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An analysis was conducted on the feasibility of using a 72-satellite constellation at an altitude of 1000 km to communicate with both low flying objects and low Earth orbiting (LEO) satellites. In addition, the satellites would be designed to communicate with ground terminal locations using radio frequency (RF) transmitters. The proposed satellite constellation would provide comprehensive communications coverage across the continental United States. The results of this analysis have shown the impracticability of such a solution at this time due to the satellite’s power budget.

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Design for a Low-Cost K-Band Communication Satellite Constellation

Sponsor: Missile Defense Agency
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Author: Peyton D. Strickland

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Abstract

MITRE funded analysis was conducted on the feasibility of using a 72-satellite constellation at an altitude of 1000 km to communicate with both low flying objects and low Earth orbiting (LEO) satellites. In addition, the satellites would be designed to communicate with ground terminal locations using radio frequency (RF) transmitters. The proposed satellite constellation would provide comprehensive communications coverage across the continental United States. The results of this analysis have shown the impracticability of such a solution at this time due to the satellite's power budget. For this design, the satellite is required to use a heritage radio frequency data transmitter to communicate resulting in a cost of almost $5 billion for the satellite constellation with a design life of 13 years. Comparison to the historical example of the FireSat II satellite provided the validation for the sizing and design of the proposed satellite. The satellite sizing models used equations found in *Space Mission Engineering: The New SMAD* [1]. Lastly, the cost estimate was provided by the 2010 version of the Small Satellite Cost Model developed by The Aerospace Corporation.

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1 Introduction

In August 2018, The MITRE Corporation requested a first-cut estimate of the size, weight, power, and cost of using a constellation of 72-satellites at an altitude of 1000 km to communicate with both low flying objects and low Earth orbiting (LEO) satellites. In addition, the satellites would be designed to communicate with ground terminal locations using radio frequency (RF) transmitters. The area of interest requiring comprehensive coverage as defined by MITRE spans from a minimum latitude of 18 degrees North and a maximum latitude of 90 degrees North with a minimum longitude of 180 degrees West and maximum longitude of 312 degrees West. This results in comprehensive coverage of the continental United States (CONUS) and its surroundings. As part of this analysis, MITRE established several design parameters. First, MITRE defined the constellation configuration using six orbital planes with each plane consisting of 12 satellites at an 85-degree inclination for a total of 72 satellites. Furthermore, MITRE established that the communication payloads for each satellite would use RF transmitters with a range constraint of 2000 km and laser communication units for satellite-to-satellite cross link capability. Lastly, MITRE provided locations for ground stations located across CONUS. The satellite design approach described in this technical report was adapted from *Space Mission Engineering: The New SMAD.*

2 Approach

2.1 Introduction

To begin this analysis, a LEO satellite constellation of 72 satellites distributed throughout six orbital planes at an altitude of 1000 km and inclination of 85 degrees with a thirteen-year design life was established. Once the orbit was established, a coverage model tool was created for the proposed satellite constellation using Analytical Graphics Incorporated (AGI)'s Systems Tool Kit (STK), a physics-based software package that allows engineers and scientists to perform complex analysis of ground, sea, air, and space platforms, through a MATLAB interface. This coverage model tool allows the user to record the percentage of area satisfied by time for the given transmitter configuration which can then be used through an iterative process to determine the number of transmitters needed per satellite as well as their location. After the optimal number of transmitters were determined, an initial spacecraft bus dry mass and power estimate were calculated based on the equations in *Space Mission Engineering: The New SMAD* [1]. With these first-order power and mass budgets, a power system model, propulsion system model, and an attitude, determination, and control system (ADCS) model were created in MATLAB to size the different subsystems of the satellite. Lastly, the Small Satellite Cost Model (SSCM) [2] was used to establish a cost model for the proposed satellite design.

2.2 Constellation Parameters

The orbital altitude of 1000 km was selected due to communication range limitations of the radio used on-board the satellite. An inclination of 85 degrees to enable access to ground terminals at high latitude regions. Furthermore, a total of 72 satellites in six orbital planes was derived through an iterative process; this process balanced the revisit time of satellites over ground terminals and the ability for cross communications between all satellites with line-of-sight using laser systems. These numbers are consistent with commercially implemented systems such as the Iridium constellation.

2.3 Coverage Model

2.3.1 Purpose

Once the constellation parameters and provided ground terminal locations were both known, a coverage model generated to determine the optimal number of RF transmitters needed per satellite and their respective orientations to provide an acceptable percentage coverage over CONUS and its surroundings at all times. To implement this coverage model, STK was used through a MATLAB interface to allow for an iterative approach to be employed.

2.3.2 Methodology

First, a scenario was created in STK lasting for 24 hours to ensure redundancy of results through multiple completed orbits. Following this, the provided ground terminal locations from MITRE were modeled as facilities in the scenario. Next, the constellation of 72 satellites at 1000 km with an inclination of 85 degrees was modeled. To do this, a seed satellite was modeled by inserting a default satellite at 1000 km with an 85-degree inclination. Subsequently, a generic RF transmitter with a maximum range constraint of 2000 km and a half cone angle of 45 degrees as prescribed by MITRE was attached to each satellite. Lastly, the Walker Tool in STK was utilized to create a Walker constellation, a group of satellites that are in circular orbits and have the same period and inclination, of 72 satellites which consisted of six planes with each plane having twelve replica satellites of the seed satellite. In addition to being optimized for high-latitude coverage, this constellation design also was optimized to allow for line-of-sight laser communication. Line-of-sight laser communication decreased the number of satellites needed due to the increase range provide when compared to the performance of the RF transmitters. Furthermore, the laser communication provided this range improvement at a fraction of the power cost associated with the RF transmitters.

After the satellite constellation was modeled, STK constellations can be utilized which allows a user to group a set of related objects, such as a group of facilities or satellites, into a single unit called a “constellation”. A facility, satellite, and transmitter constellation are all created. This makes it significantly easier to carry out a coverage analysis using a chain, a list of objects (either individual or grouped into constellations) in order of access.

The path of the chain modeled herein was facilities to satellite, satellite to satellite, and satellite to transmitter. In other words, the ground station must be able to see a satellite that can see and communicate with another satellite using laser communication. If this chain was valid, STK determined the percentage of the area covered given the transmitter’s range constraint of 2000 km and half cone angle of 45 degrees. This chain was chosen due to another requirement established by MITRE that mandated that information must be able to be communicated to a low flying object in a timely manner regardless of proximity to ground terminals so that if the low flying object is not close to the ground terminals, information can still be transmitted to the rest of the area of interest through satellite cross-linking.

Once the chain was modeled, a coverage grid is created in STK to assign the area of interest where coverage is important. CONUS and its surroundings were designated as the area of interest with the grid bounds consisting of a minimum latitude of 18 degrees North and a maximum latitude of 90 degrees North with a minimum longitude of 180 degrees West and maximum longitude of 312 degrees West. Furthermore, the coverage points are placed at an altitude of 1000 degrees to coincide with the altitude of the satellites. Lastly, the chain is assigned to the coverage grid as the asset of interest. As a result, when the chain is valid, the satisfied area is calculated as a function of time by determining the area in which the transmitter can communicate, considering range and field of view constraints.

Finally, a "Satisfied by Time Report" was generated in STK and exported to an Excel file detailing the percentage of area covered as well as the total area covered in meters squared in single minute blocks during the 24-hour analysis period. This process is repeated from one transmitter pointing nadir to six transmitters (one on each side of the rectangular shaped satellite pointed orthogonally). Originally, a single script with a "for loop" adding one transmitter to each iteration was utilized; however, due to run time and memory constraints, the original script had to be altered. The single script was separated into six separate scripts to avoid run time constraints while STK's No Graphics mode was utilized to free more memory for object and coverage analysis storage.

2.3.3 Coverage Model Output

With the coverage model finished, the program was run to determine the percentage of coverage over the area of interest as a function of time. This process was completed six times to find the optimal number of transmitters and their orientation. An example "Satisfied by Time Report" is shown below.

1	Time Satisfied	Percent Satisfied	Area Satisfied (km^2)
1417	11/19/2018 5:35	36.92949409	32710234.87
1418	11/19/2018 5:36	35.97296631	31862992.06
1419	11/19/2018 5:37	34.7110073	30745214.07
1420	11/19/2018 5:38	37.52561757	33238250.2
1421	11/19/2018 5:39	34.38901909	30460013.57
1422	11/19/2018 5:40	38.52358749	34122200.31
1423	11/19/2018 5:41	34.25605248	30342238.64
1424	11/19/2018 5:42	39.71626929	35178616.13
1425	11/19/2018 5:43	35.15143456	31135321.75
1426	11/19/2018 5:44	36.56535893	32387702.79
1427	11/19/2018 5:45	33.94081763	30063019.91
1428	11/19/2018 5:46	34.85784701	30875277.09
1429	11/19/2018 5:47	37.41263091	33138172.46
1430	11/19/2018 5:48	37.39251278	33120352.86
1431	11/19/2018 5:49	36.66428375	32475325.27
1432	11/19/2018 5:50	33.76251056	29905084.74
1433	11/19/2018 5:51	37.16968839	32922986.55
1434	11/19/2018 5:52	35.36436133	31323921.26
1435	11/19/2018 5:53	39.22217602	34740973.89
1436	11/19/2018 5:54	38.60724646	34196301.1
1437	11/19/2018 5:55	36.79616904	32592142.44
1438	11/19/2018 5:56	39.74768351	35206441.22
1439	11/19/2018 5:57	35.42441127	31377110.39
1440	11/19/2018 5:58	34.45706311	30520283.45
1441	11/19/2018 5:59	30.52975091	27041673.53
1442	11/19/2018 6:00	31.78483688	28153363.73
1443			
1444	Average	32.89048744	29132691.79

Figure 2-1 Coverage Model Output

2.4 First-Order Satellite Power Estimate

Once the coverage model was completed and the optimal number of transmitters per satellite was selected, a first-order power estimate was generated. There are two approaches for determining a spacecraft's mass and power covered in *SME: The New SMAD*. For the first approach, as described in section 14.7.1 "SCS Example" on page 432, one can begin with a target mass for the entire system and then determine the mass and power available for the spacecraft. Conversely, as described in section 14.7.2 "FireSat II Example" on page 435, one can start with the payload and then determine the mass and power for the spacecraft. While the first method is great for flexible mission objectives, the FireSat II example is used in this analysis since the payloads of RF transmitters and laser communication units have already been defined by MITRE and the results of the coverage model.

Using Table 2-1, the Low-Earth-Orbit with propulsion spacecraft section is used to generate a total power estimate (page 424 of *SME: The New SMAD* [1]).

