranfield Aerospace Solutions	Carbonite-1 (80 kg)	2.3	650 km 98 deg	2015	Future (in-orbit)	2	(16)	7-9
DOM	Cranfield Aerospace Solutions	ESEO (45 kg)	0.5	572 km × 588 km 97.77 deg	2018	Future (in-orbit)	0.5	(16)	7-9
Terminator Tape	Tethers Unlimited, Inc.	Prox-1 (71 kg)	0.808	717 km 24 deg	2019	2019	10.5	(17)	7-9
DragSail	Surrey Space Centre	InflateSail (3.2 kg)	N/A	505 km 97.44 deg	2017	2017	10	(18)	7-9
Exo-Brake	NASA	TechEdSat 5 (3.4 kg)	N/A	405 km 51.5 deg	2014	2015	0.35	(19)	7-9
Exo-Brake	NASA	TechEdSat 7 (3 kg)	N/A	485 x 513 km 60.7 deg	2021	2021	1.2	(20)	8-9
Exo-Brake	NASA	TechEdSat 13 (4 kg)	N/A	499 x 509 km 45 deg	2022	2022	0.083	(20)	8-9
Exo-Brake	NASA	TechEdSat 15 (4.5 kg)	N/A	215 x 285 km 137 deg	2022	2022	0.087	(20)	8-9
removeDebris	Surrey Space Centre	removeDebris (100 kg)	N/A	405 km 51.5 deg	2018	2019	16	(21)	7-9
CanX-7	UTIAS-SFL	3U CubeSat (3.6 kg)	0.8 (4 modules of 0.200)	688 km 98 deg	2016	2017	4	(22)	7-9

Table 13-2: Drag Sail Missions									
Product/Mission	Manufacturer	Mission host and launch mass	Device mass (kg)	Initial orbit (alt and inc.)	Launch Year	Deployment Year	Drag Area (m^2)	Ref.	TRL
NABEO-1	HPS	1U CubeSat (attached to Rocket Lab Kick Stage)	0.85	500 km	2018	2018	2.5	(23)	8-9
ADEO-2	HPS	1U CubeSat (attached to the D-orbit ION carrier)	3.4	N/A	2021	2022	3.6	(24)	9
ADEO-Cube series	HPS	1-50 kg	0.5	LEO	N/A	N/A	2	(25)	7
ADEO-N series	HPS	20-250 kg	0.8	LEO	N/A	N/A	5±2	(26)	9
ADEO-M series	HPS	100-700 kg	4	LEO	N/A	N/A	15±5	(27)	6
ADEO-L series	HPS	500-1500 kg	9.5	LEO	N/A	N/A	20±100	(25)	7
ARTICA (ALPHA)	NPC Spacemind	1U CubeSat	0.285 (0.3U)	5865 Km, 70.16 deg	2020	2020	2.2	(28)	7-9
ARTICA (FUTURA SM 3)	NPC Spacemind	6U CubeSat	0.285 (0.3U)	N/A	2023	N/A	2.2	(29)	7-9
ARTICA (DANTESAT)	NPC Spacemind	3U CubeSat	0.285 (0.3U)	415 km	2022	2022	2.2	(29)	7-9
ARTICA (URSA MAIOR)	NPC Spacemind	3U CubeSat	0.285 (0.3U)	450 km, 97.1 deg	2017	2019	2.2	(28)	7-9
ARTICA (1KUNS)	NPC Spacemind	1U CubeSat	0.285 (0.3U)	N/A	2018	2019	2.2	(28)	7-9
LightSail - 2	The Planetary Society	3U CubeSat	N/A	720 km	2019	2022	32	(30)	9
ACS3	NASA	12U CubeSat	1 (6U)	1000 km SSO	2024	2024	81	(31)	8
Gama ALPHA	Gama	6U CubeSat	N/A	550 km	2023	N/A	73.3	(32)	8-9
NANO dragsail	Frond Space Systems	2U-12U CubeSat	0.25	<650 km	2025	N/A	1	(33)	5-6
MICRO dragsail	Frond Space Systems	50-100 kg spacecraft	<1600	<650 km	2025	N/A	10	(34)	5-6

13.3.2 Deployable Booms

Deployable booms, while not strictly a deorbit device themselves, compose a vital part of many deorbit systems. They are structural components that can be stowed during launch, then deployed once in space to provide the support structure required for various drag sail designs. More specific information regarding deployable booms can be found in the *Structures, Materials, and Mechanisms* chapter.

