ME430: SENIOR DESIGN PROJECT III

TESSERACT

Final Project Report

Edgar Uribe
Vanessa Faune

Table of Contents

List of Figures

List of Tables

Executive Summary

Tesseract is PolySat's first attempt to develop a 3U structure with deployable panels that will be able to provide more power generation for future payloads. The project is funded by the Warren J Baker Endowment Fund. The goal of Tesseract is to provide a modular structure that can deploy eight solar panels with sixteen solar cells mounted on each panel. Polysat continues to use the knowledge gained from previous structures to improve the manufacturability and assembly of certain subsystems. This structure is to be used as a template for future missions requiring high power draws.

The Top Hat was redesigned to improve accessibility to the electrical stack and to minimize the amount of features that made manufacturing difficult. The new design allows for the hat to be removed by removing the four screws that hold the hat in place. This new design presents huge benefits because all connections can be made and checked entirely without the hat in place.

With the incorporation of the deployable panels, the structure was redesigned to keep all components within CubeSat size specifications, while maintaining a maximum inner volume and adequate mounting points. The structure consists of two large rail panels and support beams that provide a recessed volume for the deployable solar panels. The rails also provide mounting points for the hinges from the deployment system.

The torque required for the deployment of the solar panels is provided by spring loaded hinges. During launch, the deployment system will be in its stowed configuration, where the deployable panels are being constrained by the release mechanism. Once in space, the release mechanism will receive the signal to release the deployable panels, so the structure can transition to its deployed configuration.

When the design was complete, PolySat built a 3D printed prototype using the 3D printing facility in the Cal Poly Mechanical Engineering Department. After initial testing, there were issues identified that could prevent the success of the project. The main issues are the rigidity of the deployable panels in their stowed configuration and the ability of the deployment system to remain in its deployed configuration.

The issues were resolved by making changes to deployment interfaces, mainly in having two release mechanisms for every panel, which constrained the panels in two points rather than one in the stowed configuration. In addition, the recessed panel now has six solar cells to allow room for the two release mechanisms. The solar cell placement was also modified.

After resolving the issues, the team began manufacturing at the Cal Poly machine shops. Most of the components were manufactured on campus with the exception of the smaller hinge parts. The structure was assembled and measured to ensure that it passed CubeSat size specifications. The structure went through standard vibration test, a thermal deployment test, and measurement checks to verify that it met all requirements of the project.

1) Introduction

Sponsor Background and Needs

PolySat is a student-run, Cal Poly research program in which students develop small satellites, known as CubeSats, to be sent into space. Since the start of the program in 2000, PolySat has developed eight 10cm x 10cm x10cm CubeSats. Recently, the team has developed two satellites of double, and triple, that size for NASA-KSC & AI-Solutions and the National Science Foundation. The recent volumetric expansion has been driven by high demand for further satellite functionality, which necessitates large power generation capabilities. To remain competitive in the growing CubeSat industry, PolySat must develop a platform that can provide enough power to the increasingly complex systems residing inside the spacecraft. To accomplish this task, the PolySat team wishes to develop a mechanical structure that will facilitate this increased power requirement.

Formal Problem Definition

Typical CubeSats utilize solar cells, placed on each side of the satellite, to generate power for the payload. Therefore, the amount of surface area available on the outside of the satellite determines the number of solar cells able to be placed on the satellite. To support more power generation, a new satellite structure is needed that allows for more surface area for solar cell placement. With funding from the Warren J Baker Proposal, the team has developed a structure that allows for increased surface area through deployable solar panels. The initial concept of the deployable structure is shown in Figure 1.1.

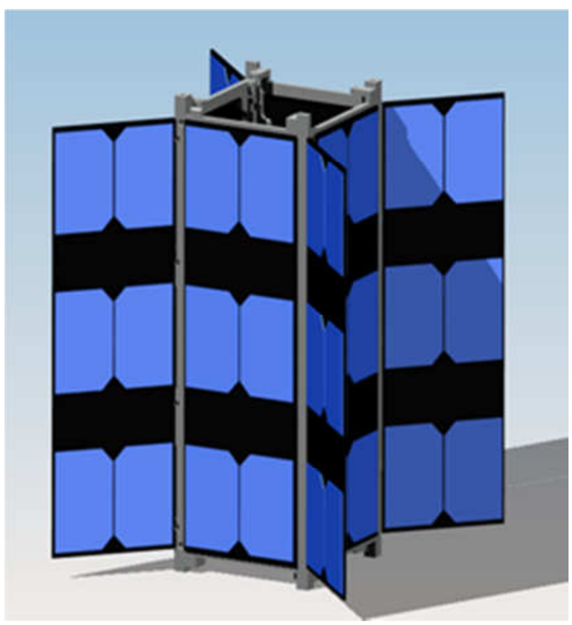


Figure 1.1 Initial Concept Design of Tesseract Structure

The new structure additionally improves the design of specific subsystems. Tesseract provides increased in-house manufacturing by ensuring that all features can be made in the Mustang ’60 machine shop. The new structure serves as a modular platform - one that will allow easy integration of future payloads.

Objective/Specification Development

The main objective of Tesseract is to provide a modular structure that accounts for payloads with high power demands. Along with the increased power generation requirement, the new structure accommodates for the following requirements laid out by CubeSat, PolySat, and a potential mission partner.

The customer requirements listed below were created to ensure that our team meets all the needs the PolySat team requested of this project. To generate more power, this new structure needs deployable panels to increase the surface area where solar cells can be mounted. Due to CubeSat size specs, the outer dimensions of the satellite cannot be larger than they have been in previous satellites. Even with the addition of new features and components, the inner volume of the satellite has to compete with that of previous PolySat satellites. Maximizing the inner volume attracts payload customers who may need a larger volume for their payload.

Additional customer requirements listed below were created to give PolySat members an easier time during assembly and testing. Often during testing, the team needs to reach components within the structure after the full assembly. Tesseract is designed such that only the recessed panel mounting screws must be removed to access the internals of the satellite. Requirements with a "✓" have been completed while those with an "×" are still in development.

Customer Requirements

- ✓ 1st deployable solar panel
- ✓ 2nd deployable solar panel
- ✓ High power generation
- ✓ Maximize available volume for payload
- ✓ Low Mass
- ✓ Make assembly as easy as possible
- ✓ Ease of access for internal components once assembled
- ✓ Redesign Top Hat to facilitate electrical stack access
- × Design dipole antenna mount and release mechanism

The dipole antenna mount is in development by the PolySat team. Additional room has been allocated above the Hat for the dipole antenna mount design.

Table 1.1 Technical Specifications of Tesseract

Spec #	Parameter	Requirement	Tolerance	Risk	Compliance
1	1st Deployable Panel	130 degrees	$\pm 10\ degrees$	H	I,A,T
2	2nd Deployable Panel (deploying from 1st deployable panel)	180 degrees	$\pm 10\ degrees$	H	I,A,T
3	Mass for Payload	2.5 kg	Max	M	I, A
4	Cost	$5000	Max	M	A

5	CubeSat specs	All	Meet	H	I, A
6	Continuous Volume for Payload	$2140\ cm^3$	$\pm 200\ cm^3$	H	I, A
7	Parts Machined on Campus	98 %	$\frac{+2\%}{-15\%}$	M	I
8	Power Generation	21.1 W	$\pm 10\ W$	H	I, A, T

*H=high, M=medium, L=low

*A=Analysis, I=Inspection, S=Similarity to Existing Designs, T=Testing

Table 1.1 is an overview of the Tesseract engineering specifications. These specifications apply quantitative goals for the Tesseract team to meet. The specifications are based off previous designs and the CubeSat specification. The QFD: House of Quality used to reveal the relationships between the engineering specifications and the customer specifications is attached in Appendix A. In populating the QFD, the team identified areas in the design which lacked the qualitative objectives necessary in evaluating the overall success of the project.

With the QFD, the team determined which requirements are crucial to the goal of the project and the relationship between requirements. Of course, the purpose of Tesseract is to maximize power generation by having deployable solar panels, so those requirements are considered significant. Having deployable solar panels increases the time to assemble the structure, the number of parts, and the cost of the project (all of which we were aiming to decrease). Although the incorporation of deployable panels was predicted to decrease ease of assembly and access to the inside of the satellite, the team was able to design deployable solar panels which can be easily removed with the rail panels of the main structure.

Method of Approach

The senior project team consists of two Mechanical Engineers, Vanessa Faune and Edgar Uribe, and the PolySat team provided additional support throughout the process. The end goal was to provide a fully functional 3U satellite structure by May 31, 2015. The entire process began by gathering experienced PolySat members, so that they can identify areas of improvement from previous satellites. These areas of improvement were treated as additional customer requirements.

All PolySat members had the chance to present their concept designs for individual subsystems to the rest of the team. A decision matrix quantified the advantages and disadvantages of each design. We moved forward with the design that best met the requirements for the specific subsystem. After each subsystem had its own concept design, we created detailed designs and evaluated them through further team design reviews.

After the initial design of the system, we used the ME 3D printer to create rapid prototype models of each component. The components were assembled to ensure that all hole-patterns and features fit together properly. The deployment with the release mechanism was tested at this stage. An issue with the deployable panels was discovered and motivated a redesign of the release mechanism. After the issue was redesigned, the team moved on to the aluminum manufacturing process.

Once the structure manufactured and assembled, we performed vibration, thermal deployment, and measurement tests to ensure that Tesseract could meet all the requirements laid out by the team.

2) Background

CubeSat

The CubeSat project was established in 1999 as a collaborative effort between the Multidisciplinary Space Technology Laboratory of Cal Poly and the Space System development Laboratory of Stanford University. The goal was to develop a satellite standard that would ultimately reduce the cost and development time of a satellite. The CubeSat specification for a 3U satellite is attached in Appendix B. CubeSat developed a deployment system called the Poly-Picosatellite Orbital Deployer (P-POD) that holds up to three 1U CubeSats and serves as an interface between the launch vehicle and the CubeSats. A 1U CubeSat (such as IPEX shown in Figure 2.1) is a 10cm x 10cm x 10 cm picosatellite. A 2U CubeSat is the size of two 1U CubeSats stacked on top of each other and a 3U CubeSat is the size of three 1U CubeSats stacked on top of each other. CubeSats must meet the CubeSat specifications to ensure successful deployment from the P-POD.

PolySat

The PolySat program is Cal Poly's CubeSat development team. PolySat has developed eight 1U CubeSats, one 2U CubeSat, and one 3U CubeSat. The 3U CubeSat, ExoCube (CP10), launched on January 31, 2015 and is still in orbit. LEO (CP9), the 2U CubeSat is still in development and is expected to launch in 2016.

Figure 2.1 Image of IPEX (CP8)

Tesseract

Previous Problems or Concerns

The CubeSat structure, developed by the PolySat team, has seen many improvements and adjustments since the start of the program. Major milestones for the structure's development include HyperCube, and the unique ExoCube structure. While these structures solved some of the issues seen with their predecessors, there is still room for improvement. To kick-off the Tesseract project, PolySat team members listed problems they encountered with the previous satellites, features they hoped to have in the future, and suggestions for the approach. The full spreadsheet used to list all points is attached in Appendix C. The major mechanical points we aimed to address in Tesseract are listed in Table 2.1 below.

Table 2.1 Mechanical issues listed by the PolySat team.

Problem/Concern	Description
Mounting space	There is no extra space to drill in mounting holes.
Top Hat	It is difficult to access the electrical stack without stripping a majority of the satellite.
Axis confusion	It is difficult to determine part orientation relative to other parts of the satellite.
Deployable's hinge	Previous hinges for deployables were very close to the protrusion limit

Tesseract aimed alleviate the issues discussed above, while also incorporating the deployable solar panels. By taking into account the problems encountered in previous satellite developments and analyzing the methods used by external developers, the Tesseract team produced an improved design for the modular PolySat satellite structure.

Mounting Space

The addition of mounting holes after the start of manufacturing has proven difficult in the past two designs. In many cases, additional components must be added to the satellite, or component position must be adjusted. Without space for additional mounting holes, any changes in the later phases of satellite development become less feasible.

Top Hat

The electrical stack in CubeSats is the "brains" of the satellite. The stack merges all the electrical components and provides a communication route between the engineers and the electronics on board. When a component on the stack is damaged, it is important to be able to repair it without compromising the functionality of the other components. In the previous satellites, the stack connections were intertwined with the structure and the solar panel boards. When changes to the stack were necessary, the panels, the connections, and the structure, all had to be disassembled. This created more issues as the integrity of connectors and components declined with each disassembly.

Axis Confusion

Many parts of the satellite can be mounted in several orientations, while only one orientation will allow for the correct connections and a smoother assembly. A problem with orientation may not be noticed until a further point in assembly, where a majority of components are integrated and testing can occur. At this point, swapping out components may require some disassembly.

Deployable's Hinge

The CubeSat size specification does not allow for any parts of the satellite to extend beyond 6.5mm from the rails of the structure. The deployables' hinges in the ExoCube structure barely met this requirement as they protruded exactly 6.5mm from the rails. For future missions, it is desirable to increase the gap between the furthest extrusion and the CubeSat limit.

Dipole Antenna

On previous missions, such as IPEX and Exocube, the PolySat team has used a monopole antenna to transmit data to and from the satellite. With the new structure, the team wants to develop a dipole antenna design that can be integrated on the structure, which will provide better data transmission. Previously the antenna was wrapped around a route and mounted onto the Top Hat. The antenna was held in this wrapped position by securing it with Spectra Line burn wire. Nichrome wire was wrapped around the Spectra Line. Upon deployment, a voltage ran through the Nichrome wire, until the Spectra Line was cut. The main issue with this procedure was that reassembly was difficult and time consuming. With the new dipole antenna design, we would like to focus on making the design easy to assemble and reassemble.

Deployable Solar Panels

Even though this is the first time we try to integrate deployable solar panels on to a structure, the PolySat team has developed a couple of successful deployables on previous satellites. Exocube had deployable Gravity Gradient Booms and an antenna. The purpose of the Gravity Gradient Booms was to align the spacecraft to facilitate pointing requirements for Exocube. Figure 2.2 below shows an image of the deployed booms on Exocube. The boom has a hinge assembly that is loaded with a torsional spring. When the boom is in the stowed position, the spring is compressed. A release mechanism deploys the boom. This method of storing energy on CubeSats has proven to be very successful. It is space conservative, lightweight, and inexpensive.

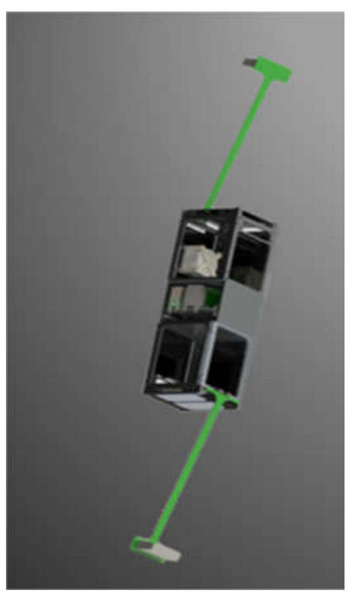


Figure 2.2 Early Exocube CAD model showing Gravity Gradient Boom deployment.

Unlocking Mechanisms

There are a number of ways that the deployable solar panels can be released. Previous methods include burn circuitry and linear actuators. Although burn circuitry is simple to integrate and is inexpensive, it often is more difficult to assemble. This increases the amount of time spent setting up during the testing phase. Linear actuators or pin pullers use shape memory alloy that will compress when a current is applied. They are more expensive, but they can also be easier to reassemble.

3) Design Development

The design of the new structure was divided up into four subsystems: Top Hat, Release Mechanism, Hinges, and overall structure. We began with brainstorming sessions for each subsystem to create a maximum number of concepts for each subsystem. We applied a controlled convergence method to each subsystem to determine which concept best met the project requirements. Because most of the subsystems are interrelated, we have to modify each subsystem to integrate with the rest of the structure.

Deployment Configuration (Power analysis)

When the PolySat team first began working on this new satellite structure, there were several different concepts for Solar Panel Deployment configurations. Most of these were a variation of two main concepts - the Dart configuration and the Cross configuration. Because the other subsystems would be dependent on the deployment configuration, the deployment configuration was the first decision we had to make. The deployment configuration would be chosen based off of which configuration would produce the most power.

Deployment Configuration Concepts

Dart Configuration

The Dart configuration deploys the solar panels by rotating them about x- and y-axis of the satellite. A couple variations that arose were to have deployables only on the top, only on the bottom, or both. With any of these variations, we could remove panels to have less than four on either the top or the bottom of the satellite, but that would be determined based on the mission. We decided to design for a structure that can support the most deployable solar panels because it would be easier to remove solar panels to a design than to add them after we determine how much actual power would be needed for a specific mission. Figure 3.1 shows the dart configuration with the maximum amount of solar panels for this configuration. The satellite structure is the grey body and the panels are the blue rectangles.

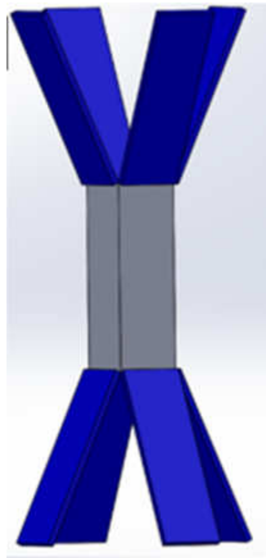


Figure 3.1 Dart Configuration with Deployable panels on both the +z and -z of the satellite

Cross Configuration

The cross configuration deploys the solar panel by rotating them about the z axis. A variation to this configuration was to add a second set of panels that will deploy from the first. This configuration is shown in Figure 3.2. Similar to the Dart configuration, panels can be removed to have deployable panels only on some faces, but that would be determined based on the mission.

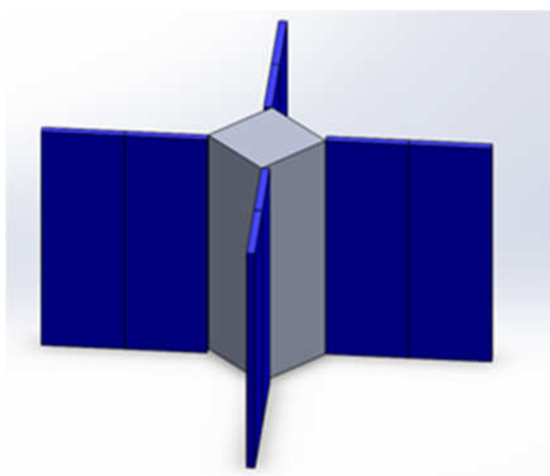


Figure 3.2 Cross Configuration with a second Set of Deployable Panels

Deployment Configuration Selection

Robert Potter, a PolySat member, did a power analysis to compare the power generation between configurations. All configurations were placed in the same simulation orbit, which was the orbit that Exocube (CP10) will be in, to determine which configuration would provide the most power. The analysis assumed that the satellite did not offer a directional control. From this analysis, we determined that the configuration that provides the most power in this type of orbit is the Cross configuration with two deployable panels (shown in Figure 3.2).

Solar Cell Configuration

Tesseract aims to be the PolySat structure for power-hungry payloads. In PolySat's previous 3U design, the structure was only capable of supporting up to six solar cells on each face of the satellite. PolySat has requested that the structure be able to support eight solar cells on each of the four sides of the satellite and eight solar cells on both sides of each deployable panel. The solar cells utilized in the PolySat lab are SpectroLab Ultra Triple Junction (UTJ) solar cells. These solar cells have an extensive space-flight history, and offer adequate performance at a competitive price. While placing eight cells on each panel does not initially seem like a difficult task, the further we progressed into detailed design, the more problems arose.

The side panel and rail panel are two terms frequently used throughout this report. Figure 3.3 shows the distinction between the two terms. The side panel is defined as the electrical board the solar cells are mounted onto, while the rail panels are the large panels used to make up the structure of the satellite. For the permanent side panel, the solar cell configuration must leave room for screws to mount the side panel onto the satellite. Additionally, the CubeSat restrictions for protrusions from the rails of satellite act as a driving factor for the permanent side panel dimensions.

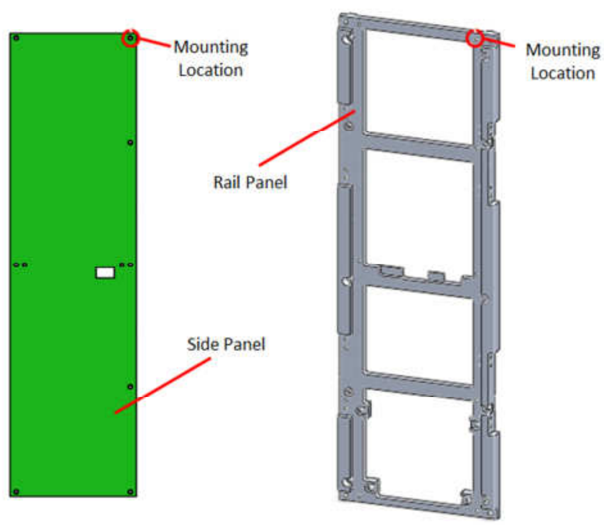


Figure 3.3 Side panel versus rail panel.

The permanent side panel dimensions are limited to fit within the recess of the rail panels to allow room for two sets of deployable panels. With this limitation, along with the demand for eight SpectroLab UTJ solar cells, PolySat proposed a zipper-style configuration (Figure 3.4).

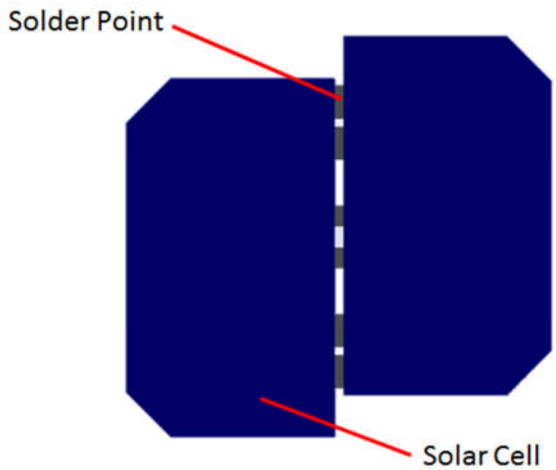


Figure 3.4 Zipper-style solar cell configuration.

In this configuration, adjacent solar cells can be packed closer together without the overlap in the solder points. This configuration requires that the structure be able to support the side panel at the points not occupied by the solar cells. The positions where mounting material is required on the side panel is nearly determined by the implementation of the zipper style.

Top Hat

Top Hat issues

The top hat is a major problem source for the PolySat team. The HyperCube Top Hat, used on recent PolySat satellites is shown in Figure 3.5. It is a fairly complex part to manufacture while also being problematic in the electrical-mechanical interface. The purpose of the Top Hat is to house the electrical stack and to provide the necessary features for meeting the CubeSat specifications (Appendix A). The Tesseract Top Hat design is driven by many factors, including the development of a new dipole antenna, manufacturability of the top hat, ease of assembly, and development of a hub board.

In the HyperCube Top Hat design, the electrical stack must be popped into place at an unnerving angle. The assembly team has difficulty integrating the electrical stack into the Top Hat without the risk of breaking boards or nearby components. The Top Hat has been modified to allow for simpler stack integration.

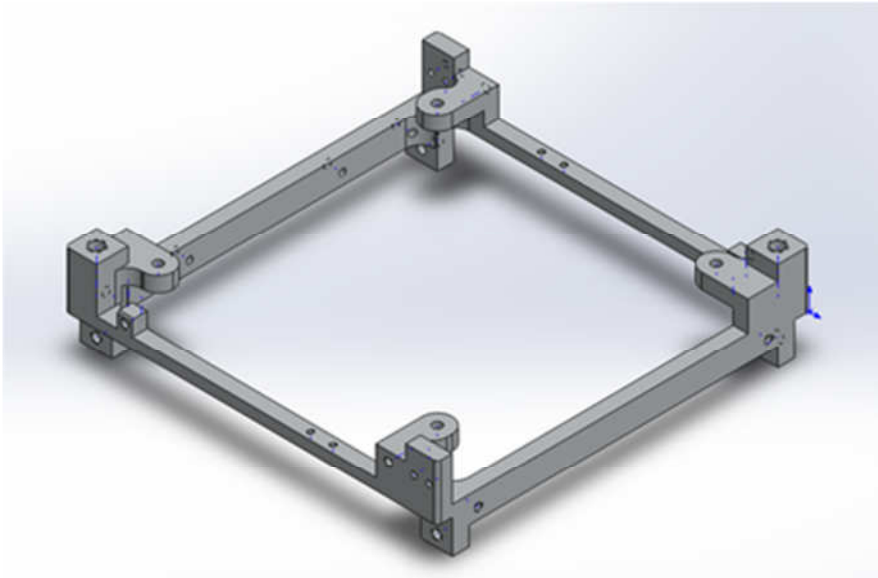


Figure 3.5 HyperCube Top Hat design.

To further aid in assembly, the PolySat team also developed a hub board. The hub board acts as the link between the electrical stack and all electrical boards outside of the stack. The Top Hat must allow for an inter-board connection between the system board and the hub board once the Top Hat is secured onto the structure. The Top Hat must also support a new dipole antenna. The new dipole antenna will be mounted on top of the electrical stack. Given the thickness of the antenna, the Top Hat must ensure that the antenna does not extend beyond the CubeSat specification.

Top Hat Design

With considerations for manufacturing, the new hub board, the dipole antenna, and ease of assembly, the team will proceed with the Top Hat design shown in Figure 3.6.

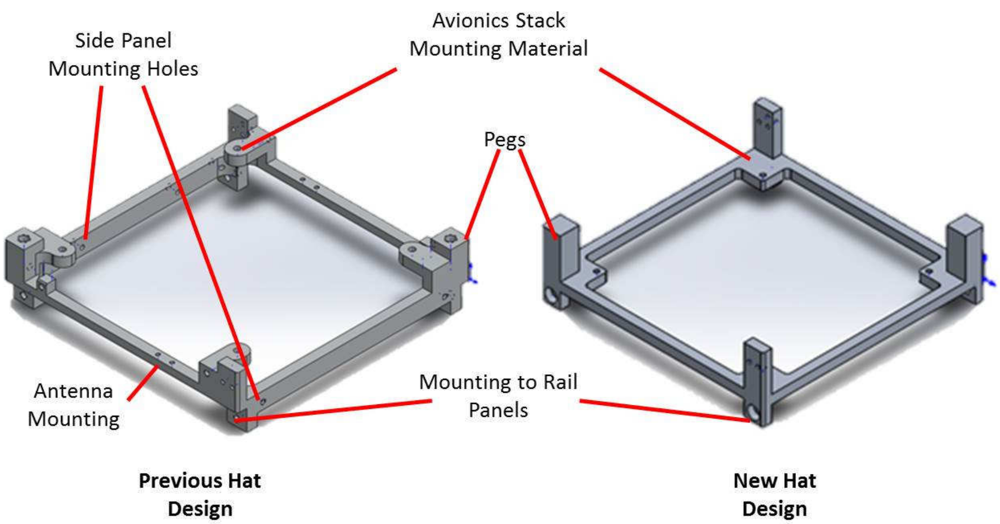


Figure 3.6 HyperCube (left) and Tesseract (right) Top Hat design.

In the Tesseract design, the mounting points for the stack have been lowered such that the stack no longer has to be angled into place. The electronics stack is mounted above the internal extrusions to remove the risk involved with squeezing boards into the hat. No side panels are mounted onto the hat, to allow for stack removal at any point of the assembly. As seen in Figure 3.6, the mounting to the rail panels have been moved outward to allow for the recess in the new structure. Additionally, the top legs of the hat have been extended to provide additional room for the antenna, which no longer has to be mounted onto the hat. This design removes most of the complex features while improving functionality of the top hat subsystem.

Hinges

During background research, the team discovered different methods of connecting the deployable panels. Some of these methods include regular hinges, flexible panels, or a scissor joint. Due to the time frame of the project, we decided to pursue designs that used regular hinges because they are simpler. The hinge must provide the required torque to deploy the panels, stop the panels when fully deployed, have a method of powering solar cells on deployable panels, and provide verification of deployment. Although the hinge between the structure and the first deployable panel could be very similar to the hinge between the first deployable panel and the second deployable panel, there will be small differences due to different space constraints. The basic components of the hinges are two leafs (or a stationary base instead of one leaf), a pin, and a torsional spring.

Hinge concepts

Mounting

The hinges can be mounted onto the panels by using screws. Because of the Solar cell configuration, there will be small amounts of areas available to mount the leaves onto the panels. Figure 3.7 is an image of the deployable side panels highlighting possible mounting points for the leaves on the deployable panels. For the first hinge, there were two mounting possibilities proposed to mount the hinge onto the structure. The first is to mount to the recessed panels. The benefit is that we can probably make the hinges the same, so that we only need one manufacturing procedure for both the first and second hinges. The drawback is that there is a possibility that the panel might hit the rail upon deployment because of the recession. The second choice for mounting is that we can mount the first hinge on the rail. We would have to make a cutout for the hinge, so that the design will meet the CubeSat P-POD requirements. Figure 3.8 shows the first hinge mounted onto the rail of the structure.

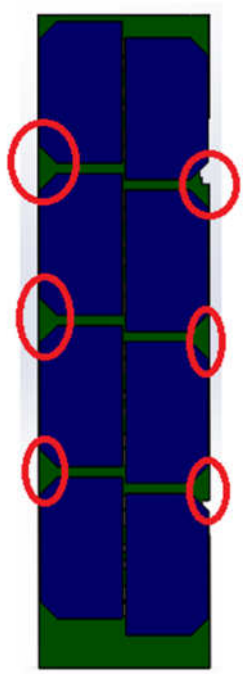


Figure 3.7 Possible mounting points for hinge leafs on deployable panels

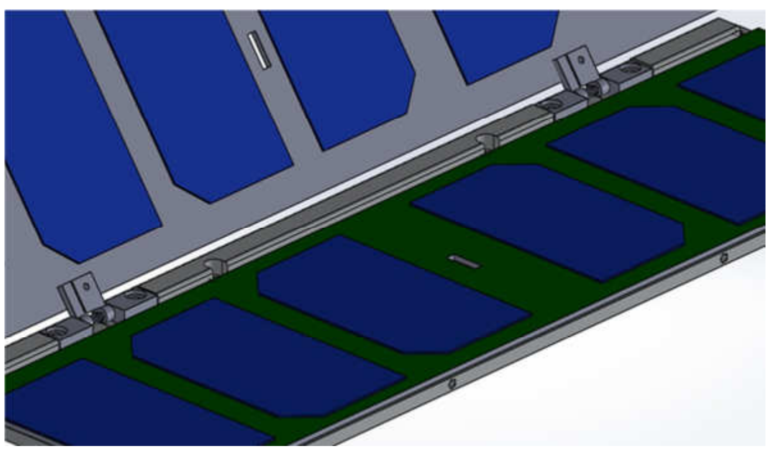


Figure 3.8 CAD model showing the first hinge integrated into the Rail Panel.

Powering Solar Cells on Deployable Panels

Two methods for powering the solar cells on the deployable panels were proposed. The first method is to use flat flex cables to connect to the panels. Flat flex cables are ideal because they can bend without damage while the panels are in their stowed position. The second method is to use the springs on the hinges as a connection between panels. At first, this method appeared to be ideal, but as we further explored this concept, we quickly realized that the amount of work required to make this concept possible was not worth the benefits and that we would probably not be able to complete this task within the timeline of the project.

Deployment Verification

In a previous mission, the PolySat team used cameras to verify the deployment of a set of gravity gradient booms. We explored this method, but due to the number of Solar cells, we found it difficult to find a place for the cameras such that they are able to capture images of the panels fully deployed.

A second option is to have an aluminum tab on the deployable panels that will hit a switch once the panels are fully deployed. The switch must be small enough and placed in a position where it won't interfere with the deployment motion.

Hinge Concept Selections

Our design for the hinges was constrained by other subsystems and the amount of space available. There were not as many concepts to choose from for this subsystem, but we can consider the design a success if it can fulfill the following requirements:

- Provide enough torque to deploy panels
- Stop Panels once fully deployed
- Ensures solar cells will not crack from bending or contact
- Verify deployment

The hinges have torsional springs that will provide the necessary torque for deployment. Once the panels deploy, there are not many forces acting against the spring. Because the panels will deploy in space, we do not need to take drag into consideration. Thus, the torque provided by the spring does not need to be very large.

We will design the hinges such that they stop the panels once they are fully deployed. Ideally, we want to design the hinges, so that the hinge leaves come into contact once the desired deployment angle is reached. The torque from spring should provide a constant force against the stop, so that the panel stays at the desired deployment angle once deployed.

To ensure that the solar cells are not damaged, we have decided to mount the first hinge on the rail of the structure. The panel will deploy by rotating about the outside edge of the rail, such that it will not contact the structure. A minimum distance of 1 millimeter between the panels is maintained to prevent the solar cells on the two panels from contacting each other. Spacers are placed between all three panels to prevent solar cells from clashing into each other.

To verify deployment, we chose to pursue the switch method instead of the camera method. The greatest decision factor was that it will be a lot easier finding a place to mount switch than to find a place for cameras. The switch method would be less expensive and easier to implement as well.

As stated earlier, we chose to use flat flex cables to power the solar cells on the deployable panels instead of powering them through conductive springs. Flat flex cables have the capability of bending without getting damage, so they should be able to handle the motions of the deployable panels. They have been used on previous missions, so the electrical engineers from the PolySat team have experience successfully integrating them on the satellite.

We designed for manufacturability to ensure that the hinges can be manufactured in house. One challenge that we foresee in the hinges is that the hinges are relatively small parts. The hinges are relatively small and it has close tolerances due to the tight space constraint imposed by the CubeSat specifications.

Release Mechanism

The release mechanism must keep the panels in their stowed position until given the signal to deploy. The torque required to deploy the solar panels will be provided by the hinge design, so the design of the release mechanism is focused on maintaining the panels in stowed position and releasing them. Various methods were explored during the first stages of concept development. Although many actuation methods were researched, most of the concepts used either burn circuitry or linear actuators because these methods have successfully been integrated into previous PolySat missions.

Release Mechanism Concepts

The concepts developed during the brainstorming sessions fell into one of two categories, which were to deploy all side panels with one release mechanism or to deploy side panels individually. Only the top four concepts, which are the Center wheel, Bar-Linkage, Miga Motor on Recessed, and Burn Circuitry on Recessed, will be discussed because they display the main features seen throughout the other concepts.

Center Wheel

The center wheel was the first concept that deploys all solar panels at once. A wheel large enough to reach the interior of all four faces of the satellite is placed on the interior of the satellite. “Hooks” attached to the second deployable panels hook on to the center wheel when they are in stowed position. A Miga Motor would release the panels by having a pin rotate the wheel a few degrees. A concept model is shown in Figure 3.9.

Some drawbacks to this method would be that it takes up too much of the inner volume and it would have to be placed either on top or bottom of the satellite because if we placed it in the center, it would cut the inner volume in half.

