A Systems Analysis of CubeSat Constellations with Distributed Sensors

by

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Abstract

This thesis explores the use of CubeSat constellations as "gap fillers" and supplements to traditionally complex, multi-sensored satellites, increasing resiliency of the system at very low cost. In standard satellite acquisitions, satellites can take years and billions of dollars to reach operational status. Should there be delays in schedule or on-orbit failures, gaps in data integral to US operations can be lost. CubeSats present a low cost temporary solution. In this thesis, the weather sensing satellite, JPSS-1, is used as a reference case for a traditional multi-sensored satellite. The sensors from JPSS-1 are paired with state-of-the-art CubeSat sensors of similar functions. These CubeSats are used to make up three different constellation architectures which are examined for the revisit times and coverage they offer. These architectures are based on some of the common methods of launching and implementing a CubeSat constellation, a single mass launch, a series of available launches, and a planned configuration. This analysis shows that in some areas, like radiometry, CubeSat sensors are comparable with operational heritage sensors. In the other cases, like optical imaging and hyperspectral imagers, CubeSats have not yet advanced enough or cannot be advanced much more based on the limitations of their size and power. A practical use of a CubeSat constellation is to supplement and augment a traditional system, increasing the overall redundancy and providing data over larger geographic regions and with lower revisit times for a approximately 4.5% of the cost of a traditional satellite.

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List of Acronyms

ADCS Attitude Determination and Control System
ATMS Advanced Technology Microwave Sounder
CERES Clouds and the Earth's Radiant Energy System
COTS Commercial Off The Shelf
CrIS Cross-track Infrared Sounder
CIRAS CubeSat Infrared Radiometer Sounder
CSD Cannisterized Satellite Dispenser
CYNGSS Cyclone Global Navigation Satellite System
DMSP Defense Meteorological Satellite Program
DSCM Demonstration Satellite Cost Model
EON Earth Observing NanoSatellite
EPS Electric Power System
ESA European Space Agency
FOV Field of View
GAO Government Accountability Office
GEO Geosynchronous Earth Orbit
GPS Global Positioning System
GPSRO Global Positioning System Radio Occultation
IR Infrared
ISS International Space Station
JPL Jet Propulsion Laboratory
JPSS Joint Polar Satellite System
LEO Low Earth Orbit
LTDN Local Time of Descending Node
MetOp Meteorological Operational Satellite
MicroMAS Micro-sized Microwave Atmospheric Satellite
MiRaTA Microwave Radiometer Technology Acceleration
MIT Massachusetts Institute of Technology
NEdT Noise Equivalent Differential Temperature
NOAA National Oceanic and Atmospheric Administration
NPOESS National Polar-orbiting Operational Environmental Satellite System
NRCSD NanoRack CubeSat Deployer System
OMPS Ozone Mapping and Profiler Suite
P-POD Poly Picosatellite Orbital Deployer
PICASSO Pico-Satellite for Atmospheric and Space Science Observation
RAAN Right Ascension of the Ascending Node
RAVAN Radiometer Assessment using Vertically Aligned Nanotubes
S-NPP Suomi National Polar-orbiting Partnership
SBIRS Space Based Infrared System
SSCM Small Satellite Cost Model
STK Systems Tool Kit
VIIRS Visible Infrared Imaging Radiometer Suite

Chapter 1

Introduction and Motivation

1.1 Introduction

The use of satellite constellations has become a way of life that affects the average person daily. Many do not realize that much of the information they use regularly comes from satellite constellations. Cell phone location detection services used in navigation and even social media applications make use of the Global Positioning System (GPS) satellite constellation. Weather predictions come from constellations of weather satellites in an international effort to provide current weather data.

These, and most satellite constellations, make use of traditional satellite architectures, where large, complex individual satellites can cost upwards of two billion of dollars and can take more than 5 years to be designed and built [1]. Such constellations are generally designed with extra satellites in orbit, so should one become defunct, another can take its place. There are also spares on the ground that can be launched if on-orbit spares become unavailable. However, in the case of multiple satellite failures, the services provided by these constellations will fail. Because of the slow and expensive procurement process for traditional satellites, returning a constellation to its original state could take years.

GPS, weather, and communications are all services provided by traditional satellite constellations that are used not only by the public, but also by the US military for national defense. The impact of the loss of a single satellite, the time and cost it takes to replace satellites, and the overall lack of resiliency inherent with traditional satellite architectures is a weak point in the US military space presence. This thesis aims to help solve this problem by using CubeSat constellations to supplement traditional satellites with a focus on weather sensing applications.

CubeSats are small, inexpensive satellites, that generally take only a couple years to go from conception to mission operations [1]. The increase in miniaturized instruments and components allows CubeSats to support sensors comparable to those on traditional large satellites. The low cost, short procurement process, and easy access to orbit associates with CubeSats, make CubeSat constellations extremely resilient.

CubeSats can be used to quickly and cost-effectively fill the hole in coverage left by one defective large satellite. The loss of one CubeSat in a constellation does not cause the same disruption as the loss of one large satellite and it is much easier to replace the lost CubeSat. A CubeSat supplement to a traditional satellite constellation will make the constellation more resilient, better securing the continued service that constellation provides.

This thesis demonstrates examples of improvements in resiliency using three CubeSat constellation architectures to augment the first satellite of the Joint Polar Satellite System (JPSS-1). The CubeSat constellations are made up of five different CubeSats that best match the function of the five instruments on JPSS-1. These five instruments and their paired CubeSat, respectively, are:

- Advanced Technology Microwave Sounder (ATMS) / Micro-sized Microwave Atmospheric Satellite (MicroMAS-2)
- Ozone Mapping and Profiler Suite (OMPS) / Pico-Satellite for Atmospheric and Space Science Observations (PICASSO)
- Clouds and the Earth's Radiant Energy System (CERES) / Radiometer Assessment using Vertically Aligned Nanotubes (RAVAN)
- Visible Infrared Imaging Radiometer Suite (VIIRS) / Dove CubeSat

- Cross-track Infrared Sounder (CrIS) / CubeSat Infrared Radiometer Sounder (CIRAS)

To compare these sensors and constellation architectures, sensor specifications of each CubeSat are compared to their corresponding JPSS-1 sensor. The sensor specifications analyzed include nadir/spectral resolutions, number/frequency of channels, and radiometric accuracy. A cost analysis of a comparison of total program cost between the constellation architectures and the single JPSS-1 satellite also supplements this analysis. Finally, revisit times for each of the CubeSats in the three constellation architectures are compared to the same metrics from the JPSS-1 satellite senors.

This thesis shows that while current CubeSat sensors can comparably match ATMS, CrIS, and CERES of JPSS-1, they still require development to compare to OMPS and VIIRS. This analysis finds that CubeSats are currently better suited as supplements to large satellites rather than a replacement for them, increasing the resiliency of the system and acting as "gap-fillers" for approximately 4.5% of the cost.

1.2 Satellite Constellations

Satellite missions can generally be described as one of two types of architectures: monolithic or distributed systems. Monolithic systems use one satellite to complete mission goals. Distributed systems use more than one satellite to successfully complete the mission. While there are a number of definitions of distributed systems, *Shaw, et al.* gives gives one that is particularly appropriate to this thesis: a distributed satellite system refers to a system of many satellites that are distributed in space to satisfy coverage requirements where the key characteristic is that more than one satellite is used to satisfy the overall coverage requirements [2], [3].

Distributed systems of satellites can come in different forms and classifications. One such classification is by the configuration of the system architecture. The most commonly recognized architectures of distributed space systems are rendezvous and docking missions, formation flying, swarms, and constellations [2], [3]. Rendezvous and docking missions include two spacecraft flying in close proximity to each other. In formation flying, spacecrafts are 10s to 100s of kilometers apart, keeping their position with respect to each other. Swarms are hundreds of satellites launched in a group. Finally, in constellations, spacecraft remain in their own orbital plane and are spread for global coverage [2], [4]. This thesis focuses on Earth observing satellite constellations for weather sensing.

1.2.1 Constellation Architectures

Satellite constellations can come in many shapes and sizes, from the GPS satellite constellation made up of thirty-one 1665 kg satellites in medium Earth orbit to the Planet Flock constellation made up of approximately a hundred 4 kg nano satellites in low Earth Orbit (LEO) [5], [6]. Constellations generally provide world wide coverage or near constant coverage of a certain region.

There are two primary constellation architectures that have been designed for optimal global coverage [7], [8], [9], [3]. These are the Walker constellation and the streets of coverage constellation. Based on the work of J.G. Walker, in the Walker Delta Pattern, satellite orbits are at a common altitude and inclination with evenly distributed ascending nodes. Satellites are distributed evenly throughout the orbital planes [10], [8]. This is shown in Figure 1-1. The Walker method is useful for measurements focused on a band between two latitudes [7].

Figure 1-1: A Walker constellation of 9 satellites, three satellites evenly spaced per plane, three planes with evenly spaced RAANs, all planes at a 30° inclination.

The streets of coverage method uses polar, or nearly so, orbits with right ascensions of the ascending node that are evenly spaced around one hemisphere of the Earth as seen in Figure 1-2 [8]. This makes all the satellites on that hemisphere ascending with all satellites in the other hemisphere descending. A modified version of this is the Polar Non-symmetric design, which uses the streets of coverage idea except that the right ascensions of the ascending node are no longer evenly spaced and are distributed to focus coverage in one area [7]. The streets of coverage method is best used for polar orbits [7].

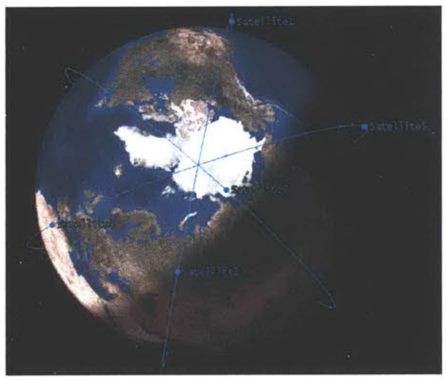

Figure 1-2: A Streets of Coverage constellation of 9 satellites, three satellites evenly spaced per plane, three planes with RAANs evenly spaced around one hemisphere, all planes at a 90° inclination.

Some other constellation architectures that focus more on regional coverage are Geosynchronous, Ellipso, and String of Pearls [3], [8]. Geosynchronous constellations can achieve global equatorial coverage with three satellites in geosynchronous orbit. The Ellipso constellation uses a combination of satellites in elliptic orbits to focus coverage on a region for a period of time. The String of Pearls architecture is currently used by the A-train weather satellite constellation and involves multiple satellites in the same orbital plane, each closely followed by another just a few minutes apart [11].

1.2.2 Cost and Schedule

Although traditional constellations have enabled capabilities that have changed society (GPS, advanced weather monitoring, global communications, global imaging etc), there are a number of issues with the current system of implementing and deploying satellite constellations. The issues on which we focus are cost, schedule, and resiliency.

Traditionally complex satellite constellation programs generally cost billions of dollars [12], [13], [14], [15]. Table 1.2 gives general estimates of expected costs and schedules for different sizes of satellites [1], [16].

Table 1.1: Table of Typical Satellite Costs and Schedules by Size [1], [16]

Satellite Type	Cost (USD)	Time of Development
Conventional Large	0.1-2 B	>5 years
Mini	10-100 M	4 years
Micro	2-10 M	3 years
Nano	0.2-2 M	1 year
Pico	20-200 k	<1 year

Unfortunately, as programs develop, these estimates on both cost and schedule are often exceeded [14]. To keep within budget, many programs implement descope options, leaving gaps in expected constellation capabilities.

For example, the Space Based Infrared System (SBIRS) program was started by the Department of Defense (DoD) in 1999. Initially planned with four satellites in geosynchronous Earth orbit (GEO), SBIRS was intended to replace and improve the United States' missile detection system. In the initial proposal, SBIRS was expected to be operational by 2004 at a cost of approximately \$4.2 B [12]. Instead, SBIRS has grown to require six satellites to complete its mission, and is now estimated to have cost approximately \$17 B as of the end of FY2015. It still has not become fully operational with only three of the six planned satellites currently in orbit [12], [15].

The National Polar-orbiting Operational Environmental Satellite System, NPOESS, is a program that has tried to solve its schedule and cost over runs with program descopes. In May 1994, the National Oceanic and Atmospheric Administration (NOAA) and the DoD were instructed to combine their weather monitoring satellite systems into a new constellation, NPOESS. The first NPOESS satellite was to launch into its sun-synchronous orbit in 2008. When an operational NPOESS satellite still had not launched by 2009, what was to be only a demonstration satellite, Suomi National Polar-orbiting Partnership (S-NPP) was instead made operational to minimize gaps in weather monitoring coverage. As schedule delays and costs increased, the NOAA and DoD programs were separated in 2010, ending the NPOESS constellation. During this time approximately $2.9 B was spent on the NPOESS program [13].

The DoD portion of the project became the Defense Weather Satellite System. This was canceled in 2012 when it was decided that new technology was not yet needed. It was more cost effective to use two satellites from the previous DoD weather constellation (Defense Meteorological Satellite Program) that were still in storage on the ground [17]. The NOAA portion of NPOESS was turned into the Joint Polar Satellite System, JPSS. This is a system that plans to launch a single weather satellite every five years, at the end of the previous satellite's operational lifetime. Currently 4 JPSS satellites are planned, but only two have been funded through the original proposal. Since its inception, JPSS has had a number of instruments and satellites descoped as cost overruns became unavoidable. According to the original proposal, the first JPSS satellite should have been operational in 2015 [13]. It is currently planned for launch in late 2017. The JPSS constellation is expected to cost $11.3 billion, including the $2.9 billion spent previously on NPOESS.

These cost and schedule overruns are just two examples of what has become an unfortunate pattern in the DoD space acquisitions process. The satellites that usually cause such overruns are those of extreme complexity and customization. It is the complexity of the multi-sensored satellite, requiring customization, that leads to budget and schedule overruns. To make up for these overruns, many space programs have had to be massively descoped. Descope options are usually concerned with reducing the number of satellites, reducing their capabilities, or ending the particular program altogether. In a report to Congress, these descoped missions were found to have left capability gaps in national security areas like missile warning (SBIRS), military communications, and weather monitoring (NPOESS/JPSS) [14].

1.2.3 Resiliency

In January 2007, China successfully conducted an anti-satellite missile test, destroying one of their own satellites with a ground based missile. Not only did this event harm the space environment with the debris created, but it represented a shift in the way the US must protect its space assets. This test demonstrated that satellites are no longer safe from ground interference [18]. US satellite systems are extremely vulnerable in this regard because of their large, complex, multi-sensord nature. With increased launch costs, programs are more frequently trying to fit many capabilities on one satellite to minimize the number of launches necessary to achieve the same mission goals [14], [18], [13]. This can make the loss of even one satellite debilitating.

The inflated cost and schedule of traditional satellite constellations make it difficult to replace a malfunctioning satellite; in other words, many US space assets have poor resiliency. It can take years to replace a satellite, leaving a gap in data during that period [1], [16]. Some constellations are planned with spare satellites, sent to an orbit where they can be moved replace malfunctioning satellites if necessary. However, in many cases, the spares are positioned in such a way that it could take months to a year to move them to the correct orbit [19].

This was the case with the original Iridium constellation. A commercially owned LEO constellation that became operational in 1998, Iridium uses 66 satellites to provide worldwide coverage and communications services to its customers, including the Pentagon. By 2016, due to unexpected satellite failures, the Iridium constellation was down to 64 out of the minimum 66 needed operational satellites for global coverage. This left a gap in coverage of several minutes [19]. Even though there were spares in orbit, their positions were such that they could not fill the needed gap before the launch of the next generation Iridium satellites. The timing of the launch of the new Iridium satellites in the beginning of January 2017 luckily filled the gap in coverage last which had only lasted a few months. Had the operational satellites begun to fail earlier or had the new satellites been delayed, the gap would have persisted until the spare satellites could be moved into position, upwards of a year.

The JPSS satellite also gives cause for resiliency concern. The JPSS constellation is to replace the NOAA POES (Polar Operational Environmental Satellites) constellation. The POES constellation is a series of satellites that were launched beginning in 1960 until the last of the series was launched in 2009 [13]. Three of the final satellites are still in commission today. Since each was only built for a mission lifetime minimum of 2 years, some on-orbit systems have begun to fail [20]. On NOAA-15, of the three Atmospheric Microwave Sounding Units, 2 are experiencing major problems and have lost several channels. Its Advanced Very High Resolution Radiometer and High Resolution Infrared Radiation Sounder have also experienced serious problems along with the communication and attitude determination system [20]. NOAA-18 is in slightly better shape having only experienced severe problems with the Solar Backscatter Ultraviolet Radiometer and the High Resolution Infrared Radiation Sounder [20]. NOAA-19 is still fully functional although its High Resolution Infrared Radiation Sounder has experienced minor problems [20].

The first installment of the JPSS constellation, S-NPP, was originally a demonstration satellite for this constellation mission. S-NPP was turned operational when it became apparent the operational JPSS satellites would not be ready for many years. Launched in 2011, S-NPP was designed for a 5 year lifetime although it contains 7 years of resources. Its replacement, JPSS-1, is currently due to launch in September 2017. While all systems of S-NPP are currently operational, any delay or failure of JPSS-1 could extend the schedule of a replacement beyond S-NPP's lifetime [13].

The degraded state of the POES constellation and the age of S-NPP, both having exceeded their design lifetimes, bring worries that if JPSS is not launched before a major failure of S-NPP, there could be a gap in weather monitoring data as shown in Figure 1-3. Should S-NPP fail completely before the launch of JPSS, NOAA will lose at least 45% of the global coverage it needs for weather models [13].

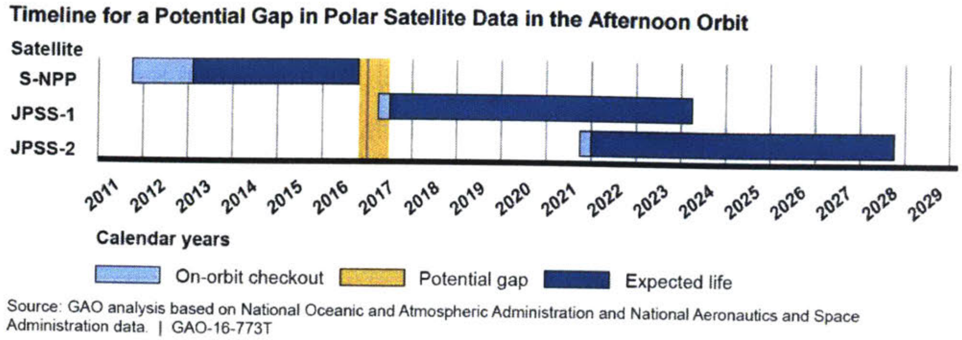

Figure 1-3: A potential gap in weather coverage data exists between the S-NPP design lifetime and JPSS launch [13].

To date, the US has been relatively lucky with the timing and number of satellite failures as there has not been a major disruption of space assets. However, it is clear that the current trend of making constellations out of few multi-sensored satellites creates a system that is susceptible to catastrophic failures. When the loss of a single weather sensing satellite can account for almost half of the coverage needed to accurately predict weather, there is a clear vulnerability.

The extensive time is takes from program inception to mission operations for these types of large custom satellites creates more problems than just the long wait for replacements. Any technology must have significant heritage to be chosen for use on these satellites to minimize risk and protect the large investment of time and money. For a new satellite, this technology is proposed at the beginning of the program, and then that technology is only brought to fruition five or ten years later. With the necessary heritage and acquisitions process, that technology is no longer at the cutting edge when put to operational use. It is actually even 20 years behind the state of the art [18]. The slow satellite acquisitions process delays the process of satellite technology development. A quicker turn around of satellites would allow updates and improvements to the next round of launches for a constellation.

1.2.4 Distributed Sensor Systems

Many of the issues addressed in the previous two sections could be addressed if, instead of one large multi-sensor satellite, the satellite system were distributed into smaller satellites, each with less complexity. Smaller satellites generally cost less and are faster to build, to be discussed in Section 1.3. The distributed architecture of the system would make it a more difficult target, less susceptible to attack. Even if one satellite malfunctions, others in the system will continue to provide data that would otherwise have been lost if the system were integrated as one monolithic satellite. The faster schedule and turn around of smaller satellites would also allow replacements to the constellation be made cheaply and quickly. The faster schedule can also make revisions to the satellite design possible, improving the technology of the satellite with each launch.

