

Feasibility Investigation of a Cubesat Modular and Rotatable Solar Array

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Master of Science in Aerospace Engineering

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Variables, Notations, Acronyms and Constants

ACS - Active Magnetic Attitude Control System

AFRL/RV – Air force Research Lab, RV unknown

COTS - Commercially Available off The Shelf

CPU - Central Processing Unit

GEO - Geosynchronous Equatorial (earth) Orbit

GN&C - Guidance Navigation & Control

GPS - Global positioning system

K - degrees Kelvin

M - meters

NASA - National Aeronautics and Space Administration

PCB - Printed Circuit Board

P-POD - Poly picosatellite orbital deployer

LEO - Low Earth Orbit (200-1000 km altitude, -50 to +50 latitude)

OBC - On Board Computer

TLE - Two Line Element

TT&C - Tracking Telemetry and Command

VHF-Very High Radio Frequency 30 MHz to 300 MHz

W - watts

Summary

The first part of the report is concentrated on the feasibility of a 3 unit CubeSat rotary deployed Solar Array. A sketch is provided of a modular Cubesat with a six panel modular array. It consists of an analysis using Orbital mechanics to find the power provided by such an array. This theoretical power supply is compared to a past mission, showing feasibility. The second part of the report reduces the six panel array to a two pedal array to enable initial design of the deployment mechanism. The mechanism is found to require a novel slip ring configuration that reduces the previous relative Solar panel area. This reduced panel area is again compared to an Electrical Power Supply of a previous mission to assess feasibility. The system is found to be feasible but the weight increase is large compared to standard Solar panel designs.

Introduction

A CubeSat is a nanosatellite providing relatively low cost payloads to conduct research or demonstrate technology in space. In this project a CubeSat is limited to a low earth orbit (LEO) this is an orbit around Earth with an altitude between 160 kilometers (99 mi), (orbital period of about 88 minutes), and 2,000 kilometers (1,200 mi) (about 127 minutes).

California Polytechnic State University at San Luis Obispo (Cal Poly SLO) and Stanford University have developed a widely accepted educational CubeSat standard. [1]. This specification is included in appendix A. The size and weight of the CubeSat was dictated by its launchers deployment system:

A 10cm x 10cm x 10cm CubeSat is referred to as a “1U” CubeSat see figure 2 below. They may be stacked such that a 2U CubeSat is 10cm x 10cm x 20cm and a 3U is 10cm x 10cm x 30cm. The spring loaded Cubesat launcher is named a P-Pod seen in figure 3. A CubeSat fits into the P-Pod for deployment. It must have a mass of 1Kg or less.



Figure 2: CubeSat [1]



Figure 3: P-Pod [1]

Legacy dictates a minimum of six subsystems are included in the CubeSat

- Structural
- Electrical Power (EPS)
- Bus-Data Handling,
- Communications (Comm)

- Attitude Determination and Control (ADCS)
- Thermal

The Avionics have risen to use an rs-422: serial interface standard applicable to windows (IBM architecture) to allow these nanosatellites to be programed from a personal computer.

The Satellite Bus also has a basic architecture that has become a commercial standard. This standard is PC/104. “PC/104 is a standard which specifies form factors and computer buses. It is intended for specialized environments where a small, rugged computer system is required. The standard is modular, and allows consumers to stack together boards from a variety of COTS manufacturers to produce a customized embedded system.” [2] In addition to standards that provide a systemic form, NASA-STD-4005: Low Earth Orbit Spacecraft Charging Design Standard is usually applied to the avionics and Electrical Power System. In regard to Structural standards these are dictated in joint venture between the designers and their launch provider.

Structural Modularity historically has comprised a frame made of 7075-T73 aluminum, some modular configurations follow:

Figure 1 below was presented in a PowerPoint presentation via Department of Defense as the first modular Small Satellite, not a “Cubesat but none the less the first modular design of a satellite.

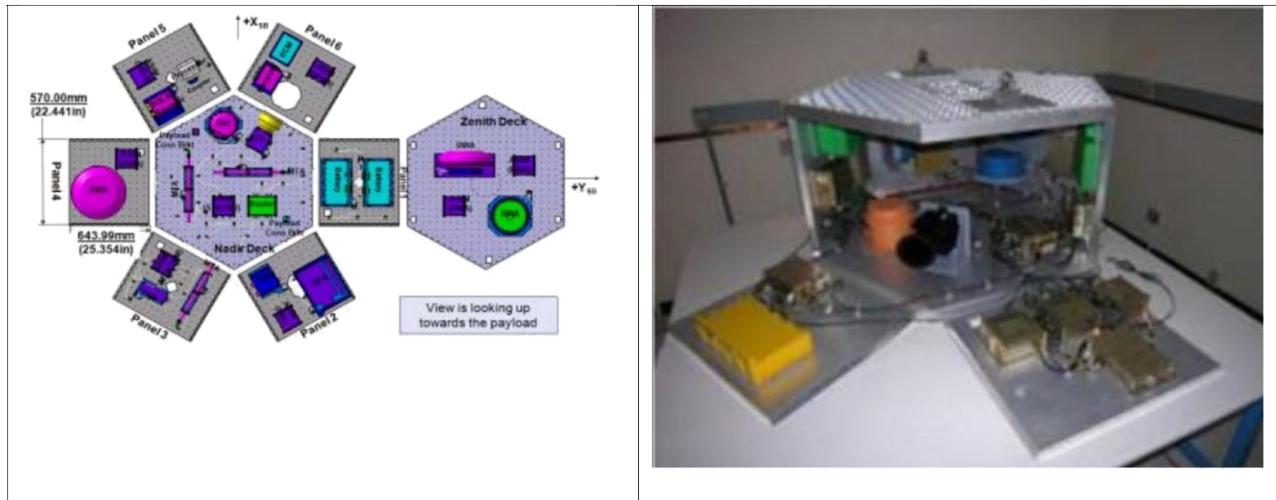


Figure 1: 1st Modular Satellite [4]

Addressing Cubesat modularization, there have been intensive attempts to make structures that could be interchangeable and expandable. One truly modular design was accomplished by the Air Force Research Laboratory's Space Vehicle Directorate (AFRL/RV). The electronic boards (nanaomodules) fit into facets on the modular structural panels and fold into a cube see figure 2. Alternately all COTS providers have defined by legacy a structural standard shown in figure 3. The boards are stacked within a structural space frame per Cal Poly specs.

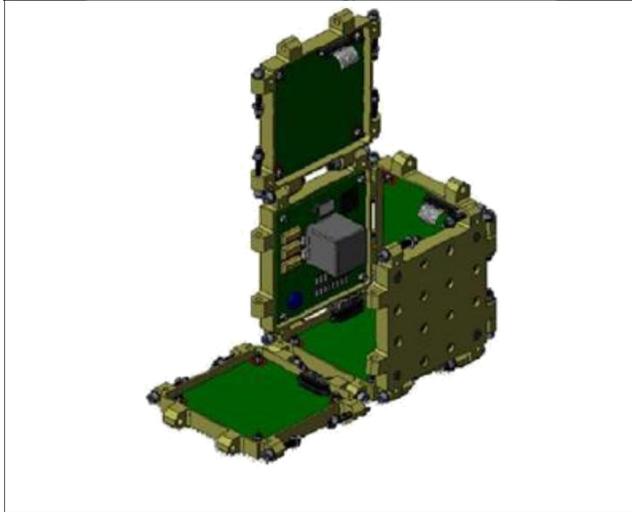


Figure 2: Nano modular CubeSat [3]

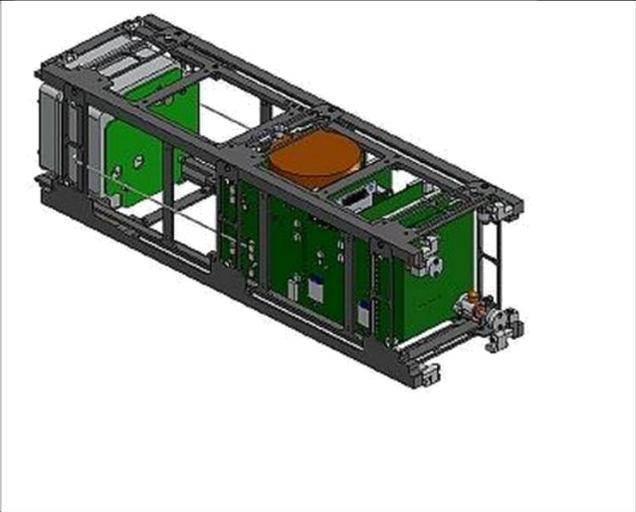


Figure 3: Standard Modular space frame [4]

A satellite Bus is the infrastructure of a spacecraft. It is the collection of the subsystems (modules) less the payload, their relative position and way of mating. One standard that is repeatedly referenced in papers on CubeSats is the PC104 standard from the computer industry. It seems that its Form Factor which is defined to be 3.550×3.775 inches (90×96 mm), with mounting holes at all four corners of the board serves the CubeSat designer quite well. However the specifications also allow for a 0.5 inches (13 mm) area beyond the edge of the PCB for I/O connectors which seemingly would not allow a COTS motherboard to be useable but so close as to be a basis for design.

There are other standards from the computer industry that may serve as a basis for a CubeSat depending on one's needs. Some designers have sought their own form attempting to create a standard. AFRL/RV proposed and implemented three architectures in an attempt to establish a standard for Bus, GNC and TT&C to include plug and play interfacing for attitude control but in the six years after that, till now there is not a required standard.

See appendix B for an actual data sheet for a commercially available 3U nanosatellite Bus.

