Phase A

Structure and Configuration

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FOREWORD

This semester project consists of a written thesis reporting the background, design processes and outcomes of a project conducted at the EPFL under the supervision of Mrs. Larissa Sorensen. It began March 16 2006 and will finish June 26 2006 with the Phase A Review. The students engaged in this project are: Diana Arce, 3rd year bachelor in Materials, Benjamin Jutzeler, 3rd year bachelor in Materials, and Guillaume Röthlisberger, 1st year master in Microengineering. For 3rd year materials the semester project is around 13% of the time semester and for the 1st year master is 40% of the time.

The report documents the investigation into the design and analysis of developing the structural subsystem of a picosatellite capable of carrying a scientific payload into orbit. The design of the satellite is constrained by the specifications defined by the CubeSat Standards.

This report is divided into 8 chapters. The first chapter introduces the reader to the CubeSat program. The project objectives are stated in chapter 2, and specific requirements for this type of satellite are given in chapter 3. Chapter 4 is dedicated to the diverse approaches and assumptions have that were used. The material, fastening, structural and configuration trades are found in chapter 5. In chapter 6 the baseline design is described and provides a detailed design of the SwissCube structural subsystem, while chapter 7 contains the baseline properties like its static behavior and physical properties. Finally, chapter 8 outlines the current progress of the project and details the areas of proposed future development. Acknowledgements, references and appendices are located at the end.

We hope this report will be a small, but useful, contribution to the development of space activities at the EPFL, by bringing the first Swiss-built satellite one step closer to realization.

TERMS, DEFINITIONS AND ABBREVIATED TERMS

ADCS	Attitude Determination and Control System
CalPoly	California Polytechnic Institute
CDMS	Command and Data Monitoring System
CDR	Critical Design Review
C.o.M.	Centre of Mass
COTS	Commercial off The Shelf
CSDS	CubeSat Design Specification Document [CalPoly]
CTE	Coefficient of thermal Expansion
CVCM	Collected Volatile Condensable Material
ECSS	European Cooperation for Space Standardization
EPS	Electrical Power System
FEA	Finite Element Analysis
FOS	Factor of Safety
GaAs	Gallium Arsenide
IC	Integrated Circuit
ICD	Interface Control Document
IRD	Interface Requirements Document
LV	Launch Vehicle
MDD	Mission Description Document
MOS	Margin of Safety
OBC	On-Board Computer
P-POD	Poly Picosatellite Orbital Deployer
PCB	Printed Circuit Board
PDR	Preliminary Design Review
RF	Radio Frequency
RML	Recovered Mass Loss
SSO	Sun-Synchronous Orbit
TBC	To be confirmed
TBD	To be defined
TML	Total Mass Loss

1 INTRODUCTION

The purpose of this report is to present the development of the baseline design for the structural subsystem of the picosatellite SwissCube. The SwissCube is the first entirely Swiss picosatellite program. The SwissCube project is based on the CubeSat program started by Stanford University and California Polytechnic State University (CalPoly).

1.1 CubeSat

The CubeSat project is a joint venture between California Polytechnic State University San Luis Obispo and Stanford University's Space Systems Development Laboratory. Started in 1999 it is the purpose of the CubeSat project to provide a conventional standard for the design and development of picosatellites such that a common deployer can be used [1]. The project attempts to reduce the cost and development time generally associated with satellite design, consequently increasing the accessibility to space for educational purposes. Currently there are more than 80 institutions around the world taking part or took part in the development of CubeSats.

The fundamental defining feature of the CubeSat standard is its dimensions. The standard specifies that the satellite must have the geometry of $10cm^3$ cube with a mass of no more than 1kg and that the center of gravity must be within 2cm of the geometrical center. The standard also specifies several other important guidelines that must be followed, which will be dealt with as the design progresses. The standards are outlined in the CubeSat Specification Document [2]. It is the purpose of the specification document to ensure that each satellite developed will integrate properly with the deployer and will not interfere with other satellites, payloads or the launch vehicle. Figure 1 is an example of a CubeSat design. It has been included to give an understanding of the basic external geometry of a typical CubeSat.

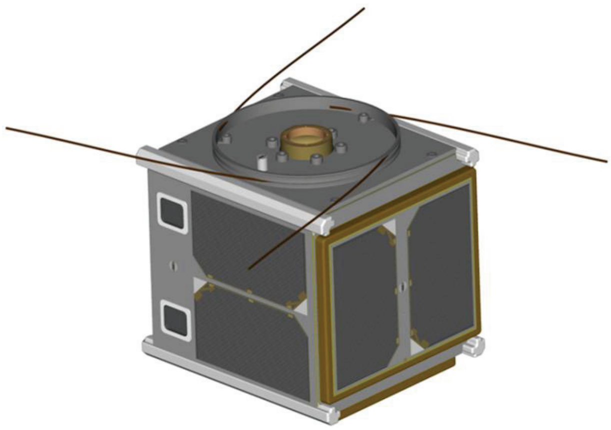

Figure 1 Example of a CubeSat (Aalborg University)

1.2 CubeSat Deployer

Figure 2 Model of the P-POD (CalPoly).

A unique feature of the CubeSat Program is the use of a standard deployment system [3]. Through the Poly Picosatellite Orbital Deployer, or P-POD (see Figure 2), standardization provides the interface between the launch vehicles and the CubeSats, thereby reducing mission costs and accelerating development time. The deployment solution was developed at California Polytechnic University (CalPoly). The P-POD has been fully qualified according to NASA worst case levels in both vibration and thermal vacuum environments.

The current P-POD is capable of containing and subsequently deploying three single CubeSats measuring 10 cm^3 and weighing 1 kg. The P-POD's design is extremely simple, and purposefully so. It is an Al 7075 T-73 box with a spring, a door, and a mechanism to open that door. CubeSats are stacked inside the P-POD and constrained by a set of hard anodized, Teflon-impregnated rails. These rails provide a low-friction surface for the CubeSats to slide against during deployment.

By providing the developers the option of building a double or triple CubeSat, the current design can accommodate three single CubeSats, a double plus a single CubeSat, or one triple CubeSat. These satellites are respectively double and triple the length and weight of a single CubeSat.

1.3 Subsystems Description

The functional requirements of the subsystems refer to the physical constraints that each of the individual subsystems impose on the design of the structure of the spacecraft. However, as the SwissCube structural configuration is limited by the CubeSat specification, it is the structural design that imposes physical constraints on the subsystems. Parameters such as volume and weight of each subsystem are required at this stage to ensure that the SwissCube meets with the CubeSat specifications. One way of remaining within the maximum mass specification of one kilogram is to produce a mass budget and assign each subsystem a maximum allocation of weight and volume. The interactions of the various subsystems with the structure are described briefly below.

The structural subsystem is unique as it has interactions with all the others subsystems. All of the subsystems will have hardware and electronics mounted internally and/or externally and the structure must provide a "safe" environment for their operation. In order to construct a structural subsystem which is satisfying, the other subsystems must then be physically defined [4; 5]. The internal configuration must both depend on the dimensions and the weight of the different subsystems and consequently has to impose restrictions on these same subsystems.

1.3.1 Power

Power is supplied by two means: solar cells and two batteries. This subsystem requires the mounting of solar cells on the external structure. The structural subsystem must provide a safe environment for the mounting and operation of these solar cells, i.e. ensure that faceplate vibrations (accelerations and displacements) are tolerable for the operation of the solar cells and that the temperature remains in a certain range. Concerning the batteries, it needs to be maintained warm in order to keep sufficient efficiency.

The batteries ensure the power supply during the part of the orbit which is during the night, when the solar cells are unusable. During the part of the orbit in sunlight the solar cells will provide enough energy to supply the system and charge the batteries.

1.3.2 Payload

The payload will probably be the most important subsystem in terms of volume, depending on the question whether a telescope (which seems more probable) or a camera is chosen. At the moment both options remain. The preliminary dimensions of the payload are 30mm of diameter and 65mm on length (TBC). The payload will be contained with its sensor in a specific structure, the mounting frame, and then fixed to the frame of the satellite. The mounting frame slightly increases the occupied volume of the payload subsystem.

1.3.3 Thermal

This subsystem shall be in charge of monitoring the internal component's temperatures. The temperature in space ranges from 120°C in direct sunlight to –100°C in Earth's shadow (TBD). Unfortunately, the satellite's systems will not operate within this temperature range, so its thermal environment must be controlled.

Thermal control is provided primarily through passive measures. Thermal coatings and tapes are used on the external spacecraft surfaces to keep the orbit average satellite temperature in an acceptable range. Thermal control of individual components is achieved using a variety of techniques including thermal isolation and heat sinks.

First simulations conduced by the thermal team say that the external temperature of the satellite will be between -23°C to 10°C

1.3.4 Attitude Control and Determination

The ACD subsystem will require external mounting surfaces for the location of magnetic torquers. Three magnetic torquers, one on each axis, and a inertial wheel will be required to provide control of roll pitch and yaw. Spatial orientation will be measured using three magnetometers and the solar cells as sun sensors.

1.3.5 Control and Data Management

The control and data management (CDMS) subsystem has two main functions. The first is to supply computing services aboard the satellite. The second is to provide communication between the satellite and the ground station for the purposes of command and control, obtaining spacecraft health and systems status as well as sensor data transfer.

1.3.6 Telecom

Two antennas will provide uplink and downlink communications with an earth-based ground station. One of the antennas will be a dipole and the other a monopole. They will be fixed on one external face of the satellite. The satellite's antennas and their release mechanism are a critical parts of the communication system, since without the antennas, communication between the satellite and the ground station would not be possible. There are several key components that compose the telecom system: antenna element, power divider/combiner, and RF circuit board.

2 PROJECT OBJECTIVES

2.1 SwissCube

SwissCube is the picosatellite being designed by students and staff at the Swiss Federal Institute of Technology Lausanne (EPFL) to be developed and launched in line with the CubeSat specifications. The primary objective of developing this satellite is to provide a dynamic and realistic learning environment for undergraduates, graduates and staff in the development of small satellite technology [4]. As a secondary objective it is hoped that the picosatellite will be able to house a science payload with the aim to take optical measurements and characterize the Nightglow phenomena (see Figure 3) over all latitudes and longitudes for at least a period of 3 months, with an extended science mission of duration up to 1 year (TBC) [6] .

Figure 3 The Nightglow phenomena [4].

In the design of the SwissCube, each of the subsystems like ADCS, EPS, etc., is being treated as an individual component and managed by a specific group of the SwissCube Team. However, although each subsystem is being designed independently it is important to remember that each component is only one part of the complete satellite. Therefore to maintain a high level of integration between the various subsystems continuous communication and discussion is maintained between the designers of the individual subsystems. This report focuses on the structural design and configuration of the picosatellite, but may at times make references to other aspects of the satellite that we deem important.

2.2 Structure and Configuration

The purpose of the structural subsystem for the SwissCube is to provide a simple sturdy structure that will survive launch loads and a suitable environment for the operation of all subsystems throughout all phases of the mission life, while providing an easily accessible data and power bus for debugging and assembly of components. Moreover the structural subsystem shall carry, support, and mechanically align the spacecraft equipment. It shall also cage and protect folded components during boost.

Structural design shall aim for simple load paths, a maximization in the use of conventional materials, simplified interfaces and easy integration. Due to the size of the satellite and small expense budget, this was done with the philosophy of maximizing usable interior space, while minimizing the complexity and cost of the design. Due to the weight constraints, the structure must be the lightest possible to allow more margins for the other subsystems.