Shaw, et al. studies these distributed systems, explaining that there are two conditions for a system that may make distributing it more cost effective [2]. The conditions are when the components to be distributed:

1) Are a large part of the system cost

2) Drive the replacement needs of the system [2].

The components to be distributed for the weather sensing case are the main sensors of the system, ATMS, OMPS, CERS, CrIS, and VIIRS. The sensors do drive replacement needs; with the loss of the primary sensors, the spacecraft will need to be replaced for the capability to be maintained. The monolithic satellites discussed earlier make themselves prime candidates for distribution according to the conditions set here.

A report by Commander of the Air Force's Space and Missile Systems Center, Gen Ellen Pawlikowski, suggests that sensor distributed architectures have military advantages. The smaller satellite size will increase launch opportunities while decreasing launch costs. This could reduce complexity allowing for more streamlined and predictable program baselines. The smaller size also reduces operational and economic consequences of losing a vehicle [18]. Overall, such distributed architecture should allow for less expensive risk management and mission assurance. Distributed architectures are suggested here to combat the growing problems of poor resilience, fragile constellations, and escalating costs.

1.3 CubeSats

The "small" part of small satellites can be defined by volume, power consumption, and mass, although mass is the attribute most commonly used in the definition. Small satellites are broken down in different categories based on their masses as defined in Table 1.2. There are multiple standards across agencies on how to define the categories of small satellites. The definition presented here comes from a standard set by the International Academy of Astronautics [21].

Table 1.2: Table of Types of Small Satellites [21]

Small Satellite Type	Mass (kg)
Mini	<1000
Micro	<100
Nano	<10
Pico	<1

As seen in Table 1.2, nanosatellites are generally defined as a satellite with a mass between 1.0 and 10.0 kg [22]. A CubeSat refers to a class of nanosatellites that conforms with the CubeSat Standard developed by Jordi Puig-Suari of California Polytechnic State University and Bob Twiggs of Stanford University in 1999 [23]. This standard was developed to keep satellite costs and development time low to benefit university programs. With the development of the CubeSat Standard, a student could conceivably design, build, test, and launch a satellite during undergraduate or graduate programs. The CubeSat standard defines a standard container size, mass limits, and power limits of the nanosatellite. A set standard facilitates the use of existing commercial off the shelf (COTS) components as well as motivates development or additional components that provide desired functions but that conform to the standard such as standard launchers and deployers.

The CubeSat Standard describes a 1 unit, 1U, CubeSat as a 10 cm^3 cube volume with a maximum mass of 1.33 kg [24]. These units can be added together like blocks stacked on top of each other to create larger satellites. The most common sizes of

CubeSats range from 1U to 3U as shown in Figure 1-4. For reference, a 3U CubeSat is about the size of a loaf of bread. These are currently the sizes supported by most orbital deployers, and therefore are the most popular. Larger sizes such as 6U and 12U CubeSats are gaining popularity as CubeSat missions become increasingly complex and new deployers become available [25].

Figure 1-4: CubeSat Sizes: 1U, 2U, 3U, 6U, and 12U [26]

CubeSats are generally launched as secondary payloads, smaller payloads that share a launch with a larger, primary payload. This drastically cuts down the cost to get to orbit. The limits set by the CubeSat Standard keep CubeSats as a low risk to the primary payload. However, there are launch vehicles that have been designed specifically for sending up small satellites as their primary payloads [27]. A few of these are the Air Force's Super Strypi, Virgin Galactic's LauncherOne, and Orbital's Pegasus Launcher.

Once launched, there are a few different CubeSat specific deployment systems that can be used to deploy a CubeSat from its launch vehicle. These options include the Poly Picosatellite Orbital Deployer (P-POD) from Cal Poly, the Cannisterized Satellite Dispenser from Planetary Systems Corporation, the Naval Post Graduate School's CubeSat Launcher, and an international version, ISIPOD from ISIS Space.

CubeSats can also be sent to the International Space Station, ISS, and launched from there using the NanoRacks CubeSat Deployer System (NRCSD).

The benefits of small satellites are well known [21]. The most important benefit is the low cost of a CubeSat. This primarily comes from both the short time between project conception and mission operations and the use of commercial components and (often) student labor. A CubeSat can take approximately a year to design, build, and test [16]. This reduced development time requires acceleration of the design process and a reduction in the amount of documentation and testing. There is little environmental testing performed on CubeSat components, especially those bought commercially. Universities can expect to spend between \$50k and \$200k on hardware. Labor costs will add another \$80k/year for each student. This gives a final program cost of \$1M -\$2M to design, build, test, and launch a CubeSat [28]. Including custom scientific instrumentation requires additional funds for the development of the payload, which can run program costs in excess of \$2 million. This is still a relatively low price compared to large satellites which can cost over \$2 billion [1].

The development of the CubeSat standard has allowed the space industry to design and commercialize low-cost parts for CubeSats, supporting a plug-and-play approach which reduces costs through mass production of parts. Companies like Clyde Space, Blue Canyon, and Pumpkin Inc, provide parts ranging from batteries, solar panels, electric power systems (EPS), radios, attitude determination and control systems (ADCS), and structures that meet CubeSat specifications. Some of these companies have moved toward commercializing the entire CubeSat bus, and some have reduced or eliminated their a la carte component options as a result, in an attempt to encourage purchases of the complete bus instead of the individual components. This is an unfortunate development for researchers and innovators that require and benefit from having diversity in available components. It also greatly reduces the opportunity for student education in building nanosatellites.

Components designed for CubeSats will not have long lifetimes or be hardened against radiation [29]. This keeps costs for parts low and is in line with the CubeSat concept of creating satellites quickly and cheaply for short missions. COTS components have significantly reduced many of the costs associated with designing and building custom parts for each CubeSat mission.

Another reason small satellites are becoming more popular is because of increased mission opportunities [21], [30]. Since CubeSats are secondary payloads, they can be launched on a number of different vehicles. CubeSat missions do not have to plan their own launches, they need only find one that meets their schedule and orbit requirements (if any). The use of ride sharing also greatly reduces the cost of the launch, as the primary payload absorbs most of the cost. For instance, the purchase of one SpaceX Falcon 9 launch to a GEO transfer orbit is \$62 million, while the average cost of a 3U CubeSat to the same type of orbit is around \$915k [31]. Even for a small satellite of 300 kg, the price is still lower at \$11.2 million because multiple payloads can share the cost of the ride [31].

Despite the benefits of nanosatellites, because CubeSats are generally designed and built quickly, they are minimally tested and see a high failure rate of about 28%, where failure is defined as not achieving any of the primary mission objectives [16]. In comparison, only about 11% of all satellites launched between 1957 and 2009 could be classified with the same type of failure [16]. It should be noted that this discrepancy in failure rates could come from the high percentage of CubeSats that are designed by university students lacking the necessary experience.

Most rideshare launch opportunities put CubeSats in a Low Earth Orbit (LEO). A LEO orbit is defined as the region from a 160 to 2000 km altitude. Within that region, most CubeSats are launched within 400 to 600 km altitude giving the spacecraft a relatively short lifetime before reentry due to the increased drag effects of Earth's atmosphere [32]. Depending on the orbit and the shape of the CubeSat, typical missions in a LEO orbit can have a lifetime of 6 months to 6 years. Because of their small size, most CubeSats 3U or under cannot support a propulsion system to maintain orbit. Still, the low cost and short time between conception and launch can justify certain CubeSat missions with such a short lifetime [25].

1.3.1 CubeSat Evolution

The CubeSat Standard was originally designed as an inexpensive way to teach graduate students about satellite design and construction through a hands-on experience that would allow them to see the results of their work launch and provide on orbit data. The first CubeSats were launched in June 2003. The launch included 6 CubeSats, all from universities, with missions ranging from early earthquake prediction with Stanford's QuakeSat, to Earth imaging with the AAU-Sat [23].

As the benefits of CubeSats have become recognized, different organizations have become involved in the production of CubeSats as shown in Figure 1-5. They have expanded the primary use of CubeSats from mostly educational to operational and scientific. A survey done of nano- and picosatellites from 1997 to 2010 found that 52% were used for education by universities, 71% were technology demonstrations, and 52% had operational use, note that these categories are not exclusive [22].

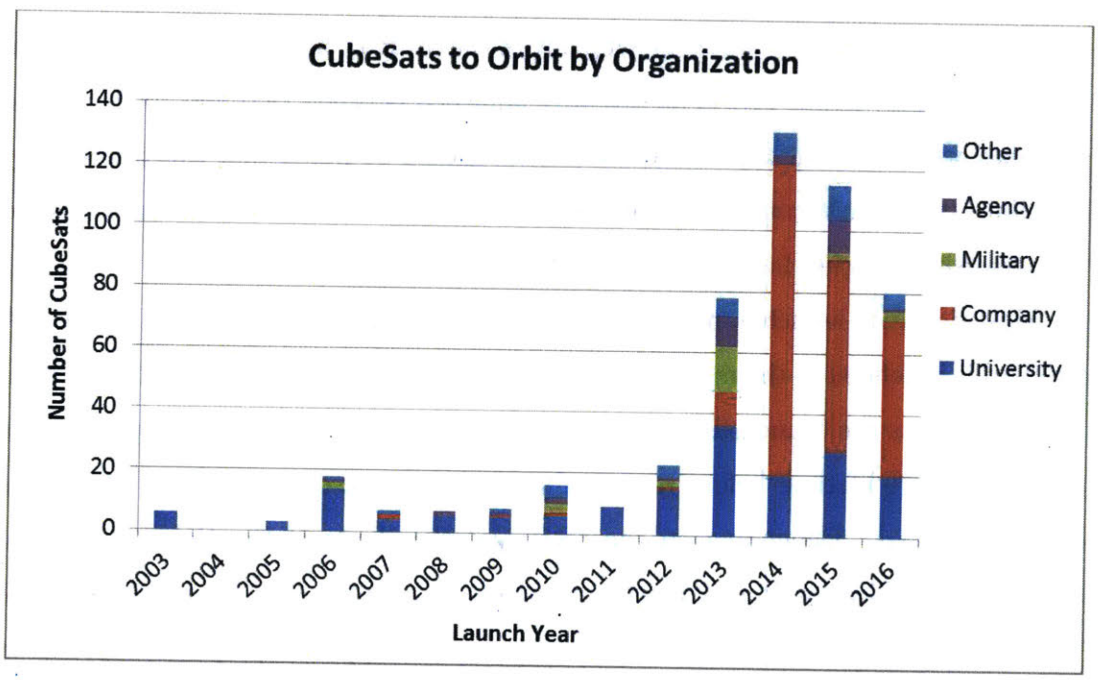

Figure 1-5: CubeSats launched over time by organization [25]

Data on numbers and trends of CubeSats come from Erik Kulu's NanoSat Database [25]. In Figure 1-5, the organizations sponsoring the CubeSat refer to:

1) University
2) Company: A private, for-profit company
3) Military: A military related satellite for research or operations
4) Agency: A government agency (i.e. NASA, NOAA)
5) Other: High Schools, Non-Profits, Institutes

It becomes clear looking at Figure 1-5 that there is a new trend in the past four years with organizations other than universities moving towards using CubeSats. Most notable in Figure 1-5 is the jump in CubeSats launched by companies in 2014-2016. This is largely due to the company Planet Labs, Inc which launched a total of 180 CubeSats through the end of 2016 for Earth imaging [25].

As more organizations become involved with CubeSats, and the possibilities for more advanced missions expands, CubeSats are becoming more complicated and growing in size as shown in Figure 1-6 [33], [25].

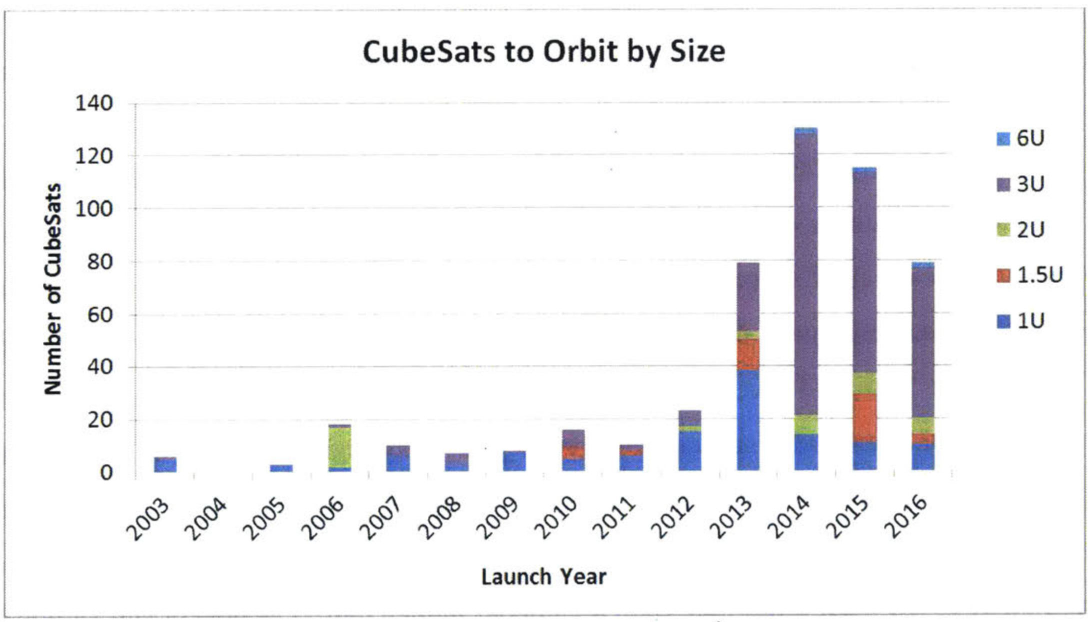

Figure 1-6: CubeSats launched over time by size [25]

Figure 1-6 shows the way CubeSat sizes have been growing over time with the 3U currently the most popular size. This is largely because it is the largest size that can fit in Cal-Poly's P-POD deployer. The P-POD is designed to deploy a combination of "3Us." Combinations can include one 3U, one 2U and one 1U, or three 1Us [23]. As developers have recognized the benefits of the small satellite standard, larger CubeSats are becoming more common. The first 6U CubeSats, Perseus M1 and M2, were launched in 2014 as a pathfinder mission for the Russian-American company Dauria Aerospace conducting maritime surveillance [34]. Larger still, the first 12U CubeSat was launched in April of 2017 by Northwestern Polytechnic University to demonstrate polarized light navigation and microgravity measurements. There are more plans for 12U satellites in development. The Earth Observing NanoSatellite (EON), by MIT Lincoln Laboratory is planned to be a 12U CubeSat that uses a microwave radiometer for weather sensing [35]. Also the 12U Iodine Satellite, set to launch in 2017, from NASA will demonstrate iodine Hall thruster technology [36].

With the shift towards these larger CubeSats, new deployers are under development to accommodate them. The NRCSD on the ISS is currently the only deployer to have deployed a 6U. A new deployer, the Cannisterized Satellite Dispenser (CSD) from Planetary Systems Corporation, is available in 3U, 6U, and 12U sizes [37]. To date, only the 3U CSD deployer has been successfully used in flight [37].

The popularity of CubeSats and the shift towards larger and commercial CubeSats has increased launch costs to accommodate the demand. This has created a push towards a standard for even smaller satellites. To return to the original purpose of CubeSats, making space affordable for students and perhaps even individuals, the PocketQube Standard has been proposed. Created in 2009 by Bob Twiggs, one of the CubeSat Standard creators, the PocketQube is an eighth the size of a CubeSat as a 5 cm^3 cube [38]. This allows as many as eight times as many satellites to be launched in a P-POD at a time, making launch costs of a PocketQube approximately a quarter of the cost, if not less, than a CubeSat launch [38]. The goal of this satellite is to allow more Universities, small groups, or even individuals, to create and launch their own affordable satellites. As of 2016, four PocketQubes have been launched, with plans for more to come [38].

1.3.2 CubeSat Constellations

CubeSats have been recognized for their usefulness in constellations and over the past few years. CubeSat constellations have been implemented for Earth Observation missions where they are used for imaging, weather monitoring, and disaster monitoring.

The company Planet Labs, Inc. has led the way for CubeSat Constellations with the creation of its Flock Constellations. The Planet Flock is made up of 3U CubeSats used for commercial imaging [39]. The constellation has been built by launching large numbers of CubeSats at each launch opportunity. After successful proof of concept missions, the first group of 28 Dove CubeSats were launched in 2014 [25]. That same year, over the course of three launches, another 66 Dove CubeSats were launched [25]. By of the end of 2016, Planet had launched 170 total CubeSats although it is not clear how many are still in operation [25]. In early 2017, Planet made history by launching 88 CubeSats in a single launch. By launching large numbers of satellites on each launch and having a variety of launches, the Flock constellation has daily coverage of Earth's landmass [6]. Spacing out the groups of CubeSats launched, even if just by a few months, allows for updates to be made to the next batch of satellites based on the on-orbit experience of the currently operational satellites.

Spire is another company using a constellation of 3U CubeSats for the tracking of ships and aircraft as well as providing GPS Radio Occultation data to NOAA (Nation Oceanic and Atmospheric Administration) for weather prediction [40]. Spire has launched 21 of its Lemur CubeSats in four different launches as of the end of 2016 [25]. While Spire's constellation does not use as many CubeSats as Planet and cannot boast the same revisit rate, it is still able to provide worldwide coverage [40].

There are also plans for smaller CubeSat constellations in the works. Massachusetts Institute of Technology's Lincoln Laboratory is working on deploying a 12 3U CubeSat constellation sponsored by NASA. This constellation, TROPICS (Time-Resolved Observations of Precipitation structure and storm Intensity with a Constellation of SmallSats), uses a twelve channel microwave radiometer for weather monitoring. This constellation will have four CubeSats in each of three evenly distributed orbital planes [41]. Each plane will have an inclination of 30°, an altitude of 600 km, with the right ascension of the ascending node (RAAN) evenly spaced between each plane [41]. Unlike the previous constellations mentioned, TROPICS currently plans to be launched as the primary payload on three different launches so that it can achieve this specific orbit configuration.

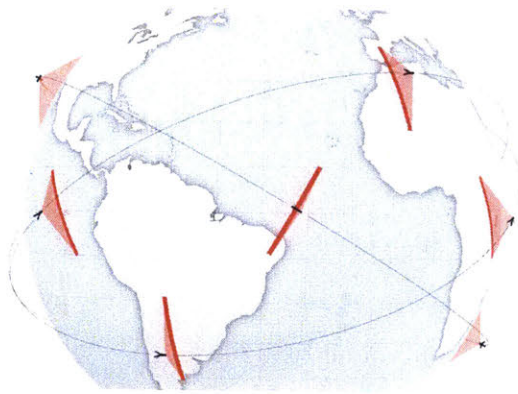

Figure 1-7: TROPICS constellation will have four CubeSats in each of 3 evenly spaced orbital planes [41]

1.4 Contributions

This thesis analyzes how CubeSat constellations can be used as a distributed system of sensors for a traditional monolithic satellite using the JPSS-1 satellite and its sensors as a reference case. The use, benefits, and feasibility of CubeSat constellations have been well documented [3], [39],[40], [41], [42], [43]. This thesis targets one specific use of CubeSat constellations as an augmentation to a large, complex, multi-sensored satellite as a way to reduce cost and increase resiliency, while maintaining the quality of data provided.

This question will be addressed by comparing the revisit time, cost, and quality of science data gathered by CubeSat constellations. These constellations are made up of existing or in development CubeSats that measure approximately the same atmospheric characteristic as a corresponding sensor of JPSS-1. The MicroMAS-2 CubeSat will be paired with JPSS-1's ATMS, the RAVAN CubeSat with CERES, CIRAS with CrIS, PICASSO with OMPS, and Dove with VIIRS.

Revisit time will be gathered through constellation simulations using Analytical Graphics Systems Tool Kit (STK). The quality of science data gathered will be determined by comparing each instrument on JPSS-1 with its paired CubeSat, looking specifically at nadir/spectral resolution, number/frequency available channels, and radiometric accuracy. The cost of the system will be determined as total program cost using heritage data from other CubeSat programs as estimates and will be compared to the total program cost for JPSS-1.