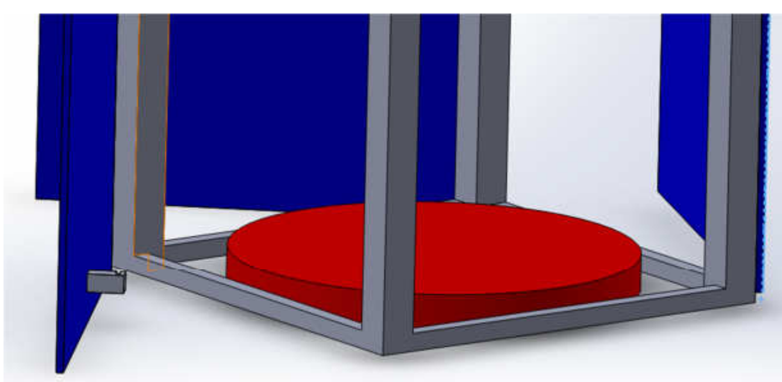


Figure 3.9 Concept CAD model of “Center Wheel” Release Mechanism Concept

Miga Motor on Recessed Panel

The Miga Motor on Recessed Panel concept is probably the most straightforward and easiest to implement into the structure. A Miga Motor will be mounted onto the inner face of each recessed Solar Panel. A "Hook" will be attached to the second deployable panel. The hook will come in through a rectangular cut on the recessed panel pushing a pin that is connected to the Miga Motor to the side. Because the Miga Motor has a spring it will push back and the pin will fall into the slot of the hook causing the hook to remain there until it is released. When the Miga Motor receives a voltage, the shape memory alloy will compress, which will pull the pin out of the hook, thus releasing the deployable panels. Figure 3.10 shows this concept without the pin on the Miga Motor.

The drawback to this method is that we will need a Miga Motor for each set of deployable panels, which would be a total of four Miga Motors. This concept would be the most expensive.

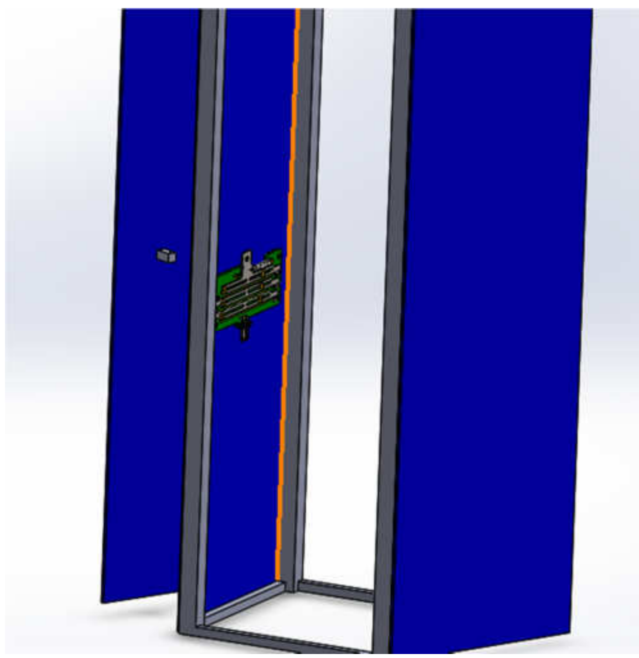


Figure 3.10 Concept CAD model of "Miga Motor on Recessed Panel" Release Mechanism Concept

Bar Linkage

Another concept that deployed all panels at once was the Bar Linkage concept. This concept arose from the Center wheel concept. The linkage is made up of four rectangular cross rails and four L-shaped levers. The linkage sits on a plate that is mounted on to the rest of the structure. The "Hooks" are attached to the deployable panels and they lock into the cross rails from the Bar linkage. Figure 3.11 shows a CAD model of this concept. A Miga Motor pulls the lever arm in the center to release the panels.

The drawbacks to this concept are similar to the drawbacks for the center wheel concept. It takes up too much of the inner volume and it would have to be placed either on top or bottom of the satellite because if we placed it in the center, it would cut the inner volume in half. An improvised version was developed to fix some of the drawbacks of this method.

The new concept was to use a similar bar linkage method, but without the center plate. The release mechanism can be mounted anywhere on the inside of the satellite and it would not take up much of its inner volume. The L-Shaped lever could be mounted on an extrusion from the structure and the Miga Motor could pull one of the cross rails directly. A model of this concept is shown in Figure 12.

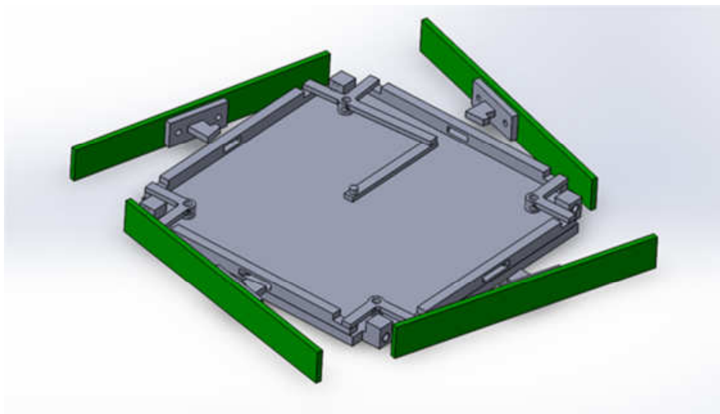


Figure 3.11 Concept CAD model of "Bar Linkage" Release Mechanism Concept

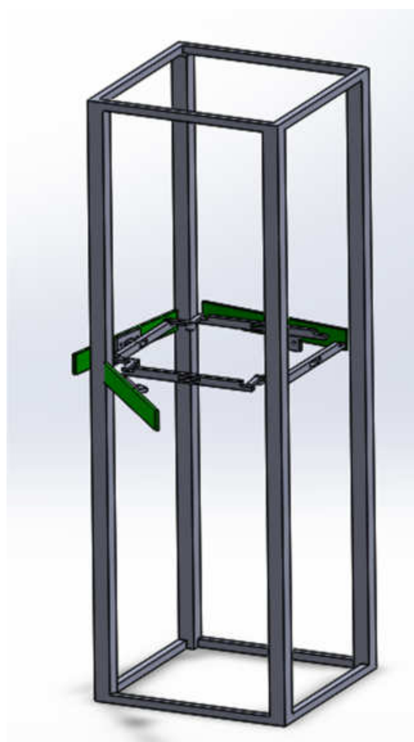


Figure 3.12 Concept CAD model of "Bar Linkage without Center Plate" Release Mechanism Concept

Burn Circuitry on Recessed Panel

This concept is the same as the Miga Motor on the Recessed Panel except it uses burn circuitry instead of a Miga Motor to release the side panels. The deployable panels would have two tabs that would protrude through two holes on the recessed panel. On the inside of the recessed panel there is a carbon rod that is used as a heat source. When the panels are stowed, we will attach some spectra line with two springs on each end to the tabs. The Spectra line will go under the carbon rod and it will be pushing on the carbon rod because the springs are providing tension. When it is time to deploy, current will run through the carbon rod causing it to heat up, thus cutting the spectra line and deploying the panels. A sketch of how the burn circuitry is set up is shown in Figure 3.13.

Some drawbacks to this method are that it will be difficult set up the burn circuitry once all the components are on the structure because it will have to be done inside the satellite. It will also cause more difficulty during testing because we would have to reassemble this release mechanism after every deployment.

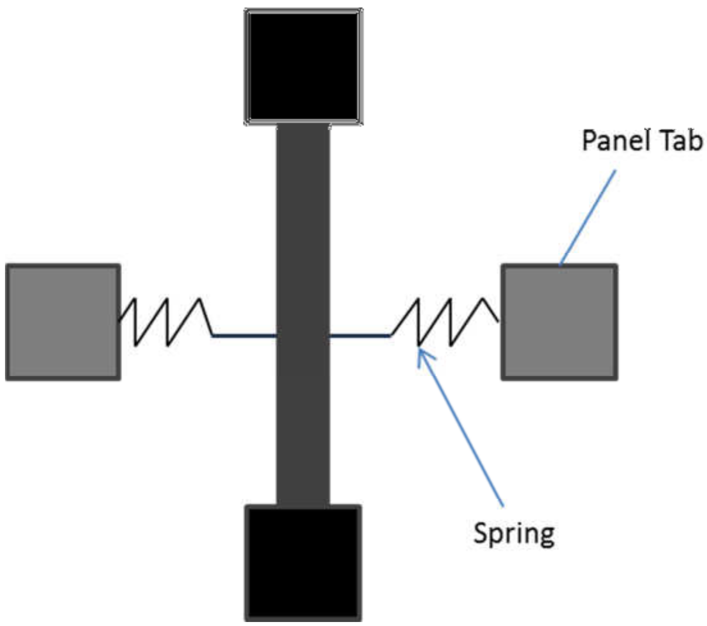


Figure 3.13 Burn circuitry set up on the inside of recessed solar panel.

Release Mechanism Concept Selection

To determine the release mechanism that best fit the project needs, we created the following criteria:

- Resettable (reset time)
- Maximizes inner volume
- Cost
- Reliability
- Complexity of Manufacture
- Low Manufacturing Time

- Number of Parts
- Placement Versatility
- Power Consumption

In previous missions, deployables have been time consuming to integrate and test, especially once the electronics are on the satellite. Resettable is the capability of the release mechanism to reset fairly easy once integrated and deployed. The amount of time it takes for the panels to go from their deployed position to their stowed position is one parameter. The amount of parts that have to be removed or added is another parameter. The PolySat team holds ease of assembly and accessibility as a couple of their overall criteria. The number of parts to be removed from an assembly to access the release mechanism plays a role in the ease of assembly and accessibility.

Since Tesseract currently does not have a mission and we do not know the size of the payload, it is best to design for maximum inner volume, so that a wide range of payloads can be adapted to the structure. Uninterrupted inner volume is preferred.

Cost is determined by the amount of material and off the shelf parts. We aim to manufacture a large percent of this structure to help maintain the cost low. The amount of aluminum stock needed for these concepts is very similar, so the number of actuators may be the deciding factor in this category.

The release mechanism needs to be very reliable. This system is meant to deploy in space, so once the satellite enters the P-POD, we can no longer make any changes to it. The mechanism needs to successfully deploy the panels every time for the mission to be successful. The number of parts moving, the complexity of parts, and degrees of freedom are all parameters of this requirement.

One of our goals for the project is to manufacture a large percent of the satellite. To do this, we need to keep the designs of the parts simple enough to match our manufacturing abilities. The time required to manufacture the entire concept is also a parameter because we need to finish manufacturing by the end of winter quarter.

The number of parts is a factor because it will decrease manufacturing. The number of moving parts is also a factor because having more moving parts decreases the chances of successful deployment.

It would be ideal to be able to mount the mechanism to various places on the satellite. When we do find a payload for Tesseract, we may need to adjust a couple components, so the placement of the release mechanism may change. Currently the largest restricting factor is the solar cell configuration because the hook will have to go through the recessed panel.

Once we knew our requirements for the release mechanism, we began generating different concepts and applying controlled convergence methods to determine which concept best fulfilled the requirements. We created three Pugh Matrices to determine the strength and weaknesses of each concept. We attempted to improve the weakness of the concepts, which led to new concepts or to removing concepts.

We compared the requirements to each other to determine their weight. A decision matrix is shown in Table 3.1. From our decision matrix, we determined that the “Miga Motor on Recessed

Panel" concept fulfilled our criteria. Supporting documents of our decision process for the Release Mechanism can be found in Appendix A. The drawback to this method is that it would require four Miga Motors. The Miga Motors are about $30 each, so this method would be about $90 more expensive than the other concepts shown. This drawback is balanced by the fact that this method is the most reliable and takes up the least amount of inner volume.

Table 3.1 Decision Matrix for Release Mechanism

Criteria	Resettablility	Max Inner Volume	Cost	Reliablility	Complexity of Manufacture	Low Manufacturing Time	# of Parts	Placement Versatility	Power Consumption	Overall Satisfaction
Weight Factor / Alternatives	0.25	0.25	0.1	0.1	0.05	0.05	0.05	0.1	0.05	1
1- Center Wheel	90% / 22.5	25% / 6.25	75% / 7.5	90% / 9.0	75% / 3.75	75% / 3.75	100% / 5	10% / 1	90% / 4.5	63.25
2- Bar Linkage	90% / 22.5	25% / 6.25	75% / 7.5	75% / 7.5	90% / 4.5	75% / 3.75	35% / 1.75	10% / 1	90% / 4.5	59.25
3- Bar Linkage No Center Plate	90% / 22.5	90% / 22.5	75% / 7.5	60% / 6	90% / 4.5	80% / 4	35% / 1.75	90% / 9	90% / 4.5	82.25
4- Migamotor on Recessed	100% / 25	100% / 22.5	35% / 3.5	100% / 10	100% / 5	80% / 4	90% / 4.5	90% / 9	10% / .5	84
5- Burn Circuitry on Recess	15% / 3.75	100% / 22.5	85% / 8.5	90% / 9	90% / 4.5	90% / 4.5	80% / 4	90% / 9	90% / 4.5	70.25

Structure

Recessed Rail Panels

To accommodate for double deployables on the four sides of the satellite, PolySat made the decision to have recessed rail panels. The recessed feature allows two deployable panels to be stacked above a permanent panel while conforming to the CubeSat specifications. It also keeps the deployables fairly isolated from the payload that will be held inside the structure.

Build up Style

PolySat's satellite structure has gone through several revisions over the course of the program. Most changes have been made to accommodate for mission specific payloads. Tesseract aims to be a modular structure for supporting payloads with high power demands. The following concepts illustrate several approaches to allow for this modular structure.

Structure Concepts

Concept 1: HyperCube

The HyperCube configuration for Tesseract is an inverse of the current HyperCube configuration. A CAD model of the current HyperCube rail panel for a 2U and the inverted HyperCube rail panel for a 3U are shown in Figure 3.14 and 3.15 below. The rail panel in the current HyperCube structure has more material toward the outer faces of the structure while the inverted HyperCube rail panel has the material shifted toward the inside. The change creates the recess desired in the rail panel.

Figure 3.14 2U HyperCube rail panel.

Figure 3.15 Inverted 3U HyperCube rail panel.

The HyperCube build-up style (shown in Figure 3.16) has been used in nearly all of PolySat's satellites. It has proven structurally sound in the 1U and 2U sizes while also providing adequate access to satellite internals. Although the HyperCube structure has been used widely in PolySat, the modified structure will still require extensive testing because of the modifications made to create the recess. The use of four identical panels saves some manufacturing time and expenses. However, the structure does not allow for much variability in terms of mounting locations. Since all the panels are identical, the position of the horizontal sections of the panel cannot be varied from panel to panel. Additionally, once manufacturing is complete, there is little room for changes in the design.

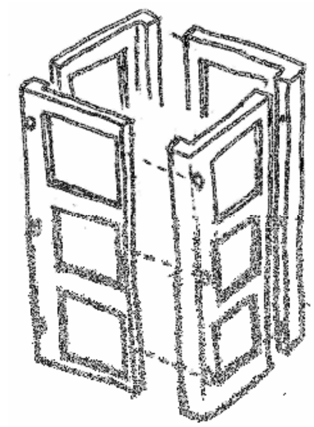


Figure 3.16 HyperCube build up style.

Concept 2: 2 Panels + Beam Supports

The concept shown in Figure 3.17 utilizes two identical rail panels and a series of beam supports to create the overall structure. The use of only two large identical panels cuts down the time and cost associated with manufacturing these large parts. The beam supports between the two rail panels require less material to manufacture. The beam supports can be repositioned at any point in the design phase to allow for changes in mounting. Additional beam supports can also be added if more mounting points are desired.

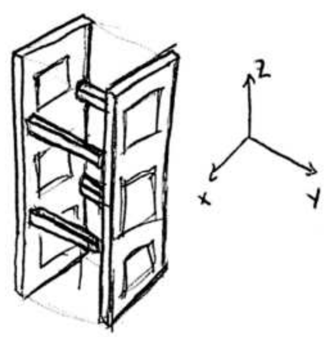


Figure 3.17 2 Panels + Beam Supports build up style.

While this concept provides the modularity desired in the structure, it is predicted to take more time to assemble than the HyperCube design with the addition of several smaller parts. Additionally, the beams will need an additional constraint to prevent beam rotation.

Concept 3: 2 Panels + Rectangular Vertical Supports

Concept 3, shown in Figure 3.18 utilizes two identical panels and rectangular vertical supports. The hat and the shoe will act as supports at the +Z and -Z sides of the satellite. This configuration provides more mounting points on the non-panel sides. The rectangular beam will have to be approximately centered along the Z-axis of the satellite to be an adequate support along the satellite while also acting as a possible mounting point for the electrical panels and the payload.

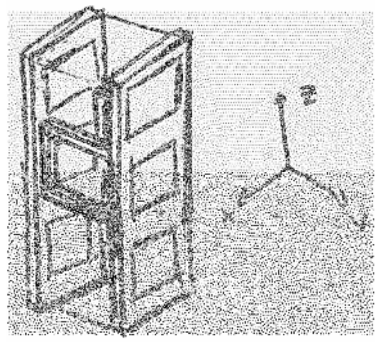


Figure 3.18 2 Panels + Rectangular Vertical Supports build up style.

Concept 4: 2 Panels + Rectangular Horizontal Supports

The concept shown in Figure 3.19 utilizes two identical panels and rectangular horizontal supports. The hat and the shoe will again act as supports at the +Z and -Z sides of the satellite. In this configuration, the internal volume left for the payload is less than in the other designs. The manufacturability of this concept will be the same as Concept 3, but this concept does not allow for the additional mounting points on the non-panel faces.

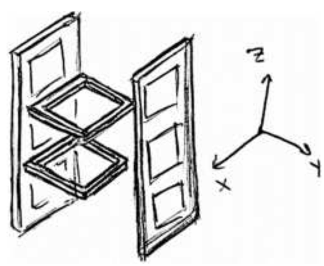


Figure 3.19 2 Panels + Rectangular Horizontal Supports build up style.

Concept 5: 3 Panels + Beam Supports

Concept 5 uses three panels and a few beam supports, as shown in Figure 3.20. This concept focuses on achieving greater ease of assembly. The material required to manufacture this design is less than HyperCube but more than the other concepts. Assembly will be easier in this configuration as three panels can be assembled upright while the internals can be inserted and supported by fasteners on the three sides.

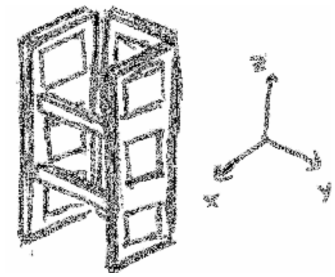


Figure 3.20 3 Panels + Beam Supports build up style.

Structure Concept Selection

PolySat discussed past issues, desirable features, and processes to consider in the design of the structure. From the discussion, the following criteria topics were created for evaluating the structure:

- Material Cost/Manufacturing
- Visual Appeal
- Variability
- Mounting Options/Space
- Internal Volume
- Ease of Assembly

Each criterion was given a decision weight from 1 to 5 based on its importance toward PolySat. A decision weight of 1 means that the criterion has relatively little influence on customer satisfaction. A decision weight of 5 means that the criterion has a high influence on customer satisfaction toward the structure. The criteria listed above were used to evaluate each of the design concepts. The results of the evaluation are summarized in the decision matrix below (Table 3.2).

Table 3.2 Structure build-up style decision matrix.

CONCEPT / CRITERA	Weight	HyperCube Panels	2 Panels + Beam Supports	2 Panels + Vertical Square Supports	2 Panels + Horizontal Square Supports	3 Panels + Beam Supports
Manufacturing	2	0	1	1	1	1
Visual Appeal	1	0	0	-1	1	-1
Adjustability	3	0	1	-1	1	0
Mounting Space	4	0	1	1	1	-1
Internal Volume	5	0	0	0	-1	0
Ease of Assembly	4	0	-1	0	-1	1
	Raw Total	0	2	0	2	0
	Weighted Total	0	5	2	0	1

Material Cost/Manufacturing

Material cost is the monetary cost to manufacture the structure components. If the component is being sent out to be manufactured, the time and labor costs associated with the job would be included as well. For Tesseract, the aim is to have all parts manufactured in the Cal Poly machine shops. The manufacturability of these components with the resources provided on campus drives whether or not the Tesseract team will be able to make all parts. For this criterion, lower values, meaning lower cost and lower difficulty levels, are desirable.

Visual Appeal

The appearance of the satellite structure is important for attracting potential mission partners. A structure that does not appear stable, does not appear capable of supporting the potential payload, and is not pleasing to eye has a lower probability of obtaining support. PolySat put this criterion lower on the priority list, giving it a decision weight of 1.

Variability

Variability is the ability to make modifications to the satellite structure at different stages in development. In the early design phase, variability would be in the ability to move or adjust a component to meet desired needs. For example, the buildup style with the support beams has high variability since the beams can be adjusted to different heights as well as added in to provide the desired mounting points and support. High variability is desirable because of the uncertainties associated with the potential payload. Variability was given a decision weight of 3.

Mounting Options/Space

In the past, the lack of additional mounting space has been an issue. Changes made to the payload or other internals in the testing phase necessitated the remanufacturing of the structure to provide mounting points at a shifted location. Another reason to have additional mounting space is to be able to add weights at different points on the satellite to ensure the center of mass is within tolerance. Designing for more mounting options during the early phases of development removes some limitations on future changes. This criterion was given a decision weight of 4 because its effect on other parts of design.

Internal Volume

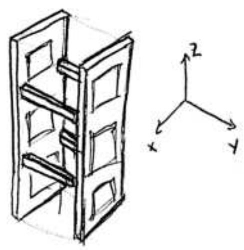
Internal volume is the space within the structure that is allocated toward the payload. Maximizing internal volume is important for attracting potential mission partners and supporting the payload needs. Designing for a payload that requires a large amount of space puts the team in a worst-case scenario. From this point, it will be easier to provide for payloads requiring less space. Internal volume was given a decision weight of 5 because of its close relation to the payload.

Ease of Assembly

Assembly is very important to the PolySat team. While the design may be sound in all other features, but if the assembly team cannot put the satellite together, the structure design is a failure. Many of the PolySat problems dealt with assembly and the electrical-mechanical interfaces. In a modular structure, it is desirable to have a greater ease of assembly to prevent broken components and to prevent unnecessary time and effort on the PolySat team. The ease of assembly was given a decision weight of 4 because of its impact on the PolySat team.

Structure Decision

Based on the above criteria, the build-up style with **two panels and support beams** produced the most beneficial results. It requires much less material than the other structures while still providing adequate mounting space. This concept has high variability, with the ability to add or reposition the support beams along the Z axis of the satellite. The internal volume is maximized in this configuration, but the ease of assembly is predicted to suffer. Overall, this design was chosen to be further developed because of its desirable features.



Chosen Concept Summary

Based on the project requirements and on subsystem requirements, we chose our top concepts for each subsystem or category. Table 3.3 below summarizes our concept selection and provides our reasoning for the selection of each concept.

Table 3.3 Concept Selection Summary

Category	Chosen Concept	Reason	Requirements Met
Deployment Configuration	Double Cross	• Highest Power Generation	• Maximize Power Generation
Solar Cell Configuration	Zipper Style	• Only eight solar cells would fit on recessed panels.	• Eight Solar cells on all panels
Top Hat	Simplified Top Hat	• Removes complex features • Able to remove the electrical stack from satellite by removing Top Hat. • Does not need to be angled into the satellite	• Improve Manufacturability • Ease of assembly
Hinges	• Aluminum Hinge • Mount First Hinge on Rail Panel • Flat Flex Cable to power solar cells on deployable	• Simple • Prevent deployable panel from colliding with satellite • Flat Flex cables are easier to implement than conductive	• Provide require torque for deployment • Maintain Solar Cells safe • Verification of Deployment

	panels • Electrical circuit method to verify deployment	hinges • Verification method will not require much room	• Power Solar cells on deployable panels
Release Mechanism	Miga Motor on Recessed Panel	• Most reliable • Maximizes inner volume • Easily Resettable • Easy to manufacture • Can be placed on different areas of the satellite	• Resettable • Maximizes inner volume • Reliability • Complexity of Manufacture • Number of Parts • Placement Versatility
Structure	Two Panels and Support Beams	• Requires less material • Provides adequate mounting space • Ability to reposition support beams along Z axis • Maximized inner volume	• Material Cost • Variability • Mounting Options/Space • Internal Volume

4) Description of the Final Design

Overall System Description

The full assembly of our final design is shown in Figure 4.1 (stowed configuration) and Figure 4.2 (deployed configuration). Figure 4.3 shows a 3D printed rapid prototype we made using the 3D printing facility in the Cal Poly Mechanical Engineering Department. The following is an overall description of Tesseract.

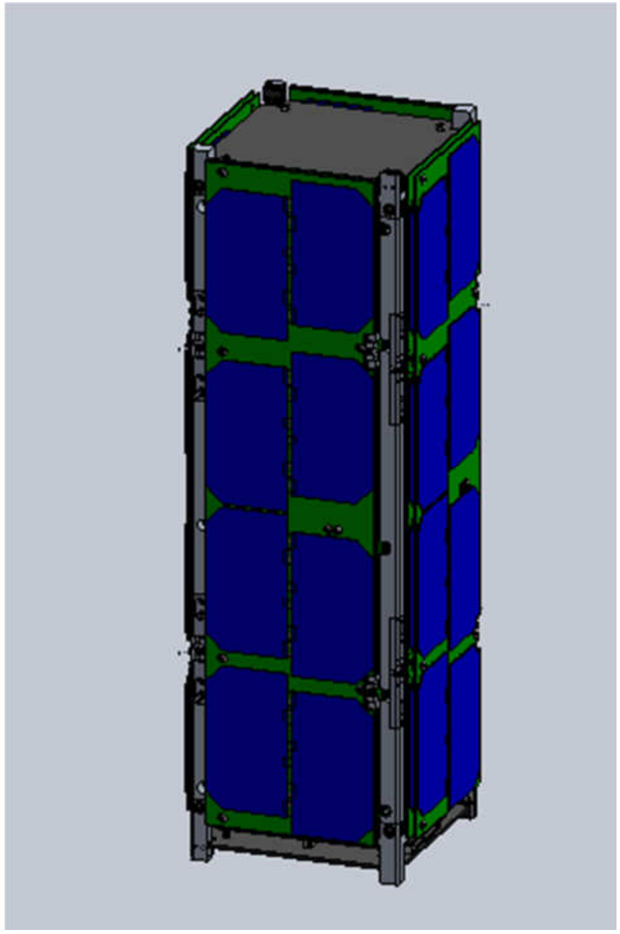


Figure 4.1 Tesseract Assembly in Stowed Configuration

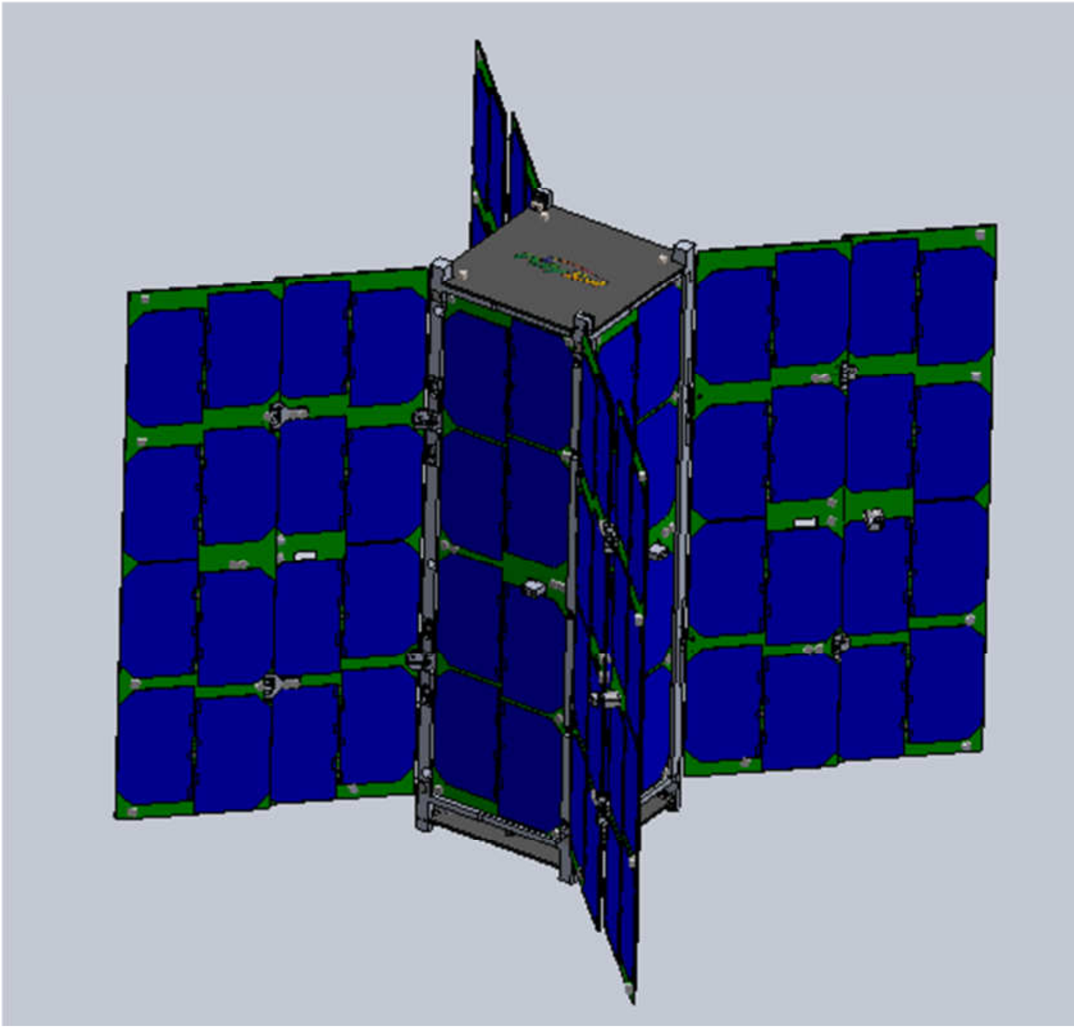


Figure 4.2 Tesseract Assembly Deployed Configuration

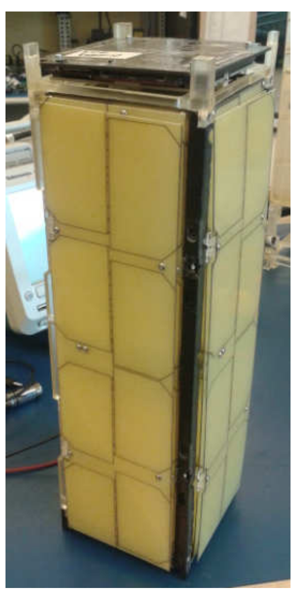


Figure 4.3 Printed Prototype of Tesseract Structure.

As stated earlier, the release mechanism has the capability of holding the system in the stowed position during launch and once given the signal, it should release the solar panels, so they can open up to their deployed configuration. The hinges provide the necessary torque to deploy the solar panels and also, maintain the solar cells safe throughout the deployment motion. The Top Hat holds the electrical stack. All the subsystems and components are held together by the main structure, which will also carry the payload once there is a customer and a mission. Figure 4.4 shows the four main subsystems of Tesseract.

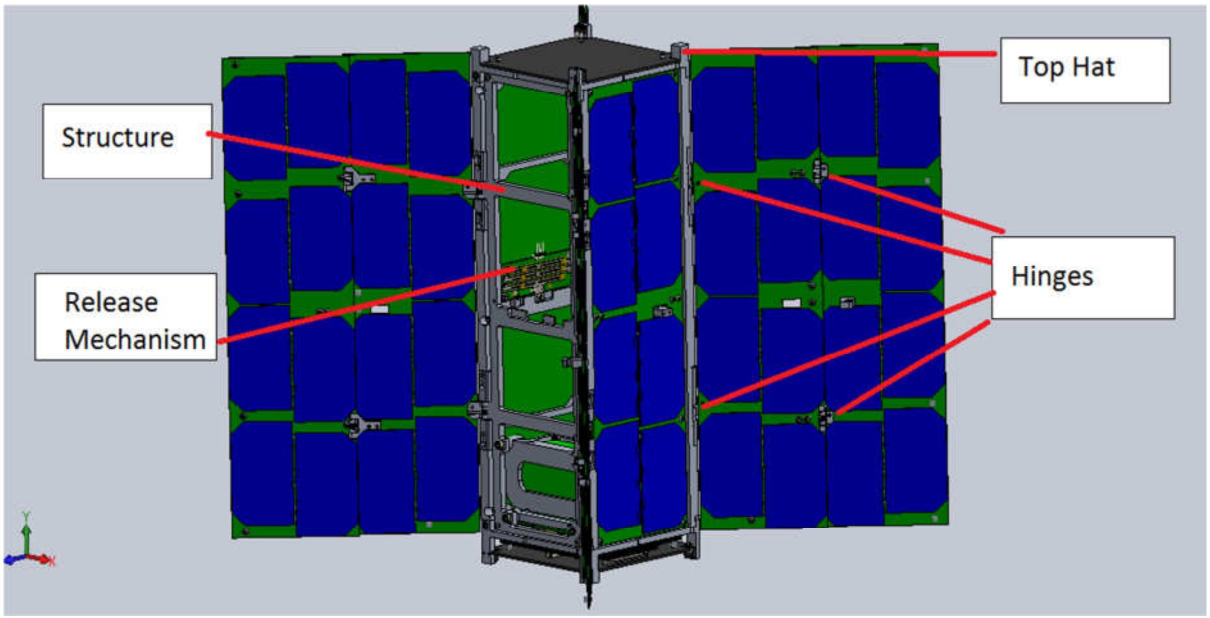


Figure 4.4 Overall System

Solar Panels

The solar panels are made from a material called FR-4. It is a composite material composed of woven fiberglass cloth with an epoxy resin binder. Some characteristics that make FR-4 a great selection for solar panels are that it has a good strength to weight ratio, it has near zero water absorption, and it is an electrical insulator. FR-4 is often used for printed circuit board and Polysat has used it numerous times in previous missions for solar panels.

There are a total of three different sets of solar panels on Tesseract and each set contains four panels (one for each side). For the rest of the document, we will refer to the stationary panel, which does not deploy, as the Recessed Panel. The other two panels will be referred to as the First Deployable Panel and the Second Deployable Panel, where the Second Deployable Panel will be the solar panel furthest from the satellite in the deployed configuration.

Ordinarily, the solar panels are 1.6 millimeters thick, but because the solar panels will be stacked upon each other in the stowed configuration, we had to use the 0.9 millimeter thick solar panels, to not violate the CubeSat 6.5 millimeters protrusion limit from the rail. Figure 4.5 shows a section of the top view for Tesseract to display that it is within the specified limit. As can be seen in the figure, the top of the screw head, which is the furthest component out from the satellite, is 6.4 millimeters from the plane of the rails.

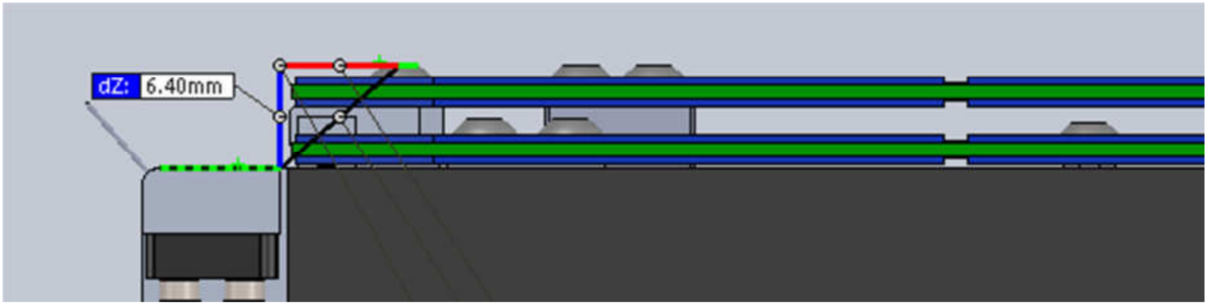


Figure 4.5 Top view of Tesseract displaying how the solar panels stay within the 6.5mm Cubesat Protrusion limit.