Table 2-1 Average Power by Subsystem for 4 Types of Spacecraft

Subsystem (% of Total Power)	No Prop	LEO Prop	High Earth	Planetary
Payload	43%	46%	35%	22%
Structure and Mechanisms	0%	1%	0%	1%
Thermal Control	5%	10%	14%	15%
Power (Incl. harness)	10%	9%	7%	10%
Telemetry, Tracking, & Command (TT&C)	11%	12%	16%	18%
On-board Processing	13%	12%	10%	11%
Attitude Determination and Control	18%	10%	16%	12%
Propulsion	0%	0%	2%	11%
Average Power (W)	299	794	691	749

According to the chart, 46% of the total power is used by the payload. With a total payload power consumption already known, the total power for the spacecraft is estimated by dividing the payload power by 0.46. This first-order estimate proves to be highly effective in that the final estimate for the power consumption of the satellite is significantly close to the final estimate that will be derived in the results section. Lastly, in order to accurately model the situation that the satellite will not always be operating at full power, it is assumed that the satellite operates a third of the day at full-power, a third of the day at half-power, and a third of the day at no-power.

Furthermore, it is assumed that you can only communicate at full-power during sunlight periods where the satellite is able to use power from both batteries and the solar panels.

2.5 Power System Model

2.5.1 Purpose

With a power requirement computed, a system to meet the power demand can be developed. The power system is comprised of two components: solar cells and batteries. There is a plethora of factors that influence the size and type of solar panels to be used. The first factor that affects the size needed is that the surface of the solar array may be eclipsed for extended durations of time depending on the satellite's altitude and inclination. Subsequently, a MATLAB program developed by the author was used to calculate the sunlight and eclipse periods for the satellite's defined orbit to determine how much time the satellite's solar panels will have to collect sunlight. Furthermore, there are three types of solar cells that are generally used for satellite applications (Gallium Arsenide, Multijunction, and Silicon). These three types of solar cells provide varying efficiency levels (higher efficiency, less area) and cost. As a result, the surface area of a solar panel needed to produce enough power to fulfill the satellite's power requirements require calculation for all three types to compare the surface area needed and the cost for each case to ensure that the optimal solution is being selected. Lastly, the batteries must be designed to store the energy derived from the solar panels. To accomplish this task, a MATLAB graphical user interface (GUI) linked to STK was created. The following design method was derived from Section 21.2 "Power" on page 641 of *SME: The New SMAD* [1].

2.5.2 Power System GUI Design

The first step in creating this model was to design a GUI that is easy to use. Designing the GUI first also defines the outputs for the program in an orderly manner which streamlines the coding process. Keeping in mind that conducting trade-studies is a key goal for this model, the GUI allows any user to quickly view results for various power budgets, orbits, design life, and battery quantities. The GUI developed for this analysis is shown below.

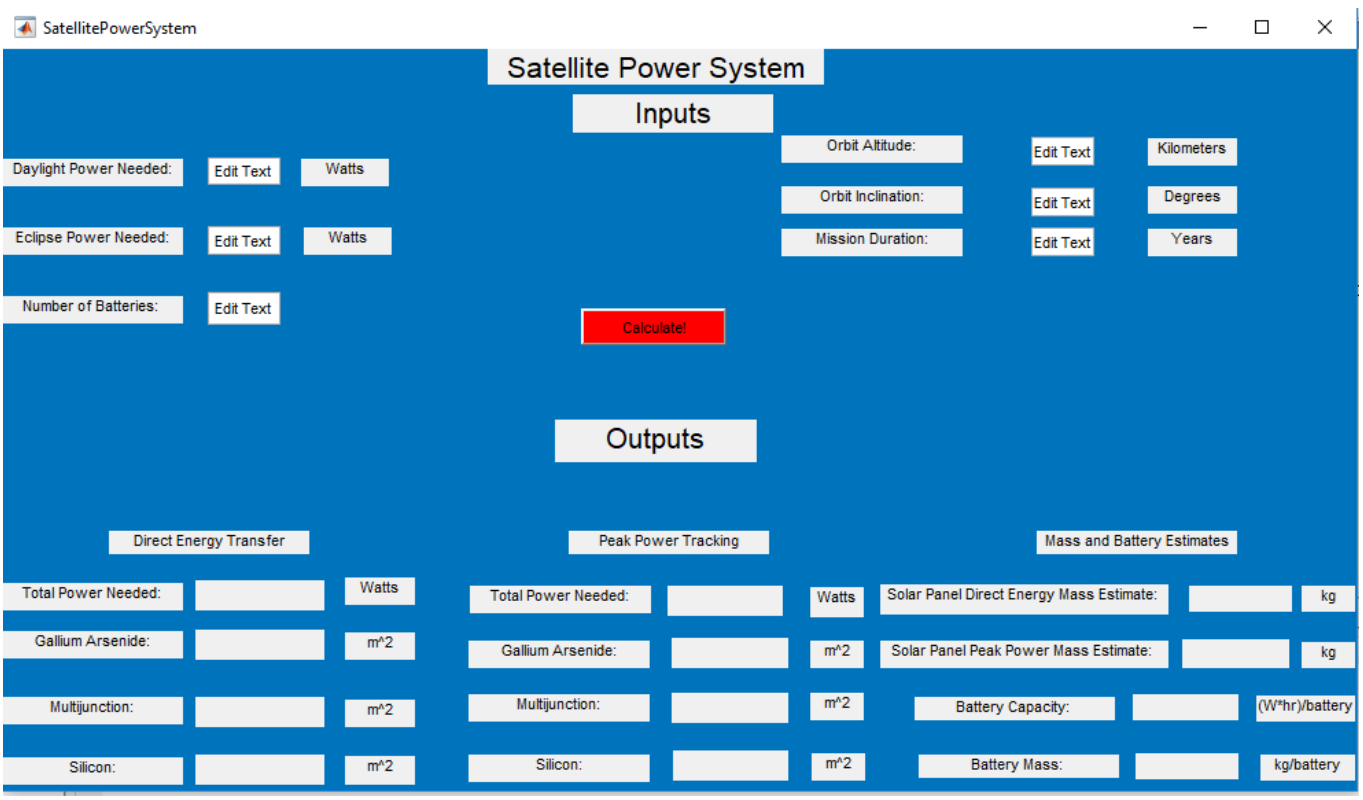

Figure 2-2 Satellite Power System GUI

2.5.3 Solar Panel Sizing

For the program to calculate the solar panel area needed, the user first must input the following information: power needed during eclipse, power needed during daylight, orbit altitude, orbit inclination, and mission duration. Mission duration and the average power requirements are the two key design considerations in sizing the solar array because photovoltaic systems are sized at end-of-life (EOL) to ensure that adequate power can be supplied for the entire duration of the mission.

Once the following design parameters have been input, the power the solar array must provide during daylight to power the spacecraft along with recharging the batteries must be calculated.

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d}$$

$\boldsymbol{P_e}$ = ***Eclipse Power Needed*** $(\boldsymbol{W})$
$\boldsymbol{P_d}$ = ***Daylight Power Needed*** $(\boldsymbol{W})$
$\boldsymbol{T_e}$ = ***Length of Eclipse per Orbit*** $(\boldsymbol{s})$
$\boldsymbol{T_d}$ = ***Length of Daylight per Orbit*** $(\boldsymbol{s})$
$\boldsymbol{X_e}$ = ***Eclipse Path Efficiency***
$\boldsymbol{X_d}$ = ***Daylight Path Efficiency***

Figure 2-3 Solar Array Power Equation

For direct energy transfer, the eclipse path efficiency and daylight path efficiency were approximated as 0.65 and 0.85, respectively; for peak-power tracking, eclipse path efficiency is estimated at 0.60 while daylight path efficiency is estimated at 0.80 (page 643 of *SME: The New SMAD* [1]). STK was used to calculate the length of the eclipse and daylight periods per orbit, and the required power production from the solar array was calculated.

Generally, the third step in the solar array design process is the selection of the type of solar cell; however, since the model for this analysis is conducting a trade-study to determine the best solar cell balancing area and cost, all three solar cells were included. Table 2-2 details solar cell efficiencies for each type (page 645 of *SME: The New SMAD* [1]).

Table 2-2 Performance Comparison for Photovoltaic Solar Cells

Cell Type	Silicon	Gallium Arsenide	Multijunction
Theoretical Efficiency	29%	23.5%	40+%
Achieved Efficiency:			
Production Efficiency	22%	18.5%	30.0%
Best Laboratory	24.7%	21.8%	33.8%

While silicon cells are mature in their development and can have lower cost in environments where radiation is not a concern, multijunction cells have become the standard for space applications despite their high cost. What makes them the #1 choice is their high efficiency resulting in less area required to produce the same amount of power comparative to silicon cells. Using the best laboratory efficiencies provided above, the power output for each solar cell is calculated by multiplying the efficiency by the solar constant 1,368 W/m^2.

Next, the beginning-of-life (BOL) power per unit area is determined using the following equation:

$$P_{BOL} = P_o I_d \cos\theta \qquad \begin{array}{ll} \boldsymbol{P_o} & = \boldsymbol{Solar\ Cell\ Power\ Output\ (W/m^2)} \\ \boldsymbol{I_d} & = \boldsymbol{Inherent\ Degradation} \\ \boldsymbol{\theta} & = \boldsymbol{Sun\ Incidence\ Angle\ (deg)} \end{array}$$

Figure 2-4 Beginning-of-Life Power Equation

The solar cell power output calculation is provided in the previous step when the solar cell type is selected. Inherent degradation quantifies the loss in performance and is assumed to be 0.77 (page 644 of *SME: The New SMAD* [1]). Lastly, the sun incidence angle is the angle between the vector normal to the surface of the array and the Sun line. Although, the solar panel is configured to minimize this cosine loss, it is assumed that theta is equal to 23.5 degrees in order to model an industry standard worst-case Sun angle assumption to ensure power production requirements are always met (page 647 of *SME: The New SMAD* [1]).

The last step is the calculation of the EOL power per unit area which can then be used in conjunction with the solar array power calculated in the first step to calculate the area required.

$$L_d = (1 - D)^L$$

$$P_{EOL} = P_{BOL} L_d$$

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$

$\boldsymbol{L_d}$ = ***Lifetime Degradation***
$\boldsymbol{D}$ = ***Degradation per Year***
L = ***Satellite Lifetime (Years)***
$\boldsymbol{P_{EOL}}$ = ***End of Life Power*** $(\boldsymbol{W/m^2})$
$\boldsymbol{P_{BOL}}$ = ***Beginning of Life Power*** $(\boldsymbol{W/m^2})$
$\boldsymbol{A_{sa}}$ = ***Solar Array Area*** $(\boldsymbol{m^2})$
$\boldsymbol{P_{sa}}$ = ***Solar Array Power*** $(\boldsymbol{W})$

Figure 2-5 End-of-Life Power and Solar Array Area Equation

Several factors degrade a solar panel´s performance. Lifetime degradation of the solar panel occurs due to thermal cycling, material degradation, and space debris impact among others. First, the lifetime degradation can be calculated by the first equation in Figure 2-5 for which the degradation per year for silicon, gallium arsenide, and multijunction are 3.75%, 2.75%, and 0.5%, respectively (page 647 of *SME: The New SMAD* [1]). Next, the end-of-life power was calculated using the beginning-of-life power and lifetime degradation (2nd equation in Figure 2-5). Finally, the solar array area can be calculated by dividing the solar array power by the end-of-life power, and the mass of the solar array is estimated by multiplying the solar array power by .04.