Chapter 2 discusses background information on and compares JPSS-1 and its instruments with the CubeSat instruments of comparable capabilities. Chapter 2 also explores CubeSat launch infrastructure and the resulting types of constellation architectures. Chapter 3 describes the simulation setup, assumptions, cost analysis approach, and the three different case studies that will be analyzed in this thesis. These three case studies represent three different CubeSat constellation architectures, a single mass launch, three available launches, and three planned launches. Chapter 4 presents the results from the simulation and cost analysis. Chapter 5 summarizes the findings from the study as well as discusses future work and recommended actions.

Chapter 2

Background

2.1 Chapter Overview

In this chapter, detail on JPSS-1 science functionality is described. This leads to a discussion of the approach toward fielding a constellation of CubeSats that host instrument payloads with similar functions to the JPSS-1 science instruments. In the distributed architecture, each JPSS-1 science instrument will be represented by a separate CubeSat, distributing JPSS-1 into at least five CubeSats as shown in Figure 2-1. Because the availability of launches, flexibility in selection of orbit planes, and spacing throughout orbit planes will affect the CubeSat constellation architectures, this chapter also includes some additional detail on the launch services currently planned or available for CubeSats.

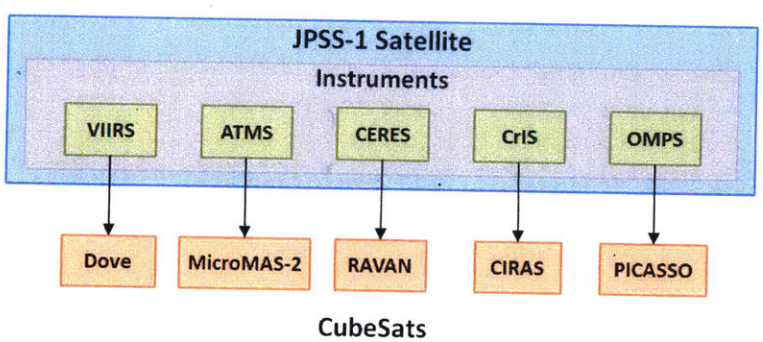

Figure 2-1: JPSS-1 will be distributed into five CubeSats. Acronym definitions can be found in Section 1.1.

2.2 JPSS-1

The Joint Polar Satellite System (JPSS) is a next generation, polar orbiting, NOAA weather monitoring mission built in collaboration with NASA. The JPSS system will be made up of five satellites, including the demonstration mission (S-NPP). Each has a designed operational lifetime of seven years with a new satellite to be launched approximately every five years. The current expected launch date of the first JPSS satellite, JPSS-1, is September 2017 [13]. This satellite was preceded by the demonstration satellite S-NPP, which became an operational mission to mitigate the delayed launch of JPSS-1. Figure 2-2 shows the progression of the NOAA polar weather satellite schedule.

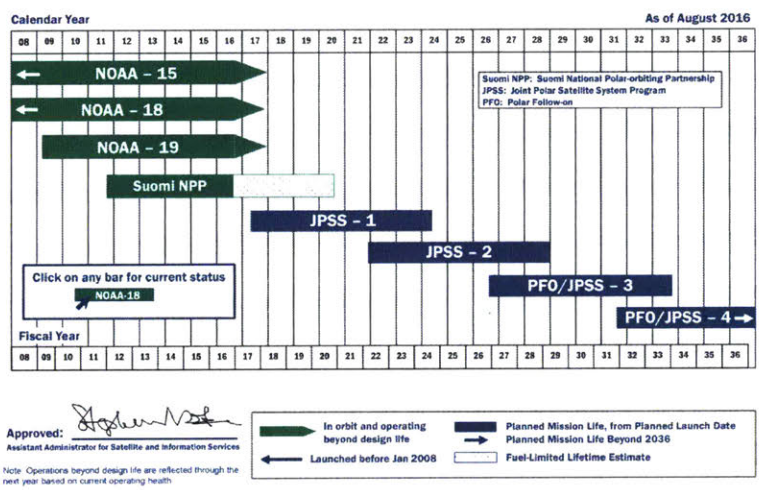

Figure 2-2: NOAA Schedule for Polar Satellite Programs [44]

The JPSS-1 orbit is planned to be an 824 km sun synchronous orbit with a local time of descending node (LDTN) of 1330, giving JPSS-1 the afternoon weather sensing orbit, similar to the orbit of S-NPP [44]. Morning weather data is provided by the Defense Meteorological Satellite Program (DMSP) satellites and a European Space Agency (ESA) satellite. DMSP-19 has a LTDN of 0530. Mid morning weather data is provided by DMSP-18 with a LTDN of 0800. MetOp, Meteorological Operational Satellite from the ESA, provides mid-morning data as well with a LTDN of 0930 [45], [13]. This is depicted in Figure 2-3 from the Government Accountability Office [13].

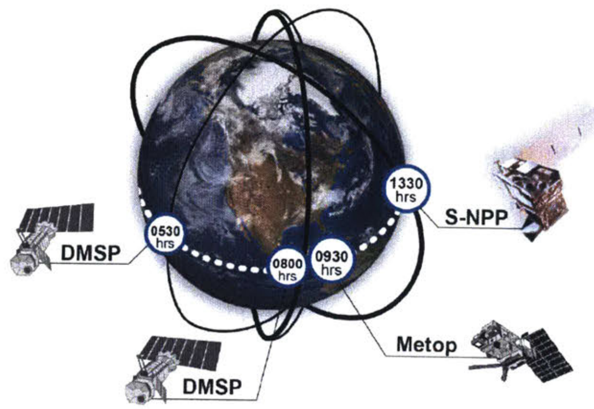

Figure 2-3: Polar satellites provide continuous weather observations [13]

JPSS-1 will communicate using Ka band and X band. It will also have S Band capability to communicate through NASA's Tracking and Data Relay Satellite System as backup. Ground stations for JPSS are located in Svalbard, Norway and McMurdo Station, Antartica with a backup station in Fairbanks, Alaska [45]. JPSS-1 will be primarily controlled by the NOAA Satellite Operations Facility in Suitland, Maryland which is also where data will be initially processed. The JPSS satellite bus was built by Ball Aerospace and Technology Corporation from their 2000 spacecraft series [45]. To complete its weather sensing and environmental monitoring mission, JPSS-1 hosts five instruments [44]. Previous versions of all of the instruments are currently flying on S-NPP. The instruments and the organizations who built them are:

1. Advanced Technology Microwave Sounder (ATMS) - Northrop Grumman Electronic Systems, Azusa, California

2. Clouds and the Earth's Radiant Energy System (CERES) - Northrop Grumman Aerospace Systems, Redondo Beach, California

3. Cross-track Infrared Sounder (CrIS) - Harris, Fort Wayne, Indiana

4. Ozone Mapping and Profiler Suite (OMPS) - Ball Aerospace & Technology Corporation, Boulder, Colorado

5. Visible Infrared Imaging Radiometer Suite (VIIRS) - Raytheon Company, El Segundo, California

Figure 2-4 shows the location of these instruments on an drawing of JPSS-1. Sections 2.3.1 through 2.3.5 provide detail on each instrument.

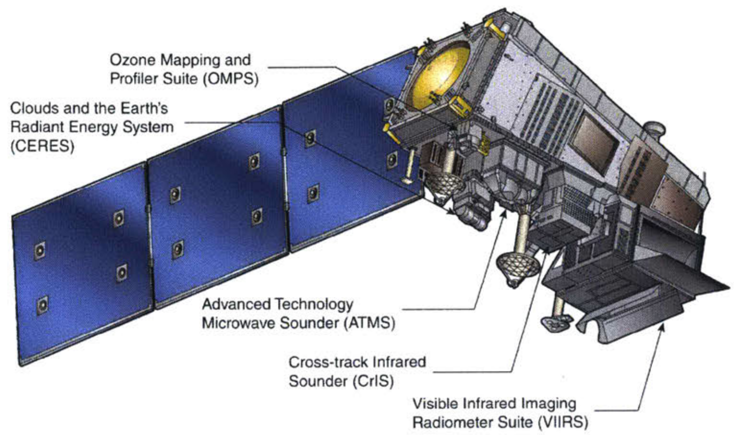

Figure 2-4: JPSS-1 Five Instruments [45]

2.3 JPSS-1 and CubeSat Sensor Comparison

This section discusses and compares each of the JPSS sensors and their corresponding CubeSats. For the most part, CubeSats cannot match the performance of the JPSS-1 sensors. One of the (several) reasons for this is because CubeSats do not have the resources to support high powered sensors. For instance, the JPSS-1 sensors require power in the hundreds of Watts. 3U CubeSats with double-deployed, double-sided solar panels can only support limited operations of a payload that requires between 10 and 20 Watts. A 3U CubeSat ideally would require only about 7 W orbit average power. This comes from the lack of area for solar cells and resulting minimal power generation ability.

Because of their size, CubeSats are also limited by the maximum aperture diameter and available volume for optical relays. To maximize the size of the aperture of a CubeSat, it must be placed on a face where there are no rails, where the CubeSat could have a maximum aperture diameter of about 9 cm. In comparison, the aperture diameters on the JPSS-1 sensors can be as large as 20 cm. Because of limitations in aperture size the spatial resolution of these imagers cannot match the heritage sensor performance. Also, spectrometers have not yet been miniaturized enough to meet CubeSat size and power limitations while maintaining the same spectral resolution as heritage sensors. Current CubeSats cannot generally meet the same spectral resolution as the larger sensors.

2.3.1 ATMS and MicroMAS-2

Advanced Technology Microwave Sounder (ATMS)

The Advanced Technology Microwave Sounder is a cross-track microwave sounder that measures temperature and moisture levels in the atmosphere for use in weather and climate monitoring. Designed and produced by Northrop Grumman Electronic Systems, Azusa, California, ATMS combines the capabilities of earlier generations of microwave sounders that fly on NOAA's POES satellites to give more channels, better resolution, and wider swath [44]. Microwave sounders generally play the largest role in weather prediction accuracy. Gases (as well as solids and liquids) in the atmosphere emit and radiate microwaves. The amount and frequency of microwaves measured by microwave radiometers can give details about atmospheric characteristics. Because clouds do not block microwaves leaving the atmosphere, microwave radiometers can measure atmospheric characteristics in all weather, giving weather information from under the clouds.

The improved accuracy of ATMS in short and medium term forecasting gained with these upgrades make ATMS crucial to weather forecasting. Not only does ATMS play a role in weather forecasting, but it is used in farming, aircraft flight planning, ship routing, and extreme weather preparedness [44], [45], [46].

The ATMS channels listed in Table 2.1 are separated into two sets, the 15 lower frequency channels below 60 GHz, and the 7 higher frequency channels above 60 GHz. There are two antennas, one for each set of channels. Two large apertures are used for lower frequency microwave collection. A single smaller aperture is used for the higher frequency collection.

Table 2.1: Key ATMS Specifications [44]

Wavelength Coverage	22 Channels: 23 GHz - 183 GHz
Nadir Resolution	15.8 km - 74.8 km
Swath	2600 km
Average Data Rate	32,000 bps
Average Power	130 W
Mass	85 kg

Of the 22 ATMS channels, three at 31, 89, and 166 GHz are used for measuring precipitable water, liquid water, and rain rates. Thirteen channels ranging from 50 - 57 GHz are used for measuring temperatures from the Earth's surface to an altitude of 45 km (upper stratosphere) with a vertical resolution of 3 to 6 km and an accuracy of 0.75 K. The remaining six channels at 22 and 183 GHz look at atmospheric water vapor from Earth's surface to 10 km (troposphere), again with the same vertical resolution of 3 to 6 km and a temperature accuracy of 0.6 K to 0.9 K [45], [47], [46].

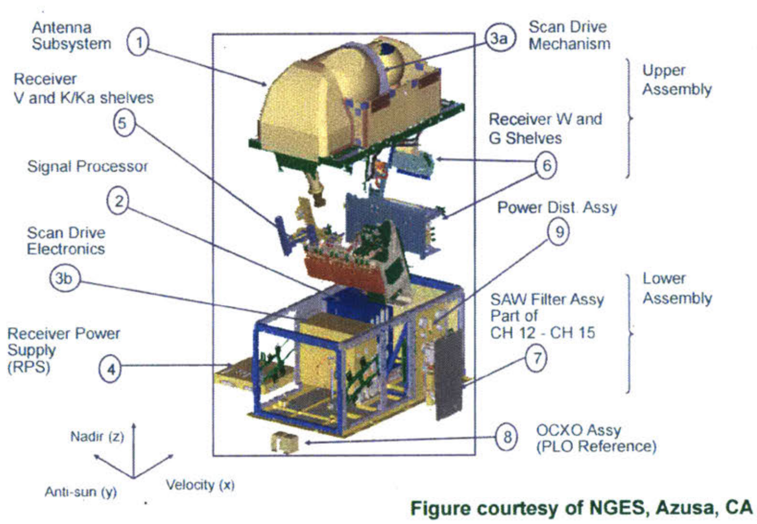

Figure 2-5: A detailed, exploded view of the ATMS sensor [48].

The ATMS instrument completes three 360° scans every 8 seconds, at which point it synchronizes its clock with the JPSS computer. Each of the three scans consists of three segments: an Earth scan, a cold calibration, and a hot calibration. During the

Earth scan portion, 96 samples are taken at 96 different angles evenly spaced ±52.725° symmetric from the nadir direction as the antenna scans the Earth at constant speed. Next, the antenna points towards space to take four cold calibration measurements. Finally, the hot calibration is done by pointing the antenna at an internal hot target During the calibration scans, the scanner accelerates to its calibration positions [46], [45]. The on board calibration system for ATMS includes two blackbody warm targets whose temperatures are monitored and using space as the cold target. The data is calibrated on orbit before being sent to the ground for processing.

MiRaTA and MicroMAS

The Microwave Radiometer Technology Acceleration CubeSat, MiRaTA, and the Micro-sized Microwave Atmospheric Satellite, MicroMAS, are two 3U CubeSats developed by MIT Lincoln Laboratory that use microwave radiometers for weather sensing. Both missions are technology demonstration missions meant to increase the TRL of CubeSat sized microwave radiometers.

Figure 2-6: MiRaTA Space Vehicle

MiRaTA is due to be launched in the fourth quarter of FY17 on the same launch as JPSS-1, putting it into an approximately 824 km polar sun synchronous orbit. Its passive microwave radiometer provides 10 channels near 60, 183, and 206 GHz. The radiometer is split into two bands, a V-band spectrometer which measures frequencies between 52-58 GHz, and a G-band broadband mixer which measures 175-191 GHz and 206 GHz [49]. These channels measurement of temperature, humidity, and cloud ice by the 60 GHz, the 183 GHz, and the 206 GHz bands respectively [49]. Calibration of the radiometer will be augmented by using the temperature profiles from the on-board GPS radio occultation (GPSRO) experiment, using the Compact TEC (Total Electron Count)/Atmosphere GPS Sensor (CTAGS) instrument from The Aerospace Corporation, El Segundo, California.

MicroMAS-1 was launched to the ISS in July 2014 and deployed from the ISS in March 2015. Unfortunately there were only three MicroMAS-1 overpasses where data was received before a communications failure occurred. It has since been determined that the most likely cause of this failure was from an unsuccessful deployment of a solar panel and the tape-spring radio antenna, where the two stayed in contact rather than separating during the deployment process [50]. This caused the solar panel to affect the load match of the antenna, reflecting power back through the power amplifier of the radio, eventually damaging it. A follow-up mission, MicroMAS-2 will be launched in summer 2017.

Figure 2-7: MicroMAS-2 CAD illustration. The upper 1U of the CubeSat contains the rotating microwave radiometer.

Both the MicroMAS-1 and MicroMAS-2 CubeSats support a cross-track-scanning passive radiometer [51]. MicroMAS-2, the more advanced radiometer, features four bands with 10 channels. The four bands are located at 89 GHz, 118.75 GHz, 183, and 207 GHz. The 89 and 207 GHz water vapor continuum bands measure water vapor burden, cloud liquid water, cloud ice, snow cover, precipitation, rain rate, and sea ice. The 118.75 oxygen absorption line gives data on temperature, pressure, and precipitation in the atmosphere. The 183.31 GHz water vapor absorption line measures water vapor (humidity) and precipitation.

At these frequencies MicroMAS-2 can make moisture measurements, temperature profiles, and precipitation imaging. These measurements can be used to observe convective thunderstorms, tropical cyclones, and hurricanes. The upper 1U of MicroMAS hosts the payload and spins separately from the rest of the CubeSat at a rate of approximately one rotation per second. It has a beam width of 3° and a swath of 2500 km. Because of MicroMas-2's channel selection and wide swath, it makes the best CubeSat match to JPSS-1's ATMS. It is due to launch in September 2017 on a Polar Satellite Launch Vehicle (PSLV).

Comparison

JPSS-1's ATMS and the MicroMAS-2 CubeSat are the first two sensors that will be compared in this section. The key specifications of each sensor are shown in Table 2.2. Both ATMS and MicroMAS-2 are microwave radiometers that measure temperature, precipitation, water vapor, cloud liquid water, and pressure.

Table 2.2: Key specs of JPSS-1's ATMS sensor and the MicroMAS-2 CubeSat [45],[51]

	ATMS **JPSS-1**	**MicroMAS-2** **3U CubeSat**
Scan Range	Cross Track: 2.2° – 6.3° Along Track: 1.1° – 5.2° Swath: 2600 km	FOV: 5° Scan Angle: 115° Swath: 2590 km
Nadir Resolution	15.8 - 74.8km	20 km
Total Channels	22	10
Spectral Bands	23.8 GHz, 31.4 GHz, 50-55 GHz (7 channels), 57.26 GHz (6 channels), 88 GHz, 165 GHz, 183 GHz (5 channels)	89 GHz, 118 GHz (5 channels), 183 GHz (3 channels), 206 GHz
NEdT @300 K	0.5-3.0 K	0.1 - 0.6 K
Mass	85 kg	3.8 kg
Power	130 W	9.1 W
Max Data Rate	32 kbps	16 kbps

ATMS and MicroMAS-2 share some of the same spectral bands, measuring the same atmospheric properties. They share the 183 GHz and the 89 GHz band, which measure humidity/precipitation and cloud liquid water respectively. ATMS has more channels, measuring similar properties in addition to those measured by MicroMAS-2. Both sensors have similar nadir resolutions although there is much more variation in ATMS resolution depending on the band; its resolution improves as the bands increase in frequency. The radiometric accuracy for both is also quite similar, 0.7-3.0 K for ATMS and 0.1 - 0.6 K for MicroMAS-2, showing that the data obtained is of the same quality from both sensors. Taking into account the orbit altitude, MicroMAS has a similar swath and field of view to ATMS.

MicroMAS gives the same quality of data as ATMS, for lower SWaP. Because of its size, it sacrifices the number of bands and channels it is capable of measuring. However the bands that it does measure give data of the same quality of ATMS. To completely capture the capabilities of ATMS, two types of CubeSats would be needed to cover all the bands with the second CubeSat covering the bands that are not covered by MicroMAS. MiRaTA covers one of these spectral bands, leaving only two center frequencies that would need to be developed on a new CubeSat radiometer.

2.3.2 CERES and RAVAN

Clouds and the Earth's Radiant Energy System (CERES)

The Clouds and the Earth's Radiant Energy System is a scanning broadband radiometer that measures reflected sunlight and thermal radiation emitted by the Earth. Built by Northrop Grumman Aerospace Systems, Redondo Beach, California, this will be the sixth version of CERES. CERES primarily allows scientists to look at the effect the atmosphere and cloud cover have on absorbed and reflected radiative energy. This aids in developing an understanding on how cloud cover affects the heating and cooling of the planet. CERES, with data from VIIRS, can also measure cloud properties such as quantity, height, thickness, particle size, and phase. The ability to measure radiation also allows CERES to be useful in studying the effects environmental disasters like drought, flood, or volcanic eruptions can have on clouds and the climate [44], [45], [52].

Table 2.3: Key CERES Specifications [44]

Wavelength Coverage	3 Channels: 0.3-5 μm, 8-12 μm, 0.3->50 μm
Nadir Resolution	20 km
Average Data Rate	10,520 bps
Accuracy	0.3 to 1%
Average Power	55 W
Mass	54 kg

CERES maps the radiation budget at the top of the atmosphere, within the atmosphere, and at the Earth's surface using three different channels listed in Table 2.3. The 8-12 μm channel focuses on radiation from the Earth's surface, although cloud cover does affect this measurement. The 0.3-5 μm channel measures shortwave radiation and the 0.3-50 μm channel measures broadband radiation [53]. The three apertures for these channels share 98% of their field of view. A scanning sequence is completed every 6.6 seconds which includes two limb to limb scans of the Earth. Calibration is completed using internal calibration sources: black bodies and a Shortwave Internal Calibration Source [45], [53].

