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THESIS

SMALLER SATELLITE OPERATIONS NEAR GEOSTATIONARY ORBIT

by

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September 2007

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With the ongoing miniaturization of components, the utility of smaller satellites is increasing. Many believe in the near future that small satellites will be able to perform all functions that larger satellites currently perform today. It has been suggested that these satellites will be less expensive, thus offer a lower risk to the consumer in case they fail before their mission design life. This paper looked at the ability to build and operate smaller satellites with current technology to perform covert Space Control and Space Situational Awareness missions near geostationary orbit. The investigation determined if space qualified Commercial Off The Shelf (COTS) components and current technology could be used to build covert smaller satellites. The largest satellite was sized to be undetectable from earth based sensors. Subsequent CubeSat sizes were selected to determine how small a satellite could be built with COTS components and current technology to perform the assigned missions. A comparative analysis was then performed to determine how these satellites could be cost effectively launched to orbit. A cost estimate was performed to determine the entire life cycle cost for each satellite size excluding launch and integration segments. Using that information, the best satellite size was determined.

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SMALLER SATELLITE OPERATIONS NEAR GEOSTATIONARY ORBIT

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Submitted in partial fulfillment of the
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ABSTRACT

With the ongoing miniaturization of components, the utility of smaller satellites is increasing. Many believe in the near future that small satellites will be able to perform all functions that larger satellites currently perform today. It has been suggested that these satellites will be less expensive, thus offer a lower risk to the consumer in case they fail before their mission design life. This paper looked at the ability to build and operate smaller satellites with current technology to perform covert Space Control and Space Situational Awareness missions near geostationary orbit. The investigation determined if space qualified Commercial Off The Shelf (COTS) components and current technology could be used to build covert smaller satellites. The largest satellite was sized to be undetectable from earth based sensors. Subsequent CubeSat sizes were selected to determine how small a satellite could be built with COTS components and current technology to perform the assigned missions. A comparative analysis was then performed to determine how these satellites could be cost effectively launched to orbit. A cost estimate was performed to determine the entire life cycle cost for each satellite size excluding launch and integration segments. Using that information, the best satellite size was determined.

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I. INTRODUCTION

With the continual miniaturization of components, satellite sizes also continue to decrease. Some missions that are currently conducted by larger satellites may now be able to be performed by smaller sized satellites. Table 1 lists the categories of satellites; each type is characterized by their overall mass. Although smaller sized satellites have been proposed as a viable tool to perform operations at all orbital regimes, to date these smaller satellite types have predominately been operated in Low Earth Orbit (LEO). Their practical use to perform certain missions may also extend throughout all orbital regimes around earth to include all inclinations and altitudes greater than geostationary orbit.

Category	**Mass range [kg]**
large satellite	>1,000
medium-sized satellite	500-1,000
minisatellite	100-500
microsatellite	10-100
nanosatellite	1-10
picosatellite	0.1-1
femtosatellite	<0.1

Table 1. Satellite sizes categorized by mass.[1]

Few seem to remember that the use of smaller satellites was mandated due to the limited carrying capacity of rockets during the beginning of the space race. In 1994, well after rocket technology had matured and was able to deliver large satellites all the way out to geostationary orbit, the Naval Research Laboratory built the components, manufactured, integrated and then operated the minisatellite Clementine. At that time when “bigger was considered better”, Clementine was built and orbiting the moon within 22 months. A timeframe that was unheard of then and still is today. Not only was the

[1] “satellite mass categories sizes.” 12 July 2007. 12 July 2007. <http://www.daviddarling.info/encyclopedia/S/satellite_mass_categories.html>.

mission successful at mapping the moon, but it also cost the United States Government (USG) a fraction of the cost of normal extra-planetary missions which use much larger and more expensive spacecraft.

Clementine is an example of a successful mission that utilized a smaller spacecraft to perform the assigned mission successfully at a reasonable cost. In the recent past the Air Force (AF) has funded Experimental Satellite System (XSS) 10 and 11. Each experiment was encompassed in a smaller satellite. Under the AF, the Air Force Research Laboratory (AFRL) has run each of these programs to further research and to develop technology so that these types of satellites could be used "to conduct "proximity operations," maneuvers around other satellites. Some have said the XSS satellites could be used to inspect, service, or attack other satellites."[2] To this extent, it is reasonable to assume that smaller satellites are a viable option to perform certain missions that are essential to fulfilling US Space Control Requirements.

One particular mission involves Space Situational Awareness (SSA), "which provides the foundation of Space Superiority."[3] The USG currently has no means for space object identification that can see what systems and their physical characteristics that are stationed in the geostationary belt. Rumors derived from open source publications have stated that China has placed small satellites near satellites that the USG considers vital to it national security. The assumed mission for these small satellites is to neutralize the USG's high value satellites when they are directed. With current ground observing radars and optical systems, the smallest non-mirrored object that can be identified by long dwell imaging is greater than one half meter in any dimension. If the USG had a spacecraft that could drift through the geostationary belt, then the USG would be able to observe these satellites at a resolution that would not only allow the detection of these satellites, but also the ability to classify the satellite's payload, mission and possibly secondary missions.

[2] Hui Zhang. Action/Reaction: U.S. Space Weaponization and China. December 2005. 20 September 2007. <http://www.armscontrol.org/act/2005_12/Dec-cvr.asp>.

[3] John Brock. Operational Utility of Small Satellites. SAB Summer Session, 28 June 2007. Slide 13.

Figure 1 depicts a view from earth that was collected using long dwell imagery techniques. Figure 1 illustrates the problem of identifying smaller objects from earth that are in or near the geostationary belt. The picture was taken over seven hours and forty minutes. In that time the observer was able to collect enough reflected light from the sun off of each satellite labeled to be able to clearly identify it. The imaged captured a portion of the geostationary belt beginning with Galaxy 13 stationed at 127.0 degrees West Longitude all the way to Galaxy 3C stationed at 95.0 degrees West Longitude. This image captures 11.25 percent of the entire geostationary belt. Even the objects that are in a geosynchronous orbit can be seen as if they are still in the night's sky. In the image, background stars appear as streaks due to earth rotating about its axis, while the geostationary satellites appear clear and distinct. This photograph was taken with a small telescope, utilizing long dwell imagery techniques on a cloudless night. It shows how easy it can be to see large satellites covered in reflective thermal insulation and other reflective surfaces. Although it is easy to image large reflective satellites from earth, features of the satellites are not captured. Knowing satellite positions are very important, but determining their capabilities is vital to SSA. To accomplish this imaging with a fine enough resolution to identify satellite components is crucial

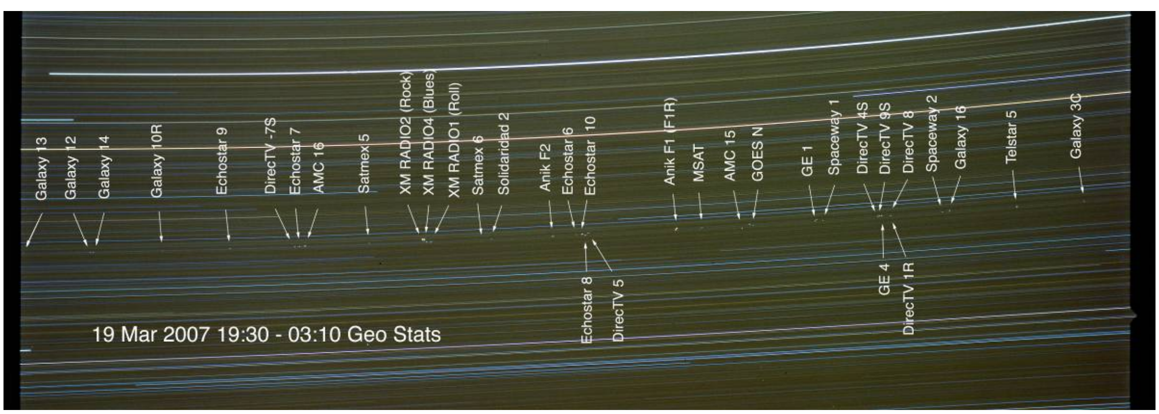

Figure 1. Time delayed photograph of a portion of the geostationary belt as viewed from earth with satellites labeled.[4]

[4] David Dolling. "Earth Science Picture of the Day, Geostationary Satellites." 9 April 2007. 17 June 2007. < http://epod.usra.edu/archive/epodviewer.php3?oid=379872>.

If a satellite could drift through the geostationary belt, it could easily observe the fine detail of each satellite while relaying its captured image back to its ground controllers. With such information, each object in geostationary orbit could be accurately imaged, cataloged and monitored. No nation, rogue state, or terrorist organization could place something into that orbit without the USG knowing about it. A satellite that drifted through the geostationary belt would be the perfect solution to the SSA mission of interest.

A smaller satellite operating in this same manner would also be able to fulfill another attractive mission that involves Space Control. A smaller satellite operating in close proximity to geostationary orbit should be able to effectively jam or inject false signals into any geostationary communications satellite. If the smaller satellite could jam the communication satellite at a high enough power, it would also have the ability to damage sensitive receiver equipment on the communications satellite. This ability could effectively neutralize the targeted satellite's receiving capability at that frequency band. These missions would allow the USG to disrupt, deny, deceive and possibly destroy enemy satellite communications relayed or broadcast from geostationary orbit.

Unfortunately a satellite can not merely drift through the geostationary belt; it would have to use considerable propulsion utilizing numerous orbit transfers to move about. A satellite can however pass by the other satellites in the geostationary belt if it has an altitude that is slightly lower or higher than 35,776.9 kilometers. If a satellite is at a lower altitude, it will orbit the earth slightly faster than the satellites in the geostationary belt, and if the satellite is higher it will orbit the earth slower. Another way to look at it is, that if a satellite is at a lower altitude, then it will seem to overtake the satellites in the geostationary belt, whereas a satellite at a higher altitude will appear to be overtaken by the geostationary belt satellites. Of these two choices, the lower altitude is more appealing since it will allow a satellite with an optical payload to capture images of the targeted satellite's payloads that are pointing toward the earth at the closest point of approach (CPA) between the two satellites. A sub-geostationary altitude will also allow the satellite to capture an image of the target's side sections with images captured during the approach and departure from the CPA. With these images, USG personnel will be able to accurately characterize each satellite in the geostationary belt.

One aspect that must be considered is that not all satellites are actually located exactly in the geostationary belt; many have drifted into what is referred to as a geosynchronous orbit. This orbit has the same period as the geostationary belt satellites, but they do not have an inclination of zero degrees. To this extent, a single satellite may not be able to reach a near enough CPA with these satellites to capture any images at a resolution that could be used to characterize these satellites. Not being able to image satellites in geosynchronous orbit may force the design of special orbits for single high interest satellites to allow these satellites to be imaged properly.

If a satellite could be manufactured small enough without jeopardizing the function of its payload, then it would likely be undetectable by known, current ground based surveillance systems. To further enhance this capability, the smaller satellite could employ techniques such as using low-reflectance materials and onboard Attitude Determination and Control System (ADCS) algorithms that will orient the satellite to avoid reflecting sunlight back to earth. Therefore covert use of a smaller satellite such as a microsatellite, or even a nanosatellite near geostationary orbit is a very attractive means to perform sensitive missions that must be conducted near the geostationary belt. At a relatively low cost to produce, the potential to develop satellites of these sizes is very attractive and lends itself to the idea of producing fleets of these vehicles. The large number of smaller satellites per fleet constructed would make up for the assumed lower reliability of these satellites and would be less of a concern in the event of a satellite failure.

Smaller satellites could be designed to operate using current USG ground control facilities and software. To further enhance their utility and cost savings; they could be made to operate as autonomously as USG officials felt comfortable. The greater the autonomy, the smaller their support workforce would need to be. These two features have the potential to keep the costs much lower for a program that would operate utilizing smaller satellites.

Opponents to such a deployment of these smaller satellites assert that the satellites lack sufficient redundancy to operate for any predictable amount of time on orbit. They also claim that these satellites do not have the capability to perform any mission at the standard that is required. These satellites cannot be hardened sufficiently to prevent single event upsets (SEUs) or micro-meteorite impacts. There is also no standard method that is currently utilized to deliver smaller satellites to a near geostationary orbit. For these reasons, many space professionals doubt that these smaller sized satellites will actually have a viable operational role at geostationary orbit.

In this thesis, I will explore through first level spacecraft design and Satellite Tool Kit (STK) simulations the viability of smaller satellite operations near geostationary orbit. I will examine the plausibility and practicality of employing a smaller satellite to perform various attractive missions at an altitude slightly lower than geostationary orbit. I will design a Half-Meter-Cube, a 5U-CubeSat and a 1U CubeSat using Commercial-Off-The-Shelf (COTS) components to the fullest extent possible. Each satellite will be designed to perform an optical survey mission. In these designs, I will try to incorporate the most capability into each satellite as possible. After determining the capabilities of these satellites, I will use simulations in STK to determine the degree to which these missions may be applicable to the capabilities of these satellites. I will also discuss the potential that these types of satellites may have to perform a service denial mission and a satellite component neutralization mission targeting geostationary satellites.

This thesis is theoretical in nature. Analytical calculations, simulations utilizing STK along with an examination of scientific literature, are the main research methodologies. Analysis will be mathematical in nature concentrating on the laws of physics. Once a basic approach has determined if this approach is indeed possible, I will continue with a more refined approach to the topic. Once this approach has confirmed that this application is indeed possible, I will approach applicable questions concentrating on satellite and constellation properties, CONOPS, financial, and then US moral reservations of the possible use of this application. When discussing these areas I will also apply a common sense approach to areas of that study when a purely mathematical approach is not possible or too cumbersome to emphasize a point clearly. I will approach the discussion with the assumption that the reader has a basic understanding of mathematics, physics and basic satellite operations principles.

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II. PAST SMALLER SATELLITE DESIGNS

A. CLEMENTINE

Construction began in 1992 at the Naval Research Laboratory (NRL) on the first United States mini-spacecraft, designed to map the moon. Twenty-two months later, the octagonal prism shaped spacecraft was launched. Clementine measured 1.8 meters in height and 1.14 meters in width. It had a mass of 227 kilograms and a nominal operating power of 360 watts. The mini-spacecraft “mapped 100% of the lunar surface in 11 spectral bands with greater than 99% coverage.”[5] In less than two months, from a lunar orbit, Clementine transmitted more than 1.8 million images of the moon’s surface. Unfortunately a malfunction caused the spacecraft to prematurely expend the onboard fuel supply preventing completion of its secondary mission to pass within 100 kilometers of the asteroid Geographos. Subsequent to the malfunction, the spacecraft was placed within the Van Allen radiation belts to test the effects that increased radiation would have on the spacecraft’s components. During the mission, the spacecraft qualified 23 advanced lightweight technologies for spaceflight.

Clementine remains one of the shining examples of how a mission can be performed faster and inexpensive by a smaller spacecraft if a level of risk is allowed by the program’s managers. In addition to its technical success, the public supported the mission and was amazed at the images produced by the spacecraft. Working with NASA, NRL increased the public support for the mission by releasing all imagery of the moon obtained by the spacecraft. To this day, NRL maintain a database of moon imagery similar to a Google Earth (TM of Google) that is accessible to the public from NRL’s website for Clementine.

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[5] J. Schaub. “Clementine The Deep Space Program Science Experiment Advanced Technology Demonstration.” 25 July 2007. 25 July 2007. <http://code8200.nrl.navy.mil/clementine.html.>

B. XSS-10

The Experimental Satellite System (XSS) program began in 1997 when Boeing was awarded the contract under a project funded by the Air Force Research Laboratory (AFRL). XSS-10 was planned as the first in a series of very small satellites that would eventually lead to the development of microsatellites used "for inspection, rendezvous, and docking and close-up maneuvering around other space objects."[6] This satellite's mission requirements were to when ordered semi-autonomously rendezvous with another object in low earth orbit. During the satellite's rendezvous and following proximity operations it continuously relayed imagery to its ground station through the AFSCN.[7] XSS-10 was the first project to take on such a rendezvous mission with an unmanned spacecraft. The spacecraft's maximum dimensions were constrained by the excess volume and mass available as a secondary payload onboard a Delta II rocket, sharing a ride with a Global Positioning System (GPS) satellite.

To satisfy all mission requirements, XSS-10 had a "lightweight propulsion system; guidance, navigation & control (GNC); miniaturized communications system; primary lithium polymer batteries; integrated camera and star sensor."[8] Only three years after the project began, "Boeing's Space and Intelligence Systems and Rocketdyne Propulsion and Power divisions designed, developed and built the 31-kilogram (68-pound) spacecraft[.]"[9] XSS-10 was integrated into the Delta II launch vehicle in September 2001, awaiting a launch date. The spacecraft was launched on January 29th, 2003 from Cape Canaveral Air Force Station, Fla.

After a few hours on orbit the micro-satellite began operations. Throughout the mission the microsatellite streamed live video from an onboard camera to the ground

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6 "XSS-10 Micro Satellite", Fact Sheet. February 2005. 20 August 2007. <http://www.kirtland.af.mil/shared/media/document/AFD-070404-107.pdf>.

7 Thomas M. Davis. "XSS-10 Micro Satellite Flight Demonstration." 2005 Space Systems Engineering Conference. 11 October 2005. 5.

8 "XSS-10 Micro Satellite", Fact Sheet. February 2005. 20 August 2007. <http://www.kirtland.af.mil/shared/media/document/AFD-070404-107.pdf>.

9 Boeing Demonstrates Capabilities of Micro-Satellite. Satellite Today. Potomac: Feb 5, 2003. Vol. 2, Iss. 18; 1.

control center. During the first 12-hour test mission, it "traveled within 100 meters (328 feet) of the second-stage booster of the Delta II rocket to take photographs and transmit the images back to ground from a low-Earth orbital position 800 kilometers (497 miles) above the equator."[10] During the second 12 hour period of operations, additional operations, requiring more demanding maneuver control were attempted, allowing the microsatellite to travel closer to its target. Unfortunately, communications with XSS-10 were lost while the microsatellite was performing a close survey of its target. XSS-10 was later determined that an onboard guidance error caused the spacecraft to collide with the Delta II rocket's second stage ending the spacecraft's mission.

> XSS-10's greatest accomplishment was the development of microsatellite hardware, software and operations procedures for the autonomous inspection of residence space objects. This included a number of firsts. Perhaps the most noteworthy of the operational firsts was demonstration of a relative navigation scheme for close-in inspection based on camera-derived RSO [Resident Space Object] centroid information.[11]

XSS-10 mission provided ground breaking technology to build future XSS missions upon and the confidence that these types of missions could be performed with even smaller satellites. The success of the project generated excitement in the military and aerospace sectors but was met with mixed feelings in the civilian media. Many reports echo this comment made by Bruce DeBlois, "In January 2003, the U.S. Air Force demonstrated its XSS-10 microsatellite, which repeatedly maneuvered to within 35 meters of a target to take photographs. Had it been equipped with a gun instead of a camera, it could have destroyed the target."[12] His comments show the way with which this mission was viewed by opponents to the weaponization of space. Even with mixed opinions from the public about the relevance of this mission, all concerned parties were pleased with the mission's overall results. These accomplishments became the foundation for the next mission to build upon.

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[10] Boeing Demonstrates Capabilities of Micro-Satellite. Satellite Today. Potomac: Feb 5, 2003. Vol. 2, Iss. 18; 1.

[11] Thomas M. Davis. "XSS-10 Micro Satellite Flight Demonstration." 2005 Space Systems Engineering Conference. 11 October 2005. 17.

[12] Bruce DeBlois. IEEE Spectrum Star-Crossed. June 2004. 18 September. <http://www.spectrum.ieee.org/print/1585.>

C. XSS-11

In 2001, with the assumed success of XSS-10, AFRL drafted a more ambitious set of requirements with a shorter timeline for XSS-11. This microsatellite would be required to conduct rendezvous and close-proximity operations with semi-autonomous guidance. After certain milestones were accomplished the microsatellite would carry out its mission with fully autonomy. Not only was it required to observe its spent launch vehicle's upper stage, but following those operations, it was required to continue through its orbit to conduct rendezvous operations with other objects. At the time, this was considered a very difficult task, due to the complexity involved with creating computer code to autonomously perform rendezvous and proximity missions. To complete these requirements the microsatellite required a propulsion system with enough propellant to change orbital altitude and planes around LEO. The maneuvers required the spacecraft to be slightly larger than its predecessor, but still remain in the microsatellite category. The design and construction had to be accomplished in less than four years with a 21 million dollar budget for the microsatellite. In August 2001, the contract was awarded to Lockheed-Martin and a dedicated Minotaur-1 launch vehicle was selected to launch the microsatellite out of Vandenberg Air Force base in California.

Before the satellite was even launched the media proposed the actions surreptitious purpose of XSS-11;

> designed for "rendezvous and proximity operations"—that is, meeting with other satellites to perform inspections, maintenance, and the like. However, as an unnamed U.S. defense official candidly acknowledged in an interview with Inside the Pentagon in December 2003, the XSS-11 could also be used as an antisatellite weapon.[13]

Comments critical of the mission were common place throughout the satellite's development and even still to this day.

The microsatellite was launched on April 11, 2005. After separating from its Minotaur launch vehicle, XSS-11 began its mission. Within the first few hours, it

[13] Bruce DeBlois. IEEE Spectrum Star-Crossed. June 2004. 18 September. <http://www.spectrum.ieee.org/print/1585>.

successfully rendezvoused and began proximity operations about the Minotaur I upper stage. "As of fall 2005, it has accomplished more than 75 natural motion circumnavigations of the expended launch vehicle. During its projected 12 to 18-month flight, the spacecraft will conduct rendezvous and proximity maneuvers with several US-owned, dead or inactive resident space objects near its orbit, as well as will exhibit more autonomy as the project continues."[14] These projected operations were successfully accomplished. As of fall 2006, the satellite had rendezvoused with at least three other orbiting objects before the mission was terminated due to re-entry fuel requirements.

After the mission was deemed successful, public debate intensified over the use of such technology, due to its potential use as a co-orbital anti-satellite. The following excerpt is from a citation that typifies the sediment between the military and public opinion:

> But that short preparation time and zero-g agility also could make the microsatellites ideal weapons for disabling other countries' orbiters, note Pentagon space critics, including Theresa Kitchens, vice president of the Center for Defense Information. XSS-11's predecessor was an experimental missile defense satellite called Clementine 2. "That history makes me suspicious," Hitchens says. In the 2004 report titled "Counterspace Operations," the Air Force declared that the "freedom to attack, denying space capability to the adversary" has become a "crucial first step in any military operation." The Defense Department plans to spend about $10 million over four years to develop small satellite payloads that could take out other orbiters. The Air Force says the XSS-11 itself "is not a weapon and it has no military mission or application." Hitchens agrees that the "current experiments are benign." It's the future potential of the mini sat that has caught her attention. To which [Harold] Baker [XSS-11 program manager at the Air Force Research Lab] replies, "Name me a technology that can't be used for the military somehow.[15]

As the citation suggests, XSS-11 demonstrated capabilities that have elevated the interest of those resistant to the USG's Space Control program. However, all rendezvous and servicing capabilities are readily extended to space control missions.

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[14] "XSS-11 Micro Satellite", Fact Sheet. December 2005. 20 August 2007. <http://www.kirtland.af.mil/shared/media/document/AFD-070404-108.pdf>.

[15] Noah Shachtman. "Smaller, Smarter Satellites Spark Debate." Popular Mechanics. New York: Jul 2005. Vol. 182, Iss. 7; 29.

As of January 2006, the total budget for the project topped 80 million dollars including the launch vehicle and the cost to operate the microsatellite. The experiment successfully met all requirements and validated technology which enables autonomous rendezvous and close proximity operations between objects orbiting earth. From the successful results of this experiment, I will assume this same technology will allow satellites to perform similar maneuvers in geostationary orbit.

D. ORBITAL EXPRESS

The Orbital Express (OE) mission was a joint effort between the Defense Advanced Research Projects Agency (DARPA), NASA and the Air Force. The program's mission was to demonstrate autonomous satellite servicing techniques between two cooperative spacecraft operating in LEO. To accomplish this mission the program kicked off in 1999 and was able to gather lessons learned from XSS-10, XSS-11 and NASA'a failed Demonstration of Autonomous Rendezvous Technology (DART) mission.

The program began in 1999 by investigating "robotic technologies enabling the on-orbit upgrade of electronics and refueling and reconfiguration of satellites, both military and commercial at low and high orbits."[16] By the end of 2000 OE had gained enough momentum to fund technology demonstrations by many leading aerospace developers. By the end of summer of 2001, "Spectrum Astro, BAE Systems and Boeing each received contracts worth about $6 million for the first phase, the study and analysis period, of the Orbital Express program."[17] In March 2002, DARPA awarded Boeing the 99 million dollar contract to complete the second phase of the program building the Autonomous Space Transport Robotic Operations (Astro) and NextSAT. Boeing then choose to partner with Ball Aerospace; Northrop Grumman Space Technology; MacDonald, Dettwiler and Associates; Charles Stark Draper Laboratory; and Starsys Research to built and integrate the satellites for the program. On April 4th, 2006, Boeing

[16] Bryan Bender. "DARPA kickstarts R&D on sensors, space robotics." Jane's Defence Weekly. Horley: Nov 10, 1999. Vol. 032, Iss. 019, 1

[17] Robert Wall. "Darpa Pursues Refueling, Electronic Upgrades for Sats." Aviation Week & Space Technology. New York: May 14, 2001. Vol. 154, Iss. 20, 80.

announced that it had completed the autonomous rendezvous and docking milestones through laboratory based testing. Later that year, Ball Aerospace delivered NextSAT to Boeing for final testing and integration. By the end of 2006, Boeing was involved in the integration of the OE mission stack into the ATLAS V 401 expendable launch vehicle being utilized for the Space Test Program One (STP-1) launch. The OE mission was launched on March 8th, 2007 out of the Cape Canaveral Air Force Station in Florida.

As soon as OE was delivered to orbit it ran into problems. One of the reaction wheels in Astro was coded in the opposite direction than the geometry used for the spacecraft's ADCS. This caused Astro to steer it body in an orientation that was away from the sun. Over two progressive orbits the Boeing team frantically tried to figure out the problem, to no avail. The Ball Aerospace team was then allowed to try to steer the entire stack with NextSAT's reaction wheels although the reaction wheels were sized to control that spacecraft alone, not the entire "stack". To the relief of the program's members, NextSAT was able to properly orient the "stack" so that both NextSAT and Astro could charge their batteries allowing the mission to continue. The coding error was identified and corrected a few weeks later. With both satellites performing properly the stack was separated on May 5th, beginning the OE mission.

> [The mission] went on to conduct numerous automated rendezvous and docking maneuvers using a three-fingered capture mechanism and a relative navigation system consisting of infrared and optical cameras keying off retro-reflector targets on NextSAT. During the mission, the two spacecraft also demonstrated component swapping in which a robotic arm on Astro built by MacDonald, Dettwiler and Associates Ltd. of Canada passed a battery capable of powering NextSAT back and forth between similar bays on the two spacecraft. They demonstrated over a dozen autonomous transfers of hydrazine monopropellant as well.[18]

On May 28 the second anomaly of the mission occurred when a single event upset (SEU) caused the navigation system onboard NextSAT to shut down during a maneuver that was supposed to place the spacecraft no more than 30 meters from Astro. Instead

[18] Jefferson Morrison. "Full Service: Pioneering Orbital Express Offers Lessons for Satellite Servicing; Pioneering Orbital Express mission offers many lessons for future satellite servicing". Aviation Week & Space Technology. New York: Jul 23, 2007. Vol. 167, Iss. 4, 57.

NextSAT drifted out to six kilometers from Astro when ground operators were able to return the spacecraft to normal operations. With the help of NASA expertise derived from Gemini and Apollo missions, NextSAT was bought back to Astro and successfully mated. Following the error, the program halted rendezvous operations while the cause of NextSAT's guidance failure was identified and remedy procedures were drafted that could be used incase of a future failure.

Operations re-commenced on June 22nd. After the unexpected six kilometer rendezvous, the OE mission validated its 30 meter and 100 meter rendezvous requirements and proceeded to chase more ambitious program goals. OE completed the first successful autonomous capture of another spacecraft with a robotic arm by an unmanned spacecraft. The mission completed autonomous rendezvous and docking procedures from distances ranging from a few meters out to seven kilometers.

With the mission's final demonstrations complete, the program completed all of the mission objectives and set a new standard for other programs. The OE mission cost "$267-million mission--to which Boeing is adding substantial in-house funding"[19] The technology created and then demonstrated by OE should be able to incorporated into smaller spacecraft and satellites with further miniaturization of electronics and solid state avionics.

E. CUBESAT

The CubeSat configuration is a cube-shaped, stackable spacecraft structure, 10-cm on a side. This configuration is credited to Bob Twiggs at Stanford University and has been implemented in the educational and research programs of a number of universities and government agencies throughout the world. The California Polytechnic State University (Cal Poly) and Stanford University are leading a collaboration of 40 universities, high schools, and private firms as part of an international CubeSat

[19] Michael A. Taverna. "Rethinking Recovery: Europe Cools to On-Orbit Servicing; Despite looming Orbital Express launch, Europe backs off from on-orbit servicing." Aviation Week & Space Technology. New York: Feb 19, 2007. Vol. 166, Iss. 8, 60

partnership. These two universities have developed a group facilitated by the Internet and bi-annual conferences, accelerating technology development, enhancing CubeSat capabilities.

This class of satellite is designed primarily for space development education. CubeSats are cubical in shape, ten centimeters in each dimension. They can also be stacked into configurations that maintain two dimensions of ten centimeters and the third dimension is increased in multiples of ten, up to 50 centimeters. A 1U-CubeSat is ten centimeters in all dimensions, while a 3U-CubeSat is 30 centimeters in one dimension and ten centimeters in the other two dimensions. This size variation allows the CubeSat standard to be much more useful and appropriate to a much wider variety of missions. With this standard in place, vendors are able to build components that are specifically designed to be utilized in CubeSats. Pumpkin Incorporated and Clyde Space are leading developers of COTS equipment specifically designed for use in CubeSats.

Various methods have been used to deploy CubeSats from launch vehicles. The most popular method is the Poly Picosatellite Orbital Deployer (P-POD), developed and manufactured by Cal Poly. A P-POD can carry and dispense 3U-worth of CubeSats. The 3U of CubeSats can be in the form of three 1U-CubeSats, a 2U-CubeSat and a 1U-CubeSat or a single 3U-CubeSat. The P-POD has been used to deploy CubeSats in LEO, but there is no reason why it would not be able to operate properly at any orbital altitude. Cal Poly is currently working to develop and construct an extended P-POD that will have the capability to deploy 5U-worth of CubeSats.

CubeSats can be designed to perform various types of missions. CubeSat developers are currently concentrating on space education and technology demonstration. Standardized COTS hardware and software can be purchased which helps educational institutions to build these satellites. In addition to satellite components, COTS ground station hardware and software are also available. With a marginal amount of effort, low tech CubeSats can be built and their ground stations erected. The only difficulty CubeSat developers have is getting their satellites launched into orbit. Standard educational

CubeSat payloads are ham radio transponders, radio frequency beacons, and space science experiments. Industry is beginning to build and launch CubeSats to demonstrate their technology capabilities.

Most notable, Aerospace Corporation has launched two 1U-CubeSats in the past two years. Their CubeSat development lab has designed, built and operated each CubeSat since its establishment in 2000. Desiring a short turn around, the developers concentrated on using COTS equipment that suited their requirements. When this equipment was not available, they developed the miniaturized components themselves. Their designs concentrate on ease of construction and reproducibility. With these guiding principles, Aerospace has been able to take great steps which will facilitate the CubeSat standard to become immediately useful.

AeroCube-2 was launched out of a P-Pod onboard a Russian Dnepr rocket on April 17, 2007. "That CubeSat was constructed in-house using equipment created by Aerospace and their standard providers."[20] Significantly innovative equipment aboard AeroCube-2 were a patch type antenna, five cameras and an inflatable balloon (for de-orbiting). In addition, the satellite used a passive thermal control system that utilized satellite thermal coatings to trap enough heat to allow the satellite to operate at optimum temperatures without the use of heaters. Each C328-7640 JPEG Compression VGA Module (cameras utilized) was positioned to view out of a single face of the satellite. Positioning the cameras on each face enabled the satellite to take pictures in five of six directions, giving the satellite the ability to take at least one picture of its intended target 80 percent of the time as it free tumbled in its orbit. The deployable balloon was mounted on the remaining open surface. The balloon would have inflated via an electrically operated valve that could have filled it using the cold gas stored onboard the nanosatellite. However, due to a power system design problem, the valve was never actuated and the balloon could not be inflated.

Even with the EPS failure the mission was still a success. AeroCube-2 validated the patch strip antenna design, which gave the satellite spherical coverage with the

[20] David A. Hinkley. "Teleconference between David A. Hinkley and Matthew T. Erdner" 6 August 2007.

exception of one axis which effectively reduced the complete coverage area by only 10 percent. This antenna and its transceiver allowed AeroCube-2 to transmit data at about 100k baud. This is compared to the nominal 18k baud limitation of most CubeSat, when developers utilize dipole type antennae on their CubeSats.

Aerospace is planning to build on the success of AeroCube-2 with another 1U-CubeSat, AeroCube-3, which they plan to launch in the spring of 2008. Aerospace's differential GPS unit is on schedule to be completed by October 2007. When they complete this component, it will be the first at a small enough size that can deliver the velocity rates that CubeSat needs to determine it's own position. They also have their eyes set on the largest engineer challenge milestone of CubeSats, attitude control. "The holy grail of 1U-CubeSat development is reliable 3-axis stabilization. Once we have that, the sky is the limit for the 1U-CubeSat's usefulness."[21] Aerospace is currently attempting attitude control by incorporating an electromagnetic coil for north and south attitude control and a single modified Maxon motor for attitude control normal to the north-south plane. In addition to these milestones, AeroCube-3 will also possess deployable solar arrays and a balloon for de-orbit that will be inflated by sublimation.

Aerospace is enthusiastically leading the way together with other organizations to make the 1U-CubeSat a viable satellite standard that may have a powerful role beyond education and technology demonstration. CubeSat capabilities will always be limited to their overall dimensions but through the continual miniaturization of components, the usefulness for CubeSats will be realized allowing them to formation flying, form sparse apertures and execute other inventive implementations.

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[21] Jordi Puig-Suari. "Conversation between Jordi Puig-Suari and Matthew T. Erdner". 15 August 2007.

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III. DISCUSSION OF GEOSTATIONARY ORBIT

A. GENERAL DESCRIPTION OF GEOSTATIONARY ORBIT

Geostationary orbit is located along the plane extending out from earth's center to through the equator to an altitude of 35776.9 kilometers above mean earth sea level. At this altitude, a satellite that has an inclination of zero degrees appears to hover directly over that geographical longitude along earth's equator. This is the uniquely advantageous altitude that allows the satellite to travel at the same velocity at which the earth revolves about its own axis. The geostationary orbit is commonly referred to as a Clarke orbit due to his proclamation in the mid 1940's that only three satellites in this type of orbit would be necessary to provide worldwide communications.

In 1964, Syncom became the first communications satellite to be placed in geostationary orbit. Since that time, the slots in this orbital regime have become extremely sought after by all countries that are capable of deploying their own or purchasing satellites to operate in this region of space. The missions for geostationary satellites have commonly fallen under three categories. The first mission is telecommunications; the second is ISR (Intelligence Surveillance and Reconnaissance.

B. DISCUSSION OF ENVIRONMENTAL CONDITIONS AT GEOSTATIONARY ORBIT

Any satellite placed into geostationary orbit must be able to withstand a very harsh environment. The most concerning factor to affect satellites operating at this orbit are the radiation and charged particles primarily released by the Sun. Solar weather directly factors in the mission lifetime and operation of a satellite located at geostationary orbit. The magnetopause is "the boundary between the region dominated by the geomagnetic field on one side and the region dominated by the solar wind plasma pressure on the other."[22] This region effectively rejects a large amount of the energetic

[22] Richard C. Olsen. Introduction to Space Environment. Monterey: Naval Postgraduate School. January 2005. 131.

particles released by the sun. If a solar flare occurs pointing in the direction of earth, then the earth's magnetopause could be pushed closer to earth. In this event, the area of the magnetopause that is between the earth and the Sun will be forced to move toward the earth a distance that will balance out with the force being applied from it by the energetic particles and solar wind released by the Sun. Within micro-seconds the satellite will pass through the new boundary of earth's magnetopause and into a region that has no protection from the sun's radiation. When this occurs the satellite is exposed to significantly larger amounts of energetic particles and electromagnetic radiation than normal. These particles can induce single event upsets (SEUs), into computer systems and even permanently damage equipment. Depending on the solar cycle, solar weather will be properly characterized and modeled to allow satellite designers and operators to anticipate what type of environment their satellite will operate in during its mission lifetime. With the analysis conducted, satellites designers balance risk and cost to design each satellite enough hardening and redundant components so that it will most likely operate throughout its mission design life (MDL).

For these reasons a satellite designer must robustly design satellites to withstand these environmental conditions. The satellite's design will nominally include enough margin to account for exceptionally bad solar weather periods and unexpected solar weather events such as large magnitude solar flares.

C. IMPORTANCE OF GEOSTATIONARY ORBIT

Geostationary orbit is the only type of orbit that allows a satellite to continually linger over one single location of the earth, providing the user the ability for continual access to a geographical location of interest, as long as that area is in the footprint of the satellite. A communications satellite operating in a geostationary orbit can provide continuous communications between areas in its coverage zone, which is also referred to a satellite's footprint. A communications satellite that is operating at an altitude which is greater than or less than 35776.9 kilometers cannot provide continuous coverage to one geographical area since the satellite is orbiting about earth at a rate that is either faster or slower than the earth rotates about its own axis respectively. This property of geostationary orbit gives it the persistence that is required for effective communications and weather observations.

D. RESTRICTIONS OF GEOSTATIONARY ORBIT

One of the major drawbacks to geostationary orbit is that due to the curvature of the earth's surface, satellites in this orbit will not have global access. A geostationary satellite's coverage zone will extend approximately from 60 degrees south to 60 degrees north latitude along the line of longitude in which the satellite is stationed. Therefore the satellite will never have access to the polar regions of the earth, unless the satellite's inclination is greater than five degrees, and then it is no longer in a true geostationary orbit, but a geosynchronous orbit. The capability of a satellite's sensors will determine its exact coverage area, but it will certainly not have access to earth's Polar regions.

Missions requiring true global access will depend upon augmentation for geostationary satellites. Satellites must operate in other orbits to provide access to these Polar regions. Unfortunately, there is no single orbital plane that will allow a satellite constellation to possess continuous global access.

Along with these coverage restrictions, there are also other perturbations effects that cause a satellite to change its orbital properties. These perturbations will cause a satellite to appear to wobble as well as change position, altitude or inclination. The major factors that contribute to these effects are the earth's non-spherical shape, the gravity of the moon, the Sun's gravity and other factors at geostationary orbit that are minor, yet accumulate with time. Some of these factors are more or less severe depending on a satellite's size. They must be considered and accounted for so a satellites propulsion system and ADCS can be sized properly to allow it to fully perform its assigned mission during its planned mission lifetime.

These factors will cause a satellite that was once operating in geostationary orbit to now operate in a geosynchronous orbit. For this reason, the scope of this thesis will also include satellites that are operating in what is referred to as near geostationary orbit. The orbital period for these satellites is still near to one sidereal day and their inclination is less than five degrees.

IV. COMMON MISSIONS PERFORMED BY GEOSTATIONARY SATELLITES

A. COMMUNICATIONS

From the time Arthur C. Clarke completed his calculations to determine the geostationary altitude, telecommunications system designers dreamed of placing three satellites at this altitude to deliver telecommunications worldwide.

Telecommunications is the primary mission of nearly 95 percent[23] of all operational geostationary satellites. Geostationary altitude provides the perfect orbit to deliver constant communications to every user in the satellite's stationary footprint. Transmitters and receivers alike should never have to be re-adjusted to continue to communicate with the satellite. Whereas ground stations that communicate with satellites operating at any other orbital altitude must continually track the satellite to communicate with it during each satellite pass. The passes can last a few minutes for a LEO satellite and up to several hours for a satellite positioned in a Molniya orbit. Regardless of pass duration, no other orbital regime will allow satellites to have continuous access to the same geographical region. Telecommunications is truly is an ideal mission to be performed by geostationary satellites.

B. ISR

Long dwell operations and persistent access to a geographical area by a single satellite is only possible if it is positioned at geostationary altitude. For these reasons low resolution optical imaging and non-imaging missions and Signals Intelligence (SIGINT) can be conducted at geostationary orbit.

Low resolution imagery missions are continuously conducted at geostationary orbit by weather satellites of various nations. These imaging satellites are perfectly suited to be stationed at geostationary orbit. Once imaging satellites are placed over a

[23] Eric Johnston. "List of Satellites in Geostationary Orbit." 5 August 2007. 22 August 2007. <http://www.satsig.net/sslist.htm.>.

geographic area they provide reliable imagery of the weather occurring in that region earth's atmosphere. They provide access to areas such as the world's oceans that are impractical to monitor in any other way. It would take fleets of ships or aircraft operating continuously equally spaced throughout our world's oceans to deliver similar weather products that a few weather satellites operating at geostationary orbit are able to provide. Severe weather forecasts are created from geostationary weather satellite products that account for thousands of lives and tens of millions of dollars saved each year, by providing accurate forecasts to allow areas to be evacuated before severe weather strikes. The economies of the world benefit from accurate day to day weather forecasts which allow people across the world to effectively plan their days. [24]

The first Defense Support Program (DSP) satellite was delivered to geostationary orbit in 1970 for launch warning of Russian intercontinental ballistic missiles.

> DSP satellites have provided an uninterrupted space-based early warning capability. The original DSP satellite weighed 2,000 pounds and had 400 watts of power, 2,000 detectors and a design life of 1.25 years. Throughout the life of the program, the satellite has undergone numerous improvements to enhance reliability and capability. The weight grew to 5,250 pounds, the power to 1,275 watts, the number of detectors increased three-fold to 6,000 and the design life has been increased to a goal of five years.[25]

The constellation relies on Non-Imaging Infrared (NI-IR) sensors to detect heat plumes against the earth's background temperature. This sensor allows detection of intercontinental ballistic missile launches, jet aircraft operating in after-burner and other objects that are above the cloud layer which are much hotter than their surrounding environment. Additionally, the relative protection offered by the orbit's sheer altitude is very attractive for defending US national assets. DSP's follow-on program, the Space Based Infrared System (SBIRS) has been delayed by several years, and is currently expected to start populating geostationary orbit by 2010.

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[24] Cheryl Pellerin. "Satellite Flood Forecasts Save Lives, Livelihoods in Bangladesh." 10 August 2007. 11 September 2007. < http://usinfo.state.gov/xarchives/display.html?p=washfile-english&y=2007&m=August&x=20070810172913lcnirellep0.2562372>.

[25] Air Force Space Command. "DEFENSE SUPPORT PROGRAM SATELLITES." Fact Sheet. March 2007. August 24, 2007. <http://www.af.mil/factsheets/factsheet.asp?id=96>.

Research is currently underway to develop a visual optical system that will have the ability to obtain higher resolution imagery from geostationary orbit. An imaging system with continuous access to specified geographical locations and attaining a ground resolution of one meter or less would be invaluable to the USG's intelligence agencies and the DoD. Limitations of current technology are the hurdle to overcome for a high resolution geostationary imaging satellite. This application has been a goal for engineers since the first geostationary satellite was considered. To obtain one meter ground resolution, an imaging payload must have a focal length of 429 meters and an aperture diameter of 38 meters which will only allow the optical system to have an F number of eleven[26]. Satellites such as the James Webb telescope offer promising designs, and potential capabilities for use at geostationary orbit. Unfortunately no materials in the dimensions required have been identified that can handle the thermal stresses at geostationary orbit due to earth's own albedo to remain rigid enough to deliver a constant image. This problem is unlikely to be solved for several decades.

Due to the properties of geostationary orbit, uncooperative communications is an ideal operation to be performed. Uncooperative communications is commonly referred to as SIGINT. A satellite that hovers over a same geographical area would have the ability to continuously monitor any region of that area. Persistent surveillance could be accomplished without an adversary being able to avoid it. Even if the satellite was identified and known to be conducting SIGINT missions, an adversary would never know where in its footprint the satellite was listening. Utilizing a geostationary satellite could be the perfect way to accomplish a SIGINT mission.

To perform a SIGINT mission successfully difficulties must be overcome. The satellite must have a very highly tuned receiver system that was able to operate over a several bands of interest. It would need a large enough antenna so it could receive the signals of interest. To maximize the gain, the antenna would have to be very finely

[26] Appendix G.

tuned. Since telecommunications is routinely accomplished at geostationary orbit, it is reasonable to assume that uncooperative communications can also be performed at this orbit.

V. PROPOSED MISSIONS TO BE PERFORMED NEAR GEOSTATIONARY ORBIT BY SMALLER SATELLITES

A. SURVEY MISSIONS

> If the US wishes to enjoy the advantages of space-enabled communications, navigation, precision timing, weather, and ISR in any potential conflict with China, the National Security Space community should aggressively pursue methods to defend its systems from attack. First and foremost, the Air Force, as Defense Department executive agent for space must develop better Space Situation Awareness (SSA), both in breadth and depth. In breadth, the Air Force should build and maintain an improved catalog of objects from low-Earth to geosynchronous orbits. The catalog must not only be complete, capturing increasingly smaller objects; it needs also to be timely to ensure maneuvering vehicles are discovered in time to permit defensive action. In depth, America should develop the capacity to better characterize the nature and capabilities of known satellites. The US must improve its ability to identify the existence, origin, and nature of attacks on its space assets differentiating these attacks from system or environmental anomalies. The need for depth and breadth in SSA extends to ground-based counterspace systems that might be employed against friendly forces. Passive and active defensive systems should follow and leverage SSA improvements to "close the loop" on American vulnerabilities. America stands a better chance of deterring aggression against its critical onorbit assets if it possesses the capability to recognize emerging threats, capture timely indications and warnings, and respond (defensively or offensively) when attacked. To do otherwise presents an inviting vulnerability to an adversary seeking unconventional means to neutralize or defeat a stronger foe.[27]

The optical survey mission of satellites operating in geostationary orbit will allow this important gap in SSA to be filled.

Due to aperture size constraints, satellites at geostationary orbit can only be observed at a desirable resolution from an orbit that is less than 100 kilometers from their location. Even a satellite such as the Hubble Space Telescope with an aperture the size of 2.4 meters with a focal length of 57.6 meters can only observe a geostationary satellite

[27] Martin E.B. France and Richard J, Adams. "*The Chinese Threat to US Superiority*" High Frontier Journal. Vol. 1, No. 3 (Winter 2005), 17.

from about 1200 kilometers away to obtain a 20 centimeter resolution. That may sound like a large distance, but it is truly small when compared to a geostationary satellite's altitude above earth's mean sea level of 35,780 kilometers. The maximum range for a Hubble-like optical imaging payload is only 3.3 percent of the altitude of a geostationary orbit. Even a satellite that costs slightly more than two billion dollars[28] to construct and launch would still have to be placed into an orbit that was relatively close to geostationary orbit so that it could observe geostationary satellites with a spatial resolution of 20 centimeters. Clearly this is not a feasible option from a monetary aspect, but designing smaller satellites that are much less expensive to operate near geostationary orbit to perform this mission may be feasible. If they could be built small enough to remain undetectable by earth based detection systems, then no one else would know they were operating there so they would not create or use techniques to prevent their actions.

B. SERVICE DENIAL

In an increasingly technological environment, all modern militaries are more dependant upon communications, particularly satellite based communications as they are widely used due to their continuous geographical access. A powerful military capability, and a space control mission, is to be able to deny an enemy the use of their geostationary satellite telecommunications.

With a constellation of controlled smaller satellites, a military commander or governmental operative could deny the use of these communications as desired, using a smaller satellite. Currently earth based satellite observation systems have a "limiting magnitude of about 17.5, equivalent to a size detection threshold of about 0.6 m[eters] in GEO"[29] for cooperative objects using an integration time of approximately 20 seconds. Sizing the satellites below the half meter detectable threshold and the use of non-reflectant camouflage insulating materials would allow these satellites to possess a magnitude less than 17.5 as viewed from earth. With a relative magnitude less than that

[28] "Hubble Space Telescope". 4 January 2007. 14 September, 2007. <http://www.bigpedia.com/encyclopedia/Hubble_Space_Telescope>.

[29] Heiner Klinkrad. Space Debris: Models and Risk Analysis. Chichester, UK. Springer, 2006. 32.

threshold, the satellites would be invisible to ground based optical and radar systems. This would allow their existence to be undetectable and thus deniable. To ensure that their effects are truly deniable, jamming payloads need to be designed to jam their targets without damaging the targeted satellite's communications equipment. With no permanent effects left on the targeted satellite, the operators of the jammed satellite would have no proof that they had been jammed. More likely, they would conclude that some environmental effect or temporary component failure was most likely the cause of the targeted satellite's communications outage.

A constellation of smaller satellites carrying jamming payloads would have the ability to jam communications of our enemies at will. Or would it…mission success is a bit more difficult than it may seem.

Several difficulties must be overcome to field a jamming satellite constellation. The active transmitting payload must be configured to jam the target satellite's communication package. At a minimum, the jamming electromagnetic radiation must match the target's operating frequency. A jammer will be more effective and require less transmit power if it can also match the modulation and polarization overpower the target's signal to noise (S/N) ratio. This is a simple process if you are designing a custom payload for a certain target communications satellite, but not if you are trying to design a payload that can jam every type of communications satellite that is currently operating at or near geostationary orbit or at least a large percentage of them. Should the jamming satellite target the uplink or downlink portion of the signal that is traveling to and from the satellite? In that respect, due to propagation path loss it is reasonable to assume that you can jam signals that are traveling to the target satellite not from it.

The only advantage that a small satellite really possesses is that it is positioned less than 100 kilometers from the target satellite, whereas the earth based terminals (ground, ship or air based) are positioned over 35,780 kilometers away. The propagation loss for the small jamming satellite alone will allow a few watts of transmitted power to overcome the milli to pico-watts of power the targeted satellite's communication system is designed to receive if it is delivered into the main beam of the satellite's receiver antenna. More than a few watts of power will be needed if the jamming signal will be able to overcome the targeted signal through a side lobe or even a back lobe. The increase in power will depend on which lobe is targeted. The jamming satellite will either need to have the ability to jam in any of these lobes or be optimized to jam in the main lobe of the targeted satellite's receive antenna. If the satellite goal remains to jam while not destroying fragile components in the targeted satellite's receiving equipment such as low-noise amplifiers (LNAs), then it must be able to determine the position it is to the targeted satellite, know the satellite's receiver antenna properties so it can autonomously determine the power at which it must transmit and the direction it must point the jamming energy. These factors lead to greater complexity, which is one thing that must be avoided if these smaller satellites are to remain relatively affordable. With these considerations, the most reasonable approach is to design the system to jam only in the targeted satellite's receive antenna's main lobe. Utilizing the gain of the targeted satellite's main lobe will allow the jamming satellite to effectively overcome the target's S/N while guaranteeing not to permanently damage the target.

Another consideration is the relative position of the optimized jamming satellite to the target satellites' communications antenna's boresight. If the jamming satellite is more than 30 degrees off of the target satellite's boresight, then it is very unlikely considering finely tuned parabolic antennae that the jamming energy will be able to overcome the signal energy the target is trying to receive. All of these factors make it very difficult to design a jamming satellite that can perform these tasks properly without destroying components on the targeted communications satellite.

Common operating frequencies for satellites communication span a range of "2-GHz to 18-GHz"[30], or from S-Band to Ku Band. This is a huge range over which a single oscillator could not adequately be used to reproduce jamming signals. This capability would require a very complex radio that had several oscillators that could be selected and then used to accurately generate the frequency at which the targeted satellite is receiving. There are no COTS units that can cover a frequency range of this size. Most units cover a portion of a band, such as S-band or X-band. These constraints drive

[30] Wayne Tomasi. Electronic Communciations Systems. Upper Saddle River: Pearson Education, 2004. 1041.

a requirement that the jamming payload must be comprised of several radios and a controlling unit that has the ability to select between these radios and control them individually. The other option is to have a custom radio developed and built for these jamming satellites which is not an attractive option. This would require a one-off type unit to be manufactured to suit the needs of a single satellite. There would likely be no commercial utility to sell this kind of a radio. Without that motivation, the component manufacturer would likely require the satellite develop to pay for the research and develop of the radio and then also pay for each unit that was needed for the project. This type of situation would be very costly to the program. Any perceived cost savings to the program through utilizing COTS equipment would vanish.

To complicate this matter further, not only does a jamming satellite's payload need to address different operating frequencies and modulation schemes but the jamming satellite must also address different antenna types that potential target telecommunication satellites are operating. The most common type of antenna is the parabolic reflector dish type, but others are various horn types, phased arrays, patch arrays, Yagi-Uda, log-periodic and helical to mention a few. Each type of antenna possesses its unique radiation characteristics, likely operating frequencies and operating EIRP levels. Effort to gain access to the main beam of an antenna of one type will be completely different to another antenna type.

Dealing with a wide range of targeted satellite systems and positioning the jamming satellite inside of the target's main beam represent a very complicated problem that a large satellite would struggle to meet. It is completely unreasonable to believe that any small satellite could perform this mission and remain relatively inexpensive while measuring less than a half meter in width, length and height.

This jamming mission could be accomplished by a smaller satellite if it is assigned to jam only a specific telecommunications satellite. The payload could then be designed to jam either the TT&C or the operating band of the target satellite. It is common practice for satellites to transmit and receive their TT&C on their operating band, and use the designated TT&C band only as a back-up after launch. Therefore it makes most sense to target the operating band. Knowing the antenna type and most likely its operating power, the jammer's payload can easily be sized so that it transmits at a power that will accomplish its mission without damaging components on the targeted satellite.

Disregarding spread spectrum, frequency hopping and other jam prevention techniques that are currently employed by most military communications satellites, it may be possible to jam a target commercial communication satellite of interest not employing these techniques. With the jamming satellite designed perfectly, there remains a problem of orbit selection, which will determine that amount of time available for the satellite to effectively jam its target. The closer to the target the jamming payload is placed; the longer it will be in the target's boresight to effectively jam it. Placing a satellite at a lower altitude also has a disadvantage of a large time to re-visit the targeted satellite. A satellite that is placed at 20 kilometers sub geostationary orbit will take approximately eight years to circumnavigate the geostationary belt, while a satellite placed at 70 kilometers sub-geostationary orbit will take three years. This means that a jamming satellite that is targeting a single satellite will reasonably only be able to perform its mission once.

The solutions to this problem are not very attractive for a smaller satellite. One option is to place a thruster system on the satellite to allow it to stay inside of its target's main lobe for a longer period. Adding a thruster will greatly increase the size of the satellite and detract from the payloads performance if it's even possible to design a small satellite of the desired dimensions to perform this task with a large propulsion system. The other option is to attach the small satellite to the target satellite, which involves adding a mating or docking system to the smaller satellite. The docking or mating system will greatly increase the satellite's complexity and mandate the need for a propulsion system. A docking or mating mechanism will require a more robust command and data handling (C&DH) system and the computer code to control the maneuvers that will be required to use the system. These requirements will drive the satellite's cost to a much higher level and will likely push the envelope of size of the satellite beyond the desired dimensions.