2.5.4 Battery Sizing

Energy storage plays a vital role in the electrical-power subsystem allowing the spacecraft to continue operating in eclipse periods and peak-power demands. For the satellite under consideration with a thirteen-year design life, secondary (rechargeable) batteries were selected. Although Nickel-Cadmium and Nickel-Hydrogen are commonly used secondary batteries, Lithium-Ion was selected based on its significant volumetric and energy density advantages.

The spacecraft's orbital parameters, especially altitude, determine the number of charge/discharge cycles the battery must support. According to page 650 *of SME: The New SMAD* [1], the depth-of-discharge (DOD) is limited to 30% for LEO spacecraft. As a result, the cycle life is increased but the amount of energy available from the batteries during each cycle is decreased.

To determine the size of the batteries (battery capacity), only one equation is required.

$$C = \frac{P_e T_e}{(DOD)(N)(n)}$$

$\boldsymbol{C}$ $= \boldsymbol{Battery\ Capacity}$ **(W-hr)**
$\boldsymbol{P_e}$ $= \boldsymbol{Eclipse\ Power\ (W)}$
$\boldsymbol{T_e}$ $= \boldsymbol{Length\ of\ Eclipse\ per\ Orbit\ (hr)}$
$\boldsymbol{DOD}$ $= \boldsymbol{Depth - of - Discharge\ \left(W/m^2\right)}$
$\boldsymbol{N}$ $= \boldsymbol{Number\ of\ Batteries}$
$\boldsymbol{n}$ $= \boldsymbol{Transmission\ Efficiency}$

Figure 2-6 Battery Capacity Equation

The eclipse power and length of eclipse per orbit are defined in the solar panel sizing portion of the code. Next, the DOD is estimated at 0.30 based on the LEO orbit. Lastly, the number of batteries is generally set at two or more for redundancy, and the transmission efficiency is estimated at 90% (page 653 of *SME: The New SMAD* [1]).

2.5.5 Power System Model Outputs

With all equations and variables defined, the power system model is complete. The solar array and battery information was outputted in the GUI that accomplishes the trade-study task. Figure 2-7 shows an example output.

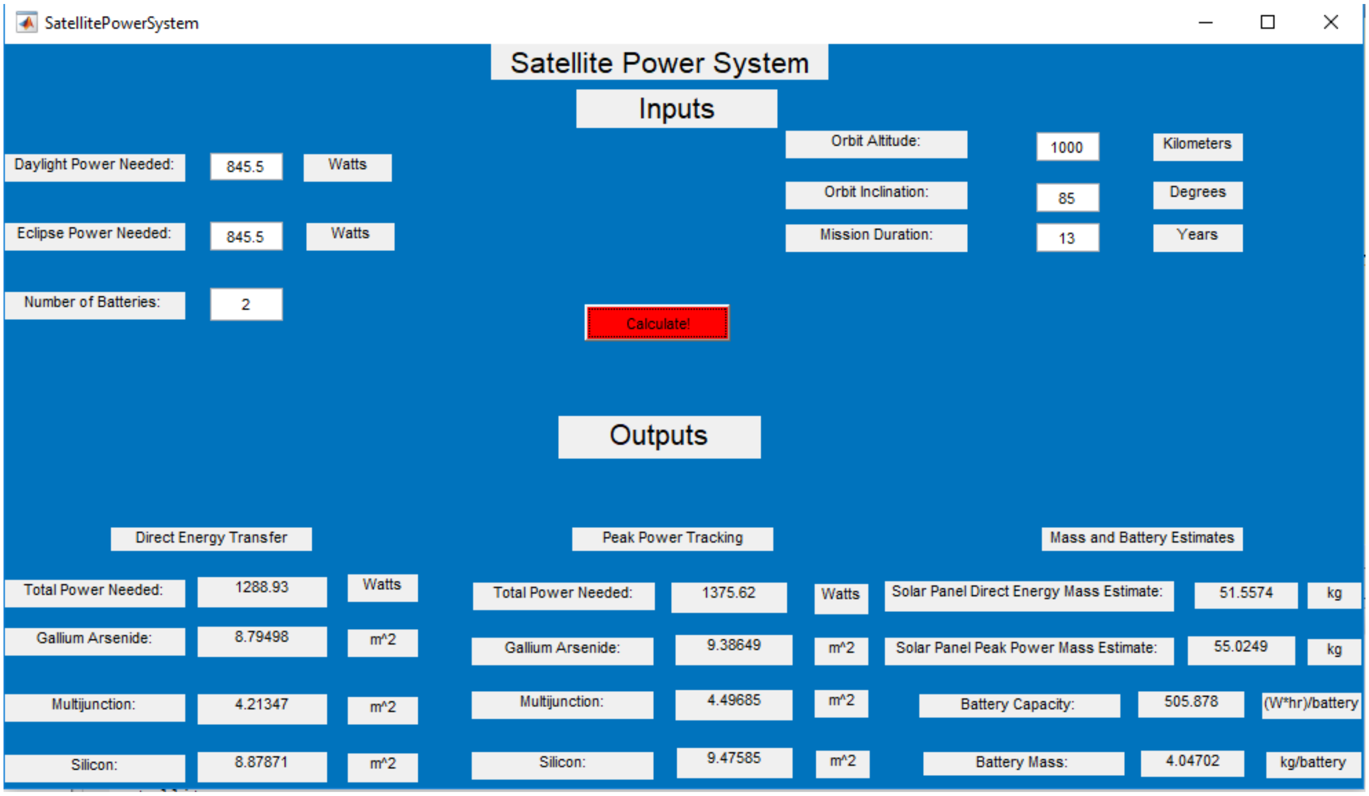

Figure 2-7 Satellite Power System Model Output

2.6 Delta-V (ΔV) Estimate

Working with the rocket equation, the ΔV budget was used to create a propellant budget and estimate the propellant mass required for the space mission. Higher orbits require more propellant for orbit acquisition and de-orbit but less propellant for on-orbit maintenance. With an altitude of 1000 km, an accurate estimation method is needed. On page 253 of *SME: The New SMAD* [1], Figure 10-16 provides ΔV budget as a function of altitude for LEO. Using this figure, at an orbit of 1000 km, the ΔV budget is estimated at 691.22 m/s for a design life of thirteen years.

2.7 Spacecraft Bus Dry Mass Estimate

With first order estimates of the payload mass, power system mass, and the ΔV budget, a first order dry mass estimate was derived so that an ADCS could be designed. First, the payload total mass was calculated by adding the weight of the RF transmitters and the laser communication units. Once the payload mass was calculated, all other first order mass estimates, excluding the power mass estimate (which has already been calculated) were derived from the usage of Table 2-3 (page 422 of *SME: The New SMAD* [1]).

Table 2-3 Average Mass by Subsystem for 4 Types of Spacecraft

Subsystem (% of Dry Mass)	No Prop	LEO Prop	High Earth	Planetary
Payload	41%	31%	32%	15%
Structure and Mechanisms	20%	27%	24%	25%
Thermal Control	2%	2%	4%	6%
Power (Incl. harness)	19%	21%	17%	21%
TT&C	2%	2%	4%	7%
On-board Processing	5%	5%	3%	4%
Attitude Determination and Control	8%	6%	6%	6%
Propulsion	0%	3%	7%	13%
Other (balance + launch)	3%	3%	3%	3%
Total	100%	100%	100%	100%
Propellant	0%	27%	72%	110%

The total dry mass was estimated by dividing the payload weight by the percent of dry mass for the payload subsystem. For this specific scenario, 64.5 kg would be divided by 0.31

(average payload percentage of dry mass for LEO spacecraft with propulsion) to calculate a total dry mass. Once this total dry mass has been estimated, the percentage for each subsystem was multiplied by the total dry mass until all subsystems' masses were calculated. The total dry estimate listed in the results is slightly greater than the estimate provided using this method due to the power mass estimate from the satellite power system model being used instead of the 21% listed in *SME: The New SMAD* [1].

2.8 Propulsion System Mass Model

The spacecraft dry mass and ΔV budget estimates allow for the estimation of the propulsion system's mass. For preliminary design, the rocket equation was utilized to estimate the propellant mass.

$$M_p = M_f[e^{\Delta V/(I_{sp} g_0)} - 1]$$

$\boldsymbol{M_p}$ $= \boldsymbol{Propellant\ Mass\ (kg)}$
$\boldsymbol{M_f}$ $= \boldsymbol{Dry\ Mass\ (kg)}$
$\boldsymbol{\Delta V}$ $= \boldsymbol{Delta - V\ (m/s)}$
$\boldsymbol{I_{sp}}$ $= \boldsymbol{Specific\ Impulse\ (sec)}$
$\boldsymbol{g_0}$ $= \boldsymbol{Gravity\ at\ Sea\ Level\ (m/s)}$

Figure 2-8 Propellant Mass Equation

The ΔV needed is within the range for monopropellant thrusters. Subsequently, hydrazine was selected as the fuel resulting in a modest I_{sp} estimate of 218 seconds. All remaining variables are known from previous calculations, and the propellant mass was computed.