RAVAN

The Radiometer Assessment using Vertically Aligned Nanotubes (RAVAN) CubeSat maps most closely to JPSS-1's CERES instrument as it also measures solar reflected radiation and Earth emissions. Developed by the Johns Hopkins University Applied Physics Laboratory, Laurel, Maryland, RAVAN was launched on November 11, 2016 into a circular, sun-synchronous 600 km orbit [54], [55]. Currently, RAVAN is still going through commissioning and is conducting on orbit solar calibrations. Preliminary data is promising, showing that the two channels are tracking each other well. Future applications of RAVAN include a constellation of these CubeSats that can measure the Earth's Radiation Budget in high fidelity and can show not only long term changes like CERES, but also short term changes which can help improve climate models.

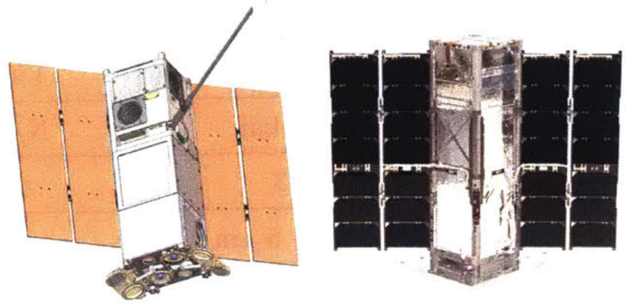

Figure 2-8: RAVAN CAD illustration (left) and Flight Model (right) [54], [55]

RAVAN consists of four individual radiometers that are set in two pairs. Each pair has two channels, one that measures all radiation from UV to far Infrared (200 nm - 200 μm) and the second that measures shortwave radiation (less than 5.5 μm) [54]. It is the shortwave channel that allows for a distinction to be made between solar reflected sunlight and the Earth's total emission. The primary pair of radiometers use vertically aligned carbon nanotubes absorbers. The secondary pair use a conical cavity design to compare results against the carbon nanotubes. The field of view of the radiometers is 130° so that they can view entirely from limb to limb of the Earth. Calibration is completed on orbit using a gallium blackbody with the sun as the primary calibration standard [54].

Comparison

JPSS-1's CERES and the RAVAN CubeSat both measure Earth's radiation. CERES measures Earth's radiation with a scanning radiometer that has three main channels, a shortwave, long wave, and total radiation channel. RAVAN measures Earth's radiation with four radiometers in two pairs. RAVAN has two spectral bands, a total radiation band that spans further than the total band for CERES and a shortwave band. Their key specifications are described in Table 2.4.

Table 2.4: **Key specs of JPSS-1's CERES sensor and the RAVAN CubeSat [45],[54]**

	CERES JPSS-1	**RAVAN 3U CubeSat**
FOV	Limb to Limb Scan	130°
Nadir Resolution	20 km	N/A
Total Channels	3	2
Spectral Bands	0.3 - 5 μm 8 - 12 μm 0.3 - 50 μm	200nm - 200 μm <5.5 μm
Accuracy	0.38 - 1.6 Wm^{-2}	0.3 Wm^{-2}
Mass	54 kg	<1 kg, PL only
Power	55 W	1.9 W, PL only
Max Data Rate	10.5 kbps	16 kbps

CERES and RAVAN differ in their mission goals in that that CERES is looking specifically at how clouds affect Earth's radiation budget and RAVAN is looking at the overall Earth's Radiation Budget. CERES gives information about cloud properties and the role they play in global climate change. To do this, CERES does limb to limb scans with a FOV of 0.7° and a nadir footprint of 20 km. RAVAN, focusing on the total Earth Radiation Budget, fits the entire Earth view from limb to limb in its field of view. This allows for measurements on the total radiation emitted by Earth to determine if Earth is emitting more or less radiation than it absorbs from the sun. RAVAN is not concerned with ground resolution because its purpose is to measure all radiation, not necessarily to tell where radiation is stronger or weaker.

RAVAN and CERES have similar accuracy values. While RAVAN and CERES do not have the same mission, they are similar enough in type and quality of data. Unlike all the other sensor comparisons, here the CubeSat has the higher maximum data rate, even it only by 5.5 kbps more. Future technology development of an Earth radiation sensing CubeSat could be applied to create a payload more similar in mission to CERES. As RAVAN exists now, not only is it different in mission but it does not contain the same specific bands as CERES. The specific separation of these bands allows for analysis of different types of radiation collected. The separate bands on CERES allow it to distinguish between radiation from the Earth's surface, shortwave radiation and broadband radiation. RAVAN can only identify broadband radiation. To completely simulate CERES's capabilities, a second CubeSat would have to be flown that covers the short and longwave IR bands specifically.

2.3.3 CrIS and CIRAS

Cross-track Infrared Sounder (CrIS)

The Cross-track Infrared Sounder (CrIS) is a Fourier transform spectrometer built by Harris, Fort Wayne, Indiana, that provides data about temperature and water vapor, supplementing ATMS data. CrIS can also provide data on the concentration of carbon dioxide and other greenhouse gases in the atmosphere, which can relate to global warming. The specifications for CrIS are listed in Table2.5.

Table 2.5: Key CrIS Specifications [44], [45]

Wavelength Coverage	1305 Spectral Channels: 3.92 μm to 4.64 μm (SWIR) 5.71 μm to 8.26 μm (MWIR) 9.14 μm to 15.38 μm (LWIR)
Nadir Resolution	14 km diameter
Scanned Width	2200 km
Average Data Rate	1.9 Mbps
Accuracy	0.3 to 1%
Average Power	245 W
Mass	175 kg

Clouds become opaque in the infrared spectrum but not in the microwave range, making CrIS sensitive to cloud cover, but not ATMS. ATMS however does not have the vertical resolution that CrIS does. Data from ATMS and CrIS are used to generate 3D high-resolution profiles of temperature, pressure, and moisture of the atmosphere [44], [56]. These profiles lead to improved weather forecasting five to seven days in advance.

CrIS has 1305 spectral channels over three wavelength ranges, long-wave infrared, mid-wave infrared, and shortwave infrared [57]. It uses interferometry to produce high resolution data with vertical resolution varying throughout the atmosphere, from 1-2 km in the troposphere to 3-5 km vertical resolution in the stratosphere [45]. The CrIS aperture measures 8 cm in diameter and takes 30 samples over its 2200 km swath scan of the Earth. Each 8 second scan includes calibration through an internal hot body and space views to act as a cold body [45], [56], [57].

CIRAS

CIRAS is the CubeSat Infrared Radiometer Sounder, a 6U CubeSat which measures midwave infrared radiation in the Earth's atmosphere developed by JPL, Pasadena California. It maps most closely to JPSS-1's CrIS which also measures infrared radiation. According to *Pagano et al.*, CIRAS uses the information from the infrared radiation measurements to develop temperature and water vapor profiles, with particular sensitivity in the lower troposphere. Developed by the Jet Propulsion Laboratory (JPL), with the spectrometer from Ball Aerospace, CIRAS is set to be launched in late 2018 [58]. It is being developed as a path finder mission for a larger nanosatellite, the Earth Observing Nanosatellite in Infrared (EON-IR). EON-IR is meant to fill the potential gap in infrared data between the end of the current IR measurement missions and the future ones to be made on JPSS.

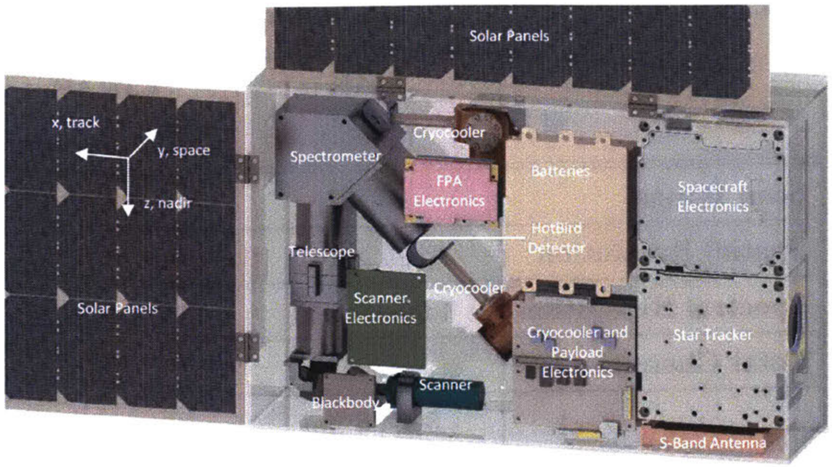

Figure 2-9: Preliminary layout of CIRAS CubeSat [58]

CIRAS hosts two main technologies for demonstration, a 2D array of High Operating Temperature Barrier Infrared Detector photosensitive material as a detector for IR imaging and a midwave infrared grating spectrometer. CIRAS's spectrometer has 625 channels around the 4.08 - 5.13 μm band. CIRAS nominally has a 13.5 km spatial resolution with a 1112 km swath, but can achieve a "zoom mode" that has a resolution of 3 km by adjusting the scan rate and the number of pixels. This, however, increases the data rate such that the swath must be lowered to 160 km in this mode. For internal calibration, CIRAS uses a silicon blackbody developed by JPL.

Comparison

Both CrIS and CIRAS measure midwave infrared radiation. CrIS uses a Fourier transform spectrometer for these measurements while CIRAS uses a grating spectrometer. The infrared radiation data gives information on temperature and water vapor profiles. Table 2.6 shows a comparison of key specifications of CrIS and CIRAS. The two values listed for the scan range and spatial resolution of the CIRAS CubeSat are due to the zoom mode which can provide high spatial resolution, but at the cost of a lower swath width.

Table 2.6: Key specs of JPSS-1's CrIS sensor and CIRAS CubeSat [58]

	CrIS JPSS-1	CIRAS 6U CubeSat
Scan Range	$\pm 43.8°$	$\pm 6.2°, \pm 41.6°$
Spatial Resolution	14 km	3 km, 13.5 km
Spectral Resolution	1.0-5.0 cm⁻1	1.3-2.0 cm⁻1
Total Channels	1305	625
Spectral Bands	3.9-4.6 μm 5.7-8.2 μm 9.1-12.0 μm 12.0-15.4 μm	4.08-5.13 μm
NEdT @250 K	0.1-1.0 K	0.2-0.6 K
Size	$0.9 \times 0.9 \times 0.7$ m^3	$0.1 \times 0.2 \times 0.3$ m^3
Mass	165 kg	14 kg
Power	117 W	40 W
Max Data Rate	1.5 Mbps	0.32 Mbps

CrIS and CIRAS have similar scan ranges, spatial resolutions, and radiometric sensitivity. They view about the same amount of Earth's atmosphere in every scan and the pictures resolved from the scan have pixels that represent approximately the same sized patch of atmosphere. Additionally, CrIS has a slightly larger scan range but CIRAS has slightly better spatial resolution. The radiometric sensitivity of the two falls on approximately the same scale as well with only 0.1-0.4 K difference.

The most significant difference in the performance of CrIS and CIRAS comes in the number of spectral bands and spectral resolution. CIRAS does not have the spectral resolution that CrIS has, meaning CIRAS has fewer spectral bands that CrIS. This is because CIRAS operates entirely in the midwave infrared region while CrIS operates in both the midwave and long wave region. The long wave region gives information on atmospheric temperature.

CIRAS will be using only midwave infrared for temperature sounding. This is not generally done because of concerns that midwave data can be contaminated by solar reflected energy. However, a heritage infrared sounder, the Atmospheric Infrared Sounder (AIRS) from NASA JPL, has successfully done temperature sounding using only the midwave wavelengths. With this heritage algorithm, CIRAS is expected to be successful at temperature sounding [58]. CIRAS also does not make measurements at the 5 μm absorption band for water vapor as CrIS does, CIRAS uses longer wavelengths for water vapor sounding.

CIRAS can supply some of the data CrIS can at the same quality. However, the smaller CIRAS comes at the cost of about 700 channels that can provide relevant data. CIRAS is not an exact replacement for CrIS but it does provide equivalent science data and could be a is a candidate to supplement or augment CrIS data. To truly cover the same capabilities as CrIS, at least two types of CubeSats would be needed to cover all the CrIS bands.

2.3.4 OMPS and PICASSO

Ozone Mapping and Profiler Suite (OMPS)

The Ozone Mapping and Profiler Suite (OMPS) on JPSS-1, built by Ball Aerospace & Technology Corporation, Boulder, Colorado, is a system of two spectrometers that make measurements on the amount of ozone in the atmosphere. It contains a nadir pointing mapper and a nadir profiler. While OMPS is not for weather monitoring like the other JPSS instruments, measurements of the ozone layer help predict the amount of harmful UV rays penetrating the atmosphere, supplementing data for the UV index. OMPS also has the ability to measure particles that come from volcanic eruptions like sulfur dioxide and ash [44]. OMPS makes its ozone measurements by measuring the backscatter of UV rays it collects. As a function of wavelength and backscatter intensity, the altitude and concentration of the ozone in the atmosphere can be retrieved. The wavelengths of radiation OMPS collects are not affected by clouds, allowing for measurements to be taken in all weather [45], [59].

Table 2.7: Key OMPS Specifications [44]

Wavelength Coverage	**Mapper**: 0.3-0.38 nm **Profiler**: 0.25-0.31 nm
Nadir Resolution	**Mapper**: 50 km **Profiler**: 250 km
Average Data Rate	109.6 kbps
Swath	**Mapper**: 2800 km
Average Power	120 W
Mass	56 kg

OMPS has two total bands, one for its Profiler, and one for the Mapper. Each of these has a different focal plane, one for the mapper to make total column ozone observations and the second for the profiler to make profile ozone observations. The system consists of a push broom telescope that has two spectrometers using charge coupled device (CCD) detectors and a reflective calibration diffuser to measure the UV backscatter from solar radiation. [59], [45].

PICASSO

The PICASSO CubeSat, Pico-Satellite for Atmospheric and Space Science Observations, is a 3U CubeSat developed by the Belgian Institute for Space Aeronomy in conjunction with the European Space Agency. PICASSO observes Earth's atmospheric limb and studies the ozone levels in the atmosphere, monitoring the state of the ozone layer and mapping most closely to OMPS.

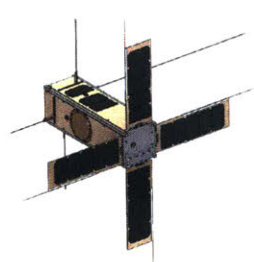

Figure 2-10: CAD Model of PICASSO CubeSat [60]

PICASSO uses a Sweeping Langmuir Probe and a miniaturized hyperspectral imager. The Langmuir Probe is used for measuring ion saturation, retardation, and electron saturation regions throughout the space it is flying through. To make ozone measurements, PICASSO uses its hyperspectral imager which operates between 430-800 nm, the visible to near-infrared spectrum [60]. During solar occultation, PICASSO observes the Earth's atmospheric limb using the Sun as a light source as shown in Figure 2-11.

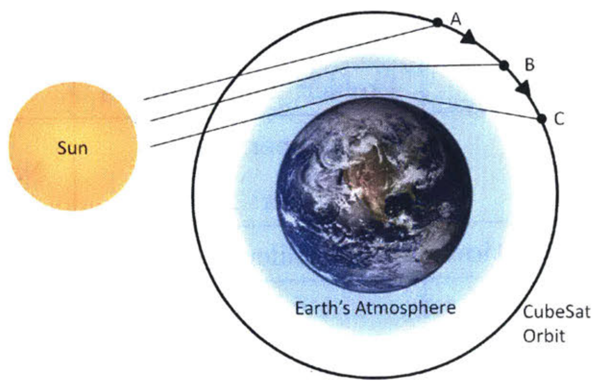

Figure 2-11: Graphical Depiction of Solar Occultation

Figure 2-11 shows the way PICASSO would collect the suns rays during a sunset event. When the sun's rays travel through the atmosphere at different heights, they are not only bent slightly by the atmosphere, but are partially absorbed as well. The wavelengths of the absorbed light, particularly within the Chappuis absorption band between 375 and 650 nm in the visible range, give information about the amount of ozone in that part of the atmosphere. This results in vertical profiles of ozone concentration throughout the upper atmosphere [60], [61].

Comparison

JPSS-1's OMPS and PICASSO are both ozone measuring sensors. The version of OMPS on JPSS-1 contains two spectrometers, a nadir column spectrometer and a profile spectrometer. PICASSO uses a hyper-spectral imager that takes 2D snapshots. OMPS nadir spectrometers measure backscattered solar UV radiation dispersed through the atmosphere. PICASSO uses limb-scattered solar radiation for ozone level determination which it measures during solar occultation events. Table 2.8 gives details of the specifications of these two sensors. Due to the early stage in PICASSO's design process, some information is not available.

Table 2.8: Key specs of JPSS-1's OMPS sensor and the PICASSO CubeSat [45],[60]

	OMPS JPSS-1	**PICASSO 3U CubeSat**
Swath/FOV	Mapper: 2800 km Swath Profiler: 250 km Swath	2.5° x 2.5° FOV
Resolution	Mapper: 50 km Profiler: 250 km	Not Available
Vertical Resolution	Profiler: 8 km	2 km
Total Channels	2	1
Spectral Resolution	Mapper: 1.0 nm Profiler: 1.0 nm	<10 nm
Spectral Bands	Mapper: 0.3-0.38 μm Profiler: 0.25 - 0.31 μm	0.43-0.8 μm
Mass	56 kg	0.5 kg, PL only
Power	120 W	Not Available
Max Data Rate	409.6 kbps	Not Available

These two sensors are different in how they measure ozone in the atmosphere. The OMPS sensor on S-NPP contains a third spectrometer that measures limb-scattered ozone similar to the method which PICASSO uses. However, on JPSS-1, OMPS only uses its nadir mapper and profiler to measure atmospheric ozone. The biggest difference between the OMPS and PICASSO sensors is the field of view. The PICASSO field of view is too small to achieve global coverage within a reasonable time; it will have maximum revisit times on the order of two to three weeks. The OMPS sensor is much more suited to measuring the ozone layer on a global basis because of its large field of view and swath.

The vertical resolution of PICASSO and OMPS are on the same order of magnitude. PICASSO's resolution is slightly better, but OMPS's profiler has notable resolution compared to its swath width. PICASSO has a significantly worse spectral resolution than OMPS, 10 nm compared to 1.0 nm. The two also use different spectral bands to make their measurements. Overall, while PICASSO may return information similar to that provided by OMPS, it does not view nearly enough of the Earth's atmosphere well enough to make the types of measurements and return the quality of data that OMPS does.

To fully represent the capabilities of OMPS, CubeSats would have to be developed with spectrometers along the same bands that OMPS uses. There may be CubeSats that are currently host payloads close to these bands, but they are not involved in ozone monitoring. Overall, it appears that there has not been much technology development in the field of ozone monitoring on CubeSats. To be able to capture the capabilities of OMPS on CubeSats, much more development in this area is needed.

2.3.5 VIIRS and Dove

Visible Infrared Imaging Radiometer Suite (VIIRS)

The Visible Infrared Imaging Radiometer Suite (VIIRS), much like its name suggests, takes images in the visible and infrared spectrum. The global images of land, atmosphere, and ocean provided by VIIRS give data on environmental factors like snow cover, ice cover, fog, fire, aerosols, even vegetation health. Built by Raytheon Company, El Segundo, California, VIIRS is a combination and improvement upon the three instruments: The Advanced Very High Resolution Radiometer, the Moderate Resolution Imaging Spectroradiometer, and the Operational Linescan System. VIIRS's higher resolution and larger swath than its heritage sensors improves weather forecasting, maritime and agriculture monitoring, and ocean monitoring (for water quality and temperature) [44], [45].

Table 2.9: **Key VIIRS Specifications** [44]

Wavelength Coverage	22 Spectral Bands: 412 nm to 12 μm
Nadir Resolution	400 m
Average Data Rate	7,674,000 bps
Swath	3000 km (max)
Average Power	319 W
Mass	280 kg

VIIRS's scanning radiometer has 22 channels with frequencies that span the visible and IR spectrum to give three distinct channels: (i) the Visible/Near IR, Day/Night, (ii) midwave infrared, and (iii) longwave infrared. Calibration sources are located internal to the VIIRS instrument and include a solar reflective and IR emissive body. VIIRS's nadir resolution is about 400 m, increasing to about 750 m at the end of its FOV. [45],[62].