The smallest satellite that has flown with an autonomous docking system is NextSAT of the OE mission. NextSAT was designed to dock specifically with Astro with their docking interface. This is not an interface that could be used to mate to an existing communications satellite. NextSAT is double the size that appeals for the smaller satellite's maximum dimensions. The docking mechanism itself would encompass the volume of the desired satellite. There are designs that the European Space Agency (ESA) is pursuing to develop this type of technology, but the sizes of their docking mechanisms remain too large for this application. In the future, this type of mating system may be possible.

Due to these considerations a jamming mission is not suitable to a smaller satellite of the desired dimensions and cost. A larger satellite could be built to accomplish this mission, but its existence would be known to earth based observers. Once it began its operations, it's target satellite's operators would easily be able to determine the source of the jamming signals and pursue diplomatic actions. The negative media would likely cause sufficient uproar in the US for the program to be shut down. The only time that Americans may support this type of program would be during a time of all out declared war, and even then there would be opponents to jamming satellites and the militarization of space.

C. SATELLITE NEUTRALIZATION

Due to the nature of most Americans, this mission in itself is the most controversial of all the missions analyzed. Proponents and opponents to this type of activity have debated in various forums since the beginning of manned spaceflight over five decades ago. To that extent the United Nation's Committee on the Peaceful Uses of Outer Space (COPUOS), the international agreement which governs the use of space, has outlawed this use for a spacecraft or satellite. That said this mission has the potential to be easily accomplished at a fairly low cost barring launch costs.

Covert ASAT missions likely would not be supported by the US population. The USG feels the need to preserve the inherent right of self-defense in all areas in which its forces operate. The US feels that it has the right to position objects into space that could be used to defend their own space assets. For this reason US officially has voted against or abstained from voting on each "Prevention of an arms race in outer space" resolution presented through COPUOS[31]. Some US citizens would likely agree to ratify such a resolution not realizing that such an agreement would not allow the US to protect its own interests in space should they become attacked by our enemies. Every defense spacecraft program that has been created to perform proximity operations around another satellite has been surrounded by negative publicity. Americans that fear this technology refer to these programs as only "really" being developed so that an ASAT weapon can be built and used to weaponize space. From XSS-10 to DART (a NASA servicing mission to validate technology that if successful would have led to a Hubble Space Telescope service vehicle) critics have raised their voices in the media and on the floors of Congress. These opposition groups have made funding viable dual use technology development programs very difficult. With the Soviet Union's collapse satellite programs involving autonomous rendezvous and proximity operations had lost their public backing completely until the Chinese shot down one of their own satellites in January 2007. Sentiment has not changed greatly in the US since January 2007, but the tide may be turning as China continues to print articles on how it can jam telecommunications satellites during warfare.

It is true that technology developed under these programs could be used to construct a very capable co-orbiter type ASAT, but XSS-10, XSS-11, and OE have only proven this technology at LEO.

> Although microsatellites are perceived primarily as a threat to satellites in LEO, they could be adapted to attack assets in geosynchronous orbit as well. A space mine would be effective only if it were orbiting very close to its quarry, in an almost identical orbit. The space mine would not need to be deployed covertly; there would be no means of destroying or

[31] "Index of Online General Assembly Resolutions Relating to Outer Space: Recorded Votes on Resolutions." 12 December 2006. 28 August 2007. <http://www.unoosa.org/oosa/SpaceLaw/gares/gavotes.html#ARES_56_23>.

disabling the mine without also risking the destruction of its much more valuable target, so the mine poses a similar threat whether its presence is known or unknown.[32]

Various types of mission can be envisioned, an anti-satellite satellite (ASAT) can take multiple forms and still accomplish its objective. Perhaps the easiest is to jam a satellite at such a high power that you damage very fragile components of the targeted satellite's receiver system such as low noise amplifiers (LNAs). Alternatively, the satellite could be no more than a space mine, that when directed, detonates and destroys satellites within range. More sophisticated satellites could be designed with propulsion systems that allow them to intercept and collide with a target satellite. Taking that notion to the next level would be to grapple an ASAT to its target. Once the ASAT is connected to the target, it can use various methods to degrade or disable the target satellite. With the effects delivered the satellite could move on to its next target, stay attached or move itself off into a final orbit where it would remain unobservable.

A kinetic effect is any means that is used to hit a target with another object possessing a different velocity vector. This type of effect could be delivered by a spacecraft shooting a projectile at a target or smashing a spacecraft directly into the target. For either of these applications, a kinetic effect will damage and likely destroy the target satellite. With the target destroyed, the portions of both spacecraft will be spread throughout that region of geostationary orbit. Some fragments depending on their initial velocity vectors will be sent to super and sub geostationary regions. The pieces of the spacecraft will range from large to small sizes. Their momentum will be large enough to damage any satellite they happen to meet. These components will effectively make that region of geostationary orbit unusable. This may be advantageous by denying a particularly useful orbit slot to our enemies, but it would prevent its use by the USG and our allies. The use of a kinetic effect at geostationary orbit would effectively deny the use of this orbit to everyone for a very long time.

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[32] Bruce DeBlois. IEEE Spectrum Star-Crossed. June 2004. 18 September. <http://www.spectrum.ieee.org/print/1585>.

The disadvantages of a kinetic effect ASAT are well founded. Our military functions most effectively with public support. As the Chinese ASAT demonstration in January of 2007 showed, this type of activity is not openly supported by the USG and the US public.[33] Making an orbit unusable by scattering debris in it may also be unwise. Out of the proposed ASAT architectures, only the rendezvousing type ASAT seems to be appropriate.

The rendezvousing ASAT must be sized to avoid detection before it arrived at geostationary orbit, while it was operating and also after it operated. Therefore, the cross-sectional length of the satellite would need to be 0.7 meters or less[34]. If a cube type satellite was to be constructed for simplicity sake, no dimensions could be larger than a half meter to remain unobservable by current earth based sensors. In addition to the physical dimension constraint, all surfaces must employ low reflectant materials for camouflage. These materials will prevent large amounts of light from being reflected off the satellite to the earth.

The ASAT would require a large amount of propellant or electrical power to allow it to reach the target from a seed geostationary orbit. For instance a satellite with a mass of 14.5 kilograms requires approximately 1,800 meters per second of velocity change to complete two Hohmann transfers to change its equatorial longitudinal position at geostationary orbit by fifteen degrees in six days.[35] Depending on the performance of the satellite's propulsion system this could be anywhere from five tenths of a kilogram of xenon for a Hall Effect thruster to forty-five kilograms of cold gas for a mono-propellant thruster with an ISP of 300 seconds. Forty-Five kilograms of cold gas would amount to about three times the satellite's original mass causing the resulting satellite to mass to be about 60 kilograms. Five tenths of a kilogram of xenon is certainly feasible, but a Hall Effect Thruster (HET) requires 1,400 watts of continuous power at a minimum

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[33] Antoaneta Bezlova. "Missile test gives new life to 'China threat'." 25 January 2007. 11 September 2007. <http://www.atimes.com/atimes/China/IA25Ad01.html>.

[34] Heiner Klinkrad. <u>Space Debris: Models and Risk Analysis</u>. Chichester, UK. Springer, 2006. 32.

[35] Appendix B.

throughout such a maneuver[36], which could not be supported by a smaller satellite's EPS. Not to mention that the mass of the propulsion system's engine would also have to be factored into the mass equation, thus causing the need for even more propellant.[37] A smaller satellite will not have the available volume or power to support either type of propulsion system.

Even if the ASAT had a propulsion system to allow it to rendezvous, it would also require the computing power to handle onboard navigation and possess a precise ADCS so it could dock with the target satellite. Docked to the target, the ASAT would need to have a mechanism that it could be used to attack components or the bus of the target satellite. With these abilities, the ASAT would have the ability to interrupt operations, damage components or destroy the target. After these effects were delivered, the ASAT could remain attached, or if it had enough propulsion it could maneuver away from the target to conduct another mission.

Unfortunately, there is no propulsion system that could give a smaller satellite of half meter dimensions the performance necessary to perform the maneuvers required to successful conduct a rendezvous ASAT mission. No known mechanisms exist to allow the ASAT to damage the target, their development would likely be expensive. The last problem is a docking system. No standard system currently exists. OE used a docking system between Astro and NextSAT, but that system only worked between those two satellites. Which leads to another new system to be developed that would have the ability to mate the ASAT to any type of target satellite. This would induce more complexity and expense into the ASAT's development.

With these findings based on current smaller satellite technology and American ethics, a geostationary ASAT is not currently practical. In the event of a declared war, this type of weapon built in a small satellite sized or larger structure is completely

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[36] Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 703.

[37] IBID, 687.

feasible. To that extent, building such a few such satellites for the US’s tactical inventory would be a useful step to ensure preparation for the next war, which may likely involve warfare that is conducted outside of earth’s atmosphere.

VI. CONCEPT OF OPERATIONS (CONOPS)

A. GENERAL OPERATIONS

The following metrics were generated to analysis the potential that smaller satellites may possess to conduct a covert optical survey mission. They were not adapted from any existing SSA requirement, nor were they based on an existing space program. For the purpose of this analysis a covert optical survey mission could be conducted successfully if it met the follow metrics.

These metrics are:

- Image all satellites operating at geostationary orbit at a maximum spatial resolution of one half meter.
- Image 95 percent of the satellites operating within 0.05 degrees inclination of the geostationary belt at a maximum spatial resolution of one half meter.
- 30 day re-visit rate for geostationary satellites.
- Altitude determined by the separation distance required for a minimum target spatial resolution of 20 centimeters.
- Relay all imagery through the Air Force Satellite Control Network (AFSCN) or the Tracking and Data Relay Satellite System (TDRSS) within two hours of collection.
- Remain undetectable from earth based sensors.
- Two year mission life.
- Ability to perform station keeping maneuvers.
- COTS equipment incorporated to the maximum extent without degrading performance.

While each satellite designed will strive to remain as inexpensive as possible, no funding limit has been imposed. The goal is to determine if this mission can be accomplished in the size satellite selected, not constraining capabilities due to fiscal limits. The listed performance metrics will drive the design requirements for each satellite size, in size increments listed.

To meet these performance criteria, a constellation of satellites will be needed. The satellites in each constellation need to be separated by equal amounts of equatorial longitude, so their collective work will meet the 30 day re-visit rate. The altitude of each constellation will be driven by the performance of their respective optical payload. Each constellation will be stationed at a distance from the geostationary belt that will allow them to image the satellites at the Closest Point of Approach (CPA) at a spatial resolution of 20 centimeters. With the nominal altitude determined by the optical payload, the payload must also have the ability to image the targets with a wide enough spectral range to make accurate, discernable conclusions about the objects in the image. Using IKONOS's monochromatic imaging properties as a guide; the spectral range requires that each pixel has eleven bits to store its captured spectral information.[38]

The optical systems will need to utilize the most advanced square matrix Focal Plane Arrays (FPAs) currently produced. Kodak produces COTS square matrix FPAs with various pixel pitch sizes. Kodak is currently evolving their production process allowing the pixel size in the FPAs to shrink with each revision. Pixel size is also referred to as pixel pitch. Each pixel is its own detector making up the FPA's imaging sensors. Effectively the smaller the pixel's pitch, the better the performance of an optical system. With each pixel's size decreasing, Charge-Coupled Devices (CCDs) can be constructed with more pixels within a given area. Modern CCD matrix imagers contain thousands to millions of pixels per CCD. Each image captured of a targeted satellite using a modern CCD matrix imager will likely contain a large amount of non-useful background. Autonomous optical system post processing techniques will need to "crop" around the target in the images to remove 90 percent of these non-useful pixels. The resulting useful, "cropped" image will then be compressed using lossless compression techniques to limit the size of any image to a few kilobytes of data. These reduced image sizes will allow data transmission to occur over shorter timeframes which will allow the satellite to orient itself in the most advantageous orientation to remain undetectable, absorb solar energy and release thermal energy.

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[38] S. Kilston. "Ikonos-2, Block-1." Sharing Earth Observation Resources. 16 April 2007. 1 May 2006. <http://directory.eoportal.org/pres_Ikonos2Block1.html>.

Patch array antennae that possess appropriate size and performance characteristics for use on a smaller satellite possess a narrow boresight which allows the antenna to have a relatively high directivity. This directivity is needed so that the smaller satellite can complete its link budget to earth or a space based relay. It is therefore likely that during imaging operations, a satellite will not be in an appropriate geometry to directly relay imagery. Due to the size constraints levied on the satellites, it is unrealistic to assume a gimbaled antenna could be incorporated into the satellites' design. Therefore the satellites will need to have the ability to store and then forward their data. Once the satellites have completed the imaging, they will have to relay the data to their ground operators through either the TDRSS constellation or the AFSCN. To transmit data to TDRSS, the satellites will need to transmit their TT&C and data over S-Band. This will drive the selection of an S-Band capable antenna and radio system. The size of the satellites will also dictate the use of a directional antenna to deliver the required performance to complete the link budget. The operation of a directional antenna combined with the payload drive the pointing requirements for satellite.

The satellites will need an embedded 3-axis stabilization system to orient them properly to conduct their missions. This drives an ADCS that has the ability to sense the satellites' position and orientation. They require a sensor package to determine these aspects and also a system to steer the satellites' to the desired orientation. For these satellites to perform their missions, they will require a miniaturized ADCS that has a capability equivalent to the ADCS of modern satellites. Smaller satellites will need to have a star tracker, sun sensors and earth sensor to feed the ADCS allowing it to accurately determine the satellite's attitude and position for imaging operations. For attitude control they will need to use small Reaction Wheels (RWs) or miniature Control Moment Gyros (CMGs) that are sized to move the satellites about their axes for imaging operations, communications transmission and to overcome environmental induced torques acting on the satellite. With these systems in place the ADCS will be able to orient the satellite so that it can accomplish its mission.

Due to orbital perturbations at geostationary orbit the satellites will need to have a Reaction Control System (RCS) that can actively adjust the satellite's orbit so that it maintains a zero degree inclination at its assigned altitude above earth. The RCS will need to utilizes inputs from the satellite's ADCS system and be able to autonomously make these maneuvers when its ground operators permit such operations. The satellite's propulsion system must be sized to conduct these maneuvers throughout the MDL. As well as accounting for these maneuvers, the propulsion system must have the performance to conduct a single maximum longitude change of fifteen degrees utilizing two Hohmann-like transfers. This ability is needed for possible orbit injection methods and may be necessary for satellite re-phasing in the event of satellite failure causing extended constellation coverage outages of high value targets.

Given the harsh environment of geostationary orbit, each satellite requires a computer and a secondary, back-up processor that are radiation hardened to withstand the expected radiation levels over the MDL. With two processors operating simultaneously, it is unlikely that the satellite will ever experience dual SEUs that would have the ability to shut down each processor at the same time, halting satellite operations. While this may double the mass, volume and power requirements for the satellite's C&DH, it will also provide necessary redundancy in this critical area.

For these components to operate properly, a thermal control system (TCS) will need to maintain the satellites' components in their safe operating temperature ranges. Each satellite will require its own unique TCS that utilizes active and passive thermal control components designed to operate during periods of eclipse at the equinoxes. The TCS components will also need to leave the smallest footprint on the satellites' volume, power and mass as possible. To do this the satellites will need to employ a spread thermal control technique. This technique encompasses placing components around the satellite's interior to effectively spread the heat produced by these components throughout the satellite. This type of technique allows passive heating of the satellite. Employing this technique should allow the TCS to only need to run one heater at a time when active heating is necessary. This and other constraints are imposed in the design of the TCS to minimize power requirements during any satellite operation. Each satellite will utilize a passive radiator to dissipate excess heat to deep space. When possible, proper operation of the radiator requires it to be pointed away from earth and the other celestial objects. This ensures that thermal energy is released by the cooling system and not induced through the radiator into the satellite.

To operate all of these required systems, the satellites require a custom EPS capable of regulating and distributing power during operations. Critical to the performance of the EPS is the area of the solar arrays and their orientation to the sun. The solar arrays will be limited in their dimensions by the half-meter requirement constraint. To this extent, the maximum solar array area in any orientation is a quarter of a square meter. With this available solar array area, the type of solar cells selected for each satellite will need to have the End of Life (EOL) performance to convert enough solar energy to sufficiently power the satellite's EPS. The satellite's size will constrain the solar arrays to be hard mounted to the satellite's structure; gimballing sun-tracking mechanisms for the solar arrays will not be possible due to technological and power budget constraints. The ADCS must orient the satellite in the most advantageous geometry relative to the sun whenever possible to produce the maximum amount of solar energy. Along with the area constraints, the batteries will require sufficient energy storage density to minimize battery volume while retaining the ability to power the satellite's EPS through an eclipse cycles at the EOL performance with only half of the battery cells operating properly. This requirement will allow enough redundancy for the satellite to continue to operate properly if half of the cells fail before MDL.

With the satellites properly designed, they will have the ability to completely perform their assigned mission. Autonomous imaging operations for each satellite require three pictures of all targets to be captured. A picture will be taken before CPA, one at the CPA and one after CPA with the target satellite. When the satellite has the ability to transfer data, it will transmit all stored data and TT&C information and then receive an updated set of operation orders. These orders will list targets and imaging time sequences for the next two weeks. If one minute passes after completion of the satellite's data transmission without beginning to receive an updated set of orders, the satellite will break the communication's link and execute its current set of orders. When the satellite is not imaging it will orient itself in the most advantageous attitude to maintain geometry between itself, the sun and earth to prevent sun light from reflecting off of it back to earth. While maintaining a “concealing” attitude, the satellite must also optimize its attitude for energy production and pointing its radiator away from celestial heat sources. A satellite will only be able to operate its propulsion system when not imaging or transmitting or receiving data. To this extent satellite operations are simplified to give the greatest opportunity for the satellites to successfully complete their mission.

VII. CONSTELLATION DESIGN

A. METHOD UTILIZED TO CONDUCT PRELIMINARY SATELLITE DESIGNS

The goal of this analysis was to determine if smaller satellites could perform the optical survey mission covertly. The Half-Meter-Cube satellite was chosen based on the maximum dimensions that a satellite could be built to have a cross sectional length less than 0.7 meters, the current benchmark size for earth based sensors to detect objects near geostationary orbit using long-dwell imagery techniques. Due to the growing interest in the CubeSat standard, the smallest and largest standard sizes, 1U and 5U respectfully, were chosen for this analysis. Each CubeSats' cross sectional length is less than 0.7 meters.

Designing the three primary sizes of satellites involved multiple iterations. The first phase in satellite design was to develop a set of requirements that each satellite size would be designed to meet to perform the assigned mission. After those requirements had been determined, then they were scrutinized thoroughly to assure they were comprehensive, yet realistic to accomplish the stated mission. The requirements are listed in Table 2. With an acceptable set of requirements, the preliminary design of the satellite began.

In most missions, a payload is selected or created that meets the mission's requirements. The satellite is then designed around the payload. This ensures that the payload can function properly which should ensure mission success for the satellite. In this assigned mission it was equally important for the satellites to remain un-observed as it was for them to complete their imaging mission. The requirement to remain undetectable drove the design process to begin with the satellite's overall dimensions to be the starting point.

Size	Maximum Length 0.5 meters.
	Maximum Width 0.5 meters.
	Maximum Height 0.5 meters.
	Maximum mass of 100 kilograms.
Mission	Image 95% of all satellites operating in the Geostationary belt with a 30 day re-visit rate
	Image target satellites at a spatial resolution of 20 centimeters.
	Onboard position determination system.
	Downlink all data through AFSCN ground station network or TRDSS system
	Semi-Autonomious operations via ground cueing received by satellite
	2 year Mission Design Life (MDL)
	COTS with emphasis on flown in space or designed for space operation at a minimum
Equipment	Transmit and receive S-Band communications
	Ability to store and then forward TT&C and data
	Downlink images within two hours of capture
	Reasonable cost

Table 2. Overarching Smaller Satellite Design Requirements.

The overall preliminary design process involved using the physical dimensions of the satellite as the constraining factor, not the payload. The largest satellite designed is the Half-Meter-Cube. This size was chosen to maximize the volume available in the satellite to allow the largest possible optical payload to be incorporated. The medium size is the 5U-CubeSat measuring ten centimeters by ten centimeters by 50 centimeters. The smallest size satellite chosen was the 1U-CubeSat. The overall dimensions of each satellite were fixed constituting the maximum size for the Structure and Mechanisms Systems (SMS) of the satellite.

Once the volume available for equipment had been determined, the search for proper COTS equipment began. Most of the searching involved internet searches of aerospace corporations known to build satellite components. When those initial searches were exhausted, the research continued with seed information from the CubeSat community via the CubeSat Workshop in April of 2007 and meeting vendors at the Small Satellite Conference in August 2007. COTS equipment was selected based on whether it had flown in space or was space qualified, and would meet or exceed the satellite's objectives. The next considerations were related to the component's volume, power consumption, safe operating temperature range, reliability and cost.

After the equipment was selected, the methodology described in SMAD was used to characterize the Electrical Power System (EPS). Solar array sizes were maximized based on the 0.5 meter satellite dimensions for body mounted panels for the Half-Meter-Cube satellite. The 0.5 meter size requirement also limited the size of deployable solar wings for the 5U-CubeSat and the 1U-CubeSat. The overall solar array dimensions were used to determine the maximum amount of solar energy that could be generated to power the satellite. This calculation was limited to determine the worst case illumination geometry with the sun only illuminating one side of any of the satellites at a 30 degree angle of incidence[39]. The EPS was furthered constrained to the performance of the solar array at its End of Life (EOL) efficiency of 23 percent. The expected power produced was then applied to simple model of the expected operations while the satellite was orbiting the earth. This established a rough power budget for use during both illuminated and eclipse operations. These parameters were used to determine power available to operate the satellite, most importantly the payload in a worst case scenario.

Optical payloads are point designs for particular missions, therefore it is extremely difficult, and in some cases impossible to use COTS components for optical payloads. The 1U-CubeSat, limited severely by available volume was limited to a C328-7640 JPEG Compression VGA Module (COTS board type camera) payload that operated onboard AeroCube-2. Due to aperture and component size constraints, no COTS optical payloads could be identified for use in the Half-Meter-Cube or the 5U-CubeSat. A custom optical payload would need to be manufactured for each of these satellites. After researching several payload designs that have flown, or are currently operating on orbit, the Kodak Model 1000 Camera System (currently operating onboard IKONOS I block

[39] Appendices A and B.

II)[40] was chosen. Each payload was then "sized" in mass, volume and power requirements using SMAD's optical payload sizing equations[41] which are based on the desired diameter of the optical payload.

The focal length, diameter and operating wavelength for these payloads were analyzed to determine the maximum distance at which each satellite would operate to observe targets with a 20 centimeter spatial resolution. The maximum distance was used as the constellation's seed value for its altitude determination. Orbital altitude for the constellation was set to the geostationary altitude minus this seed value. The properties of the payloads were used to determine the distance at which an object could be observed at a spatial resolution of 50 centimeters. This distance was then set to the maximum operating parameter for the payload.

Through this analysis it was determined that the COTS payload for the 1U-CubeSat would require the satellites to be positioned 500 meters[42] below geostationary orbit. At this orbital altitude the 1U-CubeSat constellation would require at least 2,000 satellites. The costs would be unreasonable to produce and operate this number of satellites. With all other COTS optical system alternatives available to the 1U-CubeSat delivering poorer results, the design of the 1U-CubeSat was determined to be not possible for this mission. The design for the 1U-CubeSat stopped at this point.

With the payload selected for the Half-Meter-Cube and the 5U-CubeSat, an in-depth design of the EPS was worked through several iterations as satellite components were verified for use. These iterations involved looking at each operation the satellites would be expected to perform per orbit for the maximum duration expected under normal operations. With these iterations complete, the requirements for each satellite's battery were determined. These requirements led to selecting four SAFT MP 176065

[40] S. Kilston. "Ikonos-2, Block-1." Sharing Earth Observation Resources. 16 April 2007. 1 May 2006. <http://directory.eoportal.org/pres_Ikonos2Block1.html>.

[41] Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 285.

[42] Appendix C.

Intrgrat*ion*™ lithium ion batteries which are operating at geostationary orbit on the communications satellite, W3M. Complete component selection is detailed the appendix for each satellite.

With the EPS and satellite's dimensions set, components were chosen based on their performance criteria, their physical dimensions and power requirements. After these components were chosen they were placed into the satellite's structure in a method to spread mass and heat production throughout the satellite's available volume. The moments of inertia were determined based on the center of mass of each component in relation to the coordinate system chosen for the satellite. The satellite's Center of Gravity (COG) and moments of inertia were determined. Using these properties the pitch error related to Gravity gradient and Solar radiation torques were calculated. The values of the maximum expected torques and moments of inertia were used to verify that the COTS equipment selected for the ADCS and the Reaction Control System (RCS) would appropriately provide 3-axis stability for each satellite. With a properly functioning, 3-axis stabilization system, the satellite's optical payload would be able to perform its assigned mission.

With a properly functioning ADCS, each satellite could also utilize a small directional type antenna. This type of antenna could greatly increase the satellite's capability by its small size and relatively high directional gain helping the satellite complete its overall link budget. Following involved searches of the Institute of Electrical and Electronics Engineers (IEEE) online catalog maintained by the Dudley Knox Library several antenna types were identified. Of the antenna identified, two versions were chosen due to their overall dimensions and directional gain. These antenna types were identified in papers detailing conceptual satellite antenna designs that suited each satellite's needs. A 2x2 Microstrip Array Antenna[43] was selected as the antenna for

[43] James A. Nessel, Kory, C.L. Lambert, K.M. Acosta, R.J. and F. A.Miranda. "A Microstrip Patch-Fed Short Backfire Antenna for the Tracking and Data Relay Satellite System–Continuation (TDRSS-C) Multiple Access (MA) Array." <u>Antennas and Propagation Society International Symposium EEE2006</u>, (9-14 July 2006), 894.

the Half-Meter-Cube. A Microstrip Patch-Fed Short Backfire Antenna (SBA) Array[44] was selected for the 5U-CubeSat. The antennae are used to transmit and receive data presenting a single mode of failure for each satellite. These antennae allow the satellites to complete their link budget to a TDRSS satellite and the AFSCN. With closed link budgets, the satellites are now able to perform their missions. Effort was then directed into creating a thermal control system.

The thermal control system was designed to maintain the satellites minimum and maximum equipment operating temperatures. A first level thermal analysis was conducted for each satellite to determine the expected satellite temperatures at their operating altitudes during the equinoxes. The worst case cold temperature was used to determine the number of active COTS Kapton heaters and their sizes, then their placement throughout the satellite. The worst case hot temperature was used to determine the area that was needed for the passive radiator for each satellite. With these requirements known a COTS Minco CT325 Thermal Control Module was selected to be used to regulate the temperature on each satellite. A thermal coating would be applied to each satellite's exterior. That thermal coating would then be covered by six layers of Multiple Layer Insulation (MLI) that was painted with 3M Black Velvet[45] spacecraft paint. For the purposes of this examination, the assumption was made that critical components would be mounted to heat plates. Each heat plate would have a miniaturized gas feed heat pipe that would transfer heat via conduction to the passive radiator once the heat plate reached a seed temperature. The seed temperatures were selected to be within ten percent of the maximum operating temperatures for the equipment. If a prototype design followed, the thermal system would be refined through testing to determine the optimum TCS.

With the TCS design complete, all major systems and subsystems were then complete. Unfortunately both the Half-Meter-Cube and the 5U-CubeSat satellites'

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[44] Ajay K. Sharma, S.K. Agrawal, D.S. Rajpurohit, R. Singh. and A. Mittal. "A Wideband Microstrip Array Antenna With Unique Dumbbell Shaped Aperture Coupled Radiating Elements." Antennas and Propagation Society International Symposium EEE2006, (9-14 July 2006), 891.

[45] Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 436.

designs were flawed. Neither satellite's payload was designed with a sunshade. A sunshade is required to reduce the potential for stray light from entering the aperture during imaging operations. Relatively long re-visit periods drive sunshades to be incorporated in each smaller satellite's design to reduce expected glare periods, especially during target satellite imaging opportunity windows. The solution chosen for each satellite is detailed in the following sections.

B. HALF-METER-CUBE SATELLITE PAYLOAD RE-DESIGN

The largest size small satellite was chosen to meet the largest sized object that could operate near geostationary orbit while remaining undetected by current earth based observation systems[46]. The maximum dimensions for this satellite are 0.5 meter in length, 0.5 meter in width and 0.5 meter in height. Through the design process, the satellite's structure was held to these dimensions to allow the maximum sizing of the payload to be incorporated into the satellite along with all of its support systems' equipment. The custom optical payload possessed a 40 centimeter diameter aperture with a 2.23 meter focal length that was folded five times. The payload was scaled off the "The [Kodak] Model 1000 camera system consists of the following elements: OTS (Optical Telescope Unit), FPU (Focal Plane Unit), DPU (Digital Processing Unit), PSU (Power Supply Unit), and CU (Cabling Unit)."[47] A Kodak KAF-39000-AAA-DD-AE CCD monochrome CCD with a 6.8 micron pixel pitch was incorporated into the scaled custom payload to replace the Focal Plane Unit (FPU) to deliver better optical performance. The entire payload was hermetically seal to allow the CCD to operate at a pressure of one atmosphere. It was determined to have a mass of 8.8 kilograms and consume a maximum of 18 watts of power while imaging. A twenty centimeter spatial resolution could be obtained when the custom payload was positioned 64 kilometers from a target. This resolution established an operating altitude of approximately 35720 kilometers.

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[46] Heiner Klinkrad. Space Debris: Models and Risk Analysis. Chichester, UK. Springer, 2006. 32.

[47] S. Kilston. Ikonos-2, Block-1. 1 May 2006. Accessed 16 April 2007. <http://directory.eoportal.org/pres_Ikonos2Block1.html>.

With the payload defined, the satellite's subsystems and their components were selected. The resulting satellite has a mass of 37.5 kilograms. This satellite met all requirements, but it failed to incorporate a sunshade into the payload's design. The sun shade for IKONOS's optical system was provided by the satellite's structure.

Glare is induced in an optical system when the geometry between the sun, the target satellite and imaging payload are oriented in a way that allows light from the sun to enter the optical system. These rays will effectively distort the images captured by the CCD, ruining each image in which glare is induced. The sunshade would effectively prevent off-axis light rays from the sun from entering the optical system at certain geometries, but not when the sun was directly in the optical system's Field of View (FOV). A sunshade would give the satellites the ability to mitigate some of these situations, which is very attractive due to the 30 day target satellite re-visit rate. To incorporate a sunshade, the custom optical payload's design and the Half-Meter-Cube satellite had to be re-designed.

A sun shade must meet two parameters to prevent the introduction of off axis glare due to light incident upon the focal array. The sun shade must be at least the same diameter as the optical system's aperture and at least equal to one-half of the aperture's diameter in its overall length.

Two methods were analyzed for sun shade incorporation to determine which offered the best characteristics to the satellite's overall design. The first and easiest approach was to add a sun shade of the proper dimensions to the satellite's external structure. The second method is to add the sun shade into the focal plane array and re-size the components of the optical payload to fit into their new smaller volume.

Utilizing the first method, the first implementation considered was a telescoping sun shade that could deploy during imaging operations and then retract during non-imaging operations. Adding a deployable unit would induce more complexity concerning power requirements, SMS considerations and unknown operational consequences if the mechanism jammed while deploying or retracting. There consideration would lower the satellite's reliability. To ensure that the satellite would be able to meet its MDL, this option was not pursued any further.

A simple add-on method was analyzed next that mounted a solid sunshade directly to the satellite's structure. The required length of the sun shade was calculated to be 0.2 meters; therefore the overall length of the satellite in the x-axis would have been 0.7 meters. That would have caused the satellite to be out of specifications by 40 percent of the required maximum length along the satellite's x-axis. Violating the requirements was not possible so the satellite's entire payload was re-designed.

The optical payload was then re-designed utilizing the second method which incorporated a sun shade into the optical payload's 45 centimeter focal plane. The sunshade is 201 millimeters in diameter and 125 millimeters in overall length. This limited the payload's folded focal length from 2.230 to 1.742 meters and its aperture from 40 to 25 centimeters. The overall performance of the optical payload decreased as well from a 20 centimeters spatial resolution achieved from approximately 64 kilometers to 51 kilometers. Figure 2 show the satellite's design incorporating each method.

The performance of this optical system is reduced compared to the original optical properties, but it will allow the satellite to meet requirements. With this new optical payload, the modified half-meter cube satellite to have a CPA range of 51 kilometers from target satellites to obtain a spatial resolution of 20 centimeters. The first level satellite design for the Half-Meter-Cube was now complete.

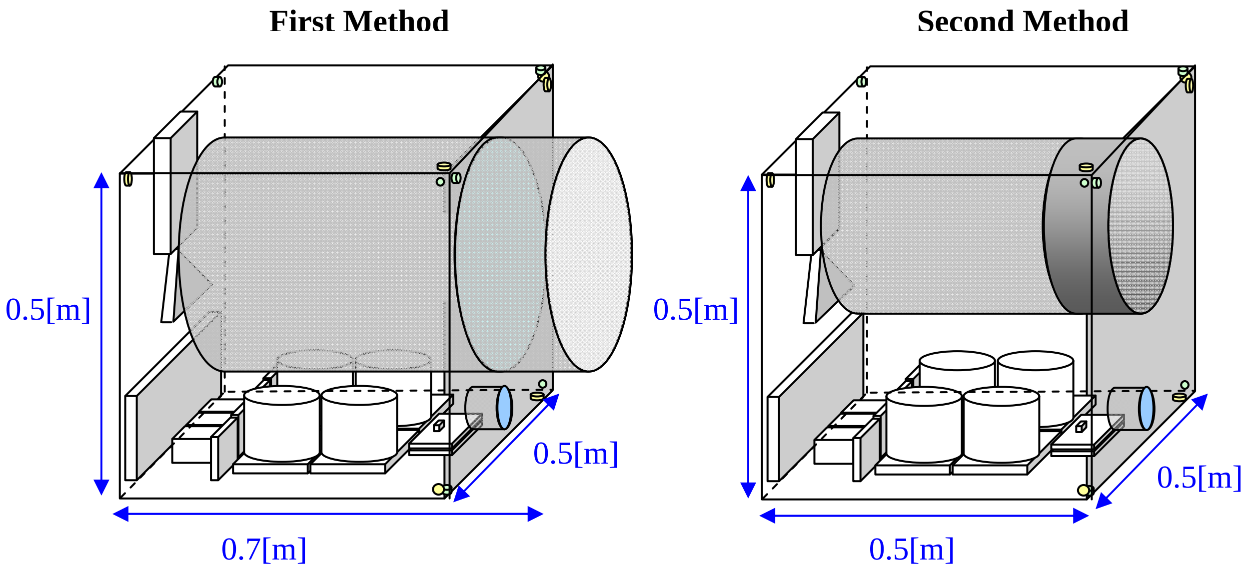

Figure 2. Illustrations of Half-Meter Cube Satellite with Sun Shade incorporated using each method.

The first approach allows the maximum size aperture for the optical payload that can fit in the satellite's overall structure. It also has a major drawback, since the optical payload is carried along an axis of the satellite that already is at the maximum size of a half meter, the additional length of a sun shade makes this dimension longer than the half meter length requirement. Violating the original constraining dimension requirement for the satellite by simply adding a sun shade the satellite's external structure, is not a viable option. Therefore the second, less desirable approach is mandated. With the second method utilized, the overall payload performance of the half-meter cube satellite was defined.[48]

C. 5U-CUBESAT PAYLOAD RE-DESIGN

The 5U-CubeSat was the "middle-size" satellite type design to be analyzed for this thesis. Its dimensions are designed to meet the standards set forth by California

48 Appendix A.

Polytechnic Institute and Stanford University for CubeSat standard dimensions, but not the mass. The standard dimensions do not exceed the mission's requirements.

Through the design process, the satellite's structure was held to these dimensions to allow the maximum sizing of the payload to be incorporated into the satellite along with all of its support systems' equipment. The satellite is 50 centimeters in length (x-axis) and ten centimeters the y and z-axes as seen in Figure 3. Due to stabilization consideration, the center of the satellite's x-axis was reserved for the 3-axis stabilization subsystem. This space allocation allowed a maximum of fifteen centimeters to be allocated for the satellite's payload along the x-axis.

Using the constraint of a maximum fifteen centimeters payload length, the custom optical payload was scaled using a nine centimeter diameter aperture. Scaled off Kodak's Model 1000 Camera System, it incorporated a Kodak KAF-38300 Monochrome CCD with a 5.4 micron pixel pitch vice the FPU normally employed. The entire payload was hermetically seal to allow the CCD to operate at a pressure of one atmosphere. It was determined to have a mass of 0.9971 kilograms and a 553 millimeter focal length. The focal length was folded five times to a linear length of 111 millimeters, which is inside the fifteen centimeters allotted for the payload. The payload was also designed to function as a star tracker like the payload flown in XSS-10. The payload was determined to consume 2.04 watts of power while imaging or functioning as a star tracker. A twenty centimeter spatial resolution could be obtained when this payload was positioned 20.5 kilometers from a target. This resolution established an operational altitude of 35765 kilometers.

With the payload defined, the satellite's subsystems and their equipment were selected. The resulting satellite has a mass of 14.48 kilograms. This satellite met all requirements, but it failed to incorporate a sunshade into the payload's design. Fortunately there was spare room inside of the satellite's structure so that the payload could be simply recessed to create a self-contained sunshade. A simple change to the 5U-CubeSat's structure and re-calculating the satellite's moments of inertia and torques would correct this problem.

The re-design recessed the payload by 45 millimeters into the satellite's structure. The added mass of the aluminum cylinder (used as the sunshade) was assumed to be included in the ten percent overall mass margin added to the satellite's mass. With the sunshade included, the payload's overall length was determined to be 156 millimeters. This length was only six millimeters longer than the length originally allocated for the payload. Due to this small increase in size and the spare five centimeters between equipment in the satellite's x-axis, the 3-axis stabilization system was shifted by one centimeter in the positive x-axis. The center of the satellite's mass only shifted by one millimeter in the x-axis and did not change in the y or x-axes.

Incorporating the re-design of the 5U-CubeSat had no effect on the satellite's performance. The design meets all requirements for the assigned mission.[49]

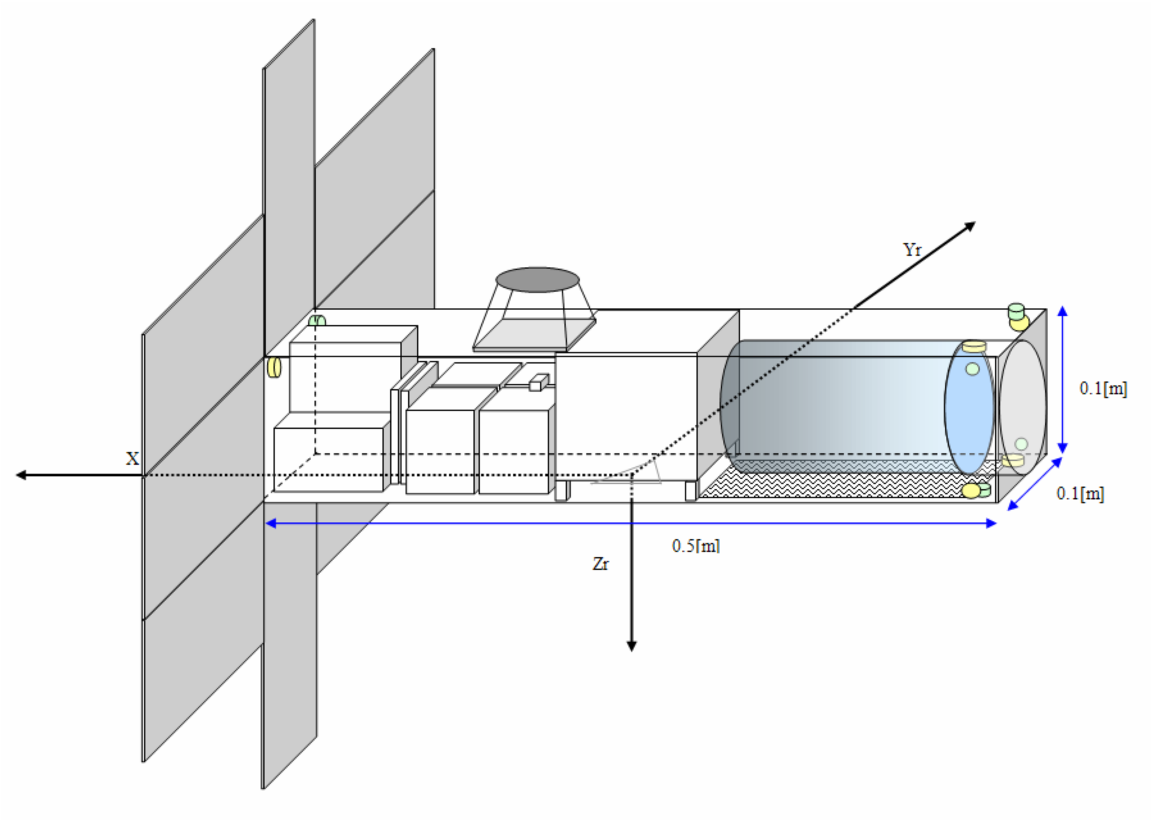

Figure 3. 5U-CubeSat Design Illustration.

[49] Appendix B.

D. FAILED 1U-CUBESAT DESIGN

The 1U-CubeSat design began as each previous satellite with the exception of the optical payload. Aerospace Corporation built and launched AeroCube-2 using five C328-7640 JPEG Compression VGA Modules as the payload. For that reason, the 1U-CubeSat was designed to carry one of these cameras.

The 1U-CubeSat design began by incorporating one of these units as the entire payload. Through mathematical analysis the performance of this payload was analyzed for the assigned mission. The 1U-CubeSat would be required to orbit 500 meters below geostationary orbit to image the satellites operating there at a 20 centimeter spatial resolution. At this altitude it would take the satellite 462 years to circumnavigate the geostationary belt. This drives a constellation of more than 2,000 satellites equally spaced in relation to equatorial longitude along the same orbital plane to posses a 30 day re-visit rate. This analysis determined that this payload could not realistically meet the requirements for this mission due the small aperture size and focal length which required an extremely large number of satellites to constitute the constellation. The large number of satellites would cause the program to be extremely expensive to build and operate.

All other COTS camera-on-a-board type alternatives analyzed performed worse. These results determined that current optical payloads for 1U size CubeSats could not meet the mission objectives. The design for the 1U-CubeSat was terminated at this point.[50]

The only other option was to build a custom deployable aperture similar to the aperture currently being built for the James Webb Space Telescope (JWST) by Ball Aerospace. While it presents a concept to be able to utilize a useful size aperture, it is not currently feasible. The technology for the JWST has just been developed for large satellite and it will take years, if not decades to miniaturize that technology to a level that could be used on a 1U-CubeSat.

[50] Appendix C.

Even if it were possible to design a JWST type aperture for a 1U-CubeSat, which could give the satellite a 26 centimeter aperture other problems still exist. A mechanism would need to be developed to deploy an effective sunshade. Deployable solar arrays would need to be mounted to produce the energy required to operate the satellite. A 3-axis stabilization system and attitude determination system does not exist in a size small enough to fit into the 1U-CubeSat with all the other sub-systems that also needed to operate within the satellite's internal volume. These concerns all prove that a 1U-CubeSat can not be built to meet the requirements set by the assigned optical survey mission at this time or in the foreseeable future.

E. STK SIMULATIONS

Simulation was chosen as a tool to evaluate each satellite constellation's performance to complete the assigned optical survey mission. STK was chosen to conduct the simulations due to the capabilities of the software. Another attractive feature is that the software is continually updated to provide the user a wide array of current tools to analyze satellite operations and up to date two-line element (TLE) sets for cataloged satellites. STK version 8.1 was utilized for all simulations.

The first approach to model this mission was to create the current geosynchronous and geostationary orbits by adding all satellites known to be currently operating in those regions. The simulation period was set to begin at noon on 1 July 2007 and run until noon of 30 July 2007. This period of time allowed the simulation period to cover 30 days exactly.

A constellation of various operating communications satellites was created for the simulation using the illustration created by Boeing Corporation as a guide to the operational communication satellites[51] positioned in geostationary and geosynchronous orbits. Satellites were chosen by their orbital properties and their position in longitude around the geostationary belt. After the initial 40 satellites were added to the simulation, ten more were selected to fill the large longitudinal gaps, with the exception of the central

[51] "Commercial Communications Satellites Geosynchronous Orbit". 30 June 2006. 12 June 2007. <http://www.boeing.com/defense-space/space/bss/launch/980031_001.pdf>.

Pacific Ocean where no satellites are known to currently operate. The simulation contains the majority of the AMC, APSTAR, ARABSAT, HOT_BIRD, INTELSAT, SINOSAT, SUPERBIRD, and ZHONGXING constellations. The resulting large gaps between these satellites were filled by adding CHINASTAR-1, ECHOSTAR-2, NSS-5, PAS-9 and TELSTAR-10. The resulting constellation's largest gap between satellites is 24 degrees between APSTAR-5 and SUPERBIRD-B2 over the central Pacific Ocean.

This simulation set-up was used for both the development of the Half-Meter-Cube satellite constellation and the 5U-CubeSat constellation. Each satellite type was placed at the altitude at which their respective optical payload could image geostationary satellites at the spatial resolution design limit of 20 centimeters. With the simulation set-up, the satellite to be modeled was created and a sensor matching its optical payload was added to the satellite. A "GEOSTAsats constellation" was added to the simulation and all of the communication satellites were added to this constellation. The satellite's payload was then assigned to target the "GEOSTAsats constellation. The simulation was run to confirm proper satellite operation. Once the satellite in the simulation was determined to operate properly a constellation consisting of that type of satellite was created.

Each constellation began with a "seed" number of satellites that was determined on the "Constellation Planning" worksheet[52] for each satellite. This "seed" number was used with STK's "Walker" satellite tool. This tool allows the user to create a type of Walker constellation by defining the number of satellites, the number of orbital planes, satellites per plane, inter plane spacing and RAAN spread. "Delta" type Walker constellations were created using the "seed" number of satellites on the same plane to create the constellations for each simulation. Once each constellation was created, the constellation was run to determine if the number of satellites allowed the constellation to demonstrate a 30 day re-visit. If the constellation did not, the number of "seed" satellites was adjusted and the simulation was run again. After multiple iterations each

[52] Appendices A and B.

constellation design was finalized. The Half-Meter-Cube constellation required 15 satellites while the 5U-CubeSat constellation required 33 satellites all placed at their optimum imaging altitudes.[53]

1. Half-Meter-Cube Constellation Coverage Properties

The Half-Meter-Cube constellation performed very well. It was able to image every communications satellite operating at geostationary orbit. In fact the constellation was able to image every communications satellite with the exception of ARABSAT-2A, SINOSAT-2 and SUPERBIRD-6. Of the satellites not observed SINOSAT-2 has the lowest inclination, 0.3235 degrees. Even at a relatively low inclination, the smallest slant range a Half-Meter-Cube could achieve with the elliptically orbiting SINOSAT-2 was 476 kilometers at the target satellite's apogee or perigee. That slant range is more than three times larger than Half-Meter-Cube's maximum imaging range of 128 kilometers. The only opportunity one of these satellites would have to image that satellite is if they passed by while it was ascending or descending the node of its orbit. Even though five separate Half-Meter-Cube satellites passed by the communications satellite, none of them were able to meet SINOSAT-2 at these two critical locations in its orbit. Each Half-Meter-Cube could have taken a picture of SINOSAT-2 anytime they passed it, however those images would have had a spatial resolution greater than the required maximum performance requirements. While those images would be better than no images, they were still outside of the performance requirements for this analysis. This incident reinforced the fact that even if a target satellite is in a near geostationary orbit, if its orbit is slightly inclined or ellipticity is not zero the slant range between the imaging satellite may be too great for an image to be taken with a spatial resolution less than a half meter.

Even though this perfect meeting for targets to be imaged can only occur twice during each of their orbits, it did occur for some satellites. The optical payload for the Half-Meter-Cube satellite is limited to 128 kilometers maximum range. Using Euclidean geometry, this means that the satellite will only be able to image target satellites with a maximum inclination of 0.086 degrees operating with an altitude of 35,785.9

[53] Appendices D, E and F.

kilometers[54]. With trying to optimize the constellation to image higher inclined targets, the Half-Meter-Cube constellation was still able to image INTELSAT-603, with an inclination of 4.679 degrees as seen in Figure 4.

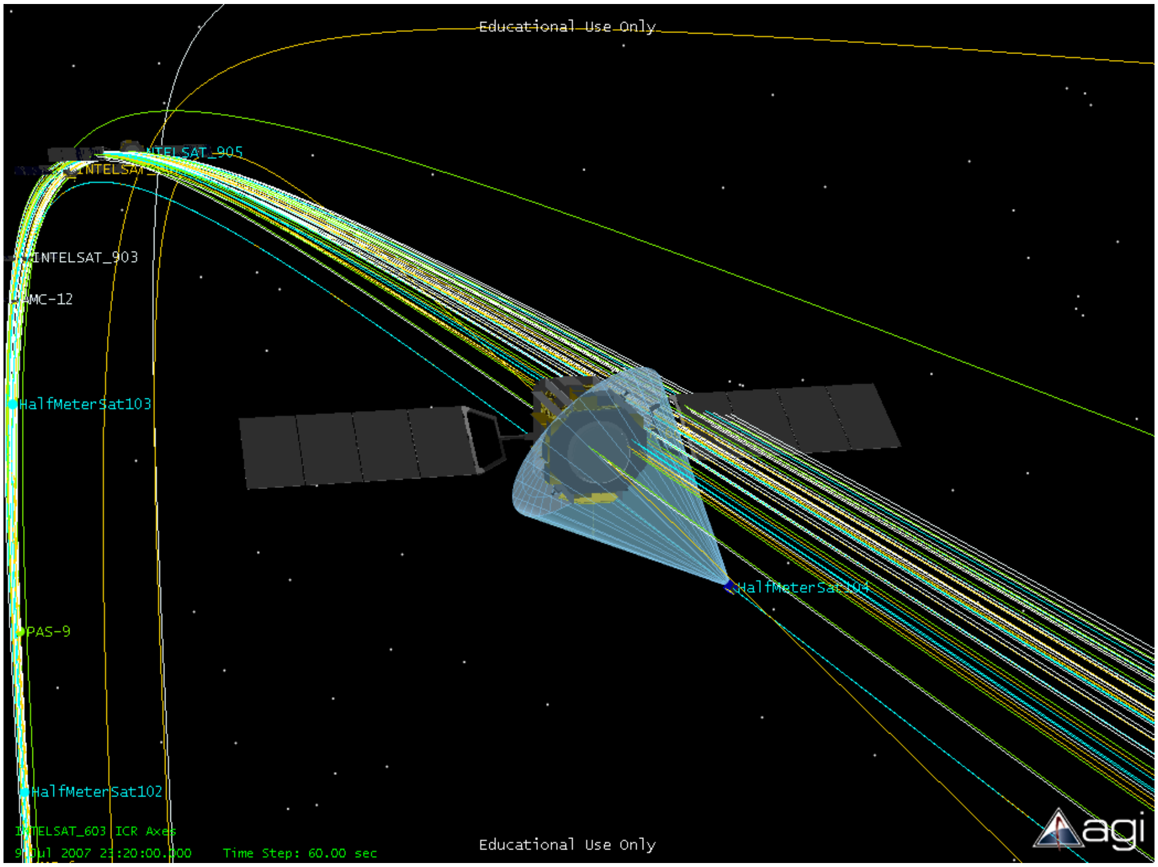

Figure 4. INTELSAT-603 at it's ascending node being imaged by a Half-Meter-Cube.

This was an impressive result. The satellites in the constellation were able to image geosynchronous satellites when they happened to pass by the targeted satellite while it was ascending or descending the node of its orbit. The constellation had a 100 percent coverage rate for geostationary satellites and a 94 percent coverage rate for all communications satellites during the 30 day period of the simulation[55].

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[54] Appendix G.

[55] Appendices D and F.

2. 5U-CubeSat Constellation Coverage Properties

The 5U-CubeSat constellation performed well. It was able to image every communications satellite operating at geostationary orbit. The constellation was no able to image AMC-10, AMC-12, APSTAR-5, ARABSAT-2A, SINOSAT-1, SINOSAT-2 or SUPERBIRD-6. Lowest inclined of the satellites not imaged was AMC-12 with an inclination of only 0.0032 degrees. This was an unexpected result. Further research revealed that AMC-12 has an eccentricity of 0.0004 and a RAAN of 319.4 degrees. The variation from a non-perfect circular orbit allows the satellite's orbit to be slightly longer at its apogee and perigee and slightly smaller 90 degrees from either of these positions. AMC-12 remained unobserved due to the time at which the 5U-CubeSat passed by it, which occurred when AMC-12 was near its own perigee as seen in Figure 5. The constellation was able to image some of the satellites that were operating in geosynchronous orbit when they happened to pass by the targeted satellite while it ascended or descended the node of their orbits. Of these satellites, INTELSAT-603 with an inclination of 4.679 degrees was imaged for 186 seconds.

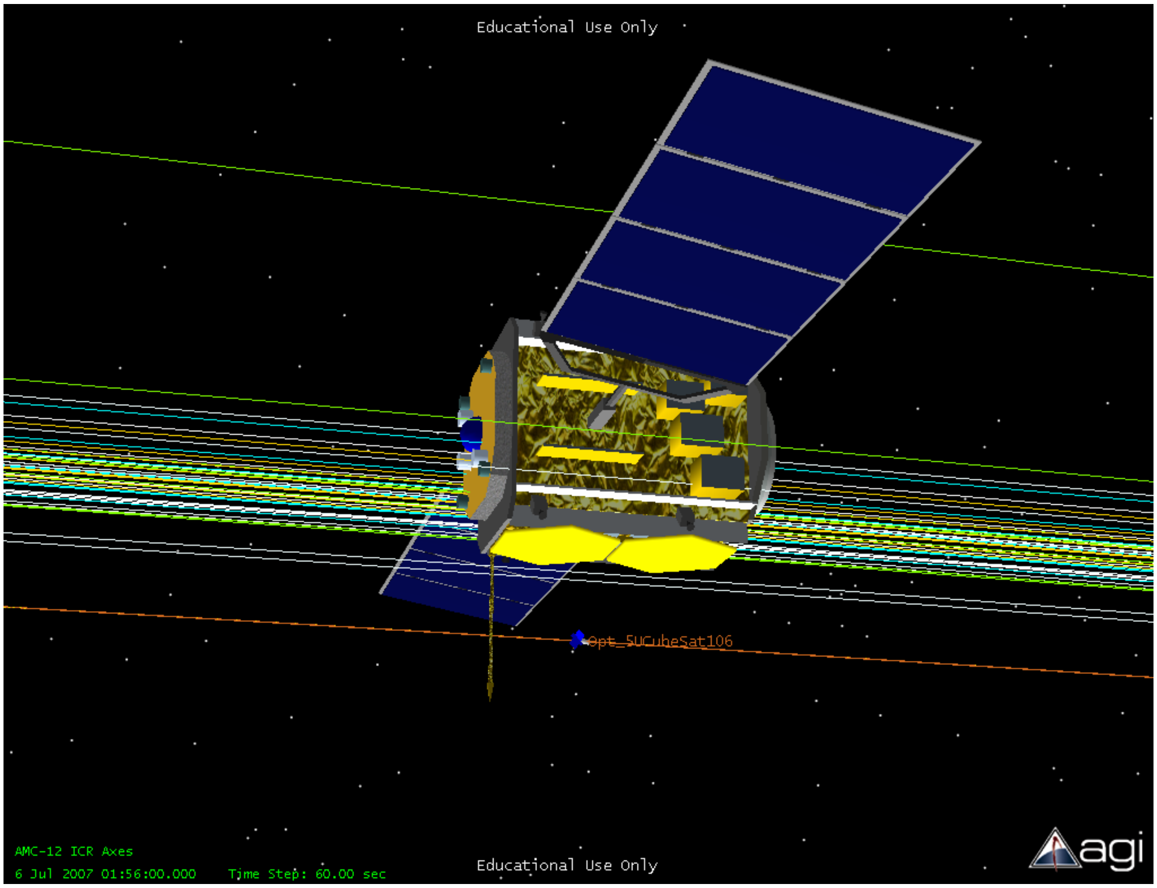

Figure 5. AMC-12 with a 56 kilometer CPA with a 5UCubeSat.

