The HAVEN Project

Helios Advanced Violent Eruption Notification Project

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Dear Commercial Space Entrepreneur:

On 02.III.2009 the University of Michigan's Aerospace 483 design course was commissioned the task to create two satellite architectures to observe the Sun and provide a warning if a solar particle event (SPE) occurs. SPEs present a unique problem to the human presence in space because of the danger they pose through the ejection of plasmas at high speeds.

Team Blue's study has produced one of the two mission architecture. Our study has found one feasible mission architecture. This report will delve further into the problems associated with SPEs, our satellite design and showing why our satellite is not only needed but is the best option for the Space Hotel Entrepreneur.

HAVEN-1 will provide in-situ (in place) measuring of accelerated particles and send back alerts and warnings when appropriate, this will aid the Space Hotel in securing the clientele from SPEs. As an added business opportunity the HAVEN Project architecture is capable of adapting to new instruments and can be readily deployed to other space locations. We would like to think of this as a medium size analogue to the CubeSat architecture.

Without our system your customers will be put in great peril with massive amounts of radiation released in a SPE. Therefore we recommend the following:

1. Invest in the HAVEN architecture to provide in-situ measurements to allow greater space access to your tourist.
2. Subscribe to NOAA Space Weather Prediction Center for the remote sensing capabilities for augmentation.

On behalf of Team Blue, I wish to express our appreciation to the Commercial Space Entrepreneur for reviewing our proposal.

Respectfully Submitted,

Richard B. Choroszucha,
Project Manager
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<tr>
<td>AFWA</td>
<td>Air Force Weather Agency (US)</td>
</tr>
<tr>
<td>APR</td>
<td>Array Power Regulator</td>
</tr>
<tr>
<td>BCDR</td>
<td>Battery Charge/Discharge Regulator</td>
</tr>
<tr>
<td>BCR</td>
<td>Battery Charge Regulator</td>
</tr>
<tr>
<td>BMA</td>
<td>Body Mounted Array</td>
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<tr>
<td>CME</td>
<td>Coronal Mass Ejection</td>
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<tr>
<td>COTS</td>
<td>Commercial off the shelf</td>
</tr>
<tr>
<td>CPTA</td>
<td>combination photovoltaic/thermovoltaic array</td>
</tr>
<tr>
<td>CPTA</td>
<td>Combinational Photovoltaic and Thermovoltaic Array</td>
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<tr>
<td>DET</td>
<td>Direct Energy Transfer</td>
</tr>
<tr>
<td>DMSP</td>
<td>Defense Meteorological Satellites Program</td>
</tr>
<tr>
<td>DoD</td>
<td>Department of Defense</td>
</tr>
<tr>
<td>DRA</td>
<td>Deployable Rigid Array</td>
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<tr>
<td>EOCV</td>
<td>End of charge voltage</td>
</tr>
<tr>
<td>EODV</td>
<td>End of discharge voltage</td>
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<tr>
<td>EP</td>
<td>Electric Propulsion</td>
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<tr>
<td>FDA</td>
<td>Flexible Fold-out Array</td>
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<tr>
<td>FMEA</td>
<td>Failure Mode and Effect Analysis</td>
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<tr>
<td>FRA</td>
<td>Flexible Roll-out Array</td>
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<tr>
<td>$I_{sp}$</td>
<td>Specific impulse</td>
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<tr>
<td>LFHC</td>
<td>Low-Frequency/ High-Consequence</td>
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<tr>
<td>LH2</td>
<td>Liquid Hydrogen ($H_2$)</td>
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<tr>
<td>LOX</td>
<td>Liquid Oxygen ($O_2$)</td>
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<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
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<tr>
<td>NOAA</td>
<td>National Oceanic and Atmospheric Administration</td>
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<tr>
<td>NSF</td>
<td>National Science Foundation</td>
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<tr>
<td>NSWP</td>
<td>National Space Weather Program</td>
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<tr>
<td>PCU</td>
<td>Power Conditioning Unit</td>
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<tr>
<td>PDM</td>
<td>Power Distribution Module</td>
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<tr>
<td>PMAD</td>
<td>Power Management</td>
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<tr>
<td>PPT</td>
<td>Point to Point Transfer</td>
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<tr>
<td>PRU</td>
<td>Power Regulator Unit</td>
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<tr>
<td>RTG</td>
<td>Radioisotope Thermoelectric generator</td>
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<tr>
<td>SKCC</td>
<td>Station-keeping &amp; course corrections stage</td>
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<tr>
<td>SPE</td>
<td>Solar Particle Event</td>
</tr>
<tr>
<td>SWPC</td>
<td>Space Weather Prediction Center</td>
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<td>TEG</td>
<td>Thermal electric generator</td>
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Executive Summary

As innovation pushes the limits of space exploration, space tourism continues to grow increasingly popular, potentially into a lucrative industry. The expansion and development of future commercial space ventures such as space hotels are limited only by our imagination and our ability to protect human life in an unpredictable space environment. Large solar storms and highly energetic particles in particular can endanger both human life and space based infrastructure.

With the ultimate goal of enabling a safer and increased human presence in space, Team Blue has designed a low-cost, operational alert system to warn space tourist of dangerous and oncoming solar storm events. This system monitors the Sun, detects adverse solar events, and delivers an accurate and prompt alert to Earth and its near-space operations. This alert system architecture has been designated as the HAVEN Project, or the Helios Advanced Violent Eruption Notification Project.

Once operational, the HAVEN Project alert system will utilize a combination of remote sensing resources within the Earth's near-space environment as well as a single deployable spacecraft to be placed at the first Sun-Earth Lagrange point ($L_1$). Earth-based remote sensors will be able to detect adverse solar disturbances as soon as they occur. A spacecraft placed at $L_1$ will then allow the direct sampling of oncoming highly energetic particles as they accelerate towards the Earth. Such an in-situ measurement confirms the oncoming storm and adds an additional degree of confidence and accuracy not available through remote sensing alone.

The HAVEN Project alert system can reliably warn of dangerous particles and give humans in space a thirty minute operational reaction time to seek shelter and protection from dangerous energetic material during large storm events.

At a time when there are no concrete plans to replace, reinforce, or enhance current alert capability, the HAVEN Project offers distinct advantages over other proposed alert systems. The HAVEN Project launches an operational solar sentinel spacecraft with in-situ sensing capability to $L_1$ for $25$ million dollars, a small cost relative to what has historically been spent on $L_1$ solar sentinel missions. Moreover, the HAVEN Project provides a 24-hour, uninterrupted space weather warning capability directly to space-based customers. This unprecedented ability makes the HAVEN project the ideal choice for start-up space tourist industries. With the success of the HAVEN Project comes the confidence and reliability needed to expand business ventures in space.
Part I

Introduction to the Mission
Chapter 1

Introduction

1.1 The Design Process

With the ultimate goal of enabling safer human presence in space, Team Blue set forth to design a spacecraft that could effectively monitor solar activity, detect adverse solar events, and finally deliver a prompt and accurate alert to Earth and various applications within its near space environment.

Although there are many inherent risks and engineering challenges associated with simply designing and launching a spacecraft into space, there were other considerable challenges to keep in mind at the start of our design process. Not only did we need to determine the motivation and high level objectives behind such a mission, but also how such a spacecraft and alert system would ultimately contribute to the future of operational alert capability in space. We wanted to pursue a mission that could potentially compete with, complement, or even replace, the dozens, if not hundreds, of government and commercial solar-studying spacecraft/ground stations that already comprise today’s space weather prediction infrastructure.

To both successfully and innovatively address those design issues, it was crucial to first establish and familiarize ourselves with how any new solar sentinel mission would fit into the arena of global space weather prediction. Specifically, we identified:

1. Currently available space weather products (solar data, services, and forecasts)
   - What types of alert systems are being used now?
   - What resources (i.e. spacecraft, sensors, and ground stations) are used to provide such products?

2. The organizations (both government and commercial) that provide these products
   - Who is responsible for operating and maintaining our current space weather prediction infrastructure?
   - Given their past experience, do they have recommendations to improve the future of alert capability?
3. How such products are delivered and made available to customers

Armed with the knowledge of current alert capability, we were able to see which related aspects were most important and which areas could be improved to better meet customers’ current and future needs. Once we recognized a specific setting in which a new spacecraft could contribute to future alert capability, our team focused on designing a single deployable spacecraft and an alert system to fit into that setting.

While this report briefly presents our findings on the current state of space weather prediction, it primarily outlines the subsystem details behind the design and launch of our proposed solar sentinel spacecraft.

1.2 Understanding Solar Particle Events

“Space weather refers to conditions on the Sun and in the space environment that can influence the performance and reliability of space-borne and ground based technological systems, and can endanger human life or health.”

-National Space Weather Program definition

A solar particle event (SPE), which is also sometimes referred to as a solar radiation storm, is only one of the many atmospheric phenomena encompassed by the term space weather. (Several other aspects of space weather include solar UV/EUV/X-rays, solar radio noise and solar wind, geomagnetic and ionospheric disturbances). The ability to understand and predict space weather comprehensively, in particular SPEs, has become increasingly important because its effects directly impact the operations of our Nation's commercial and government infrastructure.

Solar particle events generally refer to two forms of solar activity, coronal mass ejections (CME) and solar flares, both of which are characterized by a rapid release of energy from the Sun's magnetic field. This energy comes in the form of light, coronal material, and high energy particles. Although both CMEs and solar flares can sometimes occur independent of each other, both are commonly observed at the beginning of solar magnetic storms.
Although only a small portion of our Nation's annual budget (approximately $5-6 million dollars\textsuperscript{1}) is spent on researching space weather prediction and reporting efforts, there are hundreds of academic, commercial, and government research reports and journals that demonstrate a thorough understanding of the physical processes behind SPEs.

In summary of these reports, SPEs (both solar flares and CMEs) can drive magnetic storms in space by releasing and accelerating massive amounts of plasma, coronal material, and high energy particles. These SPE driven magnetic storms can reach velocities of over 1000 kilometers per second and can generate strong magnetic fields with orientations opposite that of the Earth. Moreover, since these magnetic storms are travelling faster than its surrounding space, they have also been observed to create massive shock waves that accelerate ions along its path to relativistic velocities near the speed of light.

\textsuperscript{1}Severe Space Weather Events-Understanding Societal and Economic Impacts Workshop Report, National Committee on the Societal and Economic Impacts of Severe Space Weather Events, 2008
The ability to understand and predict the effects of such solar particle events thus becomes important for several reasons:

First, these SPE driven magnetic storms have a wide range of effects that impact the Earth as well as entire industries within our Nation's government and commercial sector. SPEs, in particular, are well documented to have impacted (both negatively and directly) our planet's aviation, electric power, GPS, spacecraft development and launch (including human presence in space), and military industries over the past century. Massive amounts of monetary and human resources have been used to prevent and react to this wide range of effects. An improved ability to predict SPEs can help mitigate these effects.

Secondly, these SPE driven magnetic storms can start reaching the Earth within less than one hour of the start of the SPE. This limits the operational reaction time for any human presence and infrastructure in the Earth's near space environment. Improved prediction and alert methods can maximize this operational reaction time. Thirdly, as a society we cannot develop actual operational ventures with real-time national space weather prediction responsibility without a more accurate understanding of SPEs and more reliable prediction techniques. Until a more comprehensive understanding of solar activity and an operational warning capability is established, no substantial advances in future consumer/investment confidence can be made.
1.3 Today’s Space Weather Prediction Infrastructure

Sometimes referred to as a low-frequency/high-consequence event (LFHC), solar particle events can have a huge impact on our society even though they do not happen very often. Especially since we are currently in a period of relatively low solar activity, our society is not forced to develop major plans for reacting to such an event.

Nonetheless, if we wish to effectively and realistically improve the area of SPE prediction/ alert capability, we must take a step back and look at the entire space weather prediction infrastructure. It is important to understand and get a complete sense of this infrastructure because it encompasses all the technologies and operations that predict and react to the effects of solar activity. It dictates how any new solar sentinel mission will fit into the arena of today’s space weather prediction.

1.3.1 Organization

Within the United States, all space weather activities are coordinated through two central programs - the National Space Weather Program (NSWP) and the US Air Force Weather Agency (AFWA). The NSWP and the AFWA are the two single points of responsibility in the US government for space weather forecasting and predicting for the civilian and military communities respectively. Together, the two agencies are responsible for space weather planning, understanding user needs and requirements, as well as implementing the initiatives that will satisfy those current and future user needs.

The National Space Weather Program (NSWP) is an interagency initiative whose goal is to “achieve an active, synergistic, interagency system to provide timely, accurate, and reliable space weather warnings, observations, specifications, and forecasts.”

It has eight total participating agencies: NASA, the Department of Commerce (NOAA), the Department of Defense, the National Science Foundation, the Department of the Interior, the Department of Energy, the Department of State, and the Department of Transportation.

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The United States Air Force Weather Agency (AFWA) is a DoD agency whose mission is to “deliver accurate, relevant, and timely environmental information, products, and services, anywhere in the world.”

The National Space Weather Program primarily relies on the National Oceanic and Atmospheric Administration (NOAA) and its Space Weather Prediction Center (SWPC) in Boulder, Colorado as its operational arm for actually providing space weather situational awareness, alerts, and forecasting services. The SWPC, in turn, heavily relies on NSWP’s participating agencies (in particular NASA and the AFWA) for the scientific research and critical space weather data to provide such alerts and forecasting services.

Figure 1.3: Overview of the relationship between the NSWP and its participating agencies

1.3.2 A Snapshot of Today’s Current Capability

In general, today’s current space weather prediction capability is driven by the combination of remote and in-situ measurements provided by the many satellites and ground stations throughout the world. Earth-orbiting spacecraft (as well as spacecraft at \( L_1 \)), take critical measurements of space weather effects in Earth’s magnetosphere and ionosphere, while ground-based observatories provide data for characterizing space weather conditions and effects.

Although NOAA and its SWPC rely heavily on agencies such as NASA and the AFWA for such data with which to base its alerts and forecasts, NOAA’s SWPC is the single entity responsible for actually tailoring that data to civilian customer needs. The SWPC manages more than 1400 different types of data from the worldwide satellites and ground stations operated by NOAA, NASA, and the AFWA into several types of space weather alert and forecasting products to customers:

- **Available via NOAA’s website, anonymous FTP server, and email**
- **Either graphical or textual in nature**
- Some products available via NOAA Weather Wire, the National Weather Service direct broadcast system.
- **Provides a view of the current situation, near-term forecasts (hours to days), and long-term forecasts (months)**

![Figure 1.4: Current available space weather products](image)

Figure 1.4: Current available space weather products

These products include textual watches, warnings, alerts, as well as current-event and post-event summaries. Watches are used for making long-term predictions of adverse space weather. Warnings are used to alert customers that a specific space weather event is imminent. Alerts indicate that space conditions have crossed a certain threshold and that a space weather event has already started and summaries inform customers once the event has ended. Additional information about the operational capabilities of today’s current solar-studying spacecraft and ground stations can be found through the National Space Weather Program and Air Force Weather Agencies main websites.
1.4 The HAVEN Project, Helios Advanced Violent Eruption Notification Project

“If [adverse solar] phenomena are not observed, they can't be predicted. The Space Weather Prediction Center's ability to observe is going to make the difference between what we can predict and what we can’t. That prediction is the key to the future and is the answer to helping customers make good business decisions and maintain their continuity of operations.”

- Thomas J. Bogdan, current space weather program manager and director of the Space Weather Prediction Center (SWPC)

The HAVEN Project is a low-cost, operational alert system that can warn any human presence in space of dangerous and oncoming solar storm events. This system can reliably warn of incoming dangerous particles and give humans in space at least 10 minutes operational reaction time to seek shelter and protection from dangerous energetic material during large storm events.

Once operational, the HAVEN Project alert system, summarized in Figure 5, utilizes both remote sensing instruments within the Earth's near-space environment as well as a single deployable spacecraft to be placed at the first Sun-Earth Lagrange point ($L_1$). Earth-based remote sensors will be able to detect adverse solar disturbances as soon as they occur. A spacecraft placed at $L_1$ will then allow the direct sampling of oncoming highly energetic particles as they accelerate towards the Earth. As soon as particles are detected, the information must be relayed to the customer. Alerting the space hotel of the potentially hazardous storm advancing will allow for the safety of tourists therefore enhancing the industry.

![Figure 1.5: HAVEN warning system](image)

Annual workshops with representatives from the various industries and agencies look at plans for providing space weather prediction in the near future. These workshops are based on
the demand put forth by current and future customers as well as on what resources are currently available. If a certain area is lacking, they collaborate and determine what course of action will fill in the informational gap or how that area can be improved. Over the past 13 years, agencies have identified specific scientific objectives and recommended activities necessary for improving space weather predictive capabilities.

The placement and operation of an operational $L_1$ sensor is currently one of the many recommended activities necessary for the development of future operational alert capability.

More specifically, the HAVEN Spacecraft:

1. Contributes to maintaining situational awareness and the ability to know everything in the environment by providing a continuous, real-time awareness of space weather

2. contributes to a reliable 24-hour forecast of oncoming solar energetic particles

3. Space weather prediction capability heavily depends on current research assets, specifically the SOHO and ACE spacecraft, which are quickly approaching their end of life. Placement of the HAVEN spacecraft comes at a time where there is no solid plan for replacement.
Chapter 2

Mission Overview

The purpose of the HAVEN (Helios Advanced Violent Eruption Notification) project is to deliver a warning system architecture that is inexpensive and fast to produce. It must allow for a number of payloads to be delivered to the first Earth-Sun Lagrange point ($L_1$), thereby replacing current sensors that are subject to failure at any time. The following sections outline the requirements for producing a feasible system architecture for the HAVEN mission. Also included in the mission overview is a brief description of the associated cost, mass, and power budgets. Further details are discussed throughout the sections of this report devoted to each subsystem.

2.1 Mission Requirements

A successful mission is contingent upon a number of different variables. Requirements are set to manage these variables and ensure that mission objectives are met. Three tiers of requirements have been developed for this mission.

![Requirements Hierarchy](image)

Figure 2.1: Requirements Hierarchy
The mission objectives have been defined as the top most level with each level being derived from that above it down to the specific quantitative low level subsystem requirements. All requirements can be traced back up to the mission objectives that govern the warning system and its capability. The second tier of requirements is upper level stipulations for each subsystem. The lowest level of objectives consists of detailed, quantitative measures that trace through the second tier and back to the top level objectives. As an entrepreneur the concern is mainly with the top level requirements, however the other tiers are necessary to articulate in order to complete these goals and ensure feasibility of the system.

2.1.1 Top Level Requirements

The most desirable warning system satellite architecture must be governed by several top level requirements that the mission must meet to be worth the risk in investment. The HAVEN project is comprised of four primary goals as shown in Figure 2.2 and the following Table.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>MISS-01</td>
<td>Capability</td>
<td>Spacecraft must provide a warning signal of an incoming solar particle event to a space hotel in LEO orbit</td>
</tr>
<tr>
<td>MISS-02</td>
<td>Timeline</td>
<td>First operational alert must be achieved by 2014</td>
</tr>
<tr>
<td>MISS-03</td>
<td>Cost</td>
<td>Cost must be &lt; $25M</td>
</tr>
<tr>
<td>MISS-04</td>
<td>Sustainability</td>
<td>Mission architecture must have be sustainable for ≥ 5 years</td>
</tr>
</tbody>
</table>

This mission was designed with the intention of alerting future space hotels that solar particle events (SPEs) are approaching them so they can take appropriate action to protect their customers. The idea is that the space hotel will be a place for people to go on vacation in space. They will be able to stay in space short term and would like to be able to enjoy their time there while knowing they are protected against these dangerous Sun anomalies.
Starting off a new company or investing in something new like a space hotel requires a large amount of funding. Being able to provide a system that is cheap and quickly operational is therefore very desirable in order to reduce the overall cost to the industry. It is therefore required that this mission be relatively cheap such that it meets the cost requirement (MISS-03). The system should also be made operational by 2014 in order to provide a capstone to the space hotel industry. This allows for about a 5 year development time and allows for a launch time to be procured with a margin for delays. Not only is it important to have a system that is fast to launch and relatively low cost, it is also important that the system be sustainable. A lifetime requirement of five years has therefore been imposed based on the maximum lifetimes of the components utilized and the costs associated with maintaining and operating the system. This lifetime is adequate for the initial establishment of the space hotel industry ensuring they are properly warned without excessive maintenance costs.

2.1.2 Upper Level Subsystem Requirements

Stemming from the top level objectives are more specific high level subsystem requirements. These requirements specify an overall design guideline for each subsystem. Each requirement has been given a specific identifier, parameter, and trace to allow integration between subsystems and relation back to the overarching objectives.

The HAVEN-1 spacecraft is centered around the payload which must provide the means of sensing the approaching SPEs. The HAVEN project decided the optimal measurement technique was through in-situ measurement due to the fact the spacecraft is then able to feel the full intensity of the storm and have full confirmation it is on its way to Earth. Following are the upper level requirements of the payload.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>PAYLHL-01</td>
<td>Detection</td>
<td>Must detect solar particle events (SPEs)</td>
<td>MISS-01</td>
</tr>
<tr>
<td>PAYLHL-02</td>
<td>Compatibility</td>
<td>Must be compatible with satellite structure</td>
<td>MISS-01</td>
</tr>
<tr>
<td>PAYLHL-03</td>
<td>Data Handling</td>
<td>Must send data to the C&amp;DH subsystem</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

It is important that the data received by the payload sensor be processed and handled efficiently and properly such that no information is lost and the hotel industry receives their alerts through the quickest means possible. The requirements for command and data handling are detailed as follows.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>CDHHL-01</td>
<td>Command Receiving</td>
<td>Must receive commands from the ground and perform bit error checks</td>
<td>MISS-04</td>
</tr>
<tr>
<td>CDHHL-02</td>
<td>Command Execution</td>
<td>Commands must be either executed or stored for later execution</td>
<td>MISS-03</td>
</tr>
<tr>
<td>CDHHL-03</td>
<td>Pre-determined Commands</td>
<td>Must execute pre-determined commands based on the results of the instrument and spacecraft health checking system</td>
<td>MISS-01</td>
</tr>
<tr>
<td>CDHHL-04</td>
<td>Data Collection</td>
<td>Must collect the instrument bit stream and the spacecraft housekeeping data and store them on-board</td>
<td>MISS-04</td>
</tr>
<tr>
<td>CDHHL-05</td>
<td>Data Transfer</td>
<td>Must provide the instrument data and housekeeping data to the communication subsystem for downlink</td>
<td>MISS-04</td>
</tr>
</tbody>
</table>
Once the payload receives the data from an incoming solar particle event, the information must be transmitted to the space hotel as an alert. The following are the communication requirements for basic functionality in this system.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>COMHL-01</td>
<td>Transmit time</td>
<td>Must transmit real time information to the ground</td>
<td>MISS-01</td>
</tr>
<tr>
<td>COMHL-02</td>
<td>Receiving commands</td>
<td>Must receive commands from the ground</td>
<td>MISS-01</td>
</tr>
<tr>
<td>COMHL-03</td>
<td>Warning Notifications</td>
<td>Provide warning notification of SPEs to those in LEO with adequate time to take shelter</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

Maintaining the proper orbit, communications, power and payload pointing are all controlled by the guidance, navigation and control (GNC) system on-board the spacecraft. The requirements for this subsystem are based on attitude and pointing accuracy.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>GNCHL-01</td>
<td>Attitude</td>
<td>Must provide sun sensor and star data to the ground for attitude processing</td>
<td>GNCHL-02</td>
</tr>
<tr>
<td>GNCHL-02</td>
<td>Sun Pointing</td>
<td>Must (continuously) point the solar panels and payload at the Sun</td>
<td>MISS-01</td>
</tr>
<tr>
<td>GNCHL-03</td>
<td>Earth Pointing</td>
<td>Must (continuously) point the high gain antenna at the Earth</td>
<td>COMHL-01</td>
</tr>
</tbody>
</table>

The power system has two basic requirements: regulate voltage and provide a backup system if the primary power system (solar cells) fails to function or do not have direct access to the Sun. These are detailed as follows.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>POWHL-01</td>
<td>Voltage Regulation</td>
<td>Provide regulated output voltage of 28 V for 5 years</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWHL-02</td>
<td>Backup Power</td>
<td>Batteries must provide power to the spacecraft during launch prior to deploying the solar cells</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

The final orbit of the HAVEN-1 spacecraft will be centered around the gravitational balance point ($L_1$) between the Sun and Earth. The propulsion system must allow the satellite to reach this point from the low Earth orbit (LEO) that the launch vehicle inserts it into.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>PROPHL-01</td>
<td>Orbit Achievement</td>
<td>Must allow the spacecraft to reach $L_1$ from LEO</td>
<td>MISS-01</td>
</tr>
<tr>
<td>PROPHL-02</td>
<td>Maneuverability</td>
<td>Must allow station keeping</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

The HAVEN project requires that the orbits selected for the mission be based on simple trajectories with prior heritage to reduce risks. It is important that the final orbit be sustainable to meet our mission requirement of five years.
### ID Parameter Requirement Trace

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>ORBHL-01</td>
<td>Path</td>
<td>Must place the spacecraft on a path to the observing location</td>
<td>MISS-01</td>
</tr>
<tr>
<td>ORBHL-02</td>
<td>Location</td>
<td>Must place the spacecraft in a location that is consistently visible to the Sun</td>
<td>MISS-01</td>
</tr>
<tr>
<td>ORBHL-03</td>
<td>Simplicity</td>
<td>Trajectory should be as simple as possible to avoid any possible error in satellite deployment</td>
<td>MISS-04</td>
</tr>
<tr>
<td>ORBHL-04</td>
<td>Heritage</td>
<td>Architecture should be proven through heritage missions</td>
<td>MISS-04</td>
</tr>
<tr>
<td>ORBHL-05</td>
<td>Maintain</td>
<td>Final orbit must be maintained for ≥ 5 years</td>
<td>MISS-04</td>
</tr>
</tbody>
</table>

The structure requirements stem from being able to integrate and protect all components to meet the mission objectives. Thermal and radiation requirements have also been imposed to ensure the system is sustainable.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>STRUHL-01</td>
<td>Payload</td>
<td>Must provide a platform for the payload and instruments</td>
<td>MISS-01</td>
</tr>
<tr>
<td>STRUHL-02</td>
<td>Launch Vehicle</td>
<td>Must spatially fit in the launch vehicle fairing</td>
<td>MISS-01</td>
</tr>
<tr>
<td>STRUHL-03</td>
<td>Lifetime</td>
<td>Must protect the subsystems during its lifetime</td>
<td>MISS-01</td>
</tr>
<tr>
<td>STRUHL-04</td>
<td>Radiation</td>
<td>Must provide protection from radiation</td>
<td>MISS-01</td>
</tr>
<tr>
<td>STRUHL-05</td>
<td>Thermal</td>
<td>All components must operate at a temperature &lt;= 45 deg.</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

Finally, a launch vehicle must be chosen to safely deliver the satellite to space. The requirements for this vehicle are described as follows.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>LAUHL-01</td>
<td>Capability</td>
<td>Must place the spacecraft into its initial orbit and orient it at the Sun</td>
<td>MISS-01</td>
</tr>
<tr>
<td>LAUHL-02</td>
<td>Compatibility</td>
<td>Spacecraft must be compatible with the launch vehicle, both mechanically and electrically</td>
<td>MISS-01</td>
</tr>
<tr>
<td>LAUHL-03</td>
<td>Ascent Survivability</td>
<td>Must protect the spacecraft during the ascent phase into orbit placement</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

As described, these are only the upper level requirements of the various subsystems associated with the spacecraft. More specific, quantitative requirements can be found in each subsystem's section of the report.

### 2.2 Mission Architecture Overview

The selected mission architecture is based on the decision to place the spacecraft at the first Earth-Sun Lagrange point \((L_1)\) to take in-situ particle measurements. \(L_1\) is located between the Earth and the Sun at the position where their gravitational pulls equilibrate such that the spacecraft remains somewhat stationary thereby requiring a low level of station keeping. This location is ideal for a solar sentinel mission since it allows for the satellite to experience the solar event to fully understand the intensity and magnitude of the incoming storm through particle sensing. Several options were considered while determining this location including the other
Lagrange points and low Earth orbit. All of these options did not provide the storm sensing magnitude, simplicity and communication benefits that $L_1$ does to make the mission feasible given the current set of restraints.

Designing around the fact in-situ measurements at $L_1$ have been chosen for this architecture, the total cost is just under the $25$ million mark at about $24,961,731$ with a total spacecraft mass of approximately $900$ kg, both fitting within the given constraints. This system architecture is designed to give an average warning time of around 30 minutes, allowing tourists to get indoors and space hotels to take proper precautions to protect their investment against the incoming SPE. It is predicted that the warning time can be improved as the sensors and technology improve to provide up to an hour of warning time. This mission will provide a platform that will be quickly operational and can easily be replaced, if necessary.

2.3 Budgets

The following is a general overview of the mass, power and cost budgeted for each subsystem. A full breakdown of each is detailed in the subsystem sections.

2.3.1 Mass Budget

The total mass of the system is 900.07 kg, which includes 250.19 kg of dry mass. A total mass of 1010 kg has been allotted based on the launch vehicle restrictions for getting a payload to low Earth orbit. The spacecraft fits nicely within this limit with some additional margin. Additional mass and space can be sold to other companies as a secondary payload. Cubesats being small and lightweight would also be a considerable addition to help offset the launch cost. They could be sent to either $L_1$ or low Earth orbit, however they would require a payload adapter ring to be purchased. The spacecraft mass budget can be seen in Table 2.1.

<table>
<thead>
<tr>
<th>Subtotal Mass [kg]</th>
<th>818.24</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Mass [kg] (w/10% contingency)</td>
<td>900.07</td>
</tr>
<tr>
<td>Total Dry Mass [kg]</td>
<td>250.19</td>
</tr>
<tr>
<td>Allocated Total [kg]</td>
<td>1010</td>
</tr>
<tr>
<td>Net Difference [kg]</td>
<td>109.93</td>
</tr>
</tbody>
</table>

Table 2.1: Summarized Mass Budget

To ensure the system remains below the maximum payload mass, a 10% systems contingency was placed on the subtotal to account for any uncertainties and provide an additional margin. A summary of the subsystem totals can be seen in Table 2.2.
## Table 2.2: Total Mass Budget: Broken down by subsystem

<table>
<thead>
<tr>
<th>Subsystem Name</th>
<th>Subtotal Mass [kg]</th>
<th>Percentage of Allocated</th>
</tr>
</thead>
<tbody>
<tr>
<td>Communications</td>
<td>7.4</td>
<td>0.82</td>
</tr>
<tr>
<td>GNC</td>
<td>9.53</td>
<td>0.99</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>3.04</td>
<td>0.32</td>
</tr>
<tr>
<td>Payload</td>
<td>3.55</td>
<td>0.4</td>
</tr>
<tr>
<td>Power</td>
<td>8.87</td>
<td>0.93</td>
</tr>
<tr>
<td>Propulsion Components</td>
<td>41.33</td>
<td>7.88</td>
</tr>
<tr>
<td>Transfer Propellant</td>
<td>590.8</td>
<td>58.5</td>
</tr>
<tr>
<td>Structures/Thermal</td>
<td>104.38</td>
<td>11.18</td>
</tr>
</tbody>
</table>

Within each subsystem budget, as seen in Appendix B, component contingencies account for the risk level associated with each item. Since some of our components have a lower technology readiness level (TRL) due to the lack of heritage, since they are being custom built, the contingencies on some components are much larger than those that are off-the-shelf. As shown in the preceding breakdown by subsystem, propulsion utilizes the bulk of the total mass. Structures/thermal and power account for just over a quarter of the dry mass while the other subsystems have minimal masses in comparison. Propulsion mass is mostly due to the propellant needed to get the spacecraft to $L_1$.

### 2.3.2 Power Budget

The mission has been divided into four main operation phases: ascent, transfer orbit, cruise, and on-orbit phases. All phases except the ascent phase are capable of being operated only by the solar panels with the battery serving as a back-up. The battery will be used during the ascent phase to power the communications, command and data handling, and thermal systems. The total power for each phase is given in Table 2.3.

<table>
<thead>
<tr>
<th>Subtotal Power [W]</th>
<th>Total Power w/ 10% Contingency [W]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Ascent Power</td>
<td>48.51</td>
</tr>
<tr>
<td>Total Transfer Orbit Power</td>
<td>83.16</td>
</tr>
<tr>
<td>Total Cruise Phase Power</td>
<td>118.36</td>
</tr>
<tr>
<td>Total On-Orbit Power</td>
<td>120.4</td>
</tr>
</tbody>
</table>

Table 2.3: Summarized Power Budget: Broken down by phase

The total power each subsystem uses for each phase is given in Table 2.4. As shown, communications, guidance, navigation and control (GNC), command and data handling (C&DH), and thermal are the key contributors to the power. Communications and C&DH needs to be able handle the data received by the sensors and send the warnings back to LEO quickly in the event of a SPE. GNC is high power due to the stringent pointing requirement. Thermal heating is required to keep some of the more sensitive instruments operational.
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Communications</td>
<td>1.1</td>
<td>4.95</td>
<td>40.15</td>
<td>40.15</td>
</tr>
<tr>
<td>GNC</td>
<td>0</td>
<td>30.8</td>
<td>30.8</td>
<td>30.8</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>23.65</td>
<td>23.65</td>
<td>23.65</td>
<td>23.65</td>
</tr>
<tr>
<td>Payload</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>2.035</td>
</tr>
<tr>
<td>Power</td>
<td>1.76</td>
<td>1.76</td>
<td>1.76</td>
<td>1.76</td>
</tr>
<tr>
<td>Propulsion</td>
<td>0</td>
<td>0</td>
<td>10.56</td>
<td>10.56</td>
</tr>
<tr>
<td>Thermal</td>
<td>22</td>
<td>22</td>
<td>22</td>
<td>22</td>
</tr>
</tbody>
</table>

Table 2.4: Total Power Budget: Broken down by subsystem

The maximum power required by the spacecraft is 132.43 W during the on-orbit phase however the solar panels have been sized to accommodate up to 230 W. At this point all systems are fully functioning, however not necessarily all operating at the same time. Communications, for example, will downlink housekeeping telemetry once a day requiring a large amount of power during that time only. The GNC system requires a large amount of power due to the high amount of precision needed to keep the spacecraft pointed at the Sun. The solar panels have been sized accordingly to account for the maximum amount of power needed at any given time. Battery power will then serve as back-up power in case the solar panels can't receive full power or an unforeseen event occurs. For a breakdown of when each system is operating and the budgets at that time see Appendix B.

2.3.3 Cost Budget

The total cost for the HAVEN project system architecture is just below the allocated $25 million at $24.96 million which includes all research and development, component costs, integration, launch operations, maintenance, and any additional associated costs to achieve the warning system capability and first operational alert. The totals are summarized in Table 2.5.

<table>
<thead>
<tr>
<th>Subtotal Cost [USD]</th>
<th>$22692482.5</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total Cost [USD] (w/10% contingency)</td>
<td>$24961730.75</td>
</tr>
<tr>
<td>Allocated Total [USD]</td>
<td>$25000000</td>
</tr>
<tr>
<td>Net Difference [USD]</td>
<td>$38269.25</td>
</tr>
</tbody>
</table>

Table 2.5: Summarized Cost Budget

A 10% contingency was placed on the subtotal to account for any uncertainties and provide an additional margin to ensure the system can be delivered at the proposed price. To reduce the costs and risks of the spacecraft architecture, heritage and off-the-shelf components have been chosen if available. A summary of the subsystem totals can be seen in 2.6.
<table>
<thead>
<tr>
<th>Subsystem Name</th>
<th>Total Cost [USD]</th>
<th>Percentage of Allocated</th>
</tr>
</thead>
<tbody>
<tr>
<td>Communications</td>
<td>$271200</td>
<td>1.1%</td>
</tr>
<tr>
<td>Guidance, Navigation and Control</td>
<td>$1623000</td>
<td>6.5%</td>
</tr>
<tr>
<td>Command and Data Handling</td>
<td>$273787.5</td>
<td>1.1%</td>
</tr>
<tr>
<td>Launch</td>
<td>$9572250</td>
<td>38.3%</td>
</tr>
<tr>
<td>Payload</td>
<td>$5175000</td>
<td>20.7%</td>
</tr>
<tr>
<td>Power</td>
<td>$480418.5</td>
<td>1.9%</td>
</tr>
<tr>
<td>Propulsion</td>
<td>$2579173</td>
<td>10.3%</td>
</tr>
<tr>
<td>Structures/ Thermal</td>
<td>$307653.5</td>
<td>0.12%</td>
</tr>
<tr>
<td>Integration, Operations, etc.</td>
<td>$2410000</td>
<td>0.96%</td>
</tr>
</tbody>
</table>

Table 2.6: Total Cost Budget- Broken down by subsystem

The launch cost is high as expected, and the payload cost is high since it is customized for our mission. The chosen launch vehicle is the Falcon 1E, which has not yet been flown. For an additional $3,387,000 (over the allotted $25M) launch insurance can be purchased allowing the money spent on the satellite to be reimbursed in case of a launch vehicle failure. While this is advised as a safety precaution, there is a high confidence level that this launch vehicle will do the job.

As described in an earlier section, the payload being used is based on the EPHIN system which has previous flight heritage. Since this sensor is not currently available off-the-shelf, some research and development costs have been included to rebuild the sensor and update the technology. Approximately 20% of the cost budget has therefore been allotted to the payload due to its high level of importance in the mission’s success.

All remaining subsystems have chosen heritage components minimizing their costs. Both a component contingency and a systems contingency have been placed on each subsystem to account for research and development and integration costs as well as any uncertainty. Costs such as integration into the launch vehicle, spacecraft maintenance, and operations have all been included on top of component costs.
Part II

Subsystems
Chapter 3

Payload

Solar observing satellites utilize a variety of different sensors to observe and investigate the sun and the harmful particles it emits. The sensors may be grouped into two measurement categories: remote sensing and in-situ measurements. Remote sensing devices include those that gather light of various wavelengths whereas in-situ sensors collect on-board energetic particle data for immediate analysis and communication. We have discovered that there is a variety of different platforms, both space and Earth based, currently in use or in production that focus on remote sensing of the Sun. We have therefore decided to use and develop an in-situ sensor that is unique to our mission. In-situ sensors will allow us to utilize an innovative forecasting technique to increase current warning times up to 30 minutes.

3.1 Background

The HAVEN payload consists of in-situ sensors that provide a state of the art warning system for orbiting space hotels as well as other entrepreneurial and governmental interests. Our instrument is capable of detecting incoming SPE particles and relaying their energy levels and intensities back to Earth. We are using in-situ measurement rather than remote sensing due to its capability of providing information not found on other satellite architectures. In-situ sensors are usually custom designed for a particular satellite. Thus, the HAVEN instrument payload will be unique compared to other satellites and provide data useful to providing an alert system capability and post-event feedback, key to our warning system. Table 3.1 below displays existing and in-development remote sensing devices observing the sun.
Due to the multitude, similarity, and easy data access of the above and other unlisted remote sensing missions, a remote sensing instrument on the HAVEN platform cannot be justified as fulfilling a unique need. Having investigated the advantages of in-situ instrumentation, we have arrived at a novel warning system capable of providing a warning on average 30 minutes before any other satellite.

### 3.2 Warning System Method

It is necessary to provide a warning system to give space visitors an adequate amount of time, at least 10 minutes, to retreat to a safe location where they will not be exposed to incoming harmful particles. Dr. Arik Posner, a research scientist at Southwest Research Institute, has introduced an innovative warning system approach. It has been well documented that relativistic electrons accelerated by the shock of the CME in the range of 0.3 MeV to 1.6 MeV precede the more energetic, harmful ions in the >20MeV range. In addition, there has never been a documented event in which high-energy ions have arrived at Earth before the relativistic electrons. Through measurements using SOHO’s Electron Proton Helium Instrument (EPHIN) sensor and the GOES-8 satellite, Dr. Posner has created a predictive matrix, shown in Figure 3.1 based on the energies of the relativistic electrons.
According to estimates using the matrix we can provide an average 30-minute warning time to the hotels in orbit around earth. Further developments in this model will increase prediction accuracy of incoming SPE particles by covering all possible scenarios. The electron increase parameter is a function of electron rise time and flux. The correlations between the electron intensity, flux and rise time have been used to create the above forecasting matrix. The colors are the predicted ion intensities arriving in one hour. When electron intensities increase, proton intensities also increase as displayed by the color bar in Figure 3.1. By knowing the electron increase parameter and the electron intensity, the matrix can be used to estimate the intensity of the protons arriving within the hour.

Figure 3.2 below shows the prediction of the energetic ions in 2003 using this forecasting method.
The red observed and black predicted intensities of 30 MeV-50 MeV protons are shown above. The blue diamonds indicate a hazardous proton. As seen from the figure, the forecasting method was accurate for every event. It did not produce any false negatives, which is an important characteristic of our method. Currently there are insufficient statistics, which is evident by the blank areas in the lower right hand corner of Figure 3.1. NASA is currently considering using this method of warning for its upcoming lunar missions, and further research into this method will provide longer warning times and additional predictions. By 2014, this method will be greatly improved by filling in the blank areas and satisfy our mission requirements.

The fastest recorded storm traveled at approximately 1/3 the speed of light. Therefore, in the event of an intense storm, it will take around 24 minutes to reach the Earth, and the relativistic electrons will take about 10 minutes. This will give around a 10-minute warning period, taking into account time for signal processing and transceiving. In the worst-case scenario, the space hotel guests have 10 minutes to react and reach a sheltered portion of the hotel.

### 3.3 Requirements

According to Posner's Forecasting method, the satellite sensors must be able to detect the relativistic electrons and ions that the CME and its corresponding shock wave emit and accelerate. In-situ measurements of these electrons and ions have greater value, as remote sensing is widely used and remote data is easily available from other missions. Table 3.2 below summarizes the low-level requirements of the payload subsystem.
The payload system must measure the flux, rise time, and intensity of relativistic electrons in the 0.3-1.6 MeV energy range. The payload system must measure the flux and intensity of energetic ions with energies between 20 MeV-50 MeV. The payload subsystem must notify the Command and Data Handling subsystem when particle flux goes above 100 electrons/(cm$^2$ sec).

Table 3.2: Low Level Payload Requirements

<table>
<thead>
<tr>
<th>Identifier</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>PAYLOAD-01</td>
<td>Electron Detection</td>
<td>The payload system must measure the flux, rise time, and intensity of relativistic electrons in the 0.3-1.6 MeV energy range.</td>
<td>PAYLHL-01</td>
</tr>
<tr>
<td>PAYLOAD-02</td>
<td>Ion Detection</td>
<td>The payload system must measure the flux and intensity of energetic ions with energies between 20 MeV-50 MeV.</td>
<td>PAYLHL-01</td>
</tr>
<tr>
<td>PAYLOAD-03</td>
<td>Processor</td>
<td>The payload subsystem must notify the Command and Data Handling subsystem when particle flux goes above 100 electrons/(cm$^2$ sec).</td>
<td>PAYLHL-03</td>
</tr>
</tbody>
</table>

PAYLOAD-01: The range of relativistic electron energy levels that must be detected by the sensor in order to forecast an incoming proton event. This range lies from 0.3-1.6 MeV, according to Dr. Posner. Relativistic electron spectrums only get to these magnitudes when followed by the proton event, which generally consist of ions with energies greater than 30 MeV. In order for the forecasting matrix to be used, the electron flux, rise time, and intensity data must be measured.

PAYLOAD-02: The range of ion energy levels the payload subsystem has to detect is between 20 MeV and 50 MeV. This range includes particles that are harmful to humans and satellites alike. Ions may reach higher energies; however, it is not necessary for our sensor to be able to measure energies this high since ions with higher energies will still emit a signal in the range of 20 MeV to 50 MeV.

PAYLOAD-03: The processor requirement ensures that our sensor will transmit a notification of the detection of these particles to the Command and Data Handling Subsystem. This implies that the sensor is able to determine at what point the energy readings are considered too high, signifying that a CME is occurring. According to published Solar ion level data, a flux of electrons about 100 electrons/(cm$^2$ sec) indicates an incoming event.

### 3.4 Sensor Selection

In order to determine an appropriate set of sensors to analyze and provide a warning time for incoming solar energetic particles, previous solar observation and space weather missions, along with their in-situ sensors, were studied. The primary missions of focus are the Advanced Composite Explorer (ACE), Solar and Heliospheric Observatory (SOHO), Geostationary Operational Environmental Satellites (GOES), Polar Operational Environmental Satellites (POES), and Solar Terrestrial Relations Observatory (STEREO). The in-situ instruments on these missions were researched for sensors that would be suitable for our mission. The trade study of the instruments and sensors can be viewed in Appendix C. Our new sensors will be modeled after existing sensors for heritage, and then tailored to our specific needs and requirements.

The first step in narrowing down the list of sensors is to eliminate all in-situ sensors that do not analyze the flux, rise time, and intensity of electrons or ions. Our instrument needs to detect relativistic electrons, which arrive before harmful ions. The sensor also needs to detect...
the intensity of ions following the electrons in order to confirm the strength of the solar storm.

After eliminating the sensors that do not detect electrons and ions, the list is further narrowed down by determining which sensors are capable of measuring the range of energies of the electrons and ions that are being accelerated ahead of the CME’s shock. The electron sensor needs to be able to measure electrons with energies between 0.3 and 1.6 MeV while the ion sensor needs to measure proton energies from 20 to 50 MeV, according to the requirements in Section 3.3. From these two requirements a list of five sensors were determined capable for the mission. These sensors are listed in Table 3.3 below.

<table>
<thead>
<tr>
<th>Instrument</th>
<th>Satellite</th>
<th>Orbit</th>
<th>Energy Range [MeV]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Electron</td>
<td>EPS</td>
<td>GOES</td>
<td>LEO</td>
</tr>
<tr>
<td>Sensors</td>
<td>MEPED</td>
<td>POES</td>
<td>GEO</td>
</tr>
<tr>
<td>Proton</td>
<td>HED</td>
<td>SOHO</td>
<td>L1</td>
</tr>
<tr>
<td>Sensors</td>
<td>HET</td>
<td>STEREO</td>
<td>L4/L5</td>
</tr>
<tr>
<td></td>
<td>EPHIN</td>
<td>SOHO</td>
<td>L1</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 3.3: Sensor Trade Study

From the table above, the Electron Proton Helium Instrument (EPHIN) was chosen as both the electron sensor and the proton sensor. There are several reasons for choosing EPHIN over the other sensors in each category. EPHIN is chosen over the Energetic Particle Sensor (EPS) as the dedicated electron sensor because the Energetic Particle Sensor (EPS) can only detect electrons as low as 0.6 MeV instead of the required 0.3 MeV. EPHIN can detect electrons in the energy range of 0.2 - 8.7 MeV which satisfies the relativistic electron energy requirement. It was chosen over the Medium Energy Proton and Electron Detector (MEPED) because of its heritage of use at L1 and since Dr. Posner used this sensor in his forecasting method.

The High Energy Telescope (HET) was not chosen as the dedicated proton sensor because it can only detect ions up to 6 MeV, failing to satisfy the requirements in Section 3.3. The High Energy Detector (HED) detects ions up to a range far greater than our requirements. EPHIN was chosen over HED in order to simplify the system. Using one instrument instead of two will reduce the mass, power and complexity of the payload. EPHIN is currently operating on board the Solar and Heliospheric Observatory (SOHO). Therefore, the risks of utilizing the sensor as our payload are greatly mitigated. SOHO is not expected to be operational in 2014, thus our sensor will provide an excellent replacement to continue its capabilities.

3.5 EPHIN Sensor Background

The EPHIN sensor is part of the Comprehensive Suprathermal and Energetic Particle Analyzer (COSTEP) on board the SOHO spacecraft. It was launched in 1995 and developed at the University of Kiel in Kiel, Germany. The CHANDRA mission, equipped with the SOHO flight space, was launched in 1999. The entire research, development and creation cost was $5.2 million in
1990. This value is equivalent to $8.6 million in 2009. However, since research and development has been completed, we estimate a cost of roughly $4.5 million for the completed product.

### 3.6 Payload Architecture

An improved EPHIN sensor has been chosen for the HAVEN satellite following a rigorous trade study of comparable sensors, due to the telescope's low mass and wide range of detectable electrons. It is capable of measuring electrons and ions in the energy ranges stated in Table 3.3, fulfilling payload requirements. A depiction of the telescope is shown in Figure 3.4. The HAVEN instrument improves EPHIN's operational capabilities and optimizes its functionality as an interplanetary telescope by updating its electronics and tailoring it to the required energy ranges. The telescope will be used to provide an early warning of approaching solar energetic particles giving hotel guests a safe amount of time to retreat to a sheltered portion of the hotel.

![Figure 3.3: EPHIN sensor](image)

A schematic depiction of the payload sensor is shown in Figure 3.4.
As shown in Figure 3.4, there is a Kapton and titanium foil in front of the actual sensor. The titanium foil keeps light inside the sensor and closes the electrical shielding of the sensor. The Kapton foil functions to help to maintain the thermal environment within the sensor. The entrance of the sensor allows for an 83° full conical field of view.

The EPHIN sensor consists of five silicon detectors surrounded by an anticoincidence shield of plastic scintillator. Plates A and B are divided into six segments as shown in Figure 3.4. This construction creates a self-adaptive geometric factor. For example, if the detector registers a high particle count in A0, the logic will disable the other segments of the A and B plates. By allowing only A0 and B0 to function, the flux can be measured more precisely without losses.

Plates C, D, and E stop electrons up to 10 MeV and Hydrogen and Helium ions up to 53 MeV. Plate F allows particles that do not penetrate through the telescope to be distinguished from penetrating particles. Finally, G is the plastic scintillation detector. A scintillation detector is an instrument that converts a fraction of a particle’s energy to light. The photomultiplier depicted in Figure 3.4 converts the light to a current signal. Charged particles entering the sensor induce an electrical signal when striking the plates. The amplitude of the signal is determined by the energies of the particles deposited into the plates.

Measurements from the electron sensor will aid in the determination of a possible SPE impacting the space hotel, as well as the harmfulness of the approaching event. The rate at which a particle deposits energy into the sensor gives the electron rise-time, or how fast an electron dissipates energy. Higher rise times correlate to events that are more energetic. The sensor also measures electron counts, or how many electrons travel through the detector. Electron flux can be determined by taking the electron counts per unit time per unit area. A sample graph of electron flux is shown in Figure 3.5. From the plot it is evident that once the flux exceeds 100 electrons/(cm² sec), an SPE and magnetic storm is coming.
The HAVEN instrumentation suite will also include the capability of confirming the approach of an SPE. EPHIN is capable of measuring energies up to 53 MeV, which will be capable of measuring the energies of passing ions. However, the spacecraft will also benefit from the improved operational capabilities that come from the addition of a new processor. The primary purpose of ion detection will be to confirm the passing of an SPE and therefore inform hotel guests when it is safe to resume space activities.

The physical properties of the HAVEN instrument suite will resemble those of EPHIN. These properties can be seen in Table 3.4.

<table>
<thead>
<tr>
<th>Dimension</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dimensions</td>
<td>35.5 cm x 21.9 cm x 19.1 cm</td>
</tr>
<tr>
<td>Mass</td>
<td>3.55 kg</td>
</tr>
<tr>
<td>Field of View (half-angle)</td>
<td>LED: 83°, conical</td>
</tr>
<tr>
<td>Alignment</td>
<td>±15° with magnetic field line</td>
</tr>
<tr>
<td></td>
<td>45° West of Spacecraft-Sun line</td>
</tr>
<tr>
<td>Power ON</td>
<td>1.85 W</td>
</tr>
<tr>
<td>Data Rate</td>
<td>172 bit/s</td>
</tr>
</tbody>
</table>

Table 3.4: EPHIN Sensor Specifications

However, these properties represent a conservative estimate, as significant upgrades will be made to the processor used by EPHIN that should greatly enhance the operational capabilities
of the sensor. Figure 3.6 shows an example of ion particle identification process in confirming a SPE. An identification number is assigned to certain energies and the sensor measures how many ions occur for that identification number.