Missions

The University of Florida has developed the Drag Deorbit Device (D3) 2U CubeSat which provides attitude stabilization and modulation of the satellite drag area at the same time, making the overall solution an alternative to regular ADCS units. Four 3.7 m long tape spring booms form the D3, which can deorbit a 15 kg satellite from an altitude of 700 km. A final design has already been tested and simulated, including thermal vacuum and fatigue testing (51)(52). Figure 13.11 shows two images of the final design. The mission was selected by NASA through the CubeSat Launch Initiative, and on September 6, 2022, D3 was successfully placed in orbit (53).
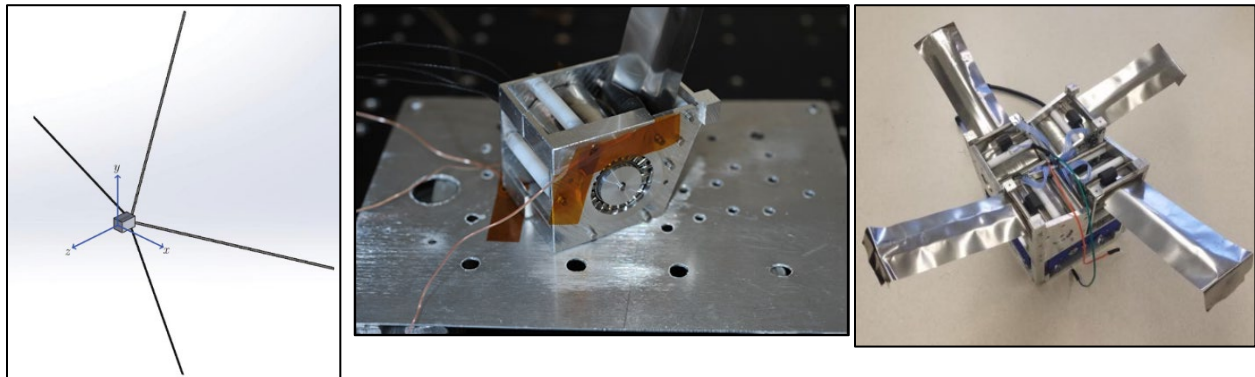

Figure 13.11: D3 CAD design (left), boom inside thermal vacuum chamber (center), and prototype design (right). Credit: Omar et al., 2019, and Martin et al., 2019.

13.3.3 Electromagnetic Tethers

In addition to drag sails, an electromagnetic tether has proven to be an effective deorbit method. This technology uses a conductive tether to generate an electromagnetic force as the tether system moves relative to Earth's magnetic field.

Figure 13.12: Image of the NSTT (left) and the CSTT modules. Credit: Tethers Unlimited.

Missions

Tethers Unlimited (now Amergint Technologies) developed terminator tape that uses a burn-wire release mechanism to actuate the ejection of the terminator's cover, deploying a 70 m long conductive tape at the conclusion of the small spacecraft mission. There are currently two main modules. The first, NSTT for NanoSats has a mass of 0.808 kg. The second, CSTT, is made for CubeSats and has a mass of just 0.083 kg. Figure 13.12 shows an image of both systems respectively. The 70 m long NSTT has been implemented in the 71 kg Prox-1 satellite, launched in mid-2019 by AFRL (17). DragRacer, an experiment jointly developed by Tethers Unlimited, Millennium Space Systems, RocketLab, and TriSept Corp., consisted of a satellite (Alchemy) with the terminator tape, and another satellite (Augury) without it, to characterize the tape performance (56). Alchemy reentered in July 2021 while Augury is still in orbit.

13.3.4 Natural Deorbiting Highways

Finally, recent analysis has shown that there exists a natural network of what may be called 'deorbiting highways/corridors' (55). Judicious injection into specific orbits in LEO may result in natural, relatively rapid orbital decay. There exist dynamical resonances associated with solar radiation pressure, luni-solar perturbations and the Earth's gravitational harmonics. In turn, these dynamical resonances are associated with natural deorbiting corridors; furthermore, the efficiencies of these deorbiting corridors are proportional to the spacecraft's area-to-mass ratio. These LEO dynamical resonances are expressed as a rich network of specific orbital initial conditions. For example, for a nearly initially circular orbit at an altitude of 700 km, and assuming a typical value of the area-to-mass ratio, a spacecraft with an initial inclination of approximately 41 degrees will have a shorter lifetime in orbit than an identical spacecraft but with an initial inclination of 20 degrees. The former example is taking advantage of a naturally existing corridor associated with the dynamical resonances. The downside of using this approach is that small satellites (a) often go as 'piggyback' rides to larger missions and do not have the luxury of picking an orbit, and (b) often go to sun-synchronous orbits, which for typical area-to-mass ratio values are not dynamically close to these natural deorbiting corridors.

13.4   State-of-the-Art – Active Deorbit Systems

While it is still less common to design an active deorbiting system on a spacecraft directly due to mass, size, and power constraints, there are other methods enabling active decommissioning. Several companies, such as Astroscale and ClearSpace, offer active spacecraft-based deorbit services and have already started initial technology demonstration missions to advance their deorbit systems. These deorbit systems consist of separate, dedicated spacecraft that attach to decommissioned satellites to place them into decaying or graveyard orbits. This niche is quite appropriate for the growing number of constellations that are planned to launch in the next few years.