Next, M_{p_usable} was defined as the propellant mass plus the fuel needed for attitude control. The fuel needed for attitude control is much smaller than the propellant mass needed to meet the ΔV, so for a first order estimate, it was assumed to be 9% of the total propellant mass. Not all the propellant loaded into a tank is usable however. As a result, a 3% margin is applied to the usable propellant to account for propellant trapped in the tank, feed lines, or valves [3]. Furthermore, there is a measurement uncertainty of about 0.5% on propellant loaded. Thus, the total propellant loaded is

$$M_{p_{loaded}} = M_{p_{usable}}(1.0 + 0.03 + 0.005)$$

Figure 2-9 Loaded Propellant Mass Equation

With the true value of loaded propellant mass calculated, the propellant tank and feed system can be sized.

$$V_{p\_Loaded} = {M_{p\_loaded}}/{\rho_{fuel}}$$

$$V_{p\_usable} = 0.97(V_{p\_loaded})$$

$$V_{p\_ullage} = {V_{p\_usable}}/{(B-1)}$$

$$V_{total} = 1.2(V_{p\_loaded} + V_{p\_ullage})$$

$$M_{tank} = 2.7086e^{-8}{V_{total}}^3 - 6.1703e^{-5}{V_{total}}^2 + 6.629e^{-2}V_{total} + 1.3192$$

$$M_{pressurant} = \left(1.2V_{p_{ullage}}\right)(.01\rho_{pressurant})$$

$$M_{feed\_sys} = 0.15\left(M_{p_{loaded}} + M_{thrusters} + M_{pressurant}\right) - M_{tank}$$

$\boldsymbol{V_{p\_loaded}}$ $= \boldsymbol{Volume\ of\ Fuel\ (L)}$
$\boldsymbol{V_{p\_usable}}$ $= \boldsymbol{Usable\ Fuel\ Volume\ (L)}$
$\boldsymbol{B}$ $= \boldsymbol{Blow-Down\ Ratio}$
$\boldsymbol{V_{p\_ullage}}$ $= \boldsymbol{Ullage\ Volume\ (L)}$
$\boldsymbol{V_{total}}$ $= \boldsymbol{Total\ Tank\ Volume\ (L)}$
$\boldsymbol{M_{tank}}$ $= \boldsymbol{Tank\ Mass\ (kg)}$
$\boldsymbol{M_{pressurant}}$ $= \boldsymbol{Pressurant\ Mass\ (kg)}$
$\boldsymbol{M_{feed\_sys}}$ $= \boldsymbol{Feed\ System\ Mass\ (kg)}$
$\boldsymbol{M_{thrusters}}$ $= \boldsymbol{Thrusters\ Mass\ (kg)}$

Figure 2-10 Propellant Tank and Feed System Sizing Equations

First, the volume of fuel was calculated by dividing the loaded amount of propellant by the density of the fuel. The density of the fuel (Hydrazine) is 1.01 kg/L at 293 K (Table 18-8 of *SME: The New SMAD* [1]). Naturally, not all the fuel will be consumed. It is estimated that 97% of the volume is usable [3]. Next, the ullage volume was calculated using the usable volume and the blow-down ratio. The blow-down ratio, defined as the final ullage volume over the initial ullage volume or the initial pressurant pressure over the final pressurant pressure, was assumed to be four, a typical value for modern blow-down propellant tanks. The final total volume was calculated adding the loaded volume and the ullage volume and adding a customary 20% margin. Finally, the tank mass is now sized using the final total volume using a curve fit of commercially available propellant management devices (PMD) propellant tanks.

After computing the mass of the tank and the propellants, the mass of the pressurant and feed system require calculation. Currently, the initial operating pressure for the tank is unknown; however, an estimation was made from the operating pressure ranges of two candidate thrusters: MRE-1.0 and Monarc 445. The operating range for the MRE-1.0 is 0.055 to 3.9 MPa while the operating range for the Monarc 445 is 0.5 to 3.1 MPa [4] [5]. A 20% margin over the 3.9 MPa was carried so the initial pressure was estimated at 4.7 MPa and the initial temperature at 323 K. The pressurant selected was Helium (He). According to the NIST Chemistry WebBook, the density for He under those conditions is assumed to be 6.87 kg/m^3 [6]. The equations for the mass of the pressurant and the feed system can now be solved.

To complete the propulsion system model, the mass of the thrusters needed was estimated. For this specific scenario, four thrusters for unloading momentum from the reaction wheels and one main thruster for primary propulsion were selected following the example of the FireSat II. For the attitude control maneuvers, the candidate thruster selected was the flight-proven MRE-1.0. The MRE-1.0 has a mass of 0.5 kg, average thrust of 3.4 N, and maximum thrust of 5.0 N. For the primary propulsion thruster, the potential candidate thruster is the Monarc-445 which has a mass of 1.6 kg and steady state thrust of 445 N. While these thrusters should meet thrust requirements, a final decision can be made after more information on thrust levels is gathered from the completion of the ADCS model.

2.9 First Order Estimate of Satellite Dimensions

A first order estimate of the satellite's dimensions and mass minus the attitude and control system are needed to size the ADCS. With the first order mass estimates finalized and the size of several subsystems known a priori, the satellite's dimensions were calculated through a first order approximation process. Although, the ADCS cannot be sized prior to the calculation, it constitutes a very small percentage of the overall size and mass of the satellite. As a result, the satellite was sized with a small portion of the volume reserved. To create an estimate of the satellite's dimensions, the author utilized an Excel spreadsheet to document the dimensions of each of the parts needed. With an Excel spreadsheet listing the dimensions of each part, SolidWorks, a solid modeling computer-aided design and computer-aided engineering computer program, can be utilized to create a 3D model to estimate the overall dimensions of the satellite body needed to store all the parts.

2.10 ADCS Model

2.10.1 Introduction

The final subsystem to be sized was the Attitude Determination and Control Subsystem (ADCS). First, the determination transmitters were selected based off the pointing requirements. Once the determination transmitters were selected, the MATLAB model was initialized. The MATLAB model required vehicle, orbital, and Earth properties as input to generate the cyclical and secular angular momentum per orbit, a single reaction wheel mass, the fuel required for momentum dumping over the satellite lifetime, and the thrust required for each momentum dumping event.

2.10.2 Determination Transmitters

The driving force behind the selection of the determination transmitters was the pointing requirements of the payload. The RF transmitter described by MITRE has a 45-degree half-angle cone; however, the laser communication system does not have this ability afforded to it. Laser communication systems often require pointing accuracy to $\mp 1^{\circ}$. As a result, a star tracker was selected due to its ability to meet this accuracy requirement. Furthermore, a sun sensor was also selected in conjunction with the star tracker for redundancy of data and for its ability to help with determination during maneuvers such as detumbling.

2.10.3 Attitude and Control System (ACS) Initiation File

The purpose of the initiation file was to establish satellite properties and to call the functions that will determine the torques on the satellite and subsequently size the reaction wheels and thrusters accordingly. The vehicle properties utilized by the program included physical dimensions, center of gravity, mass, surface material code, and the lifespan of the vehicle. Next, the orbital elements for the proposed satellite were input. The last information input was the properties of Earth. Finally, the ACS Sizing function was called by the MATLAB script, and the torques and the ACS' size were estimated.

2.10.4 ACS Sizing Function

2.10.4.1 Introduction

The ACS Sizing function requires inputs of the orbital elements, vehicle parameters, and planet parameters to compute the cyclical and secular angular momentum, the reaction wheel mass, the ACS fuel mass required, and the thrust required per momentum dump. Several steps are required to accomplish this task. First, the vehicle's 2^{nd} moment of inertia and the time for one orbit were calculated thereby enabling calculation and summation of the solar radiation, aerodynamic, magnetic, and gravity torques. This process was completed for the entire angular range of the orbit (0 to 360 degrees). Using the computed torques, the maximum torques around the orbit were integrated to find the total angular moment which was then used to find the cyclical and secular angular momentums. Lastly, the reaction wheel was sized using the cyclical angular momentum while the ACS propellant mass and thrust required for momentum dumping were sized using the secular momentum dumping. The calculated torques were displayed on a polar graph for visualization purposes.

2.10.4.2 Solar Radiation Pressure (SRP) Torque Calculation

Sunlight (i.e. photons) has momentum, and therefore exerts pressure when it strikes an object. The more absorptive the material, the more momentum absorbed resulting in a certain pressure force. If the sunlight is reflected, the pressure force felt is twice as much as the pressure force if all the sunlight is absorbed. While it is extremely difficult to estimate the true SRP because of the varying surfaces used on a satellite, a good first-order estimate can be made by assuming uniform reflectance. With uniform reflectance assumed, the following equation is appropriate:

$$T_s = \frac{\varphi}{c} A_s (1 + q)(cp_s - cm)\cos\theta$$

T_s = SRP Toque (Nm)
φ = Solar Constant (W/m^2)
c = Speed of Light (m/s)
A_s = Sunlit Surface Area (m)
q = Reflectance Factor
cp_s = Center of SRP (m)
cm = Center of Mass (m)
θ = Sun Incidence Angle (°)

Figure 2-11 SRP Torque Equation

The solar constant was assumed to be 1,366 W/m^2. The surface material number was one of the inputs gathered at the beginning of the ACS Model. Using an Excel spread sheet, the MATLAB function gathered the reflectance factor (q) corresponding to the surface material number input by the user. Lastly, the angle of incidence of the sun was assumed to be zero degrees (worst-case). Listed below are the reflectance factors for possible spacecraft surface materials.

Table 2-4 Reflectance Factors for Commonly Used Spacecraft Materials [7]

Name	Material Number	Absorptance	Reflectance
Aluminized FEP	1	0.16	0.84
Aluminized Teflon	2	0.163	0.837
Aluminum Tape	3	0.21	0.79
Black Paint	4	0.95	0.05
Goldized Kapton	5	0.25	0.75
Optical Solar Reflector	6	0.07	0.93
Polished Beryllium	7	0.44	0.56
Quartz over Silver	8	0.077	0.923
Silver Coated FEP	9	0.08	0.92
Silver Paint	10	0.37	0.63
Silvered Teflon	11	0.08	0.92
Solar Cells, GaAs	12	0.88	0.12
Solar Cells, Silicon	13	0.75	0.25
Titanium (Polished)	14	0.6	0.4
White Paint (Silicate)	15	0.12	0.88
White Paint (Silicone)	16	0.26	0.74

2.10.4.3 Atmospheric Drag Torque Calculation

Just as photons have momentum and create pressure upon impacting a spacecraft, air particles also have momentum and create pressure when they impact a spacecraft. The density of air and pressure decrease exponentially with increasing altitude. As a result, only spacecraft in

LEO encounter enough particles to justify an atmospheric drag torque calculation. The atmospheric drag was estimated by:

$$T_a = \frac{1}{2}\rho C_d A_r V^2 (cp_a - cm)$$

$\boldsymbol{T_a}$ = ***Atmospheric Drag Toque*** $(\boldsymbol{Nm})$
ρ = ***Density of Air*** $(\boldsymbol{kg/m^3})$
$\boldsymbol{C_d}$ = ***Drag Coefficient*** $(\boldsymbol{m})$
$\boldsymbol{A_r}$ = ***Ram Area*** $(\boldsymbol{m^2})$
$\boldsymbol{V}$ = ***Velocity*** $(\boldsymbol{m^2})$
$\boldsymbol{cp_a}$ = ***Aerodynamic Pressure Center*** $(\boldsymbol{m})$
$\boldsymbol{cm}$ = ***Center of Mass*** $(\boldsymbol{m})$

Figure 2-12 Atmospheric Drag Torque Equation

Like SRP, when the center of atmospheric pressure, determined by the spacecraft area exposed to the atmosphere in the direction of the orbital velocity (i.e. ram direction), is not aligned with the center of mass, a torque occurs. The density of air is estimated using an upper atmospheric model derived by NASA:

$$T = -131.21 + .00299h$$

$$P = 2.488\left(\frac{(T + 273.1)}{216.6}\right)^{-11.388}$$

$$\rho = \frac{P}{(.2869(T + 273.1))}$$

$\boldsymbol{T}$ = ***Temperature*** (℃)
P = ***Pressure*** $(\boldsymbol{kPa})$
$\boldsymbol{\rho}$ = ***Density of air*** $(\boldsymbol{kg/m^3})$

Figure 2-13 Upper Atmospheric Air Model Equations [8]

The drag coefficient was estimated at a constant 2.2 (as is common practice for a LEO flying satellite). The ram area was calculated for the worst-case scenario by determining the largest area of all the sides. Lastly, the maximum distance from the aerodynamic center of pressure to the center of mass was calculated.