![Particle Identification Test](image)

**3.7 Data Processing and Operations**

Our sensor will operate in the modes described in Table 3.5 below.

<table>
<thead>
<tr>
<th>Mode</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Off</td>
<td>Power Off</td>
</tr>
<tr>
<td>Upload</td>
<td>Programs uploaded from LEO or ground stations</td>
</tr>
<tr>
<td>Standby</td>
<td>Takes measurements every hour to check if SPE has passed</td>
</tr>
<tr>
<td>Maintain</td>
<td>Testing and configuration</td>
</tr>
<tr>
<td>On</td>
<td>Normal operating mode</td>
</tr>
<tr>
<td>Calibrate</td>
<td>Calibration</td>
</tr>
</tbody>
</table>

Table 3.5: Sensor Modes

Data processing will occur on-board with the digital control unit processing all scientific and housekeeping data. Calibration occurs on the ground before launch. During normal non-SPE operations the sensor is taking readings continuously. When the relativistic electrons flux increases above 100 particles/(cm$^2$ sec) an alert will be issued that high-energy particles are incoming. When particles reach 180 kRad (Si), which is 90% of the maximum dosage of our electronics, the sensor will switch to safe mode in order to keep from being damaged. The most harmful portion of an SPE typically lasts 12 hours and therefore, while in safe mode, the sensor will take measurements approximately every 30 minutes in order to determine if the SPE has passed.
3.8 Electronic Improvements

One of the technical improvements of the EPHIN type sensor will be an upgrade of the processing unit. The EPHIN sensor used the radiation-hardened MAS281 microprocessor, which was built taking into account the requirements of MIL-STD-1750A. Since MIL-STD-1750 was declared inactive for military projects, it is necessary to upgrade the processor used for our sensor as well.

We are using the RAD750 single-board computer, manufactured by BAE Systems. Using the RAD750 instead of the MAS281 microprocessor, we reduce size, weight, and increase the radiation tolerance of our system. The RAD750 processor meets the current quality control requirements from Military Standards and is currently the most advanced space computer available.

A trade study of available processors is shown below in Table 3.6. We also researched the following stand-alone processors: the MA31750 (Dynex Semiconductor), the P1750A (Pyramid Semiconductor), the PPC750FX (IBM), and the RAD6000 (IBM). The RAD750 was chosen over these because its radiation tolerance was the highest and its bit error/day comparable to the others.

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>MA31750</td>
<td>3000 Rad (Si) 1x10⁻¹¹ errors/day</td>
<td>-55 to 125 Operating, -65 to 150 Storage</td>
<td>37.9 mm x 37.9 mm x 2.667 mm</td>
<td>16000</td>
<td>NO</td>
<td></td>
<td></td>
</tr>
<tr>
<td>P1750A</td>
<td>100000 Rad (Si) 9x10⁻¹⁰ errors/day</td>
<td>-65 to 150 Storage, -55 to 125 Operating</td>
<td>14.98 mm x 42.24 mm x 9.52 mm</td>
<td>9x10⁻¹⁰ errors/day</td>
<td>-55 to 125 x 9.52 mm</td>
<td></td>
<td></td>
</tr>
<tr>
<td>RAD6000</td>
<td>7.4x10⁻¹⁰ errors/day</td>
<td>2.5 to 20 Storage, -25 to 105 Operating</td>
<td>40 mm x 30 mm</td>
<td>0.09</td>
<td>YES</td>
<td></td>
<td></td>
</tr>
<tr>
<td>RAD750</td>
<td>200000 Rad (Si) 1x10⁻¹⁰ errors/day</td>
<td>-55 to 125 Operating</td>
<td>25 mm x 25 mm x 6.22 mm</td>
<td>0.09</td>
<td>YES</td>
<td></td>
<td></td>
</tr>
<tr>
<td>MAS281 (EPHN)</td>
<td>30000 Rad (Si) 1x10⁻¹⁰ errors/day</td>
<td>-55 to 125 Operating, -65 to 150 Storage</td>
<td>81.2 mm x 28 mm x 6 mm</td>
<td>YES</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 3.6: Processor Trade Studies
Chapter 4

Orbits

4.1 Background

In order to design the best possible orbit for the Lagrangian point mission research was conducted on similar previous missions to gain an understanding of what properties are desirable as well as feasible for an orbit around the first Lagrangian point ($L_1$). Newton's three-body problem is particularly applicable to the HAVEN Project and is examined here.

4.1.1 Mission Heritage

Initially several heritage missions which orbited out to $L_1$ were researched. Initial study focused on four Lagrangian point missions: the Advanced Composition Explorer (ACE), the International Sun-Earth Explorer 3 (ISEE-3), the Solar and Heliospheric Observatory (SOHO), and WIND. The team additionally researched other Lagrangian point missions to facilitate further comparisons for our mission design. Among those were Triana, the Next Generation Space Telescope (NGST) and the Wilkinson Microwave Anisotropy Probe (WMAP). The primary focus of the background investigations were the parking orbit, transfer trajectory, injection orbit and final Lissajous orbits for these missions. Table 4.1 summarizes the orbit information for these missions.
We chose the ISEE-3 mission to be the main reference for transfer trajectory techniques, as its mass was similar to anticipated values for the HAVEN-1 spacecraft. Based on closely aligned project goals, we modeled our final halo orbit after the SOHO mission.

### 4.1.2 The Restricted Three-Body Problem

Lagrangian points exist in any system where the body being monitored is influenced by two other significantly more massive bodies; these points are locations where the smaller body can theoretically be kept stationary relative to the other two bodies. For this mission design, the two larger bodies are the Earth and the Sun. By taking into account the gravitational field of the Sun, more complicated orbital mechanics can be used to both reduce the ∆V of the transfer orbit to a location, as well as create a semi-stable orbit between the bodies. Figure 4.1 shows the relative locations of these points compared to the two attracting bodies.

![Figure 4.1: Lagrangian Points in a Three Body System](image_url)
$L_4$ and $L_5$ are stable points, where a disturbance in any direction will be damped out. The other three Lagrangian points lie along the Earth-Sun axis and are semi-stable, with any disturbance along the Earth-Sun axis causing eventual instability in the orbit.

By placing the spacecraft in an orbit around $L_1$, it will remain in a fixed orientation directly between the Earth and Sun, providing a constant location to monitor solar activity without any eclipse. In the Earth-Sun system, $L_1$ occurs at a distance of $\approx 1,500,000$ kilometers from the Earth, or approximately 0.01 Astronomical Units (AU). The orbital mechanics of such an orbit are quite complex compared to the traditional two-body problem. Analytical, closed form, solutions to the three-body problem are not readily available. However, the equations that describe the motion of a body in this system are well understood due to their heritage.

The principle of conservation of momentum cannot be used to characterize the three-body system since the spacecraft is constantly being acted upon by two separate forces (the gravitational pull from the Earth and Sun.) However, conservation of energy still applies. Equation 4.1 gives the potential energy of the system, where $r$ is the position of the spacecraft, $\mu_1$ and $\mu_2$ are the gravitational parameters of the two large bodies, and $r_1$ and $r_2$ are the distance between the spacecraft and the respective body.

$$V(r) = -\frac{\mu_1}{r_1} - \frac{\mu_2}{r_2} \quad (4.1)$$

Equation 4.2 gives the kinetic energy of the system, where $m$ and $p$ are the spacecraft's mass and linear momentum respectively.

$$K = \frac{p^2}{2m} \quad (4.2)$$

The sum of the kinetic and potential energy of the system is the total energy of the system, which remains constant independent of the location of the spacecraft on the orbit. By analyzing the energy conservation problem, it is possible to calculate the orbit parameters for the stable or semi-stable orbits previously mentioned, which are known as Lissajous orbits; this Lissajous class of orbits includes traditional Lissajous orbits as well as “halo” orbits.

The main difference between Lissajous and halo orbits is that in a halo orbit the in-plane and out-of-plane frequencies of the spacecraft are equal, allowing the spacecraft to have a periodic orbit; halo orbits require a Y-axis amplitude of at least 654,276 kilometers in order to obtain this periodic motion.

A third type of orbit can be employed around Lagrangian points which are entirely within the plane of the two bodies, referred to as Lyapunov orbits. However, due to communications and attitude determination and control limitations created by lying within the plane of the Sun, this family of orbit is largely ignored when designing an $L_1$ mission orbit.

Equations 4.3 through 4.5 describe the simplified linearized equations of motion of the spacecraft near the Sun-Earth $L_1$ point, with $X, Y, Z$ representing the position vector of the spacecraft, $A_x, A_y$ and $A_z$ representing the amplitudes of the Lissajous orbit, $k = 0.310$ is a constant, $\omega_{xy}$ and $\omega_{xz}$ are constant at 2.086 and 2.015 respectively, and $\phi$ represents the phase angle.
\[ X_n = kA_y \sin(\omega_{xy}t + \phi_{xy}) \]  
\[ Y_n = kA_x \cos(\omega_{xy}t + \phi_{xy}) \]  
\[ Z_n = A_z \sin(\omega_z t + \phi_z) \]  
(Equation 4.5)

Equation 4.5, which describes motion along the Z-axis, is independent of the X- and Y-axes. The only constraint on selecting the Z-axis amplitude of the orbit is the beam width constraint of the spacecraft. The X and Y plane motions of the spacecraft are coupled, meaning that the X and Y amplitudes of the final orbit need to be carefully chosen to obtain a desirable final orbit \( \phi_z \) and \( \phi_{xy} \). Further analysis and appropriate calculations are handled in MATLAB and are discussed in Section 4.2.4.

4.2 Mission Design

The following section outlines our approach for the mission design phase, taking all research into account.

4.2.1 HAVEN Orbital Requirements

The feasibility of the orbit is determined by the ability of the chosen orbit to satisfy the high level mission requirements outlined in Section 2.1, as well as the capacity of the HAVEN-1 spacecraft subsystems to satisfy the quantitative requirements particular to each orbit architecture. The subsystem requirements in Table 4.2 describe the design criteria and constraints for the orbits.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>ORBIT-01</td>
<td>Location</td>
<td>Spacecraft must be placed in Lissajous orbit around ( L_1 )</td>
</tr>
<tr>
<td>ORBIT-02</td>
<td>Transfer</td>
<td>Propulsive burn of ( \Delta V \approx 3.2 \text{ km/s} ) to transfer from LEO to Lissajous orbit</td>
</tr>
<tr>
<td>ORBIT-03</td>
<td>Station Keeping</td>
<td>Maneuvers performed every three months</td>
</tr>
<tr>
<td>ORBIT-04</td>
<td>Station Keeping</td>
<td>Total ( \Delta V &lt; 10 \text{ m/s} ) every year</td>
</tr>
<tr>
<td>ORBIT-05</td>
<td>SEZ Avoidance</td>
<td>Strategic selection of orbit parameters</td>
</tr>
</tbody>
</table>

Table 4.2: Orbit low level requirements

ORBIT-01: The spacecraft must be placed in an orbit around the Lagrangian point \( L_1 \) in order to allow for constant access to the Sun. Because \( L_1 \) is an unstable location, the only feasible orbit category is a Lissajous orbit.

ORBIT-02: The \( \Delta V \) necessary for a propulsive burn to transfer from a LEO parking orbit or a phasing loop for a lunar assist (detailed later in Section 4.2.3) is 3.2 km/s.

ORBIT-03: Due to the unstable nature of \( L_1 \), the orbit must be periodically corrected with propulsive maneuvers, known as station keeping. As the spacecraft gains or loses energy (i.e.
speeds up or slows down), the parameters of the orbit deviate from the intended design. It is not possible to predict the exact alterations that will occur, thus prediction techniques are utilized. Following the "orbital energy balancing" loose control strategy utilized by the SOHO mission, station keeping maneuvers are expected to occur about every three months.

**ORBIT-04:** Along with the ORBIT-03 requirement, the predicted $\Delta V$ required during a year is around 2 m/s. As a result of the unpredictable nature of the orbit deviations, the $\Delta V$ required for each maneuver will be different. The SOHO satellite, prior to the June 25, 1998 anomaly, performed station keeping maneuvers with $\Delta V$ magnitudes ranging from 0.041 m/s to 1.89 m/s.

**ORBIT-05:** The solar exclusion zone (SEZ) is a region between the Sun and the Earth where communication is limited due to interference noise generated by the Sun (further detailed in Section 4.2.3). The amplitude of the selected orbit can be rigorously determined such that the spacecraft does not enter the SEZ.

### 4.2.2 Final Lagrangian Point Orbit Design

One of the main drivers for selecting a final orbit is the necessity of avoiding the SEZ. This region is defined by the angle between the Sun, Earth, and vehicle (SEV), and is illustrated in Figure 4.2.

![Figure 4.2: Illustration of the Solar Exclusion Zone](#)

The amount of noise created by having the Sun in the background in this region makes communication with the spacecraft very difficult. For example, the orbit of the ACE mission as it crosses into the solar exclusion zone is shown in Figure 4.3.
ACE was originally designed to avoid the SEZ at all times, however, to increase the mission lifetime by saving fuel, the station keeping maneuver used to keep the spacecraft out of the SEZ stopped being performed in July of 2001.

### 4.2.3 Transfer Trajectory Design and Optimization

The following section outlines the direct injection and lunar gravity assist transfer methods along with the advantages and disadvantages for each. The ACE, SOHO, and ISEE-3 missions used the direct injection method. In a direct injection, one burn is performed from a low-Earth parking orbit, with enough $\Delta V$ obtained to place the spacecraft in a trajectory towards the selected Lagrangian point. The WIND and WMAP missions both performed lunar gravity assist transfer orbits. For these missions, the spacecraft was initially placed into a highly elliptical parking orbit, and after several “phasing loops,” the spacecraft rendezvoused with the Moon and used the lunar gravity to gain a small amount of $\Delta V$ necessary to reach the spacecraft's destination.

#### 4.2.3.1 Direct Injection Transfer

Figure 4.4 illustrates both the transfer and final halo orbit of the ISEE-3 mission. The figure also shows the timeline of the transfer maneuver, with insertion into the halo orbit occurring 106 days after launch.
The particular transfer trajectory used by ACE, SOHO, and ISEE-3 were all very similar and follow basically the same course as shown in Figure 4.4. This route allows for the lowest amount of $\Delta V$ while keeping a relatively short timeline of around three months. An alternate path can be used to reduce the time spent in transfer down to approximately 35 days. This “fast” direct injection requires about 25% more $\Delta V$ for mid-course corrections (MCC) and injection into the final orbit, and has not been used by any of the missions researched. A comparison of the two trajectories can be seen in Figure 4.5.
4.2.3.2 Lunar Gravity Assist Transfer

Figure 4.6 shows the transfer trajectory used by the WMAP mission. Lunar gravity assist missions use phasing loops for two main reasons, to increase launch windows as well as reduce the impact of parking orbit errors occurred during launch. Without the use of phasing loops, launch windows would be limited to the one day a month when the relative orientation of the Earth, Moon, and Sun were properly aligned. By planning on using phasing loops, the spacecraft can be launched when conditions are appropriate, and idle in phasing loops until the Moon orbits into the necessary location. WMAP spent the first 25 days of its mission in these phasing loops, increasing the duration of the transfer over a direct insertion by about a month. After this time, the spacecraft passed slightly behind the moon, and performed a “slingshot” maneuver to gain the required $\Delta V$ to reach its final orbit.
The use of a lunar gravity assists reduces the $\Delta V$ required by approximately 80 m/s. However, additional complexities arise when planning a mission around a lunar assist. The launch window decreases to anywhere from 14 days a month (in two consecutive seven day periods), down to one day a month if no phasing loops are used. The transfer orbit increases in time by close to an additional month, and the complexity of the orbit design greatly increases. These trade-offs need to be weighed with the $\Delta V$ savings when selecting an appropriate transfer trajectory.

4.2.4 MATLAB Applications

The restricted three-body problem describes the motion of a relatively small object, such as the spacecraft, in the presence of two massive, primary objects. In this case, the two primary bodies in which we concern ourselves with are the Sun and the Earth. In $L_1$ orbit design, we generally ignore the perturbation effects of the Moon, as they are small in magnitude when compared to the dynamic effects inherent to an $L_1$ halo orbit.

For the mission, we examined the circular restricted three-body problem (CRTBP). This follows a general reduction of the three body problem in which $m_1$ is the primary body or the Sun, $m_2$ is the secondary primary body or Earth, and $m_3$ represents the HAVEN-1 spacecraft.
We can reduce the general equations of motion for these bodies in three-dimensional space by making some simplified assumptions and consequently solve for them to obtain relatively accurate models for the system's dynamics. First, it helps to neglect the Earth's orbital eccentricity about the Sun and thus assume a perfectly circular orbit. This is not too inaccurate as the Earth's actual orbital eccentricity is 0.0167; in our model we will assume it to be zero. Second, we will assume that the two primary bodies are in circular motion about their centers of mass, or barycenters. Lastly, we state the previous assumption in which \( m_3 << m_1, m_2 \) such that it does not affect their respective motion. Thus, we can reduce the equations of motion for the three body system

\[
\begin{align*}
\ddot{r}_1 &= G\left(\frac{m_2}{|r_{12}|^3}r_{12} + \frac{m_3}{|r_{13}|^3}r_{13}\right) \\
\ddot{r}_2 &= G\left(\frac{m_1}{|r_{21}|^3}r_{21} + \frac{m_3}{|r_{23}|^3}r_{23}\right) \\
\ddot{r}_3 &= G\left(\frac{m_1}{|r_{31}|^3}r_{31} + \frac{m_2}{|r_{32}|^3}r_{32}\right)
\end{align*}
\]

Where \( r_{i,j} = r_i - r_j \) is the vector pointing from the tip of \( r_i \) to the tip of \( r_j \). Because we assume that \( m_3 << m_1, m_2 \) we can essentially assume that \( m_3 \) is negligible and allow it to be zero, focusing on Equation (4.8). Also, because of our circular orbit assumption, \( r_1(t), \dot{r}_1(t), r_2(t) \) and \( \dot{r}_2(t) \) are known, reducing to:

\[
\ddot{r}_3 = G\left(\frac{m_1}{|r_1(t) - r_3|^3} (r_1(t) - r_3) + \frac{m_2}{|r_2(t) - r_3|^3} (r_2(t) - r_3)\right)
\]

Where \( r_1(t) \) and \( r_2(t) \) are known functions of time. Effectively normalizing these equations and creating a dimensionless problem, we simplify the solution methods and can obtain reasonable estimates for final halo orbit parameters as well as lowest-energy transfer trajectory solutions.

In order to solve the CRTBP, we examine the equations of motion and perform operations to translate them into the rotating frame about the barycenter of the Earth-Sun system. Additionally we nondimensionalize the equations. The result yields the following equations for the vector field of the gravitational field:

\[
\begin{bmatrix}
\dot{x} \\
\dot{y} \\
\dot{z} \\
\ddot{x} \\
\ddot{y} \\
\ddot{z}
\end{bmatrix} =
\begin{bmatrix}
u \\
v \\
w \\
2\dot{y} + \frac{\partial U}{\partial x} \\
-2\dot{x} + \frac{\partial U}{\partial y} \\
\frac{\partial U}{\partial z}
\end{bmatrix}
\]

In which we define:
\[ p = (x, y, z) \]
\[ q = \dot{p} = (\dot{x}, \dot{y}, \dot{z}) = (u, v, w) \]
\[ \mu = \frac{m_2}{m_1 + m_2} \]
\[ U(x, y, z) = \frac{1}{2} (x^2 + y^2) + \frac{1 - \mu}{\sqrt{(x + \mu)^2 + y^2 + z^2}} + \frac{\mu}{\sqrt{(x + (1 - \mu))^2 + y^2 + z^2}} \]

(4.11)

To solve for the motion of the third body in the CRTBP, it helps to solve for the conserved Jacobi Integral \( C \), using Equation 4.12.

\[ C = \left( x^2 + y^2 \right) + 2 \frac{1 - \mu}{\sqrt{(x + \mu)^2 + y^2 + z^2}} + 2 \frac{\mu}{\sqrt{(x + (1 - \mu))^2 + y^2 + z^2}} - (\dot{x} + \dot{y} + \dot{z}) \]

(4.12)

This quantity corresponds to a set of constant velocity curves in the CRTBP. That is to say, if an energy level is fixed as well as an initial position, then the velocity of the spacecraft will remain fixed.

In order to solve for the location of the Lagrangian points, set \( u = v = w = 0 \) because, at the Lagrangian point, there is equilibrium in gravitational forces acting upon the spacecraft. It is also true that the solution exists on the line between the barycenters of the two primaries in the system, that is to say \( y = z = 0 \). We can then linearize the system about the \( L_1 \) Lagrangian point and proceed to study the dynamics and stability of the space surrounding the Lagrangian point. This method was employed in obtaining the MATLAB solutions for the CRTBP. Figures 4.7-4.9 illustrate families of solutions to the CRTBP obtained in MATLAB.

Figure 4.7: Family of halo orbits centered about the Lagrangian point \( L_1 \)
Figure 4.8: XY-plane view of direct transfer trajectory

Figure 4.9: Sample direct transfer trajectory into final halo orbit
4.3 Orbit Design

This section outlines our final orbit design specifications and the subsequent model using Satellite Tool Kit (STK).

4.3.1 STK/Astrogator Model

STK is the most widely used orbital simulation tool in the aerospace industry. One of the reasons that the program is so widely used is for a feature that it contains called Astrogator, an iterative orbital solver that helps to narrow down orbital parameters without having to manually numerically solve all of the orbital parameters that go into a simulation. This STK feature was used to simulate all three simulations conducted for this mission. These simulations include halo orbit simulation with instantaneous satellite placement, orbital transfer, and orbital transfer into orbit insertion and propagation.

4.3.1.1 Halo Orbit Simulation with Instantaneous Satellite Placement

This orbital simulation was used to understand the power absorption, downlink schedule and coverage simulations. For this situation a satellite was placed at the inflection point of the halo orbit. After creating the initial state, the orbit was propagated from that point. An image of this simulation can be seen in Figures 4.10a and 4.10b.

![Orbit Top View](image1)

![Dynamic Orbit View](image2)

Figure 4.10: The Orbit

This simulation varied the initial velocity values at $L_1$ to put the spacecraft into a halo orbit with the desired parameters. As Astrogator varied the values for the initial velocity, it matched it with the given SOHO orbit values that we used for the initial simulations. These values were
the orbit amplitudes: maximum x-value of 206,448 km, maximum y-value of 666,672 km, and maximum z-value of 120,000 km. The program varied the initial velocity until it could put all three axis values at their proper number at each of the four inflection points on the ellipse.

4.3.1.2 Orbital Transfer

This orbital simulation was used to outline the launch and orbital transfer, and how they would affect the parking time, and burn values. From this simulation the HAVEN-1 satellite is launched from the launch site of Kwajalein Atoll. The launch time is set at 1st September 2009, however the simulation has been constructed such that the launch date can be changed, and the simulation will re-run setting up the new parking time values and burn values. For the Astrogator values, the parking orbit duration time, and the burn intensity are varied until the satellite reaches the final orbit values for the closest point on orbit to the Earth. The transfer can be seen in Figures 4.11a and 4.11b.

![Figure 4.11: The Transfer](image)

The satellite crosses the Earth-Sun line 206,448 km away from \( L_1 \) along the x-axis and 120,000km away from \( L_1 \) along the z-axis. This is the closest earth point to the desired orbit that we are trying to create for the mission. This burn value for the orbital transfer from the simulation comes out to 3.115 km/sec. This is very close to our estimated values using research and MATLAB calculations.

4.3.1.3 Orbital Transfer into Orbit Insertion and Propogation

This simulation is intended to fully simulate the satellite from launch to orbit insertion to the station keeping maneuvers. Working from the orbital transfer simulation that has already been created it is possible to do one or two orbit insertion burns that will take the satellite from orbital transfer directly into the halo orbit. Astrogator has been used to attempt this by varying the burn
values and trying impose the first quadrant orbit parameters onto that simulation. This method will not work in order to simulate this orbit, there are too many variables when the satellite is being placed in an orbit with 6 degrees of freedom; 3 position, and 3 velocity. Further work will need to be conducted at later design stages for the spacecraft. This work will include more detailed MATLAB calculations that can give precise values that can be used to propagate the astrogator simulation.

4.3.2 Orbital Timeline for HAVEN Mission

Our selection of the direct transfer orbit has minimized the time spent in transit to our final halo orbit. Depending on the actual launch time, the HAVEN-1 spacecraft will begin its transfer trajectory insertion (TTI) maneuver within the first 90 minutes of launch. Upon the initial TTI burn, the spacecraft will begin its transit to its $L_1$ destination. HAVEN-1 will be in the vicinity of $L_1$ within 30 days of the TTI maneuver, with final insertion into the halo orbit in approximately 100 days, depending on accuracy of the TTI burn, and magnitude of midcourse correction maneuvers. Thus, it is fully anticipated that HAVEN-1 will be in its final halo orbit within 110 days of launch.
Chapter 5

Guidance, Navigation and Control

The Haven project must provide continuous accurate and reliable information on solar particle events for space weather forecasting. To accomplish this task the craft must be able to determine and control its attitude for the entire mission lifetime. It is the task of the Guidance, Navigation and Controls (GNC) subsystem to determine the proper mission architecture to accomplish this feat within the spacecraft’s cost, mass, and power margins.

5.1 GNC Requirements

<table>
<thead>
<tr>
<th>GNC-01</th>
<th>Sensors</th>
<th>Assure spacecraft is sun pointing within sun sensor FOV</th>
<th>POWHL-01</th>
</tr>
</thead>
<tbody>
<tr>
<td>GNC-02</td>
<td>Sensors</td>
<td>Have dual redundancy in star tracker system</td>
<td>MISS-04</td>
</tr>
<tr>
<td>GNC-03</td>
<td>Sensors</td>
<td>Have dual redundancy in sun sensor system</td>
<td>MISS-04</td>
</tr>
<tr>
<td>GNC-04</td>
<td>Orientation</td>
<td>Knowledge of satellite pointing vector (within 2 Deg. Half Cone)</td>
<td>PAYLOAD-01</td>
</tr>
<tr>
<td>GNC-05</td>
<td>Orientation</td>
<td>Ability to modify satellite pointing vector (within 10° Half Cone)</td>
<td>PAYLOAD-01</td>
</tr>
<tr>
<td>GNC-06</td>
<td>Operation</td>
<td>Ability to accurately point High Gain Antenna (within 30° Half Cone)</td>
<td>COMM-01</td>
</tr>
</tbody>
</table>

Table 5.1: Low Level GNC Requirements

GNC-01: In order for the mission to be successful, the payload must keep a continuous view in the direction of incoming solar particle events. It is GNC’s primary responsibility of insuring the spacecraft is always pointed correctly to assure proper payload operations.

GNC-02 and GNC-03: It is of utmost importance that the GNC system not fail, thus we require GNC to have redundancy on all essential sensors and controls.

GNC-04: Knowledge of the spacecraft attitude is necessary for both proper control of the spacecraft as well as extrapolating data from the payload sensor.

GNC-05: The spacecraft must be pointed within a 10°. Half Cone in order for the power system and the payload to function properly.

GNC-06: The High Gain Antenna must be pointed at the earth in order for proper communications to take place. The antenna cannot be directly mounted to the spacecraft because the craft
is required to point at the Sun, so a pointing mechanism is required to direct the antenna.

All subsystems are influenced by the orientation of the spacecraft. The three primary GNC requirements have been established from the subsystems that the spacecraft must achieve:

- 10° half cone Sun pointing accuracy for solar panels at all times (Power)
- 32° half cone Earth pointing accuracy for high gain antenna at all times (Communication)
- 2° spacecraft attitude accuracy in any orientation at all times to allow effective feedback (Controls)

Our mission requires that the payload be pointing at the sun so that we will be able to constantly monitor the sun for SPEs. The pointing requirements for the Payload subsystem is a 15° half cone to the Sun at all times. The Power subsystem requires the solar panels point at the Sun with 10° half cone which also satisfies the Payload requirements. In addition to this, the communication system will need to be able to stay in constant communication with Earth due to the fact SPEs could occur at unpredictable times.

### 5.2 GNC System Architecture

To maintain all requirements, the GNC subsystem will contain: (2) Bradford Engineering Fine and Coarse Sun Sensors for quick response attitude determination, (2) Terma HE-5AS Star Trackers for precision attitude determination in any orientation, (4) Surrey Reaction Wheels for majority of attitude control, (12) Thrusters for Orbit Station Keeping and Secondary Attitude control and Reaction Wheel Desaturation, and (1) Antenna Pointing Mechanism to decouple Earth communication pointing requirements from the spacecraft attitude on approach and final orbit.

### 5.3 Attitude Sensor Suite

To maintain the pointing requirements, the satellite requires the ability to measure its attitude in space in any orientation. The sensors must provide reliable and accurate orientations with a sample time that is practical for online feedback control. The performance of the sensors have been validated by the successful GNC Spacecraft attitude simulation model discussed later.

#### 5.3.1 Attitude Sensor Selection

Based on the mission requirements that were defined for the attitude determination system, trade studies for the sensor suite were performed. A system composed of both star trackers and sun sensors for coarse and fine attitude sensing has been defined. During our mission, the sun sensor will provide coarse measurements of the sun for general pointing of our spacecraft.
The sun sensor will be used in tandem with the star tracker, which will provide precise attitude and location measurements based on visible star patterns. The star tracker uses its built-in processor and star library to locate and identify stars and then output a position based on the reference of those star patterns. The reason this combination was used is because the sun sensor provides only the coarse measurements for general sun pointing with a quicker acquisition time, while the star tracker provides the precise attitude information but takes longer to reacquire attitude. The Sun Sensor will provide immediate notification if the sun pointing requirements are not met and will force the satellite into the Emergency Sun Reacquisition (ESR) mode. Following are the trade studies that we performed for both sun sensor and star tracker selection. The final selections are discussed in depth afterwards.

<table>
<thead>
<tr>
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<tbody>
<tr>
<td>Ball Aerospace</td>
<td>283.2 H</td>
<td>0.049</td>
<td>9 W max</td>
<td>500,000</td>
<td>-30 to 50</td>
<td>7.8 x 7.8</td>
<td>10 Hz</td>
<td>Yes</td>
<td>Nemo, Mitex, Roadrunner</td>
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<tr>
<td>CT-602</td>
<td>198.12 H</td>
<td></td>
<td>8 W typical</td>
<td></td>
<td></td>
<td></td>
<td></td>
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<tr>
<td>CT-633</td>
<td>134.62 H</td>
<td>0.49</td>
<td>8 W max</td>
<td>475,000</td>
<td>-25 to 45</td>
<td>17.5 x 17.5</td>
<td>5 Hz</td>
<td>Up to 5 stars</td>
<td></td>
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<tr>
<td>CT-633</td>
<td>142.24 H</td>
<td></td>
<td>8 W typical</td>
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<tr>
<td>Galileo Avionica</td>
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<td>3</td>
<td>13.5 W max</td>
<td>375,000</td>
<td>-30 to 60</td>
<td>16.4 x 16.4</td>
<td>10 Hz</td>
<td>Up to 10 stars</td>
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<td>8.9 W</td>
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<td>A-STR</td>
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<td></td>
<td>8.9 W</td>
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<tr>
<td>A-STR</td>
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<td>6.8 W</td>
<td>375,000</td>
<td>-40 to 20</td>
<td>22 x 22</td>
<td>4 Hz</td>
<td>Yes</td>
<td></td>
</tr>
<tr>
<td>Terma HE-5AS</td>
<td>62 (up to 346)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Terma HE-5AS</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 5.2: Star Tracker Trade Study

We have selected the Terma He-5AS Star Tracker as opposed to other star trackers because it is heritage technology (Nemo, Mitex, Roadrunner, and more) that meets our mission requirements and gives our mission more flexibility in terms of weight and cost over other options. Additionally, it only draws 6.8W of power at a weight of 2.2 kg.

Figure 5.1: Terma HE-5AS Star Tracker
Table 5.3: Sun Sensor Trade Study

The Bradford Engineering Fine and Coarse Sun Sensor (Figure 5.2) was chosen because it is also heritage technology (Alphabus, SPOT, XMM, and more), and it meets the mission requirements providing better overall system performance parameters in terms of operating temperature, field of view, and reliability. In addition to this, it contains 2 levels of fidelity in the data (coarse and fine) to further improve our measurements and accuracy. Also, the sun sensor draws less than 0.2 Watts of power and weighs under half of a kilogram. This sun sensor was selected because it provides the highest field of view and widest operating temperature range.

In conclusion, the spacecraft will be outfitted with (2) Terma He-5AS Star Trackers (shown in Figure 5.1) in tandem with (2) Bradford Engineering Fine and Coarse Sun Sensors (shown...
This system will provide the spacecraft with attitude data as well as serve as feedback in the control system for the spacecraft. The star trackers provide higher orientation accuracy, but require a longer cold start acquisition time (45 seconds). The sun sensors provide immediate acquisition time and serves as a warning for the system to go into an Emergency Sun Reacquisition (ESR) mode.

This attitude sensor configuration has extensive heritage on many missions and will provide the precision and reliability our mission requires. Genesis, ACE, and SOHO are all past missions that have used this sensor configuration for attitude determination and control. It is a reliable configuration that not only provides precise attitude data, but can also be used to autonomously recover the spacecraft from a “lost in space” condition with its onboard star library and processor.

5.3.2 Actuation Suite

Using the following control mechanisms, the spacecraft will be able to point its sensors, solar panels, and communications antenna at all of their respective targets. Four reaction wheels are used as the primary pointing mechanism. Three axis pointing control using reaction wheels eliminates thruster chatter, and requires a smaller fuel budget for attitude corrections when compared to a spin stabilized platform, which require large thruster burns to correct for spin nutations.

5.3.2.1 Actuators

The spacecraft will have a set of (4) Surrey reaction wheels to control its attitude throughout the mission. The wheels will be placed in a four reaction wheel pyramid configuration for a redundant, long lifetime design. These wheels were chosen due to their flight heritage and fall within the stringent weight and power budgets of the project. These wheels will provide the spacecraft with a reliable and economical attitude control system, while maintaining proper control characteristics. (Seen in Figure 5.3).
The Microwheel 10SP was chosen after a trade study of the existing COTS attitude control solutions for small spacecraft. Five of the primary choices that the team considered are profiled in Table 5.4 below. The choices were narrowed to the Microwheel and the SunSpace wheels because of their small size and adequate momentum storage. The adequacy of the momentum storage was calculated by extrapolating the maximum slew rate from the momentum wheel at the maximum reaction wheel momentum based on a model satellites’ moment of inertias. The model satellite was a 100kg 1.5m long, 0.5 radius cylinder. The Microwheel was chosen because of its longer design lifetime. The actual spacecraft control rates were later validated using the final satellites’ moments of inertia.

<table>
<thead>
<tr>
<th>Wheel</th>
<th>Surrey Microwheel 10SP UM</th>
<th>Bradford Engineering W05</th>
<th>SunSpace Reaction Wheel</th>
<th>Valley Forge Composites VF MR 4.0</th>
<th>Vectronic Aerospace RW-01</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass [kg]</td>
<td>1.1</td>
<td>3.2</td>
<td>2</td>
<td>2.6</td>
<td>1.8</td>
</tr>
<tr>
<td>Power [W] (max-nominal)</td>
<td>3.5-0.7</td>
<td>73-16</td>
<td>1.8-0.3</td>
<td>19-3</td>
<td>25-4</td>
</tr>
<tr>
<td>Design Life [yr]</td>
<td>7.5</td>
<td>10</td>
<td>1.9</td>
<td>10</td>
<td>4.6</td>
</tr>
<tr>
<td>Momentum [Nms]</td>
<td>0.42</td>
<td>10</td>
<td>5</td>
<td>10</td>
<td>4.6</td>
</tr>
<tr>
<td>Torque [mNm]</td>
<td>10</td>
<td>100</td>
<td>50</td>
<td>20</td>
<td>20</td>
</tr>
<tr>
<td>Heritage</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>–</td>
</tr>
<tr>
<td>Max Satellite rotation [°/s]</td>
<td>1.2</td>
<td>20.3</td>
<td>1.9</td>
<td>11.6</td>
<td>2.9</td>
</tr>
</tbody>
</table>

Table 5.4: Reaction Wheel Trade Study

5.3.2.2 Reaction Wheel Configuration

A pyramid reaction wheel configuration (shown below in Figure 5.4) allows three axis control of the spacecraft while adding redundancy. Only three of the four wheels are necessary for complete spacecraft control; which means that any one of the four can act as a backup in case of
a wheel failure. Losing the ability to use reaction wheels to maintain orientation would greatly reduce the lifetime of the spacecraft due to the fuel usage of the attitude sensors.

![Four Reaction Wheel Pyramid Configuration](image)

**Figure 5.4: Four Reaction Wheel Pyramid Configuration**

### 5.3.2.3 Expected Mission Behaviour

Based on sun sensor and star sensor data, the control system will manipulate the spacecraft’s attitude to keep the payload facing the sun. The spacecraft needs to keep a 0.5° tilt with respect to the Sun-Earth line to stay pointed at the sun. The actuation suite will provide this attitude manipulation during the orbit based on the data from the attitude determination sensors. The expected perturbations and necessary controls corrections are projected to be quite small, and so the small torques and momentum storage of these reaction wheels is more than adequate to perform the spacecraft’s mission. Thrusters will be used to desaturate the reaction wheels stored momentum when necessary and will also serve as a backup in case of complete reaction wheel system failure.

### 5.3.2.4 Heritage

The Surrey 10SP reaction wheels have flown on the following small satellite missions; all are cube shaped craft ranging from 90-120 kg such as the Deimos-1 (2008), UK-DMC2 (2 units, 2008), NigeriaSat-2 (2009), and the NX (2009).

Our spacecraft weight is expected to be about 140kg, making it heavier than all of the heritage satellites. Due to our low slew rate requirements our momentum wheels will be used to reduce pointing chatter and maintain pointing orientation. All of these heritage spacecraft have been developed and tested by Surrey Satellite Technology Ltd, based in the UK.
A high fidelity simulink dynamics and controller model (seen in Figure 5.5) was designed to estimate the performance of the GNC component selection. A nonlinear satellite dynamics model takes the applied moments from the reaction wheels and calculates the new attitude and angular velocity. This fed back to a controller which determines the necessary moments the reaction wheels must produce based on the reference commands.
5.4.1 Modeling Considerations

To increase the accuracy of the model, several characteristics of a real system were taken into account. The moments of inertias about the three axes were calculated by using a 3D CAD assembly model of the spacecraft and their appropriate dimensions/weights. Three PD controllers were designed and tuned to determine the moments required from the reaction wheels. The reaction wheels output torques were saturated at the specifications given by their manufacturer. Finally, the state of the system (Roll, Pitch, and Yaw) was discretized based on the sampling rate of the sensors. Future work would consider the possibility of momentum dumping through the thrusters, noise characteristics of the attitude sensors, limitations in the reaction wheels' output delay, and the recovery from a reaction wheel failure.

5.4.2 Model Results

In this example, the reference attitude (Roll: Yellow, Pitch: Purple, and Yaw: Cyan) were given as $\phi = 45^\circ$, $\theta = 90^\circ$, and $\psi = 45^\circ$. Figure 5.6 shows the spacecraft reorienting to the desired orientation from the initial state of $\phi = 0^\circ$, $\theta = 0^\circ$, and $\psi = 0^\circ$. Since this graph is a reflection on what actually occurs, the attitude information is passed through a zero order hold with the sampling time of the onboard sensors. These results show that the sensor and actuator outfit will satisfy the requirements for the mission. The following graphs show how the error for any required pointing angle drops to zero within a time period that is acceptable by our mission.

Figure 5.6: Spacecraft Attitude: Initial State [$\phi = 0$, $\theta = 0$, $\psi = 0$]° to Final State [$\phi = 45$, $\theta = 90$, $\psi = 45$]°
Figure 5.7: Actual Attitude Error: Initial State \([\phi = 0, \theta = 0, \psi = 0] \, ^\circ\) to Final State \([\phi = 45, \theta = 90, \psi = 45] \, ^\circ\)

Figure 5.8 shows the actual attitude error while Figure 5.9 shows the attitude error in the discrete form the controllers would receive from the sensor limitations. Figure 5.8 shows the angular velocity the spacecraft achieves. Finally, Figures 5.10 - 5.12 show the Spacecraft Attitude vs. Angular velocity. These results show that the sensor and actuator outfit will satisfy the requirements for the mission.
Figure 5.8: Discrete Attitude Error: Initial State $[\phi = 0, \theta = 0, \psi = 0] \, ^\circ$ to Final State $[\phi = 45, \theta = 90, \psi = 45] \, ^\circ$

Figure 5.9: Angular Velocity: Initial State $[\phi = 0, \theta = 0, \psi = 0] \, ^\circ$ to Final State $[\phi = 45, \theta = 90, \psi = 45] \, ^\circ$
Figure 5.10: Roll Angle vs. Angular Velocity

Figure 5.11: Roll Angle vs. Angular Velocity
Figure 5.12: Yaw Angle vs. Angular Velocity
Chapter 6

Command and Data Handling

The Command and Data Handling subsystem (C&DH) includes the required computers and network components for processing, storage, and data handling methods of the satellite. The subsystem includes two Micro Space Inc Proton200k processing boards, one for GNC processing and the other for C&DH processing. The boards are compatible with our network architecture and ample processing power to run all satellite operations. We chose a space rated TCP/IP networking protocol to link all the components of the satellite together using a relatively new SpaceWire technology. Below is an outline of the requirements, components chosen, and system architecture.

6.1 Requirements

<table>
<thead>
<tr>
<th>CDH-01</th>
<th>CDH</th>
<th>Uplink and stored command management (180 Mb/day)</th>
<th>COMM-08</th>
</tr>
</thead>
<tbody>
<tr>
<td>CDH-02</td>
<td>CDH</td>
<td>Autonomous Fault Protection</td>
<td>MISS-04</td>
</tr>
<tr>
<td>CDH-03</td>
<td>CDH</td>
<td>Subsystem Intercommunication</td>
<td>MISS-04</td>
</tr>
</tbody>
</table>

Table 6.1: Low Level C&DH Requirements

CDH-01: The communications system data throughput limits the telemetry data that the CDH system can downlink to less than 180Mb/day

CDH-02: The CDH system must be able to protect itself against unforeseen faults, this will ensure that the spacecraft cannot be lost due to a software or memory fault.

CDH-03: CDH must gather system health information as well as mission specific data from subsystems. This will allow fault detection and proper mission functionality.

The primary function of the C&DH subsystem is to provide data processing and storage to the satellite while minimizing single event upsets (SEU) such as bit flips. An SEU in critical data can cause the satellite to falsely register its current state jeopardizing the mission. In the event that a SEU should occur, the C&DH subsystem should be capable of responding in a manner
that is not harmful to the satellite and allows the satellite to return to a safe and fully operational state. Each sensor should also be capable of communicating with the C&DH computer as well as any other component on the satellite that requires sensor telemetry. In order to ensure proper communications within the satellite, an effective and redundant network architecture should be implemented to ensure proper data routing. Finally, C&DH subsystem must interface with the telecommunications subsystem and determine data packaging and transmission.

6.2 C&DH System Architecture

The C&DH system architecture is key to the storage and processing of all data and telemetry throughout the satellite.

6.2.1 C&DH Board

In order to drive our C&DH subsystem, our overall system architecture includes two processors board. The board that was chosen for both the C&DH and GNC is the Space Micros Inc.'s Proton200k. The first is dedicated solely to the GNC subsystem and is in charge of all station keeping, attitude maintenance, and kinematics calculations. The second is dedicated solely to C&DH and is in charge of acting as a watch dog for the satellite, reporting the telemetry and status of all sensors, and deciphering/composing all command messages.

Figure 6.1: Space Micro Inc Proton200k

6.2.1.1 Specs

The Proton200k is a high speed rad-hardened processor capable of running at 300 MHz with a total ionizing dose (TID) of over 100 krad and experiencing 1 SEU per 1000 days (in GEO). The
board runs at a low 5 to 7 W, comes with an option for 512 MB of rad-hardened flash, and comes equipped with technology for SEU and Single-event Functional Interrupts detection and mitigation. The Proton200k is SpaceWire compatible to allow for easy integration into the networking scheme described in the next section. Everything used in the C&DH subsystem is currently in use on other missions and has been developed. We predict the procurement of hardware, network and messaging protocol setup, and testing to take no longer than 1 year.

6.2.1.2 Heritage

Space Micro Inc boards have been used on several missions since it was founded in 2002. While the Proton boards are relatively new, NASA, DARPA, ESA, and the Air Force have used them or is planning on using them in space for purposes such as the Space Station Medical Computer, Lockheed Martin's ANGELS nanosat, and DARPA C&DH Nanosat Computer.

6.2.1.3 Alternatives

The BAE Systems RAD6000 Space Computer was used as a baseline since it had been suggested from past University of Michigan C&DH boards. The Proton200k was chosen over the RAD6000 because of its faster processor, lower power consumption, and its SEU prevention design.

6.2.2 Network Architecture

The chosen network architecture is the star configuration using IEEE 1355, also known as SpaceWire. SpaceWire is similar to a standard ground based TCP/IP computer network. Each sensor or component connects to a centralized SpaceWire router by its own individual connection through the equivalent of a network interface card (NIC), designated by SMCS1665SpW. It is the routers duty to forward all incoming messages to the addressed subsystem or component. Figure 6.2 below shows the expected SpaceWire network architecture.
6.2.2.1 Alternatives

The most common network topology used in satellites is an IEEE 1553 Bus line. It is used for most of the computer racks on the International Space Station (ISS) as well as being used on the sun STEREO mission. All components communicate to each other through a standard bus line consisting of a single resistor terminated wire. A downside to using a bus line is communication streams are slow since a sensor must first verify no one is transmitting on the cable before a transmission is sent. Every component must read all signals passed to verify if it is addressed to them. This uses up processing time and energy. There is a possibility for data collisions to occur if two components on the network attempt to transmit before the transmission time has passed. By implementing the SpaceWire topology we can eliminate the data collisions and noise over a wire as well as increasing the rate at which data can be taken. The router will make sure that the correct component gets the correct data packets so there will not be any processing power lost on decoding messages that are not needed by a component.

6.2.2.2 Specs

6.2.2.2.1 SpaceWire Router Each router, shown in Figure 6.3, is capable of handling up to 8 SpaceWire compatible modules which is more than sufficient for the 5 required by our network architecture. The router draws 2.4 Watts of power and is capable of transferring data at 100-400 Mbps.
6.2.2.2 NICS In order to interface with the network certain components require a NIC, shown in Figure 6.4 to translate the output of the component into a SpaceWire compatible format. Each NIC draws a 0.7 W at its max processing load of 200 Mbps.
6.2.2.3 Heritage

While the SpaceWire IEEE standard is a relatively new technology, it has been implemented on many current missions such as NASA’s Lunar Reconnaissance Orbiter, the Swift Gamma-ray Explorer, and is planned to be used on the James Webb Space Telescope. It has also been adopted by the ESA and is currently used on the ISS European Drawer Rack, the Mars Observer and is planned to be used on Gaia which is an up scaled version of Hubble.

6.2.3 Data Budget- includes all data collected and transmitted

A data budget, shown in Appendix E was created to insure the C&DH architecture will fulfill all mission requirements. Our communications subsystem is capable of transmitting 180 MB/day therefore we need out data sampling from a given day to be less than 180 MB. The satellites estimated data production is 72.88 MB/day. Including a 50% overhead we estimate the satellite to produce 109.32 MB/day which is well within the given communications constraints.
Chapter 7

Communications System

The $L_1$ satellite communication system is responsible for transmitting satellite telemetry and payload data warning of Solar Particle Events (SPE’s), receiving uplinked commands from Earth ground stations, and allowing for satellite tracking, telemetry and control during satellite insertion from low Earth orbit to $L_1$ and during operations at $L_1$.

7.1 System Requirements

The communications subsystem has requirements that must be achieved in order to have a successful overall mission. One requirement is for it to provide at least 10 minutes warning notification of SPE’s to those in LEO to have an adequate amount of time to take shelter. Another requirement for the communication subsystem is that it must be able to support 65 kbps transmitting. This ensures that all down linking of a day’s worth of telemetry data can be done within one pass of the $L_1$ satellite over a ground station. Furthermore, the communications system should be robust against mechanical and electrical failures. This is taken care of in redundant radio systems as well as primary and secondary antenna systems discussed below. Furthermore, system communications should be robust against major types of interference such as Earth atmospheric attenuation, man-made interference, and galactic noise. This is taken care of by using frequencies under 10 GHz, signal conditioning, and the use of multiple ground stations on Earth.
### Table 7.1: Low Level Communication Requirements

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>COMM-01</td>
<td>Coverage</td>
<td>Provide warning notification of CME to those in LEO with adequate time to take shelter</td>
</tr>
<tr>
<td>COMM-02</td>
<td>Coverage</td>
<td>No communication gaps greater than 20 minutes</td>
</tr>
<tr>
<td>COMM-03</td>
<td>System</td>
<td>Communication robust against earth weather changes, galactic noise, and man-made noise</td>
</tr>
<tr>
<td>COMM-04</td>
<td>System</td>
<td>System must be robust against mechanical/electrical failure</td>
</tr>
<tr>
<td>COMM-05</td>
<td>Lifetime</td>
<td>System designed to last for 5 years</td>
</tr>
<tr>
<td>COMM-06</td>
<td>Power</td>
<td>Peak power consumption no more than 40 Watts</td>
</tr>
<tr>
<td>COMM-07</td>
<td>Mass/Size</td>
<td>Comm system will have 10 kg max mass and fit within designated section of S/C</td>
</tr>
<tr>
<td>COMM-08</td>
<td>Data Transfer</td>
<td>Must be able to support TBD bps Receive and 65 kbps transmit</td>
</tr>
<tr>
<td>COMM-09</td>
<td>Cost</td>
<td>Total allocation for system of mission $25 million</td>
</tr>
<tr>
<td>COMM-10</td>
<td>Signal Strength</td>
<td>Maintain minimum link margin of 3 dB</td>
</tr>
</tbody>
</table>

7.2 **System Drivers**

The primary communication driver of the HAVEN project is cost. The high performance requirements of the HAVEN satellite telemetry and the need for near constant coverage of both the $L_1$ satellite and customer satellites in LEO demand a network of multiple ground stations interfaced over the internet to provide real-time, simultaneous communication of warning messages to LEO satellites over several different parts of the Earth. However, currently available commercial ground station networks can be extremely expensive costing in excess of $1,000,000 per year depending on time scheduled and amount of data transferred with them. Thus, to minimize cost, scheduled time of telemetry down linking should be minimized. This places a premium on a system that can maximize data rate while still maintaining the required 3 dB link margin in system requirements. Subsequent sections will show that all satellite telemetry data can be communicated in one pass over a chosen ground station.
7.3 System Architecture

There are several different components to the communications system. The major components of the communication system architecture discussed here are transceiver, antennas, signal processing and ground station network. The purpose of the overall communications system is to provide telemetry downlink and telecommand uplink in order to transmit payload feedback from the spacecraft and to receive instructions from the ground. The transceiver is the device that relays the information between the C&DH of the payload, through the antenna, to the ground. For this mission, we are using two types of antennas. The primary antenna system is the parabolic high gain antenna (HGA). The HGA is used for sending high uplink/downlink data rates (greater than 1 kbps). Six low gain antennas will be placed radially on the spacecraft. These low gain antennas are used for low data rate transmissions and possible emergency contact situations such as loss of spacecraft attitude control or HGA pointing failure. A network of multiple ground stations on Earth will be used to minimize communication gap times from $L_1$ and to low Earth orbit.