I refer the reader to the ALL-STAR (Agile Low-cost Laboratory for Space Technology Acceleration and Research) 3U CubeSat which was designed between 2010 -2012 in joint venture between Colorado Space Grant Consortium (COSGC) and Lockheed Martin. This is an excellent specimen of Bus modularity to a PC/104, RS - 422 standard and Figure 4 shows a matching architecture.

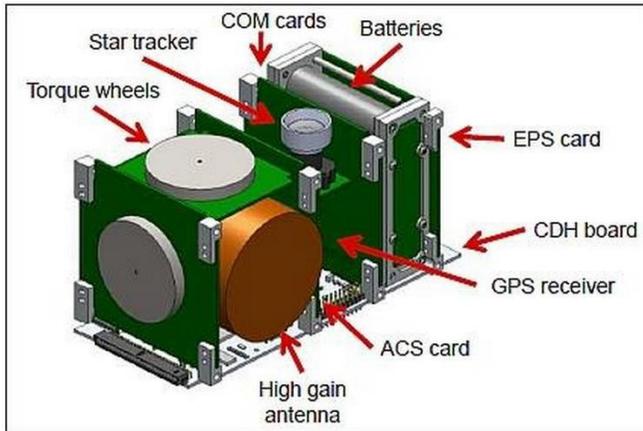


Figure 4: Bus electronics layout (image credit: COSGC) [4]

Further Figure 5 shows the entire ALL-STAR Satellite with horizontal rotary deployable Solar panels.

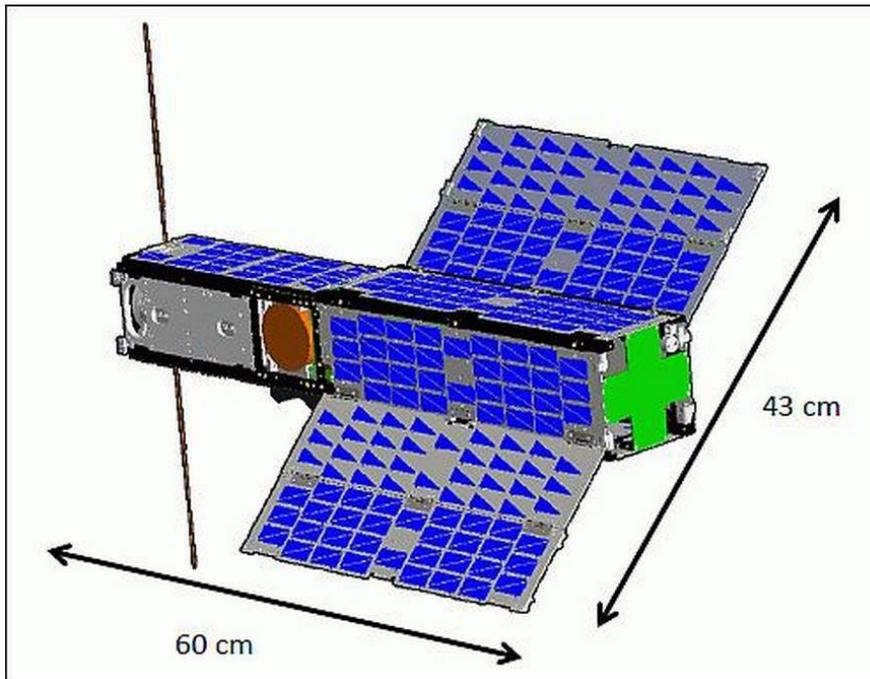


Figure 5: Illustration of the fully deployed All-Star nanosatellite (image credit: COSGC) [4]

Modular Bus technology has become the standard. It is described by these, :

1. Computer standard PC104 coupled with RS-422 previously described
2. Space Plug-and-play Avionics (SPA), see following description, and
3. Modular Open System Architecture (MOSA)

“A Modular Open Systems Approach (MOSA) is an integrated business and technical strategy for developing flexible and standards-based architectures to achieve affordable, interoperable, and sustainable systems. As a business strategy, MOSA aims at reducing the total system ownership costs using the latest products and state-of-the-art technologies from multiple sources”. [5]

SPA is a set of principles that facilitate the automatic resource discretion, resource discovery, network self-organization of of systems, and facilitates the automatic management of components(“care and feeding”) and relationships between those components.[6]

Core technologies of SPA Space Plug-and-play(SPA) is:

- A Set pf technologies
- A Brand of plug-and play (PnP) focused on shortening the time to construct a complex system.

Key technology elements which are:

- Hardware that is self-describing components and self-organizing networks
- Software consisting of Electronic datasheets(“XTEDS”) and their vocabulary enabling automatic component discovery.

- Tools for push-button tool flow and test bypass

Figure 6 shows an SPA satellite data model that would be available on a computer network.

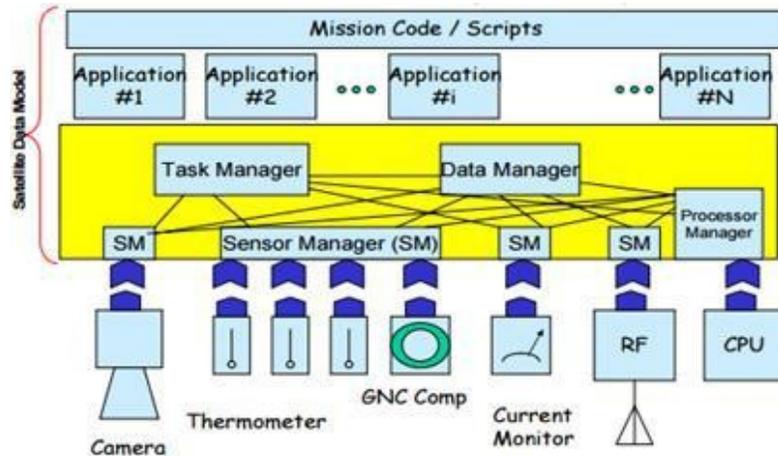


Figure 6: SPA Satellite data model [6]

The EPS provides electrical energy to the Satellite systems. It consists of solar cells, a rechargeable battery pack, and power regulation board. The solar cells are the primary source of energy. The photovoltaic Solar cells convert light into electrical energy. The secondary Lithium batteries provide power during the eclipse and when power draw is more than the Solar Cells can provide. The power regulation board provides power to the systems and to the battery.

Solar panel power configurations which is the topic of this report historically have been in two accepted formats. The first is solar panels on the sides of the CubeSat (fig.7) or secondly, panels that fold flat to the side(s) of the CubeSat vertically (fig. 8) or, horizontally (fig. 9).

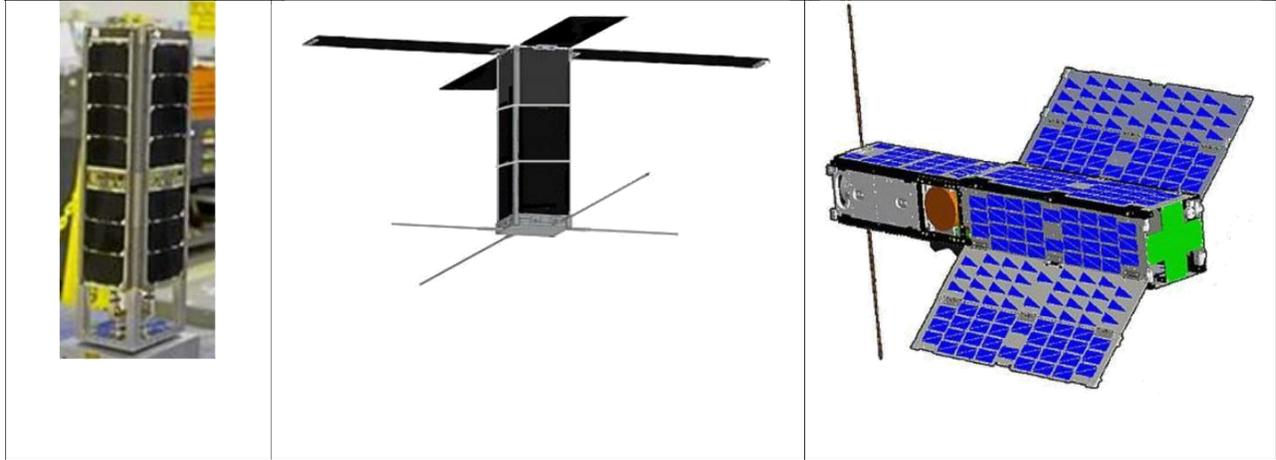


Figure 7:Side panels[1]

Figure 8: Vertically foldable solar panels

Figure 9: Horizontally deployed Panels [5]

Figures 8 and 9 show modular rotary deployable Solar Panel Arrays and though of slightly different orientation if a patent were granted in re one of the designs it would cover in likely development of 5 years the other.

The total delivered power of the 3U panels is in the range of 22 to 56 W. The system described in this paper unlike the aforementioned panels has a frame just like a cubesat and the panels rotate into position. The panels at all times, stowed or deployed are perpendicular to their axis of rotation.

1.0 Literature Review

There were two instances of a rotary deployable 3U CubeSat (nanosatellite) found while searching for a preexisting like design.