Recessed Panel

Figure 4.6 shows an outside and inside view of the recessed panel. On the outside view, the mounting points are shown on the corners and in between solar cells. The recessed panels are mounted using #2-56 x ¼" button-head socket cap screws. There is also a rectangular cutout (visible on the outside view) for the hook used in the release mechanism. The hook is mounted on the Second Deployable Panel and it goes through the cutout on the Recessed Panel, where it gets hooked onto a pin attached to the Miga Motor. The mounting of the Miga Motor can be seen on the inside view in Figure 4.6.

Using the solar cell zipper style configuration was the only way that the solar cells would fit in the width of the solar panel. With this configuration used on the deployable panels, we were able to fit eight solar cells on each side of the deployable panels, a task many members of the CubeSat industry claimed to be impossible.

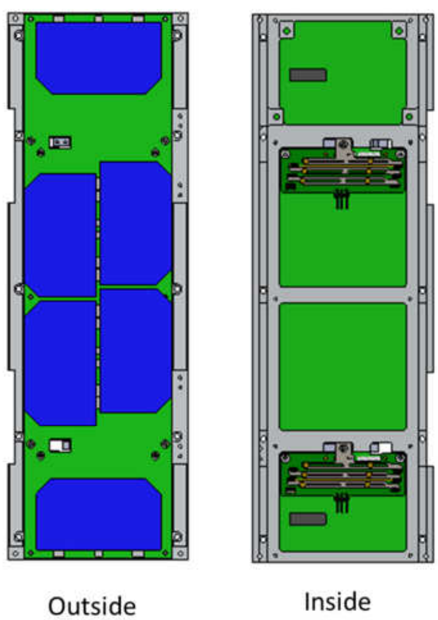


Figure 4.6 Outside and Inside View of Recessed Panel

First Deployable Panel

Figure 4.7 shows an outside and inside view of the First Deployable Panel. On the inside view, the hinge leaves are visible for both the first and second hinge (second hinge is the longer pair on the left side of the inside view and the first hinge is the smaller pair on the right side). There is also ⅛" long spacers mounted on the First Deployable panel to prevent the solar cells from clashing into the Recessed Panel. Again, the rectangular cutout is for the hook mounted on the Second Deployable Panel that passes through both the First Deployable Panel and Recessed Panel.

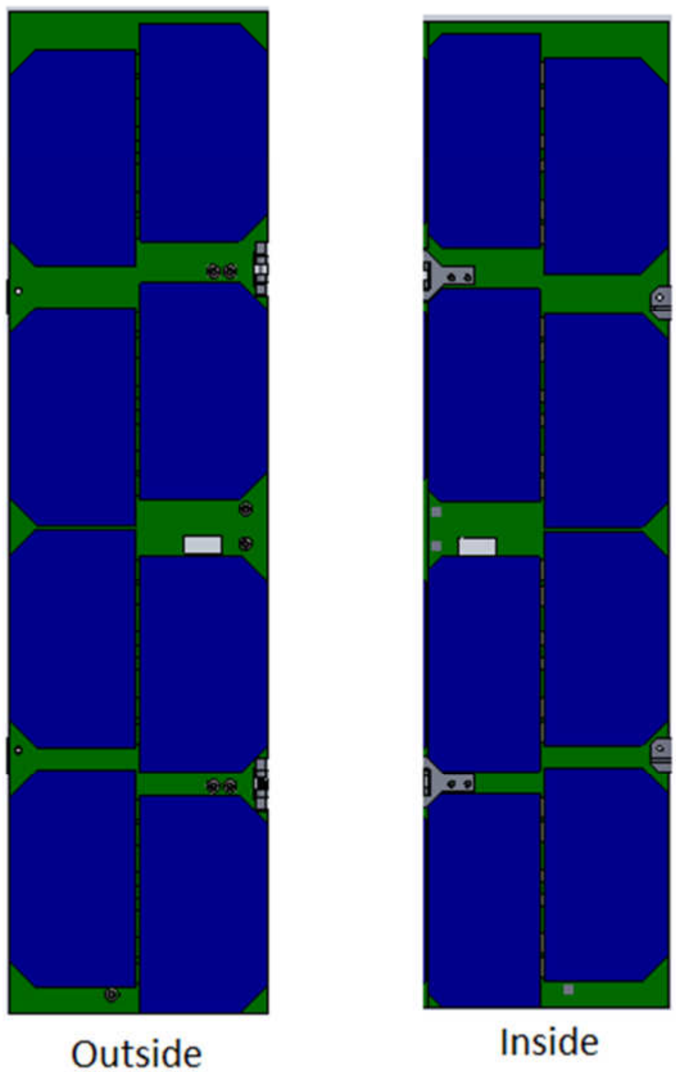


Figure 4.7 Outside and Inside View of First Deployable Panel

Second Deployable Panel

Figure 4.8 shows an outside and side view of the Second Deployable Panel. On the outside view, the cutouts for the hinges are shown on the right. There are 2.7 millimeter standoffs mounted along the left side that prevent the solar cells on the inside from clashing into the First Deployable

Panel. On the side view, the hook for the release mechanism is visible. Drawings showing the dimensions of the solar panels can be found in Appendix B.

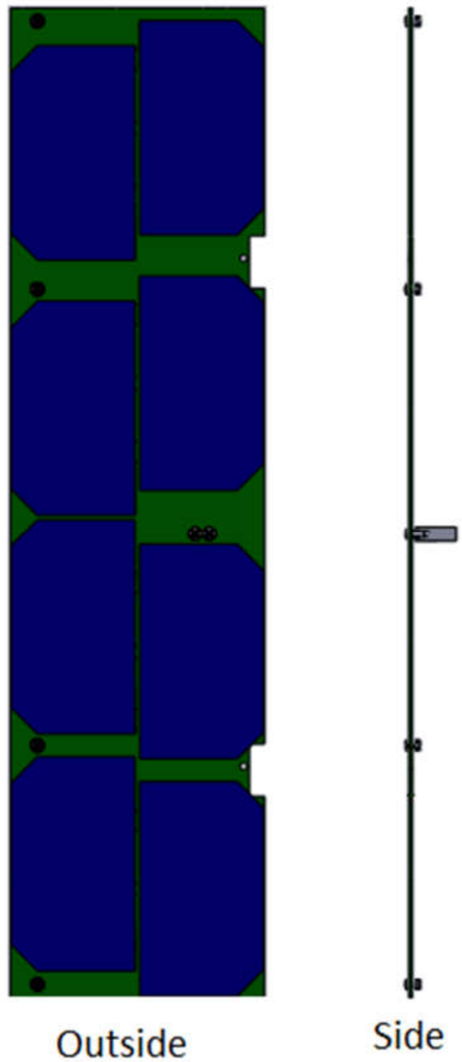


Figure 4.8 Outside and Side View of Second Deployable Panel

Release Mechanism

The release mechanism shown in Figure 4.9 consist of three main components; a Miga motor, a hook, and a pin. We chose this design for its simplicity because it would result in a more reliable method. Also, this design is easily resettable. From the deployment configuration, the panels just need to be folded back up and pushed until the hook clicks into place (shown in Figure 4.10). Each of the four sides of the satellite will have its own release mechanism.

A Miga Motor is a shape memory alloy actuator sold by Miga Motor Company. They are very light weight and use Nickel-Titanium muscle wires that contract when they are heated. In the release mechanism, we attached a pin, which gets pulled by the Miga Motor when an electrical current heats up the Nickel-Titanium muscle wires, thus releasing the hook and the deployable panels.

When there is no longer an electrical current, the muscle wire cools down and returns to its original shape. The release mechanism design uses a Dash4 Miga Motor, which provides a stroke of 0.23 inches and 1.75 pounds of actuation force. We chose the Dash4 over the other models because it was small enough to fit in the recessed panel and it provided a large enough stroke to allow enough contact between the pin and the hook.

The Hook and the pin (also known as the Talon) will be made out of Aluminum 6061. We will manufacture them on campus. Drawings showing the dimensions of the parts can be found in Appendix B. The Talon will slide across the Deployment Support Beam, which is part of the main structure. The Miga Motor will be mounted on the Recessed Panel using #2-56 screws and the Talon will be secured on the Miga Motor using a #2-56 screw as well. The Hook will be secured on the Second Deployable Panel using two #2-56 screws. The Hook will go through cutouts in the First Deployable Panel and the Recessed Panel before it locks with the Talon.

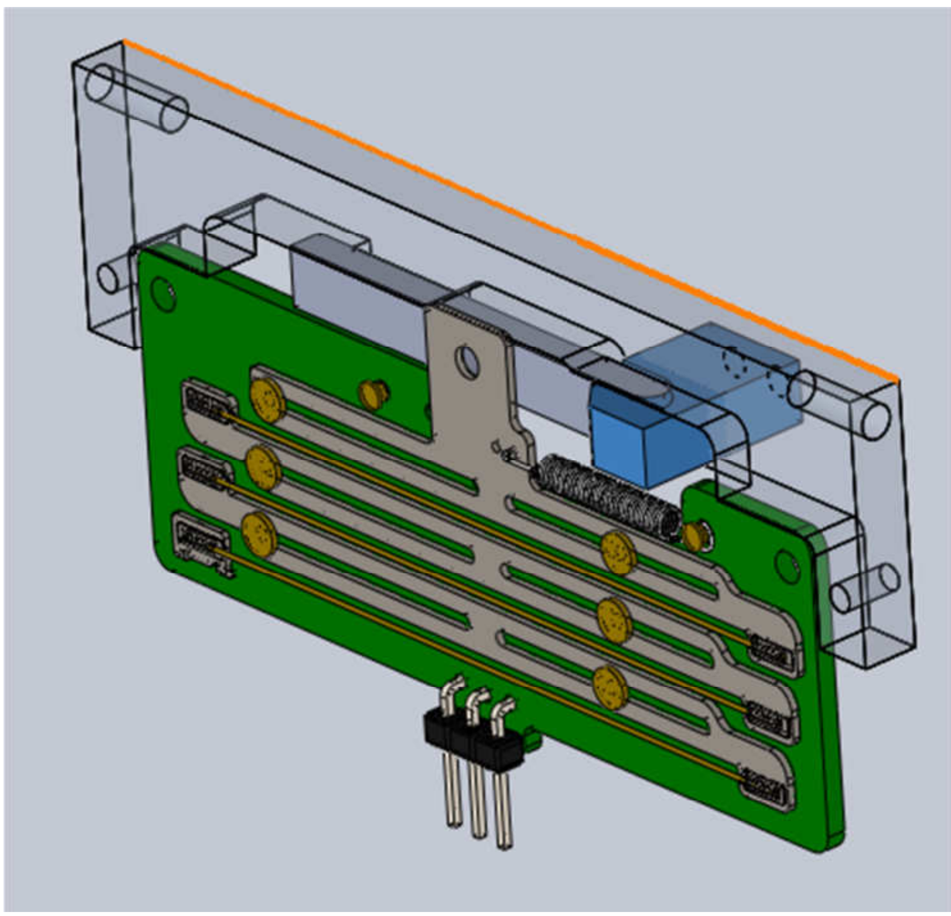


Figure 4.9 Release Mechanism

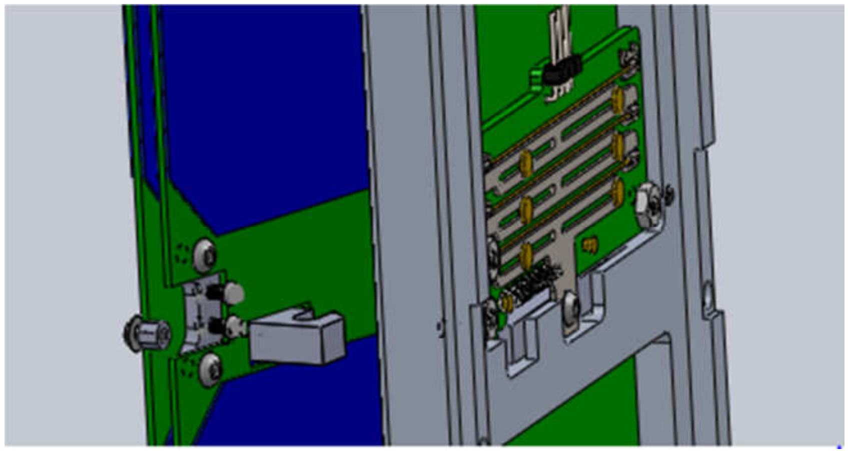


Figure 4.10 Resetting Release Mechanism

We built a small prototype of the release mechanism using an old Miga Motor we found in the Polysat lab and we manufactured the Hook and Talon out of wood. We tested to see if the angle on the Hook and the fillet on the Talon would actually be enough to push the Talon, so the Hook can click into place. The geometry worked fine on our first prototype. In our RP model, the release mechanism worked very smoothly. It clicked into place just as expected and deployed the panels when given an electrical current.

Hinges

The design of the hinges provides the required torque to deploy the solar panels, stops the the solar panels once fully deployed, keeps the solar cells safe through the deployment motion, and verifies deployment. The biggest challenges with the design of the hinges was the tight space constraint due to the solar cell configuration, the space constraint due to the Cubesat protrusion limit, and keeping enough distance between the panels during deployment. There are two different design of hinges; one to deploy the First Deployable Panel and one to deploy the Second Deployable Panel.

First Hinge

The first hinge is made up of five main components; Hinge Block Left, Hinge Block Right, First Leaf, a pin, and a spring. The CAD model of the first hinge assembly in both stowed and deployed configuration is shown in Figure 4.11. The spring and the pin are COTS parts and the other three parts are custom parts that we will manufacture ourselves. They are made of Aluminum 6061 because that is the material that we use for most of our structure, due to its availability and mechanical properties. Drawings showing the dimensions of the part are in Appendix B.

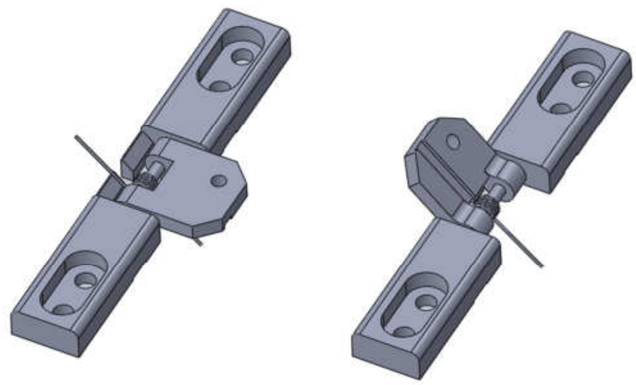


Figure 4.11 First Hinge Assembly. Left is in stowed configuration and right is in deployed configuration

The first hinge is mounted on the cutout on the rail. The top face of the hinge, when it is in the stowed configuration, must be flush with the rails of the structure to not violate Cubesat specification. The pin needs to go through the leaf and spring first, then inserted into the two blocks. After the Hinge is assembled like the figures above, 2 #2-56 screws will be used to secure each block to the rail. The leaf falls in between the Recessed Panel and the First Deployable Panel. The First Deployable Panel is secured to with a #2-56 screw. There are two First Hinge assemblies per rail, so a total of 8 on Tesseract. Figure 4. 12 shows the properties of the torsional spring used in the first hinge.

Wind Direction	Right-Hand
Deflection Angle	180° Deflection Angle/Degrees of Rotation
Spring OD	0.109"
Wire Diameter	0.012"
Leg Length	0.375"
Maximum Rod OD	0.067"
Number of Coils	5.00
Torque in.-lbs.	0.047
Spring Length @ Torque	0.090"
Additional Specifications	Type 302 Stainless Steel
RoHS	Compliant

Figure 4.12 First Hinge Torsional Spring Properties from McMaster-Carr

Maintaining Solar Cells Safe

By mounting the hinge on to the rail, we avoided the possibility of damaging the solar cells with the structure. As described earlier, there will be ⅛ inch long standoffs in between the First Deployable Panel and the Recessed Panel. This gives a clearance of about 2.25 millimeters between the solar cells. The First Hinge will deploy the First Deployable Panel 135 degrees before coming into contact with a hard stop on the Hinge Blocks. Figure 4.13 shows a side view of the fully deployed first hinge assembly. This figure highlights the faces that will come into contact.

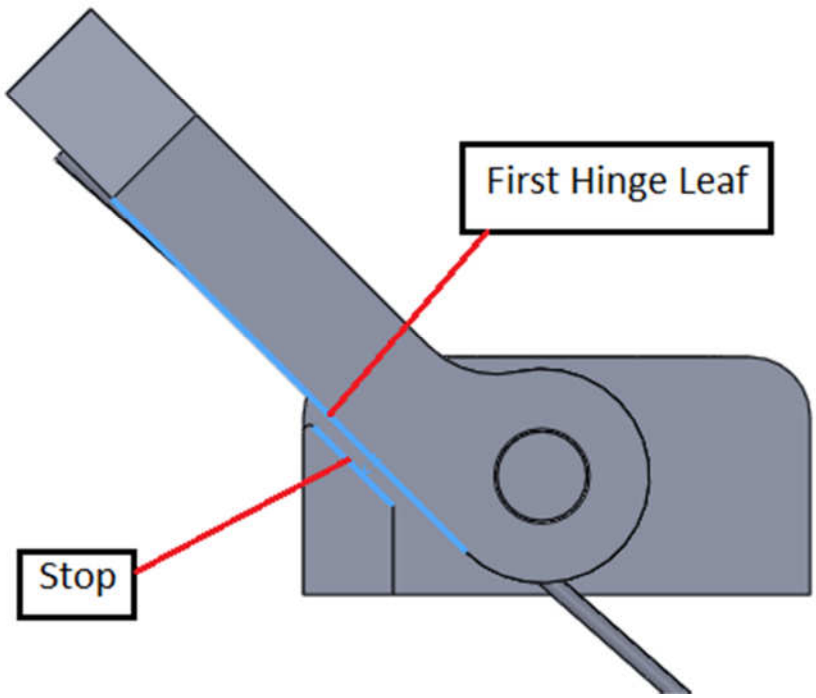


Figure 4.13 Full Deployment of First Hinge showing Hard Stop

Because Tesseract will be deploying in space, we did not take drag into consideration and we assumed friction to be small. The largest stresses on the hinge would be when the leaf hits the stop. We performed analysis to ensure that the stop and the pin would survive the impact because those are the thinnest parts on the hinge. Complete hand calculations are shown in Appendix E. Using the work-energy principle, we calculated the angular velocity at 135 degree (right before impact) to be 3.66 rad/s. Then, we applied the linear impulse-momentum principle and assumed that the collision would be inelastic because the spring constant of the torsional spring is small. We calculated the impact force to be about 2.6 lbf per hinge. We used this force to perform stress analysis on the pin and stop to ensure they would not yield. We calculated a safety factor of 3 for the pin and a safety factor of about 10 for the stop.

Second Hinge

Initially, we tried to keep the design of the second hinge really similar to the first hinge, but we steered away from that as the design process moved on. The hinges had a few different constraints on them and little functionality differences, but these small differences became big changes on such small components. The second hinge assembly consist of a Second Bottom Leaf, a Second Top Leaf, a torsional spring, and a pin. The spring and the pin are COTS parts and both leaves are custom parts, which will be made of Aluminum 6061. Figure 4.14 shows CAD models of the second hinge assembly in both stowed and deployed configuration. Drawings showing the dimensions of the part are in Appendix B.

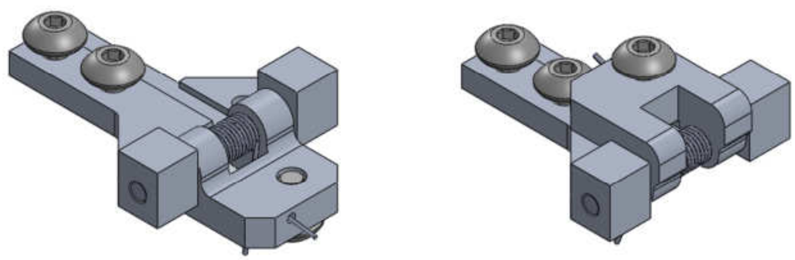


Figure 4.14 Second Hinge Assembly. Right is in stowed configuration and left is in deployed configuration.

The layout of the hinges is shown in Figure 4.15. This is a top view of just one rail and all three solar panels. As can be seen in the image, both the First Leaf and the Second Bottom Leaf are in between the First Deployable Panel and the Recessed Panel. The Second Top Leaf is in between the Second Deployable Panel and the First Deployable Panel. The red arrows show the direction each hinge will rotate once deployed. Placing the hinges in this stacked configuration was the only way we can fit all the components within the 6.5 millimeter Cubesat protrusion limit.

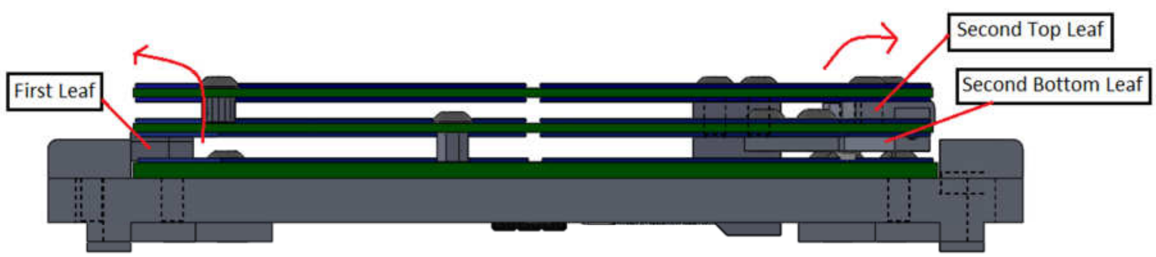


Figure 4.15 Top View of Deployable Panels in stowed configuration.

Similar to the First Hinge Assembly, the Second Hinge will have to be assembled before being mounted onto the solar panels. First, we must insert one of the legs of the torsional spring into the hole of the Second Top Leaf. Then, we must align the holes of both leaves, so that they are concentric and insert the pin through both the leaves and the spring. Finally, we can mount the Second Bottom Leaf to the First Deployable Panel and the Second Top Leaf to the Second Deployable Panel using #2-56 x ⅛” button-head socket cap screws. Figure 4.16 demonstrates the order of assembly. Figure 4.17 shows the properties of the torsional spring.

Figure 4.16 Order of Assembly for Second Hinge

CSC #	TO-5000RS
OD (in)	0.1030
OD (mm)	2.62
ID (in)	0.0790
ID (mm)	2.01
Rate (in-lbs deg)	0.00017
Total No of Coils	9.750
Wire Dia (in)	0.0120
Sugg Mandrel Dia (in)	0.06
Sugg Max Defl (deg)	215
Sugg Max Load (in-lbs)	0.037
Body Length (in)	0.14
Leg Length (mm)	9.5
Wind Dir	RIGHT

Figure 4.17 Second Hinge Torsional Spring Properties from Century Springs

There are 2.7 millimeter spacers between the First Deployable Panel and the Second Deployable Panel to prevent the Solar Cells from clashing with each other during launch (see Figure 4.15; spacer in on the top left). The clearance between the solar cells is 1.78 millimeters. Using the dynamic clearance tool in SolidWorks, we determined that the closest the solar cells came to clashing with each other during deployment was 0.82 millimeters. This can be seen in Figure 4.18, where the Second Deployable Panel is about 80 degrees deployed from the First Deployable Panel.

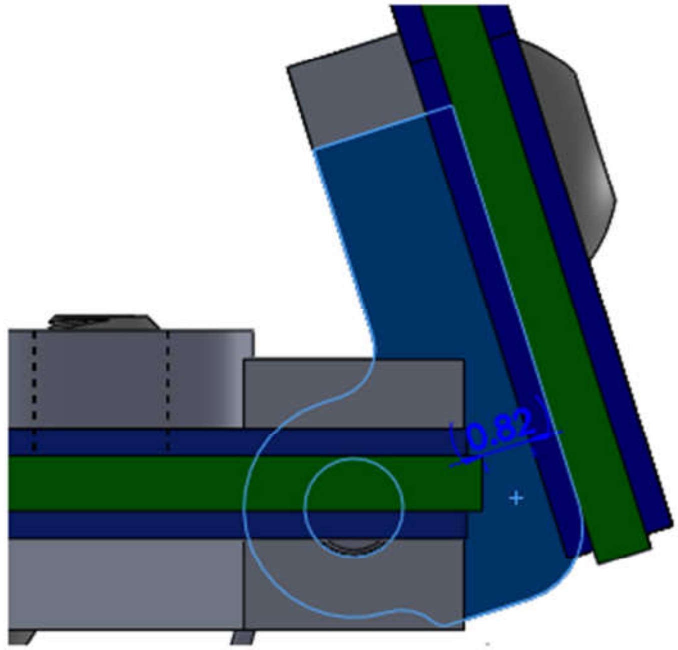


Figure 4.18 Minimum Clearance between Solar Cells during Deployment.

Although the Second Hinge is small, it has a couple key features that are crucial to its functionality. The spring hole and the spring slot shown in Figure 4.19 keep the spring constrained, while the spring pushes on the hinges to provide the necessary torque for deployment. The back of both the top and bottom leaf collide when the hinge opens up to 180 degrees providing the stop once fully deployed. The clearance radii are there for manufacturability and to allow the hinge to open a full 180 degrees. We do not have the capability of making rectangular cutouts in the shop, so we need a fillet on the inside corners the size of the radius of our tool. If we made a regular quarter-circle fillet, there is a possibility that the hinge would not open 180 degrees because the top leaf would make contact with the fillet before stop.

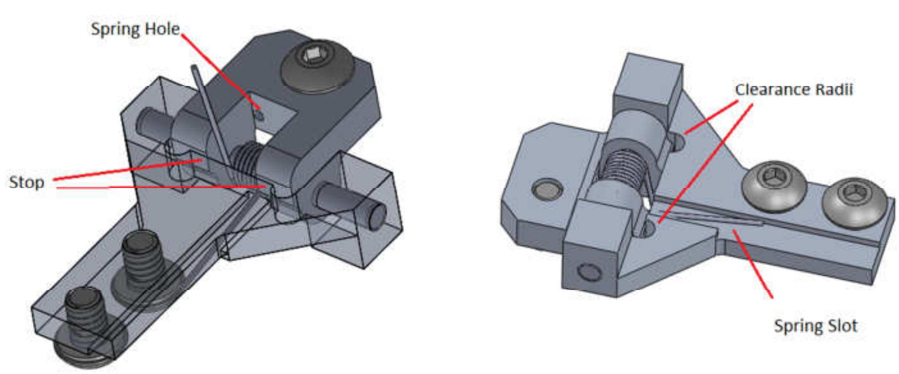


Figure 4.19 Some Key Features of Second Hinge

Deployment Verification

The deployment verification design should send a signal to electrical stack, when the panels are fully deployed. We decided to go with a switch instead of cameras because there may not be enough space to fit the cameras somewhere where they can capture the solar panels being fully deployed. The switch we are using is very small. It is 7mm x 8mm x 2.7mm. It is the same type of switch used on the Top Hat to verify that the Satellite has deployed from the P-POD.

The First Deployable Panel and the Second Deployable Panel overlap about 4mm without interfering when they are fully deployed. In the center of the First Deployable Panel near the hook cutout, there is enough room to mount the switch. A standoff is mounted to the Second Deployable Panel to come into contact and close the switch. A CAD model of the design is shown in Figure 4.20. The switch is the black part that is mounted to the aluminum plate. We will manufacture the aluminum plate as well.

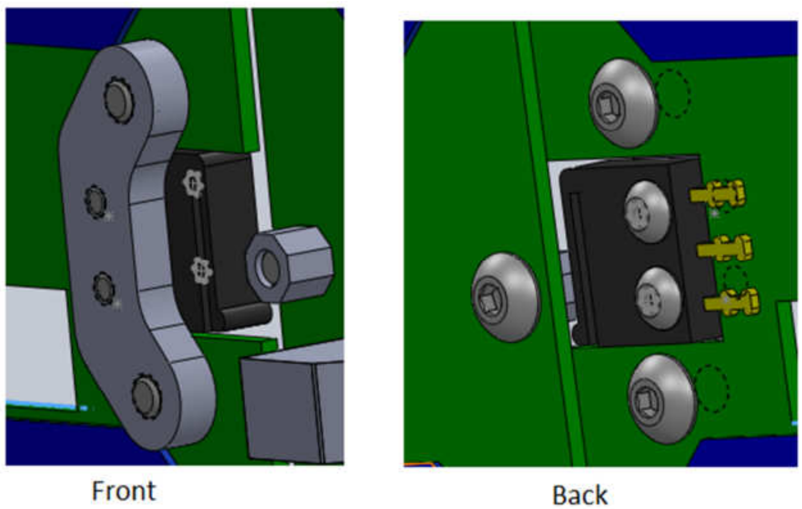


Figure 4.20 Deployment Verification Design

Structure

Tesseracts structural design was driven by the CubeSat specifications and the need for mounting. Nearly all parts contributed to the final design of Tesseract's skeletal structure. In this section we will discuss how each part or the CubeSat specification contributed to the rail panel and beam build-up style. The major components in this section include:

- Rail Panel
- Beams
- Top Hat/Shoe
- Stack Stabilizer
- Battery Mount

The Tesseract Rail Panel, shown in Figure 4.21, is the current design for the rail panel. It includes mounting locations for the numerous components of the satellite while providing a framework for a CubeSat-approved satellite.

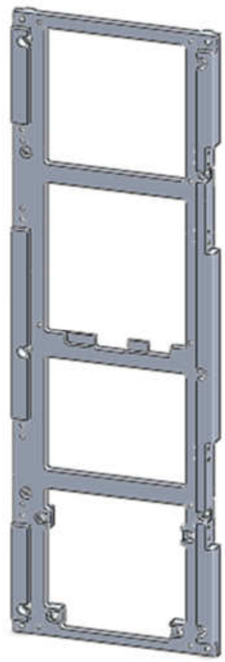


Figure 4.21 Design for the Tesseract Rail Panel.

The general rail panel dimensions are determined by the 3U CubeSat specifications attached in Appendix B. The satellite must conform to these specifications to launch as a CubeSat in the PPOD. The driving specifications for the rail panel dimensions are outlined below:

- Each side of the CubeSat must be 100mm from one edge of the rail to the other (Figure 4.22)
- Each rail must be sized to a minimum of 8.5mm x 8.5mm (Figure 4.22)
- The height of the CubeSat from the top of the hat to the bottom of the shoe must be 340.5mm (Figure 4.23)

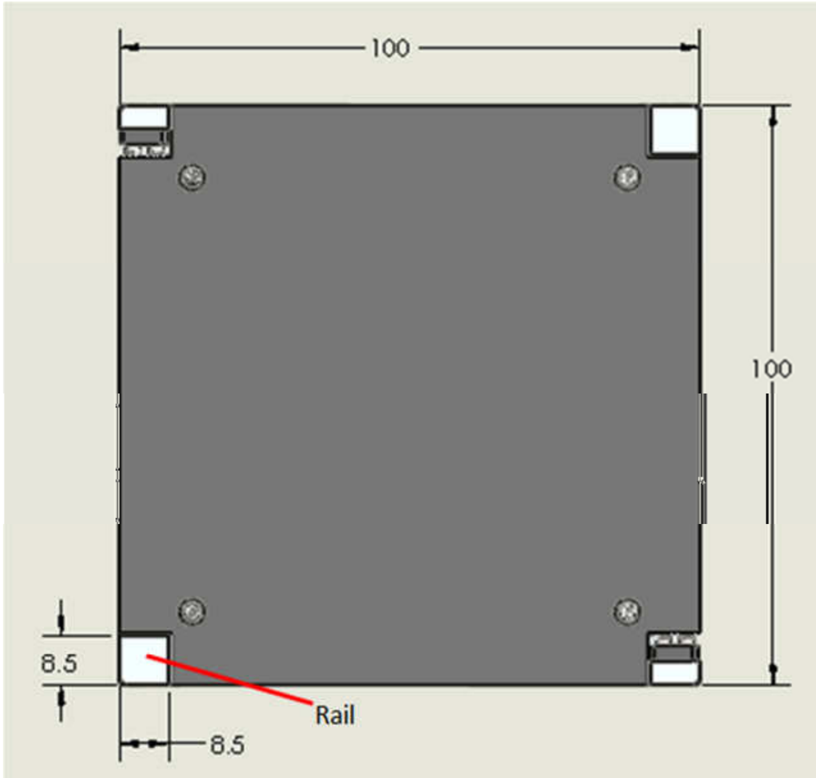


Figure 4.22 CubeSat 100mm square profile and 8.5mm square rail viewed from Z-axis.

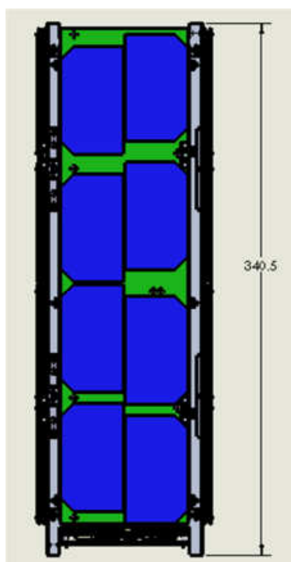


Figure 4.23 CubeSat height requirement of 340.5 mm from edge of hat to edge of shoe.

The recess of the rail panel was sized to fit permanent side panel. The width of the structure was limited to 100mm and with 8.5mm rails on each side, the recess was fixed at 83mm. Unlike the width dimension, the height dimension had some variability. The height of the rail panels is dependent on the heights of the hat and shoe. The hat and shoe together added a height of 36mm to the rail panel height. This locked the rail panel height dimension at 340.5mm.

In addition to conforming to CubeSat specifications, the rail panel also includes mounting locations for the support beams, the hinges, the battery mounts, the stack stabilizer, the top hat and shoe. An assembly with all these components is illustrated in Figure 4.24. An exploded view with the structural components can be found in Appendix B.