2.10.4.4 Magnetic Torque Calculation

The Earth's liquid core is a dynamo that induces a magnetic field. This magnetic field is strong enough to generate effects on the space surrounding Earth, so strong in fact that it interacts with the satellite's weak magnetic residual moment. When the satellite's residual moment is not aligned to the local magnetic field from Earth, the satellite experiences a magnetic torque that attempts to align the two. The magnitude of this magnetic torque can be calculated using the equation presented in Figure 2-14.

$$T_m = DB = D\frac{M}{R^3}\lambda$$

$$\begin{aligned} \boldsymbol{T_g} &= \boldsymbol{Gravity - Gradient\ \ Torque}\ (\boldsymbol{N \cdot m}) \\ \boldsymbol{\mu} &= \boldsymbol{Gravitational\ Constant}\ (\boldsymbol{m^3/s}) \\ \boldsymbol{R} &= \boldsymbol{Distance\ to\ Earth's\ Center}\ (\boldsymbol{m}) \\ \boldsymbol{I_z} &= \boldsymbol{Moment\ of\ Inertia\ about\ Z}\ (\boldsymbol{kg \cdot m^2}) \\ \boldsymbol{I_y} &= \boldsymbol{Moment\ of\ Inertia\ about\ y}\ (\boldsymbol{kg \cdot m^2}) \\ \boldsymbol{\theta} &= \boldsymbol{Angle\ between\ Vertical\ and\ Z\ Axis} \end{aligned}$$

Figure 2-14 Magnetic Torque Equation

This equation models the Earth's magnetic field as a dipole. The spacecraft's dipole moment is assumed to be 1 for a small, uncompensated spacecraft. Earth's magnetic constant is estimated at 7.18×10^{15} Tm^3. Lastly, lambda is a unitless function of the magnetic latitude that ranges from 1 at the magnetic equator to 2 at the magnetic poles.

2.10.4.5 Gravity-Gradient Torque Calculation

The final torque requiring computation was the gravity-gradient torque. Gravity-gradient torques arise when the spacecraft's center of gravity is not aligned with its center of mass with respect to the local vertical. The gravity-gradient torque increases as a function of the angle between the local vertical and the spacecraft's principal axes with the gravity-gradient torque always trying to align the minimum principal axis with the local vertical. Figure 2-15 provides a simplified equation for the gravity-gradient torque for a spacecraft with the minimum principal axis in its Z-direction is shown below. Earth's gravitational constant of $3.986\times 10^{14}\ \frac{\mathrm{m}^3}{\mathrm{s}}$ and a theta value equal to 45 degrees (i.e. worst-case scenario) are selected.

$$T_g = \frac{3\mu}{2R^3}|I_z - I_y|\sin 2\theta$$

$\boldsymbol{T_g}$ $= \boldsymbol{Gravity - Gradient\ Torque\ (N \cdot m)}$
$\boldsymbol{\mu}$ $= \boldsymbol{Gravitational\ Constant\ (m^3/s)}$
$\boldsymbol{R}$ $= \boldsymbol{Distance\ to\ Earth's\ Center\ (m)}$
$\boldsymbol{I_z}$ $= \boldsymbol{Moment\ of\ Inertia\ about\ Z\ (kg \cdot m^2)}$
$\boldsymbol{I_y}$ $= \boldsymbol{Moment\ of\ Inertia\ about\ y\ (kg \cdot m^2)}$
$\boldsymbol{\theta}$ $= \boldsymbol{Angle\ between\ Vertical\ and\ Z\ Axis}$

Figure 2-15 Gravity-Gradient Torque Equation

2.10.4.6 Reaction Wheel Sizing

The reaction wheels were sized for cyclical momentum storage. To determine the mass of a reaction wheel, data was collected on several commercially available reaction wheels. With the data gathered in MATLAB, a fourth-order polynomial curve was fit to the data to compare the momentum storage capabilities of the reaction wheels and their weight in kilograms. From the polynomial curve, an approximate mass was selected based off the momentum storage capabilities.

2.10.4.7 ACS Propulsion Sizing

The ACS propulsion sizing requires five inputs: the mass of the spacecraft, the center of gravity, satellite lifetime, the saturation point of a reaction wheel (secular angular momentum), and the rate of saturation of a reaction wheel. These were used to compute the momentum dumping fuel mass required over the satellite lifetime and the thrust required for a single momentum dump. Although the proposed design includes four thrusters for momentum dumping and one main thruster, this code utilizes six thrusters (one on each side) and chooses the shortest moment arm which will require the largest thrust to design for the worst-case scenario. The author assumed that the thruster required for momentum dumping will fire for one second.

With the worst-case moment arm calculated, the thrust required to dump the momentum is calculated (per pulse) using the following equation.

$$T = \frac{h}{Lt}$$

$\boldsymbol{T}$ $= \boldsymbol{Thrust\ (N)}$
$\boldsymbol{h}$ $= \boldsymbol{Stored\ Wheel\ Momentum\ (m^3/s)}$
$\boldsymbol{L}$ $= \boldsymbol{Thruster\ Moment\ Arm\ (m)}$
$\boldsymbol{t}$ $= \boldsymbol{Burn\ Time\ (s)}$

Figure 2-16 Thrust Per Momentum Dump Equation

The propellant mass required for ACS over the satellite lifetime is then calculated with these equations.

$$\sum_{Pulses} = \frac{3 \cdot Life_{Sat} \cdot 365.25}{Sat.Rate}$$

$$m_{ACSfuel} = \frac{T \cdot \sum_{Pulses} \cdot t}{I_{sp} \cdot g}$$

$\boldsymbol{\sum_{Pulses}} = \boldsymbol{\#\ of\ Thruster\ Firings}$
$\boldsymbol{Life_{Sat}} = \boldsymbol{Satellite\ Lifespan\ (years)}$
$\boldsymbol{Sat.Rate} = \boldsymbol{Wheel\ Saturation\ Rate\ (\frac{day}{sat.})}$
$\boldsymbol{m_{ACSfuel}} = \boldsymbol{ACS\ Fuel\ (kg)}$
$\boldsymbol{T} = \boldsymbol{Thrust\ (N)}$
$\boldsymbol{t} = \boldsymbol{Time\ (s)}$
$\boldsymbol{I_{sp}} = \boldsymbol{Specific\ Impulse\ (s)}$
$\boldsymbol{g} = \boldsymbol{Gravity\ at\ Sea\ Level\ (m/s^2)}$

Figure 2-17 ACS Fuel Required Equation

$\sum_{Pulses}$ is the total number of thruster firings required throughout the lifetime of the spacecraft to ensure that the reaction wheels can control the attitude of the spacecraft. The numerator of the first equation in Figure 2-17 has a factor of 3 because three reaction wheels will need to be desaturated each time a momentum dump is required. Lastly, since hydrazine was selected as the fuel of choice, the I_{sp} is 218 seconds.

2.10.4.8 ACS Model Outputs

This completes the ACS model process. In the Command Window of MATLAB, the following items are printed: cyclical angular momentum per orbit, secular angular momentum per orbit, reaction wheel mass, the fuel required for momentum dumping, and the thrust required per momentum dump. In addition, a polar graph showing the individual and total torques over the entire orbit is displayed. An example output screen is shown in Figure 2-18.

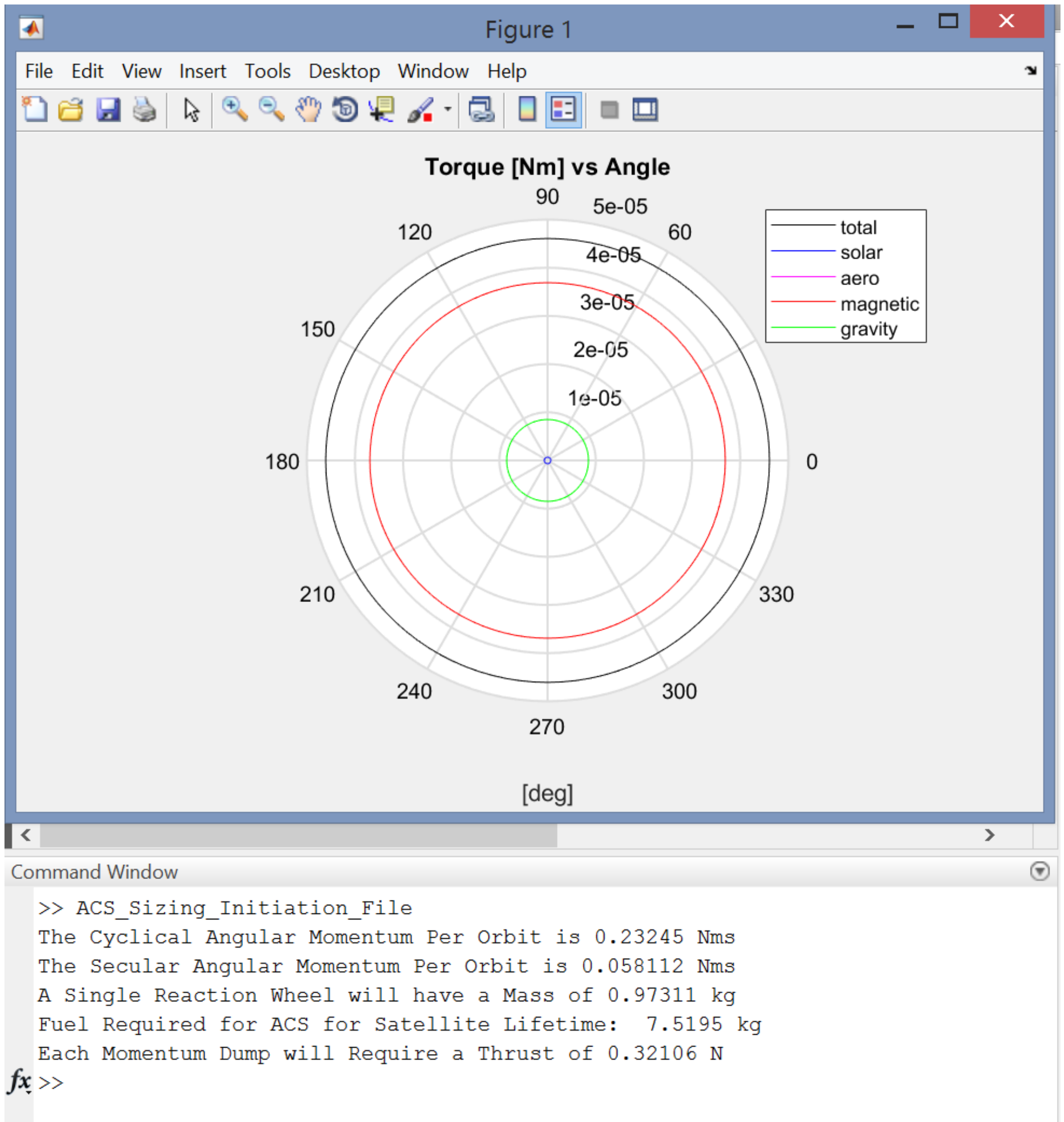

Figure 2-18 ACS Model Output

2.11 Cost Model

With the satellite designed, a parametric cost estimating model can be selected to price the constellation. A publicly available special purpose model was selected due to its free nature. While *SME: The New SMAD* [1] presents several publicly available special purpose cost estimating models, the SSCM is selected because of its usefulness in pricing spacecraft weighing less than 500 kg. Developed by The Aerospace Corporation, it is assumed that these cost estimating relationships (CERs) include the cost of contractor program management, systems engineering, product assurance, and I&T. The equations used to determine the CERS are shown below, and FY10 dollars have been adjusted to FY18 when utilized.