This constellation had a 100 percent coverage rate for the geostationary orbit and an 86 percent coverage rate for the all communications satellites the 30 day period of the simulation.[56]

F. COST ANALYSIS

In a satellite's design, the determination of the cost is perhaps the most frustrating and difficult task. When a constellation of satellites is desired, total system cost becomes more involved and difficult to determine. Ground work first needs to be laid by which a grounded cost estimation can be forged and remain consistent throughout the process. In this examination, a complete cost model will not be developed. Instead a sense for the overall constellation and operation cost will be estimated.

[56] Appendices D and F.

The cost estimation process for the Half-Meter-Cube microsatellite and 5U-CubeSat constellations were identical. Neither program was restricted in any fiscal aspect; an overall cost for each constellation was determined using processes detailed in SMAD with the inclusion of actual component cost when known. Not all component costs were known forcing a hybrid method of a cost estimation to be employed. When cost quotes were known they were converted into fiscal year 2000 (FY00) dollars. Inflation rate forecasts to convert cost to and from FY00 dollars were used from SMAD's table 20-1. Since these two constellations were designed to be operated by DoD and USG employers, the use of government equipment was considered to be free of charge, although the price to pay the work force was estimated. Overall constellation cost determination was utilized using the learning curve percentage method presented in SMAD. All contractor fees, bonuses and potential cost of work stoppages for any reason were not considered. Constellation operations were estimated over the MDL, but constellation re-constitution in the event of earlier satellite failure was not estimated. With this ground work established, a hybrid parametric cost estimation was used to determine over all constellation cost.

Parametric cost estimation is a method of using a "series of mathematical relationships that relate cost to physical, technical, and performance parameters that are known to strongly influence costs. An equation called the Cost Estimating Relationship, or CER, expresses the cost as a function of parameters."[57] This method provides a top-down approach for estimating a system's cost. Applying actual costs when they are known is a method to enhance this process. This hybrid method was applied to keep the cost estimation process "grounded" where it could be applied.

A common sense method for smaller satellite construction is to incorporate COTS equipment which sells regularly, demonstrating a predictable and current cost. Unfortunately, cost quotes, when provided, are a factor of the moment they were delivered. Another consideration is that a satellite may be designed based on an existing COTS component. By the time the satellite is to be built, that component may no longer

[57] Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 787.

be produced. If there is a more advanced unit, it can not be assumed that it will be fully compatible. A relationship between the vendor and the satellite designer must be established early. This relationship may ease the component procurement thorough the satellite's development.

Many manufacturers refuse to quote cost per unit due to various reasons, among which are propriety information, competition, and unwillingness to provide a quote to a non-purchaser. If a quote can not be obtained, then the price must be estimated utilizing a CER. SMAD provides three types of CERs to estimate cost. These are "Estimating Subsystem RDT&E Cost"[58], "Estimating Subsystem Theoretical First Unit (TFU) Cost"[59], and "Cost-Estimating Relationships for Earth-orbiting Small Satellites Including RDT&E and Theoretical First Unit"[60]. The CERs provided by SMAD were developed by the US Air Force and NASA. CERs from these tables were used where appropriate in the cost estimation process.

The first stage used in the cost estimation was to determine the cost of the TFU for each constellation. TFU cost was determined by summing the cost of all known components and estimated component costs including computer code. Once the TFU's cost had been determined, then the costs of subsequent satellites were determined. The cost will depend on the number of satellites to be built, recurring and non-recurring factors. Once these areas are determined, "total production costs for all flight units are computed by multiplying the TFU cost by the learning curve factor"[61]. This allowed the cost for the satellite constellation to be derived. With the cost of the constellation determined, the cost to operate it needed to be estimated over the constellation's MDL.

The operational costs were simplified due to the assumption that the constellations will be operated using current DoD and USG facilitates and systems. A conceptual ground operations scheme was estimated using a common sense approach.

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58 Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 795.

59 Ibid , 796.

60 Ibid, 797.

61 Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 798.

This approach assumed that three government employees and two contractors would be needed to operate 15 satellites during daily eight hour shifts. From this approach, the daily workforce was determined to be six contractors and nine government employees (including military operators) for every fifteen satellites in the constellation. This satellite ground force would need to be employed over a minimum of the constellation's MDL. The estimation assumed that holidays and vacations would not affect the cost of this workforce. The estimation did not consider the time required or extra cost associated with training this workforce. The metrics of this workforce were used to determine the cost for personnel needed to operate and support the constellation throughout its MDL. The life cycle cost was determined by summing the constellation's total cost, ground segment operations, maintenance costs and the cost for launch vehicle integration for each satellite. It did not include potential launch costs, ground station, AFSCN or TDRSS usage fees or storage fees for completed satellites that are awaiting launch.

1. Half-Meter Cube Constellation Cost

The satellite is designed with 75 percent COTS components. For 75 percent of these COTS components I was able to obtain a price quote. I applied the inflation index for FY07 to current the cost of this equipment into FY00 dollars. CER's were used to calculate the price in FY00 dollars for the remaining 25 percent of the COTS components. The custom satellite components, the satellite's structure and computer code were determined solely through the use of CERs.

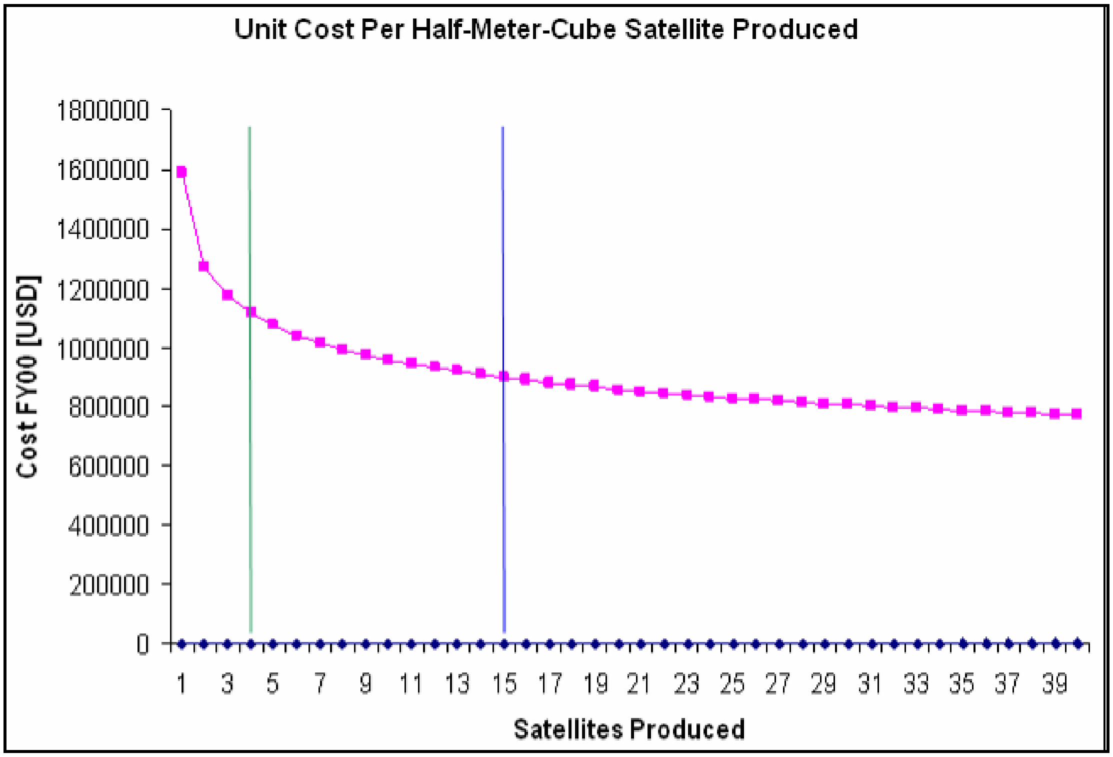

Figure 6. Unit Cost for the Half-Meter-Cube satellite illustrated out to 40 units.

The TFU cost was determined to be 1.6 Million FY00 dollars. A learning curve slope of 90 percent[62] was applied over the cost to build the fifteen satellite constellation. This curve can be seen in Figure 6 is plotted over a projected 40 unit project build. The "knee-in-the-curve" shows the point in the production run in which the cost difference to produce following unit is nearly equal to the cost difference to produce that unit compared to one previous to it. In the figure the green line highlighting the "knee-in-the-curve" and the blue line showing the total number of satellites that need to built to field the constellation.

The total constellation production cost was estimated to be 16 Million FY00 dollars, with an average satellite production cost of slightly more than one Million FY00 dollars. The cost to operate the constellation per year was determined to slightly less than

[62] Wiley J. Larson and James R. Wertz. SPACE MISSION ANALYSIS AND DESIGN. 3rd ed. El Segundo: Microcosm Press, 2005. 809.

two Million FY00 dollars. Overall USG cost for the constellation over its MDL is slightly less than 22 Million FY00 dollars, which translates into about 24.7 Million FY07 dollars.

2. 5U-CubeSat Constellation Cost

The cost determination for the 5U-CubeSat constellation mimicked the method used for the Half-Meter Cube constellation. Of the satellite's components, 73 percent were COTS. Of those COTS components, twenty percent were determined through the use of CERs, while the rest were quoted prices. The custom components were calculated solely through the use of CERs.

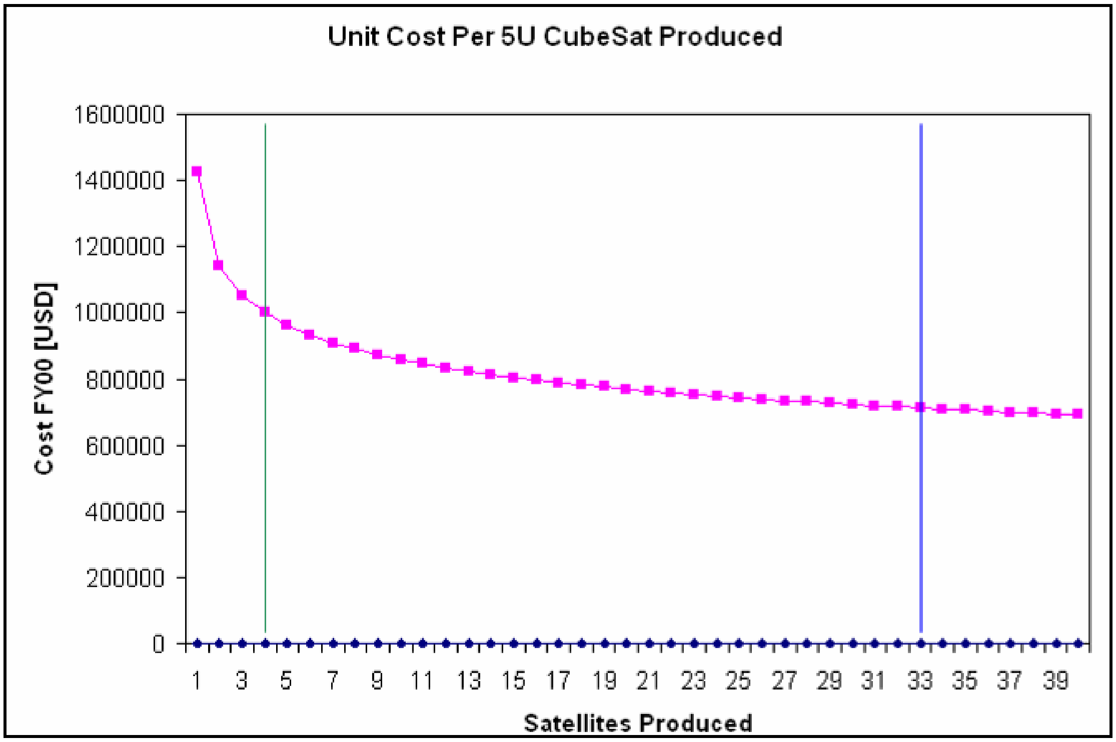

Figure 7. Unit Cost for the 5U-CubeSat illustrated out to 40 units.

The TFU cost was determined to be about 1.5 Million FY00 dollars. A learning curve slope of 90 percent was applied over the cost to build the 33 satellite constellation. This learning curve can be seen in Figure 7 is plotted over a projected 40 unit project build. In the figure the green line highlighting the "knee-in-the-curve" and the blue line showing the total number of satellites that need to built to field the constellation.

The total constellation production cost was slightly more than 33 Million FY00 dollars, with an average satellite production cost of just less than 900 thousand FY00 dollars. The cost to operate the constellation per year was determined to about 4.3 Million FY00 dollars. Overall USG cost for the constellation over its MDL is slightly less than 42 Million FY00 dollars, which translates into just less than 46.5 Million FY07 dollars.

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VIII. POSSIBLE ORBIT INJECTION METHODS

A. PRIMARY MISSION OF A "MOTHER-SHIP" COMMUNICATIONS SATELLITES OPERATING AT GEOSTATIONARY ORBIT

A "Mother-ship satellite" is a satellite designed to carry smaller spacecraft that will eventually be deployed and operate separately. While the mother-ship's primary mission is to carry and then deploy its smaller satellites, it remains disguised as a normal satellite. Communications is a logical disguise for a mother-ship satellite operating at geostationary orbit

The mother-ship orbit injection method has many advantages. A dedicated launch for a Mother-ship satellite could be secured if it carried enough smaller satellites to make it worthy of the launch cost. Disguised as a communications satellite, it could offer an added benefit as a communications relay for the smaller satellites it carries. If the smaller satellites needed to be completely undetected, they could communicate only to the "Mother-ship satellite" over a carrier band frequency that is absorbed by the earth's atmosphere. Even the deployment mechanism for the smaller satellites can be designed cleverly enough so it could double as a minor station keeping propulsion mechanism for the mother-ship. All of these advantages make the "Mother-ship satellite" concept very appealing, but it is not perfect.

The main drawback to this design is that the mother-ship will be stationed at a specific longitude in geostationary orbit. The smaller satellites will therefore need a significant propulsion system to deploy and constitute the desired constellation that is needed to fulfill the optical survey mission. The mother-ship could deploy the Half-Meter-Cubes at their regular 30 day re-visit interval over the four and a half years it takes these satellites to circumnavigate geostationary orbit at their assigned altitude. This timeframe needed to constitute the constellation is more than double the Half-Meter-Cube satellite's MDL. It would take almost six times the MDL to constitute the 5U-CubeSat constellation by this method. To this extent, each smaller satellite would require a kick motor to propel them to form the constellation in a matter of weeks vice years.

This is not possible since each smaller satellite type does not have the power or volume available to support a kick motor and remain undetectable.

Another option is to deploy several "Mother-ship satellites" equally spaced around geostationary orbit. Each would require its own launch vehicle, but they could carry the number of smaller satellites needed to operate in their sector plus a few spares. When all of the "Mother-ship satellites" were operating on orbit they could begin to deploy their smaller satellites until the constellation was established. Four Mother-ship satellites would be needed to deploy the Half-Meter-Cube constellation in just over a year and six Mother-ship satellites to deploy the 5U-CubeSat constellation in just under the two year MDL.

The Mother-ship satellite method does seem possible, though it does have a few drawbacks.

B. SECONDARY PAYLOAD ON A GEOSTATIONARY LAUNCH VEHICLE

STP-1 launched in March of 2007 demonstrating the ability to launch secondary payloads to LEO utilizing the EELV Secondary Payload Adapter (ESPA) ring shown in Figure 8. Each small satellite mated to the ESPA ring via a standard interface called a light-band. The spring loaded light-band also gave the small satellites the necessary velocity to safely separate from the launch vehicle's ESPA ring. On that flight four small satellite secondary payloads were deployed without endangering the primary payload (OE mission) or the launch vehicle. This launch demonstrated that spare mass and volume on a launch vehicle could be made available to secondary payloads in a standard inexpensive manner to LEO.

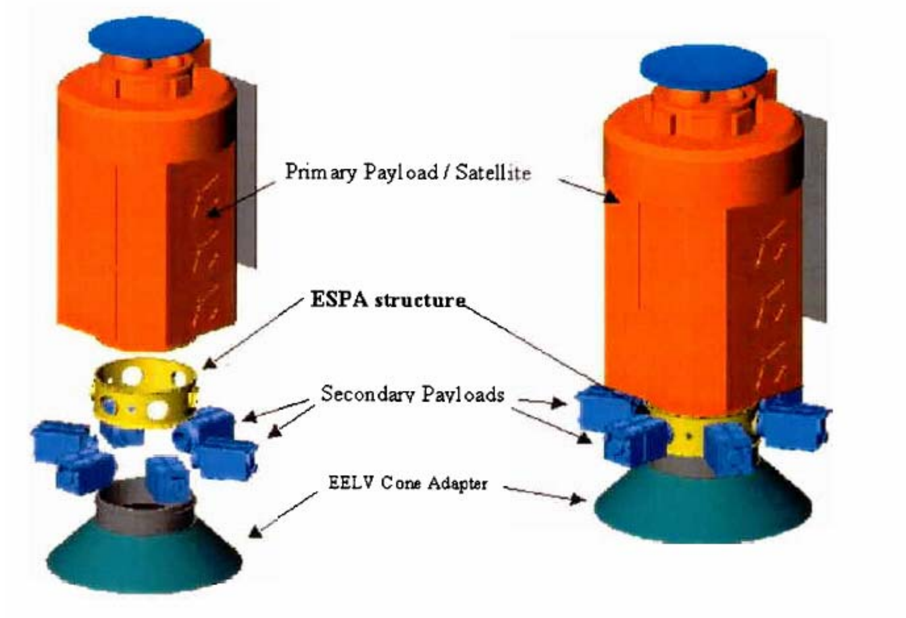

Figure 8. EELV with a fully loaded ESPA ring of six secondary payloads.

Applying such a system to launch vehicles with the performance to deliver large satellites to geostationary orbit seems reasonable. Many launches bound for geostationary orbit likely have mass margin and a fair amount of excess volume inside of the fairing. If this space could be made available to launch secondary payloads, an inexpensive ride to geostationary orbit could be possible. A major obstacle is convincing the program management of the launch vehicle and the primary payload that this type of implementation is an intelligent use of resources and convincing them that it would not endanger the primary payload's ability to reach its assigned orbit.

This type of implementation seems like common sense and a wise use of limited launch infrastructure resources, but we are dealing with a very risk adverse space industry. Each launch that fails in some manner to deliver its payload to the proper orbit, costs the parties involved hundreds of millions, and in some cases billions of dollars. For these fiscal reasons, insurance for these flights are mandatory for commercial launches, and those rates can cost up to 25 percent of the insured cost of the satellite (payload)[63].

[63] Charles M. Racoosin. "Conversation between Charles M. Racoosin and Matthew T. Erdner". 10 September 2007.

The USG self-insures its own dedicated launches. Every launch that fails will cause these rates to increase as well. Launch failures may also cause budget cuts in current space programs forcing their schedules to slip.[64] Risk is a real concern to these professionals, but it also causes advancements in technology and space operations to take a very long time to occur. This society contains a large portion of men and women whom feel best when something is done because it has been done successfully that way in the past.[65] That mentality is not sound reasoning but it does allow them to feel better about the operations they are planning to conduct in the future. New methods are always over analyzed to determine any faults and usually not employed to their fullest extent the first few times they are used. Change in the space industry occurs at a very slow rate and incentives must be used to convince commercial organizations of why they should change for it to occur at all.

A very good reason needs to be identified to change anything in the commercial space field; most of these reasons involve increased revenue. This is not the case in the military space industry. If a satellite project shows that it can deliver a needed space product fulfilling a requirement for a relatively low cost it most likely will get funded. If this project is funded, then a way to place the satellites into orbit will be determined. The most cost effective way would be to lunch them as secondary payloads on geostationary bound launch vehicles with excess mass and volume. This may make the respective program managers unhappy, but if these smaller satellites were to accomplish a critical SSA mission, then the military leaders will determine if this is allowed to happen. They will analyze the entire benefit of the primary and secondary satellites' missions and determine if this is acceptable. What ever their decision the primary satellites program manager will execute the order.

[64] Wayne Eleazer. "The third shoe." 20 February 2006. 11 September 2007. <http://www.thespacereview.com/article/561/1>.

[65] Michael D. Griffin. "Human Space Exploration: The Next 50 Years." 14 March 2007. 11 September 2007. <http://aviationweek.typepad.com/space/2007/03/human_space_exp.html>.

In the 2009 to 2016 timeframe, the military is planning to launch four new communications satellite systems.[66] Each communications system will be composed of at least three geostationary communications satellites. Each satellite will have a dedicated launch to geostationary orbit and it is likely that each launch will have excess mass and volume margin. These launches could be the method to deploy one of the smaller satellite constellations designed.

Assuming that the available mass and volume aboard each of these launches were made available to launch one of these constellations, a few obstacles would still need to be overcome. A standard integration and deployment method would need to be developed that was compatible to these launch vehicles and their primary payload. Assuming that the launch vehicle was a Delta IV or Atlas V EELV, a Centaur could be utilized to deliver the payloads to geostationary orbit. The secondary payloads could be integrated onto the Centaur.[67]. The method could be in the form of an ESPA ring another method that made use of the spare volume inside of the fairing. If this was possible, a very attractive method to launch one of these constellations would exist.

Several launch vehicles will be required to space these smaller satellites around geostationary orbit to allow them to deploy the smaller satellites in an effective manner before the smaller satellites' MDL expired. This could be accomplished if the integrating method packaged the smaller satellites in bundles. Each bundle will need to possess the ability to shield the satellites from the space environment effects while maintaining a safe storage environment until they needed to be deployed and when signaled deploy them. This adds another layer of complexity to the constellation design and development costs.

If this method would be acceptable to the DoD program managers and they made mass and volume available to an acceptable interface method, the constellation's time to orbit (TOO) would be dependant on the schedules of these primaries. It is common knowledge that every recent DoD communications satellite program has faced various

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[66]"Special Report: The USA's Transformational Communications Satellite System (TSAT)." 23 July 2007. 28 August 2007. <http://www.defenseindustrydaily.com/special-report-the-usas-transformational-communications-satellite-system-tsat-0866/>.

[67] Wiley J. Larson and James R. Wertz. <u>SPACE MISSION ANALYSIS AND DESIGN</u>. 3rd ed. El Segundo: Microcosm Press, 2005. 718.

problems that have forced them to delay their launches. This trend does not seem to be getting any better. If this launch method did become available to deliver one of the smaller satellite constellations to geostationary orbit, they would be forced to wait until their respective primaries were ready to launch. This could result in a portion of the smaller satellite constellation having to wait several years on orbit until the remaining components of the constellation could be launched. This would mandate a robust storage system to prevent the components on these smaller satellites from failing before the smaller satellites could operate through their MDL.

This method could be accomplished. Several key factors will need to be overcome for it to be possible, but this method certainly is not impossible. Primary satellite program managers will need to be convinced that the secondary satellites will not induce any risk into their primary satellite's launch. It will add a few more layers of complexity and risk to the program but they can be mitigated through sound engineering. These layers will increase the time to develop the smaller satellites and their eventual deployment will depend on the primary satellites' launch schedules, but it is a viable method to launch one of the smaller satellite constellations.

C. MODULAR ADD-ON TO A GEOSTATIONARY BOUND SATELLITE

Many satellites are launched to orbit with excess mass and volume. If a secondary payload could be notified early enough to prepare for a launch of opportunity, then the launch vehicle's excess performance could be put to use instead of being wasted. When the secondary payload was ready for launch it could then be encased in a standard, modular containment and deployment mechanism. If sufficient volume was available on a primary satellite's bus modular secondary satellite storage and deployment mechanisms could then be bolted to it. Once the launch vehicle had delivered the primary satellite to orbit, the primary could choose to deploy the secondary satellites when it was directed. This method would optimize each launch and help smaller satellites with small budgets to get launched.

This method applied to all launches could allow smaller satellites to ride with the primary satellites all the way out to geostationary orbit. These smaller satellites would not be restricted in only being able to ride with primaries that flew on launch vehicles delivered payloads to geostationary, but also ones that were delivered to geosynchronous transfer orbit (GTO). Riding with the primary would allow these smaller satellites to be delivered to geostationary orbit by the primary satellite's kick motor. Once the primary had reached geostationary orbit it could deploy the smaller satellites when it was directed. Once each satellite was deployed it would use its propulsion system to execute the Hohmann transfers necessary to deliver it to its assigned altitude and assigned RAAN within the constellation.

In the commercial space industry the greatest inhibitor to this method of utilizing this excess mass and volume is the primary satellite's management's aversion to risk. Their primary goal is to place their satellite, the rocket's primary payload, into the satellite's assigned orbit so it can start making them money. To this extent, they do not want to do anything that might jeopardize their goal. Any risk, however small, must be mitigated to some acceptable level. You must convince them of how your satellite will benefit them, which is extremely difficult. This obstacle is not as difficult to deal in the DoD. If your program can deliver the desired results, a program manager for a primary could be ordered to integrate the secondary payload. The risks will be analyzed and methods to mitigate the identified risks will be mitigated to the fullest extent. The secondary payloads will then be fail-safe to the primary and launched.

In the same manner a mind set in the commercial space industry must develop that evaluates the overall benefit of using this excess capacity in a safe manner that delivers the smaller satellites to orbit for this method to be realized. If the benefit of this mission is analyzed in a separate way aside from monetarily, , then it could appear as a worthwhile option. That would let the management of the primary satellite decide if they want to accept this added risk during launch and orbit insertion to obtain the benefit of the mission the smaller satellites and how it would enhance the nation's SSA.

As mentioned earlier, aversion to risk is a programmatic trait that has been bred into US satellite program managers due to past failures and the zero tolerance attitudes for failure which predominate the USG's leadership today.[68] With this type of mentality, architecture to launch smaller satellites in an add-on modular mechanism bolted to a primary satellite's bus, no matter how safe or beneficial is not viewed as a good option. Even if this method has the more benefits than risks, it still induces more risks to the primary satellite. These reasons may mean this is not a desired option for the commercial space industry but maybe in the DoD space industry. The modular add-on method must be accepted by the space community's higher echelon to be employed to deliver secondary payloads to geostationary orbit. It is a viable method, but will depend upon excess volume that primary satellites have available on their bus and the methods utilized to mitigate risk to the primary satellite. This method will invariably induce several more development layers into the satellite program and make its constitution on orbit dependant upon the primary satellites' launch schedules.

[68] Wayne Eleazer. "The third shoe." 20 February 2006. 11 September 2007. <http://www.thespacereview.com/article/561/1>.

IX. CONCLUSION

A. OVERVIEW

This study involved determining if satellites of established dimensions, not a particular payload size, could perform an optical survey mission of the satellites operating near geostationary orbit. The Half-Meter-Cube satellite was chosen since it was just smaller than the smallest object that could be detected at geostationary orbit by current earth based sensors. The 5U and 1U sized CubeSats were chosen due to the increasing appeal of these smaller satellites in space technology research and development fields. Each satellite was designed using a hybrid spacecraft design method derived from the method described by SMAD. The equations in SMAD were still used but applied in a different order. No laws of physics were violated.

The satellites were designed around a set structure size instead of sizing the entire satellite off of the payload needed to conduct the mission. Working through these designs, early in the process it was determined that the 1U-CubeSat could not be designed to perform its assigned mission. The determination was made due to the limited volume of the satellite and resulting inability to include a payload and all necessary sub-systems. The only types of payloads that would fit into the 1U-CubeSat were computer board mounted cameras. Of the COTS payloads analyzed, the C328-7640 JPEG Compression VGA Module performed best, but still would have mandated an orbital altitude 500 meters below geostationary orbit. The only other option was to build a custom deployable aperture similar to the aperture currently being built for the James Webb Space Telescope (JWST) by Ball Aerospace. While it presents a concept to be able to utilize a useful size aperture, it is not currently feasible, nor will it be for the foreseeable future.

The results of this study have shown that two of the three satellite sizes selected could indeed perform the optical survey mission of geostationary satellites. These two constellations were then analyzed to determine if they could meet all mission objectives and also how much each constellation would cost to develop.

For comparison purposes of this conclusion all costs have been translated into current fiscal year dollars. The TFU cost for the Half-Meter-Cube was determined to be about 1.8 million dollars, while the TFU cost for the 5U-CubeSat was around 1.6 Million dollars. The 5U-CubeSat constellation requires 33 satellites, while the Half-Meter-Cube constellation only requires 15 satellites. The overall cost savings to build the Half-Meter-Cube constellation instead of the 5U-CubeSat constellation is almost 22 Million dollars, a huge cost difference for a small program.

Acknowledging the life-cycle cost benefits of the Half-Meter-Cube constellation, each constellation's performance was determined utilizing STK simulation. Each simulation had an identical target set of fifty satellites and their respective smaller satellite constellation. Through these simulations the Half-Meter-Cube constellation was able to imaging 100 percent of the satellites with an inclination less than 0.086 degrees and 94 percent of all the satellites in the simulation. The 5U-CubeSat constellation was able to image 100 percent of the geostationary satellites, while only 86 percent of all the satellites in the simulation. The larger optical payload in the Half-Meter-Cube satellite constellation gave them a true advantage allowing them to out perform the 5U-CubeSat constellation.

From the analysis conducted, the Half-Meter-Cube constellation and the 5U-CubeSat constellation both met all requirements set forth to effectively conduct the optical survey mission. The value of the Half-Meter-Cube constellation was superior to the 5U-CubeSat constellation in its ability to conduct the optical survey at a lower constellation cost. The only metric that was not used to compare them was how exactly these constellations could be injected into their assigned orbits.

B. ORBIT INJECTION METHODS

The most difficult problem facing the constitution of either satellite constellation is getting the satellites to orbit. With the "Launch infrastructure currently tailored for large spacecraft."[69]it is very difficult to launch smaller satellites. For this reason

[69] John Brock. Operational Utility of Small Satellites. SAB Summer Session, 28 June 2007. Slide 21.

launching smaller satellites to geostationary orbit is less practical than large more expensive satellites. Launching smaller satellites to geostationary orbit is nearly impossible utilizing commercial launch services for secondary payloads, but DoD launches could be made available if the program can prove its value. If the program is deemed worthy any of the methods discussed earlier could be impossible.

The mother-ship method seems to be the most appealing method due to the nature of the launch. It requires a few dedicated launches for the mother-ships and their secondary payloads, the smaller satellites. The mother-ship is the primary payload for the launch vehicle, there is not another primary satellite to be endangered. The launch schedules could be slated to deliver the mother-ship to their assigned geostationary orbital slots in a timely manner that would allow them to deploy their smaller satellite payloads in an effective manner to constitute the smaller satellite constellation with an optimal timeframe. The smaller satellites could then begin to operate on their own mission's timeframe.

Of the two remaining methods to deliver the smaller satellites to geostationary orbit, the secondary payloads on a geostationary bound launch vehicle seem to be the most promising. That method constitutes the lowest risk to a primary satellite since it is only integrated into the launch vehicle. Therefore the risk it needs to mitigate only exists during launch. The secondary payloads could even be partitioned away from the primary so there is no way a secondary could ever bump into the primary. The modular add-on method being bolted to the primary's bus would induce numerous potential risks to the primary satellite. The methods to mitigate these risks would be more involved than the just integrating to the launch vehicle, so this method is less likely to occur but not impossible. Each of these methods is much more likely to be accepted by the DoD to deliver viable secondary payloads to orbit than in the commercial space industry.

The Air Force's goal is to have "~20 domestic Small Sat launches/payloads per year by 2015."[70] The Air Force does not state its exact method for these launches, but has made it their goal. This goal seems achievable by 2015 if the development of non-

[70] John Brock. Operational Utility of Small Satellites. SAB Summer Session, 28 June 2007. Slide 27.

traditional launch service providers such as SPACEX and Rocketplane Kistler mature at their projected current schedule. This will open up more launch sites and launch vehicle options that will hopefully solve some of the launch problem that currently exists in the US, however, these solutions do not address the problem of establishing a method by which smaller satellites can reach geostationary orbit without a dedicated launch.

Even though the Mother-ship concept would need its own dedicated launch, it may be the reason the program would not be funded. For this type of small program, the launch vehicle would cost four to five times the cost of the entire smaller satellite constellation. From that standpoint, it would be very difficult to convince the JROC to approve it unless they were able to view the operational gains of the project instead of the launch cost eclipsing the cost to build the constellation of smaller satellites. Cheaper satellite than their launch vehicles have been launched in the past, but it still could be a hard sell.

For this reason, the modular add-on method for orbit injection seems the most viable option at this time. It offers several launch opportunities per year onboard geostationary communications and weather satellites that are launched with excess mass and volume margin. This method does not involve modifying the launch vehicle or the primary's deployment in any way. For these reasons, the launch vehicle's overall reliability and operation will not be impacted. Although risk will be mitigated to it fullest extent, the only party that would be impacted is the primary satellite. Even though this method seems to be the most simplified from a common sense approach, it will not be easy to impose.

The true obstacle that must be overcome is the protective nature of the primary satellite's management. This can be overcome within the DoD, but not easily in the commercial sector. If those programs can be convinced that this type of project is worth the added risk, then this could be a very viable option to populate a geostationary belt observation constellation. In the near future there should be a large variety of satellites to ride along with.

> The TCA envisions a Global Information Grid (GIG) that includes the Wideband Global SATCOM (WGS) for unprotected wideband, the Mobile User Objective System (MUOS or next generation narrowband) scheduled for launch in 2009, the Advanced Extremely High Frequency (AEHF next generation protected, a.k.a. Milstar III) to be launched between 2008-2011, an Advanced Polar System for various strategic missions, and the Transformational Communications Satellite (TSAT) system that could be launched from 2013 as a major upgrade, instead of deploying AEHF #4 & 5.[71]

Any or all of these satellites could be the vehicle to transport a constellation of smaller satellites to operate near geostationary orbit. The DoD would simply have to instruct those respective program managers to integrate the necessary secondary payloads.

C. POSSIBLE SURVEY MISSIONS VARIANTS

The requirements for the survey mission were established to allow the entire geostationary belt of satellites to be imaged every 30 days over two years. A primary requirement in addition to the survey mission was that the imaging satellites were to remain undetectable by earth based optical and radar sensors. While these overarching requirements require a complete constellation, many benefits can be obtained by launching a partial constellation, a single satellite with a specific mission or even launch a complete constellation over time and using the satellites as they arrive to orbit.

The major drawback is the cost in schedule and dollars for launching a complete constellation. If the constellation was launched in a segmented fashion, partial effects could be delivered as spacecraft are launched. It would allow immediate results as soon as some of the satellites could reach their operating altitudes. This method may prevent the entire constellation from ever being established during the satellites' MDL. It does provide a reliable means to gather some results as quickly as possible. The launch

[71] "Special Report: The USA's Transformational Communications Satellite System (TSAT)." 23 July 2007. 28 August 2007. <http://www.defenseindustrydaily.com/special-report-the-usas-transformational-communications-satellite-system-tsat-0866/>.

segment and satellite insertion remains the most difficult problem, but this method allows immediate operations, vice a potential long storage on orbit until the entire constellation could be constituted.

A partial constellation deployed from a single launch can produce some results that have the potential to be very useful. A partial constellation may consist of two or more satellites that could have an identical mission to the one described previously with the exception of the 30 day target re-visit rate. This would require the satellites to possess the same spacing required previously for a completer constellation. A semi-constellation of this type could be used to image a section of high interest targets over their two year mission life.

From a purely hypothetical approach, assume the high interest targets were operating over Asia, with a number of satellites operating from 87.5 to 122.0 degrees east longitude along the geostationary belt. That region represents 34.5 degrees, or slightly less than ten percent of the geostationary belt. If the orbital spacing and altitudes were the same as for the complete constellations, then seven Half-Meter-Cube satellites and six 5U-CubeSats would be required to complete this mission.[72] Each respective partial constellation would need to be seeded properly so that the first satellite in each would complete its inspection of the final satellite of interest at the end of the 30 day re-visit period. At that point the next satellite in the partial constellation should be at the position in which the first satellite started its operations. These partial constellations would be able to perform this type of mission, but delivering them to their specific orbital slots would still be difficult.

A single satellite of each constellation type would be able to perform two different variants of the survey mission. This application would also alleviate the need for multiple launches. A single satellite could be launched in any of the previously mentioned methods and would not encounter the potential difficulties of having to wait for years until the entire constellation could reach orbit for it to begin operations. The first variety is to survey a single satellite. It could be launched into an orbit of any

[72] Appendix G.

altitude and inclination that was needed to observe the target of interest. The other method is an extension of the single target satellite option, which could include a targeted region only using one imaging satellite. This single satellite could be inserted into an orbit to observe as many targets as possible until it could no longer operate. A satellite of this design could certainly have some utility to the DoD.

Another option is to openly conduct the mission. This type of mission would no longer carry the dimension size constraint, which allows for larger optics to be carried by the satellite's payload. In that manner a small satellite utilizing COTS equipment could be build on a small budget. An IKONOS-like satellite could be built and then operated at a 350 kilometers sub-geostationary orbit to obtain a 20 centimeter spatial resolution of targets. At this altitude the IKONOS-like satellite would circumnavigate the geostationary belt in 233 days. Four of these satellites would be needed to constitute an equally spaced, non-inclined constellation at a 30 day re-visit rate. This type of constellation would also discourage our enemies from placing convert objects in the geostationary belt, or at a minimum let them know the USG's capabilities to observe such an action. A project such as this may gain public support, without it though, it may be impossible to build. Regardless this type of satellite would be a solution to the current SSA need.

D. FOLLOW-ON RESEARCH

As with any study, this one could certainly be continued and improved upon. The following sections list a few avenues that could be pursued to determine a better solution for this area of SSA.

1. Realistic STK Simulation

The foundations for the survey mission were established to allow the entire geostationary belt of satellites to be imaged every 30 days over two years. These requirements were the foundation for a STK simulation. The STK simulation created and utilized for this study was sufficient, but it was not a realistic model. Simplification was required for the simulation to run on computing hardware that was available for this study. The simplified STK simulation was contained to about one-fifth of the satellites[73] that are known to operate near geostationary.

If a realistic simulation could be conducted using a more powerful computer then it would be possible to create a scenario that would include every known geosynchronous and geostationary satellite using the most up to date TLEs. A simulation of this type could allow the researcher to determine the actual coverage of the constellation and then optimize the constellation's orbital characteristics. An optimization of this would provide an accurate number of images capture per day per satellite. That information would give a complete understanding of the link budget demands for each satellite of the selected constellation. With this information an accurate impact to the AFSCN and TDRSS systems could be determined. Then the true impact upon these systems could be determined. This would determine if this type of imaging constellation could be supported by the AFSCN and TDRSS in their current form or if another data relay or ground station network would need to be utilized, augmented or created.

2. Satellite Re-Design

An individual could explore satellite designs utilizing a customized component approach to the design of each subsystem. This may produce some interesting results from a performance and cost analysis.

Following current technology development trends components should continue to shrink and more useful in the future. This may allow each size of satellite to become much more useful and even incorporate redundancy in key system to increase MDL. A complete satellite re-design in the future may be necessary to take advantages of these technologies. Such a re-design may come to different conclusions than the author's current conclusion and offer a better solution to this area of SSA.

[73]"Commercial Communications Satellites Geosynchronous Orbit". 30 June 2006. 12 June 2007. <http://www.boeing.com/defense-space/space/bss/launch/980031_001.pdf>.

3. Other Missions That Could Be Performed

With technology evolving newly developed COTS components may enable other missions to be conducted by smaller satellites near geostationary orbit. Missions such as service denial and anti-satellite missions are very appealing. Science missions such as space physics and space weather may not be as glamorous but are equally practical and important.

Sparse apertures are becoming more relevant today at they continue to attract attention and research. In the not so distanced future, smaller satellites will be used in this construct. Most likely initially demonstrated at LEO they will ultimately be deployed in geostationary orbit. Such applications to smaller satellites may one day enable the USG to possess high resolution, long dwell imagers are geostationary orbit.

A Space Operations student could even begin now by devising a CONOPS for any such mission. Even though the technology is not currently developed, a CONOPS can help determine if the validity of the application. Researchers may gain insight from such a study, which may allow them to avoid potential pitfalls exploring equipment that could never be utilized or employed.

4. Casting, Development of Launch Architecture Given Improved STK Analysis

The largest challenge to fielding a constellation of smaller satellites near geostationary orbit is getting the satellites to orbit. This challenge is due to fiscal constraints and our current launch infrastructure. With a realistic STK simulation created, and an optimized smaller satellite constellation devised a method could be formulated on how to properly populate this constellation. The method would need to detail which orbital regime would be populated, the method to transport these smaller satellites and the manner in which the smaller satellites would reach their assigned positions in the constellation. After these characteristics are determined a comparison between partial and full constellation performance would determine how best to utilize these satellites as they reached orbit. If that comparison determined that only the complete constellation effects are desirable, then a determination of the maximum on-orbit storage time would need to be estimated so that the satellite's MDL would not decease. If a partial constellation's effects were desirable, then a cost versus benefit evaluation would allow the number of useful satellites be determined. With these evaluations, the cost to develop a constellation of smaller satellites and operate them over their MDL could be accurately estimated.

5. The Unforeseen

With any study that has been completed, the future is unknown. Someone reading this thesis may be inspired in someway that the author can not predict. It is the author's hope that this study will inspire a reader to look at the aspect of SSA in a new innovational light that will allow the USG to better accomplish this mission. Whatever that could be, the author looks forward to its development.

APPENDIX A. HALF-METER-CUBE-SATELLITE DESIGN EXCEL WORKBOOK

Mission Reqmts	N/A	~~N/A~~	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Sub-GEOSTA Altitude [km] 35734.64	L/V	~~N/A~~							Sep mass [kg] 33.71
~~GEO station~~	~~N/A~~	~~AKM~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~Propellant [kg]~~
Thrust for GEO/Inc. and deorbit	N/A	~~N/A~~	RCS	3-axis stabalized		despin on station & S/C pointing			Total S/C mass [%] 15.00
BER 10E-6 TM & Data @ 300kbps with BPSK modulation	Sub-GEOSTA	~~N/A~~	Station keeping	P/L & TT&C		S/C Point Reqs. [degrees] **0.81**			Total S/C mass [%] 9.60
Li-ion 80%DOD 2 year life, %eff 70	Elect. Interface req'd	~~N/A~~	50W µVAT actuation 10.6W R.W's + Rate Sensor	10.9Vdc @ 26.4Vdc 15W TT&C and Data, 5W C&DH	EPS	power for sensors and computers	4 body solar panels each 0.25m² produce 57.5W - 4 batteries (105Wh)		Total S/C mass [%] 44.50
Mom-bias with minimum Zcp [cm] -0.15	N/A	~~N/A~~	ISP [sec] 1500	3-axis stab mom.	Point solar arrays perp to sun to charge batteries	ADCS	N/A	N/A	Total S/C mass [%] 3.00
Weight << Shuttle GTO sep mass 2 year MMD		~~N/A~~	Dynacon MicroWheel200 X-axis, Y-axis, Z-axis, Spare		Body Mounted Solar Panels	Star Tracker, Sun and Earth sensors inputs to onboard orbit propagator	SMS	Radiator space [m²] 0.121	Total S/C mass [%] 25.00
Worst Case Hot 333[K] Worst Case Cold 273[K]	N/A	~~N/A~~	Reaction Wheels and Rate Sensor 243K to 333K	CCD Operating Limits 273K to 333K	Batteries parameters (must not freeze or overheat) 253K to 333K	243K to 353K	N/A	TCS	Total S/C mass [%] 7.80
0.5m x 0.5m x 0.5m >=90[kg] sep mass **L/V dependant adapter fairing**	High risk if failure of L/V, but high reliability L/V	~~N/A~~	Reaction Wheels and Rate Sensor [kg] **3.40**	Custom Optics Package +CCD [kg] **5.34**	Solar arrays and Batteries [kg] **2.73**	Star Trk, Sun & Earth Sensors [kg] **1.14**	Aluminum 0.5m cube [kg] **4.50**	Temp Ctlr, Kapton Htrs [kg] **1.83**	Total Mass with Margin [kg] **33.71**
Cost	**Launch Vehicle** **$0**	~~**Apogee Kick Motor**~~ **$0**	**Reaction Control System** **$23,065**	**Paylaod & TT&C** **$703,275**	**Electrical Power System** **$110,610**	**Attitude Control System** **$460,563**	**Structures and Mechanisms** **$1,983**	**Thermal Control Systems** **$9,270**	**Average Build Cost per Satellite** **$1,052,928**

MicroSAT		**Units**	**Comments**
Overall Mission	Optical Survey		
Desired CPA from Target	51.2	km	
Orbital Radius	42112.8	km	Adjust N2 chart when this is changed; GEOSTA radius is 42155[km]
Orbital Altitude	35734.6	km	
SC Orbit Insertion Inclination	0.0	deg	
Targets' Altitude	35785.9	km	Assuming all targets will be at exactly Geostationary orbit
Targets' Inclination	0.0	deg	Assuming all targets will be at exactly Geostationary orbit
Max S/C Mass (Estimate)	33.7	kg	
Mission Design Life (MDL)	2	years	

Major System components

Payload -	Custom Optical Telescope Package using Kodak sqaure matrix CCD KAF-39000
Batteries -	SAFT MP176065 Integration
Solar Cells -	Spectrolab UTJ (GaInP2/GaAs/Ge)
Command & Data Handling -	AFRL RAD6000 Computer (Microprocessor)
Propulsion/Thrusters -	Micro Aerospace Solutions Vacuum Arc Thrusters (VAT)
Stability Control -	Dynacon MicroWheel 200 (3.2[W] max with rate sensor)
Data Transciever -	AeroAstro Modular S-Band Radio System
Star Tracker -	AeroAstro Miniature Star Tracker
Inertial Control Unit -	*Imbedded into Command & Data Handling CPU using inputs from rate sensor, cmds to RWs.*
Power Control Unit -	Reconfigurable Fault Tolerant Processor configured for power management, back-up photo processing and attitude control.
Earth Sensor -	Optical Energy Tech
Sun Sensor -	Optical Energy Tech
Thermal Control System -	Minco CT325 Thermal Control Module, Thermal coatings, Radiator, Heat Pipes, Multi-layer Insulation (MLI), Kapton Heaters
Antenna -	2x2 Microstrip Array Antenna
TT&C Transciever -	AeroAstro Modular S-Band Radio System

Estimated Spacecraft Design Characteristics:

Parameter	MicroSAT	Units	Equation	Comments
Earth radius (Er)	6378.137	km	base*height*width	
Orbit Radius (Or)	42113	km		Selected
Earth Angular Radius (OAr)	0.15	Rad	ASIN(Er/Or)	
Payload:				
Mass (P/Lm)	5.34	kg		P/L scaled from IKONOS on Optical P/L sheet
Power (P/Lp)	10.94	W		P/L scaled from IKONOS on Optical P/L sheet
Spacecraft:				
Dry Mass (Dm)	19.50	kg	P/Lm/0.274	
Average Power (Ap)	24.31	W	P/Lp/0.45	
Orbit period (Op)	23.89	hr	2*π*SQRT(Or^3/μ)/3600	
Eclipse Period (Te)	1.16	hr	Op*ACOS(COS(EAr)/COS(0))/π	
Solar Array Power	32	W	(Ap*Te/0.6+Ap*(Op-Te)/0.8)/(Op-Te)	Estimate of needed power
Solar Array Design	E-W Trkg			S/C uses reaction wheels to track sun
Control Approach	*3- axis nadir pointing*			
Propellant:				
ΔV ~m/s	10.00	m/s		<-This is only accounting for S/C disposal
Mp~kg	0.01	kg	Dm*(EXP(ΔV/Isp/g)-1)	
Attitude control + residuals	0.00	kg	Mp*0.07	
Margin	0.00	kg	(Mp+AttitudeCtrl)*0.15	
Total propellant	**0.02**	kg	Mp+AttitudeCtrl+Margin	
Propulsion Isp	1500	sec		Propulsion system property'
Spacecraft loaded mass	19.52	kg	TotalPropellant+Dm	
Spacecraft size and MOI				
Volume	0.125	m^3	base*height*width	S/C Property since it's a cube
Linear dimensions	0.500	m		Choosen
Body x-sectional area	0.250	m^2	Ld^2	
MOI	**1.415**	kg*m^2		
		N.m*s^2		

Constants

μ = 398600.44

g = 9.80665

Ref: SMAD p.303-317

Spacecraft Mass Estimation

Element	Estimated % of Dry SC %	Payload %	Est mass based on Dry SC kg	Actual Mass based on Selected Equipment kg	
Payload	20.00	100.00	5.34	5.34	
Structures	25.00	125.00	4.88	4.50	Aluminum Cube Structure, no bulkheads.
Thermal	7.80	39.00	1.52	1.83	TempCtrlModule+KaptonHtrs+Radiator+MLI+Coatings
Power	8.90	44.50	1.74	2.73	Batteries+SolarArrays
TT&C	9.60	48.00	1.87	3.06	Antenna+Transceiver+MiscWiring(etc)
ACS	3.00	15.00	0.59	1.14	SunSensors+EarthSensors+StarTracker
Prop (dry)	13.00	65.00	2.54	4.00	
Reaction Control System	3.00	15.00	0.59	3.40	ReactionWheels+MEMs(RateSensors)
Margin [kg]	9.70		0.45		
SC dry [kg]			19.50	26	
Prop mass [kg]			0.02	0	
SC loaded [kg]			19.52	26	SC no Margin
Margin % Dry SC			2.30	28.60	SC Mass + 10%Margin

Spacecraft Selected Equipment

Element	Actual Dry Mass [kg]	Actual % for Dry [%]	
Batteries (1-4)	0.58	1.9%	4 Batteries, 146[g] each
Command & Data Handling	1.50	4.9%	AFRL RAD6000 Computer with margin to account for harnessing and other structures
Reaction Wheels	3.08	10.1%	Dynacon MicroWheel 200
Data & TT&C Radio System	0.80	2.6%	AeroAstro Modular S-Band Radio: Rx/Tx/HPA/Pwr&Inter Modules
Data Handling & TT&C Antenna	1.75	5.7%	2x2 MicroStrip Array
Inertial Reference Unit	0.32	1.0%	MEMS embedded into Dynacon MicroWheel 200
Payload	5.34	17.4%	Custom Optical package
Power Control Unit/RFTP	0.28	0.9%	ThermoFoil CT325 Miniture DC Controller
Solar Arrays	1.98	6.5%	4 body mounted UTJ solar arrays
Sun Sensors	2.40	7.8%	Includes 6 Sensors at 0.04 kg each
Earth Sensors	0.60	2.0%	Includes 6 Sensors at 0.001 kg each
Propulsion Unit	4.00	13.1%	Micro Aerospace solutions μVAT
Star Tracker	0.30	1.0%	AeroAstro Miniature Star Tracker
Radiator	0.40	1.3%	Area*3.3[kg/m2]; SMAD Table 11-49
Structures	4.50	14.7%	Aluminum Cube with no bulkheads
TT&C Miscellaneous	1.28	4.2%	Coax cables, Filters, Switchers & Diplexers
Thermal	1.43	4.7%	4% of 500 kg spacecraft
UDI	0.10	0.3%	Uniformly Distributed Items
SC (No Margin)	30.64	100.0%	
Margin	0.10	10.0%	
Mass Margin	3.06	10.0%	
SC (With Margin)	**33.71**	110.0%	

Spacecraft Mass of Subsystem Categories

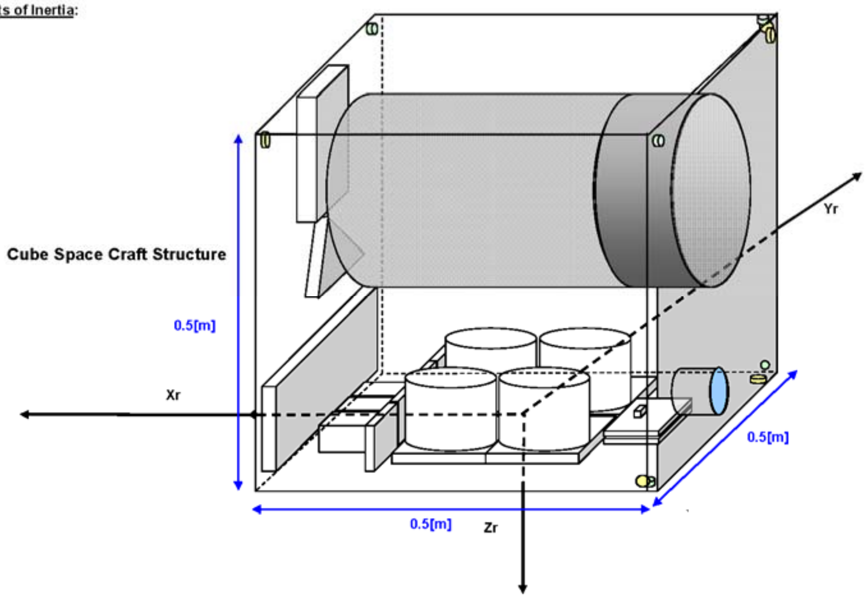
	[kg]	[% of SC]
RCS Subsystem	4.00	11.87%
Payload Subsystem	5.34	15.85%
Comms & TT&C Subsystems	5.33	15.80%
Electrical Power Subsystem	2.84	8.43%
ACS Subsystem	6.70	19.88%
Structures	4.50	13.35%
Thermal Control Subsystem	1.83	5.43%
Uniformly Distributed Items	0.10	0.30%
10% Margin (of dry spacecraft)	3.06	10.00%
SC Mass with Margin	33.71	110%

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p.341

Moments of Inertia:



1. Identify S/C components and their mass
2. Find component MOI's WRT their individual mass centers Ixxo,Iyyo,........
3. Establish a convenient XrYrZr REFERENCE frame and find Xr, Yr, Zr of each components mass center
4. Find the S/C center of mass WRT the Reference frame (Xcm, Ycm, Zcm)
5. Find: Xcm-Xr, Ycm-Yr, Zcm-Zr for each component (i.e., component distance from S/C cm)
6. Use the MOI parallel axis x-fer ; e.g., Ixxcm=Ixxo+M*[(Ycm-Yr)^2+(Zcm-Zr)^2]
7. Sum S/C MOIs