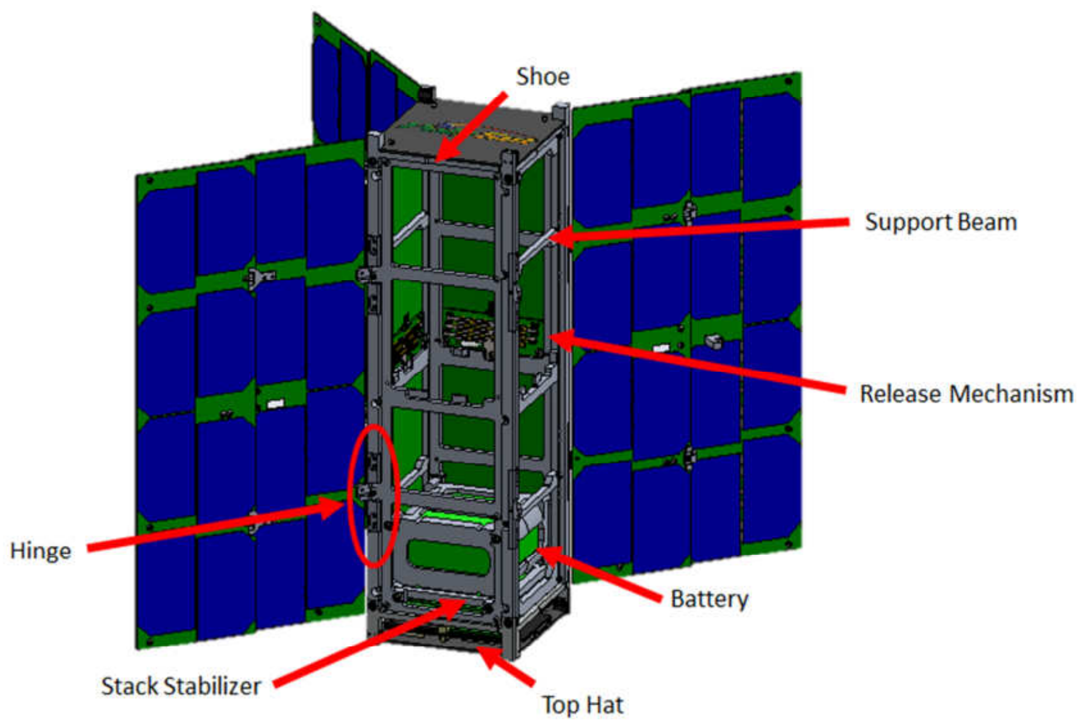


Figure 4.24 Components supported by the structure

The rail panel has beam and top hat mounting points within the rail of the structure. The mounting locations for the beams were designed such that the recess for the permanent side panel is the same depth on all sides. The vertical locations of the beams were designed to adequately secure the side panel to the structure, and, in the case of the deployment beam, to be utilized in the release mechanism.

The support beams are the support between the two rail panels, making up the two other walls of the satellite. The beams, like the rail panel, offer mounting points for components such as the side panels. The beams used for the Tesseract structure are shown in Figure 4.25.

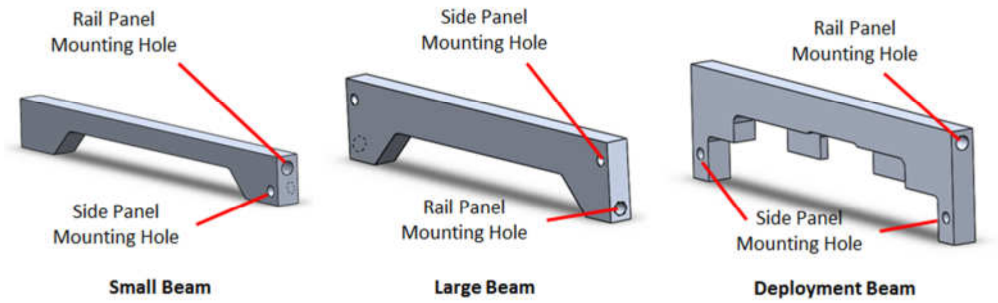


Figure 4.25 Tesseract beams.

Tesseract began with three identical beams evenly distributed along the structure. However, as designs of the release mechanism, solar cell configuration, and top hat progressed, three different beams emerged. The small beam was the original beam designed for Tesseract. A critical load analysis was performed on this beam to determine if it would buckle under the NASA GEVS loads. The calculation, attached in Appendix E reveal that the small beam is able to support 170lbm (77kg) at the 14g NASA GEVS level - a safety factor of 17 if the whole weight of the satellite rested on a single beam. Additional beams were added along the structure for the purpose of mounting rather than for support of a heavier load. The small beam features a mounting hole for the side panel, along with a hole for mounting onto the rail panel. The two holes must be vertically offset from each other to prevent the rail panel mounting screw from clashing with the side panel mounting screw. The vertical offset is featured in all three beams. The deployment beam is designed as part of the deployable panels' release mechanism. The features in the beam, restrict excess movement of mechanism components. The large beam is a modification of the small beam. The height of the beam had to be adjusted such that the screw for mounting the beam to the rail panel would not interfere with the screw for mounting the hat or shoe to the rail panel. While the cross section of the small beam is already able to support any expected loads, the increase in cross sectional was a designer's aesthetic choice, not driven by structural strength. Additional analysis could result in additional efficiencies in mass and strength. Figure 4.26 shows the top hat mounting points onto the rail panel.

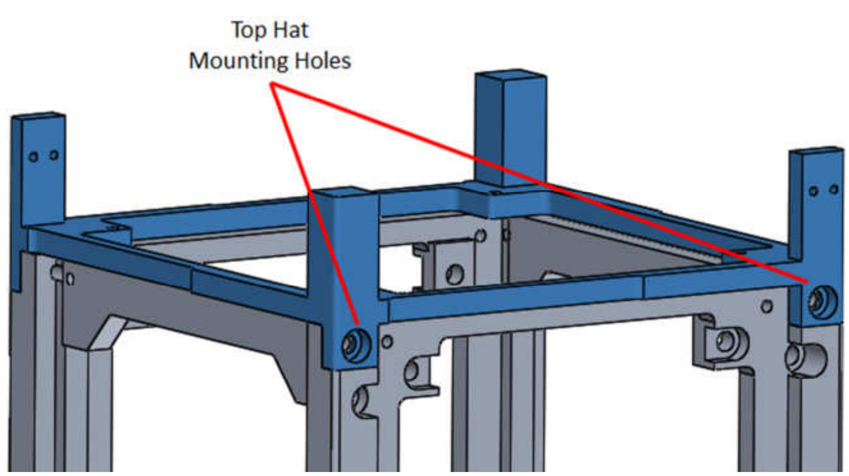


Figure 4.26 Top hat (blue) mounting points onto the rail panel (grey).

The hat and shoe are used for holding the avionics stack and the +Z-board, respectively, and do not feature the recess. The mounting points for each were positioned approximately centered on the rails at each end of the structure. The hat was modified to allow for easy access to the stack at any point during assembly. The hat's taller pegs accommodate the boards of the avionics stack, while allowing room for an antenna. Figure 4.28 is an assembly of the top hat with the avionics stack.

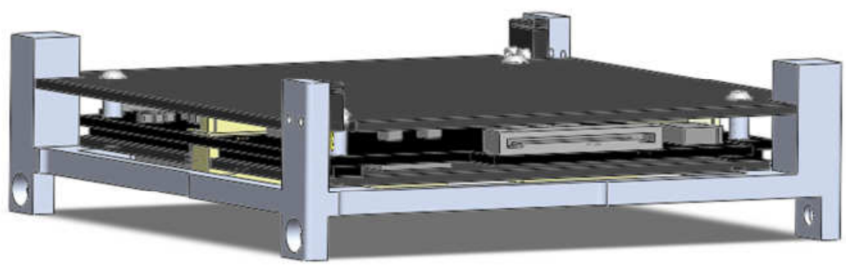


Figure 4.27 Tesseract top hat and avionics stack.

The Tesseract top hat was revised by Oliver Woolsoncroft of the PolySat team. The new hat places all boards of the avionics stack above the mounting material of the hat. This relieves the issue PolySat members had of forcing electrical boards of the assembled stack into position on the top hat. With the new hat, the avionics stack can be easily separated from the rest of the structure when the top hat is removed. This feature was made possible with the addition of a hub board.

The hub board is a new board developed by the PolySat team. All flat-flex cables, which are typically routed onto boards on the stack, are now plugged into the hub board. The hub board then connects to the avionics stack through an inter-board connector which runs across the board. A stack stabilizer, shown in Figure 4.28, is used to secure the stack to the structure.

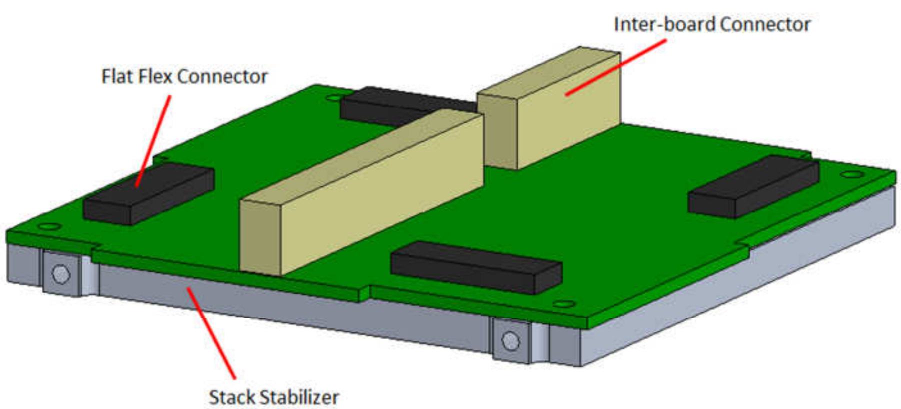


Figure 4.28 Stack stabilizer with hub board showing connectors.

The hub board mounts to the rail panels, right below the avionics stack. Its mounting holes, along with the battery bracket mounting holes, protrude into the "windows" - the square cutouts in the rail panels. The holes mounting the stack to the structure were horizontally offset inward such that their screws did not interfere with the screws mounting the hub to the stabilizer. The battery bracket assembly, along with the stack stabilizer, are shown mounted onto the structure in Figure 4.29.

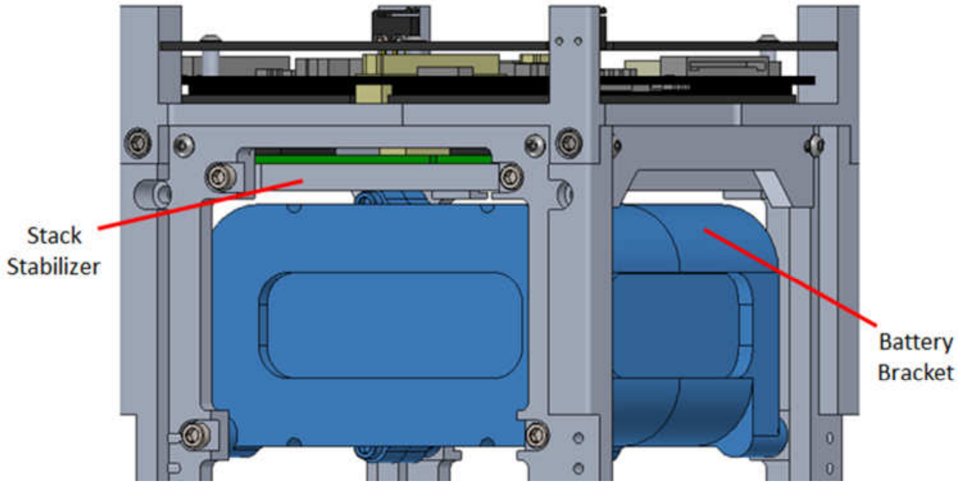


Figure 4.29 Battery bracket and stack stabilizer mounted to the rail panels

The battery bracket, designed by Wesley Williams from PolySat, was designed to fit eight batteries. The battery bracket is able to slide into and out of the structure from either of the Z sides. The mounting points of the battery bracket to the structure are shifted inward to allow for simplified features for manufacturing.

All structural and heavy components are secured by 4-40 screws. While each 4-40 screws provides more than enough support to each component, PolySat has requested that these screws be used on the structure because they are easier to tap in the manufacturing phase. Smaller and lighter components, such as the hinges and electrical boards, utilized 2-56 screws because of size restrictions. The only component utilizing 0-80 screws were the required deployment switches. Since the deployment switches are required, they were not swapped out. However, access to the 0-80 locations was improved, which will allow for smoother manufacturing and assembly.

The structural design appears to be an overall success. Further testing will be required to ensure that the structure meets all CubeSat requirements and to prove its reliability. For more details on the dimensions of each component, see Appendix B.

Changes to Final Design

There were issues with the design that we noticed after building the 3D printed model. The biggest issue is the lack of rigidity in the deployable panels in both stowed and deployed configurations. In the stowed configuration, the deployable panels are constrained by the hook, but because the solar panels are so thin, they have large deflection away from the hook, especially at the top and bottom of the panels. There are standoffs in between the panels that will prevent the solar cells from clashing into each other, but there is nothing preventing the Second Deployable from deflecting outwards during vibrations. If the First and Second Deployable panels are rattling too much due to the large accelerations from launch, there is a high probability that the solar cells will get damaged.

We had brainstorming sessions with the rest of the PolySat team, but most of the ideas required that we redesign many other components. There was one idea that did stand out in its simplicity, but the team was unsure if it would actually fix the problem or just decrease the severity of the problem. The idea was to have two miga motors on the top and bottom of every recessed panel instead of one in the center. Having the panels constrained near the top and the bottom would decrease the amount that the panels would deflect during launch. To verify that this new configuration would work, Oliver Woolsoncroft decided to analyze deflection in the panels using Finite Element Analysis and the NASA GEVS Random Vibration Levels (discussed later in Vibration Testing section). Although the model used was a simplified version of the panels, the results gave us the confidence to make the changes to Tesseract.

Figure 4.30 shows the image of the revised Tesseract structure. The placement of the Miga motors can be seen in the structure. Because there are now two hooks going through the recessed solar panel, there are now only six solar cells attached to the recessed solar panel instead of eight. The solar cells on the deployable panels had to be squeezed toward the center, to make room for the hook, so deployment switch was removed. Another method of verifying deployment would be to look at the power being generated from the solar cells. The amount of power being generated should increase significantly when the panels are deployed.

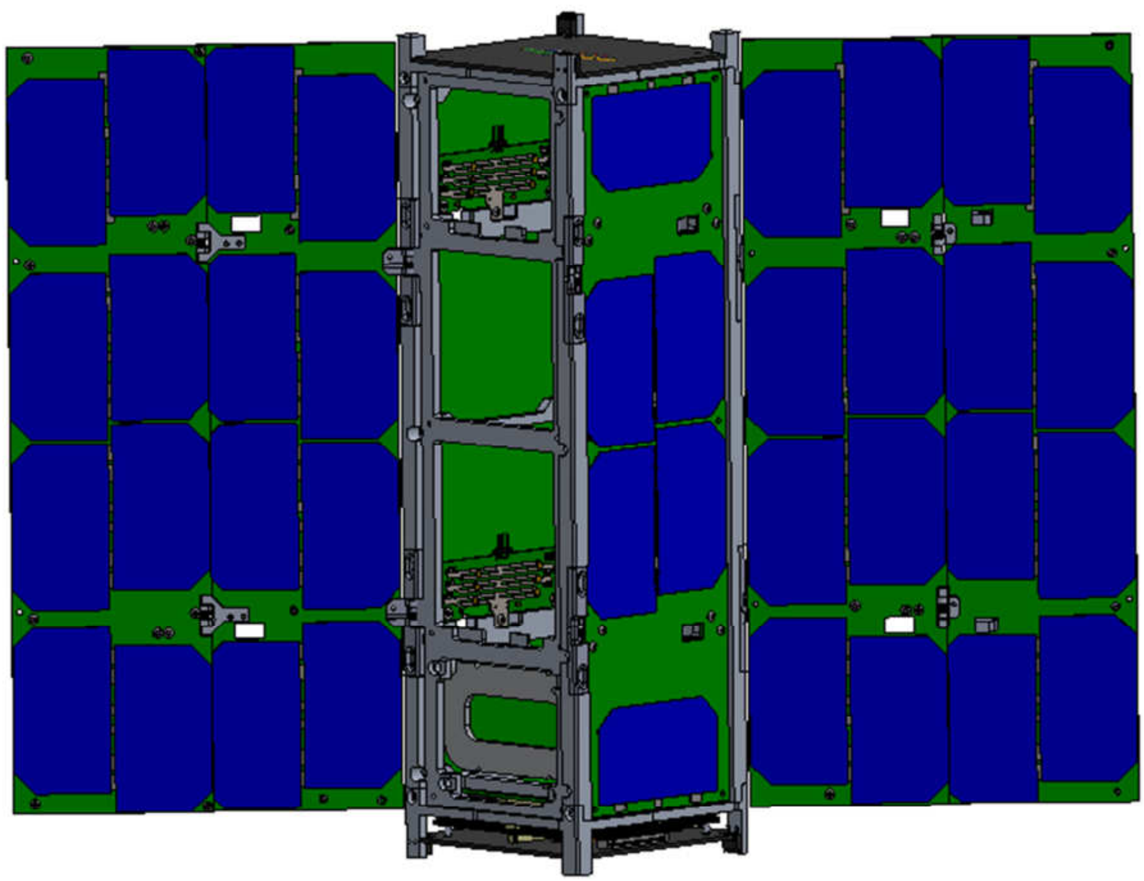


Figure 4.30. Final Tesseract design (one recessed panel removed to view internals).

The rail panels had to be modified as well to incorporate two Miga motors. Figure 4.31 shows an image comparing the rail panel divisions. The new panel (left) has two Deployable Beams, one Large Beam, and two Small beams. These small changes added a few extra components to our structure, but the amount was minimal when compared to the changes that would have been made if a different solution was pursued.

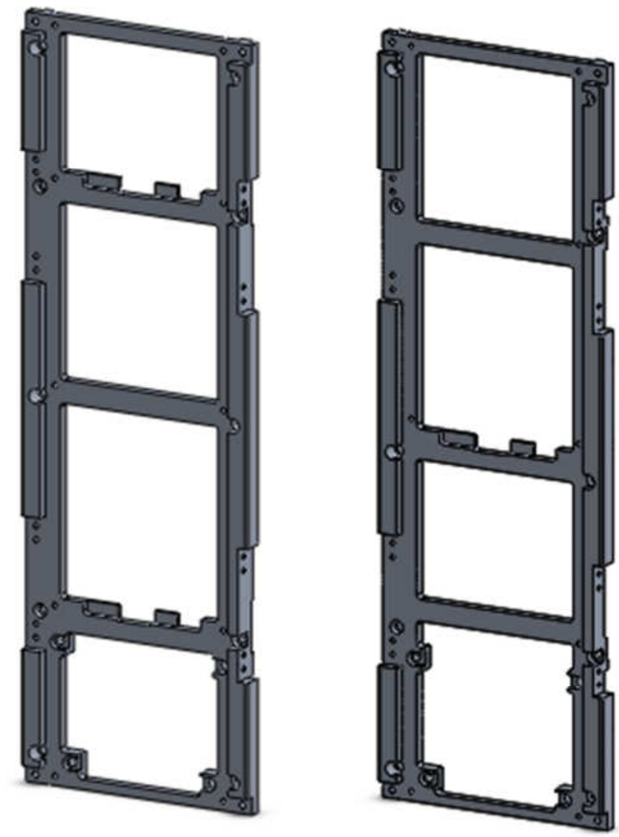


Figure 4.31. New panel design (left) and previous panel design (right).

Another issue that was concerning was that the panels on the RP model were very flimsy during deployment. Some analysis was conducted on the motion of the Panels to investigate the effect of different torsional springs. Fortunately, the FR-4 solar panels that would be used on the engineering test unit (ETU) arrived early. After mounting them on the RP model and deploying them, it was obvious that these solar panels did not swing back and forth. The FR-4 boards used on the RP model were slightly different from those used on the ETU. The RP model boards lacked the inner copper layer which actually allowed electronic traces to be made in the board. The new solar panels, used on the ETU, included the thin copper layer which made the panels stiffer.

Cost Analysis

Tesseract aimed to be machined and manufactured almost entirely at Cal Poly. The PolySat team machined some components listed Table 4.1 in the Cal Poly machine shops. The estimated cost for stock material, along with the cost of parts that were sent out is included in the table.

Table 4.1 Cost Analysis on stock.

Part	Off-the-Shelf Material Cost (Sent out Cost)
Rail Panels + Soft Jaws	$178.64
Battery Brackets	$15.00
Beams	$10.00
Stack Stabilizer	$6.19
Top Hat	$30.00
Shoe	$30.00
Hooks and Talons	$6.19 ($388.78)
Hinge Components	$6.19 ($1,184)
Total	**$282.21 ($1,854)**

During the final stage of manufacturing, Tesseract had some components machined on campus while some parts were machined off campus due to time constraints. Future PolySat members will have access to all the machining files to manufacture all above components on campus.

5) Description of the Final Design

Manufacturing Processes

One of the main goals for Tesseract was to design the components so that the majority could be machined using the CNC mills on campus. Table 5.1 is a list of all the parts that needed to be machined for Tesseract. Due to the limited time that we were able to use the CNC mills on campus, a few of the parts were manufacture by First Cut. CAM programs were created for each part using HSMWorks, even for the parts that were not manufactured on campus, so that future Polysat members will have access in case they need to remanufacture a specific component. The parts that were not machined on campus were the smaller parts such as the Hinge Blocks, the Talons, and the Hinge Leafs. The battery bracket and the stack stabilizers were not manufactured in time for the senior expo, but they will be manufactured during the summer by Polysat members.

Table 5.1 Machined components.

Parts	Qty
Rail Panel	2
Small Beam	4
Large Beam	2
Deployment Beam	8
Battery Bracket A	1
Battery Bracket B	1
Hinge Block A	8
Hinge Block B	8
1st Leaf	8
2nd Bottom Leaf	8
2nd Top Leaf	8
Hook	8
Talon	8
Stack Stabilizer	1
Top Hat	1
Shoe	1

Rail Panel Manufacturing

Manufacturing and assembly were major considerations in the Tesseract Rail Panel design. Even with considerations for manufacturing, the Rail Panel still includes complex features to accommodate the deployable subsystem and minimize mass. For this design, soft jaws were created to hold the panels during machining. Figure 5.1 is the panel in the CNC machine.

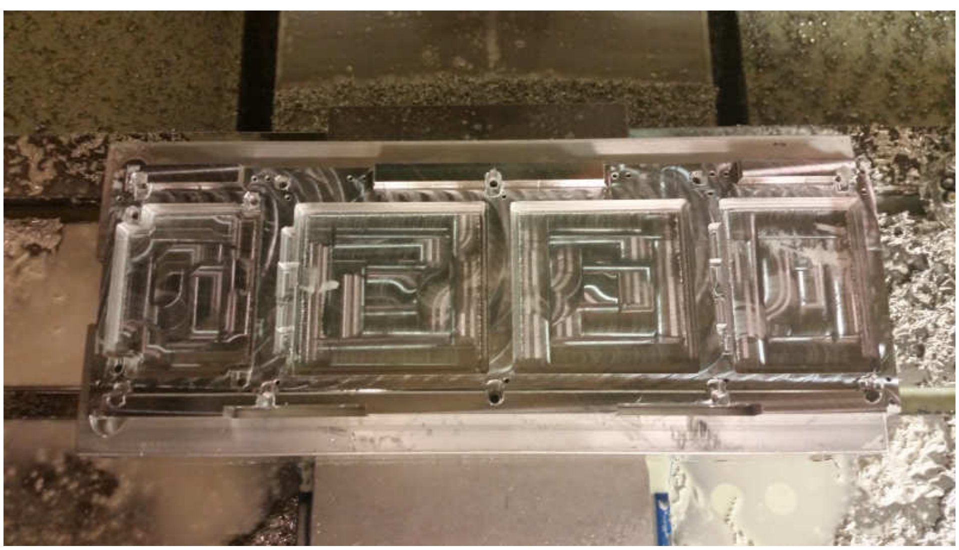


Figure 5.1 Rail Panel after the second machining operation.

Figure 5.1 shows images from HSMWorks after each operations for the Rail Panel. Features for the side hinges are machined out in Operation 1. The outer features, which include the recess, counterbores, tapped panel mounting holes, tapped hat and shoe holes, deployment features, and the windows are created in Operation 2. Operation 3 finishes the part by removing extra material behind the windows and developing the interface to the beams, battery bracket, and stack stabilizer.

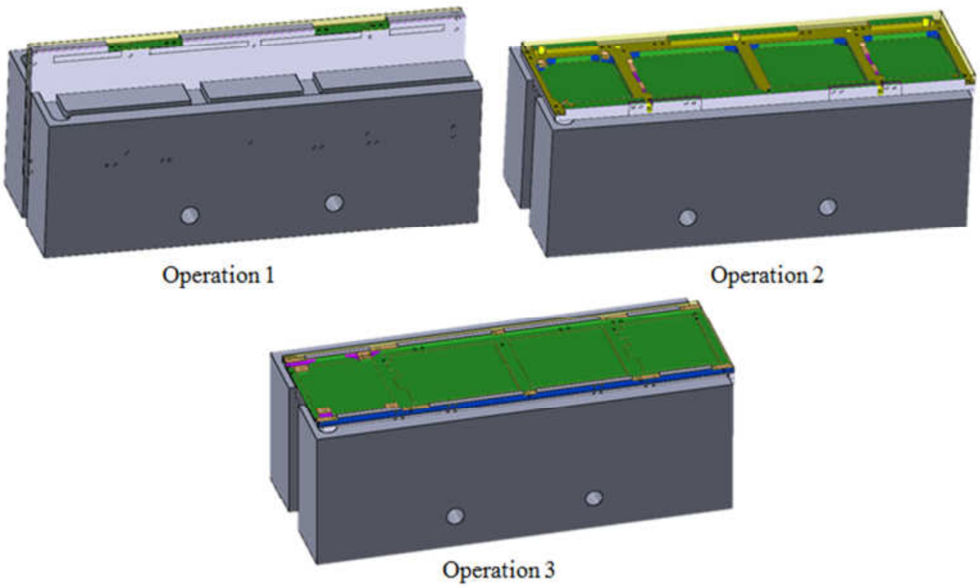


Figure 5.2. Rail Panel manufacturing operations.

One time-saving tool that was used for the Tesseract panel was the chamfer tool. The chamfer tool removed most burs on the part, and created smooth edges. In addition, power tapping was utilized for all tapped holes. For future manufacturing, an interface plate which the panel can mount to after drilling a few holes will help keep all features within desired limits and reduce any bending during the machine operations.

Top Hat/Shoe Manufacturing

The new design of the Top Hat really simplified the manufacturing process, but there were still a few features that require attention. The Top Hat and the Shoe are two parts that are nearly identical except the pegs on the Top hat are about 8mm longer than the pegs on the Shoe. The beams on the Top Hat are only 3mm in thickness, so having a fixture that will provide support and minimize deflection is ideal. Figure 5.3 shows the fixture created to manufacture the Top Hat and the Shoe. The features on the fixture were designed to support the the beams when machining the outside profile of the Top Hat. The Top Hot is secured onto the fixture using four #4-40 screws that go through the counterbore holes on the sides of the Top Hat.

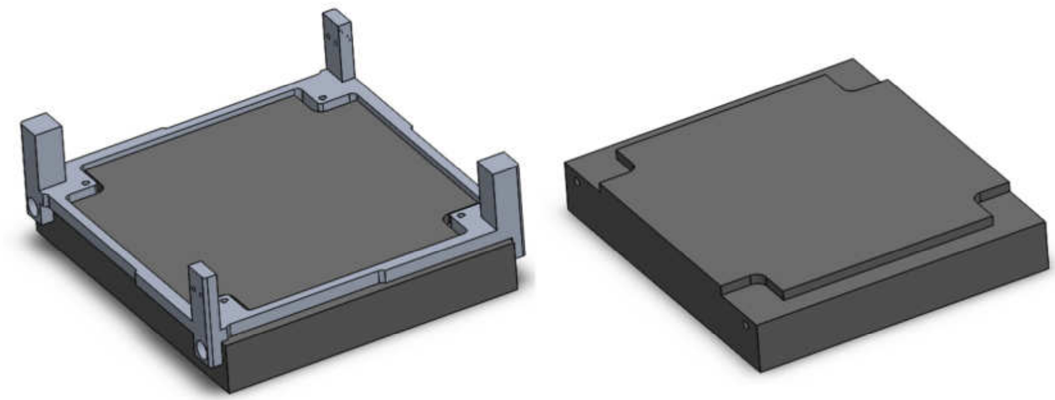


Figure 5.3 Top hat and fixture.

Figure 5.4 shows images from HSMWorks after each operations for the Top Hat. The Top Hat is mounted to the fixture after operation 1 as shown in Figure 5.5.

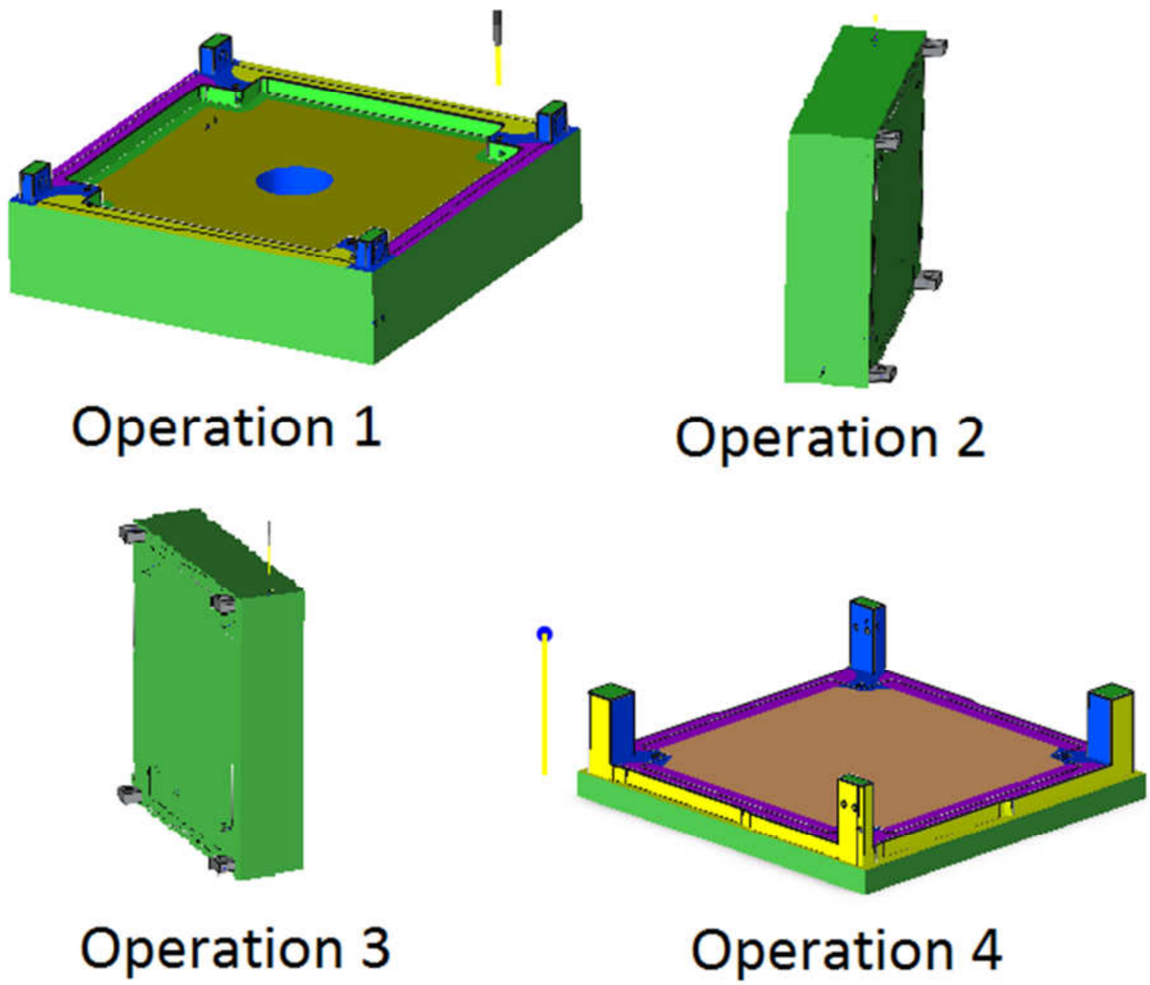


Figure 5.4. Top Hat machining operations.

Figure 5.5 Top hat with bronze shims for spacing.

There were some issues that we ran into when machining the shoe that were corrected before we began machining the Top Hat. The main one resulted from the clearance of 0.02in allowed between the fixture and the Top Hat/Shoe as shown in Figure 5.6. A clearance of 0.004 inches would have been more appropriate. The screws produced a compressive force where the red arrows are located, which caused the beams of the Shoe to bow (as demonstrated by the red line) as more material was removed. As a result, bowed portions of the hat are thinner than designed. Fortunately, this defect did not have a major impact on the part and could be overlooked. The mistake was corrected on the Top hat by adding bronze shims in between the fixture and the top hat as can be seen in Figure 5.5.

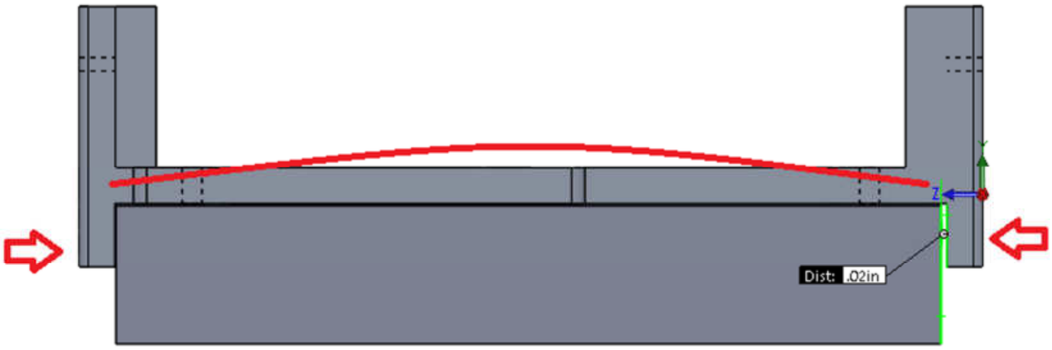


Figure 5.6 Clearance between fixture and Top Hat.

Beams

The deployable beams, the Large beams, and the Small beams were all manufactured using similar operations. The features were different for each part, but the set up process was similar for all the beams. Figure 5.7 shows images from HSMWorks after each operation for the Deployable Beam. Figure 5.8 shows an image of the Deployable beam in CNC mill after the first operation.

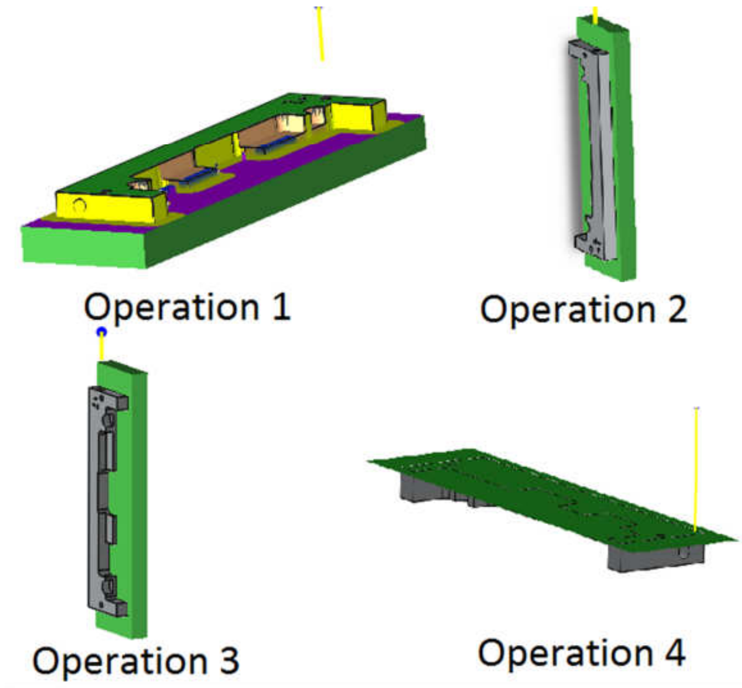


Figure 5.7 Deployment Beam Operations.