Dove

Mapping a single CubeSat to VIIRS is more difficult than with the other instruments on JPSS-1. While there are a number of Earth imaging CubeSats like Aerospace Corporation's AeroCube-4 and -5 and Planet's Doves, there are none that carry imaging multi-spectral radiometers like VIIRS [63], [64]. Current Earth imaging CubeSats generally use cameras that typically operate in the visible range; few can image in the near infrared.

A good comparison of CubeSat imagers with VIIRS comes from *Pack et al.* of The Aerospace Corporation. They do a direct comparison of a nighttime image by their AeroCube-4 camera with one from VIIRS, showing that, while AeroCube-4 picks up lights, it has a much more difficult time distinguishing between water and sand in the image of the Persian Gulf in Figure 2-12 [64]. It does not have the spectral or spatial resolution that VIIRS does.

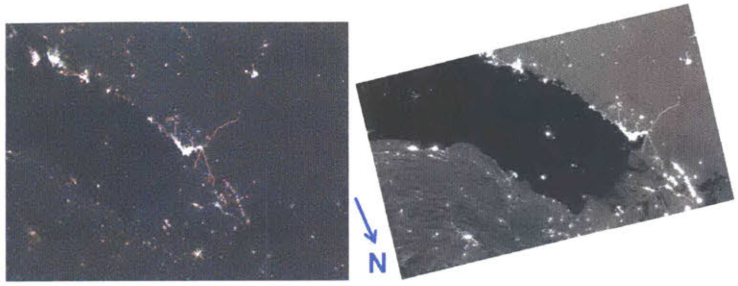

Figure 2-12: The Persian Gulf at night. Left: Image from AeroCube-4, Right: Image from VIIRS [64]

Planet's Dove CubeSats have better spatial resolution than the AeroCube-4 imager. The Planet imagers do have multiple filters in the visible band and can be used as a proxy for the visible band of VIIRS. The Dove CubeSat has four bands: red, blue, green, and near-IR. The Planet instrument shown in Figure 2-13, shows the Dove imager payload. This payload includes a telescope and a frame CCD camera with Bayer-mask filter.

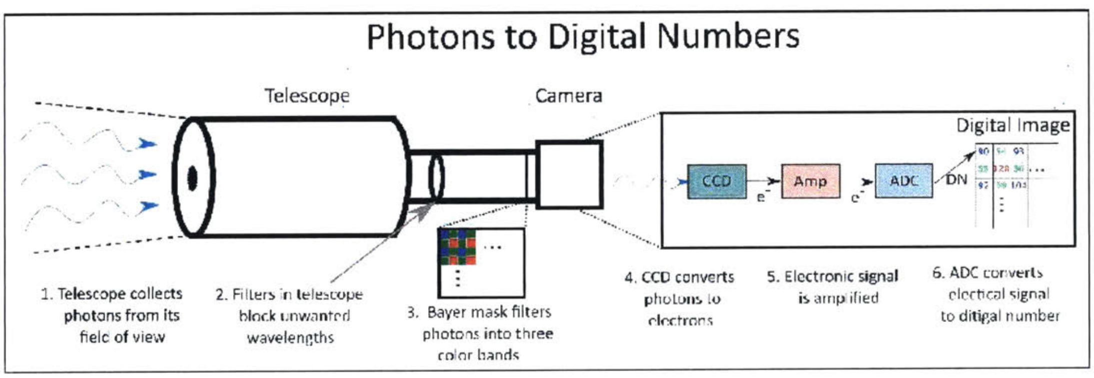

Figure 2-13: Drawing of Planet Imaging payload [65]

The four bands from the imager are combined to get a final picture as shown in Figure 2-14. The tricolor image has been color corrected to look similar to what the human eye would see. All images are corrected to remove terrain distortions and to account for differences in latitude and time of acquisition. The ground sample distance, the distance between each pixel in an image, is 3.7 km (at an altitude of 475 km). At that altitude, each image has a nominal size of 24 x 7 km [63]. Calibration of the imager is done using deep space stares and on board calibration methods.

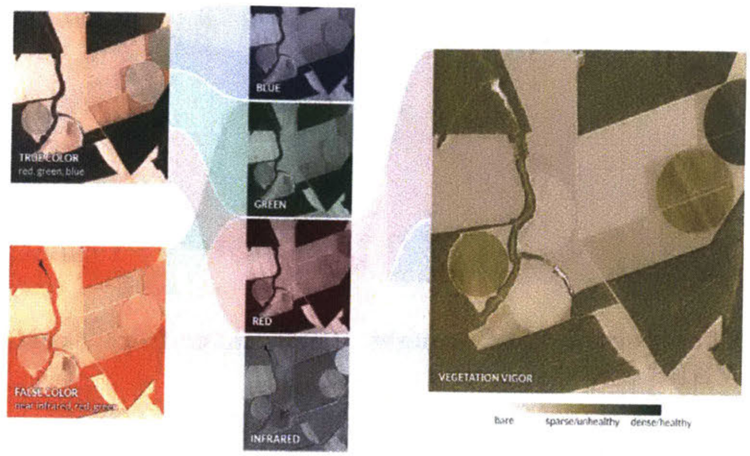

Figure 2-14: The four imaging bands on the Dove CubeSats are combined into one picture [63]

Each launch of multiple Dove satellites is called a Flock and is given a number and letter designation. Figure 2-15 shows a flight model of a Dove CubeSat showing the solar panels and the aperture cover deployed.

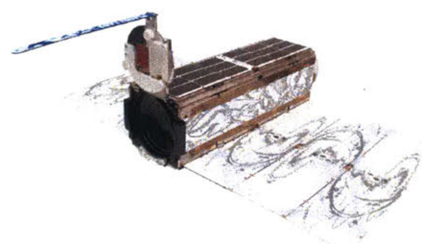

Figure 2-15: Image of a Dove CubeSat from Planet [6]

Comparison

JPSS-1's VIIRS is a spectrometer while Planet's Dove CubeSat does multi band photometry. Of all the sensors compared in this thesis, these two are the most dissimilar. The Dove CubeSat does not provide capability that is comparable to the results from VIIRS. The biggest reason for this it that CubeSats are constrained by aperture size and volume. The Dove CubeSat has a 9 cm aperture, the biggest a 3U CubeSat can support without deployables. The VIIRS instrument has a 19.1 cm aperture, over twice the size of the Dove CubeSat. It is also hard to fit a high enough resolution spectrometer on a 3U or even 6U CubeSat. Table 2.10 compares the key specifications of VIIRS and the Dove CubeSat from Planet. For some of the specifications mentioned, data is not available for the Dove CubeSat due to its proprietary nature but best estimates are made.

Table 2.10: Key specs of JPSS-1's VIIRS sensor and the Dove CubeSat [62],[63], [45]

	VIIRS **JPSS-1**	**Dove** **3U CubeSat**
Swath	3000 km	24.6 km $\times$ 16.4 km at 475 km alt
Nadir Resolution	400 m	3.7 GSD at 475 km alt
Total Channels	22	4
Spectral Bands	0.412 - 12 μm	Blue: 455-515 nm Green: 500 - 590 nm Red: 590 - 670 nm NIR: 780 - 860 nm
Mass	280 kg	5 kg
Power	319 W	~10 W
Max Data Rate	7.674 Mbps	~15kbps

The first key difference that jumps out from this table is the size of the individual images from these sensors. Each image from the Dove CubeSat is 24.6 km x 16.4 km while VIIRS can image areas up to 3000 km. The small footprint of the Dove CubeSat means that it can take months for a single CubeSat to revisit the same area.

VIIRS has 22 channels covering a wider spectral range and at higher resolution than does the Dove CubeSat. VIIRS is especially effective in its use of the near-infrared to infrared band which supports nighttime imaging. The Planet Dove satellite does not have this capability. Dove CubeSats are most effective at taking pictures during daylight. Their use of the near infrared (NIR) band can provide information on vegetation like agricultural crops, but this band on Dove is not typically used for low-light imagery.

While the Dove CubeSat represents CubeSat imaging capabilities and take images of a similar quality to VIIRS, the size and spectral content of the images it does take are not similar to the VIIRS sensor. To truly cover all the 22 bands that VIIRS covers, at least another 4 CubeSats would be necessary, each with bands cooresponding to VIIRS bands. One of these CubeSats would need to be a near IR imager to match VIIRS's Day/Night bands, ideally this would be a spectrometer but still needs significant development.

2.4 CubeSat Constellation Orbital Insertion

2.4.1 Launch and Deployment

CubeSats are generally launched as secondary payloads on launch vehicles. They can be easily supported as secondary payloads because of their standardized size and deployment configuration. There are also a number of new launch vehicles in development that are built specifically for taking small satellites, like CubeSats, to orbit as primary payloads. This allows mission planners to choose which orbit planes to send the small satellites to.

Table 2.11: CubeSat Launch Vehicles through 2016 [25]

Launch	Country	CubeSats Launched
Falcon 9	US	109
Antares	US	81
Atlas V	US	78
H-II (A and B)	Japan	37
Dnepr	Ukraine	26
PSLV (-G, -XL)	India	26
Soyuz (-U, -2.1a, -2.1b)	Russia	16
Long March (6, 7, 2D, 11)	China	12
Super Strypi	US	12
Delta II	US	4

Table 2.11 lists all the launch vehicles that launched CubeSats through the end of 2016. The PSLV has now launched well over a hundred CubeSats after is launched Planet's Flock of 88 CubeSats in February 2017. On all of these launches except for the Super Strypi, CubeSats were secondary payloads. The Super Strypi rocket is a three stage rail launched small satellite rocket with a payload capacity of 250 kg sponsored by the US Air Force. However, upon its first operational launch, in which it carried 12 CubeSats, it experienced a malfunction of the first stage motor and exploded before making it to orbit. After that failure, the project was disbanded [27].

A 2015 survey of the state of small satellite launch vehicles by *Niederstrasser et al.* found twenty-two different launch vehicles in development for small satellite specific launches. Of these, only two have since had operational launches, Super Strypi as mentioned above, and a Pegasus XL rocket from Orbital ATK Inc, Dulles, Virginia. Pegasus rockets have been in operation since the 1990s, and the XL version can carry 468 kg of payload to LEO. They are air launched from a Lockheed L1011 TriStar airplane. In late 2016, a Pegasus XL rocket successfully launched 8 small satellites making up the weather sensing constellation Cyclone Global Navigation Satellite System (CYNGSS) into orbit. However, through the end of 2016, it had not yet launched any CubeSat missions.

The rest of the rockets surveyed are at different stages of development. Of the twenty-two launchers, eight use air launches from planes, thirteen launch from land, and one launches from a balloon. The average payload capacity of these launchers is $\sim$200 kg to LEO. The rage of capacity is large however, where the smallest launcher, Small Air Launch Vehicle to Orbit, can only take one CubeSat at 4 kg to LEO and the largest, Athena Inc, can take 760 kg to LEO [27]. Some of the most likely small satellite launch vehicles to make it to an operational status are LauncherOne from Virgin Galactic and Electron from RocketLab. Both are scheduled for test flights throughout 2017 with the goal to begin commercial operations in the following years. LauncherOne can carry 225 kg of payload while Electron carries about 100 kg of payload [27].

CubeSats are integrated into launch vehicles within a deployer that will eject the CubeSat from the launch vehicle. The exception to this approach is the NanoRacks system, NRCSD, on the ISS. In this system, the CubeSat is sent to the ISS on a resupply mission and deployed from the ISS through NRCSD. The NRCSD has six deployer pods, where each deployer can contain two 3U CubeSats stacked on top of each other (or an equivalent total of smaller CubeSats).

The P-POD is a more conventional type of deployer which deploys a CubeSat directly from the launch vehicle. P-PODs come in 1U x 3U and 2U x 3U sizes to fit any capacity CubeSat from 1U to 6U. The P-POD is Cal-Poly developed, by the same group that created the CubeSat specification. The CubeSat is integrated into the P-POD in such a way that it compresses a spring at the back of the P-POD and the door to the P-POD is sealed. When that door is released after launch, the CubeSat is pushed out of the P-POD by the spring. There are access ports in the P-POD so that CubeSat charging and data ports can be reached once the CubeSat is integrated in the P-POD [66].

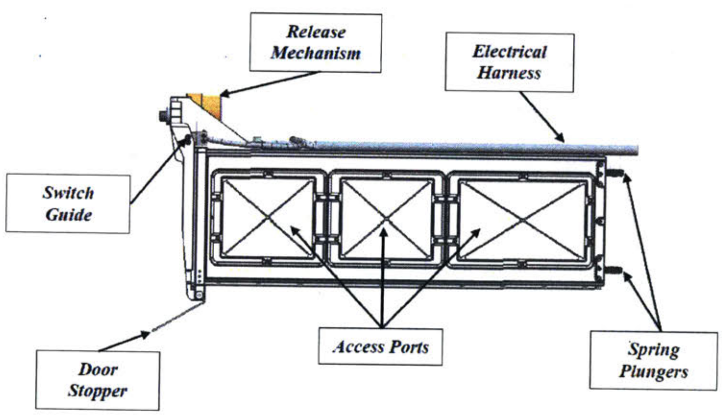

Figure 2-16: Cal-Poly's P-POD [66]

The most popular international version of a Cubesat deployer is the European ISIPOD from Innovative Solutions In Space. This version comes in 1U, 2U, and 3U sizes with a 6U in development. It works much like a P-POD but is qualified to deploy heavier CubeSats than the P-POD. For instance, the P-POD maximum mass for a 3U is 4.4 kg, while for the ISIPOD, 6 kg is the maximum mass for a 3U

[67]. There are other deployers that have been developed by universities, such as the University of Tokyo's Tokyo Pico-satellite Orbital Deployer, or the University of Toronto's Experimental Push Out Deployer but they are not used as often as the P-POD, ISIPOD, or NRCSD.

2.4.2 Differential Drag

It is extremely difficult to put propulsion systems on CubeSats for a few reasons. The first is that CubeSats are generally secondary payloads. As a secondary payload, a CubeSat has to demonstrate to the primary payload that it does not pose a hazard to the mission. Propulsion systems, that contain hazardous, combustible fuels and pressurized vessels, can pose significant risk to the primary payload.

A second reason is that it is physically difficult to fit a propulsion system on a CubeSat of a 3U size and smaller because of the size, weight, and power (SWaP) the systems require. CubeSats are tightly packed systems due to their limited volume. Even volume used by cabling must be accounted for in models to ensure there is sufficient room. For example, a typical distribution of volume in a 3U CubeSat is approximately 1U for the payload and corresponding electronics, 1.5U for the bus, and 0.5U for the attitude control system actuators. This distribution varies by the type of payload and required bus support systems, but generally CubeSats are sized and built in extremely dense forms.

Propulsion systems require significant resources. They can take up a large amount of volume within a CubeSat to account for fuel tanks, piping, controllers, electronics, and thrusters. They also have high power requirements during operating modes. Table 2.12 lists some of the available CubeSat thrusters from Busek, a provider of space propulsion systems to give a general idea about the specifications of COTS CubeSat propulsion systems today. These volumes typically do not contain the necessary electronics or fuel tanks.

Table 2.12: **Sample of CubeSat Propulsion System Options** [68]

Thruster Type	**Volume**	**Power**	**Mass**	**Delta-V**	**Isp**
Micro Resistojet	810 cm^3	3 - 15 W	1.25 kg	60 m/s	150 s
Electrospray	433.5 cm^3	15 W	1.15 kg	151 m/s	1300 s
PF Ion Thruster	27.9 cm^3	10 W	0.053 kg	Not Available	2150 s

There are smaller options like the Scalable ion-Electrospray Propulsion System from MIT that can feasibly be supported by a CubeSat but can currently only provide enough delta-v to counter the effects of drag on a CubeSat and to extend the lifetime of the satellite. This is currently one of the less resource-intensive implementations of propulsion on a CubeSat. In the context of constellations, propulsion would be most useful if it could move CubeSats enough within their orbit planes to achieve the proper constellation configuration. Multiple launches would still be required if the constellation had multiple planes.

Current propulsion options are generally not practical for CubeSats 3U or below. Thrusters with a higher Isp take less fuel to obtain a certain delta-v than thrusters with a lower Isp, the higher the Isp, the more efficient the system. These efficient systems provide lower thrust, taking longer to achieve the necessary delta-v. The Micro Resistojet therefore provide the fastest change in velocity, but overall provides less delta-v in a larger volume. These thruster systems provide such little delta-v over such a long period of time they are impractical to use to maneuver into a constellation configuration. This makes using differential drag to implement constellation architecture an appealing method.

Differential drag involves manipulating the attitude of a satellite to affect the ballistic coefficient and rates of atmospheric decay of the satellite, ultimately changing the rate of the CubeSat's mean motion. By correctly using differential drag, the orbit of a satellite can be changed as desired. Differential drag is most effective at altitudes less that 700 km as the atmosphere is more dense and the changes in attitude of the spacecraft will have a greater effect on the orbit [69], [70].

Differential drag only applies a change in velocity in the tangential direction of the satellites orbit. An acceleration in this direction can affect the semi-major axis, the eccentricity, the argument of perigee, and mean anomaly of a satellite. With near circular orbits, the change in argument of perigee and eccentricity is essentially negligible. It is the change in the semi major axis which causes the change in mean anomaly, allowing satellites to be separated throughout nearly the same orbit [69].

This is the primary method that Planet uses to install its CubeSats in the desired orbital plane slots, equal spacing of all satellites over the orbital plane. Planet first tested this method with ten 3U CubeSats that they spaced in a single orbit over about 4 months. Figure 2-17 shows the successful results from this first test.

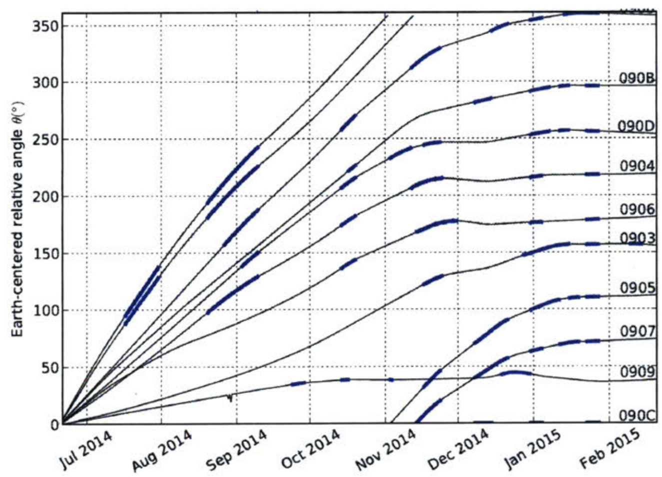

Figure 2-17: Ten 3U Cubesats equally spaced in orbit over four months using differential drag [70]

Both Planet and AeroCube-4 have shown that differential drag is a viable option for implementing constellation architectures for CubeSats. The use of differential drag in establishing a constellation will be considered feasible throughout this analysis [70], [69].

2.4.3 Constellation Configurations

Because CubeSats to date have been secondary payloads, there are difficulties in establishing a well-organized CubeSat constellation. It is difficult, but not impossible, to launch CubeSats that would achieve near global coverage and low revisit rates with existing launches. To do this, the developer would need to have many flight vehicles ready for integration on multiple launch vehicles and plan for their mission lifetimes to overlap sufficiently [3]. With most CubeSat lifetimes in LEO ranging from 6 months to 6 years depending on altitude, to establish a constellation, the sequence of launches may have to be within a short period of time. This thesis considers three main ways of establishing a CubeSat constellation: (i) being the primary payload on a series of launches, the dedicated launch case, (ii) through mass deployment on a single launch, the Planet Case, and (iii) an ad hoc series of launches, the ad hoc Case.

Planet is known for their massive launches of CubeSats [25]. In early 2017 they deployed 88 CubeSats from a single PSLV. This was the largest deployment of satellites from a single launch to date. As discussed Section 2.4.2, differential drag is then used to place the CubeSats in the desired orbit. This method ensures that all satellites in the constellation are launched within the same time period. This is the best method for Planet to feasibly launch large numbers of satellites. However, it does not provide as large a variety of orbits as separate launches would which could decrease revisit times and increase coverage areas.