7.3.1 Transceiver Selection

Four major transceiver manufacturers, General Dynamics, AeroAstro, Surrey Space, and L-3 Communications were identified as having available COTS transceivers fitting with communication system requirements. An emphasis was placed on using COTS products for the satellite transceiver to reduce uncertainty in this part of the system design. Specifications of these transceivers can be seen in Table 7.2. The chosen unit, the AeroAstro Modular S-Band Radio, is shown in Figure 7.1.

The AeroAstro Modular S-Band Radio was chosen based on its comparable output power with lower mass and input power than other identified candidates. The output power directly...
couples with maximum data rate. Thus, this unit minimizes system mass without penalizing the time required for down link of satellite data. Furthermore, the unit comes equipped with built-in signal filtering components, reducing the need to place extra signal conditioning components on the spacecraft. All of the transceivers considered operate in the 2 GHz frequency range so the operating frequency, in this case, wasn’t a driving factor in our decision. Furthermore the operating temperature range for all the transceivers is around -20 to 60°C, so this is also not a decision criterion.

### Table 7.2: COTS transceiver trade study

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>General</td>
<td>Multi-Mode S-Band Transceiver</td>
<td>2.025-2.12 : Rx 7 : Rx</td>
<td>2.3</td>
<td>512 : Rx</td>
<td>-20° to 65°</td>
<td></td>
</tr>
<tr>
<td>Dynamics</td>
<td>Band Transceiver</td>
<td>2.2-2.3 : TX 36 : Tx</td>
<td>&lt;6000 : Tx</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>AeroAstro</td>
<td>Modular S-Band Radio</td>
<td>2.025-2.12 : Rx 1 : Rx</td>
<td>0.9</td>
<td>1-4 : Rx</td>
<td>-20° to 60°</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Transceiver</td>
<td>2.2-2.3 : TX 14 : Tx</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Surrey Space</td>
<td>S-Band Receiver / Transmitter</td>
<td>2.025-2.12 : Rx 1.3 : Rx</td>
<td>1.4</td>
<td>19.6 : Rx</td>
<td>-20° to 50°</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Band Radio</td>
<td>2.2-2.3 : TX 6.3 : Tx</td>
<td></td>
<td></td>
<td>8000 : Tx</td>
<td></td>
</tr>
<tr>
<td>L-3 Communications</td>
<td>CXS-610 STDN/USB</td>
<td>2.025-2.12 : Rx 5 : Rx</td>
<td>2.5</td>
<td>0.3-2 : Rx</td>
<td>-24° to 61°</td>
<td></td>
</tr>
<tr>
<td></td>
<td>DSN Space Transponder</td>
<td>2.2-2.3 : TX 35 : Tx</td>
<td></td>
<td>128-256 : Tx</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

7.3.2 Antenna Selection

The selection of the antenna system needs to include both a primary antenna (high gain, directional) and secondary antennas (low gain, isotropic) for reasons of redundancy and varying data rate as described in Section [7.3.1]. Both the primary and secondary antennas need to be low in mass and should have dimensions that are compatible with the spacecraft structure.

### Table 7.3: Antenna Trade Study

<table>
<thead>
<tr>
<th>Antenna Type</th>
<th>Pros</th>
<th>Cons</th>
</tr>
</thead>
<tbody>
<tr>
<td>Parabolic</td>
<td>-High gain</td>
<td>-Narrow antenna beam width requires pointing mechanism</td>
</tr>
<tr>
<td>Directional</td>
<td>-High gain allows strong connections (high speed, and low latency) at large distances</td>
<td>-Must be properly aligned with its intended target or there will be no communications transmitted/received</td>
</tr>
<tr>
<td>Dipole</td>
<td>-Half the space of dipole since monopole length is λ/4</td>
<td>-Gain typically not as strong as directional antenna</td>
</tr>
<tr>
<td>Omnidirectional</td>
<td>-Has high gain in 2 directions</td>
<td>-Symmetry of antenna makes alignment with target easier</td>
</tr>
<tr>
<td>Patch</td>
<td>-Does not require alignment with intended target</td>
<td>-Limited transmission range</td>
</tr>
<tr>
<td>Fractal</td>
<td>-Radiation pattern allows for less accurate alignment with target</td>
<td>-Gain not as strong as directional/dipole</td>
</tr>
<tr>
<td>Evolved</td>
<td>-Small surface area</td>
<td>-Signal can be noisy or weak if antenna not properly oriented</td>
</tr>
<tr>
<td>Quadrifilar Helix</td>
<td>-No thin weak structures protruding</td>
<td>-Custom made</td>
</tr>
<tr>
<td></td>
<td>-Multiband/wideband capabilities to allow transmissions through the same antenna</td>
<td>-Only made by one company</td>
</tr>
<tr>
<td></td>
<td>-Medium gain (2-4 dB) in wider angle range for omnidirectional antenna</td>
<td>-No existing space-rated component</td>
</tr>
<tr>
<td></td>
<td>-Can be constructed for different frequency bands</td>
<td>-Custom made</td>
</tr>
<tr>
<td></td>
<td>-Requires less alignment</td>
<td>-Aproximately 1-6 months to fabricate and test</td>
</tr>
<tr>
<td></td>
<td>-Design optimized for specific environment</td>
<td>-Gain not as high as directional antenna</td>
</tr>
<tr>
<td></td>
<td>-Wide bandwidth</td>
<td>-Flight heritage ST5 mission</td>
</tr>
<tr>
<td></td>
<td>-Larger radiation distribution pattern ideal for mobile communications</td>
<td>-Large trade between beam width and gain</td>
</tr>
</tbody>
</table>

Table 7.3: Antenna Trade Study

Where the Parabolic dish is our primary transmission device and the Patch antenna is our secondary.
7.3.2.1 Primary Antenna-High Gain Antenna

Given the high gain requirements for the desired primary communication link (<25 dBi), the only feasible option for the $L_1$ satellite which still meets the communications system mass requirement of less than 10 kg is a parabolic dish. To achieve desired data transmit rates (50-75 kbps), this dish should be approximately 1.1 meters in diameter similar to the dish in Figure 7.2. This antenna will allow for the desired telemetry data rates of the communication system and should provide a robust communication link for SPE warning messages.

![Figure 7.2: High Gain Antenna (HGA), Image Courtesy of NASA](image1)

7.3.2.2 Secondary Antennas-Low Gain Patch

The communication subsystem includes the use of six low gain microstrip patch antennas to provide complete coverage around the spacecraft. These patch antennas are used for communication of data rates of up to 800 bps with a maximum gain of 6 dBi. Each patch has a mass of approximately 0.08 kg and dimensions of 10 cm x 10 cm x 2 cm.

![Figure 7.3: Secondary Patch Antenna, Image Courtesy of Antenna Development Corporation](image2)
These low gain antennas shown in Figure 7.3, made by the Antenna Development Corporation, have a total mass of 0.5 kg which accounts for roughly 7% of the overall communications mass. Antenna Development Corporation has these patch antennas for about $2000 each, so the total cost of the low gain antennas is about $12,000.

7.3.3 Signal Conditioning

Methods of optimizing the received and transmitted signals from the $L_1$ satellite have been investigated on a high level to determine necessary components. In the context of this report, signal conditioning can involve signal filtering, amplification, converting, modulation, and other ways to make the data output able to be processed faster and with higher resolution.

7.3.3.1 Signal Filtering

Signal filtering allows the communications system to extract important frequencies from incoming data while eliminating the remaining extraneous frequencies. An antenna diplexer built into the AeroAstro S-Band Radio is the primary filtering component in this system. The diplexer allows a receiver and transmitter to use the same antenna, and even allows multiple receivers and transmitters use of the same antenna, which this system uses to provide a redundant radio system.

![Figure 7.4: Example of an RF diplexer that uses a low/high pass filter](image)

This diplexer employs a bandpass filter, to first select its desired band of frequencies, in this case the S-Band. The transmitter also uses an extra low-pass filter to aid in downlink modulation of data (discussed in Section 7.3.3.2). Further low pass or high pass filtering can also be added during system prototyping to further refine signals to the transmitter and receiver. Following the bandpass filter, the signal is further refined by a combination of demodulation and analog to digital conversion.
7.3.3.2 Signal Modulation

Signal modulation is employed by both the $L_1$ satellite transceiver as well as the ground stations. Signal modulation allows for compression of data to enhance data rates on the satellite and also for receiving or transmitting of multiple sources on the ground.

The AeroAstro transceiver has preset modulation modes of binary phase shift keying (BPSK) and quadriphased phase shift keying (QPSK). QPSK modulation was chosen, despite its more complicated operating mode, since it provides the resolution of two BPSK channels without any penalty to the required link budget margin.

Signal modulation also plays a very important role in the transmission of warning messages to the selected ground network stations (Section 7.3.4). Since warning message transmissions are not scheduled one of two things must happen, either a commercial ground station must be equipped to be constantly monitoring the $L_1$ satellite and ready to transmit messages to $L_1$ or a new separate ground station must be constructed to perform this task. Both of these options can prove extremely costly and time intensive to achieve, which puts a strain on meeting the overall project cost budget of less than $25$ million and achieving first operational readiness by 2014. However, a third low cost, low complexity solution can be achieved through use of code division multiple access (CDMA) phase modulation. With this technique, low data rate messages can be received and transmitted from multiple sources simultaneously allowing the HAVEN project to work with a commercial ground station without interrupting its normal operating activities.

7.3.4 Ground Station Selection

Given an estimated $L_1$ satellite footprint, three well placed Earth ground stations should provide nearly complete coverage of the satellite. This will be demonstrated in the following discussion. With this fact known, more important criteria for ground station selection becomes operational cost, networking capability and ground station transmit and receive capabilities.
Given the large footprint of the $L_1$ satellite as shown in Figure 7.6 above, ground station location becomes much more important for communicating with the chosen customer satellites in LEO. Current in-orbit (or planned) tourist spacecraft are at altitudes of approximately 550 km and between 0 and 63 degrees inclination. Thus, maximizing communication windows and attempting to shrink communication gap times with these satellites to zero will be most important in ground station location selection. Figure 7.7 illustrates the locations of ground stations for the Universal Space Network (USN), European Space Agency (ESA), and Bigelow networks. Another important selection criterion of ground stations is their ability to work within a worldwide network, which will be discussed in Section 7.3.4.2.
To decide which ground stations have the most benefit to use, they were broken down amongst five different coverage regions. The five major ground station coverage regions for the USN, ESA, and Bigelow networks are shown in Figure 7.8 below and include:

i. Limestone, ME; Kourou, French Guiana

ii. North Pole, AK; Kiruna, Sweden

iii. Maspalomas, Canary Islands; Villafranca, Spain; Malindi, Kenya; Sturup, Sweden; Redu, Belgium

iv. Perth, Western Australia; Dongara, Western Australia

v. Na’alehu, HI; Las Vegas, NV; South Point, HI
Coverage analysis within Satellite Tool Kit (STK) has shown that regions (i) and (ii) are useful for some combinations of ground stations, but are not always necessary to obtain zero gap time for $L_1$. The best combinations (producing the fewest gaps in $L_1$ coverage) of ground stations include a ground station from each of regions (iii), (iv), and (v). However, a ground station from each of region (iv) and (v) with either Maspalomas and Malindi all provide zero gaps in $L_1$ coverage. These results are all based on the assumption that it is possible to combine ground stations from different networks.

The following analysis shows the capability of monitoring the selected tourism spacecraft in LEO using each commercial network, and finally the potential capability of a system combining ground stations from two or more of these networks. In each analysis of ground station combinations, a plot was generated showing satellite communication gap time over one full period of the Halo orbit of the $L_1$ satellite. The larger blue lines reflect higher gap times at the given date on the plot.

### 7.3.4.1.1 USN

USN stations are located at the North Pole, AK; Dongara, Western Australia; Kiruna, Sweden; South Point, HI; and Sturup, Sweden. Using all of these stations together provides the coverage illustrated in Figure 7.9.
Figure 7.9: $L_1$ Satellite Gap Times Using the Universal Space Network

The minimum USN $L_1$ gap in coverage is 36 sec, while the maximum is about 3.3 hours.

7.3.4.1.2 ESA

The ESTRACK network operated by ESA includes sites at Kiruna, Sweden; Kourou, French Guiana; Malindi, Kenya; Maspalomas, Canary Islands; Perth, Western Australia; Redu, Belgium; Villafranca, Spain. Using all of these stations together provides the coverage illustrated in Figure 7.10.

Figure 7.10: Satellite Gap Times Using ESTRACK

The minimum ESTRACK gap time for $L_1$ is 41 sec, while the maximum is approximately 1.2 hours.
7.3.4.1.3 Bigelow

The Bigelow Aerospace Corporation, who is currently the leading firm in LEO space tourism development, has ground station located in Las Vegas, NV; Limestone, ME; Na‘alehu, HI; North Pole, AK. Using all of these stations together provides the coverage illustrated in Figure 7.11.

Figure 7.11: $L_1$ Satellite Gap Times Using Bigelow Network

7.3.4.1.4 Combining Networks to Achieve Zero Gaps in $L_1$ Coverage

Figure 7.12 shows coverage provided by three ground stations; two operated by ESA and one by Bigelow in the three regions. The chosen ground stations are located at Perth, Western Australia; Las Vegas, NV; and Malindi, Kenya. As shown by the coverage plot in Figure 7.13 this ground station configuration minimizes communication $L_1$ coverage gaps, thereby maximizing possible warning time to LEO.
7.3.4.2 Ground Station LEO Performance

To model potential LEO satellites needing coverage of these warning times, three satellites were simulated with roughly the same orbit parameters of the Bigelow spacecraft currently in orbit or proposed for the future. The three simulated satellites are all at 550 km altitude with orbit inclinations of 0, 40 and 60° respectively. Initial analysis has shown that it is not possible with current available ground station network resources to achieve complete coverage of any of these satellites. Thus, there becomes a tradeoff in ground stations used and average gap time in orbit.
The results of each individual network's capability for LEO communication gap management are summarized in Table 7.4.

<table>
<thead>
<tr>
<th>% of time LEO</th>
<th>Bigelow</th>
<th>ESA</th>
<th>USN</th>
</tr>
</thead>
<tbody>
<tr>
<td>satellite has gap time below X minutes</td>
<td>60° inclin.</td>
<td>40° inclin.</td>
<td>0° inclin.</td>
</tr>
<tr>
<td>95</td>
<td>436.633</td>
<td>384.8</td>
<td>95.317</td>
</tr>
<tr>
<td>90</td>
<td>436.75</td>
<td>384.683</td>
<td>95.3</td>
</tr>
<tr>
<td>85</td>
<td>343.517</td>
<td>439.3</td>
<td>95.283</td>
</tr>
<tr>
<td>80</td>
<td>342.6</td>
<td>384.617</td>
<td>95.3</td>
</tr>
<tr>
<td>75</td>
<td>88.417</td>
<td>438.85</td>
<td>95.267</td>
</tr>
<tr>
<td>70</td>
<td>87.367</td>
<td>439.3</td>
<td>95.25</td>
</tr>
<tr>
<td>65</td>
<td>65.2</td>
<td>72</td>
<td>36.833</td>
</tr>
<tr>
<td>60</td>
<td>63.783</td>
<td>70.617</td>
<td>36.817</td>
</tr>
<tr>
<td>55</td>
<td>49.8</td>
<td>65.15</td>
<td>36.8</td>
</tr>
<tr>
<td>50</td>
<td>43.917</td>
<td>44.233</td>
<td>36.783</td>
</tr>
<tr>
<td>45</td>
<td>40.133</td>
<td>42</td>
<td>36.633</td>
</tr>
<tr>
<td>40</td>
<td>36.467</td>
<td>39.15</td>
<td>36.45</td>
</tr>
<tr>
<td>35</td>
<td>33.1</td>
<td>32.383</td>
<td>36.217</td>
</tr>
<tr>
<td>30</td>
<td>30.833</td>
<td>30.683</td>
<td>36.183</td>
</tr>
<tr>
<td>Max gap time</td>
<td>219.183</td>
<td>221.983</td>
<td>62.93</td>
</tr>
<tr>
<td>Avg. gap time</td>
<td>95.4883</td>
<td>110.85</td>
<td>95.25</td>
</tr>
</tbody>
</table>

Table 7.4: LEO satellite coverage of simulated tourism spacecraft with single ground station network

The results of this simulation show that clearly none of these networks by themselves afford the HAVEN project with an acceptable amount of gap time as the average SPE will reach Earth within 30 minutes after detection at \( L_1 \). Thus, a solution of combine ground station networks must be used. The results of a combining network simulation are shown in Table 7.5. Gap times that allow for a fulfillment of the 10 minutes warning time to seek shelter are bolded. Gap times less than 30 minutes but do not fulfill the 10 minute requirement are italicized.

<table>
<thead>
<tr>
<th>% of time LEO</th>
<th>Bigelow/ESA</th>
<th>Bigelow/ USN</th>
<th>ESA/USN</th>
</tr>
</thead>
<tbody>
<tr>
<td>satellite has gap time below X minutes</td>
<td>60° inclin.</td>
<td>40° inclin.</td>
<td>0° inclin.</td>
</tr>
<tr>
<td>95</td>
<td>77.433</td>
<td>174.933</td>
<td>36.85</td>
</tr>
<tr>
<td>90</td>
<td>75.45</td>
<td>90.233</td>
<td>36.833</td>
</tr>
<tr>
<td>85</td>
<td>65.2</td>
<td>72</td>
<td>36.833</td>
</tr>
<tr>
<td>80</td>
<td>63.783</td>
<td>70.617</td>
<td>36.817</td>
</tr>
<tr>
<td>75</td>
<td>49.8</td>
<td>65.15</td>
<td>36.8</td>
</tr>
<tr>
<td>70</td>
<td>43.917</td>
<td>44.233</td>
<td>36.783</td>
</tr>
<tr>
<td>65</td>
<td>40.133</td>
<td>42</td>
<td>36.633</td>
</tr>
<tr>
<td>60</td>
<td>36.467</td>
<td>39.15</td>
<td>36.45</td>
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<tr>
<td>55</td>
<td>33.1</td>
<td>32.383</td>
<td>36.217</td>
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<tr>
<td>50</td>
<td>30.833</td>
<td>30.683</td>
<td>36.183</td>
</tr>
<tr>
<td>Max gap time</td>
<td>229.183</td>
<td>221.983</td>
<td>62.93</td>
</tr>
<tr>
<td>Avg. gap time</td>
<td>95.4883</td>
<td>110.85</td>
<td>95.25</td>
</tr>
</tbody>
</table>

Table 7.5: LEO satellite coverage of simulated tourism spacecraft with multiple ground station network

Based on the results of this simulation it is obvious that the best ground station network to pursue is a combination of the Bigelow and ESA networks as this combination is the only one that demonstrates the ability to reach LEO with a warning message on the average for all three simulated satellites before the SPE reaches Earth. Currently, based on these results a space tourism venture could start within an equatorial space station for space walk’s as this system demonstrates that over 65% of the time the space station would have acceptable gap times in coverage to respond to an SPE warning. Furthermore, analysis shows that similar results can
be yielded using a combination of just six ground stations between the Bigelow and ESA networks. The locations of these ground stations and results of the simulation are shown in Table 7.6. Using this solution yields fewer points of failure in the ground station network, and less data sources to manage for the central command hub to manage when distributing warning messages. Thus, this solution is preferable.

<p>| Las Vegas, NaŠalehu, Kourou, Maspalomas, Malindi, Perth |</p>
<table>
<thead>
<tr>
<th>% of time LEO satellite has gap time below X minutes</th>
<th>60° inclination angle [mins]</th>
<th>40° inclination angle [mins]</th>
<th>0°inclination angle [mins]</th>
</tr>
</thead>
<tbody>
<tr>
<td>95</td>
<td>144.117</td>
<td>174.933</td>
<td>36.85</td>
</tr>
<tr>
<td>90</td>
<td>89.65</td>
<td>90.233</td>
<td>36.833</td>
</tr>
<tr>
<td>85</td>
<td>87.2</td>
<td>77.483</td>
<td>36.817</td>
</tr>
<tr>
<td>80</td>
<td>86.8</td>
<td>76.283</td>
<td>36.817</td>
</tr>
<tr>
<td>75</td>
<td>76.717</td>
<td>65.383</td>
<td>36.8</td>
</tr>
<tr>
<td>70</td>
<td>74.233</td>
<td>49.667</td>
<td>36.783</td>
</tr>
<tr>
<td>65</td>
<td>47.417</td>
<td>43.917</td>
<td>19.233</td>
</tr>
<tr>
<td>60</td>
<td>44.183</td>
<td>40.3</td>
<td>19.233</td>
</tr>
<tr>
<td>55</td>
<td>42.367</td>
<td>32.967</td>
<td>19.217</td>
</tr>
<tr>
<td>50</td>
<td>36.55</td>
<td>31.683</td>
<td>19.2</td>
</tr>
<tr>
<td>Max gap time</td>
<td>146.367</td>
<td>176.717</td>
<td>36.85</td>
</tr>
<tr>
<td>Avg. gap time</td>
<td>39.083</td>
<td>29.583</td>
<td>23.2</td>
</tr>
</tbody>
</table>

Table 7.6: LEO coverage of simulated tourism spacecraft with six key chosen ground stations

As a final note to this discussion, many of these large gap times limiting ground network coverage were caused by a lack of ground stations identified ground stations in South East Asia and in the south pacific closer to the coast of South America. Placing ground stations in these two locations would drastically reduce LEO communication gap times.

7.3.4.3 Networking between Ground Stations

Networking between ground stations will be accomplished via a secure internet connection. Using this network a single ground station can disperse a message from $L_1$ to several satellites in low Earth orbit around various parts of the Earth simultaneously. An important part of this will be in developing proper network security over which to send warning message data, potentially between multiple ground station developers. This should be feasible, however, granted that commercial ground networks today grant their users internet access already. Thus, interconnecting multiple stations would involve establishing a central command hub for our customer at which it is possible to gather and distribute information from across the chosen ground stations.
7.3.4.4 Link Budget Analysis

This section includes link budgets of the high gain antenna to Earth ground station and low gain patch antenna to Earth communication systems. Both links support a margin greater than 3 dB. Full Link budgets can be found in Appendix [ ]. It should be noted that achieving the max data rates shown the low losses in implementation and internal signal transmission. A more conservative max data rate estimate for the HGA is about 65 kbps and about 700 bps for the low gain patch antenna.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Frequency</td>
<td>2.2 GHz</td>
</tr>
<tr>
<td>Transmit Power</td>
<td>5.0 W</td>
</tr>
<tr>
<td>Antenna Diameter</td>
<td>1.1 m</td>
</tr>
<tr>
<td>Data Rate</td>
<td>75 kbps</td>
</tr>
<tr>
<td>Link Margin</td>
<td>3.2 dB</td>
</tr>
</tbody>
</table>

Table 7.7: The high gain antenna link budget reflects a rough estimate of the maximum data rate. The low gain patch antenna link budget

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Frequency</td>
<td>2.2 GHz</td>
</tr>
<tr>
<td>Transmit Power</td>
<td>5.0 W</td>
</tr>
<tr>
<td>Max Gain</td>
<td>6 dBi</td>
</tr>
<tr>
<td>Data Rate</td>
<td>800 bps</td>
</tr>
<tr>
<td>Link Margin</td>
<td>3.4 dB</td>
</tr>
</tbody>
</table>

Table 7.8: The high gain antenna link budget reflects an rough estimate data rate assuming a uniform distribution of the gain pattern.

7.4 Space Communication Regulation (International Telecommunication Union)

Radio regulations for the International Telecommunication Union (ITU) assigns frequencies to services and has set constraints and rules to allow the services to operate without causing harmful interference to each other. The request must show that radio usage would not cause harmful interference to other communications systems. ITU frequency allocation requires that you have to secure specific frequencies by filing a request with the ITU.

The ITU has a three-step process for satellites

1. Advanced Publication of Information (API)
   This is to give other countries the possibility of looking at the information and communicating with the publishing state within four months if they think that there might be risk of interference.
2. Frequency Coordination Request (FCR)

The publishing nation sends the information to the requested nation and the ITU Radio-
communication Bureaus. The Bureau helps the publishing nation and, if necessary, sends
the request for coordination to the country concerned.

3. Notification

The ITU processes notifications and requests, and they coordinate radio frequencies and
orbital locations so as to avoid harmful interference.

Initial frequency allocation process is accomplished through ITU entities called, World Ra-
dio Conference (WRC) and the Regional Radio Conference (RRC). If frequency bands are allo-
cated for new use, the waiting period may be several years before the re-allocation process is
complete. Regulations require ITU to process each application in 4 months, but in actuality,
the average application delay is nearly 30 months. A big reason for this is because of so called
“paper satellites” which are satellites that only exist on paper but reserve potential orbital and
frequency rights. Once bands are approved, they must register on the Master International Fre-
quency Registry (MIFR). This is actually carried out within the Radio-communications Bureau
of ITU. It is recommended that because of such delay of paper processing with the ITU, our
mission must apply for a frequency allocation as soon as possible to secure the necessary fre-
quencies.

Thus, for the HAVEN system design to become a reality by 2014, work toward agreements
with the ITU and the identified commercial ground station networks should begin almost im-
mediately to ensure adequate time is allotted for negotiations, legal processing, and integration
with the established space communications community.

7.5 Concept of Operations Summary

Ultimately, this communications system architecture supports two different modes of opera-
tion. The first being regularly scheduled, daily down linking of satellite telemetry data. The sec-
ond being immediate communications downlink of SPE warning messages to the Earth ground
station network for relay to LEO.

Down linking of regular telemetry data will occur over either one chosen customer ground
station such as the Bigelow Las Vegas facility or with an agreed station from one of the identified
commercial ground stations depending on current customer ground station performance ca-
pability. Given estimated maximum data rates of the $L_1$ satellite and total estimated telemetry
accumulated by the C&DH system each day, one daily transfer of all telemetry data should take
roughly four hours. This yields a robust method of telemetry data transfer in the case of missed
passes, inclement weather, or other technical difficulties incurred by the ground station facility.

The warning message mode operates within three major steps:

1. Warning message sent from $L_1$ and beaconed to ground station network on Earth until
confirmation that message is received is sent back.
2. Warning message is distributed amongst all stations in ground network via secure internet connection.

3. Ground stations communicate message to all customer satellites in LEO simultaneously.

7.6 Impact of future technology and ground station network infrastructure development recommendations

Several groups, such as the NASA Space Communications Architecture Working Group (SCAWG) are currently researching into new methods to improve performance and efficiency of current deep space communication technology. Possible developments include development of large antenna arrays, more orbital relay networks, advancements in data modulation, and advancements in space-rated transceiver hardware. This technological growth should be monitored closely as it has the potential to raise data rate performance as much as 1-2 orders of magnitude within the next decade and also potentially lower user costs as much as an order of magnitude with the increase in user traffic.

If the HAVEN project is to be extended indefinitely for future commercial use, it would make sense to build a separate ground station network for the system. Have such an infrastructure would allow for easier upgrades to the system and in the long-run is more cost effective as the cost of each facility would likely be commensurate with the cost of using a commercial ground station network for about a year. Furthermore, with independently run ground station facilities, our customer could begin marketing the service more freely to other satellite operators in LEO. This could lead to substantial returns on the investment of building permanent facilities.
Chapter 8

Power

The electrical power subsystem (EPS) generates, stores, distributes, and controls the spacecraft’s electrical power. The most important sizing requirements and design drivers are the spacecraft’s peak electrical power and the orbital profile. As introduced in Section 2, the power budget was divided into 4 phases: ascent, cruise, transfer orbit, and on-orbit. In Section 8.2, the power requirement of each phase will be described. During the ascent phase, the launch vehicle provides the necessary power. During the rest of the mission, the spacecraft’s electrical power system is responsible for providing the required power. A thorough analysis of the electrical power system will be presented throughout this section.

8.1 Power Sources

The power subsystem has the main requirement of generating the distributing power. There are several ways in which they system architecture can satisfy that requirement. For power generation, four methods were studied: nuclear, radioisotope thermal electric generator, solar photovoltaic, and thermo electric generator. In the nuclear option, the spacecraft has a large nuclear reactor that it uses to generate power. The advantage of this technology is a large power output and long lifetime. However, the drawbacks include difficulty to attain fuel, radiation damage to components, the large size of the system, expensive cost, and waste by products. A solar photovoltaic system utilizes sun light and converts it into energy using a series of layers of different material to facilitate the flow of electrons through a cell. The advantages of this system are low mass, simplicity, and heritage. The disadvantages of this system include degradation over time, and the need for a large area for large wattage production.

Radioisotope thermoelectric generator (RTG) uses a thermal gradient created by a radioactive isotope. The heat from the fuel heats one side while the other side of the thermoelectric generator is cool, which creates energy through thermoelectric effects. The advantages of this technology are long lifetime, large power outputs, and heritage. The drawbacks include difficulty in attaining fuel, a large system mass and volume, and very expensive cost. Thermoelectric generators can be used without the radioactive isotope if they have some other heat source and are able to maintain a thermal gradient. Using other thermoelectric generation devices could provide all of the advantages of RTGs without the radioactive drawbacks.
8.2 Energy Storage

In addition to power generation, the power system must be responsible for managing energy storage. Energy storage is an integral part of the spacecraft since it provides back-up power and is a source for any instantaneous power requirement that may increase peak power demands. Three energy storage methods were studied: primary battery cells, secondary battery cells, and fuel cells. Primary battery cells have high energy density but are not rechargeable. Therefore are not suitable for long missions. Secondary batteries have lower energy densities but are rechargeable for thousands of cycles. Both primary and secondary batteries have great flight heritage. Fuel cells produce electricity from fuel and an oxidant which react in environment with the presence of electrolyte. The reactants flow into the device and the products flow out while the electrolyte remains in the device. Fuel cells can operate continuously as long as the necessary flows are sustained. They have high energy densities and are rechargeable but massive and voluminous. Fuel cells are almost exclusively used on human space flight missions.

8.3 Power Requirements

The requirements for power are shown in Table 8.1. These requirements stem from the main requirement to power the all of the subsystems.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>POWER-01</td>
<td>Total Power</td>
<td>Maximum total power must be ≤ 132 W</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWER-02</td>
<td>Rating</td>
<td>All components must be rated for solar maximum, summer solstice and performance in space</td>
<td>MISS-02</td>
</tr>
<tr>
<td>POWER-03</td>
<td>Commands</td>
<td>Power system must be compatible with the communications system and provide telemetry</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWER-04</td>
<td>Voltage Protection</td>
<td>Provide for overvoltage protection of 30V</td>
<td>POWHL-01</td>
</tr>
<tr>
<td>POWER-05</td>
<td>Total Power</td>
<td>Support power requirement of 132 W until end of life</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWER-06</td>
<td>Reuse of Batteries</td>
<td>Provide for discharging and charging of batteries for reconditioning</td>
<td>POWHL-02</td>
</tr>
<tr>
<td>POWER-07</td>
<td>Energy Dissipation</td>
<td>Dissipate excess energy through shunt elements</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWER-08</td>
<td>Sensor Maintenance</td>
<td>Maintain sensors to measure system health including charge, voltage, and temperature</td>
<td>MISS-01</td>
</tr>
<tr>
<td>POWER-09</td>
<td>Error Overrides</td>
<td>No single error aborts Mission</td>
<td>MISS-01</td>
</tr>
</tbody>
</table>

Table 8.1: Power Subsystem Requirements

POWER-01: The minimum power requirement to ensure mission success is 90W and the peak power requirement is 132W. Therefore, the power subsystem must provide at least 132W of power at the spacecraft end of life (EOL). Our system provides 232 W of power, satisfying both the minimum power and peak power requirement and allows for less complicated tasking orders due to power availability of excess power.

POWER-02: The power subsystem services all of the other subsystems on the spacecraft including the payload. In light of this, the power system must protect all of the components from damage due to improper electrical supply. A power conditioning and distribution unit has been selected to satisfy this requirement by providing over-voltage and over-current protection.

POWER-03: In the case of emergency, a secondary battery is necessary to provide power to critical systems. The battery should be able to charge, recharge and be reconditioned for several
thousand discharge cycles. The power system utilizes a Lithium Ion (Li-Ion) battery to satisfy this requirement.

POWER-04: There are times during the mission where the power generation system will be producing more power than the system requires. At these times it is necessary to find a way to expel the energy so that it does not damage the system. The selection of a direct energy transfer architecture enables excess power dissipation via electrical shunts.

POWER-05: In order to provide maximum power and protect power subsystems, sensors will be put into place to measure vital telemetry. Using this data, the spacecraft will be able to check on the status of power subsystems and relay any problem that needs to be fixed.

POWER-06: Redundancy and the power architecture allow the mission to have fewer mission ending failure modes. The wiring of the solar panel arrays and thermoelectric generators will be such that a failure to any single section will not result in a failure overall. Doing this will mitigate risk and increase the chances for a successful mission.

POWER-07: The durability of our components needs to be great enough to survive the lifetime of the mission in the temperature and radiation of space. Due to the environment, components will be space rated and for their expected lifetime performance.

8.4 Electrical Power System Architecture

In order to satisfy the requirements outlined in Section 8.3, this mission will employ the power architecture shown in Figure 8.1. In this report, we refer to the power generation and storage components as primary, and the conditioning and distribution components as secondary. Power generation and storage components include solar cells, thermoelectric generator (TEG) cells, and secondary batteries. Conditioning and distribution components include the battery charge regulator (BCR), power conditioning module (PCM), power distribution module (PDM), shunts, and the telemetry and telecommand system. All of the secondary components consist of electronics integrated into one single unit: the power conditioning and distribution unit (PCDU). Trade studies supporting the choices of components and a thorough analysis of the power calculation methods will be presented in the rest of the report.

8.4.1 Power Generation and Storage

Solar and TEG cells are used to power the spacecraft producing a total of 232 Watts (W). The power generation system produces an extra 75% of required power; however the extra 70% is produced by the TEGs. The solar cells themselves meet the total power requirement with an additional 5% of required power.

The existence of the TEGs provides flexibility in the power management system which yields a very large redundancy and mitigates any mission failures caused by insufficient power generation. TEG cells have extremely high energy to mass and cost ratios and therefore their addition is almost negligible when compared to the total mission cost and spacecraft mass. The TEG cost and TEG mass are less than 0.5% of the mission cost and spacecraft mass respectively. Success-
ful implementation of the TEGs will serve as a proof-of-concept of the technology and raise the technology readiness level (TRL), which will yield an extremely efficient power source for similar future missions. The current layout of our TEG cell array is patentable and could generate additional sources of funding.

The array panels will be divided into several solar cell and TEG cell sections. The power distribution will be initially configured so 75% of peak power is produced from the solar cells and 25% is produced from the TEGs. In the case of unexpected decrease in performance, different combinations of solar cell or TEG cell sections will be turned on in order to keep the total generated power constant.

For additional mitigation of risk of long-term failure from short term energy loss, the spacecraft will implement a secondary battery that is compatible with our power distribution system. A diagram of our system is shown in Figure 8.1.

![System Diagram](image)

**Figure 8.1: System Diagram**

### 8.4.2 Power Distribution

This section outlines the process by which the power system components were selected. The selection process as described in the proceeding section involves in depth research, trade studies, and eventually the selection of a component that satisfies the requirements and needs of the system. This section will analyze the selections of power management and distribution system, the voltage bus, secondary components (power management components), and primary
components (power generation components).

8.4.2.1 Power Management and Distribution

Generated power from various on-board energy collection devices must be properly conditioned and transmitted throughout the satellite. The power management and distribution (PMAD) system is the heart and soul of the power subsystem architecture. Our selection of the PMAD layout is highly mission dependent, with requirements heavily based on the overall mission constraints and requirements. The general system selection, bus selection, voltage, and overall power capabilities of our architecture were made based on the aforementioned process.

8.4.2.2 System Selection

Due to our small spacecraft size, limited maneuverability, simple functionality, and low overall power requirements (>200 W), we have chosen the path of a single centralized PMAD system. Because of the satellite's limited size and volume, a small system permits power regulation hardware and electronics to fit into a single box. Larger, higher power systems involve transmission of signals and power in the multi-kilowatt range, over distances of one meter or more through satellite wiring. Practically speaking, a low line voltage is also the best decision, in the 28 V range. Additionally, most satellite electronics run on the 28 V level and would require step-down converters if a larger voltage was chosen.

The energy generated on board the satellite from the photovoltaic and thermovoltaic cells enters into the centralized system as DC. The system was chosen for this reason to operate on DC power. The electronic components of our particular spacecraft all accept DC input as well. While AC power has the possibility to provide higher system efficiency at a lower mass, power is often lost if converted between AC and DC. Similar to the power transmission systems on Earth, AC is only necessary for power transmission over long distances (such as in large satellites).

8.4.2.3 Direct Energy Transfer vs. Peak Power Tracking

Further breaking down the power system architecture, it was determined that our satellite will use a direct energy transfer (DET) system, rather than peak power tracking (PPT). DET systems are less complex with fewer parts; offering a higher system efficiency at a lower cost. Linear shunts (transistors) are connected in parallel to the solar/TEG arrays to regulate or 'dissipate' excess, unneeded power. PPT, or 'non-dissipative' systems, extracts power at specific points of the solar array when the load increases. PPT requires extra power converters and decreases efficiency. However, it has been shown to obtain higher solar array power at spacecraft EOL. It must be noted that PPT systems leave excess power on the solar cells when energy is unneeded, raising the array temperature. Thus, to avoid the adverse effects of overheating in an already sun-baked L1 mission, in addition to the cost and simplicity of DET systems, we have chosen direct energy transfer as the main architecture. DET is further subdivided into two bus classes: fully regulated and unregulated busses.
8.4.2.4 DET Bus Selection

Several factors were considered when selecting a bus, mainly the EOL power requirement, voltage level, heritage equipment, and the sun-steady non-eclipse orbit of the spacecraft. The main choices for a bus system were between fully regulated and unregulated buses. A comparison of these bus systems is provided in Table 8.2.

Regulated bus systems are quite complex and involve extra equipment such as switching regulators, although they are lower in mass than the other bus systems. An unregulated bus and regulated bus both have the standard battery charge regulator to monitor the rate and level of battery charge during sun exposure. Inherently, unregulated and fully regulated busses are similar, except that unregulated busses have no battery discharge converter in the main power regulator unit (PRU). Therefore, unregulated busses only allow discharge from the battery; leaving the battery disconnected from the bus during sunlight when shunts are regulating the bus voltage.

Since our satellite does not plan to be in eclipse, it will require the use of the battery for events where sudden peak power is required, while the Sun is still in view (for sudden communication bursts, for instance, or station keeping burns). Without a discharge converter in the PRU, the bus voltage would remain the same as, and fluctuate with the battery voltage while in use. Upon further review, some aspects of our mission may warrant the need for a battery discharge converter.

![Table 8.2: Architecture Trade Study](image)

In comparison, a fully regulated bus is most often found in spacecraft requiring a load power above 3 kilo-watts (kW), where an unregulated bus is found in low power (>1 kW) spacecraft, similar to our own. However, a fully regulated bus allows for several enticing advantages. A higher capacity battery cell can be used, since the discharge rate is regulated. In addition, regulated bus harnesses are usually lower in mass, work at a more consistent voltage level than unregulated, and often have lower mass load power converter units for other instrumentation. The unregulated bus often experiences problems with voltage level fluctuation.

Overall, a fully regulated bus provides direct and simple system specifications and interfaces, operational flexibility, completely autonomous overload control, and is easily user-controlled. In no way does it limit the size, load, or type of battery we have selected. Thus, we have chosen a unique power conditioning unit (PCU), incredibly small in size for spacecraft such as ours, but which also contains a battery charge and discharge regulator; cementing our selection of a fully regulated bus.
Overall, a fully regulated bus provides direct and simple system specifications and interfaces, operational flexibility, completely autonomous overload control, and is easily user-controlled. In no way does it limit the size, load, or type of battery we have selected. Thus, we have chosen a unique power conditioning unit (PCU), incredibly small in size for spacecraft such as ours, but which also contains a battery charge and discharge regulator; cementing our selection of a fully regulated bus.

8.4.3 System Operational Modes

Within a fully regulated bus, the PCDU has the task of supplying the correct voltages to all loads through the use of MOSFET switches and fuses. Within the distribution circuitry resides the bus voltage controller. The BVC interfaces with sensors which monitor voltage outputs to all instruments and reference them to the reference voltage. The BVC runs the error signal amplifier with this data, and outputs this signal to the mode controller. The mode controller regulates the entire bus voltage with the specified limits of the signal amplifier.

8.4.3.1 Mode Controller

When the mode controller receives the signal inputs, it updates the error processing system to most correctly respond to the error signal. There exist sub-controllers within the mode controller; the mode controller activates certain sub-controllers as needed by switching a magnetic latching relay. A small shunt voltage detector, essentially a simple voltage gate, is set to 3 detected limits by power system engineers. The limits include 0 to 1 V, 1 to 3 V, and 3+ V.

- **Shunting Mode**
  During the bulk of its lifetime, the spacecraft will remain in shunting mode. This means that for any extra solar/thermoelectric power produced, the mode controller diverts this unneeded power to the shunt units for dissipation. If this does not occur, the voltage of the voltage bus will rise above the nominal 28 volts we have set. Due to the sensitive electrochemistry of the Li-Ion battery, it is carefully charged with an orchestrated and alternating trickle/full charge rate during this time. If the shunt voltage detector is over 3 V, shunting mode is activated.

- **Battery Charge Cut-Back Mode**
  In order to control the battery temperature when maximum charge is approached, the rate of charge is reduced. Once a full charge is attained, trickle charge will continue for the Li-Ion battery. In the event that primary power production surpasses the load requirement, but not enough to sufficiently charge the battery (as in Shunting mode), battery charge cut-back mode is activated. If the shunt voltage detector reads between 1 and 3 V, battery charge cut-back mode is activated.

- **Discharge Mode**
  If for any reason primary power production is sent off-line, or when the spacecraft has not yet been positioned in a sun-facing direction such as in the cruise-phase, the battery
will be the primary source of power. Discharge mode regulates the battery discharge; by increasing the boost ratio used to keep a steady voltage supplied to the bus. As the battery drains, the voltage naturally drops, and the discharge converter must properly correct this difference. This mode re-directs the power to the control and regulation portion of the PCDU. If the shunt voltage detector reads less than 1 V, discharge mode is activated.

- Power Regulation Unit Bypass Mode

In load control emergencies, such as faults with any load within the spacecraft, the corresponding fuse for the troublesome load must be tripped as soon as possible to keep the bus voltage constant. Our battery is connected directly to a bypass diode, also linked to the PRU bypass mode sub-controller. This device, along with the PCDU, detects the fault and directs battery power to the fuse system so that it may be tripped. This task requires a separate sub-controller mode because the power regulation electronics alone experience a high delay; the signal must travel through the internal control loops before error is detected. The PRU bypass mode also blocks any charge flow to the battery in an emergency.

Figure 8.2 shows an example of the operating modes in use around our nominal 28 volt bus value.

Figure 8.2: Bus voltage operating modes

8.4.4 Control Circuit

Our selected PCDU comes equipped with a digital control circuit, with several analog elements. Although a full analog control loop has shown better dynamic responses to correcting transient voltages in the power bus, it requires a high bandwidth to reduce the noise it creates.

Full digital control enables smaller shunt dissipaters to control the bus voltage more accurately. A digital system can handle more shunts, as well as more advanced switching algorithms to control power flow. A digital PCDU makes it flexible to change the system to our exact mission requirements. Most importantly, we are able to program a variety of charging scenarios;
changing the discharge/charge rate and control to specific events in the mission by uploading commands from Earth.

The same control is exercised over our combinational photovoltaic/thermovoltaic array (CPTA), where we can change power gathering configurations. We are able to reduce cost and mass because of the flexibility of having a computerized circuit do the integration for us.

### 8.5 Array Design

This section discusses different solar array designs of mounted or deployable solar arrays, with the emphasis on the structural design and deployment mechanisms. The selection is made based on the area, mass, cost, and complexity. The purpose of this section is to provide enough information behind the selection of the exact solar array configuration and array deployment mechanisms. It will also discuss the layout of the array panel, the integration of it with the spacecraft, and its functionality in a system level.

#### 8.5.1 Array Area and Mass

This section outlines the methods and equations used to calculate the required solar cell and TEG array area and mass. The produced power of the satellite is dependent on time. An array is sized to meet power requirements at EOL. Because the array will experience performance degradation over its lifetime it is oversized for power requirements at beginning of life (BOL).

#### 8.5.1.1 Solar Cell Array Calculations

To estimate the solar array area we first need to determine how much power the solar array must provide, $P_{sa}$, during daylight to power the spacecraft.

$$\frac{P_e T_e + P_d T_d}{X_d T_d} \quad (8.1)$$

Where $P_e$ and $P_d$ are the power requirements during eclipse and daylight respectively, and $T_e$ and $T_d$ are the time durations of these periods. During the mission the spacecraft will not ever be in eclipse and therefore the above equation is simplified.

$$P_{sa} = \frac{P_d}{X_d} \quad (8.2)$$

The term is the path efficiency from the solar array to the individual loads and batteries. The value depends on the type of power regulation. As mentioned in Section 8.4.2.3 the system chosen is direct energy transfer and the efficiency is about $X_d = 0.85$ \textsuperscript{[102]}. Next we calculate
the ideal solar cell power output per unit area, \( P_0 \), which depends on the solar cell energy conversion efficiency, \( e \). The power input value is the solar illumination intensity, \( J \), at the different phases of the mission.

The term \( X_d \) is the path efficiency from the solar array to the individual loads and batteries. The value depends on the type of power regulation. As mentioned in Section 2.2 the system chosen is direct energy transfer and the efficiency is about \( X_d = 0.85 \) [102]. Next we calculate the ideal solar cell power output per unit area, \( P_0 \), which depends on the solar cell energy conversion efficiency, \( e \). The power input value is the solar illumination intensity, \( J \), at the different phases of the mission.

\[
P_0 = e \times J \tag{8.3}
\]

Next, we determine the realistic solar array power output per unit area. An assembled solar array is less efficient than a single solar cells due to design inefficiencies and temperature variations, referred to as inherent degradation, \( I_d \). Another source of loss is the Sun incidence angle, \( \theta \), between the vector normal to the array surface and the Sun rays.

Finally, thermal degradation is also taken into account. The power lost with severe temperature change, \( P_T \), is dependent on the surface temperature of the array. According to the Stefan-Boltzmann law, an object’s temperature in space which receives radiation, in our case from the light and heat of the sun, can be described with the equation:

\[
T = \left[ \frac{I_L}{\sigma} \times \frac{a}{\epsilon} \right]^{\frac{1}{4}} \tag{8.4}
\]

Where \( T \) is absolute surface temperature in Kelvin, \( I_{L_1} \) is the Solar Flux at \( L_1 \) in W/m\(^2\), \( \sigma \) is the Stefan-Boltzmann constant of 5.67 \( \times \) \( 10^{-8} \) W/m\(^2\)K\(^4\), \( a \) is the material absorptivity, and \( \epsilon \) is the surface emissivity. This equation also assumes that the spacecraft solar cell surface is radiating to 3\(^\circ\)K, the temperature of space. According to readings from previous \( L_1 \) missions, the incident solar energy, or solar flux, is governed by:

\[
I_{L_1} = \frac{I_{Earth}}{R_{L_1}^2} \tag{8.5}
\]

Where \( I_{Earth} = 1358 \) W/m\(^2\), the solar flux in Earth orbit and \( R_{L_1} = 0.99 \), distance from the sun in astronomical units (AU). \( I_{L_1} \) is found to be 1385.57 W/m\(^2\). For sizing calculations, the power is calculated using the solar flux of low-Earth orbit, \( I_{LEO} = 1367 \) W/m\(^2\), where our cells will begin powering the spacecraft as it makes its way to \( L_1 \). Spectrolab states that the absorptivity and emissivity of the UTJ solar cells is 0.92 and 0.85 respectively.

Applying this to Equation 8.13, it is found that the surface temperature of the solar cells is approximately = 401.92 K or 128.769\(^\circ\)C. Spectrolab specifies that the array output decreases by 0.335 mW/m\(^2\)-\(^\circ\)C above 75\(^\circ\)C. Therefore, 128.769 - 75 = 53.769\(^\circ\)C \( \times \) 0.335 mW/\(^\circ\)C = \( P_T = -0.0180 \) total Watts lost.
Therefore, the realistic solar array power output per unit area at BOL, \( P_{BOL} \) is given by the following equation:

\[
P_{BOL} = P_o \times I_d \times \cos(\theta) + P_T
\]  

(8.6)

Lastly, we must consider other factors besides temperature that degrade the solar array’s performance during its lifetime. Life degradation, \( L_d \), occurs because of the severe thermal and radiation environment. The value of \( L_d \) is a function of the solar cell degradation factor and the spacecraft’s lifetime.

\[
L_d = \left(1 - \frac{\text{degradation}}{\text{year}}\right)^{\text{spacecraft life}}
\]  

(8.7)

We can now calculate the EOL solar array output per unit area, \( P_{EOL} \), the actual solar array area, \( A_{sa} \), and mass, \( M_{sa} \). The mass is a is a function of the solar cell mass per unit area, \( m \); Spectrolab produces finished panel structures which are 2.06 kg/m\(^2\). With the preceding variables and equations in hand, a MATLAB code is run to process Equations 8.8 through 8.10.

\[
P_{EOL} = P_{EOL} \times L_d
\]  

(8.8)

\[
A_{sa} = \frac{P_{sa}}{P_{EOL}}
\]  

(8.9)

\[
M_{sa} = A_{sa} \times m
\]  

(8.10)

This process is repeated to size the entire array based on maximum area, minimum mass, and desired solar array power production. The MATLAB program used to size the CPTA array is found in Appendix H.

The solar cell area is 0.581 m\(^2\), which is also the maximum area possible for solar cell allocation on the spacecraft front face. This area produces approximately 100% (104% exact) of the spacecraft required power at EOL. The solar array alone yields 137.74 W of power with a mass of 1.197 kg.

### 8.5.1.2 TEG Cell Array Calculations

In comparison to the solar array sizing method, the MATLAB algorithm used to determine the number of TEG modules in the CPTA is quite similar. It is assumed that a nearly constant thermal gradient will be maintained for maximum power production, discussed in Section 8.6.

The implementation goal for the mission is to have 25% of initial on-board required power produced by the TEG modules for the first 3 years of the mission. If this successful step is achieved, then throughout the remaining mission lifetime, the TEG power dependence will be increased by 15% every year; From 25%, to 40%, then 55%, and finally to 70%. Ultimately, a 70% power generation goal would establish this new technology as feasible as a substantial power source in future missions.
In order to produce 70% of required power, an array of five TEG modules was selected. 70% of 132 W yields approximately 92.4 W. Five TEG modules, each theoretically producing 19 W of power, will generate 95 W of power for the spacecraft. Each TEG module is 56.25 cm$^2$, amounting to a total TEG area of 281.25 cm$^2$ with a total mass of 0.575 kg.

8.5.1.3 Heat conduction within TEG modules

Heat Loss and gain through the walls of a TEG can be modeled by the equation:

$$\frac{Q}{t} = A \times K \left[ \frac{T_{hot} - T_{cold}}{d} \right]$$  \hspace{1cm} (8.11)

Where $\frac{Q}{t}$ is the heat lost or gained in Watts, also known as the heat conduction over time. $d$ is the thickness of insulation in meters, $K$ is the thermal conductivity of the insulation material in W/m°C, $A$ is the outside surface area of the container in meters squared, $T_{hot} - T_{cold}$ is the difference between the outside hot temperature in °C, and the rear side temperature in °C.