Table 2-5 SSCM Earth Orbiting Total Non-Recurring Cost Equations (Development Plus One Protoflight Unit)

SME-SMAD WBS Element	CER Y = total non-recurring cost of development plus one protoflight flight unit in FY10 $K	Cost Driver(s)	Cost Driver Input Range	Standard Error of Estimate (absolute) FY10 $K
1.1 Spacecraft Bus (Alternate CER if no component information available)	$Y = 1{,}064 + 35.5X^{1.261}$	X = Spacecraft Bus Dry Weight (kg)	2-400 kg	3,696
1.1.1 Structure	$Y = 407 + 19.3X + \ln X$	X= Structure Weight (kg)	5-100 kg	1,097
1.1.2 Thermal Control	$Y = 335 + 5.7X^2$	X = Thermal Control Weight (kg)	5-12 kg	119
1.1.3 ADCS	$Y = 1{,}850 + 11.7X^2$	X = ADCS Dry Weight (kg)	1-25 kg	1,113
1.1.4 Electrical Power Supply (EPS)	$Y = 1{,}261 + 539X^{0.72}$	X = EPS Weight (kg)	7-70 kg	910
1.1.5 Propulsion (Reaction Control)	$Y = 89 + 3.0X^{1.261}$	X = Spacecraft Bus Dry Weight (kg)	20-400 kg	310
1.1.6a Telemetry, Tracking, & Command (TT&C)	$Y = 486 + 55.5X^{1.35}$	X = TT&C Weight (kg)	3-30 kg	629
1.1.6b Command & Data Handling (CD&H)	$Y = 658 + 75X^{1.35}$	X = CD&H Weight (kg)	3-30 kg	854
1.2 Payload	Y = 0.4X	X = Spacecraft Bus Total Cost ($K)	2,600-69,000 ($K)	
1.3 Integration, Assembly, & Test	Y = 0.139X	X = Spacecraft Bus Total Cost ($K)	2,600-69,000 ($K)	
4.0 Program Level	Y = 0.229X	X = Spacecraft Bus Total Cost ($K)	2,600-69,000 ($K)	
5.0 Launch & Orbital Operations Support (LOOS)	Y = 0.061X	X = Spacecraft Bus Total Cost ($K)	2,600-69,000 ($K)	
6.0 Ground Support Equipment (GSE)	Y = 0.066X	X = Spacecraft Bus Total Cost ($K)	2,600-69,000 ($K)	

In addition to providing the equations, *SME: The New SMAD* [1] also provides an interactive Excel sheet for the SSCM that is ready to use. Figure 2-19 presents a screenshot of the Excel file used to estimate the cost of the FireSat II spacecraft.

User Inputs in Orange

Computation of Spacecraft computer source lines of code				
Kwords	bits/Kwords	Bytes/bit	SLOC/Byte (C code)	Source Lines of Code Converted from Kwords of Code:
781.35	16,384	0.125	0.02381	38,100

Learning Curve	
Number of Units Manufactured, N	2
Learning Curve Slope S =	0.95
B = 1 - (ln(1.0/S)/ln(2))	0.926
Learning Curve Multiplication Factor, L	1.90

Design Parameter	Mass (kg)	Average Power (W)	Percentage of Dry Mass
Payload	26.0	65.0	30.2%
S/C Subsystems	60.0	76.0	69.8%
ADCS	6.0	14.0	7.0%
C&DH	4.0	17.0	4.7%
Power	18.0	13.0	20.9%
Propulsion	4.0	0.0	4.7%
Structure	23.0	1.0	26.7%
Thermal	2.0	14.0	2.3%
TT&C (Comm)	3.0	17.0	3.5%
Margin	14.0	42.3	
Spacecraft Dry Mass	100.0	183.3	

Spacecraft Power Margin
30%

Small Spacecraft Cost Model											
Cost Component	CER Input Parameter	Value	Units	RDT&E plus 1st Unit Cost (FY10$K)	2nd Unit Cost (FY10$K)	Total Cost 2010 (FY10$K)	Total Cost 2012	Std Error ($K) 2010 (FY10$K)	Std Error ($K) 2012	Std Error Percentage**	SEE (FY$2010)
1.1 Spacecraft Bus											$3,696
1.1.1 Structure*	Mass	26.7	kg	$2,103	$1,893	$3,996	$4,119	$2,084	$2,149		$1,097
1.1.2 Thermal*		2.3	kg	$366	$329	$695	$716	$226	$233		$119
1.1.3 ADCS*	Mass	7.0	kg	$2,419	$2,178	$4,597	$4,739	$2,115	$2,180		$1,113
1.1.4 Electrical Power System*	Mass	20.9	kg	$3,553	$3,198	$6,752	$6,960	$1,729	$1,782		$910
1.1.5 Propulsion*	Spacecraft Bus Dry Mass	69.8	kg	$723	$651	$1,373	$1,416	$589	$607		$310
1.1.6a TT&C*	Mass	3.5	kg	$786	$707	$1,493	$1,539	$1,195	$1,232		$629
1.1.6b Command & Data Handling	Mass	4.7	kg	$1,255	$1,130	$2,385	$2,459	$1,623	$1,673		$854
1.1.7 Integration, Assembly, & Test (IA&T)	Spacecraft Bus Dry Mass	$11,206	FY10$K	$1,558	$1,402	$2,960	$3,051	$888	$915	30%	
1.1.9 Flight Software†	Source Lines of Code	38,100	N/A	$20,955	$0	$20,955	$21,601	$6,287	$6,480	30%	
Spacecraft Bus Total Cost				$33,719	$11,487	$45,206	$46,599	$7,522	$7,753		
1.2 Payload											
1.2.2 IR Sensor	Spacecraft Bus Total Cost	$11,206	FY10$K	$4,482	$4,034	$8,517	$8,779	$2,555	$2,634	30%	
2. Launch Segment	2 Minotaur Launches	$50,000	FY10$K	$25,000	$25,000	$50,000	$51,541	$5,000	$5,154	10%	
4. Program Level	Spacecraft Bus Total Cost	$11,206	FY10$K	$2,566	$2,310	$4,876	$5,026	$1,463	$1,508	30%	
5. Launch and Orbital Ops Support (LOOS)	Spacecraft Bus Total Cost	$11,206	FY10$K	$684	$615	$1,299	$1,339	$390	$402	30%	
6. Ground Support Equipment (GSE)	Spacecraft Bus Total Cost	$11,206	FY10$K	$740	$666	$1,405	$1,449	$422	$435	30%	
Total Space Segment Cost to Contractor				$41,507	$18,497	$60,004	$61,853				
10% Contractor Fee				$4,151	$1,850	$6,000	$6,185				
Total Space Segment Cost to Government				$45,658	$20,346	$66,004	$68,038				
Total Cost of Deployment						**$116,004**	**$119,580**	**9,517**	**$9,810**		

Figure 2-19 FireSat II SSCM Excel File

To estimate the cost of the entire constellation (total lot cost), the following equation can be used.

$$Total\ Lot\ Cost = T1 \cdot N^{\left(1+\frac{\ln S}{\ln 2}\right)}$$

$$\begin{aligned} &\boldsymbol{Total\ Lot\ Cost} = \boldsymbol{Constellation\ Cost}\ (\$) \\ &\boldsymbol{T1} = \boldsymbol{Theoretical\ First\ Unti\ Cost}\ (\boldsymbol{S}) \\ &\boldsymbol{N} \quad = \boldsymbol{Number\ of\ Satellites} \\ &\boldsymbol{S} \quad = \boldsymbol{Learning\ Curve\ Slope} \end{aligned}$$

Figure 2-20 Total Lot Cost Equation

The theoretical first unit cost is predicted by the SSCM Research, Development, Testing, and Evaluation (RDT&E) plus 1st Unit Cost estimate. The number of satellites is equal to the number of satellites in the constellation. Lastly, the learning curve slope is estimated at 95% based on traditional learning theory estimates.

While the SSCM predicts the development plus one protoflight unit cost, it does not predict the launch cost. As this research effort design a constellation of 72 satellites, predicting the cost to launch all satellites is complicated. Table 11-23 “Historical Launch Vehicle Costs” in *SME: The New SMAD* [1] provides the mass capacity for several launch vehicles. The largest mass capacity of a vehicle still in use is the Delta IV Heavy which boasts a mass capacity of 22,560 kg. Using the mass capacity, a rough estimate can be made to determine how many satellites a single Delta IV Heavy or similar launch vehicle could carry; however, caution should be used with this method. Volume constraints also become a concern when dealing with such a high quantity of satellites. After adding the SSCM estimate and the launch vehicle(s) cost, the design-to-orbit cost is complete.

3 Validation Efforts

3.1 Introduction

While the entire design cannot be validated due to its specific mission objectives, several of the developed models can be validated against the FireSat II spacecraft, a textbook example from *SME: The New SMAD* [1]. Overall, the satellite power system model, the propulsion system mass model, and the ADCS Model were validated.

3.2 Satellite Power System Model

An example test run was executed using the data provided by Table 21-12 “Solar Array Design Process” and Table 21-19 “Steps in the Energy Storage Subsystem Design” in *SME: The New SMAD* [1] for the FireSat II spacecraft. The results of the validation test are shown in Table 3-1.

Table 3-1 FireSat II Example Vs Satellite Power System Model

	FireSat II	Model	% Difference
Solar Array Area (m^2)	2.0	1.99	0.5
Solar Array Mass (kg)	9.6	9.42	1.88
Battery Capacity (W·hr)	119	116.34	2.24

With a maximum difference of 2.24 %, the satellite power system has been verified to be accurate.

3.3 Propulsion System Mass Model

To verify the propulsion system mass model, data provided in section 18.8 of *SME: The New SMAD* [1] was utilized. Table 3-2 contains the results of the comparison between the FireSat II spacecraft and the model.

Table 3-2 FireSat II Example Vs Propulsion System Mass Model

	FireSat II	Model
Tank (kg)	8.2	8.2
Feed System (kg)	3.6	3.6
Propellant (kg)	41.5	41.5
Pressurant (kg)	0.13	0.13

As expected, no difference exists between the two because the same equations were used and no values were dynamically computed unlike the satellite power system where STK had to be used to solve for a daylight and eclipse period length that changes constantly based on the time period for the scenario of interest. Subsequently, the propulsion system mass model has been verified as being accurate when given the correct dry mass, ACS fuel estimate, and ΔV values.