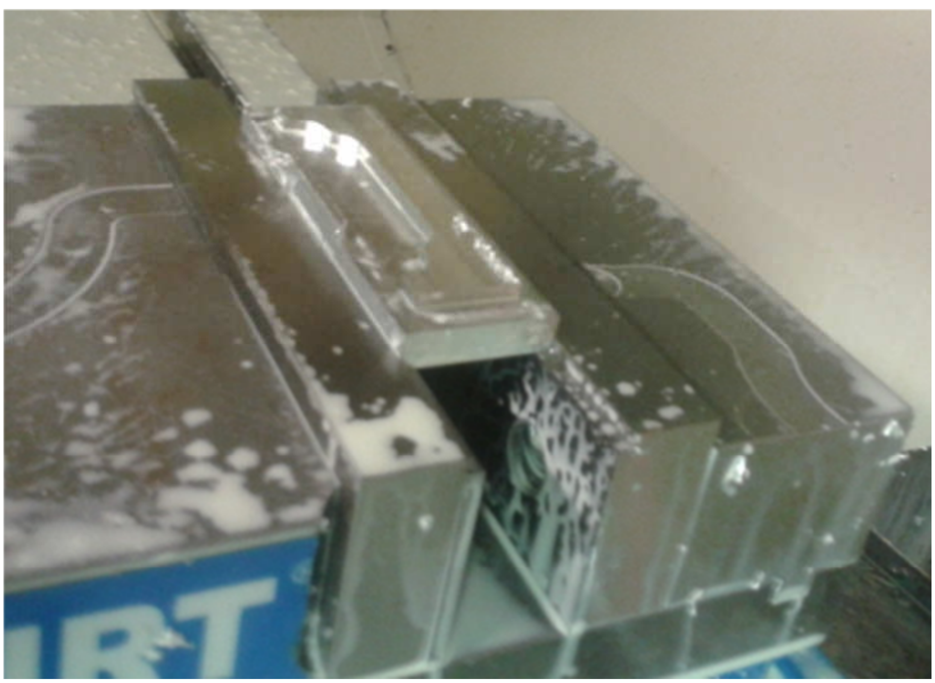


Figure 5.8 Deployment beam after the first operation.

Assembly

Tesseract assembly was another major consideration of the design. PolySat requested that the structure be easy to assemble and disassemble whenever necessary. Overall, the structure was relatively simple to put together, even with the large number of parts.

To ensure that Tesseract had a feasible design for assembly, procedures were created with the rapid prototype model. The team created six subassemblies and an overall assembly to break down the assembly process. The subassemblies are for the first set of hinges, the recessed side panels, the deployable panels, the batteries, the top hat, and shoe. Figure 5.9 shows the product of each subassembly procedure.

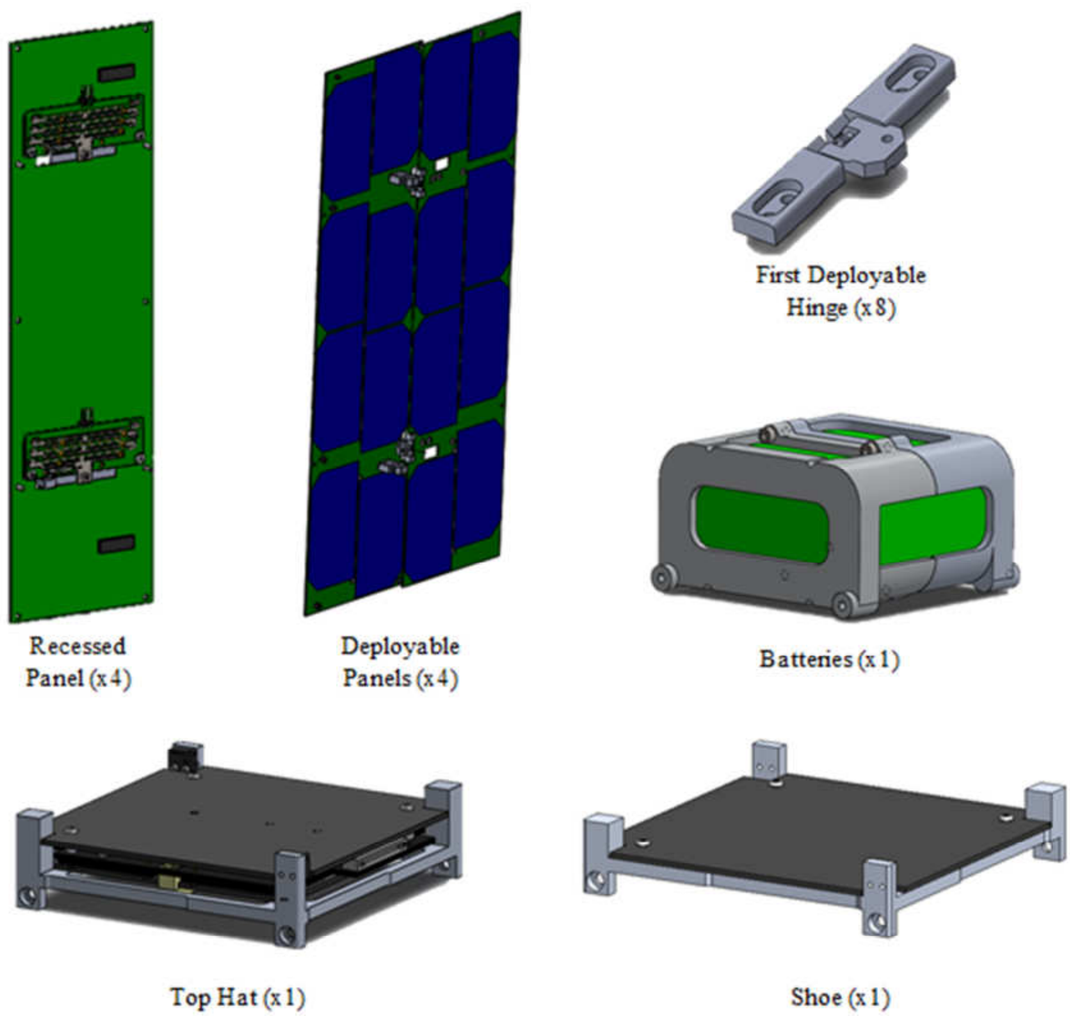


Figure 5.9 Subassemblies.

Once the subassemblies are complete, the overall assembly involves integrating above subassemblies to the beam and rail panel structure. During the assembly run-though, the team ran into a slight obstacle. The rail panel to beam interface was designed at too small of a tolerance. The beams were not able to fit within the cutouts in the rail while also allowing a screw to be threaded in. To solve the issue, the cutout was widened by filing down material. Figure 5.10 shows the beam-panel interface.

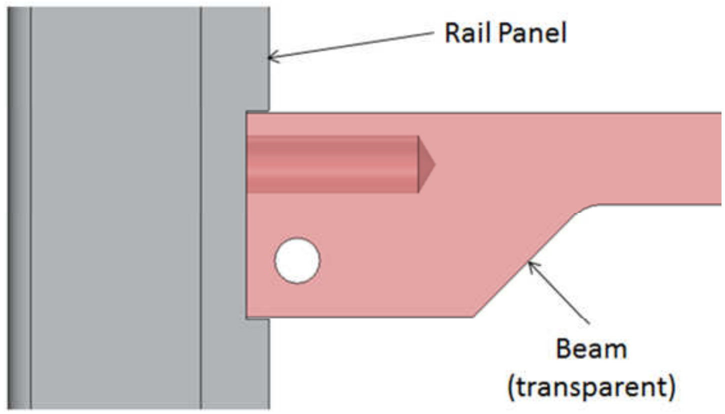


Figure 5.10 Rail Panel and beam interface.

Once more material was removed, the structure and the subassemblies were integrated. The structure alone provided easy access to the internals. Electronic components can easily be removed without much risk of damaging connections.

6) Design Verification Plan (Testing)

The following is a discussion of our Design Verification Plan. A portion of our DVPR is shown in Figure 6.1. The full document can be found in Appendix A. Our DVPR outlines the required testing to verify that our design meets our project needs. The two tests that did not meet the requirement are the inner volume check and the rail panel protrusion check.

DVP&R Tesseract													
Report Date	20-Jun-15	Sponsor	Polysat						Component/Assembly		REPORTING ENGINEER:	Edgar Uribe and Vanessa Fuane	
TEST PLAN										TEST REPORT			
Item No	Specification or Clause Reference	Test Description	Acceptance Criteria	Test Responsi	Test Stage	SAMPLES		TIMING		TEST RESULTS			NOTES
						Quantity	Type	Start date	Finish date	Test Result	Pass	Quantity Fail	
1	Deployment of First Panel	Proper Deployment Test	135°±10° from recessed panel	Edgar	DV	10	B	1/8/2015	1/12/2015	130	PASS		
2	Deployment of Second Panel	Proper Deployment Test	180°±10° from first deployable panel	Edgar	DV	10	B	1/8/2015	1/12/2015	180	PASS		
5	Structure Weight	Weigh Structure with Electrical Components	Max 1.5 kg	Edgar	PV	1	C	3/30/2015	5/1/2015	.71 kg	PASS		
6	Inner Volume	Measure Inner Volume available for Payload	min 2140cm cubed	Vanessa	PV	1	C	3/30/2015	5/1/2015	2015cm cubed		FAIL	There is still plenty of room for a wide range of payloads.
7	No Solar Cell Damage	Solar Cell Damage Test	Solar Cells cannot collide with anything	Edgar	DV	10	B	1/8/2015	1/12/2015	N/A	PASS		
10	Vibration Testing	Vibration Testing	Cubesat Specs	Vanessa	PV	1	C	3/30/2015	5/1/2015	N/A	PASS		
11	Thermal Chamber Testing	Deployment Test at Max and Min Temperatures	Deploy at 140 F and -20 F	Vanessa	PV	1	C	3/30/2015	5/1/2015	100%	PASS		
12	Rail Plane Protrusion limit	Measure Outmost component of second deployable panel	Max 6.5mm	Edgar	PV	1	C	3/30/2015	5/1/2015	6.6mm		FAIL	Still able to fit inside the PPOD. Ensure all screws tight and all interfaces are clean before assembly.

Figure 6.1 Test Plan from DVPR

Vibration Testing

Tesseract was tested under the NASA General Environmental Verification Standard (GEVS) Random Vibration Acceptance profile, shown in Table 6.1, to verify structural soundness of the design. During the rapid prototype stage, the deployable panels exhibited large amounts of deflection when shaken by hand. The survivability of the solar cells mounted on the deployable solar panels became a key focus of the test.

Table 6.1 NASA GEVS Random Vibration Acceptance Levels.

Frequency (Hz)	ASD Level (g^2/Hz)
20	0.013
20-50	+6 dB/oct
50-800	0.08
800-2000	-6 dB/oct
2000	0.013
Overall	10.0 G_{rms}

The availability and cost per solar cell limited the number of cells put on Tesseract to two cells for the test. The most useful mounting location for the cells was determined to be the points of maximum deflection. Finite element analysis of the deployables was used to find positions of max displacement, and the solar cells were placed accordingly, shown in Figure 6.2.

Figure 6.2 Solar cells on deployable solar panels.

Tesseract was placed in a TestPOD for the vibration test, which took place in the Mechanical Engineering Vibration Laboratory (Bldg. 13 Rm. 101). While the test's main objective was to check survivability of the cells, accelerometer response data was additionally collected during the vibration. The vibration test set-up is shown in Figure 6.3. Tesseract did not have all four sets of solar panels placed on the structure for the vibration test. A single set was tested in the worst-case configuration. The results for the tested set of solar panels are predicted to be the same for all sets in the full assembly.

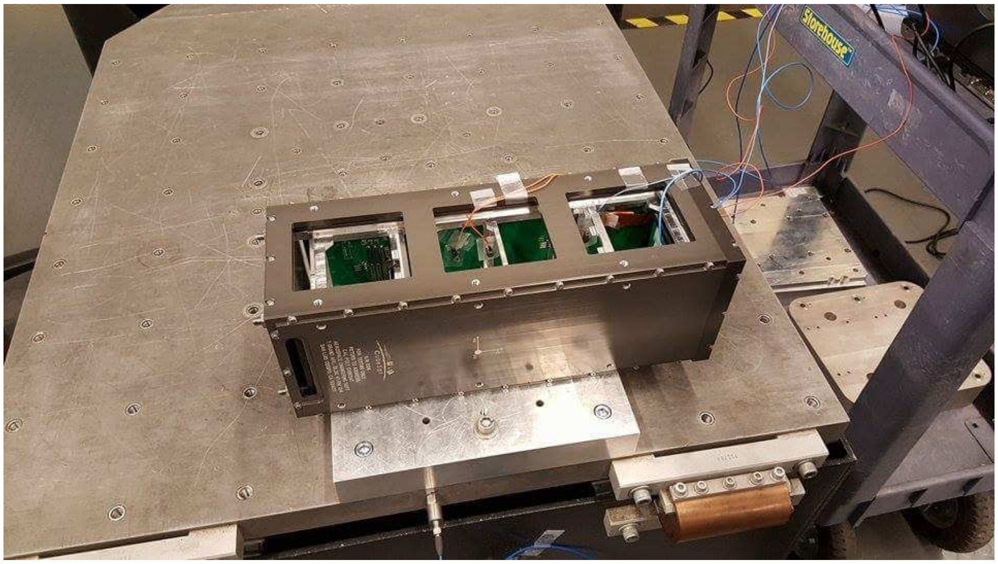


Figure 6.3 Tesseract vibration test set-up.

The condition of the cells, placed in the most high-risk area, will determine whether or not Tesseract can support both sets of deployable solar panels. Broken cells constitute a redesign of the structure and deployment system, while in-tact cells validate the design. The requirements for passing the vibration test are shown in Table 6.2.

Table 6.2 Requirements for passing the vibration test.

Pass Requirement	Pass? (Y/N)
Deployable panels do not deploy during the vibe (or any time prior to the deploy signal).	Y
All screws and nuts remain in their proper location, without backing out.	Y
All structural components do not displace from their proper positions, and are not damaged from the vibe.	Y
All electrical components and cabling do not displace from their proper positions and are not damaged in any way.	Y
The test solar cells are not damaged in any way.	Y
The Miga Motors are able to pull the talon and relase the deployable solar panels.	Y

Tesseract passed all requirements for the vibration test. The test is a success with no broken solar cells and with response levels below any material yield conditions.

Deployment of Panels (Thermal chamber)

Tesseract was additionally tested in a thermal chamber. The purpose of this test is to check that the satellite is still operational in normal flight conditions, as well as to determine if it can handle the two extremes of the temperature range. Testing, such as deployment in a thermal chamber, will be crucial in determining the reliability Tesseract deployables. Figure 6.4 shows the internals of the Thermal Chamber in the Polysat lab.

Figure 6.4 PolySat Thermal Chamber with Tesseract inside.

The structure underwent a deployment test at the highest and lowest temperatures that the interior of the satellite may experience once in orbit. The objective of the test was to ensure that the solar panels deploy when current is provided to the Miga motor under maximum and minimum temperatures.The testing occurred at 140F and -20F. The input voltage to the Miga motor was 3.3V.

Tesseract was first tested at 140 F. The thermal chamber was ramped up to 140F and allowed to stabilize. When the temperature on a digital thermometer (which had a thermocouple taped to the recessed panel) read 140 F, 3.3V were delivered to the Miga motor using a power source. The panels were deployed five times under these conditions. The experiment was then repeated at -20 F and deployed five more times.

The panels deployed successfully every time that current was sent to the Miga motor. We noticed that the panels took longer to deploy at the minimum temperature, but that was expected because it takes the shape memory alloy on the Miga motor longer to heat up to the required temperature under these conditions. Slower deployment would not be an issue because timing of deployment is not a factor for successful deployment.

Measurement Checks

The internal volume of the satellite must be measured, once everything is assembled because one of our goals was to make a modular structure that would be able to carry a wide variety of payloads. The interval volume of the structure available for the payload is 2015 cm cubed. Although this number is smaller than the minimum reported on the DVPR, we find it to be acceptable nonetheless. The minimum volume reported on the DVPR was 2140 cm cubed, which was the inner volume of ExoCube. Our aim was to increase this value, but with the incorporation of the deployable solar panels, the rail panels had to be recessed, which minimized the inner volume. The Tesseract structure still presents plenty of room for a wide variety of payloads.

The mass of the entire structure must be no more than 1.5kg, so that the payload can have a mass of 2.5kg, if needed. After assembling the machined structure and adding the electrical boards, the weight of the entire structure was 0.71kg. This is about half of the maximum desired weight and leaves 3.1kg available for the payload.

Finally, we must check that the deployable panels or any other component do not protrude more than 6.5 millimeters from the plane of the rails. The design only allows approximately 0.1mm of clearance between the top of the outermost screw head and the 6.5mm limit. During the measurement check before integration into the TestPOD, the outermost screw head measured about 6.6mm. The CubeSat representative allowed integration into the TestPOD because the extrusion was still within the additional tolerable limits. A hardware review in which all screws are torqued and all interfaces are cleaned must be made before future assemblies to ensure the tightest interfaces.

7) Project Management Plan

Due to the small size of our team, the organization is more of a partnership. We both overlook each other's work to ensure that we stay on schedule and all tasks meet the requirements. While gathering background information, Vanessa focused on the main structure, including the Top Hat, and Edgar focused on the deployables. Other responsibilities throughout the scope of the project were assigned during weekly meetings.

The Senior Project team met once a week on Wednesday afternoons and the Polysat team met every two weeks. A team contract has been signed by each member that specifies the type of conduct and attitude to bring to these meetings. All meetings are held in the PolySat lab and attendance is mandatory. Table 1.2 below shows important dates throughout the scope of the project.

Table 7.1 Timeline of Tesseract Senior Project

Deliverable/Task	Start Date	End Date
Design Fixtures for Manufacturing	1/12/15	2/15/15
Cut Stock	1/12/15	2/15/15
CAM Parts	1/26/15	2/15/15
Machine Hinge Components	2/2/15	2/9/15
Machine Top Hat and Shoe	2/9/15	2/16/15
Machine Beams	2/16/15	2/23/15
Machine Hooks, Talons, and Battery Brackets	2/16/15	2/23/15
Machine Rail Panels	2/23/15	3/2/15
Compose Detailed Assembly Procedures	2/16/15	3/13/15
Compose Testing Procedures	3/13/15	3/20/15
Assembly	3/30/15	4/6/15
CubeSat Specifications Checklist	4/6/15	4/6/15
Internal Component & Stack Access Test	4/6/15	4/13/15
Vibration Testing	4/13/15	4/27/15

Thermal Testing	4/27/15	5/11/15
Final Report	5/11/15	5/31/15

The above schedule was pushed back further than anticipated as the team struggled with changes to the design.

8) Conclusions and Recommendations

At this point the Tesseract senior project is complete. Most requirements were met and we are mostly satisfied with the final outcome. The project has been well documented, so that future PolySat members can use it as a resource for future missions. Many of the design decisions are explained in this report and other design review presentations. CAD models and drawings have been created for every part as well as manufacturing procedures. Testing and assembly procedures have also been documented.

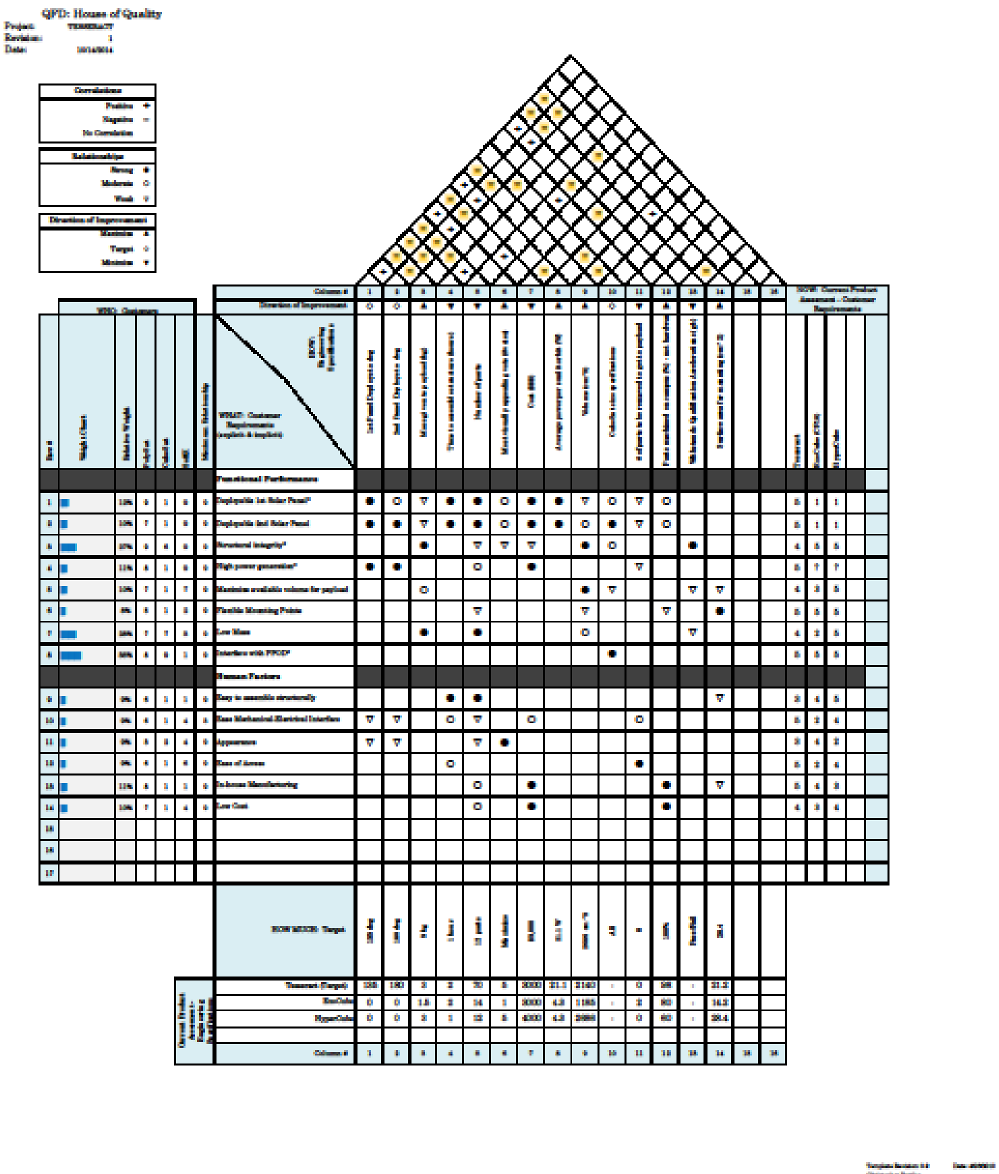
Based on our experience with this project, we recommend that future senior project teams or PolySat members put in extra effort to create a reasonable schedule and to ensure that they follow that schedule. There are many issues that are not anticipated during the design phase that come to light after building a prototype. If the issues are critical to the design objectives, extra time must be allotted to redesign and fix the problems. Extra time must also be allotted for manufacturing. After speaking to a few experienced shop technicians, many suggest to conservatively calculate the amount of time it will take to machine a part and double it.

Parts of the design that should continue to be explored include the dipole antenna mount and the extrusion limit for the deployables. Tesseract is still getting close to the 6.5mm limit. Research and development to decrease the extrusion should be looked into further.

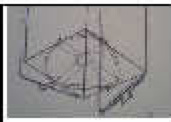
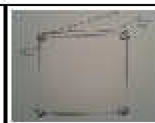
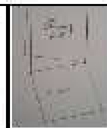
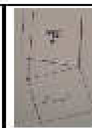
Overall, the team learned more about designing and manufacturing CubeSats. Tesseract was deemed a success, with only a few improvements to be made. We hope to see the Tesseract structure being used as a baseline for future missions.

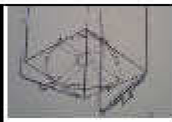
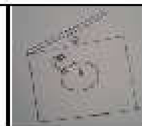
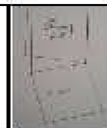
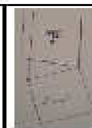
Appendix A

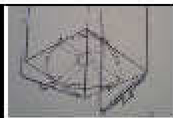
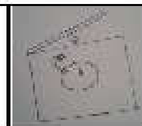
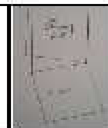
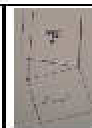
Quality Function Development



Release Mechanism Decision Process

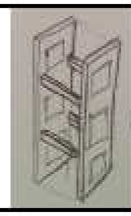
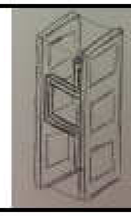
Criteria \ Concept	 1- Center Wheel	 2- Bar Linkage	 3- Bar Linkage no center plate	 4- Migamotor on Recessed	 5- Burn Circuitry on Recess
Resettable	S	S	S	D	-
Allows Max Inner Volume	-	-	S		S
Cost	+	+	+	A	+
Reliable	S	-	-		S
Complex of Manufacture	S	S	S	T	+
Time to Manufacture	S	-	S		+
# of Parts	+	-	-		S
Placement Versatility	-	-	S	M	S
Power Consumption	+	+	+		S
∑+	3	2	2		3
∑-	2	5	2		1
∑S	4	2	5		6

Concept / Criteria						
	1- Center Wheel	2- Bar Linkage	3- Center Wheel with Pawl	4- Migamotor on Recessed	5- Burn Circuitry on Recess	6- B. C All Panels
Resettable	D	S	S	S	-	-
Allows Max Inner Volume		S	S	+	+	+
Cost	A	S	-	-	+	+
Reliable		-	-	+	-	-
Complex of Manufacture	T	+	S	+	+	+
Time to Manufacture		S	-	+	+	+
Low Mass	U	S	S	S	S	S
# of Parts		-	-	-	-	-
Placement Versatility	M	S	S	+	+	+
Power Consumption		S	S	-	-	S
∑+		1	0	5	5	5
∑-		2	4	3	4	3
∑S		7	6	2	1	2

Concept / Criteria						
	1- Center Wheel	2- Bar Linkage	3- Center Wheel with Pawl	4- Migamotor on Recessed	5- Burn Circuitry on Recess	6- B. C All Panels
Resettable	D	S	S	S	-	-
Allows Max Inner Volume		S	S	+	+	+
Cost	A	S	-	-	+	+
Reliable		-	-	+	-	-
Complex of Manufacture	T	+	S	+	+	+
Time to Manufacture		S	-	+	+	+
Low Mass	U	S	S	S	S	S
# of Parts		-	-	-	-	-
Placement Versatility	M	S	S	+	+	+
Power Consumption		S	S	-	-	S
∑+		1	0	5	5	5
∑-		2	4	3	4	3
∑S		7	6	2	1	2

Tesseract
ME428 – Senior Project Design I
Vanessa Faune, Edgar Uribe
11/13/14

Structure Style Pugh Matrix

Concept / Criteria					
	HyperCube	Beam Supports	2 Panels + Vertical Square Supports	Panels + Horizontal Square Supports	3 Panels + Beam Supports
Material Cost	D	+	+	+	+
Visual Appeal	A	S	-	S	-
Variability	T	+	-	+	S
Mounting Options/Space	U	+	+	+	-
Internal Volume	M	S	S	-	S
Ease of Assembly		-	S	-	+
Sum (+)		3	2	3	2
Sum (-)		1	2	2	2
Sum (S)		2	2	1	2

Failure Mode and Effects Analysis (FMEA)

Potential
Failure Mode and Effect Analysis
Deployables DFMEA

___ System
___ Subsystem
___ Component

Design Responsibility:

FMEA Number:

Page 1 of 1

Satellite Bus: **Tesseract**

Key Date:

Prepared By: **Vanessa Faune**

Core Team: **V. Faune ; E. Uribe**

FMEA Date (Orig.) **25-Nov**

Item / Function	Potential Failure Mode	Potential Effect(s) of Failure	Sev	Potential Cause(s) / Mechanism(s) of Failure	Occur	Crit	Recommended Action(s)	Responsibility & Target Completion Date	Action Results: Actions Taken	Sev	Occur	Crit
Secure all side panels within 6.5mm from the rails of the structure	Panels are not held down within 6.5mm from the rails of the structure	Unable to be integrated into the PPOD	9	Inadequate securing mechanism to keep within tolerance	7	63	Design a tolerance between furthest component and 6.5mm limit	Edgar & 12/29/14				
Support side panels onto the structure while allowing release	Hinges components bind and do not allow panel deployment	Loss in power generation	7	Inadequate coating on components; corrosion from multiple test runs	4	28	Test coating on components and analyze success	Edgar & 1/16/15				
Deploy 1st side panel 135 deg from initial state	Incomplete draw of deployment pin does not let panels deploy			Electrical current supplied does not pull pin far enough in a set amount of time	4	28	Test pin pull in close-to-actual mission environment	Edgar & 1/12/15				
Deploy 2nd side panel 180 deg from initial state	Hinge springs warp/bind and do not allow full deployment			Incorrect material at desired temperatures; repeated use causes fatigue and permanent deformation	1	7	Test deployment several times in close-to-actual mission environment	Edgar/Vanessa & 4/15/15				
Support solar cells	Solar cells break upon deployment			Did not account for elastic panels; interference during deployment	9	63	Spacers between panels; space between panels during deployment; secure during launch	Edgar/Vanessa & 1/12/15				
Structure												
Structure is able to secure payload within the CubeSat specification	Structure does not comply with CubeSat spec; fails to keep payload safe	Unable to launch; mission failure	10	Not enough support; failure to implement all CubeSat specs	2	20	Dynamic analysis under launch conditions; Undergo testing	Vanessa & N/A				
Can be assembled by PolySat members	Extremely difficult to assemble	Cannot be integrated into PPOD; cannot support payload	6	Lack of foresight; did not practice assembly	7	42	Practice assembly with high fidelity prototype and detailed assembly procedures	Vanessa & 1/16/15				
Supports and secures all electronics	Not enough support on each board	Broken electronics; broken boards; broken solar cells	8	Lacking analysis to characterize board dynamics	8	64	Dynamic analysis under launch conditions; Undergo testing	Vanessa & 4/15/15				

Analysis Plan

Analysis Plan

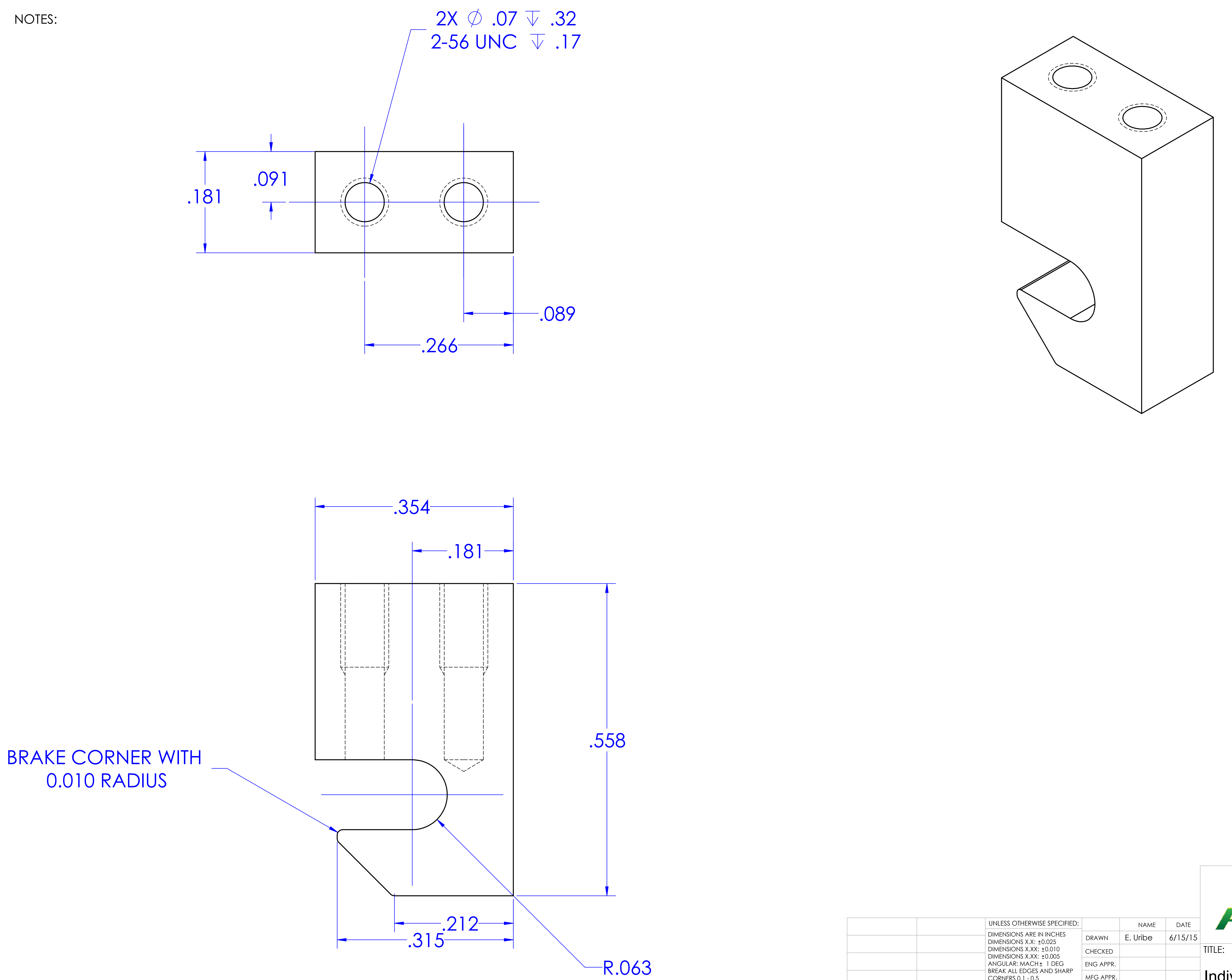
Priority	Analysis	Calculations or Test	Reason	Estimate Time to Complete
1	FEA on Structure	Calcs	Verify Stucture can survive vibes	3 Weeks
2	Screw Size Calculations	Calcs	Verify Screws can survive vibes	2 Hours
3	Momentum Calculations	Calcs	Determine impact force from deployment stop	3 Hours
4	Impact Deployment Test	Test	Verify impact force from Calculations	1 Day
5	Hinge leaf Stress Analysis	Calcs	Determine size of leaf based off of impact force	2 Hours
6	Hinge pin Stress Analysis	Calcs	Determine size of pin based off of impact force	30 Minutes
7	Hinge Stop Stress Analysis	Calcs	Determine size of stop based off of impact force	1 Hours
8	Solar Panel Deflection Analysis	Calcs	Determine maximum deflection of solar panels when stowed and deployed	1 Day
After Rapid Prototype				
1	Deployment Test	Test	Verify Proper Deployment	1 Day
2	Solar Cell Damage Deployment Test	Test	Verify no Solar Cell Damage from Deployment	1 Day
3	Deployment Verification Test	Test	Verify Deployment Verification method will trigger everytime deployed	1 Day

Design Verification Plan (DVP)

DVP&R Tesseract													
Report Date		20-Jun-15	Sponsor	Polysat					Component/Assembly			REPORTING ENGINEER:	Edgar Uribe and Vanessa Fuane
TEST PLAN										**TEST REPORT**			
Item No	Specification or Clause Reference	Test Description	Acceptance Criteria	Test Responsi	Test Stage	SAMPLES		TIMING		TEST RESULTS			NOTES
						Quantity	Type	Start date	Finish date	Test Result	Pass	Quantity Fail	
1	Deployment of First Panel	Proper Deployment Test	135°±10° from recessed panel	Edgar	DV	10	B	1/8/2015	1/12/2015	130	PASS		
2	Deployment of Second Panel	Proper Deployment Test	180°±10° from first deployable panel	Edgar	DV	10	B	1/8/2015	1/12/2015	180	PASS		
5	Structure Weight	Weigh Structure with Electrical Components	Max 1.5 kg	Edgar	PV	1	C	3/30/2015	5/1/2015	.71 kg	PASS		
6	Inner Volume	Measure Inner Volume available for Payload	min 2140cm cubed	Vanessa	PV	1	C	3/30/2015	5/1/2015	2015cm cubed		FAIL	There is still plenty of room for a wide range of payloads.
7	No Solar Cell Damage	Solar Cell Damage Test	Solar Cells cannot collide with anything	Edgar	DV	10	B	1/8/2015	1/12/2015	N/A	PASS		
10	Vibration Testing	Vibration Testing	Cubesat Specs	Vanessa	PV	1	C	3/30/2015	5/1/2015	N/A	PASS		
11	Thermal Chamber Testing	Deployment Test at Max and Min Temperatures	Deploy at 140 F and -20 F	Vanessa	PV	1	C	3/30/2015	5/1/2015	100%	PASS		
12	Rail Plane Protrusion limit	Measure Outmost component of second deployable panel	Max 6.5mm	Edgar	PV	1	C	3/30/2015	5/1/2015	6.6mm		FAIL	Still able to fit inside the PPOD. Ensure all screws tight and all interfaces are clean before assembly.