Alternatively, CubeSats can be launched into an ad hoc constellation, meaning that a number of the CubeSats in the constellation are launched on a series of available launches. Whenever a CubeSat launch is available, one or or more of the CubeSats in the constellation will be on the manifest. The performance of this type of launch system was analyzed by *Marinan, 2013*. In this research it was found that the ad hoc constellations generally had better revisit times than a planned Walker constellation. However, the Walker constellation outperforms the ad hoc constellation in response and percent coverage [3]. This method of CubeSat constellation formation is most useful when the CubeSat will have a long lifetime so that no part of the constellation will have de-orbited before the entire constellation has been installed.

A third method, which is the most desirable with the appropriate funding, is for a CubeSat constellation to purchase their own launches and be inserted into the desired orbits at the desired times from the beginning of the mission. This method may be used by MIT Lincoln Laboratory TROPICS CubeSat Constellation, allowing the CubeSats to launch with the correct timing between launches and into the orbits they desire. The drawback of this method is that as the primary payload on multiple missions, the cost of the mission will increase significantly. The development of small satellite launchers may make this method more of a reality in the years to come.

These three CubeSat constellation installation methods motivate the approach for the cases studied in this thesis. The constellation type defined in each of the three case study is based on the configuration resulting from one of these three possible methods of launching a CubeSat constellation.

Chapter 3

Simulation Methods

This chapter discusses the simulation approach and describes the metrics used to compare a distributed system of CubeSats to the JPSS-1 satellite. The CubeSat system is set up as three different constellations: (i) being the primary payload on a series of launches, the Dedicated Launch Case, (ii) through mass deployment on a single launch, the Planet Case, and (iii) an ad hoc series of launches, the Ad Hoc Case. This is done through simulation which provides information on revisit times and coverage of the constellation. A cost analysis on each constellation is performed, taking into account development, production, launch, mission operations, and any required spares or expected replacements.

3.1 Simulation

This thesis presents three different case studies: (i) the Dedicated Launch case (ii) the Planet Case, and (iii) the Ad Hoc case. Each case has a different constellation configuration based on different possible launch configurations. The system of five different types of CubeSats represents the bare minimum, if even that, to distribute the JPSS-1 system. It is clear that none of the CubeSat instruments can currently replicate the same performance as one of the JPSS-1 instruments hosted on a single 3U CubeSat. To supplement the lack of science data, the constellations in these case studies will have 15 CubeSats, three of each of MicroMAS, RAVAN, CIRAS, PICASSO, and Dove. This will allow for improved revisit times and coverage area, better for observing transient local phenomena, as well as overall increased resiliency to the system.

Simulations of constellations will be done in Analytical Graphics, Inc. Systems Tool Kit (STK). The cases that will be simulated are briefly described in Table 3.1.

Table 3.1: Table of simulation cases

Case	Description
Reference Case	The JPSS satellite
Case 1	A planned series of 3 CubeSat launches into three orbital planes
Case 2	A single mass launch of CubeSats into one orbital plane
Case 3	Three ad hoc launches of CubeSats into three orbital planes

For the Reference Case, the Dedicated Launch Case, and the Planet Case the simulation will be started on 23 September 2017, the launch date for JPSS-1 as of Spring 2016. The Ad Hoc Case will being on the date of the last expected ad hoc launch, September 30 2017. The simulation will be run over six months to determine the maximum revisit times and the percentage of time covered by the constellation using specifications from the JPSS-1 and CubeSat sensors. Coverage and revisit times are calculated in STK using a grid of points as placeholder observations with a point located every 5 degrees of latitude and longitude.

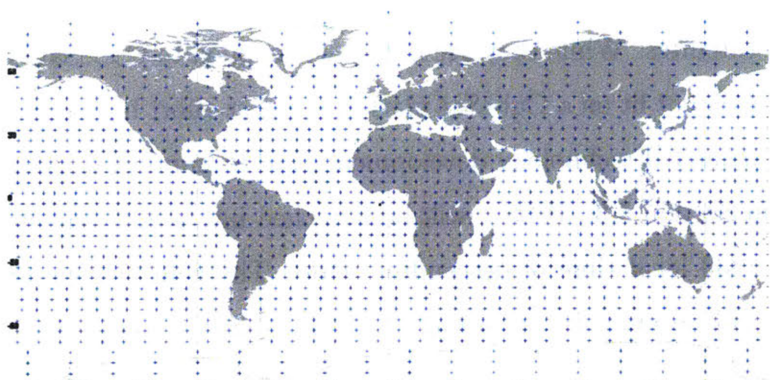

Figure 3-1: 2D view of the Earth with coverage analysis grid of simulation shown

Revisit time is defined as the time between coverage of a grid point when access to that point is not available. For each of the different constellation cases, the average revisit time at each point pictured in Figure 3-1 will be shown for each type of CubeSat sensor. These data points give an good idea of which areas the CubeSat constellation will be able to cover the longest and which it will revisit the most often.

In determining the orbital dynamics of the CubeSats, the same satellite cross-sectional area will be assumed for each CubeSat so that all satellites in the same orbits will have approximately the same lifetimes. The J2 effect, the astrodynamic effect of the Earth's oblateness, is also considered. The oblateness of the Earth causes the line of nodes to regress for prograde orbits and progress for retrograde orbits. As long as the inclination of all orbital planes are the same however, the RAAN for each orbit should regress at the same rate. The gravitational effects of third bodies, like the Moon, are not considered in determining the CubeSat orbits. STK uses an SPG4 propagator to predict the orbital dynamics.

3.2 Case Studies

3.2.1 Reference Case: JPSS-1

JPSS-1 is the reference mission for this work. During its development, JPSS-1 has experienced cost and schedule overruns that have led to a potential gap in 45% of weather data. JPSS-1 also has multiple independent sensors that can be distributed to individual satellites. While they are not easily mapped to CubeSats because these sensors simply do not fit on a CubeSat platform, a best-effort has been made to do so here. A recommended technology development plan to improve CubeSat technology is discussed in Section 5.3. JPSS-1 will be launched into a 824 km sun synchronous orbit with a 1330 local time of descending node. The field of view (FOV) of each JPSS sensor will be modeled according to their published specifications. It will be assumed that each sensor can be turned on whenever necessary to view an area of interest, giving the possible revisit rates to each location.

Table 3.2: **JPSS-1 Sensor FOV as modeled in the Reference Case** [45]

Sensor	Vertical Half Angle	Horizontal Half Angle
VIIRS	61.2°	13.6°
CERES	63.0°	0.7°
OMPS	59.5°	1.7°
ATMS	57.6°	2.6°
CrIS	43.8°	0.5°

Figure 3-2 shows the footprint of the JPSS sensors given their FOV and swath width (vertical half angle); see Table 3.2 for a reference on the field of view of each sensor.

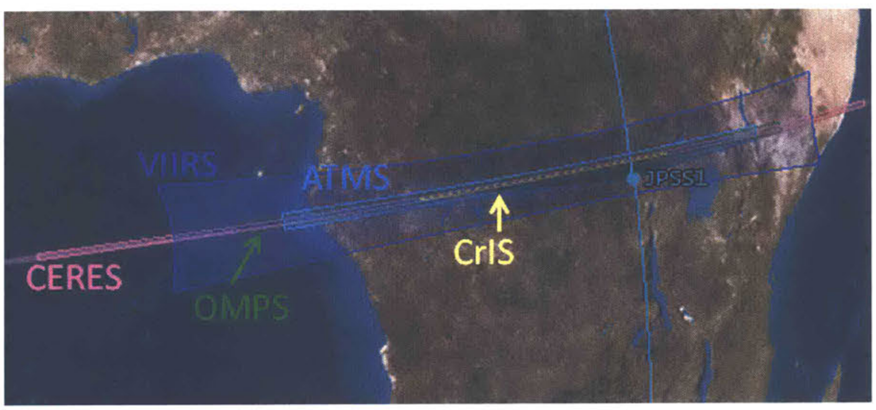

Figure 3-2: JPSS Sensor field of view shown in STK as used in the JPSS-1 Reference Case

3.2.2 Case 1: Planned 3-Plane Constellation

The Case 1 dedicated launch constellation will represent a planned constellation where the orbits of the satellites are selected by the CubeSat provider and configured in a desired arrangement. This planned constellation will be a combination of a streets of coverage and string of pearls constellation, based on the A-train constellation of weather satellites. In the A-train constellation, eight weather and atmospheric sensing satellites fly in a line on the same polar orbital track, crossing the same spot within seconds to minutes of each other. This allows for viewing of a location with different sensors in quick succession during the coverage of a single string. By having three strings in different planes, the revisit time to an area between different strings should be minimized.

Applying this configuration to the CubeSat system, there will be three orbital planes of the same inclination, eccentricity, and semi-major axis with equally spaced RAANs and five CubeSats per plane. One of each of the five CubeSats will be located in the orbital planes spaced thirty seconds apart from each other. These orbits will be based on the JPSS-1 orbit with an altitude of 824 km and an inclination of 97°. They will have equally spaced RAANs, with the first RAAN of 205° corresponding to an LTDN of 1330 and the other two, 85° and 325°, spaced 120° from the first orbital plane. This is shown in Figure 3-3.

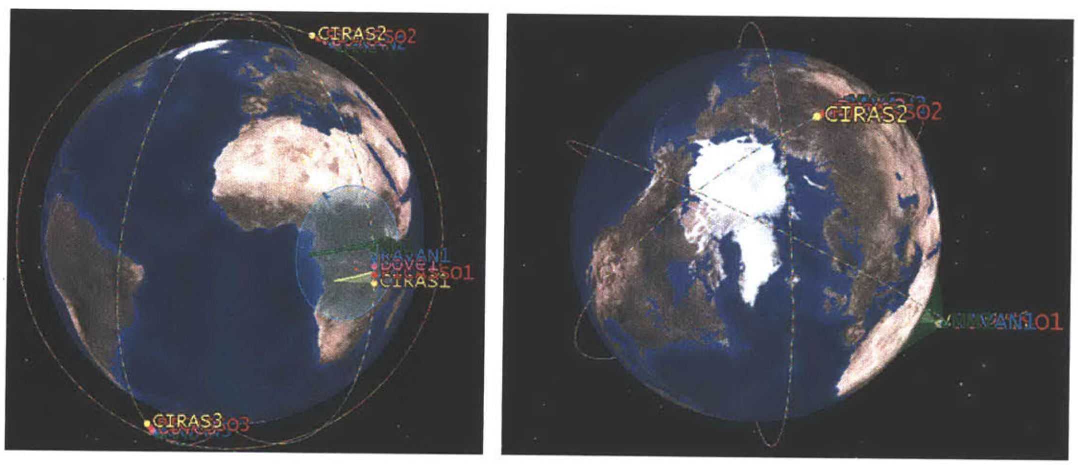

Figure 3-3: The 15 CubeSats in a String of Pearls/Streets of Coverage configuration.

The string of pearls design, with the CubeSats left in clusters allows for fast revisit times to a spot while the cluster is passing overhead. This is useful for studying phenomena that change significantly at a fast pace. It allows the same area to be viewed five times in a ten to twenty minute period. However, that specific area may then go unmonitored for hours.

This constellation configuration must be achieved through three dedicated launches for each orbit plane. For this type of dedicated launch, the CubeSats will have to be the primary payloads to obtain their desired orbits. The small satellite launchers will still have a lower cost than if three full sized launchers were used. However, the selection of small satellite launchers may be limited by the required orbit; like those in this case at 824 km, altitudes that are about twice that of the ISS, the higher side of a LEO orbit. This may be challenging to reach for some of the small sat dedicated launchers.

3.2.3 Case 2: Single Launch

The Case 2, Planet case, constellation is based on the constellation launching on a single rocket, similar to how Planet installs their constellations. The single rocket launch for this case will be considered to be the same launch as JPSS-1. This puts all fifteen CubeSats into an approximately 824 km sun-synchronous orbit with an LTDN of 1330. When ejected from their launch vehicle the CubeSats will be clustered in slightly different orbits, with the average of the orbits assumed to be the JPSS-1 orbit. After deployment, the CubeSats will use differential drag to place themselves equally spaced throughout the orbit. This is shown in Figure 3-4 where the slight differences in the orbits can be seen.

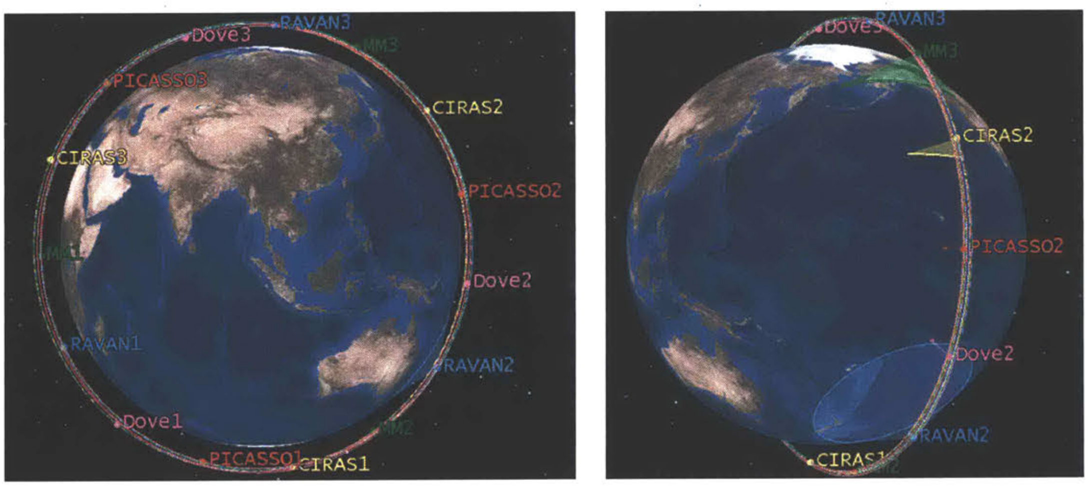

Figure 3-4: Left: The 15 CubeSats evenly spaced through a single orbital plane. Right: Shows the footprint of each CubeSat sensor in this constellation configuration

To get an idea of the possible differences caused by ejection from the deployers, the orbits of the Planet 88 CubeSat deployment in early 2017 are considered. The results are not expected to be the same because the Planet deployment had far more satellites and they deployed over a longer period of time. As time progressed, the differences in orbit parameters between satellites grew for the Planet Labs deployment. Table 3.3 gives the statistical results from the analysis of the Planet CubeSats.

Table 3.3: Distribution of 88 Planet CubeSat orbits from a single rocket [25]

Orbital Element	Average	Std. Dev 1-σ
Inclination	97.5°	0.0013°
RAAN	133.7°	0.19°
Eccentricity	8.8×10^{-4}	7.0×10^{-5}
Argument of Perigee	167.5°	9.4°
Semi-Major Axis	6878.6 km	0.64 km

The most affected orbital element, based on this analysis, appears to be the Argument of Perigee with a 1-σ standard deviation of approximately 9.4°. However, with eccentricities so near to zero, the location of the argument of perigee is almost negligible in analysis. The argument of perigee is defined as the angle between the orbit's perigee and its ascending node and so defines the location of apogee and perigee. Orbits like these that have such low eccentricity are essentially circular; the average difference between apogee and perigee altitude in this case is only about 12 km. For comparison, the Earth's oblateness causes the difference in radius between the equator and the poles to be about 30 km.

The differences in RAAN, while small, separate the orbits to put the CubeSats in slightly different locations as they cross the equator. The difference in semi-major axis, 0.64 km, is a negligible amount as it does not significantly change the coverage area of any sensors despite the slight change in altitude of the orbit. The inclination change affects which latitudes the satellites can image as they will only travel as far north or south as their inclination. However, again the changes in inclination of 0.0013° are almost too small to affect the coverage. Finally, the changes in eccentricity affect the difference between apogee and perigee by about 0.5 km, again a negligible amount on its own when the swath width of these sensors can be thousands of kilometers long.

However the combination of these small differences can compound and create orbits that are different enough to have slight effects on the revisit times. For the sake of completeness, the orbits from this case were normally distributed using this data to create a variety of slightly different orbits that better represent the selection of orbits that might result if fifteen CubeSats were deployed from the JPSS-1 launch. The slight differences in these orbits are shown in Figure 3-4.

3.2.4 Case 3: Ad Hoc Constellation

The third case represents launching a constellation using a series of three different available launches, giving this case five CubeSats per orbital plane. Only the 2017 launch manifest is considered for possible launch options. This narrows the time between launches to a year at maximum which ensures that if high enough orbits are selected, the lifetime of the CubeSats will be long enough to have the full constellation in operation.

Current launch manifests show that there are eleven launches scheduled for 2017 that will carry CubeSats. Of these, five are US launches and the rest are international [71]. It will be assumed that when implementing this CubeSat constellation, only US launches can be considered. While this assumption used to hold true in the past when US based companies had trouble exporting satellites, it does not always apply any more. Planet launched their 88 CubeSats from India, and MicroMAs-2 will also launch from India. However, most large NOAA/NASA satellites launch on US launches. Since these CubeSats would be part of the JPSS-1 program, it will be assumed they also fly on US launches.

Of the five US launches available in 2017, two are slated for ISS resupply missions. While CubeSats deploy from the ISS all the time, ISS orbits are not sufficient for this mission as the CubeSats will deorbit within about 6 months from the low 400 km altitude orbit. This mission requires at least 2 years before deorbit.

That leaves exactly three launches that meet the minimum acceptable standards for this mission. The first launch is on May 31st from a Minotaur-C where the main mission is to launch six SkySat satellites into approximately 500 km sun-synchronous orbits [72]. The next launch is the JPSS-1 launch on September 23 from a Delta II rocket into an 824 km sun-synchronous orbit [72]. The final is a launch on the Falcon Heavy rocket from SpaceX on September 30. Unfortunately this is a rather risky launch as it is the first operational launch of this rocket. The main payload of this launch is the USAF STP-2 satellite which will be launched into an approximately 641 x 652 km 72° inclination orbit [72]. The RAAN values for the first and second launches have not been specified so a best case scenario of equally spaced RAANs will be assumed.

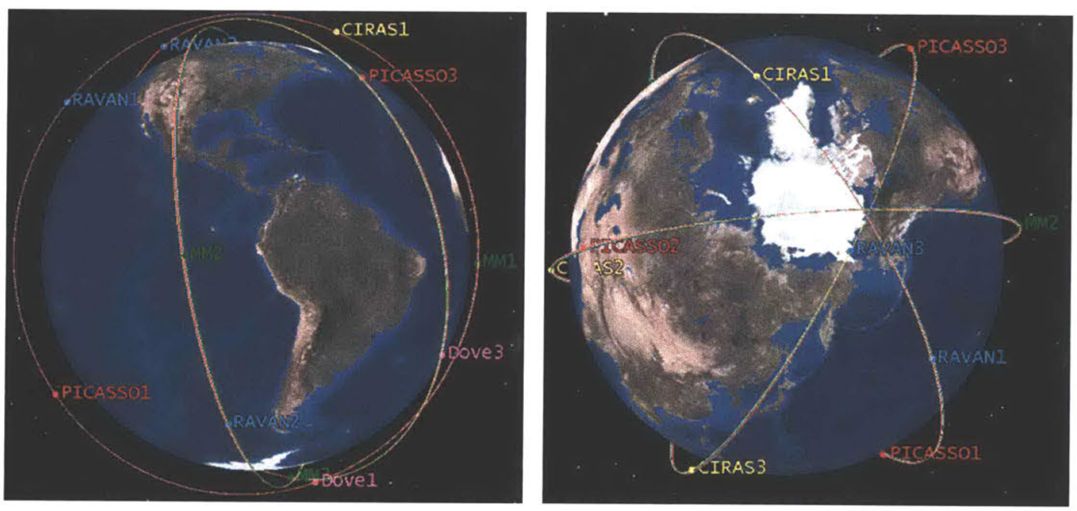

Figure 3-5: Case 3: The 15 CubeSats are spread through three orbits from ad hoc launches.

Figure 3-5 shows the orbits for Ad Hoc Case and the placement of the CubeSats in their orbits. As in the Planet Case, the CubeSats are equally spaced throughout the orbits using differential drag.

3.3 Cost Analysis Approach

Over the past few decades, data have been recorded on the larger satellite systems, resulting in cost modeling tools that have proven somewhat accurate for less complex satellites. With the emerging field of CubeSats, that type of data is not as available. Unfortunately, it is not possible to scale CubeSat costs in proportion to their mass [73]. There are a few different cost models that try to deal with this problem of small satellite cost modeling including the Small Satellite Cost Model and the Demonstration Satellite Cost Model [74], [73],[16].