The NASA STD-8719.14C document stipulates all NASA-sponsored spacecraft using controlled reentry processes must have a designed trajectory to ensure that any remaining debris does not impact landmasses. Specifically, debris with a kinetic energy exceeding 15 Joules must stay at least 370 km away from any foreign landmass, or within 50 km from any territory of the United States or the permanent ice pack of Antarctica (3). This requirement implies the spacecraft must be equipped with improved active deorbit technology. There have been advances on deorbit technologies that can either be used on a specific mission or with a commercial service.

This section highlights some of the main stakeholders that are working towards the implementation of active space debris removal services, and some promising technologies that can potentially be used for actively deorbiting spacecraft in the future.

13.4.1 Active Debris Removal Hardware

This section focuses on efforts to develop technologies for active debris removal. These include the active control of the Exo-Brake on the TechEdSat series for targeted re-entry, as well as potential and upcoming missions dedicated to actively removing orbital debris using drag sails, deployable booms and electromagnetic tethers. In addition, some upcoming missions include orbiting vehicles to execute active operations to remove small satellites at the end of life.

TechEdSat (Active) Exo-Brake Series

The Exo-Brake design has morphed from a fully passive device to active control capability that can target a specific reentry location by adjusting the drag device for the optimal ballistic coefficient based on the satellite's orbital determination. The TES-6 mission first implemented this technology with a 3.5U CubeSat (3.51 kg mass) and deployed the Exo-Brake from the rear and

targeted a reentry over NASA Wallops Flight Facility. Although the target area was overshot, analysis shows that a low $4 - 5$ kg m^2 ballistic coefficient configuration would have yielded suitable results if placed at 300 km (see Figure 13.13). TES-6 successfully demonstrated the reentry experiment and the command/control capability by overflying Wallops right before reentering.

To date, the Exo-Brake targeting technology has been demonstrated on TES-6 and TES-10 missions and was going to be implemented on TES-8 except a power system failure occurred after Exo-Brake deployment but before the targeting process. The Exo-Brake on TES-10 was an improved version of the previous TES-5 and TES-6 devices with a wider ballistic coefficient range ($6 - 18$ kg m^2) to enable better control authority for a targeted deorbit flight test over Wallops (58). TES-11 also incorporated this improved Exo-Brake design (35) and was launched July 2024. TES-13 and TES-15 used variations of the TES-7 design for active, controlled mission deorbiting. Table 13-3 provides a comprehensive overview on the different TES-n missions.

Table 13-3: Active TechEdSat-N Spacecraft
TES-n mission (Volume)	Deorbit Technology Type (Active or Passive)	Deployment (month, year)	Deorbit Duration, Reentry Year	Nominal Drag Area (m^2)	Deployed Orbit Alt. & Inc.	Technology Tested Notes	Ref.
TES-8 (6U)	Active	January 2019	April 2020	0.5	406 km, 51.6 deg	Hot Exo-Brake; improved version of TES-5 and -6 designed for continued operation in high temperature environment	(39)
TES-10 (6U)	Active	July 2020	March 2021	0.564	416 km, 51.6 deg	largest iteration Exo-Brake and will perform targeting experiment	(40)
TES-13 (3U)	Active	January 2022	July 2024	0.087	504 km 45 deg	Autonomous navigation and targeted reentry;	(42)
TES-15 (3U)	Active	October 2022	October 2022	0.087	250 km, 137 deg	Hot Exo-Brake re-entry test; unable to complete full test	(40)
TES-22 (1U)	Pending	January 2025	TBD	2.00	~500 km	Rapid Disposal (Exo-Brake /Heliophysics Test Probe)	

*at the time of writing, spacecraft is still on orbit.

RemoveDebris Consortium Partners

The RemoveDebris mission focused on testing several active debris removal experiments with different technologies on mock targets in LEO. In total, the RemoveDebris spacecraft carried two 2U CubeSats, a net, harpoon, laser ranging instrument, and a dragsail. One experiment deployed a 5 m net at simulated space debris using one of the CubeSats, DebrisSat 1, and a balloon. Both DebrisSat 1 and balloon were captured at ~11 m and maneuvered to a lower altitude to re-enter in March 2019 (21). Another active debris technology used a harpoon and a deployable target where the target platform attached to a 1.5 m boom was deployed from the main spacecraft and the harpoon fastened by a tether was fired at 19 m/s to hit the platform in the center. Once that occurred, the boom that connected both harpoon and target platform were joined on one end. However, a tether secured the target in place, avoiding the creation of new debris. This resulted in the first space demonstration of a harpoon technology. The harpoon target assembly had a dry mass of 4.3 kg (21).

13.4.1 Active Debris Removal / Spacecraft Reentry Services

This section focuses on commercial efforts to provide space debris removal services. This commonly involves a 'servicer' spacecraft released from a host-spacecraft that rendezvous with an object, captures it via attachment or docking method, and releases it in a lower altitude for reentry. Several core technologies, such as autonomous navigation, rendezvous and proximity operations (RPO) and robotics, are being developed by NASA and international space agencies, academics and researchers, and the commercial industry to boost the active debris removal service. Some deorbit removal services require the spacecraft to be fitted with a deorbit fixture for their servicer spacecraft to attach to. In response to the surge of small spacecraft constellations, deorbit fixtures are now being integrated on several new spacecraft constellations.