3 DESIGN REQUIREMENTS

The following chapter gives a complete list of the preliminary requirements for the structural subsystem established during phase A of the SwissCube project. Requirements are grouped in five different categories: physical requirements launch environment, space environment, testing requirements and design requirements.

The complete list of constraints imposed on the SwissCube by the CubeSat standard is given by the CubeSat Design Specifications document [2] (CDS) in Appendix A - SwissCube Requirements and Specifications (Further information can be found at http://cubesat.calpoly.edu). Where no other reference is indicated, requirements are taken from the CDS. Any deviations from CDS must be discussed with Cal Poly/Stanford launch personnel before the final CubeSat design is approved for launch. A final check of specifications will be conducted prior to launch.

Besides the requirements imposed by the CubeSat Design Specifications several additional requirements for the structural subsystem were established. Data (i.e.: temperature range) are preliminarily based on values estimated by other CubeSat missions and will be renewed as soon as additional information is provided by the other subsystems.

3.1 Physical requirements

- Each single CubeSat may not exceed 1kg mass.
- Center of mass must be within 2 cm of its geometric center.
- The use of Aluminum 7075 or 6061-T6 is suggested for the main structure. If other materials are used, the thermal expansion must be similar to that of Aluminum 7075-T73 (P-POD material) and approved by Cal Poly launch personnel.

Strength

The structure shall be of adequate strength to withstand the design loads without yielding, failing or exhibiting excessive deformations that can endanger the mission objectives, i.e. brooking the fragile solar cells.

Buckling

- The stability (no buckling) of the structure shall be verified for the design loads.
- Local buckling shall only be tolerated if it is reversible and on the condition that the resulting stiffness and deformations remain in conformance with the structural requirements without risk of general buckling being induced by local instability.

For composite materials micro-buckling of fibers shall not be accepted.

Stiffness

The structure shall be designed to meet the requirements for stiffness under the specified load and boundary conditions. Stiffness is often expressed in terms of a minimum natural frequency requirement and is therefore related to the overall mass.

The stiffness of sub-assemblies and components shall be such that the structural and functional performance requirements are met, avoiding excessive deformations, leading to violations of specified envelopes, gapping at joints or the creation of inefficient load paths.

Dynamic behavior

The resonant frequencies of the structure shall be restricted to specified bandwidths which have been chosen to prevent dynamic coupling with major excitation frequencies (e.g. launch vehicle fundamental frequencies).

Structural / vibrational requirements

- Preliminary mass allocation for the structure (including primary structure, secondary structure, deployment switches and separation springs) is 202g.
- The structure has to resist typical maximal launch accelerations of 10g (see §7.2).
- Vibration testing has to be performed considering the vibration spectra of the launch vessel (DNEPR, VEGA, SOYOUZ), as specified in the Cal Poly Safety Compliance Requirements. The Cal Poly Test Pod has to be used for vibration testing [9; 10].
- Structural rigidity has to assure that the values of the fundamental mode frequencies are between 20 Hz and 45 Hz for the longitudinal axis and superior to 15Hz for the lateral axis [11].

3.2 Launch environment

CubeSat behavior in the P-POD

- CubeSats must not present any danger to neighboring CubeSats in the P-POD, the LV or primary payloads:
- All parts must remain attached to the CubeSats during launch, ejection and operation. No additional space debris may be created.
- CubeSats must be designed to minimize jamming in the P-POD.

RF cross-interference

- No electronics may be active during launch to prevent any electrical or RF interference with the launch vehicle and primary payloads. CubeSats with rechargeable batteries must be fully deactivated during launch or launch with discharged batteries.

Cross-contamination

- NASA approved materials should be used whenever possible to prevent contamination of other spacecraft during integration, testing and launch.

Deployables

- Deployables must be constrained by the CubeSat. The P-POD rails and walls are not to be used to constrain deployables.

Remove before flight pin

- A remove before flight (RBF) pin is required to deactivate the CubeSats during integration outside the P-POD. The pin will be removed once the CubeSats are placed inside the P-POD. RBF pins must fit within the designated data ports (Attachment 1). RBF pins should not protrude more than 6.5 mm from the rails when fully inserted.

Deployment switch

- One deployment switch is required (two are recommended) for each CubeSat. The deployment switch should be located at designated points.

launcher rail interface

- Rails must be smooth and edges must be rounded to a minimum radius of 1 mm.
- At least 75% (85.125 mm of a possible 113.5mm) of the rail must be in contact with the P-POD rails. 25% of the rails may be recessed and no part of the rails may exceed the specification.
- All rails must be hard anodized to prevent cold-welding, reduce wear, and provide electrical isolation between the CubeSats and the P-POD.

Separation springs

- Separation springs must be included at designated contact points in Figure 56. Spring plungers are recommended (McMaster-Carr P/N: 84985A76 available at http://www.mcmaster.com). A custom separation system may be used, but must be approved by Cal Poly launch personnel.

3.3 Testing requirements

- Random vibration testing at a level higher than the published launch vehicle envelope outlined in the MTP.
- Thermal vacuum bakeout to ensure proper outgassing of components. The test cycle and duration will be outlined in the MTP.
- Visual inspection of the CubeSat and measurement of critical areas as per the CubeSat Acceptance Checklist (CAC) [2]

3.4 Space environment

Due to the harsh environment of space, the satellite must be designed to withstand certain conditions not experienced on the ground. It must handle radiation, debris, extremes in temperature and outgassing. The principal requirements regarding the space environment are as follows:

Thermal requirements

- Operating temperature range is expected to be -40°C to 70°C [12].
- Temperature range for non-controlled environment: -150°C to 150°C (no active thermal control) [4].
- Thermal vacuum testing has to be performed at a minimum vacuum level of 5 x 10-4 Torr and at a temperature of 70°C [9].

3.5 Design requirements

Accessibility

- The lay-out and design of the subsystem hardware shall provide sufficient accessibility to allow for easy integration, removal, inspection and if required maintenance of subsystem items during the course of the on-ground project activities.

Maintainability

- The overall design shall require a minimum of special tools and test equipment to perform assembly, integration, repair and maintenance activities
- The design shall minimize the maintenance required during storage and ground life.

In addition to the established accessibility and maintainability requirements, several SwissCube specific specifications will equally impose limitations on the structural design. These include:

- Keeping the cost of the structural subsystem as minimal as possible
- Providing a light and simple structural subsystem.
- Using COTS materials and components where possible. This will reduce both cost and design time.
- Realizing a modular design that can be fabricated easily.

4 Design Assumptions and Approaches

The following chapter outlines the approach followed during the design of the SwissCube's structural subsystem, as well as the assumptions made about the launch environment and the different other subsystems.

4.1 Design approach

The "Aerospace Design Engineers Guide" of the AIAA [13] provides a comprehensive discussion of the stages involved in the structural design of a spacecraft. These stages are described schematically in Figure 4. In this section we consider the preliminary structural design, including the development of a preliminary mass budget and an initial structural configuration while considering all engineering constraints.

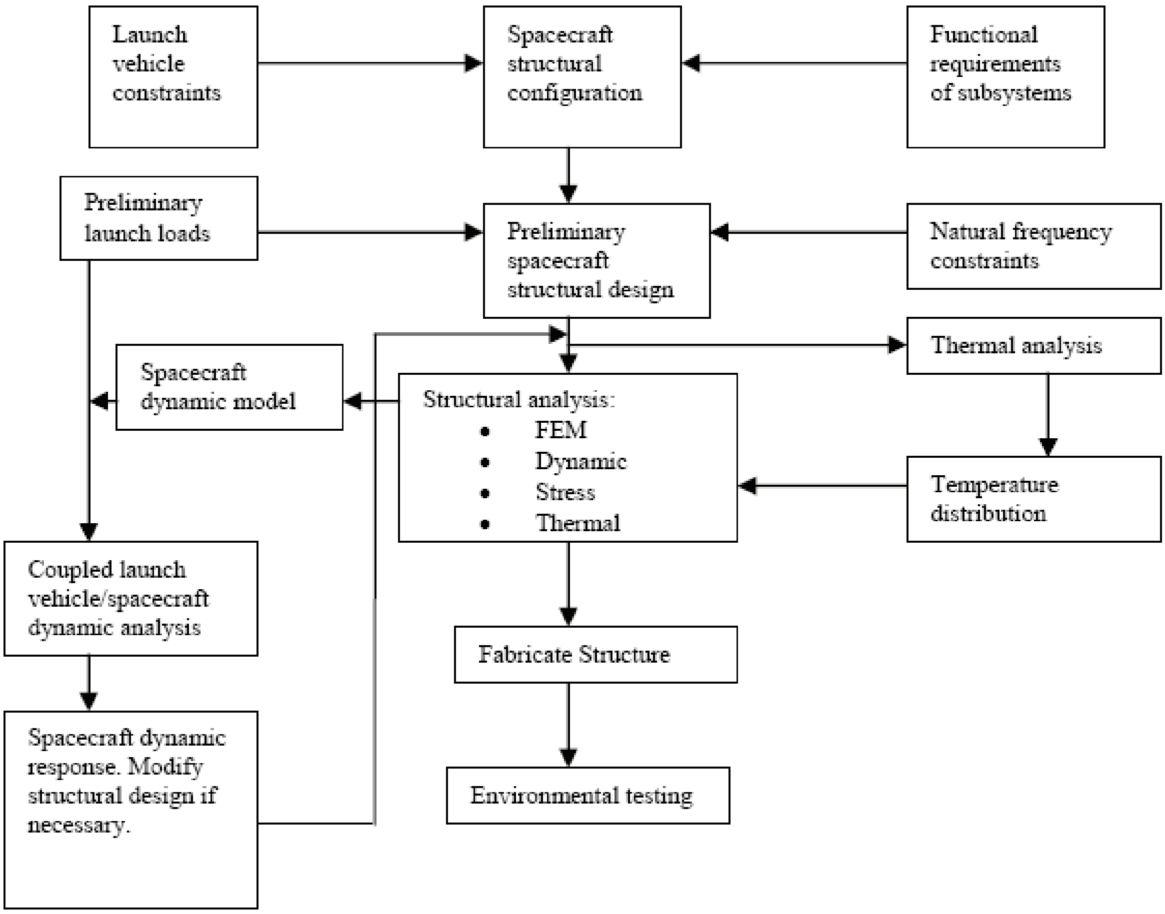

Figure 4 Spacecraft structural design procedure.

4.1.1 Launch vehicle constraints and launch loads

At this stage the type of launch vehicle is known (VEGA launcher) the specific loads imposed on the structure as a result of the launch can be found in the VEGA user's manual [11]. Information about DNEPR launch vehicle is equality given as a second option in case of a unforeseeable chance. This information is tabulated in Table 1 and Table 2.More detailed information can be found in Appendix A - SwissCube Requirements and Specifications.

Table 1 Accelerations induced on structure during launch.

Launch Vehicle	Axial Acceleration (g)	Lateral Acceleration (g)
Dnepr	7,5	0,8
Vega	5,5	0,9

Table 2 Low frequency vibration of our possible launch vehicles.