3.4 ADCS Model

Verification of the ADCS model was made difficult due to the FireSat II spacecraft's lack of information directly related to using thrusters for momentum dumping. Upon researching other potential verification methods, *Space Mission and Design* (SMAD) [9] had information on the cyclical angular momentum per orbit and the fuel required for ACS for the original FireSat spacecraft. The numbers used for verification of the model are found in Table 11-13 "Simplified Equations for Preliminary Sizing of Thruster Systems".

Table 3-3 FireSat Example Vs ADCS Model

	FireSat Example	ADCS Model	% Difference
Cyclical Angular Momentum Per Orbit (Nms)	0.4	0.3815	4.625
ACS Fuel Required (kg)	2.43	2.38	2.06

While this verification has the highest percent difference, less than five percent difference is more than acceptable.

4 Results

4.1 Coverage Model

The results from the six iterations are shown below.

Table 4-1 Average % Coverage Over CONUS Per Number of Transmitters on Each Satellite

Number of Transmitters	Transmitter Orientation(s)	Average Percent Coverage (%)	Average Area Covered (km²)
1	Down Towards Earth	***0.011944***	***10,579.33***
2	Down Towards Earth and Left Face of Satellite	**32.89**	**29,132,691.79**
3	Down Towards Earth and Left and Right Face of Satellite	**55.12**	**48,822,630.89**
4	Down Towards Earth and Left, Right, and Front Face of Satellite	**76.014**	**67,329,614.69**
5	Down Towards Earth and Left, Right, Front, and Back Face of Satellite	**94.69**	**83,870,816.93**
6	All Sides	**94.69**	**83,870,816.93**

Initial estimates of a reasonable percentage coverage over CONUS to justify the cost of this program were placed at a minimum of 90 percent by MITRE personnel. Based on the numbers gathered through the analysis shown above, five transmitters are the optimal number. After reviewing the power consumption requirements per transmitter (389 Watts), it quickly became apparent that five transmitters on each satellite is not effective due to the increased size of solar panels needed to produce enough power for a five sensor orientation. As the solar panels grow in scale, the rest of the satellite would also have to grow to support the added weight and thermal needs. As a result, a two-transmitter orientation with one transmitter looking down and one transmitter on one of the sides of the satellite was selected. Because of this orientation, the satellite would require an orientation maneuver to communicate with any space object that is not in the field of view of one of the two RF transmitters at a given instance to provide an acceptable percentage coverage.

4.2 Satellite Power Budget

Using the equations from *SME: The New SMAD* [1], a first order power estimate for the satellite was placed at 2,387 Watts. This estimate is reasonably close to the following final power budget which was derived through the design iteration process.

Table 4-2 Final Power Budget

Item	Watts (W)	Quantity Needed	Total Power Required (W)
RF Transmitter	389	2	778
RF Transmitter Computer	178	1	178
ConLCT (Laser) [10]	80	4	320
Reaction Wheels (Blue Canyon Tech RWP500) [11]	6	3	18
Star Tracker VST-41M [12]	2.5	1	2.5
Fine Sun Transmitter [13]	0.25	1	0.25
Main Thruster (Monarc-445)	58	1	58
Momentum Dumping Thrusters (MRE-1.0)	15	4	60
Thermal Control	238.7	1	238.7
On-Board Processing	286.44	1	286.43
Power	214.83	1	214.83
Structure and Mechanism	23.87	1	23.87
Total			**2178.58**

As previously discussed, in order to accurately model the situation that the satellite will not always be operating at full power, it is assumed that the satellite operates a third of the day at full-power, a third of the day at half-power, and a third of the day at no-power. Furthermore, it is assumed that you can only communicate at full-power during sunlight periods where the satellite is able to use power from both batteries and the solar panels. Thus, the power used for all subsequent calculations for the final design is 1,089.3 Watts.

4.3 Satellite Power System Results

Using the 1,089.3 Watts estimate, the Satellite Power System model is used to calculate the solar array area, solar array mass, battery capacity, and battery mass. Figure 4-1 shows the results of the satellite power system model.

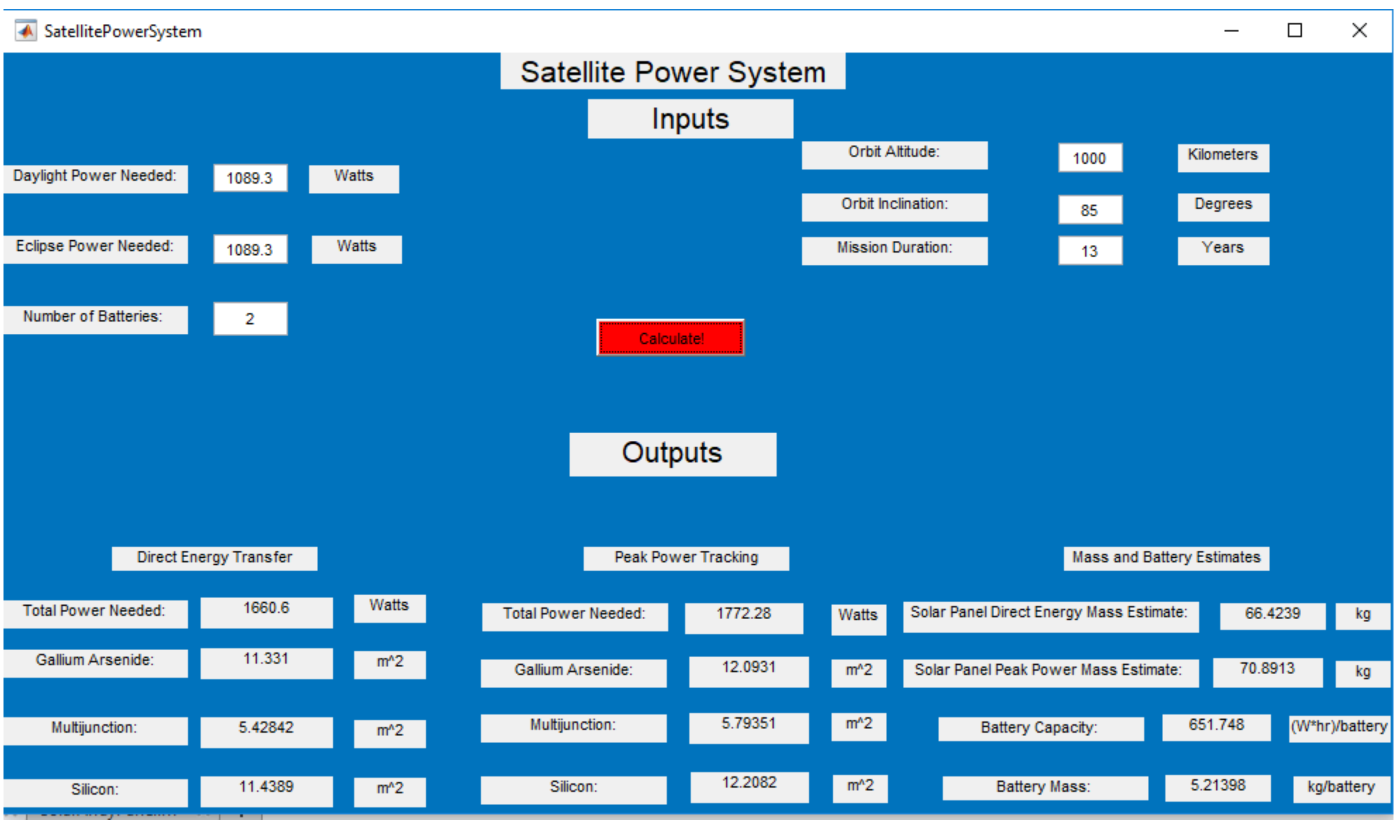

Figure 4-1 Satellite Power System Results

Based on these results, multi-junction cells using direct energy transfer were selected for the final design. An estimated 5.43 m^2 of solar array weighing 66.42 kg is required to meet the daylight and eclipse power requirements. Furthermore, to store the energy from the solar array, a battery with a capacity of 1,303.5 W·hr weighing 10.43 kg is needed.

4.4 Satellite Mass Results

With the results from the coverage model and the satellite power system model, the spacecraft bus dry mass results using the equations from *SME: The New SMAD* [1] are shown below for the satellite designed to complete MITRE's proposed mission.

Table 4-3 Spacecraft Bus Dry Mass Estimate Using *SME: The New SMAD* [1] Equations

	Mass (kg)
Payload	64.5
Structure	56.25
Thermal	4.16
Power	76.85
TT&C	4.16
On-Board Processing	10.41
ADCS	12.49
Propulsion	6.24
Other	6.24
Total Dry	**241.3**

Again, this dry mass estimation method proves extremely effective compared to the final dry mass estimate derived from a part list created after multiple iterations of the design process. Table 4-4 shows the final dry and wet mass estimates.

Table 4-4 Satellite Dry and Wet Mass Results

Item	Quantity	Mass (kg)	Total Mass (kg)
RF Transmitter	2	2.27	4.54
RF Computer	1	2.27	2.27
ConLCT (Laser)	4	15	60
Reaction Wheels (Blue Canyon Tech RWP500)	3	0.75	2.25
Star Tracker VST-41M	1	0.9	0.9
Fine Sun Transmitter	1	0.375	0.375
Main Thruster (Monarc-445)	1	1.59	1.59
Momentum Dumping Thrusters (MRE-1.0)	4	1	4
Surface Tension Propellant Tank OST 33/0 [14]	1	13.5	13.5
Propellant Feed System	1	5.87	5.87
Solar Panels	1	66.42	66.42
VES 180 Batteries [15]	7	1.11	7.77
VES 100 Battery [15]	1	0.81	0.81
Structure	1	56.21	56.21
Thermal Control	1	4.16	4.16
TT&C	1	4.16	4.16
On-Board Processing	1	10.41	10.41
Other (Balance+Launch)	1	6.25	6.25
Total Dry Mass			**251.49**
Propellant for ΔV Budget	1	95.54	95.54
Propellant for Momentum Dumping	1	7.59	7.59

Total Wet Mass			**103.13**
Total Mass			**354.62**

With a total mass of 354.62 kg, the satellite remains in the SSCM range.

4.5 Satellite Dimensions

During the creation of the satellite's part list, the dimensions of each part is also recorded. Using this part list, a SolidWorks assembly was created to estimate the overall dimensions of the satellite, leaving a small portion of volume for parts whose size could not be easily estimated. The dimensions of the various parts and a picture of the SolidWorks assembly are shown below (Note: The solar panels are not shown).