Some companies, like Momentus, are more focused on in-orbit servicing capabilities as a whole, that include deorbit/removal applications. Those companies with a deeper affiliation with in-orbit servicing are featured in the Orbital Transfer / Maneuvering Vehicles (OTV/OMV) section of the *Integrated, Launch, Deployment, and Orbital Transfer* chapter.

Astroscale Holdings Inc.

Astroscale aims to provide services that will address the end-of-life (EOL) scenario of newly launched satellites and be proactively engaged in active debris removal. They are involved in collaborations with a variety of governmental and international organizations (such as the US government, ESA, the European Union, or the United Nations) that aim to address space debris removal. Since 2022, they are responsible for the Cleaning Outer Space Mission through Innovative Capture (COSMIC) mission design as part of the Active Debris Removal (ADR) mission with the UK Space Agency that will remove two defunct British spacecraft in 2026 (59). In 2023, Astro Digital partnered with Astroscale to begin incorporating their Generation 2 Docking Plate on to Astro Digital's modular satellite bus for all future missions (60). Astroscale is also working with national space agencies to incorporate solutions to remove critical debris such as rocket upper stages or defunct satellites. As part of JAXA's Commercial Removal of Debris Demonstration (CRD2) initiative, which focuses on the removal of large Japanese-made space debris in two mission phases, Astroscale will advance ground testing of hardware and software for close proximity operations and refine the design of the capture mechanism. Satellite providers will need to have this plate onboard their spacecraft for the Phase II CD2R mission which will rendezvous and the capture a JAXA rocket upper stage and aims to launch in 2026 (61). Phase I consisted of the 150-kg Active Debris Removal by Astroscale-Japan (ADRAS-J) satellite that identified and took several images of the upper stage. It was launched July 2024 and maintained a constant distance of 50 m (65).

As part of their EOL campaign, the End-of-Life Services by Astroscale demonstration (ELSA-d) mission launched on March 22, 2021 and has successfully demonstrated EOL technologies in LEO (63). ELSA-d consisted of two spacecraft: a 'servicer' and 'client' where the client spacecraft represented a piece of space debris that the servicer rendezvous with and eventually attached to. With respective launch masses ~175 and ~17 kg, the servicer and client spacecraft repeatedly performed several complex demonstrations including magnetic capture and rendezvous operations of both tumbling and non-tumbling cases. In January 2022, the servicer spacecraft successfully released the client counterpart and initiated autonomous relative navigation over the course of multiple orbits as part of the mission plan. As of 2024, the servicer spacecraft is expected to re-enter in 3.5 years.

Planned for launch in 2026, the ELSA-M demonstration mission will leverage lessons learned for magnetic docking and autonomous navigation technologies demonstrated in the precursor mission to ultimately dock with and remove a OneWeb spacecraft. This capture method requires the spacecraft to be equipped with a compatible magnetic capture mechanism such as the

Astroscale Docking Plate (64). OneWeb plans to have Astroscale's docking plate on their future spacecraft (66). An important note is that several science missions are undertaking extensive efforts to make their spacecraft magnetically neutral, which may be a concern for magnetic docking methods.

ClearSpace

Another company that is solely focused on the design and execution of space debris removal is ClearSpace. By advancing robotics to support in-orbit servicing applications such as disposal, transport, inspection, assembly, manufacturing, repair, and recycling, ClearSpace will help establish a market for in-orbit servicing and debris removal. Collaborating with ESA, ClearSpace-1 is a space debris removal mission designed to locate and capture a non-cooperative, tumbling object via a four-armed robotic device (67). The object to be removed was originally a 100 kg VEGA upper stage, however in August 2023 it had collided with another piece of space debris (69). The mission will now rendezvous with Project for On-Board Autonomy (PROBA)-1, which is ESA's first spacecraft with fully autonomous technologies launched 20 years ago, capture it with a group of robotic arms, and then both spacecraft will be deorbited together to a lower orbit for final disintegration in the atmosphere (68).

ClearSpace entered the preliminary design review in October 2023 of the Clearing of the LEO Environment with Active Removal (CLEAR) mission currently projected to launch in the second half of 2026. This study is designed to remove at least two UK non-operational satellites that have been in orbit for more than 10 years in a ~700 km orbit and have a natural deorbit time longer than 100 years (68). The CLEAR spacecraft features unfolding arms used to capture and release target objects.