Appendix B

NOTES:



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NEXT ASSY	USED ON
APPLICATION	

UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
DIMENSIONS X.X: ±0.025
DIMENSIONS X.XX: ±0.010
DIMENSIONS X.XX: ±0.005
ANGULAR: MACH± 1 DEG
BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5

INTERPRET GEOMETRIC TOLERANCING PER:

MATERIAL ALUMINUM 6061

FINISH 32 MICRONS

DO NOT SCALE DRAWING

	NAME	DATE
DRAWN	E. Uribe	6/15/15
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		

COMMENTS:

PolySat

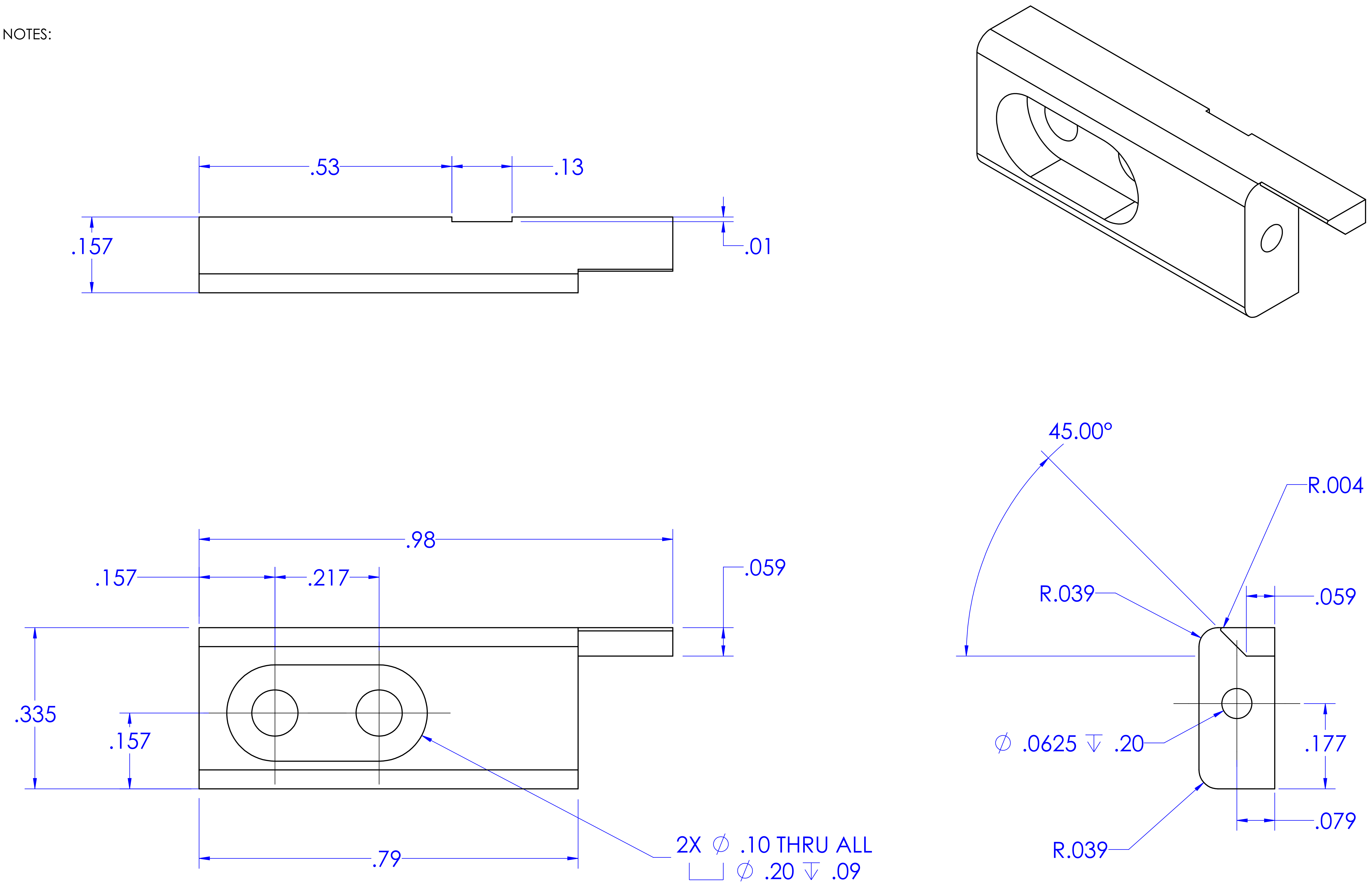
TITLE:

Individual_Panels_
Retaining pin_v3.0

SIZE	DWG. NO.	REV
C	1	3

SCALE: 12:1 WEIGHT: N/A SHEET 1 OF 1

NOTES:



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UNLESS OTHERWISE SPECIFIED:		NAME / DATE
DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XX: ±0.005 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	E. Uribe 1/29/15
	CHECKED	
	ENG APPR.	
	MFG APPR.	
INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.	
MATERIAL ALUMINUM 6061	COMMENTS:	
FINISH 32 MICRONS		
DO NOT SCALE DRAWING		

NEXT ASSY	USED ON
APPLICATION	

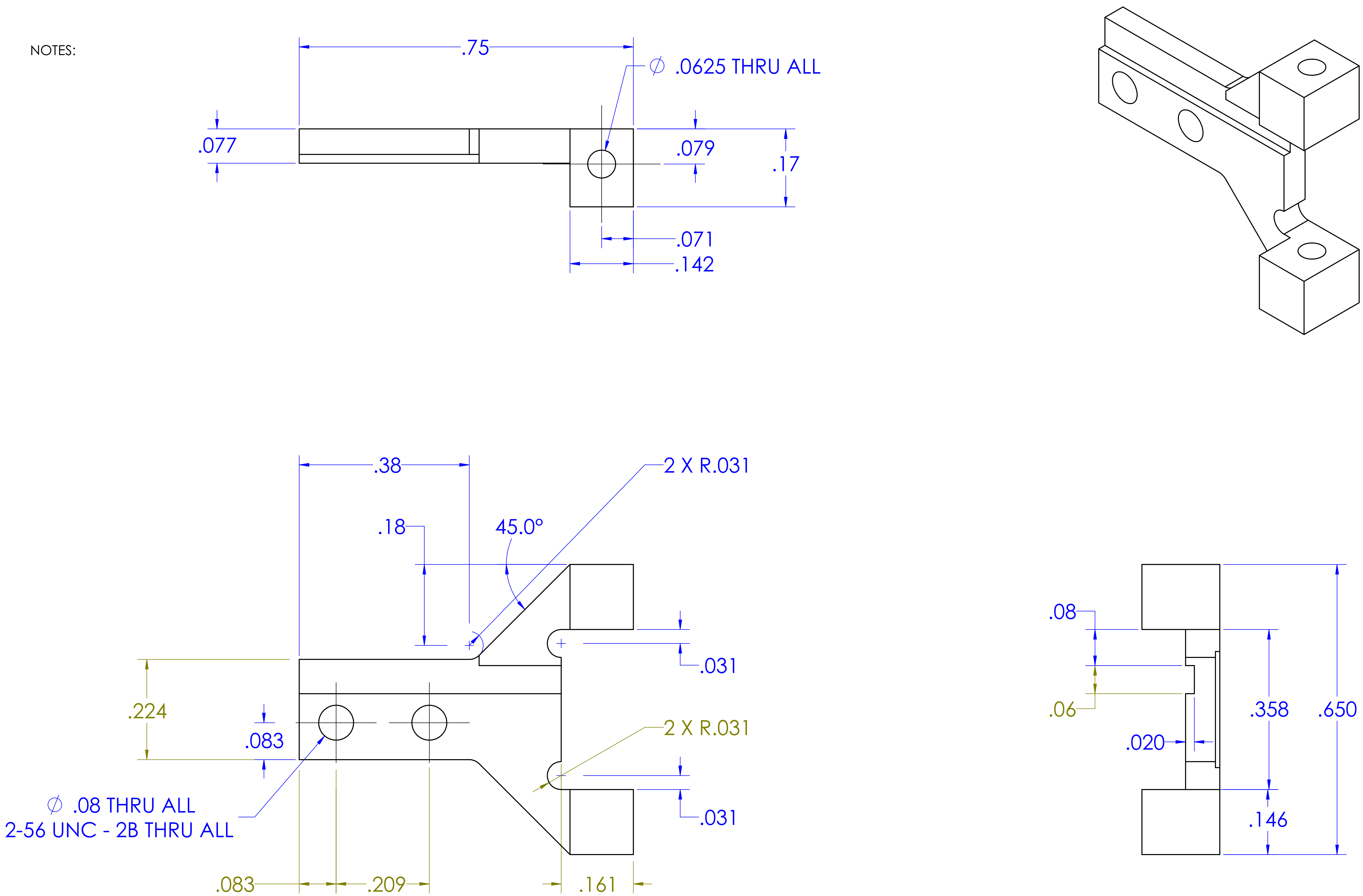
PolySat

TITLE:

TesseractHingeBlockLeft

SIZE	DWG. NO.	REV
C	1	1
SCALE: 5:1	WEIGHT: N/A	SHEET 1 OF 1

NOTES:



		UNLESS OTHERWISE SPECIFIED:		NAME	DATE
		DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XXX: ±0.004 ANGULAR: MACH ± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	E. Uribe	1/29/15
			CHECKED		
			ENG APPR.		
			MFG APPR.		
		INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.		
		MATERIAL ALUMINUM 6061	COMMENTS:		
NEXT ASSY	USED ON	FINISH 32 MICRONS			
APPLICATION		DO NOT SCALE DRAWING			

PROPRIETARY AND CONFIDENTIAL

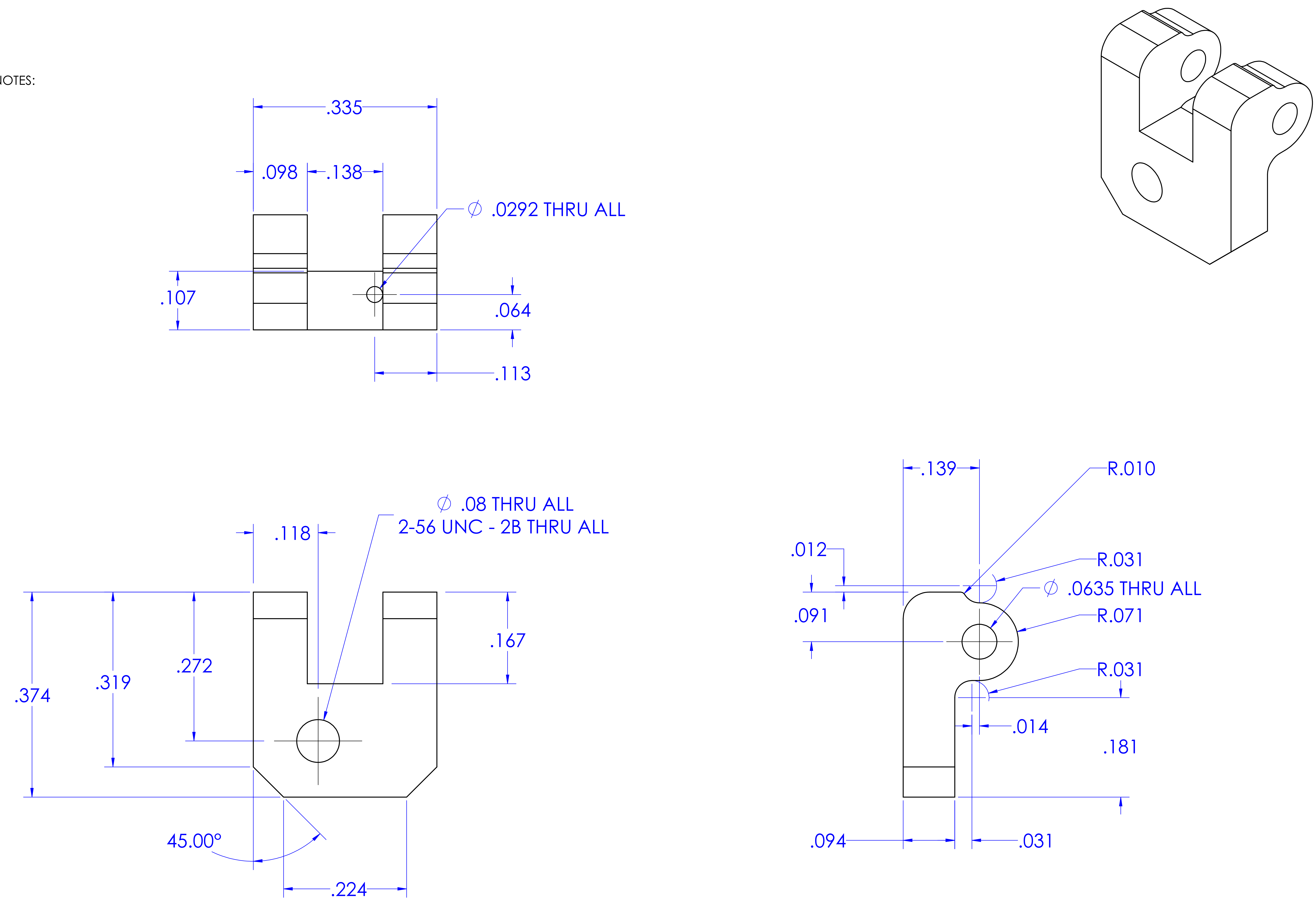
TITLE:

SmallBottomLeaf

SIZE	DWG. NO.	REV
C	1	9

SCALE: 6:1 WEIGHT: N/A SHEET 1 OF 1

NOTES:



UNLESS OTHERWISE SPECIFIED:	
DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XXX: ±0.004 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	
INTERPRET GEOMETRIC TOLERANCING PER:	
MATERIAL	ALUMINUM 6061
FINISH	32 MICRONS
DO NOT SCALE DRAWING	

	NAME	DATE
DRAWN	E. Uribe	1/29/15
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		
COMMENTS:		

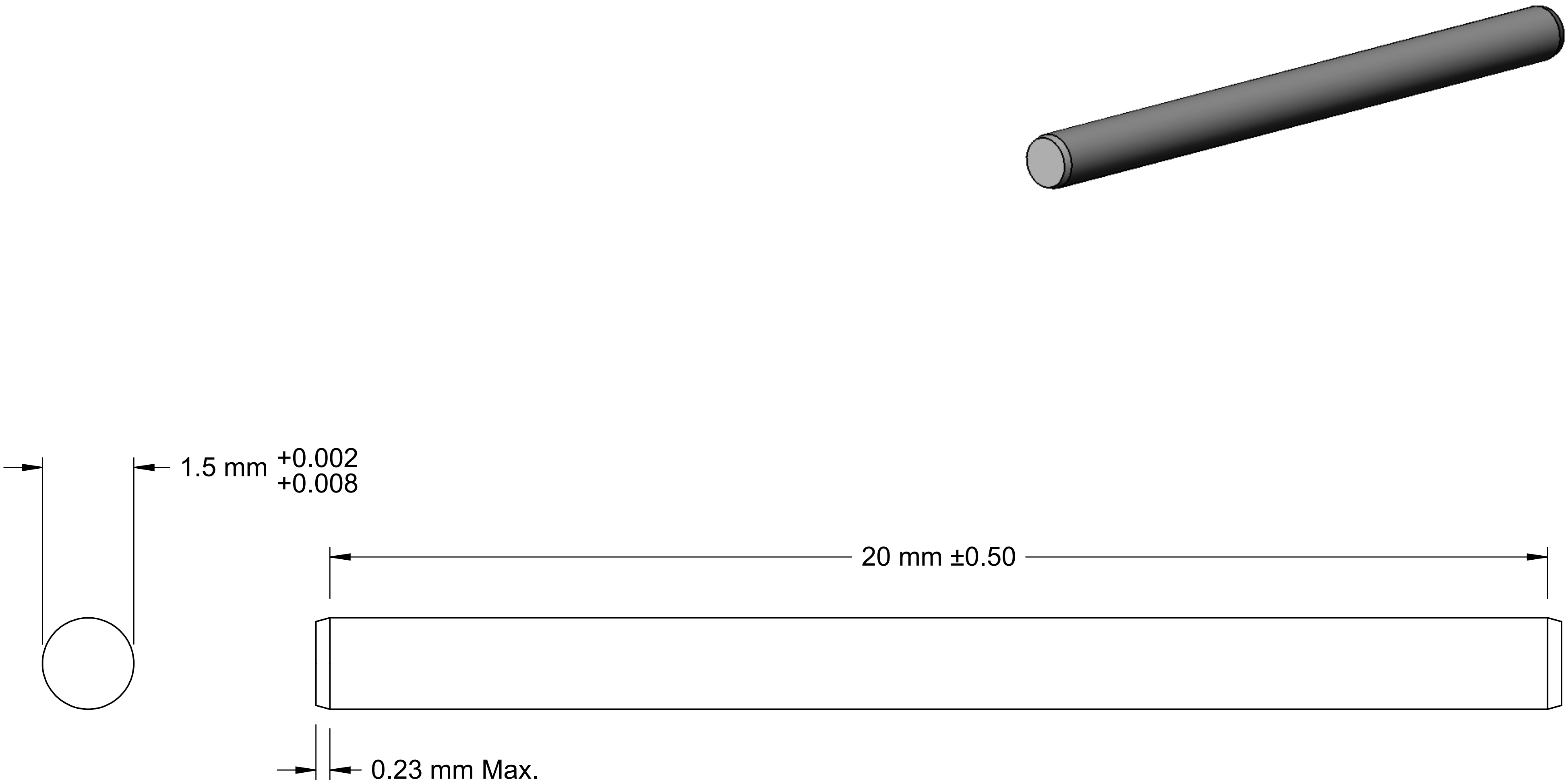
NEXT ASSY	USED ON
APPLICATION	

PROPRIETARY AND CONFIDENTIAL

TITLE:

SmallTopLeaf

SIZE	DWG. NO.	REV
C	1	9
SCALE: 8:1	WEIGHT: N/A	SHEET 1 OF 1



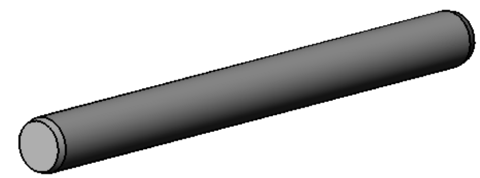
McMASTER-CARR® CAD

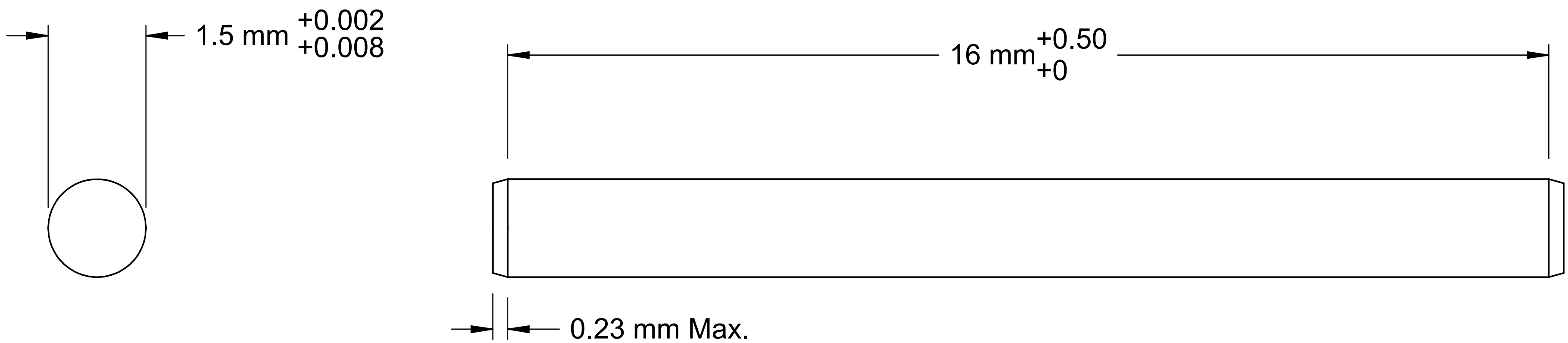
PART NUMBER 91585A014

http://www.mcmaster.com

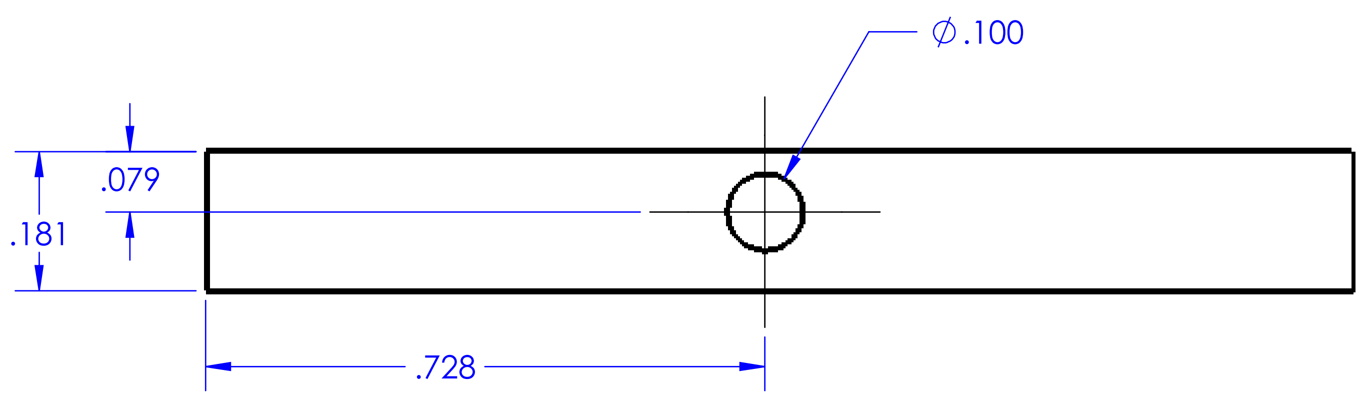
Information in this drawing is provided for reference only.

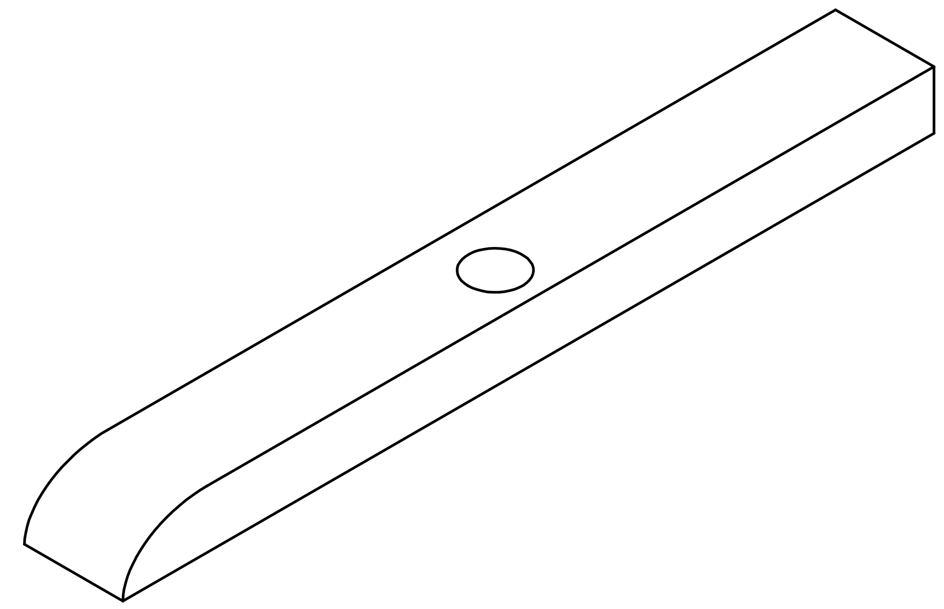
18-8 Stainless Steel Dowel Pin

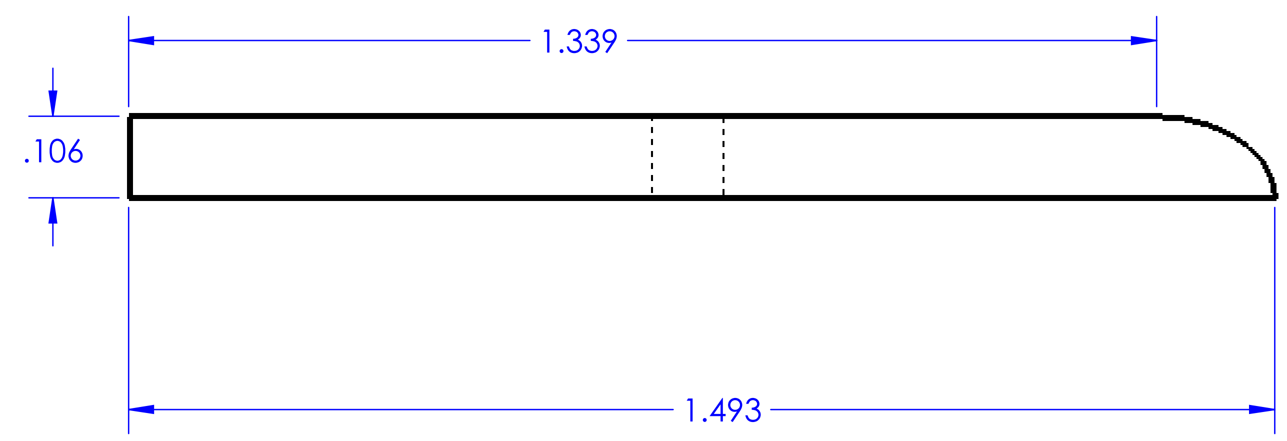




McMASTER-CARR® CAD	PART NUMBER	93600A064
http://www.mcmaster.com © 2014 McMaster-Carr Supply Company Information in this drawing is provided for reference only.	Type 316 Stainless Steel Dowel Pin	







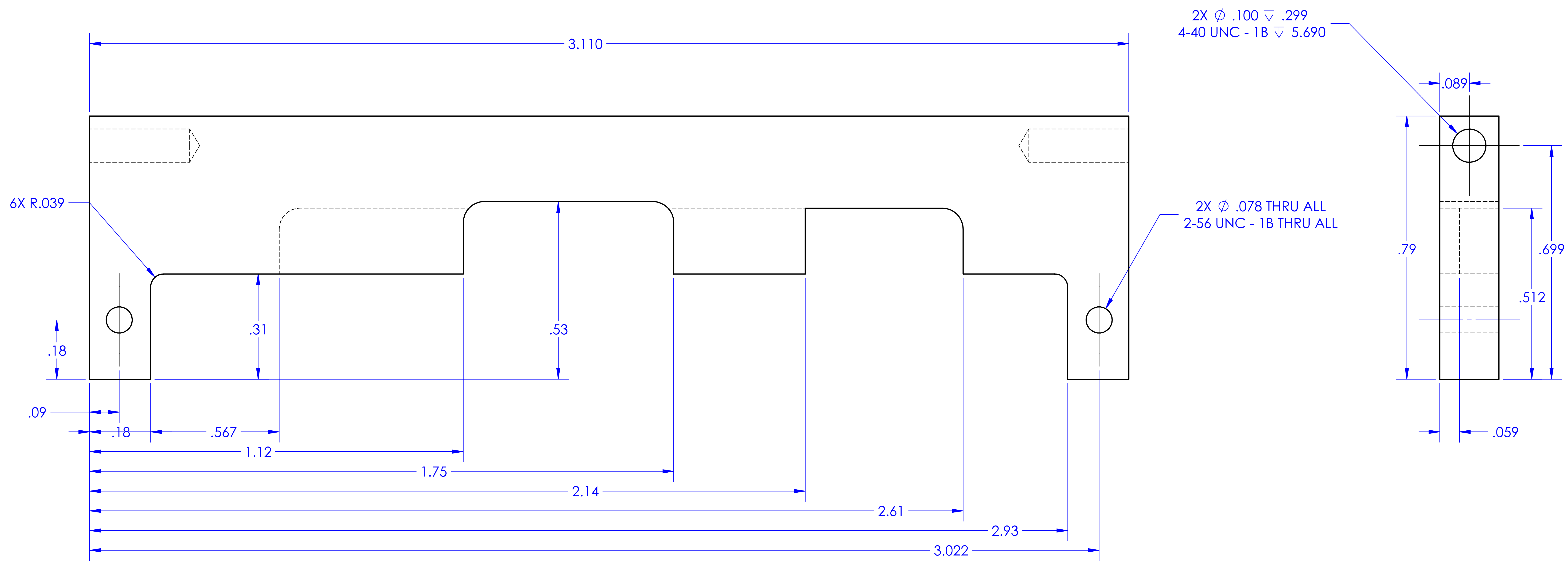
UNLESS OTHERWISE SPECIFIED:		NAME	DATE
DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XX: ±0.005 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	V.FAUNE	1/30/15
	CHECKED		
	ENG APPR.		
	MFG APPR.		
INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.		
MATERIAL ALUMINUM 6061	COMMENTS:		
FINISH 32 MICRONS			
DO NOT SCALE DRAWING			

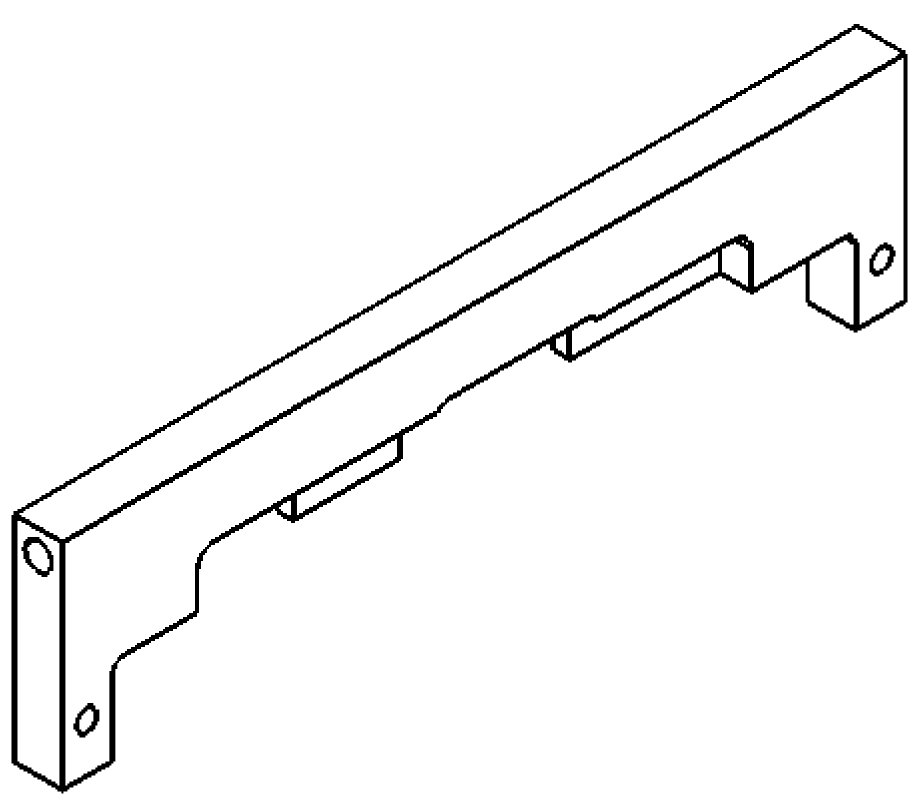
NEXT ASSY	USED ON
APPLICATION	

PolySat

TITLE: TALON_REV2

SIZE	DWG. NO.	REV
C	1	2
SCALE: 4:1	WEIGHT: N/A	SHEET 1 OF 1





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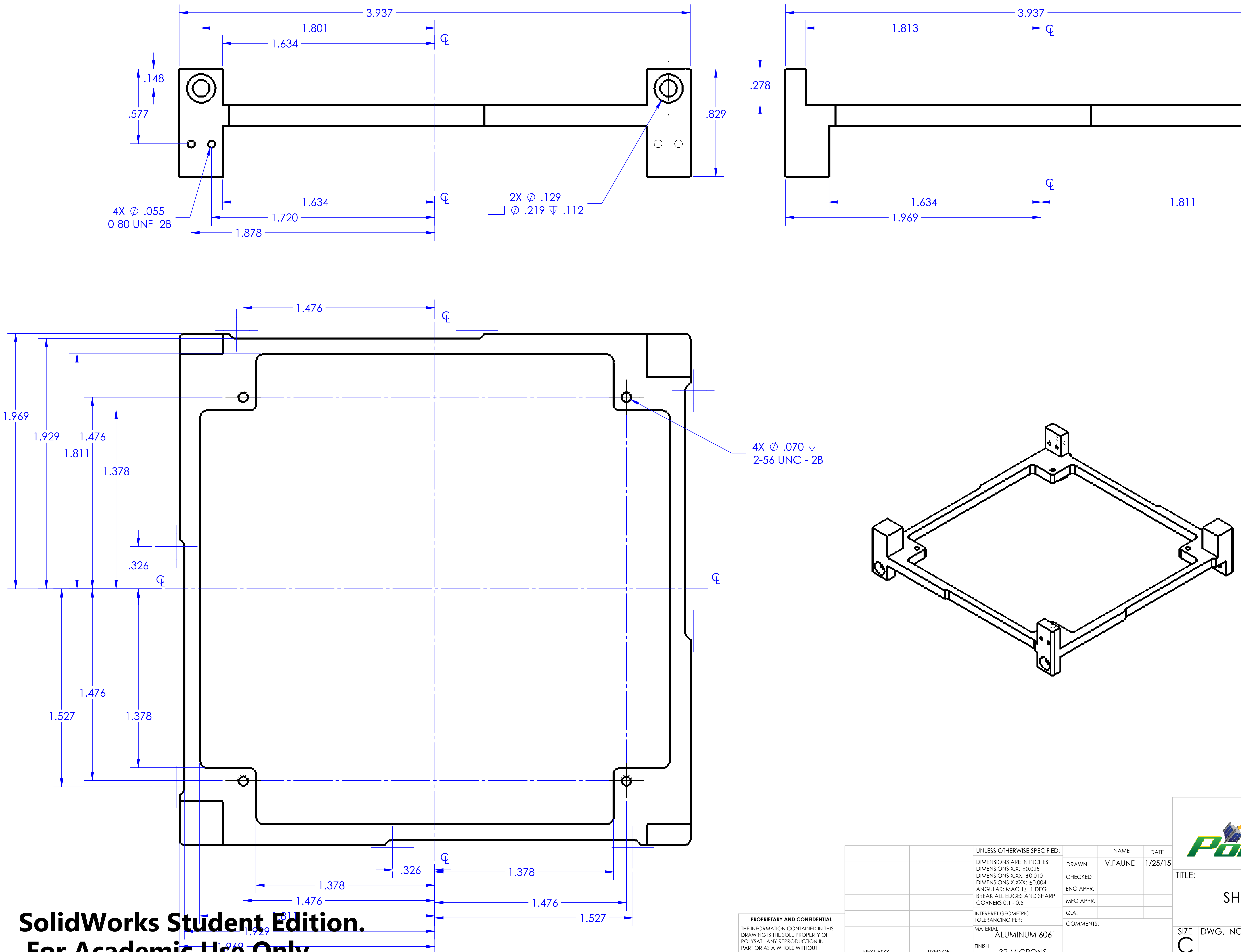
UNLESS OTHERWISE SPECIFIED:		NAME	DATE	
DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XXX: ±0.004 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	V.FAUNE	1/15/15	TITLE: DEPLOYMENT_BEAM
	CHECKED			
	ENG APPR.			
	MFG APPR.			
INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.			
MATERIAL ALUMINUM 6061	COMMENTS:			SIZE C · DWG. NO. 1 · REV 4
FINISH 32 MICRONS				
DO NOT SCALE DRAWING				SCALE: 4:1 · WEIGHT: N/A · SHEET 1 OF 1