Step 1 - Major Component Masses

Item	Name	Shape	Mass		Component MOI equations
1	Battery 1	box	0.146	kg	for boxes Depth (d) in x, Width (w) in y, Height (h) in z
2	Battery 2	box	0.146	kg	Ixo= Iyo =(1/12)*M*(3(R^2 + r^2) + h^2); Izo=(1/2)M(R^2 + r^2)
3	Battery 3	box	0.146	kg	Ixo= Iyo =(1/12)*M*(3R^2+h^2); Izo=(1/2)MR^2
4	Battery 4	box	0.146	kg	Ixo=(1/12)*M*(w^2+h^2); Iyo=(1/12)*M*(d^2+h^2);Izo=(1/12)*M*(d^2+w^2)
5	Side 1 (front w/Ap)	solid sheet	0.366	kg	0.5x0.5[m] Aluminum 7075-T73 - 0.4[m] Aperature for P/L +5% margin
6	Side 2 (back w/Rad)	solid sheet	0.491	kg	0.5x0.5[m] Aluminum 7075-T73 - Radiator AreaµVAT area - MicroStrip Antenna +5% margin
7	Side 3 & Solar Array	solid sheet	1.254	kg	0.5x0.5[m] Aluminum 7075-T73 +0.24[m^2] Square Spectrolab UTJ solar array+ 5% margin
8	Side 4 & Solar Array	solid sheet	1.254	kg	0.5x0.5[m] Aluminum 7075-T73 +0.24[m^2] Square Spectrolab UTJ solar array+ 5% margin
9	Side 5 & Solar Array	solid sheet	1.254	kg	0.5x0.5[m] Aluminum 7075-T73 +0.24[m^2] Square Spectrolab UTJ solar array+ 5% margin
10	Side 6 & Solar Array	solid sheet	1.254	kg	0.5x0.5[m] Aluminum 7075-T73 +0.24[m^2] Square Spectrolab UTJ solar array+ 5% margin
11	C&DH Proc	box	1.500	kg	AFRL RAD6000 Computer with margin to account for harnessing and other structures
12	RFTP/Power Control Unit	box	0.280	kg	RFTP Processor stacked on top of PowerPC
13	"X" Reaction Wheel&IRU	box	3.400	kg	Dynacon MicroWheel 200 with IRU
14	"Y" Reaction Wheel&IRU	box	3.400	kg	Dynacon MicroWheel 200 with IRU
15	"Z" Reaction Wheel	box	3.400	kg	Dynacon MicroWheel 200
16	Spare Reaction Wheel	box	3.400	kg	Dynacon MicroWheel
17	Temp Controller	box	0.028	kg	CT325 Miniature DC Controller
18	C&DH/TT&C Antenna	box	1.750	kg	Custom 2x2 Wideband MicroStrip Array Antenna
19	Payload	solid cyl	5.344	kg	Custom Optical P/L utilizing Kodak KAF-39000 CCD with integrated sunshade
20	Star Tracker	solid cyl	0.300	kg	AeroAstro Miniture Star Tracker
21	Sun Sensor 1	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
22	Sun Sensor 2	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
23	Sun Sensor 3	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
24	Sun Sensor 4	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
25	Sun Sensor 5	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
26	Sun Sensor 6	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
27	Earth Sensor 1	solid cyl	0.100	kg	OET Earth Sensor
28	Earth Sensor 2	solid cyl	0.100	kg	OET Earth Sensor
29	Earth Sensor 3	solid cyl	0.100	kg	OET Earth Sensor
30	Earth Sensor 4	solid cyl	0.100	kg	OET Earth Sensor
31	Earth Sensor 5	solid cyl	0.100	kg	OET Earth Sensor
32	Earth Sensor 6	solid cyl	0.100	kg	OET Earth Sensor
33	Transciever&HPA Module	box	0.400	kg	AeroAstro Modular S-Band Radio
34	Receiver Module	box	0.300	kg	AeroAstro Modular S-Band Radio
35	Interface/Power Module	box	0.300	kg	AeroAstro Modular S-Band Radio
36	µVAT	solid tri	4.000	kg	Micro Aerospace Solutions µVAT sized for 50[kg] SC
37	Radiator	solid rec	0.400	kg	Sized from Thermal needs; mass calculated using SMAD table 11-49
38	TT&C Miscellaneous	UDI	1.275	kg	Includes Coax cables, Filters, Switchers and Diplexers
39	Thermal Miscellaneous	UDI	1.430	kg	Evenly distributed: Heat pipes, heaters and coatings
40	Miscellaneous	UDI	0.101	kg	Uniformly distributed Items throughout SC
		Σ SC Mass =	40.47	kg	*Slightly lighter than SC Mass due to bulkhead calculations of actual size.
		Σ SC Mass w/ Margin at 10% =	44.51	kg	

Step 2 - Component Moment of Inertia about Center of Mass R is outter radius, r is inner radius

Item	Name	Shape	d (x) or R	w (y) or r	h (z)	Ixxo	Iyyo	Izzo
1	Battery 1	box	0.0192	0.0600	0.0684	0.00010	0.00006	0.00005
2	Battery 2	box	0.0192	0.0600	0.0684	0.00010	0.00006	0.00005
3	Battery 3	box	0.0192	0.0600	0.0684	0.00010	0.00006	0.00005
4	Battery 4	box	0.0192	0.0600	0.0684	0.00010	0.00006	0.00005
5	Side 1 (front w/Ap)	solid sheet	0.0100	0.5000	0.5000	0.01523	0.00762	0.00762
6	Side 2 (back w/Rad)	solid sheet	0.0100	0.5000	0.5000	0.02046	0.01023	0.01023
7	Side 3 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.02614	0.05226	0.02614
8	Side 4 & Solar Array	solid sheet	0.5000	0.5000	0.0100	0.02614	0.02614	0.05226
9	Side 5 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.02614	0.05226	0.02614
10	Side 6 & Solar Array	solid sheet	0.5000	0.5000	0.0100	0.02614	0.02614	0.05226
11	C&DH Proc	box	0.0700	0.1200	0.0100	0.00181	0.00063	0.00241
12	RFTP/Power Control Unit	box	0.0700	0.1200	0.0100	0.00034	0.00012	0.00045
13	"X" Reaction Wheel&IRU	box	0.0940	0.1020	0.0890	0.00519	0.00475	0.00545
14	"Y" Reaction Wheel&IRU	box	0.0940	0.1020	0.0890	0.00519	0.00475	0.00545
15	"Z" Reaction Wheel	box	0.0940	0.1020	0.0890	0.00519	0.00475	0.00545
16	Spare Reaction Wheel	box	0.0940	0.1020	0.0890	0.00519	0.00475	0.00545
17	Temp Controller	box	0.0038	0.0254	0.0277	0.00000	0.00000	0.00000
18	C&DH/TT&C Antenna	box	0.1900	0.0100	0.1900	0.00528	0.01053	0.00528
19	Payload	solid cyl	0.235	0.0001	0.25	0.00208	0.00392	0.00184
20	Star Tracker	solid cyl	0.054	0.054	0.076	0.00029	0.00029	0.00019
21	Sun Sensor 1	solid cyl	0.0001	0.015	0.01	0.00001	0.00000	0.00001
22	Sun Sensor 2	solid cyl	0.0001	0.015	0.01	0.00001	0.00000	0.00001
23	Sun Sensor 3	solid cyl	0.015	0.0001	0.01	0.00000	0.00001	0.00001
24	Sun Sensor 4	solid cyl	0.015	0.01	0.0001	0.00000	0.00001	0.00001
25	Sun Sensor 5	solid cyl	0.015	0.0001	0.01	0.00000	0.00001	0.00001
26	Sun Sensor 6	solid cyl	0.015	0.01	0.0001	0.00000	0.00000	0.00000
27	Earth Sensor 1	solid cyl	0.0001	0.0135	0.02	0.00000	0.00000	0.00000
28	Earth Sensor 2	solid cyl	0.0001	0.0135	0.02	0.00000	0.00000	0.00000
29	Earth Sensor 3	solid cyl	0.0135	0.0001	0.02	0.00000	0.00000	0.00000
30	Earth Sensor 4	solid cyl	0.0135	0.02	0.0001	0.00000	0.00000	0.00000
31	Earth Sensor 5	solid cyl	0.0135	0.0001	0.02	0.00000	0.00000	0.00000
32	Earth Sensor 6	solid cyl	0.0135	0.02	0.0001	0.00001	0.00001	0.00002
33	Transciever&HPA Module	box	0.0889	0.0508	0.0279	0.00011	0.00029	0.00035
34	Receiver Module	box	0.0889	0.0508	0.0279	0.00008	0.00022	0.00026
35	Interface/Power Module	box	0.0889	0.0508	0.0279	0.00008	0.00022	0.00026
36	µVAT	solid tri	0.0100	0.0500	0.0450	0.00151	0.00071	0.00087
37	Radiator	solid rec	0.0350	0.4000	0.2665	0.00771	0.00241	0.00538
38	TT&C Miscellaneous	UDI						
39	Thermal Miscellaneous	UDI						
40	Miscellaneous	UDI						

Step 3 - Component Center of Mass Xr, Yr, Zr values from Reference (0,0,0)

Item	Name		Xr	Yr	Zr	
1	Battery 1	box	0.1200	-0.0900	-0.0342	m
2	Battery 2	box	0.1200	-0.0300	-0.0342	m
3	Battery 3	box	0.1200	0.0300	-0.0342	m
4	Battery 4	box	0.1200	0.0900	-0.0342	m
5	Side 1 (front w/Ap)	solid sheet	-0.2500	0.0000	-0.2500	m
6	Side 2 (back w/Rad)	solid sheet	0.2500	0.0000	-0.2500	m
7	Side 3 & Solar Array	solid sheet	0.0000	-0.2500	-0.2500	m
8	Side 4 & Solar Array	solid sheet	0.0000	0.0000	-0.5000	m
9	Side 5 & Solar Array	solid sheet	0.0000	0.2500	-0.2500	m
10	Side 6 & Solar Array	solid sheet	0.0000	0.0000	0.0000	m
11	C&DH Proc	box	-0.1400	0.0000	-0.0400	m
12	RFTP/Power Control Unit	box	-0.1400	0.0000	-0.0800	m
13	"X" Reaction Wheel&IRU	box	0.0500	-0.0520	-0.0445	m
14	"Y" Reaction Wheel&IRU	box	-0.0500	-0.0520	-0.0445	m
15	"Z" Reaction Wheel	box	0.0500	0.0520	-0.0445	m
16	Spare Reaction Wheel	box	-0.0500	0.0520	-0.0445	m
17	Temp Controller	box	-0.1250	0.0000	-0.0939	m
18	C&DH/TT&C Antenna	box	0.2450	0.0000	-0.3150	m
19	Payload	solid cyl	0.0000	0.0000	-0.3550	m
20	Star Tracker	solid cyl	-0.2250	0.0000	-0.0450	m
21	Sun Sensor 1	solid cyl	-0.2300	0.2300	-0.4800	m
22	Sun Sensor 2	solid cyl	0.2300	-0.2300	-0.4800	m
23	Sun Sensor 3	solid cyl	-0.2300	-0.2300	-0.0200	m
24	Sun Sensor 4	solid cyl	-0.2300	-0.2300	-0.4800	m
25	Sun Sensor 5	solid cyl	-0.2300	0.2300	-0.4800	m
26	Sun Sensor 6	solid cyl	-0.2300	0.2300	0.0000	m
27	Earth Sensor 1	solid cyl	-0.2300	-0.2300	-0.4800	m
28	Earth Sensor 2	solid cyl	0.2300	0.2300	-0.4800	m
29	Earth Sensor 3	solid cyl	-0.2300	-0.2300	-0.4800	m
30	Earth Sensor 4	solid cyl	-0.2300	0.2300	-0.4800	m
31	Earth Sensor 5	solid cyl	-0.2300	0.2300	-0.2000	m
32	Earth Sensor 6	solid cyl	-0.2300	-0.2300	0.0000	m
33	Transciever&HPA Module	box	0.2006	0.0000	-0.0190	m
34	Receiver Module	box	0.2006	0.0558	-0.0190	m
35	Interface/Power Module	box	0.2006	-0.0558	-0.0190	m
36	µVAT	solid tri	0.2450	0.0000	-0.2500	m
37	Radiator	solid rec	0.2375	0.0000	-0.1383	m
38	TT&C Miscellaneous	UDI				
39	Thermal Miscellaneous	UDI				
40	Miscellaneous	UDI				

Step 4 - Spacecraft Center of Mass

Item	Name		M*Xr	M*Yr	M*Zr	
1	Battery 1	box	0.02	-0.01	0.00	kg*m
2	Battery 2	box	0.02	0.00	0.00	kg*m
3	Battery 3	box	0.02	0.00	0.00	kg*m
4	Battery 4	box	0.02	0.01	0.00	kg*m
5	Side 1 (front w/Ap)	solid sheet	-0.09	0.00	-0.09	kg*m
6	Side 2 (back w/Rad)	solid sheet	0.12	0.00	-0.12	kg*m
7	Side 3 & Solar Array	solid sheet	0.00	-0.31	-0.31	kg*m
8	Side 4 & Solar Array	solid sheet	0.00	0.00	-0.63	kg*m
9	Side 5 & Solar Array	solid sheet	0.00	0.31	-0.31	kg*m
10	Side 6 & Solar Array	solid sheet	0.00	0.00	0.00	kg*m
11	C&DH Proc	box	-0.21	0.00	-0.06	kg*m
12	RFTP/Power Control Unit	box	-0.04	0.00	-0.02	kg*m
13	"X" Reaction Wheel&IRU	box	0.17	-0.18	-0.15	kg*m
14	"Y" Reaction Wheel&IRU	box	-0.17	-0.18	-0.15	kg*m
15	"Z" Reaction Wheel	box	0.17	0.18	-0.15	kg*m
16	Spare Reaction Wheel	box	-0.17	0.18	-0.15	kg*m
17	Temp Controller	box	0.00	0.00	0.00	kg*m
18	C&DH/TT&C Antenna	box	0.43	0.00	-0.55	kg*m
19	Payload	solid cyl	0.00	0.00	-1.90	kg*m
20	Star Tracker	solid cyl	-0.07	0.00	-0.01	kg*m
21	Sun Sensor 1	solid cyl	-0.09	0.09	-0.19	kg*m
22	Sun Sensor 2	solid cyl	0.09	-0.09	-0.19	kg*m
23	Sun Sensor 3	solid cyl	-0.09	-0.09	-0.01	kg*m
24	Sun Sensor 4	solid cyl	-0.09	-0.09	-0.19	kg*m
25	Sun Sensor 5	solid cyl	-0.09	0.09	-0.19	kg*m
26	Sun Sensor 6	solid cyl	-0.09	0.09	0.00	kg*m
27	Earth Sensor 1	solid cyl	-0.02	-0.02	-0.05	kg*m
28	Earth Sensor 2	solid cyl	0.02	0.02	-0.05	kg*m
29	Earth Sensor 3	solid cyl	-0.02	-0.02	-0.05	kg*m
30	Earth Sensor 4	solid cyl	-0.02	0.02	-0.05	kg*m
31	Earth Sensor 5	solid cyl	-0.02	0.02	-0.02	kg*m
32	Earth Sensor 6	solid cyl	-0.02	-0.02	0.00	kg*m
33	Transciever&HPA Module	box	0.08	0.00	-0.01	kg*m
34	Receiver Module	box	0.06	0.02	-0.01	kg*m
35	Interface/Power Module	box	0.06	-0.02	-0.01	kg*m
36	μVAT	solid tri	0.98	0.00	-1.00	kg*m
37	Radiator	solid rec	0.10	0.00	-0.06	kg*m
38	TT&C Miscellaneous	UDI				
39	Thermal Miscellaneous	UDI				
40	Miscellaneous	UDI				

		Xcm	Ycm	Zcm	
Composite		0.023	0.000	-0.151	m (from 0,0,0)
	Mass =	44.51	kg		

Step 5 - Component Center of Mass distance from Spacecraft Center of Mass (Wet)

Item	Name		Xcm-Xr	Ycm-Yr	Zcm-Zr
1	Battery 1	box	-0.10	0.09	0.12
2	Battery 2	box	-0.10	0.03	0.12
3	Battery 3	box	-0.10	-0.03	0.12
4	Battery 4	box	-0.10	-0.09	0.12
5	Side 1 (front w/Ap)	solid sheet	0.27	0.00	-0.10
6	Side 2 (back w/Rad)	solid sheet	-0.23	0.00	-0.10
7	Side 3 & Solar Array	solid sheet	0.02	0.25	-0.10
8	Side 4 & Solar Array	solid sheet	0.02	0.00	-0.35
9	Side 5 & Solar Array	solid sheet	0.02	-0.25	-0.10
10	Side 6 & Solar Array	solid sheet	0.02	0.00	0.15
11	C&DH Proc	box	0.16	0.00	0.11
12	RFTP/Power Control Unit	box	0.16	0.00	0.07
13	"X" Reaction Wheel&IRU	box	-0.03	0.05	0.11
14	"Y" Reaction Wheel&IRU	box	0.07	0.05	0.11
15	"Z" Reaction Wheel	box	-0.03	-0.05	0.11
16	Spare Reaction Wheel	box	0.07	-0.05	0.11
17	Temp Controller	box	0.15	0.00	0.06
18	C&DH/TT&C Antenna	box	-0.22	0.00	-0.16
19	Payload	solid cyl	0.02	0.00	-0.20
20	Star Tracker	solid cyl	0.25	0.00	0.11
21	Sun Sensor 1	solid cyl	0.25	-0.23	-0.33
22	Sun Sensor 2	solid cyl	-0.21	0.23	-0.33
23	Sun Sensor 3	solid cyl	0.25	0.23	0.13
24	Sun Sensor 4	solid cyl	0.25	0.23	-0.33
25	Sun Sensor 5	solid cyl	0.25	-0.23	-0.33
26	Sun Sensor 6	solid cyl	0.25	-0.23	0.15
27	Earth Sensor 1	solid cyl	0.25	0.23	-0.33
28	Earth Sensor 2	solid cyl	-0.21	-0.23	-0.33
29	Earth Sensor 3	solid cyl	0.25	0.23	-0.33
30	Earth Sensor 4	solid cyl	0.25	-0.23	-0.33
31	Earth Sensor 5	solid cyl	0.25	-0.23	-0.05
32	Earth Sensor 6	solid cyl	0.25	0.23	0.15
33	Transciever&HPA Module	box	-0.18	0.00	0.13
34	Receiver Module	box	-0.18	-0.06	0.13
35	Interface/Power Module	box	-0.18	0.06	0.13
36	μVAT	solid tri	-0.22	0.00	-0.10
37	Radiator	solid rec	-0.21	0.00	0.01
38	TT&C Miscellaneous	UDI			
39	Thermal Miscellaneous	UDI			
40	Miscellaneous	UDI			

Step 6 - Moments Of Intertia wrt Spacecraft Center of Mass

Item	Name		Ixx	Iyy	Izz	Ixy	Ixz	Iyz
1	Battery 1	box	0.00	0.00	0.00	0.0014	0.0017	-0.0015
2	Battery 2	box	0.00	0.00	0.00	0.0005	0.0017	-0.0005
3	Battery 3	box	0.00	0.00	0.00	-0.0003	0.0017	0.0006
4	Battery 4	box	0.00	0.00	0.00	-0.0012	0.0017	0.0016
5	Side 1 (front w/Ap)	solid sheet	0.02	0.04	0.03	0.0152	0.0175	0.0076
6	Side 2 (back w/Rad)	solid sheet	0.03	0.04	0.04	0.0205	-0.0008	0.0102
7	Side 3 & Solar Array	solid sheet	0.12	0.07	0.11	0.0189	0.0551	0.0573
8	Side 4 & Solar Array	solid sheet	0.18	0.18	0.05	0.0261	0.0362	0.0523
9	Side 5 & Solar Array	solid sheet	0.12	0.07	0.11	0.0334	0.0551	-0.0050
10	Side 6 & Solar Array	solid sheet	0.05	0.06	0.05	0.0261	0.0218	0.0523
11	C&DH Proc	box	0.02	0.06	0.04	0.0018	-0.0264	0.0024
12	RFTP/Power Control Unit	box	0.00	0.01	0.01	0.0003	-0.0031	0.0005
13	"X" Reaction Wheel&IRU	box	0.05	0.05	0.02	0.0100	0.0145	-0.0133
14	"Y" Reaction Wheel&IRU	box	0.05	0.06	0.03	-0.0077	-0.0216	-0.0133
15	"Z" Reaction Wheel	box	0.05	0.05	0.02	0.0004	0.0145	0.0242
16	Spare Reaction Wheel	box	0.05	0.06	0.03	0.0181	-0.0216	0.0242
17	Temp Controller	box	0.00	0.00	0.00	0.0000	-0.0002	0.0000
18	C&DH/TT&C Antenna	box	0.05	0.14	0.09	0.0053	-0.0533	0.0053
19	Payload	solid cyl	0.23	0.23	0.00	0.0021	0.0291	0.0018
20	Star Tracker	solid cyl	0.00	0.02	0.02	0.0003	-0.0076	0.0002
21	Sun Sensor 1	solid cyl	0.06	0.07	0.05	0.0233	0.0333	-0.0303
22	Sun Sensor 2	solid cyl	0.06	0.06	0.04	0.0191	-0.0273	0.0303
23	Sun Sensor 3	solid cyl	0.03	0.03	0.05	-0.0233	-0.0132	-0.0120
24	Sun Sensor 4	solid cyl	0.06	0.07	0.05	-0.0233	0.0334	0.0303
25	Sun Sensor 5	solid cyl	0.06	0.07	0.05	0.0233	0.0334	-0.0303
26	Sun Sensor 6	solid cyl	0.03	0.03	0.05	0.0233	-0.0152	0.0139
27	Earth Sensor 1	solid cyl	0.02	0.02	0.01	-0.0058	0.0083	0.0076
28	Earth Sensor 2	solid cyl	0.02	0.02	0.01	-0.0048	-0.0068	-0.0076
29	Earth Sensor 3	solid cyl	0.02	0.02	0.01	-0.0058	0.0083	0.0076
30	Earth Sensor 4	solid cyl	0.02	0.02	0.01	0.0058	0.0083	-0.0076
31	Earth Sensor 5	solid cyl	0.01	0.01	0.01	0.0058	0.0013	-0.0011
32	Earth Sensor 6	solid cyl	0.01	0.01	0.01	-0.0058	-0.0038	-0.0034
33	Transciever&HPA Module	box	0.01	0.02	0.01	0.0001	0.0096	0.0003
34	Receiver Module	box	0.01	0.01	0.01	-0.0029	0.0072	0.0025
35	Interface/Power Module	box	0.01	0.01	0.01	0.0031	0.0072	-0.0019
36	μVAT	solid tri	0.04	0.24	0.20	0.0015	-0.0876	0.0009
37	Radiator	solid rec	0.01	0.02	0.02	0.0077	0.0035	0.0054
38	TT&C Miscellaneous	UDI						
39	Thermal Miscellaneous	UDI						
40	Miscellaneous	UDI						

Step 7 - Sum Spacecraft Moments of Inertia

Ixx	Iyy	Izz	Ixy	Ixz	Iyz
1.50	1.86	1.26	0.21	0.12	0.21

Ref: SMAD p.466(material properties, 476-477, 924 (coversation factors)

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97E+24	kg
G =	6.67E-20	$km^3/kg*s^2$
μ Earth =	398600.4	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Pitch Error:

1. Sum S/C MOIs,
2. Find center of pressure,
3. Calculate pitch error (GG & Aero torque equilibrium)

Orbit Parameters

orbital rate (ω_0) =	0.000073	rad/sec
velocity (v) =	3076.535636	m/s
atmospheric density (ρ) =	0.00	kg/m³

Component Characteristics

Component	Shape	d (x) or R [m]	w (y) or r [m]	h (z) [m]	X-Section Area [m^2]
Battery 1	box	0.0600	0.0192	0.0684	0.0012
Battery 2	box	0.0600	0.0192	0.0684	0.0012
Battery 3	box	0.0600	0.0192	0.0684	0.0012
Battery 4	box	0.0600	0.0192	0.0684	0.0012
Side 1 (front w/Ap)	solid sheet	0.5000	0.0100	0.5000	0.0050
Side 2 (back w/Rad)	solid sheet	0.5000	0.0100	0.5000	0.0050
Side 3 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.0050
Side 4 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.0050
Side 5 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.0050
Side 6 & Solar Array	solid sheet	0.5000	0.0100	0.5000	0.0050
C&DH Proc	box	0.1200	0.0100	0.0700	0.0012
RFTP/Power Control Unit	box	0.1200	0.0100	0.0700	0.0012
"X" Reaction Wheel&IRU	box	0.0940	0.1020	0.0890	0.0096
"Y" Reaction Wheel&IRU	box	0.0940	0.1020	0.0890	0.0096
"Z" Reaction Wheel	box	0.0940	0.1020	0.0890	0.0096
Spare Reaction Wheel	box	0.0940	0.1020	0.0890	0.0096
Temp Controller	box	0.0038	0.0254	0.0277	0.0001
C&DH/TT&C Antenna	box	0.1900	0.0100	0.1900	0.0019
Payload	solid cyl	0.225	0.0001	0.4	0.2827
Star Tracker	solid cyl	0.054	0.054	0.076	0.0041
Sun Sensor 1	solid cyl	0.015	0.0001	0.01	0.0005
Sun Sensor 2	solid cyl	0.015	0.0001	0.01	0.0005
Sun Sensor 3	solid cyl	0.015	0.0001	0.01	0.0005
Sun Sensor 4	solid cyl	0.015	0.0001	0.01	0.0005
Sun Sensor 5	solid cyl	0.015	0.0001	0.01	0.0005
Sun Sensor 6	solid cyl	0.015	0.0001	0.01	0.0005
Earth Sensor 1	solid cyl	0.0135	0.0001	0.02	0.0008
Earth Sensor 2	solid cyl	0.0135	0.0001	0.02	0.0008
Earth Sensor 3	solid cyl	0.0135	0.0001	0.02	0.0008
Earth Sensor 4	solid cyl	0.0135	0.0001	0.02	0.0008
Earth Sensor 5	solid cyl	0.0135	0.0001	0.02	0.0008
Earth Sensor 6	solid cyl	0.0135	0.0001	0.02	0.0008
Transciever&HPA Module	box	0.0890	0.0250	0.0510	0.0022
Receiver Module	box	0.0890	0.0250	0.0510	0.0022
Interface/Power Module	box	0.0890	0.0250	0.0510	0.0022
μVAT	solid tri	0.0500	0.0100	0.0450	0.0005
Radiator	solid rec	0.4000	0.0350	0.2665	0.0140

Sum Spacecraft Moments of Inertia

Ixx	Iyy	Izz	Ixy	Ixz	Iyz
1.500	1.861	1.257	0.21	0.12	0.21

Spacecraft Center of Mass

	X_{cm}	Y_{cm}	Z_{cm}	
Composite	0.023	0.000	-0.151	m (from 0,0,0)

Center of Mass (Z_{cm}) = -0.151 m

Center of Pressure

Component	Shape	Area	cp (Zr)	Area*cp
Battery 1	box	0.0012	-0.0342	-0.000039
Battery 2	box	0.0012	-0.0342	-0.000039
Battery 3	box	0.0012	-0.0342	-0.000039
Battery 4	box	0.0012	-0.0342	-0.000039
Side 1 (front w/Ap)	solid sheet	0.0050	-0.2500	-0.001250
Side 2 (back w/Rad)	solid sheet	0.0050	-0.2500	-0.001250
Side 3 & Solar Array	solid sheet	0.0050	-0.2500	-0.001250
Side 4 & Solar Array	solid sheet	0.0050	-0.5000	-0.002500
Side 5 & Solar Array	solid sheet	0.0050	-0.2500	-0.001250
Side 6 & Solar Array	solid sheet	0.0050	0.0000	0.000000
C&DH Proc	box	0.0012	-0.0400	-0.000048
RFTP/Power Control Unit	box	0.0012	-0.0800	-0.000096
"X" Reaction Wheel&IRU	box	0.0096	-0.0445	-0.000427
"Y" Reaction Wheel&IRU	box	0.0096	-0.0445	-0.000427
"Z" Reaction Wheel	box	0.0096	-0.0445	-0.000427
Spare Reaction Wheel	box	0.0096	-0.0445	-0.000427
Temp Controller	box	0.0001	-0.0939	-0.000009
C&DH/TT&C Antenna	box	0.0019	-0.3150	-0.000599
Payload	solid cyl	0.2827	-0.3550	-0.100374
Star Tracker	solid cyl	0.0041	-0.0450	-0.000185
Sun Sensor 1	solid cyl	0.0005	-0.4800	-0.000226
Sun Sensor 2	solid cyl	0.0005	-0.4800	-0.000226
Sun Sensor 3	solid cyl	0.0005	-0.0200	-0.000009
Sun Sensor 4	solid cyl	0.0005	-0.4800	-0.000226
Sun Sensor 5	solid cyl	0.0005	-0.4800	-0.000226
Sun Sensor 6	solid cyl	0.0005	0.0000	0.000000
Earth Sensor 1	solid cyl	0.0008	-0.4800	-0.000407
Earth Sensor 2	solid cyl	0.0008	-0.4800	-0.000407
Earth Sensor 3	solid cyl	0.0008	-0.4800	-0.000407
Earth Sensor 4	solid cyl	0.0008	-0.4800	-0.000407
Earth Sensor 5	solid cyl	0.0008	-0.2000	-0.000170
Earth Sensor 6	solid cyl	0.0008	0.0000	0.000000
Transciever&HPA Module	box	0.0022	-0.0190	-0.000042
Receiver Module	box	0.0022	-0.0190	-0.000042
Interface/Power Module	box	0.0022	-0.0190	-0.000042
μVAT	solid tri	0.0005	-0.2500	-0.000125
Radiator	solid rec	0.0140	-0.1383	-0.001936

Total Area 1 = 0.393 m²

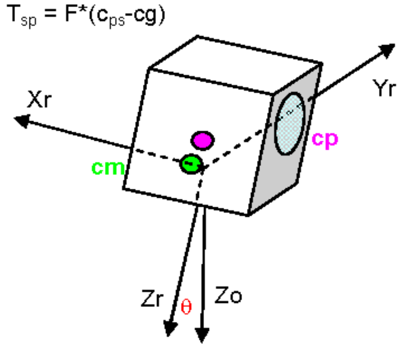
Center of Pressure 1(Z_{cp1}) = -0.294 m

θ Error 1

Drag Coeff (C_d) =	2.5		
Atmospheric Drag (F_a) =	0.00	N	$F_a = \rho * C_d * A * V^2$
Aero Torque (T_a) =	0.00	N*m	$T_a = (1/2) * F_a (C_{pa} - cg)$
θ =	0.00	rad	$\theta = T_a / [3 * \omega_o^2 (I_{xx} - I_{zz})]$
θ =	0.00	deg	
Max allowable deivation of Z-axis (θz) =	8.73E-03	rad	Operational Restriction
Grav Gradient Torque (T_g) =	8.44E-11	N*m	$Tg = ((3*\mu)/(2*R^3))*\lvert Izz-Iyy\rvert*\sin(2*\theta z)$
θ =	2.17E-02	rad	
θ =	1.24E+00	deg	

	X_{cm}	Y_{cm}	Z_{cm}	
cm =	0.023	0.000	-0.151	m
Vector from Area's center to SCcm (s_c)=	-0.023	-0.250	0.099	m

Angle of incidence of the Sun (i) =	30.000	deg	
Solar Vector to SC (F) =	1.283E-06	N	$F = (F_s/c)*A*(1+q)*\cos(i)$
Solar Radiation Torque (T_{sp}) =	1.275E-07	N*m	$T_{sp} = F*(c_{ps}-cg)$
θ =	2.395E-03	rad	
θ =	1.37E-01	deg	
Maximux θ Error =	1.244E+00	deg	



Ref: SMAD p322-324, 366

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Optical Payload:

Orbit Parameters

SC Altitude (H_{sc}) =	35734.64	km	given
SC Orbit Period (P_{sc}) =	1433.73	min	$(\mu/(R_e+H_{sc})^3)^{1/2}$
Target's Altitude (H_{tar}) =	35785.86	km	Assuming GEOSTA
Target's Obital Period (P_{tar}) =	1436.35	min	$sqrt(\mu/(R_e+H_{tar})^3$
SC ω (ω_{sc}) =	7.30E-05	Rad/sec	$\omega_{sc} = (2*\pi)/P_{sc}$
Target ω (ω_{tar}) =	7.29E-05	Rad/sec	$\omega_{tar} = (2*\pi)/P_{tar}$
SC orbital Radius (R_{sc}) =	42118.37	km	$(\mu_{Earth}/\omega_{sc}^2)^{1/3}$
Target orbital Radius (R_{tar}) =	42169.61	km	$(\mu_{Earth}/\omega_{tar}^2)^{1/3}$
SC Velocity (V_{sc}) =	3.08	km/sec	$V_{sc} = \omega_{sc}*R_{sc}$
Target's Velocity (V_{tar}) =	3.07	km/sec	$V_{tar} = \omega_{tar}*R_{tar}$
Closing Velocity (CV[km/sec]) =	1.87E-03	km/sec	V_{tar}-V_{sc}
Closing Velocity (CV) =	1.87	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	51.23	km	H_{tar}-H_{sc}
Target's Orbital Circumference (Cir_{tar}) =	264869.00	km	$2*\pi*(H_{tar}*6378.137)$
Distance SC travels relative to GEOSTA orbit =	161.51	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	1640.26	days	Cir_{tar}/Distance SC travels
Time for SC to traverse GEOSTA orbit =	4.49	years	(Cir_{tar}/Distance SC travels)*365.25

Kodak KAF-39000-AAA-DD-AE Image Sensor Properties

Pixel Height =	6.8	μm	CCD Data Sheet
Pixel Width =	6.8	μm	CCD Data Sheet
# of Horizontal pixels (PxlH) =	7216	pxl	CCD Data Sheet
# of Vertical pixels (PxlV) =	5412	pxl	CCD Data Sheet
Maximum Data Rate (DR) =	2.40E+07	Hz	CCD Data Sheet
Bits/pxl (Nb) =	1.10E+01		
Line Readout Time (Tlrout) =	1.81E-04	sec/line	CCD Data Sheet
Pixel Period (1 count) =	4.20E-08	sec	CCD Data Sheet
#sec/pxl =	3.34E-08	sec/pxl	Tlrout/PxlV
#pxls/sec (Z) =	2.99E+07	pxl/sec	1/#sec/pxl
Operating Temp Range =	-20 to 70	°C	CCD Data Sheet
Guaranteed Operating Temp Range =	0 to 60	°C	CCD Data Sheet

Target Parameters

In track Elevation angle (e) =		deg	Depends on target's relative postion
Slew angle (h) =		deg	Depends on target's relative postion
Slant range (Rs) =		km	Depends on target's relative postion
# Active Pixels (Zact) =	39052992	pixels	CCD characteristic
Pixel Integration Time (Tipxl) =	5.00E-04	sec	CCD characteristic
Relative Motion During pxl image capture(Blur) =	9.35E-04	m	CV*Tipxl
#pxls/sec (Z) =	7.81E+10	pxls/sec	Zact/Tipxl
Bits/pxl (Nb) =	11	bits/pixel	Choosen
DataRate (DR) =	429.58	Mbps	Zact*Nb
Megabytes per Picture (Psz) =	53.70	MBytes	CCD characteristic
Compressed Image Size (cPsz) =	4.47	MBytes	Psz/12 ->Need to verify for jpeg format

Optic System

Aperature (Ap) =	0.25	m	S/C property
CPA Seperation (CPA Sep) =	51229.5	m	Htar-Hsc; will differ when e≠ 90 since SC is orbiting lower than Target
Lambda (λ) =	4.0000E-07	m	Choosen
Resolution (X) =	0.200	m	(2.44*λ*Sep)/Ap
Resolution of Target at 50[km] (X_{50km}) =	0.195	m	(2.44*λ*50000[m])/Ap
Resolution of Target at 100[km] (X_{100km}) =	0.390	m	(2.44*λ*100000[m])/Ap
0.5[m] Resolution of Target =	0.50	m	(2.44*λ*100000[m])/Ap
Distance for 0.5[m] Resolution of Target ($D_{0.5m}$) =	128073.8	m	Determined by goal seeking 0.5[m] Resolution
Detector width *(square pixel width)* (d) =	6.80E-06	m	CCD Data Sheet
Quality factor (Q) =	1.10E+00		0.5<Q<2 selected
Operating wavelength (λw) =	4.00E-07	m	selected
Focal length (f) =	1.742	m	Sep*d/X
Folded folcal length (f/5) =	0.348	m	f/5
Diffraction-limited aperature diameter (D) =	0.275	m	2.44*λw*f*Q/d
F-number (F#) =	6.334	#	f/D

Scaling Estimate SMAD method (sec. 9.5.3 using IKONOS)

Aperture ratio (R) =	0.25		Ai/Ao (aperature of LEO/aperture of IKONOS)
Linear dimensions (Li) =	0.38	m	R*Lo (Lo=linear dimension of IKONOS)
Surface area (Si) =	0.15	m^2	Li^2
Volume (Vi) =	0.06	m^3	Li^3
Weight (Wi) =	5.34	kg	K*R^3*Wo (K=2 if R<0.5, else 1)
Power (Pi) =	10.94	W	K*R^3*Po

IKONOS-2 using Kodak Model 1000 Camera System Data:
Assembly size: 1.524[m] by 0.787[m] (1[m^3] volume)
10[m] focal length; f/14.3; 0.7[m] Primary mirror aperature diameter.
171[kg] total mass; 350[W] total power

MOI Estimate Budianto method (Eq. 4.16)

Moment of Inertia (I) =	1.38E+00	kg*m^2	(1/12)*Wi*(3(D/2)^2+f^2) assuming cylindrical assembly

Maximum allowable error in optical component construction

Max allowable angular deviation($\Delta\Theta$) =	3.90E-06	deg	X/h

Pointing Requirements

Y_{min} (Basic Calc) =	14.43	deg	h*d/f
Y_{min} (Diffraction limit) =	0.18	deg	2.44*λ*h/D
Y_{min} (Image Blur) =	9.05E-07	deg	10*(Scan Velocity)*Δt
Pointing Requirement =	14.43	deg	Larget Y_{min} calculated for Spatial resolution desired.

Thermal Requirements

Operating Temperature =	273-333	K	Based on CCD restriction
Operating Temperature =	0-60	C	Based on CCD restriction

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Comments

Distortion is not a worry due to limited size of target object.
Entire optical formulae for pointing out into space vice down toward earth -> use stop and stare method at each choosen distance in closing path.
Velocity formulae need to be corrected
Closing velocity will dictate shutter speed needed

Ref: SMAD p.247-91; 364-369(ACS constraints)

Constellation Estimating:

Orbit Parameters

SC Altitude (Hsc) =	35734.64	km	given
SC Orbit Period (Psc) =	1433.73	min	(m/(Re+Hsc)3)1/2
Target's Altitude (Htar) =	35785.86	km	Assuming GEOSTA
Target's Obital Period (Ptar) =	1436.35	min	sqrt(m/(Re+Htar)3
SC ω (ωsc) =	0.00	Rad/sec	ωsc = (2*π)/Psc
Target ω (ωtar) =	0.00	Rad/sec	ωtar = (2*π)/Ptar
SC orbital Radius (Rsc) =	42118.37	km	(μEarth/ωsc2)1/3
Target orbital Radius (Rtar) =	42169.61	km	(μEarth/ωtar2)1/3
SC Velocity (Vsc) =	3.08	km/sec	Vsc = ωsc*Rsc
Target's Velocity (Vtar) =	3.07	km/sec	Vtar = ωtar*Rtar

Coverage and Access Factors

Closing Velocity (CV[km/sec]) =	0.00187	km/sec	Vtar-Vsc
Closing Velocity (CV) =	1.87	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	51.23	km	Htar-Hsc
Target's Orbital Circumference (Cirtar) =	264869.00	km	2*π*(Htar*6378.137)
Distance SC travels relative to GEOSTA orbit =	161.51	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	1640.26	days	Cirtar/Distance SC travels
Time for SC to traverse GEOSTA orbit =	4.49	years	(Cirtar/Distance SC travels)*365.25

Coverage and Access Considerations

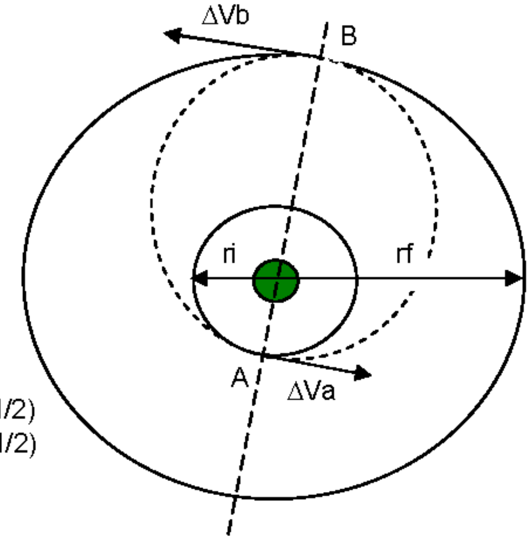
Desired Constellation re-visit rate =	30	days	Choosen
Planes of SC desired =	1	Planes	Majority of GEOSTA at i≈0°
Distance SC can travel during re-visit rate =	7973850.38	km	(Re-visit rate)* Vsc*60*60*24
Distance Targets travel during re-visit rate =	7969004.96	km	(Re-visit rate)* Vtar*60*60*24
Difference in distances traveled =	4845.42	km	(SC travel)-(Tar travel)
Separation distance for SC =	9690.84	km	2*SepDist
Number of SC needed =	27.00		(Circumference of target's orbit)/(Sep distance) rounded up

Modeled in STK

Seeed number of SC =	22.00	SC	From estimation
STK optimized SC number through trial and error =	15	SC	Evenly spaced SC simulated in STK to give complete GEOSTA coverage in desired re-visit rate.
SC separation =	18919.21	km	(2*π*(Hsc+6378.137))/14

Ref: SMAD p.190-196

Orbit Transfer Calculations:



		MicroSAT	Units	Comments
μ earth	398600.4418		km^3/s^2	constant
Re	6378.137		km	constant
hpark		42164	km	given
	ri	42164	km	given
	rf	42113	km	given
Step				
1	atx	42138	km	(ri+rf)/2
2	Via	3.07	km/s	(μ earth/ri)^(1/2)
3	Vfb	3.08	km/s	(μ earth/rf)^(1/2)
4	Vtxa	3.07	km/s	(μ earth*(2/ri)-(1/atx))^(1/2)
5	Vtxb	3.08	km/s	(μ earth*(2/rf)-(1/atx))^(1/2)
6	ΔVa	0.00	km/s	|Vtxa-Via|
7	ΔVb	0.00	km/s	|Vfb-Vtxb|
8	ΔVtotal	0.00	km/s	ΔVa+ΔVb
9	x-fer time	43043	s	P/2 = π*SQRT(a^3/μ)
		11.96	hr	

Plane change in parking orbit followed by Hohmann transfer:

θ	15.00	deg
ΔVpc	0.8031	km/s
ΔVa+ΔVb+ΔVpc	0.8013	km/s

Plane change at B combined with Hohmann transfer:

ΔVcmd	0.8033	km/s
ΔVa+ΔVcmd	0.8023	km/s

RAAN change utilitizing two Hohmann transfers

ΔV	1.6246	km/s

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p.147-152 & Table 6-5; FS p.340

*Ortibal Change to meet a target SC: SMAD p.152

Wait Time (WT) = (φi-φf+2kπ)/(ωint-ωtar)

ΔV Estimation:

Basic Data	**MicroSAT**	**Units**	**Comments**
Initial Radius	42164	km	Choosen
Initial inclination	0	deg	Orbit insertion inclination
Mission Radius	42113	km	Choosen
Mission inclination	0	deg	Choosen to match target's inclination
Mission Duration	2	yr	Choosen
Orbit Maintenance Req	Various		Mission Dependant
Drag Parameters	0		Orbital Regime Property
m/CdA	N/A		Orbital Regime Property
Max Atmospheric Density (ρmax)	N/A	kg/m^3	Orbital Regime Property
Orbit Manuever Req	Unknown		Mission Dependant
Final Conditions	Unknown		Mission Dependant
ΔV Budget [m/s]			
Orbit Transfer			
1st Burn	-1	m/s	ΔVa*1000
2nd Burn	803	m/s	ΔVcmd*1000
Altitude Maintenance (LEO)	N/A	m/s	Property of orbit
North/South Stationkeeping	103	m/s	Formula constants need to be analysized
East/West Stationkeeping	3	m/s	Formula constants need to be analysized
Orbit Manuevers	-	m/s	Maybe required as per mission
Rephasing, Rendezvous	Unknown		Maybe required as per mission
Node or Plance Change	Unknown		Maybe required as per mission
Spacecraft disposal	10	m/s	Usual Reqt's
Total ΔV	**919**	**m/s**	

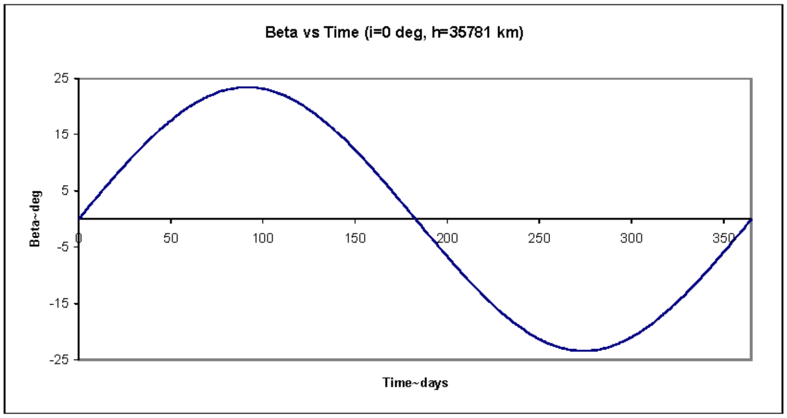
Key:
Coefficient of Drag (Cd)
SC Cross-sectional Area (A)
SC Mass (m)
SC velocity (Vsc)

Constants

Radius Earth =	6378.137	km
mass Earth =	5.9733E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.442	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p. 147-151

Basic Data	**Value**	**Units**	**Remarks**
uo=RA of sun in ecliptic	0		
wo=RA of AN of Orbit	0		
h (altitude)	35734.64	km	Orbit property
i (inclination)	0	deg	Orbit property
E (eccentricity)	0		Orbit property
wdot = nodal reg'n rate*	-0.01347	rate	-(9.96390003*(R/(R+h))^3.5*COS(i*π/180))/(1-E²)²
e (Earth axis tilt)	23.44241	deg	Earth property
R (Earth radius)	6378.137	km	Earth property
R/(R+h)	0.151454	*no unit*	R/(R+h)
MSD (mean solar day)	0.985647	sidereal day	Earth property
Earth g const	398601.2	kg^3/s^2	Orbit property
Orbital rate	7.31E-05	Rad/s	(Earth g const/(R+h)^3)^(1/2)
Orbital period	1433.442	min	2*π/Orbital rate/60
Earth angular radius	8.711182	deg	ASIN(R/(R+h))*180/π



[-9.96390003(R/(R+h))^3.5*cos(i)]/(1-E^2)^2 [deg/MSD]

Ref: SMAD p.107-110

day	u [deg]	w [deg]	β [deg]	Eclp Ang[deg]	Te/To
0	0	0	0	17.42236492	0.048395
1	0.985647	-0.0134713	0.392100859	17.4048431	0.048347
2	1.971295	-0.02694259	0.784104039	17.35218571	0.048201
3	2.956942	-0.04041389	1.175911848	17.26411414	0.047956
4	3.942589	-0.05388519	1.567426571	17.14015453	0.047612
5	4.928237	-0.06735648	1.958550455	16.97962273	0.047166
6	5.913884	-0.08082778	2.349185699	16.78160182	0.046616
7	6.899531	-0.09429908	2.739234444	16.54491025	0.045958
8	7.885179	-0.10777037	3.128598756	16.26805811	0.045189
9	8.870826	-0.12124167	3.517180623	15.94918732	0.044303
10	9.856473	-0.13471296	3.904881939	15.5859899	0.043294
11	10.84212	-0.14818426	4.291604493	15.1755948	0.042154
12	11.82777	-0.16165556	4.677249967	14.7144087	0.040873
13	12.81342	-0.17512685	5.061719918	14.1978869	0.039439
14	13.79906	-0.18859815	5.44491578	13.62019376	0.037834
15	14.78471	-0.20206945	5.826738846	12.97368096	0.036038
16	15.77036	-0.21554074	6.207090272	12.24804879	0.034022
17	16.756	-0.22901204	6.585871065	11.42891792	0.031747
18	17.74165	-0.24248334	6.962982081	10.49520925	0.029153
19	18.7273	-0.25595463	7.338324022	9.413826182	0.02615
20	19.71295	-0.26942593	7.711797433	8.127176642	0.022575
21	20.69859	-0.28289723	8.083302702	6.516273408	0.018101
22	21.68424	-0.29636852	8.45274006	4.227695657	0.011744
23	22.66989	-0.30983982	8.820009585	0	0
24	23.65554	-0.32331112	9.185011198	0	0
25	24.64118	-0.33678241	9.547644679	0	0
26	25.62683	-0.35025371	9.907809662	0	0
27	26.61248	-0.363725	10.26540565	0	0
28	27.59813	-0.3771963	10.62033203	0	0
29	28.58377	-0.3906676	10.97248806	0	0
30	29.56942	-0.40413889	11.32177291	0	0
31	30.55507	-0.41761019	11.66808568	0	0
32	31.54071	-0.43108149	12.01132538	0	0
33	32.52636	-0.44455278	12.35139099	0	0
34	33.51201	-0.45802408	12.68818145	0	0
35	34.49766	-0.47149538	13.02159572	0	0
36	35.4833	-0.48496667	13.35153276	0	0
37	36.46895	-0.49843797	13.67789161	0	0
38	37.4546	-0.51190927	14.00057135	0	0
39	38.44025	-0.52538056	14.31947123	0	0
40	39.42589	-0.53885186	14.6344906	0	0
41	40.41154	-0.55232316	14.94552904	0	0
42	41.39719	-0.56579445	15.25248632	0	0
43	42.38284	-0.57926575	15.55526252	0	0
44	43.36848	-0.59273705	15.853758	0	0
45	44.35413	-0.60620834	16.14787351	0	0
46	45.33978	-0.61967964	16.43751021	0	0
47	46.32542	-0.63315093	16.7225697	0	0
48	47.31107	-0.64662223	17.00295414	0	0
49	48.29672	-0.66009353	17.27856625	0	0
50	49.28237	-0.67356482	17.5493094	0	0
51	50.26801	-0.68703612	17.81508764	0	0
52	51.25366	-0.70050742	18.07580582	0	0
53	52.23931	-0.71397871	18.33136959	0	0
54	53.22496	-0.72745001	18.58168555	0	0
55	54.2106	-0.74092131	18.82666122	0	0
56	55.19625	-0.7543926	19.06620521	0	0
57	56.1819	-0.7678639	19.30022725	0	0
58	57.16755	-0.7813352	19.52863825	0	0
59	58.15319	-0.79480649	19.75135043	0	0
60	59.13884	-0.80827779	19.96827735	0	0
61	60.12449	-0.82174909	20.17933402	0	0
62	61.11013	-0.83522038	20.38443698	0	0
63	62.09578	-0.84869168	20.58350437	0	0
64	63.08143	-0.86216297	20.77645605	0	0
65	64.06708	-0.87563427	20.96321362	0	0
66	65.05272	-0.88910557	21.14370058	0	0
67	66.03837	-0.90257686	21.31784237	0	0
68	67.02402	-0.91604816	21.48556646	0	0
69	68.00967	-0.92951946	21.64680245	0	0
70	68.99531	-0.94299075	21.80148215	0	0
71	69.98096	-0.95646205	21.94953964	0	0
72	70.96661	-0.96993335	22.09091141	0	0

Solar Beta Angle

73	71.95226	-0.98340464	22.22553637	0	0
74	72.9379	-0.99687594	22.35335599	0	0
75	73.92355	-1.01034724	22.47431432	0	0
76	74.9092	-1.02381853	22.58835813	0	0
77	75.89484	-1.03728983	22.69543693	0	0
78	76.88049	-1.05076113	22.79550308	0	0
79	77.86614	-1.06423242	22.8885118	0	0
80	78.85179	-1.07770372	22.97442132	0	0
81	79.83743	-1.09117501	23.05319286	0	0
82	80.82308	-1.10464631	23.12479071	0	0
83	81.80873	-1.11811761	23.18918233	0	0
84	82.79438	-1.1315889	23.24633832	0	0
85	83.78002	-1.1450602	23.29623252	0	0
86	84.76567	-1.1585315	23.33884204	0	0
87	85.75132	-1.17200279	23.37414729	0	0
88	86.73697	-1.18547409	23.40213198	0	0
89	87.72261	-1.19894539	23.4227832	0	0
90	88.70826	-1.21241668	23.43609142	0	0
91	89.69391	-1.22588798	23.44205046	0	0
92	90.67955	-1.23935928	23.44065759	0	0
93	91.6652	-1.25283057	23.43191344	0	0
94	92.65085	-1.26630187	23.41582205	0	0
95	93.6365	-1.27977317	23.39239087	0	0
96	94.62214	-1.29324446	23.36163073	0	0
97	95.60779	-1.30671576	23.3235558	0	0
98	96.59344	-1.32018706	23.27818363	0	0
99	97.57909	-1.33365835	23.22553507	0	0
100	98.56473	-1.34712965	23.16563425	0	0
101	99.55038	-1.36060094	23.09850856	0	0
102	100.536	-1.37407224	23.0241886	0	0
103	101.5217	-1.38754354	22.94270812	0	0
104	102.5073	-1.40101483	22.85410398	0	0
105	103.493	-1.41448613	22.75841612	0	0
106	104.4786	-1.42795743	22.65568746	0	0
107	105.4643	-1.44142872	22.54596386	0	0
108	106.4499	-1.45490002	22.42929405	0	0
109	107.4356	-1.46837132	22.30572957	0	0
110	108.4212	-1.48184261	22.17532469	0	0
111	109.4069	-1.49531391	22.03813633	0	0
112	110.3925	-1.50878521	21.89422403	0	0
113	111.3781	-1.5222565	21.74364977	0	0
114	112.3638	-1.5357278	21.58647802	0	0
115	113.3494	-1.5491991	21.42277555	0	0
116	114.3351	-1.56267039	21.25261139	0	0
117	115.3207	-1.57614169	21.07605677	0	0
118	116.3064	-1.58961298	20.893185	0	0
119	117.292	-1.60308428	20.70407138	0	0
120	118.2777	-1.61655558	20.50879314	0	0
121	119.2633	-1.63002687	20.30742935	0	0
122	120.249	-1.64349817	20.10006081	0	0
123	121.2346	-1.65696947	19.88677	0	0
124	122.2203	-1.67044076	19.66764096	0	0
125	123.2059	-1.68391206	19.44275926	0	0
126	124.1916	-1.69738336	19.21221184	0	0
127	125.1772	-1.71085465	18.97608698	0	0
128	126.1629	-1.72432595	18.73447424	0	0
129	127.1485	-1.73779725	18.48746433	0	0
130	128.1342	-1.75126854	18.23514904	0	0
131	129.1198	-1.76473984	17.97762122	0	0
132	130.1054	-1.77821114	17.71497464	0	0
133	131.0911	-1.79168243	17.44730396	0	0
134	132.0767	-1.80515373	17.17470464	0	0
135	133.0624	-1.81862502	16.89727289	0	0
136	134.048	-1.83209632	16.61510559	0	0
137	135.0337	-1.84556762	16.32830025	0	0
138	136.0193	-1.85903891	16.03695494	0	0
139	137.005	-1.87251021	15.7411682	0	0
140	137.9906	-1.88598151	15.44103906	0	0
141	138.9763	-1.8994528	15.13666691	0	0
142	139.9619	-1.9129241	14.8281515	0	0
143	140.9476	-1.9263954	14.51559288	0	0
144	141.9332	-1.93986669	14.19909136	0	0
145	142.9189	-1.95333799	13.87874744	0	0
146	143.9045	-1.96680929	13.55466182	0	0
147	144.8902	-1.98028058	13.22693532	0	0
148	145.8758	-1.99375188	12.89566888	0	0
149	146.8615	-2.00722318	12.56096348	0	0
150	147.8471	-2.02069447	12.22292017	0	0