In building a cost model for CubeSats there are problems other than the lack of flight mission data. The CubeSat industry is ever changing and evolving. The large range in types of missions CubeSats can do make building a standardized model difficult. The lack of standard procedure in documenting CubeSat costs and development/testing methods also means that each CubeSat builder will perform their own range of tests and may or may not accurately track costs. This lack of standardization also developing an accurate cost model difficult.

One model made for small satellites, not specifically CubeSats, is called the Small Satellite Cost Model (SSCM) from The Aerospace Corporation. This model focuses on satellites that are less than 1000 kg. It uses heritage data from other programs to establish cost estimating relationships. The model has separate options for satellites greater than and less than 100 kg, but the model for satellites less than 100 kg is not as robust and detailed as is the model for larger satellites due to lack of data.

Another possible cost model for small satellites is the Demonstration Satellite Cost Model (DSCM) created by the National Reconnaissance Office. This model is specifically for small satellites that are demonstrating new technologies. These satellites accept much more risk than do higher level satellites that are intended for operational use [73]. In this thesis, the CubeSats are not treated as technology demonstration missions. They are to be used operationally and are assumed to be less accepting of risk. This analysis assumes that the CubeSat technologies will have already been demonstrated sufficiently on precursor missions and are at a least at technology readiness level of 7.

To estimate the cost of designing, building, testing, launching, and operating these CubeSat constellations, estimates are based on available heritage information from the MicroMAS, MiRaTA, and MicroMAS-2 satellites [16]. Specific cost information on the other CubeSats studied here is not currently available and so best estimates are made. The cost analysis will takes into account a two year life time for each CubeSat and the need to replace the aging constellation at least three times to meet the operational life time of JPSS-1 (seven years). It also leverages the benefits from multiple productions of the same satellite.

The major costs that are considered in this analysis are: staff, hardware, launch, and operator costs. Staff costs are the costs associated with paying all personnel involved with the program, including management, engineering staff, and technical staff. Students can also be included in this category as they are compensated in fellowships or stipends. Hardware costs refer to all the hardware purchases and repairs that support the program, including not only flight and engineering model components, but also ground support equipment. Launch costs include the cost to purchase a launch opportunity on a rocket and the launch integration service. Operator costs are the costs for operators working at the ground stations supporting satellite operations.

For staffing costs, each CubeSat Type will have its own team. Each team will be assumed to have eleven full staff members paid an annual salary of $200,000 from project kickoff to Mission Readiness Review. For hardware costs, each satellite will be assumed to have approximately the same bus system needs. The power system will include a battery, electric power system board, and solar panels. The attitude determination and control system will include three reaction wheels, three torque rods, an inertial measurement unit, six sun senors, and two earth horizon sensor assemblies. The command and data handling system will be made up of two electronics boards. The communication system will include two radios. The structure will be assumed to cost approximately the same whether the CubeSat is a 3U or 6U. The total bus hardware cost is assumed to be $350,000. Because of lack of available data, the payload hardware cost for each CubeSat will be assumed at $650,000 [16].

Operator costs have been assumed at $200 a day in other CubeSat cost analysis reports and that figure will be used in this analysis [16]. Operator services will be used every federal work day, 250 days per year, throughout the lifetime of the constellations. Each CubeSat type will have two operators assigned to them. Launch costs are generally not publicly available. Estimates of small satellite specific launcher costs range from \$20k-\$56k per kilogram in orbit [27]. However, in each of these cases small satellite launchers are not used. The Delta II, Falcon Heavy, and Minotaur-C are the launch vehicles used throughout these cases. The company Spaceflight Industries, Inc which buys excess space on launch vehicles such as the Falcon 9 estimates their cost for a 3U CubeSat as \$295k and \$545k for a 6U CubeSat [31]. These values from Spaceflight Industries will be used in the Planet and Ad Hoc Cases where the CubeSats are secondary payloads.

For Case 1, the Dedicated Launch Case, the constellation launches as the primary payloads on a small satellite launch vehicle such as Pegasus, Minotaur I, or Minotaur C. Each of these launches is approximated to cost \$40 million [27]. It can be assumed that even as the primary payloads, this CubeSat program will be able to share the cost of the launch with secondary payloads. It will be assumed that because of secondary payloads, this program only has to pay 50% of the full cost of a launch. \$20 million per launch will be used to analyze cost of launch in the Dedicated Launch Case.

When determining the cost of the constellation using these cost estimates, the Planet case and the Ad Hoc Case will have the same cost numbers. This is because they use the same launch methods as secondary payloads. The Dedicated Launch Case will be different because each launch must be purchased entirely by the CubeSat program because in this case the constellation is the primary payload.

In reality the Planet Case launch may be cheaper than the Ad Hoc Case because all of the CubeSats are going up on a single launch, so there may be some sort of bulk discount. However, trying to estimate what that discount might be is outside the scope of this analysis. The results for the Ad Hoc Case may also vary because this analysis approximates that each launch vehicle charges the same amount for a CubeSat ride. However, different launch vehicles, as we have in the Ad Hoc Case, likely charge different rates. Since that data is not publicly available, it must be assumed that that the cost to launch a CubeSat as a secondary payload is essentially the same across all launch vehicles.

JPSS-1 has a seven year lifetime. The CubeSats are expected to have 2 year lifetimes. To account for this, the constellation must be "refreshed" with new CubeSats every two years. To match the seven year lifetime, this means that there must be at least three rounds of replacements to the aging CubeSats. In this analysis, it will be assumed that the total 60 CubeSats (four installations of fifteen) are built at the beginning of the program. This will be assumed to take 3 years with 11 full time staff members working on each type of CubeSat from program kickoff until first launch. These are considered the Design/Build staff. This cost analysis classifies the type of staff into three categories, Design/Build Staff, Data Analysis Staff, and Operators. After launch, the team for each type of CubeSat will be reduced to two full time staffers who will plan operations and analyze the data from the CubeSats over the lifetime. The operators will handle communications and control with the CubeSat during overpasses, collecting all data for the Data Analysis team. There are two operators assigned to each CubeSat.

Overall, this cost analysis is very much an estimate. The number of necessary staff and hardware may vary by CubeSat type instead of being held constant as they are in this analysis. Schedules may be delayed or accelerated. Launches may be more expensive or cheaper than they are assumed to be here. Inflation has not been taken into account and can add to this uncertainty. To account for this, the budgets hold a 20% margin applied overall. Appendix A shows details on the cost analysis calculations and data.

Chapter 4

Results

This chapter details the results from the cost analysis and constellation simulation analysis. The first section, cost analysis results, shows the differences in cost breakdown between all three cases. It also highlights the relatively small cost of the constellation budget relative to JPSS-1's overall budget. Finally, it goes into some detail on how adjusting the assumptions of this simulation would change the final cost. The second section, the constellation analysis, presents the revisit times provided by each sensor and compares these results between the different constellation architectures and sensors.

4.1 Cost

This section gives the results from the cost estimation analysis done on the fifteen CubeSat constellation baseline for each Case studied. Figure 4-1 shows the breakdown of the total cost to build, launch, and operate these constellations. For Case 1, the Dedicated Launch Case, the approximate cost is $433.2 million. For the Planet and Ad Hoc Cases, Cases 2 and 3, the approximate total cost constellation is $170.0 million. Note that this number includes three "refreshments" in addition to the original installation of the constellation to replace the aging CubeSats every two years. With the three years design and build time, this gives a total mission life of 11 years.

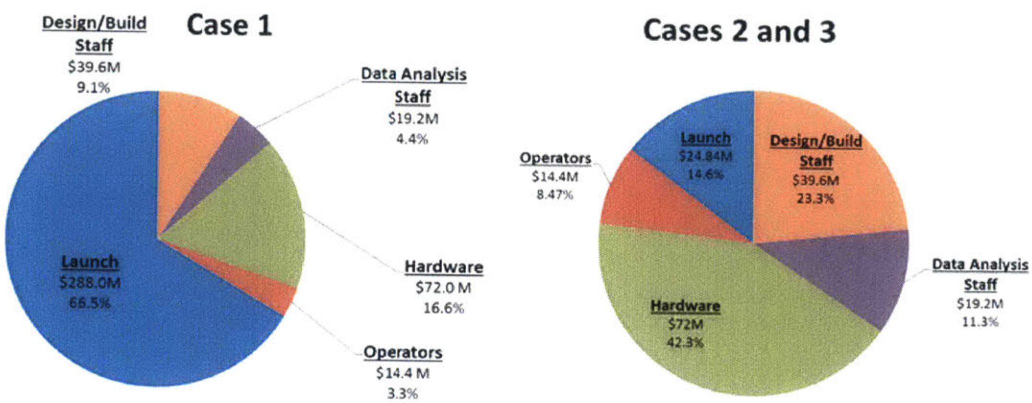

Figure 4-1: Left: Cost breakdown for a Case 1 constellation of 15 CubeSats, 3 orbital planes. Right: Cost breakdown for a Case 2 (15 CubeSats, 1 orbital plane) or Case 3 (15 CubeSats, 3 orbital planes) constellation.

The Dedicated Launch and Ad Hoc Cases, Cases 2 and 3, have the same cost estimate since they each assume the same launch cost per CubeSat as a secondary payload. Case 1, the Dedicated Launch Case, buys three launch vehicles so that the constellation can be the primary payload on each launch and so is dominated by launch costs. Case 1 is assumed to pay 50% of the cost of a full launch due to secondary payloads helping to offset the cost of the launch. The Dedicated Launch and Ad Hoc Cases (Cases 2 and 3) show a more realistic CubeSat budget since CubeSats are generally launched as secondary payloads. Figure 4-1 shows that for Cases 2 and 3, hardware and staff make up the most of the constellation's budget. While all hardware is combined into one category, staff costs are separated. Staff costs combined make up almost 43% of the total cost of this constellation.

The JPSS project was allotted \$11.3 billion for S-NPP, JPSS-1, and JPSS-2. Assuming that each satellite is approximately the same (S-NPP and JPSS-1 have the same sensors), JPSS-1's budget is approximately \$3.77 billion. The CubeSat constellations cost approximately 11.5% of JPSS-1's budget for Case 1, the Dedicated Launch Case, and only 4.5% for Cases 2 and 3, the Dedicated Launch and Ad Hoc Cases. The constellations explored in Cases 2 and 3, compared to the overall JPSS-1 budget, cost very little. At only a 4.5% budget increase, this constellation would be within the noise of the budget. Case 1 is a more significant percent increase at around

11.5%. Achieving the desired configuration of constellation, as in Case 1, may not be worth the cost for the launches. The benefits of each constellation configuration will be explored in the following section, Section 4.2.

4.2 Revisit Time Results

This section details the results from each of the cases studied. The maximum and minimum revisit times shown in the table are averages of the maximum and minimum revisit times to each of the points defined on the grid shown in Figure 3-1. The average revisit time is the average revisit time across all points of the grid.

4.2.1 ATMS and MicroMAS-2

MicroMAS-2 and ATMS revisit times are presented in Table 4.1. These values represent the revisit times for the ATMS sensor alone and the combination of the three MicroMAS-2 CubeSats in the different constellations. MicroMAS-2 has a similar, if slightly smaller, field of view to ATMS. Because of this, even just two of the MicroMAS-2 CubeSats could be sufficient to meet the revisit time of the ATMS. With the three CubeSats used in each of these constellation configurations, MicroMAS-2 outperforms ATMS in all categories. However, it should be noted that since MicroMAS-2 does not cover all of the spectral bands of ATMS, another CubeSat with the missing radiometer bands would be necessary to fully match the capabilities of ATMS.

Table 4.1: MicroMAS-2 and ATMS Revisit Results

	Revisit Time (Hrs)		
	Avg	**Max**	**Min**
ATMS	7.6	12.2	6.4
Dedicated Launch Case: MM-2	2.7	4.7	0.8
Planet Case: MM-2	2.6	11.5	4.6 sec
Ad Hoc Case: MM-2	3.3	9.9	0.7

The Dedicated Launch Case provides the most consistent revisit times while the Planet Case provides the widest ranges. With the Dedicated Launch, revisit times to each point can be expected on a more regular basis. With the Planet Case, revisit times will be in quick succession of the three CubeSats with a longer wait until until there is another pass.

While Table 4.1 is a summary of these revisit results, Figure 4-2 shows in detail the revisit times for ATMS and MicroMAS-2 at each grid point across the globe. This figure shows that because the orbits in this case are near polar, the best coverage is near the poles, making the worst coverage at the equator. It should also be noted that a different scale is used for the ATMS sensor because the difference in revisit times is so great between ATMS and MicroMAS-2 in these configurations.

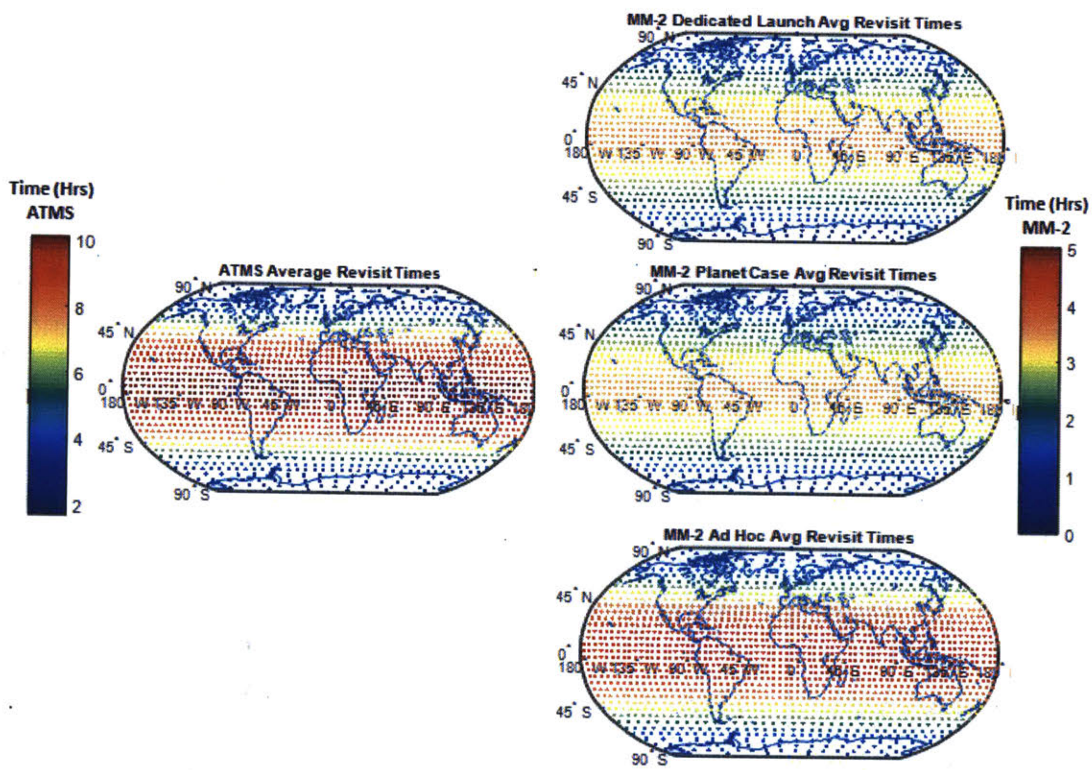

Figure 4-2: Left: ATMS average revisit times. Right: MicroMAS-2 average revisit times for each case.

4.2.2 CERES and RAVAN

The large field of view for the RAVAN CubeSat makes a significant difference in a comparison of RAVAN with CERES for revisit time. Even one of the RAVAN CubeSats would have better revisit times than CERES because of its larger field of view. In these constellations, the revisit time of the RAVAN CubeSats alone and the CERES instrument are summarized in Table 4.2.

Table 4.2: RAVAN and CERES Revisit Results

	Revisit Time (Hrs)		
	Avg	**Max**	**Min**
CERES	4.1	9.9	1.1
Dedicated Launch Case: RAVAN	1.2	2.4	0.2
Planet Case: RAVAN	1.8	9.6	1.5 sec
Ad Hoc Case: RAVAN	1.6	4.1	0.1

Table 4.2 shows that RAVAN has better revisit times than CERES by about three hours, as expected. Figure 4-3 shows these revisit times in more detail over the globe. For RAVAN, the Dedicated Launch case is clearly the better performer at giving consistent coverage globally, especially near the equator where it performs better by about 1.5 hours. The Ad Hoc Case is here the second best performer, getting better revisit times at the equator than the Planet case by about an hour. Because of the large field of view, the RAVAN architecture benefited from being spread more evenly across the globe.

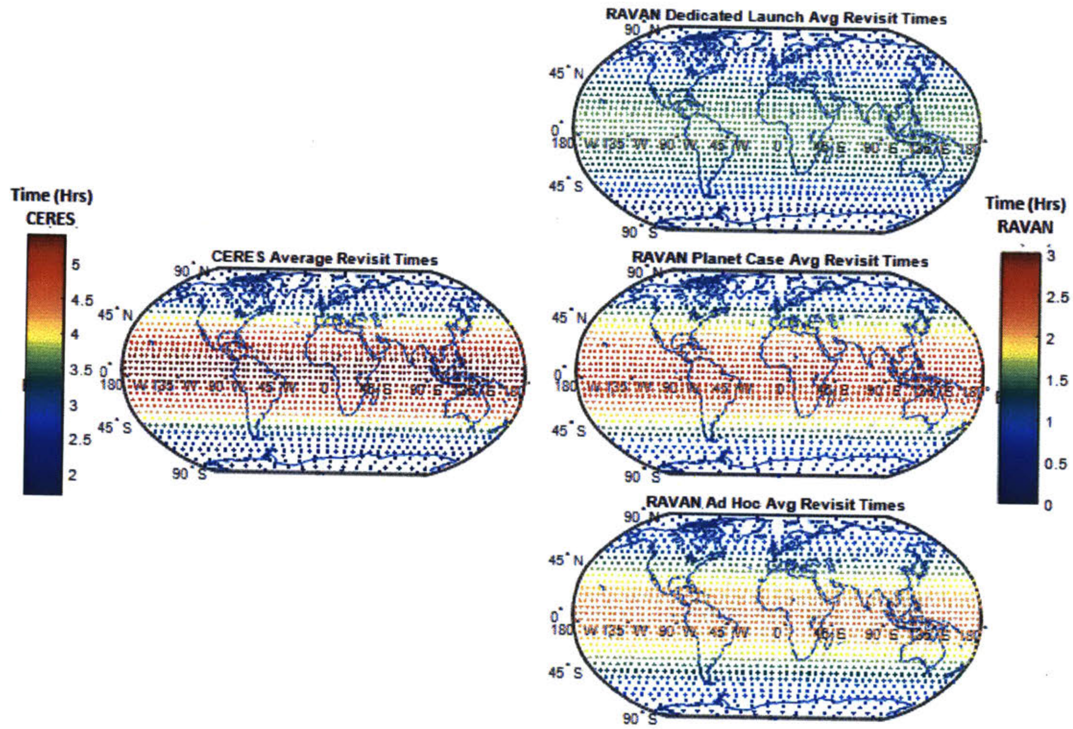

Figure 4-3: Left: CERES average revisit times. Right: RAVAN average revisit times for each case.

4.2.3 CrIS and CIRAS

CrIS and CIRAS do not see all points on Earth. Because these orbits are not perfectly polar and CrIS and CIRAS have relatively small fields of view compared to ATMS/MicroMAS-2 and CERES/RAVAN, they do not see locations at the very top of the poles in some configurations. CIRAS and CrIS have similar FOV with CrIS's just a couple of degrees larger. Since the sensor views are so similar, it is clear that having a constellation of three CIRAS CubeSats will have better revisit times than a single CrIS sensor as summarized in Table 4.3. One CIRAS CubeSat would most likely be sufficient to obtain comparable revisit times to the one CrIS sensor. However, the CIRAS CubeSat does not cover all the bands that CrIS does, multiple CubeSats would be needed to truly match CrIS's spectral capabilities.

Table 4.3: CIRAS and CrIS Revisit Results

	Revisit Time (Hrs)		
	Avg	Max	Min
CrIS	15.0	36.0	4.2
Dedicated Launch Case: CIRAS	5.5	11.7	1.5
Planet Case: CIRAS	5.5	19.9	6.0 sec
Ad Hoc Case: CIRAS	6.7	29.4	1.6

From Table 4.3 and Figure 4-4 show the Dedicated and Planet Cases provide similar revisit times for CIRAS. Of course, based on their architecture, again the Planet Case will provide a broader range of revisit times. Overall, it appears the averages between these two architectures are very similar with only a a few minutes difference in some regions. The Ad Hoc architecture however has revisit times near the equator that are at least an hour greater than those of the other cases. Still, this is better than CIRAS revisit times and so can still be considered to meet revisit requirements.