Vehicle	Frequency Range (Hz)	Acceleration (g)
Dnepr	2 - 5	0,2 - 0,5 lateral
	5 - 10	0,5 lateral
		0,5 longitudinal
	10 - 15	0,5 - 0,1 lateral
		0,6 longitudinal
	15 - 20	0,5 longitudinal
Vega	5 - 45	1,0 axial
(qualification levels)	45 - 100	1,25 axial
	5 - 25	1,0 lateral
	25 - 100	0,62 lateral

Mechanisms shall be designed to meet the mechanical performance requirements and to withstand the specified environment during launch without damage or degradation. Mechanisms shall conform to the specified stiffness, strength and safety requirements derived from the launcher and the spacecraft structural requirements. The factors of safety is a coefficient by which the design loads are multiplied in order to account for uncertainties in the statistical distribution of loads, uncertainties in structural analysis, manufacturing process, material properties and failure criteria

In the computation of safety margins the following minimum factors of safety shall be used for standard metallic materials:

- yield stress factor of safety 1,25
- ultimate stress factor of safety 1,5
- minimum fatigue factor (cycles) 4

4.1.2 Functional block diagram

The main functions to be fulfilled by the structural subsystem of the SwissCube are given by the functional block diagram in Figure 5:

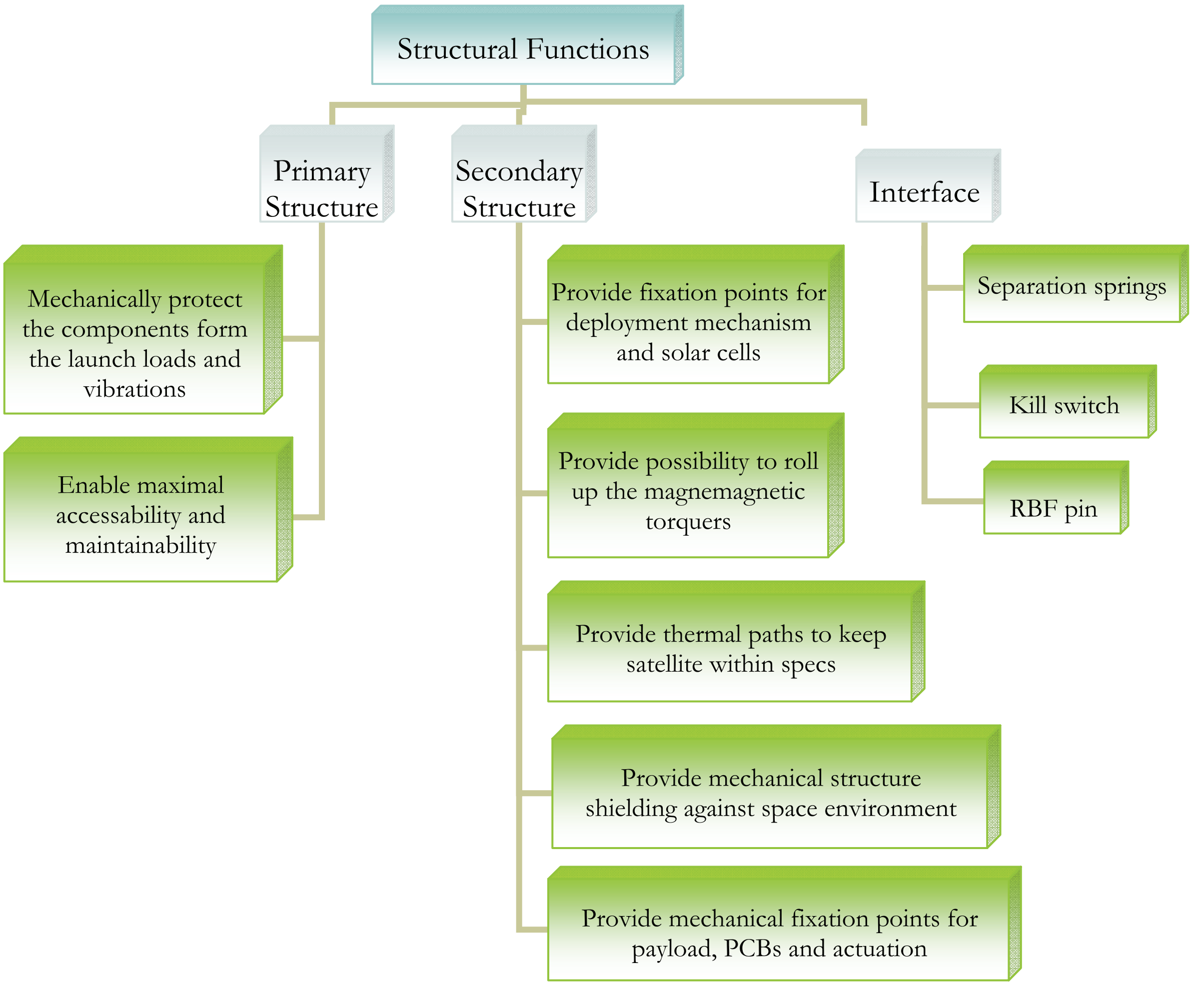

Figure 5 Functional block diagram

4.2 Design principles

The design of the structural subsystem progressed through a number of iterative stages. The first stage involved finding several initial concepts for the design of the structure. These initial configurations had to take into account the CubeSat requirements as well as all of the specific SwissCube requirements as discussed previously.

4.2.1 Principles used by other CubeSat developers

Because of the public nature of the CubeSat idea, many interesting documents on the structural configuration of CubeSats are available on the internet (see Internet website addresses in References). From these documents, other development team's ideas, and even detailed drawings are available.

Figure 6: Several designs from other CubeSat developers (StenSat, CalPoly, Tokyo Institute of Technology, Nihon and Design Massif).

Most developers rely on designs based on relatively massive side panels or rails to build a rigid structure. Most of the designs are assembled by the use of screws, while a few are riveted. Most designs feature single sheet side panels in thickness of up to 1.5 mm or rib-constructions. The commercially available design from OSSS is constructed of 7 frames that stack on top of each other to form the body of the satellite. Most of the available designs are rather simple, and only few have masses below 300 g (the allocated weight budget of the SwissCube structure is 202 g).

Generally the commercially available designs look like a shell structure, in which there are space and attachment possibilities for one or several PCBs. The designs developed by universities and other participating organizations are constructed specifically for their mission.

From the study of other available designs, the following ideas can be developed:

- A construction with stacked frames is very simple as most of the parts are identical. The drawbacks are high mass, due to the fact that the frames must be interconnected, and low maintainability, as the body is not easily disassembled after attachment of, for example. the solar cells.

- A box and lid construction (as the Tokyo Institute of Technology construction) combined with a motherboard type electronic backbone makes the maintenance of the electronic circuits easy, as they can be slid freely in and out of the structure. Here the drawback is the difficulty of assembly of the systems accommodated in the box, as they must be installed inside the small cavity, and are not easily disassembled again.

4.2.2 The principle of fewest possible parts

Every joint in a structure requires fasteners in the form of clamps, screws, adhesives, grooves and the like and is most likely making the construction both weaker and heavier. To avoid this problem, a design featuring as few joints as possible might be developed.

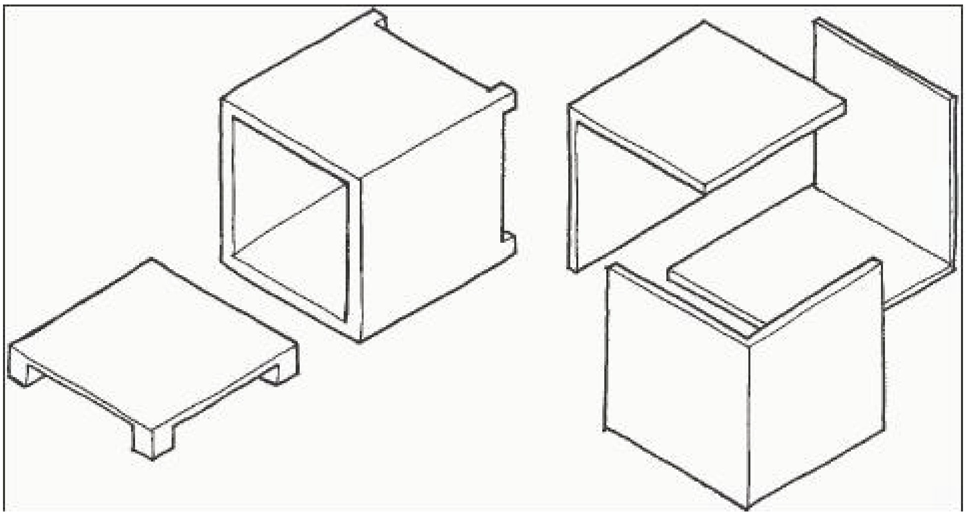

Figure 7 Examples of the principle of fewest possible parts.

Considering the design goal of multiple assemblies and disassemblies, it is clear that the solution must feature a lid or some other possibility of opening the shell structure to access the internal parts. One advantage of this principle is the improved tolerance stack-up that is gained by reducing the number of assembled parts. If all the internal frames and beams can be machined from a single piece of raw material, the resulting end tolerance will be small compared to a design made up of many parts. The other advantage is the optimal thermal conductivity if the body is designed of a single metal piece.

The disadvantages of the principle are the manufacturing problem and the low flexibility, making late design changes difficult. The machining of three-dimensional shapes inside the very small envelope of the satellite body is extremely difficult and must be conducted with adequate forethought and care. The low flexibility is also a major problem, as not all parts of the satellites other subsystems are yet fully defined, and changes of design therefore might be necessary.

4.2.3 The principle of maximum flexibility

If the satellite body is constructed using a large number of pieces (like an IKEA kit), each designed to accomplish a single task, the flexibility of the design is maximized. If one of the subsystems is exchanged, or the quantitative structure of the satellite must be changed for other reasons, all non-affected parts of the design can remain untouched. Another advantage is that the manufacturing complexity is much lower, mainly because of the simpler part geometry.

The main disadvantage is the poor thermal and electrical properties of the structure, and the assembly task that will require a detailed plan and a high level of precision. Furthermore a large number of fastening devices are needed and consequently the weight of the overall structure is heavier and this large number of joints or fasteners is also detrimental to vibrational robustness.

4.3 Design assumptions

4.3.1 Assumptions about mechanical loads

In the initial design phase, mechanical loads and vibration spectra imposed on the satellite are not clearly defined, since neither the orientation of P-POD in the launch vehicle, nor the position of the SwissCube in the P-POD are defined. In order to develop design trades of the structure, worst case assumptions of the launch load as well as of possible vibration frequency ranges were established based on the information given for the both types of possible launch vehicles that are VEGA and DNEPR. Generally, maximal estimated launch acceleration is 7.5g (DNEPR case) axially with largely inferior lateral launch accelerations. The worst case positioning of the satellite is in the lowest position of a vertically posed P-POD, since the SwissCube then has to support the entire load of the CubeSats above (see remarks in §7.2.1). In order to account for uncertainties in the statistical distribution of loads a factor of safety must be used. The yield stress factor of safety recommended by ECSS [17] is 1.25, so the resultant acceleration is 9.375 g and rounded to 10 g.

4.3.2 Assumptions on other subsystems

Several design assumptions had to be established in order to develop different design trade-offs for the Swiss Cube's primary structure and internal configuration. Given that the development of a design for the structural subsystem is a process dependent on the knowledge of physical parameters such as the mass and dimensions of each of the other subsystems, a design of a preliminary internal layout would be impossible without these assumptions. In order to obtain information on other subsystems mass and dimensions, information was retrieved by questionnaires and then used for the assignment of available volume to each subsystem and to evaluate the number of PCBs. From the different possibilities for the payload, preliminary design trades where established considering a cylindrical camera with a lens diameter of 25mm.