In our TEG application, the surface area is 281.25 cm$^2$, consisting of 5 TEG modules, with a thickness of 0.0508 cm and a thermal conductivity of 2.4 W/m°C. The side is 230°C and the side is held at 30°C. Finally, the conduction heat loss rate across the TEG series is equivalent to 2657.48 Watts.

Therefore, it has also been determined that the heat pipes must remove 2657.48 W for the TEG module to work effectively.

8.5.2 Types of Array Configuration

There are different types of solar array designs which can be differentiated from each other by the placement of the solar cells. The two basic configurations are body mounted array (BMA) and deployable rigid array (RDA). Further on an RDA can be divided into Flexible Roll-out Array (FRA) and flexible fold-out array (FDA) based on the stiffness and rigidity of the substrate. Each configuration has several advantages and disadvantages. A summary of these advantages and disadvantages are found in Table 8.3.

<table>
<thead>
<tr>
<th>Deployment</th>
<th>Heritage</th>
<th>Stowage Volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>BMA</td>
<td>none</td>
<td>high</td>
</tr>
<tr>
<td>DRA</td>
<td>low</td>
<td>high</td>
</tr>
<tr>
<td>FRA</td>
<td>high</td>
<td>low</td>
</tr>
<tr>
<td>FFA</td>
<td>medium</td>
<td>low</td>
</tr>
</tbody>
</table>

Table 8.3: Summary of design array characteristics

With a BMA the absence of deployment mechanisms keeps attitude control relatively simple since the moments of inertia are kept constant and keeps the overall system’s complexity and cost down. In addition the space taken in the launch vehicle is reduced. However, the power
generation is limited because only part of the spacecraft can be constantly sunlit. In our mission since our in situ instruments have continuous sun pointing requirements a BMA can be applicable if the spacecraft's area which will be pointing at the sun is larger than the required solar cell area. If the area is sufficient then this choice of array design is the most cost and mass efficient.

A DRA is the most commonly used design because the required array area usually exceeds the available sun-lit spacecraft area. If combined with the correct turning mechanisms, the RDA has very low pointing requirements. The most common design consists of two identical wings; one on either side of the spacecraft which are attached symmetrically with a yoke in order to minimize the disturbances on the center of gravity during deployment. This type of design has an extensive heritage and development and production costs are low. However, the folded position requires a lot of space which limit the spacecraft's maximum size in the launch fairing. In addition, deployment of the array panels can greatly influence the attitude determination control system (ADCS) and the ADCS needs to be designed accordingly.

As mentioned previously, the DRA can be subdivided into a roll-out and fold-out array design. These types have been developed in an effort to decrease stowage space while increasing the array area. In both cases, rigid panels are interconnected with a flexible foil substrate which is called a blanket. The blanket can offer thermal and radiation protection and can also increase the energy generation area if covered with flexible solar cells. The differentiation between a roll-out and a fold-out comes directly from the definition of the naming. One is rolled for packing and rolled-out during deployment and the other is folded for packing and folded-out during deployment. These types have small heritage and are related with higher development costs. Moreover, the deployment of the non-rigid blanket involves complicated mechanisms, coupled with the fact that non-rigid materials exert higher radiation degradation and have a smaller lifetime.

Power calculations, shown in Appendix [H], are used to determine the array area necessary to produce 132 W for the lifetime duration of the mission, $0.558 \text{ m}^2$. As indicated by structures, this area is available at the base of the spacecraft. The spacecraft's base will constantly point to the sun during the cruise, transfer, and on-orbit phase of the mission. With our desired area constantly sun lit, we have decided to use a BMA. A BMA is the most mass and cost efficient choice since it requires the least number of components and structural support. The absence of any deployment mechanisms keeps the overall system complexity low.

### 8.5.3 Array Layout

A solar cell generates an electrical voltage with contributions from photovoltaic effects, while TEG cells utilize the thermoelectric effect. In order to harness the maximum amount of incident solar energy, both photovoltaic and thermovoltaic technology must be integrated. We have designed a unique array. The array will incorporate the use of both solar cells and TEGs and is therefore called a combinational photovoltaic and thermovoltaic array (CPTA). The CPTA panel will be in the form of a sandwich plate structure integrated with the wall of the spacecraft structure. The spacecraft will be made out of aluminum honeycomb core which has a high stiffness-to-weight ratio per unit area. The array panel will offer enhanced radiation protection to the spacecraft as it will absorb most of the sun's radiation and therefore a BMA provides
another advantage to the spacecraft’s structure. The top of the panel will be covered by the selected Spectrolab Ultra Triple Junction Gallium Arsenide solar cells. Mounted in a linear row alongside the solar cells at a specified location are five Hi-Z Technologies Inc. HZ-20 TEG cells. Under each TEG resides a heat sink integrated with heat pipes in order to maintain a low temperature at the cold side. This is done to preserve the necessary thermal gradient between the TEG’s top and bottom faces. The TEG’s hot side will be kept hot by the heat absorbed from solar radiation. Figure 8.3 shows a clear diagram of the CPTA panel.

A solar cell generates an electrical voltage with contributions from photovoltaic effects, while TEG cells uses the thermoelectric effect. In order to harness the maximum amount of incident solar energy, both photovoltaic and thermovoltaic technology must be integrated. We have designed a unique array, called the combinational photovoltaic/thermovoltaic array (CPTA), which linearly aligns Hi-Z HZ-20 TEG modules alongside state of the art Spectrolab Ultra Triple Junction GaAr solar cells.

For the achievement of a substantial thermoelectric voltage contribution to the power system, the sun-face TEG module is naturally heated to an elevated temperature consistent with, but not detrimental to its efficient operation. The absorptivity and emissivity characteristics of a selected molybdenum TEG coating transfer this heat through the device, creating the thermal gradient which produces power.

This special combinational array can be mounted to receive concentrated sunlight and thermal radiation on its front side. The rear face of the TEG is mounted to the spacecraft structure where a continuous and constant thermal gradient can be ensured with the placement of heat pipes to remove heat from the cool side of the TEG modules.

The overall ratio of power output to mass of the array has been researched, and an improvement by as much as 50 percent can be seen over a conventional GaAr solar cell-only array. Accelerated experiments with TEG cells have been performed by NASA JPL, with the addition of a protective coating and have demonstrated the equivalent of 11 to 13 years of operation without any apparent degradation.

8.5.3.1 Mounting TEG Modules in CPTA

In modern applications, only three viable mounting methods for thermoelectric modules exist: Adhesive bonding, compression assembly, and solder mounting.

- Mounting with Adhesive Bonding
  
  For terrestrial applications, thermally conductive adhesive (such as epoxy) is used to adhere TEG cells to either heat sink or cold plate. Only epoxies with the highest thermal conductivity can be used to ensure an optimum thermal system, and often have severe outgassing problems in extreme vacuum environments. However, adhesives are commonly used in solar cell mounting for space applications.

- Mounting with the Compression Method
  
  Compression assembly is the most common method; However, it requires precise placement and careful assembly. Essentially, a TEC is compressed between a cold plate and a
heat sink and held in such a compression with mechanical fasteners. If thermal gradient
control is needed, various block extensions with different material properties can be used
to alter the distance between either the cold plate or hot plate.

• Mounting with Solder

Solder mounting has the potential to offer the greatest heat transfer capacity, but also
proves to be the most complex assembly method. Metallization, or metal plating on the
hot and/or cold faces of the TEG, allows soldering as a means of mounting. Solder is
best to use for minimal outgassing, and when the TEC is smaller than 23 mm on a side.
Also, solder mounting is best used when a high-strength junction and the highest thermal
conductivity is required. The phenomenon known as ‘tin whiskers’ has been observed in
previous space missions involving solder. This growth of microscopic solder crystals un-
der vacuum may interfere in electronic components, but is unlikely to be a risk in simple
structural bonding.

A combination of these methods can be used, such as solder mounting the TEG bottom and
epoxy mounting the TEG top. These methods have been examined, and it has been determined
that soldering is the best choice for a high heat, space vacuum environment.

8.5.3.2 Combinational Photovoltaic/Thermovoltaic Array Structure

We are using a traditional method of construction, where solar cells are mounted upon a rigid
substrate consisting of aluminum honeycomb with various insulating sheets. Spectrolab Inc.,
our selected solar cell supplier, has agreed to produce a custom panel to fit our spacecraft; in-
cluded are all required layers, photovoltaic cells, and panel assembly.

The majority of the array is protected from external space environment by a coverglass sub-
stance known as fused silica microsheeting. A black thermal paint enhances the light absorp-
tion to thermal emissivity ratio, which actually enhances the light absorption and controls the
temperature. A silicon adhesive is applied to fix the cells onto the paneling. Table 8.4 shows the
thickness of each layer in the CPTA and Figure 8.3 offers a visual representation of the structure.

<table>
<thead>
<tr>
<th>CPTA Layers (starting at top layer)</th>
<th>Component Thickness (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Silica coverglass</td>
<td>0.25</td>
</tr>
<tr>
<td>Coverglass adhesive</td>
<td>0.05</td>
</tr>
<tr>
<td>Spectrolab UTJ photovoltaic cell</td>
<td>0.05</td>
</tr>
<tr>
<td>Solder adhesive</td>
<td>0.1</td>
</tr>
<tr>
<td>Kapton/ fiberglass insulation</td>
<td>0.05</td>
</tr>
<tr>
<td>Solder adhesive</td>
<td>0.005</td>
</tr>
<tr>
<td>Face Skin</td>
<td>0.1</td>
</tr>
<tr>
<td>Adhesive</td>
<td>0.1</td>
</tr>
<tr>
<td>Honeycomb</td>
<td>7 → 12</td>
</tr>
<tr>
<td>Adhesive</td>
<td>0.1</td>
</tr>
<tr>
<td>Face Skin</td>
<td>0.1</td>
</tr>
<tr>
<td>Backing thermal paint</td>
<td>0.05</td>
</tr>
</tbody>
</table>

Table 8.4: CPTA Structure
The top layer of the structure is composed of processed coverglass, which protects the solar cell against harmful solar and thermal radiation. Also, it is designed to limit the ultra-violet flux entering the adhesive layer and the rest of the cell. The coverglass allows suitable wavelength selection via optical coupling; creating the best distance of free-space between glass and adhesive.

The substrate underneath the solar cell is a Kapton coating with glass or carbon reinforcement. In addition to this, adhesives fix the solar cell to the CPTA and fix the cover glass upon the cell. Special adhesives are also used to maximize electrical conductivity while minimizing thermal conductivity.

Finally, the previously mentioned honeycomb panel physically supports the cells and transfers any heat away from the cells. This component is ultimately a selection of the structural sub-system; compatible with any thermal gradients the spacecraft may face.

### 8.6 Primary Components

Primary components include power generation and storage components. Power generation and storage components include solar cells, thermoelectric generator cells, and secondary batteries.

#### 8.6.1 Solar Cells

Solar cells are devices that convert sunlight into electricity using the photoelectric effect. They are the lifeline of a majority of long-term space missions as they are a feasible method of powering spacecraft for long periods of time due to their low cost and mass. Because the HAVEN mission has a lifetime requirement of 5 years, and because it will be in constant view of the sun, solar cells are the obvious primary power source selection for our satellite.

In recent years, solar cell technology has advanced at an unprecedented rate. Triple junction solar cells have shown the most promise and have achieved efficiencies as high as 42.8% during laboratory testing. It is, however, worth noting that all solar cells with efficiencies greater than 28.3% have not been tested adequately and have low TRLs.
Spectrolab, a subsidiary of The Boeing Company, has been contracted to provide solar cells for the HAVEN mission. The company has a solid track record and has placed more than 675 kW of high efficiency, multi-junction arrays, in orbit. They are also the only solar cell vendors that produce solar cells with an efficiency of greater than 28% on a large scale.

Three types of solar cells from Spectrolab were considered as the primary power source for the HAVEN mission, i.e. Spectrolab ’Next Triple Junction’ cells, ’Ultra Triple Junction’ cells, and single junction cells. A trade study between these three cells can be seen in Table 8.5 below. Flight heritage, cell mass, cell efficiency and TRL were the major design drivers considered while selecting the appropriate solar cell for the mission. As can be seen from Table 8.5, the Ultra Triple Junction cell has been selected for the HAVEN mission as it strikes a suitable balance between cell efficiency, cells mass and TRL. The Next Triple Junction cell is still in the testing stages. It is expect to have a TRL of 5 by mid-2009. Using a cell with low TRL and no flight heritage introduces unnecessary risk into the HAVEN mission. The single junction cell mentioned in the table below does have a lower mass than the Ultra Triple Junction cell. However, the high efficiency of the Ultra Triple Junction cell outweighs the potential mass savings from the single junction cell.

<table>
<thead>
<tr>
<th>Type</th>
<th>BOL Efficiency @ AM0</th>
<th>Mass w/o Substrate</th>
<th>TRL</th>
</tr>
</thead>
<tbody>
<tr>
<td>NeXt Triple Junction (GaInP&lt;sub&gt;2&lt;/sub&gt;)</td>
<td>0.299</td>
<td>84 mg/cm&lt;sup&gt;2&lt;/sup&gt; @ 140 µm</td>
<td>5</td>
</tr>
<tr>
<td>Ultra Triple Junction (GaInP&lt;sub&gt;2&lt;/sub&gt;)</td>
<td>0.283</td>
<td>84 mg/cm&lt;sup&gt;2&lt;/sup&gt; @ 140 µm</td>
<td>9</td>
</tr>
<tr>
<td>Single Junction (GaAs)</td>
<td>0.19</td>
<td>80 mg/cm&lt;sup&gt;2&lt;/sup&gt; @ 140 µm</td>
<td>9</td>
</tr>
</tbody>
</table>

Table 8.5: Solar cell trade study

The Ultra Triple Junction solar cell consists of a Germanium cell, Gallium Arsinide (GaAr) cell and Gallium Indium Phosphide cell mounted of a Germanium substrate. A cross section of the cell can be seen in Figure 9.4 below. These cells have a higher radiation tolerance than single junction cells due to their structure. Each cell has a 0.15mm thick Ceria doped coverslide to provide additional protection against radiation. According to the specifications provided by Spectrolab, these cells provide 350W/m<sup>2</sup> and have an area specific mass of 2.06kg/m<sup>2</sup> without an aluminum honeycomb support structure.

The cost of the UTJ cells, pre-mounted on a solar panel is $400 per Watt of power generated by the solar cell. Thus, for our power requirement of 132 W, the total cost of assembled solar arrays will be $67,000. Each cell will have an area of 32 cm<sup>2</sup> and will be sized to custom dimensions as determined by the structures subsystem.
Figure 8.4: Ultra Triple Junction cell cross section

8.6.2 TEG Selection

Discovered by Thomas Seebeck in 1822, the conversion of heat directly into electricity was observed in such a way that electric voltage is generated when two dissimilar yet electrically conductive materials are joined in a closed circuit. One material is called N-type and the other is P-type; engineered so that the N-type material has an abundance of electrons, while P-type with an electron deficiency. When the junctions are kept at different temperatures, electrons bridge this gap and current is produced.

The materials are connected in series with metallic strips and imbedded between electrically insulating plates. However, the plates are also thermally conductive. Thus, as heat moves from one 'hot' side to the opposite 'cold' side, charge carriers are carried with the heat.

When dozens or hundreds of N/P couples are linked together, a practical and useable voltage is produced. Since electrons are the main transport media for the energy production, thermoelectric devices require no moving parts, fluids, or maintenance. They are lightweight and produce no vibration.

8.6.2.1 Thermoelectrics in Space Applications

When dozens or hundreds of N/P couples are linked together, a practical, sizeable, and useable voltage is produced. Since electrons are the main transport media for the energy production, thermoelectric devices require no moving parts, fluids, or maintenance. They are lightweight,
produce no vibration, nor are sensitive to vibration.

Energy conversion in similar space systems are have existed for generations with radioisotope thermalelectric generator (RTG) technology on board dozens of US and Russian spacecraft. The only difference between RTG units and solar TEG units is the source of the heat energy. The RTG heat source comes from the nuclear radiation of a suitable radioisotope fuel, while heat in the solar thermoelectric generator comes from solar radiation.

8.6.2.2 Thermoelectrics in Space Applications

The power systems for a near-sun spacecraft operating at temperatures higher than that of GEO or LEO cannot effectively use photovoltaic cells because of severe temperature degradation in performance. In such missions, NASA has proposed and researched the prospect of collecting solar heat and directing it onto an array of thermoelectric generators able to not only withstand the heat, but produce substantial power. This technology is well developed and flight-proven for a few hundred Watts. Although expensive, solar heated TEGs may be considered in regions of intense radiation and/or missions with nuclear burst threats.

Energy conversion in similar space systems have existed for decades with RTG technology on-board dozens of US and Russian spacecraft. The only difference between RTG units and solar TEG cells is the source of the heat energy. The RTG heat source comes from the nuclear radiation of a suitable radioisotope fuel, while heat in the solar TEG comes from solar radiation.

8.6.2.3 Model and Manufacturing

While the research and design of TEG technology has been active for more than 50 years, consumer use and industrial application of such devices has only recently developed. Several large manufacturers of thermodynamic technologies such as Lockheed Martin, have cut back their programs in the 1980’s and 90’s; producing only custom cells for NASA and the Department of Defense (DOD). However, according to the International Thermoelectric Society, the largest and most sophisticated company in the United States producing off-the-shelf thermoelectric technology is Hi-Z Technologies of San Diego California. The output wattage and vacuum operation of Hi-Z’s TEG modules are unparalleled throughout the industry, and are also used by NASA and the DOD.

Therefore, we have selected Hi-Z technologies’ highest mass-to-power TEG module, the HZ-20, to operate as the on-board thermovoltaic power generator.

The HZ-20 is designed to efficiently operate in temperature gradients of -30°C to 300°C, while the satellite may see extreme temperature differences from -50°C to 260°C. It is proven that the HZ-20 module works especially well in this temperature range. The cold side of the module can be regulated and controlled with heat sinks as discussed in the following section.

Hi-Z module semiconductors are manufactured with Bismuth Telluride alloys, sandwiched between bonded metal conductors; enabling the modules to operate continuously for up to 10 years or longer. Other performance characteristics of the HZ-20 are found in Table 8.6.
HZ-20 TEG Module

<table>
<thead>
<tr>
<th>Watt/Kg</th>
<th>165.22 W/kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Watt /m²</td>
<td>3.377 kW/m²</td>
</tr>
<tr>
<td>Price/Watt</td>
<td>$11.00/W</td>
</tr>
<tr>
<td>m³/Watt</td>
<td>1.5039 x 10⁻⁶ m³/W</td>
</tr>
</tbody>
</table>

Table 8.6: HZ-20 Characteristics

The power of the modules increases approximately with the square of the temperature difference, up to an operating temperature of 225°C. From 225°C to the maximum operating temperature of 250°C, the power will not continue to follow the square and does not increase as rapidly at higher temperature.

### 8.6.2.4 Increasing Efficiency

Primary cell components are heavily affected by heat flow and their thermal properties. While this may fall under the responsibility of the thermal control sub-system, the proper functionality of solar cells and TEG cells almost completely relies on the understanding of their thermal properties and environment.

The magnitude of the thermoelectric voltage contribution can be increased in several ways. Reducing the coefficient of thermal conductivity of the TEG cell material is one important step. Additionally, increasing the light intensity and thus the heat input to the front side of the solar cell, as well as cooling the cold side of the TEG cell all increase power generation potential. In order to increase efficiency, it is crucial to understand the short and long term effects of solar flux in our integrated array.

To determine the amount of power absorbed by the TEG, the energy emitted by the sun must be taken into account. According to readings from previous L₁ missions, the incident solar energy, or solar flux, is governed by:

\[ I_{L_1} = \frac{I_{Earth}}{R_{L_1}^2} \]  

Where \( I_{Earth} = 1358 \text{ W/m}^2 \), the solar flux in Earth orbit and \( R_{L_1} = 0.99 \), distance from the Sun in astronomical units (AU). \( I_{L_1} \) is found to be 1385.57 W/m².

The Stefan-Boltzmann law, discussed in Section 8.6.1, best describes an object’s temperature in space which receives heat, in our case from the sun, and also radiates heat away. Revisiting Equation 8.13 we again assume that the spacecraft surface is radiating to 3°C.

\[ T = \left[ \frac{I_L}{\sigma} \times \frac{\alpha}{\epsilon} \right]^{\frac{1}{4}} \]  

Reiterating, the temperature, T is equivalent to the absolute surface temperature in Kelvin. \( I_{L_1} \) is the Solar Flux at \( L_1 \) in W/m², \( \sigma \) is the Stefan-Boltzmann constant of \( 5.67 \times 10^{-8} \text{ W/m}^2 \text{K}^4 \),
\(\alpha\) is the material absorptivity, and \(\epsilon\) is the surface emissivity. In order to maximize the thermal gradient for the TEG, the minimum cold side temperature must be 30°C, and the ideal hot side surface temperature must be 230°C. Therefore, in order for \(T = 230^\circ\text{C}\), we re-arrange Equation 8.13 and calculate that the absorptivity to emissivity ratio \(\frac{\alpha}{\epsilon}\) must be:

\[
\frac{\alpha}{\epsilon} = \frac{[230 \times 273.1]^4}{\frac{1385.57}{5.67 \times 10^{-8}}} = 2.263
\]

According to a NASA study on solar absorptance and thermal emittance of common spacecraft thermal-control coatings, it is determined that two candidates exist for the proper coating of the TEG modules: Vapor deposited coating of molybdenum, with an \(\frac{\alpha}{\epsilon}\) ratio of 2.666, and an electroless Nickel application, with an \(\frac{\alpha}{\epsilon}\) ratio of 2.60.

A Molybdenum coating, with slightly higher \(\frac{\alpha}{\epsilon}\) ratio, will effectively raise the exterior TEG temperature 4° higher than electroless Nickel. This provides an effective temperature of 232.1°C, just over the TEG specified ideal hot side temperature (not the operational maximum, however). Heat pipes underneath the TEG, operated by the thermal subsystem, will ensure the cold-side temperature will remain 30°C, thereby obtaining a near-perfect thermal gradient and consistent 19 Watt output from each TEG module.

### 8.6.3 Secondary Power

#### 8.6.3.1 Electrochemical Cell vs. Fuel Cell

We chose to implement an electrochemical-cell battery system, rather than a new fuel cell technology system for a secondary power system. Lockheed Martin Space Systems recently performed an in-depth power system trade study for very small satellites with low power requirements, having peaks of less than 1 kW. The study investigated the benefits and disadvantages of a mission using a fuel cell power system versus a solar array-plus battery power system.

The main metric comparison was power system mass versus peak power for a range of mission lifetimes. It was found that solar array/battery electrical power subsystems have a significant mass advantage over fuel cell and regenerative fuel cell systems for the range of application for periods of more than one week. However, for short duration missions of less than a week, fuel cell systems have a mass advantage over solar-battery systems. For longer duration missions, fuel cell systems with liquid storage tanks have significant mass savings when integrated with an on-board ADCS system that uses liquid hydrogen and oxygen propellant. It has been shown that fuel cells have the potential to store greater amounts of energy versus conventional batteries.

On the other hand, fuel cells are more expensive in their current state of development and less efficient than batteries. Our spacecraft will consume and rely on secondary power far less than conventional Earth-orbiting satellites which experience frequent eclipses. Furthermore, it must be realized that the power distribution system only relies on secondary power during the launch phase, in addition to acting as an emergency backup for unexpected moments of peak power loading. If the mission relied heavily on a secondary power system, fuel cells would
perhaps be a stronger option. However, for these reasons, mass and cost are reduced while risk is mitigated with the selection of electrochemical-cells within the secondary battery.

8.6.3.2 Primary Vs. Secondary Batteries

In secondary energy storage, auxiliary power sources are meant for two purposes. Most commonly, they are used to power spacecraft loads during eclipse, but also used to meet a higher demand for power when the primary power generation is exceeded. In our spacecraft application, we will be using this auxiliary power source almost exclusively for meeting the need of excess power.

In primary batteries, once an electrochemical reaction takes place, no power remains within the unit. These batteries are only used on short duration missions, and are almost always used as the primary source of power. Primary batteries contain specific energy with an order of magnitude higher than that of secondary batteries, but are massive and sometimes ejected after use.

Secondary power devices are also electrochemical in nature; However, the reaction is reversible. These battery units can be recharged after each discharge, as long as they are powered by an external primary source of power. They are best suited in missions with durations greater than several days. Therefore, our mission is utilizing a secondary power device.

8.6.3.2.1 Re-chargeable Battery

In the re-chargeable battery realm, respective chemical technologies are advantageous for different applications; A battery must be selected only on the specific parameters of the satellite mission pertaining especially to mass and cost requirements. Therefore, in accordance with our $L_1$ mission architecture, have selected a 480 Watt-hour (W-hr) Lithium-Ion MicroSat module from the Saft Battery Corporation.

8.6.3.2.2 Chemical Cell Trade Study

Battery design and selection always begins with the selection of the proper electrochemical cell, and number of cells based on mission requirements. There are dozens of available capable chemical-electric cell technologies for use in recharageable batteries, and more than 10 were carefully considered for use aboard our spacecraft. Nickel-Cadmium (NiCd) battery, along with SuperNiCd, Nickel Hydrogen (NiH2), Lithium Ion (Li-Ion), Lithium Polymer and Nickel Metal Hydride have emerged as front-runners in secondary space power. The workhorse battery of the spacecraft industry of the last 60 years has been the NiCd battery, although many new technologies have emerged over the last 25 years. Also, the NiH2’s depth of discharge is much deeper for the same life cycle as the NiCd, which translates to a lower mass battery. Moreover, Lithium based battery models are currently emerging as the front-runner in the space industry, with
two to five times the specific energy of NiH2 models. Tables 8.7 and 8.8 weigh the merits and disadvantages of several battery technologies.

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>NiCd</td>
<td>40-50</td>
<td>50-100</td>
<td>150-200</td>
<td>300-500</td>
</tr>
<tr>
<td>NiH2</td>
<td>45-65</td>
<td>35-50</td>
<td>150-200</td>
<td>200-300</td>
</tr>
<tr>
<td>NiMH</td>
<td>50-70</td>
<td>140-180</td>
<td>150-200</td>
<td>300-500</td>
</tr>
<tr>
<td>Li-Ion</td>
<td>90-150</td>
<td>150-250</td>
<td>200-220</td>
<td>400-500</td>
</tr>
<tr>
<td>Lithium-Polymer</td>
<td>100-200</td>
<td>150-300</td>
<td>&gt;200</td>
<td>&gt;400</td>
</tr>
</tbody>
</table>

Table 8.7: Power and energy densities of electrochemical technologies

<table>
<thead>
<tr>
<th>Electro-chemistry</th>
<th># of lifecycle discharges possible at 25°C</th>
<th>Calendar life in years</th>
<th>Self discharge, % per month at 25°C</th>
<th>Cost in $ per kilowatt</th>
</tr>
</thead>
<tbody>
<tr>
<td>NiCd</td>
<td>1000-2000</td>
<td>2009-10-15</td>
<td>20-30</td>
<td>1500</td>
</tr>
<tr>
<td>NiH2</td>
<td>2000-4000</td>
<td>2009-10-15</td>
<td>20-30</td>
<td>1500</td>
</tr>
<tr>
<td>Li-Ion</td>
<td>500-1000</td>
<td>2009-08-10</td>
<td>2009-05-10</td>
<td>3000</td>
</tr>
<tr>
<td>Lithium Polymer</td>
<td>500-1000</td>
<td>In Development</td>
<td>2009-01-02</td>
<td>&gt;3000</td>
</tr>
</tbody>
</table>

Table 8.8: Life and cost of electrochemical technologies

- **NiCd**
  The positive aspects of the NiCd battery have long been forgotten, as it is shown to provide relatively low specific energy compared to today’s spacecraft. Also, it has a shorter life cycle and very temperature sensitive compared to the NiH2.

- **SuperNiCd**
  The DOD created a new NiCd technology called SuperNiCd, which competes with NiH2 at power levels between 1 and 2 kW, and reduces battery mass. For lower power applications such as our own, the SuperNiCd proves to excel in low current applications, and in the high-current extremes. However, large problems still remain with heat build-up within the battery, which often involve expensive and complex solutions.

- **Nickel Hydrogen**
  The NiH2 battery is the most widely used secondary battery system; Essentially, it combines the best technology features of NiCd and fuel cells. The NiH2 are rugged and designed for heavy abuse. On the negative side, NiH2 battery system was not selected for our mission because of a combination of factors. First, it has a low energy density, has been known to rupture due to pressure handling, has a high self-discharge rate, and has a high loss of capacity on storage. Also, due to the required pressure vessel housing of a NiH2 battery assembly, other technologies which reduce mass and volume have developed.
• Nickel Metal Hydride

Nickel Metal Hydride offers improvement in specific energy over NiCd and carries promising properties, such as having a negligible memory effect. The technology holds several disadvantages as well, such as frequent over-charging and the inability to deliver high-peak power; Crucial in powering intermittent communication loads and emergency systems. It cannot handle high temperatures, which our battery may experience at $L_1$, but is less sensitive to lower regimes. Unlike other batteries, it produces very high heat during its charging cycle. Compared to other available technologies, it is currently more expensive and is not compatible with our budget.

• Lithium-Ion

The new development in Lithium-Ion technology offers higher specific energy, charge efficiency, and energy density over previous battery technologies. It can operate in a wide range of temperatures, and exceptional at delivering peak power for short bursts of time. In addition, it is cost-competitive with NiH$_2$ in small volume production. Lithium-Ion batteries can perform poorly at low temperatures, which is why our selected Saft model is built with integrated heaters. The battery also requires elaborate charging and discharging circuitry with other extra electronics management due to the cell sensitivity.

• Others

Other technologies were examined but deemed inadequate in our trade study because conflict between their characteristics and our own requirements are too severe. Lithium Polymer and Silver Zinc are both high energy-dense options, but were omitted due to the extremely high cost and short life-span, respectively.

8.6.3.3 Model Selection

A variety of battery systems have been evaluated for use within our satellite. Several prominent manufacturers’ satellite specific battery models have been examined, and are listed in Table 8.9.

<table>
<thead>
<tr>
<th>Unit</th>
<th>Power Consumption</th>
<th>Cost</th>
<th>Mass</th>
<th>Dimensions</th>
<th>Heritage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Complete Secondary</td>
<td>1.6 [W]</td>
<td>TBD [$]</td>
<td>1.5  [kg]</td>
<td>300 [mm]</td>
<td>Used on over 150 missions</td>
</tr>
<tr>
<td>from Clyde Space</td>
<td></td>
<td></td>
<td>x 150</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Complete Secondary</td>
<td>5.3 [W]</td>
<td>$2.5 million</td>
<td>8.6  [kg]</td>
<td>312 [mm]</td>
<td>Used on 6 missions 430</td>
</tr>
<tr>
<td>Power Package</td>
<td></td>
<td></td>
<td>x 70</td>
<td></td>
<td></td>
</tr>
<tr>
<td>from Terma Space</td>
<td></td>
<td></td>
<td>x 430</td>
<td></td>
<td>Contracted for 158</td>
</tr>
</tbody>
</table>

Table 8.9: Li-Ion battery model trade study

After reviewing the selection the best battery system for the spacecraft is the Saft Li-Ion MicroSat Module. What makes this battery the best option is the large range of operating temperature, small mass, and large battery capacity. Given the power needs for the spacecraft, we believe that the Yardney and Mars Exploration batteries provide more power than necessary and they are too massive. The Saft VES has the lowest mass, but the capacity would only give...
the mission a little over an hour to while running all systems. The MicroSat Module gives the most flexibility through a compromise between mass and capacity. In addition, the MicroSat Module has several advantageous features, including:

- Flight Heritage
- Autonomous Heating Circuit
- Thermally Insulated
- Over Charge Protection
- Over Pressure Protection
- Thermal Monitoring Sensors

Ultimately, the Saft model meets all necessary battery power requirements in the following section.

### 8.6.3.4 Battery Sizing

In general, total mission efficiency is used to calculate the assorted approximate sizes and ratings of the battery system. On average, Li-Ion efficiency is between 85-95%, and the maximum DOD must be lower than a particular value which is influenced by the number of charge and discharge cycles the battery will experience in its lifetime. Based on the mission power budget, the battery must be able to power the entire spacecraft during cruise phase and during any emergency outages. Cruise phase is estimated to take several hundred seconds. In order to mitigate risk, the worst case duration for cruise phase is estimated to increase by one order of magnitude, from approximately 1000 seconds to 10,000 seconds. Therefore, in the worst-case scenario, the Li-Ion battery must output an absolute maximum of 131 Watts for 2.77 hours, plus an additional 10% contingency factor. This makes the battery power dependency time 3.0 hours.

Keeping this in mind, the battery parameters were derived from the following:

1. The required minimum possible voltage needed for burst-power loading corresponds to the strength of battery cells at their DOD at EOL when wired in series. Our Saft model contains 24 cells with minimum voltage of 0.77 V/cell at EOD, which provides 480 W-hr with a voltage of 21.6 V at EOD.

2. The number of watt-hours discharged corresponds to the eclipse load in Watts multiplied by the eclipse duration in hours. In our $L_1$ mission application, we see no eclipse. Therefore the variable for eclipse load is used to represent ascent and cruise phase when the spacecraft is not sun-facing, and the eclipse duration is the time duration for which the spacecraft is not sun-facing. To maintain 131 Watts of continuous power for 3.0 hours, 393 W-hr discharged is needed. Our Saft MicroSat battery can supply 130 Watts of maximum power for 3 hours and 42 minutes in a single drainage event.
3. The number of amp-hours discharged corresponds to the number of W-hr discharged divided by the average cell voltage, multiplied by the number of battery cells in series. 393 W-hr discharged/average cell voltage of 1.25 V multiplied by 24 cells produces a required 13.1 Amp-hours (A-hr) discharged.

4. The amp-hour capacity corresponds to the number of amp-hours discharged divided by the maximum allowable DOD for the spacecraft life duration. For 13.1 A-hr discharged/80% charge level after discharge, a capacity of 16.37 A-hr is needed. Our Saft model provides a total A-hr capacity of 16.8 A-hr.

5. The W-hr charge equates to the number of W-hr discharged divided by the round trip efficiency estimate of 85%. For 393 W-hr discharged, the original charge should be at least 462 W-hr. Our Saft model provides an initial charge of 480 W-hr.

6. Finally, the charge power equals the W-hr charge amount divided by the sunlight exposure time. Saft claims that the Li-Ion charge rate is C/5, or the Amp-hour capacity divided by 5. Therefore, the charging rate is 17 A-hr/5 = 3.4 Amps per hour. The battery will be charged within 5 hours to maximum capacity, but will only require 3.36 hours of sunlight to reach mission-dependent maximum charge.

### 8.6.3.5 Thermal Control

For many batteries, including current Li-Ion types, initial electrochemical reactions during the charge process are endothermic; Heat is absorbed by the battery’s surroundings. However, after an electrical balance point is achieved, and full charge is approached, the battery begins to generate heat and therefore becomes exothermic. This temperature rise during charge phase is important and worth addressing. During charge, the cell temperature decreases until a full charge state is reached; Usually around a voltage of 1.55 V. After this tipping point, electrochemical reactions cease and energy directed into the battery turns to heat. This internal power loss within the battery causes all units to self-heat, which can result in extreme damage or explosion. The heat produced by this process can be predicted with the equation:

\[
H_c = -I_c [3.6 - E_c]
\] (8.14)

Where 3.6 is the ratio of the enthalpy in Joules equivalent to the Faraday’s constant for Li-Ion, \(\eta\) is the instantaneous charge efficiency, \(E_c\) is the charge voltage, and \(I_c\) is the charge current.

At the start of charging the Saft battery is expected to absorb \(-5.6 \text{ amps} \left\{3.6 \times 1 - \frac{21.6 \text{ V}}{24 \text{ cells}}\right\}\) = -15.2 Watts, while producing \(-5.6 \text{ amps} \left\{3.6 \times 0 - \frac{32.8 \text{ V}}{24 \text{ cells}}\right\}\) = 7.92 Watts at the end of charge.

Also, in the reverse scenario during discharge, only about 80% of the stored power is released as electricity, while 15-20% is released as heat and work energy. Thus, heat is generated during discharge and modeled by the equation:

\[
H_d = I_d [E_m - E_d]
\] (8.15)
Where \( I_d \) is the discharge current, \( E_m \) is the maximum charge voltage, and \( E_d \) is the discharge voltage. Our Saft battery is expected to produce \( 4 \text{ amps} \left[ \frac{32.8 \text{ V}}{24 \text{ cells}} - \frac{21.6 \text{ V}}{24 \text{ cells}} \right] = 44.8 \text{ Watts.} \) The Li-Ion discharge currents are presented in Figures 8.5 and 8.6.

---

**Figure 8.5:** Li-Ion lifetime cycles at 25 for current discharge capacity

**Figure 8.6:** Li-Ion lifetime cycle at 25 for current capacity

Because of the aforementioned dangerous heating phenomenon, it is important that our selected unit is under careful thermal control. The Saft MicroSat model is equipped with an integrated thermal control system which mitigates much of the risk associated with heating. In addition to this, the selected ClydeSpace PCDU includes charge and discharge regulation electronics suited for Li-Ion technologies, which also monitor the thermal environment of the battery casing. This information is crucial for the thermal subsystem, where values such as heat generation rate, thermal conductivity, and specific heat are needed for accurate analysis.
Since our battery is used primarily for short-duration high-burst loading once in position at \( L_1 \) (in addition to emergency backup power), the adiabatic temperature will increase for higher specific energy cells which experience higher temperature rise during this kind of discharge. This sudden temperature change is modeled by:

\[
\Delta T = \frac{W \times H_d}{M \times C_p} \left[ 1 - \eta + \frac{E_d}{E_0} \right]
\]

(8.16)

Where \( \Delta T \) equals the adiabatic temperature rise of the battery in \( ^\circ \text{C} \), \( WH_d \) equals the Watt-hour of energy discharged, \( M \) is the mass of the battery, \( C_p \) is the specific heat of the battery in \( W\text{-hr/Kg}^\circ \text{C} \), \( \eta_v \) is the voltage efficiency factor during discharge, \( E_d \) is the average cell entropy per coulomb during discharge (power loss per amp in \( W/A \)) and \( E_0 \) is the average open cell circuit voltage in volts.

The Saft Battery can be expected to jump \( \frac{131 \text{ W-hr}}{4.5 \text{ kg} \times 0.297 \text{ W-hr/kg}^\circ \text{C}} \left[ 1 - 0.85 + \frac{28.23}{28 \text{ V}} \right] = 14.82^\circ \text{C} \) during a peak loading event. The utilization of the integrated Saft thermal control is crucial to keep the battery at a maximum specific energy without overheating, as shown in Figure 8.7.

![Figure 8.7: Performance curves of Li-Ion batteries at different temperatures](image)

8.6.3.6 Cycle Lifetime

Due to our unique orbit situation at \( L_1 \), coupled with our constant sun-facing requirements, our mission utilizes the secondary battery system in a unique way. Free of traditional satellite eclipse time, our battery will be in a perpetual state of readiness and constant charge. Incorporated for mainly short-duration high-load bursts, our battery will be used in a way which affects its lifetime differently than that of traditional LEO and GEO satellite batteries. Because of the incredible importance of this mission, the battery lifetime must be seriously addressed.

If even one electrochemical cell voltage level falls below 1.0 V prior to its discharge capacity, the entire battery has technically failed. Although a worst-case battery failure definition, it must
be realized that failure is not sudden, but occurs slowly over time. Therefore, it is important
that if even one cell degrades below 1.0 V, the remaining healthy cells must provide a voltage
greater than 1.0 V to maintain the minimum required voltage to ensure mission success for the
spacecraft lifetime.

The number or discharge/charge cycles the battery experiences depends on the number
of peak-load events it will encounter during the mission. For long life cycles requirements,
it is best to limit the battery to a low depth of discharge, of about 30%, rather than the 80%
requirement for GEO satellites. This design inevitably commands a larger battery system, and
further mandates that the lifetime survival at a certain temperature is inversely proportional to
the depth of discharge of the battery. Because the battery cycle lifetime is mostly influenced by
the post-discharge DOD and battery temperature, data presented in Figures 8.8, 8.9 and 8.10
most accurately predict our Saft battery's performance in $L_1$ orbit.

Figure 8.8: End-of-discharge voltage data for lithium-ion batteries
Essentially, it can be seen that the DOD and lifetime cycle relationships are fairly linear and
constant for general approximation. Therefore, when the lifetime cycle is multiplied by the DOD value, this product decreases as temperature increases. The total energy in Watt-hours which the battery can provide thus remains fairly constant through its lifetime. In addition, this consistency is crucial in powering all instruments and electronics onboard the satellite which require a nearly constant voltage supply, both at BOD and EOD. Figure 8.11 shows the consistency obtained only by Li-Ion batteries.

![Figure 8.11: Li-Ion lifetime cycle for EOD voltage at 0°](image)

8.6.3.7 High-Load Power Burst

Our battery will be meeting pulse power demands throughout the mission, used for fast discharge applications such as communications once in orbit at $L_1$. The peak power that can be delivered at any time is important to verify and can be derived with Equation 8.17, the maximum power transfer equation for electrical circuits:

$$P_{MAX} = \frac{E_i^2}{4R_i}$$

(8.17)

The maximum power transferred from source to load occurs only when the battery internal impedance equals the load conjugate impedance, $R_i = R_L$. The internal electrochemical voltage of the battery is $E_i$. The power transfer efficiency is found to be only 50% for theoretical calculations, but may not matter for such short load durations. Also, the impedance of the load electronics changes this relationship as well. Our Saft Li-Ion MicroSat contains electrochemical cells, each with an internal resistance of 35 mΩ and internal voltage of 4.1 V at peak charge. This
equates to a maximum single burst of 120.07 Watts at peak charge, and a maximum single burst of 52.07 Watts at EOD.

No single instrument in the spacecraft requires a peak charge of over 32 Watts during any single time. This means that even with the battery at EOD, it can still provide a high-load power burst sufficient for the mission parameters.

### 8.7 Secondary Components

Secondary components include power conditioning and distribution components. Conditioning and distribution components include the battery charge regulator (BCR), power conditioning module (PCM), power distribution module (PDM), shunts, and the telemetry and telecommand system. All of the secondary components are part of one unit: the power conditioning and distribution unit (PCDU).

The secondary components for the power system are determined by the architecture selected by the mission. As stated in Section 8.4.2 above, the HAVEN mission will utilize a fully regulated power system architecture. Two companies, Terma Space and Clyde Space, provide complete custom made secondary power system packages for small satellites. Each of the two power systems were evaluated on the basis of their power consumption, cost, mass and dimensions and heritage. A summary of the trade study can be seen in Table 8.10 below.

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
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<th></th>
<th></th>
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<tr>
<td>Yardney Model 9522</td>
<td>125</td>
<td>11.3</td>
<td>22x30x15</td>
<td>-20 to +50</td>
<td>864</td>
<td>&gt; 800</td>
<td>NextSat</td>
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<tr>
<td>Yardney Model 9492</td>
<td>125</td>
<td>9.17</td>
<td>6x10x7</td>
<td>-20 to +50</td>
<td>864</td>
<td>&gt;2000</td>
<td>XSS-11</td>
</tr>
<tr>
<td>Mars Exploration Rover</td>
<td>50</td>
<td>7.9</td>
<td>10x41x8</td>
<td>-40 to +65</td>
<td>604.5</td>
<td>&gt;2000</td>
<td>Mars Rover</td>
</tr>
<tr>
<td>SAFT Li-ion</td>
<td>N/A</td>
<td>4.5</td>
<td>22x17x9.5</td>
<td>-50 to +65</td>
<td>480</td>
<td>N/A</td>
<td>LEO and GEO satellites</td>
</tr>
<tr>
<td>microSat module</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>SAFT VES 180</td>
<td>N/A</td>
<td>1.11</td>
<td>25 Height 4.3 Diameter</td>
<td>10 to 35</td>
<td>175</td>
<td>N/A</td>
<td>Galileo IOV, Optus D3</td>
</tr>
</tbody>
</table>

Table 8.10: Li-Ion lifetime cycles at 25 for current discharge capacity

Clyde Space has been selected as the preferred vendor for all secondary components as its SmallSat system has low power consumption, low cost, low mass, compact design and adequate flight heritage. Furthermore, acquiring all secondary components from a single vendor will reduce potential integration problems. The system itself is compatible with a fully regulated satellite architecture and finds application in small satellites requiring upto 300W of average on-orbit power. The SmallSat PCDU power system consists of the following components:

1. **Battery Charge Regulator (BCR):** Power from each solar panel feeds into a dedicated BCR. The output then feeds directly onto a battery bus. This configuration allows the use of different cell technologies of varying string lengths on each panel. This configuration also allows the maximum power point of each individual panel to be tracked. The SmallSat power system is equipped with four BCR’s, which allows the HAVEN mission to utilize up to four solar panels.
2. Power Conditioning Module (PCM): The PCM provides a regulated 5V supply from the raw battery voltage. It also serves to provide autonomous battery over-discharge protection. The SmallSat system is equipped with a dual redundant PCM.

3. Power Distribution Module (PDM): The PDM interfaces the PCM with the rest of the spacecraft using 40 MOSFET switches. The MOSFET switches vary the frequency with which they turn on and off, thereby serving as DC-DC voltage converters. The PDM also protects vital system components from current surges via fuses.

4. Shunts: The SmallSat system comes equipped with shunts to dissipate excess power generated by the solar panels.

5. Telemetry and Telecommand: The system comes equipped with sensors to monitor voltage and current per string. A dedicated serial data bus interfaces this performance telemetry from the spacecraft. This interface is also used to issue telecommands. Radiation hardened wiring is another integral part of the secondary components of the power subsystem. Wiring with double insulation will be used to prevent single point failure. By scaling the power requirement and size of the SSETI ESMO mission, it has been determined that the mass of wiring required for the HAVEN project is 1 kg.

Radiation hardened wiring is another integral part of the secondary components of the power subsystem. Wiring with double insulation will be used to prevent single point failure. By scaling the power requirement and size of the European Space Agency’s SSETI ESMO mission, it has been determined that the mass of wiring required for the HAVEN project is 1 kg.

8.8 Cost Matrix

Primary cost estimation has been conducted using both the analogy based estimation method and the parametric estimation method to verify the power subsystem cost budget. The analogy based estimation method involves cost estimation of items similar to those being used on the HAVEN power subsystem and scaling the costs to fit mission requirements. The parametric estimation method involves usage of mathematical relationships called Cost Estimating Relationships (CERs) to relate performance parameters to system costs.

The total cost for the entire power sub-system is found to be $480,568 using the parametric estimation method and $425,750 using the analogy estimation method. Both methods provide a cost within 11% of each other. To ensure adequate funding for the power sub-system, the higher of the two values has been budgeted. For detailed cost calculations, refer to Appendix I.
Chapter 9

Propulsion

The mission requires the use of a $L_1$ transfer stage and a station-keeping and course corrections, or SKCC stage. A $\Delta V$ of 3160 m/s must be achieved from low Earth orbit in order to ensure a proper transfer to $L_1$. Once this $\Delta V$ is provided, the transfer stage will be jettisoned from the spacecraft. During the transfer stage, three course corrections are necessary and in total amount to a $\Delta V$ of 100 m/s. These corrections will be accomplished by the control thrusters that make up the SKCC stage. In order to ensure that the spacecraft makes it to $L_1$ along the desired trajectory, the control thrusters will be on stand-by for the duration of the transfer stage. When the spacecraft reached the $L_1$ halo orbit, any perturbations in the orbit will be mitigated by the control thrusters. The following discussion outlines the propulsion system architecture in detail.

9.1 Background

There are three categories of propulsion that have the potential to be feasible for the HAVEN project mission. They are solid rocket motors, chemical rocket thrusters, and electric propulsion (EP) systems. Each has distinct advantages and disadvantages over the other systems. The transfer stage requires a high thrust so that the burn time during the transfer stage is minimized. This requirement is based on the astrodynamical constraints of the chosen transfer orbit. A high $I_{sp}$ is also desired so that the payload mass is maximized. In contrast, the SKCC stage requires a low thrust such that the spacecraft does not experience high vibration forces during normal operations. The three propulsion types were evaluated based on these criteria.

Solid rockets allow for very high thrust and reasonable specific impulse, or $I_{sp}$ at approximately $\approx 290$ seconds. These engines are typically very heavy, and cannot be throttled. Once a solid rocket motor is turned on, it is extremely hard to turn it off. This propulsion system is only considered for the mission transfer stage.

Chemical rockets have a wide range of $I_{sp}$ and can be throttled. These engines are typically lighter than their solid rocket counterparts, but they require heavy fuel tanks, and piping systems that outweigh the solid rockets in the end. These engines can be used with a variety of propellants. Hypergolic propellants, fuels that ignite upon contact with an oxidizer, are both
the most reliable and the most dangerous in the case of a failure. Liquid Oxygen (LOX) and Liquid Hydrogen (LH2) propellants have very high $I_{sp}$ ($\approx$400 seconds) but require higher tank volume, frequently leak, and need to be stored cryogenically. Monopropellants like hydrazine have very low $I_{sp} \approx$200 seconds but are often used for their simple propellant feed system.

Electric propulsion (EP) systems are very power intensive, drawing on the order of 1000's of Watts. While they provide extremely high $I_{sp} \approx$1000s of seconds, they also provide very low thrust, typically on the order of single Newtons or less. These thrusters were mainly only considered for station-keeping. The main reason for ruling out this type of propulsion for the transfer stage so quickly was that back of the envelope calculations showed that the transfer to $L_1$ would take on the order of tens of years.

The table below shows a trade study for each propulsion category. For the transfer stage, the best option turned out to be the solid rocket motor. For the SKCC stage, it was decided that simple Hydrazine monopropellant thrusters would be used. The main reasons for these choices were mass, performance ($I_{sp}$), thrust, and heritage. Solid rocket propulsion best satisfies the requirements of the mission transfer stage.

### 9.2 Propulsion Requirements

The main requirements of the propulsion subsystem are outlined in the [9.1](#).