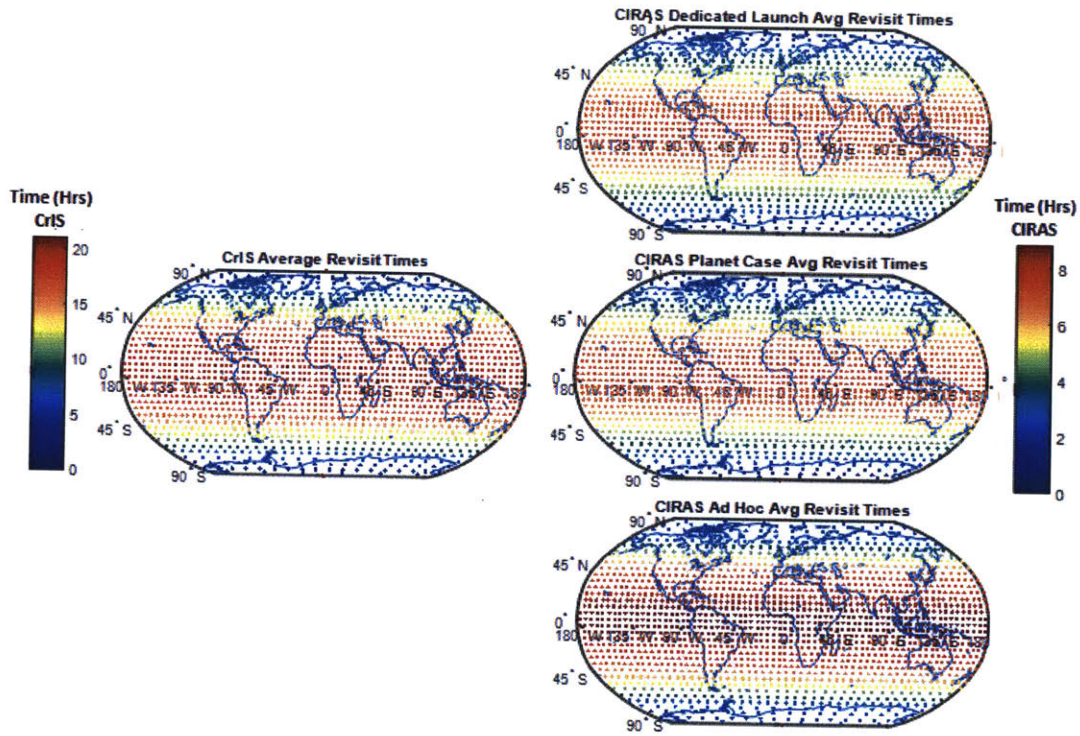

Figure 4-4: Left: CrIS average revisit times. Right: CIRAS average revisit times for each case.

4.2.4 OMPS and PICASSO

In comparing OMPS to PICASSO, the difference between the sizes of the sensors' FOVs becomes clear. Since PICASSO sees smaller areas, in these constellation configurations, it does not see all points on the Earth over the course of the simulation. The points it does not view are located at the poles because these architectures do not have perfectly polar orbits as can be seen in Figure 4-5. In all the cases presented here, the average revisit time is over 110 hours, or over 4.5 days. Because the revisit time is a requirement to be equivalent to this sensor, many additional PICASSO CubeSats would be necessary to meet the same requirements as OMPS. Here, the Launch and Planet Cases perform similarly while the Ad Hoc Case stands out as the lowest performer for almost all metrics.

Table 4.4: **PICASSO and OMPS Revisit Results**

	Revisit Time (Hrs)		
	Avg	Max	Min
OMPS	6.6	12.0	1.2
Dedicated Launch Case: PICASSO	4.6 days	14.1 days	24.1
Planet Case: PICASSO	4.6 days	13.4 days	4.8
Ad Hoc Case: PICASSO	5.5 days	20.8 days	6.4

Table 4.4 and Figure 4-5 show that PICASSO is not well suited to being used to observe the ozone layer at all parts of the atmosphere. Note that the scale for PICASSO in Figure 4-5 is in days (while OMPS is in hours) because of the length of these revisit times. Where OMPS has maximum revisit times of half a day, PICASSO has average revisit times of 4-5 days.

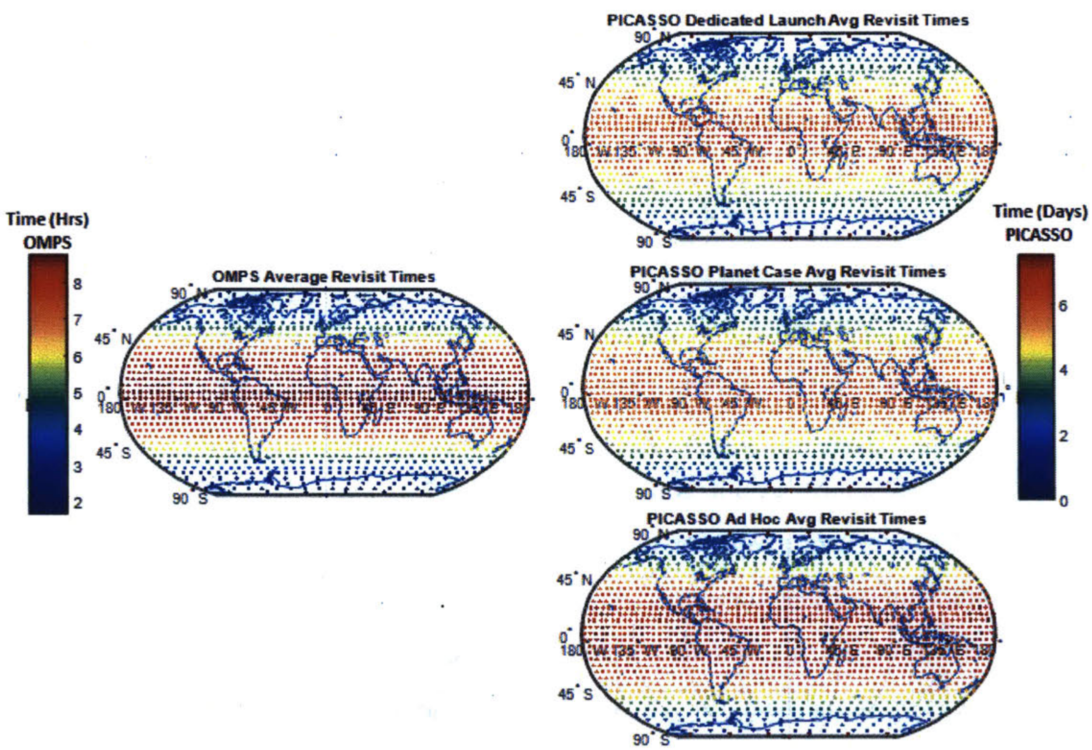

Figure 4-5: Left: OMPS average revisit times. Right: PICASSO average revisit times for each case.

For the PICASSO CubeSat to match the worldwide coverage of OMPS, it would need over 100 CubeSats. With even three PICASSO CubeSats in this constellation, revisit times are still on average between 4 and 5 days and in some locations can be upwards of 20 days. This is not useful in trying to give information on the UV index, a metric that OMPS can measure and is updated daily.

4.2.5 VIIRS and Dove

Dove has a much smaller aperture and field of view than VIIRS. This affects the metrics shown in Table 4.5 and means that Dove also does not points at the poles as can be seen in Figure 4-6. Table 4.5 shows how much longer the revisit times are using three Dove CubeSats as compared to one VIIRS sensor. As Planet, the maker of the Dove CubeSat notes, it would take 105 Dove CubeSats to achieve daily global coverage which is provided by the one VIIRS instrument.

Table 4.5: Dove and VIIRS Revisit Results

	Revisit Time (Hrs)		
	Avg	**Max**	**Min**
VIIRS	5.2	11.3	0.6
Dedicated Launch Case: Dove	5.9 days	30.8 days	64.6
Planet Case: Dove	4.0 days	14.1 days	0.8
Ad Hoc Case: Dove	5.2 days	23.9 days	8.7

Dove revisit times do not come close to VIIRS as expected and shown in Table 4.5 and Figure 4-6. For weather monitoring, revisit times on the average of 4-6 days is not sufficient to meet mission requirements. The three Dove CubeSats used here are not enough in terms of science data or revisit time to replace VIIRS. In Figure 4-6 it should be noted that the VIIRS scale is in hours and the Dove scale is in days to compensate for the extreme difference in revisit times.

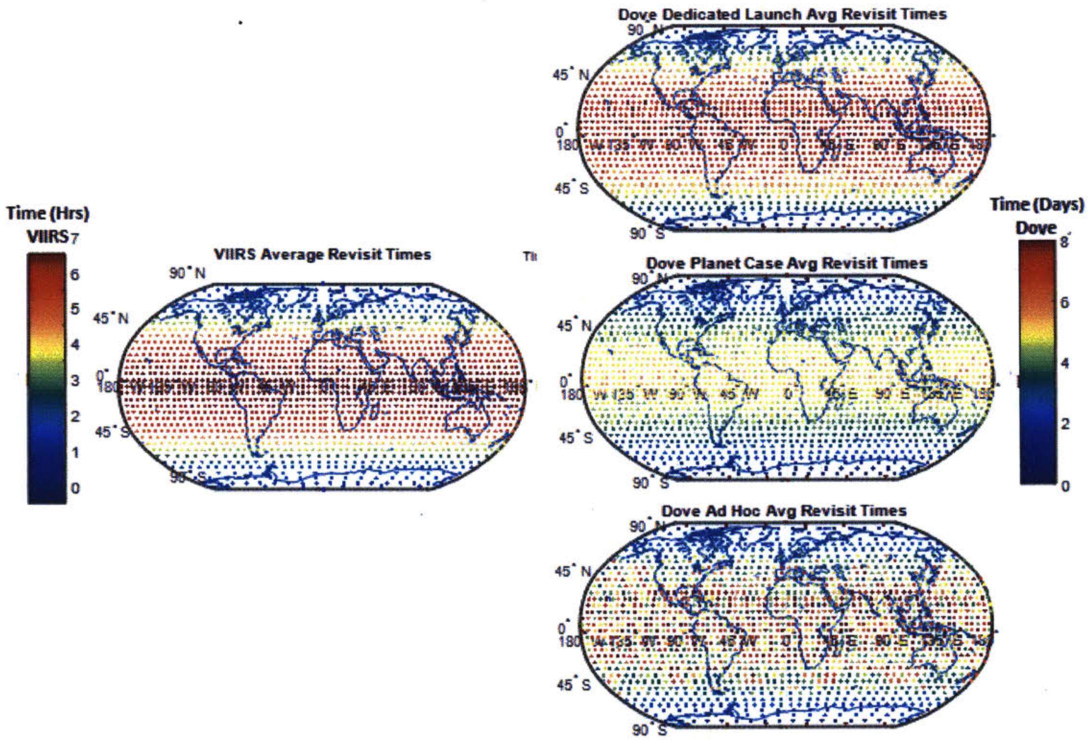

Figure 4-6: Left: VIIRS average revisit times. Right: Dove average revisit times for each case.

What is interesting in Figure 4-6, yet also somewhat expected, is that unlike any of the other sensors, Dove performs much better in the Planet Case. This makes sense as this is the constellation configuration in which they are intended to be used. This is the configuration into which Planet launches all of its Flock constellations. For some sensors, the single mass launch may provide the best constellation architecture and not just be a cost reduction method.

Chapter 5

Conclusions

5.1 Summary

During the implementation of the JPSS mission, there have been many delays, as is common in large satellite missions. These delays turned what was supposed to be a demonstration mission, S-NPP, into an operational mission. More delays have created a gap between the end of S-NPP's design lifetime and the launch of its replacement, JPSS-1. While there has not yet been a gap, this is still a large risk that could result in the loss of 45% of weather data. This could be mitigated by using CubeSat constellations as gap fillers.

This thesis examines the specific case of using CubeSats in a constellation to supplement a traditional satellite, JPSS-1. The three constellation architectures studied were: (i) Dedicated Launch Case, (ii) Planet Case, and (iii) Ad Hoc Case. The results from this simulation show that there are some CubeSat sensors that can be considered comparable to the JPSS-1 instruments, there are sill many gaps and limitations. Most notably these gaps exist with near IR and IR CubeSat imagers as well as ozone measuring CubeSats.

Of the five sensors on JPSS-1, ATMS, CERES, CrIS, OMPS, and VIIRS, three of them have CubeSat analogs with reasonably similar capability. Weather sensing through the microwave radiometry by MicroMAS-2, Earth's emitted radiation measurements from RAVAN, and atmospheric sensing through CIRAS all compare with their mapped sensors, ATMS, CERES, and CrIS, respectively. The difference in the quality of data from these systems comes in the number of bands measured by the sensor, which could be addressed by developing separate sensors that cover the missing bands hosted on additional CubeSats. Moving up to a larger CubeSat platform (such as to a 6U from a 3U) and increasing the instrument SWaP of the satellite could improve the ability of a CubeSat to host a completely comparable sensor. While MicroMAS-2, RAVAN, and CIRAS do not give as much data as the heritage JPSS-1 sensors, they give quality data that can be used as a supplement to the JPSS sensors, a gap filler between JPSS installments or in the event of a failure.

However, for any sensor type where aperture size or power requirements are the limiting factors on the system, CubeSats will always be at a disadvantage. The JPSS-1 imager, VIIRS, creates images that CubeSats are unlikely to be able to replicate without deployables or a larger form factor because of its large aperture and amount of available power. As a supplement to heritage sensors and satellites, they can provide useful information, but they are not yet a replacement.

The work presented here shows that for about 4.5% of the cost of the JPSS-1 satellite, a constellation of fifteen CubeSats can be designed, built, and launched. While these CubeSats are limited in their capability and cannot produce the same amount and type of data as the JPSS-1 sensors, they serve well as supplements to the traditional satellite creating comparable quality of weather data. While this does not make them ideal as a permanent replacement for heritage sensors, they can augment the availability of weather data in the event of a delay or failure in traditional satellites.

5.2 Simulation Considerations

There are many assumptions made in this work that could be further explored or revised. The simulation could be refined to make this a more realistic representation of the operations of a CubeSat constellation. For example, one way of adjusting this simulation would be to choose the number of each type of CubeSat based on requirements for revisit time. This would lower the number of CubeSats with larger fields of view, or where higher revisit time are not necessary, and increase the number of CubeSat types where faster revisit times are required. This would optimize the constellation to be closer to the requirements of the large satellite that is being supplemented by the constellation.

Additional work could be done to better design a CubeSat constellation to match the capabilities of JPSS-1. Exactly one CubeSat per JPSS-1 sensor is not a perfect comparison since for most JPSS-1 sensors, multiple CubeSats are necessary to provide all the same capabilities. For even the best performing of the CubeSats, MicroMAS-2, RAVAN, and CIRAS, one CubeSat was not enough to cover all spectral bands provided by the JPSS-1 sensors. At least one more CubeSat would have to be designed for each sensor to include all the spectral bands that JPSS does.

It may be that with current technology limitations, JPSS-1 sensors should be best fit on a 6U or 12U CubeSat. The larger form factor would be more appropriate to housing larger sensors and providing the necessary resources. These do not quite exist yet, although the 6U CIRAS is a good first step. Technology development funding directed towards these larger CubeSats would help promote advanced sensor development on CubeSats. Until this point where a CubeSat can host a JPSS-1 sensor, a beneficial study that could show the usefulness of these CubeSat sensors would be to use science data quality metrics to help determine what the value is of the investment in the CubeSat constellation.

The operations of CubeSats could also be treated in more detail, rather than making assumptions about always-on operations, which are not accurate. For example, power generation abilities of CubeSats have been mentioned multiple times as a limiting factor for the performance of CubeSat systems. All simulation in this thesis assumed that the CubeSat payloads would be available at any time to make measurements. In reality, this is not the case. On CubeSats, most science gathering cannot be completed on solar power alone and will draw the necessary remaining power from the battery. After collecting data, time is required to charge the battery before another science data gathering session can take place, limiting the amount of measurements than can be made. Development of CubeSat power generation technologies would help to make fitting more advanced sensors on the CubeSat frame.

Ground and satellite to satellite communications are another area that should be treated in more detail. Scheduling ground communications with these CubeSats can be complicated with so many in orbit. If they are close together, as they are in the first case, this means that the ground station must be capable of supporting multiple access. Creating the schedule for each CubeSat, when it is to make measurements, when it is to send down data, is necessary for smooth operations and data acquisition. However, communications with the satellites are also power intensive, limiting how quickly science can be gathered after a ground contact. Satellite to satellite communications can optimize scheduling of when scans should occur and which CubeSats should complete them, but this is also power intensive, limiting use of payloads. A simulation that takes into account the timing of ground contact and the power state of the CubeSat could give higher fidelity estimates for revisit times [75]. Two algorithms by *Kennedy et al.* that have been developed for this type of mission planning based on resource constraints are the Resource-Aware SmallSat planner and the Limited Communications Constellation [75]. The application of these algorithms to this study could give better information as to realistic revisit times for these CubeSats.

5.3 Future Considerations

This thesis highlights the areas in which CubeSat development is necessary to compare to heritage sensors. Specifically, work needs to be done to develop a high resolution IR and near IR imager for a CubeSat to even begin to match the capabilities of the VIIRS sensor. Additionally, almost no development has been done on ozone sensing CubeSats. There is a wide field that has yet to be explored in using CubeSats to study the ozone layer along the same bands that OMPS uses. The development of these fields along with improvement to the current ones may someday allow a CubeSat constellation to match the full capabilities of a larger multi-sensored satellite.

Demonstrations of CubeSat constellations, like those studied here, will show how inexpensive CubeSat constellations can be used to provide fast revisit times, comparable data, and redundancy for filling gaps in weather data. Planet's constellation of over 100 on-orbit satellites that provide daily imagery of the entire land mass of the Earth is a notable first step to having large, but capable, CubeSat constellations of Earth observing instruments. The upcoming TROPICS constellation of CubeSats, which are intended to be similar to MicroMAS-2 will provide additional insight about how a CubeSat constellation with an instrument based on the heritage instrument ATMS from JPSS can compare with ATMS data. The RAVAN CubeSat project aims at creating a constellation that will map the Earth's radiation budget, similar to CERES. Projects like these pave the way for new uses of CubeSat constellations that can provide protection and redundancy for US space assets.

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Appendix A

Cost Analysis Values

Item	Quantity	Time	Number of CubeSats	Cost		Total (mil)	% of Budget
Case 1							
Design/Build Staff	11 per Type	3 years	5 CubeSat Types	$200,000	per person, year	$39.60	9.14%
Data Analysis Staff	2 per Type	2 years	5 CubeSat Types	$200,000	per person, year	$19.20	4.43%
Hardware	1 per CubeSat		15 Total CubeSats	$1,000,000	per CubeSat	$72.00	16.62%
Operators	2 per CubeSat	500 days	15 Total CubeSats	$200	per person, year	$14.40	3.32%
Launch	3 total		12 CubeSats	$20,000,000	per launch	$288.00	66.48%
						TOTAL COST:	$433.20
						% of JPSS Cost:	11.50088%
Cases 2 and 3							
Design/Build Staff	11 per Type	3 years	5 CubeSat Types	$200,000	per person, year	$39.60	23.29%
Data Analysis Staff	2 per Type	2 years	5 CubeSat Types	$200,000	per person, year	$19.20	11.29%
Hardware	1 per CubeSat		15 Total CubeSats	$1,000,000	per CubeSat	$72.00	42.34%
Operators	2 per CubeSat	500 days	15 Total CubeSats	$200	per person, year	$14.40	8.47%
Launch	1 per 3U CubeSat		12 3U CubeSats	$295,000	per 3U Cubesat	$24.84	14.61%
	1 per 6U CubeSat		3 6U CubeSats	$545,000	per 6U CubeSat		
						TOTAL COST:	$170.04
						% of JPSS Cost:	4.51434%

Figure A-1: The spreadsheet showing all values used in the CubeSat constellation cost analysis described in Section 3.3.