D-Orbit

Known for their transportation services onboard their ION CubeSat carrier platform, D-Orbit also provides an external solid motor booster specifically for deorbiting purposes. This independent module, known as D-Orbit Decommissioning Device (D3) shown in Figure 13.13, is a solution that is optimized for end-of-life maneuvers that was first demonstrated in 2017 (52). This technology would need to be added prior to launch and is activated from the ground or host spacecraft.
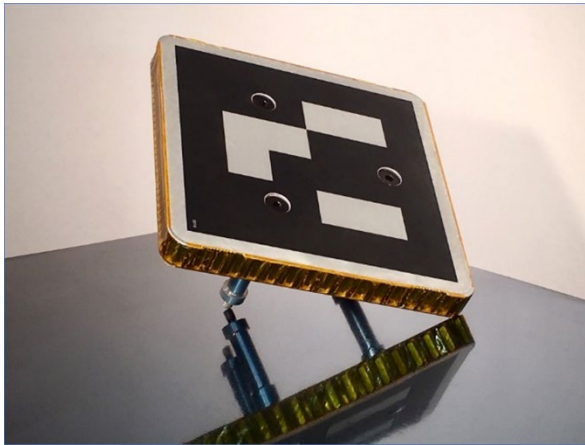
Figure 13.13: D-Orbit D3 module. Credit: D-orbit.

Voyager Space

The main objective of Voyager Space is advancing humanity's presence in space and on Earth. They focus on space exploration and infrastructure, microgravity research, and technological innovations that will contribute to a sustainable space economy and enhance humanity's future. As part of their in-space servicing and assembly, they developed the DogTag, which is an inexpensive, lightweight, versatile, and advanced grappling interface. This fixture enables small spacecraft to be captured by various mechanisms such as magnetic grapples, mechanical arms, electrostatic or gecko adhesive, and even harpoons. This interface allows for spacecraft servicing or decommissioning and is compatible with other techniques to accommodate other customers (72)(73). Figure 13.14 includes an image of the flight DogTags and a table of its main features. In 2019, OneWeb signed a partnership with Altius Space Machines (acquired by Voyager Space in 2020) to include a grappling fixture on all their future launched satellites to make space more sustainable. The first batch of DogTags were launched into space on OneWeb satellites January 2021, and in February 2022; 34 OneWeb

satellites were equipped with Altius DogTags to mitigate future space debris (73). In total, over 500 DogTags have already been launched to space.

Kall Morris Inc.

Bounding Volume	150mm x150mm x 65mm
Total Mass	250g
Mounting Interface	3x M5x0.8 threaded inserts on an 84.5mm bolt-hole circle
Compatible Gripping Methods	Magnetic Capture Adhesive Capture - Electrostatic - Gecko - Hot-Melt - Chemical Mechanical Capture - Pinch-Grasp - Snare Penetrating Capture (Harpoon)

Figure 13.14: DogTag prototype. Credit: Voyager Space.

Another orbital debris research and solution development company with a focus on active debris removal techniques is Kall Morris Inc (KMI). They are developing a commercially viable system designed to detumble an object, whether a client satellite at its end of life or unprepared debris and release it into a deorbit altitude (74). Its Laelaps spacecraft will rendezvous with and attach itself to a debris object, and then release the object to a deorbit altitude. Laelaps is equipped with a mechanically, multi-armed articulated robotic device called REACCH, shown in Figure 13.15 as the tentacle-like structure, that can dock to unprepared space debris. Gecko adhesion is used on the REACCH tentacles to attach to the debris object and easily releases the object when it retracts.

REACCH has undergone ground testing with the assistance of the University of Southern California and is sponsored for ISS testing through the Center for the Advancement of Science in Space (CASIS) for an Astrobee REACCH Demonstration scheduled to launch end of 2024 (74). Laelaps testing will occur after the Astrobee demonstration and KMI aims for in-space testing in 2027, with component ground testing leading up to that time.

Figure 13.15: REACCH interpretation. Credit: Kall Morris Inc.

In 2027, the Space Machines Company's orbital transport vehicle will enter lunar orbit to rendezvous with and capture space junk orbiting the Moon. The Australian-built Optimus orbital transport vehicle (OTV) will rendezvous with, capture, and return dangerous debris from lunar orbit to Earth. Along the way, it will carry Australian payloads to lunar orbit for testing and development of lunar exploration communications and situational awareness infrastructure, before bringing them back to Earth.

TransAstra was awarded a Phase 2 Small Business Innovation Research contract from NASA to keep developing their inflatable capture bag technology for space debris remediation. This device will be capable of enveloping and subsequently removing a non-cooperative object in orbit. A collaboration with ThinkOrbital, a company based in Colorado, includes the transport of the removed debris to an on-orbit processing plant.

13.5   Summary

Space debris regulations are becoming more stringent. Consequently, several deorbit technologies have matured significantly over the course of the last few years. Traditionally passive systems have been more common, have flown on various missions, and have increased to TRL 9 after successful technology demonstrations. Drag sails are the main technology for passive systems, and several companies have already commercialized and sold these products. Other systems such as electromagnetic tethers, deployable booms, or the NASA TechEdSat series Exo-Brake have also already been prototyped and demonstrated in space, now with navigation capabilities and increased reliability. The investment in active systems has also grown significantly. Several companies are offering transfer vehicles to remove debris or deorbit spacecraft at the end of their mission, and compatible systems for spacecraft rendezvous and removal are being developed in parallel as well. As an example, the RemoveDebris mission has successfully tested two different active methods: a net and a harpoon, for future implementation in active debris removal operations. Companies such as Astroscale or ClearSpace are developing missions to remove defunct satellites and are launching precursor technology demonstration spacecraft in the initial stages of their roadmaps. In conclusion, the various deorbit technologies have seen a significant TRL increase since the last iteration of this report and the robustness of the technologies is expected to grow even further as demand for deorbiting services increases with additional launches and new regulations.