Solar Beta Angle

151	148.8327	-2.03416577	11.88164001	0	0
152	149.8184	-2.04763707	11.53722405	0	0
153	150.804	-2.06110836	11.18977328	0	0
154	151.7897	-2.07457966	10.83938866	0	0
155	152.7753	-2.08805095	10.48617109	0	0
156	153.761	-2.10152225	10.13022135	0	0
157	154.7466	-2.11499355	9.771640138	0	0
158	155.7323	-2.12846484	9.410528024	0	0
159	156.7179	-2.14193614	9.046985457	0	0
160	157.7036	-2.15540744	8.681112748	1.451923425	0.004033
161	158.6892	-2.16887873	8.313010065	5.225529328	0.014515
162	159.6749	-2.18235003	7.942777426	7.177637281	0.019938
163	160.6605	-2.19582133	7.570514698	8.64427858	0.024012
164	161.6462	-2.20929262	7.196321589	9.843828208	0.027344
165	162.6318	-2.22276392	6.820297652	10.86422157	0.030178
166	163.6175	-2.23623522	6.442542283	11.75143836	0.032643
167	164.6031	-2.24970651	6.06315472	12.53310265	0.034814
168	165.5888	-2.26317781	5.68223405	13.22732622	0.036743
169	166.5744	-2.27664911	5.299879209	13.84671816	0.038463
170	167.56	-2.2901204	4.916188988	14.40044584	0.040001
171	168.5457	-2.3035917	4.53126204	14.89539322	0.041376
172	169.5313	-2.31706299	4.145196885	15.33685738	0.042602
173	170.517	-2.33053429	3.758091915	15.72898969	0.043692
174	171.5026	-2.34400559	3.370045409	16.07508732	0.044653
175	172.4883	-2.35747688	2.981155535	16.37779212	0.045494
176	173.4739	-2.37094818	2.591520363	16.63923024	0.04622
177	174.4596	-2.38441948	2.201237875	16.861112	0.046836
178	175.4452	-2.39789077	1.810405973	17.0448044	0.047347
179	176.4309	-2.41136207	1.419122496	17.1913843	0.047754
180	177.4165	-2.42483337	1.027485223	17.30167723	0.04806
181	178.4022	-2.43830466	0.635591891	17.37628544	0.048267
182	179.3878	-2.45177596	0.243540207	17.41560741	0.048377
183	180.3735	-2.46524726	-0.14857215	17.41985034	0.048388
184	181.3591	-2.47871855	-0.54064749	17.38903646	0.048303
185	182.3448	-2.49218985	-0.93258814	17.32300374	0.048119
186	183.3304	-2.50566115	-1.3242964	17.22140094	0.047837
187	184.3161	-2.51913244	-1.71567454	17.08367665	0.047455
188	185.3017	-2.53260374	-2.10662478	16.90906164	0.04697
189	186.2873	-2.54607503	-2.49704932	16.69654303	0.046379
190	187.273	-2.55954633	-2.88685026	16.44482836	0.04568
191	188.2586	-2.57301763	-3.27592965	16.15229626	0.044867
192	189.2443	-2.58648892	-3.66418942	15.81692928	0.043936
193	190.2299	-2.59996022	-4.05153145	15.43622149	0.042878
194	191.2156	-2.61343152	-4.43785747	15.00705003	0.041686
195	192.2012	-2.62690281	-4.82306913	14.52549292	0.040349
196	193.1869	-2.64037411	-5.20706794	13.98656409	0.038852
197	194.1725	-2.65384541	-5.58975527	13.38381571	0.037177
198	195.1582	-2.6673167	-5.97103237	12.70871731	0.035302
199	196.1438	-2.680788	-6.35080034	11.94963734	0.033193
200	197.1295	-2.6942593	-6.72896013	11.09006354	0.030806
201	198.1151	-2.70773059	-7.10541253	10.10522357	0.02807
202	199.1008	-2.72120189	-7.4800582	8.954886888	0.024875
203	200.0864	-2.73467319	-7.85279761	7.565182934	0.021014
204	201.0721	-2.74814448	-8.2235311	5.767295415	0.01602
205	202.0577	-2.76161578	-8.59215883	2.881031507	0.008003
206	203.0433	-2.77508708	-8.95858083	0	0
207	204.029	-2.78855837	-9.32269696	0	0
208	205.0146	-2.80202967	-9.68440694	0	0
209	206.0003	-2.81550096	-10.0436104	0	0
210	206.9859	-2.82897226	-10.4002067	0	0
211	207.9716	-2.84244356	-10.7540952	0	0
212	208.9572	-2.85591485	-11.1051753	0	0
213	209.9429	-2.86938615	-11.4533459	0	0
214	210.9285	-2.88285745	-11.7985061	0	0
215	211.9142	-2.89632874	-12.1405551	0	0
216	212.8998	-2.90980004	-12.4793916	0	0
217	213.8855	-2.92327134	-12.8149147	0	0
218	214.8711	-2.93674263	-13.1470234	0	0
219	215.8568	-2.95021393	-13.4756165	0	0
220	216.8424	-2.96368523	-13.8005932	0	0
221	217.8281	-2.97715652	-14.1218526	0	0
222	218.8137	-2.99062782	-14.439294	0	0
223	219.7994	-3.00409912	-14.7528168	0	0
224	220.785	-3.01757041	-15.0623207	0	0
225	221.7706	-3.03104171	-15.3677055	0	0
226	222.7563	-3.044513	-15.6688714	0	0
227	223.7419	-3.0579843	-15.965719	0	0

Solar Beta Angle

228	224.7276	-3.0714556	-16.2581491	0	0
229	225.7132	-3.08492689	-16.546063	0	0
230	226.6989	-3.09839819	-16.8293626	0	0
231	227.6845	-3.11186949	-17.1079503	0	0
232	228.6702	-3.12534078	-17.3817288	0	0
233	229.6558	-3.13881208	-17.650602	0	0
234	230.6415	-3.15228338	-17.9144742	0	0
235	231.6271	-3.16575467	-18.1732504	0	0
236	232.6128	-3.17922597	-18.4268367	0	0
237	233.5984	-3.19269727	-18.6751401	0	0
238	234.5841	-3.20616856	-18.9180684	0	0
239	235.5697	-3.21963986	-19.1555307	0	0
240	236.5554	-3.23311116	-19.3874371	0	0
241	237.541	-3.24658245	-19.6136991	0	0
242	238.5267	-3.26005375	-19.8342293	0	0
243	239.5123	-3.27352504	-20.0489419	0	0
244	240.4979	-3.28699634	-20.2577525	0	0
245	241.4836	-3.30046764	-20.4605781	0	0
246	242.4692	-3.31393893	-20.6573376	0	0
247	243.4549	-3.32741023	-20.8479514	0	0
248	244.4405	-3.34088153	-21.032342	0	0
249	245.4262	-3.35435282	-21.2104334	0	0
250	246.4118	-3.36782412	-21.382152	0	0
251	247.3975	-3.38129542	-21.5474259	0	0
252	248.3831	-3.39476671	-21.7061856	0	0
253	249.3688	-3.40823801	-21.8583638	0	0
254	250.3544	-3.42170931	-22.0038954	0	0
255	251.3401	-3.4351806	-22.1427179	0	0
256	252.3257	-3.4486519	-22.274771	0	0
257	253.3114	-3.4621232	-22.3999973	0	0
258	254.297	-3.47559449	-22.5183419	0	0
259	255.2827	-3.48906579	-22.6297524	0	0
260	256.2683	-3.50253708	-22.7341796	0	0
261	257.254	-3.51600838	-22.8315768	0	0
262	258.2396	-3.52947968	-22.9219004	0	0
263	259.2252	-3.54295097	-23.0051097	0	0
264	260.2109	-3.55642227	-23.0811672	0	0
265	261.1965	-3.56989357	-23.1500383	0	0
266	262.1822	-3.58336486	-23.2116916	0	0
267	263.1678	-3.59683616	-23.266099	0	0
268	264.1535	-3.61030746	-23.3132356	0	0
269	265.1391	-3.62377875	-23.3530796	0	0
270	266.1248	-3.63725005	-23.3856128	0	0
271	267.1104	-3.65072135	-23.4108202	0	0
272	268.0961	-3.66419264	-23.4286901	0	0
273	269.0817	-3.67766394	-23.4392143	0	0
274	270.0674	-3.69113524	-23.4423878	0	0
275	271.053	-3.70460653	-23.4382093	0	0
276	272.0387	-3.71807783	-23.4266806	0	0
277	273.0243	-3.73154913	-23.4078071	0	0
278	274.01	-3.74502042	-23.3815976	0	0
279	274.9956	-3.75849172	-23.348064	0	0
280	275.9813	-3.77196301	-23.3072219	0	0
281	276.9669	-3.78543431	-23.2590901	0	0
282	277.9525	-3.79890561	-23.2036906	0	0
283	278.9382	-3.8123769	-23.1410489	0	0
284	279.9238	-3.8258482	-23.0711935	0	0
285	280.9095	-3.8393195	-22.9941563	0	0
286	281.8951	-3.85279079	-22.9099722	0	0
287	282.8808	-3.86626209	-22.8186792	0	0
288	283.8664	-3.87973339	-22.7203183	0	0
289	284.8521	-3.89320468	-22.6149337	0	0
290	285.8377	-3.90667598	-22.5025723	0	0
291	286.8234	-3.92014728	-22.3832839	0	0
292	287.809	-3.93361857	-22.2571211	0	0
293	288.7947	-3.94708987	-22.1241391	0	0
294	289.7803	-3.96056117	-21.9843959	0	0
295	290.766	-3.97403246	-21.837952	0	0
296	291.7516	-3.98750376	-21.6848702	0	0
297	292.7373	-4.00097505	-21.525216	0	0
298	293.7229	-4.01444635	-21.3590569	0	0
299	294.7086	-4.02791765	-21.1864629	0	0
300	295.6942	-4.04138894	-21.007506	0	0
301	296.6798	-4.05486024	-20.8222602	0	0
302	297.6655	-4.06833154	-20.6308017	0	0
303	298.6511	-4.08180283	-20.4332082	0	0
304	299.6368	-4.09527413	-20.2295596	0	0

Solar Beta Angle

305	300.6224	-4.10874543	-20.0199373	0	0
306	301.6081	-4.12221672	-19.8044245	0	0
307	302.5937	-4.13568802	-19.5831056	0	0
308	303.5794	-4.14915932	-19.3560669	0	0
309	304.565	-4.16263061	-19.1233957	0	0
310	305.5507	-4.17610191	-18.8851809	0	0
311	306.5363	-4.18957321	-18.6415124	0	0
312	307.522	-4.2030445	-18.3924814	0	0
313	308.5076	-4.2165158	-18.1381801	0	0
314	309.4933	-4.22998709	-17.8787017	0	0
315	310.4789	-4.24345839	-17.6141404	0	0
316	311.4646	-4.25692969	-17.3445909	0	0
317	312.4502	-4.27040098	-17.0701492	0	0
318	313.4359	-4.28387228	-16.7909118	0	0
319	314.4215	-4.29734358	-16.5069756	0	0
320	315.4071	-4.31081487	-16.2184385	0	0
321	316.3928	-4.32428617	-15.9253987	0	0
322	317.3784	-4.33775747	-15.627955	0	0
323	318.3641	-4.35122876	-15.3262064	0	0
324	319.3497	-4.36470006	-15.0202527	0	0
325	320.3354	-4.37817136	-14.7101935	0	0
326	321.321	-4.39164265	-14.3961291	0	0
327	322.3067	-4.40511395	-14.0781599	0	0
328	323.2923	-4.41858525	-13.7563864	0	0
329	324.278	-4.43205654	-13.4309093	0	0
330	325.2636	-4.44552784	-13.1018297	0	0
331	326.2493	-4.45899914	-12.7692483	0	0
332	327.2349	-4.47247043	-12.4332663	0	0
333	328.2206	-4.48594173	-12.0939847	0	0
334	329.2062	-4.49941302	-11.7515045	0	0
335	330.1919	-4.51288432	-11.4059267	0	0
336	331.1775	-4.52635562	-11.0573524	0	0
337	332.1632	-4.53982691	-10.7058826	0	0
338	333.1488	-4.55329821	-10.3516179	0	0
339	334.1344	-4.56676951	-9.99465927	0	0
340	335.1201	-4.5802408	-9.63510727	0	0
341	336.1057	-4.5937121	-9.27306244	0	0
342	337.0914	-4.6071834	-8.90862519	0	0
343	338.077	-4.62065469	-8.54189577	3.430772932	0.00953
344	339.0627	-4.63412599	-8.17297428	6.0495762	0.016804
345	340.0483	-4.64759729	-7.8019607	7.773706892	0.021594
346	341.034	-4.66106858	-7.42895483	9.124064634	0.025345
347	342.0196	-4.67453988	-7.05405631	10.24848547	0.028468
348	343.0053	-4.68801118	-6.67736465	11.21428749	0.031151
349	343.9909	-4.70148247	-6.29897918	12.05889725	0.033497
350	344.9766	-4.71495377	-5.91899907	12.80565721	0.035571
351	345.9622	-4.72842506	-5.53752336	13.47026246	0.037417
352	346.9479	-4.74189636	-5.15465093	14.06383832	0.039066
353	347.9335	-4.75536766	-4.77048051	14.59458193	0.040541
354	348.9192	-4.76883895	-4.38511071	15.06871098	0.041858
355	349.9048	-4.78231025	-3.99864	15.49104627	0.043031
356	350.8904	-4.79578155	-3.61116672	15.86538716	0.044071
357	351.8761	-4.80925284	-3.22278912	16.19476286	0.044985
358	352.8617	-4.82272414	-2.83360532	16.48160591	0.045782
359	353.8474	-4.83619544	-2.44371335	16.72787482	0.046466
360	354.833	-4.84966673	-2.05321117	16.93514248	0.047042
361	355.8187	-4.86313803	-1.66219666	17.10466057	0.047513
362	356.8043	-4.87660933	-1.27076763	17.23740681	0.047882
363	357.79	-4.89008062	-0.87902184	17.33411932	0.04815
364	358.7756	-4.90355192	-0.48705701	17.39532125	0.04832
365	359.7613	-4.91702322	-0.09497084	17.42133749	0.048393

Solar Array	MicroSAT	Units	Comments
Solar constant	1367	watts/m²	SMAD p.333
β (Beta)	45	deg	
rho	1.162316332	Radians	ASIN(μ Earth/(μ Earth+Alt))
Pwr Reqt Daylight (Pd)	32.44211143	watts	From Power Budget Page
Pwr Reqt Eclipse (Pe)	32.44211143	watts	From Power Budget Page
Altitude	35734.6	km	
Orbital period (To)	1433.4	min	2*π/SQRT(μ Earth/(Rearth+Altsc)^3)/60
Period of Eclipse (Te)	1009.6	min	To*ACOS(COS(rho/COS(β)))/π
Period of Daylight (Td)	423.9	min	To-Te
Int pwr x-fer eff Eclipse (Xe)	0.6		Power system property
Int pwr x-fer eff Daylight (Xd)	0.8		
Req Solar Array Pwr (Psa)	169.3	watts	(Pe*Te/Xe+Pd*Td/Xd)/Td
Cell Efficiency (EOL)	0.243		At 15 years mission duration, not 2yrs
Power Output (Po)	332.2	watts/m²	EOL*Solar constant
			Property of UTJ
Inherent Degradation (Id)	0.96		solar cells
Theta (θ)	45	deg	0 for 1-axis gimballed
Pwr @ beg of life (Pbol)	225.5	watts/m²	Po*Id*COS(θ*π/180)
Design Life	2.0	years	Mission Property
Lifetime Degradation (Ld)	0.9973		(1-Annual Degradation/100)^Design Life
Annual Degradation	0.13	%	DL*(1/15)
Pwr @ end of life (Peol)	224.9	watts/m²	Pbol*Ld
Req Solar Array 1 Area (Asa1)	0.753	m²	Psa/Peol
Req Solar Array 2 Area (Asa2)	0.510	m²	Psa/solar constant/EOL
Solar Array Mass	**1.760**	**kg**	4*SAarea*1.76[kg/m^2]

Batteries	MicroSAT	Units	Comments
Battery Specific Energy Density	40	W*hr/kg	Li-Ion property
Voltage	15.00	volts	Determined by goal seeking cell M8
Current	2.16	Amps	15A Max continuous discharge(Required Power/Voltage)
Depth of Discharge (DOD)	70.00	%	
Number of Batteries (N)	4.00	#	
Transmission efficieny (n)	0.95		Battery Property
Battery Capability (Cr)	105.00	Watt-hrs	Eclipse Power Reqt
Battery Capability (Cr)	7.0	Amp-hrs	Nominal is 7[Amp*hr]
Individual Battery Mass	0.146	kg	Battery Property
Total Mass (kg)	**0.584**	**kg**	Total SC battery mass
Total Power System Mass	**2.58**	**kg**	**Solar Arrays, Batteries and cabling**

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	km³/kg*s²
μ Earth =	398600.4415	km³/s²
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Comparative Data

Cell Type	Efficiency	(link)
Silicon	0.148	1
GaAs	0.185	
Multijunction	0.22	
Ultra Triple Junction	0.283	

Ref: SMAD p.109, 333, 422; SAFT MP 176065 Integrat*ion* Batteries;
Spectrolab Ultra Triple Junction (UTJ GaInP2/GaAs/Ge) Solar cells

Power Budget:

Mission Design Data

GEO sep mass* [kg]	34
dsgn life [yr]	2
station lat [deg]	45
closest stbl pt [deg]	75
g [m/s²]	9.80665

*i=28.4[deg], 185[km x GEO]
Shuttle/IUS (pg 728)

Propellant & Dry Mass Calculation

Item	ΔV[m/s]	ISP[s]	Efficiency	RCS [kg]	SCM [kg]	SC [kg]
GEO sep mass						33.7065034
pre-burn RCS				0.010		33.6965034
GEO to Sub-GEO	803.26	1500	0.99		1.8	31.9
Post RCS to Sub-GEO				0.010		31.9
N-S Sta Kpg	102.76	1500	0.99	0.224		31.7
E-W Sta Kpg	2.9704671	1500	0.99	0.006		31.6
Sta reposition'g	0	1500	0.99	0.000		31.6
ACS				0.830		30.8
Deorbit	7	1500	0.99	0.015		30.8
RCS margin @ 10%				0.1		**30.7**

SCM -> Station Changing Motor

Mass Summary

Dry mass [kg]	**30.7**
RCS Prop [kg]	1.2
SCM Prop [kg]	1.8
GEO sep mass [kg]	33.7

PL RF Power Allocation

Max Power System mass [kg]	1.7	
Pwr gen capability needed [W]	32.4	SC Design Page
PWR gen from 1 ATJ array [W]	57.5	Worst case: 1 ATJ Solar array area illuminated.
Pwr Avail. w/ 10% margin [W]	51.8	PWR gen minus margin
PL Budget @ 70% Avail. [W]	36.2	(pg 316,345 Tables10-9,-35)
RF Pwr@ 35% efficiency [W]	0.33	System Characteristic (=Pta/Ptin)

Mission Design Data

Dry Mass =	28.60	kg	Propellant for μVAT is dry mass
Mission Duration =	2	years	
Station Longitude =	45	deg	Continuously changing
nearest stable point =	75	deg	At GEOSTA

Power Budget

Subsystem Peak Power Requirements

Command & Data Handling =	7.5	watts	During maximum computing
Attitude Control =	9.6	watts	10% during transmitting and imaging
Station Keeping =	50	watts	µVAT Propulsion
TT&C and Data Transciever =	32	watts	Maximum power output
Inertial Control Unit =	1	watts	MEMs Rate Sensor imbedded in Dynacon200
Payload =	10.94	watts	During picture taking
Power Control Unit =	4	watts	Parasitic power for solar array and battery control
Earth Sensor =	0.3	watts	
Sun Sensor =	0.3	watts	0.05 watts each for 6
Star Tracker =	2	watts	Continuous use, for constant attitude control
Thermal Control System =	80	watts	Maximum if all heaters are on at once
Total =	197.6375	watts	

Battery Information ***At full charge***

Battery Type :	SAFT MP 176065 Intrgration	
Voltage =	3.75	V
Current Capacity =	7	Ah
Charge Rate =	2 to 3	h
Charge/2 Rate =	3 to 4	h
Charge/5 Rate =	6 to 7	h
Power =	26.25	Wh
Max Continuous Discharge =	15	A
Pulse Discharge Current =	30	A
Total Number of batteries =	4	
Discharge Cutoff Voltage =	2.5	V
Battery Power Stored =	105	Wh

PL RF Power Allocation ***Detailed Worst Case Power Anaylsis***

	Bathing	Eclipse	At XMIT	Slewing	Sta Keeping	Imaging	Units
Max Pwr Sys mass =	2.6	2.6	2.6	2.6	2.6	2.6	kg
Solar Panel Pwr gen cap =	**51.8**	**0**	**0**	**0**	**0**	**0**	W
Pwr Avail w/ 10% margin =	46.6	0.0	0.0	0.0	0.0	0.0	W
Command & Data Handling =	1.1	3.0	3.8	7.5	3.8	7.5	W
Attitude Control =	1.0	1.0	1.0	9.6	9.6	1.9	W
Station Keeping =	0.0	0.0	0.0	0.0	50.0	0.0	W
TT&C and Data Transciever =	0.0	0.0	32.0	0.0	0.0	0.0	W
Inertial Control Unit =	1.0	1.0	1.0	1.0	1.0	1.0	W
Payload Power Use =	0.1	0.1	0.1	0.1	0.1	10.9	W
Power Control Unit =	4.0	4.0	4.0	4.0	4.0	4.0	W
Space Station Knowledge =	2.60	2.60	2.60	2.60	2.60	2.60	W
Thermal Control System =	20.0	40.0	0.0	20.0	0.0	20.0	W
Time Spent per Orbit =	**80469**	**4162**	**859**	**480**	**30**	**6**	**sec**
Power Used =	**29.8**	**51.7**	**44.4**	**44.8**	**71.1**	**48.0**	**W**
Solar cell Pwr NOT used =	16.78	-51.67	-44.42	-44.81	-71.06	-47.96	W
Fully Charged Battery =	105.00	105.00	105.00	105.00	105.00	105.00	Wh
System Power Available =	**121.78**	**53.33**	**60.58**	**60.19**	**33.94**	**57.04**	**W**

Power Budget

PL RF Power Allocation	***Likely Case Power Anaylsis***						
	Bathing	Eclipse	At XMIT	Slewing	Sta Keeping	Imaging	Units
Max Pwr Sys mass =	2.6	2.6	2.6	2.6	2.6	2.6	kg
Solar Panel Pwr gen cap =	**51.8**	**0.0**	**41.4**	**41.4**	**41.4**	**41.4**	W
Pwr Avail w/ 10% margin =	46.6	0.0	37.3	37.3	37.3	37.3	W
Command & Data Handling =	1.1	3.0	3.8	7.5	3.8	7.5	W
Attitude Control =	1.0	1.0	1.0	9.6	9.6	1.9	W
Station Keeping =	0.0	0.0	0.0	0.0	50.0	0.0	W
TT&C and Data Transciever =	0.0	0.0	32.0	0.0	0.0	0.0	W
Inertial Control Unit =	1.0	1.0	1.0	1.0	1.0	1.0	W
Payload Power Use =	0.1	0.1	0.1	0.1	0.1	10.9	W
Power Control Unit =	4.0	4.0	4.0	4.0	4.0	4.0	W
Space Station Knowledge =	2.60	2.60	2.60	2.60	2.60	2.60	W
Thermal Control System =	5.0	20.0	0.0	0.0	0.0	5.0	W
Time Spent per Orbit =	**80469**	**4162**	**859**	**480**	**30**	**6**	**sec**
Power Used =	**14.8**	**31.7**	**44.4**	**24.8**	**71.1**	**33.0**	**W**
Solar cell Pwr NOT used =	31.78	-31.67	-7.16	12.45	-33.80	4.30	W
Fully Charged Battery =	105.00	105.00	105.00	105.00	105.00	105.00	Wh
System Power Available =	**136.78**	**73.33**	**97.84**	**117.45**	**71.20**	**109.30**	**W**

Ref: SMAD p.334, 314-316, 412, 418-422, 423-425

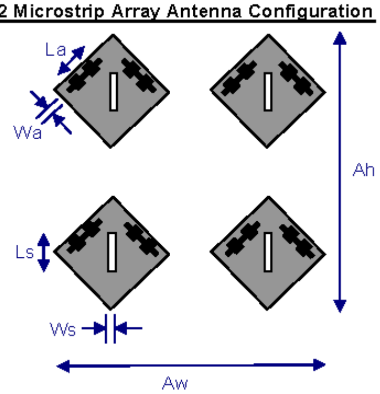
Constants			
	dius Earth =	6378.137	km
	mass Earth =	5.97333E+24	kg
	G =	6.673E-20	$km^3/kg*s^2$
	μ Earth =	398600.4415	km^3/s^2
	g =	9.80665	m/s
	MSD (mean solar day) =	0.985647	
	Earth axial tilt =	23.44241	deg

Link Budget:

Transmitter

	SAT to AFSCN		SAT to TDRSS		
Transmit Frequency (f) =	**2.1064**	**Ghz**	**2.1064**	**Ghz**	Standard to XMIT to TDRSS
Transmit Wavelength (λ) =	0.142		0.142	m	λ = c/f
Power Budget Allocation in watts (Ptin) =	15	watts	15	watts	Power Budget Allocation
Rectangular Slot Width (Ws) =	0.0028		0.0028	m	Ws = λ/50 -> Sharma
Rectangular Slot Length (Ls) =	0.0178		0.0178	m	Ls = λ/8 -> Sharma
Aperture Width (Wa) =	0.008		0.008	m	Given Sharma
Aperture Length (La) =	0.020		0.020	m	Given Sharma
Max Matching Stub Length (Sl) =	0.0352		0.0352	m	Sharma: Sl less than λ/4 -> (λ/4)-(λ*0.01/4)
Square Patch Side Length (Pl) =	0.0408		0.0408	m	Given Sharma
Inter-element Spacing Restriction =	0.1409		0.1409	m	Sharma: Less than λ -> λ-(λ*0.01)
Array Length (Al) =	0.1900		0.1900	m	Given Sharma: Elevation inter-element spacing 0.65λ
Array Height (Ah) =	0.1900		0.1900	m	Given Sharma: Azimuth inter-element spacing 0.8λ
Array area (Aa) =	0.0361		0.0361	m²	Aa = Al*Ah
Transmitter Efficiency (ηdc) =	0.33		0.33		Pta/Ptin
Available Transmit Power (Pta) =	5	watts	5	watts	Equipment Property
Transmitter Power in Decibels (Pt) =	6.990	dBw	6.990	dBw	10*LOG(Pta)
Antenna plus Transmitter Mass =	1.5	kg	1.5	kg	SMAD pg. 394, Table 11-26, Scaled to meet design
Transmitter Line Loss (Ll) =	-1	dB	-1	dB	
Transmit Antenna Beamwidth (θbt) =	33	deg	33	deg	**Adjust to meet ground target requirement**
Transmit Antenna Pointing Error (θet) =	1	deg	1	deg	
Assumed Antenna Efficiency (η) =	0.7		0.7		Sharma
Transmit Antenna Gain (Gt) =	14.2	dBi	14.2	dBi	Sharma (Assuming same results for f=2.1064 as f=2.6[GHz]
Equiv. Isotropic Rad. Pwr. (EIRP) =	**20.190**	**dBw**	**20.190**	**dBw**	Pt+Ll+Gt
Bandwidth (BW) =	**1053.2**	**MHz**	**1053.2**	**MHz**	Sharma: BW = f*0.01

2x2 Microstrip Array Antenna Configuration



Spatial Geometry for Satellite to Terrestrial

Sat Xmt Ant Max Cvg Ang (η°) =	0.288	rad	η° = 0.5*θbt
Earrth Central Angle (λ) =	-0.288	rad	λ = 180-(η-acos[sin(η)/(Re/Ro)]+90)
ECA (λ) in degrees =	-16.497	degrees	**This can reach the req ground targets w/ no slew**
Slant Range (S) =	36042.745	km	S = SQRT[(Ro-Re*cos(λ))^2 + (Re*sin(λ))^2]

Spatial Geometry for SAT to TDRSS Satellite Constellation (if necessary)

Sat #1 Orbit Radius^2 =			1773485438.5	km^2	SAT Radius^2
Sat #2 Orbit Radius^2 =			1777814701.9	km^2	TDRSS SC Operating Radius^2
Maximum Sat - Sat Distance =			59592.8	km	SQRT(SAT Rad^2+TDRSS Rad^2) Worst case seperation & earth would block line
Max Constellation SC - SC Distance =			126.0	deg longitude	Great TDRSS SC seperation (F-3 & F-4)
Seperation Arc Length (L) =			92723.5	km	L = θ*r
Max half-Sep between TDRSS SC =			46361.7	km	L/2
Slant Range at Max Sep =			48679.9	km	Linear Geometry Est + 5% margin: S = SQRT(OrbitSep^2+HalfSep^2)+5%
Coverage footprint Diameter =	20473.385	km			2*(S*SIN(η°))/(SIN(π/2)): plane geometry estimate
Coverage footprint in NM =	11054.748	NM			Coverage Footprint diameter*0.539957
Power Flux Density (PFD) =	-141.939	dB	-144.549	dB	PFD = EIRP/(4pS^2)
PFD/4kHz band =	-177.959	dB	-180.570	dB	PFD/4000
Space (path) Loss (Ls) =	-190.057	dB	-192.244	dB	Ls = 147.55-20log(S~m)-20log(f~Hz)
Propagation & Polarization Loss (La) =	-0.3	dB	-0.3	dB	SMAD Table 13-13

Receiver

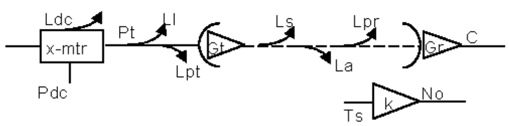
Assumed Antenna Efficiency (η) =	0.65		0.7		SMAD Figure of Merit p 553
Receiver Antenna Diameter (Dr) =	7	m	4.572	m	Smallest AFSCN dish size & TDRSS RCVR respectfully
Peak Receiver Antenna Gain (Gpr) =	41.912	dB	38.534	dB	G = -159.59+20*LOG(Dt)+20*LOG(f[GHz])+10*LOG(η)
Receiver Antenna Beamwidth (θbr) =	1.424	deg	2.181	deg	θ = 21/(D*f)
Receiver Antenna Pointing Error (θer) =	0.812	deg	1.190	deg	θer = θbr/2+0.1
Receiver Antenna Pointing Loss (Lθr) =	-3.902	dB	-3.576	dB	Lθ = -12*(θet/θbt)^2
Receiver Antenna Gain (Gr) =	38.010	dB	34.958	dB	Gpr+Lpr

Link Design Equations

System Noise Temperature (Ts) =	135	K	614	K	SMAD Table 13-10
Data Rate (R) =	3.00E+05	bps	3.00E+05	bps	SMAD pg. 385, Table 11-19
Eb/No (1) =	16.466	dB	4.976	dB	Eb/No = EIRP+Lpr+Ls+La+Gr+228.6-10LogTs-10LogR
Carrier-to-Noise Density Ratio (C/No) =	71.238	dB-Hz	59.747	dB-Hz	C/No = Eb/No+ 10*logR
Bit Error Rate (BER) =	1.00E-06	-------	1.00E-06	-------	BPSK Viterbi for TT&C, BPSK Reed-Soloman for Data Link
Required Eb/No (2) =	5.2	dB	2.8	dB	SMAD Figure 13-9
Implementation Loss (3) =	-2	dB	-2	dB	Estimate
Margin =	**9.266**	**dB**	**0.176**	**dB**	(1)-(2)+(3)

Alternate Approach

Carrier (C) =	-136.1	-141.0	dB	C = Pt*Ll*Gt*Ls*La*Gr
Noise (No) =	-207.3	-200.7	dB	No = k*Ts
Carrier/Noise Density Ratio (C/No) =	71.2	59.7	dB	C/No = C-No
Error Bits/Noise Ratio (Eb/No) =	16.5	5.0	dB	Eb/No = C/No-10*logR



Note: C=10^(-135.3/10)=~0.03 picowatts

Ref: Tomasi p.551-552; Sharma A wideband Microstrip Array Antenna with Unique Dumbell Shaped Aperture Coupled Radiating Elements
TDRSS info: http://msl.jpl.nasa.gov/QuickLooks/tdrssQL.html & http://msp.gsfc.nasa.gov/tdrss/tconst.html &http://msp.gsfc.nasa.gov/tdrss/scraft.html

Constants

Radius Earth (Re) =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	km^3/kg*s^2
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Spherical SC Analysis:

Item	Symbol	Small SAT	Units	Source	Comments
Surface Area	A	0.25	m^2	Geometry	Square Surface Area
Diam. Of equiv. Sphere	D	0.28	m^2	Geometry	
Max power dissipation	Qwmax	71.06	W	Power Budget	
Min power dissipation	Qwmin	9.79	W	Power Budget	
Altitude	H	35734.64	km	Calculated	Orbital Radius - Earth's Mean Sea Level Radius
Earth Radius	Re	6378.14	km	Given	
Earth angular radius	rho	0.15	radians	Eq 5-16	
Albedo correction	Ka	0.74		Eq 11-28	
Max Earth IR emission @ surface	qmax	258	W/m^2	pg 447	
Min Earth IR emission @ surface	qmin	216	W/m^2	pg 447	
Direct solar flux	G	1399	W/m^2	pg 447	
Albedo	al	35	%	pg 447	
Emmisivity	ε	0.84		Table 11-46	3M Black Velvet paint
Absorptivity (solar)	α	0.97		Table 11-46	3M Black Velvet paint; BOL value
Stefan-Boltzmann constant	σ	5.67E-08	$W/(m^2.K^4)$	Biven	
Earth view factor=(1-cosρ)/2	vf	0.0058		pg 448	
Sphere x-section area=πD^2/4	Acx	0.0625	m^2	Geometry	
Solar input	Acx*G*α	84.81	W	calc	
Earth input	A*vf*qmax*ε	0.31	W	calc	
Albedo input	A*vf*G*al*af*Ka/100	0.51	W	calc	
Worst case hot temp	Tmax	65.70	C	eq 11-34	
Worst case cold temp	Tmin	-102.53	C	eq 11-35	
Upper temp limit	Tu	60	C	Equipment Data Sheets	
Lower temp limit	Tl	0	C	Equipment Data Sheets	
Radiator area (WC hot)	Ard	0.121	m^2	eq 11-19	
Radiator temp (WC cold)	Tr	-2.05	C	eq 11-20	
Heater power for lower limit	Qn	5.89	W	eq 11-21	

Ref: SMAD p.446-456

Equipment Temperature Limits

Spacecraft Internal Units	Temp Range [°K]	
Worst Case Envelope	273	333
Payload		
Optical Sensors (CCD most temp sensitive)	273	333
Onboard Computer	233	358
TT&C Units	243	333
Electrical Power		
Batteries	253	333
Solar Arrays	168	383
Attitude Control		
Earth Sensors	243	353
Sun Sensors	243	353
Star Tracker	243	353
Inertial Measurement Unit (IMU)	243	333
Reaction Wheels	243	333
Propulsion	213	353
Processors		
AFRL RAD6000 Computer (microprocessor)	253	333
Fault Tolerant Reconfigurable Processor	253	333
Thermal Control		
MLI	113	523
Radiators	178	333
Heaters, thermostats, heat pipes	238	333
Minco CT325 Thermal Controller	233	343
Antennas		
2x2 Microstrip Array	233	358

Ref: SMAD p.428, Various Equipment Data Sheet

Thermal Hardware Mass-Power

Hardware Properties - Mass and Power:

Hardware	Mass [kg]	Power [W]	Comments		
MLI	0.9782	0	.73kg/m^2*(As/c-Arad)	SMAD p.457, Table 11-49	Based on 15 layers
Heaters (3)	0.035	80	3 Kapton heaters sized for components	(2) KH-202/(*)-P, (1) KH-404/(*)-P	
Thermostats					
Thermisters					
Adhesives/Paints	0.120	0	Spread throughout the spacecraft		
Heat Pipes (NH3)	0.225	0	.15kg/m*1.5m	SMAD p.457, Table 11-49	
Radiator Panels	0.400	0	3.3 kg/m^2*Area	SMAD p.457, Table 11-49	
Electronic Controllers	0.024	0	.2kg/ and 1-3W/	SMAD p.457, Table 11-49	
Radiative Coupler	0.005	0			
TOTALS	**1.788**	**80**	***If all heaters are on at the same time, which they will not be.***		

Ref: SMAD p.457

Cost Estimation:

COTS System components

Components	Quantity	Cost FY2007 Dolars	Cost FY2000 Dolars	
Batteries -	4	$1,600.00	$1,393.73	SAFT MP176065 Integration
Solar Cells -	1 [m^2]	$52,896.00	$46,076.66	Spectrolab UTJ (GaInP2/GaAs/Ge) *($ est using CER SMAD p.797)*
Command & Data Handling -	1	$500,000.00	$435,540.07	AFRL RAD6000 Computer (Microprocessor)
Propulsion/Thrusters -	1	*$20,236.76*	$17,627.84	Micro Aerospace Solutions Vacuum Arc Thrusters (VAT) *($ est using CER SMAD p.797)*
Stability Control -	4	$180,000.00	$156,794.43	Dynacon MicroWheel 200 (3.2[W] max with rate sensor)
Data Transciever -	1	$150,000.00	$130,662.02	AeroAstro Modular S-Band Radio System
Star Tracker -	1	$75,000.00	$65,331.01	AeroAstro Miniature Star Tracker (without AeroAstro baffle; custom baffle incorporated into structure)
Inertial Control Unit -	1	$0.00	$0.00	*Imbedded into Command & Data Handling CPU using inputs from rate sensor, cmds to RWs.*
Power Control Unit -	1	$60,000.00	$52,264.81	Reconfigurable Fault Tolerant Processor configured for power management, back-up photo processing and attitude control.
Earth Sensor -	6	$90,000.00	$78,397.21	Optical Energy Tech
Sun Sensor -	6	$240,000.00	$209,059.23	Optical Energy Tech
TT&C Transciever -	1	$150,000.00	$130,662.02	AeroAstro Modular S-Band Radio System

Custom System Components (Cost Estimating for Small Satellites including RDT&E and Theoretical First Unit)

Components	Quantity	Parameter	Cost FY2000 Dolars	
Optical Payload -	1	0.25	$132,723.40	SMAD CERs for visible paylaod (p.795&6) plus cost of Kodak sqaure matrix CCD KAF-39000
Antenna -	1	5.33	$12,626.82	SMAD CERs for communications subsystem (p.795&6)
Structure -	1	33.71	$1,982.68	SMAD CER for structures (p.797), using total S/C mass with 10% margin included
Thermal Control -	1	1.83	$9,270.07	SMAD CER for thermal (p.797) plus control cost with estimate for mini readiator.
All Computer Code -	1	250000	$108,750.00	SMAD CER (p.800); One Time Incurred Cost
Satellite per Constellation(Csn) -	15			
TFU Cost -	**1**		**$1,589,162.00**	Sum of all COTS and Custom components.
Learning Curve Slope (S) -		90%		Recommended S percent for 10 to 50 units to be build by SMAD, p.809.
Learning Cuvre Factor (L) -	15	9.94		(Number of satellites)^(1-((LN(100%/S))/(LN(2)))); SMAD p.809.
Cnstl Production Cost (C_{pc}) -	1		$15,793,919.50	$L*C_{sn}$
Average Satellite Cost -	15		$1,052,927.97	C_{pc}/C_{sn}
Integration cost per satellite(S_{int}) -	1		$244,272.41	Assuming 10% of satellite's construction cost to integrate into a free/shared launch
Cstl Integration Cost (C_{int}) -	1		$2,427,706.45	$S_{int}*C_{sn}$
Cstl Launch Cost -	1		$0.00	Assuming free/shared launch
Cost to field Cstl -	1		$18,221,625.95	$C_{pc}+C_{int}+$(Cstl Launch Cost)

Constellation Operations and Support Cost Determination

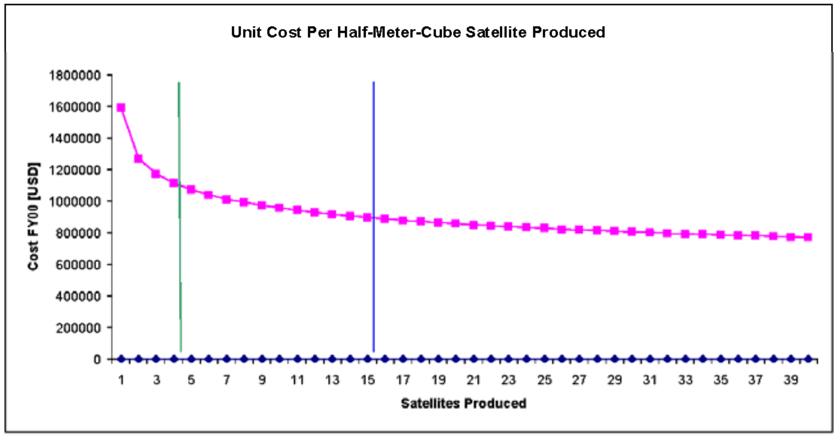
Contractor Labor -	6	1 [year]	$960,000.00	SMAD, p.801
Government/Military Labor -	9	1 [year]	$990,000.00	SMAD, p.801
Maintenance Labor -	0	1 [year]	$0.00	Free due to assumption that constellation will utilize existing facilities.
Cost to operate Cstl per year -	1		$1,950,000.00	6 Contractors and 9 Military employees needed per 15 satellites per year

	Cost FY2000 Dolars	Cost FY2007 Dolars	
TFU Cost -	**$1,589,162.00**	**$1,771,597.79**	
Total Government Cost -	**$22,121,625.95**	**$24,661,188.61**	**To field constellation and operate for only two years.**

Cost Est.

Determination of Unit Cost Curve over Satellite Constellation Production

Unit Number	Production Cost	Average Cost	Unit Cost
1	$1,589,162.00	$1,589,162.00	$1,589,162.00
2	$2,860,491.59	$1,430,245.80	$1,271,329.60
3	$4,034,275.18	$1,344,758.39	$1,173,783.59
4	$5,148,884.87	$1,287,221.22	$1,114,609.68
5	$6,221,463.71	$1,244,292.74	$1,072,578.84
6	$7,261,695.33	$1,210,282.55	$1,040,231.62
7	$8,275,775.37	$1,182,253.62	$1,014,080.04
8	$9,267,992.76	$1,158,499.09	$992,217.39
9	$10,241,483.43	$1,137,942.60	$973,490.67
10	$11,198,634.67	$1,119,863.47	$957,151.24
11	$12,141,321.13	$1,103,756.47	$942,686.46
12	$13,071,051.59	$1,089,254.30	$929,730.46
13	$13,989,064.97	$1,076,081.92	$918,013.38
14	$14,896,395.67	$1,064,028.26	$907,330.69
15	$15,793,919.50	$1,052,927.97	$897,523.83
16	$16,682,386.97	$1,042,649.19	$888,467.47
17	$17,562,447.92	$1,033,085.17	$880,060.96
18	$18,434,670.17	$1,024,148.34	$872,222.25
19	$19,299,553.80	$1,015,765.99	$864,883.63
20	$20,157,542.41	$1,007,877.12	$857,988.61
21	$21,009,031.98	$1,000,430.09	$851,489.58
22	$21,854,378.03	$993,380.82	$845,346.05
23	$22,693,901.37	$986,691.36	$839,523.34
24	$23,527,892.86	$980,328.87	$833,991.49
25	$24,356,617.36	$974,264.69	$828,724.50
26	$25,180,316.95	$968,473.73	$823,699.60
27	$25,999,213.76	$962,933.84	$818,896.80
28	$26,813,512.20	$957,625.44	$814,298.45
29	$27,623,401.06	$952,531.07	$809,888.85
30	$28,429,055.09	$947,635.17	$805,654.04
31	$29,230,636.58	$942,923.76	$801,581.49
32	$30,028,296.54	$938,384.27	$797,659.96
33	$30,822,175.86	$934,005.33	$793,879.32
34	$31,612,406.26	$929,776.65	$790,230.40
35	$32,399,111.14	$925,688.89	$786,704.87
36	$33,182,406.31	$921,733.51	$783,295.17
37	$33,962,400.68	$917,902.72	$779,994.37
38	$34,739,196.84	$914,189.39	$776,796.16
39	$35,512,891.57	$910,586.96	$773,694.73
40	$36,283,576.33	$907,089.41	$770,684.76



Ref: SMAD p.784- 802.

APPENDIX B. 5U-CUBESAT DESIGN EXCEL WORKBOOK

Mission Reqmts	N/A	~~N/A~~	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Sub-GEOSTA Altitude [km] 35765.37	L/V	~~N/A~~							Sep mass [kg] **14.33**
~~GEO station~~	~~N/A~~	~~AKM~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~	~~Propellant [kg]~~
Thrust for GEO/Inc. and deorbit	N/A	~~N/A~~	RCS	3-axis stabalized		despin on station & S/C pointing			Total S/C mass [%] 10.91
BER 10E-6 TM & Data @ 300kbps with BPSK modulation	Sub-GEOSTA	~~N/A~~	Station keeping	P/L & TT&C		point reqs. [degrees] **7.07**			Total S/C mass [%] 9.60
Li-ion 80%DOD 2 year life, %eff 70%	Elect. Interface req'd	~~N/A~~	50W µVAT actuation 4.2W R.W's + Rate Sensor	15Vdc @ 26.4Vdc 15W TT&C and Data, 5W C&DH	EPS	power for sensors and computers	Solar panel wings & body mounted pane produce 39.1W & 4 batteries (105Wh)		Total S/C mass [%] 32.36
Mom-bias with minimum Zcp [cm] -0.034	N/A	~~N/A~~	ISP [sec] 1500	3-axis stab mom.	Point solar arrays perp to sun to charge batteries	ADCS	N/A	N/A	Total S/C mass [%] 3.00
Weight << Shuttle GTO sep mass 2 year MMD		~~N/A~~	Pumpkin 3-Axis Stabilized ACS X-axis, Y-axis, Z-axis, No Spare		Body Mounted and wing mounted deployable Solar Panels	Star, Sun and Earth sensor inputs to onboard orbit propagator	SMS	Radiator space $[m^2]$ 0.019	Total S/C mass [%] 19.00
Worst Case Hot 333[K] Worst Case Cold 273[K]	N/A	~~N/A~~	Reaction Wheels 243K to 333K	CCD Operating Limits 273K to 333K	Batteries parameters (must not freeze or overheat) 253K to 333K	Earth and Sun Sensors 243K to 353K	N/A	TCS	Total S/C mass [%] 12.00
0.5m x 0.5m x 0.5m >=90[kg] sep mass **L/V dependant adapter fairing**	High risk if failure of L/V, but high reliability L/V	~~N/A~~	Reaction Wheels and Rate Sensor [kg] **0.93**	Custom Optics Package +CCD [kg] **0.85**	Solar arrays and Batteries [kg] **1.80**	Sun & Earth Sensors [kg] **2.34**	Aluminum 0.5m cube [kg] **0.67**	Temp Ctlr, Kapton Htrs [kg] **0.49**	Total Mass with Margin [kg] **14.33**
Cost	**Launch Vehicle** **$0**	**~~Apogee Kick Motor~~** **$0**	**Reaction Control System** **$130,809**	**Paylaod & TT&C** **$522,128**	**Electrical Power System** **$192,034**	**Attitude Control System** **$131,103**	**Structures and Mechanisms** **$1,295**	**Thermal Control Systems** **$4,257**	**Average Build Cost per Satellite** **$837,621**

5U CubeSAT		Units	Comments
Overall Mission	Optical Survey		
Desired CPA from Target	20.5	km	
Orbital Radius	42143.5	km	Adjust N2 chart when this is changed; GEOSTA radius is 42155[km]
Orbital Altitude	35765.4	km	
SC Orbit Insertion Incl.	0.0	deg	
Targets' Altitude	35785.9	km	Assuming all targets will be at exactly Geostationary orbit
Targets' Inclination	0.0	deg	Assuming all targets will be at exactly Geostationary orbit
Max S/C Mass (Estimate)	14.3	kg	
Mission Design Life (MDL)	2	years	

Major System components

Payload -	Custom Optical Telescope Package using Kodak sqaure matrix CCD KAF-383000
Batteries -	SAFT MP176065 Integration
Solar Cells -	Spectrolab UTJ (GaInP2/GaAs/Ge)
Command & Data Handling -	AFRL RAD6000 Computer (microprocessor)
Propulsion/Thrusters -	Micro Aerospace Solutions Vacuum Arc Thrusters (VAT)
Stability Control -	Pumpkin Miniature 3-Axis Reaction Wheel & Attitude Determination and Control System
Data Transciever -	AeroAstro Modular S-Band Radio System
Inertial Control Unit -	*Imbedded into Command & Data Handling CPU using inputs from rate sensor, cmds to RWs.*
Power Control Unit -	Reconfigurable Fault Tolerant Processor configured for power management, back-up photo processing and attitude control.
Earth Sensor -	Optical Energy Tech
Sun Sensor -	Optical Energy Tech
Thermal Control System -	Minco CT325 Thermal Control Module, Thermal coatings, Radiator, Heat Pipes, Multi-layer Insulation (MLI), Kapton Heaters
Antenna -	Microstrip Patch-Fed Short Backfire Antenna Array
TT&C Transciever -	AeroAstro Modular S-Band Radio System

Estimation of Spacecraft Design Characteristics:

Parameter	5U CubeSAT	Units	Equation	Comments
Earth radius (Er)	6378.137	km	base*height*width	
Orbit Radius (Or)	42144	km		Selected
Earth Angular Radius (OAr)	0.15	Rad	ASIN(Er/Or)	
Payload:				
Mass (P/Lm)	0.85	kg		P/L scaled from IKONOS on Optical P/L sheet
Power (P/Lp)	1.75	W		P/L scaled from IKONOS on Optical P/L sheet
Spacecraft:				
Dry Mass (Dm)	3.12	kg	P/Lm/0.274	**This mass is too low, SMAD eq must not model correctly at this size.**
Average Power (Ap)	3.89	W	P/Lp/0.45	
Orbit period (Op)	23.92	hr	2*π*SQRT(Or^3/μ)/3600	
Eclipse Period (Te)	1.16	hr	Op*ACOS(COS(EAr)/COS(0))/π	
Solar Array Power	5	W	(Ap*Te/0.6+Ap*(Op-Te)/0.8)/(Op-Te)	Estimate of needed power
Solar Array Design	E-W Trkg			S/C uses reaction wheels to track sun
Control Approach	*3- axis nadir pointing*			
Propellant:				
ΔV ~m/s	10.00	m/s		<-This is only accounting for S/C disposal
Mp~kg	0.0021	kg	Dm*(EXP(ΔV/Isp/g)-1)	
Attitude control + residuals	0.0001	kg	Mp*0.07	
Margin	0.0003	kg	(Mp+AttitudeCtrl)*0.15	
Total propellant	**0.0026**	kg	Mp+AttitudeCtrl+Margin	
Propulsion Isp	1500	sec		Propulsion system property'
Spacecraft loaded mass	3.12	kg	TotalPropellant+Dm	
Spacecraft size and MOI				
Volume	0.125	m^3	base*height*width	S/C Property since it's a cube
Linear dimensions	0.500	m		Choosen
Body x-sectional area	0.250	m^2	Ld^2	
MOI	**0.067**	kg*m^2		
		N.m*s^2		

Constants

μ = 398600.4418

g = 9.80665

Ref: SMAD p.303-317

Spacecraft Mass Estimation:

Element	Estimated % of Dry SC %	Payload %	Est mass based on Dry SC kg	Actual Mass based on Selected Equipment kg	
Payload	27.50	100.00	0.858	0.855	
Structures	19.00	69.09	0.593	0.670	Aluminum 5U Structure+ArrayWings, no bulkheads.
Thermal	12.00	43.64	0.374	0.489	TempCtrlModule+KaptonHtrs+Radiator+MLI+Coatings
Power	8.90	32.36	0.278	1.803	Batteries+SolarArrays+PowerctrlModule+RFTP
TT&C	9.60	34.91	0.300	1.415	Antenna+Transceiver+MiscWring(etc)
ACS	3.00	10.91	0.094	2.340	SunSensors+EarthSensors+Processor&MountingStruct
Prop (dry)	13.00	47.27	0.406	2.000	Propellant+µVAT
Reaction Control System	3.00	10.91	0.094	0.930	ReactionWheels+MEMs(RateSensor)
Margin [kg]	4.00		0.12		
SC dry [kg]			3.12	11	
Prop mass [kg]			0.00	0	
SC loaded [kg]			3.12	11	SC no Margin
Margin % Dry SC			4.00	11.55	SC Mass + 10%Margir

Space Craft Selected Equipment

Element	Actual Dry Mass [kg]	Actual % for Dry [%]	
Batteries (1-4)	0.584	4.5%	4 Batteries, 146[g] each
Command & Data Handling	1.500	11.5%	AFRL RAD6000 Computer + Misc mounting structure
Reaction Wheels	0.910	7.0%	Pumpkin Miniature 3-Axis Reaction Wheel & ADCS
Data & TT&C Radio System	0.800	6.1%	AeroAstro Modular S-Band Radio: Rx/Tx&HPA/Pwr&Inter Modules
Data Handling & TT&C Antenna	0.500	3.8%	Microstrip Patch-Fed Short Backfire Antenna Array
Inertial Reference Unit	0.020	0.2%	Pumpkin Magnetometer (packaged on top of RWs' housing)
Payload	0.855	6.6%	Custom Optical package
Power Control Unit/RFTP	1.030	7.9%	ThermoFoil CT325 Miniture DC Controller
Solar Arrays	0.355	2.7%	1 body mounted and 4 mounted wing type UTJ solar arrays
Sun Sensors	2.400	18.4%	Includes 6 Sensors at 0.04 kg each
Earth Sensors	0.600	4.6%	Includes 6 Sensors at 0.001 kg each
Propulsion Unit	2.000	15.4%	Micro Aerospace solutions µVAT
Radiator	0.059	0.5%	Area*3.3[kg/m2]; SMAD Table 11-49
Structures	0.670	5.1%	Aluminum Cube with no bulkheads
TT&C Miscellaneous	0.260	2.0%	Coax cables, Filters, Switchers & Diplexers
Thermal	0.430	3.3%	KaptonHtrs-Radiator+TempCtrlModule+MLI+Coatings
UDI	0.051	0.4%	Uniformly Distributed Items
SC (No Margin)	13.025	100.0%	
Margin	0.100	10.0%	
Mass Margin	1.302	10.0%	
SC (With Margin)	**14.327**	110.0%	