The first instance is by Fabio Santoni and his team from the University of Rome published in IAA 2014, titled, An orientable solar panel system for nanospacecraft, in which is sited, “ An orientable deployed solar Array system for 1-5 kg weight nanospacecraft is described enhancing the achievable performance of these typically power-limited systems. The system is based on deployable solar panel system, previously developed with cooperation between Laboratorio di Sistemi Aerospaziali of University

of Roma”la Sapienza” and the company IMT(Ingegneria Marketing Tecnologia). The system is modular one, and suitable in principle for the 1U,2U and 3U CubeSats. The size of each solar panel is the size of a lateral CubeSat surface. A single degree of freedom maneuvering capability is given to the deployed solar array, in order to follow the apparent motion of the sun.....” [7]. Though the fore mentioned novel solar panel system is modular, the panel(s) are hinged not strictly pivoted as in this papers explored design.

The second instance is by Nathan K. Walsh, College of Engineering, University of Hawai’i at Mānoa, titled, DEVELOPMENT OF A DEPLOYABLE 3U CUBESAT SOLAR PANEL ARRAY, in which is sited, “The primary goal of this project is to design, fabricate, and test a deployable solar array for a 3U CubeSat. The deployable mechanisms will adhere to the design restrictions of the standardized 3U CubeSat. The mechanisms will consider the capabilities of the Attitude Determination and Control System (ADCS) to ensure a smooth deployment.....”, [8] Both Solar panel designs are for practical purposes exactly the same and shown in figure 10



Figure 10: Modular deployable Solar panels[8]

Loads on the CubeSat must be accounted for in the forthcoming design investigation,

The following graphic shows the Cubesat modular and rotatable solar Array under investigation.

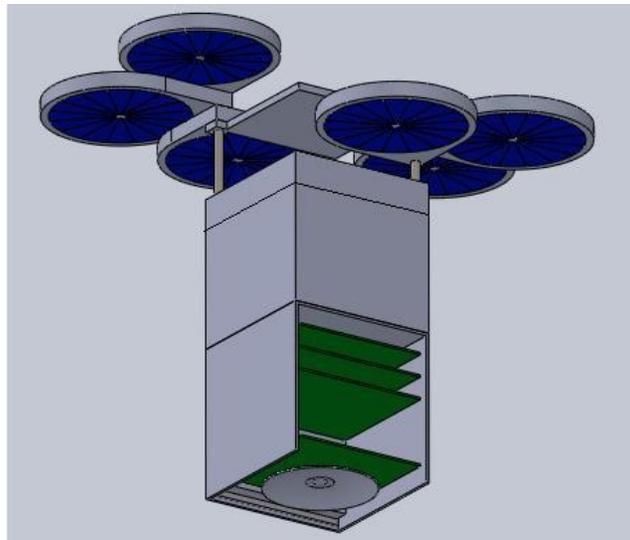


Figure 11: Cubesat modular and rotatable solar Array under investigation

2.0 Rotary Solar panel electrical analysis

As previously stated Cube satellites are governed by a standard created by Stanford and Cal Poly. The requirements restrict any material from protruding from the surface of the cube to 6.5 mm which makes deployable solar panel arrays a much more difficult option. The 6.5mm constraint means that the stack of solar panel be impossibly thin or a second CubeSat type module containing the stack be added. Due to the deployment mechanism the later choice is made.

In regard to Solar array power output, for comparison I site a typical 3U CubeSat solar array output Power referenced in, “Electrical power system for a 3U CubeSat nanosatellite incorporating peak power tracking with dual redundant control by Bester published in PRZEGLĄD ELEKTROTECHNICZNY (Electrical Review), ISSN 0033-2097, R. 88 NR 4a/2012.” [11]

“ A typical 3U CubeSat solar array configuration is two cells in series with

three such groups in parallel, giving a power output of: ... $P_{out} = 6.0675W$ " [10]

The proposed array consists of 6 solar panels each having two sides. The panels are photovoltaic Silicon, Gallium-Arsenide. Each side has a circular array of solar cells.

- Solar cell area per pedal per side is min: .0064 m², max: 0.00785398 m².
- Thickness: 140 [μm] • Weight per pedal side: .52[g]
- Advanced triple junction InGaP/GaAs, Ge substrate cell
- Efficiency (BOL) = min. 27.5 [%]
- Efficiency (EOL) = min. 25 [%]
- Open circuit voltage each: 2.616 [V]
- Short circuit current each: 462 [mA]
- Degradation of GaAs Cells per year = 2.75% [SMAD 417]

Upper and lower solar panel area is then 384 cm²

Two sources of energy are available to the solar panels, Sun solar radiation 1353 W/m², Albedo of the earth 406 W/m². I assume the top Solar panels are illuminated from the sun and the lower panels illuminated from Earth Albedo. The satellite is in Low Earth Orbit with the inclination of 96 degrees and height of approx. 600 km. Velocity of the satellite on orbit is estimated to be 27000 km/h. Based on these parameters, revolution time is calculated.

Given Earth's radius (equatorial) = 6978.1 Km

$$\text{Radius of orbit from earth center} = 6978.1 \text{ Km} + 600 \text{ Km} = 7578.1 \text{ Km} \quad (2.1)$$

$$\text{Circumference of circular orbit:} = 2\pi r = 47614.607 \text{ Km} \quad (2.2)$$

$$\text{Velocity of Satellite} = 27000 \text{ Km/h} \quad (2.3)$$

$$\text{Revolution time} = \frac{47614.607}{27000} = 1.764 \text{ h} = 105.81 \quad (2.4)$$

For solar panels to achieve 100% efficiency they need two degrees of freedom or to articulate. Since the assembly is static it can be assumed the panels will be 90% efficient at maximum illumination. To determine the duration of direct sun illumination on the upper panels we need the duration of the satellite eclipsed by earth when it passes through the earth's shadow. The shadow is assumed cylindrical. Computation of the time the satellite is in eclipse is a function of orbital mechanics explained as follows: The following calculations are based on explanations from a text book, the reference is: [12] and the process is explained in Appendix D.

2.1 Orbital Mechanics

$$= -1[\quad (\quad - \quad) +] \quad (2.1.1)$$

Where,

2015) = , . The date is chosen as a median value to liberate an average result (March 21

$$= 0 \text{ degrees } 00 \text{ minutes}$$

i = inclined orbit angle as referenced to equatorial plane

$$\Omega = h = 96 \text{ degrees}$$

$$= 0 \text{ degrees}$$

=

$$= 0 \quad = 0 \text{ at vernal equinox, March 21}$$

$$* = -1 [0 \quad 96 \quad (0 - 0) + \quad 0 \quad 96]$$

$$* = -1 [\lambda + h]$$

Where,

$$R = \text{earth equatorial radius} \quad (2.1.2)$$

$$= 6978.1 \text{ Km}$$

$$h = \text{satellite altitude above earth} \quad (2.1.3)$$

$$= 600 \text{ Km}$$

$$* = 1.170 \quad -1 [6978.1 \text{ Km} / (6978.1 \text{ Km} + 600 \text{ Km})]$$

$$= \frac{2\pi}{360} = \frac{1}{180} = \frac{1}{180} \text{ rad} = .0175 \text{ rad}$$

$$= 1 = .318$$

Computed Orbit time = 105.81 min

Eclipse time is then 105.81 * .318 = 33.68 min = 2020.8 s

$$\text{Sun Time} = 72 \text{ min} = 4328 \text{ s} \quad (2.1.4)$$

$$(2.1.5)$$

$$(2.1.6)$$

2.2 Power Calculation

Power of the top panel is computed:

Beginning of life[BOL]

End of life [EOL]

$$P_{\text{Upper Panel total power}} = \overline{P} \times 6 \quad (2.2.1)$$

$$P_{\text{Upper Panel total power}} = 1367 \left[\frac{\text{W}}{\text{m}^2} \right] \times .28 \times 6 \times .0064 \left[\text{m}^2 \right] \times .6 = 8.8 \text{ W} \quad (2.2.2)$$

Lower panel total power

$$P_{\text{Lower panel total power}} = 406 \left[\frac{\text{W}}{\text{m}^2} \right] \times .28 \times 6 \times .0064 \left[\text{m}^2 \right] \times .6 = 2.6 \text{ W} \quad (2.2.3)$$

$$\text{Total BOL Power} = 8.8\text{W} + 2.6 \text{ W} = 11.4 \text{ W} \quad (2.2.4)$$

$$\text{BOL Energy per cycle [J]} = \text{BOL Power [W]} \times \text{Sun time [s]} \quad (2.2.5)$$

$$= 11.4\text{W} \times 4328 \text{ s} = 49339 \text{ J}$$

The Cubesat is only using energy in eclipse so the amount of energy will remain the same, but the time to use the energy will be shorter. Calculating the Eclipse power available from the battery it is then:

$$P_{\text{Eclipse}} = \frac{E_{\text{BOL}}}{t_{\text{Eclipse}}} \quad (2.2.5)$$

$$[] = \frac{49339}{2024.1} = 24.4 \quad (2.2.6)$$

$$[] = [] * (1 -) \quad (2.2.7)$$

$$[] = 24.4 * (1 - .0275) \quad (2.2.8)$$

$$= 23.7 \text{ W}$$

We see the array with a battery as an EPS is in the ballpark but transmission of electricity through deployment has not been addressed. So it will be reduced in size. The calculations are repeated for only 2 pedals. and compared against the following 1U previous CubeSat missions

a. DC/PIP

DC/PIP mission	Budgeted		Estimate				
	Percentage of Total	Allocated Power (W)	Standby Power (W)	Peak Power (W)	Peak On Time (%)	Average Power (W)	Variance (W)
Thermal	11%	0.14	0.002	2.01	7%	0.14	0.000
C&DH	11%	0.14	0.05	0.30	30%	0.13	0.02
Communications	16%	0.21	0.10	1.00	5%	0.15	0.06
Science - Board	42%	0.55	0.00	1.50	17%	0.26	0.29
Science - Magnetometer	5%	0.07	0.00	0.025	100%	0.03	0.04
Total Allocated	85%	1.11	0.15			0.67	0.44
Contingency	15%	0.20				0.20	
Total Power	100%	1.30				0.87	0.43

b. GPS

GPS mission	Budgeted		Estimate				
	Percentage of Total	Allocated Power (W)	Standby Power (W)	Peak Power (W)	Peak On Time (%)	Average Power (W)	Variance (W)
Thermal	11%	0.14	0.002	2.01	7%	0.14	0.000
C&DH	11%	0.14	0.05	0.30	30%	0.13	0.02
Communications	16%	0.21	0.10	1.00	5%	0.15	0.06
Science - Board	42%	0.55	0.00	2.00	27%	0.54	0.01
Science - Magnetometer	5%	0.07	0.00	0.025	100%	0.03	0.04
Total Allocated	85%	1.11	0.15			0.95	0.15
Contingency	15%	0.20				0.20	
Total Power	100%	1.30				1.15	0.15