For feedback solicitation, please email: arc-sst-soa@mail.nasa.gov. Please include a business email so someone may contact you further.

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Summary

This report provides an extensive overview and assessment of the state-of-the-art (SoA) for small spacecraft technologies publicly available as of September 2024. The reader should be aware that the pace of SmallSat technology advancement overall is rapidly accelerating and varies per subsystem chapter. Technology maturation and miniaturization continues to expand small spacecraft capabilities with the rise in complex SmallSat mission designs. These improved capabilities have broadened the common SmallSat platform resulting in larger CubeSats and smaller SmallSats; the traditional CubeSat platforms of 1U and 3U volume now include up to 16U form factors. SmallSats once designed as <400 kg are now <100 kg with similar capability for less cost.

While still fairly dominated by the traditional CubeSat form factor, this SoA report is starting to reflect increased interest in the larger, more capable SmallSat platforms. This and the rise in SmallSat accessibility and improved state-of-the-art technology indicate the transition into the next-generation SmallSat paradigm. There has been a significant surge in launch opportunities and services for these larger, more capable SmallSat platforms such as rideshares, hosted payloads, and dedicated launchers. Between 2022 and 2024, there was particular focus on missions actively working on rideshares (or dedicated rides) to LEO destinations as stepping stone towards deep space presence and science data collection capabilities of SmallSats. Hosted payload services are also increasingly more availabile for larger SmallSats and other commercial satellites that can accommodate additional instrumentation or technologies for demonstration. Dedicated launches provide rapid integration and greater mission design flexibility, allowing spacecraft designers to better dictate mission parameters, and a wide variety of integration and deployment systems are now available to facilitate access to space for small spacecraft.

This SoA edition reports specific subsystem growth with recent flight demonstrations of innovative technologies on SmallSat missions, enhanced ground station support, improved technical efficiency, emerging sensor technology, and in rideshare and hosted payload opportunities. Recent SmallSat efforts have demonstrated enhanced technical capability which is the stepping stone needed for our deep space understanding and exploration. The successful flights of PTD-3 and CLICK have initiated optical communication technology on spacecraft and the surplus of optical ground station networks support this shift in data collection. In development for small spacecraft, LiDAR sensor technology has applications for improved altimetry and relative navigation that can be applicable for rendezvous, docking, and formation flying. The ongoing Solid-tate Architecture Batteries for Enhanced Rechargeability and Safety (SABERS) project at NASA GRC is paving the way for solid-state batteries with significantly higher energy than the current state-of-the-art lithium-ion batteries and with less hazardous qualities. In the past few years, there has been particular consideration on deployment mechanisms for multiple small spacecraft subsystems such as antenna booms, gravity gradient, stabilization, sensors, sails, and solar panels, and these technologies are gaining space heritage through operations. New deployable structures and solar sail technologies on the ACS3 mission are on-orbit currently for characterization. There is a spike in position, navigation, and timing technology progression in inertial sensors and atomic clocks, and magnetic navigation for near-Earth environments.

CubeSats have demonstrated current state-of-the-art subsystem technology in a variety of science missions and technology demonstrations in the last decade, giving the SmallSat community a realistic expectation of their advanced capabilities. The common CubeSat form factor is evolving from its initial 1U cube design to accommodate more specific and complex science missions; PocketQubes are now used more widely, and the anticipated DiskSat structure will make its debut in the next few years as an alternative to the "CubeSat" concept. The launch of Artemis I provided a closer step towards a deeper understanding of CubeSat capabilities, achievable destinations, and what the deep space CubeSat form factor may look like in the future.

In addition to the technological advancements SmallSats have achieved, launch and hosted payload services are becoming more widespread and several commercial providers are under contract to facilitate the delivery of SmallSat payloads to their destinations. NASA's Launch Services program's new Indefinite Delivery/Indefinite Quantity (IDIQ) mechanism through the Venture Class Acquisition of Dedicated and Rideshare (VADR) launch services to accommodate very low complexity CubeSats (up to more complex Class D missions) and provide FAA licensed launch services capable of delivering payloads to a variety of orbits (1). NASA is working with several American companies to deliver science and technology to the lunar surface through the Commercial Lunar Payload Services (CLPS) initiative. Under the Artemis program, these commercial deliveries present SmallSat designers with opportunities to perform science experiments, test technologies and demonstrate capabilities to help NASA explore the Moon and prepare for human missions. Through CLPS, NASA has awarded nine task orders to five providers to deliver over 40 payloads to the surface of the Moon between 2024 and 2026. The NASA CLPS program began delivering science payloads to the Moon in 2023. CLPS contracts are IDIQ contracts with a cumulative maximum contract value of $2.6 billion through 2028. Companies of varying sizes can work with selected vendors and are encouraged to fly commercial payloads in addition to the NASA payloads (2).