5 Design Trades

In this chapter, the various trades of our subsystem like material, fastening, structural and configuration are introduced. These trade-offs stem from the requirements defined previously and can be interpreted as degrees of freedom in the design of the global satellite.

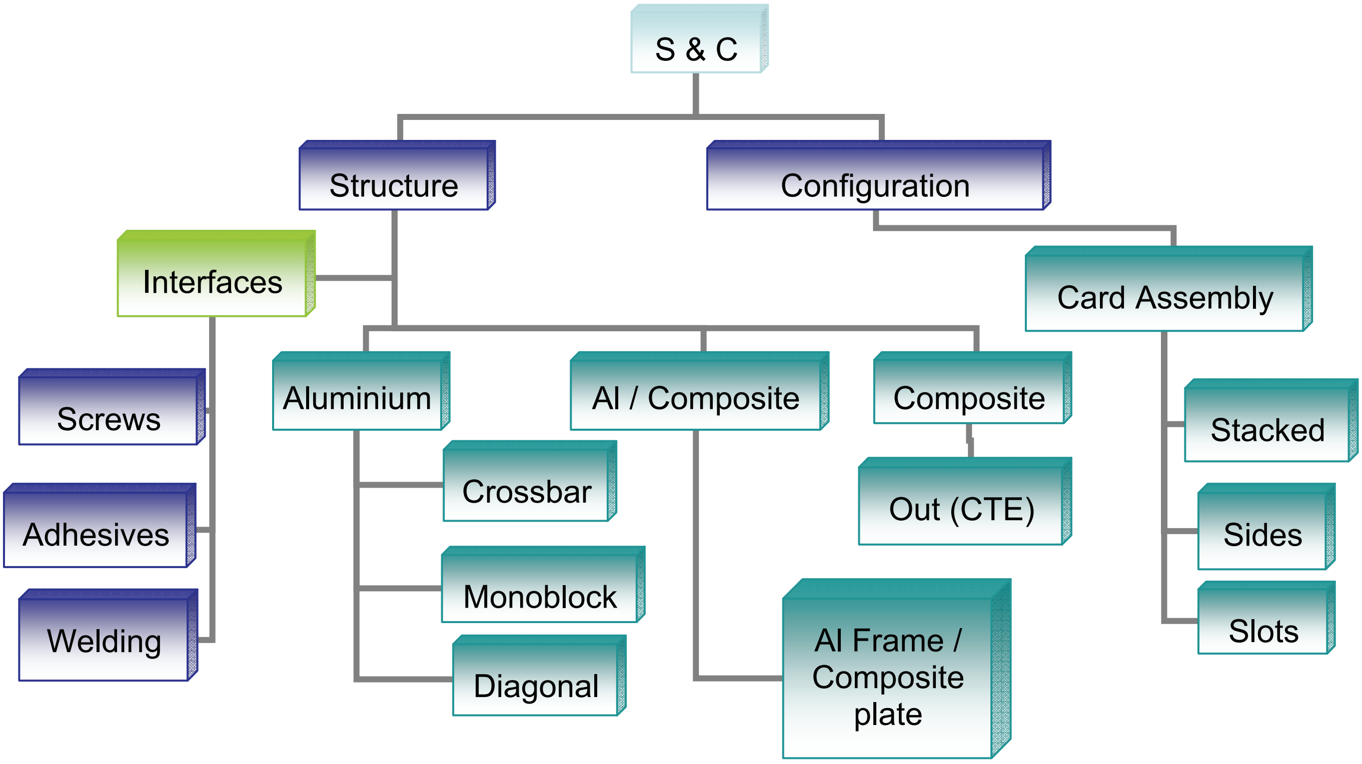

Figure 8 Trades-Tree

5.1 Material Trades

The selection of the materials to be used for the fabrication of the SwissCube structural subsystem is an important initial step in the design process. It will influence the fastening and the shape of the elements. When discussing which materials to use, it is important to know what kind of effects might influence the decision. Several materials were considered before selecting the final material. The criteria for selection were based on characteristics listed below:

- Strength
- Weight
- Coefficient of thermal expansion

- Machineability
- Cost

The Table 3 lists several materials along with their density, strength, coefficient of thermal expansion and machineability [8].

Table 3 Selected material properties data.

Material	Density [g/cm^3]	Yield Strength [Mpa]	CTE linear [µm/m-°C]	Machineability
Stainless Steel	7.76	790	-	Easy
Titanium	4.43	900	-	Hard
Al-6061-T6	2.85	320	23.6	Easy
Al-7075-T6	2.80	340	23.6	Easy
Al-7075-T73	2.81	435	23.6	Easy
Al-7022-T651	2.76	460	23.6	Easy
Reinforced composites	1.4 – 1.7	100 - 800	11 - 18	No applicable

The CubeSat standard specifies that the satellite must be constructed of a material with a similar thermal expansion coefficient to the materials used for the construction of the P-POD. The P-POD has been fabricated from the aluminum alloy Al–7075–T73. The specifications recommend the use of either Al-7075-T73 or Al-6061-T6 [2]. To reduce the weight of the structure, we consider making panels out of composite materials which have a density lower than aluminum, even though the value is variable from one composite to other. The density indicated in Table 3 is an average value of some carbon fibers reinforced composites. Early on in the design process, it was decided to make the rails in aluminum (probably Al-7075-T73) such as to comply with the constraining requirements. A problem with the rails during launch could end the mission, and possibly, depending on the place of the satellite in the P-POD, block other CubeSats. Consequently, at least for this first SwissCube, no other material will be discussed for the rails.

5.1.1 Aluminum

Aluminum alloys are some of the basic building materials of existing spacecraft and appear in many subsystems. They are used for favorably in primary and secondary structures.

- The material with the best conductivity to weight ratio is aluminum.
- Aluminum has the desired thermal expansion coefficient.
- Its thermal conductivity is very favorable.

All these advantages are not found in other materials at the same price and availability. Furthermore certain alloys properties can be advantageous for specific uses.

5.1.2 Other metals

For this project, due the strict limitation on weight, steel and titanium are not considered for the main structure because of their excessive density. Other light metals like magnesium and beryllium, are difficult to machine, very expensive to obtain, and are thus not recommended.

5.1.3 Composites

We consider here only fiber reinforced polymer composites in prepegs and sandwich panels with honeycomb-core. The reinforced composites have very different mechanical and physical properties dictated by the fiber reinforcement (material and form), the reinforcement content and orientation and the polymer matrix used to support the fibers. Honeycombs sandwich panels have a smaller density and a better rigidity than prepegs, and a good impact resistance. Reinforced composites are used to made the substrate of electronic printed-circuit boards [16].

For fiber reinforced polymer composite, there are different choices of fibers, e.g. glass, boron, carbon, metallic. The most appropriate types of fibers are carbon and aramid fibers because they can achieve a lower density. Carbon fibers are conductive, whereas aramid fibers are not. The polymer matrices are usually a thermosetting resin, e.g. epoxies, cyanate esters. Only a limited number of high-performance thermoplastics were evaluated and commercialized, but the thermoset resins are much more extent. Under space conditions, thermosetting plastics are in general quite stable if the recommendations of the ECSS standards [16] are take in account for the selection. For structural applications, epoxies and cyanate esters are the most common resins. Generally cyanate ester resins are preferred to epoxies despite they are more expensive because they have smaller water absorption. But with a picosatellite it's easy to put it in vacuum chamber until the launch and like this the water absorption is limited.

Fiber composites are extremely lightweight, but they will be a poor choice for the main structure, due to generally low coefficients of expansion, and their poor thermal and electric conducting capabilities. For the interior design and sides panels, density is the most important factor, as consequence of the satellite's mass requirement that conduce to reduce the mass of the structure the most possible. The main problems for composite material are the outgassing (leaded by vacuum), water absorption, micro-cracking and micro-buckling. The outgassing does not generally degrade the properties of the polymer, but can raise contamination problems in the vicinity. If subjected to thermal cycling, micro-cracks are introduced because of differences in thermal expansion coefficient for each layer (ply) normal and parallel to the fibers. Water absorption and outgassing properties must be checked before materials are bought. Normally these parameters are hard to test, and one must rely on the information provided by the supplier.

5.2 Fastening methods

This subchapter describes the various methods of fastening and their advantages and disadvantage in the case of a space application.

5.2.1 Structural adhesives

Structural adhesives present different major advantages over more traditional fastening methods. Firstly, adhesives are the most lightweight method for joining different mechanical parts. Additionally, they only create a weak stress concentration at the interface. Adhesives can be used to join different kinds of materials which in some cases is beneficial since they provide stress relief through deformation for materials with different thermal expansion coefficients.

The major disadvantage of adhesive bonding is that disassembly is generally impossible once parts are attached. In addition to this, adhesives that are subject to thermal cycling may degrade and thus become brittle. Different types of adhesives (silicones…) are prohibited for space use since contamination is risked in vacuum conditions. Adhesives also provide a weak thermal and electrical conductivity, which might certainly lead to problems for joining parts where heat dissipation is required.

5.2.2 Mechanical fasteners (screws, rivets…)

Compared to adhesives, mechanical fasteners present the advantage that parts can be disassembled and reassembled numerous times during the testing phase. The mechanical properties of the joints are also superior to adhesive bonded joints, even though there is a risk of stress concentration whereas for adhesives the stress is evenly distributed. Mechanical fasteners are also environmentally stable except for minor risks of corrosion during long term use. For the final assembly, screws have to be glued in order to prevent the risk of disassembly during launch due to vibrations.

5.2.3 Welding

Welding is the least expensive joining method and in general provides mechanically strong interfaces. Nevertheless, disassembly of welded parts is impossible and welding is only applicable for joining of metallic surfaces. Additionally, welding can lead to inhomogeneous microstructures leading to the formation of fragile intermetallics and increasing the risk of selective and intergranulary corrosion. Moreover, welding of aluminum is an extremely difficult procedure due to its weak melting temperature. For all of these reasons, welding will probably not find an application for joining SwissCube structural components.

5.3 Structural Trades

The design of the structural subsystem progressed through a number of iterative stages. The first stage was to find several initial concepts for the design of the structure. These initial configurations had to take into account the CubeSat requirements as well as all of the specific SwissCube requirements, discussed previously.

5.3.1 Trade-offs for Primary Structure Design Concepts

From the literature four basic design concepts are considered applicable to our project. They include:

- Monoblock
- Double monoblock
- Structure in panels
- IKEA kit (several pieces)

5.3.1.1 Monoblock

AAUSat designers created a monoblock design for a frame of aluminum which is milled from one solid block of aluminum. This way a very strong, robust and at the same time extremely light frame can be achieved. Assembly consists of fixing components to the frame, directly on the internal face of the various sides of the frame. The subsystems for the satellite should be inserted/removed in a certain order.