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>PROP-01</td>
<td>Component Safety</td>
<td>Spacecraft shall operate without causing damage to components</td>
<td>MISS-02</td>
</tr>
<tr>
<td>PROP-02</td>
<td>Power</td>
<td>Propulsion system shall be able to run entirely off batteries</td>
<td>SYS-01</td>
</tr>
<tr>
<td>PROP-03</td>
<td>Reliability</td>
<td>Few single points of failure</td>
<td>SYS-01</td>
</tr>
<tr>
<td>PROP-04</td>
<td>Complexity</td>
<td>Take advantage of simple, high TRL systems</td>
<td>MISS-04</td>
</tr>
<tr>
<td>PROP-05</td>
<td>Attitude</td>
<td>Attitude shall be controlled by thrusters</td>
<td>PROPHL-02</td>
</tr>
<tr>
<td>PROP-06</td>
<td>Station Keeping</td>
<td>Provide Station Keeping support for &gt;5 years</td>
<td>SYS-01</td>
</tr>
<tr>
<td>PROP-07</td>
<td>$\Delta V$</td>
<td>Shall provide at least 100 m/s $\Delta V$ for course correction</td>
<td>MISS-01</td>
</tr>
<tr>
<td>PROP-08</td>
<td>Timeline</td>
<td>Must not inhibit the spacecraft to achieve operation by 2014</td>
<td>MISS-03</td>
</tr>
<tr>
<td>PROP-09</td>
<td>Cost</td>
<td>Overall SKCC vehicle cost must be $\leq$ $2,000,000$</td>
<td>MISS-04</td>
</tr>
<tr>
<td>PROP-10</td>
<td>Heritage</td>
<td>Must exhibit at least 75% heritage success rate at the time of launch</td>
<td>MISS-04</td>
</tr>
<tr>
<td>PROP-11</td>
<td>Sustainability</td>
<td>Must not have an established retirement date</td>
<td>STRUHL-05</td>
</tr>
<tr>
<td>PROP-12</td>
<td>Solar Access</td>
<td>Must be capable of pointing satellite to the sun for entire mission lifetime</td>
<td>PROP-09</td>
</tr>
<tr>
<td>PROP-13</td>
<td>Orbit Achievement</td>
<td>Must insert the spacecraft into a $L_1$ Halo Orbit and maintain orbital stability for entire mission lifetime</td>
<td>PROP-11</td>
</tr>
<tr>
<td>PROP-14</td>
<td>Satellite Pointing</td>
<td>Must be able to control all six degrees of freedom during mission lifetime</td>
<td>PROPHL-02</td>
</tr>
<tr>
<td>PROP-15</td>
<td>Timeline</td>
<td>Reach $L_1$ within a year</td>
<td>MISS-02</td>
</tr>
<tr>
<td>PROP-16</td>
<td>System Mass</td>
<td>High $I_{sp}$ in order to maximize payload mass</td>
<td>MISS-01</td>
</tr>
<tr>
<td>PROP-17</td>
<td>Astrodynamics</td>
<td>Minimal burn time</td>
<td>PROPHL-01</td>
</tr>
</tbody>
</table>

Table 9.1: Propulsion Subsystem Requirements

PROP-01: Sudden movement as a result of the propulsion systems can have a negative effect on
some of the more sensitive subsystems. In order to mitigate this problem, the selection of SKCC and Transfer stage components will take into account maximum g-force that the spacecraft can withstand.

PROP-02: Propulsion systems must be able operate in case of emergency therefore it should be able to run on battery power. The components selected have a maximum power consumption of 9.6 Watts.

PROP-03: Failure is mitigated through redundancy of as many systems as possible. In this case, only SKCC systems can be made redundant because of cost and mass constraints. Redundancy includes two propellant tanks to hold hydrazine instead of one large tank, and individual thruster cutoffs.

PROP-04: Complexity in a system presents problems with increases risk modes and mass. In light of this, a simple solid rocket motor was selected over other options such as liquid engines. Hydrazine thrusters were selected for the SKCC stage due to their extensive heritage and established architectures.

PROP-05: Propulsion will provide a means of adjusting attitude in coordination with Attitude Determination and Controls Systems and provide a means of desaturating the momentum wheels.

PROP-06: High level requirements state that the spacecraft be operational for at least five years. Therefore, propulsion subsystem must be prepared to meet that requirement with a system and components that will last the lifetime. The mission lifetime requirement is five years, therefore enough hydrazine must be stored to perform maneuvers for the duration of the mission. In addition, the system must be able to withstand the corrosive nature of hydrazine. For this reason, off the shelf thrusters and tanks with known lifetime have been selected.

PROP-07: In order to have proper insertion into the $L_1$ halo orbit, the orbits team has determined that several corrections have to be made by the hydrazine thrusters. The maximum burn requires 100 m/s of $\Delta V$, therefore given the duration of the burn a 5 N hydrazine thruster was selected.

PROP-08: The satellite must achieve full functionality by 2014, and the selected SKCC system and Transfer system must be functional and available to achieve that.

PROP-09: Based from several trade studies, SKCC is comprised of several expensive components. As a system, trade studies discovered that the lower estimate of SKCC cost is approximately $1,000,000 USD in 2009 dollars. With a 20% contingency to account for fluctuations outside of the study and the general lack of concrete knowledge on SKCC costs leads to a less than $1,200,000 USD requirement. The Transfer stage was analyzed and optimized to mitigate cost, with a system cost of $1,500,000 USD in 2009 dollars. Given the modest budget for this mission, it is determined necessary to comply with this requirement for mission success.

PROP-10: At the time of launch, the selected SKCC and Transfer systems must exhibit at least a 75% success rate. SKCC systems are typically chosen based on success rates much greater than 7%. The Transfer Stage system has flown on several missions. Trade studies found that all components have extensive heritage.

PROP-11: The ultimate goal is to attain a sustainable architecture well into the future.
plish this, the selected SKCC system must exhibit potential to operate well past the expected mission lifetime.

PROP-12: The SKCC system must be able to consistently point the satellite toward the sun in order to allow the power system to generate power during the mission lifetime.

PROP-13: The SKCC system must safely transition the satellite during insertion into orbit, and during the maintenance of orbital stability.

PROP-14: The SKCC system must be able to control all six degrees of freedom, allowing for desaturation of momentum wheels and accurate pointing of the satellite body for telemetry and communication.

PROP-15: The time restriction is imposed because of the 5 year operational deadline imposed by high level requirement. In order to reach \( L_1 \) from LEO a maneuver must be made to put the satellite on course. Because of this, only the choices of liquid and solid rockets were available because of the need for high thrust. Our system uses the STAR 30C/BP solid rocket that provides 35000N of thrust.

PROP-16: Mass is a great concern because it is very limited based on the cost constraints of the mission. The propulsion system must make efficient use of propellant in order to maximize the payload while maintaining an overall system mass that the launch vehicle can successfully launch. The \( I_{sp} \) should be high in order to achieve this. For this reason, the STAR 30C/BP solid rocket motor with an \( I_{sp} \) of 290 sec was selected.

PROP-17: Burn time is vital for accurate transfer trajectory and orbits calculation. In order for the orbit calculations to be used, the burn should be nearly instantaneous. For this reason liquid rockets were not selected because of their long burn time. The STAR 30C/BP rocket was selected due to the fact that it was the system with the lowest burn time that fit the rest of the requirements of the mission.

### 9.3 Propulsion System Selection

#### 9.3.1 Transfer Stage

Transfer Stage As stated previously, three different propulsion types were considered for the cruise stage. EP was ruled out because of the extremely long transit time due to the low thrust as well as the high power requirements which in turn would greatly increase the transit time to \( L_1 \). A liquid propellant rocket engine was ruled out because of the high mass and complexity of the components that accommodate liquid engine systems such as fuel tanks, valves, and piping. A solid rocket was chosen because it maximized the payload mass due to its high system thrust to weight ratio. Other reasons for choosing it include its high thrust within a short burn period, and its system simplicity.

Choosing a solid rocket motor for the transfer stage left very few options to choose from that meet the needs of the mission due to the mass and \( \Delta V \) requirements. The motors that were considered for the \( L_1 \) transfer stage are shown in Table 9.2 below. The solid motor that was
chosen for this mission is the STAR 30C/BP, allowing for a payload mass of 251.55 kg.

<table>
<thead>
<tr>
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<tr>
<td>STAR 30E</td>
<td>290.4</td>
<td>35186</td>
<td>673.9</td>
<td>42.5</td>
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<td>STAR 30BP</td>
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<td>542.81</td>
<td>37.69</td>
<td>213.41</td>
<td>243.77</td>
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</table>

Table 9.2: Solid Rocket Motor Trade Study

9.3.2 Station Keeping and Course Correction Stage

For station keeping, past missions have determined that approx. 2.38 m/s of $\Delta V$ per year is required to maintain orbit stability. The ACE mission used 3 lbm/yr of fuel for orbit corrections, and 6lbm/yr for attitude corrections. From this data, and from a carefully designed MATLAB code, we concluded that our station keeping for an eight year mission requires 17 kg of Hydrazine. To accommodate this fuel and maintain a measure of reliability through a backup system, this spacecraft mission will use two custom titanium propellant tanks. Each of these tanks can carry 17 kg of Hydrazine. These tanks are lightweight and are of a simple, reliable, pressure blow down design.

<table>
<thead>
<tr>
<th>Thruster</th>
<th>Propellant</th>
<th>$I_{sp}$</th>
<th>Thrust [N]</th>
<th>Mass [kg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>CHT 0.5</td>
<td>Hydrazine</td>
<td>227.3</td>
<td>0.5</td>
<td>0.195</td>
</tr>
<tr>
<td>CHT 1</td>
<td>Hydrazine</td>
<td>200-223</td>
<td>0.320-1.1</td>
<td>0.29</td>
</tr>
<tr>
<td>CHT 2</td>
<td>Hydrazine</td>
<td>210-227</td>
<td>0.6-2.0</td>
<td>0.2</td>
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<tr>
<td>CHT 5</td>
<td>Hydrazine</td>
<td>216-228</td>
<td>1.85-6.0</td>
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<td>CHT 10</td>
<td>Hydrazine</td>
<td>220-230</td>
<td>3.0-10.0</td>
<td>0.24</td>
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<tr>
<td>CHT 20</td>
<td>Hydrazine</td>
<td>224-230</td>
<td>7.9-24.6</td>
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<td>CHT 400</td>
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<td>Xenon</td>
<td>1000-1600</td>
<td>0.005-0.020</td>
<td>&gt;1</td>
</tr>
<tr>
<td>T-140</td>
<td>Xenon</td>
<td>1800-2200</td>
<td>0.160-0.300</td>
<td>&gt;1</td>
</tr>
<tr>
<td>T-220</td>
<td>Xenon</td>
<td>1500-2500</td>
<td>0.5-1.0</td>
<td>&gt;1</td>
</tr>
<tr>
<td>LT-1N SP</td>
<td>Hydrazine</td>
<td>210 @ 22 bar</td>
<td>1.1-0.3</td>
<td>0.25</td>
</tr>
<tr>
<td>LT-5N SP</td>
<td>Hydrazine</td>
<td>200</td>
<td>6-1.5</td>
<td>0.28</td>
</tr>
<tr>
<td>ACT-45[25?]</td>
<td>Hydrazine</td>
<td>210 @ 6 bar, 220 @ 24 bar</td>
<td>45-16</td>
<td>0.46</td>
</tr>
<tr>
<td>ST-200N</td>
<td>Hydrazine</td>
<td>225</td>
<td>180</td>
<td>1.25</td>
</tr>
<tr>
<td>MR-104A</td>
<td>Hydrazine</td>
<td>239</td>
<td>572.5</td>
<td>1.86</td>
</tr>
<tr>
<td>MR-104C</td>
<td>Hydrazine</td>
<td>223</td>
<td>204.6</td>
<td>1.86</td>
</tr>
</tbody>
</table>

Table 9.3: Station keeping Thruster Trade Study
To accommodate the necessary $\Delta V$ for this mission lifetime the LP-5N SP Hydrazine monopropellant thruster has been chosen to provide orbital station keeping. An ideal station keeping thruster must have low thrust, low mass, and high $I_{sp}$ characteristics. These characteristics will allow for our spacecraft to maintain its orbit for the full eight year mission at the smallest cost to the overall design. This engine is lightweight (280 grams per engine), and has an $I_{sp}$ that is greater than 200 seconds while providing up to 6 Newtons of thrust. This engine will require 9.6 W of power during firing, and has been used on the Ofeq and EROS satellite families. This heritage has demonstrated the reliable success of the LT-5N SP thruster in station keeping maneuvers.

9.4 Propulsion System Architecture

The propulsion system architectures are designed to be as robust as possible in fulfilling the requirements of the mission. The architectures of the transfer stage and the SKCC stage are outlined in the following sections.

9.4.1 Transfer Stage

There are several issues that must be addressed when working with a solid rocket motor. The most important problem is balancing the $I_{sp}$ of the motor, the initial mass, and the final mass of the system such that the correct $\Delta V$ is achieved. The solid motor needs a spark to ignite the rocket fuel. Once this has been done, the rocket will continue to thrust until it runs out of fuel, not until it achieves the correct $\Delta V$. Therefore, our spacecraft mass and $\Delta V$ requirement must be matched accordingly. Since the Falcon 1E launch vehicle maximum payload is 1000 kg, the $I_{sp}$ of the motor and the $\Delta V$ necessary for the orbit transfer will determine exactly how much payload mass the rocket will be capable of transferring to the $L_1$ Halo Orbit. The relationship between these quantities is characterized by the rocket Equation 9.1 shown below.

$$\frac{m_o}{m_f} = 1 = e^{\frac{-\Delta V}{g^* I_{sp}}}$$

(9.1)

Using this equation along with the specifications of the solid rocket motors in the trade study in Section 9.2 the payload mass can be calculated. It is very important that the payload mass be exactly the calculated value or the solid rocket motor will overshoot the target transfer orbit in the case of an under mass payload. It will undershoot the target transfer orbit in the case of an over mass payload and the spacecraft will drift back to Earth. The mass of the HAVEN payload is 251.55 kg and so it is expected to follow the exact trajectory as outlined in Section 4 (Orbits section). In depth calculations were done using the MATLAB code shown in Appendix L. From these calculations the STAR 30C/BP was selected as the transfer stage engine. A diagram of the STAR series solid rocket motors is shown in Figure 9.1.
9.4.2 Transfer Stage Component Specifics

The following discussion outlines each of the component specifics of the solid rocket motor in more detail. Solid Rocket Grain.

The solid grain in the STAR series motors is heterogeneous. This means that solid fuel and solid oxidizer are mixed together and embedded within a polymerized liquid monomer binder. Fuels are typically a metal substance such as aluminum, beryllium, and boron. Possible oxidizers include Potassium perchlorate, ammonia perchlorate, lithium nitrate, potassium nitrate, and ammonia nitrate. Asphalt, rubber, resins, and polymers are used for binders. ATK has not disclosed which fuel, oxidizer crystals, or binder is used to make the STAR 30C/BP, their COTS (Commercial Over The Shelf) rocket motor, but it is most likely that boron was used due to the high $I_{sp}$ of the motor as mentioned in the component specifications sheet. The use of boron in solid rocket motors typically yields an $I_{sp}$ of approximately 290 seconds, and the $I_{sp}$ of the STAR 30C/BP is 291.8 seconds.

9.4.2.1 Internal Burning Cavity

Once a solid rocket motor has been started, it cannot be shut off. Instead, the shape of the solid grain must be designed such that a desirable thrust profile is achieved. The STAR 30C/BP features an eight point star profile, which ultimately yields a relatively constant thrust during the course of the burn. The actual thrust profile of the STAR 30C/BP is shown below in Figure
9.2. Note that the curve is not in SI units. For reference, 1 pound per square inch (psia) is 6894.75729 Pa and 1 pound (lb) is 0.45359237 kg.

Figure 9.2: Thrust vs. time profile of the STAR 30C/BP

9.4.2.2 Ignition System

The rocket igniter is essentially an explosive charge. It is made up of an initiation system and an energy release system. The ignition system is provided instantaneous power. The energy release system supplies the necessary energy to ignite the propellant inside the motor. The energy necessary for ignition is released in the form of heat. This system serves to start the burning of the solid grain propellant.

9.4.2.3 Mounting Flange

The mounting flange serves to mount the solid rocket motor to the spacecraft. A model image of the spacecraft inside of the Falcon 1E fairing can be found in Chapter 10. The motor attaches to the aluminum structure at the mounting flange, but in addition to this, the structure includes a redundant restraint near the leading edge of the rocket motor. In Figure 9.3, the leading edge corresponds to the left side of the diagram.
9.4.2.4 Insulation Layer

The insulation layer in the solid rocket motor is a layer of material that acts as a heat-barrier and is placed between the internal surface of the case and the solid propellant. The main function of the insulation is to prevent the case from losing structural integrity due to high temperatures. Insulation also inhibits undesirable burning on certain propellant grain surfaces, buffers the transmission of case strain into the propellant, bars the mitigation of mobile chemical species within the motor, prevents impingement of combustion products on the case, seals the case, joints, and fittings. This prevents damage and loss of pressure from hot combustion products and guides combustion products into the nozzle in laminar flow. While not all insulator layers satisfy secondary purposes, they all should adequately satisfy the primary purpose. The optimal design of the insulation has already been calculated and integrated into the system upon delivery.

The choices for insulation layers are separated into two categories: elastomers and thermosetting plastics. The selected rocket motor uses the elastomer ethylene propylene diene monomer (EPDM). Elastomers are elastomeric materials that are relatively soft, have low elastic modulus, withstand strains of up to 900%, return quickly to original length after the strain, and are noncrystalline. Elastomers filled with other materials are useful in internal insulation because of their thermal properties and high strain capabilities. For rocket motor uses, there are several different types of elastomers that have been filled with materials like asbestos, carbon, silica, or boric acid.

9.4.2.5 Nozzle

The carbon nozzle throat insert is an advanced design feature that changes the area ratio during the solid rocket burn. The area ratio, or the ratio of the exit of the nozzle to the inlet of the nozzle, is highest upon ignition of the solid rocket. As the fuel is exhausted, the intense heat that is created ablates the carbon nozzle throat insert, thereby decreasing the area ratio. As seen in Figure 9.2, the desired thrust profile increases at a high rate near the beginning of the burn, but then levels off; this is due to the nozzle ablation effect stated above. The exhaust nozzle is a highly specified and designed aspect of the rocket motor. There are two main types of nozzle designs: submerged nozzle and the external nozzle shown in Figure 9.3. There are also two main exit designs: conical and contoured, shown in Figure 9.4. Another design choice to note is the insulator for the nozzle. The rocket motor selected has already been tested and the optimal configuration features the external contoured nozzle with carbon phenolic liner in order to produce the desirable specifications.
Figure 9.3: Submerged and External Nozzles
The Submerged and the External Nozzles each have their own limitations and strengths. The term submerged nozzle refers to the fact that the nozzle entry, throat, and part or the entire exit is cantilevered into the combustion chamber. The submerged nozzle uses space more effectively for a volume-limited system, but it tends to be more complex because both inner and outer surfaces of the submerged portion are exposed to pressure forces and hot gases. In the external system, the components that are in the combustion chamber in the submerged design are now connected outside of the chamber. This makes the rocket motor larger, but much less complex. This decreased complexity translates to this mission as cost and risk benefits. This is one of the reasons the STAR 30C/BP is the preferred motor.

The contoured nozzle design turns flow so that the exhaust products exit in a more nearly axial direction, which reduces divergence loss more effectively than a conical exit. Contoured nozzles are subject to increased erosion, but the design can potentially decrease the length of the system and increase the specific impulse. The main driver for the shape of the nozzle is the cost of creating the specific contoured shape. The conical shape is much easier to machine and therefore has the lowest cost. The contoured nozzle on the other hand has a more complex process that could end up increasing the price of the system. Given the ability to choose the nozzle type, a contoured nozzle is the clear choice because it delivers superior performance. The STAR 30C/BP employs the contoured design, which is the ideal choice for our system.
Thermal design is necessary for the nozzle in order to maintain proper aerodynamics and structural integrity. In order to assure optimal performance throughout the exit, the rocket motor must have a thermal liner that will form the aerodynamic contour. This material is exposed to the exhaust-product flow and has an insulating material behind it to serve as a barrier to protect the structure. Carbon liners offer four main advantages over the other main choice, pyrolytic graphite: less change from initial contour, less severe erosion, less surface roughness in the eroded condition, and no char layer. The rocket motor selected uses the carbon phenolic liner because of its superior performance; the use of this material decreases the risk of failure due to nozzle structure.

9.4.2.6 Casing

The titanium case is lightweight and corrosion resistant. Corrosion resistance is desirable so that the motor can be stored for a period of time during system integration and preparation. The casing contains all components of the solid rocket motor necessary for functionality and prevents any contact of these components with the rest of the spacecraft.

9.4.3 Transfer Stage Alternative

Although a solid rocket motor was chosen for the transfer stage of the mission, a liquid bipropellant rocket was strongly considered for a time. A trade study was conducted on liquid propellant rockets and is shown below in Table 9.4. The rocket that was originally recommended is the R-40B made by Aerojet. It uses NTO and MMH hypergolic propellants. The engine itself has a mass of only 6.8 kg but a thrust of 4000 N and an $I_{sp}$ of 293 sec. The mission can be completed with a liquid thruster, but in the end, the liquid system was replaced with a solid rocket motor. There are several reasons for this; these reasons are outlined in the following discussion.

<table>
<thead>
<tr>
<th>Thruster</th>
<th>Manufacturer</th>
<th>Propellants</th>
<th>Thrust [N]</th>
<th>Isp [s]</th>
<th>Mass [kg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>R-42</td>
<td>Aerojet Redmond</td>
<td>NTO/MMH</td>
<td>890</td>
<td>303</td>
<td>4.53</td>
</tr>
<tr>
<td>R-40B</td>
<td>Aerojet</td>
<td>NTO/MMH</td>
<td>4000</td>
<td>293</td>
<td>6.8</td>
</tr>
<tr>
<td>MUT</td>
<td>Rocketdyne</td>
<td>NTO/MMH</td>
<td>5640</td>
<td>292</td>
<td>2</td>
</tr>
<tr>
<td>RL-10</td>
<td>Pratt and Whitney</td>
<td>LOX/LH2</td>
<td>66700</td>
<td>410</td>
<td>131</td>
</tr>
<tr>
<td>AJ10-190</td>
<td>Aerojet</td>
<td>NTO/MMH</td>
<td>26700</td>
<td>316</td>
<td>118</td>
</tr>
<tr>
<td>XLR-132</td>
<td>Rocketdyne</td>
<td>NTO/MMH</td>
<td>16700</td>
<td>340</td>
<td>54</td>
</tr>
<tr>
<td>R4-D</td>
<td>Marquardt (Aerojet)</td>
<td>NTO/MMH</td>
<td>489</td>
<td>316</td>
<td>3.76</td>
</tr>
<tr>
<td>HiPAT</td>
<td>Aerojet</td>
<td>NTO/MMH</td>
<td>445</td>
<td>323</td>
<td>5.2</td>
</tr>
<tr>
<td>R-40A</td>
<td>Marquardt (Aerojet)</td>
<td>NTO/MMH</td>
<td>3870</td>
<td>306</td>
<td>10.25</td>
</tr>
<tr>
<td>MR-104A</td>
<td>“Rocket Research”</td>
<td>Hydrazine</td>
<td>573</td>
<td>239</td>
<td>1.86</td>
</tr>
</tbody>
</table>

Table 9.4: Liquid propellant brief trade study

The liquid propulsion system allowed for a lesser payload mass than the solid rocket motor system by approximately 20 kg. The additional components required by the liquid system for functionality outweigh the mass of the empty solid rocket casing. Given the mass budget of the
system, the liquid configuration could still support the mission. However, a lesser payload mass would be supported with the liquid configuration than the solid configuration. Furthermore, the liquid configuration yielded a higher in-fairing mass which left less mass to potentially sell to secondary payload missions. From a mass standpoint, it is best to go with the solid rocket motor.

The liquid system takes up more volume than the solid rocket system due to the need for propellant tanks. The propellant necessary to make it to the \(L_1\) transfer orbit only fits in the Falcon 1E fairing if custom tanks were made. Four tanks of approximately 145 liters each are needed to propel a 333 kg dry mass (including dry masses of all propulsion components) to \(L_1\). The volumes of the propellants were calculated using the fuel mixture ratio of the R-40B engine and the molecular weights of the MMH and NTO. Calculations are done in MATLAB and the code can be found in Appendix L. From a volume standpoint, it is best to use the solid rocket motor.

Burn time is another variable that is affected by the change from a liquid to a solid rocket. Burn time is inversely proportional to the thrust of the rocket. Since the thrust of the solid rocket motor is roughly an order of magnitude greater than the liquid engine, the burn time of the solid rocket motor is roughly an order of magnitude less than that of the liquid rocket engine. The burn time on the solid rocket motor is only 51 seconds. The orbits subsystem requires a short burn time so that the correct transfer orbit can be achieved easily. The liquid rocket burn time is 501 seconds, and would require more assistance from the SKCC stage in order to maintain the correct transfer orbit. From a thrust-burn time standpoint, the solid rocket motor is more desirable. The solid rocket system also costs less as a whole than the liquid system. The liquid configuration requires many parts, and the cost of the sum of these parts is greater than that of the solid rocket motor system cost. The amount of money saved by choosing the solid is on the order of millions of dollars. The R-40B engine alone costs $2 million while the cost of the STAR 30C/BP motor is only $1.5 million. The system also needs additional components which add up to be an estimated $1-2 million more. From a cost standpoint, it is better to go with a solid rocket motor.

**9.4.4 Station Keeping and Course Correction Stage**

The Station Keeping and Course Correction system has been designed to support an five year mission at the \(L_1\) equilibration point between the Earth and the Sun. This system will provide final halo orbit insertion and five years worth of orbital corrections. This architecture is built to include as many redundant opportunities as reasonably possible. As seen in the diagram, the two fuel tanks are able to supply any and all of the 12 thrusters. If only one tank is viable, the other tank is able to supply fuel to the 12 control thrusters. In the event that one of the thrusters fails, the fuel supply can be cut off in order to prevent further fuel losses. This system is operated by a complex architecture of 21 Hydrazine fuel valves. Each of the propellant fuel valves have been extensively tested with Hydrazine and have a certified lifetime of 15 years, whereas propellant tank lifetimes are estimated with a service life of 10 years. The thrusters have been used on a variety of satellites with 10 year lifetimes, proving its reliability. The lifespan for these thrusters has been determined by the satellite lifetime that they have been used on. A diagram of the SKCC system is shown in Figure 9.5.
Twelve thrusters are necessary to control all six degrees of freedom. By controlling all 6 degrees of freedom, the momentum wheels can be desaturated and the satellite can be adjusted for orbital corrections, solar pointing, and telemetry adjustments. Each thruster requires 9.6 W @ 28VDC and 20°C. To maintain a moderate temperature of 20°C, each thruster comes with three heaters to ensure that the propellant and thrusters operate under less than desirable conditions. The three heaters each require 3 W of power, for a total of 9 W in addition to the intermittent 9.6 W firing power. (9W constant power as needed when temperatures go below 20°C, plus 9.6 W firing power needed when thrusters are required).

The thrusters will use monopropellant Hydrazine as the fuel. This fuel is commonly used for station keeping purposes due to its reliability and availability. This fuel is corrosive by nature, so measures have been taken to ensure that corrosion will not negatively impact the lifetime of any component or overall mission. Each component is specifically designed with corrosive resistant materials like titanium, have been extensively tested, and have broad heritage with Hydrazine fuels.
Chapter 10

Structures, Thermal and Radiation

This section includes a description of the structure, thermal, and radiation subsystems. The items discussed in this section are the overall requirements, a description of the structure and the analysis performed on it, logistics behind component placement, thermal protection, radiation protection, a rough timeline, and risks associated with the subsystem.

10.1 Requirements

The main function of our structure is to house, support and protect the components of the spacecraft. The overall structure needs to be strong enough to withstand the launching g-forces, loads and frequencies, while trying to minimize the mass added to the spacecraft. In addition to the support of the structure, this subsystem is also responsible for radiation protection and thermal management. Table 10.1 is the structures/thermal requirements matrix.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>STRUCT-01</td>
<td>Radiation</td>
<td>Will sufficiently protect all components up to 1 GeV of radiation up to 8 years</td>
</tr>
<tr>
<td>STRUCT-02</td>
<td>Cost</td>
<td>Maximum cost for structure/thermal must be $\leq$ $2 \text{ M}$</td>
</tr>
<tr>
<td>STRUCT-03</td>
<td>Structure Mass</td>
<td>Maximum structure mass must be $\leq 20%$ of the wet mass</td>
</tr>
<tr>
<td>STRUCT-04</td>
<td>Orientation</td>
<td>All components must be able to access necessary information from placement on structure</td>
</tr>
<tr>
<td>STRUCT-05</td>
<td>Solar Panels</td>
<td>Structure must accommodate necessary solar panels</td>
</tr>
<tr>
<td>STRUCT-06</td>
<td>Thermal</td>
<td>Structure must keep components within 2-45°C</td>
</tr>
<tr>
<td>STRUCT-07</td>
<td>Vibration</td>
<td>The first vibrational mode must be above 25 HZ</td>
</tr>
<tr>
<td>STRUCT-08</td>
<td>Stress</td>
<td>Must be able to withstand a 7.7 g axial load and a 0.75g lateral load without yielding or buckling</td>
</tr>
</tbody>
</table>

Table 10.1: Structures, Thermal, and Radiation requirements

STRUCT-01: Particles at $L_1$ have a range of energies from several hundred KeV to MeV. During an SPE, these energies can spike as high as 1 GeV. So in order to ensure that none of the compo-
nents fail, we will protect the satellite from the 1GeV of electrons for the duration of our mission, five years.

STRUCT-02: According to SMAD, the cost of the structure and thermal components of a satellite is approximately 8% of the total budget. $2M is 8% of $25M

STRUCT-03: According to SMAD, the mass of a satellite is approximately 20% of the wet mass.

STRUCT-04: All components must be placed so that they can function properly within the structure. See Section 10.2.1 for further details.

STRUCT-05: The geometry of the sun facing side of the satellite must be able to accommodate the solar panels necessary to power the satellite.

STRUCT-06: The Hydrazine in the control thruster tanks freezes at 2°C, and the star trackers cease to function correctly at 45°C. As shown later in Figure 10.9, these are the two components that define the minimum and maximum temperatures that the satellite can operate at.

STRUCT-07: During launch, the spacecraft will be shaken at 25 Hz. To avoid having the spacecraft shaken apart, we must have the first vibration mode above this frequency.

STRUCT-08: During launch the spacecraft will be subjected to at most 7.7 g of loading in the axial direction and 0.75 g in the lateral direction. So it must be seen that the structure can withstand these loads, with the addition of a safety factor, without yielding or buckling.

### 10.2 Structure Selection

The shape of the spacecraft is based on several factors. Since the solar cells are body-mounted on the top of the satellite, the satellite needs to have the largest cross section area as possible, while still fitting into the Falcon 1E launch vehicle fairing. A circular cross section would be best, however, the flat nature of the aluminum honeycomb panels rule this out as a possibility. On the other extreme, a rectangular cross section will create wasted space. To optimize our design we choose a hexagonal cross section. The structure will be created using hollow, circular 7075 aluminum tubes as a skeleton and honeycomb aluminum panels between them for extra structural stiffening, as well as thermal and radiation protection for the spacecraft. The height of the spacecraft is then minimized so another payload could possibly be flown in the fairing to mitigate launch costs.

#### 10.2.1 Satellite Component Placement

The component placing is mainly determined by restrictions set by the launch vehicle, radiation protection, and pointing needs of the spacecraft. The launch vehicle requires that the center of mass of the spacecraft be aligned within 25 mm of launch vehicle's centerline so as not to interfere with its balance. The designed symmetry of the satellite puts the center of mass within 2 mm in both directions, thus meeting this requirement. The pointing constraints require that the parabolic dish must be on the dark side of the satellite while the sensors and solar cells must be on the sun facing side.
The solid rocket motor and its support structure will be the most massive items, so they will be placed near the bottom of the launch vehicle so as not to crush any of the other components. Since the electrical components, such as the computers, are sensitive to radiation they will be put as close to the dark side of the satellite as possible and the less sensitive components will be placed between them and the sun to act as a shield. The transceiver will be placed at the bottom of this stack because it is by far the most sensitive to radiation. Finally, station keeping thrusters and backup antennae are placed along all major axes and oriented for full control of the spacecraft.

### 10.2.2 Structural Material

Aluminum honeycomb panels will be used on the external surfaces of the HAVEN-1 spacecraft to provide the necessary structural support, thermal protection, and radiation shielding. It is a lightweight core material that offers high strength and good thermal conductivity properties. The panel is formed through bonding a hexagonal honeycomb core to aluminum face material. A section of aluminum honeycomb is shown below in Figure 10.1. The honeycomb is sandwiched between two faces of an aluminum alloy. Aluminum alloy 5056 will be used for our honeycomb panels because the material has been approved for aerospace applications.

![Figure 10.1: A section of aluminum honeycomb used for aerospace applications](image)

Through our use of aluminum honeycomb, the structure will be stiffened and strengthened without adding unnecessary mass. The two sheets of aluminum surrounding the aluminum honeycomb will substantially increase the structure's relative stiffness and strength. The relative strength and stiffness relationship is demonstrated in Figure 10.2. Through the use of a sheet of aluminum honeycomb our structural stiffness increases by a factor of 7 and the strength increases by a factor of 3.5, with only a 3% increase in mass. There is also heritage of the use of aluminum honeycomb in orbit around Lagrange point 1 on SOHO and ACE spacecrafts.
10.2.3 Overall Structure

The satellite's structure consists of a hexagon with a 0.57 m side and is 0.80 m tall as shown in Figure 10.3. The overall structure stands at 3.07 m and is 1.22 m at its widest point as shown in Figure 10.4.

![Figure 10.2: Relative stiffness and strength of a structure with honeycomb](image)

<table>
<thead>
<tr>
<th></th>
<th>Solid Metal Sheet</th>
<th>Sandwich Construction</th>
<th>Thicker Sandwich</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Relative Stiffness</strong></td>
<td>100</td>
<td>700</td>
<td>3700</td>
</tr>
<tr>
<td></td>
<td></td>
<td>7 times more rigid</td>
<td>37 times more rigid!</td>
</tr>
<tr>
<td><strong>Relative Strength</strong></td>
<td>100</td>
<td>350</td>
<td>925</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3.5 times as strong</td>
<td>9.25 times as strong!</td>
</tr>
<tr>
<td><strong>Relative Weight</strong></td>
<td>100</td>
<td>103</td>
<td>106</td>
</tr>
<tr>
<td></td>
<td></td>
<td>3% increase in weight</td>
<td>6% increase in weight</td>
</tr>
</tbody>
</table>

![Figure 10.3: Dimensions of the satellite](image)
The total mass for the structure, radiation, and thermal portion of the spacecraft is 134.5 kg. This mass includes 123.7 kg for the entire structure and fasteners, 7.8 kg for thermal protection and 3 kg for radiation protection. The x, y, and z moments of inertia for our spacecraft are 1.8 kg-m$^2$, 2.1 kg-m$^2$ and 3.0 kg-m$^2$ respectively. Also, the spacecraft occupies a volume of about 2.6 m$^3$ which fits in the Falcon 1E fairing as shown in Figure 10.5.
10.3 Stress and Modal Analysis

We have conducted analyses on the stress, buckling and dynamic response of the structure that will support the spacecraft during and after launch. The purpose of these analyses was to find an initial size, mass, and shape for our structure that would meet all of our stress, stiffness, and buckling requirements. This section contains specifics about the structure design, FEA (finite element analysis) results, and back of the envelope calculations.

10.3.1 Results Summary

We have designed a structure shape that is feasible, desirable, and meets all of our criteria. For launch, our structure is required to be able to withstand a 7.7 g axial load and a 0.75 g lateral load without yielding or buckling, while being stiff enough to keep its first resonant frequency above 25 Hz. Of these three criteria, the stiffness requirement of 25 Hz was our primary design driver.

Figure 10.6: Satellite structure

We have designed a structure that utilizes a series of triangular struts to support HAVEN-1 (Figure 10.6). The triangular supports provide our structure with stiffness. The structure is composed of four different sized tubes, which are described in Table 10.2. The “bottom struts” are the tubes that run from the bottom of the fairing to the rocket support. The “rocket support tube” is the tube that will be bolted to the solid rocket motor. The “middle struts” are the tubes.
that connect the bottom struts to HAVEN-1 (the top of the space craft). Lastly the “HAVEN-1 struts” are the tubes that make up the satellite that will be operating at $L_1$.

<table>
<thead>
<tr>
<th>Tube</th>
<th>Outer Diameter</th>
<th>Inner Diameter</th>
<th>Thickness</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bottom Struts</td>
<td>70</td>
<td>59</td>
<td>5.5</td>
</tr>
<tr>
<td>Middle Struts</td>
<td>35</td>
<td>30</td>
<td>2.5</td>
</tr>
<tr>
<td>HAVEN-1 Struts</td>
<td>30</td>
<td>27</td>
<td>1.5</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Length</th>
<th>Width</th>
<th>Thickness</th>
</tr>
</thead>
<tbody>
<tr>
<td>60</td>
<td>45</td>
<td>3</td>
</tr>
</tbody>
</table>

Table 10.2: Dimensions of tubes that make up the structure. Length and width represent dimensions in the rocket support tube cross-section.

Most of the mass of our structure was needed to meet the stiffness requirement. The first resonant mode of our structure was 27.5 Hz, providing a 10% margin from the 25 Hz requirement. Our structure has a buckling safety factor of 7.7, with the HAVEN-1 struts as the most critical component. Lastly, our structure had a stress safety factor of 2.1, with the rocket support tube as the critical component. All of the tubes in our support structure have a total mass of 55 kg, which does not include the honeycomb panels, brackets, or fasteners. With further analysis it should be possible to further reduce the mass of the structure.

10.3.2 FEA Assumptions/Mesh

It is important to understand the simplifications and assumptions we used when creating the FEA model we used for analysis. We modeled the satellite support structure using the FEA software package, ANSYS. The material properties of aluminum 7075 were applied to tube elements, which were used to model the structure (Table 10.3). The six points that would be touching the bottom of the rocket fairing were constrained to have zero displacement.

The forces from the components, which are shown in red on Figure 10.6 were modeled as distributed loads on the beams. The masses were modeled by one point mass representing the motor and one point mass representing the HAVEN-1. The point mass representing the rocket motor was assumed to be 633 kg, while the point mass representing HAVEN-1 was assumed to be 155 kg. While performing the analysis we assumed that the honeycomb panels, which will be placed onto HAVEN-1, were not load bearing.
### Material Properties of 7075 aluminum used in analysis

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Young's Modulus</td>
<td>71.7 GPa</td>
</tr>
<tr>
<td>Yield Strength</td>
<td>448 MPa</td>
</tr>
<tr>
<td>Ultimate Strength</td>
<td>524 MPa</td>
</tr>
<tr>
<td>Density</td>
<td>2800 kg/m³</td>
</tr>
<tr>
<td>Poisson's Ratio</td>
<td>0.33</td>
</tr>
</tbody>
</table>

Table 10.3: Material Properties of 7075 aluminum used in analysis

#### 10.3.3 Modal Analysis

Stiffness was the primary driver for our structure design. To ensure that our structure had enough stiffness to meet the requirement of the Falcon 1E launch vehicle, we performed a modal analysis using FEA, backed up by back of the envelope calculations.

The Falcon 1E launch vehicle required our structure to have its first resonance mode greater than 25 Hz. We desired a 10% margin above this criterion to ensure that the mode was completely damped out at 25 Hz.

#### 10.3.3.1 FEA

The FEA results show that our structure's first resonant mode occurred at 27.5 Hz, meeting the requirement of the Falcon 1E launch vehicle with a 10% margin. The shape of the first resonant mode is shown in Figure 10.7 below. To ensure that the results we found using FEA analysis made sense, we performed back of the envelope calculations.
10.3.3.2 Back of the Envelope

Back of the envelope calculations were done to prove that the FEA solution was reasonable. To characterize the stiffness of the structure, we modeled it as a cylinder with a distributed load. The radius of this cylinder was 0.493 m. We used the equations below and solved for $I$.

\[ f = 0.25 \sqrt{\frac{EI}{mbL}} \quad (10.1) \]

Where $f$ is the first resonant frequency, $E$ is Young’s modulus, $I$ is the area moment of inertia, $m_b$ is the distributed mass, and $L$ is the length of the cylinder. Using the frequency requirement of 25 Hz from the Falcon 1E launch vehicle, we solved for the moment of inertia. Next, we found a bar structure with an equivalent moment of inertia, utilizing the parallel axis theorem:

\[ I = I_{cm} + Ad^2 \quad (10.2) \]

Where $I_{cm}$ is the moment of inertia around the center of the bar, $A$ is the cross-sectional area of the bar, and $d$ is the distance from the center of the bar to the center of our spacecraft. Because $I_{cm}$ is small compared to the $Ad^2$ term, we neglected it. Assuming the distance from the center of the bar to the center of the spacecraft is 0.493 m, we found the total mass needed to
keep our structure stiff enough to meet the 25Hz criterion. The mass needed to keep our satellite stiff enough to meet the launch requirement is approximately 30kg. When cross-supports, and the fact that the structure is not a straight cylinder are taken into account our 55kg mass appears reasonable.

10.3.4 Buckling Analysis

Buckling is the main failure mode for long slender beams. Therefore we did an analysis and back of the envelope calculations to ensure that our structure would not fail from buckling. There are two types of buckling that we were concerned with: global buckling and local buckling. Global buckling occurs when the whole beam bends, and local buckling occurs when the beam crushes, or the tube wall buckles. We found that our design will be able to withstand the effects of both global buckling and local buckling.

10.3.4.1 Global Buckling FEA

We used FEA to run a global buckling analysis. Figure 10.8 shows that in the event of a failure HAVEN-1 would be the first section to buckle, but it has a safety factor of 7.7. Thus global buckling is not a major concern in this design.

Figure 10.8: The 1st buckling mode shape shows that HAVEN-1 struts are the first to buckle
10.3.4.2 Back of Envelope

There are three sections in our structure design that could buckle: the bottom struts, middle struts, and the vertical HAVEN-1 struts. Each of these sections was checked with back of the envelope calculations to ensure that our FEA results were accurate. To check the buckling loads of the struts we assumed they were vertical columns with one end glued and one free, giving a very conservative estimate. The equation for the buckling load for this case is:

\[ F = \frac{\pi^2 EI}{(2L)^2} \]  

(10.3)

Where \( F \) is the critical buckling load, \( E \) is the Young’s modulus, \( I \) is the area moment of inertia and \( L \) is the length of the beam. Using this equation, we calculated the critical buckling loads shown in Table 10.4.

<table>
<thead>
<tr>
<th>Section</th>
<th>Critical Buckling Load [N]</th>
<th>Expected Load [N]</th>
<th>Safety Factor</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bottom Struts</td>
<td>103000</td>
<td>5300</td>
<td>19</td>
</tr>
<tr>
<td>Middle Struts</td>
<td>6000</td>
<td>976</td>
<td>6.1</td>
</tr>
<tr>
<td>HAVEN-1 Struts</td>
<td>2400</td>
<td>1300</td>
<td>1.8</td>
</tr>
</tbody>
</table>

Table 10.4: Back of the envelope calculations for buckling

10.3.4.3 Local Buckling

We have designed the tubes used to be able to withstand local buckling. A research paper from the University of Bristle showed that the following relationship could be used to characterize the local buckling limits of a tube with a circular cross-section.

\[ L' = \sqrt{\frac{l^2 t}{r^3}} \]  

(10.4)

Where \( l \) is the length of the tube, \( r \) is the radius of the tube, and \( t \) is the wall thickness of the tube. If \( L' \) is greater than three, the tube is in the global buckling range. If \( L' \) is less than three, the tube is in the local buckling range. Because the tests done using this relationship were in a laboratory environment under ideal conditions, we would like our tubes to have a minimum \( L' \) value of six. This would give us a safety factor of two. The \( L' \) values for our beams are shown in Table 10.5 below.
<table>
<thead>
<tr>
<th>Section</th>
<th>$L'$</th>
<th>Local Buckling?</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bottom Struts</td>
<td>11.8</td>
<td>NO</td>
</tr>
<tr>
<td>Middle Struts</td>
<td>18.2</td>
<td>NO</td>
</tr>
<tr>
<td>HAVEN-1 Struts</td>
<td>16.9</td>
<td>NO</td>
</tr>
</tbody>
</table>

Table 10.5: Local Buckling occurs when $L'$ is less than three

### 10.3.5 Stress Analysis

To ensure our spacecraft will survive launch we designed a structure that is capable of carrying 7.7 g in the axial direction and 0.75 g in the lateral direction without yielding. We used FEA to perform our stress analysis, and backed up the results with back of the envelope calculations. We found that meeting yield stress was not a design driver, thus most of our structure had very high safety factors. There were a few areas that had higher stresses. These were the tube that supported the rocket motor and HAVEN-1. While these spots had higher stresses than the rest of the spacecraft, they still maintained a safety factor of 2.1.

Figure 10.9: Stresses in our structure when under a 7.7 g axial load and a 0.75 g lateral load.
10.3.5.1 Back of Envelope

To ensure the FEA results made sense, we compared them to back of the envelope calculations. The total mass of our spacecraft is supported by the twelve bottom struts. Assuming each of these struts supports equal weight, we found that each strut should support approximately 70 kg. This equates to 5,300 N. Each of these bottom spars has a cross-sectional area of 0.0011 m$^2$, so using the equation below, where $\sigma$ is stress, $F$ is the force acting on the strut, and $A$ is the cross-sections area of the strut, each spar has a stress of 4.76 MPa.

$$\sigma = \frac{F}{A}$$ (10.5)

This is close to the values we saw from the FEA model. The middle struts support approximately 155 kg. Using the same method as we did for the bottom struts, we found that the stress in the middle struts should be approximately 0.051 MPa. This number is comparable to the FEA results in Figure 10.9.

10.3.6 Conclusion

We have designed a structure that is feasible, desirable, and meets all of the criteria. Our launch vehicle required that our structure be able to withstand a 0.75 g in the lateral direction and 7.7 g in the axial direction, while having enough stiffness to keep the first resonant frequency below 25 Hz. The main design driver was the stiffness criteria, which we met with a 10% margin. Stresses and buckling have large safety factors and are not a concern. There is room to further optimize the structural design and shed off extra mass if desired in the future.

10.4 Thermal Protection

During launch, the spacecraft can experience temperatures as high as 93.3°C, but the solid rocket can only withstand temperatures of up to 38°C. Because of this, the solid rocket will be covered with Felt Reusable Surface Insulation (FRSI) to thermally protect it. FRSI are white Nomex felt blankets that can withstand temperatures up to 371°C. The blankets are bonded directly to the surface with a silicon adhesive. This material is currently used in the Space Shuttle's thermal protection system.

The HAVEN-1 spacecraft itself will utilize both passive and active thermal control mechanisms to ensure that all subsystem components are within operating temperatures for the entire lifetime of the mission. The operating temperatures of our spacecraft components are shown in Figure 10.10. We anticipate a heat transfer of 2657 Watts into the spacecraft which will need to be dissipated for regulation of the internal spacecraft environment.
For passive control, we will use a system combining heat pipes, radiators, thermal paint, and Multilayer Insulation (MLI) blankets. There will be a total of two radiators located on the side panels of the spacecraft. Heat pipes will be used to route heat accumulated from the sun side of the spacecraft and internal instruments to radiators located on the surface of the structure. The heat will then be dissipated into space to help regulate the temperature of the spacecraft.

For the radiator design, we will use a Carbon-Carbon (C-C) composite material with an aluminum honeycomb core. The radiators will serve a dual purpose of being a structural support and thermal management system. Instead of being mounted to the external surface of the spacecraft it will be integrated into the structure. The spacecraft side panels are composed of an aluminum honeycomb core with a top and bottom faceplate of an aluminum alloy. For our purposes, we will replace the aluminum faceplates with a C-C composite material that is highly conductive. Each radiator will be 0.4 m by 0.5 m and placed on the side panels.

The amount of heat rejected by the radiators can be predicted using Equation 10.6 below. In this equation, the variables represent the following: \(\epsilon\) is the emissivity of the material, \(\sigma\) is Boltzmann’s constant, \(A\) is the area of the radiator surface, and \(T\) is the temperature. At a temperature of 20°C, the heat dissipated is approximately 75.4 W. This value will increase as the temperature rises.

\[
Q = \epsilon \sigma A T^4
\]  

(10.6)

Constant conductance heat pipes (CCHPs) will be used to help route heat from the spacecraft to the radiator panels. At minimal temperature differences, CCHPs can transfer heat from a heat source to a heat sink. Heat can be transported in either direction but are typically used
to transfer heat from thermal loads to a radiator panel. The CCHPs consist of a sealed metal tube with a capillary structure on the inside and are filled with a working fluid determined by operating temperature range. Typically, fluorocarbons are used for manned missions and hydrocarbons are used for low temperatures. For our purposes, ammonia will be adequate with an aluminum seal. At one end of the heat pipe, heat is absorbed by evaporation of the working fluid and it is then released at the other end by condensation. As the fluid condenses, it is transported back to the evaporator by capillary forces. This is shown below in Figure 10.11.

![Figure 10.11: Schematic on how heat pipes operate](image)

The heat pipes are lightweight at a mass of 0.5 kg/m with a diameter of 0.952 cm. The CCHPs will be placed underneath the TEGs located under the solar cells and routed down to the radiators on the side panels. Below, Figure 10.12 shows an image of CCHPs. Each CCHP has a heat transport capacity of 423 W with a heat flux of 1574 W/m.
For active control within our thermal system, Polyimide Thermofoil (Kapton) patch heaters will be used to maintain the operating temperatures of particularly sensitive instruments located within the spacecraft. Kapton heaters are flexible, thin-foil heaters ideal for aerospace applications due to their durability and lightweight characteristics.

Four patch heaters will be used with the following specifications: 0.25 mm thick, weigh 0.04 grams per squared centimeter, and run at an operating temperature range of -200°C to +200°C. Two of heaters, at 0.258 grams each, will be placed on either side of the power components box to ensure that the temperature at all times stays at 20°C Celsius. The last two heaters will be one on each of the two thruster propellant tanks. Hydrazine has a freezing point of 2°C Celsius. The tank heaters weigh 0.0244 grams each and will be operating to keep the propellant tanks at 40°C Celsius. The heaters have a minimum lifetime of 20 years in space environment with care to not go over their power limits. Thermistors will be placed on each instrument with the heaters to monitor temperatures and relay when the heaters need to be turned on. Each of the heaters will be running at 5 Watts.

By using Equation [10.7] the warm-up power can be calculated for our system. In order to raise our power components by 2° of our set 20° Celsius using two heaters at 5 Watts each, we calculate that it would take approximately 40 minutes.

\[
P = \frac{mC_p(T_{init} - T_{final})}{t}
\]

For added thermal protection, MLI blankets and thermal paint will be used help protect internal components from the external environment. The MLI blankets will be placed as shielding around the external spacecraft structure with cutouts for each radiator. Thermal paint will protect the spacecraft from heat and radiation by reflecting and absorbing the solar energy emitted. Two coatings of YB-71 white paint will be applied to the sun side of the craft. YB-71 is a ceramic, non-specular zinc orthotitanate (ZOT) white coating with a low solar absorptivity (α) of 0.12 and an emissivity of 0.91. On the dark side of our spacecraft, we will be using a black
Chemglaze Z306 flat surface finish. The Chemglaze Z306 is ideal in space applications providing high thermal absorptivity properties of 0.95 and normal emittance of 0.90.

10.5 Radiation Protection

This section describes what is needed to protect the spacecraft from radiation and how this protection will be accomplished.

10.5.1 Space Environment

The space environment is very dangerous, especially for electronics. Particles that are found in the space environment include electrons and protons. Electrons can cause bit flips and other errors in electrical components that do not usually occur. These are very hard to protect against due to the small size of electrons. The dangerous particles that we will protect against are protons, mainly hydrogen which makes up 90% of the protons. Particles at $L_1$ have a range of energies from several hundred KeV to MeV. During an SPE, these energies can spike as high as 1 GeV.

10.5.2 Protection

Radiation shielding will be needed to protect the components within the spacecraft from galactic rays, solar events, and the radiation belts. There are several types of events that can freeze up or bring down electronic systems. The first kind is known as Single Event Upsets (SEU). These are usually bit flips in which electrons, flowing as electricity, cause a small amount of data to be lost. This is usually of little concern to spacecraft designers as the severity of these events is very low. The second type of event is known as a Single Event Latch up (SEL). These usually occur when a large enough particle strikes an electrical component causing an electron to interfere with the positive side of a component. This component then starts to draw excess current from the component, stopping the flow of electrons and preventing the component's functioning. If found fast enough, a component can be turned off and then on again to correct the problem. The most severe event is known as a Single Event Burnout (SEB). This is when a component is hit hard enough and at the right moment that the component fails completely. A SEB is the main disaster we want to protect against. Our goal is to protect the instruments to allow them to work for our entire mission lifetime. Table [10.6] below shows information about the space environment we are protecting against.
Fluence is a measure of the amount of particles passing through a surface. This can be used to determine the exposure a spacecraft will see. As is evident from Table 10.6, the sheer number of particles at each energy range increases ten-fold during SPEs.