This report will be updated annually as emerging technologies mature and become state of the art. Any current technologies that were inadvertently overlooked in this version may be included in subsequent editions. Updates to technologies listed in this report could be also modified in subsequent revisions. This report is also available online at: https://www.nasa.gov/smallsat-institute/sst-soa. Technology inputs, updates, or corrections can be made by reaching out to the editor of this report at arc-sst-soa@mail.nasa.gov.

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**NASA
Procedural
Requirements**

NPR 7123.1D
Effective Date: July 05, 2023
Expiration Date: July 05, 2028

COMPLIANCE IS MANDATORY FOR NASA EMPLOYEES

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Responsible Office: Office of the Chief Engineer

Appendix E. Technology Readiness Levels

TRL	Definition	Hardware Description	Software Description	Success criteria
1	Basic principles observed and reported.	Scientific knowledge generated underpinning hardware technology concepts/applications.	Scientific knowledge generated underpinning basic properties of software architecture and mathematical formulation.	Peer reviewed documentation of research underlying the proposed concept/application.

Examples:

a. Initial Paper published providing representative examples of phenomenon as well as supporting equations for a concept.

b. Conference presentations on concepts and basic observations presented within the scientific community.

| 2 | Technology concept and/or application formulated. | Invention begins, practical application is identified but is speculative, no experimental proof or detailed analysis is available to support the conjecture. | Practical application is identified but is speculative; no experimental proof or detailed analysis is available to support the conjecture. Basic properties of algorithms, representations, and concepts defined. Basic principles coded. | Documented description of the application/concept that addresses feasibility and benefit. |

		Experiments performed with synthetic data.		

Example:
Carbon nanotube composites were created for lightweight, high-strength structural materials for space structures.

TRL	Definition	Hardware Description	Software Description	Success criteria
3	Analytical and experimental proof-of-concept of critical function and/or characteristics.	Research and development are initiated, including analytical and laboratory studies to validate predictions regarding the technology.	Development of limited functionality to validate critical properties and predictions using non-integrated software components.	Documented analytical/experimental results validating predictions of key parameters.

Examples:

a. High efficiency Gallium Arsenide solar panels for space application is conceived for use over a wide temperature range. The concept critically relies on improved welding technology for the cell assembly. Samples of solar cell assemblies are manufactured and submitted to a preliminary thermal environment test at ambient pressure for demonstrating the concept viability.

b. A fiber optic laser gyroscope is envisioned using optical fibers for the light propagation and Sagnac Effect. The overall concept is modeled including the laser source, the optical fiber loop, and the phase shift measurement. The laser injection in the optical fiber and the detection principles are supported by dedicated experiments.

c. In Situ Resource Utilization: Demonstrated the application of a cryofreezer for CO_2 acquisition and microwave processor for water extraction from soils.

TRL	Definition	Hardware Description	Software Description	Success criteria
	Component and/or breadboard validation in a laboratory environment.	A low fidelity system/component breadboard is built and operated to demonstrate basic functionality in a laboratory environment.	Key, functionality critical software components are integrated and functionally validated to establish interoperability and begin architecture development. Relevant environments defined and performance in the environment predicted.	Documented test performance demonstrating agreement with analytical predictions. Documented definition of potentially relevant environment.

4

Examples:

a. Fiber optic laser gyroscope: A breadboard model is built including the proposed laser diode, optical fiber and detection system. The angular velocity measurement performance is demonstrated in the laboratory for one axis rotation.

b. Bi-liquid chemical propulsion engine: A breadboard of the engine is built and thrust performance is demonstrated at ambient pressure. Calculations are done to estimate the theoretical performance in the expected environment (e.g., pressure, temperature).

c. A new fuzzy logic approach to avionics is validated in a lab environment by testing the algorithms in a partially computer-based, partially bench-top component (with fiber optic gyros) demonstration in a controls lab using simulated vehicle inputs.

d. Variable Specific Impulse Magnetosphere Rocket (VASIMR): 100 kW magnetoplasma engine operated 10 hours cumulative (up to 3 minutes continuous) in a laboratory vacuum chamber.

TRL	Definition	Hardware Description	Software Description	Success criteria
5	Component and/or brassboard validated in a relevant environment.	A medium-fidelity component and/or brassboard, with realistic support elements, is built and operated for validation in a relevant environment so as to demonstrate overall performance in critical areas.	End-to-end software elements implemented and interfaced with existing systems/simulations conforming to target environment. End-to-end software system tested in relevant environment, meeting predicted performance. Operational environment performance predicted. Implementations.	Documented test performance demonstrating agreement with analytical predictions. Documented definition of scaling requirements. Performance predictions are made for subsequent development phases.