Figure 12: Sample 1U power budgets from LEO-Based Earth Science Missions12. [13]

In both budgets the allocated peak power is 1.3 Watts, with a maximum of 1.15 watts. This 1.15 watts then needs to be provided by the revised solar array for the modular redesign to be a valid configuration. The average output of the new standalone array is calculated as:

$$P = \frac{1367 \text{ W}}{2} * .28 * 2 * .0064 \text{ [}^2\text{]} * .6 \quad (2.2.9)$$

$$P = 1367 \text{ [}^2\text{]} * .28 * 2 * .0064 \text{ [}^2\text{]} * .6$$

$$= 2.1 \text{ W} \quad (2.2.9.1)$$

Lower panel total power

$$P = 406 \text{ [}^2\text{]} * .28 * 2 * .0064 \text{ [}^2\text{]} * .6$$

$$= .873 \text{ W} \quad (2.2.9.2)$$

$$\text{Total BOL Power} = 2.1\text{W} + .873 \text{ W} = 2.98 \text{ W} \quad (2.2.9.3)$$

$$\text{BOL Energy per cycle [J]} = \text{BOL Power [W]} * \text{Sun time [s]} \quad (2.2.9.4)$$

$$= 2.98\text{W} * 4328 \text{ s} = 12923 \text{ J}$$

The CubeSat is only using energy in eclipse so the amount of energy will remain the same, but the time to use the energy will be shorter. Calculating the Eclipse power available from the battery it is then:

$$P = \frac{12923 \text{ J}}{2000 \text{ s}} \quad (2.2.9.5)$$

$$P = \frac{12923 \text{ J}}{2000 \text{ s}} = 6.395 \quad (2.2.9.6)$$

$$\begin{aligned} [] &= [] * (1 -) \\ [] &= 6.395 * (1 - .0275) \end{aligned}$$

$$= 6.2 \text{ W} \quad (2.2.9.7)$$

The Solar Array Module 2 panel design is shown in figure 13, below, enabling novel rotary deployment. The problem is that it cannot be made to fit onto a 1U design package per PPOD specs. To comply with the P-POD deployment spec an attachable module of CubeSat form is made able to slide out of the PPOD conforming to the same specification as the CubeSat seen in Figure 13.

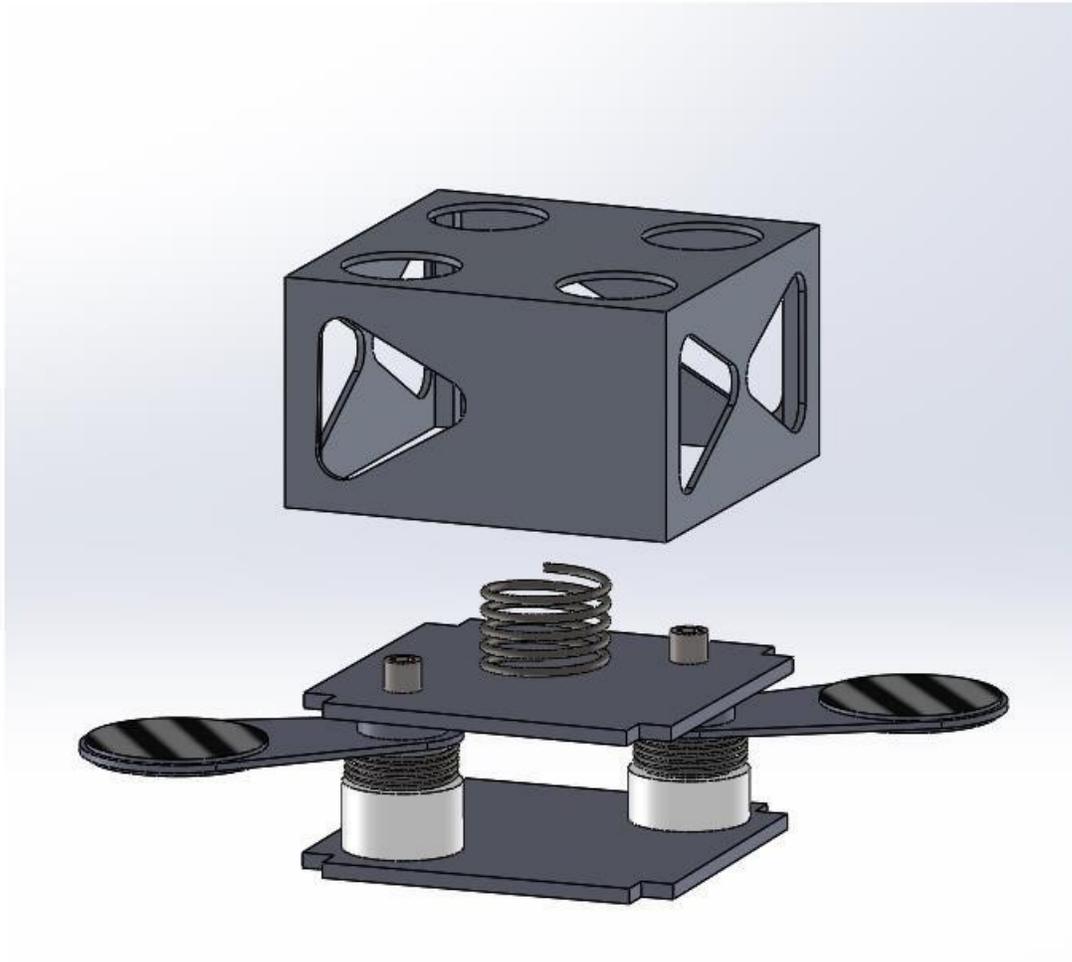


Figure 13: modular rotating Solar panel assembly

The deployable Solar Panels are constructed from printed circuit boards (E-Glass) conforming to standard IPC-4101B/21, the frame is Aluminum and springs are cold drawn steel (music wire).

Electricity from the solar panels is transferred down through the base plate via slip rings and spring loaded carbon brushes insulated from each other seen in Figure 16.

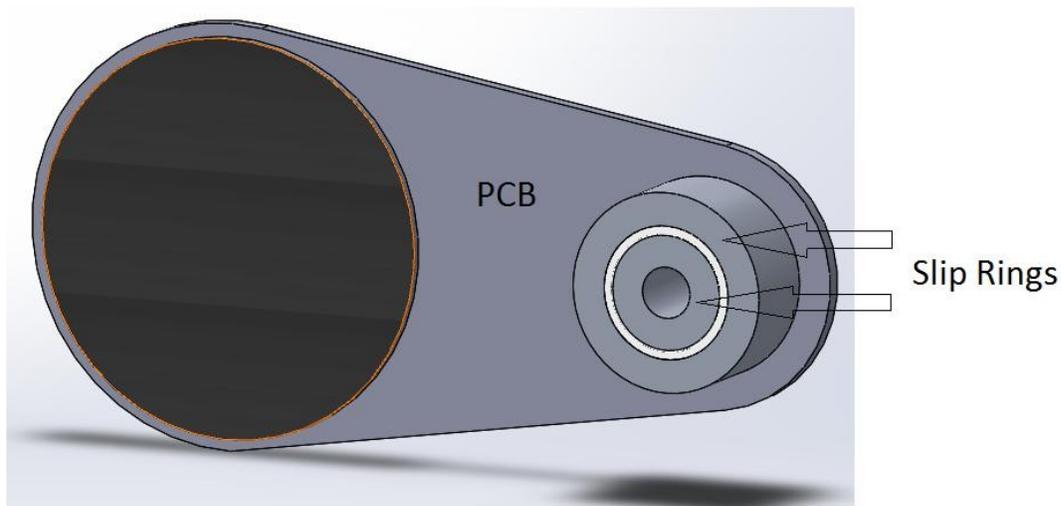


Figure 14: Solar Panel with electrical Slip Rings

3.0 Mechanical/Stress Analysis

A Static Stress Analysis was run in Solidworks 2014. The deployment of the solar Panels does not represent a stress mode of concern for analysis as there is no hard stop to the event. The panels upon deployment would slowly oscillate with reducing frequency till reaching a full stop. Likewise the spring loaded ejection bridge was designed robust enough that it too is neglected. Of concern is the solar panel deflection during stowed launch. The solar panel is thought to be the most likely candidate for failure, as such it is chosen for analysis. The solar panel in the assembly during the analysis is considered fixed.

Note: The Solidworks graphic of deflection is exaggerated, the actual deflection via the scale is .02mm not enough to drive the material past the elastic range.