Space Craft Mass of Subsystem Categories

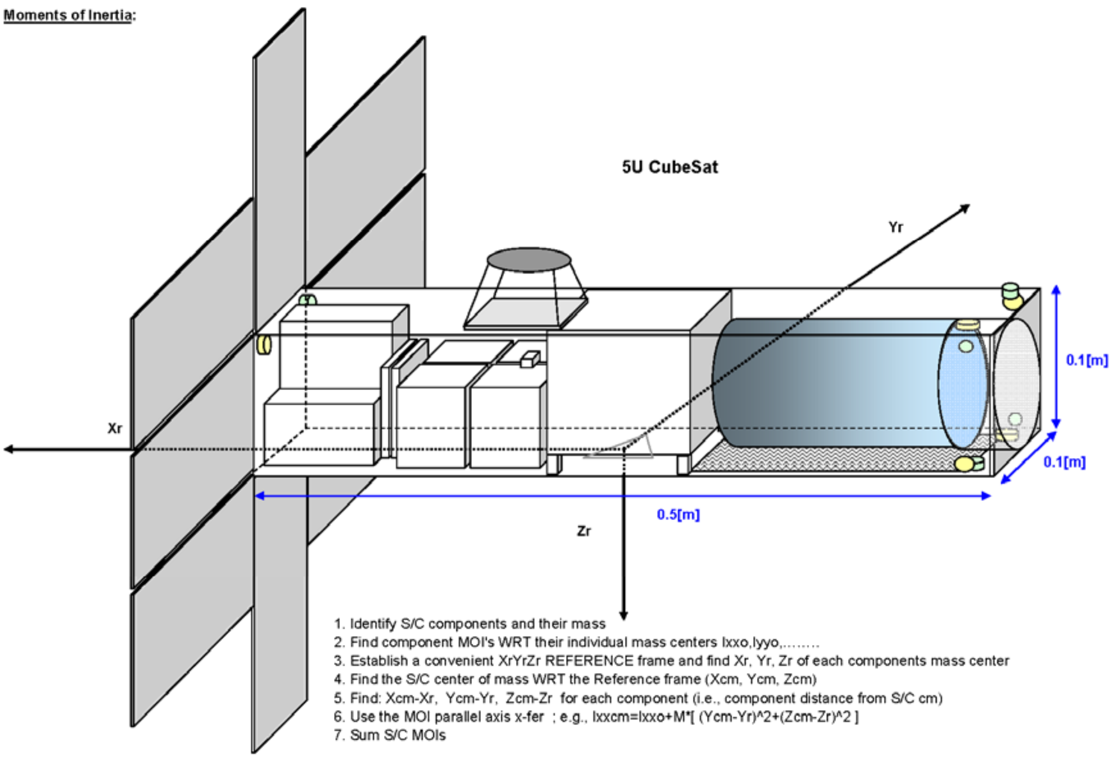
	[kg]	[% of SC]
RCS Subsystem	2.00	13.96%
Payload Subsystem	0.85	5.97%
Comms & TT&C Subsystems	3.06	21.36%
Electrical Power Subsystem	1.97	13.75%
ACS Subsystem	3.93	27.43%
Structures	0.67	4.68%
Thermal Control Subsystem	0.49	3.42%
Uniformly Distributed Items	0.05	0.35%
10% Margin (of dry spacecraft)	1.30	10.00%
SC Mass with Margin	14.33	110%

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
µ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p.341

Moments of Inertia:



1. Identify S/C components and their mass
2. Find component MOI's WRT their individual mass centers Ixxo,Iyyo,........
3. Establish a convenient XrYrZr REFERENCE frame and find Xr, Yr, Zr of each components mass center
4. Find the S/C center of mass WRT the Reference frame (Xcm, Ycm, Zcm)
5. Find: Xcm-Xr, Ycm-Yr, Zcm-Zr for each component (i.e., component distance from S/C cm)
6. Use the MOI parallel axis x-fer ; e.g., Ixxcm=Ixxo+M*[(Ycm-Yr)^2+(Zcm-Zr)^2]
7. Sum S/C MOIs

Step 1 - Major Component Masses

Item	Name	Shape	Mass		Component MOI equations
1	Battery 1	box	0.146	kg	for boxes Depth (d) in x, Width (w) in y, Height (h) in z
2	Battery 2	box	0.146	kg	Ixo= Iyo =(1/12)*M*(3(R^2 + r^2) + h^2); Izo=(1/2)M(R^2 + r^2)
3	Battery 3	box	0.146	kg	Ixo= Iyo =(1/12)*M*(3R^2+h^2); Izo=(1/2)MR^2
4	Battery 4	box	0.146	kg	Ixo=(1/12)*M*(w^2+h^2); Iyo=(1/12)*M*(d^2+h^2);Izo=(1/12)*M*(d^2+w^2)
5	Side 1 (front w/Ap)	solid sheet	0.007	kg	0.1x0.1[m] Aluminum 7075-T73 - 0.098[m] Aperature for P/L +5% margin
6	Side 2 (back w/Solar Array)	solid sheet	0.049	kg	0.1x0.1[m] Aluminum 7075-T73 -0.1x0.1[m] Square Spectolab solar array +5% margin
7	Side 3	solid sheet	0.147	kg	0.1x0.5[m] Aluminum 7075-T73 +0.24[m^2] + 5% margin
8	Side 4	solid sheet	0.087	kg	0.1x0.5[m] Aluminum 7075-T73 +0.24[m^2] - Radiator Area - µVAT area + 5% margin
9	Side 5	solid sheet	0.147	kg	0.1x0.5[m] Aluminum 7075-T73 +0.24[m^2] + 5% margin
10	Side 6	solid sheet	0.123	kg	0.1x0.5[m] Aluminum 7075-T73 +0.24[m^2] - Antenna Array area + 5% margin
11	Solar Array Wing 1	solid sheet	0.124	kg	0.2x.1[m] Spectolab solar array wing + 5% margin
12	Solar Array Wing 2	solid sheet	0.043	kg	0.2x.1[m] Spectolab solar array wing + 5% margin
13	Solar Array Wing 3	solid sheet	0.124	kg	0.2x.1[m] Spectolab solar array wing + 5% margin
14	Solar Array Wing 4	solid sheet	0.043	kg	0.2x.1[m] Spectolab solar array wing + 5% margin
15	C&DH Proc	box	1.500	kg	AFRL RAD6000 Computer with margin to account for harnessing and other structures
16	RFTP/Power Control Unit	box	1.030	kg	RFTP Processor stacked on top of RAD6000
17	Reaction Wheels & IRU	box	0.930	kg	Pumpkin Miniature 3-Axis Reaction Wheel & Attitude Determination and Control System
18	Temp Controller	box	0.028	kg	CT325 Miniature DC Controller
19	C&DH/TT&C Antenna	box	0.500	kg	Microstrip Patch-Fed Backfire Antenna Array
20	Antenna Reflector	solid cyl	0.150	kg	Reflector and mounting structure
21	Payload	solid cyl	0.855	kg	Custom Optical P/L utilizing Kodak KAF-39000 CCD with deployable sun shade
22	Sun Sensor 2	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
23	Sun Sensor 3	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
24	Sun Sensor 4	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
25	Sun Sensor 5	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
26	Sun Sensor 6	solid cyl	0.400	kg	OET Model 0.05 Sun Sensor
27	Earth Sensor 2	solid cyl	0.100	kg	OET Earth Sensor
28	Earth Sensor 3	solid cyl	0.100	kg	OET Earth Sensor
29	Earth Sensor 4	solid cyl	0.100	kg	OET Earth Sensor
30	Earth Sensor 5	solid cyl	0.100	kg	OET Earth Sensor
31	Earth Sensor 6	solid cyl	0.100	kg	OET Earth Sensor
32	Transciever&HPA Module	box	0.400	kg	AeroAstro Modular S-Band Radio
33	Receiver Module	box	0.300	kg	AeroAstro Modular S-Band Radio
34	Interface/Power Module	box	0.300	kg	AeroAstro Modular S-Band Radio
35	µVAT	solid tri	2.000	kg	Micro Aerospace Solutions µVAT sized for 50[kg] SC
36	Radiator	solid rec	0.059	kg	Sized from Thermal needs; mass calculated using SMAD table 11-49
37	TT&C Miscellaneous	UDI	0.260	kg	Includes Coax cables, Filters, Switchers and Diplexers
38	Thermal Miscellaneous	UDI	0.430	kg	Evenly distributed: Heat pipes, heaters and coatings
39	Miscellaneous	UDI	0.051	kg	Uniformly distributed Items throughout SC
	Σ SC Mass =		12.77	kg	*Slightly lighter than SC Mass due to bulkhead calculations of actual size.
	Σ SC Mass w/ Margin at 10% =		14.05	kg	

Step 2 - Component Moment of Inertia about Center of Mass (R is outter radius, r is inner radius)

Item	Name	Shape	d (x) or R	w (y) or r	h (z)	Ixxo	Iyyo	Izzo
1	Battery 1	box	0.0450	0.0192	0.0684	0.000061	0.000081	0.000029
2	Battery 2	box	0.0450	0.0192	0.0684	0.000061	0.000081	0.000029
3	Battery 3	box	0.0450	0.0192	0.0684	0.000061	0.000081	0.000029
4	Battery 4	box	0.0450	0.0192	0.0684	0.000061	0.000081	0.000029
5	Side 1 (front w/Ap)	solid sheet	0.1000	0.0050	0.1000	0.000006	0.000012	0.000006
6	Side 2 (back w/Solar Array)	solid sheet	0.1000	0.0050	0.1000	0.000041	0.000081	0.000041
7	Side 3	solid sheet	0.5000	0.0050	0.1000	0.000123	0.003185	0.003063
8	Side 4	solid sheet	0.5000	0.0050	0.1000	0.000073	0.001895	0.001822
9	Side 5	solid sheet	0.5000	0.0050	0.1000	0.000123	0.003185	0.003063
10	Side 6	solid sheet	0.5000	0.0050	0.1000	0.000103	0.002669	0.002567
11	Solar Array Wing 1	solid sheet	0.0025	0.2000	0.3000	0.001348	0.000933	0.000415
12	Solar Array Wing 2	solid sheet	0.0025	0.1000	0.2000	0.000179	0.000143	0.000036
13	Solar Array Wing 3	solid sheet	0.0025	0.2000	0.3000	0.001348	0.000933	0.000415
14	Solar Array Wing 4	solid sheet	0.0025	0.1000	0.2000	0.000179	0.000143	0.000036
15	C&DH Proc	box	0.0950	0.0050	0.0700	0.000616	0.001741	0.001131
16	RFTP/Power Control Unit	box	0.0150	0.0050	0.0700	0.000423	0.000440	0.000021
17	Reaction Wheels & IRU	box	0.0150	0.9500	0.0700	0.070324	0.000397	0.069961
18	Temp Controller	box	0.0038	0.0254	0.0277	0.000003	0.000002	0.000002
19	C&DH/TT&C Antenna	box	0.0530	0.0530	0.0112	0.000122	0.000122	0.000234
20	Antenna Reflector	solid cyl	0.0230	0.0000	0.0020	0.000000	0.000038	0.000038
21	Payload	solid cyl	0.1501	0.0000	0.0980	0.000320	0.001071	0.000751
22	Sun Sensor 2	solid cyl	0.0150	0.0000	0.0100	0.000003	0.000011	0.000008
23	Sun Sensor 3	solid cyl	0.0150	0.0000	0.0100	0.000003	0.000011	0.000008
24	Sun Sensor 4	solid cyl	0.0150	0.0000	0.0100	0.000003	0.000011	0.000008
25	Sun Sensor 5	solid cyl	0.0150	0.0000	0.0100	0.000003	0.000011	0.000008
26	Sun Sensor 6	solid cyl	0.0150	0.0000	0.0100	0.000001	0.000003	0.000002
27	Earth Sensor 2	solid cyl	0.0135	0.0100	0.0200	0.000004	0.000005	0.000002
28	Earth Sensor 3	solid cyl	0.0135	0.0100	0.0200	0.000004	0.000005	0.000002
29	Earth Sensor 4	solid cyl	0.0135	0.0100	0.0200	0.000004	0.000005	0.000002
30	Earth Sensor 5	solid cyl	0.0135	0.0100	0.0200	0.000004	0.000005	0.000002
31	Earth Sensor 6	solid cyl	0.0135	0.0100	0.0200	0.000017	0.000019	0.000009
32	Transciever&HPA Module	box	0.0790	0.0250	0.0450	0.000088	0.000276	0.000229
33	Receiver Module	box	0.0790	0.0250	0.0450	0.000066	0.000207	0.000172
34	Interface/Power Module	box	0.0790	0.0250	0.0450	0.000066	0.000207	0.000172
35	µVAT	solid tri	0.0500	0.0100	0.0450	0.000354	0.000754	0.000433
36	Radiator	solid rec	0.2000	0.0900	0.0100	0.000041	0.000198	0.000238
37	TT&C Miscellaneous	UDI						
38	Thermal Miscellaneous	UDI						
39	Miscellaneous	UDI						

Step 3 - Component Center of Mass Xr, Yr, Zr values from Reference (0,0,0)

Item	Name		Xr	Yr	Zr	
1	Battery 1	box	0.0285	-0.1000	-0.0342	m
2	Battery 2	box	0.0285	0.1000	-0.0342	m
3	Battery 3	box	0.0775	-0.1000	-0.0342	m
4	Battery 4	box	0.0775	0.1000	-0.0342	m
5	Side 1 (front w/Ap)	solid sheet	-0.2500	0.0000	-0.0500	m
6	Side 2 (back w/Solar Array)	solid sheet	0.2500	0.0000	-0.0500	m
7	Side 3	solid sheet	0.0000	-0.0500	-0.1000	m
8	Side 4	solid sheet	0.0000	0.0000	-0.0500	m
9	Side 5	solid sheet	0.0000	0.0500	-0.0500	m
10	Side 6	solid sheet	0.0000	0.0000	0.0000	m
11	Solar Array Wing 1	solid sheet	0.2500	-0.1500	-0.0500	m
12	Solar Array Wing 2	solid sheet	0.2500	0.0000	-0.2000	m
13	Solar Array Wing 3	solid sheet	0.2500	0.1500	-0.0500	m
14	Solar Array Wing 4	solid sheet	0.2500	0.0000	0.1000	m
15	C&DH Proc	box	0.1800	0.0000	-0.0350	m
16	RFTP/Power Control Unit	box	0.1700	0.0000	-0.0350	m
17	Reaction Wheels & IRU	box	0.0250	0.0000	-0.0400	m
18	Temp Controller	box	0.0715	0.0000	-0.0822	m
19	C&DH/TT&C Antenna	box	0.0950	0.0000	-0.0950	m
20	Antenna Reflector	solid cyl	0.0950	0.0000	-0.1600	m
21	Payload	solid cyl	-0.0751	0.0000	-0.0500	m
22	Sun Sensor 2	solid cyl	0.2450	-0.0450	-0.0200	m
23	Sun Sensor 3	solid cyl	-0.2200	-0.0450	-0.0200	m
24	Sun Sensor 4	solid cyl	-0.2200	-0.0450	-0.0900	m
25	Sun Sensor 5	solid cyl	-0.2200	0.0450	-0.0800	m
26	Sun Sensor 6	solid cyl	-0.2200	0.0450	0.0000	m
27	Earth Sensor 2	solid cyl	0.2450	0.0450	-0.0200	m
28	Earth Sensor 3	solid cyl	-0.2200	-0.0450	-0.0800	m
29	Earth Sensor 4	solid cyl	-0.2200	0.0450	-0.0900	m
30	Earth Sensor 5	solid cyl	-0.2200	0.0450	-0.2000	m
31	Earth Sensor 6	solid cyl	-0.2200	-0.0450	0.0000	m
32	Transciever&HPA Module	box	0.1805	0.0000	-0.0675	m
33	Receiver Module	box	0.1805	0.0125	-0.0225	m
34	Interface/Power Module	box	0.1805	-0.0125	-0.0225	m
35	µVAT	solid tri	0.0000	0.0000	-0.0050	m
36	Radiator	solid rec	-0.1250	0.0000	-0.0050	m
37	TT&C Miscellaneous	UDI				
38	Thermal Miscellaneous	UDI				
39	Miscellaneous	UDI				

Step 4 - Spacecraft Center of Mass

Item	Name		M*Xr	M*Yr	M*Zr	
1	Battery 1	box	0.0042	-0.0146	-0.0050	kg*m
2	Battery 2	box	0.0042	0.0146	-0.0050	kg*m
3	Battery 3	box	0.0113	-0.0146	-0.0050	kg*m
4	Battery 4	box	0.0113	0.0146	-0.0050	kg*m
5	Side 1 (front w/Ap)	solid sheet	-0.0018	0.0000	-0.0004	kg*m
6	Side 2 (back w/Solar Array)	solid sheet	0.0122	0.0000	-0.0024	kg*m
7	Side 3	solid sheet	0.0000	-0.0074	-0.0147	kg*m
8	Side 4	solid sheet	0.0000	0.0000	-0.0044	kg*m
9	Side 5	solid sheet	0.0000	0.0074	-0.0074	kg*m
10	Side 6	solid sheet	0.0000	0.0000	0.0000	kg*m
11	Solar Array Wing 1	solid sheet	0.0311	-0.0187	-0.0062	kg*m
12	Solar Array Wing 2	solid sheet	0.0107	0.0000	-0.0086	kg*m
13	Solar Array Wing 3	solid sheet	0.0311	0.0187	-0.0062	kg*m
14	Solar Array Wing 4	solid sheet	0.0107	0.0000	0.0043	kg*m
15	C&DH Proc	box	0.2700	0.0000	-0.0525	kg*m
16	RFTP/Power Control Unit	box	0.1751	0.0000	-0.0361	kg*m
17	Reaction Wheels & IRU	box	0.0233	0.0000	-0.0372	kg*m
18	Temp Controller	box	0.0020	0.0000	-0.0023	kg*m
19	C&DH/TT&C Antenna	box	0.0475	0.0000	-0.0475	kg*m
20	Antenna Reflector	solid cyl	0.0143	0.0000	-0.0240	kg*m
21	Payload	solid cyl	-0.0642	0.0000	-0.0427	kg*m
22	Sun Sensor 2	solid cyl	0.0980	-0.0180	-0.0080	kg*m
23	Sun Sensor 3	solid cyl	-0.0880	-0.0180	-0.0080	kg*m
24	Sun Sensor 4	solid cyl	-0.0880	-0.0180	-0.0360	kg*m
25	Sun Sensor 5	solid cyl	-0.0880	0.0180	-0.0320	kg*m
26	Sun Sensor 6	solid cyl	-0.0880	0.0180	0.0000	kg*m
27	Earth Sensor 2	solid cyl	0.0245	0.0045	-0.0020	kg*m
28	Earth Sensor 3	solid cyl	-0.0220	-0.0045	-0.0080	kg*m
29	Earth Sensor 4	solid cyl	-0.0220	0.0045	-0.0090	kg*m
30	Earth Sensor 5	solid cyl	-0.0220	0.0045	-0.0200	kg*m
31	Earth Sensor 6	solid cyl	-0.0220	-0.0045	0.0000	kg*m
32	Transciever&HPA Module	box	0.0722	0.0000	-0.0270	kg*m
33	Receiver Module	box	0.0542	0.0038	-0.0068	kg*m
34	Interface/Power Module	box	0.0542	-0.0038	-0.0068	kg*m
35	µVAT	solid tri	0.0000	0.0000	-0.0100	kg*m
36	Radiator	solid rec	-0.0074	0.0000	-0.0003	kg*m
37	TT&C Miscellaneous	UDI				
38	Thermal Miscellaneous	UDI				
39	Miscellaneous	UDI				

			Xcm	Ycm	Zcm	
	Composite		0.0319	-0.0010	-0.0343	m (from 0,0,0)
			Mass =	14.05	kg	

Step 5 - Component Center of Mass distance from Spacecraft Center of Mass

Item	Name		Xcm-Xr	Ycm-Yr	Zcm-Zr
1	Battery 1	box	0.0034	0.0990	0.0001
2	Battery 2	box	0.0034	-0.1010	0.0001
3	Battery 3	box	-0.0456	0.0990	0.0001
4	Battery 4	box	-0.0456	-0.1010	0.0001
5	Side 1 (front w/Ap)	solid sheet	0.2819	-0.0010	-0.0157
6	Side 2 (back w/Solar Array)	solid sheet	-0.2181	-0.0010	-0.0157
7	Side 3	solid sheet	0.0319	0.0490	-0.0657
8	Side 4	solid sheet	0.0319	-0.0010	-0.0157
9	Side 5	solid sheet	0.0319	-0.0510	-0.0157
10	Side 6	solid sheet	0.0319	-0.0010	0.0343
11	Solar Array Wing 1	solid sheet	-0.2181	0.1490	-0.0157
12	Solar Array Wing 2	solid sheet	-0.2181	-0.0010	-0.1657
13	Solar Array Wing 3	solid sheet	-0.2181	-0.1510	-0.0157
14	Solar Array Wing 4	solid sheet	-0.2181	-0.0010	0.1343
15	C&DH Proc	box	-0.1481	-0.0010	-0.0007
16	RFTP/Power Control Unit	box	-0.1381	-0.0010	-0.0007
17	Reaction Wheels & IRU	box	0.0069	-0.0010	-0.0057
18	Temp Controller	box	-0.0396	-0.0010	-0.0479
19	C&DH/TT&C Antenna	box	-0.0631	-0.0010	-0.0607
20	Antenna Reflector	solid cyl	-0.0631	-0.0010	-0.1257
21	Payload	solid cyl	0.1070	-0.0010	-0.0157
22	Sun Sensor 2	solid cyl	-0.2131	0.0440	0.0143
23	Sun Sensor 3	solid cyl	0.2519	0.0440	0.0143
24	Sun Sensor 4	solid cyl	0.2519	0.0440	-0.0557
25	Sun Sensor 5	solid cyl	0.2519	-0.0460	-0.0457
26	Sun Sensor 6	solid cyl	0.2519	-0.0460	0.0343
27	Earth Sensor 2	solid cyl	-0.2131	-0.0460	0.0143
28	Earth Sensor 3	solid cyl	0.2519	0.0440	-0.0457
29	Earth Sensor 4	solid cyl	0.2519	-0.0460	-0.0557
30	Earth Sensor 5	solid cyl	0.2519	-0.0460	-0.1657
31	Earth Sensor 6	solid cyl	0.2519	0.0440	0.0343
32	Transciever&HPA Module	box	-0.1486	-0.0010	-0.0332
33	Receiver Module	box	-0.1486	-0.0135	0.0118
34	Interface/Power Module	box	-0.1486	0.0115	0.0118
35	µVAT	solid tri	0.0319	-0.0010	0.0293
36	Radiator	solid rec	0.1569	-0.0010	0.0293
37	TT&C Miscellaneous	UDI			
38	Thermal Miscellaneous	UDI			
39	Miscellaneous	UDI			

Step 6 - Moments Of Intertia wrt Spacecraft Center of Mass

Item	Name		Ixx	Iyy	Izz	Ixy	Ixz	Iyz
1	Battery 1	box	0.0015	0.0001	0.0015	0.0000	0.0001	0.0000
2	Battery 2	box	0.0015	0.0001	0.0015	0.0001	0.0001	0.0000
3	Battery 3	box	0.0015	0.0004	0.0018	0.0007	0.0001	0.0000
4	Battery 4	box	0.0015	0.0004	0.0018	-0.0006	0.0001	0.0000
5	Side 1 (front w/Ap)	solid sheet	0.0000	0.0006	0.0006	0.0000	0.0000	0.0000
6	Side 2 (back w/Solar Array)	solid sheet	0.0001	0.0024	0.0024	0.0000	-0.0001	0.0000
7	Side 3	solid sheet	0.0011	0.0040	0.0036	-0.0001	0.0035	0.0035
8	Side 4	solid sheet	0.0001	0.0020	0.0019	0.0001	0.0019	0.0018
9	Side 5	solid sheet	0.0005	0.0034	0.0036	0.0004	0.0033	0.0029
10	Side 6	solid sheet	0.0002	0.0029	0.0027	0.0001	0.0025	0.0026
11	Solar Array Wing 1	solid sheet	0.0041	0.0069	0.0091	0.0054	0.0005	0.0007
12	Solar Array Wing 2	solid sheet	0.0014	0.0034	0.0021	0.0002	-0.0014	0.0000
13	Solar Array Wing 3	solid sheet	0.0042	0.0069	0.0092	-0.0027	0.0005	0.0001
14	Solar Array Wing 4	solid sheet	0.0010	0.0030	0.0021	0.0002	0.0014	0.0000
15	C&DH Proc	box	0.0006	0.0346	0.0340	0.0004	0.0016	0.0011
16	RFTP/Power Control Unit	box	0.0004	0.0201	0.0197	0.0003	0.0003	0.0000
17	Reaction Wheels & IRU	box	0.0704	0.0005	0.0700	0.0703	0.0004	0.0700
18	Temp Controller	box	0.0001	0.0001	0.0000	0.0000	-0.0001	0.0000
19	C&DH/TT&C Antenna	box	0.0020	0.0040	0.0022	0.0001	-0.0018	0.0002
20	Antenna Reflector	solid cyl	0.0024	0.0030	0.0006	0.0000	-0.0012	0.0000
21	Payload	solid cyl	0.0005	0.0111	0.0105	0.0004	0.0025	0.0007
22	Sun Sensor 2	solid cyl	0.0009	0.0183	0.0189	0.0038	0.0012	-0.0002
23	Sun Sensor 3	solid cyl	0.0009	0.0255	0.0262	-0.0044	-0.0014	-0.0002
24	Sun Sensor 4	solid cyl	0.0020	0.0266	0.0262	-0.0044	0.0056	0.0010
25	Sun Sensor 5	solid cyl	0.0017	0.0262	0.0262	0.0046	0.0046	-0.0008
26	Sun Sensor 6	solid cyl	0.0013	0.0259	0.0262	0.0046	-0.0035	0.0006
27	Earth Sensor 2	solid cyl	0.0002	0.0046	0.0048	-0.0010	0.0003	0.0001
28	Earth Sensor 3	solid cyl	0.0004	0.0066	0.0065	-0.0011	0.0012	0.0002
29	Earth Sensor 4	solid cyl	0.0005	0.0067	0.0066	0.0012	0.0014	-0.0003
30	Earth Sensor 5	solid cyl	0.0030	0.0091	0.0066	0.0012	0.0042	-0.0008
31	Earth Sensor 6	solid cyl	0.0003	0.0065	0.0065	-0.0011	-0.0008	-0.0001
32	Transciever&HPA Module	box	0.0005	0.0095	0.0091	0.0000	-0.0017	0.0002
33	Receiver Module	box	0.0002	0.0069	0.0068	-0.0005	0.0007	0.0002
34	Interface/Power Module	box	0.0001	0.0069	0.0068	0.0006	0.0007	0.0001
35	μVAT	solid tri	0.0021	0.0045	0.0025	0.0004	-0.0011	0.0005
36	Radiator	solid rec	0.0001	0.0017	0.0017	0.0000	-0.0001	0.0002
37	TT&C Miscellaneous	UDI						
38	Thermal Miscellaneous	UDI						
39	Miscellaneous	UDI						

Step 7 - Sum Spacecraft Moments of Inertia

Ixx	Iyy	Izz	Ixy	Ixz	Iyz
0.1093	0.2949	0.3624	0.0791	0.0258	0.0847

Ref: SMAD p.466(material properties, 476-477, 924 (coversation factors)

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97E+24	kg
G =	6.67E-20	$km^3/kg*s^2$
μ Earth =	398600.4	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Pitch Error:

1. Sum S/C MOIs,
2. Find center of pressure,
3. Calculate pitch error (GG & Aero torque equilibrium)

Orbit Parameters

orbital rate (ω0) =	0.000073	rad/sec
velocity (v) =	3075.413783	m/s
atmospheric density (ρ) =	0.00	kg/m^3

Component Characteristics

Component	Shape	d (x) or R [m]	w (y) or r [m]	h (z) [m]	X-Section Area [m^2]
Battery 1	box	0.0450	0.0192	0.0684	0.0009
Battery 2	box	0.0450	0.0192	0.0684	0.0009
Battery 3	box	0.0450	0.0192	0.0684	0.0009
Battery 4	box	0.0450	0.0192	0.0684	0.0009
Side 1 (front w/Ap)	solid sheet	0.1000	0.0050	0.1000	0.0005
Side 2 (back w/Solar Array	solid sheet	0.1000	0.0050	0.1000	0.0005
Side 3	solid sheet	0.5000	0.0050	0.1000	0.0025
Side 4	solid sheet	0.5000	0.0050	0.1000	0.0025
Side 5	solid sheet	0.5000	0.0050	0.1000	0.0025
Side 6	solid sheet	0.5000	0.0050	0.1000	0.0025
Solar Array Wing 1	solid sheet	0.0025	0.2000	0.3000	0.0005
Solar Array Wing 2	solid sheet	0.0025	0.1000	0.2000	0.0003
Solar Array Wing 3	solid sheet	0.0025	0.2000	0.3000	0.0005
Solar Array Wing 4	solid sheet	0.0025	0.1000	0.2000	0.0003
C&DH Proc	box	0.0950	0.0050	0.0700	0.0005
RFTP/Power Control Unit	box	0.0150	0.0050	0.0700	0.0001
Reaction Wheels & IRU	box	0.0150	0.9500	0.0700	0.0143
Temp Controller	box	0.0038	0.0254	0.0277	0.0001
C&DH/TT&C Antenna	box	0.0530	0.0530	0.0112	0.0028
Antenna Reflector	solid cyl	0.0230	0.0000	0.0020	0.0017
Payload	solid cyl	0.1501	0.0000	0.0980	0.0462
Sun Sensor 2	solid cyl	0.0150	0.0000	0.0100	0.0005
Sun Sensor 3	solid cyl	0.0150	0.0000	0.0100	0.0005
Sun Sensor 4	solid cyl	0.0150	0.0000	0.0100	0.0005
Sun Sensor 5	solid cyl	0.0150	0.0000	0.0100	0.0005
Sun Sensor 6	solid cyl	0.0150	0.0000	0.0100	0.0005
Earth Sensor 2	solid cyl	0.0135	0.0100	0.0200	0.0008
Earth Sensor 3	solid cyl	0.0135	0.0100	0.0200	0.0008
Earth Sensor 4	solid cyl	0.0135	0.0100	0.0200	0.0008
Earth Sensor 5	solid cyl	0.0135	0.0100	0.0200	0.0008
Earth Sensor 6	solid cyl	0.0135	0.0100	0.0200	0.0008
Transciever&HPA Module	box	0.0790	0.0250	0.0450	0.0020
Receiver Module	box	0.0790	0.0250	0.0450	0.0020
Interface/Power Module	box	0.0790	0.0250	0.0450	0.0020
μVAT	solid tri	0.0500	0.0100	0.0450	0.0005
Radiator	solid rec	0.2000	0.0900	0.0100	0.0180

Sum Spacecraft Moments of Inertia

Ixx	Iyy	Izz	Ixy	Ixz	Iyz
0.109	0.295	0.362	0.079	0.026	0.085

Spacecraft Center of Mass

	Xcm	Ycm	Zcm	
Composite	0.032	-0.001	-0.034	m (from 0,0,0)

Center of Mass (Zcm) = -0.034 m

Center of Pressure

Component	Shape	Area[m^2]	cp (Zr)[m]	Area*cp [m^3]
Battery 1	box	0.0009	-0.0342	-0.000030
Battery 2	box	0.0009	-0.0342	-0.000030
Battery 3	box	0.0009	-0.0342	-0.000030
Battery 4	box	0.0009	-0.0342	-0.000030
Side 1 (front w/Ap)	solid sheet	0.0005	-0.0500	-0.000025
Side 2 (back w/Solar Array	solid sheet	0.0005	-0.0500	-0.000025
Side 3	solid sheet	0.0025	-0.1000	-0.000250
Side 4	solid sheet	0.0025	-0.0500	-0.000125
Side 5	solid sheet	0.0025	-0.0500	-0.000125
Side 6	solid sheet	0.0025	0.0000	0.000000
Solar Array Wing 1	solid sheet	0.0005	-0.0500	-0.000025
Solar Array Wing 2	solid sheet	0.0003	-0.2000	-0.000050
Solar Array Wing 3	solid sheet	0.0005	-0.0500	-0.000025
Solar Array Wing 4	solid sheet	0.0003	0.1000	0.000025
C&DH Proc	box	0.0005	-0.0350	-0.000017
RFTP/Power Control Unit	box	0.0001	-0.0350	-0.000003
Reaction Wheels & IRU	box	0.0143	-0.0400	-0.000570
Temp Controller	box	0.0001	-0.0822	-0.000008
C&DH/TT&C Antenna	box	0.0028	-0.0950	-0.000267
Antenna Reflector	solid cyl	0.0017	-0.1600	-0.000266
Payload	solid cyl	0.0462	-0.0500	-0.002311
Sun Sensor 2	solid cyl	0.0005	-0.0200	-0.000009
Sun Sensor 3	solid cyl	0.0005	-0.0200	-0.000009
Sun Sensor 4	solid cyl	0.0005	-0.0900	-0.000042
Sun Sensor 5	solid cyl	0.0005	-0.0800	-0.000038
Sun Sensor 6	solid cyl	0.0005	0.0000	0.000000
Earth Sensor 2	solid cyl	0.0008	-0.0200	-0.000017
Earth Sensor 3	solid cyl	0.0008	-0.0800	-0.000068
Earth Sensor 4	solid cyl	0.0008	-0.0900	-0.000076
Earth Sensor 5	solid cyl	0.0008	-0.2000	-0.000170
Earth Sensor 6	solid cyl	0.0008	0.0000	0.000000
Transciever&HPA Module	box	0.0020	-0.0675	-0.000133
Receiver Module	box	0.0020	-0.0225	-0.000044
Interface/Power Module	box	0.0020	-0.0225	-0.000044
μVAT	solid tri	0.0005	-0.0050	-0.000003
Radiator	solid rec	0.0180	-0.0050	-0.000090

Total Area 1 = 0.113 m²

Center of Pressure 1(Zcp1) = -0.044 m

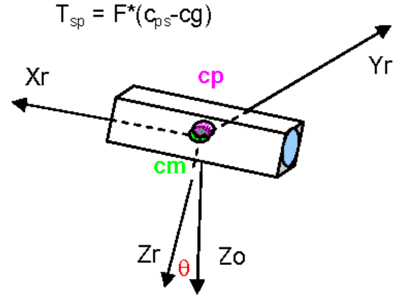
1 = Area with solar panels normal to motion

θ Error 1

Drag Coeff (Cd) =	2.5		
Atmospheric Drag (Fa) =	0.00	N	$Fa = \rho*Cd*A*V^2$
Aero Torque (Ta) =	0.00	N*m	Ta = Fa(Cpa-cg)
θ =	0.00	rad	$\theta = Ta/[3*\omega o^2(Ixx-Izz)$
θ =	0.00	deg	
Max allowable deivation of Z-axis (θz) =	8.73E-03	rad	Operational Restriction
Grav Gradient Torque (Tg) =	9.41E-12	N*m	$Tg = ((3*\mu)/(2*R^3))*\lvert Izz-Iyy\rvert*\sin(2*\theta z)$
θ =	2.33E-03	rad	
θ =	1.33E-01	deg	

	X_{cm}	Y_{cm}	Z_{cm}	
cm =	0.032	-0.001	-0.034	m
Vector from Area's center to SCcm (s)=	-0.032	-0.251	0.216	m

Angle of incidence of the Sun (i) =	30.000	deg	
Solar Vector to SC (F) =	1.283E-06	N	$F = (F_s/c)*A*(1+q)*\cos(i)$
Solar Radiation Torque (T_{sp}) =	2.766E-07	N*m	$T_{sp} = F*(c_{ps}-cg)$
θ =	4.992E-03	rad	
θ =	2.86E-01	deg	
Maximux θ Error =	2.86E-01	deg	



Ref: SMAD p322-324, 366

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Optical Payload:

Orbit Parameters

SC Altitude (Hsc) =	35765.37	km	given
SC Orbit period (Psc) =	1435.30	min	(μ/(Re+Hsc)^3)^(1/2)
Target's altitude =	35785.86	km	Assuming GEOSTA
Target's Obital Period (Ptar) =	1436.35	min	sqrt(μ/(Re+Htar)^3)
SC ω (ωsc) =	7.30E-05	Rad/sec	ωsc = (2*PI)/Psc
Target ω (ωtar) =	7.29E-05	Rad/sec	ωtar = (2*PI)/Ptar
SC orbital Radius (Rsc) =	42149.11	km	(μ Earth/ω^2)^(1/3)
Target orbital Radius (Rtar) =	42169.61	km	(μ Earth/ω^2)^(1/3)
SC velocity (Vsc) =	3.08	km/sec	Vsc = ωsc*Rsc
Target's velocity (Vtar) =	3.07	km/sec	Vtar = ωtar*Rtar
Closing Velocity (CV[km/sec]) =	7.48E-04	km/sec	Vtar-Vsc
Closing Velocity (CV) =	0.75	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	20.50	km	Htar-Hsc
Target's Orbital Circumference (Cir_{tar}) =	264869.00	km	2*π*(H_{tar}*6378.137)
Distance SC travels relative to GEOSTA orbit =	64.59	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	4101.48	days	Cir_{tar}/Distance SC travels
Time for SC to traverse GEOSTA orbit =	11.23	years	(Cir_{tar}/Distance SC travels)*365.25

Kodak KAF-38300 Image Sensor (Monochrome) Properties

Pixel Height =	5.4	μm	CCD Data Sheet
Pixel Width =	5.4	μm	CCD Data Sheet
# of Horizontal pixels (PxlH) =	3358	pxl	Effective # of pxl from CCD Data Sheet
# of Vertical pixels (PxlV) =	2536	pxl	Effective # of pxl from CCD Data Sheet
Maximum Data Rate (DR) =	2.40E+07	Hz	CCD Data Sheet
Bits/pxl (Nb) =	1.10E+01		
Readout Time (Tlrout) =	3.70E-01	sec/image	CCD Data Sheet
Pixel Period (1 count) =	4.20E-08	sec	CCD Data Sheet
#sec/pxl =	1.46E-04	sec/pxl	Tlrout/PxlV
#pxls/sec (Z) =	6.85E+03	pxl/sec	1/#sec/pxl
Max Operating Temp Range =	-10 to 70	°C	CCD Data Sheet
Guaranteed Operating Temp Range =	0 to 60	°C	CCD Data Sheet

Target Parameters

In track Elevation angle (e) =		deg	Depends on target's relative postion
Slew angle (h) =		deg	Depends on target's relative postion
Slant range (Rs) =		km	Depends on target's relative postion
# Active Pixels (Zact) =	8515888	pixels	CCD characteristic
Pixel Integration Time (Tipxl) =	5.00E-04	sec	CCD characteristic
Relative Motion During pxl image capture(Blur) =	3.74E-04	m	CV*Tipxl
#pxls/sec (Z) =	1.70E+10	pxls/sec	Zact/Tipxl
Bits/pxl (Nb) =	11	bits/pixel	Choosen
DataRate (DR) =	93.67	Mbps	Zact*Nb
Megabytes per Picture (Psz) =	11.71	MBytes	CCD characteristic
Compressed Image Size (cPsz) =	0.98	MBytes	Psz/12 ->Need to verify for jpeg format

Optic System

Aperature (Ap) =	0.095	m	S/C property
CPA Seperation (CPA Sep) =	20491.8	m	Htar-Hsc; will differ when e≠ 90 since SC is orbiting lower than Target
Lambda (λ) =	4.0000E-07	m	Choosen
Resolution (X) =	2.1053E-01	m	(2.44*λ*Sep)/Ap
Resolution of Target at 50[km] (X_{50km}) =	5.1368E-01	m	(2.44*λ*50000[m])/Ap
Resolution of Target at 100[km] (X_{100km}) =	1.03E+00	m	(2.44*λ*100000[m])/Ap
0.5[m] Resolution of Target =	0.53	m	(2.44*λ*100000[m])/Ap
Distance for 0.5[m] Resolution of Target ($D_{0.5m}$) =	51229.5	m	Determined by goal seeking 0.5[m] Resolution
Detector width *(square pixel width)* (d) =	5.40E-06	m	CCD Data Sheet
Quality factor (Q) =	1.10E+00		0.5<Q<2 selected
Operating wavelength (λw) =	4.00E-07	m	selected
Focal length (f) =	0.526	m	Sep*d/X
Folded folcal length (f/5) =	0.105	m	f/5
Diffraction-limited aperature diameter (D) =	0.105	m	2.44*λw*f*Q/d
F-number (F#) =	5.030	#	f/D

Scaling Estimate SMAD method (sec. 9.5.3 using IKONOS)

Aperture ratio (R) =	0.1357		Ai/Ao (aperature of SC/aperture of IKONOS)
Linear dimensions (Li) =	0.2068	m	R*Lo (Lo=linear dimension of IKONOS)
Surface area (Si) =	0.0428	m^2	Li^2
Volume (Vi) =	0.0088	m^3	Li^3
Weight (Wi) =	0.8549	kg	K*R^3*Wo (K=2 if R<0.5, else 1)
Power (Pi) =	1.7497	W	K*R^3*Po

IKONOS-2 using Kodak Model 1000 Camera System Data:
Assembly size: 1.524[m] by 0.787[m] (1[m^3] volume)
10[m] focal length; f/14.3; 0.7[m] Primary mirror aperature diameter.
171[kg] total mass; 350[W] total power

MOI Estimate Budianto method (Eq. 4.16)

Moment of Inertia (I) =	2.03E-02	kg*m^2	(1/12)*Wi*(3(D/2)^2+f^2) assuming cylindrical assembly

Maximum allowable error in optical component construction

Max allowable angular deviation($\Delta\Theta$) =	1.03E-05	deg	X/h

Pointing Requirements

Y_{min} (Basic Calc) =	7.07	deg	h*d/f
Y_{min} (Diffraction limit) =	0.19	deg	2.44*λ*h/D
Y_{min} (Image Blur) =	1.85E-03	deg	10*(Scan Velocity)*Δt
Pointing Requirement =	7.07	deg	Larget Y_{min} calculated for Spatial resolution desired.

Thermal Requirements

Operating Temperature =	273-333	K	Based on CCD restriction
Operating Temperature =	0-60	C	Based on CCD restriction

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Comments

Distortion is not a worry due to limited size of target object.
Entire optical formulae for pointing out into space vice down toward earth -> use stop and stare method at each choosen distance in closing path.
Velocity formulae need to be corrected
Closing velocity will dictate shutter speed needed

Ref: SMAD p.247-91; 364-369(ACS constraints)

Constellation Estimating:

Orbit Parameters

SC Altitude (Hsc) =	35765.37	km	given
SC Orbit Period (Psc) =	1435.30	min	(m/(Re+Hsc)3)1/2
Target's Altitude (Htar) =	35785.86	km	Assuming GEOSTA
Target's Obital Period (Ptar) =	1436.35	min	sqrt(m/(Re+Htar)3
SC ω (ωsc) =	0.0000730	Rad/sec	ωsc = (2*π)/Psc
Target ω (ωtar) =	0.0000729	Rad/sec	ωtar = (2*π)/Ptar
SC orbital Radius (Rsc) =	42149.11	km	(μEarth/ωsc2)1/3
Target orbital Radius (Rtar) =	42169.61	km	(μEarth/ωtar2)1/3
SC Velocity (Vsc) =	3.08	km/sec	Vsc = ωsc*Rsc
Target's Velocity (Vtar) =	3.07	km/sec	Vtar = ωtar*Rtar

Coverage and Access Factors

Closing Velocity (CV[km/sec]) =	0.00075	km/sec	Vtar-Vsc
Closing Velocity (CV) =	0.75	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	20.50	km	Htar-Hsc
Target's Orbital Circumference (Cirtar) =	264869.00	km	2*π*(Htar*6378.137)
Distance SC travels relative to GEOSTA orbit =	64.59	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	4101.48	days	Cirtar/Distance SC travels
Time for SC to traverse GEOSTA orbit =	11.23	years	(Cirtar/Distance SC travels)*365.25

Coverage and Access Considerations

Desired Constellation re-visit rate =	30	days	Choosen
Planes of SC desired =	1	Planes	Majority of GEOSTA at i≈0°
Distance SC can travel during re-visit rate =	7970942.73	km	(Re-visit rate)* Vsc*60*60*24
Distance Targets travel during re-visit rate =	7969004.96	km	(Re-visit rate)* Vtar*60*60*24
Difference in distances traveled =	1937.77	km	(SC travel)-(Tar travel)
Separation distance for SC =	3875.54	km	2*SepDist
Number of SC needed =	68.00		(Circumference of target's orbit)/(Sep distance) rounded up

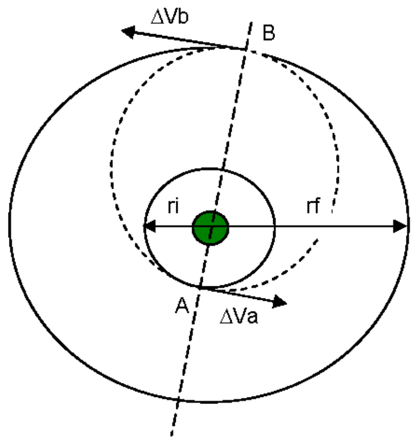
Modeled in STK

Seeed number of SC =	48.00	SC	From estimation
STK optimized SC number through trial and error =	33	SC	Evenly spaced SC simulated in STK to give complete GEOSTA coverage in desired re-visit rate.
SC separation =	18919.21	km	(2*π*(Hsc+6378.137))/14

Ref: SMAD p.190-196

Orbit Transfer Calculations:

		5U CubeSAT	Units	Comments
μ earth	398600.4418		km^3/s^2	constant
Re	6378.137		km	constant
hpark		42164	km	given
	ri	42164	km	given
	rf	42144	km	given
Step				
1	atx	42154	km	(ri+rf)/2
2	Via	3.07	km/s	(μ earth/ri)^(1/2)
3	Vfb	3.08	km/s	(μ earth/rf)^(1/2)
4	Vtxa	3.07	km/s	(μ earth*(2/ri)-(1/atx))^(1/2)
5	Vtxb	3.08	km/s	(μ earth*(2/rf)-(1/atx))^(1/2)
6	ΔVa	0.00	km/s	|Vtxa-Via|
7	ΔVb	0.00	km/s	|Vfb-Vtxb|
8	ΔVtotal	0.00	km/s	ΔVa+ΔVb
9	x-fer time	43066	s	P/2 = π*SQRT(a^3/μ)
		11.96	hr	



Plane change in parking orbit followed by Hohmann transfer

	θ	15.00	deg
	ΔVpc	0.8028	km/s
	ΔVa+ΔVb+ΔVpc	0.8021	km/s

Plane change at B combined with Hohmann transfer

	ΔVcmd	0.8029	km/s
	ΔVa+ΔVcmd	0.8025	km/s

RAAN change utilitizing two Hohmann transfers

	ΔV	1.6250	km/s

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p.147-152 & Table 6-5; FS p.340
*Ortibal Change to meet a target SC: SMAD p.152
Wait Time (WT) = (ϕi-ϕf+2kπ)/(ωint-ωtar)

ΔV Budget:

Basic Data	**5U CubeSAT**	**Units**	**Comments**
Initial Radius	42164	km	Choosen
Initial inclination	0	deg	Orbit insertion inclination
Mission Radius	42144	km	Choosen
Mission inclination	0	deg	Choosen to match target's inclination
Mission Duration	2	yr	Choosen
Orbit Maintenance Req	Various		Mission Dependant
Drag Parameters	0		Orbital Regime Property
m/CdA	N/A		Orbital Regime Property
Max Atmospheric Density (ρmax)	N/A	kg/m^3	Orbital Regime Property
Orbit Manuever Req	Unknown		Mission Dependant
Final Conditions	Unknown		Mission Dependant
ΔV Budget [m/s]			
Orbit Transfer			
1st Burn	0	m/s	ΔVa*1000
2nd Burn	803	m/s	ΔVcmd*1000
Altitude Maintenance (LEO)	N/A	m/s	Property of orbit
North/South Stationkeeping	103	m/s	Formula constants need to be analysized
East/West Stationkeeping	3	m/s	Formula constants need to be analysized
Orbit Manuevers	-	m/s	Maybe required as per mission
Rephasing, Rendezvous	Unknown		Maybe required as per mission
Node or Plance Change	Unknown		Maybe required as per mission
Spacecraft disposal	10	m/s	Usual Reqt's
Total ΔV	**919**	**m/s**	

Key:

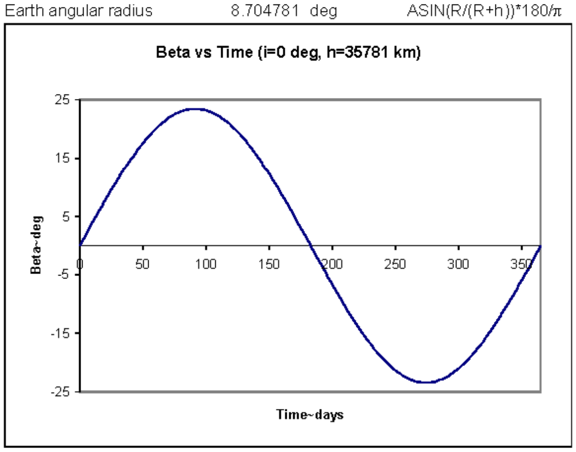
Coefficient of Drag (Cd)
SC Cross-sectional Area (A)
SC Mass (m)
SC velocity (Vsc)

Constants

Radius Earth =	6378.137	km
mass Earth =	5.9733E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.442	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Ref: SMAD p.147-151

Basic Data		**Units**	**Remarks**
uo=RA of sun in ecliptic	0		
wo=RA of AN of Orbit	0		
h (altitude)	35765.37	km	Orbit property
i (inclination)	0	deg	Orbit property
E (eccentricity)	0		Orbit property
wdot = nodal reg'n rate*	-0.01344	rate	-(9.96390003*(R/(R+h))^3.5*COS(i*π/180))/(1-E²)²
e (Earth axis tilt)	23.44241	deg	Earth property
R (Earth radius)	6378.137	km	Earth property
R/(R+h)	0.151343	*no unit*	R/(R+h)
MSD (mean solar day)	0.985647	sidereal day	Earth property
Earth g const	398601.2	kg³/s²	Orbit property
Orbital rate	7.3E-05	Rad/s	(Earth g const/(R+h)³)^(1/2)
Orbital period	1435.011	min	2*π/Orbital rate/60
Earth angular radius	8.704781	deg	ASIN(R/(R+h))*180/π



[-9.96390003(R/(R+h))^3.5*cos(i)]/(1-E^2)^2 [deg/MSD]

Ref: SMAD p.107-110

day	u [deg]	w [deg]	β [deg]	Eclp Ang[deg]	Te/To
0	0	0	0	17.40956248	0.04836
1	0.985647	-0.01343695	0.392100859	17.39202756	0.048311
2	1.971295	-0.0268739	0.784104039	17.33933065	0.048165
3	2.956942	-0.04031084	1.175911848	17.25119244	0.04792
4	3.942589	-0.05374779	1.567426571	17.12713788	0.047575
5	4.928237	-0.06718474	1.958550455	16.96648107	0.047129
6	5.913884	-0.08062169	2.349185699	16.76830269	0.046579
7	6.899531	-0.09405864	2.739234444	16.53141798	0.045921
8	7.885179	-0.10749558	3.128598756	16.25433283	0.045151
9	8.870826	-0.12093253	3.517180623	15.93518371	0.044264
10	9.856473	-0.13436948	3.904881939	15.57165547	0.043255
11	10.84212	-0.14780643	4.291604493	15.16086759	0.042114
12	11.82777	-0.16124338	4.677249967	14.69921407	0.040831
13	12.81342	-0.17468033	5.061719918	14.18213287	0.039395
14	13.79906	-0.18811727	5.44491578	13.60376398	0.037788
15	14.78471	-0.20155422	5.826738846	12.95642379	0.03599
16	15.77036	-0.21499117	6.207090272	12.22975919	0.033972
17	16.756	-0.22842812	6.585871065	11.40930559	0.031693
18	17.74165	-0.24186507	6.962982081	10.47383753	0.029094
19	18.7273	-0.25530201	7.338324022	9.389980452	0.026083
20	19.71295	-0.26873896	7.711797433	8.099527888	0.022499
21	20.69859	-0.28217591	8.083302702	6.481736011	0.018005
22	21.68424	-0.29561286	8.45274006	4.174231974	0.011595
23	22.66989	-0.30904981	8.820009585	0	0
24	23.65554	-0.32248675	9.185011198	0	0
25	24.64118	-0.3359237	9.547644679	0	0
26	25.62683	-0.34936065	9.907809662	0	0
27	26.61248	-0.3627976	10.26540565	0	0
28	27.59813	-0.37623455	10.62033203	0	0
29	28.58377	-0.38967149	10.97248806	0	0
30	29.56942	-0.40310844	11.32177291	0	0
31	30.55507	-0.41654539	11.66808568	0	0
32	31.54071	-0.42998234	12.01132538	0	0
33	32.52636	-0.44341929	12.35139099	0	0
34	33.51201	-0.45685623	12.68818145	0	0
35	34.49766	-0.47029318	13.02159572	0	0
36	35.4833	-0.48373013	13.35153276	0	0
37	36.46895	-0.49716708	13.67789161	0	0
38	37.4546	-0.51060403	14.00057135	0	0
39	38.44025	-0.52404098	14.31947123	0	0
40	39.42589	-0.53747792	14.6344906	0	0
41	40.41154	-0.55091487	14.94552904	0	0
42	41.39719	-0.56435182	15.25248632	0	0
43	42.38284	-0.57778877	15.55526252	0	0
44	43.36848	-0.59122572	15.853758	0	0
45	44.35413	-0.60466266	16.14787351	0	0
46	45.33978	-0.61809961	16.43751021	0	0
47	46.32542	-0.63153656	16.7225697	0	0
48	47.31107	-0.64497351	17.00295414	0	0
49	48.29672	-0.65841046	17.27856625	0	0
50	49.28237	-0.6718474	17.5493094	0	0
51	50.26801	-0.68528435	17.81508764	0	0
52	51.25366	-0.6987213	18.07580582	0	0
53	52.23931	-0.71215825	18.33136959	0	0
54	53.22496	-0.7255952	18.58168555	0	0
55	54.2106	-0.73903214	18.82666122	0	0
56	55.19625	-0.75246909	19.06620521	0	0
57	56.1819	-0.76590604	19.30022725	0	0
58	57.16755	-0.77934299	19.52863825	0	0
59	58.15319	-0.79277994	19.75135043	0	0
60	59.13884	-0.80621688	19.96827735	0	0
61	60.12449	-0.81965383	20.17933402	0	0
62	61.11013	-0.83309078	20.38443698	0	0
63	62.09578	-0.84652773	20.58350437	0	0
64	63.08143	-0.85996468	20.77645605	0	0
65	64.06708	-0.87340163	20.96321362	0	0
66	65.05272	-0.88683857	21.14370058	0	0
67	66.03837	-0.90027552	21.31784237	0	0
68	67.02402	-0.91371247	21.48556646	0	0
69	68.00967	-0.92714942	21.64680245	0	0
70	68.99531	-0.94058637	21.80148215	0	0
71	69.98096	-0.95402331	21.94953964	0	0
72	70.96661	-0.96746026	22.09091141	0	0
73	71.95226	-0.98089721	22.22553637	0	0