Examples:

a. A 6.0-meter deployable space telescope comprised of multiple petals is proposed for near infrared astronomy operating at 30K. Optical performance of individual petals in a cold environment is a critical function and is driven by material selection. A series of 1m mirrors (corresponding to a single petal) were fabricated from different materials and tested at 30K to evaluate performance and to select the final material for the telescope. Performance was extrapolated to the full-sized mirror.

b. For a launch vehicle, TRL 5 is the level demonstrating the availability of the technology at subscale level (e.g., the fuel management is a critical function for a re-ignitable upper stage). The demonstration of the management of the propellant is achieved on the ground at a subscale level.

c. ISS Additive Manufacturing Facility: Characterization tests compare parts and material properties of polymer specimens printed on ISS to copies printed on the ground.

TRL	Definition	Hardware Description	Software Description	Success criteria
6	System/sub-system model or prototype demonstration in a relevant environment.	A high-fidelity prototype of the system/subsystems that adequately addresses all critical scaling issues is built and tested in a relevant environment to demonstrate performance under critical environmental conditions.	Prototype implementations of the software demonstrated on full-scale, realistic problems. Partially integrated with existing hardware/software systems. Limited documentation available. Engineering feasibility fully demonstrated.	Documented test performance demonstrating agreement with analytical predictions.

Examples:

a. A remote sensing camera includes a large 3-meter telescope, a detection assembly, a cooling cabin for the detector cooling, and an electronics control unit. All elements have been demonstrated at TRL 6 except for the mirror assembly and its optical performance in orbit, which is driven by the distance between the primary and secondary mirrors needing to be stable within a fraction of a micrometer. The corresponding critical part includes the two mirrors and their supporting structure. A full-scale prototype consisting of the two mirrors and the supporting structure is built and tested in the relevant environment (e.g., including thermo-elastic distortions and launch vibrations) for demonstrating the required stability can effectively be met with the proposed design.

b. Vacuum Pressure Integrated Suit Test (VPIST): Demonstrated the integrated performance of the Orion suit loop when integrated with human-suited test subjects in a vacuum chamber.

TRL	Definition	Hardware Description	Software Description	Success criteria
	System prototype demonstration in an operational environment.	A high-fidelity prototype or engineering unit that adequately addresses all critical scaling issues is built and functions in the actual operational environment and platform (ground, airborne, or space).	Prototype software exists having all key functionality available for demonstration and test. Well integrated with operational hardware/software systems demonstrating operational feasibility. Most	Documented test performance demonstrating agreement with analytical predictions.

TRL	Definition	Hardware Description	Software Description	Success criteria
7			software bugs removed. Limited documentation available.	

Examples:

a. Mars Pathfinder Rover flight and operation on Mars as a technology demonstration for future micro-rovers based on that system design.

b. First flight test of a new launch vehicle, which is a performance demonstration in the operational environment. Design changes could follow as a result of the flight test.

c. In-space demonstration missions for technology (e.g., autonomous robotics and deep space atomic clock). Successful flight demonstration could result in use of the technology in a future operational mission

d. Robotic External Leak Locator (RELL): Originally flown as a technology demonstrator, the test article was subsequently put to use to help operators locate the likely spot where ammonia was leaking from the International Space Station (ISS) External Active Thermal Control System Loop B.

TRL	Definition	Hardware Description	Software Description	Success criteria
8	Actual system completed and "flight qualified" through test and demonstration.	The final product in its final configuration is successfully demonstrated through test and analysis for its intended operational environment and platform (ground, airborne, or space). If necessary*, life testing has been completed.	All software has been thoroughly debugged and fully integrated with all operational hardware and software systems. All user documentation, training documentation, and maintenance documentation completed. All functionality successfully demonstrated in simulated operational scenarios. Verification and Validation completed.	Documented test performance verifying analytical predictions.

Note:
*"If necessary" refers to the need to life test either for worn out mechanisms, for temperature stability over time, and for performance over time in extreme environments. An evaluation on a case-by-case basis should be made to determine the system/systems that warrant life testing and the tests begun early in the technology development process to enable completion by TRL 8. It is preferable to have the technology life test initiated and

completed at the earliest possible stage in development. Some components may require life testing on or after TRL 5.

Examples:

a. The level is reached when the final product is qualified for the operational environment through test and analysis. Examples are when Cassini and Galileo were qualified, but not yet flown.

b. Interim Cryo Propulsion Stage (ICPS): A Delta Cryogenic Second Stage modified to meet Space Launch System requirements for Exploration Mission-1 (EM-1). Qualified and accepted by NASA for flight on EM-1.

TRL	Definition	Hardware Description	Software Description	Success criteria
9	Actual system flight proven through successful mission operations.	The final product is successfully operated in an actual mission.	All software has been thoroughly debugged and fully integrated with all operational hardware and software systems. All documentation has been completed. Sustaining software support is in place. System has been successfully operated in the operational environment.	Documented mission operational results.

Examples:

a. Flown spacecraft (e.g., Cassini, Hubble Space telescope).

b. Technologies flown in an operational environment.

c. Nanoracks CubeSat Deployer: Commercially developed and operated small satellite deployer on-board the ISS.

Note: In cases of conflict between NASA directives concerning TRL definitions, NPR 7123.1 will take precedence.

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