Property	Measured	Manufacturers values	Units
Young's modulus (x axis)	4.02×10^{10}	3.50×10^{10}	N/m ²
Young's modulus (y axis)	4.56×10^{10}	3.50×10^{10}	N/m ²
Density	2481	2400	kg/m ³
Shear modulus	1.21×10^{10}	Not given	N/m ²
Poisson ratio	0.3	0.3	
Thickness	1.54-1.66	1.65	mm

Figure 15:PCB Material Values [11]

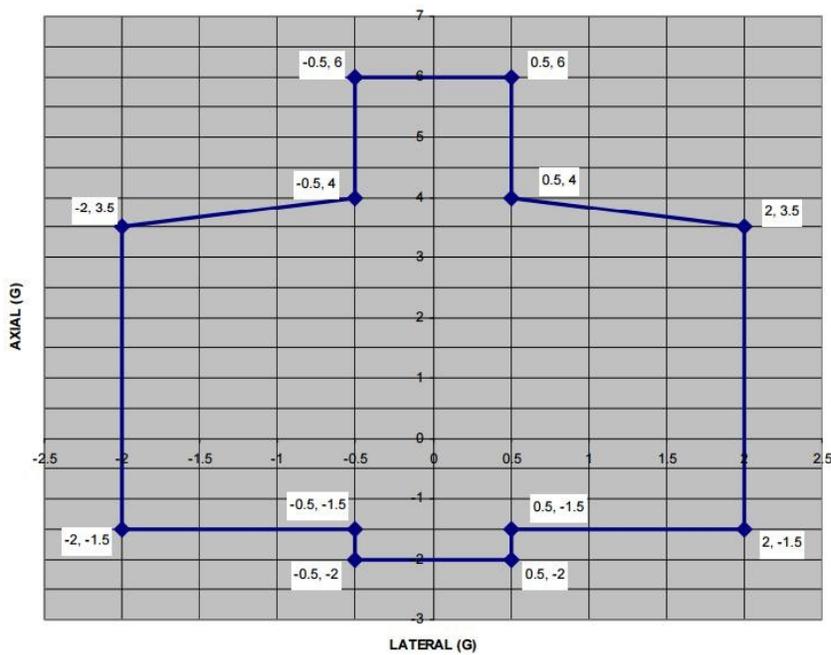


Figure 16: Falcon 9 load factors via SpaceX [14]

A static stress analysis was conducted with Solidworks the results follow,

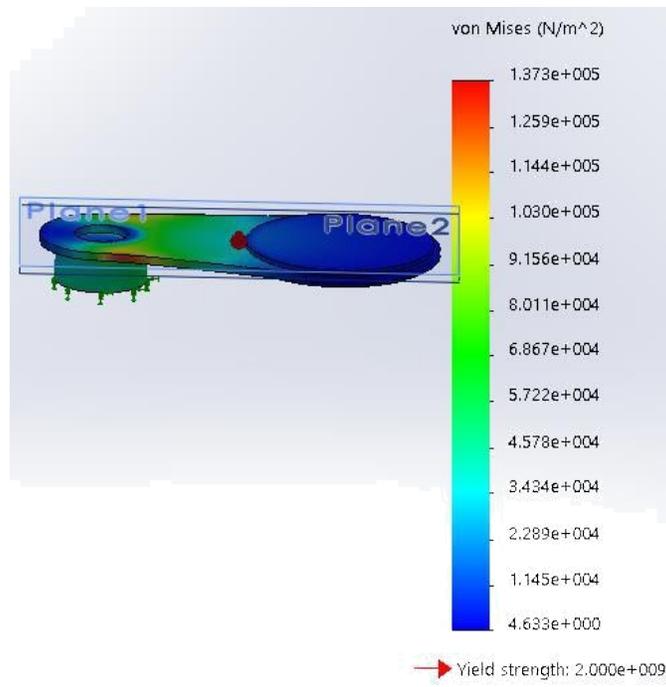


Figure 17: Solar Panel Von Mises static nodal Stress

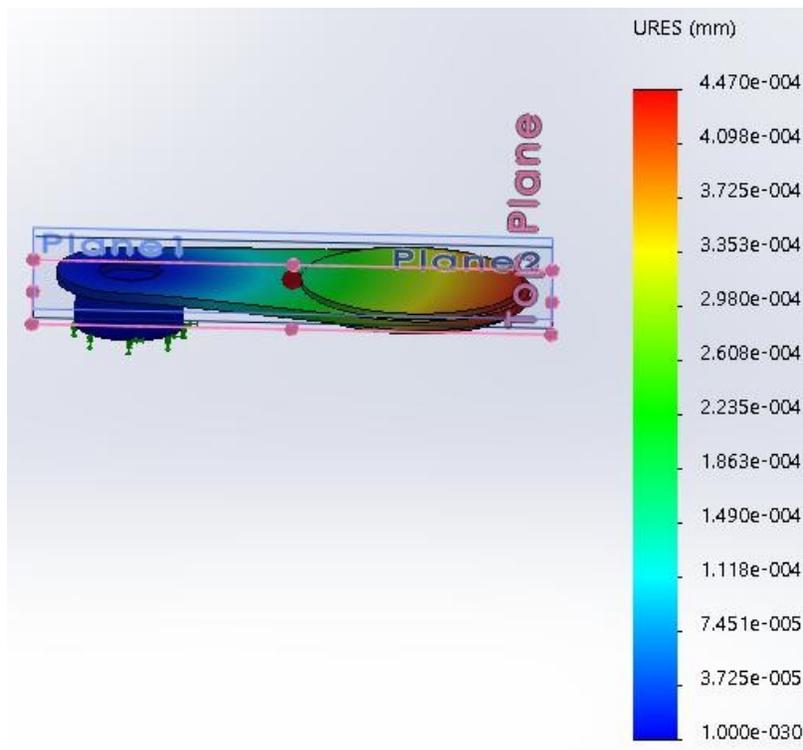


Figure 18: Solar Panel deflection

Torsion Spring calculation:

A stock torsion spring shown in figure 19 was chosen for the mechanism to deploy the array. The spring chosen is .072 in diameter made of music wire(cold drawn carbon steel). The wound OD is .593 in. and it has 5 turns. The following calculations are provided to check, 1) Torque and subsequent force on the restraining box that holds the Panels in the stowed position, 2) Angular deflection to ensure the panels swivel out enough and 3) the reduction in diameter of the loaded spring allows the pin that it sits around to be used without breaking the spring.

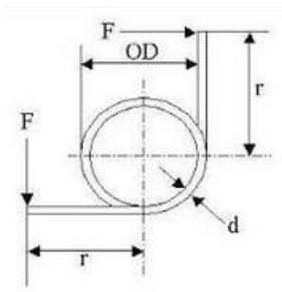


Figure 19: Solar array type torsion spring

References for the following calculations are from McGraw Hill, Mechanical Engineering Design 5th Ed., 1989. [15]

1) First calculating the torque:

Where: A= Spring intercept/min tensile strength referenced from McGraw hill Table 10-5 pg 422

M= exponent from McGraw hill Table 10-5 pg 422

D_i = reduction in spring diameter due to winding

N=Number of turns

m = Spring Exponent from McGraw hill Table 10-5 pg 422

= Ultimate tensile strength
 = yield strength

$$= \frac{186}{0.78} = 286 \quad (3.1)$$

The mean coil diameter is $D=0.593-0.072 = 0.521$ (3.2)

$C=OD/d = 7.24$ (3.3)

The stress concentration factor on a fiber on the inside of the coil is then (3.4)

The maximum torque F_r is given by: (3.5)

(3.6)

No safety factor has been used because the value of S_y used is an allowable value.

$$= \frac{4}{3} = 27.974 \quad (3.7)$$

Thus the torque of $F_r = 7.33$ lb per turn, which is good because a torque of 7.33 will wind the spring which is a relatively low value of force against the Solar Panel restraining box cover used to stow the panels for flight.

The number of actual turns to wind the spring to the max torque value is n:

$$= \frac{1}{3.1} = .262 \quad (3.8)$$

2) Calculating angular deflection Θ :

$$\theta = .262(360^\circ) = 94.32^\circ \quad (3.9)$$

the angular deflection is good just what it needs to be.

3) Calculating reduction in diameter D_i^* from spring being wound up:

$$D_i = .593 - 2(0.072) = .449 \quad (3.9.1)$$

$$D_i = .427 \quad (3.9.2)$$

My inner spring pin is .433 so my spring is safe to be wound up.