74	72.9379	-0.99433416	22.35335599	0	0
75	73.92355	-1.00777111	22.47431432	0	0
76	74.9092	-1.02120805	22.58835813	0	0
77	75.89484	-1.034645	22.69543693	0	0
78	76.88049	-1.04808195	22.79550308	0	0
79	77.86614	-1.0615189	22.8885118	0	0
80	78.85179	-1.07495585	22.97442132	0	0
81	79.83743	-1.08839279	23.05319286	0	0
82	80.82308	-1.10182974	23.12479071	0	0
83	81.80873	-1.11526669	23.18918233	0	0
84	82.79438	-1.12870364	23.24633832	0	0
85	83.78002	-1.14214059	23.29623252	0	0
86	84.76567	-1.15557753	23.33884204	0	0
87	85.75132	-1.16901448	23.37414729	0	0
88	86.73697	-1.18245143	23.40213198	0	0
89	87.72261	-1.19588838	23.4227832	0	0
90	88.70826	-1.20932533	23.43609142	0	0
91	89.69391	-1.22276228	23.44205046	0	0
92	90.67955	-1.23619922	23.44065759	0	0
93	91.6652	-1.24963617	23.43191344	0	0
94	92.65085	-1.26307312	23.41582205	0	0
95	93.6365	-1.27651007	23.39239087	0	0
96	94.62214	-1.28994702	23.36163073	0	0
97	95.60779	-1.30338396	23.3235558	0	0
98	96.59344	-1.31682091	23.27818363	0	0
99	97.57909	-1.33025786	23.22553507	0	0
100	98.56473	-1.34369481	23.16563425	0	0
101	99.55038	-1.35713176	23.09850856	0	0
102	100.536	-1.3705687	23.0241886	0	0
103	101.5217	-1.38400565	22.94270812	0	0
104	102.5073	-1.3974426	22.85410398	0	0
105	103.493	-1.41087955	22.75841612	0	0
106	104.4786	-1.4243165	22.65568746	0	0
107	105.4643	-1.43775344	22.54596386	0	0
108	106.4499	-1.45119039	22.42929405	0	0
109	107.4356	-1.46462734	22.30572957	0	0
110	108.4212	-1.47806429	22.17532469	0	0
111	109.4069	-1.49150124	22.03813633	0	0
112	110.3925	-1.50493818	21.89422403	0	0
113	111.3781	-1.51837513	21.74364977	0	0
114	112.3638	-1.53181208	21.58647802	0	0
115	113.3494	-1.54524903	21.42277555	0	0
116	114.3351	-1.55868598	21.25261139	0	0
117	115.3207	-1.57212293	21.07605677	0	0
118	116.3064	-1.58555987	20.893185	0	0
119	117.292	-1.59899682	20.70407138	0	0
120	118.2777	-1.61243377	20.50879314	0	0
121	119.2633	-1.62587072	20.30742935	0	0
122	120.249	-1.63930767	20.10006081	0	0
123	121.2346	-1.65274461	19.88677	0	0
124	122.2203	-1.66618156	19.66764096	0	0
125	123.2059	-1.67961851	19.44275926	0	0
126	124.1916	-1.69305546	19.21221184	0	0
127	125.1772	-1.70649241	18.97608698	0	0
128	126.1629	-1.71992935	18.73447424	0	0
129	127.1485	-1.7333663	18.48746433	0	0
130	128.1342	-1.74680325	18.23514904	0	0
131	129.1198	-1.7602402	17.97762122	0	0
132	130.1054	-1.77367715	17.71497464	0	0
133	131.0911	-1.78711409	17.44730396	0	0
134	132.0767	-1.80055104	17.17470464	0	0
135	133.0624	-1.81398799	16.89727289	0	0
136	134.048	-1.82742494	16.61510559	0	0
137	135.0337	-1.84086189	16.32830025	0	0
138	136.0193	-1.85429883	16.03695494	0	0
139	137.005	-1.86773578	15.7411682	0	0
140	137.9906	-1.88117273	15.44103906	0	0
141	138.9763	-1.89460968	15.13666691	0	0
142	139.9619	-1.90804663	14.8281515	0	0
143	140.9476	-1.92148358	14.51559288	0	0
144	141.9332	-1.93492052	14.19909136	0	0
145	142.9189	-1.94835747	13.87874744	0	0
146	143.9045	-1.96179442	13.55466182	0	0
147	144.8902	-1.97523137	13.22693532	0	0
148	145.8758	-1.98866832	12.89566888	0	0
149	146.8615	-2.00210526	12.56096348	0	0
150	147.8471	-2.01554221	12.22292017	0	0
151	148.8327	-2.02897916	11.88164001	0	0
152	149.8184	-2.04241611	11.53722405	0	0
153	150.804	-2.05585306	11.18977328	0	0
154	151.7897	-2.06929	10.83938866	0	0

Solar Beta Angle

155	152.7753	-2.08272695	10.48617109	0	0
156	153.761	-2.0961639	10.13022135	0	0
157	154.7466	-2.10960085	9.771640138	0	0
158	155.7323	-2.1230378	9.410528024	0	0
159	156.7179	-2.13647474	9.046985457	0	0
160	157.7036	-2.14991169	8.681112748	1.287906737	0.003578
161	158.6892	-2.16334864	8.313010065	5.182380378	0.014396
162	159.6749	-2.17678559	7.942777426	7.146304143	0.019851
163	160.6605	-2.19022254	7.570514698	8.618294669	0.02394
164	161.6462	-2.20365948	7.196321589	9.821031404	0.027281
165	162.6318	-2.21709643	6.820297652	10.84358132	0.030121
166	163.6175	-2.23053338	6.442542283	11.73236881	0.03259
167	164.6031	-2.24397033	6.06315472	12.51523286	0.034765
168	165.5888	-2.25740728	5.68223405	13.21040327	0.036696
169	166.5744	-2.27084423	5.299879209	13.83056005	0.038418
170	167.56	-2.28428117	4.916188988	14.38491594	0.039958
171	168.5457	-2.29771812	4.53126204	14.88038546	0.041334
172	169.5313	-2.31115507	4.145196885	15.322287	0.042562
173	170.517	-2.32459202	3.758091915	15.71478733	0.043652
174	171.5026	-2.33802897	3.370045409	16.06119492	0.044614
175	172.4883	-2.35146591	2.981155535	16.36416013	0.045456
176	173.4739	-2.36490286	2.591520363	16.62581559	0.046183
177	174.4596	-2.37833981	2.201237875	16.84787653	0.0468
178	175.4452	-2.39177676	1.810405973	17.03171378	0.04731
179	176.4309	-2.40521371	1.419122496	17.17840705	0.047718
180	177.4165	-2.41865065	1.027485223	17.28878403	0.048024
181	178.4022	-2.4320876	0.635591891	17.36344849	0.048232
182	179.3878	-2.44552455	0.243540207	17.40279992	0.048341
183	180.3735	-2.4589615	-0.14857215	17.40704602	0.048353
184	181.3591	-2.47239845	-0.54064749	17.37620908	0.048267
185	182.3448	-2.48583539	-0.93258814	17.31012667	0.048084
186	183.3304	-2.49927234	-1.3242964	17.20844667	0.047801
187	184.3161	-2.51270929	-1.71567454	17.07061628	0.047418
188	185.3017	-2.52614624	-2.10662478	16.89586429	0.046933
189	186.2873	-2.53958319	-2.49704932	16.68317512	0.046342
190	187.273	-2.55302013	-2.88685026	16.43125276	0.045642
191	188.2586	-2.56645708	-3.27592965	16.13847121	0.044829
192	189.2443	-2.57989403	-3.66418942	15.80280695	0.043897
193	190.2299	-2.59333098	-4.05153145	15.42174612	0.042838
194	191.2156	-2.60676793	-4.43785747	14.99215531	0.041645
195	192.2012	-2.62020488	-4.82306913	14.51009828	0.040306
196	193.1869	-2.63364182	-5.20706794	13.97056932	0.038807
197	194.1725	-2.64707877	-5.58975527	13.36709267	0.037131
198	195.1582	-2.66051572	-5.97103237	12.69109679	0.035253
199	196.1438	-2.67395267	-6.35080034	11.93088684	0.033141
200	197.1295	-2.68738962	-6.72896013	11.06984694	0.03075
201	198.1151	-2.70082656	-7.10541253	10.08302072	0.028008
202	199.1008	-2.71426351	-7.4800582	8.929810325	0.024805
203	200.0864	-2.72770046	-7.85279761	7.535465745	0.020932
204	201.0721	-2.74113741	-8.2235311	5.728234807	0.015912
205	202.0577	-2.75457436	-8.59215883	2.8019699	0.007783
206	203.0433	-2.7680113	-8.95858083	0	0
207	204.029	-2.78144825	-9.32269696	0	0
208	205.0146	-2.7948852	-9.68440694	0	0
209	206.0003	-2.80832215	-10.0436104	0	0
210	206.9859	-2.8217591	-10.4002067	0	0
211	207.9716	-2.83519604	-10.7540952	0	0
212	208.9572	-2.84863299	-11.1051753	0	0
213	209.9429	-2.86206994	-11.4533459	0	0
214	210.9285	-2.87550689	-11.7985061	0	0
215	211.9142	-2.88894384	-12.1405551	0	0
216	212.8998	-2.90238078	-12.4793916	0	0
217	213.8855	-2.91581773	-12.8149147	0	0
218	214.8711	-2.92925468	-13.1470234	0	0
219	215.8568	-2.94269163	-13.4756165	0	0
220	216.8424	-2.95612858	-13.8005932	0	0
221	217.8281	-2.96956553	-14.1218526	0	0
222	218.8137	-2.98300247	-14.439294	0	0
223	219.7994	-2.99643942	-14.7528168	0	0
224	220.785	-3.00987637	-15.0623207	0	0
225	221.7706	-3.02331332	-15.3677055	0	0
226	222.7563	-3.03675027	-15.6688714	0	0
227	223.7419	-3.05018721	-15.965719	0	0
228	224.7276	-3.06362416	-16.2581491	0	0
229	225.7132	-3.07706111	-16.546063	0	0
230	226.6989	-3.09049806	-16.8293626	0	0
231	227.6845	-3.10393501	-17.1079503	0	0
232	228.6702	-3.11737195	-17.3817288	0	0
233	229.6558	-3.1308089	-17.650602	0	0
234	230.6415	-3.14424585	-17.9144742	0	0
235	231.6271	-3.1576828	-18.1732504	0	0
236	232.6128	-3.17111975	-18.4268367	0	0
237	233.5984	-3.18455669	-18.6751401	0	0

238	234.5841	-3.19799364	-18.9180684	0	0
239	235.5697	-3.21143059	-19.1555307	0	0
240	236.5554	-3.22486754	-19.3874371	0	0
241	237.541	-3.23830449	-19.6136991	0	0
242	238.5267	-3.25174143	-19.8342293	0	0
243	239.5123	-3.26517838	-20.0489419	0	0
244	240.4979	-3.27861533	-20.2577525	0	0
245	241.4836	-3.29205228	-20.4605781	0	0
246	242.4692	-3.30548923	-20.6573376	0	0
247	243.4549	-3.31892618	-20.8479514	0	0
248	244.4405	-3.33236312	-21.032342	0	0
249	245.4262	-3.34580007	-21.2104334	0	0
250	246.4118	-3.35923702	-21.382152	0	0
251	247.3975	-3.37267397	-21.5474259	0	0
252	248.3831	-3.38611092	-21.7061856	0	0
253	249.3688	-3.39954786	-21.8583638	0	0
254	250.3544	-3.41298481	-22.0038954	0	0
255	251.3401	-3.42642176	-22.1427179	0	0
256	252.3257	-3.43985871	-22.274771	0	0
257	253.3114	-3.45329566	-22.3999973	0	0
258	254.297	-3.4667326	-22.5183419	0	0
259	255.2827	-3.48016955	-22.6297524	0	0
260	256.2683	-3.4936065	-22.7341796	0	0
261	257.254	-3.50704345	-22.8315768	0	0
262	258.2396	-3.5204804	-22.9219004	0	0
263	259.2252	-3.53391734	-23.0051097	0	0
264	260.2109	-3.54735429	-23.0811672	0	0
265	261.1965	-3.56079124	-23.1500383	0	0
266	262.1822	-3.57422819	-23.2116916	0	0
267	263.1678	-3.58766514	-23.266099	0	0
268	264.1535	-3.60110208	-23.3132356	0	0
269	265.1391	-3.61453903	-23.3530796	0	0
270	266.1248	-3.62797598	-23.3856128	0	0
271	267.1104	-3.64141293	-23.4108202	0	0
272	268.0961	-3.65484988	-23.4286901	0	0
273	269.0817	-3.66828683	-23.4392143	0	0
274	270.0674	-3.68172377	-23.4423878	0	0
275	271.053	-3.69516072	-23.4382093	0	0
276	272.0387	-3.70859767	-23.4266806	0	0
277	273.0243	-3.72203462	-23.4078071	0	0
278	274.01	-3.73547157	-23.3815976	0	0
279	274.9956	-3.74890851	-23.348064	0	0
280	275.9813	-3.76234546	-23.3072219	0	0
281	276.9669	-3.77578241	-23.2590901	0	0
282	277.9525	-3.78921936	-23.2036906	0	0
283	278.9382	-3.80265631	-23.1410489	0	0
284	279.9238	-3.81609325	-23.0711935	0	0
285	280.9095	-3.8295302	-22.9941563	0	0
286	281.8951	-3.84296715	-22.9099722	0	0
287	282.8808	-3.8564041	-22.8186792	0	0
288	283.8664	-3.86984105	-22.7203183	0	0
289	284.8521	-3.88327799	-22.6149337	0	0
290	285.8377	-3.89671494	-22.5025723	0	0
291	286.8234	-3.91015189	-22.3832839	0	0
292	287.809	-3.92358884	-22.2571211	0	0
293	288.7947	-3.93702579	-22.1241391	0	0
294	289.7803	-3.95046273	-21.9843959	0	0
295	290.766	-3.96389968	-21.837952	0	0
296	291.7516	-3.97733663	-21.6848702	0	0
297	292.7373	-3.99077358	-21.525216	0	0
298	293.7229	-4.00421053	-21.3590569	0	0
299	294.7086	-4.01764748	-21.1864629	0	0
300	295.6942	-4.03108442	-21.007506	0	0
301	296.6798	-4.04452137	-20.8222602	0	0
302	297.6655	-4.05795832	-20.6308017	0	0
303	298.6511	-4.07139527	-20.4332082	0	0
304	299.6368	-4.08483222	-20.2295596	0	0
305	300.6224	-4.09826916	-20.0199373	0	0
306	301.6081	-4.11170611	-19.8044245	0	0
307	302.5937	-4.12514306	-19.5831056	0	0
308	303.5794	-4.13858001	-19.3560669	0	0
309	304.565	-4.15201696	-19.1233957	0	0
310	305.5507	-4.1654539	-18.8851809	0	0
311	306.5363	-4.17889085	-18.6415124	0	0

Solar Beta Angle

312	307.522	-4.1923278	-18.3924814	0	0
313	308.5076	-4.20576475	-18.1381801	0	0
314	309.4933	-4.2192017	-17.8787017	0	0
315	310.4789	-4.23263864	-17.6141404	0	0
316	311.4646	-4.24607559	-17.3445909	0	0
317	312.4502	-4.25951254	-17.0701492	0	0
318	313.4359	-4.27294949	-16.7909118	0	0
319	314.4215	-4.28638644	-16.5069756	0	0
320	315.4071	-4.29982338	-16.2184385	0	0
321	316.3928	-4.31326033	-15.9253987	0	0
322	317.3784	-4.32669728	-15.627955	0	0
323	318.3641	-4.34013423	-15.3262064	0	0
324	319.3497	-4.35357118	-15.0202527	0	0
325	320.3354	-4.36700813	-14.7101935	0	0
326	321.321	-4.38044507	-14.3961291	0	0
327	322.3067	-4.39388202	-14.0781599	0	0
328	323.2923	-4.40731897	-13.7563864	0	0
329	324.278	-4.42075592	-13.4309093	0	0
330	325.2636	-4.43419287	-13.1018297	0	0
331	326.2493	-4.44762981	-12.7692483	0	0
332	327.2349	-4.46106676	-12.4332663	0	0
333	328.2206	-4.47450371	-12.0939847	0	0
334	329.2062	-4.48794066	-11.7515045	0	0
335	330.1919	-4.50137761	-11.4059267	0	0
336	331.1775	-4.51481455	-11.0573524	0	0
337	332.1632	-4.5282515	-10.7058826	0	0
338	333.1488	-4.54168845	-10.3516179	0	0
339	334.1344	-4.5551254	-9.99465927	0	0
340	335.1201	-4.56856235	-9.63510727	0	0
341	336.1057	-4.58199929	-9.27306244	0	0
342	337.0914	-4.59543624	-8.90862519	0	0
343	338.077	-4.60887319	-8.54189577	3.364659756	0.009346
344	339.0627	-4.62231014	-8.17297428	6.012352939	0.016701
345	340.0483	-4.63574709	-7.8019607	7.74479222	0.021513
346	341.034	-4.64918403	-7.42895483	9.099456227	0.025276
347	342.0196	-4.66262098	-7.05405631	10.2265953	0.028407
348	343.0053	-4.67605793	-6.67736465	11.19429666	0.031095
349	343.9909	-4.68949488	-6.29897918	12.04031815	0.033445
350	344.9766	-4.70293183	-5.91899907	12.78817136	0.035523
351	345.9622	-4.71636878	-5.53752336	13.45364786	0.037371
352	346.9479	-4.72980572	-5.15465093	14.04793241	0.039022
353	347.9335	-4.74324267	-4.77048051	14.57926104	0.040498
354	348.9192	-4.75667962	-4.38511071	15.05387797	0.041816
355	349.9048	-4.77011657	-3.99864	15.4766228	0.042991
356	350.8904	-4.78355352	-3.61116672	15.85130856	0.044031
357	351.8761	-4.79699046	-3.22278912	16.18097458	0.044947
358	352.8617	-4.81042741	-2.83360532	16.46806104	0.045745
359	353.8474	-4.82386436	-2.44371335	16.71453232	0.046429
360	354.833	-4.83730131	-2.05321117	16.92196576	0.047005
361	355.8187	-4.85073826	-1.66219666	17.09161648	0.047477
362	356.8043	-4.8641752	-1.27076763	17.22446476	0.047846
363	357.79	-4.87761215	-0.87902184	17.32125064	0.048115
364	358.7756	-4.8910491	-0.48705701	17.38249858	0.048285
365	359.7613	-4.90448605	-0.09497084	17.40853428	0.048357

Solar Array and Battery Sizing Estimation:

Solar Array	**5U CubeSAT**	**Units**	**Comments**
Solar constant	1367	watts/m²	SMAD p.333
β (Beta)	45	deg	
rho	1.162152912	Radians	ASIN(μ Earth/(μ Earth+Alt))
Pwr Reqt Daylight (Pd)	5.18972667	watts	From Power Budget Page
Pwr Reqt Eclipse (Pe)	5.18972667	watts	From Power Budget Page
Altitude	35765.4	km	
Orbital period (To)	1435.0	min	2*π/SQRT(μ Earth/(Rearth+Altsc)^3)/60
Period of Eclipse (Te)	1010.5	min	To*ACOS(COS(rho/COS(β)))/π
Period of Daylight (Td)	424.5	min	To-Te
Int pwr x-fer eff Eclipse (Xe)	0.6		Power system property
Int pwr x-fer eff Daylight (Xd)	0.8		
Req Solar Array Pwr (Psa)	27.1	watts	(Pe*Te/Xe+Pd*Td/Xd)/Td
Cell Efficiency (EOL)	0.243		At 15 years mission duration, not 2yrs
Power Output (Po)	332.2	watts/m²	EOL*Solar constant
Inherent Degradation (Id)	0.96		Property of UTJ solar cells
Theta (θ)	45	deg	0 for 1-axis gimballed
Pwr @ beg of life (Pbol)	225.5	watts/m²	Po*Id*COS(θ*π/180)
Design Life	2.0	years	Mission Property
Lifetime Degradation (Ld)	0.9973		(1-Annual Degradation/100)^Design Life
Annual Degradation	0.13	%	DL*(1/15)
Pwr @ end of life (Peol)	224.9	watts/m²	Pbol*Ld
Req Solar Array 1 Area (Asa1)	0.120	m²	Psa/Peol
Req Solar Array 2 Area (Asa2)	0.082	m²	Psa/solar constant/EOL
Solar Array Mass	**0.355**	**kg**	SC Mass Page

Batteries	**5U CubeSAT**	**Units**	**Comments**
Battery Specific Energy Density	40	W*hr/kg	Li-Ion property
Voltage	15.00	volts	Determined by goal seeking cell M8
Current	0.35	Amps	15A Max continuous discharge(Required Power/Voltage)
Depth of Discharge (DOD)	70.00	%	
Number of Batteries (N)	4.00	#	
Transmission efficieny (n)	0.95		Battery Property
Battery Capability (Cr)	105.00	Watt-hrs	Eclipse Power Reqt
Battery Capability (Cr)	7.0	Amp-hrs	Nominal is 7[Amp*hr]
Individual Battery Mass	0.146	kg	Battery Property
Total Mass (kg)	**0.584**	**kg**	Total SC battery mass
Total Power System Mass	**1.03**	**kg**	**Solar Arrays, Batteries and cabling**

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	km³/kg*s²
μ Earth =	398600.4415	km³/s²
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Comparative Data		
Cell Type	Efficiency	(link)
Silicon	0.148	1
GaAs	0.185	
Multijunction	0.22	
Ultra Triple Junction	0.283	

Ref: SMAD p.109, 333, 422; SAFT MP 176065 Integrat*ion* Batteries;
Spectrolab Ultra Triple Junction (UTJ GaInP2/GaAs/Ge) Solar cells

Power Budget Allocation:

Mission Design Data

GEO sep mass* [kg]	14
dsgn life [yr]	2
station lat [deg]	45
closest stbl pt [deg]	75
g [m/s²]	9.80665

*i=28.4[deg], 185[km x GEO]
Shuttle/IUS (pg 728)

Propellant & Dry Mass Calculation

Item	ΔV[m/s]	ISP[s]	Efficiency	RCS [kg]	SCM [kg]	SC [kg]
GEO sep mass						14.32702706
pre-burn RCS				0.010		14.31702706
GEO to Sub-GEO	802.89	1500	0.99		0.8	13.5
Post RCS to Sub-GEO				0.010		13.5
N-S Sta Kpg	102.76	1500	0.99	0.095		13.4
E-W Sta Kpg	2.9704671	1500	0.99	0.003		13.4
Sta reposition'g	0	1500	0.99	0.000		13.4
ACS				0.830		12.6
Deorbit	7	1500	0.99	0.006		12.6
RCS margin @ 10%				0.1		**12.5**

SCM -> Station Changing Motor

Mass Summary

Dry mass [kg]	**12.5**
RCS Prop [kg]	1.1
SCM Prop [kg]	0.8
GEO sep mass [kg]	14.3

PL RF Power Allocation

Max Power System mass [kg]	0.28	SC Design Page -> Modeled uncorrectly
Pwr gen capability needed [W]	5.19	SC Design Page -> Modeled uncorrectly
Solar Array Area [m^2]	0.17	
PWR gen from 1 ATJ array [W]	39.10	ATJ Solar array area illuminated.
Pwr Avail. w/ 10% margin [W]	35.19	PWR gen minus margin
PL Budget @ 70% Avail. [W]	24.63	(pg 316,345 Tables10-9,-35)
RF Pwr@ 35% efficiency [W]	0.33	System Characteristic (=Pta/Ptin)

Mission Design Data

Dry Mass =	11.55	kg	Propellant for μVAT is dry mass
Mission Duration =	2	years	
Station Longitude =	45	deg	Continuously changing
nearest stable point =	75	deg	At GEOSTA

Subsystem Peak Power Requirements

Command & Data Handling =	7.5	watts	During maximum computing
Attitude Control =	3.2	watts	10% during transmitting and imaging
Station Keeping =	50.0	watts	μVAT Propulsion
TT&C and Data Transciever =	32.0	watts	Maximum power output
Inertial Control Unit =	1.0	watts	MEMs Rate Sensor imbedded in Dynacon200
Payload =	1.7	watts	During picture taking
Power Control Unit =	4.0	watts	Parasitic power for solar array and battery control
Earth Sensor =	0.25	watts	0.05 watts each for 4
Sun Sensor =	0.25	watts	0.05 watts each for 5
Thermal Control System =	40.0	watts	Maximum if both heaters are on at once
Total =	139.9	watts	

<u>Battery Information</u> ***At full charge***

Battery Type :	SAFT MP 176065 Intrgrat*ion*	
Voltage =	3.75	V
Current Capacity =	7	Ah
Charge Rate =	2 to 3	h
Charge/2 Rate =	3 to 4	h
Charge/5 Rate =	6 to 7	h
Power =	26.25	Wh
Max Continuous Discharge =	15	A
Pulse Discharge Current =	30	A
Total Number of batteries =	4	
Discharge Cutoff Voltage =	2.5	V
Battery Power Stored =	105	Wh

<u>PL RF Power Allocation</u> ***Detailed Worst Case Power Anaylsis***

	Bathing	Eclipse	At XMIT	Slewing	Sta Keeping	Imaging	Units
Max Pwr Sys mass =	1.0	1.0	1.0	1.0	1.0	1.0	kg
Solar Panel Pwr gen cap =	**35.2**	**0**	**0**	**0**	**0**	**0**	W
Pwr Avail w/ 10% margin =	31.7	0.0	0.0	0.0	0.0	0.0	W
Command & Data Handling =	1.1	3.0	3.8	7.5	3.8	7.5	W
Attitude Control =	0.3	0.3	0.3	3.2	3.2	0.6	W
Station Keeping =	0.0	0.0	0.0	0.0	50.0	0.0	W
TT&C and Data Transciever =	0.0	0.0	32.0	0.0	0.0	0.0	W
Inertial Control Unit =	1.0	1.0	1.0	1.0	1.0	1.0	W
Payload Power Use =	0.0	0.0	0.0	0.0	0.0	1.7	W
Power Control Unit =	4.0	4.0	4.0	4.0	4.0	4.0	W
Spacecraft Station Sensing =	0.50	0.50	0.50	0.50	0.50	0.50	W
Thermal Control System =	0.0	20.0	0.0	0.0	0.0	0.0	W
Time Spent per Orbit =	**81234**	**4164**	**187**	**480**	**30**	**6**	**sec**
Power Used =	**7.0**	**28.8**	**41.6**	**16.2**	**62.5**	**15.4**	**W**
Solar cell Pwr NOT used =	24.71	-28.84	-41.59	-16.22	-62.47	-15.39	W
Fully Charged Battery =	105.00	105.00	105.00	105.00	105.00	105.00	Wh
System Power Available =	**129.71**	**76.16**	**63.41**	**88.78**	**42.53**	**89.61**	**W**

Ref: SMAD p.334, 314-316, 412, 418-422, 423-425

<u>Constants</u>

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	$km^3/kg*s^2$
μ Earth =	398600.4415	km^3/s^2
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

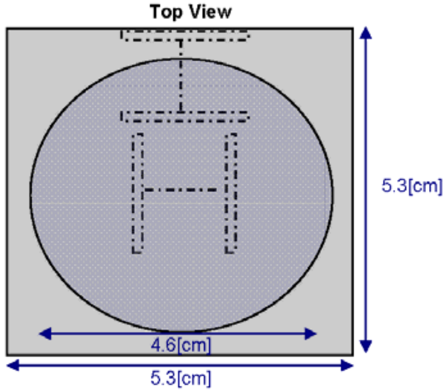
Link Budget:

Transmitter

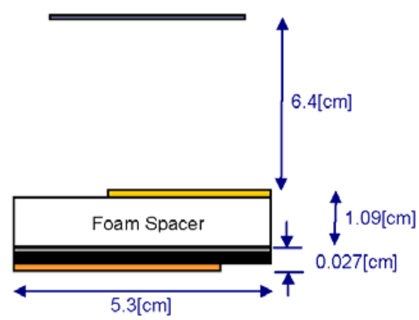
	SAT to AFSCN		SAT to TDRSS		
Transmit Frequency (f) =	**2.1064**	**2.2875**	**2.1064**	**Ghz**	Standard to XMIT to TDRSS
Transmit Wavelength (λ) =	0.142	0.131	0.142	m	λ = c/f
Power Budget Allocation in watts (Ptin) =	15	watts	15	watts	Power Budget Allocation
Patch Array Length (Al) =	0.0530		0.0530	m	Given Nessel
Patch Array Height (Ah) =	0.0530		0.0530	m	Given Nessel
Array area (Aa) =	0.0028		0.0028	m^2	Aa = Al*Ah
Sub-Reflector (SR) Diameter (Dsr) =	0.0463		0.0463		Nessel: 0.325*λ
Distance between SR and Patch Array =	0.0640		0.0640		Nessel: 0.45*λ
Transmitter Efficiency (ηdc) =	0.33		0.33		Pta/Ptin
Available Transmit Power (Pta) =	5	watts	5	watts	Equipment Property
Transmitter Power in Decibels (Pt) =	6.989700043	dBw	6.989700043	dBw	10*LOG(Pta)
Antenna Mass =	0.5	kg	0.5	kg	SMAD pg. 394, Table 11-26, Scaled to meet design
Transmitter Line Loss (Ll) =	-1	dB	-1	dB	
Transmit Antenna Beamwidth (θbt) =	33	deg	33	deg	**Adjust to meet ground target requirement**
Transmit Antenna Pointing Error (θet) =	1	deg	1	deg	
Assumed Antenna Efficiency (η) =	0.7		0.7		Given Nessel
Transmit Antenna Gain (Gt) =	15.2	dBi	15.2	dBi	Assuming same results for f=2.1064 as f=2.2875[GHz]
Equiv. Isotropic Rad. Pwr. (EIRP) =	**21.18970004**	**dBw**	**21.18970004**	**dBw**	Pt+Ll+Gt
Bandwidth (BW) =	**250**	**MHz**	**250**	**MHz**	Nessel: BW = 250[Mhz] maintaining at least 15[dBi]

Short Backfire Antenna (SBA) Array

Top View



Side View



Spatial Geometry for Satellite to Terrestrial

Sat Xmt Ant Max Cvg Ang (η°) =	0.287979327	rad	η° = 0.5*θbt
Earrth Central Angle (λ) =	-0.287935436	rad	λ = 180-{η-acos[sin(η)/(Re/Ro)]+90}
ECA (λ) in degrees =	-16.49748527	degrees	**This can reach the req ground targets w/ no slew**
Slant Range (S) =	36073.43558	km	S = SQRT[(Ro-Re*cos(λ))^2 + (Re*sin(λ))^2]

Spatial Geometry for SAT to TDRSS Satellite Constellation (if necessary)

Sat #1 Orbit Radius² =			1776074592.3	km²	SAT Radius^2
Sat #2 Orbit Radius² =			1777814701.9	km²	TDRSS SC Operating Radius^2
Maximum Sat - Sat Distance =			59614.5	km	SQRT(SAT Rad^2+TDRSS Rad^2) Worst case seperation & earth would block line
Max Constellation SC - SC Distance =			126.0	deg longitude	Great TDRSS SC seperation (F-3 & F-4)
Seperation Arc Length (L) =			92723.5	km	L = θ*r
Max half-Sep between TDRSS SC =			46361.7	km	L/2
Slant Range at Max Sep =			48679.8	km	Linear Geometry Est + 5% margin: S = SQRT(OrbitSep^2+HalfSep^2)+5%
Coverage footprint Diameter =	20490.81848	km			2*(S*SIN(η°))/(SIN(π/2)): plane geometry estimate
Coverage footprint in NM =	11064.16087	NM			Coverage Footprint diameter*0.539957
Power Flux Density (PFD) =	-140.9461487	dB	-143.5493797	dB	PFD = EIRP/(4pS^2)
PFD/4kHz band =	-176.9667486	dB	-179.5699796	dB	PFD/4000
Space (path) Loss (Ls) =	-190.064567	dB	-192.2440112	dB	Ls = 147.55-20log(S~m)-20log(f~Hz)
Propagation & Polarization Loss (La) =	-0.3	dB	-0.3	dB	SMAD Table 13-13

Receiver

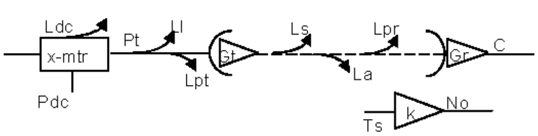
Assumed Antenna Efficiency (η) =	0.65		0.7		SMAD Figure of Merit p 553
Receiver Antenna Diameter (Dr) =	7	m	4.572	m	Smallest AFSCN dish size & TDRSS RCVR respectfully
Peak Receiver Antenna Gain (Gpr) =	41.91191129	dB	38.53392176	dB	G = -159.59+20*LOG(Dt)+20*LOG(f[GHz])+10*LOG(η)
Receiver Antenna Beamwidth (θbr) =	1.424230915	deg	2.180581016	deg	θ = 21/(D*f)
Receiver Antenna Pointing Error (θer) =	0.812115458	deg	1.190290508	deg	θer = θbr/2+0.1
Receiver Antenna Pointing Loss (Lθr) =	-3.901718946	dB	-3.575548991	dB	Lθ = -12*(θet/θbt)^2
Receiver Antenna Gain (Gr) =	38.01019234	dB	34.95837277	dB	Gpr+Lpr

Link Design Equations

System Noise Temperature (Ts) =	135	K	614	K	SMAD Table 13-10
Data Rate (R) =	3.00E+05	bps	3.00E+05	bps	SMAD pg. 385, Table 11-19
Eb/No (1) =	17.45905617	dB	5.975616409	dB	Eb/No = EIRP+Lpr+Ls+La+Gr+228.6-10LogTs-10LogR
Carrier-to-Noise Density Ratio (C/No) =	72.23026872	dB-Hz	60.74682896	dB-Hz	C/No = Eb/No+ 10*logR
Bit Error Rate (BER) =	1.00E-06	--------	1.00E-06	--------	BPSK Viterbi for TT&C, BPSK Reed-Soloman for Data Link
Required Eb/No (2) =	5.2	dB	2.8	dB	SMAD Figure 13-9
Implementation Loss (3) =	-2	dB	-2	dB	Estimate
Margin =	**10.25905617**	**dB**	**1.175616409**	**dB**	(1)-(2)+(3)

Alternate Approach

Carrier (C) =	-135.1	-140.0	dB	C = Pt*Ll*Gt*Ls*La*Gr
Noise (No) =	-207.3	-200.7	dB	No = k*Ts
Carrier/Noise Density Ratio (C/No) =	72.2	60.7	dB	C/No = C-No
Error Bits/Noise Ratio (Eb/No) =	17.5	6.0	dB	Eb/No = C/No-10*logR



Note: C=10^(-135.3/10)=~0.03 picowatts

Ref: Tomasi p.551-552; Sharma <u>A wideband Microstrip Array Antenna with Unique Dumbell Shaped Aperture Coupled Radiating Elements</u>
TDRSS info: http://msl.jpl.nasa.gov/QuickLooks/tdrssQL.html & http://msp.gsfc.nasa.gov/tdrss/tconst.html &http://msp.gsfc.nasa.gov/tdrss/scraft.html

Constants

Radius Earth (Re) =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	km³/kg*s²
μ Earth =	398600.4415	km³/s²
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Spherical Spacecraft Analysis:

Item	Symbol	5U CubeSAT	Units	Source	Comments
Surface Area	A	0.05	m^2	Geometry	Rectangular Surface Area
Diam. Of equiv. Sphere	D	0.13	m^2	Geometry	
Max power dissipation	Qwmax	62.47	W	Power Budget	
Min power dissipation	Qwmin	6.96	W	Power Budget	
Altitude	H	35765.37	km	Calculated	Orbital Radius - Earth's Mean Sea Level Radius
Earth Radius	Re	6378.14	km	Given	
Earth angular radius	rho	0.15	radians	Eq 5-16	
Albedo correction	Ka	0.74		Eq 11-28	
Max Earth IR emission @ surface	qmax	258	W/m^2	pg 447	
Min Earth IR emission @ surface	qmin	216	W/m^2	pg 447	
Direct solar flux	G	1399	W/m^2	pg 447	
Albedo	al	35	%	pg 447	
Emmisivity	ε	0.84		Table 11-46	3M Black Velvet paint
Absorptivity (solar)	α	0.97		Table 11-46	3M Black Velvet paint; BOL value
Stefan-Boltzmann constant	σ	5.67E-08	$W/(m^2.K^4)$	Biven	
Earth view factor=(1-cosρ)/2	vf	0.0058		pg 448	
Sphere x-section area=πD^2/4	Acx	0.0125	m^2	Geometry	
Solar input	Acx*G*α	16.96	W	calc	
Earth input	A*vf*qmax*ε	0.06	W	calc	
Albedo input	A*vf*G*al*af*Ka/100	0.10	W	calc	
Worst case hot temp	Tmax	154.57	C	eq 11-34	
Worst case cold temp	Tmin	-40.03	C	eq 11-35	
Upper temp limit	Tu	60	C	Equipment Data Sheets	
Lower temp limit	Tl	0	C	Equipment Data Sheets	
Radiator area (WC hot)	Ard	0.019	m^2	eq 11-19	
Radiator temp (WC cold)	Tr	50.48	C	eq 11-20	
Heater power for lower limit	Qn	0.91	W	eq 11-21	

Ref: SMAD p.446-456

Equipment Temperature Limits

Spacecraft Internal Units	Temp Range [°K]		
Worst Case Envelope	273	333	Based on equipment properties
Payload			
Optical Sensors (CCD most temp sensitive)	273	333	
Onboard Computer	233	358	
TT&C Units	243	333	
Electrical Power			
Batteries	253	333	
Solar Arrays	168	383	
Attitude Control			
Earth Sensors	243	353	
Sun Sensors	243	353	
Inertial Measurement Unit (IMU)	243	333	
Reaction Wheels	243	333	
Propulsion	213	353	
Processors			
AFRL RAD6000 Computer (microprocessor)	253	333	
Fault Tolerant Reconfigurable Processor	253	333	
Thermal Control			
MLI	113	523	
Radiators	178	333	
Heaters, thermostats, heat pipes	238	333	
Minco CT325 Thermal Controller	233	343	
Antennas			
Microstrip Patch-Fed Short Backfire Antenna	213	348	

Ref: SMAD p.428, Various Equipment Data Sheets

Thermal Hardware Mass and Power Requirements

Hardware Properties - Power and Mass:

Hardware	Mass [kg]	Power [W]	Comments		
MLI	0.9782	0	.73kg/m^2*(As/c-Arad)	SMAD p.457, Table 11-49	Based on 15 layers
Heaters (3)	0.035	80	3 Kapton heaters sized for components	(2) KH-202/(*)-P, (1) KH-404/(*)-P	
Thermostats					
Thermisters					
Adhesives/Paints	0.120	0	Spread throughout the spacecraft		
Heat Pipes (NH3)	0.225	0	.15kg/m*1.5m	SMAD p.457, Table 11-49	
Radiator Panels	0.062	0	3.3 kg/m^2*Area	SMAD p.457, Table 11-49	
Electronic Controllers	0.024	0	.2kg/ and 1-3W/	SMAD p.457, Table 11-49	
Radiative Coupler	0.005	0			
TOTALS	**1.449**	**80**	***If all heaters are on at the same time, which they will not be.***		

Ref: SMAD p.457

Cost Estimation

Cost Estimation:

COTS System components

Components	Quantity	Cost FY2007 Dolars	Cost FY2000 Dolars	
Batteries -	4	$1,600.00	$1,393.73	SAFT MP176065 Integr*ation*
Solar Cells -	0.17	*$52,896.00*	$46,076.66	Spectrolab UTJ (GaInP2/GaAs/Ge) *($ est using CER SMAD p.797)*
Command & Data Handling -	1	$500,000.00	$435,540.07	AFRL RAD6000 Computer (Microprocessor)
Propulsion/Thrusters -	1	*$20,117.60*	$17,524.04	Micro Aerospace Solutions Vacuum Arc Thrusters (VAT) *($ est using CER SMAD p.797)*
Stability Control -	4	$180,000.00	$156,794.43	Dynacon MicroWheel 200 (3.2[W] max with rate sensor)
Data Transciever -	1	$150,000.00	$130,662.02	AeroAstro Modular S-Band Radio System
Star Tracker -	0	$0.00	$0.00	Star Tracker integrated into Payload
Inertial Control Unit -	1	$0.00	$0.00	*Imbedded into Command & Data Handling CPU using inputs from rate sensor, cmds to RWs.*
Power Control Unit -	1	$60,000.00	$52,264.81	Reconfigurable Fault Tolerant Processor configured for power management, back-up photo processing and attitude control.
Earth Sensor -	5	$75,000.00	$65,331.01	Optical Energy Tech
Sun Sensor -	5	$200,000.00	$174,216.03	Optical Energy Tech
TT&C Transciever -	1	$150,000.00	$130,662.02	AeroAstro Modular S-Band Radio System

Custom System Components (Cost Estimating for Small Satellites including RDT&E and Theoretical First Unit)

Components	Quantity	Parameter	Cost FY2000 Dolars	
Optical Payload -	1	0.09	$86,587.60	SMAD CERs for visible paylaod (p.795&6) plus cost of Kodak sqaure matrix CCD KAF-383000
Antenna -	1	3.93	$2,938.67	SMAD CERs for communications subsystem (p.795&6)
Structure -	1	0.67	$1,295.19	SMAD CER for structures (p.797), using total S/C mass with 10% margin included
Thermal Control -	1	0.49	$4,257.01	SMAD CER for thermal (p.797) plus control cost with estimate for mini readiator.
All Computer Code -	1	275000	$119,625.00	SMAD CER (p.800); One Time Incurred Cost
Satellite per Constellation(Csn) -	33			
TFU Cost -	**1**		**$1,425,168.28**	Sum of all COTS and Custom components.
Learning Curve Slope (S) -		90%		Recommended S percent for 10 to 50 units to be build by SMAD, p.809.
Learning Cuvre Factor (L) -	33	19.40		(Number of satellites)^(1-((LN(100%/S))/(LN(2)))); SMAD p.809.
Constellation Production Cost (C_{pc}) -	1		$27,641,478.66	$L*C_{sn}$
Average Satellite Cost -	33		$837,620.57	C_{pc}/C_{sn}
Integration cost per satellite (S_{int}) -	1		$226,385.79	Assuming 10% of satellite's construction cost to integrate into a free/shared launch
Cstl Integration Cost (C_{int}) -	1		$4,390,806.39	$S_{int}*C_{sn}$
Cstl Launch Cost -	1		$0.00	Assuming free/shared launch
Cost to field Cstl -	1		$32,032,285.05	$C_{pc}+C_{int}$+(Cstl Launch Cost)

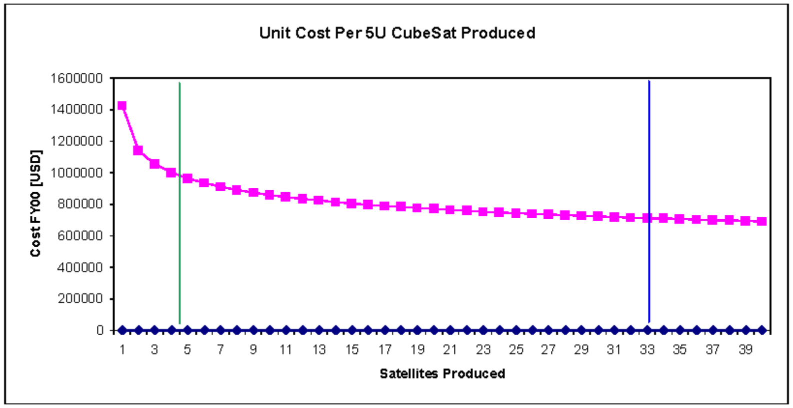
Constellation Operations and Support Cost Determination

Contractor Labor -	13	1 [year]	$2,080,000.00	SMAD, p.801
Government/Military Labor -	20	1 [year]	$2,200,000.00	SMAD, p.801
Maintenance Labor -	0	1 [year]	$0.00	Free due to assumption that constellation will utilize existing facilities.
Cost to operate Cstl per year -	1		$4,280,000.00	6 Contractors and 9 Military employees needed per 15 satellites per year

	Cost FY2000 Dolars	Cost FY2007 Dolars	
TFU Cost -	**$1,425,168.28**	**$1,588,777.60**	
Total Government Cost -	**$40,592,285.05**	**$45,252,279.37**	**To field constellation and operate for only two years.**

Determination of Unit Cost Curve over Satellite Constellation Production

Unit Number	Production Cost	Average Cost	Unit Cost
1	$1,425,168.28	$1,425,168.28	$1,425,168.28
2	$2,565,302.91	$1,282,651.45	$1,140,134.63
3	$3,617,957.79	$1,205,985.93	$1,052,654.89
4	$4,617,545.23	$1,154,386.31	$999,587.44
5	$5,579,439.20	$1,115,887.84	$961,893.97
6	$6,512,324.03	$1,085,387.34	$932,884.83
7	$7,421,755.99	$1,060,250.86	$909,431.96
8	$8,311,581.42	$1,038,947.68	$889,825.42
9	$9,184,612.63	$1,020,512.51	$873,031.21
10	$10,042,990.56	$1,004,299.06	$858,377.93
11	$10,888,396.41	$989,854.22	$845,405.84
12	$11,722,183.25	$976,848.60	$833,786.84
13	$12,545,462.16	$965,035.55	$823,278.91
14	$13,359,160.78	$954,225.77	$813,698.62
15	$14,164,064.56	$944,270.97	$804,903.77
16	$14,960,846.55	$935,052.91	$796,781.99
17	$15,750,089.54	$926,475.86	$789,242.99
18	$16,532,302.73	$918,461.26	$782,213.19
19	$17,307,934.62	$910,943.93	$775,631.89
20	$18,077,383.01	$903,869.15	$769,448.40
21	$18,841,003.05	$897,190.62	$763,620.03
22	$19,599,113.53	$890,868.80	$758,110.49
23	$20,352,002.18	$884,869.66	$752,888.65
24	$21,099,929.85	$879,163.74	$747,927.67
25	$21,843,134.05	$873,725.36	$743,204.20
26	$22,581,831.89	$868,532.00	$738,697.84
27	$23,316,222.57	$863,563.80	$734,390.67
28	$24,046,489.41	$858,803.19	$730,266.85
29	$24,772,801.71	$854,234.54	$726,312.30
30	$25,495,316.21	$849,843.87	$722,514.49
31	$26,214,178.42	$845,618.66	$718,862.21
32	$26,929,523.79	$841,547.62	$715,345.37
33	$27,641,478.66	$837,620.57	$711,954.88
34	$28,350,161.17	$833,828.27	$708,682.50
35	$29,055,681.96	$830,162.34	$705,520.79
36	$29,758,144.92	$826,615.14	$702,462.95
37	$30,457,647.70	$823,179.67	$699,502.78
38	$31,154,282.31	$819,849.53	$696,634.61
39	$31,848,135.55	$816,618.86	$693,853.24
40	$32,539,289.43	$813,482.24	$691,153.88



Ref: SMAD p.784- 802.

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APPENDIX C. 1U-CUBESAT DESIGN EXCEL WORKBOOK

Optical Payload Design

Orbit Parameters

SC Altitude (Hsc) =	35785.365	km	given
SC Orbit period (Psc) =	1436.321	min	(m/(Re+Hsc)^3)^(1/2)
Target's altitude =	35785.863	km	Assuming GEOSTA
Target's Obital Period (Ptar) =	1436.346	min	sqrt(m/(Re+Htar)^3)
SC ω (ωsc) =	7.291E-05	Rad/sec	ωsc = (2*PI)/Psc
Target ω (ωtar) =	7.291E-05	Rad/sec	ωtar = (2*PI)/Ptar
SC orbital Radius (Rsc) =	42169.111	km	(μ Earth/ω^2)^(1/3)
Target orbital Radius (Rtar) =	42169.609	km	(μ Earth/ω^2)^(1/3)
SC velocity (Vsc) =	3.074480	km/sec	Vsc = ωsc*Rsc
Target's velocity (Vtar) =	3.074462	km/sec	Vtar = ωtar*Rtar
Closing Velocity (CV[km/sec]) =	0.000018	km/sec	Vtar-Vsc
Closing Velocity (CV) =	0.018156	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	0.498	km	Htar-Hsc
Target's Orbital Circumference (Cirtar) =	264869	km	2*π*(Htar*6378.137)
Distance SC travels relative to GEOSTA orbit =	1.569	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	168879.697	days	Cirtar/Distance SC travels
Time for SC to traverse GEOSTA orbit =	462.367	years	(Cirtar/Distance SC travels)*365.25

C328-7640 JPEG Compression VGA Modules Parameters

Pixel Height =	9	μm	Data Sheet
Pixel Width =	9	μm	Data Sheet
# of Horizontal pixels (PxlH) =	160	pxl	Effective # of pxl from Data Sheet
# of Vertical pixels (PxlV) =	128	pxl	Effective # of pxl from Data Sheet
Maximum Data Rate (DR) =	115200	Hz	Data Sheet
Bits/pxl (Nb) =	8		
Readout Time (Tlrout) =	0.17	sec/image	Data Sheet
Pixel Period (1 count) =	0.001041667	sec	Data Sheet
#sec/pxl =	0.001302083	sec/pxl	Tlrout/PxlV
#pxls/sec (Z) =	768	pxl/sec	1/#sec/pxl
Max Operating Temp Range =	0 to 60	°C	Assumption
Guaranteed Operating Temp Range =	10 to 50	°C	Assumption

Target Parameters

In track Elevation angle (e) =		deg	Depends on target's relative postion
Slew angle (h) =		deg	Depends on target's relative postion
Slant range (Rs) =		km	Depends on target's relative postion
# Active Pixels (Zact) =	20480	pixels	Data Sheet
Pixel Integration Time (Tipxl) =	0.001041667	sec	Data Sheet
Relative Motion During pxl image capture(Blur) =	1.89129E-05	m	CV*Tipxl
#pxls/sec (Z) =	19660800	pxls/sec	Zact/Tipxl
Bits/pxl (Nb) =	8	bits/pixel	Choosen
DataRate (DR) =	163840	bps	Zact*Nb
Megabytes per Picture (Psz) =	0.16384	MBytes	Data Sheet
Compressed Image Size (cPsz) =	0.1152	MBytes	Psz/12 ->Need to verify for jpeg format

Optic System

Aperature (Ap) =	0.01	m	S/C property
CPA Seperation (CPA Sep) =	498.0	m	Htar-Hsc; will differ when e≠ 90 since SC is orbiting lower than Target
Lambda (l) =	4.0E-07	m	Choosen
Resolution (X) =	0.049	m	(2.44*l*Sep)/Ap
Resolution of Target at 50[km] (X50km) =	4.880	m	(2.44*l*50000[m])/Ap
Resolution of Target at 100[km] (X100km) =	9.760	m	(2.44*l*100000[m])/Ap
0.5[m] Resolution of Target =	0.50	m	(2.44*l*100000[m])/Ap
Distance for 0.5[m] Resolution of Target (D0.5m) =	5123.0	m	Determined by goal seeking 0.5[m] Resolution
Detector width (square pixel width) (d) =	9.00E-06	m	Data Sheet
Quality factor (Q) =	7.00E-01		0.5<Q<2 selected
Operating wavelength (lw) =	4.00E-07	m	selected
Focal length (f) =	0.00463	m	Sep*d/X
Diffraction-limited aperature diameter (D) =	0.00035	m	2.44*lw*f*Q/d
F-number (F#) =	2.800	#	Data Sheet

Thermal Requirements

Operating Temperature =	273-333	K	Based on CCD restriction
Operating Temperature =	0-60	C	Based on CCD restriction

Constants

Radius Earth =	6378.137	km
mass Earth =	5.97333E+24	kg
G =	6.673E-20	km³/kg*s²
μ Earth =	398600.4415	km³/s²
g =	9.80665	m/s
MSD (mean solar day) =	0.985647	
Earth axial tilt =	23.44241	deg

Comments

Distortion is not a worry due to limited size of target object.
Entire optical formulae for pointing out into space vice down toward earth -> use stop and stare method at each choosen distance in closing path.
Velocity formulae need to be corrected
Closing velocity will dictate shutter speed needed

Ref: SMAD p.287-91

Constellation Estimating:

Orbit Parameters

SC Altitude (Hsc) =	35785.37	km	given
SC Orbit Period (Psc) =	1436.32	min	(m/(Re+Hsc)3)1/2
Target's Altitude (Htar) =	35785.86	km	Assuming GEOSTA
Target's Obital Period (Ptar) =	1436.35	min	sqrt(m/(Re+Htar)3
SC ω (ωsc) =	0.0000729	Rad/sec	ωsc = (2*π)/Psc
Target ω (ωtar) =	0.0000729	Rad/sec	ωtar = (2*π)/Ptar
SC orbital Radius (Rsc) =	42169.11	km	(μEarth/ωsc2)1/3
Target orbital Radius (Rtar) =	42169.61	km	(μEarth/ωtar2)1/3
SC Velocity (Vsc) =	3.07448	km/sec	Vsc = ωsc*Rsc
Target's Velocity (Vtar) =	3.07446	km/sec	Vtar = ωtar*Rtar

Coverage and Access Factors

Closing Velocity (CV[km/sec]) =	0.00002	km/sec	Vtar-Vsc
Closing Velocity (CV) =	0.02	m/sec	CV[km/sec]*1000[m/km]
Closest Point of Approach (CPA) =	0.50	km	Htar-Hsc
Target's Orbital Circumference (Cirtar) =	264869.00	km	2*π*(Htar*6378.137)
Distance SC travels relative to GEOSTA orbit =	1.57	km/day	CV[km/sec]*60*60*24
Time for SC to traverse GEOSTA orbit =	168879.70	days	Cirtar/Distance SC travels
Time for SC to traverse GEOSTA orbit =	462.37	years	(Cirtar/Distance SC travels)*365.25

Coverage and Access Considerations

Desired Constellation re-visit rate =	30	days	Choosen
Planes of SC desired =	1	Planes	Majority of GEOSTA at i≈0°
Distance SC can travel during re-visit rate =	7969052.02	km	(Re-visit rate)* Vsc*60*60*24
Distance Targets travel during re-visit rate =	7969004.96	km	(Re-visit rate)* Vtar*60*60*24
Difference in distances traveled =	47.06	km	(SC travel)-(Tar travel)
Separation distance for SC =	94.12	km	2*SepDist
Number of SC needed =	2814.00		(Circumference of target's orbit)/(Sep distance) rounded up

Modeled in STK

Seeed number of SC =	2000.00	SC	Low estimation from cell C31.
STK optimized SC number through trial and error =	UNKNOWN	SC	Evenly spaced SC simulated in STK to give complete GEOSTA coverage in desired re-visit rate.
SC separation =	UNKNOWN	km	(2*π*(Hsc+6378.137))/14

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APPENDIX D. HALF-METER-CUBE STK SIMULATION COVERAGE REPORTS

Page 1

1:16:03 21 Aug 2007 1
Educational Use Only
Coverage Intervals

Coverage for HalfMeterSat101-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	15 Jul 2007 08:26:57.409	15 Jul 2007 16:59:39.491	30762.082	AMC-3
	2	17 Jul 2007 15:05:17.661	18 Jul 2007 00:26:43.566	33685.905	AMC-16
	3	17 Jul 2007 17:47:50.947	17 Jul 2007 17:54:56.487	425.540	AMC-2
	4	18 Jul 2007 00:12:07.536	18 Jul 2007 02:39:12.386	8824.850	AMC-2
	5	20 Jul 2007 03:53:07.186	20 Jul 2007 10:34:42.250	24095.064	AMC-9
	6	24 Jul 2007 19:34:41.755	25 Jul 2007 05:06:32.067	34310.312	AMC-5

Global Statistics

Min Duration	3	17 Jul 2007 17:47:50.947	17 Jul 2007 17:54:56.487	425.540	AMC-2
Max Duration	6	24 Jul 2007 19:34:41.755	25 Jul 2007 05:06:32.067	34310.312	AMC-5
Mean Duration				22017.292	
Total Duration				132103.753	