We will be using COTS (Commercial Off The Shelf) hardware in our satellite. Radiation hardened materials could cost over 5% of the satellite budget. Hardware is considered radiation hardened if “the total dose that it can take is two or more times that which will be experienced in the natural space environment”[139]. With that in mind, certain components are available radiation hardened off the shelf. These devices include our payload and sun sensors. These sensors are designed to operate in harsh space conditions. Found below is a table of our electronic components, the total radiation dose they can handle, and how much shielding is needed to keep them under these limits.

<table>
<thead>
<tr>
<th>Component</th>
<th>Radiation Limit</th>
<th>Shielding Needed [mm of Al]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Comm. Transceiver</td>
<td>10 kRad (Si)</td>
<td>4</td>
</tr>
<tr>
<td>Star Tracker</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Sun Sensor</td>
<td>300 kRad (Si)</td>
<td>0.75</td>
</tr>
<tr>
<td>GNC Processing Board</td>
<td>200 kRad (Si)</td>
<td>1.5</td>
</tr>
<tr>
<td>C&amp;DH Router</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Proton 200k</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Payload Sensor</td>
<td>200 kRad (Si)</td>
<td>1.5</td>
</tr>
<tr>
<td>Battery Unit</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Power Conditioner Unit</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Power Distribution Unit</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
<tr>
<td>Primary/Secondary Battery</td>
<td>100 kRad (Si)</td>
<td>1.75</td>
</tr>
</tbody>
</table>

Table 10.7: Satellite components, the radiation limits, and amount of shielding of aluminum needed
a graph generated through radiation analysis, shown in the radiation testing section, and is based on an 8 year mission design for safety. Our spacecraft will utilize three different methods of radiation shielding, allowing us to effectively shield our spacecraft without adding excessive mass or cost. The methods are the following: using solar panels on the front of the spacecraft, using smart component placement, and utilizing different levels of shielding. The main radiation design driver is to protect the communication transceiver, the most sensitive instrument on our spacecraft. If we can ensure this will be working, the other instruments will certainly work as well.

### 10.5.3 Solar Panels

Placing solar panels on the sun-side of our spacecraft will help shield our spacecraft from radiation coming from the sun. Solar panels can provide 4mm of aluminum worth of shielding. Thus they can help protect our spacecraft while providing power.

### 10.5.4 Smart Component Placement

While some components are very sensitive to radiation, others are not, and can be used to shield the more sensitive components. When placed around the sensitive components, the non-sensitive components will provide shielding with minimal mass or cost penalty. The most sensitive component is the communications transceiver. At a total radiation tolerance of 10 kRad (Si) it is by far the most sensitive component on our satellite (refer to Table[10.7] for a complete listing of our components and their radiation tolerance). The transceiver is placed behind almost every component that is capable of providing shielding including a battery, the payload, solar cells, and the aluminum honeycomb structure. These components act as if there were an extra 9 mm of aluminum shielding.

![Diagram](image.png)

**Figure 10.13:** Other components, such as a battery pack, can be placed to help protect radiation sensitive components

#### 10.5.4.1 Shielding

To decrease mass, different levels of shielding will be used only in the places where smart component placement and the solar panels on the sun-side of the spacecraft fail to add adequate
protection. Three different levels of shielding will be used: local shielding, whole board shielding, and whole box shielding. Local shielding covers one radiation sensitive IC (integrated circuit), whole board shielding covers an entire radiation sensitive board, and whole box shielding surrounds a radiation sensitive instrument.

For shielding, we will use a sandwich of tantalum - a high-Z material - and aluminum everywhere we see fit based on our simulations. This can be seen in Figure 10.14 below.

A high-Z material is a material with a large number of protons per atom. They are useful for stopping particle radiation. High-Z materials help to protect against large particles but may also cause secondary interactions. This is when space particles interfere with the larger particles found in the high-Z material thus causing more protons to be shot off (like a domino effect). For this reason, this material should be sandwiched between low-Z materials.

10.5.4.2 Radiation Testing

In order to decide how much additional shielding is needed we used SPENVIS4 (website), a free program offered through the European Space Agency. The program works by inputting orbital information. The output will give you information such as the electron and proton fluence and how much shielding is needed for your satellite to withstand a certain radiation dose. Specifications used in our simulation can be found below.

- Mission duration = 8 years
- Orbit type = Near-Earth interplanetary
- Start date = January 1, 2014
- Distance from sun = 0.99 AU

Our results can be seen below in Figure 10.15.
Figure 10.15: SPENVIS radiation results showing total predicted yearly dose versus thickness of aluminum for our satellite.

Based on our testing, we need 4 mm of aluminum to provide adequate protection to our most sensitive equipment. Using the existing protection, we have the equivalent of 11 mm of protection. We will add a sandwich of aluminum and tantalum, as is seen in Figure 10.15 above, consisting of 1 mm of aluminum and 0.2 mm of tantalum to help block out the larger protons that may not get blocked by the other components. Tantalum is often used because it is easily manufactured and is relatively inexpensive. We end up with the equivalent of 13 mm of aluminum covering the communications transceiver allowing for a total radiation does for all 8 years of 1.50 kRad (Si). The next most sensitive instruments require 1.75 mm of aluminum protection to keep in operation. With just the 4 mm that will be guaranteed by the solar panels, these instruments will receive a total dose of about 11.3 kRad (Si) over the mission duration, leaving them with leeway for up to 9 times the expected dose.
Chapter 11

Launch Vehicle

The launch vehicle is an integral part to a mission’s success, responsible for safely transporting the satellite from the ground, through the atmosphere and to its desired orbit. The launch vehicle is therefore an expensive piece of the mission puzzle, and also one that cannot be compromised.

The selected launch vehicle for this solar sentinel mission is responsible for transporting the satellite from the ground to a 185 kilometer lower Earth parking orbit where a separate propulsion system will provide the necessary means to place the satellite in its desired \( L_1 \) orbit.

11.1 Requirements

The low level requirements in Table 11.1 describe the design criteria and constraints for our launch vehicle design. These requirements are derived and traced back to the high level subsystem requirements detailed in Section 2.1.

<table>
<thead>
<tr>
<th>ID</th>
<th>Parameter</th>
<th>Requirement</th>
<th>Trace</th>
</tr>
</thead>
<tbody>
<tr>
<td>LAUNCH-01</td>
<td>Vibration</td>
<td>Must not interfere with the vibrational modes and frequencies of the spacecraft</td>
<td>MISS-01</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>LAUHL-02</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>LAUHL-03</td>
</tr>
<tr>
<td>LAUNCH-02</td>
<td>Timeline</td>
<td>Must not inhibit the spacecraft to achieve operation by 2014</td>
<td>MISS-02</td>
</tr>
<tr>
<td>LAUNCH-03</td>
<td>Cost</td>
<td>Overall launch vehicle cost must be ( \leq $30,000 ) USD/kg</td>
<td>MISS-03</td>
</tr>
<tr>
<td>LAUNCH-04</td>
<td>Heritage</td>
<td>Must exhibit at least 50% heritage success rate at the time of launch</td>
<td>MISS-04</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>LAUHL-03</td>
</tr>
<tr>
<td>LAUNCH-05</td>
<td>Sustainability</td>
<td>Must not have an established retirement date</td>
<td>MISS-04</td>
</tr>
<tr>
<td>LAUNCH-06</td>
<td>Temperature</td>
<td>Must maintain a temperature inside the fairing casing during the ascent phase to lower Earth parking orbit placement that does not affect component performance</td>
<td>STRUHL-05</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>LAUHL-02</td>
</tr>
<tr>
<td>LAUNCH-07</td>
<td>Orbit Achievement</td>
<td>Must place the spacecraft into a lower Earth parking orbit with an altitude between 180 km and 800 km</td>
<td>LAUHL-01</td>
</tr>
</tbody>
</table>

Table 11.1: Launch Vehicle Low Level Requirements

LAUNCH-01: The launch vehicle must safely house the satellite during ascent into orbit, ensuring its natural frequencies do not interfere with those of the satellite and its components.
LAUNCH-02: The satellite must achieve full functionality by 2014, and the selected launch vehicle must be functional and available to achieve that.

LAUNCH-03: Futron Corporation conducted a study in 2002 on the rising costs of launch vehicles using the metric of dollars per kilogram. The study discovered that the lower estimate of launch cost is approximately $25,000 USD/kilogram in 2009 dollars. With a 20% contingency to account for fluctuations outside of the study and the general lack of concrete knowledge on launch costs leads to a less than $30,000 USD/kilogram requirement. Given the modest budget for this mission, it is determined necessary to comply with this requirement for mission success.

LAUNCH-04: At the time of launch, the selected launch vehicle must exhibit at least a 50% success rate. Launch vehicles are typically chosen based on success rates much greater than 50%. However, our budget put a constraint on establishing a more stringent requirement. Furthermore, the nominal purchase of insurance on the satellite helps relieve the risk.

LAUNCH-05: The ultimate goal is to attain a sustainable architecture well into the future. To accomplish this, the selected launch vehicle must exhibit potential to exist well into the future, with no present plans for retirement or replacement.

LAUNCH-06: The launch vehicle must safely house the satellite during ascent into orbit, maintaining a fully-operational temperature for all satellite components.

LAUNCH-07: Two orbit scenarios were outlined for placement into the desired $L_1$ orbit: direct insertion via a third stage that is part of the launch vehicle, or insertion into a lower parking orbit between 180 kilometers and 800 kilometers where the satellite’s propulsion system would place the satellite into the desired $L_1$ orbit. With the $25$ million overall budget, it was quickly determined that direct insertion into the $L_1$ orbit via the launch vehicle was infeasible.

### 11.2 Trade Studies

The initial trade study in Table 11.2 was conducted to determine which of the three architectures was suitable for the solar sentinel mission.

<table>
<thead>
<tr>
<th>Architecture</th>
<th>Description</th>
<th>Pros</th>
<th>Cons</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flagship “piggyback”</td>
<td>“Piggyback” as a secondary payload on a flagship NASA, DoD, or government mission</td>
<td>-Successful heritage</td>
<td>-No control over schedule and integration</td>
</tr>
<tr>
<td>International</td>
<td>Launch as either a primary or secondary payload via an international reconditioned intercontinental ballistic missile</td>
<td>-Extremely affordable as both primary and secondary</td>
<td>-ITAR concerns and constraints</td>
</tr>
<tr>
<td>Domestic Commercial</td>
<td>Launch as either a primary or secondary payload via a domestic commercial launch vehicle</td>
<td>-Affordable as both primary and secondary</td>
<td>-Suspect heritage</td>
</tr>
</tbody>
</table>

Table 11.2: High Level Launch Vehicle Trade Study
The flagship “piggyback” launch option requires the satellite be launched as a secondary payload, which subjects the satellite to the structural attributes and environment of the primary payload and launch vehicle. Accordingly, control and design autonomy of the satellite is lost, and added constraints affect the design. Similarly, although the International launch option is affordable, it subjects the satellite to additional constraints from the International Traffic and Arms Regulations (ITAR), which controls the import and export of defense-related programs. ITAR adds significant constraints in the logistics of the satellite design, such as visa acquisitions, export licenses, control plans, shipping, and the launching nation’s red tape. This results in additional design constraints that affect the satellite.

Thus, the domestic commercial launch architecture was determined most suitable for this mission given the strict cost and schedule constraints and mission description. This architecture eliminates any and all ITAR concerns/constraints and results in the most control for the launch logistics.

The following domestic commercial launch vehicles in Table 11.3 were studied considering the main design drivers: cost, volume, mass to orbit, and schedule.

<table>
<thead>
<tr>
<th></th>
<th>Falcon 1E</th>
<th>Falcon 9</th>
<th>Taurus</th>
<th>Pegasus</th>
<th>Minotaur</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cost</td>
<td>$9.1 million</td>
<td>$40 million</td>
<td>$50 million</td>
<td>$20 million</td>
<td>$20 million</td>
</tr>
<tr>
<td></td>
<td>$9,010/kg</td>
<td>$3830/kg</td>
<td>$37040/kg</td>
<td>$43960/kg</td>
<td>$31250</td>
</tr>
<tr>
<td>Volume [m$^3$]</td>
<td>Height: 3.8</td>
<td>Height: 11.4</td>
<td>Height: 5.7</td>
<td>Height: 2.14</td>
<td>Height: 3.84</td>
</tr>
<tr>
<td></td>
<td>Diam: 1.55</td>
<td>Diam: 5.2</td>
<td>Diam: 1.98</td>
<td>Diam: 1.15</td>
<td>Diam: 1.39</td>
</tr>
<tr>
<td>Mass (185 km LEO) Schedule</td>
<td>1010 kg</td>
<td>10450 kg</td>
<td>1350 kg</td>
<td>455 kg</td>
<td>640 kg</td>
</tr>
<tr>
<td></td>
<td>Projected in service through 2014</td>
<td>Projected in service through 2014</td>
<td>Projected in service through 2014</td>
<td>Projected in service through 2014</td>
<td></td>
</tr>
</tbody>
</table>

Table 11.3: Domestic Commercial Launch Vehicle Trade Study

Anticipating a total payload of approximately 1010 kilograms, both the Pegasus and Minotaur, produced by Orbital, are infeasible for the mission cost requirements. Launching on the Falcon 9 would relegate the satellite to a secondary payload, which is something we are attempting to avoid for logistics and control purposes. The Falcon 1E, developed by SpaceX, and the Taurus, developed by Orbital, are very similar and both comply with the launch low level requirements; however, the Taurus does not meet the cost requirement. Consequently, the Falcon 1E was selected as the primary launch vehicle.

11.3  **Falcon 1E**

The Falcon 1E, developed by Space Exploration Technologies Corporation (SpaceX), is the selected launch vehicle for the solar sentinel mission. The Falcon 1E is an enhanced version Falcon 1, building on the Falcon 1’s successful multiple satellite launch heritage, and improving on its launch capabilities. The Falcon 1E, shown in Figure 11.1, is a two-stage liquid oxygen and rocket grade kerosene powered launch vehicle that is scheduled to replace the Falcon 1 in 2010 for the foreseeable future.
The first stage is an aluminum alloy casing whose design borrows from the heritage of both the Atlas II and Delta II rockets. Inside is the SpaceX Merlin engine, which powers the Falcon 1E's first stage. This first stage is unique in that it is partially reusable. Upon jettison from the vehicle during ascent, the first stage parachutes into the ocean where it is recovered for future launches.

The second stage uses the same aluminum alloy casing as the first stage, and encompasses the second stage propellant tank structure—the Kestrel 2 engine. The Kestrel 2 engine borrows from the heritage of the Space Shuttle External Tank. This stage takes over after the first stage is jettisoned and places the satellite into its desired orbit.

11.3.1 Launch Vehicle Cost

The launch cost of the Falcon 1E is $9.1 million in 2009 USD, or roughly $9,010 per kilogram for a launch to 185 kilometers at a 9.1° inclination angle—the conditions for optimal launch performance and mission effectiveness. This price includes range, standard payload integration, and third party liability insurance.

Additional satellite insurance can be purchased for approximately 15% of the total satellite cost, or approximately $3.75 million to insure the satellite during launch. The 15% average takes into account previous satellites launched by SpaceX and the industry average. The exact cost of insurance is mission specific and cannot be determined until the final mission design is established. Potential companies include Willis Inspace and International Space Brokers and can be chosen prior to launch as seen best fit at the time. The launch insurance covers any damages to the satellite that occur during the launch portion of the mission. This is to ensure that, because the Falcon 1E has yet to develop a successful heritage, funding into the development of the solar sentinel mission is not wasted.

In addition, this cost does not take into consideration transportation to the launch facility at the Kwajalein Atoll launch site. The satellite is shipped via FID International, which is a Better Business Bureau accredited shipping company. FID International accepts the satellite at the Port of Los Angeles and transports it to the Kwajalein Atoll, where a SpaceX barge finishes the shipment and delivers the satellite to the Omelek Island launch site. The shipment takes six weeks and costs $15,000.
11.3.2 Launch Vehicle Fairing

The Falcon 1E fairing in Figure 11.2 is 1.7 meters in diameter, with a usable payload diameter of 1.55 meters. The fairing has a constant usable payload height of 1.7 meters and a tapered usable payload height of 2.1 meters. Two access doors are provided standard for the Falcon 1E.

![Figure 11.2: Falcon 1E Payload Fairing](image)

During development, if it is revealed that there is unused mass and volume in the Falcon 1E fairing, the launch vehicle is equipped to launch multiple satellites. The fairing is compatible with the Secondary Payload Adapter Separation System (SPASS) developed by Space Access Technologies and the Evolved Expendable Launch Vehicle Secondary Payload Adapter (ESPA) developed by CSA Engineering, Inc. The ESPA is utilized on this mission for potential secondary payloads, and is further discussed in Section 11.3.3.

11.3.3 Evolved Expendable Launch Vehicle Secondary Payload Adapter

The Small Launch ESPA 8 in Figure 11.3 is designed for the Falcon 1E, and is a 44 kilogram, 30.48 centimeter tall, 98.55 centimeter diameter composite cylinder with a primary spacecraft isolation system, and accommodations for six secondary payloads.
The six secondary payloads are located at the ports at equidistant points around the cylinder beneath the primary payload. Each secondary payload ring can support a satellite with a mass of up to 45 kilograms. In this launch configuration there are 27.9 centimeters between the ports and the fairing. P-Pods and CubeSats are ideal for this SL ESPA 8 launch configuration and are fully supported by the six auxiliary ports. A maximum of 18 1U CubeSats can be launched as secondary payloads using the SL ESPA 8.

11.3.4 Launch Vehicle Mass Envelope

The Falcon 1E is capable of delivering 1,010 kilograms of payload into an orbit of 185 kilometers at a launch inclination angle of 9.1° as illustrated in Figure 11.4. Both the Orbits and Propulsion subsystems approved this launch scenario as feasible for effective \( L_1 \) orbit placement and have engineered their satellite subsystems accordingly.
Furthermore, it should be noted that the Falcon 1E can launch at a wide range of inclination angles to different altitudes to accommodate for a variety of future launches.

### 11.3.5 Launch Vehicle Thermal Attributes

The Falcon 1E fairing experiences a wide range of thermal environments during the launch phase. The satellite is designed to operate and survive the frigid environment in space, and also designed to survive the increased temperatures during launch. The thermal conditions within the fairing initially increase up to a maximum temperature of 93.3°C before decreasing with the decreasing ambient conditions. The satellite and its components have been designed to withstand the increased temperature during the launch phase of the mission.

### 11.3.6 Launch Vehicle Structural Attributes

The satellite structure has been engineered such that it will not interfere with the Falcon 1E frequency modes in Table 11.4. In particular, the satellite structure has been designed to avoid frequencies of approximately 5 Hz to 43 Hz to ensure no structural failure with the launch vehicle during launch.
11.3.7 Launch Vehicle Acceleration Loads

The Falcon 1E experiences both axial and lateral loads due to acceleration during launch. These loads, summarized by the different phases of the launch in Table 11.5, are also experienced by the satellite structure. The satellite is designed to survive the maximum acceleration load of 7.7 g's axially and 2.0 g's laterally.

<table>
<thead>
<tr>
<th>Flight Event</th>
<th>Axial (g)</th>
<th>Lateral (g)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ground Handling</td>
<td>0.50</td>
<td>2.00</td>
</tr>
<tr>
<td>Liftoff</td>
<td>1.60</td>
<td>0.50</td>
</tr>
<tr>
<td>Stage 1 Burnout</td>
<td>7.70</td>
<td>0.75</td>
</tr>
<tr>
<td>Stage 2 Ignition</td>
<td>6.25</td>
<td>0.25</td>
</tr>
<tr>
<td>Stage 2 Burnout</td>
<td>7.00</td>
<td>0.25</td>
</tr>
</tbody>
</table>

Table 11.5: Falcon 1E Acceleration Loads

11.4 Logistics

With cost and schedule being two of the main design drivers, launch vehicle logistics for the Falcon 1E are taken into the overall consideration to determine its feasibility for this solar sentinel mission. The launch site and facility offered by the launch vehicle customer dictate schedule, costs, and performance issues that must be taken into consideration for overall compliance to the low level design requirements.

11.4.1 Launch Site

The Falcon 1E launch facilities are located on the Omelek Island in the Kwajalein Atoll, part of the Ronald Reagan Ballistic Missile Defense Test Site. The Atoll is located approximately 2,500 miles southwest of the Hawaiian Islands at a latitude of 8.99°. The integration, operation, storage, and processing facilities are all located on Omelek Island for easy integration into the payload fairing. Because of this co-location, SpaceX requires a mere 18 days for physical integration of the satellite (e.g. the satellite must arrive on Omelek Island 18 days prior to launch). Four Falcon 1 launches have been successfully erected on Omelek Island using these existing facilities.
11.4.2 Schedule

SpaceX has designed the Falcon family of launch vehicles around the premise of streamlined, efficient, and effective satellite integration into the launch vehicle. Their trademark is an 18 day window for physical satellite integration. The final payload design including mass, volume, structural characteristics, mission, operations, and interface requirements must be submitted six months prior to launch. Anticipating a Q1 2014 launch constraint, final designs must be completed by Q2 2013, which is well within our design schedule. The standard launch integration process for the Falcon 1E is in Appendix M.

11.4.3 Shipping

Shipment of the satellite to the Kwajalein Atoll takes approximately six weeks via FID International. With SpaceX requiring the satellite to arrive at Omelek Island 18 days prior to launch, the satellite must be at the Port of Los Angeles ready to ship approximately nine weeks prior to launch.

11.4.4 Insurance

The current cost budget does not include the purchase of launch insurance. Purchasing launch insurance puts the overall budget over the $25 million USD requirement and should be purchased as appropriate. Additional funding would need to be acquired, but is advisable if the Falcon 1E launch vehicle does not clearly demonstrate a successful launch history of greater than 75%-the success rate of comparable commercial launch vehicles-at the time of launch. Not doing so puts the entire mission and investment at a tremendous risk.

11.5 Alternative Launch Vehicle

In the event that the SpaceX Falcon 1E does not comply with the heritage requirement LAUNCH-04 at the time of launch, an alternative launch option has been investigated. The alternative launch option is to launch as a primary payload aboard an Orbital Taurus rocket, which does currently satisfy the heritage requirement. The Taurus launch vehicle complies with all the other requirements except for the cost requirement LAUNCH-03.

11.5.1 Taurus

The Taurus launch vehicle is developed by Orbital Sciences Corporation. The Taurus can launch up to 1,350 kilograms of satellite mass into a 185 kilometer lower Earth orbit. Its fairing, depicted in Figure 11.5, is slightly larger than the Falcon 1E; however, the SL ESPA 8 is still functional in the Taurus. Effectively, with a Taurus rocket, there is more mass and volume available to secondary payloads.
Logistically, the Taurus launches from within the continental United States, easing the schedule in terms of shipping. And because the Taurus has established itself as one of the most reliant commercial launch vehicles, additional launch insurance does not need to be procured. However, this increased performance and reliability comes with a significantly increased cost that will effectively place the program over budget. If this route is to be taken into consideration, further funding will be needed as the cost for launch will be approximately $35-40 million.\footnote{$35-40$ million was quoted by an Orbital representative given our total mass and assuming the remaining mass is fully consumed with secondary payloads.}
Part III

Concluding Remarks
Chapter 12

Risk Analysis and Mitigation Strategies

Providing a solar warning system outside the realm of low Earth orbit is not without its risks, however mitigation strategies for our mission have been established. The following section details both general mission risks along with those specific to individual subsystems.

12.1 Mission Risks

The overall system risks have been identified to fall under five key items as detailed in Table 12.1. These risks are anywhere from component failure to false warnings.

<table>
<thead>
<tr>
<th>Callout</th>
<th>Risk</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Component Failure</td>
</tr>
<tr>
<td></td>
<td>Prior to operation</td>
</tr>
<tr>
<td></td>
<td>After hit by a SPE</td>
</tr>
<tr>
<td></td>
<td>Random occurrence during flight</td>
</tr>
<tr>
<td>2</td>
<td>Communications Loss</td>
</tr>
<tr>
<td>3</td>
<td>Pointing Error</td>
</tr>
<tr>
<td>4</td>
<td>Schedule Lapse</td>
</tr>
<tr>
<td>5</td>
<td>Warning System False</td>
</tr>
<tr>
<td></td>
<td>Positives/Negatives</td>
</tr>
</tbody>
</table>

Table 12.1: Risk

Where possible, components were selected based on heritage or off-the-shelf availability therefore reducing risks of failure. Radiation, thermal and other protection has been taken to ensure the likelihood of failure during the SPE. We have chosen our orbit and GNC systems with added contingencies such that we will maintain strict pointing accuracy and Earth access at all times. Schedule lapses may occur as issues arise during testing and production that are unavoidable.

Outlined in the risk matrix, shown in Figure 12.1 are these system risks. As indicated, the majority of the risks are low probability with a medium to high impact on failure. The likelihood
of a schedule lapse is higher probability, but has a lower impact keeping it in the green (safe) zone.

![Figure 12.1: Mission risks matrix](image)

As shown, there are no high level risks (red) apparent at this time and medium risks (yellow) have mitigation plans in place reducing the chances of failure.

## 12.2 Subsystem Component Risks

The focus of the subsystem component risk analysis is centered on the Technology Readiness Level (TRL), probability and impact in the event failure of individual subsystem components occurs. These parameters are summarized in Table 12.2 and indicated on Figure 12.2 by the callouts.

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Callout</th>
<th>Lowest TRL</th>
<th>Failure Likelihood</th>
<th>Severity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>1</td>
<td>7</td>
<td>Low</td>
<td>High</td>
</tr>
<tr>
<td>Orbits</td>
<td>2</td>
<td>N/A</td>
<td>Low</td>
<td>High</td>
</tr>
<tr>
<td>GNC</td>
<td>3</td>
<td>9</td>
<td>Medium</td>
<td>Medium</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>4</td>
<td>9</td>
<td>Low</td>
<td>High</td>
</tr>
<tr>
<td>Telecommunications</td>
<td>5</td>
<td>9</td>
<td>Low</td>
<td>Medium</td>
</tr>
<tr>
<td>Power</td>
<td>6</td>
<td>6</td>
<td>Medium</td>
<td>Medium</td>
</tr>
<tr>
<td>Propulsion</td>
<td>7</td>
<td>9</td>
<td>Medium</td>
<td>High</td>
</tr>
<tr>
<td>Launch</td>
<td>8</td>
<td>6</td>
<td>Low</td>
<td>High</td>
</tr>
</tbody>
</table>

Table 12.2: Component risk analysis based on lowest TRL
As shown in the Figure, the majority of the risks are high impact but low likelihood of occurring. The propulsion system has been flagged as a slightly higher risk due to the corrosive nature of the tanks and the massive impact of an explosion. The following discusses the individual risks of each subsystem.

### 12.3 Payload

The EPHIN sensor brings along several associated risks. One of these risks with an in-situ sensor at $L_1$ is the small possibility that the particles miss the sensor. These misses account for approximately 1.5% of the CMEs striking Earth. Another risk of placing an in-situ sensor at $L_1$ would be a fast ejection from the Sun. This would not allow for much warning time to the space hotel.

These risks are mitigated by the use of employing the Space Weather Prediction Center (SWPC) which provides remote sensing data as an initial method of detecting an incoming SPE. Risks involving the data include a bit flip or SEU. This could send a false-positive or a false-negative signal to the space hotel. This is a very small risk as we are using radiation hardened boards. When the sensor turns back “On” during a storm to check if the highly energetic particles are still coming, the electronics could be damaged by those highly energetic particles.

Another risk with the sensor is that the electrons arrive while the sensor is in the upload, calibration, or maintenance modes. The sensor will not be detecting particles at these times and would not be able to give a warning until it turned back to “On” mode. Thus, the calibration might be wrong if electrons arrive during the calibration mode.

The last risk is that the Posner prediction method is fairly new and might not be the most accurate warning method by the year 2014. The later risks will be developed and subsequently mitigated by creating better procedures in conjunction with the SWPC system.
12.4 Orbits

There are few risks that can compromise the orbit of the spacecraft. The major risks are an improper estimation of the $\Delta V$ required to transfer out to $L_1$ and to maintain the orbit. In the event that the actual $\Delta V$ exceeds what was allotted, the spacecraft will begin following an unexpected and possibly detrimental orbit path. The risk for a transfer error can be mitigated by storing enough propellant to perform one or more MCCs; imposing a contingency on the propellant mass available for station keeping, and therefore storing excess propellant available for the unexpected maneuvers, will alleviate errors in the $\Delta V$ requirements to correct the Halo orbit. However, the heritage missions generally confirm the $\Delta V$ estimates, and thus these risks are considered to be moderate.

12.5 GNC

The antenna must be pointed to Earth within its (2°) cone. To accomplish this, the spacecraft will be equipped with an antenna pointing mechanism (APM by Surrey Satellite Technology Limited) which will provide a full range of antenna pointing capability to Earth. The antenna will be mounted on the APM and our pointing requirements for satellite communications will be met throughout the mission. The APM covers $\pm 110^\circ$ elevation and $\pm 270^\circ$ azimuth at a rotation rate of $19^\circ$ within the accuracy of $1 \pm 0.162$

12.6 C&DH

The C&DH system is responsible for providing commands and communication channels for the subsystems to communicate their health status and mission specific data. Since this subsystem plays such an important role in the functioning and well being of the spacecraft, special care is taken to ensure its proper functionality. In case of a total loss of C&DH functionality, the spacecraft would be effectively lost. To ensure that this cannot happen, the primary risks for the subsystem have been identified and mitigated. To ensure against bit flips in memory, which could cause a command malfunction, the system will do regular memory checks to guard against this type of failure. In case of corrupted or faulty code, the system has been designed for redundancy and to revert to a safe mode when it encounters a sensor anomaly. New code can also be uploaded via the communications platform. Finally, the Solar Particle Events that the system is designed to warn against can cause electrical damage to the system if the circuitry is improperly designed, and if the system does not enter a proper command mode. This risk can be mitigated through proper electrical system design.

12.7 Communications

The largest points of failure in this communications system would be with the pointing mechanism, radio, and ground station network security breach. A pointing mechanism failure is one
of the greatest concerns since this event would render the spacecraft’s high gain antenna use-
less, forcing ground station workers to communicate slowly with the low gain antennas to fix
problems and manage the spacecraft. Radio failure is another potential mission ending event.
However a redundant S-Band modular radio is included with the system to handle such a fail-
ure. Lastly, is the issue of ground station network security. Each ground station will be hooked
up to a network connecting it to the rest of the team’s chosen ground station sites around the
world to minimize LEO satellite gap time and maximize time with the satellite at $L_1$. However, it
will be important to maintain strict security of the ground station sites and over the connecting
network itself. The details of maintaining this security will fall under mission management.

12.8 Power

Failures can occur in both the primary and secondary power system components. The follow-
ing list summarizes failure modes of primary and secondary power system components and
suggests methods to eliminate these failure modes.

1. Shorting of solar cells: A major problem observed in solar arrays has been shorting be-
tween adjacent solar cells after solar storms. Charging of the solar panels occurs when a
massive electron cloud emitted by a solar storm reaches the vicinity of the satellite. The
charge builds up over time between the cell cover glass and panel surface. Ordinarily such
charges are low in current; however, in regions where the voltage between two adjacent
cells exceeds a threshold value, array current is diverted and sustained into an arc that
forms between the two adjacent cells or between the cells and satellite ground. This large
current value causes local heating which damages the cell substrate and separates the so-
lar cells from the aluminum surface of the array panel. This in turn causes the formation
of a permanent low impedance path for array current.

This failure mode can be prevented by arranging the cell strings so that the voltage be-
tween two adjacent cell strings never exceeds the threshold voltage. Effective use of cell
bypass diodes can also eliminate the cell to ground failure mode. Since the SPEWS satel-
lite will never be in eclipse, an insulating barrier, such as silicone, can be inserted between
cell strings to prevent arc formation.

2. TEG Placeholder: There is a risk that the TEG will fail within the environment due to the
lower TRL level; however everything in the system can be powered solely by solar panel
meaning this is not a significant risk.

3. Single point failures in the power conditioning unit: In the event of a single event upset
which causes a single point failure in the power conditioning unit, the system is guaran-
teed by the manufacturer to recover autonomously without any bus power interruption.

4. Overload of power distribution module: In case of overload or short circuit, the output
current from the power distribution module is instantly limited to protect the upstream
main power bus.

5. Wiring: All connections in the bus will be laser welded to improve reliability. Wires with
double insulation will be used to prevent single point failures.
6. Telemetry: All telemetry not critical to the spacecraft operation (i.e. battery cell voltage monitoring) is non redundant as per standard spacecraft specifications.

12.9 Propulsion

There are a few risks associated with the transfer stage propulsion components. One failure mode of the transfer stage is the possibility that the spacecraft is not aligned properly, thereby causing the thrust of the motor to propel the spacecraft to the wrong transfer orbit. The control thrusters must be used to properly line up the spacecraft orientation such that when the rocket motor fires, it takes the spacecraft to the right place.

The tanks for station keeping and course correction stage will be filled with Hydrazine, a corrosive liquid. Missions that have used this tank have not exceeded 10 years of Specified Service Life. This may be a result of the fuel itself, or of the operation of the tank. This corrosive liquid will also affect the effectiveness of the valves and piping. We would need redundancies of the valves. We need to make sure that in the event of a single tank failure, that the other tank can provide fuel to all 12 thrusters. This means that the piping will be more complicated. Redundancy will also be necessary for the 12 thrusters. Other failure modes include fuel leaks, and valve failures. In order to mitigate these risks components selected will have heritage and extensive testing from the supplier.

In order to mitigate these risks we have designed redundancy for several of the subsystem components. Two tanks for hydrazine will be used in case there is a failure in one. In addition the system will have all welded connections and surge suppression orifices to limit leakage and failure. Heaters will be used to control the temperature of the propellant. In the transfer stage, the main risks are for pump failure, valve failure, and leaking. Leaking is the most serious problem because if the propellant leaks and reacts with the oxidizer the spacecraft will explode. In order to mitigate the risk of mixture, a special piping architecture will be researched. There will be multiple valves strategically placed in the system to account for failure.

12.10 Structures

All of the risks involved with the structures subsystem have a low probability of occurrence. Many of these components do, however, have a high impact on the mission if they fail. The structural failure points can be mitigated by constructing the spacecraft with a high safety factor and doing tests, whereas the thermal failures can be prevented by having redundancies in the thermal systems. Although many of the risks have a high impact on the spacecraft, the low probability of occurrence limits the risks to a medium level.
12.11 Launch

Launch has two major risks, the launch vehicle and the payload adapter (ESPA) with TRL levels of 7 and 6, respectively. For the use of the HAVEN-1, the launch vehicle is an evolved expendable launch vehicle secondary payload adapter and poses a medium risk with a medium level of failure probability due to the lack of heritage. The ESPA also has a medium risk level due to minimal heritage. The adapter will only be used however if we decide to fly additional payloads besides the main spacecraft.
Chapter 13

Time-Line

In order to ensure the mission is operational by 2014, the following timeline will be enforced. This schedule assumes a four year period from development to the first signal being relayed.

Since most of the components have flight heritage, the design and engineering analysis period should last around four months. Following this period there will be around four months to procure components and begin testing. Once all components are received they will undergo the critical design period, lasting around a year, where the components will undergo extensive testing and improvements. Having proven each separately they will be assembled and integrated into the spacecraft over the course of another four months. Full system acceptance testing will then last another three months. Finally, a year will be allotted for the spacecraft to be integrated into the launch vehicle, launched and placed at \( L_1 \) sending the first signal back on schedule. There will be minimal time for errors to occur and thus heritage components were a key driver in the spacecraft design.

13.1 ITU Regulation Management and Negotiations

The first important step in development of the HAVEN communications system is beginning the process of completing the steps necessary to notifying regulatory agencies such as the ITU.

Likely a legal consultant should be hired to help facilitate this process and ensure regulations do not inhibit development later on during more critical project development phases. Thus, this work should begin immediately following project proposal acceptance.
13.2 Ground Station Agreement Negotiations

In conjunction with ITU negotiations, negotiations with selected commercial ground stations should begin to understand exactly what costs are associated with what services for each set of ground stations. This should also aid in the generation of detailed communication schedules for the mission once $L_1$ is reached. Ideally, negotiations could be finished or at least agreed to in principle before the first component orders are placed.

13.3 Detailed Design, Scheduling and Link Analysis

Immediately following proposal acceptance, an aggressive program will be undertaken to create a more detailed design through analysis and simulation of the spacecraft. This period should involve analyzing the impact of other systems interaction with the communication system, especially looking into potential radiation patterns with respect to secondary antenna placement.

13.4 Production and Flight Hardware Components

Following the detailed design phase, any new engineering development to be done for ordered components based on component requirements determined by the detailed design phase. Following development of the products this also includes time for space hardening and qualification of the components, which will be the responsibility of the contracted company responsible for each component. The one year for production of all flight components should be more than adequate as the transceiver are COTS components still offered by the selected vendors. The high gain parabolic antenna will have to be custom-made, but has been done before and thus should be available within this time-frame.

13.5 Spacecraft Integration and Testing

Following arrival of flight hardware will begin spacecraft integration and testing of the components integrated with the other spacecraft flight systems. A more detailed test matrix will be worked out in the detailed design phase of the project. This phase is expected to last approximately 6-12 months for completion of integrated systems testing with the rest of the spacecraft and likely after this stage will be handed off shortly thereafter to be transported to the selected launch facility.
13.6 Integration with Commercial and Customer Ground Station Networks

In conjunction with the spacecraft integration and testing phase will be integration of selected ground station networks with the mission communications architecture. In terms of resources required by this team for this effort, most work will fall under testing with our customer's current ground station facilities. Commercial ground station networks such as USN, according to their posted policies, will supply staff to aid with integration with their network upon finishing a service contract agreement, which should place less need to allocate resources to this area when this point in the mission development cycle comes.
Chapter 14

Conclusions

Solar particle events threaten systems vital to operations on Earth and in the surrounding space. The most important of these events are large solar storms that are characterized by highly energetic particles that both endanger humans and cause mechanical malfunctions.

While the demand for more privatized space opportunities has increased, the risk of exposure to solar particles remains. Entrepreneurs with hopes of tapping into the space market must currently go through the government in order to obtain space weather information. These government agencies not only slow down the flow of information, but they utilize spacecraft that are at a high risk of failure. In order to mitigate these risks and to better cater to the increasing demand for safe access to space, we propose the first phase of the HAVEN Project (the HAVEN-1 spacecraft).

The HAVEN-1 solar sentinel will be loaded onto a Falcon 1E launch vehicle and launched into low earth orbit. After separation from the fairing, any secondary payloads will be released and the STAR 30C/BP rocket motor will ignite, propelling the HAVEN-1 spacecraft into its transfer orbit. After some minor course corrections, the spacecraft will be injected into the $L_1$ halo orbit. Once there, the EPHIN sensor will activate and start scanning for any solar particle events. If any events are detected, the alert will activate, the United Space Network will receive the transmission, and any space ventures will be alerted and given a thirty minute warning time to get to safety.

The HAVEN Project will act as the next generation of solar sentinel detection systems and pave the way for an affordable and diverse network of sensors devoted to the safety of mankind. The system shifts the stream of data from the web of government agencies to directly where the data is most vital, the space entrepreneurs themselves. By spending less than $25 million, customers will receive the advantage of in-situ measurement ability. The HAVEN-1 spacecraft will alleviate the cost of solar sentinel missions while keeping the advantage of in-situ measurement capability. Operational success will pave the way for safe ventures in the near-Earth space.
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Part IV

Appendices
Appendix A

Summary of Past and Related Missions

Solar Maximum Mission Satellite (SMM) [NASA]. A satellite launched on February 14, 1980 designed to investigate solar phenomenon (particularly solar flares). The mission ended on December 2, 1989 when the spacecraft re-entered the atmosphere and burned up.

- In 1984, the Space Shuttle Challenger docked with the SMM and performed maintenance and repairs. This increased the SMM mission life.
- Major Sensor on board: Active Cavity Radiometer Irradiance Monitor (ACRIM) - showed that the Sun is actually brighter during the sunspot cycle maximum.

Solar and Heliospheric Observatory (SOHO) [ESA, NASA]. Spacecraft launched on Atlas IIAS on December 2, 1995 to study the sun.

- Began normal operations in May 1996
- Originally planned as a two-year mission, but currently continues to operate.
- Currently the main source of near-real time solar data for space weather prediction.
- One of three spacecraft currently near \( L_1 \)
- First three-axis-stabilized spacecraft to use its reaction wheels as a virtual gyroscope.
- 12 instruments in its payload module, each capable of independent or coordinated observation of the Sun or parts of the Sun.
- Three main Scientific Objectives:
  - Investigate (through solar atmosphere remote sensing) the outer layer of the sun (consisting of the chromospheres, transition region, and corona)
  - Observe solar wind and associated phenomena in the vicinity of \( L_1 \) (through in-situ solar wind observations)
  - Probe the interior structure of the Sun
Global Geospace Science (GGS) Wind Satellite [NASA]. Spacecraft launched on Delta II on November 1, 1994 from Cape Canaveral, Florida to study the radio and plasma that occur in the solar wind and in the Earth's magnetosphere. WIND arrived at $L_1$ in 2004 and is currently still operational.

- Main Scientific Objectives:
  - Provide complete plasma, energetic particle, and magnetic field input for magnetospheric and ionospheric studies.
  - Determine the magnetospheric output to interplanetary space in the up-stream region.
  - Investigate basic plasma processes occurring in the near-Earth solar wind
  - Provide baseline ecliptic plane observations

Advanced Composition Explorer (ACE) [NASA]. Spacecraft launched on August 25, 1997 to study matter and energetic particles (through in-situ observation) from solar wind, interplanetary medium, and other sources.

- Data from ACE is used to improve forecasts and warnings of solar storms
- Is located at $L_1$
- Some of the Major Sensors on board:
  - Cosmic Ray Isotope Spectrometer (CRIS) - determines the isotope composition of galactic cosmic rays.
  - Solar Wind Ion Mass Spectrometer (SWIMS) and Solar Wind Ion Composition Spectrometer (SWICS) - Analyze the chemical and isotopic composition of solar wind and interstellar matter.
  - Ultra-Low Energy Isotope Spectrometer (ULEIS) - measures ion flux and determines the makeup of solar energetic particles.
  - Solar Energetic Particle Ionic Charge Analyzer (SEPICA)
  - Solar Isotope Spectrometer (SIS)
  - Solar Wind Electron, Proton, and Alpha Monitor (SWEPAM)
  - Electron, Proton, and Alpha-particle Monitor (EPAM)
  - Magnetometer (MAG)

Ulysses [NASA, ESA]. Spacecraft launched on October 6, 1990 to study the Sun at all latitudes, and characterize fields, particles, and dust.

- Powered by radioisotope thermoelectric generator (RTG).
- Mission originally scheduled to end in July 2008, but remains operational.
• Part of the InterPlanetary Network (IPN), helped detect gamma ray bursts.

• Scientific findings:
  – observed complex ways the Sun's magnetic field interacts with the Solar system
  – discovered that dust from deep space was 30 times more abundant than previously expected.
  – discovered that the magnetic field from the Sun's poles is much weaker than expected.

• Major Sensors on board (10 total):
  – Radio/Plasma antennas
  – Experiment boom
  – Solar Wind Observations Over the Poles of the Sun (SWOOPS)

**Solar Terrestrial Relations Observatory (STEREO).** Two nearly identical spacecraft that were launched into orbit on October 26, 2006 from Cape Canaveral, Florida on a Delta II that enabled stereoscopic imaging of the Sun and solar phenomena (such as Coronal Mass Ejections).

• Highly elliptical geocentric orbits.

• The spacecraft will continue to separate from each other and will eventually allow us to view the entire Sun.

• Major Sensors on board divided across four instrument packages:
  – Sun Earth Connection Coronal and Heliospheric Investigation (SECCHI)
  – In-situ Measurements of Particles and CME Transients (IMPACT)
  – Plasma and Suprathermal Ion Composition (PLASTIC)
  – Stereo/Waves (SWAVES)

**Geostationary Operational Environmental Satellite (GOES) [USA].** First satellite launched in 1974. Used to support weather forecasting, severe storm tracking, and meteorological research.

Yohkoh [JAXA]. Solar observatory spacecraft launched on August 30, 1991 from Kagoshima Space Center.

• 3 axis stabilized in near circular orbit.

• Major instruments on board:
  – Soft X-Ray Telescope (SXT)
  – Hard X-Ray Telescope (HXT)
  – Bragg Crystal Spectrometer (BCS)
Wide Band Spectrometer (WBS)

Burned up during re-entry on September 12, 2005

Hinode [JXA]. Launched on September 22, 2006 to explore the magnetic fields of the sun.

Major instruments:
- Solar Optical Telescope (SOT)
- X-Ray Telescope (XRT)
- Extreme Ultraviolet Imaging Spectrometer (EIS)

### A.1 Future Missions

**Solar Dynamics Observatory (SDO) [NASA].** Follow-on mission to SOHO. Part of the Living With a Star (LWS) program to develop the scientific understanding necessary to effectively address those aspects of the connected Sun-Earth system that directly affect life and society. Scheduled for launch Oct 2009 on Atlas V from Kennedy Space Center. Mission is scheduled to last 5 years 3 months, with expendables expected to last for 10 years.

- 3-axis stabilized spacecraft with 2 solar arrays and 2 high-gain antennas
- to be inserted into geosynchronous orbit at 102 degree longitude, 28.5 inclination
- Major instruments on board:
  - Extreme Ultraviolet Variability Experiment (EVE)
  - Helioseismic and Magnetic Imager (HMI) - studies solar variability and characterizes the Sun’s interior and the various components of magnetic activity.
  - Atmospheric Imaging Assembly (AIA)

**Solar Orbiter (SOLO) [ESA].** Sun-observing satellite scheduled for launch on Atlas 5 from Kennedy Space Center in May 2015. It will perform close observations of the polar regions of the Sun.

- Major instrument packages:
  - Solar Wind Plasma Package
  - Fields Package
  - Particles Package
  - Remote-Sensing Package
Below you can find some more missions planned for launch in the near future that will go to $L_1/L_2$.

**Gaia [ESA].** Mission to Sun-Earth $L_2$. The successor mission to Hipparcos, Gaia, was approved in 2000 as an ESA Cornerstone mission to be launched around 2011.