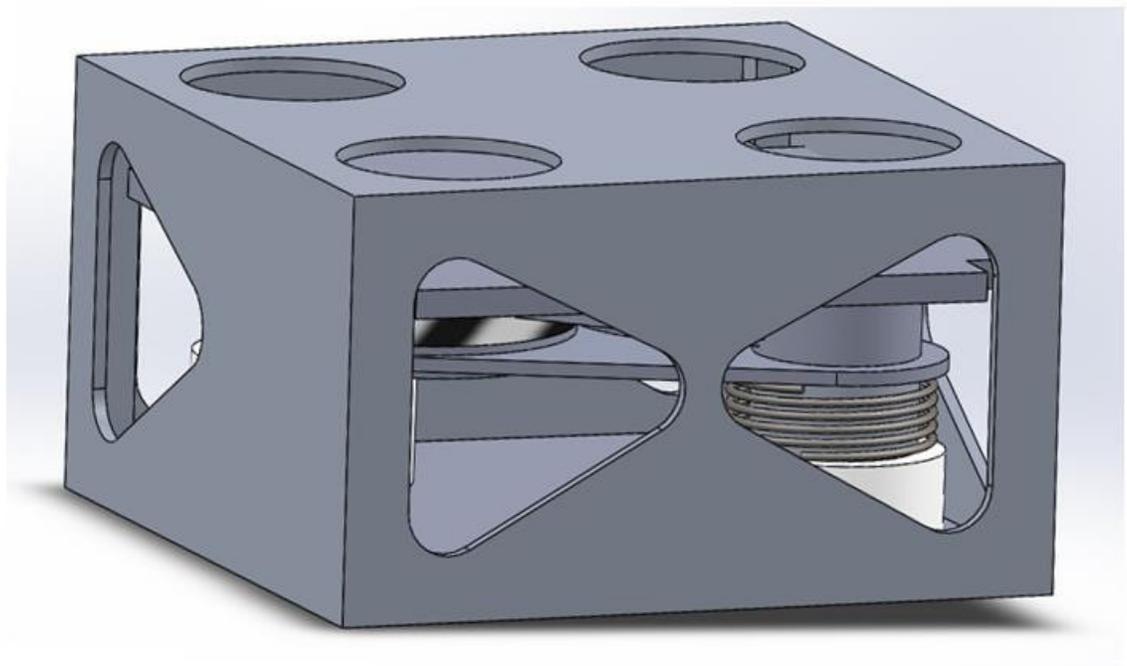


Figure 20: Closed Solar Array

Dimensions of closed box are: 3.94" X 3.94" X 1.88" or
 10cm X 10cm X 4.8 cm

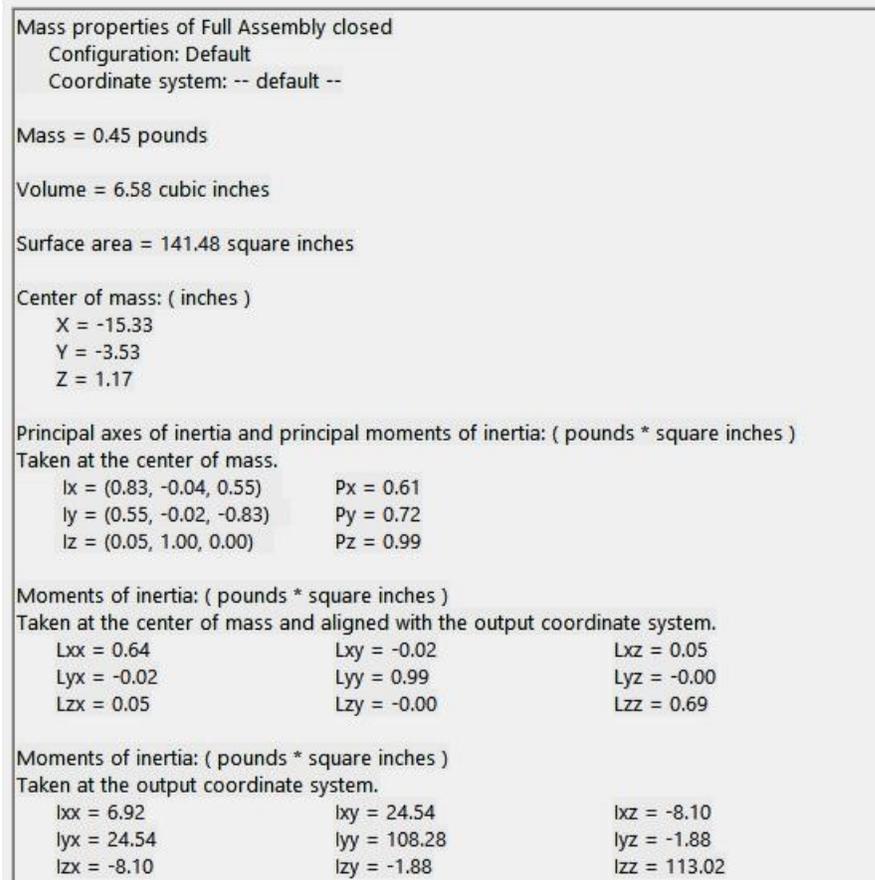


Figure 21: Mass Properties of Closed Assembly

Weight table of Assembly figure 13

Base plate	23g
2 X titanium allen bolts	7g
2 X solar panels	54g
2X torque springs 1,reverse wound	4g

Upper spring plate	23g
Compression spring	7g
Restraining cover	86
Total Mass	204g

Figure 22: Weight Table, mass parts breakdown

After performing a general mass properties calculation the array assembly is compared to existing vertically deployable panels from Clydespace weighing 100grams total for two 1U panels. Comparison yields a 104% increase in weight adding 104g of structure. Referring to the weight table fig. 25 we see the greatest increase is from the restraining cover. So effort should be in the direction of reducing its weight. An alternative for deployment may be a clamshell restraining cover design, allowing the spring tension of the closed solar panels to eject the clam shell thus deleting the compression spring and upper surface of the current restraining cover.

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APPENDICIES

Appendix A: CubeSat Collegiate Design Specification

The CubeSat: The Picosatellite Standard for Research and Education

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ABSTRACT

The development of the CubeSat standard, a picosatellite standard, has become a tool that encourages engineering collaboration, trains students with real-world satellite experience, and provides technology advancement in the aerospace industry. The Poly-Picosatellite Orbital Deployer (P-POD), in conjunction with the CubeSat standard, plays a key role in providing access to space for CubeSats. Developing satellites at the CubeSat level highlight the increasing opportunities for access to space while yielding quicker development times. The upcoming launches demonstrate the growing interest of universities, companies, and government organizations to develop CubeSats to perform valuable scientific experiments and missions. The educational benefits of CubeSat development are emphasized by providing an ideal training ground for future scientists and engineers.

I. Background

The P-POD was developed with seven primary goals¹:

- Protect the primary payload
- Protect the launch vehicle
- Protect the CubeSats
- Safely group multiple CubeSats for launch
- Eject CubeSats for safe deployment
- Increase access to space for CubeSats
- Provide a standard interface to launch vehicles

The design of the P-POD is relatively simple consisting of an aluminum box of tubular design with spring assisted ejection. A non-explosive release mechanism controls the deployment of the CubeSats to minimize shock to the launch vehicle and satellites. The P-POD has the capability to accommodate any configuration of 3 single CubeSats. The ability of the P-POD to hold multiple satellites and combine them as one single payload decreases launch costs.

A. The P-POD

1. Design of the P-POD

The Poly-Picosatellite Orbital Deployer (P-POD) started as a collaboration between California Polytechnic State University, San Luis Obispo (Cal Poly) and the Space Systems Development Laboratory (SSDL) at Stanford University in 1999. The need for consistency in the design and launching of picosatellite class CubeSat systems drove the design and development of the P-POD.



Figure 1: P-POD Mk. II

1

American Institute of Aeronautics and Astronautics

2. Revisions of the P-POD

The flexibility of the P-POD has allowed for design changes to better satisfy the requirements of the launch vehicle and the needs of CubeSat developers. Each revision to the P-POD accounts for backwards compatibility of the CubeSat Design Specification, so revisions to the P-POD do not affect developers negatively. Also, revisions to the P-POD are made to ensure that the launch vehicle mechanical mounting interface remains the same for continued P-POD to launch vehicle compatibility.

The original version of the P-POD, the Mk. I, met the basic goals of protecting the launch vehicle, the primary satellite, and the contained CubeSats. The Mk. I used a burn wire deployment system to open the door and release the satellites. Planetary Systems Corporation's Line Cutter Assembly burned through a Vectran line 30 ± 5 seconds after receiving the signal from the launch vehicle. The Mk. I contained all the power needed to burn through the Vectran line and did not require any resources from the launch vehicle, except for the standard launch vehicle deployment signal.



Figure 2: P-POD Mk. III

B. The CubeSat Standard

The CubeSat Standard states that a single CubeSat should be a 10-cm cube, and have a total mass of no

For the Mk. II design, feedback from launch providers led to the selection of a new release mechanism, as the actuation time of the line cutter mechanism lacked precision. Also, the Mk. II design incorporated a mounting bracket that could accommodate both the Qwknut and NEA release mechanisms. In addition, feedback from the CubeSat developers showed the need to have a switch on the P-POD that detected when the door was open at least 90 degrees from the closed position. This switch confirmed that the door sufficiently opened for nominal CubeSat deployment from the P-POD.

The current version of the P-POD is the Mk. III, shown in Fig. 2. Lessons learned from integrating CubeSats with the P-POD drove most of the modifications. The P-POD Mk. III offers larger access ports on two sides of the P-POD for increased access to CubeSats after integration, larger spring plungers for easier satellite integration, and door and bracket modifications to account for shear relief for the release mechanisms developed by Starsys and NEA Electronics. Also, instead of the deployment switch indicating the door opening at least 90 degrees, the switch for the Mk. III provides data to the launch vehicle that the door remains closed until the nominal P-POD deployment signal is sent.

more than 1 kg. The CubeSat specifications were derived from four basic sources¹:

- The size of available commercial off-the-shelf (COTS) components (i.e. solar cells, batteries, transceivers, etc)
- The P-POD's dimensions and features
- Launch vehicle environmental and operational requirements
- Self-imposed safety standards



Figure 3: CP1, Cal Poly's First CubeSat

The primary COTS components which drove dimensional requirements for the CubeSat Standard were solar cells and batteries. The current market

offers a number of solar cells about 30 mm x 70 mm in size. CubeSats should be able to body mount at least two solar cells per face to generate enough voltage to support common microcontrollers (3 to 5 volts). A wide variety of cylindrical and prismatic cell batteries of various chemistries are available in compatible sizes¹.