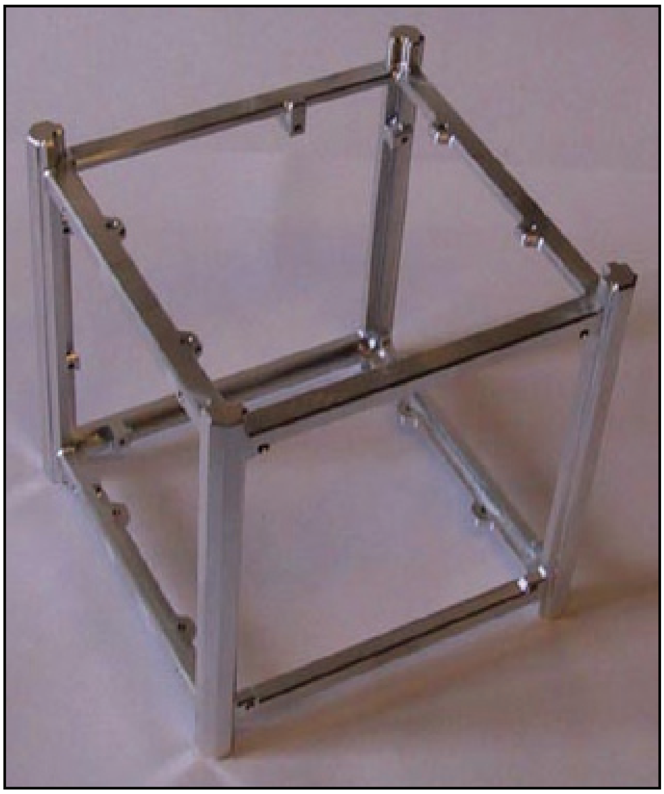

Figure 9 AAUSat (Aalborg University) option.

As mentioned previously, the advantages of this option are a reduction in the tolerance stack-up, the optimal thermal and electrical conductivity, and the saved mass because no joints are required between the various parts of the frame. But there are also disadvantages, for example the manufacturing problem, the accessibility of the subsystems, or the low flexibility, making design changes difficult.

5.3.1.2 Double monoblock

This idea comes from CalPoly for its satellite CP2. Their concept is to use a double monoblock for the frame. One of the objectives is to design a satellite that is both modular and serviceable. The triangular structure allows for easy disassembly. The entire structure can be split into two symmetrical halves by removing 8 screws and allows access to all important subsystems.

To make manufacturing easy and cost effective, the two halves are actually composed of low profile triangular pieces and four cross members. Assembly consists of sliding components into place and using fasteners to secure them.

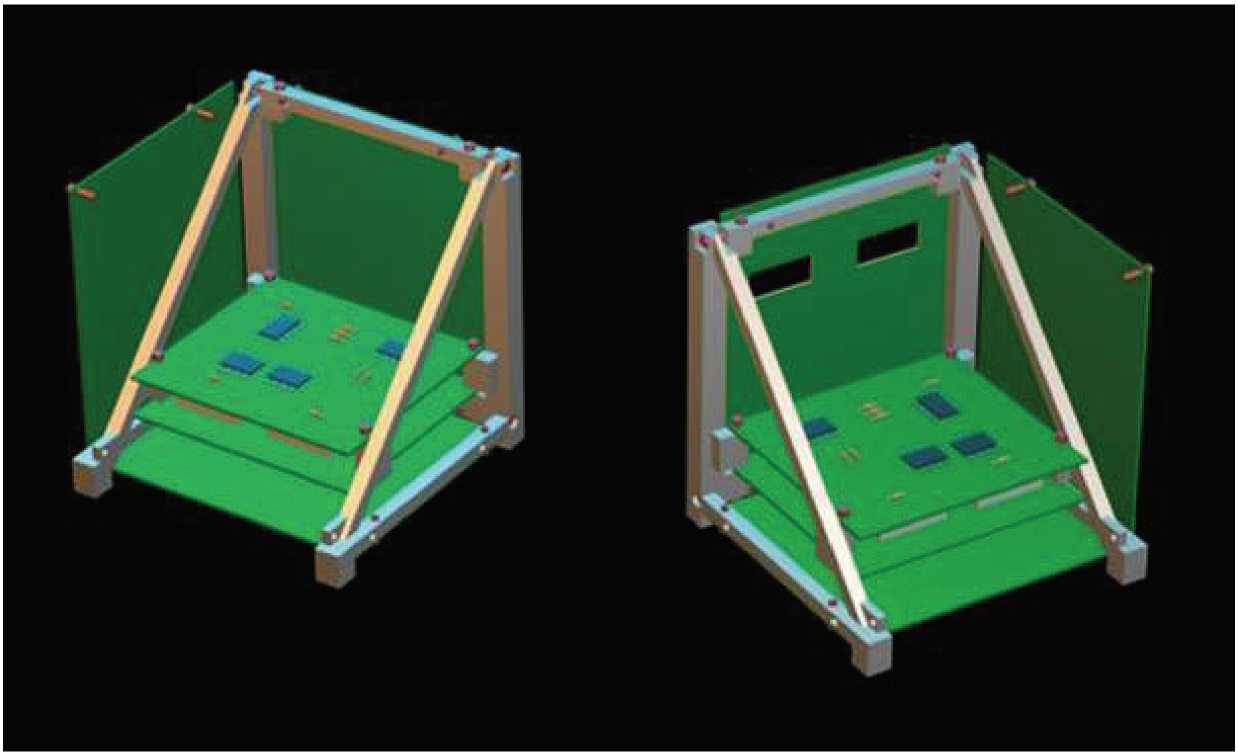

Figure 10 CP2 (CalPoly) option.

The advantage of this solution are easy serviceability of the interior components, efficient mounting of electronics and payload, or the fact that diagonal cross members offer more structural rigidity. The main disadvantage is the redundancy of the diagonal crossbars, due to their additional weight.

5.3.1.3 Structure in panels

This idea comes from the satellite of the Iowa State University (CySat). They design their CubeSat starting from panels. The two principal panels consist of two rails each; the other panels come to be fixed from above.

The advantages of this strategy are the flexibility and simplicity to reach the other subsystems. But with this option, the mass of the main structure is very high, because of the thickness of the structural panels. Moreover, the external geometrical tolerances are difficult to reach because of the assembly of several parts and the risk of non-alignment.

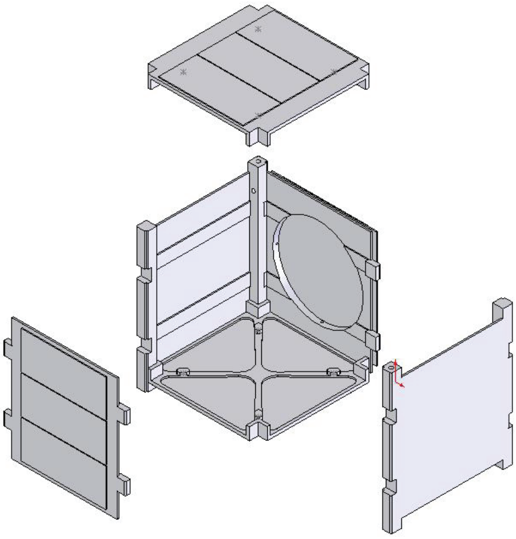

Figure 11 CySat (Iowa State University) option.

5.3.1.4 IKEA kit (several pieces)

This concept can be illustrated by the satellite of the University of Sydney (CASSat). The structure consists of four rails, eight cross-bars and 16 L-sections, allowing the crossbar to be connected to rails and also provide surfaces for the mounting of faceplates.

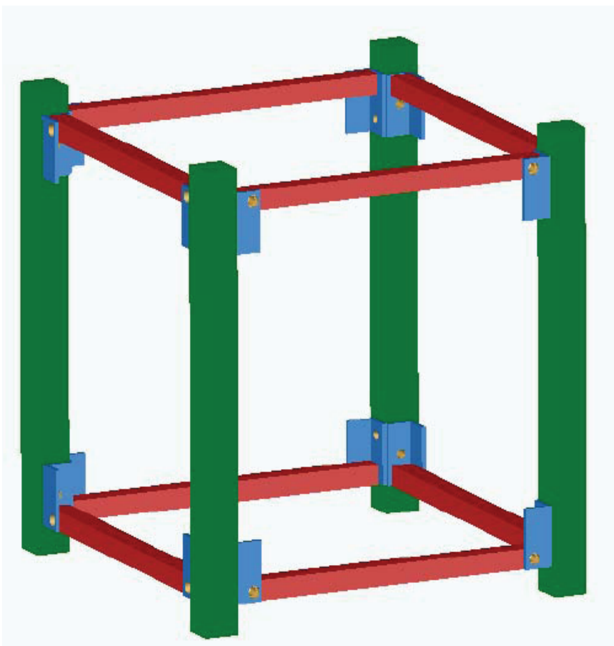

Figure 12 CASSat (Sydney University) option.

The main advantages of this design are that all pieces can be fabricated from commonly available extruded sections; they do not require expensive machining and the simple design will allow for a short design time. As before, the main disadvantage is that the external geometrical tolerances are difficult to reach. Moreover the mass of the frame is passably raised, because there are a lot of interfaces and thus many screws.

5.3.2 Initial Design Concepts

The next paragraphs describe the initial design concepts that lead to the structural baseline explained afterwards (see §6.1). It is only a brief explanation to understand the structural baseline choice.

After considering the various strategies that other universities adopted for their structures, we imagined distinct structures according to strategies described in § 5.3.1. Some of these options are illustrated by the following figures.

The use of the strategy with several pieces (maximum flexibility §4.2.3) has been rapidly dismissed because of the relative heaviness of the structure due to the oversizing of the pieces at the interfaces. Moreover the risk of misalignment during the assembly is present, and the external geometrical tolerances are difficult to reach.

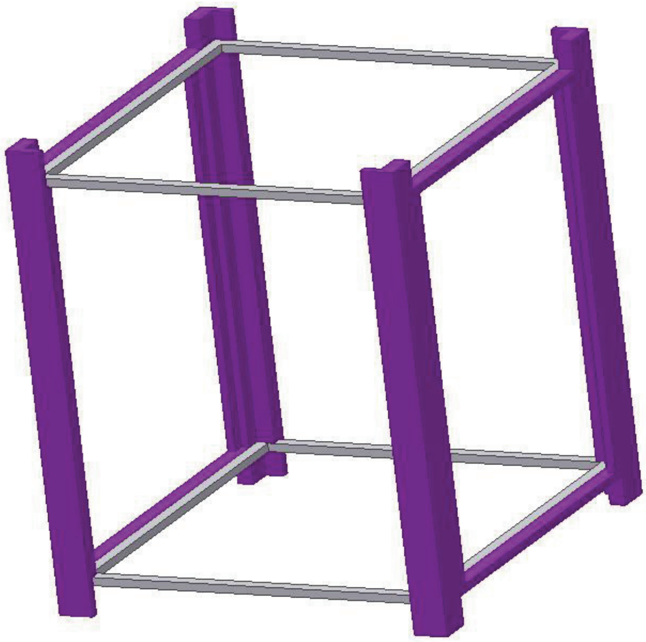

Figure 13 Structure with two panels and two crossrings.

Next a structure in panels has been investigated (see Figure 13). The idea is to have two panels, each with two rails. To connect these two main parts, the use of crossrings is envisioned. Indeed, to be rigid enough and to be able to drill and thread holes in the crossring, this piece must be relatively massive, thus making this an unfavorable strategy. In addition with this kind of structure the crossbars binding both rails are redundant and to optimize this, an alternative strategy with four crossbars to connect the violet panels instead of the two crossrings can be a solution. This is however, less robust with respect to vibrations. But this idea again requires a lot of parts and thus returns to the first strategy that is the "IKEA kit".

Finally, the option of the monoblock is chosen (see Figure 14). The major reason of this choice is that the monoblock offers the best compromise between the lightness of the structure and its robustness, compared with the other strategies stated in §4.2.