Page 1

1:17:01
Educational Use Only
Coverage Intervals

21 Aug 2007 1

Coverage for HalfMeterSat102-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	4 Jul 2007 04:23:43.723	4 Jul 2007 11:15:31.709	24707.986	AMC-6
	2	21 Jul 2007 04:41:15.377	21 Jul 2007 13:35:20.690	32045.313	PAS-9

Global Statistics

Min Duration	1	4 Jul 2007 04:23:43.723	4 Jul 2007 11:15:31.709	24707.986	AMC-6
Max Duration	2	21 Jul 2007 04:41:15.377	21 Jul 2007 13:35:20.690	32045.313	PAS-9
Mean Duration				28376.649	
Total Duration				56753.298	

Page 1

1:17:41 21 Aug 2007 1
Educational Use Only
Coverage Intervals

Coverage for HalfMeterSat103-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	17 Jul 2007 06:29:29.942	17 Jul 2007 16:54:16.002	37486.060	AMC-12
_903	2	20 Jul 2007 18:27:01.532	21 Jul 2007 02:00:14.784	27193.252	INTELSAT
_907	3	29 Jul 2007 00:17:00.309	29 Jul 2007 06:49:28.918	23548.609	INTELSAT

Global Statistics

Min Duration _907	3	29 Jul 2007 00:17:00.309	29 Jul 2007 06:49:28.918	23548.609	INTELSAT
Max Duration	1	17 Jul 2007 06:29:29.942	17 Jul 2007 16:54:16.002	37486.060	AMC-12
Mean Duration				29409.307	
Total Duration				88227.920	

1:30:41 21 Aug 2007 1
Educational Use Only
Coverage Intervals

Coverage for HalfMeterSat104-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_905	1	4 Jul 2007 01:57:31.479	4 Jul 2007 08:53:04.868	24933.389	INTELSAT
_603	2	9 Jul 2007 23:11:28.009	9 Jul 2007 23:20:28.772	540.764	INTELSAT
_901	3	11 Jul 2007 23:02:24.876	12 Jul 2007 05:05:07.351	21762.475	INTELSAT

Global Statistics

Min Duration _603	2	9 Jul 2007 23:11:28.009	9 Jul 2007 23:20:28.772	540.764	INTELSAT
Max Duration _905	1	4 Jul 2007 01:57:31.479	4 Jul 2007 08:53:04.868	24933.389	INTELSAT
Mean Duration				15745.543	
Total Duration				47236.628	

Page 1

21 Aug 2007 1

1:30:08
Educational Use Only
Coverage Intervals

Coverage for HalfMeterSat105-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_10-02	1	4 Jul 2007 02:02:47.543	4 Jul 2007 09:59:35.428	28607.885	INTELSAT
_7A	2	20 Jul 2007 05:39:07.732	20 Jul 2007 13:30:28.903	28281.171	HOT_BIRD
_8	3	20 Jul 2007 15:33:46.203	20 Jul 2007 22:48:38.016	26091.813	HOT_BIRD

Global Statistics

Min Duration _8	3	20 Jul 2007 15:33:46.203	20 Jul 2007 22:48:38.016	26091.813	HOT_BIRD
Max Duration _10-02	1	4 Jul 2007 02:02:47.543	4 Jul 2007 09:59:35.428	28607.885	INTELSAT
Mean Duration				27660.290	
Total Duration				82980.869	

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Coverage Intervals

Coverage for HalfMeterSat106-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
3A	1	7 Jul 2007 09:28:18.008	7 Jul 2007 20:37:29.681	40151.673	ARABSAT_
4B	2	7 Jul 2007 22:59:27.242	8 Jul 2007 05:54:21.879	24894.637	ARABSAT-
2C	3	8 Jul 2007 14:08:16.327	8 Jul 2007 21:45:01.320	27404.993	ARABSAT_
2B	4	13 Jul 2007 01:27:06.429	13 Jul 2007 08:47:05.891	26399.462	ARABSAT_
_802	5	15 Jul 2007 18:05:30.318	16 Jul 2007 01:05:15.832	25185.514	INTELSAT

Global Statistics

Min Duration 4B	2	7 Jul 2007 22:59:27.242	8 Jul 2007 05:54:21.879	24894.637	ARABSAT-
Max Duration 3A	1	7 Jul 2007 09:28:18.008	7 Jul 2007 20:37:29.681	40151.673	ARABSAT_
Mean Duration				28807.256	
Total Duration				144036.279	

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Coverage Intervals

Coverage for HalfMeterSat107-HalfMeterOpticalPL

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ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_706	1	8 Jul 2007 01:41:54.699	8 Jul 2007 09:55:17.078	29602.378	INTELSAT
_904	2	19 Jul 2007 01:26:40.815	19 Jul 2007 10:53:17.559	33996.744	INTELSAT
_902	3	21 Jul 2007 11:24:29.682	21 Jul 2007 19:03:38.398	27548.716	INTELSAT
_906	4	23 Jul 2007 21:26:43.963	24 Jul 2007 04:31:03.080	25459.116	INTELSAT
_704	5	26 Jul 2007 00:47:19.363	26 Jul 2007 11:08:06.161	37246.798	INTELSAT

Global Statistics

Min Duration _906	4	23 Jul 2007 21:26:43.963	24 Jul 2007 04:31:03.080	25459.116	INTELSAT
Max Duration _704	5	26 Jul 2007 00:47:19.363	26 Jul 2007 11:08:06.161	37246.798	INTELSAT
Mean Duration				30770.750	
Total Duration				153853.752	

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Coverage Intervals

Coverage for HalfMeterSat108-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
10	1	10 Jul 2007 01:24:32.956	10 Jul 2007 11:21:26.499	35813.543	TELSTAR_
R_1	2	23 Jul 2007 06:48:21.405	23 Jul 2007 14:50:43.661	28942.256	CHINASTA

Global Statistics

Min Duration R_1	2	23 Jul 2007 06:48:21.405	23 Jul 2007 14:50:43.661	28942.256	CHINASTA
Max Duration 10	1	10 Jul 2007 01:24:32.956	10 Jul 2007 11:21:26.499	35813.543	TELSTAR_
Mean Duration				32377.900	
Total Duration				64755.799	

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Coverage Intervals

Coverage for HalfMeterSat109-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
G_22A	1	5 Jul 2007 22:25:25.716	6 Jul 2007 07:07:19.092	31313.376	ZHONGXIN
G_22	2	6 Jul 2007 22:00:24.637	7 Jul 2007 06:18:20.809	29876.173	ZHONGXIN
G_20	3	13 Jul 2007 01:23:28.007	13 Jul 2007 10:28:15.150	32687.143	ZHONGXIN
1	4	22 Jul 2007 00:02:36.694	22 Jul 2007 09:37:11.865	34475.171	SINOSAT_

Global Statistics

Min Duration G_22	2	6 Jul 2007 22:00:24.637	7 Jul 2007 06:18:20.809	29876.173	ZHONGXIN
Max Duration 1	4	22 Jul 2007 00:02:36.694	22 Jul 2007 09:37:11.865	34475.171	SINOSAT_
Mean Duration				32087.965	
Total Duration				128351.862	

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Coverage Intervals

Coverage for HalfMeterSat110-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
3	1	10 Jul 2007 01:50:45.416	10 Jul 2007 07:35:51.235	20705.819	SINOSAT_
	2	21 Jul 2007 07:08:18.872	21 Jul 2007 15:07:12.957	28734.084	APSTAR_6
	3	25 Jul 2007 20:28:28.360	26 Jul 2007 06:31:28.482	36180.122	APSTAR_5

Global Statistics

Min Duration 3	1	10 Jul 2007 01:50:45.416	10 Jul 2007 07:35:51.235	20705.819	SINOSAT_
Max Duration	3	25 Jul 2007 20:28:28.360	26 Jul 2007 06:31:28.482	36180.122	APSTAR_5
Mean Duration				28540.008	
Total Duration				85620.025	

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Coverage for HalfMeterSat111-HalfMeterOpticalPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	26 Jul 2007 01:32:51.210	26 Jul 2007 09:56:37.738	30226.528	SUPERBIRD_4

Global Statistics

Min Duration	1	26 Jul 2007 01:32:51.210	26 Jul 2007 09:56:37.738	30226.528	SUPERBIR D_4
Max Duration	1	26 Jul 2007 01:32:51.210	26 Jul 2007 09:56:37.738	30226.528	SUPERBIR D_4
Mean Duration				30226.528	
Total Duration				30226.528	

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Coverage for HalfMeterSat112-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	9 Jul 2007 18:10:23.347	10 Jul 2007 03:18:41.648	32898.301	AMC-23
	2	22 Jul 2007 14:39:04.915	22 Jul 2007 23:05:53.706	30408.791	NSS-5

Global Statistics

Min Duration	2	22 Jul 2007 14:39:04.915	22 Jul 2007 23:05:53.706	30408.791	NSS-5
Max Duration	1	9 Jul 2007 18:10:23.347	10 Jul 2007 03:18:41.648	32898.301	AMC-23
Mean Duration				31653.546	
Total Duration				63307.092	

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Coverage Intervals

Coverage for HalfMeterSat113-HalfMeterOpticalPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	28 Jul 2007 11:24:22.987	28 Jul 2007 19:04:22.468	27599.481	ECHOSTAR_2

Global Statistics

Min Duration	1	28 Jul 2007 11:24:22.987	28 Jul 2007 19:04:22.468	27599.481	ECHOSTAR_2
Max Duration	1	28 Jul 2007 11:24:22.987	28 Jul 2007 19:04:22.468	27599.481	ECHOSTAR_2
Mean Duration				27599.481	
Total Duration				27599.481	

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Coverage Intervals

Coverage for HalfMeterSat114-HalfMeterOpticalPL

Asset Full Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_2	1	1 Jul 2007 12:00:00.000	1 Jul 2007 13:46:46.616	6406.616	ECHOSTAR
	2	11 Jul 2007 04:35:58.199	11 Jul 2007 12:14:04.442	27486.242	AMC-8
	3	13 Jul 2007 13:49:24.206	13 Jul 2007 22:23:32.953	30848.747	AMC-7
	4	15 Jul 2007 21:32:59.851	16 Jul 2007 05:47:56.271	29696.421	AMC-10
	5	20 Jul 2007 06:51:05.837	20 Jul 2007 14:49:45.100	28719.263	AMC-11

Global Statistics

Min Duration _2	1	1 Jul 2007 12:00:00.000	1 Jul 2007 13:46:46.616	6406.616	ECHOSTAR
Max Duration	3	13 Jul 2007 13:49:24.206	13 Jul 2007 22:23:32.953	30848.747	AMC-7
Mean Duration				24631.458	
Total Duration				123157.289	

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Coverage Intervals

Coverage for HalfMeterSat115-HalfMeterOpticalPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	22 Jul 2007 10:41:34.340	22 Jul 2007 19:37:46.363	32172.023	AMC-15
	2	22 Jul 2007 19:37:46.574	22 Jul 2007 23:13:48.412	12961.838	AMC_18
	3	24 Jul 2007 19:57:22.853	25 Jul 2007 04:46:34.491	31751.638	AMC-1
	4	27 Jul 2007 04:23:47.009	27 Jul 2007 12:06:47.050	27780.041	AMC-4

Global Statistics

Min Duration	2	22 Jul 2007 19:37:46.574	22 Jul 2007 23:13:48.412	12961.838	AMC_18
Max Duration	1	22 Jul 2007 10:41:34.340	22 Jul 2007 19:37:46.363	32172.023	AMC-15
Mean Duration				26166.385	
Total Duration				104665.541	

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APPENDIX E. 5U-CUBESAT STK SIMULATION COVERAGE REPORTS

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Coverage Intervals

Coverage for Opt_5UCubeSat101-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	18 Jul 2007 11:27:53.625	18 Jul 2007 19:10:17.781	27744.156	AMC-3
	2	23 Jul 2007 08:09:54.039	23 Jul 2007 13:26:55.468	19021.429	AMC-16
	3	23 Jul 2007 13:26:55.595	23 Jul 2007 20:18:15.231	24679.636	AMC-2
	4	28 Jul 2007 11:15:09.952	28 Jul 2007 18:03:39.336	24509.384	AMC-9

Global Statistics

Min Duration	2	23 Jul 2007 08:09:54.039	23 Jul 2007 13:26:55.468	19021.429	AMC-16
Max Duration	1	18 Jul 2007 11:27:53.625	18 Jul 2007 19:10:17.781	27744.156	AMC-3
Mean Duration				23988.651	
Total Duration				95954.605	

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Coverage Intervals

Coverage for Opt_5UCubeSat102-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	10 Jul 2007 20:15:54.518	11 Jul 2007 04:51:33.621	30939.103	AMC-5
	2	28 Jul 2007 12:43:05.138	28 Jul 2007 22:32:54.344	35389.206	AMC-6

Global Statistics

Min Duration	1	10 Jul 2007 20:15:54.518	11 Jul 2007 04:51:33.621	30939.103	AMC-5
Max Duration	2	28 Jul 2007 12:43:05.138	28 Jul 2007 22:32:54.344	35389.206	AMC-6
Mean Duration				33164.155	
Total Duration				66328.309	

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Coverage for Opt_5UCubeSat104-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	8 Jul 2007 12:34:12.492	8 Jul 2007 17:17:51.625	17019.133	PAS-9

Global Statistics

Min Duration	1	8 Jul 2007 12:34:12.492	8 Jul 2007 17:17:51.625	17019.133	PAS-9
Max Duration	1	8 Jul 2007 12:34:12.492	8 Jul 2007 17:17:51.625	17019.133	PAS-9
Mean Duration				17019.133	
Total Duration				17019.133	

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Coverage for Opt_5UCubeSat106-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	13 Jul 2007 09:00:11.774	13 Jul 2007 16:27:44.078	26852.304	INTELSAT_903

Global Statistics

Min Duration	1	13 Jul 2007 09:00:11.774	13 Jul 2007 16:27:44.078	26852.304	INTELSAT_903
Max Duration	1	13 Jul 2007 09:00:11.774	13 Jul 2007 16:27:44.078	26852.304	INTELSAT_903
Mean Duration				26852.304	
Total Duration				26852.304	

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Coverage Intervals

Coverage for Opt_5UCubeSat107-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_907	1	3 Jul 2007 08:25:33.377	3 Jul 2007 16:55:32.286	30598.909	INTELSAT
_905	2	11 Jul 2007 04:18:44.353	11 Jul 2007 07:32:22.324	11617.971	INTELSAT
_603	3	22 Jul 2007 22:23:15.361	22 Jul 2007 22:26:21.729	186.367	INTELSAT
_901	4	27 Jul 2007 07:11:53.596	27 Jul 2007 14:26:44.013	26090.417	INTELSAT

Global Statistics

Min Duration _603	3	22 Jul 2007 22:23:15.361	22 Jul 2007 22:26:21.729	186.367	INTELSAT
Max Duration _907	1	3 Jul 2007 08:25:33.377	3 Jul 2007 16:55:32.286	30598.909	INTELSAT
Mean Duration				17123.416	
Total Duration				68493.664	

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Coverage for Opt_5UCubeSat109-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	15 Jul 2007 22:02:24.057	16 Jul 2007 02:51:50.302	17366.245	INTELSAT_10-02

Global Statistics

Min Duration	1	15 Jul 2007 22:02:24.057	16 Jul 2007 02:51:50.302	17366.245	INTELSAT_10-02
Max Duration	1	15 Jul 2007 22:02:24.057	16 Jul 2007 02:51:50.302	17366.245	INTELSAT_10-02
Mean Duration				17366.245	
Total Duration				17366.245	

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Coverage Intervals

Coverage for Opt_5UCubeSat110-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_7A	1	23 Jul 2007 04:48:17.062	23 Jul 2007 09:19:34.468	16277.405	HOT_BIRD
_8	2	23 Jul 2007 20:31:34.471	24 Jul 2007 00:23:34.046	13919.574	HOT_BIRD

Global Statistics

Min Duration _8	2	23 Jul 2007 20:31:34.471	24 Jul 2007 00:23:34.046	13919.574	HOT_BIRD
Max Duration _7A	1	23 Jul 2007 04:48:17.062	23 Jul 2007 09:19:34.468	16277.405	HOT_BIRD
Mean Duration				15098.490	
Total Duration				30196.980	

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Coverage Intervals

Coverage for Opt_5UCubeSat111-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	27 Jul 2007 20:32:51.037	27 Jul 2007 22:06:01.706	5590.670	ARABSAT_3A

Global Statistics

Min Duration	1	27 Jul 2007 20:32:51.037	27 Jul 2007 22:06:01.706	5590.670	ARABSAT_ 3A
Max Duration	1	27 Jul 2007 20:32:51.037	27 Jul 2007 22:06:01.706	5590.670	ARABSAT_ 3A
Mean Duration				5590.670	
Total Duration				5590.670	

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Coverage Intervals

Coverage for Opt_5UCubeSat112-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
3A	1	2 Jul 2007 07:48:31.088	2 Jul 2007 19:17:45.067	41353.979	ARABSAT_
4B	2	3 Jul 2007 13:15:26.408	3 Jul 2007 16:28:15.030	11568.622	ARABSAT-
2C	3	4 Jul 2007 21:27:22.935	4 Jul 2007 23:40:45.324	8002.388	ARABSAT_
2B	4	13 Jul 2007 15:52:20.741	13 Jul 2007 20:28:15.361	16554.620	ARABSAT_
_802	5	19 Jul 2007 00:45:23.837	19 Jul 2007 02:26:50.823	6086.986	INTELSAT

Global Statistics

Min Duration _802	5	19 Jul 2007 00:45:23.837	19 Jul 2007 02:26:50.823	6086.986	INTELSAT
Max Duration 3A	1	2 Jul 2007 07:48:31.088	2 Jul 2007 19:17:45.067	41353.979	ARABSAT_
Mean Duration				16713.319	
Total Duration				83566.595	

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Coverage for Opt_5UCubeSat114-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	8 Jul 2007 15:56:58.033	8 Jul 2007 19:07:36.335	11438.303	INTELSAT_706

Global Statistics

Min Duration	1	8 Jul 2007 15:56:58.033	8 Jul 2007 19:07:36.335	11438.303	INTELSAT_706
Max Duration	1	8 Jul 2007 15:56:58.033	8 Jul 2007 19:07:36.335	11438.303	INTELSAT_706
Mean Duration				11438.303	
Total Duration				11438.303	

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Coverage Intervals

Coverage for Opt_5UCubeSat115-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
_904	1	5 Jul 2007 12:51:35.700	5 Jul 2007 17:42:19.226	17443.526	INTELSAT
_902	2	10 Jul 2007 04:54:25.540	10 Jul 2007 13:32:50.703	31105.163	INTELSAT
_906	3	15 Jul 2007 01:20:05.972	15 Jul 2007 06:02:57.188	16971.216	INTELSAT
_704	4	19 Jul 2007 16:20:36.458	19 Jul 2007 18:53:07.242	9150.784	INTELSAT

Global Statistics

Min Duration _704	4	19 Jul 2007 16:20:36.458	19 Jul 2007 18:53:07.242	9150.784	INTELSAT
Max Duration _902	2	10 Jul 2007 04:54:25.540	10 Jul 2007 13:32:50.703	31105.163	INTELSAT
Mean Duration				18667.672	
Total Duration				74670.689	

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Coverage Intervals

Coverage for Opt_5UCubeSat116-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	18 Jul 2007 03:03:27.172	18 Jul 2007 14:37:05.864	41618.693	TELSTAR_10

Global Statistics

Min Duration	1	18 Jul 2007 03:03:27.172	18 Jul 2007 14:37:05.864	41618.693	TELSTAR_ 10
Max Duration	1	18 Jul 2007 03:03:27.172	18 Jul 2007 14:37:05.864	41618.693	TELSTAR_ 10
Mean Duration				41618.693	
Total Duration				41618.693	

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Coverage Intervals

Coverage for Opt_5UCubeSat117-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	18 Jul 2007 19:48:08.386	19 Jul 2007 03:03:14.046	26105.660	CHINASTAR_1

Global Statistics

Min Duration R_1	1	18 Jul 2007 19:48:08.386	19 Jul 2007 03:03:14.046	26105.660	CHINASTA
Max Duration R_1	1	18 Jul 2007 19:48:08.386	19 Jul 2007 03:03:14.046	26105.660	CHINASTA
Mean Duration				26105.660	
Total Duration				26105.660	

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Coverage Intervals

Coverage for Opt_5UCubeSat118-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
G_22A	1	16 Jul 2007 02:04:33.858	16 Jul 2007 05:21:52.583	11838.725	ZHONGXIN
G_22	2	17 Jul 2007 14:09:07.427	17 Jul 2007 17:05:17.590	10570.164	ZHONGXIN

Global Statistics

Min Duration G_22	2	17 Jul 2007 14:09:07.427	17 Jul 2007 17:05:17.590	10570.164	ZHONGXIN
Max Duration G_22A	1	16 Jul 2007 02:04:33.858	16 Jul 2007 05:21:52.583	11838.725	ZHONGXIN
Mean Duration				11204.444	
Total Duration				22408.889	

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Coverage Intervals

Coverage for Opt_5UCubeSat119-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	1 Jul 2007 18:23:15.065	1 Jul 2007 23:24:38.421	18083.356	ZHONGXING_20

Global Statistics

Min Duration	1	1 Jul 2007 18:23:15.065	1 Jul 2007 23:24:38.421	18083.356	ZHONGXING_20
Max Duration	1	1 Jul 2007 18:23:15.065	1 Jul 2007 23:24:38.421	18083.356	ZHONGXING_20
Mean Duration				18083.356	
Total Duration				18083.356	

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Coverage Intervals

Coverage for Opt_5UCubeSat120-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	30 Jul 2007 10:04:06.468	30 Jul 2007 12:00:00.000	6953.532	SINOSAT_3

Global Statistics

Min Duration	1	30 Jul 2007 10:04:06.468	30 Jul 2007 12:00:00.000	6953.532	SINOSAT_ 3
Max Duration	1	30 Jul 2007 10:04:06.468	30 Jul 2007 12:00:00.000	6953.532	SINOSAT_ 3
Mean Duration				6953.532	
Total Duration				6953.532	

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Coverage Intervals

Coverage for Opt_5UCubeSat121-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	25 Jul 2007 15:23:09.003	25 Jul 2007 16:32:17.963	4148.959	APSTAR_6

Global Statistics

Min Duration	1	25 Jul 2007 15:23:09.003	25 Jul 2007 16:32:17.963	4148.959	APSTAR_6
Max Duration	1	25 Jul 2007 15:23:09.003	25 Jul 2007 16:32:17.963	4148.959	APSTAR_6
Mean Duration				4148.959	
Total Duration				4148.959	

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Coverage for Opt_5UCubeSat124-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	14 Jul 2007 02:47:06.045	14 Jul 2007 09:38:40.440	24694.395	SUPERBIRD_4

Global Statistics

Min Duration	1	14 Jul 2007 02:47:06.045	14 Jul 2007 09:38:40.440	24694.395	SUPERBIR D_4
Max Duration	1	14 Jul 2007 02:47:06.045	14 Jul 2007 09:38:40.440	24694.395	SUPERBIR D_4
Mean Duration				24694.395	
Total Duration				24694.395	

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Coverage Intervals

Coverage for Opt_5UCubeSat125-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	12 Jul 2007 01:41:05.145	12 Jul 2007 09:25:24.048	27858.903	AMC-23

Global Statistics

Min Duration	1	12 Jul 2007 01:41:05.145	12 Jul 2007 09:25:24.048	27858.903	AMC-23
Max Duration	1	12 Jul 2007 01:41:05.145	12 Jul 2007 09:25:24.048	27858.903	AMC-23
Mean Duration				27858.903	
Total Duration				27858.903	

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Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat126-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	12 Jul 2007 19:06:08.449	12 Jul 2007 23:30:04.112	15835.663	NSS-5

Global Statistics

Min Duration	1	12 Jul 2007 19:06:08.449	12 Jul 2007 23:30:04.112	15835.663	NSS-5
Max Duration	1	12 Jul 2007 19:06:08.449	12 Jul 2007 23:30:04.112	15835.663	NSS-5
Mean Duration				15835.663	
Total Duration				15835.663	

Page 1
13 Sep 2007 10:50:36
Educational Use Only
Coverage Intervals

No Accesses Found

0:51:13 13 Sep 2007 1
Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat128-5UCubeSatOptPL

Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Full Name
1	29 Jul 2007 16:46:23.187	30 Jul 2007 01:03:53.105	29849.918	ECHOSTAR_2

Global Statistics

Min Duration	1	29 Jul 2007 16:46:23.187	30 Jul 2007 01:03:53.105	29849.918	ECHOSTAR_2
Max Duration	1	29 Jul 2007 16:46:23.187	30 Jul 2007 01:03:53.105	29849.918	ECHOSTAR_2
Mean Duration				29849.918	
Total Duration				29849.918	

0:51:39 13 Sep 2007 1
Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat129-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	25 Jul 2007 01:27:13.091	25 Jul 2007 06:51:02.543	19429.452	AMC-8
	2	29 Jul 2007 21:23:25.329	30 Jul 2007 03:43:35.333	22810.004	AMC-7

Global Statistics

Min Duration	1	25 Jul 2007 01:27:13.091	25 Jul 2007 06:51:02.543	19429.452	AMC-8
Max Duration	2	29 Jul 2007 21:23:25.329	30 Jul 2007 03:43:35.333	22810.004	AMC-7
Mean Duration				21119.728	
Total Duration				42239.456	

0:52:06 13 Sep 2007 1
Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat130-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	4 Jul 2007 19:31:47.314	5 Jul 2007 02:33:37.902	25310.588	AMC-7
	2	18 Jul 2007 03:50:57.956	18 Jul 2007 07:08:11.591	11833.635	AMC-11

Global Statistics

Min Duration	2	18 Jul 2007 03:50:57.956	18 Jul 2007 07:08:11.591	11833.635	AMC-11
Max Duration	1	4 Jul 2007 19:31:47.314	5 Jul 2007 02:33:37.902	25310.588	AMC-7
Mean Duration				18572.111	
Total Duration				37144.222	

Page 1
13 Sep 2007 10:52:31
Educational Use Only
Coverage Intervals

No Accesses Found

Page 1

0:53:00 13 Sep 2007 1
Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat132-5UCubeSatOptPL

ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	27 Jul 2007 22:47:48.040	28 Jul 2007 04:46:31.665	21523.624	AMC-15
	2	28 Jul 2007 05:29:20.305	28 Jul 2007 09:20:44.576	13884.271	AMC_18

Global Statistics

Min Duration	2	28 Jul 2007 05:29:20.305	28 Jul 2007 09:20:44.576	13884.271	AMC_18
Max Duration	1	27 Jul 2007 22:47:48.040	28 Jul 2007 04:46:31.665	21523.624	AMC-15
Mean Duration				17703.948	
Total Duration				35407.895	

0:53:24 13 Sep 2007 1
Educational Use Only
Coverage Intervals

Coverage for Opt_5UCubeSat133-5UCubeSatOptPL
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ll Name	Access	Access Start (UTCG)	Access End (UTCG)	Duration (sec)	Asset Fu
	1	1 Jul 2007 14:55:24.955	1 Jul 2007 17:48:43.859	10398.904	AMC_18
	2	6 Jul 2007 01:54:28.076	6 Jul 2007 05:36:02.193	13294.118	AMC-1
	3	11 Jul 2007 00:02:25.345	11 Jul 2007 04:54:10.071	17504.726	AMC-4

Global Statistics

Min Duration	1	1 Jul 2007 14:55:24.955	1 Jul 2007 17:48:43.859	10398.904	AMC_18
Max Duration	3	11 Jul 2007 00:02:25.345	11 Jul 2007 04:54:10.071	17504.726	AMC-4
Mean Duration				13732.583	
Total Duration				41197.748	

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APPENDIX F. EXCEL WORKBOOK OF STK SIMULATION COVERAGE REPORTS FOR HALF-METER-CUBE-SATELLITE AND 5U-CUBESAT CONSTELLAIONS

Half-Meter-Cube Coverage Rpt

Half-MeterSat_101CovRpt

Access	Access Start (UTCG)	Access End (UTCG)	Duration [s]	Asset Full Name
1	26:57.4	59:39.5	30762.082	AMC-3
2	05:17.7	26:43.6	33685.905	AMC-16
3	47:50.9	54:56.5	425.54	AMC-2
4	12:07.5	39:12.4	8824.85	AMC-2
5	53:07.2	34:42.2	24095.064	AMC-9
6	34:41.8	06:32.1	34310.312	AMC-5

Half-MeterSat_102CovRpt

1	23:43.7	15:31.7	24707.986	AMC-6
2	41:15.4	35:20.7	32045.313	PAS-9

Half-MeterSat_103CovRpt

1	29:29.9	54:16.0	37486.06	AMC-12
2	27:01.5	00:14.8	27193.252	INTELSAT_903
3	17:00.3	49:28.9	23548.609	INTELSAT_907

Half-MeterSat_104CovRpt

1	57:31.5	53:04.9	24933.389	INTELSAT_905
2	11:28.0	20:28.8	540.764	INTELSAT_603
3	02:24.9	05:07.4	21762.475	INTELSAT_901

Half-MeterSat_105CovRpt

1	02:47.5	59:35.4	28607.885	INTELSAT_10-02
2	39:07.7	30:28.9	28281.171	HOT_BIRD_7A
3	33:46.2	48:38.0	26091.813	HOT_BIRD_8

Half-MeterSat_106CovRpt

1	28:18.0	37:29.7	40151.673	ARABSAT_3A
2	59:27.2	54:21.9	24894.637	ARABSAT-4B
3	08:16.3	45:01.3	27404.993	ARABSAT_2C
4	27:06.4	47:05.9	26399.462	ARABSAT_2B
5	05:30.3	05:15.8	25185.514	INTELSAT_802

Half-MeterSat_107CovRpt

1	41:54.7	55:17.1	29602.378	INTELSAT_706
2	26:40.8	53:17.6	33996.744	INTELSAT_904
3	24:29.7	03:38.4	27548.716	INTELSAT_902
4	26:44.0	31:03.1	25459.116	INTELSAT_906
5	47:19.4	08:06.2	37246.798	INTELSAT_704

Half-MeterSat_108CovRpt

Access	Access Start	Access End	Duration	Asset Full Name
1	24:33.0	21:26.5	35813.5	TELSTAR_10
2	48:21.4	50:43.7	28942.3	CHINASTAR_1

Half-MeterSat_109CovRpt

1	25:25.7	07:19.1	31313.4	ZHONGXING_22A
2	00:24.6	18:20.8	29876.2	ZHONGXING_22
3	23:28.0	28:15.1	32687.1	ZHONGXING_20
4	02:36.7	37:11.9	34475.2	SINOSAT_1

Half-MeterSat_110CovRpt

1	50:45.4	35:51.2	20705.8	SINOSAT_3
2	08:18.9	07:13.0	28734.1	APSTAR_6
3	28:28.4	31:28.5	36180.1	APSTAR_5

Half-MeterSat_111CovRpt

1	32:51.2	56:37.7	30226.5	SUPERBIRD_4

Half-MeterSat_112CovRpt

1	10:23.3	18:41.6	32898.3	AMC-23
2	39:04.9	05:53.7	30408.8	NSS-5

Half-MeterSat_113CovRpt

1	24:23.0	04:22.5	27599.5	ECHOSTAR_2

Half-MeterSat_114CovRpt

1	00:00.0	46:46.6	6406.62	ECHOSTAR_2
2	35:58.2	14:04.4	27486.2	AMC-8
3	49:24.2	23:33.0	30848.7	AMC-7
4	32:59.9	47:56.3	29696.4	AMC-10
5	51:05.8	49:45.1	28719.3	AMC-11

Half-MeterSat_115CovRpt

1	41:34.3	37:46.4	32172	AMC-15
2	37:46.6	13:48.4	12961.8	AMC_18
3	57:22.9	46:34.5	31751.6	AMC-1
4	23:47.0	06:47.0	27780	AMC-4

Half-Meter-Cube Cstl Cov Rpt

Asset Full Name	Duration [sec]	Observing Half-Meter (HMC) Sat
AMC_18	12961.838	115
AMC-1	31751.638	115
AMC-10	29696.421	114
AMC-11	28719.263	114
AMC-12	37486.06	103
AMC-15	32172.023	115
AMC-16	33685.905	101
AMC-2	425.54	101
AMC-2	8824.85	101
AMC-23	32898.301	112
AMC-3	30762.082	101
AMC-4	27780.041	115
AMC-5	34310.312	101
AMC-6	24707.986	102
AMC-7	30848.747	114
AMC-8	27486.242	114
AMC-9	24095.064	101
APSTAR_5	36180.122	110
APSTAR_6	28734.084	110
ARABSAT_2B	26399.462	106
ARABSAT_2C	27404.993	106
ARABSAT_3A	40151.673	106
ARABSAT-4B	24894.637	106
CHINASTAR_1	28942.256	108
ECHOSTAR_2	27599.481	113
ECHOSTAR_2	6406.616	114
HOT_BIRD_7A	28281.171	105
HOT_BIRD_8	26091.813	105
INTELSAT_10-02	28607.885	105
INTELSAT_603	540.764	104
INTELSAT_704	37246.798	107
INTELSAT_706	29602.378	107
INTELSAT_802	25185.514	106
INTELSAT_901	21762.475	104
INTELSAT_902	27548.716	107
INTELSAT_903	27193.252	103
INTELSAT_904	33996.744	107
INTELSAT_905	24933.389	104
INTELSAT_906	25459.116	107
INTELSAT_907	23548.609	103
NSS-5	30408.791	112
PAS-9	32045.313	102
SINOSAT_1	34475.171	109
SINOSAT_3	20705.819	110
SUPERBIRD_4	30226.528	111
TELSTAR_10	35813.543	108
ZHONGXING_20	32687.143	109
ZHONGXING_22	29876.173	109
ZHONGXING_22A	31313.376	109

5UCubeSat_101CovRpt

Access	Access Start (UTCG)	Access End (UTCG)	Duration [s]	Asset Full Name
1	27:53.6	10:17.8	27744.156	AMC-3
2	09:54.0	26:55.5	19021.429	AMC-16
3	26:55.6	18:15.2	24679.635	AMC-2
4	15:10.0	03:39.3	24509.384	AMC-9

5UCubeSat_102CovRpt

1	15:54.5	51:33.6	30939.101	AMC-5
2	43:05.1	32:54.3	35389.206	AMC-6

5UCubeSat_103CovRpt

No access during time period of simulation

5UCubeSat_104CovRpt

1	34:12.5	17:51.6	17019.133	PAS-9

5UCubeSat_105CovRpt

No access during time period of simulation

5UCubeSat_106CovRpt

1	00:11.8	27:44.1	26852.304	INTELSAT_903

5UCubeSat_107CovRpt

1	25:33.4	55:32.3	30598.913	INTELSAT_907
2	18:44.4	32:22.3	11617.981	INTELSAT_905
3	23:15.4	26:21.7	186.367	INTELSAT_603
4	11:53.6	26:44.0	26090.417	INTELSAT_901

5UCubeSat_108CovRpt

No access during time period of simulation

5UCubeSat_109CovRpt

1	02:24.1	51:50.3	17366.245	INTELSAT_10-02

5UCubeSat_118CovRpt

1	04:33.9	21:52.6	11838.725	ZHONGXING_22A
2	09:07.4	05:17.6	10570.169	ZHONGXING_22

5UCubeSat_110CovRpt

Access	Access Start	Access End	Duration	Asset Full Name
1	48:17.1	19:34.5	16277.4	HOT_BIRD_7A
2	31:34.5	23:34.0	13919.6	HOT_BIRD_8

5UCubeSat_111CovRpt

1	32:51.0	06:01.7	5590.67	ARABSAT_3A

5UCubeSat_112CovRpt

1	48:31.1	17:45.1	41354	ARABSAT_3A
2	15:26.4	28:15.0	11568.6	ARABSAT-4B
3	27:22.9	40:45.3	8002.39	ARABSAT_2C
4	52:20.7	28:15.4	16554.6	ARABSAT_2B
5	45:23.8	26:50.8	6086.99	INTELSAT_802

5UCubeSat_113CovRpt

No access during time period of simulation

5UCubeSat_114CovRpt

1	56:58.0	07:36.3	11438.3	INTELSAT_706

5UCubeSat_115CovRpt

1	51:35.7	42:19.2	17443.5	INTELSAT_904
2	54:25.5	32:50.7	31105.2	INTELSAT_902
3	20:06.0	02:57.2	16971.2	INTELSAT_906
4	20:36.5	53:07.2	9150.78	INTELSAT_704

5UCubeSat_116CovRpt

1	03:27.2	37:05.9	41618.7	TELSTAR_10

5UCubeSat_117CovRpt

1	48:08.4	03:14.0	26105.7	CHINASTAR_1

5UCubeSat_119CovRpt

Access	Access Start (UTCG)	Access End (UTCG)	Duration [s]	Asset Full Name
1	23:15.1	24:38.4	18083.356	ZHONGXING_20

5UCubeSat_120CovRpt

1	04:06.5	00:00.0	6953.532	SINOSAT_3

5UCubeSat_121CovRpt

1	23:09.0	32:18.0	4148.959	APSTAR_6

5UCubeSat_122CovRpt

No access during time period of simulation

5UCubeSat_123CovRpt

No access during time period of simulation

5UCubeSat_124CovRpt

1	47:06.0	38:40.4	24694.395	SUPERBIRD_4

5UCubeSat_125CovRpt

1	41:05.1	25:24.0	27858.903	AMC-23

5UCubeSat_126CovRpt

1	06:08.4	30:04.1	15835.663	NSS-5

5UCubeSat_127CovRpt

No access during time period of simulation

5UCubeSat_128CovRpt

Access	Access Start	Access End	Duration	Asset Full Name
1	46:23.2	03:53.1	29849.9	ECHOSTAR_2

5UCubeSat_129CovRpt

1	27:13.1	51:02.5	19429.5	AMC-8
2	23:25.3	43:35.3	22810	AMC-7

5UCubeSat_130CovRpt

1	31:47.3	33:37.9	25310.6	AMC-7
2	50:58.0	08:11.6	11833.6	AMC-11

5UCubeSat_131CovRpt

No access during time period of simulation

5UCubeSat_132CovRpt

1	47:48.0	46:31.7	21523.6	AMC-15
2	29:20.3	20:44.6	13884.3	AMC_18

5UCubeSat_133CovRpt

1	55:25.0	48:43.9	10398.9	AMC_18
2	54:28.1	36:02.2	13294.1	AMC-1
3	02:25.3	54:10.1	17504.7	AMC-4

5U Cstl Coverage Rpt

Asset Full Name	Duration [s]	Observing Half-Meter Sat
AMC_18	13884.268	132
AMC_18	10398.903	133
AMC-1	13294.118	133
AMC-11	11833.635	130
AMC-15	21523.624	132
AMC-16	19021.429	101
AMC-2	24679.635	101
AMC-23	27858.903	125
AMC-3	27744.156	101
AMC-4	17504.723	133
AMC-5	30939.101	102
AMC-6	35389.206	102
AMC-7	22810.004	129
AMC-7	25310.588	130
AMC-8	19429.452	129
AMC-9	24509.384	101
APSTAR_6	4148.959	121
ARABSAT_2B	16554.62	112
ARABSAT_2C	8002.388	112
ARABSAT_3A	5590.67	111
ARABSAT_3A	41353.979	112
ARABSAT-4B	11568.622	112
CHINASTAR_1	26105.66	117
ECHOSTAR_2	29849.914	128
HOT_BIRD_7A	16277.405	110
HOT_BIRD_8	13919.579	110
INTELSAT_10-02	17366.245	109
INTELSAT_603	186.367	107
INTELSAT_704	9150.78	115
INTELSAT_706	11438.305	114
INTELSAT_802	6086.986	112
INTELSAT_901	26090.417	107
INTELSAT_902	31105.176	115
INTELSAT_903	26852.304	106
INTELSAT_904	17443.526	115
INTELSAT_905	11617.981	107
INTELSAT_906	16971.207	115
INTELSAT_907	30598.913	107
NSS-5	15835.663	126
PAS-9	17019.133	104
SINOSAT_3	6953.532	120
SUPERBIRD_4	24694.395	124
TELSTAR_10	41618.693	116
ZHONGXING_20	18083.356	119
ZHONGXING_22	10570.169	118
ZHONGXING_22A	11838.725	118

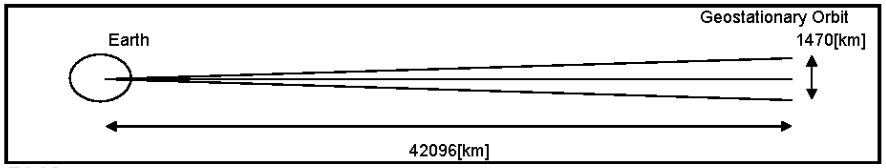
Asset Full Name	Duration [s]	Obs'ing HMC Sat (w/15SC)
AMC_18	12961.838	115
AMC-1	31751.638	115
AMC-10	29696.421	114
AMC-11	28719.263	114
AMC-12	37486.06	103
AMC-15	32172.023	115
AMC-16	33685.905	101
AMC-2	425.54	101
AMC-2	8824.85	101
AMC-23	32898.301	112
AMC-3	30762.082	101
AMC-4	27780.041	115
AMC-5	34310.312	101
AMC-6	24707.986	102
AMC-7	30848.747	114
AMC-8	27486.242	114
AMC-9	24095.064	101
APSTAR_5	36180.122	110
APSTAR_6	28734.084	110
ARABSAT_2B	26399.462	106
ARABSAT_2C	27404.993	106
ARABSAT_3A	40151.673	106
ARABSAT-4B	24894.637	106
CHINASTAR_1	28942.256	108
ECHOSTAR_2	27599.481	113
ECHOSTAR_2	6406.616	114
HOT_BIRD_7A	28281.171	105
HOT_BIRD_8	26091.813	105
INTELSAT_10-02	28607.885	105
INTELSAT_603	540.764	104
INTELSAT_704	37246.798	107
INTELSAT_706	29602.378	107
INTELSAT_802	25185.514	106
INTELSAT_901	21762.475	104
INTELSAT_902	27548.716	107
INTELSAT_903	27193.252	103
INTELSAT_904	33996.744	107
INTELSAT_905	24933.389	104
INTELSAT_906	25459.116	107
INTELSAT_907	23548.609	103
NSS-5	30408.791	112
PAS-9	32045.313	102
SINOSAT_1	34475.171	109
SINOSAT_3	20705.819	110
SUPERBIRD_4	30226.528	111
TELSTAR_10	35813.543	108
ZHONGXING_20	32687.143	109
ZHONGXING_22	29876.173	109
ZHONGXING_22A	31313.376	109

Asset Full Name	Duration [s]	Obs'ing 5UCubeSat (w/33SC)
AMC_18	13884.268	132
AMC_18	10398.903	133
AMC-1	13294.118	133
AMC-11	11833.635	130
AMC-15	21523.624	132
AMC-16	19021.429	101
AMC-2	24679.635	101
AMC-23	27858.903	125
AMC-3	27744.156	101
AMC-4	17504.723	133
AMC-5	30939.101	102
AMC-6	35389.206	102
AMC-7	22810.004	129
AMC-7	25310.588	130
AMC-8	19429.452	129
AMC-9	24509.384	101
APSTAR_6	4148.959	121
ARABSAT_2B	16554.62	112
ARABSAT_2C	8002.388	112
ARABSAT_3A	5590.67	111
ARABSAT_3A	41353.979	112
ARABSAT-4B	11568.622	112
CHINASTAR_1	26105.66	117
ECHOSTAR_2	29849.914	128
HOT_BIRD_7A	16277.405	110
HOT_BIRD_8	13919.579	110
INTELSAT_10-02	17366.245	109
INTELSAT_603	186.367	107
INTELSAT_704	9150.78	115
INTELSAT_706	11438.305	114
INTELSAT_802	6086.986	112
INTELSAT_901	26090.417	107
INTELSAT_902	31105.176	115
INTELSAT_903	26852.304	106
INTELSAT_904	17443.526	115
INTELSAT_905	11617.981	107
INTELSAT_906	16971.207	115
INTELSAT_907	30598.913	107
NSS-5	15835.663	126
PAS-9	17019.133	104
SINOSAT_3	6953.532	120
SUPERBIRD_4	24694.395	124
TELSTAR_10	41618.693	116
ZHONGXING_20	18083.356	119
ZHONGXING_22	10570.169	118
ZHONGXING_22A	11838.725	118

APPENDIX G. EXCEL WORKBOOK CONTAINING MISCELLANEOUS CALCULATIONS

Miscellaneous Calculations

Earth radius (Re~km)	6.38E+03		
Earth grav const (μ~m^3/s^2)	3.99E+14		
GEOSTA Altitude (~km)	3.5786E+04		
Orbital distance from GEOSTA (~km)	6.40E+01		
Satellite Altitude (H~km)	3.5722E+04	given	
Orbit period (P~min)	1.43E+03	sqrt(μ/(Re+H)^3)	
SC velocity (Vs~km/s)	3.08E+00	2π(Re+H)	42164
SSP velocity (Vg~km/s)	4.66E-01	2πRe/P	35716.8
Earth rotat'n/P (ΔL~deg)	3.59E+02	P/(23.93*60)	
Earth ang radius (ρ~deg)	8.71E+00	asin(Re/(Re+H))	



Appraoch 1: Thin Lens Equation

Thin lense equation: (ImageSize)/(FocalLength) = (ObjectSize)/(Range)

		Trial 1	Trial 2	Trial 3	Trial 4	Trial 5	
Object Size	[m]	1470	734.783	1470	734.783	1470	734.783
Range	[m]	20	20	2000	2000	10000	10000
FocalLength	[m]	0.5	0.5	0.5	0.5	1	1
Image Size	[m]	36.75	18.369575	0.3675	0.18369575	0.147	0.0734783
# pxl per Array Height	[#]	2161765	1080563	21618	10806	8647	4322

Height [km]	1470	1470	1470	1470
Height/2[km]	735	735	735	735
Range [km]	20	64	200	500
Theta [deg]	88.44	85.02	74.78	55.77
Elevation Angle[deg]	1.56	4.98	15.22	34.23

deltaTheta = (Range*PixelPitch)/(FocalLength)

PixelPitch	[m]	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05
deltaTheta	[Rad]	0.00048	0.00048	0.048	0.048	0.12	0.12
GSD = (deltaTheta)*(SlantRange)							
GSD	[m]	0.0096	0.0096	96	96	1200	1200

Appraoch 2: Determine Optics with a desired resolution

Range (Nadir)	[m]	96000	10000	10000	10000	2000	2000	2000	2000	20	20	20	20
Sq Pixel Pitch	[m]	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05	1.20E-05
GSD min (X)	[m]	0.02	0.1	0.01	0.01	0.1	0.1	0.01	0.01	0.1	0.1	0.01	0.01
Wavelength	[m]	4.00E-07	7.50E-07	5.00E-07	7.50E-07	5.00E-07	7.50E-07	5.00E-07	7.50E-07	5.00E-07	7.50E-07	5.00E-07	7.50E-07
Quality Factor	[#]	1.1	1.1	1.1	1.1	1.1	1.1	1.1	1.1	1.1	1.1	1.1	1.1
Focal Length	[m]	5.76E+01	1.20E+00	1.20E+01	1.20E+01	2.40E-01	2.40E-01	2.40E+00	2.40E+00	2.40E-03	2.40E-03	2.40E-02	2.40E-02
Aperature	[m]	5.15E+00	2.01E-01	1.34E+00	2.01E+00	2.68E-02	4.03E-02	2.68E-01	4.03E-01	2.68E-04	4.03E-04	2.68E-03	4.03E-03
F-Number	[f/#]	1.12E+01	5.96E+00	8.94E+00	5.96E+00	8.94E+00	5.96E+00	8.94E+00	5.96E+00	8.94E+00	5.96E+00	8.94E+00	5.96E+00

Thin lense equation:

Object Size	[m]	1470	1470	1470	1470	1470	1470	1470	1470	1470	1470	1470	1470
Range	[m]	96000	10000	10000	10000	2000	2000	2000	2000	20	20	20	20
FocalLength	[m]	5.76E+01	1.20E+00	1.20E+01	1.20E+01	2.40E-01	2.40E-01	2.40E+00	2.40E+00	2.40E-03	2.40E-03	2.40E-02	2.40E-02
Image Size	[m]	0.882	0.1764	1.764	1.764	0.1764	0.1764	1.764	1.764	0.1764	0.1764	1.764	1.764

deltaTheta:

PixelPitch	[m]	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05	1.200E-05
deltaTheta	[Rad]	0.02	0.1	0.01	0.01	0.1	1.000E-01	0.01	0.01	0.1	0.1	0.01	0.01
GSD = (deltaTheta)*(SlantRange)													
GSD	[m]	1920	1000	100	100	200	200	20	20	2	2	0.2	0.2

Miscellaneous Calculations

Determining Half Angle for Optical Sensor for use in STK Simulation

Half Meter Cube Satellite: Custom Optical Payload

CPA distance	[m]	51200.00
Cross Track View at CPA	[m]	5000.00
Angle	[deg]	2.80
Max separation for 0.5[m] Spatial Res	[km]	128.00
Max Inclination Imagable at 0.5[m] Spatial Res	[deg]	0.1741

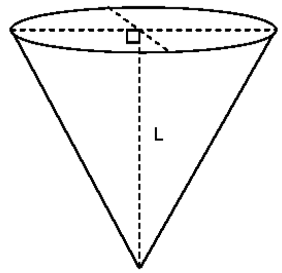
5U CubeSat: Custom Optical Payload

CPA distance	[m]	20491.00
Cross Track View at CPA	[m]	5000.00
Angle	[deg]	6.96
Max separation for 0.5[m] Spatial Res	[km]	51.20
Max Inclination Imagable at 0.5[m] Spatial Res	[deg]	0.0697

1U CubeSat: 1U CubeSat C328-7640 JPEG Compression VGA Camera Module

CPA distance	[m]	500.00
Cross Track View at CPA	[m]	800.00
Angle	[deg]	38.66
Max separation for 0.5[m] Spatial Res	[km]	1.70
Max Inclination Imagable at 0.5[m] Spatial Res	[deg]	0.0023

***Max Inclination only applies if satellite was positioned at GEOSTA orbit.**



Camera Considerations:

Monochrome, 5 to 10 Mpixel; Wide Angle Optic; Auto-focus (off the shelf)
Off the Shelf: Maybe Surri; usually problems talking to camera with processor/computer
Jpeg format for pictures; highly compressed to ~a few 100kbytes/picture
*Use step method to estimate off axis GSD.

1U-CubeSat		
aperature	0.03	[m]
height	500	[m]
lambda	5.00E-07	[m]
Spat Res	0.0203333	[m]
5U-CubeSat		
aperature	0.095	[m]
height	20000	[m]
lambda	5.00E-07	[m]
Spat Res	0.2568421	[m]
Half-Meter-Cube Satellite		
aperature	0.25	[m]
height	52000	[m]
lambda	5.00E-07	[m]
Spat Res	0.25376	[m]

Satellite height above geostionary orbit

Target Satellite	[Name]	AMC-12	Chosen
Target SAT Alt	[km]	35785.86	
Inclination	[deg]	0.0032	Orbital Property
Height	[km]	4.70	1470[km/deg]*Inclination[deg]
Satellite Alt	[km]	35765.37	
Slant Range	[km]	21.03	
Payload Max Rng	[km]	51.20	5U-CubeSat Property

Target Satellite	[Name]	AMC-10	Chosen
Target SAT Alt	[km]	35785.86	
Inclination	[deg]	0.0452	Orbital Property
Height	[km]	66.44	1470[km/deg]*Inclination[deg]
Satellite Alt	[km]	35765.37	
Slant Range	[km]	69.53	
Payload Max Rng	[km]	51.20	5U-CubeSat Property

Target Satellite	[Name]	SINOSAT-2	Chosen
Target SAT Alt	[km]	35785.86	
Inclination	[deg]	0.3235	Orbital Property
Height	[km]	475.55	1470[km/deg]*Inclination[deg]
Satellite Alt	[km]	35765.37	
Slant Range	[km]	475.99	
Payload Max Rng	[km]	128.00	Half-Meter-Cube Property

Miscellaneous Calculations

Rough determination of existing Optical Payloads

		Hubble	IKONOS	Equation/Comments
Wavelength	[m]	4.00E-07	4.00E-07	Selected
Range	[m]	1200000	350000	Changed by goal seeking GSD value
Desired Orbital Height	[m]	34576900	35426900	(GEOSTA Altitude)-Range
FocalLength	[m]	57.6	17.5	Satellite Property
Aperture Diameter (D_a)	[m]	2.4	0.7	Satellite Property
GSD	[m]	0.2	0.2	(λ/D_a)*Range
GEOSTA altitude	[m]	35776900	35776900	Orbit property
Fraction of GEO		3.4%	1.0%	Range/GEOSTA altitude

Determining Optics Needed to Observe Earth from GEO with 1[m] GSD

Range (Nadir)	[m]	3.5786E+07	Property
Sq Pixel Pitch	[m]	1.20E-05	Assumption
GSD min (X)	[m]	1	Desired Res
Wavelength	[m]	4.00E-07	Assumption
Quality Factor	[#]	1.1	Assumption
Focal Length	[m]	429.43	
Aperature	[m]	38.42	
F-Number	[f/#]	11.18	

Geostationay belt field of view (FOV) captured by earth observer

1st SAT station (Sta1)	[deg]	127.00	West longitude
2nd SAT station (Sta2)	[deg]	95.00	West longitude
Separation (Sep)	[deg]	32.00	longitude
Orbit Viewed	[%]	11.25	360/Separation
Orbit Viewed	[fraction]	0.09	Separation/360

Partial Constellation Population Calculation

1st SAT station (Sta1)	[deg]	122.00	East longitude (AsiaSat-4)
2nd SAT station (Sta2)	[deg]	87.50	East longitude (ChinaStar-1)
Separation (Sep)	[deg]	34.50	Sta1 - Sta2
Percent of Geostationary Belt (GEO%)	[%]	9.58	(Sep/360)*100%
Half-Meter-Cube Constellation (HMC)	[SC]	15	Complete Constellation
Half-Meter-Cube Altitude (HMalt)	[km]	35734.64	Orbit altitude selected for Constellation
Half-Meter-Cube Circumference of Orbital Altitude (HMcr)	[km]	264602	2*π*(HMalt+6378.137); Altitude Property
Time for HM to circumnavigate GEO belt (HMt)	[years]	4.49	Single Satellite
5U-CubeSat Constellation (5UC)	[SC]	33	Complete Constellation
5U-CubeSat Altitude (5Ualt)	[km]	35765.37	Orbit altitude selected for Constellation
5U-CubeSat Circumference of Orbital Altitude (5Ucr)	[km]	264795	2*π*(5Ualt+6378.137); Altitude Property
Time for 5U to circumnavigate GEO belt (5Ut)	[years]	11.23	Single Satellite
Half-Meter-Cubes needed to normal cover region (HMCn)	[SC]	1.44	(Sep*HMC)/360
Half-Meter-Cubes needed to normal cover region (5UCn)	[SC]	3.16	(Sep*5UC)/361
HMCn for region coverage over MDL (HMCrgn)	[SC]	7.00	((2/HMt)*HMcr)*(HMC/HMcr) -> Rounded Up
Percentage of HMC orbital belt populated	[%]	46.67	(HMCrgn/HMC)*100%
5UCn for region coverage over MDL (5Urgn)	[SC]	6.00	((2/5Ut)*5Ucr)*(5UC/5Ucr) -> Rounded Up
Percentage of 5UC orbital belt populated	[%]	18.18	(5Urgn/5UC)*100%

Ref: SMAD p.247-91

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