The following are specifications of the CubeSat Standard as a result of the P-POD interface¹:

- The center of mass of a CubeSat must be within 2 cm of its geometric center to minimize tumble and spin rates during deployment from the P-POD
- The location of the access ports on the P-POD determines where CubeSats should have diagnostic ports and remove before flight (RBF) pins
- Rails on CubeSats must be smooth, flat, and hard anodized to prevent cold welding from the launch environment and minimize friction during deployment
- Thermal expansion of the CubeSats should be similar to that of the P-POD material, Aluminum 7075-T73
- CubeSat design tolerances are based on P-POD tolerances and specifications

The CubeSat Design Specification (CDS) imposes several safety features to minimize risk to other CubeSats. These include:

- A RBF pin is required to keep the CubeSats inactive during integration
- At least one deployment switch must physically disable the electronic systems of the CubeSat when inside the integrated P-POD
- Separation springs to allow adequate separation between CubeSats after deployment from the P-POD
- A specified time delay between deployment and activation of any antennas, booms, or transmitters to ensure safety of other CubeSats

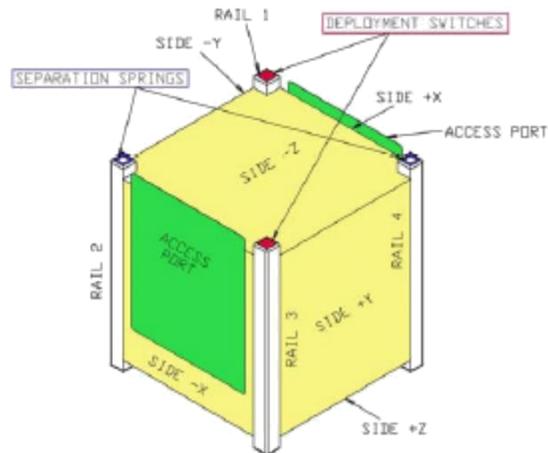


Figure 4: Schematic of the CubeSat Standard

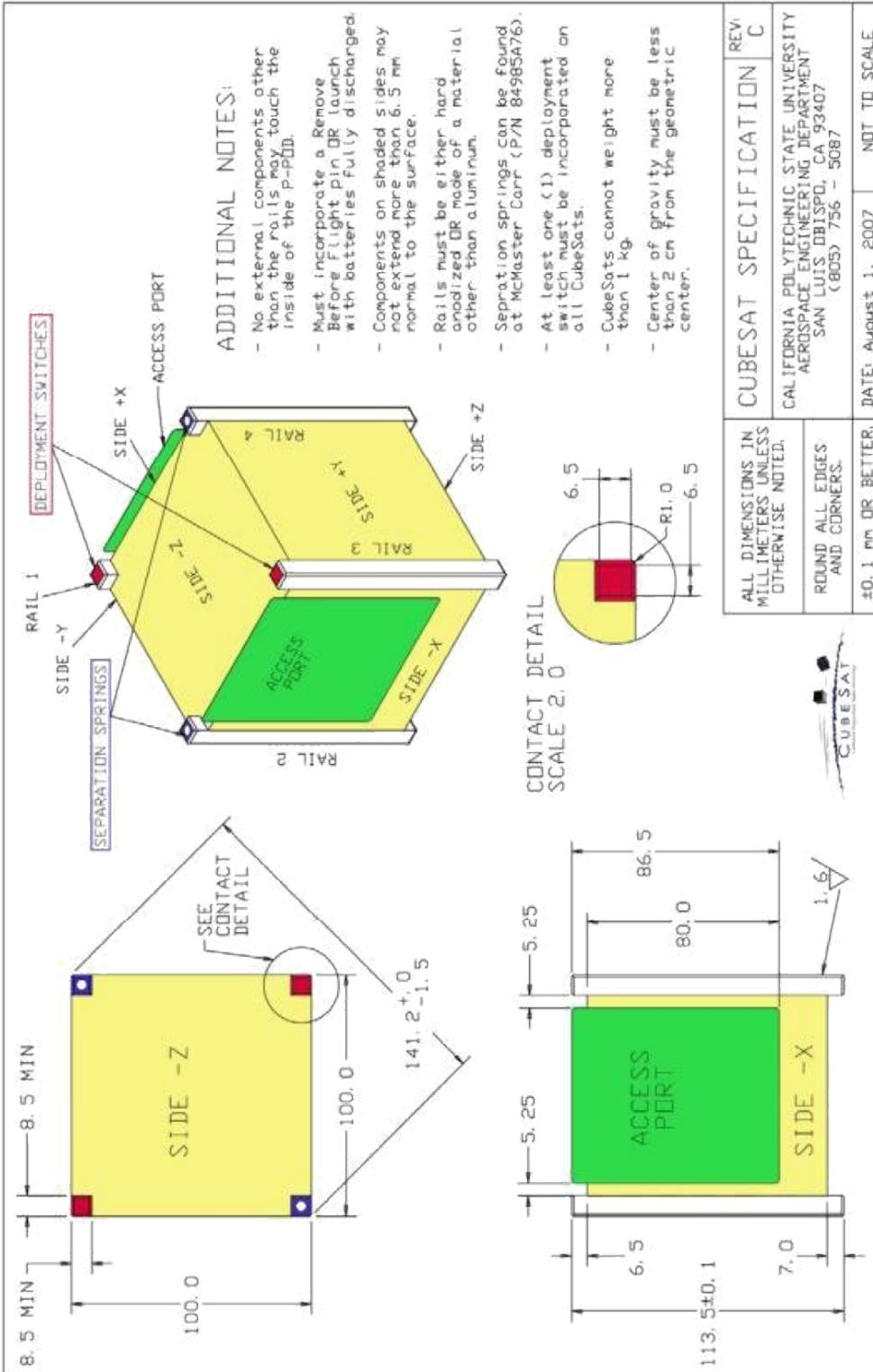
C. Cal Poly's Role

Cal Poly's current roles include:

- Maintaining the CubeSat Standard
- Developing, testing, and flying the P-POD
- Coordinating launches for CubeSats

Having an objective organization, such as Cal Poly, maintain the CubeSat Standard is vital for the successful enforcement of the standard and compliance of launch provider and Cal Poly requirements.

Traditionally, Cal Poly is responsible for coordinating with CubeSat developers and launch providers. This assures that the overall system will meet the launch provider requirements, as well as Cal Poly's requirements. In this case, as long as the developers abide by the CDS, they only need to communicate to Cal Poly, and Cal Poly will communicate with the launch provider. This greatly simplifies the communication path for the launch provider since they only need to be concerned with the mechanical and electrical interfaces of the P-POD to their launch vehicle. In addition to the technical side, Cal Poly also handles most of the export licensing process and ITAR compliance issues associated with each specific mission.



Appendix B: Pumpkin 3U Bus

http://www.cubesatkit.com/docs/datasheet/DS_MISC_300553-A.pdf



MISC 3™ 3U nanosatellite Bus

Hardware Revision: A

CubeSat-class Spacecraft Bus

Applications

- General-purpose 3U CubeSat missions for Earth-observation missions

Features

- 3U-size CubeSat
- Modular, customizable architecture
- >1300cc payload volume
- Multiple solar array configurations possible (e.g. "Propeller", "Turkey Tail", "Space Dart")
- Optional GPS
- Supports a minimum of 3 Separation Switches

Incorporated Subsystems

- Pumpkin CubeSat Kit™ Pro chassis
- Pumpkin 5th-generation PMDSAS fixed and/or deployable solar panels with up to 46 triple-junction solar cells (1W BOL each)
- Pumpkin fixed side panels with integrated Pumpkin Panel Release Mechanisms (PRMs)
- MAI MAI-400 ADCS with dual Earth-Horizon Sensors (EHS) for attitude knowledge and control
- Pumpkin Solar Interface Module (SIM)
- Pumpkin ADCS Interface Module, with:
 - AstroDev™ Lithium-2™ UHF transceiver
 - AstroDev™ UHF splitter/phaser
- Pumpkin Battery Module 1 (BM 1), with:
 - 40Wh energy storage
 - 2S2P cell configuration
- Clyde Space XUEPS 6-channel EPS, with:
 - Unregulated VBATT output
 - Regulated +5V_SYS and VCC_SYS outputs
- Pumpkin Motherboard (MB), with choice of Pumpkin Pluggable Processor Module (PPM)
- Pumpkin UHF deployable RHCP turnstyle antenna system

Also Includes

- Test & validation software



Propeller configuration shown

ORDERING INFORMATION

Pumpkin P/N 715-00553

Option Code	Configuration
/00 (standard)	standard
per factory	consult factory

Contact factory for availability of optional configurations.
Option code /00 shown.



CAUTION

Electrostatic Sensitive Devices

Handle with Care



User Customization

- End-users can customize this configuration in a variety of ways, e.g., alternate solar panel(s) configuration, alternate transceivers, alternate antennas, etc. Please consult factory for further information.

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Appendix D: Orbital Mechanics

Sun-Orbit Orientation

Frequently, thermal or power considerations of spacecraft require that the angle between the sun and the orbit plane be maintained within specified bounds for the

duration of the mission. This angle, conventionally known as the beta angle β , is illustrated in Fig. 11.10. Since β is the complement of the angle between the sun vector \hat{s} and the positive normal to the orbit \hat{n} , it follows from their scalar product that

$$\beta = \sin^{-1}(\hat{s} \cdot \hat{n}) \quad (11.9)$$

and ultimately that¹

$$\beta = \sin^{-1}[\cos \delta_S \sin i \sin(\Omega - \text{RATS}) + \sin \delta_S \cos i] \quad (11.10)$$

where β is defined to lie in the range from -90 to $+90$ deg.