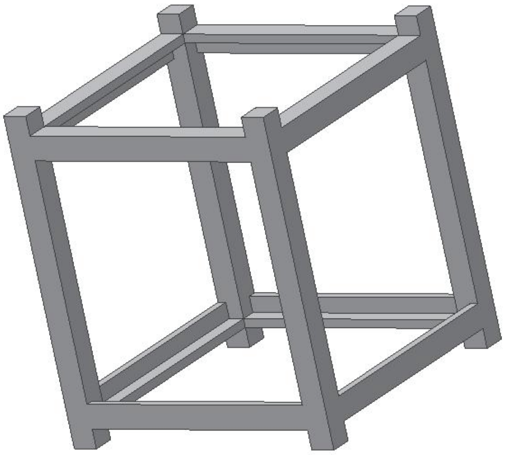

Figure 14 Monoblock option (CASSat).

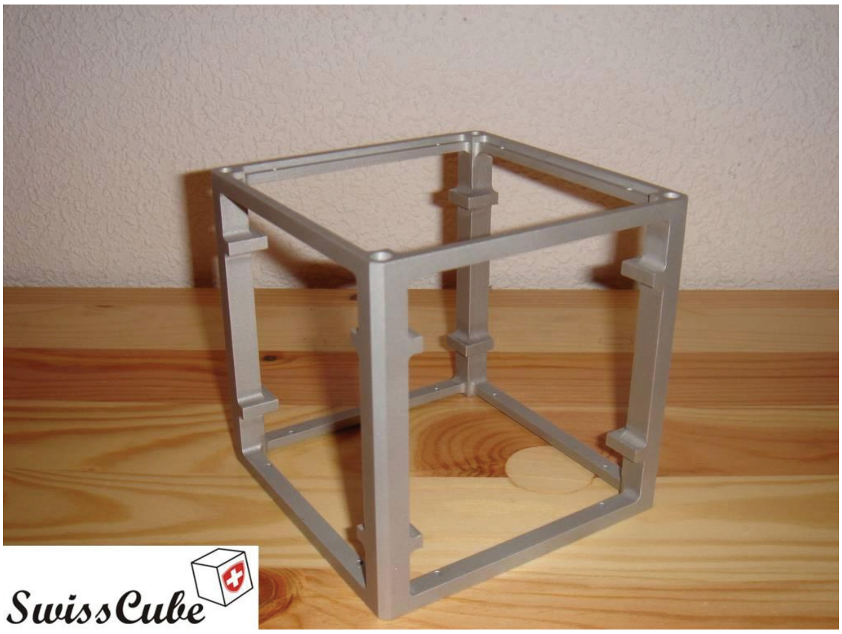

Figure 15 View of the first prototype.

To ensure the feasibility of our structural strategy, a monoblock frame has been machined from a solid block of Certal® aluminum (see Figure 15). It has been done with a CNC milling cutter. First the aluminum block was rough-machined on all six sides. Thereafter, it was mounted with one end used for attachment for the actual machining of the five sides. When they were finished, the block was mounted in the CNC miller in the thread holes, and the last side was machined. Finally the free block at the centre was removed.

For simplicity, the eight feet have not been machined. The mass of this prototype is 112g, and the manufacturing time is 12 hours (4 hours to program, 8 other to machine).

5.3.3 Design Iteration

Design is an iterative process. The necessary number of iterations is one more than the number you have currently done. This is true at any point in time.

Akin's Laws of Spacecraft Design [14]

To reduce structural redundancy and in this way the mass of structure, it is important to iterate upon the best initial design. This also allows the optimal sizing of components. Several stages of design iteration are required to arrive at an optimal structural configuration.

The design starts out with a simple monoblock which is the most weight efficient solution but problematic on the level of the assembly of the subsystems. The crossbars of the monoblock frame prevent the efficient use of the whole space inside the cube and limit the access to the subsystems during integration.

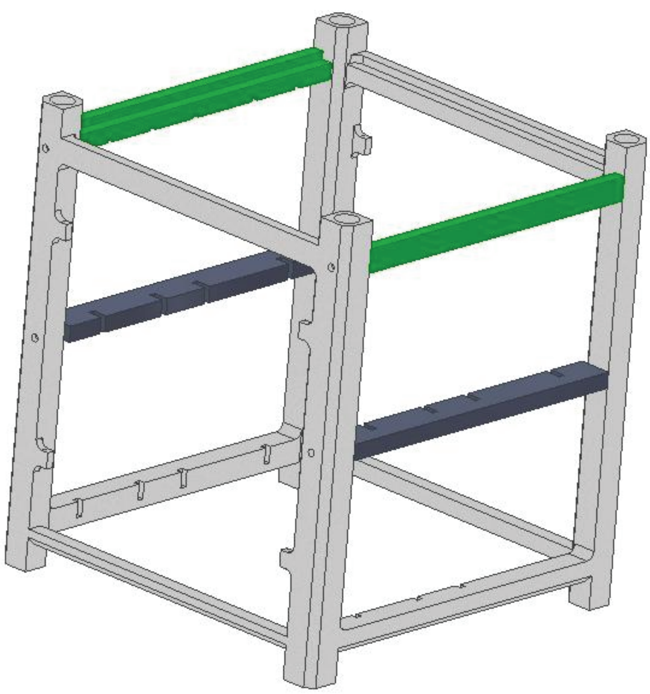

Figure 16 Monoblock with two removed top-crossbars.

In order to overcome this disadvantage, two of the four top-crossbars can be removed. In this way, it is possible to insert PCBs whose width is close to 100 mm. The removed crossbars are added to the subassembly which is composed of the various PCBs and the whole is finally introduced inside the SwissCube (see Figure 16). The disadvantages of this option are the difficulty to machine the frame and moreover the risk of deformation of the frame. The structure also becomes less robust.

Subsequently the decision to keep a "full" monoblock is made, in order to assure that the frame will be sufficiently rigid, notably from a vibrational point of view. After having to define the general shape of the main frame, its weight is investigated, and in order to satisfy our limited mass budget, weight has been minimized everywhere possible. For that, through holes of 6.5 mm diameter are bored in each rail, and the exterior and interior edges of each rail are chamfered to 2 mm In order to further reduce the mass, the crossbars are also optimized. Their sections are very small in order to save the maximum of weight while remaining sufficiently rigid.

5.4 Configuration Trade-offs

The internal layout and configuration is a very important aspect in the design of the SwissCube structural subsystem. Primarily it is important that the internal components of the individual subsystems are located in such a way that the CubeSat specifications are met and the centre of gravity of the satellite is within two centimeters of its geometric centre. It is also important that the components are located in such a way as to optimize their performance and maintain functionality throughout the mission. In our case, the requirement of the payload will dictate the location of the internal components, because of its relatively big size and its central role in the mission.

In order to reduce any oscillation and to increase the stability and control of the satellite, the centre of mass needs to be as close to the geometric centre of the satellite as possible. Therefore, we need to distribute the weight evenly throughout the interior of the satellite. This is not an easy task because depending on the different design options there are several components that must be placed in specific locations of along certain axes regardless of their weight (i.e. payload, inertial wheel).

Access to the electrical components is an important design consideration. During the development and testing phase of the CubeSat, the printed circuit boards (PCBs) will be removed and replaced with great frequency. Easy access to these components will save a significant amount of time over the entire development and launch phase.

5.4.1 Placement of the payload

The payload of the SwissCube represents the largest single unit in the construction. The dimensions of this subsystem are Ø30 x 70 mm (TBC). Due to the great influence of the payload placement on the other subsystems, the payload position must be defined at first.

We have many possibilities for positioning the payload; the direction can be: 1) parallel to the P-POD rails, 2) perpendicular to the P-POD rails and the location can be: 1) at the center of a face (see Figure 17), 2) at a corner of a face, 3) displaced from the center only along one direction of the reference frame.

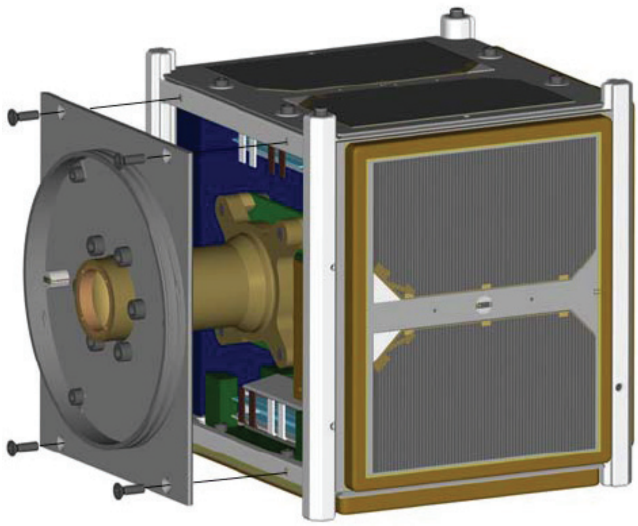

Figure 17 The case of a centre location of the payload (AAUSat).

5.4.2 Arrangement of the main PCBs

Each of the main electronic subsystems is implemented on a standard size printed circuit board (PCB). The dimensions of the standard PCB are maximized in order to use as well as possible the place available.

There are only a few specific requirements for the placement of the PCBs. Attention must be paid in order to minimize the interferences between the various electronic subsystems, as for example the RF subsystem will certainly generate a lot of disturbances. Moreover, the placement of the PCBs has to be optimized in order to minimize the connections and wires. The arrangement of the PCBs must be seen in relation to the necessary interconnections between the different printed circuit boards.

In a general way, the placement of the payload restricts the possibilities for the arrangement of the 5 to 6 main PCBs. There are basically three configurations possible:

1) An arrangement in layers or a stack is commonly used. Generally, the PCBs are arranged in stack, the whole forming a secondary structure which is fixed to the main structure (see Figure 18). An advantage is that the rigidity is higher because the PCBs are interconnected by a secondary structure. But the secondary structure can be seen like a disadvantage in a weight point of view and connections between non-neighboring PCBs are difficult to establish.

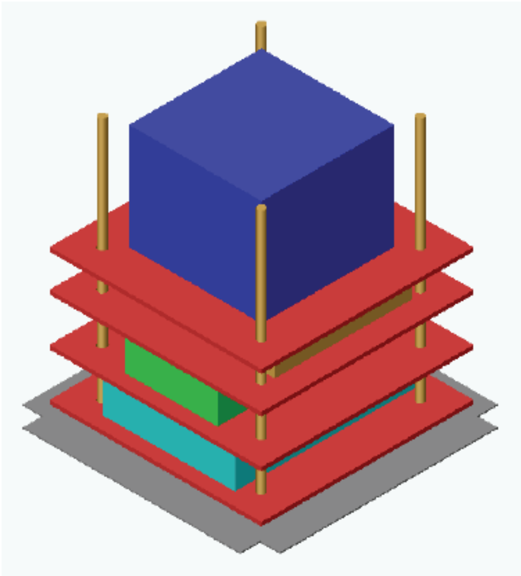

Figure 18 Schematic of PCB stack (CASSat).

2) Arrangement at the faces of the satellite (see Figure 19). The advantage of this case is that the PCBs can be fixed firmly at the main frame. Moreover, the place between the rails and crossbars are used in an optimal way, which makes it possible to have much free space in the center, for the payload for instance.

But negative points are also present: the connections between the electronic subsystems are difficult, and the use of a lot of wires makes assembly/disassembly difficult, and can be a source of error. Moreover, the subsystems for the satellite should be inserted/removed in an invariable order.