NEXT ASSY	USED ON
APPLICATION	



NEXT ASSY	USED ON
APPLICATION	

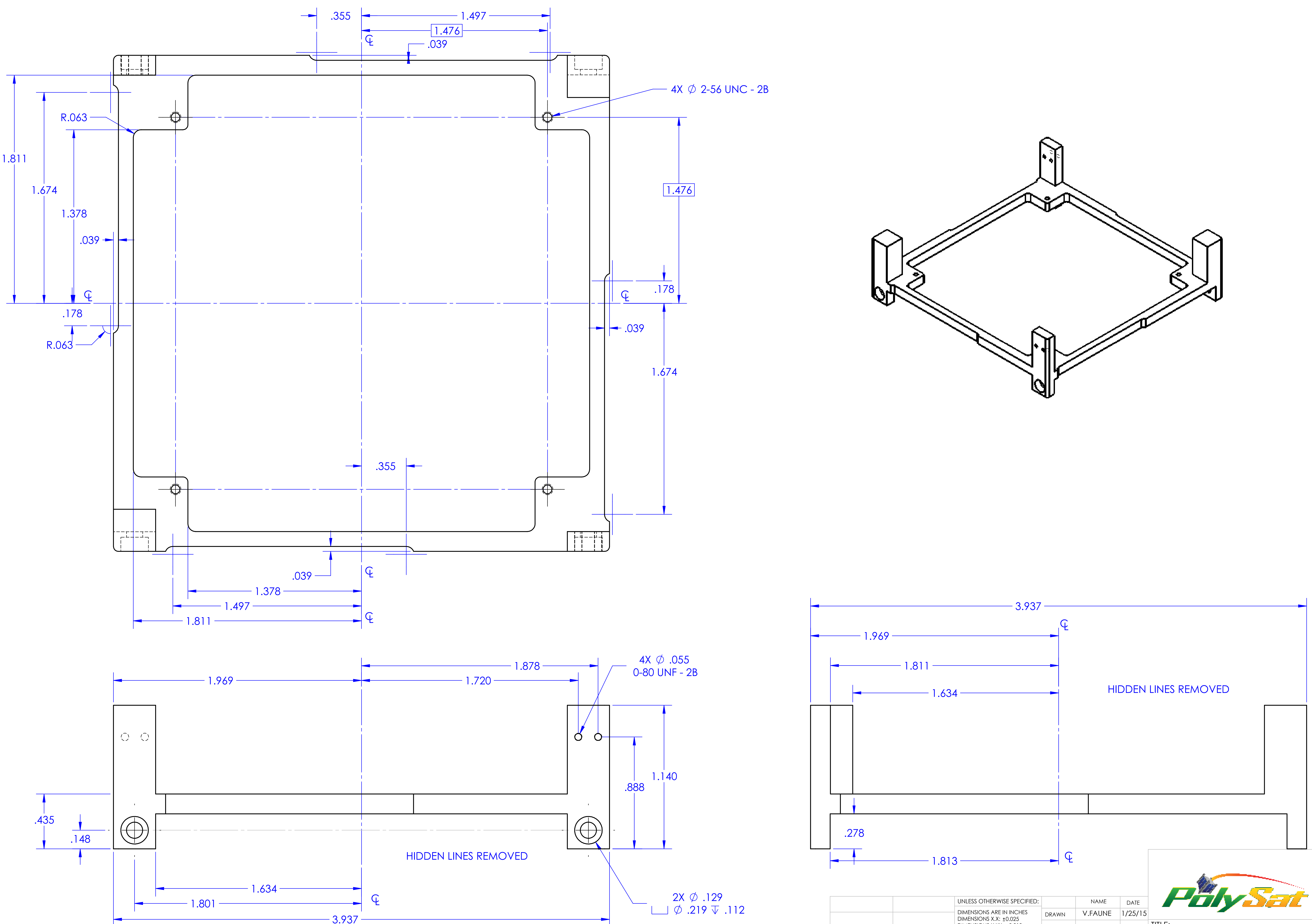
UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
DIMENSIONS X.X: ±0.025
DIMENSIONS X.XX: ±0.010
DIMENSIONS X.XXX: ±0.004
ANGULAR: MACH± 1 DEG
BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5
INTERPRET GEOMETRIC TOLERANCING PER:
MATERIAL ALUMINUM 6061
FINISH 32 MICRONS
DO NOT SCALE DRAWING

	NAME	DATE
DRAWN	V.FAUNE	1/25/15
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		
COMMENTS:		

TITLE: SHOE_REV2

SIZE	DWG. NO.	REV
C	1	2

SCALE: 2:1 WEIGHT: N/A SHEET 1 OF 1



UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
DIMENSIONS X.X: ±0.025
DIMENSIONS X.XX: ±0.010
DIMENSIONS X.XXX: ±0.004
ANGULAR: MACH± 1 DEG
BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5
INTERPRET GEOMETRIC TOLERANCING PER:

MATERIAL: ALUMINUM 6061
FINISH: 32 MICRONS

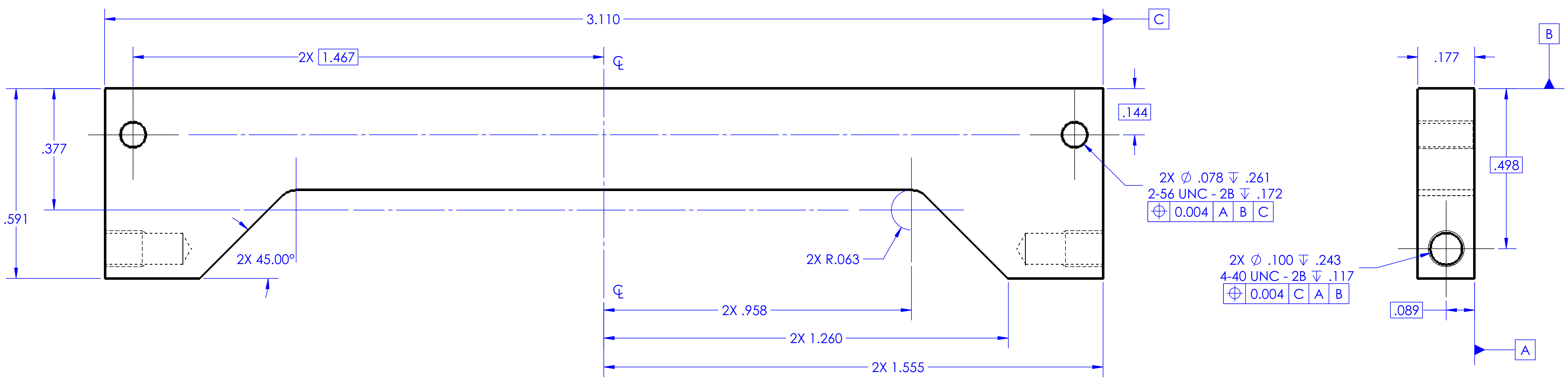
NEXT ASSY | USED ON
APPLICATION | DO NOT SCALE DRAWING

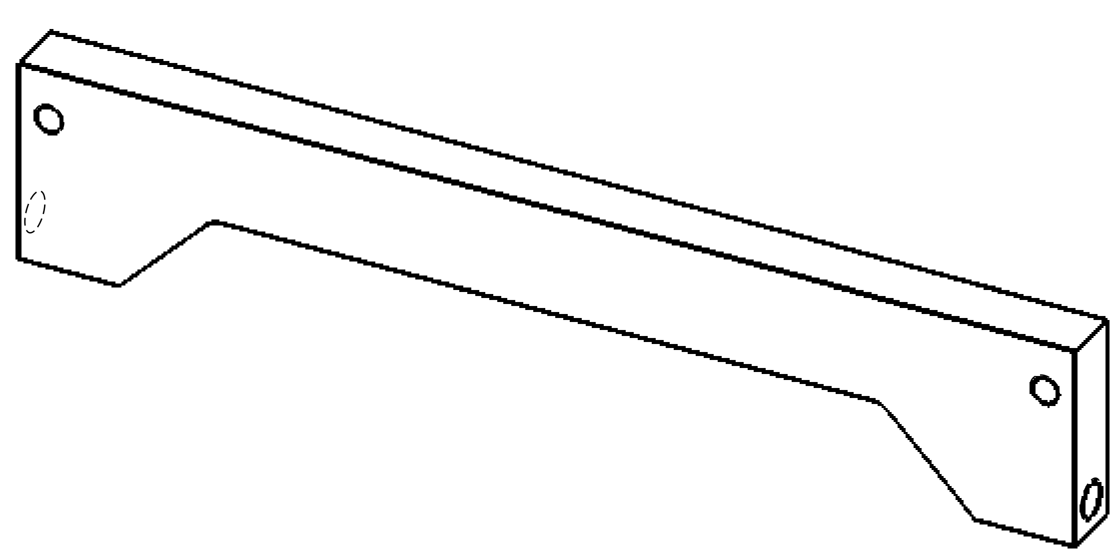
	NAME	DATE
DRAWN	V.FAUNE	1/25/15
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		
COMMENTS:		

TITLE: TOP_HAT_REV5

SIZE	DWG. NO.	REV
C	1	5

SCALE: 2:1 WEIGHT: N/A SHEET 1 OF 1





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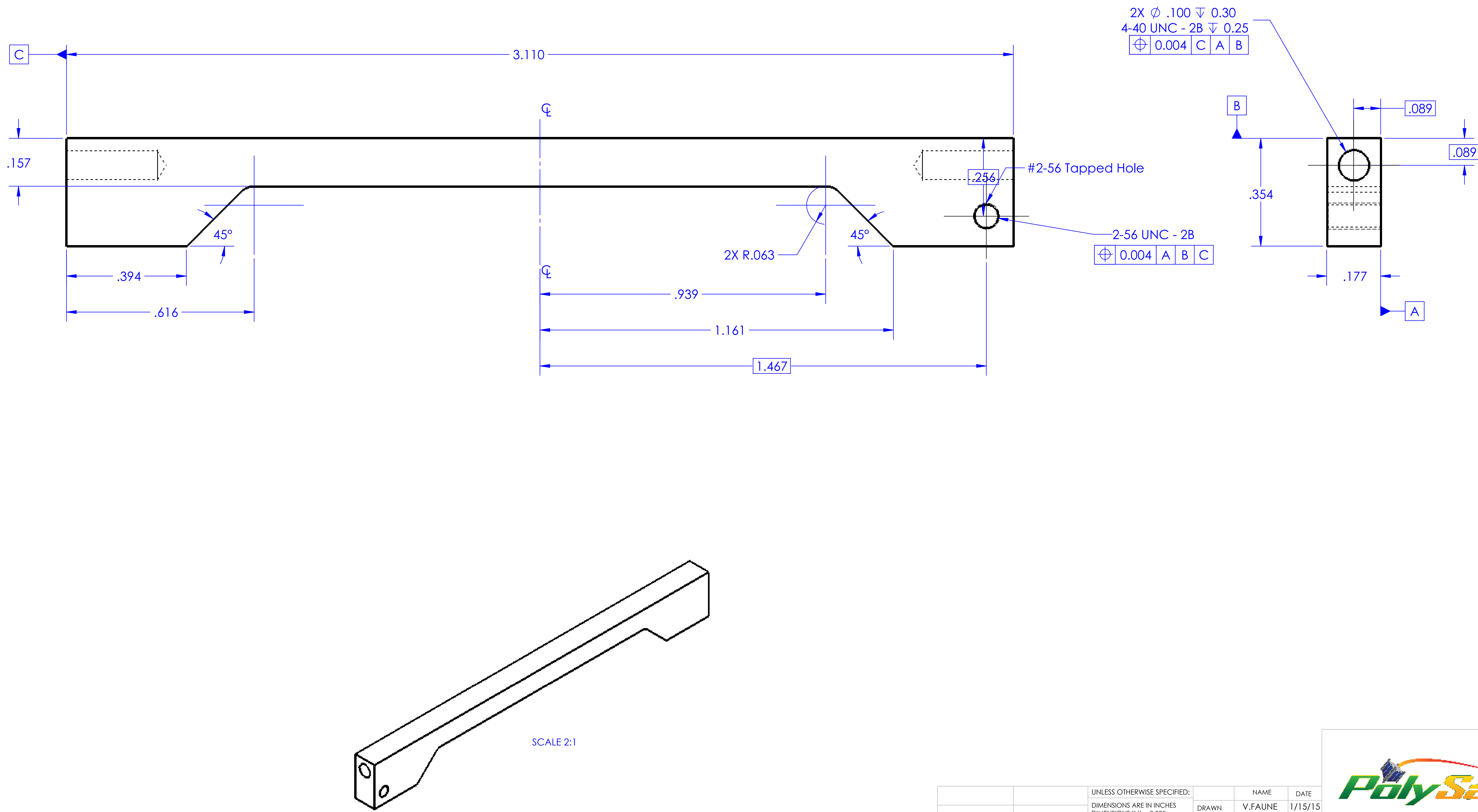
				NAME	DATE
		UNLESS OTHERWISE SPECIFIED: DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XXX: ±0.005 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	V.FAUNE	1/15/15
			CHECKED		
			ENG APPR.		
			MFG APPR.		
		INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.		
		MATERIAL ALUMINUM 6061	COMMENTS:		
NEXT ASSY	USED ON	FINISH 32 MICRONS			
APPLICATION		DO NOT SCALE DRAWING			

PolySat

TITLE: LARGE_BEAM_REV2

SIZE	DWG. NO.	REV
C	1	2

SCALE: 4:1 WEIGHT: N/A SHEET 1 OF 1



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NEXT ASSY	USED ON	
APPLICATION		

UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
DIMENSIONS X.X: ±0.025
DIMENSIONS X.XX: ±0.010
DIMENSIONS X.XXX: ±0.004
ANGULAR: MACH± 1 DEG
BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5

INTERPRET GEOMETRIC TOLERANCING PER:

MATERIAL ALUMINUM 6061

FINISH 32 MICRONS

DO NOT SCALE DRAWING

	NAME	DATE
DRAWN	V.FAUNE	1/15/15
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		

COMMENTS:

PolySat

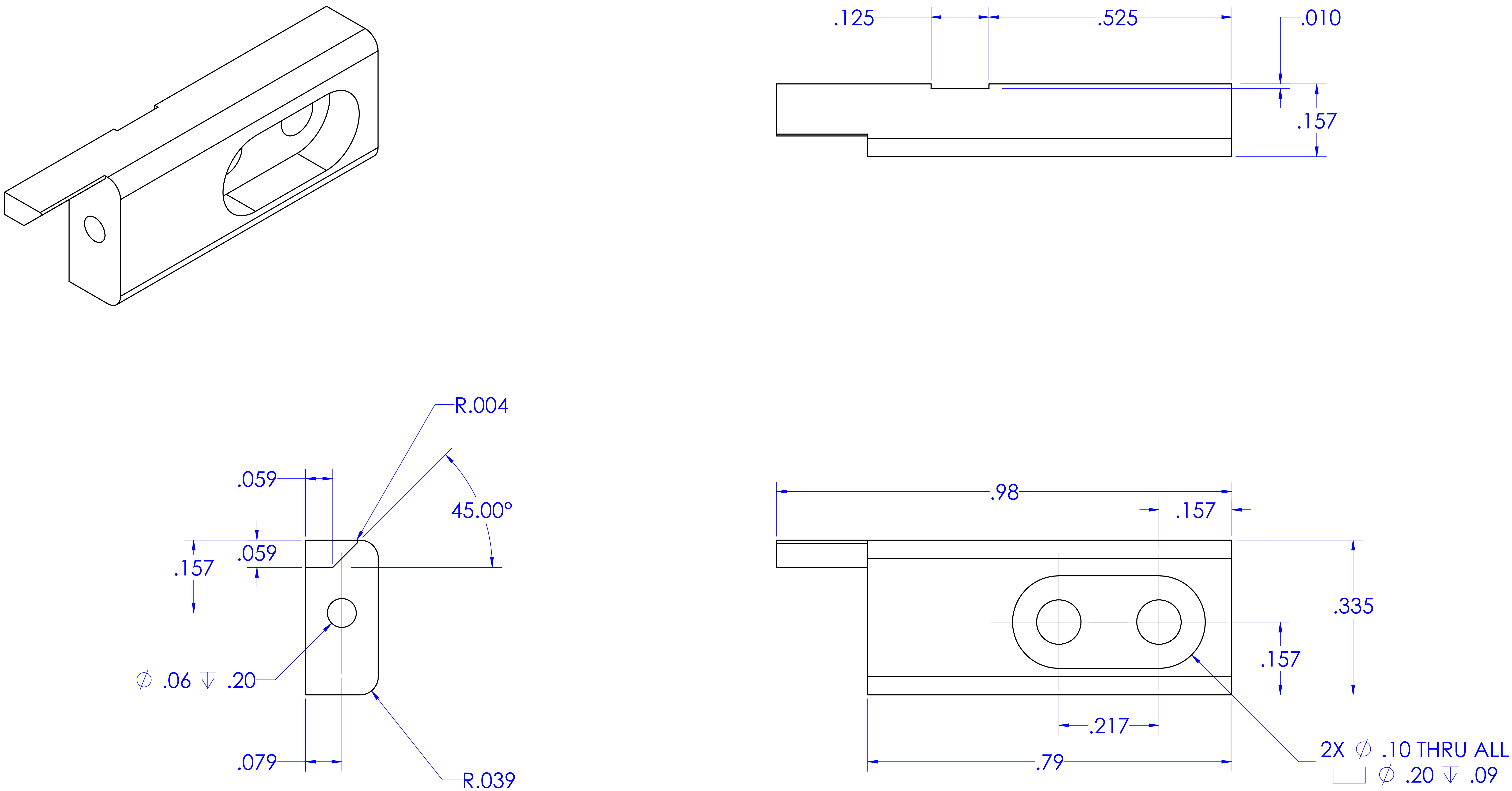
TITLE:

SMALL_BEAM_REV4

SIZE	DWG. NO.	REV
C	1	4

SCALE: 4:1 WEIGHT: N/A SHEET 1 OF 1

NOTES:



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UNLESS OTHERWISE SPECIFIED:		NAME	DATE
DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XX: ±0.005 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	E. Uribe	1/29/15
	CHECKED		
	ENG APPR.		
	MFG APPR.		
INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.		
MATERIAL ALUMINUM 6061	COMMENTS:		
FINISH 32 MICRONS			
DO NOT SCALE DRAWING			

NEXT ASSY | USED ON

APPLICATION

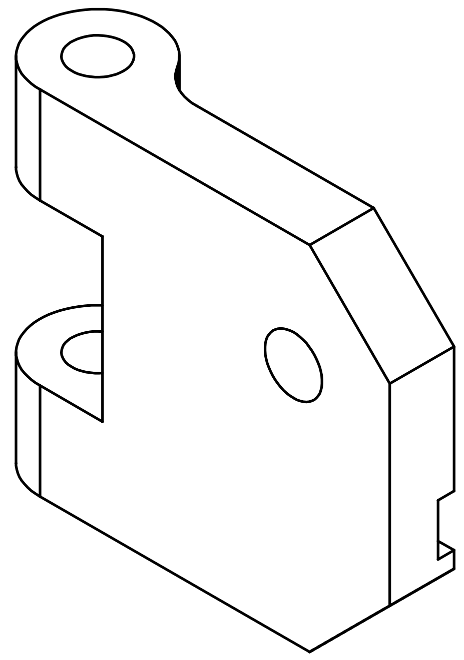
TITLE:

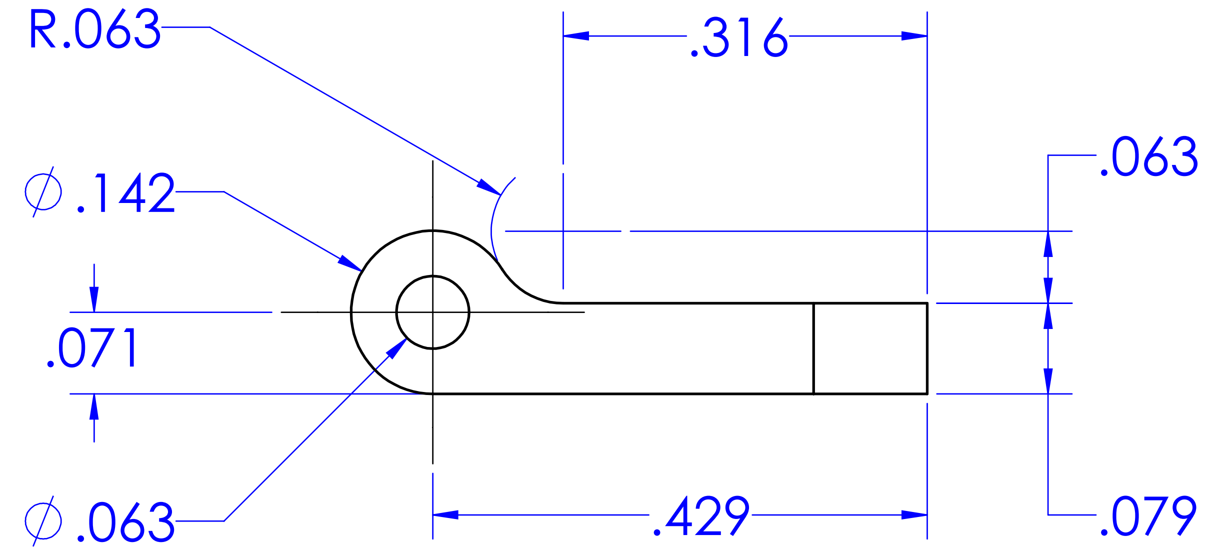
TesseractHingeBlockRight

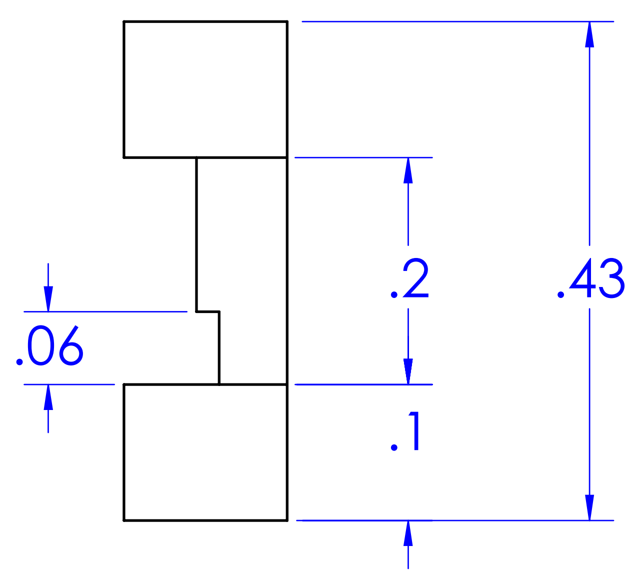
SIZE C | DWG. NO. 1 | REV 1

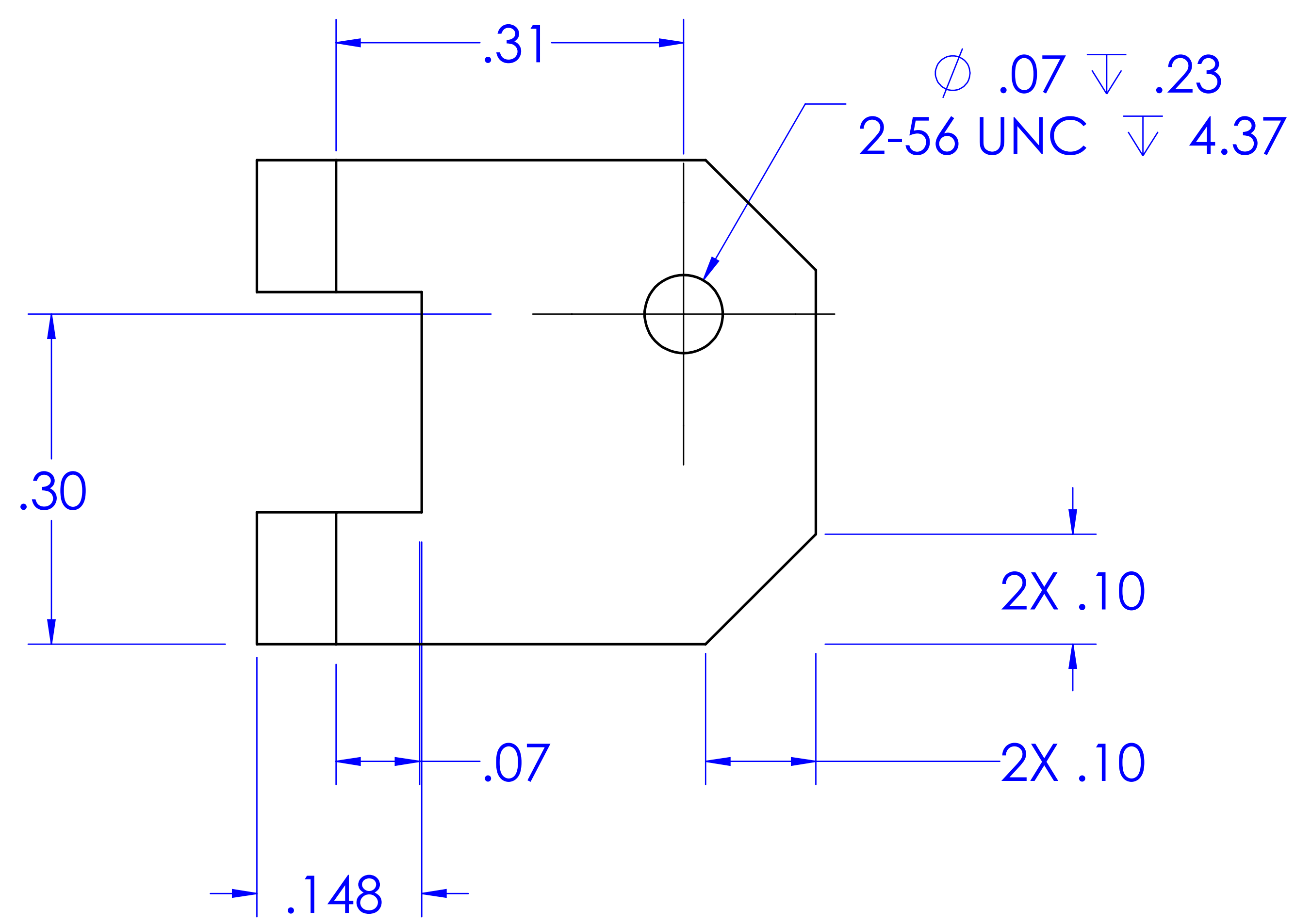
SCALE: 6:1 | WEIGHT: N/A | SHEET 1 OF 1

NOTES:









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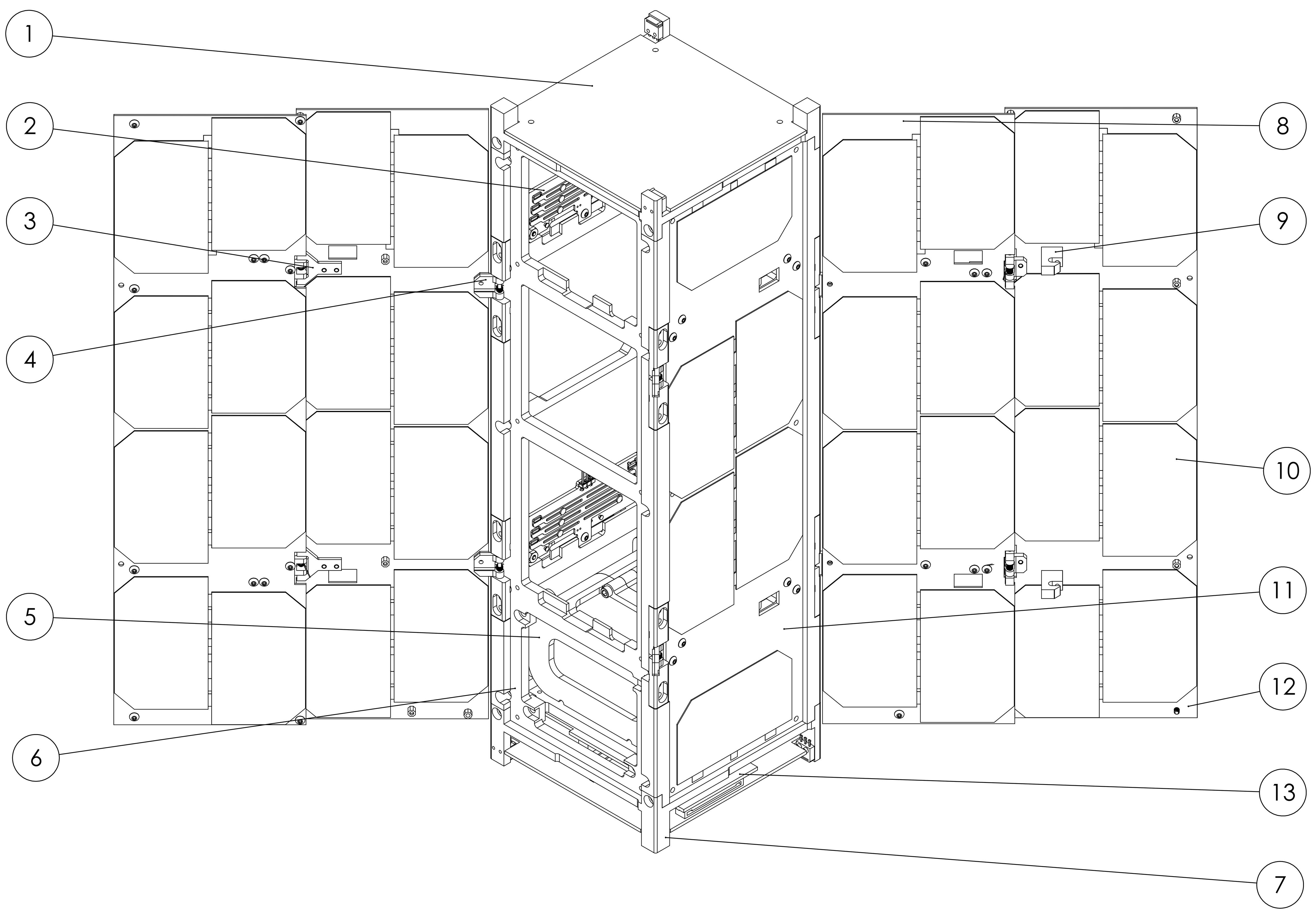
		UNLESS OTHERWISE SPECIFIED:		NAME	DATE
		DIMENSIONS ARE IN INCHES DIMENSIONS X.X: ±0.025 DIMENSIONS X.XX: ±0.010 DIMENSIONS X.XXX: ±0.004 ANGULAR: MACH± 1 DEG BREAK ALL EDGES AND SHARP CORNERS 0.1 - 0.5	DRAWN	E. Uribe	1/29/15
			CHECKED		
			ENG APPR.		
			MFG APPR.		
		INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.		
		MATERIAL ALUMINUM 6061	COMMENTS:		
NEXT ASSY	USED ON	FINISH 32 MICRONS			
APPLICATION		DO NOT SCALE DRAWING			

PolySat

TITLE:

TesseractHingeLeaf

SIZE	DWG. NO.	REV
C	1	1
SCALE: 6:1	WEIGHT: N/A	SHEET 1 OF 1



ITEM NO.	PART NUMBER	QTY.
1	Z Panel	2
2	Release Mechanism	8
3	Second Hinge	8
4	First Hinge	8
5	Battery Bracket	1
6	Recessed Rail Panel	2
7	Top Hat	2
8	First Deployable Panel	4
9	Hook	8
10	Solar Cell	152
11	Recessed Solar Panel	4
12	Second Deployable Panel	4
13	Avionics Stack	1

NEXT ASSY | USED ON
APPLICATION

UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL ±
ANGULAR: MACH ± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±
INTERPRET GEOMETRIC TOLERANCING PER:
MATERIAL
FINISH

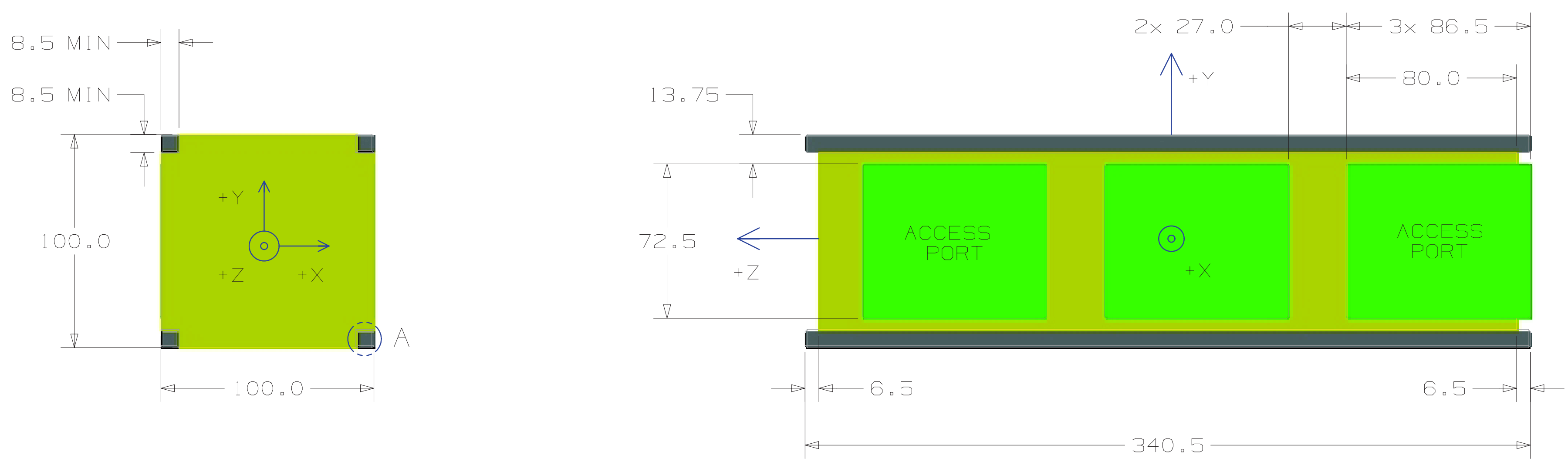
	NAME	DATE
DRAWN		
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		

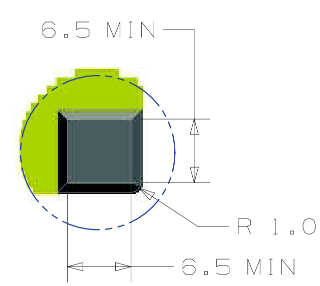
COMMENTS:

TITLE:

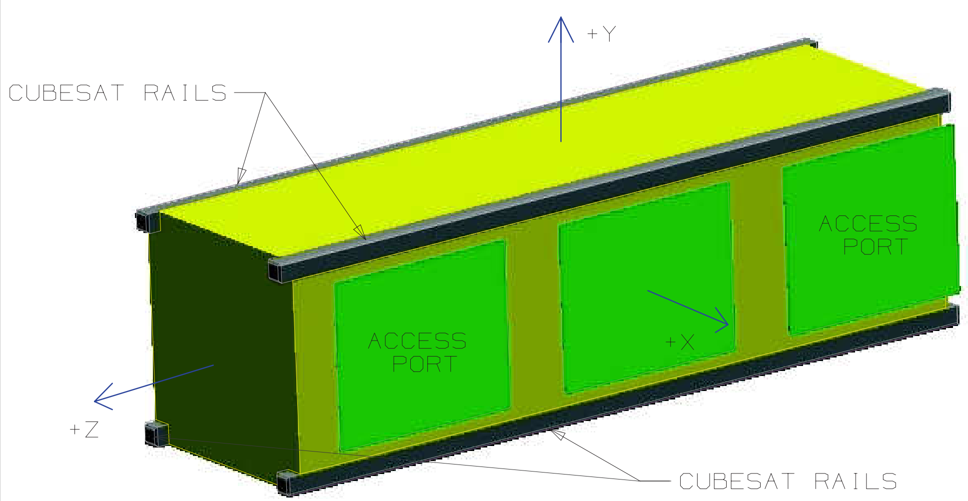
SIZE E DWG. NO. Tesseract_Assembly_Bill REV

SCALE: 1:1 WEIGHT: SHEET 2 OF 2





DETAIL A
STANDOFF CONTACT
DETAIL FOR +Z



ADDITIONAL NOTES:

- CubeSat coordinate system located in the geometric center of the CubeSat.
- No external components other than the rails shall touch the inside of the P-POD.
- All 3U CubeSats must have at least one (1) deployment switch on the -Z face. See drawing CDS-13-006, Deployment Switch and Separation Spring Location, for more details.