- Major Instruments on board:
  - Astrometry instrument (ASTRO)
  - Photometric instrument
  - High-resolution spectrometer


**LISA Pathfinder [ESA].** Mission to Sun-Earth $L_1$. Scheduled to Launch in 2010

**Kuafu.** Chinese space project to establish a space weather forecast system composed of three satellites to be completed by 2012. One of these satellites will be placed at $L_1$. 
Appendix B

Budgets

B.1 Costs Budget

<table>
<thead>
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Table B.1: Costs Budget: Communications
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<td>and Coarse Sun Sensor</td>
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Table B.2: Costs Budget: Guidance, Navigation & Control

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Table B.3: Costs Budget: Command & Data Handling
### Table B.4: Costs Budget: Launch

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### Table B.6: Costs Budget: Power

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<td>Clyde Space SmallSat system + Wire</td>
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### Subsystem Summary

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**Table B.7: Costs Budget: Propulsion**

### Subsystem Summary

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**Table B.8: Costs Budget: Structures & Thermal**
### Table B.9: Costs Budget: Additional Costs

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### B.2 Mass Budget

### Table B.10: Mass Budget: Communications

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Table B.10: Mass Budget: Communications
### Table B.11: Mass Budget: Guidance, Navigation & Control

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### Table B.12: Mass Budget: Command & Data Handling

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<td>0.075</td>
<td>2</td>
<td>0.15</td>
<td>0.05</td>
<td>0.1575</td>
</tr>
<tr>
<td>Router</td>
<td>ATMEL AT7910E</td>
<td>23.3 x 16 x 1</td>
<td>0.08</td>
<td>1</td>
<td>0.08</td>
<td>0.05</td>
<td>0.084</td>
</tr>
<tr>
<td>NIC Boards</td>
<td>ATMEL AT7912F</td>
<td>0.021</td>
<td>0.105</td>
<td>5</td>
<td>0.105</td>
<td>0.05</td>
<td>0.11025</td>
</tr>
</tbody>
</table>

### Table B.13: Mass Budget: Launch

<table>
<thead>
<tr>
<th>Component</th>
<th>Component Details</th>
<th>Dimensions (cm)</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Secondary Payload Adapter</td>
<td>SL ESPA 8</td>
<td>44</td>
<td>1</td>
<td>44</td>
<td>0.1</td>
<td>48.4</td>
<td></td>
</tr>
</tbody>
</table>
## Table B.14: Mass Budget: Payload

<table>
<thead>
<tr>
<th>Component</th>
<th>Component Details</th>
<th>Dimensions (cm)</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>EPHIN</td>
<td></td>
<td>35.5x21.9x19.1cm³</td>
<td>3.55</td>
<td>1</td>
<td>3.55</td>
<td>0.15</td>
<td>4.0825</td>
</tr>
</tbody>
</table>

## Table B.15: Mass Budget: Power

<table>
<thead>
<tr>
<th>Component</th>
<th>Component Details</th>
<th>Dimensions (cm)</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Panel</td>
<td>Hexagon w/ 0.58325m² area + hole</td>
<td>Ask Structures</td>
<td>1.1972</td>
<td>1</td>
<td>1.1972</td>
<td>0.05</td>
<td>1.2571</td>
</tr>
<tr>
<td>TEG</td>
<td>HZ-20 w/ 0.02602 m² area</td>
<td>7.5 x 7.5 x 0.508</td>
<td>0.115</td>
<td>5</td>
<td>0.575</td>
<td>0.05</td>
<td>0.60375</td>
</tr>
<tr>
<td>Clyde Space SmallSat Power System</td>
<td>Power Conditioning and Distribution Unit</td>
<td>30 x 15 x 7</td>
<td>1.6</td>
<td>1</td>
<td>1.6</td>
<td>0.05</td>
<td>1.68</td>
</tr>
<tr>
<td>Battery</td>
<td>Saft Li-Ion MicroSat Module</td>
<td>22 x 17 x 9.5</td>
<td>4.5</td>
<td>1</td>
<td>4.5</td>
<td>0.05</td>
<td>4.725</td>
</tr>
<tr>
<td>Wire</td>
<td>N/A</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>0.1</td>
<td>1.1</td>
<td></td>
</tr>
</tbody>
</table>

Subsystem Summary

- **Subtotal Mass**: 3.55 kg
- **Subsystem Mass Total**: 4.08 kg

**Table B.14: Mass Budget: Payload**

- **Subtotal Mass**: 8.87 kg
- **Subsystem Mass Total**: 9.37 kg

**Table B.15: Mass Budget: Power**
## Subsystem Summary

<table>
<thead>
<tr>
<th>Subsystem Mass Total</th>
<th>79.63</th>
</tr>
</thead>
<tbody>
<tr>
<td>Transfer Stage Propellant Mass</td>
<td>590.8</td>
</tr>
</tbody>
</table>

### Table B.16: Mass Budget: Propulsion

<table>
<thead>
<tr>
<th>Component</th>
<th>Component Details</th>
<th>Dimensions (cm)</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Engine</td>
<td>STAR 30C/BP Solid Rocket Engine</td>
<td>76.2 diameter x 168.402 length</td>
<td>41.33</td>
<td>1</td>
<td>41.33</td>
<td>0</td>
<td>41.33</td>
</tr>
<tr>
<td>Thruster</td>
<td>LT-5N SP</td>
<td></td>
<td>0.24</td>
<td>12</td>
<td>2.88</td>
<td>0</td>
<td>2.88</td>
</tr>
<tr>
<td>Hydrazine Tanks</td>
<td>Custom from Hamilton Sundstrand</td>
<td>15.38DIA</td>
<td>3.71</td>
<td>2</td>
<td>7.42</td>
<td>0</td>
<td>7.42</td>
</tr>
<tr>
<td>Hydrazine</td>
<td>8 year mission + 100 m/s of correction ΔV</td>
<td></td>
<td>28</td>
<td>1</td>
<td>28</td>
<td>0</td>
<td>28</td>
</tr>
<tr>
<td>Solid Fuel</td>
<td></td>
<td></td>
<td>590.8</td>
<td>1</td>
<td>590.8</td>
<td>0</td>
<td>590.8</td>
</tr>
<tr>
<td>Valves</td>
<td></td>
<td></td>
<td>0.015</td>
<td>21</td>
<td>0.315</td>
<td>0.2</td>
<td>0.378</td>
</tr>
<tr>
<td>Pipes</td>
<td></td>
<td></td>
<td>5</td>
<td>1</td>
<td>5</td>
<td>0.2</td>
<td>6</td>
</tr>
</tbody>
</table>

Total: 195 kg
### Subsystem Summary

<table>
<thead>
<tr>
<th>Subsystem Name</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subtotal Mass</td>
<td>104.38</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Subsystem Mass Total</td>
<td>112.9</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Component</th>
<th>Component Details</th>
<th>Dimensions (cm)</th>
<th>Mass (kg)</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Aluminum Honeycomb</td>
<td>Aluminum Honeycomb composite</td>
<td>72.5 x 300</td>
<td>21.6</td>
<td>1</td>
<td>21.6</td>
<td>0.05</td>
<td>22.68</td>
</tr>
<tr>
<td>Support Structure</td>
<td></td>
<td>55</td>
<td>1</td>
<td>55</td>
<td>0.05</td>
<td>57.75</td>
<td></td>
</tr>
<tr>
<td>Brackets and Fasteners</td>
<td></td>
<td>20</td>
<td>1</td>
<td>20</td>
<td>0.05</td>
<td>21</td>
<td></td>
</tr>
<tr>
<td>Thermal</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Radiators</td>
<td>Carbon-Carbon Radiator</td>
<td>72.7 x 71.75 x 2.2</td>
<td>0.86</td>
<td>2</td>
<td>1.72</td>
<td>0.05</td>
<td>1.806</td>
</tr>
<tr>
<td>Heat Pipes</td>
<td>Variable and Constant Conductance</td>
<td>0.31 (dia) x 300</td>
<td>0.5</td>
<td>6</td>
<td>3</td>
<td>0.05</td>
<td>3.15</td>
</tr>
<tr>
<td>Thermostats</td>
<td></td>
<td>0.0002</td>
<td></td>
<td>10</td>
<td>0.002</td>
<td>0.05</td>
<td>0.0021</td>
</tr>
<tr>
<td>Thermal Blanket and Paint</td>
<td>MLI, Hardened with black capton or Indionoxide</td>
<td>2.91</td>
<td>1</td>
<td>2.91</td>
<td>0.05</td>
<td>3.0555</td>
<td></td>
</tr>
<tr>
<td>Patch Heaters</td>
<td>Polyimide Kapton Thermofoil</td>
<td>72.5 x 300</td>
<td>0.061</td>
<td>2</td>
<td>0.122</td>
<td>0.05</td>
<td>0.1281</td>
</tr>
<tr>
<td>Cartridge/Thruster Heaters</td>
<td>for Hydrzazine Control Thrusters</td>
<td></td>
<td>0.03</td>
<td>1</td>
<td>0.03</td>
<td>0.05</td>
<td>0.0315</td>
</tr>
<tr>
<td>Radiation</td>
<td></td>
<td></td>
<td>3</td>
<td>1</td>
<td>3</td>
<td>0.1</td>
<td>3.3</td>
</tr>
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</table>

Table B.17: Mass Budget: Structures & Thermal
### B.3 Power Budget

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent</th>
<th>Transfer Orbit</th>
<th>Cruise Phase</th>
<th>On-Orbit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Computer Board</td>
<td>on (disabled)</td>
<td>on (enabled)</td>
<td>on (enabled)</td>
<td>on (enabled)</td>
</tr>
<tr>
<td>Router</td>
<td>on</td>
<td>on</td>
<td>on</td>
<td>on</td>
</tr>
<tr>
<td>NIC Boards</td>
<td>on</td>
<td>on</td>
<td>on</td>
<td>on</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent Power (W)</th>
<th>Transfer Orbit Power (W)</th>
<th>Cruise Phase Power (W)</th>
<th>On-Orbit Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Computer Board</td>
<td>14</td>
<td>14</td>
<td>14</td>
<td>14</td>
</tr>
<tr>
<td>Router</td>
<td>4</td>
<td>4</td>
<td>4</td>
<td>4</td>
</tr>
<tr>
<td>NIC Boards</td>
<td>3.5</td>
<td>3.5</td>
<td>3.5</td>
<td>3.5</td>
</tr>
<tr>
<td>Subtotal</td>
<td>21.5</td>
<td>21.5</td>
<td>21.5</td>
<td>21.5</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
<td>23.65</td>
<td>23.65</td>
<td>23.65</td>
<td>23.65</td>
</tr>
</tbody>
</table>

Table B.18: Power Budget: Command & Data Handling

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent</th>
<th>Transfer Orbit</th>
<th>Cruise Phase</th>
<th>On-Orbit</th>
</tr>
</thead>
<tbody>
<tr>
<td>EPHIN</td>
<td>off</td>
<td>off</td>
<td>off</td>
<td>on</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent Power (W)</th>
<th>Transfer Orbit Power (W)</th>
<th>Cruise Phase Power (W)</th>
<th>On-Orbit Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>EPHIN</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>1.85</td>
</tr>
<tr>
<td>Subtotal</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>1.85</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>2.035</td>
</tr>
</tbody>
</table>

Table B.19: Power Budget: Payload
### Table B.20: Power Budget: Power

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent Power (W)</th>
<th>Transfer Orbit Power (W)</th>
<th>Cruise Phase Power (W)</th>
<th>On-Orbit Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Front Solar Panels</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>PCD Unit</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
</tr>
<tr>
<td>Bus Voltage Controller</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Battery Bus</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Battery</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Subtotal</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
<td>1.76</td>
<td>1.76</td>
<td>1.76</td>
<td>1.76</td>
</tr>
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</table>

### Table B.21: Power Budget: Propulsion

<table>
<thead>
<tr>
<th>Component</th>
<th>Ascent</th>
<th>Transfer Orbit</th>
<th>Cruise Phase</th>
<th>On-Orbit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrusters</td>
<td>off</td>
<td>off</td>
<td>Controlled by Ground Control</td>
<td>Controlled by Ground Control</td>
</tr>
<tr>
<td>Tanks</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td>Component</td>
<td>Ascent Power (W)</td>
<td>Transfer Orbit Power (W)</td>
<td>Cruise Phase Power (W)</td>
<td>On-Orbit Power (W)</td>
</tr>
<tr>
<td>Thrusters</td>
<td>0</td>
<td>0</td>
<td>9.6</td>
<td>9.6</td>
</tr>
<tr>
<td>Tanks</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Subtotal</td>
<td>0</td>
<td>0</td>
<td>9.6</td>
<td>9.6</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
<td>0</td>
<td>0</td>
<td>10.56</td>
<td>10.56</td>
</tr>
<tr>
<td>System</td>
<td>Ascent Power (W)</td>
<td>Transfer Orbit Power (W)</td>
<td>Cruise Phase Power (W)</td>
<td>On-Orbit Power (W)</td>
</tr>
<tr>
<td>-----------------</td>
<td>------------------</td>
<td>----------------------------</td>
<td>------------------------</td>
<td>-------------------</td>
</tr>
<tr>
<td>Heater</td>
<td>as required</td>
<td>as required</td>
<td>as required</td>
<td>as required</td>
</tr>
<tr>
<td>Component</td>
<td></td>
<td></td>
<td></td>
<td></td>
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<td>20</td>
</tr>
<tr>
<td>Subtotal</td>
<td>20</td>
<td>20</td>
<td>20</td>
<td>20</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
<td>22</td>
<td>22</td>
<td>22</td>
<td>22</td>
</tr>
<tr>
<td>Battery</td>
<td>discharge</td>
<td>discharge</td>
<td>charge/ as required</td>
<td>charge/ as required</td>
</tr>
<tr>
<td>Component</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Front Solar Panels</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>PCD Unit</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
</tr>
<tr>
<td>Bus Voltage Controller</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Battery Bus</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Battery</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Subtotal</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
<td>1.6</td>
</tr>
<tr>
<td>Contingency</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
<td>0.1</td>
</tr>
<tr>
<td>Total</td>
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<td>1.76</td>
<td>1.76</td>
<td>1.76</td>
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</table>

Table B.22: Power Budget: Thermal
Appendix C

In-Situ Sensor Trade Study
<table>
<thead>
<tr>
<th>Mission</th>
<th>Instrument</th>
<th>Instrument Type</th>
<th>Purpose</th>
<th>Data Determined</th>
<th>Mass (kg)</th>
<th>Power (W)</th>
<th>Bit rate</th>
<th>SOURCE</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Energetic Particle Sensor (EPS)</td>
<td>Energetic Particle Sensor</td>
<td>Detect energetic electrons and protons trapped in Earth's magnetic field as well as direct solar proton, alpha particles, and cosmic rays</td>
<td>Energetic protons (4-500 MeV/n), electrons (4-4 MeV/n), alpha particles (3400 MeV/n), magnetospheric protons, magnetospheric electrons, &amp; high energy protons (3400 MeV/n)</td>
<td>42 kg</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>High Energy Proton and Alpha Particle Detector (HEPAD)</td>
<td>High Energy Proton and Alpha Particle Detector</td>
<td>Detect energetic electrons and protons trapped in Earth's magnetic field as well as direct solar proton, alpha particles, and cosmic rays</td>
<td>Energetic protons (550-700 MeV/n), electrons, alpha particles (640-850 MeV/n), magnetospheric protons, magnetospheric electrons, &amp; high energy protons</td>
<td>42 kg</td>
<td>6.5 W</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>X-ray Sensor (XRS)</td>
<td>X-ray Sensor</td>
<td>Measures solar X-ray flux</td>
<td>Solar X-ray flux (&lt;3 angstroms and 1-8 angstroms)</td>
<td>42 kg</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Extreme UV Instrument (EUV)</td>
<td>Extreme UV Instrument</td>
<td>Measures extreme ultraviolet flux from sun</td>
<td>EUV flux</td>
<td>42 kg</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Magnetometer</td>
<td>Magnetometer</td>
<td>Measure three components of Earth's magnetic field &amp; monitors variations caused by ionospheric &amp; magnetospheric currents</td>
<td>Magnetic fields strength and direction</td>
<td>2 kg</td>
<td>2 W</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td>POES</td>
<td>Total Energy Detector (TED)</td>
<td>Solid-state energetic particle detectors</td>
<td>Measure energy flux carried by charged ions and electrons. Measures magnitude and spatial extent of ion and electron intensities</td>
<td>Electron and proton intensities, Telescope 1 (30-1000 keV) / Telescope 2 (30-6000 keV) Protons (10-275 MeV)</td>
<td>4.7 kg</td>
<td>3.9 W</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Medium Energy Proton and Electron Detector (MEPED)</td>
<td>Solid-state energetic particle detectors</td>
<td>Monitor the intensities of protons and electrons</td>
<td>Density, Temperature, flow characteristics, elemental abundances of solar atmosphere</td>
<td>8.7 kg</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Coronal Diagnostic Spectrometers (CDS)</td>
<td>Double Spectrometers, 5 detector systems, Individual pointing</td>
<td>View sun in extreme ultraviolet spectrum</td>
<td>Composition, charge state, mass distribution, kinetic temperature, and speed of the more abundant solar wind ions (e.g., He, C, N, O, Ne, Mg, Si, and Fe). &lt;1 MeV</td>
<td>100</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td>Cielas</td>
<td>Charge TOF (CTOF)</td>
<td>Spectrometer, accepts solar wind through a 30 degree cone.</td>
<td>Analyze characteristics of particles in the charge range of 1 to 54 keV/n charge, which includes solar wind range. Corresponds to about 1000 km/s</td>
<td>Composition, charge state, mass distribution, kinetic temperature, and speed of the more abundant solar wind ions (e.g., He, C, N, O, Ne, Mg, Si, and Fe). &lt;1 MeV</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Mass TOF (MTOF)</td>
<td>High resolution TOF mass spectrometer</td>
<td>Measure the TOF of mostly singly ionized particles, measure the elemental and isotopic composition of the solar wind over a wide range of solar wind bulk speeds.</td>
<td>Mass of the ion</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Suprathermal TOF (STOF)</td>
<td>Multi-element array of sensors</td>
<td>Measure energy distribution. High energy suprathermal and low energy SEP</td>
<td>Energy distribution of individual charge states of various elements (He and Ne) of solar energetic particles. &lt;1 MeV. Similar to CTOF but different energies.</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Electron Photon Helium Instrument (EPHEN)</td>
<td>Multi-element array of solid-state detectors</td>
<td>Measure energy distribution</td>
<td>Energy spectra of electrons in the range 250 keV to &gt; 8.7 MeV and of hydrogen and helium isotopes in the range 4 MeV/n to &gt; 33 MeV/n</td>
<td>N/A</td>
<td>1.85</td>
<td>172 bits/s</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Low Energy Ion and Electron Instrument (LION)</td>
<td>X-ray Sensor</td>
<td>Measure energy distribution</td>
<td>Measure particle spectra in the range 44 keV to 6 MeV for protons and 44 keV to 300 keV for electrons</td>
<td>2.2</td>
<td>0.9</td>
<td>40 bits/s</td>
<td></td>
</tr>
<tr>
<td>Mission</td>
<td>Instrument Type</td>
<td>Instrument Description</td>
<td>Purpose</td>
<td>Data Determined</td>
<td>Mass (kg)</td>
<td>Power (W)</td>
<td>Bit rate</td>
<td>SOURCE</td>
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</tr>
<tr>
<td>ERNE</td>
<td>Low Energy Detector (LED)</td>
<td>detector layers D1, D2, and AC with pulse amplification and digitization electronics, particle telescope</td>
<td>High energy energy distributions</td>
<td>1 MeV/n up to hundreds of MeV/n and ions from hydrogen to zinc</td>
<td>9.305</td>
<td>7</td>
<td>768 bits/s</td>
<td></td>
</tr>
<tr>
<td></td>
<td>High Energy Detector (HED)</td>
<td></td>
<td></td>
<td></td>
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<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>STEREO/IMPACT</td>
<td>Solar Wind Electron Analyzer (SWEA)</td>
<td>spectrometer, accepts solar wind at 90° from any hemisphere</td>
<td>Measure the distribution of the solar wind core and halo electrons over a specified eV range, provide high spectral and angular resolution</td>
<td>composition, charge state, mass distribution, kinetic temperature, and speed of the solar wind core, max is several keV</td>
<td>0.97</td>
<td>0.35</td>
<td>534 bit/s</td>
<td>IMPACT Solar Wind Electron Analyzer Instrument (on Cots)</td>
</tr>
<tr>
<td></td>
<td>High Energy Detector (HED)</td>
<td></td>
<td></td>
<td></td>
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<tr>
<td></td>
<td>Suprathermal Electron Telescope (STE)</td>
<td>cooled silicon semiconductor devices, electrostatic analyzer</td>
<td>Measure electrons in the energy range 0.2 to 20 keV that extend beyond SWEA range present as a superhalo on the solar wind electrons</td>
<td>energy distribution of electrons</td>
<td>0.8</td>
<td>1</td>
<td>210 bit/s</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Suprathermal Ion Telescope (SIT)</td>
<td>mass spectrometer</td>
<td>Measures elemental composition of H-Fe ions over the 30 keV/nucleon to 2 MeV/nucleon energy range</td>
<td>elemental composition, charge state of He-Fe ions, time-of-flight to determine composition through secondary electrons</td>
<td>1.63</td>
<td>1.65</td>
<td>424 bit/s</td>
<td>IMPACT Suprathermal Ion Telescope Instrument (on Cots)</td>
</tr>
<tr>
<td></td>
<td>Low Energy Telescope (LET)</td>
<td>14 solid state detectors</td>
<td>Measure protons and helium ions from e1.5 to 13 MeV/nucleon, and heavier ions from e2 to 30 MeV/nucleon, field of view: 20 deg above to 20 deg below the ecliptic plane and 65 deg to either side</td>
<td>composition, charge, ensures detection of small SEP events</td>
<td>0.855</td>
<td>1.18</td>
<td>577 bit/s</td>
<td>IMPACT Low Energy Telescope (LET) Instrument (on Cots)</td>
</tr>
<tr>
<td></td>
<td>High Energy Telescope (HET)</td>
<td>6 solid state detectors</td>
<td>Measure protons and helium ions to 100 MeV/nucleon, and energetic electrons to 5 MeV</td>
<td>composition, charge, higher energy events</td>
<td>0.395</td>
<td>0.359-498 max</td>
<td>218 bit/s</td>
<td>STEREO High Energy Telescope Instrument (on Cots)</td>
</tr>
<tr>
<td>Mission</td>
<td>Instrument Type</td>
<td>Purpose</td>
<td>Data Determined</td>
<td>Mass (kg)</td>
<td>Power (W)</td>
<td>Bit rate</td>
<td>SOURCE</td>
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<td>-------------------------------------------------------------------------</td>
<td></td>
</tr>
<tr>
<td>ACE</td>
<td>Cosmic Ray Isotope Spectrometer (CRIS)</td>
<td>solid-state charged particle telescope</td>
<td>measure the abundances of galactic cosmic ray isotopes with energies from ≈ 100 to ≈ 600 MeV/nucleon over the energy range from ≈ 100 to ≈ 1000 MeV/nucleon</td>
<td>30.4 kg</td>
<td>12 W (nominal)</td>
<td>494 bits/sec</td>
<td><a href="http://sd-www.jhuapl.edu/ACE/ULEIS/ace_cris_descri.html">http://sd-www.jhuapl.edu/ACE/ULEIS/ace_cris_descri.html</a></td>
<td></td>
</tr>
<tr>
<td>Solar Isotope Spectrometer (SIS)</td>
<td>solid-state charged particle telescope</td>
<td>designed to provide high-resolution measurements of the isotopic composition of energetic nuclei from He to Z=2 to Z=28 over the energy range from ≈ 100 to ≈ 600 MeV/nucleon</td>
<td>energy and flux</td>
<td>22 kg</td>
<td>18 W</td>
<td>1992 bits/sec</td>
<td><a href="http://www.srl.caltech.edu/ACE/CRIS_SIS/sis.html">http://www.srl.caltech.edu/ACE/CRIS_SIS/sis.html</a></td>
<td></td>
</tr>
<tr>
<td>Magnetometer</td>
<td>twin triaxial fluxgate magnetometer system (on a boom)</td>
<td>measure the local interplanetary magnetic field (IMF) direction and magnitude and establish the large scale structure and fluctuation characteristics of the IMF at 1 AU upstream of Earth as a function of time</td>
<td>IMF direction and magnitude, continuous data at 3.4 to 6 vectors/sec, and snapshot memory data and Fast Fourier Transform data (FFT) based on 24 vectors/sec</td>
<td>Sensors 2): 450 g total, 30 g total, 300 g total, 2100 g total</td>
<td>2.4 watts, electronics regulated 28 Volts, 2% beam, 1.0 watts, beam not regulated 28 Volts</td>
<td><a href="http://www.ssg.sr.unh.edu/mag/ace/instrument.html">http://www.ssg.sr.unh.edu/mag/ace/instrument.html</a></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Solar Energetic Particle Ionic Charge Analyzer (SEPICA)</td>
<td>one high charge resolution sensor section and twolow charge resolution</td>
<td>determine the ionic charge state of solar energetic particles and particles accelerated in interplanetary space. The charge state of energetic particles contains information about the temperature at the source of the particles, as well as related acceleration transport processes.</td>
<td>ionic charge state, Q, the kinetic energy, E, and the nuclear charge, Z, of energetic ions above 0.2 MeV/Nuc</td>
<td>37.4 kg</td>
<td>18.5 W (Nominal)</td>
<td>608 bps</td>
<td><a href="http://www.ssg.sr.unh.edu/td/Missions/Ace/index.html/sepica.html">http://www.ssg.sr.unh.edu/td/Missions/Ace/index.html/sepica.html</a></td>
<td></td>
</tr>
<tr>
<td>Ultra Low Energy Isotope Spectrometer (ULEIS)</td>
<td>Isotope Spectrometer</td>
<td>studying the elemental and isotopic composition of solar energetic particles, and the mechanisms by which these particles are energized in the solar corona. ULEIS will also investigate mechanisms by which supersonic interplanetary shock waves energize ions.</td>
<td>ion fluxes over the charge range from He through Ni from about 20 keV/nucleon to 10 MeV/nucleon, thus covering both suprathermal and energetic particle energy ranges.</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://sd-www.jhuapl.edu/ACE/ULEIS/">http://sd-www.jhuapl.edu/ACE/ULEIS/</a></td>
<td></td>
</tr>
<tr>
<td>Solar Wind Ion Mass Spectrometer (SWIMS)</td>
<td>time-of-flight mass spectrometers with electrostatic analyzers</td>
<td>record the solar wind composition</td>
<td>measures the chemical and isotopic composition of the solar wind for every element between He and Ni, up to 10 keV/e.</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://solar-heliospheric.engin.umich.edu/ace/">http://solar-heliospheric.engin.umich.edu/ace/</a></td>
<td></td>
</tr>
<tr>
<td>Solar Wind Ion Composition Spectrometer (SWICS)</td>
<td>time-of-flight mass spectrometers with electrostatic analyzers</td>
<td>solar wind composition</td>
<td>determines the chemical and isotonic charge state composition of the solar wind and reaches He and H isotopes of both solar and interstellar sources also measures the distribution functions of both the interstellar doul and dust cloud pickup ions up to energies of 100 keV/e.</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://solar-heliospheric.engin.umich.edu/ace/">http://solar-heliospheric.engin.umich.edu/ace/</a></td>
<td></td>
</tr>
<tr>
<td>The Solar Wind Electron, Proton and Alpha Monitor (SWEPAM)</td>
<td>electrostatic energy per charge (E/q) analyzers (one ion one electron) followed by sets of channel electron multiplier (CEM) sensors</td>
<td>solar wind observations that enable the understanding of the ACE composition measurements in the context of previous knowledge about the solar wind</td>
<td>electron sensor measures particle energies between about 0.26 and 36 keV and the electron sensor’s energy range is between 1 and 1.350 eV</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://SWEPAM.lanl.gov/">http://SWEPAM.lanl.gov/</a></td>
<td></td>
</tr>
<tr>
<td>Mission</td>
<td>Instrument</td>
<td>Instrument Type</td>
<td>Purpose</td>
<td>Data Determined</td>
<td>Mass (kg)</td>
<td>Power (W)</td>
<td>Bit rate</td>
<td>SOURCE</td>
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</tr>
<tr>
<td>Electron, Proton and Alpha Monitor (EPAM)</td>
<td>Two Low Energy Foil Spectrometers (LEFS)</td>
<td>telescope apertures with solid state detectors</td>
<td>measure the flux and direction of electrons above 30 keV (geometry factor = 0.397 cm² sr)</td>
<td>energy and composition</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://sd-www.jhuapl.edu/ACE/EPAM/">http://sd-www.jhuapl.edu/ACE/EPAM/</a></td>
</tr>
<tr>
<td></td>
<td>Two Low Energy Magnetic Spectrometers (LEMS)</td>
<td>telescope apertures with solid state detectors</td>
<td>measure the flux and direction of ions greater than 50 keV (geometry factor = 0.48 cm² sr)</td>
<td>energy and composition</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Composition Aperture (CA)</td>
<td>telescope aperture with solid state detector</td>
<td>measure the elemental composition of the ions (geometry factor = 0.24 cm² sr)</td>
<td>energy and composition</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Defense Meteorological Satellite Program</td>
<td>SSH</td>
<td>Precipitating Electron and Ion Spectrometer</td>
<td>Determine complete energy spectrum of low energy particles. Also record high energy ions that penetrate satellite</td>
<td>Flux of electrons and ions from 30 eV to 30 KeV</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td></td>
<td>SSIES</td>
<td>Ion Scintillation Monitor</td>
<td>Measures ambient electron density and temperature, ambient ion density, average ion temperature, molecular weight</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
<td><a href="http://www.ngdc.noaa.gov/dmsp/sensors/ssies.html">http://www.ngdc.noaa.gov/dmsp/sensors/ssies.html</a></td>
</tr>
<tr>
<td></td>
<td></td>
<td>SSM</td>
<td>Magnetometer</td>
<td>Combined with SSIES and SS4 provides heating and electron density profiles</td>
<td>Measures geomagnetic fluctuations</td>
<td>N/A</td>
<td>N/A</td>
<td>N/A</td>
</tr>
</tbody>
</table>
Appendix D

MATLAB Code Used for Orbit Determination

The following code was based off of code written by J.D. Mireles James of the University of Texas-Austin. This code was based off of a personal note set written in the Fall of 2006. The code was modified as much as possible with our orbit parameters to determine necessary information and assist us in determining the solutions to the circular restricted three body problem. These were the main functions used to estimate the MATLAB solutions.

```matlab
function ydot=CRTBP(t,y,options,flag,G,mu)

%the distances
r1=sqrt((mu+y(1))^2+(y(2))^2+(y(3))^2);
r2=sqrt((1-mu-y(1))^2+(y(2))^2+(y(3))^2);

%masses
m1=1-mu;
m2=mu;

ydot=[y(4);
y(5);
y(6);
y(1)+2*y(5)+G*m1*(-mu-y(1))/(r1^3)+G*m2*(1-mu-y(1))/(r2^3);
y(2)-2*y(4)-G*m1*(y(2))/(r1^3)-G*m2*y(2)/(r2^3);
-G*m1*y(3)/(r1^3)-G*m2*y(3)/(r2^3)];

function varargout=librationPoints(mu);

%For a given value of mu, this function computes the location
%of all five libration points for the circular restricted three
%body problem. It returns them as equilibrium points in R^3
%space. Then the output is five points each with three components.
%Each point is a column in a matrix with three rows. The first column is
%L1, the second L2, and so on to the last which is L5.
```

205
% Compute the location of the libration points
% \( l = 1 - \mu \);

% L3
p_L3 = [1, 2*(\mu - l), l^2 - 4*\mu*l + \mu^2, 2*\mu*l*(1-\mu) + (1+\mu),
        \mu^2*l^2 + 2*(\mu^2 - l^2), l^3 + \mu^3];
L3roots = roots(p_L3);
% initialize L3 for loop
L3 = 0;
for i = 1:5
    if L3roots(i) < -\mu
        L3 = L3roots(i);
    end
end

% L1
p_L1 = [1, 2*(\mu - l), l^2 - 4*l*\mu + \mu^2, 2*\mu*l*(1-\mu) - (\mu + l),
        \mu^2*l^2 + 2*(l^2 - \mu^2), \mu^3 - l^3];
L1roots = roots(p_L1);
% initialize L1 for loop
L1 = 0;
for i = 1:5
    if (L1roots(i) > -\mu) & (L1roots(i) < l)
        L1 = L1roots(i);
    end
end

% L2
p_L2 = [1, 2*(\mu - l), l^2 - 4*\mu*l + \mu^2, 2*\mu*l*(1-\mu) - (\mu + l),
        \mu^2*l^2 + 2*(1^2 - \mu^2), -(\mu^3 + l^3)];
L2roots = roots(p_L2);
% initialize L2 for loop
L2 = 0;
for i = 1:5
    if (L2roots(i) > -\mu) & (L2roots(i) > l)
        L2 = L2roots(i);
    end
end

% L4
L4 = [-\mu + 0.5;
      \sqrt{3}/2];
%L5
L5=[-mu+0.5; -sqrt(3)/2];

varargout(1,1)={[L1; 0; 0]};
varargout(2,1)={[L2; 0; 0]};
varargout(3,1)={[L3; 0; 0]};
varargout(4,1)={[L4(1); L4(2); 0]};
varargout(5,1)={[L5(1); L5(2); 0]};

function C=jacobiConst(y,v,mu)

%File computes Jacobi Energy for a given state (x,v)
%the distances
r1=sqrt((mu+y(1,1))^2+(y(2,1))^2+(y(3,1))^2);
r2=sqrt((y(1,1)-(1-mu))^2+(y(2,1))^2+(y(3,1))^2);

%Compute the Jacobi Energy
C=-(v(1,1)^2 + v(2,1)^2+v(3,1)^2)/2 + (y(1,1)^2 + y(2,1)^2)/2 + (1-mu)/r1 + mu/r2;

function vel=magnitudeVelocity(y,C, mu)

%File computes magnitude of velocity for a given state (x,C)
%the distances
r1=sqrt((y(1,1)+mu)^2+(y(2,1))^2+(y(3,1))^2);
r2=sqrt((y(1,1)-(1-mu))^2+(y(2,1))^2+(y(3,1))^2);

%Compute the potential
Potential=(1/2)*(y(1,1)^2+y(2,1)^2)+(1-mu)/r1+mu/r2;

%Compute Magnitude of Velocity
vel=sqrt(2*Potential-C);

function A=stateTransCRTBP(t0, tf, x, mu)

%Compute the state transition matrix at time tf for the
%path x(t) with x(t0)=x0

tspan=[t0,tf];                  %time span over which to
%run the integration

%--------------------------------------------------------------------------
%The state transition matrix is determined by a 6X6 matrix ODE
This gives rise first to a system of 36 ODEs. However the matrix
OEDE is nonautonomous and depends on a particular solution (trajectory)
of the CRTBP. This trajectory is itself the solution of a system
of 6 ODEs. The two systems are solved simultaneously, giving an
autonomous system of 36+6=42 ODEs.

set up the initial condition, y0 for the 42 component system

y0=0;

since the initial condition for the 6X6 matrix system is the 6X6
identity matrix the initial condition vector is very sparse. The first
36 components are 0s and 1s, and the last 6 are the initial conditions
from the CRTBP.

Initialize a 6X6 identity matrix
I=eye(6);

put the entries, row by row, into y0.
for i=1:6
    for j=1:6;
        y(6*(i-1)+j)=I(i,j);
    end
end

the initial conditions for the particular orbit of the CRTBP have
been passed in as 'x'. This to the end of y.

y(37:42)=x’;

Convert to a column vector and pass the initial conditions to the
integrator
y0=y’; %inital condition
options=odeset(‘RelTol’,1e-13,’AbsTol’,1e-22); %set tolerences
[t,Y]=ode113(‘sysSolveCRTBP’,tspan,y0,options,[],mu); %integrate the system

[t,Y] is a huge matrix. It is made up of a row for each time and 42
columns. But the desired data is the state transition matrix at the final
time. Then the first 36 entries of the last row of Y must be put into a
6X6 matrix which will be passed back to the caller.

find out the row number of tf
b=size(Y); %returns the number of rows and columns as a row vector
m=b(1,1); %so ‘m’ is the number of rows
c=Y(m,1:36); %so ‘c’ is a vector containing the first
36 entries of the last row of Y
%now c has to be made into a 12X12 matrix

d=0;
for i=1:6
    for j=1:6
        d(i,j)=c(6*(i-1)+j);
    end
end

%d is the state transition matrix at the final time and is passed back to %the caller
A=d;

function GMatrix=G_CRTBP(x, mu)

%function returns the matrix 'G' for the CRTBP
%with parameter mu.

%the distances
r1=sqrt((x(1)+mu)^2+x(2)^2+x(3)^2);
r2=sqrt((x(1)-(1-mu))^2+x(2)^2+x(3)^2);

%If U is the potential for the CRTBP then this code is computing %the derivative of the gradient of U. The gradient has three %components which we will call u1, u2, u3. The differential of the %gradient is the matrix of partials of these functions. These will %be denoted by u1_x, u1_y, and so forth.

u1_x=1-(1-mu)*(1/(r1^3)-3*((x(1)+mu)^2)/(r1^5))-mu*(1/(r2^3)... -3*((x(1)-(1-mu))^2)/(r2^5));
u2_y=1-(1-mu)*(1/(r1)^3-3*x(2)^2/r1^5)-mu*(1/r2^3-3*x(2)^2/r2^5);
u3_z=(-1)*(1-mu)*(1/(r1)^3-3*x(3)^2/r1^5)-mu*(1/r2^3-3*x(3)^2/r2^5);
u1_y=3*(1-mu)*x(2)*(x(1)+mu)/r1^5+3*mu*x(2)*(x(1)-(1-mu))/r2^5;
u1_z=3*(1-mu)*x(3)*(x(1)+mu)/r1^5+3*mu*x(3)*(x(1)-(1-mu))/r2^5;
u2_z=3*(1-mu)*x(2)*x(3)/r1^5+3*mu*x(2)*x(3)/r2^5;

%equality of mixed partials gives (as all the terms are already partials %of the potential function);
u3_y=u2_z;
u2_x = u1_y;

u3_x = u1_z;

% Then (as mentioned) G is the matrix of partials

GMatrix = [u1_x, u1_y, u1_z;
u2_x, u2_y, u2_z;
u3_x, u3_y, u3_z];

function f = vectorField_CRTBP(y, G, mu)

% the distances
r1 = sqrt((mu+y(1))^2+(y(2))^2+(y(3))^2);
r2 = sqrt((y(1)-(1-mu))^2+(y(2))^2+(y(3))^2);
% masses
m1 = 1-mu;
m2 = mu;

f = [y(4);
y(5);
y(6);
y(1)+2*y(5)+G*m1*(-mu-y(1))/(r1^3)+G*m2*(1-mu-y(1))/(r2^3);
y(2)-2*y(4)-G*m1*(y(2))/(r1^3)-G*m2*y(2)/(r2^3);
-G*m1*y(3)/(r1^3)-G*m2*y(3)/(r2^3)];

function x = CRTBPpoincareNewton(t0, initialPoint, t1, finalPoint, ...
tolerance, G, mu)

%-------------------------------------------------------------------------
%-------------------------------------------------------------------------
% NOTES-------------------------------------------------------------------
%-------------------------------------------------------------------------

% INPUT:
% This function is passed data about two points, one on each side of the
% line y=0. Both are points on a trajectory, and 'initialPoint' is the one
% that occurs first. The time that each point occurs on the trajectory is
% passed into the function, but only the difference will matter. Finally, an
% error tolerance is passed in as well. This tells the function how close a
% result has to be to the x axis so that the Newton method can be considered
% to have converged. Of course the mu and G being used should be passed.

% OUTPUT:
% The purpose of the function is to find a point on the trajectory between
%'initialPoint' and 'finalPoint' which is an intersection with the x axis
%up to the allowed tolerance. This is the 'x' that the function returns.

%HOW IT WORKS:
%As the name suggests the function implements a Newton method to find the
%point. However since we want a new trajectory that begins at say
%'initialPoint' and flows to the x axis, we only want to find the time that
%'initialPoint' must be flowed. The initial guess is t1-t0. Then the
%Newton method is only one dimensional. Since it will only need the
%derivative of the flow with respect to time (and not w.r.t. initial
%conditions) all that is required is an evaluation of the vector field (and
%there is no need to integrate the variational equations).

%A STICKY POINT:
%Of the two points 'initialPoint' and 'finalPoint' one will be closer to
%the x axis than the other (possibly much closer). Then the program will
%pick the point that is closer, and use that as the initial guess for the
%crossing. The subtly is that if 'initialPoint' is closer, then we want
%to examine trajectories beginning from 'finalPoint' and flowing back to
%'initialPoint', as the one we want is one that flows not quite to
%'initialPoint' but intersects the x axis. In this case the equations of
%motion for the CRTBP are not integrated, but instead the equations for the
%reverse time problem. Then the function must code two different Newton
%methods and decide which one to use.

%------------------------------------------------------------------------
%----------------------Computation----------------------------------------
%------------------------------------------------------------------------

%Check while will count the number of times the Newton step executes within
%the algorithm. If the step executes more than say 15 times, then the
%method is probably not going to converge. In that case we do not want to
%be stuck in the Newton loop forever. If 'checkWhile' gets bigger than
%'breakWhile' then the Newton algorithm should break and return some
%default value 'newtonFailed'. If 'newtonFailed' is some value that the
%Poincare map will never otherwise take on, then it gives a good count of
%how many times the algorithm failed to converge.
checkWhile=0;
breakWhile=16;
maxTime=4; %If the Newton algorithm is not
%converging the integration times
%may get longer and longer. However
%under reasonable circumstances the
%initial time guess should be small and get smaller. Then if t_n is getting bigger than say 10, the algorithm is almost definitively diverging. Check for this as well.

newtonFailed=[-mu, 0, 0, 0, 0, 0];  %This is the location of the primary which is a singularity of the system. There is no way an iterate of point could ever reach this value, so if we count the number if times it is returned then we get a good count of how many times the algorithm failed to converge.

Regardless of which point we begin at the time it took to get from one to the other is the same. This is the initial guess for the time from start to intersection.

t_n=t1-t0;

%Determine which point is closest and find the intersection:
if abs(finalPoint(2))<=abs(initialPoint(2))
    %In this case 'finalPoint' is the initial guess as to the location of the crossing. Then we begin with trajectories starting at 'initialPoint' and use the forward time dynamics with a Newton algorithm to refine the guess.

    %The initial guess for the location of the crossing is:
    y_n=finalPoint(2);

    %The Forward Time Newton Algorithm: execute untill tolerance is met while abs(y_n) >= tolerance
    if (checkWhile < breakWhile) & (t_n < maxTime)
        %Integrate. fx_n=f_n(endTime,2) is f(x_n) in the Newton Algorithm
        tspan=[0 t_n];
        options=odeset('RelTol',1e-13,'AbsTol',1e-22);
        [t,f_n]=ode113('CRTBP',tspan,initialPoint,options,[],G,mu);
        %Compute 'endTime':
        sizef_n=size(f_n);
        endTime=sizef_n(1,1);
        %Compute fx_n
        fx_n=f_n(endTime,2);
        %Compute Dfx_n = [f(x_n')]^-1 =f_n(endTime, 5):
        Dfx_n = f_n(endTime, 5);
% Compute the Newton Step
    t_n = t_n - fx_n/Dfx_n;
% This is the new time estimate. Now compute the new crossing
% estimate:
    % First Integrate over new time;
    tspan=[0 t_n];
    options=odeset('RelTol',1e-13,'AbsTol',1e-22);
    [t,f_n]=ode113('CRTBP',tspan,initialPoint,options,[],G,mu);
% Now y_n is the second component of the end of this:
% Reompute 'endTime':
    sizef_n=size(f_n);
    endTime=sizef_n(1,1);
    y_n=f_n(endTime,2);
    crossing_n=f_n(endTime,:);
% Increment 'checkWhile'
    checkWhile=checkWhile+1;
else
    crossing_n=newtonFailed;
    y_n = 0.1*tolerence;
end
end % end of while: the Forward time Newton Algorithm
else
% In this case 'initialPoint' is closer to the x-axis so it is our first
% guess. Then we flow from 'finalPoint' under the backwards time
% dynamics, and use a Newton algorithm to refine the guess.

% The initial guess for the location of the crossing is:
    y_n=initialPoint(2);

% The Backward Time Newton Method: execute until tolerance is met
while abs(y_n) >= tolerance
    if (checkWhile < breakWhile) & (t_n < maxTime)
% Integrate. fx_n=f_n(endTime,2) is f(x_n) in the Newton Algorithm
        tspan=[0 t_n];
        options=odeset('RelTol',1e-13,'AbsTol',1e-22);
        [t,f_n]=ode113('BackwardCRTBP',tspan,finalPoint,options,[],G,mu);
% Compute 'endTime':
        sizef_n=size(f_n);
        endTime=sizef_n(1,1);
% Compute fx_n
        fx_n=f_n(endSize_n,1,1);
% Compute Dfx_n = [f(x_n)']^-1 =f_n(endTime, 5):
        Dfx_n = -f_n(endTime, 5);
% Compute the Newton Step
        t_n = t_n - fx_n/Dfx_n;
% This is the new time estimate. Now compute the new crossing
% estimate:
% First Integrate over new time;
tspan=[0 t_n];
options=odeset('RelTol',1e-13,'AbsTol',1e-22);
[t,f_n]=ode113('BackwardCRTBP',tspan,finalPoint,options,[ ],G,mu);
% Now y_n is the second component of the end of this:
% Reompute 'endTinme':
sizef_n=size(f_n);
endTime=sizef_n(1,1);
y_n=f_n(endTime,2);
crossing_n=f_n(endTime,:);
% increment 'checkWhile'
checkWhile=checkWhile+1;
else
crossing_n=newtonFailed;
y_n=0.1*tolerence;
end % end of check while if conditional
end % end of while: the backward time Newton Algorithm
end % end of the if then conditional

% Return the point of intersection to the caller
x=crossing_n;

% given an initial position of a periodic orbit this program
% computes the stable manifolds of the orbit (assuming it exist...)

% clear the workspace
clear;
% output long
format long;

%--------------------------------------------------------------------------
%----------------------NOTES-----------------------------------------------
%--------------------------------------------------------------------------

% NAME: stableEarthMoonLyap.m

% DEPENDANCIES: The program calls 'CRTBP.m', 'librationPoints.m',
% 'stateTransCRTBP.m' and 'jacobiConst.m'. The program 'stateTransCRTBP
% calls 'sysSolveCRTBP', which calls 'G_CRTBP.m'.

% These must be in the MatLab search path in order for the present
% program to run.

% USE: Once the program and it's dependencies are in the search path simply
% type:
% stableEarthMoonLyap

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% from the matlab The program also keeps all the
data points for the orbit and it’s unstable manifold. This can be
obtained by typing 'who' at the command prompt after the program has
run.command line.

The program can be run with the default data currently set, or the user
can specify the initial condition of an unstable periodic orbit and it’s
period as described below.

OUTPUT: The output is a plot of the Lyapunov orbit, it’s stable manifold
in the vicinity of the moon (secondary body), and the zero velocity curve
at the energy level of the Lyapunov orbit.

PURPOSE: similar to 'unstableEarthMoonLyap.m'. see the notes there

NOTE: The word "halo" appears in several variable names throughout the
program. This is because the program was originally developed to compute
the stable and unstable manifolds of halo orbits. However it is not
necessary that orbit under consideration be a halo orbit. In fact the
default setting of the program are for a Lyapunov orbit near L1.

--------------------------------------USER SPECIFIED--------------------------------------
----------------------------------------PARAMETERS----------------------------------------
--------------------------------------PRIMARY CONTROL VARIABLES--------------------------------------
%how far to move in the unstable direction away from the orbit before
integrating;
epsilon=2*10^(-8);

NOTE: The next data really decides what you are computing. The user
should input 'r0' and 'v0' which are the initial position and velocity
for the periodic orbit they are interested in. The user also inputs the
period of the orbit. The defaults are for a Lyapunov orbit about L1
in the earth/moon system (this problem is planar).

The initial condition for the halo orbit
r0=[ 0.83946302646687;
     0;
     0];
v0=[ 
0; 
-0.02596831282986; 
0]; 

%The period of the halo orbit is known as well (this should not be 
%changed unless the orbit is changed. 
haloPeriod= 2.69239959528586; 

%Parameterize the Manifold: These two variables are the manifold 
%coordinates. How long along the unstable holonomy and from how many 
%and what starting points along the orbit 

%how long to move along the unstable orbit (how local 
%is the computation of the manifold) 
T=3.1*haloPeriod; 

%'k' is how many points to use in order to parameterize the periodic 
%orbit. 
k=40; 

%--------------------------------------------------------------------- 
%--------------------------------------------------------------------- 
%----------------------SECONDARY PARAMETERS-------------------------- 
%--------------------------------------------------------------------- 
%--------------------------------------------------------------------- 

%for the earth moon system the value of mu is approx 0.012277471 
mu=0.012277471; 

%Compute the Jacobi constant for the given initial conditions 
C=jacobiConst(r0,v0,mu) 

%Gravational constant 
G=1; 

%We are given data for the Sun/Earth/Moon system where the sun is the 
%primary, and the earth and moon are combined into one body at their center 
%of mass. The earth moon system is referred to as the earth. Then 

GM_sun=1.327*10^-11; 
GM_earth=4.053*10^-5; 
au=1.496*10^-8; 

%NOTE: The previous parameters are not used in the present version of the
The definition of \( \mu \) gives
\[
\mu = \frac{GM_{\text{earth}}}{GM_{\text{sun}} + GM_{\text{earth}}}
\]

\[
m_1 = 1 - \mu;
\]
\[
m_2 = \mu;
\]

Once 'G' and '\( \mu \) are defined we can compute the location of the libration points and their Jacobi Integrals. These will be handy when we set the value of the Jacobi Integral for the simulation below.

AXULARY CONSTANTS:
Compute the location of the five libration points as points in three space. These can be used to set the value of the Jacobi Constant.
\[
[L_1, L_2, L_3, L_4, L_5] = \text{librationPoints}(\mu);
\]

Output location to the screen (if you want);
\[
\text{outL1} = L_1;
\]
\[
\text{outL2} = L_2;
\]
\[
\text{outL3} = L_3;
\]
\[
\text{outL4} = L_4;
\]
\[
\text{outL5} = L_5;
\]

Now compute the Jacobi Energy at each collinear libration point
\[
\text{The velocity at any equilibrium is zero.}
\]
\[
v_{\text{equil}} = [0; 0; 0];
\]

Compute the energy at each libration point
\[
CL_3 = \text{jacobiConst}(L_3, v_{\text{equil}}, \mu);
\]
\[
CL_1 = \text{jacobiConst}(L_1, v_{\text{equil}}, \mu);
\]
\[
CL_2 = \text{jacobiConst}(L_2, v_{\text{equil}}, \mu);
\]
\[
CL_4 = \text{jacobiConst}(L_4, v_{\text{equil}}, \mu);
\]
\[
CL_5 = CL_4 \quad \text{so no extra computation is needed}
\]

Compute the periodic orbit
t0=0;
numSteps=300;
tspan=linspace(0,haloPeriod+0.1*haloPeriod,numSteps);

% numerical integration
y0=[r0; v0]';

% initial condition
options=odeset('RelTol',1e-13,'AbsTol',1e-22);

[t,x_halo]=ode113('CRTBP', tspan, y0, options, [], G, mu);

% compute the monodromy matrix
Phi=stateTransCRTBP(t0, haloPeriod, [r0; v0], mu);

% compute the eigenvalues and eigenvectors

% Get eigenvalues and eigenvectors
[Phi_eigVecs, Phi_eigValOnDiag]=eigs(Phi);
PhiEigs=diag(Phi_eigValOnDiag);

% NOTE:
% the first eigen value corresponds to the unstable manifold
% the sixth (last) corresponds to the stable direction. This is because
% Matlab organizes the eigenvalues and their eigenvectors by the magnitude
% of the eigen values. Then the first eigen value is the largest and the
% last is the smallest. This assumes that the orbit has two dimensional
% stable and unstable manifolds.

figure
hold on

% get the parameterization points of the periodic orbit
h=round((numSteps-1)/k)
for n=1:k
    haloPoint(n,1:6)=x_halo((n-1)*h+1,1:6);
    haloTimes(n)=t((n-1)*h+1);
end

I=eye(6);

% The un-stable eigenvector is
deltaU=Phi_eigVecs(1:6,6);

% Compute an orbit (fiber of the unstable manifold) on the unstable
% manifold beginning at the origin of the orbit. first the initial
% condition is found:
xU_p(1,1:6)=haloPoint(1,1:6)'+epsilon*I*deltaU;

%compute the stable orbit
t0=0;
numSteps=2000;
tspan=linspace(0, T,numSteps);
%numerical integration
y0=xU_p(1,:);
options=odeset('RelTol',2.5e-13,'AbsTol',1e-22);
[t,xU_p1]=ode113('BackwardCRTBP',tspan,y0,options,[],G,mu);

for n=2:k
    %compute the initial conditions for the manifold orbits. These are
    %found by pushing the vector 'deltaU' around with the state
    %transition matrix. (This is a push forward of deltaU). The
    %justification of this is in the 5th note set.
    initialconditions=n
    perturbationVector=stateTransCRTBP(t0, haloTimes(n), [r0; v0],...
        mu)*deltaU;
    u_pV=perturbationVector/norm(perturbationVector);
    xU_p(n,1:6)=haloPoint(n,1:6)'+epsilon*u_pV;
end

%Compute and plot the unstable manifold

for n=2:k
    manifoldLoop=n
    %the positive ones;
    t0=0;
    numSteps=2000;
tspan=linspace(0, T,numSteps);
    %numerical integration
    y0=xU_p(n,1:6);
    options=odeset('RelTol',2.5e-13,'AbsTol',1e-22);
    [t,xU_pOrb]=ode113('BackwardCRTBP',tspan,y0,options,[],G,mu);

    %plot the unstable pos orbits as red lines
    plot3(xU_pOrb(:,1), xU_pOrb(:,2), xU_pOrb(:,3), 'b')
end
%Contour Plot of the zero velocity surface
[X,Y]=meshgrid(-1.1:0.002:1.2);
r1=((X + mu).^2+Y.^2).^(1/2);
r2=((X + mu-1).^2+Y.^2).^(1/2);
Z=2*[(1/2)*(X.^2 + Y.^2)+(1-mu)./r1+mu./r2];
contour(X,Y,Z,[C, C],’r’)

%plot the libration points
plot3(L3(1), L3(2), L3(3), ’k*’)
plot3(L2(1), L2(2), L2(3), ’k*’)
plot3(L5(1), L5(2), L5(3), ’k*’)

%plot the planets/primaries
plot3(1-mu, 0, 0,’g*’)
plot3(-mu, 0, 0, ’b*’)

%plot the periodic orbit in black dots
plot3(x_halo(:,1), x_halo(:,2), x_halo(:,3), ’k.’)

given an initial position close to a halo orbit this program
applies a Newton’s method to find the orbit (without a stopping time).

clear the workspace
clear;
%output long
format long;

%-----------------------------------NOTES-----------------------------------
%NAME: EarthSunHaloOrbit_NewtonMethod.m
%DEPENDANCIES: The program calls ’CRTBP.m’, ’librationPoints.m’,
’stateTransCRTBP.m’and ’jacobiConst.m’. The program ’stateTransCRTBP
calls ’sysSolveCRTBP’, which calls ’G_CRTBP.m’.

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These must be in the MatLab search path in order for the present program to run.

%USE: Once the program and it’s dependencies are in the search path simply type:
EarthSunHaloOrbit_NewtonMethod
%from the matlab command prompt.

The premise of this program is that we are given initial conditions which are in metric units and which are supposed to be "close" to a sun/earth halo orbit. We convert these to dimensionless initial conditions and proceed.

OUTPUT: The output of the program is two plots. The first plot is a planar ("overhead") view of the earth sun system and the halo orbit. The second plot is the same but is three dimensional (even thought the initial view is still overhead). In both plots the initial guess orbit is shown in black and the orbit to which the Newton Method converges is shown in blue. If the program works correctly the blue orbit is a halo orbit. In both plots the user has to zoom in to see the halo orbit which is about sun/earth L1. (The initial view contains the sun, earth, and L1-L5).