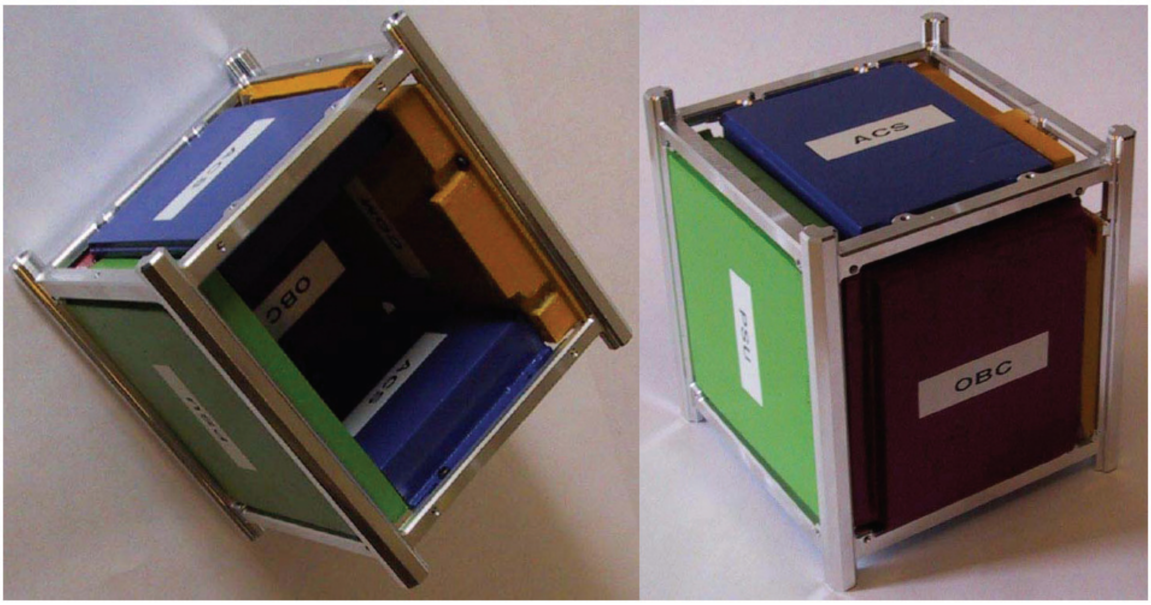

Figure 19 Arrangement of the PCB at the faces of the satellite (AAUSat).

3) Arrangement in slot with the use of a motherboard (see Figure 20). With this option, the various subsystem boards are attached to the motherboard. The main advantage of this alternative is that the connection between the electronic subsystems needs almost no wires; all the electronic connections pass through the motherboard. Thus, the risk of error during assembly is drastically reduced. Another benefit of this strategy is that the whole electronic subsystems can be disassembled from the main frame in one operation.

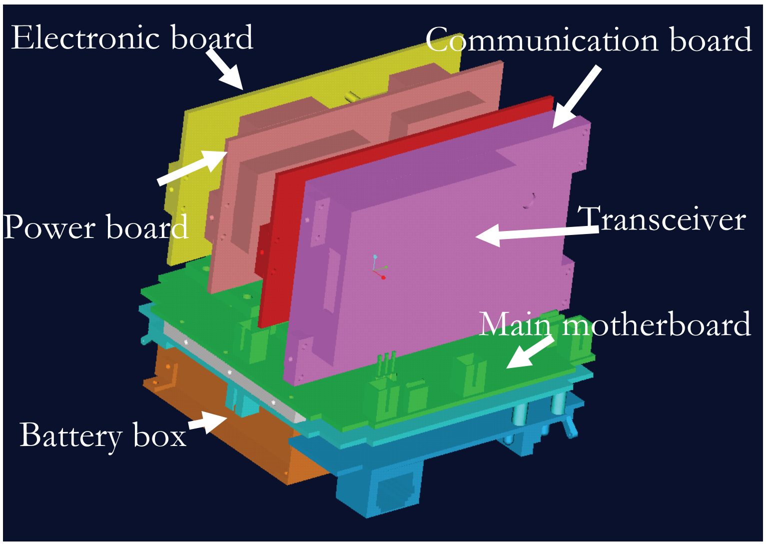

Figure 20 Arrangement of the PCB in slot (University of Tokyo).

5.4.3 Magnetic torquers and inertial wheel

The area of the coils should be as big as possible. To obtain a practically feasible design, the coils must be coplanar with the outside faces of the spacecraft.

The trades for the ADCS subsystem are the amount and position of the magnetic torquers and inertial wheel. For our mission objectives we can imagine to use one to three (or more for redundancy) magnetic torquers and inertial wheels. The limiting factor for the number of attitude control systems are the mass and also the available space. Concerning the position of the various attitude control systems, we have the possibility both internal and external location.

5.4.4 Antenna array

The trades for the placement of the antenna deployment system are the fixing place of the deployment system and the orientation of the antennas. The attachment location is generally an exterior face of the satellite, but it is also possible to use an internal deployment system. The orientation of the antennas can be in the plane or perpendicular to the fixed location of the deployment system. This second option is significantly harder to achieve.

5.4.5 Batteries

The two batteries used in the satellite are assumed to be rectangular units of 35x70x5 mm (TBC) (length, width, depth). The batteries are critical components in relation to the spacecrafts temperature. The battery temperature must be kept in the range between 0°C and 45°C (TBC), which might necessitate special insulation of the battery, depending on the thermal environment. Apart from the thermal issues, the placement of the batteries is not critical, and it might be used to optimize the mass balance.

6 BASELINE DESIGN

Having outlined various options for the choice of materials, the different fastening possibilities, the design of the principal structure and the internal configuration of the subsystems, we condensed this information in order to obtain our baseline design of the SwissCube's structure.

6.1 Structural Baseline

Figure 21 represents the structural baseline. The body structure consists of only one part, the main frame, whereas the structural subsystem consists of no more than three major structural components. These components are:

- the main frame (in grey)
- spacers (in orange)
- faceplates (side, top and "payload" panels) (in pale yellow and aqua respectively)

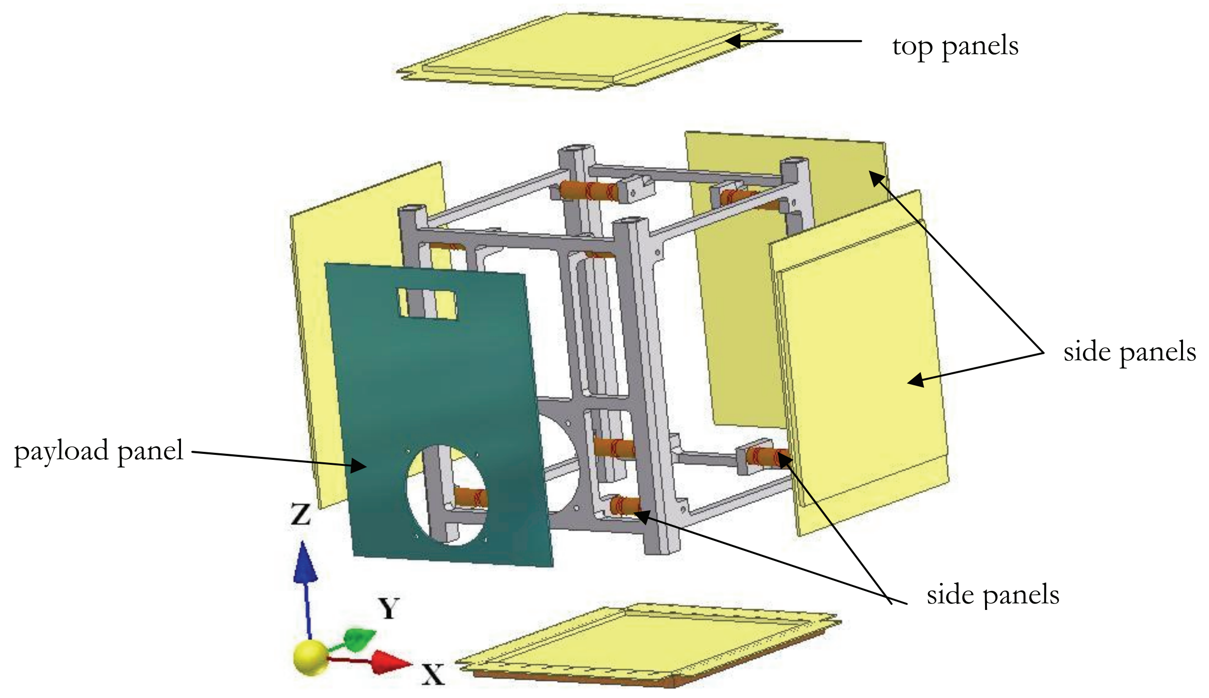

Figure 21 SwissCube body structure and Reference Coordinate System.

6.1.1 Main frame

The design of the frame has been meticulously investigated in order to minimize mass, since it was clear from the start that it would be the most mass consuming part. The frame has been altered radically but still within the boundaries of the restrictions set up by CalPoly.

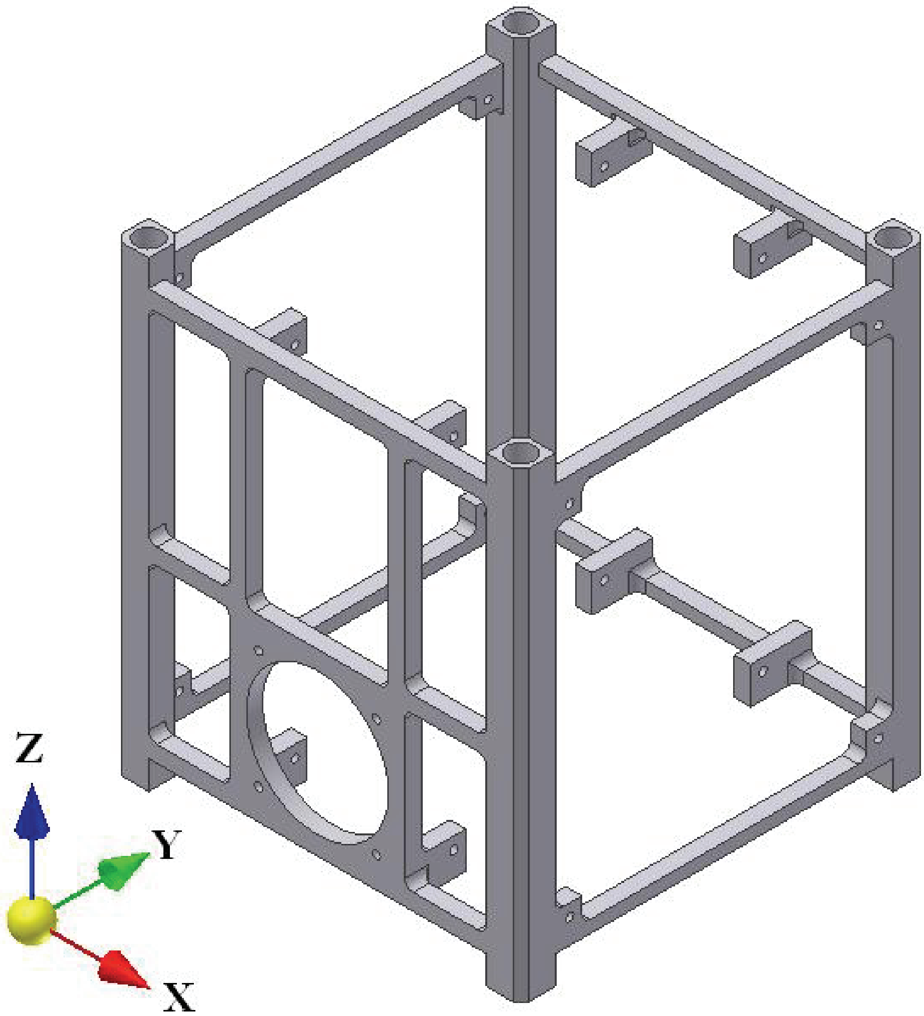

Figure 22 View of the SwissCube main frame.