CUBESAT		California Polytechnic State University CubeSat Program (805) 756-5087 San Luis Obispo, CA 93407	
ALL DIMENSIONS IN MILLIMETERS	DESIGNED BY S.FUR	PART NAME 3U CUBESAT	REV 13
TOLERANCE .X ± 0.1	DRAWN BY S.FUR		
	CHECKED BY J.CAR		
ROUND ALL EDGES AND CORNERS	APPROVED BY R.NUG	SIZE B	ASSEMBLY CUBESAT SPECIFICATION
	DRAWING # CDS-13-004	NOT TO SCALE	DATE: 04/03/12 — SHEET 1 OF 1

2. Poly Picosatellite Orbital Deployer

2.1 *Interface*

The Poly Picosatellite Orbital Deployer (P-POD) is Cal Poly's standardized CubeSat deployment system. It is capable of carrying three standard CubeSats and serves as the interface between the CubeSats and LV. The P-POD is a rectangular box with a door and a spring mechanism. Once the release mechanism of the P-POD is actuated by a deployment signal sent from the LV, a set of torsion springs at the door hinge force the door open and the CubeSats are deployed by the main spring gliding on its rails and the P-PODs rails (P-POD rails are shown in Figure 3b). The P-POD is made up of anodized aluminum. CubeSats slide along a series of rails during ejection into orbit. CubeSats will be compatible with the P-POD to ensure safety and success of the mission by meeting the requirements outlined in this document. The P-POD is backward compatible, and any CubeSat developed within the design specification of CDS rev. 9 and later will not have compatibility issues. Developers are encouraged to design to the most current CDS to take full advantage of the P-POD features.

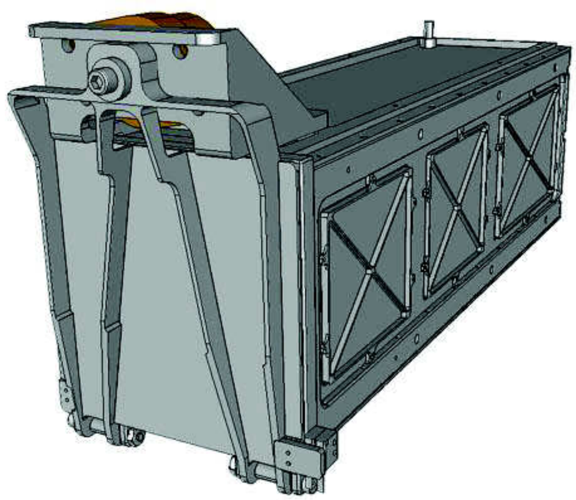
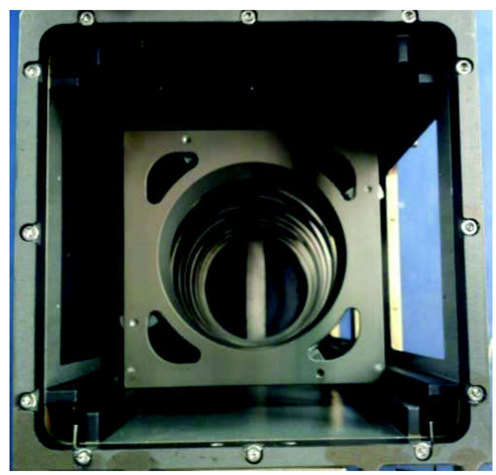



Figure 3a and 3b: Poly Picosatellite Orbital Deployer (P-POD) and cross section

3. CubeSat Specification

3.1 *General Requirements*

3.1.1 CubeSats which incorporate any deviation from the CDS will submit a DAR and adhere to the waiver process (see Section 1.3 and Appendix A).

3.1.2 All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created.

3.1.3 No pyrotechnics shall be permitted.

3.1.4 Any propulsion systems shall be designed, integrated, and tested in accordance with AFSPCMAN 91-710 Volume 3.

3.1.5 Propulsion systems shall have at least 3 inhibits to activation.

3.1.6 Total stored chemical energy will not exceed 100 Watt-Hours.

3.1.6.1 Note: Higher capacities may be permitted, but could potentially limit launch opportunities.

3.1.7 CubeSat hazardous materials shall conform to AFSPCMAN 91-710, Volume 3.

3.1.8 CubeSat materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft during integration, testing, and launch. A list of NASA approved low out-gassing materials can be found at: http://outgassing.nasa.gov

3.1.8.1 CubeSats materials shall have a Total Mass Loss (TML) $\leq$ 1.0 %

3.1.8.2 CubeSat materials shall have a Collected Volatile Condensable Material (CVCM) $\leq$ 0.1%

3.1.9 The latest revision of the CubeSat Design Specification will be the official version which all CubeSat developers will adhere to. The latest revision is available at http://www.cubesat.org.

3.1.9.1 Cal Poly will send updates to the CubeSat mailing list upon any changes to the specification. You can sign-up for the CubeSat mailing list here: www.cubesat.org/index.php/about-us/how-to-join

3.1.10 Note: Some launch vehicles hold requirements on magnetic field strength. Additionally, strong magnets can interfere with the separation between CubeSat spacecraft in the same P-POD. As a general guideline, it is advised to limit magnetic field outside the CubeSat static envelope to 0.5 Gauss above Earth's magnetic field.

3.1.11 The CubeSat shall be designed to accommodate ascent venting per ventable volume/area < 2000 inches.

3.2 *CubeSat Mechanical Requirements*

CubeSats are cube shaped picosatellites with dimensions and features outlined in the CubeSat Specification Drawing (Appendix B). The PPOD coordinate system is shown below in Figure 4 for reference. General features of all CubeSats include:

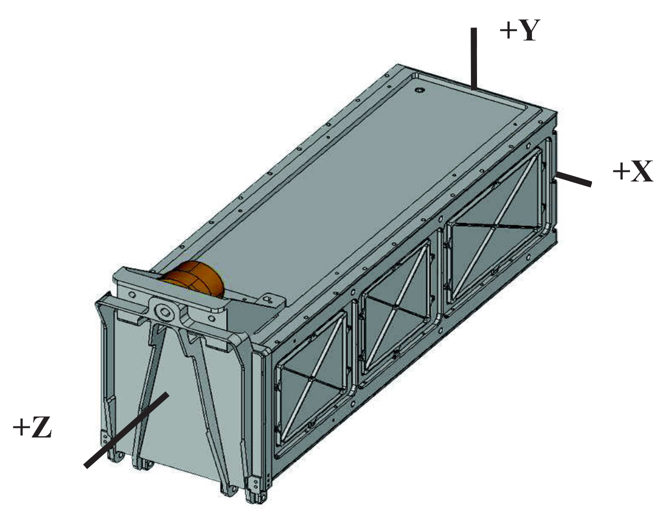


Figure 4: PPOD Coordinate System

3.2.1 The CubeSat shall use the coordinate system as defined in Appendix B for the appropriate size. The CubeSat coordinate system will match the P-POD coordinate system while integrated into the P-POD. The origin of the CubeSat coordinate system is located at the geometric center of the CubeSat.

3.2.1.1 The CubeSat configuration and physical dimensions shall be per the appropriate section of Appendix B.

3.2.1.2 The extra volume available for 3U+ CubeSats is shown in Figure 6.

3.2.2 The –Z face of the CubeSat will be inserted first into the P-POD.

3.2.3 No components on the green and yellow shaded sides shall exceed 6.5 mm normal to the surface.

3.2.3.1 When completing a CubeSat Acceptance Checklist (CAC), protrusions will be measured from the plane of the rails.

3.2.4 Deployables shall be constrained by the CubeSat, not the P-POD.

3.2.5 Rails shall have a minimum width of 8.5mm.

3.2.6 Rails will have a surface roughness less than 1.6 μm.

3.2.7 The edges of the rails will be rounded to a radius of at least 1 mm

3.2.8 The ends of the rails on the +/- Z face shall have a minimum surface area of 6.5 mm x 6.5 mm contact area for neighboring CubeSat rails (as per Figure 6).

3.2.9 At least 75% of the rail will be in contact with the P-POD rails. 25% of the rails may be recessed and no part of the rails will exceed the specification.

3.2.10 The maximum mass of a 1U CubeSat shall be 1.33 kg.

3.2.10.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.11 The maximum mass of a 1.5U CubeSat shall be 2.00 kg.

3.2.11.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.12 The maximum mass of a 2U CubeSat shall be 2.66 kg.

3.2.12.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.13 The maximum mass of a 3U CubeSat shall be 4.00 kg.

3.2.13.1 Note: Larger masses may be evaluated on a mission to mission basis.

3.2.14 The CubeSat center of gravity shall be located within 2 cm from its geometric center in the X and Y direction.

3.2.14.1 The 1U CubeSat center of gravity shall be located within 2 cm from its geometric center in the Z direction.

3.2.14.2 The 1.5U CubeSat center of gravity shall be located within 3 cm from its geometric center in the Z direction.

3.2.14.3 The 2U CubeSat center of gravity shall be located within 4.5 cm from its geometric center in the Z direction.

3.2.14.4 3U and 3U+ CubeSats' center of gravity shall be located within 7 cm from its geometric center in the Z direction.

3.2.15 Aluminum 7075, 6061, 5005, and/or 5052 will be used for both the main CubeSat structure and the rails.

3.2.15.1 If other materials are used the developer will submit a DAR and adhere to the waiver process.

3.2.16 The CubeSat rails and standoff, which contact the P-POD rails and adjacent CubeSat standoffs, shall be hard anodized aluminum to prevent any cold welding within the P-POD.

Appendix C

List of Vendors

McMaster Carr

- Aluminum stock
- Springs
- Hardware

Miga

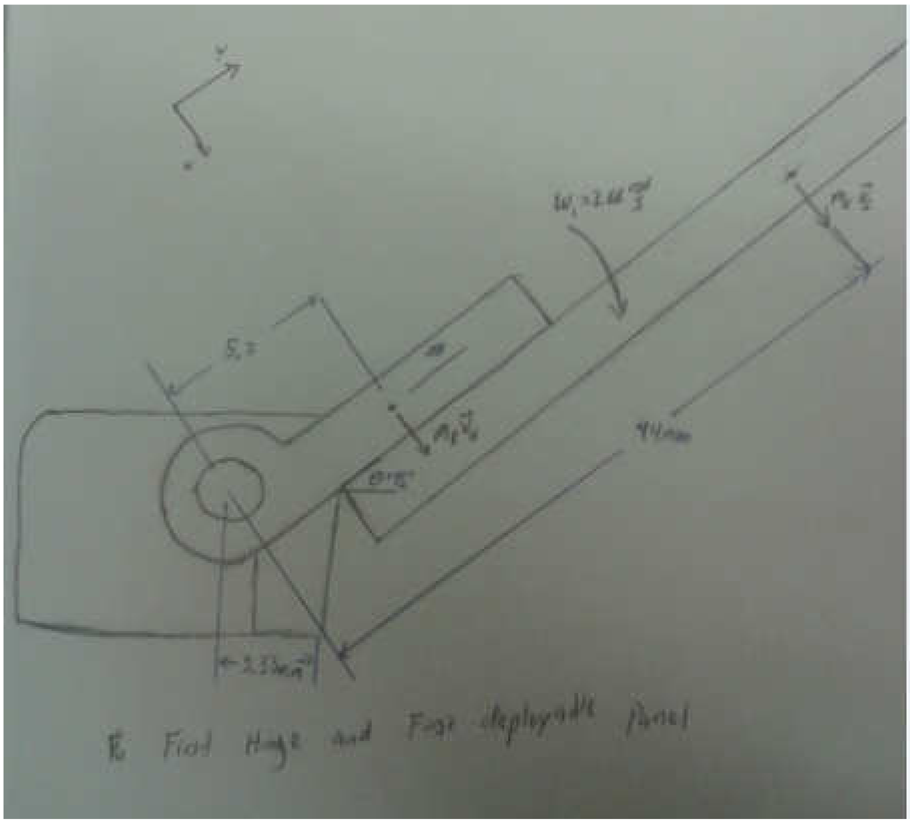
- Miga Motors

Appendix D

Dynamic Analysis and Stress on Hinges

DEPLOYABLE SOLAR PANEL DYNAMICS

(Hand Calculations)



Initially @ Rest $\quad T_1 = 0$

Subscripts
B → Hinge
S → side panel

$$T_1 + U_{1\to2} = T_2$$

$$I_O = \frac{1}{3} m L^2$$

$$\cancel{I_{OB} = \frac{1}{3}(4.808\times10^{-5}\,\text{slugs})(.65\,\text{in})^2 = 6.77\times10^{-6}\,\text{slug}}$$

$$\cancel{I_{OS} = \frac{1}{3}(.068\,\text{slugs})(5.65\,\text{in})^2 = .234\,\text{slug-in}^2}$$

I_{OB} is negligible

Length from pivot point to ~~center of mass~~ end

$$L_B = \cancel{5.2\,\text{mm}\left(\frac{1\,\text{in}}{25.4}\right) = .21\,\text{in}} = 16.5\,\text{mm}\left(\frac{1\,\text{in}}{25.4}\right) = .65\,\text{in}$$

$$L_S = \cancel{44\,\text{mm}\left(\frac{1\,\text{in}}{25.4}\right) = 1.73\,\text{in}} = 82.5\,\text{mm}\left(\frac{1\,\text{in}}{25.4}\right) = 3.25\,\text{in}$$

Mass: $m_B = .70\,\text{g}\left(\frac{6.85\times10^{-5}\,\text{slugs}}{1\,\text{g}}\right) = 4.808\times10^{-5}\,\text{slugs}$

$m_S = .068$ slugs

$$I_{OB} = \frac{1}{3}(4.808\times10^{-5}\,\text{slugs})(.65\,\text{in})^2 = 6.77\times10^{-6}\,\text{slugs-in}^2$$

I_{OB} is negligible

$$I_{OS} = \frac{1}{3}(.068\,\text{slugs})(3.25\,\text{in})^2 = .239\,\text{slug-in}^2$$

Spring Constant, $k = .00028\,\frac{\text{in-lb}}{\text{deg}}\left(\frac{360\,\text{deg}}{2\pi}\right) = 0.016\,\frac{\text{in-lb}}{\text{rad}}$

Using 3 springs $\quad K_{Total} = k_1 + k_2 + k_3 = \cancel{.032\,\frac{\text{in-lb}}{\text{rad}}}\ .048\,\frac{\text{in-lb}}{\text{rad}}$

Finding Angular Velocity right before it hits stop (135°)

$$\cancel{T_1}^{\,0} + U_{1\to2} = T_2$$

$$\int_0^{3\pi/4} \left(.048\,\frac{\text{in-lb}}{\text{rad}}\right)\theta\, d\theta = \frac{1}{2} I_{OS}\,\omega_S^2 + \underbrace{\frac{1}{2} I_{OB}\,\omega_B^2}_{0} \qquad \omega_B = \omega_S = \omega$$

$$\frac{\left(.048\,\frac{\text{in-lb}}{\text{rad}}\right)}{2}\left(\frac{3\pi}{4}\right)^2 = \frac{1}{2}\omega^2\left(\cancel{I_{OB}}^{\,0} + I_{OS}\right)$$

$$.133\,\text{in-lb} = (.1195\,\text{slug-in}^2)\,\omega^2$$

$$\left(\frac{12\,\text{in}}{1\,\text{ft}}\right) .133\,\frac{\text{in-slug-ft}}{\text{s}^2} = (.1195\,\text{slug-in}^2)\,\omega^2$$

$$1.6\ \frac{slug-in^2}{s^2} = (.119\ slug-in^2)\,\omega^2$$

$$13.45\ \frac{rad}{s^2} = \omega^2$$

$$\underbrace{\omega = 3.66\ \frac{rad}{s}\ \text{right before impact}}$$

Assume Inelastic collision

$\omega_f = 0$ so $V_{Sxf} = V_{Bxf} = V_{Sf} = V_{yf} = 0$

$$m_B V_{Bxi} + m_S V_{Sxi} + \int_{ti}^{tf} F_{Px}\,dt = m_B \cancel{V_{Bxf}}^{0} + m_S \cancel{V_{Sxf}}^{0}$$

$$m_B \cancel{V_{Byi}} + m_S \cancel{V_{Syi}} + \int_{ti}^{tf} F_{Py}\,dt = m_B \cancel{V_{Byf}} + m_S \cancel{V_{Syf}} \qquad F_{Py} = 0$$

$$V_{Bxi} = \omega_1 (5.2mm) = 3.66(5.2mm) = 19.03\ \frac{mm}{s} = .75\ \frac{in}{s} \qquad V_{Byi} = 0$$

$$V_{Sxi} = \omega_1 (44mm) = 161.04\ \frac{mm}{s} = 6.34\ \frac{in}{s} \qquad V_{Syi} = 0$$

$$F_{Px}(t_f - t_i) = m_B V_{Bxi} + m_S V_{Sxi}$$

$$= (4.808\times10^{-5}\ slugs)(.75\ \tfrac{in}{s}) + (0.068\ slugs)(6.34\ \tfrac{in}{s})$$

$$= .431\ \frac{slugs-in}{s}\left(\frac{1\,ft}{12\,in}\right)$$

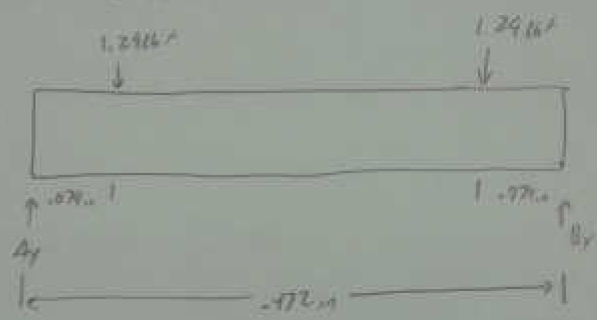
$$F_{Px}(t_f - t_i) = .036\ \frac{slugs-ft}{s}$$

$\Delta t = .007s$ Assume Impact time is .007s

$$\underbrace{F_{Px} = 5.13\ lbf \quad \text{Force due to both springs}}$$

$$\underbrace{\text{Per Hinge} \quad F_{Px} = 2.57\ lbf}$$

Stress Pin Due to shock load



@ $\Sigma M_A = 0$

$$-1.29\,lbf(.079\,in) - 1.29\,lbf(.472 - .079\,in) + B_y(.472\,in) = 0$$

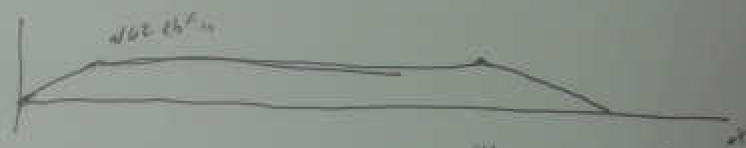
$$B_y = \frac{1.29(.079) + (1.29(.472 - .079))}{.472}$$

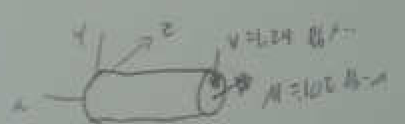
$$B_y = 1.29\,lbf$$

$\Sigma F_y = 0$

$$-1.29(2) + 1.29 + A_y = 0$$

$$A_y = 1.29\,lbf$$





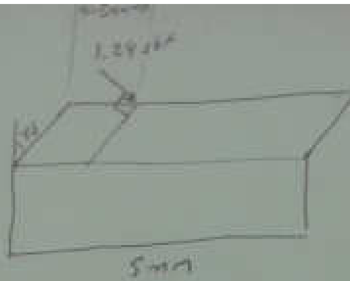
$$\sigma_{xx} = \frac{My}{I} \qquad I = \frac{\pi D^4}{64} = 1.88 \times 10^{-5}\,in^4$$

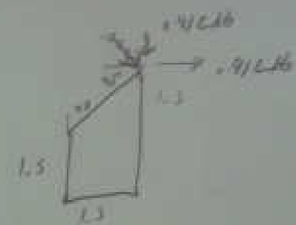
$$= \frac{(.102\,lbf\text{-}in)(1/32\,in)}{1.88 \times 10^{-5}\,in^4} = 2.128 \times 10^{3}\,psi$$

Aluminum Pin
should be OK

6061-T6
S_y = 30000 psi
6061-T4
8000

$$n = \frac{S_y}{\sigma} = \frac{8000}{2128} = 3.75$$





	Area	X_i	$X_i \cdot$ Area	Y_i	$Y_i \cdot$ Area
A_1	1.125mm	.1	1.125	2	2.25
A_2	2.25	.75	1.688	.75	1.6875
Sum	3.375		2.813		3.9375

$\bar{X} = \frac{\Sigma x_i \cdot A}{\Sigma A}$

$\bar{x} = .75$

$\bar{y} = 1.167mm$

	I_y	$A(dx)^2$	$I_y + Adx^2$
A_1	.141	.763	.844
A_2	.423		.423

$\bar{I}_y = 1.267mm^4$

A_1 $I_y = \frac{1}{36} bh^3$

A_2 $I_y = \frac{1}{12} bh^3$

$I_y = 1.267mm^4$

	I_x	$A(dy)^2$	$I_y + A(dy)^2$
A_1	.141	.781	.922
A_2	.423	.381	.814

d_y is distance from $\bar{x}$ to centroid of individual shape

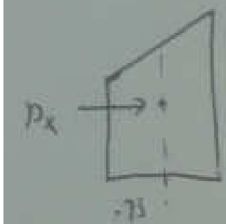
$\bar{I}_x = 1.736 m^4$

- Since the rail lies underneath the step tab, P_y should not be a problem

$M_y = P_x l = .078$ lbf·in

$$\sigma_x = \frac{Mc}{I_y} = \frac{(.078\,lbf \cdot in)(.03in)}{(1.267mm^4)(\frac{1in}{25.4mm})^4} = 707.6\,psi$$

$$n = \frac{S_y}{707.6} = \frac{8000}{707.6} = 11 \quad \text{Due to loading}$$



$$Q = \left[(.75)(1.5) + \frac{1}{2}(.75)(.75)\right] \times \left(\frac{.75}{2}\right) = 1.23\,mm^3 = 7.5\times10^{-5}\,in^3$$

$$V = P_x = .912\,lb$$

$$I = (1.267\,mm^4)\left(\frac{1\,in}{25.4\,mm}\right)^4 = 3.04\times10^{-6}\,in^4$$

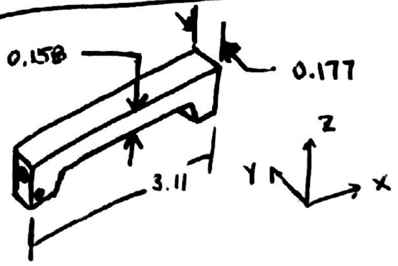
$$b = 1.5\,mm = .06\,in$$

$$\tau = \frac{VQ}{Ib} = \frac{(.912\,lb)(7.5\times10^{-5}\,in^3)}{(3.04\times10^{-6}\,in^4)(.06\,in)} = 380\,psi$$

$$n = \frac{S_y}{\tau} = \frac{8000}{380} = 21$$ Due to shear only

Appendix E

BEAM CALCULATION - CRITICAL BUCKLING LOAD (SMALL BEAM)



NOTE: DIMS IN INCHES

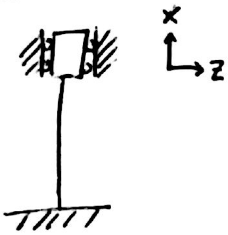
MATERIAL: ALUMINUM 6061
LENGTH, l = 3.11 INCHES
WIDTH, b = 0.177 INCHES
HEIGHT, h = 0.158 INCHES

FIND: CRITICAL LOAD~~ING~~ FOR BUCKLING

ASSUMPTIONS:

- LONG COLUMN
- COMPRESSION ALONG LENGTH THROUGH CENTER AXIS OF THIN PORTION
- WILL BUCKLE ABOUT THE y-AXIS
- FIXED-FIXED END CONDITION

ANALYSIS:



$$P_{cr} = \frac{C\pi^2 EI}{l^2}$$

EULER COLUMN EQUATION (REF: SHIGLEY'S M.E. DESIGN)

C ≡ END CONDITION CONSTANT = 4 FOR FIXED-FIXED

E ≡ MODULUS OF ELASTICITY = 10000×10^3 psi

I ≡ AREA MOMENT OF INERTIA

$$I = \frac{1}{12} bh^3 = \frac{1}{12}(0.177\ in)(0.158\ in)^3$$

$$I = 5.82 \times 10^{-5}\ in^4$$

$$P_{cr} = \frac{(4)\pi^2 (10000 \times 10^3\ psi)(5.82 \times 10^{-5}\ in^4)}{(3.11\ in)^2}$$

$$\underline{P_{cr} = 2375\ lbs_f}$$

ACCORDING TO NASA GEVS, STRUCTURE SHOULD BE ABLE TO WITHSTAND AT LEAST 14 G'S. THE BEAM ~~W~~ IS ~~PREDICTED TO~~ NOT PREDICTED TO UNDERGO MUCH LOADING — SUPPORTS THE RAIL PANEL AND MOUNTED DEPLOYABLES. THE RAIL PANEL WEIGHS ~ 0.5 lbs; ADDING ADDITIONAL MASS FOR DEPLOYABLES/OTHERS

MAX MASS TO SUPPORT: $\frac{P_{cr}}{14\ G} = \frac{2375\ \cancel{lbf}}{14(\cancel{32.174\ ft/s^2})} \times \frac{1\ lbm \cdot \cancel{32.174\ ft/s^2}}{\cancel{1\ lbf}} = 170\ lb_m$

↳ EACH SATELLITE (3U) IS ONLY ALLOWED TO BE 4 kg MASS

Appendix F

Gantt Chart

Tesseract

Start Date: 10/2/2014
End Date: 5/31/2015

Today: 1/30/2015
Senior Project Members: Edgar Uribe // Vanessa Faune

ID	WBS	Status	Task Name	Dur.	Start	Finish	Resource	Dependent	% Comp.
1	**1**	**On Schedule**	***Project Proposal Report***	**19**	**10/2/14**	**10/21/14**			**100%**
2	1.1	Complete	*Intro Letter to Sponsor*		10/2/14	10/2/14			100%
3	1.2	Complete	*Team Contract*	5	10/2/14	10/7/14			100%
4	1.3	Complete	*Background Research for QFD*	7	10/2/14	10/9/14			100%
5	1.4	Complete	*Problem Statement*	2	10/7/14	10/9/14			100%
6	1.5	Complete	*QFD House of Quality*	7	10/9/14	10/16/14			100%
7	1.6	Complete	*Compose Project Proposal Report*	5	10/16/14	10/21/14			100%
8	**2**	**On Schedule**	***Preliminary Design Report***	**24**	**10/21/14**	**11/14/14**			**14%**
9	2.1	Complete	*1st Team Evaluation*	2	10/21/14	10/23/14			0%
10	2.2	Complete	*1st Reflection*	5	10/23/14	10/28/14			0%
11	2.3	Complete	*Present Concept Models*	7	10/28/14	11/4/14			0%
12	2.4	Complete	*Pugh Matrices*	7	10/28/14	11/4/14			0%
13	2.4.1	Complete	*Structure Style Matrices*	7	10/28/14	11/4/14			0%
14	2.4.2	Complete	*Deployment Matrices*	7	10/28/14	11/4/14			0%
15	2.5	Complete	*Schedule PDR with sponsor*	2	11/4/14	11/6/14			0%
16	2.6	Complete	*Final Decision Matrix*	9	11/4/14	11/13/14			0%
17	2.6	Complete	*Structure Style Matrices*	9	11/4/14	11/13/14			0%
18	2.6	Complete	*Deployment Matrices*	9	11/4/14	11/13/14			0%
19	2.7	Complete	*Machine Shop Yellow Tag*	7	11/6/14	11/13/14			100%
20	2.8	Complete	*Preliminary Design Review & Report*	12	11/6/14	11/18/14			100%
21	2.8.1	Complete	*Create Presentation Slides*	6	11/12/14	11/18/14			100%
22	2.8.2	Complete	*Compose Preliminary Design Report*	8	11/6/14	11/14/14			100%
23	**3**	**On Schedule**	***Final Design Report***	**73**	**11/18/14**	**1/30/15**			**0%**
24	3.2	Complete	*Design Check 1*	11	11/14/14	11/25/14			0%
25	3.2	Complete	*Impact Test*	6	11/25/14	12/1/14			0%
26	3.1	Complete	*FMEA, DVP, Analysis Plan*	4	11/21/14	11/25/14			0%
27	3.2	Complete	*2nd Team Eval and Reflection*	1	12/1/14	12/2/14			0%
28	3.2	Complete	*Design Check 2*	7	11/25/14	12/2/14			0%
29	3.3	Complete	*PDR minutes*	1	12/4/14	12/5/14			0%
30	3.2	Complete	*Design Review*	4	12/8/14	12/12/14			0%
31	3.2	Complete	*Final Design Review*	23	12/28/14	1/20/15			0%
32	3.2	Complete	*Rapid Prototype/ Hardware*	3	1/5/15	1/8/15			0%
33	3.2	Late	*Proper Deployment Test*	4	1/8/15	1/12/15			0%
34	3.2	Late	*Solar Cell Damage Test*	4	1/8/15	1/12/15			0%
35	3.2	Late	*Deployment Verification Test*	4	1/8/15	1/12/15			0%
36	3.2	Complete	*Fit Check/ Assembly*	1	1/8/15	1/9/15			0%
37	3.2	Complete	*Make Corrections to 1st and 2nd Report*	5	1/6/15	1/11/15			0%
38	3.2	Late	*Rigidity Design*	12	1/6/15	1/18/15			0%
39	3.2	Complete	*Detailed Analysis*	7	1/11/15	1/18/15			0%
40	3.2	At Risk	*Part Drawings*	15	1/9/15	1/24/15			0%
41	3.2	On Schedule	*Final Report and Presentation*	13	1/11/15	1/24/15			0%
42	3.2	Complete	*Order Stock*		1/9/15	1/9/15			0%
43	3.4	Complete	*Oder long lead-time parts*	3	1/6/15	1/9/15			0%
44	3.4	At Risk	*Design Fixtures*	11	1/12/15	1/23/15			0%
45	3.5	At Risk	*Begin Cutting Stock*	18	1/12/15	1/30/15			0%
46	3.5	At Risk	*CAM Parts*	18	1/12/15	1/30/15			0%
47	3.5	On Schedule	*CNC Parts*	59	1/12/15	3/12/15			0%
48	3.5	On Schedule	*Write Up Assembly Procedures*	22	1/8/15	1/30/15			0%
49	3.5	On Schedule	*Write Up Testing Procedures*	27	2/1/15	2/28/15			0%
50	3.5	On Schedule	*Assembly*	11	3/16/15	3/27/15			0%
51	3.5	On Schedule	*Testing*	32	3/30/15	5/1/15			0%
52	3.5	On Schedule	*Internal Component Access Test*	32	3/30/15	5/1/15			0%
53	3.5	On Schedule	*Stack Access Test*	32	3/30/15	5/1/15			0%
54	3.5	On Schedule	*Weigh Structure*	32	3/30/15	5/1/15			0%
55	3.5	On Schedule	*Measure Inner Volume*	32	3/30/15	5/1/15			0%
56	3.5	On Schedule	*Shock Testing*	32	3/30/15	5/1/15			0%
57	3.5	On Schedule	*Vibration Testing*	32	3/30/15	5/1/15			0%
58	3.5	On Schedule	*TVAC Testing*	32	3/30/15	5/1/15			0%
59	3.5	On Schedule	*Measure Outermost components*	32	3/30/15	5/1/15			0%
60	3.5	On Schedule	*Final Report*	27	5/4/15	5/31/15			0%