Table 4-5 Satellite Dimensions

Item	Dimensions (m or m^2)
RF Transmitter	**Length:** 0.127; **Width:** 0.089; **Height:** 0.165
RF Computer	**Length:** 0.14; **Width:** 0.131; **Height:** 0.159
ConLCT (Laser)	**Length:** 0.76; **Width:** 0.29; **Height:** 0.435
Reaction Wheels (Blue Canyon Tech RWP500)	**Length:** 0.11; **Width:** 0.11; **Height:** 0.038
Star Tracker VST-41M	**Length:** 0.08; **Width:** 0.10; **Height:** 0.18
Fine Sun Transmitter	**Length:** 0.108; **Width:** 0.108; **Height:** 0.0525
Main Thruster (Monarc-445)	**Length:** 0.41; **Exit Diameter:** 0.148
Momentum Dumping Thrusters (MRE-1.0)	**Length:** 0.188; **Width:** 0.114
Surface Tension Propellant Tank OST 33/0	**Height:** 0.896; **Diameter:** 0.6
Solar Panels	**Area:** 5.43
VES 180 Battery	**Height:** 0.25; **Diameter:** 0.053

VES 100 Battery	**Height:** 0.185; **Diameter:** 0.054
Overall Dimensions (Structure):	**Length:** 1.143; **Width:** 0.724; **Height:** 0.635

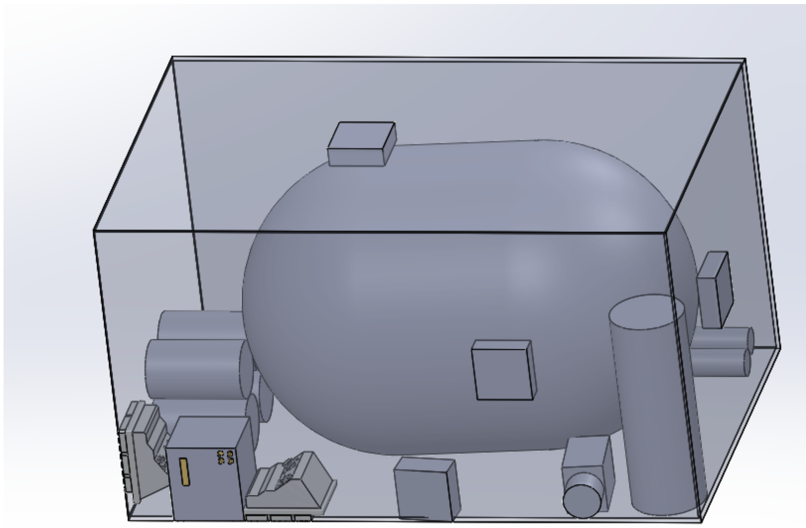

Figure 4-2 Satellite SolidWorks Assembly

Due to time constraints, some simplifications were made for a few of the part drawings. For example, the thrusters are simplified as cylinders. Lastly, as stated earlier, some of the volume is reserved for parts that could not be sized easily as shown in Figure 4-2.

4.6 Satellite Cost Model

Finally, the SSCM was utilized to price the proposed satellite constellation. The estimated cost for the research, development, test, and evaluation (RDT&E) plus the first unit was $74.245 M in FY$2019 dollars while the estimated cost for the RDT&E and the building of the entire constellation is $3.23 B in FY$2019. This cost does not include the cost of launching the constellation into orbit. Figure 4-3 displays the derivation of the costs provided above.

User Inputs in Orange

Computation of Spacecraft computer source lines of code				
Kwords	bits/Kwords	Bytes/bit	SLOC/Byte (C code)	Source Lines of Code Converted from Kwords of Code:
781.35	16,384	0.125	0.02381	38,100

Learning Curve	
Number of Units Manufactured, N	72
Learning Curve Slope S =	0.95
B = 1 - (ln(1.0/S)/ln(2))	0.925999
Learning Curve Multiplication Factor, L	52.47

Design Parameter	Mass (kg)	Average Power (W)	Percentage of Dry Mass
Payload	64.5	1098.0	26.3%
S/C Subsystems	180.3	902.6	73.7%
ADCS	7.5	80.8	3.1%
C&DH	10.4	286.4	4.3%
Power	76.8	214.8	31.4%
Propulsion	21.0	58.0	8.6%
Structure	56.2	23.9	23.0%
Thermal	4.2	238.7	1.7%
TT&C (Comm)	4.2	0.0	1.7%
Other	6.2	178.0	
Spacecraft Dry Mass	251.0	2178.6	

Spacecraft Power Margin
0%

Small Spacecraft Cost Model											
Cost Component	CER Input Parameter	Value	Units	RDT&E plus 1st Unit Cost (FY10$K)	2nd - 72nd Unit Cost (FY10$K)	Total Cost 2010 (FY10$K)	Total Cost 2019	Std Error ($K) 2010 (FY10$K)	Std Error ($K) 2019	Std Error Percentage**	SEE (FY$2010)
1.1 Spacecraft Bus											$3,696
1.1.1 Structure*	Mass	56.2	kg	$4,778	$245,910	$250,688	$298,933	$57,557	$68,634		$1,097
1.1.2 Thermal*	Mass	4.2	kg	$434	$22,318	$22,752	$27,131	$6,244	$7,445		$119
1.1.3 ADCS*	Mass	7.5	kg	$2,513	$129,313	$131,825	$157,195	$58,396	$69,635		$1,113
1.1.4 Electrical Power System*	Mass	76.8	kg	$11,019	$567,138	$578,157	$689,426	$47,745	$56,934		$910
1.1.5 Propulsion*	Spacecraft Bus Dry Mass	251.0	kg	$3,275	$168,533	$171,808	$204,873	$16,265	$19,395		$310
1.1.6a TT&C*	Mass	4.2	kg	$866	$44,584	$45,450	$54,197	$33,002	$39,353		$629
1.1.6b Command & Data Handling	Mass	10.7	kg	$2,492	$128,252	$130,744	$155,906	$44,807	$53,430		$854
1.1.7 Integration, Assembly, & Test (IA&T)	Spacecraft Bus Dry Mass	$25,376	FY10$K	$3,527	$181,540	$185,068	$220,685	$55,520	$66,205	30%	
1.1.9 Flight Software†	Source Lines of Code	38,100	N/A	$20,955	$0	$20,955	$24,988	$6,287	$7,496	30%	
Spacecraft Bus Total Cost				$49,859	$1,487,587	$1,537,446	$1,833,334	$124,599	$148,578		
1.2 Payload											
1.2.2 IR Sensor	Spacecraft Bus Total Cost	$25,376	FY10$K	$10,150	$522,419	$532,569	$635,064	$159,771	$190,519	30%	
2. Launch Segment	2 Minotaur Launches	$0	FY10$K	$0	$0	$0	$0	$0	$0	10%	
4. Program Level	Spacecraft Bus Total Cost	$25,376	FY10$K	$5,811	$299,085	$304,896	$363,574	$91,469	$109,072	30%	
5. Launch and Orbital Ops Support (LOOS)	Spacecraft Bus Total Cost	$25,376	FY10$K	$1,548	$79,669	$81,217	$96,847	$24,365	$29,054	30%	
6. Ground Support Equipment (GSE)	Spacecraft Bus Total Cost	$25,376	FY10$K	$1,675	$86,199	$87,874	$104,786	$26,362	$31,436	30%	
Total Space Segment Cost to Contractor				$67,495	$2,395,289	$2,462,784	$2,936,758				
10% Contractor Fee				$6,750	$239,529	$246,278	$293,676				
Total Space Segment Cost to Government				$74,245	$2,634,818	$2,709,063	$3,230,434				
Total Cost of Deployment						**$2,709,063**	**$3,230,433.90**	**225,181**	**$268,519**		

Figure 4-3 Satellite Cost Results

An estimate for the cost of launching the constellation was also needed. The Delta IV Heavy was selected as the launch vehicle. With the ability to take a payload mass of 22,950 kg to LEO, the Delta-IV heavy could take 70 of the 72 satellites at an estimated cost of $215 M. With only two satellites left, a Minotaur IV was selected at a cost of $22 M. Thus, the total launch cost was estimated at $237 M. Finally, the design-to-orbit cost was calculated by adding the RDT&E, manufacturing, and launch costs and was estimated at $3.467 B.

5 Conclusion

Design optimizations of satellite concepts were performed to demonstrate the communication capability over CONUS between low flying objects, LEO satellites and ground terminals. The analysis from these optimizations has shown encouraging results with respect to performance capabilities; however, a constellation design-to-orbit cost of $3.467 B is likely limiting.

The satellite design processes undertaken in this technical report provide insight into possible satellite design improvements. The primary objective has been generating useful results that can aid MITRE and its sponsors in the systems engineering process at the preliminary design and verification level. The goals of the design process have been to optimize the cost of the satellite while at the same time meeting the desired performance requirements.

Upon review of the results, the satellite design does meet the requirements provided by MITRE. Coverage over CONUS is provided at an acceptable rate; however, the cost of the constellation is limiting. The initial goal for total constellation cost was between $500 M and $1 B. This optimized satellite design employs conventional forms of subsystem designs based on heritage satellite technology. It is likely that additional improvements in technology will develop between now and implementation of such a satellite. The use of composite materials along with improvements in monopropellants could bring about better performing and more cost-effective solutions.

6 Recommended Improvements

6.1 Laser Communication

Although MITRE established a requirement that RF transmitters be used for communication between the satellite, low-flying objects, LEO satellites and the ground terminals, the ability to use the laser communication system for all communications would drastically reduce the size of the satellite. The power requirements of the RF transmitters are almost half of the total power budget. If the power budget could be reduced by half, the solar panel area will decrease greatly which will also cause a significant decrease in the mass of the satellite. This mass savings will also decrease the mass of other systems allowing for a significant cost savings to occur.

6.2 SolidWorks Drawing

The SolidWorks drawing should be updated to include more accurate renderings of a few of the parts. Time constraints resulted in some approximation methods being utilized such as thrusters being approximated by a cylinder. More accurate renderings would allow for better visualization and center of mass approximations.

6.3 SSCM

The SSCM is considered the best small satellite cost model living up to its namesake. While the latest version of *SME: The New SMAD* [1] was utilized, the SSCM provided in the textbook is the 1996 version. Since the 1996 version was released, society has seen the creation and exponential growth in the development of cubesats, microsatellites, and small satellites thus greatly decreasing the cost to design and build a satellite. The latest version of SSCM should be acquired from The Aerospace Corporation to verify and update the cost model.

7 References

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Appendix A Acronyms

ACS	Attitude and Control System
ADCS	Attitude, Determination, and Control System
BOL	Beginning-of-Life
CERs	Cost Estimating Relationships
CONUS	Continental United States
ΔV	Delta-V
DOD	Depth-of-Discharge
EOL	End-of-Life
GUI	Graphical User Interface
He	Helium
PMD	Propellant Management Devices
RF	Radio Frequency
RDT&E	Research, Development, Test, and Evaluation
STK	Systems Tool Kit
SMAD	Space Mission Analysis and Design
SSCM	Small Satellite Cost Model
SRP	Solar Radiation Pressure

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