Equation (11.10) reveals that beta angle depends on solar declination δ_S , orbit inclination i , and the difference in right ascensions of the true sun and the ascending node ($\Omega - \text{RATS}$). The first of these quantities, δ_S , depends on the date during the mission. The second quantity, i , is essentially constant during the mission. The last quantity ($\Omega - \text{RATS}$) changes because of nodal regression (induced by Earth's oblateness perturbations, as described in Chapter 8) and seasonal variation in the right ascension of the true sun.

In light of the variability of the terms on the right side of Eq. (11.10), it is clear that the beta angle cannot be held constant throughout a mission. However, it is generally possible to select conditions at the start of the mission so that the beta angle will stay within some prescribed tolerable range of values for that portion of the mission during which β is essential to performance.

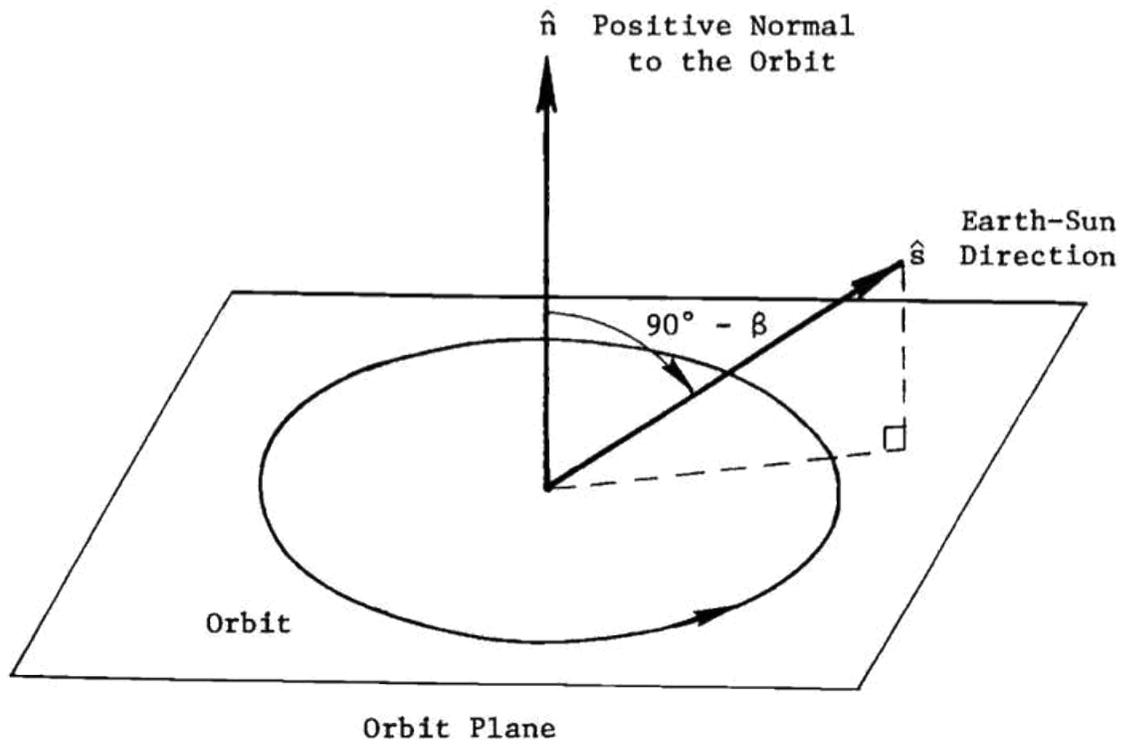


Figure 11.10

Earth Eclipsing of a Circular Orbit

It may be important to determine those occasions during its mission when a satellite is eclipsed by the Earth. Such eclipsing occurs whenever the satellite passes through the Earth's shadow, which is assumed cylindrical in this discussion.* Figure 11.11 shows that the Earth's shadow intersects the orbital sphere of a satellite at altitude h in a minor circle whose Earth-central-angular radius is β^* , where¹

$$\beta^* = \sin^{-1}[R/(R + h)], \quad 0 \text{ deg} \leq \beta^* \leq 90 \text{ deg} \quad (11.11)$$

View $A-A$ in Fig. 11.11 reveals that the orbit intersects the perimeter of the shadow circle at points E_1 and E_2 . Note that the length of the eclipsed orbital arc E_1E_2 is just twice arc CE_1 , where C is the point on the orbit of closest approach to the shadow axis A . That is, the length of arc AC is just the magnitude of β . Hence, it follows from the right spherical triangle ACE_1 that

$$\Delta u = \cos^{-1}(\cos \beta^*/\cos \beta) \quad (11.12)$$

When Eqs. (11.11) and (11.12) are combined, the eclipsed fraction of the circular orbit is found to be¹

$$f_E = \frac{2\Delta u}{2\pi} = \frac{1}{\pi} \cos^{-1} \left[\frac{\sqrt{h^2 + 2Rh}}{(R + h) \cos \beta} \right] \quad (11.13)$$

*Such an assumption is valid at low satellite altitudes, where there is no appreciable difference between the umbral and penumbral regions of total and partial eclipsing, respectively.

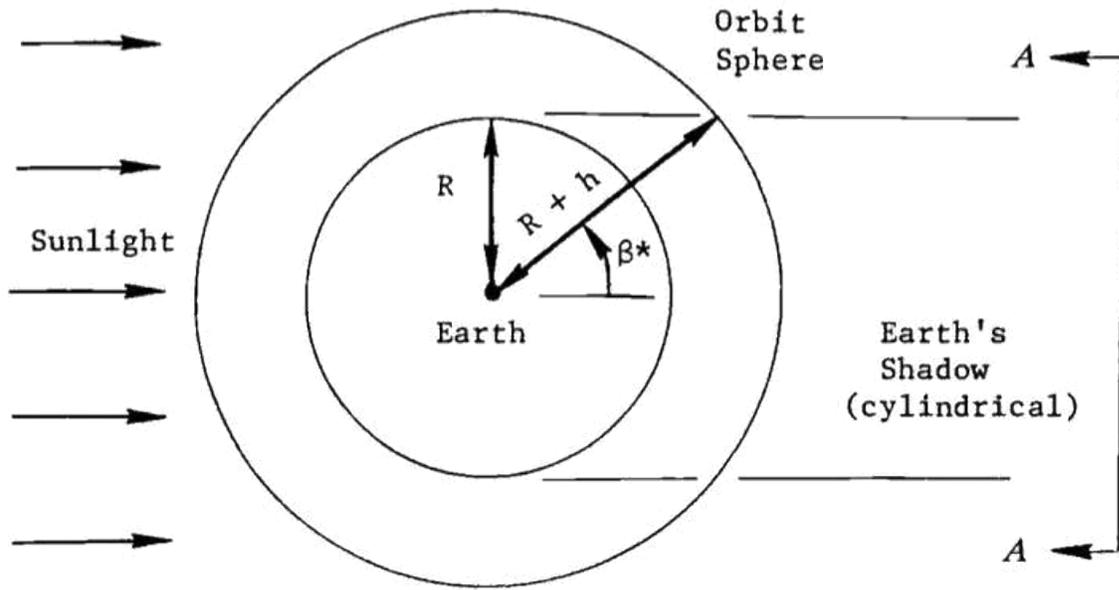
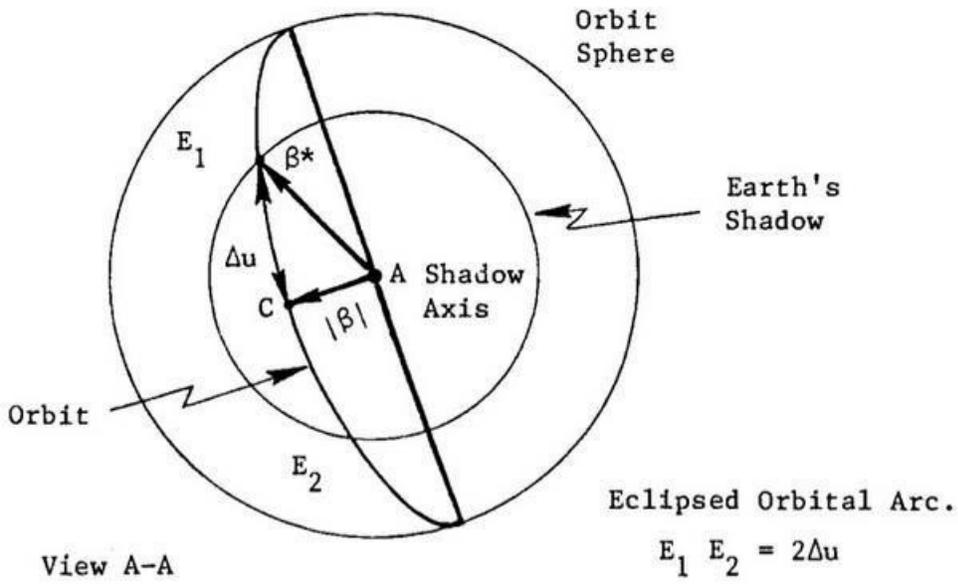


Figure xx Earth eclipse cylindrical shadow



FigureXX

Eclipse orbit fraction calculated:

Equation (11.10) reveals that beta angle depends on solar declination δ_S , orbit inclination i , and the difference in right ascensions of the true sun and the ascending node ($\Omega - \text{RATS}$). The first of these quantities, δ_S , depends on the date during the mission. The second quantity, i , is essentially constant during the mission. The last quantity ($\Omega - \text{RATS}$) changes because of nodal regression (induced by Earth's oblateness perturbations, as described in Chapter 8) and seasonal variation