PURPOSE: Given an initial condition in metric units which is close to a sun/earth halo orbit and a good initial guess for it’s period, the program uses a Newton Method to find an actual halo orbit. At present the Newton Method is not controlled by a while loop, but a for loop. The number of iterations of the for loop were selected after some experimentation. A better implementation would have the Newton loop run until some desired tolerance is met and then stop. This is in fact what happens (the tolerance is roughly 10^-9) but the number of iterations has been hard wired. For different initial data, a different number of times might be needed.

-------------------------------DATA-------------------------------

%parameters

%Gravational constant
G=1;

We are given data for the Sun/Earth/Moon system where the sun is the primary, and the earth and moon are combined into one body at their center of mass. The earth moon system is referred to as the earth. Then
\[ GM_{\text{sun}} = 1.327 \times 10^{11}; \]
\[ GM_{\text{earth}} = 4.053 \times 10^{5}; \]

% The definition of \( \mu \) gives
\[ \mu = \frac{GM_{\text{earth}}}{GM_{\text{sun}} + GM_{\text{earth}}} \]

\[ m1 = 1 - \mu; \]
\[ m2 = \mu; \]

\[ au = 1.496 \times 10^8; \]

% initial position guess

% convert physical units to the scaled units of the CRTBP
% The physical coordinates are given w.r.t. the earth, so
% a shift is also needed.
\[ r0 = [1 - \mu; 0; 0] + \left( \frac{1}{au} \right) \times [-1200000; 0; -280000] \]
% The time units in the CRTBP are not seconds, but the
% physical units were given in km/sec. So an additional const
% is needed for the velocity
\[ v0 = \left( \frac{1}{au} \right) \times \left( \frac{60 \times 60 \times 24 \times 365.25}{2 \pi} \right) \times [0; -0.350; 0] \]

\[ v0 = \left( \frac{1}{au} \right) \times \left( \frac{60 \times 60 \times 24 \times 365.25}{2 \pi} \right) \times [0; -0.3275035; 0] \]

\[ x0 = [r0; v0] \]

% initial time guess
\[ t\_initial = 1.45; \]
\[ \% t\_initial = 1.537 \]

% Compute the Jacobi constant for the given initial conditions
\[ C = \text{jacobiConst}(r0, v0, \mu) \]

%--------Libration Points---------------------------------------------

% Once 'G' and 'mu' are defined we can compute the location of the libration
% points and their Jacobi Integrals. These will be handy when we set the
% value of the Jacobi Integral for the simulation below.

% AXULARY CONSTANTS:
% Compute the location of the five libration points as points in three
% space. These can be used to set the value of the Jacobi Constant.
\[ [L1, L2, L3, L4, L5] = \text{librationPoints}(\mu); \]
%Output location to the screen (if you want);
outL1=L1;
outL2=L2;
outL3=L3;
outL4=L4;
outL5=L5;
%Now compute the Jacobi Energy at each collinear libration point

%The velocity at any equilibrium is zero.
v_equil=[0; 0; 0];

%Compute the energy at each libration point
CL3=jacobiConst(L3, v_equil, mu);
CL1=jacobiConst(L1, v_equil, mu);
CL2=jacobiConst(L2, v_equil, mu);
CL4=jacobiConst(L4, v_equil, mu);
CL5=CL4 %so no extra computation is needed

%------------------------------------------------------------------------
%Aux Matrices
%------------------------------------------------------------------------
O=zeros(3);
I=eye(3);
K=[0, 2, 0;
   -2, 0, 0;
   0, 0, 0];

%------------------------------------------------------------------------
%----------------------Newton Method; find orbits--------------------------
%------------------------------------------------------------------------

%initial guess
x_n=x0;
tau_n=t_initial
tau=tau_n;

K=14;
for i = 1:K
    out_i=i
    %We need some information from the 'reference trajectory'.
t0=0;
tf=tau_n;
numSteps=2000;
tspan=linspace(0, tf, numSteps);
%numerical integration
\[y_0 = x_n';\] %initial condition
options=odeset('RelTol',2.5e-14,'AbsTol',1e-22); %set tolerances
[t,x_ref]=ode113('CRTBP',tspan,y0,options,[],G,mu); %Start computing the differential of the Newton condition vector
%Need the STM for nth guess at nth time (tau_n is n+1 th time)
Phi=stateTransCRTBP(t0, tf, x_n, mu); %some of the terms depend on the vector field
f_x=vectorField_CRTBP(x_ref(numSteps,:),G,mu); %Construct the needed differential
a11=Phi(4,3);
a12=Phi(4,5);
a21=Phi(6,3);
a22=Phi(6,5);
a31=Phi(2,3);
a32=Phi(2,5);
%the differential of the constraint vector
DF=[a11, a12, f_x(4);
a21, a22, f_x(6);
a31, a32, f_x(2)]; %Invert
D=inv(DF);
%Compute next (or final/target) initial conditions
%compute next step
Term_1 = [x_n(3); x_n(5); tau_n];
Term_2 = -D*[x_ref(numSteps, 4); x_ref(numSteps, 6); x_ref(numSteps, 2)];
x_star= Term_1 + Term_2; %set up x_n for next iteration or output
%Put it all into x_n (which is really x_(n+1) as it occurs at the end
%if the loop
x_n=[r0(1);
  0;
  x_star(1);
  0;
  x_star(2);
  0];
tau_n=x_star(3);
end

outX=x_n
outTime=tau_n

CTargetOrbit=jacobiConst(x_n(1:3), x_n(4:6), mu)
deltaInitialEnergy=abs(C-CTargetOrbit)

\%--------------------------------HALO ORBIT------------------------------------------
% Now compute the whole orbit using symmetry
 t0=0;
 haloPeriod=2*tau_n
 numSteps=2000;
 tspan=linspace(0,haloPeriod,numSteps);
 % numerical integration
 y0=x_n';                     % initial condition
 options=odeset('RelTol',1e-12,'AbsTol',1e-12);    % set tolerances
 [t,x_halo]=ode113('CRTBP',tspan,y0,options,[],G,mu);

% Some analysis of the orbit;

% check the order of magnitude to which the orbit is periodic
periodicityCheck=norm(x_halo(1,1:3)-x_halo(numSteps,1:3))

% Compute the monodromy matrix for the periodic orbit
Phi=stateTransCRTBP(t0, haloPeriod, x_n, mu);

% determinant of Phi;
testDetPhi=det(Phi)

% Get eigenvalues and eigenvectors
[Phi_eigVecs, Phi_eigValOnDiag]=eigs(Phi);

PhiEigenValues=diag(Phi_eigValOnDiag)

% Simulate the original conditions
 tf=2*tf;
 numSteps=2000;
 tspan=linspace(0,tf,numSteps);
 % numerical integration
 y0=x0';                     % initial condition
 options=odeset('RelTol',1e-12,'AbsTol',1e-22);    % set tolerances
 [t,x]=ode113('CRTBP',tspan,y0,options,[],G,mu);

%-----------------------------2d plotting-----------------------------------
figure
hold on
% Contour Plot of the Energy Surface
 [X,Y]=meshgrid(-1.1:0.002:1.2);
 r1=($(X + mu)^2+Y^2)^{1/2}$;
 r2=($(X + mu-1)^2+Y^2)^{1/2}$;
\[ Z = 2\left[\frac{1}{2}(X.\times X + Y.\times Y) + \frac{1}{r_1} + \frac{\mu}{r_2}\right] \]

contour(X, Y, Z, [CTargetOrbit, CTargetOrbit], 'r')

% plot the initial trajectory
plot(x(:,1), x(:,2), 'k')

% plot the halo orbit
plot(x_halo(:,1), x_halo(:,2), 'b')

% plot the libration points
plot(L3, 0, 'ko', L1, 0, 'ko', L2, 0, 'ko')
plot(L4(1,1), L4(2,1), 'ko', L5(1,1), L5(2,1), 'ko')

% plot the planets (primaries)
plot(-mu, 0, 'r*', 1-mu, 0, 'g*')

%-------3d plotting-------------------------------------------

figure
hold on

% plot the libration points
plot3(L3, 0, 0, 'ko', L1, 0, 0, 'ko', L2, 0, 0, 'ko')
plot3(L4(1,1), L4(2,1), 0, 'ko', L5(1,1), L5(2,1), 0, 'ko')

% plot the planets (primaries)
plot3(-mu, 0, 0, 'r*', 1-mu, 0, 0, 'g*')

% plot the initial trajectory
plot3(x(:,1), x(:,2), x(:,3), 'k')

% TEST: Plot the stopping point of x_halo
% plot3(x_halo(1,1), x_halo(1,2), x_halo(1,3), 'r*')
% plot3(x_halo(numSteps,1), x_halo(numSteps,2), x_halo(numSteps,3), 'r*')

% plot the halo trajectory
plot3(x_halo(:,1), x_halo(:,2), x_halo(:,3), 'b')

axis([-1.2 1.2 -1.2 1.2 -0.006 0.006])
Appendix E

Data Budget
<table>
<thead>
<tr>
<th>Component</th>
<th>Measurement</th>
<th>Data Rate</th>
<th>Accuracy (Bit)</th>
<th>Data Range</th>
<th>Accuracy</th>
<th>Repetition</th>
<th>Sampling Rate</th>
<th>Storage Rate</th>
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<td>-40 to 40 Watts</td>
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<td>10</td>
<td>320</td>
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<td>10</td>
<td>32</td>
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<td><strong>Power</strong></td>
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<td>-40 to 40 Watts</td>
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<td>192</td>
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<td>Thrusters</td>
<td>Digital State</td>
<td>1</td>
<td>1</td>
<td>On/Off</td>
<td>1</td>
<td>12</td>
<td>10</td>
<td>120</td>
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<td>Radiation</td>
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<td>-40 to 65 Degrees Celsius</td>
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<td>10</td>
<td>320</td>
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<td>Temperatures</td>
<td>Analog State</td>
<td>32</td>
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<td>-40 to 65 Degrees Celsius</td>
<td>5</td>
<td>1</td>
<td>160</td>
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<tr>
<td><strong>GN&amp;C/COM</strong></td>
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<td></td>
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<tr>
<td>Star Tracker</td>
<td>Digital State</td>
<td>32</td>
<td>4294967295</td>
<td>0 to 360 Degrees</td>
<td>3</td>
<td>10</td>
<td>900</td>
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<td>Sun Sensor</td>
<td>Analog State</td>
<td>32</td>
<td>4294967295</td>
<td>0 to 128 Degrees</td>
<td>2</td>
<td>10</td>
<td>940</td>
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<tr>
<td>Antenna/Simball</td>
<td>Digital State</td>
<td>32</td>
<td>4294967295</td>
<td>+/- 110 and +/- 270</td>
<td>2</td>
<td>10</td>
<td>940</td>
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<tr>
<td>Reaction Wheel</td>
<td>Analog State</td>
<td>32</td>
<td>4294967295</td>
<td>0 to 5000 rpm</td>
<td>4</td>
<td>10</td>
<td>1290</td>
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<tr>
<td>Clock</td>
<td>Digital State</td>
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<td>4294967295</td>
<td>0</td>
<td>1</td>
<td>10</td>
<td>320</td>
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<tr>
<td><strong>Payload</strong></td>
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<td>In-Situ CME Sensor</td>
<td>Analog State</td>
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</table>

**Total:** 2060

Maximum Data Storage (bits/day): 611,386,400

Maximum Allowed Data Transmission (Mbits/day): 72.88

Overhead: 36.44

Total: 109.32
Appendix F

Link Budget
<table>
<thead>
<tr>
<th>Item</th>
<th>Symbol</th>
<th>Units</th>
<th>Source</th>
<th>Spacecraft to Ground</th>
</tr>
</thead>
<tbody>
<tr>
<td>Frequency</td>
<td>$f$</td>
<td>GHz</td>
<td>Input Parameter</td>
<td>2.2</td>
</tr>
<tr>
<td>Transmitter Power (RF)</td>
<td>$P$</td>
<td>Watts</td>
<td>$P*\eta_p$</td>
<td>5</td>
</tr>
<tr>
<td>Transmitter Power (RF)</td>
<td>$P$</td>
<td>dBW</td>
<td>10 log(P)</td>
<td>6.99</td>
</tr>
<tr>
<td>Transmitter Line Loss</td>
<td>$L_t$</td>
<td>dB</td>
<td>Input Parameter</td>
<td>-1</td>
</tr>
<tr>
<td>Transmit Antenna Beamwidth</td>
<td>$\theta_t$</td>
<td>deg</td>
<td>SMAD</td>
<td>8.678</td>
</tr>
<tr>
<td>Transmit Antenna efficiency</td>
<td>$h_t$</td>
<td>–</td>
<td>Input Parameter</td>
<td>0.55</td>
</tr>
<tr>
<td>Peak Transmit Antenna Gain</td>
<td>$G_{pt}$</td>
<td>dBi</td>
<td>SMAD</td>
<td>25.49</td>
</tr>
<tr>
<td>Transmit Antenna Diameter</td>
<td>$D_t$</td>
<td>m</td>
<td>Input Parameter</td>
<td>1.1</td>
</tr>
<tr>
<td>Transmit Antenna Pointing Error</td>
<td>$e_t$</td>
<td>deg</td>
<td>Input Parameter</td>
<td>1</td>
</tr>
<tr>
<td>Transmit Antenna Pointing Loss</td>
<td>$L_{pt}$</td>
<td>dB</td>
<td>SMAD</td>
<td>-0.159</td>
</tr>
<tr>
<td>Transmit Antenna Gain (net)</td>
<td>$G_t$</td>
<td>dBi</td>
<td>$G_{pt} + L_{pt}$</td>
<td>25.33</td>
</tr>
<tr>
<td>Equiv. Isotropic Radiated Power</td>
<td>$E_{IRP}$</td>
<td>dBW</td>
<td>$P + L_t + G_t$</td>
<td>31.32</td>
</tr>
<tr>
<td>Propagation &amp; Polarization Loss</td>
<td>$L_a$</td>
<td>dB</td>
<td>Fig. 13-10</td>
<td>0</td>
</tr>
<tr>
<td>Receive Antenna Diameter</td>
<td>$D_r$</td>
<td>m</td>
<td>Input Parameter</td>
<td>13</td>
</tr>
<tr>
<td>Receive Antenna Efficiency</td>
<td>$h_r$</td>
<td>–</td>
<td>Input Parameter</td>
<td>0.55</td>
</tr>
<tr>
<td>Peak Receive Antenna Gain</td>
<td>$G_{rp}$</td>
<td>dBi</td>
<td>SMAD</td>
<td>46.94</td>
</tr>
<tr>
<td>Receive Antenna Beamwidth</td>
<td>$\theta_r$</td>
<td>deg</td>
<td>SMAD</td>
<td>0.734</td>
</tr>
<tr>
<td>Receive Antenna Pointing Error</td>
<td>$e_r$</td>
<td>deg</td>
<td>Input Parameter</td>
<td>0</td>
</tr>
<tr>
<td>Receive Antenna Pointing Loss</td>
<td>$L_{pr}$</td>
<td>dB</td>
<td>SMAD</td>
<td>0</td>
</tr>
<tr>
<td>Receive Antenna Gain (net)</td>
<td>$G_r$</td>
<td>dBi</td>
<td>$G_{rp} + L_{pr}$</td>
<td>46.94</td>
</tr>
<tr>
<td>System Noise Temperature</td>
<td>$T_s$</td>
<td>K</td>
<td>SMAD</td>
<td>135</td>
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<tr>
<td>Effective Data Rate</td>
<td>$R$</td>
<td>bps</td>
<td>Input parameter</td>
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<tr>
<td>Eb/No (1)</td>
<td>$E_b/N_o$</td>
<td>dB</td>
<td>SMAD</td>
<td>14.81</td>
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<tr>
<td>Carrier-to-Noise Density Ratio</td>
<td>$C/N_o$</td>
<td>dB-Hz</td>
<td>SMAD</td>
<td>63.56</td>
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<tr>
<td>Bit Error Rate</td>
<td>BER</td>
<td>–</td>
<td>Input Parameter</td>
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<tr>
<td>Required Eb/No (2)</td>
<td>Req $E_b/N_o$</td>
<td>dB</td>
<td>SMAD Fig. 13-9</td>
<td>9.6</td>
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<tr>
<td>Implementation Loss (3)</td>
<td>—</td>
<td>dB</td>
<td>Input Parameter</td>
<td>-2</td>
</tr>
<tr>
<td>Rain Attenuation (4)</td>
<td>—</td>
<td>dB</td>
<td>SMAD</td>
<td>0</td>
</tr>
<tr>
<td>Margin</td>
<td>—</td>
<td>dB</td>
<td>(1)$(\bar{U}(2)+(3)+(4))$</td>
<td>3.207</td>
</tr>
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</table>

Table F.1: Link budget for HGA to Earth ground station
<table>
<thead>
<tr>
<th>Item</th>
<th>Symbol</th>
<th>Units</th>
<th>Source</th>
<th>Spacecraft to Ground</th>
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</thead>
<tbody>
<tr>
<td>Frequency</td>
<td>$f$</td>
<td>GHz</td>
<td>Input Parameter</td>
<td>2.2</td>
</tr>
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<td>Transmitter Power (RF)</td>
<td>$P$</td>
<td>Watts</td>
<td>$\eta_p$</td>
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</tr>
<tr>
<td>Transmitter Power (RF)</td>
<td>$P$</td>
<td>dBW</td>
<td>$10 \log(P)$</td>
<td>6.99</td>
</tr>
<tr>
<td>Transmitter Line Loss</td>
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<td>dB</td>
<td>Input Parameter</td>
<td>-2</td>
</tr>
<tr>
<td>Transmit Antenna efficiency</td>
<td>$\eta_t$</td>
<td>–</td>
<td>Input Parameter</td>
<td>0.55</td>
</tr>
<tr>
<td>Peak Transmit Antenna Gain</td>
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<td>dBi</td>
<td>SMAD</td>
<td>6</td>
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<td>Transmitter Line Loss</td>
<td>$L_t$</td>
<td>dB</td>
<td>Input Parameter</td>
<td>1.1</td>
</tr>
<tr>
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<td>dB</td>
<td>SMAD</td>
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</tr>
<tr>
<td>Transmit Isotropic Radiated Power</td>
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<td>dBW</td>
<td>$P + L_t + G_t$</td>
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<td>Transmitter Gain (net)</td>
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<td>Propagation Path Length</td>
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<td>dB</td>
<td>SMAD</td>
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<td>–</td>
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<td>dBi</td>
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<td>K</td>
<td>SMAD</td>
<td>135</td>
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<td>Effective Data Rate</td>
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<tr>
<td>Required $E_b/N_o$ (2)</td>
<td>$\text{Req } E_b/N_o$</td>
<td>dB</td>
<td>SMAD Fig. 13-9</td>
<td>9.6</td>
</tr>
<tr>
<td>Implementation Loss (3)</td>
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<td>dB</td>
<td>Input Parameter</td>
<td>-1</td>
</tr>
<tr>
<td>Rain Attenuation (4)</td>
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<td>0</td>
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<td>Margin</td>
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<td>(1)Ú(2)+(3)+(4)</td>
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Table E2: Low gain patch antenna to earth ground station link budget
Appendix G

Communications
S - Band Radios

The AeroAstro Modular S-Band Radio System is a miniature software/FPGA based radio providing full NASA/DSN, ESA and AFSCN interoperability, enabling a new low-cost generation of space missions.

Three 3.5” x 2.0” x 1.1” modules combine to create a powerful 5W RF transceiver and/or coherent transponder. The modular approach supports distributed placement in the small recesses of a spacecraft or UAV. With a total mass of less than 1 Kg and a volume of just over 23 in³, the AeroAstro Modular S-Band radio provides the designer more useful payload while reducing costs.

The radio modules interconnect via an EIA-485 network which can also interface with other future radio products to add flexibility and capability.

- SGLS, STDN and CCSDS variants
- Interface to MCU - 110 crypto unit
- RS-422, EIA-485 & custom I/F
- PRN ranging / coherency supported
- Telemetry uplink at 1, 2 or 10kbps
- Downlink rates to 25 Mbps
- Receiver available for stand-alone use

For more information, please contact AeroAstro at info@aeroastro.com and sales@aeroastro.com or 703-723-9800.
## Component Specifications

### S-Band Radios

<table>
<thead>
<tr>
<th>Specification</th>
<th>Details</th>
</tr>
</thead>
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<td>Flight Heritage</td>
<td>ALEXIS, HETE, MOST, NASA NM/ST - 5</td>
</tr>
<tr>
<td>Input Voltage</td>
<td>15 - 50Vdc continuous</td>
</tr>
<tr>
<td>Reverse Voltage</td>
<td>Up to -50V continuous</td>
</tr>
<tr>
<td>Output Protection</td>
<td>No damage; open or short circuit</td>
</tr>
<tr>
<td>Thermal Monitoring</td>
<td>Individual sensors and reporting from each of the three modules.</td>
</tr>
<tr>
<td>RF Input Dynamic Range</td>
<td>-130dBm to -40dBm</td>
</tr>
<tr>
<td>RX Carrier Tracking Range</td>
<td>±105kHz</td>
</tr>
<tr>
<td>RX Carrier Acquisition Threshold</td>
<td>-119dBm</td>
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<tr>
<td>RX Noise Figure</td>
<td>4dB</td>
</tr>
<tr>
<td>RX Carrier Acquisition Time</td>
<td>&lt;0.5 seconds</td>
</tr>
<tr>
<td>TX Frequency Stability</td>
<td>±20ppm over temperature</td>
</tr>
<tr>
<td>Output Power</td>
<td>Adjustable in 0.5W steps from 0.5W to 5W RF under software control</td>
</tr>
<tr>
<td>Ranging</td>
<td>B/W: 100 Hz to 1MHz (-3dB) \ Turnaround UMI: 1:1 (±10%)</td>
</tr>
<tr>
<td>Uplink Modulation Index</td>
<td>0.3Rad peak (nom.)</td>
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<tr>
<td>Interface</td>
<td>RS-422/EIA-485 software command I/F</td>
</tr>
<tr>
<td>Operating Temperature</td>
<td>-20°C - +60°C</td>
</tr>
<tr>
<td>Radiation Tolerance</td>
<td>10kRads(Si) - box level (higher levels available with shielding)</td>
</tr>
<tr>
<td>Latch-up</td>
<td>Detection and Mitigation (2μ sec response, 200m sec reset)</td>
</tr>
<tr>
<td>Dimensions</td>
<td>three modules - each 3.5” x 2.0” x 1.1” (8.9cm x 5.1cm x 2.6cm)</td>
</tr>
<tr>
<td>Mass</td>
<td>&lt;900g (total for 3 modules)</td>
</tr>
</tbody>
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---

**About AeroAstro**

AeroAstro, founded in 1980, is a leader in innovative microsatellite systems, components, and advanced communications technologies. AeroAstro was the prime contractor for STPSat-1 responsible for spacecraft design and fabrication, integration of all experiments, space vehicle testing, launch integration support, launch and early orbit operations support, and post-launch mission operations support.
Microstrip Patch Antennas

Antenna Development Corporation, Inc. (AntDevCo) employees have designed and manufactured spacecraft microstrip patch antennas for many small spacecraft programs. These antennas are capable of supporting high data rates and up to 10 Watts of transmitted power. Applications include GPS, USAF SGLS, NASA SN (Including TDRSS forward/return pairs), radar transponder, and the NASA DSN. The antennas can be supplied with LHCP, RHCP, or linear polarization with single frequency or dual frequency operation in stacked or side-by-side configurations. They are supplied in two standard form factors. Low frequency units are 4 X 4 inches (100 X 100 mm metric option). High frequency units are 2 X 2 inches (50 X 50 mm metric option). The antenna thickness depends on the polarization and number of frequencies.

All antennas are supplied with extensive testing data including principal plane radiation pattern plots, gain bounds plots, and coverage statistics. Simulations of the expected performance on your satellite can also be supplied.

The antennas may also be ordered with semi-conductive radomes for satellite applications where no exposed dielectrics are allowed.

- Space qualified
- Conformal form factor
- Low mass
- High Performance

Side-by-Side Single Frequency Patch Antenna (4 X 8 inches)

Specifications

- Gain: 6 dB nominal
- Frequency: L band, S-band, C-band, & X-band. Single and dual frequency models.
- Bandwidth: 40 MHz nominal
- HPBW: 70 degrees full width
- Impedance: 50 Ohms
- Polarization: Linear or Circular
- VSWR: < 1.3
- Axial Ratio: < 4 dB
- Connector: SMA Female
- Dimensions: 4” X 4” standard (4” X 8” pictured)
- Mass: < 80 grams
- Temperature: -100 C to +100 C
- Power: up to 10 Watts CW
- IR Emissivity = 0.90
  Absorptivity = 0.43

AntDevCo supports and is working to become ISO 9000 compliant.

www.AntDevCo.com (575) 541-9319
BBlevins@AntDevCo.com (575) 635-3528
TFgreening@AntDevCo.com (575) 644-1527
Appendix H

Power Generation MATLAB Code

clear
clc

% Written by Mark Pokora, Gerasimos Houpis, Alex Gorodetsky

%------------------- Total PowerCalc -------------------------------------
% Assuming Payload L1
Payload_P = 20;

% Attitude Control
ACDS_P = 20;  % at most 2 are on

% C&DH recent
CDH_P = 50.4;

% Propulsion recent
Prop_P = 70;  % at 28 volts  % prop power is inst and can be provided by batteries

% Communications
Comm_P = 25;

% Thermal/ Structures recent
Therm_P = 4;

% Power System recent
Power_P = 20;

TotalPower = 132.44;  % Update with most recent number from budget

%------------------------------------------------------------------------
All_cell_components_mass=zeros(1,100);
Cell_Panels_Area = zeros(1,100);
Cell_Panels_Mass = zeros(1,100);
TEGtotalPower = zeros(1,100);
TEGcells = zeros(1,100);
TEGcellMass = zeros(1,100);
TEGpanelCost = zeros(1,100);
TEGcellsArea = zeros(1,100);

%-- Power ratio ---
for i = 1:1:200

%-------------Solar Panel Array Area & Mass Calc -------------
%Incidence Angle
theta = 10/180*pi;

%Efficiency for Multi-Junction GaAs
eff = .28;

%Inherent Degradation
I = .77;

%deg per year
deg = 0.005;

%Satellite Lifetime (years)
life = 8;

%Solar intensity W/m^-2
solar_inten = 1367;

%Path efficiency for Direct Energy transfer
patheff = .85;

% Power change w/ Temperature change (starting at 75 deg C)
temp = 128.77; % C
% BOL
Jmp_BOL = 5*10000*(10^-6) ;
Vmp_BOL = -0.0067 ;
Therm_power_adjust_BOL = Jmp_BOL*Vmp_BOL*(temp-75);
% adjust to deg/meter^-2
Therm_power_adjust_BOL = Therm_power_adjust_BOL;

% Solar Cell Thermal Emissivity
therm_E=0.85;
% TEG Thermal Paint Emissivity
TEGtherm_E=1;
% Solar Cell Thermal Absorbtivity  
therm_A=0.92;  
% TEG thermal Paint Absorbtivity  
TEGtherm_A=10;  

%Weight/Area of cells (kg/m^2)  
%weight_area = .84;  
Cell_weight_area = 2.06;  

%Total Power Needed from Cell  
CellTotalPower(i) = (i/100)*(TotalPower);  
SolarPanelPower(i) = CellTotalPower(i)/patheff;  

%EOL and BOL Power are units Watts/m^2  
BOL_P = (eff*solar_inten*I*cos(theta))+(Therm_power_adjust_BOL);  
LD = (1-deg)^life;  
EOL_P = BOL_P*LD;  
Cell_Panel_Area(i) = SolarPanelPower(i)/EOL_P;  
Cell_Panel_Area_cost(i)= (BOL_P*Cell_Panel_Area(i))*400;  
Cell_Panel_Mass(i) = Cell_Panel_Area(i)*Cell_weight_area;  

%------------ TEG panel area & mass calc--------------------------  

TEGtotalPower(i) = (1-(i/100))*TotalPower;  

% Number of TEG cells needed  
TEGcells(i) = ceil(TEGtotalPower(i)/19);  

TEGcellsArea(i) = TEGcells(i)*0.005625;  
TEGcellMass = 5*0.115;  
TEGpanelCost(i) = TEGcells(i)*209;  

%---------TEG Heat Calculator. Use separately, turn off when not in use -----
% Heat gained through walls of TEG  
%TEG_area= 0.005625;  
%Therm_k= 2.4;  
%Thick= 0.00508;  
%Temp_diff=TEG_hot_side-200;  
%Q=(TEG_area*Temp_diff*Therm_k)/(Thick);  
%New_Temp_diff=((Thick*Q)/(Therm_k*TEG_area));  
%TEG_cold_side=New_Temp_diff;
% Totals and summations

All_cell_components_mass(i) = TEGcellMass + Cell_Panel_Mass(i);
All_cell_components_area(i) = Cell_Panel_Area(i) + TEGcellsArea(i);
end

[val, ind] = min(All_cell_components_area);

disp(['The lowest possible primary cell area to meet power requirements is...
', num2str(val), ' m^2, with the following properties: ']);
disp([' ', num2str(ind), '% of the total power comes from solar cells an...
d', num2str(100-ind), '% comes from TEG cells']);
disp([' Solar cell panel array size: ', num2str(Cell_Panel_Area(ind)), ' ... square meters']);
disp([' Number of TEG cells underneath: ', num2str(TEGcells(ind))]);
disp([' Cost of TEG cells: $', num2str(TEGpanelCost(ind))]);
disp([' TEG cell panel array size: ', num2str(TEGcellsArea(ind)), ' square...e meters']);
% disp([' TEG cell hot side temperature: ',num2str(TEG_hot_side),' K']);
% disp([' TEG cell cold side temperature: ',num2str(TEG_cold_side),' K']);

[val, ind] == min(All_cell_components_mass);
disp(['With num2str(ind) solar cells to meet power requirements, the minim...um mass is ', num2str(val), ' kg, with the following properties: ']);
disp([' ', num2str(ind), '% of the total power comes from solar cells an...
d', num2str(ind), '% comes from TEG cells']);
disp([' Solar cell panel array size: ', num2str(Cell_Panel_Area(ind)), ' ... square meters']);
disp([' Number of TEG cells underneath: ', num2str(TEGcells(ind))]);
disp([' Cost of TEG cells: $', num2str(TEGpanelCost(ind))]);
disp([' TEG cell panel array size: ',num2str(TEGcellsArea(ind)), ' square...meters']);

% For the scenario with 5 permanent TEG's

dot=104;  % User Entry number to manually change % of solar power of total ...
power
val=All_cell_components_area(dot);
With % of TOTAL minimum power coming from solar cells, solar/TEG primary cell area is , m^2, with the following properties: ’

% of the NEEDED minimum power comes from solar cells and % comes from TEG cells

Solar cell panel array size: square meters

Number of TEG cells: 

Cost of TEG cells: $

TEG cell panel array size: square meters

Cost of Solar cells: $

Total array size: square meters

Total power at EOL , watts

Total mass kg

%--------- In the event that we need to build our own substrate --------

Substrate Mass density of Ge (germanium)

Substrate_mass = ge_dens*ge_vol;
Total_Array_Mass = substrate_mass + CellPanelMass

%loop for graphs
%for ii = 1: 1000
  Payload_P = ii;
  ACDS_P = .15*Payload_P/.4;
  Comm_P = .05*Payload_P/.4;
  CDH_P = .05*Payload_P/.4;
  Therm_P = .05*Payload_P/.4;
  Power_P = .3*Payload_P/.4;
  TotalPower = Payload_P + ACDS_P + Comm_P + CDH_P + Therm_P + Power_P;
  SolarPanelPower = TotalPower/patheff;
  ArrayArea = SolarPanelPower/EOL_P;

%figure(1)
%hold on

240
plot(Payload_P,ArrayArea)
xlabel('Payload Power (W)')
ylabel('Solar Panel Area (m^2)')

% Solar panel dimensions
% assuming 3 panels
NumberOfPanels = 3;
PanelArea = Cell_Panel_Area / NumberOfPanels;
PanelWidth = 0.7; % set by Structures
PanelLength = PanelArea / PanelWidth;

%--- Sunshade Dimensions
% regular hexagon S/C base with t=0.7m set by structures
t=0.7;
HexagonBaseArea = 3*sqrt(3)/2*t.^2;
BigDiameter = 2*t;
SmallDiameter = t * sqrt(3);
HalfSmallDiameter = SmallDiameter/2;
radius = sqrt((PanelLength+HalfSmallDiameter).^2 +(PanelWidth/2).^2);
SunshadeArea = pi*radius.^2;

%---- Support Structures
% Support Beams
NumberOfBeams = NumberOfPanels*2;
BeamLength = sqrt( PanelLength.^2 + PanelLength);
BeamVolume = 0.02 * 0.02 * BeamLength; % 2cm is chosen "randomly"
TotalBeamVolume = NumberOfBeams*BeamVolume;
TotalBeamMass = TotalBeamVolume * 2700; % 2700 density of Aluminum

% Panel Support
PanelSupportThickness = 0.05; % 5cm is chosen "randomly"
PanelSupportVolume = Cell_Panel_Area*PanelSupportThickness;
PanelSupportMass = PanelSupportVolume * 19; % 19kg/m3 density of al honeycomb

TotalPanelSupportMass = PanelSupportMass + PanelSupportMass*0.10 ;% 10 for C... RP sheets

% Shield
ShieldArea = SunshadeArea - Cell_Panel_Area - HexagonBaseArea;
ShieldVolume = ShieldArea * 0.01 ;% 0.01 chosen randomly
ShieldMass = ShieldVolume * 1400 ;% 1400 density of vectran
% Sunshade design total mass
Springs = 1; % support structure parts, randomly chosen
SunshadeMass = Cell_Panel_Mass + TotalBeamMass + TotalPanelSupportMass + Sp... rings + ShieldMass;
TotalSunshadeMass = SunshadeMass + SunshadeMass*0.10;
%----------------------------------------------------------------------
--------------
Appendix I

Parametric Cost Estimation

The CER for estimating the cost of the Theoretical First Unit (TFU) is found from Table 20-5 in SMAD [102].

\[
TFU\ Cost\ (FY'00\$K) = 112 \times X^{0.763} \tag{I.1}
\]

Where X is the EPS mass in kg. Using an EPS mass of 9.3 kg, the TFU cost is $614,000. The SMAD states that actual costs may be 25% below or above the cost predicted by the CER. Since the HAVEN project utilizes off the shelf components for its power sub-system, the cost will be below the cost predicted by the CER. Adjust for the 25% discrepancy, the revised cost is $460,500. We then account for inflation using the inflation factor from Table 20-1 in SMAD for the 2009 fiscal year.

\[
TFU\ Cost\ (FY'09\$K) = TFU\ Cost\ (FY'00\$K) \times 1.19 \tag{I.2}
\]

The present day cost of the theoretical first unit $547,995. Final adjustments to the costs are made by adjusting for the learning curve since all of the major sub-system components have a flight heritage of at least 20 missions. Taking into account the learning curve, production cost is calculated using the following method:

\[
\text{Production Cost Per Unit} = \text{TFU} \times \frac{L}{N} \tag{I.3}
\]

Where \( L = NB \), N is the number of units produced, \( B = 1 - \frac{\ln(100/S)}{\ln(2)} \), S = 95 [SMAD]

Using these equations, the production cost per unit is $439,035. The cost of the Saft battery unit, solar panels and TEGs has been determined from individual manufacturers. To estimate the cost of the Clyde Space SmallSat power system, the costs of the known components were subtracted from the total system cost of $439,035. Individual breakdowns are available in Section B.3.
## Appendix J

### Analogy Based Cost Estimation

<table>
<thead>
<tr>
<th>Component</th>
<th>Cost</th>
<th>Quantity</th>
<th>Subtotal</th>
<th>Contingency (%)</th>
<th>Total</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Panels</td>
<td>64000</td>
<td>1</td>
<td>64000</td>
<td>0.05</td>
<td>67200</td>
</tr>
<tr>
<td>TEG</td>
<td>200</td>
<td>5</td>
<td>1000</td>
<td>0.05</td>
<td>1050</td>
</tr>
<tr>
<td>Battery Control Charger</td>
<td>35000</td>
<td>5</td>
<td>175000</td>
<td>0.1</td>
<td>192500</td>
</tr>
<tr>
<td>+ Telemetry Board</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>+ Voltage Regulator</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Battery</td>
<td>90000</td>
<td>1</td>
<td>90000</td>
<td>0.1</td>
<td>99000</td>
</tr>
<tr>
<td>DC-DC Voltage Converter</td>
<td>7000</td>
<td>4</td>
<td>28000</td>
<td>0.1</td>
<td>30800</td>
</tr>
<tr>
<td>Power Distribution Board and Voltage Regulator</td>
<td>15000</td>
<td>1</td>
<td>15000</td>
<td>0.1</td>
<td>16500</td>
</tr>
<tr>
<td>PDB Switches</td>
<td>10000</td>
<td>1</td>
<td>10000</td>
<td>0.1</td>
<td>11000</td>
</tr>
<tr>
<td>Power Distribution Distiller</td>
<td>14000</td>
<td>1</td>
<td>14000</td>
<td>0.1</td>
<td>15400</td>
</tr>
<tr>
<td>Grand Total</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>433450</td>
</tr>
</tbody>
</table>

Table J.1: Analogy Based Cost Budget

The above budget is based off primary components supplied by the vendors selected by the HAVEN project. The secondary component costs are obtained from Space Quest Ltd and have been scaled up to ensure integration with the HAVEN project power system architecture.
Appendix K

Propulsion Code: Solid

%Shane Moore
%Aero 483 - Project 3
%Team Blue
%Propulsion Sub-subsystem

% RESULTS FOR THE STAR 30E
% ***Propulsion Transfer Stage***
% The transfer stage mass ratio is 3.0334.
% The burn time of the transfer is 51.1 seconds.
% The average thrust of the motor is 35186.1847 Newtons.
% The total propellant mass is 631.4 kg.
% The total in-fairing spacecraft mass is 941.9153 kg.
% The total dry mass is 310.5153 kg.
% The mass after jettisoning the rocket shell is 268.0153 kg.
%
%
% ***Propulsion Control Stage***
% The correction mass ratio is 1.0474.
% The mass of the correction propellant is 12.14 kg.
% The mass of the station-keeping propellant is 12.8 kg.
% The total hydrazine mass is 24.94 kg.
% The remaining spacecraft mass absolutely has to be 255.8753 kg.
% The burn time of the corrections is 1309.4922 s in total.
% The volume of the station-keeping propellant is 0.024742 m^3.
RESULTS FOR THE STAR 30C/BP

***Propulsion Transfer Stage***

The transfer stage mass ratio is 3.0173.

The burn time of the transfer is 51 seconds.
The average thrust of the motor is 33147.4586 Newtons.

The total propellant mass is 590.8037 kg.
The total in-fairing spacecraft mass is 883.6742 kg.
The total dry mass is 292.8705 kg.
The mass after jettisoning the rocket shell is 251.5482 kg.

***Propulsion Control Stage***

The correction mass ratio is 1.0474.

The mass of the correction propellant is 11.3941 kg.
The mass of the station-keeping propellant is 12.8 kg.
The total hydrazine mass is 24.1941 kg.
The remaining spacecraft mass absolutely has to be 240.1541 kg.

The burn time of the corrections is 1229.0357 s in total.
The volume of the station-keeping propellant is 0.024002 m^3.

RESULTS FOR THE STAR 30C

***Propulsion Transfer Stage***

The transfer stage mass ratio is 3.0521.

The burn time of the transfer is 51 seconds.
The average thrust of the motor is 32806.6879 Newtons.

The total propellant mass is 590.8041 kg.
The total in-fairing spacecraft mass is 878.7062 kg.
The total dry mass is 287.9022 kg.
The mass after jettisoning the rocket shell is 248.5304 kg.

***Propulsion Control Stage***

The correction mass ratio is 1.0474.
The mass of the correction propellant is 11.2574 kg.
The mass of the station-keeping propellant is 12.8 kg.
The total hydrazine mass is 24.0574 kg.
The remaining spacecraft mass absolutely has to be 237.273 kg.
The burn time of the corrections is 1214.2909 s in total.
The volume of the station-keeping propellant is 0.023866 m^3.

RESULTS FOR THE STAR 30BP - won’t work
***Propulsion Transfer Stage***
The transfer stage mass ratio is 3.0116.
The burn time of the transfer is 54 seconds.
The average thrust of the motor is 26811.5505 Newtons.
The total propellant mass is 505.1205 kg.
The total in-fairing spacecraft mass is 756.2253 kg.
The total dry mass is 251.1048 kg.
The mass after jettisoning the rocket shell is 213.4113 kg.

***Propulsion Control Stage***
The correction mass ratio is 1.0474.
The mass of the correction propellant is 9.6667 kg.
The mass of the station-keeping propellant is 12.8 kg.
The total hydrazine mass is 22.4667 kg.
The remaining spacecraft mass absolutely has to be 203.7446 kg.
The burn time of the corrections is 1042.7031 s in total.
The volume of the station-keeping propellant is 0.022288 m^3.

function Prop = Solid(Isp,tb,DeltaV,Mfull,Mempty)
Prop = 5;
close all;
g = 9.806;
l = exp(DeltaV/(g*Isp)); %Rocket equation
%l = Mo/Mf
\[
\%Mp = Mo - Mf
\]

\[
\%Solving this system of equations yields
Mf = -(Mempty - Mfull)/(l - 1); \%dry mass
Mo = -l*(Mempty - Mfull)/(l - 1); \%wet mass
\]

\[
Mp = Mo - Mf; \%propellant mass
F = g*(l - 1)*Isp*Mo/(l*tb);
\]

\[
Mdry = Mf - Mempty;
\]

\[
\\
%Transfer Stage Results
\\
clc
disp('***Propulsion Transfer Stage***');
disp('');
disp(['The transfer stage mass ratio is ',num2str(l),'.']);
disp('');
disp(['The burn time of the transfer is ',num2str(tb),' seconds.']);
disp(['The average thrust of the motor is ',num2str(F),' Newtons.']);
disp('');
disp(['The total propellant mass is ',num2str(Mp),', kg.']);
disp(['The total in-fairing spacecraft mass is ',num2str(Mo),', kg.']);
disp(['The total dry mass is ',num2str(Mf),', kg.']);
disp(['The mass after jettisoning the rocket shell is ',num2str(Mdry),...'
    kg.']);
disp('');
disp('');

\]

\[
\\
%Control Stage
\\
DeltaV = 100; \%Orbits mandated that 100 m/s must be
Isp = 220;
F = 20;
\]

\[
1 = \exp(DeltaV/(g*Isp)); \%l = Mo/Mf
Mf = Mdry./l;
\]

\[
Mp = Mdry - Mf;
Tb = Isp./(F./(Mdry.*g))*(1-1/l);
\]

\[
MH25 = 40; \%Clayton said that he needs 40 kg for a 25 year mission life
M8H = MH25*8/25; \%Mass for an 8 year mission life
\]

\[
MH = M8H + Mp;
\]
rhoH = 1008; %Density of Hydrazine
VH = MH/rhoH; %Volume of Hydrazine

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Control Stage Results
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
disp('***Propulsion Control Stage***');
disp(' ');
disp(['The correction mass ratio is ',num2str(l),'.']);
disp(' ');
disp(['The mass of the correction propellant is ',num2str(Mp),' kg.']);
disp(' ');
disp(['The mass of the station-keeping propellant is ',num2str(M8H),' kg.']);
disp(['The total hydrazine mass is ',num2str(MH),' kg.']);
disp(['The remaining spacecraft mass absolutely has to be ',num2str(Mf),...'
   kg.']);
disp(' ');
disp(['The burn time of the corrections is ',num2str(Tb),' s in total.']);
disp(' ');
disp(['The volume of the station-keeping propellant is ',num2str(VH),' m^3.']);
disp(' ');
disp('clear all
end')
Appendix L

Propulsion Code: Liquid

Appendix: Propulsion Liquid Rocket Code

%Clayton McPherson  
%Shane Moore  
%Aero 483 - Project 3  
%Team Blue  
%Propulsion Sub-subsystem

%RESULTS (Phase A): Assuming a payload mass of 900 kg,

% Propulsion(281,2*3870,3200,650)  
%  
% The overall mass ratio is 3.1941.  
%  
% The propellant mass is 1426.1946 kilograms.  
%  
% The total mass of the spacecraft is 2076.1946 kilograms.  
%  
% The total burn time necessary to accomplish the transfer is 507.7338.  
%  
% The total mass of the oxidizer is 1086.6245 kg.  
%  
% The total mass of the fuel is 339.5702 kg.  
%  
% The total volume of the oxidizer is 0.7494 m^3.  
%  
% The total volume of the fuel is 0.38588 m^3.

% RESULTS (Phase B): Assuming a payload mass of 328 kg,
% Propulsion(293,4000,3160,333)
% ***Propulsion Cruise Stage***
%
% The overall mass ratio is 3.0037.
%
% The burn time of the transfer is 479.2588 seconds.
%
% The mass of the fuel is 256.624 kg.
% The mass of the oxidizer is 410.5984 kg.
% The total propellant mass is 667.2223 kg.
% The total wet mass is 1000.2223 kilograms.
%
% The volume of the fuel is 0.29162 m^-3.
% The volume of the oxidizer is 0.28317 m^-3.
%
% The cost of the fuel is $43626.0754.
% The cost of the oxidizer is $4927.1803.
%
%
% ***Propulsion Station-Keeping***
%
% The mass of the station-keeping propellant is 12.8
%
% The volume of the station-keeping propellant is 0.012698

function Prop = Propulsion(Isp,F,DeltaV,Mf)
Prop = 5;

close all;

%Transfer Stage

% R-40B cruise stage
% Isp = 293;
% F = 4000;
% DeltaV = 3200;

MMF = 0:1:2*Mf;

g = 9.806;

%Solve for:
%mass flow rate, Mdot
%overall mass ratio, l = Mo/Mf
%Initial mass of the spacecraft
%total necessary burn time, Tb
%Mass of the propellant
%Volume of the propellant

%Mdot = F/(g*Isp); %Questionable accuracy
l = exp(DeltaV/(g*Isp)); %l = Mo/Mf
Mo = Mf.*l;
MO = MMF.*l;
Tb = Isp./(F./(Mo.*g))*(1-1/l); %Very questionable
Mp = Mo - Mf;

OF = 1.6; %Oxidizer / Fuel Mass Ratio SPECIFICALLY FOR THE R-40A

%MMH (CN_2H_6) has a molecular weight of 0.046 kg/mol.
%NTO (CH_3N_2H_3) has a molecular weight of 0.092 kg/mol.

%Mp = kg of Oxidizer + kg of Fuel
%OF = kg of Oxidizer / kg of Fuel
%Solving this system of equations yields:

MF = Mp/(OF + 1);
MO = Mp*OF/(OF + 1);

rhoF = 880; %Density of Fuel (kg/m^3)
rhoO = 1450; %Density of Oxidizer (kg/m^3)

VF = MF/rhoF; %Volume of Monomethyl Hydrazine
VO = MO/rhoO; %Volume of Nitrogen Tetroxide

CostF = MF*170; %MMH fuel costs 170 $/kg
CostO = MO*12; %NTO costs 12 $/kg

Mdry = Mf - 6.8;

%MpOverMo = 1-exp(-DeltaV/(g*Isp)) %Possibly questionable
%DeltaV = g*Isp*log(l);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Results
clc
disp(’***Propulsion Cruise Stage***’);
disp(’ ’);
disp([’The overall mass ratio is ’,num2str(l),’.’]);
disp(’ ’);
disp([’The burn time of the transfer is ’,num2str(Tb),’ seconds.’]);
disp(’ ’);
disp([’The mass of the fuel is ’,num2str(MF),’ kg.’]);
disp([’The mass of the oxidizer is ’,num2str(MO),’ kg.’]);
disp(['The total propellant mass is ',num2str(Mp),' kg.']);
disp(['The total wet mass is ',num2str(Mo),' kilograms.']);
disp(['The total dry mass is ',num2str(Mf),' kilograms.']);
disp('');
disp(['The volume of the fuel is ',num2str(VF),' m^3.']);
disp(['The volume of the oxidizer is ',num2str(VO),' m^3.']);
disp('');
disp(['The cost of the fuel is $',num2str(CostF),'.']);
disp(['The cost of the oxidizer is $',num2str(CostO),'.']);
disp('');
disp('');

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Control Stage
DeltaV = 100; %Orbits mandated that 100 m/s must be
Isp = 220;
F = 20;

l = exp(DeltaV/(g*Isp)); %l = Mo/Mf
Mf = Mdry./l;

Mp = Mdry - Mf;
Tb = Isp./(F./(Mf.*g))*(1-1/l);

MH25 = 40; %Clayton said that he needs 40 kg for a 25 year mission life
M8H = MH25*8/25; %Mass for an 8 year mission life

MH = M8H + Mp;
rhoH = 1008; %Density of Hydrazine
VH = MH/rhoH; %Volume of Hydrazine

disp('***Propulsion Control Stage***');
disp('');
disp(['The correction mass ratio is ',num2str(l),'.']);
disp('');
disp(['The mass of the correction propellant is ',num2str(Mp),'.']);
disp('');
disp(['The mass of the station-keeping propellant is ',num2str(M8H),'.']);
disp(['The total hydrazine mass is ',num2str(MH),'.']);
disp(['The remaining spacecraft mass absolutely has to be ',num2str(Mf),'.']);
disp('');
disp(['The burn time of the corrections is ',num2str(Tb),'. s in total.']);
disp('');
disp(['The volume of the station-keeping propellant is ',num2str(VH),'. m^3.']);
figure(1)
plot(MMF,M0);
xlabel('Spacecraft Dry Mass (kg)');
ylabel('Spacecraft Total Mass (kg)');

clear all
end
## Appendix M

### Falcon 1E Launch Integration Process

<table>
<thead>
<tr>
<th>Timeframe</th>
<th>Activities</th>
</tr>
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</table>
| Launch – 8 months or more | Contract signing and authority to proceed  
|                  |   • Estimated payload mass, volume, mission, operations and interface requirements  
|                  |   • Safety information (Safety Program Plan; Design information: battery, ordnance, propellants, and operations)  
|                  |   • Mission analysis summary provided to the Customer within 30 days of contract  |
| Launch – 6 months | Final payload design, including: mass, volume, structural characteristics,  
|                  |   mission, operations, and interface requirements  
|                  |   • Payload to provide test verified structural dynamic model  |
| Launch – 4 months | Payload readiness review for Range Safety  
|                  |   • Launch site operations plan  
|                  |   • Hazard analyses  |
| Launch – 3 months | Verification  
|                  |   • Review of payload test data verifying compatibility with Falcon 1 environments  
|                  |   • Coupled payload and Falcon 1 loads analysis completed  
|                  |   • Confirm payload interfaces as built are compatible with Falcon 1  
|                  |   • Mission safety approval  |
| Launch – 4-6 weeks | System Readiness Review (SRR)  
|                  |   • Pre-shipment reviews have occurred, or are about to occur.  
|                  |   • Verify launch site, Range, Regulatory agencies, launch vehicle, payload, people and paper are all in place and ready to begin launch campaign  |
| Launch – 2 weeks  | Payload arrival at launch location  |
| Launch – 8-9 days | Payload mating to Launch Vehicle and fairing encapsulation  |
| Launch – 7 days   | Flight Readiness Review (FRR)  
|                  |   • Review of LV and payload checkouts in hangar. Confirmation of readiness to proceed with Vehicle rollout.  |
| Launch – 1 day    | Launch Readiness Review (LRR)  |
| Launch            | Post-Launch Reports- Quick look  |
| Launch + 4 hours  | Post-Launch Report- Final Report  |