As one can see in Figure 22 and as explained in more details in §5.3.3, holes and chamfers are machined at each rail to save mass. The diameter of the holes is 6 mm and the chamfers are 2 mm at 45°. The crossbars have a rectangular shape with a section of 3 x 4 mm. The four crossbars parallel to the X-axis have two protuberances each one in order to fix the internal subsystems by the means of the spacers and at the same time to fix the external panels (for more details see § 6.2). Between the crossbars in the Y direction and the rails, material is kept in order to have a counter fixation for the spacers at a mechanically rigid point of the main structure.

The sides of the main frame are reasonably the same, except the Y- side which contains additional structural elements in order to fix the payload is fixed. This face must be enough rigid to support the weight of the payload and since the payload subsystem is precariously balanced fixed like a cantilever beam some additionally crossbars are needed to guarantee that the payload's axis doesn't undergoes any misalignment. The final mass for the main frame is 99 g.

6.1.2 Spacers

The role of the spacers is to connect the different PCBs between each other and at the same time to fix the PCB's stack to the main frame. Additionally, the spacers serve as a thermal path between the PCBs and the aluminum frame. Therefore, the spacers will also be fabricated from the Certal aluminum alloy (Appendix B - Material Properties). A whole spacer like in Figure 23 is composed of three basic spacers which are all screwed one into the next.

The external diameter is 6 mm whereas the internal diameter is 4 mm. Both basic spacers at the extremity of an entire spacer unit have threaded holes in order to fix the stack to the main frame. The length of each individual spacer is actually not known because the distance between different PCBs depends on the dimensions of the electronic components which can fluctuate. Nevertheless, the length of the entire spacer unit is already fixed at 24.5 mm. with a mass of 2g each. A total of 8 spacers is used in the entire structural subsystem.

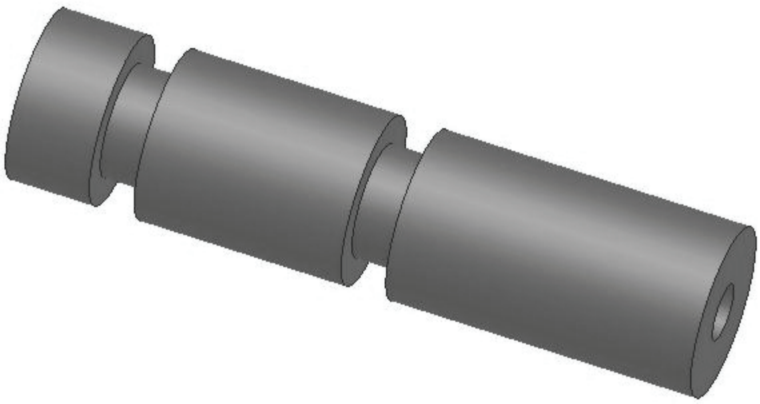

Figure 23 View of a three spacers screwed together.

6.1.3 Sides and top Panels

The face plate provides the surface for the mounting of external components (solar cells, magnetic torquers etc.).

As shown in Figure 24 the side panels is are rectangular composite plates with a size of 83mm wide by 100mm long and a thickness of 1 mm (TBC). This part has an elevated centre section (2 mm thick) in order to wind the magnetic torquers around it. Three side panels are used in the structural subsystem, with a mass of 13g for each one. In order to reduce mass, the plates could be lightened by making holes into the panels. This can be achieved since neither the solar cells, nor the panels are subject to large stresses. Thus the solar cells don't need to be glued on their entire surface.

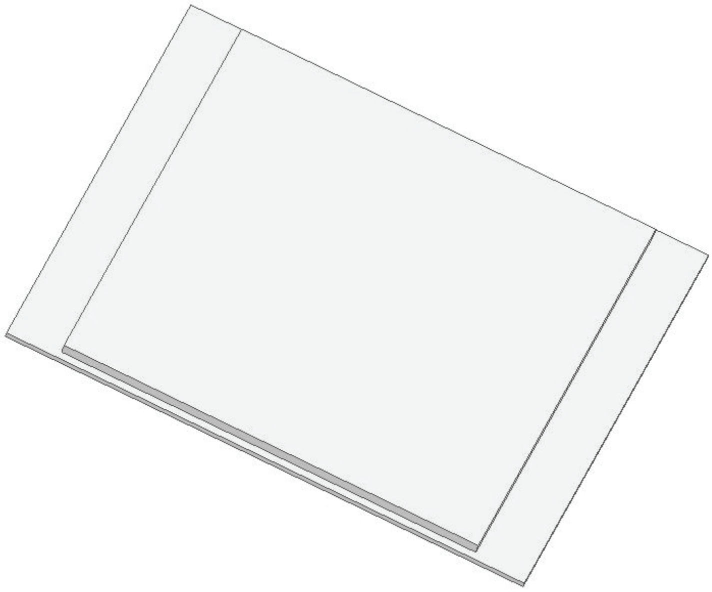

Figure 24 View of the side panel.

The top and bottom plates have a slightly different geometry; 100mm wide by 100mm long with cut-outs at the corners (see Figure 25). They are fabricated from the same 1 mm thick composite plate (TBC) as the side plates. These parts equally have elevated center sections for the same reason as mentioned before, and its mass is 13g. Two top panels are used in the structural subsystem.

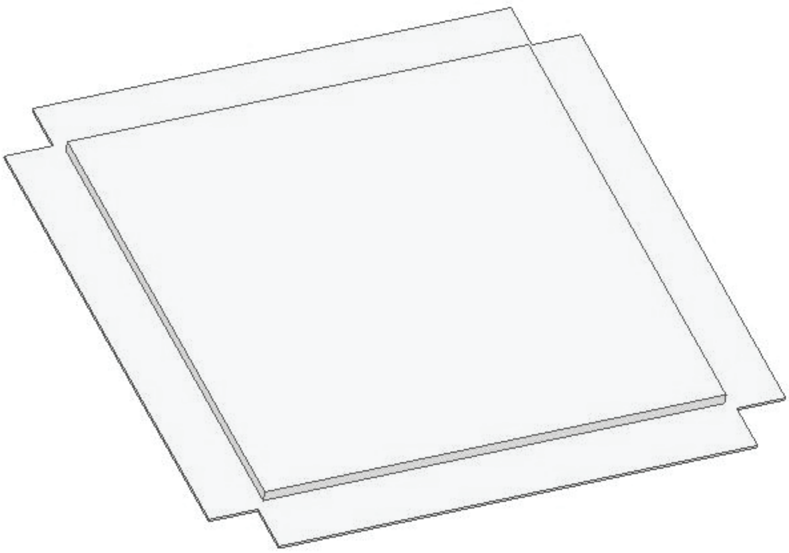

Figure 25 View of the top panel.

It must be noted that there is a possibility to have holes in some of the panels for screws or for depressurizing.

6.1.4 Payload Panel

The payload plate provides the surface for the mounting of the antennas and has two large holes, one for the payload and the other for the access port. The diameter of the payload hole is 25mm (TBC); the size of the access port is dependent on the types and dimensions of the connections (TBC). As shown in Figure 26 this part is a rectangular plate 83mm wide by 100mm long with a thickness of 1 mm (TBC). Composite material will probably be used for this panel; however, this decision must be made considering the attachment of the antenna deployment system. Its mass is 13g and one panel of this kind is used in the structural subsystem.

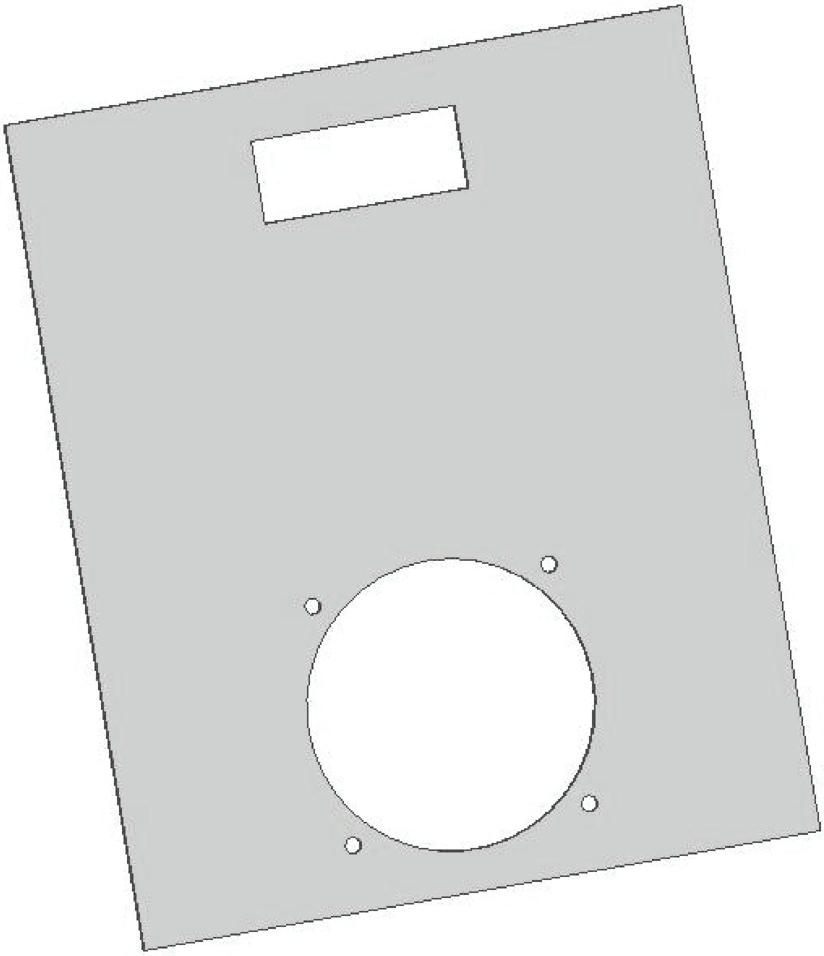

Figure 26 View of the payload panel.

6.2 Configuration Baseline

The final internal configuration is a combination of different trade-offs explained in § 5.4. The kind of internal layout is directed by two principal restrictions: the payload and the arrangement of PCBs. The ideal baseline is that which optimizes these both constraints at the same time.

Concerning the payload, the best choice for the orientation is along the direction perpendicular to the rails. In this case the camera points out of the side face which already features the access ports and a face is "saved"; t. This has the advantage, that payload, antennas and the access port are all on the same face and hence, there are five free faces free for the solar panels. The payload points in the Y- direction.

For the placement of the camera, the "full" center solution is not the best way for a key reason: when the camera is in the centre of a face, it is difficult to fix it in a solid way. The strategy to fix the payload at a corner can seem like a good idea, especially in a fastening point of view, but in this case the inertial properties of the whole satellite are unpleasant (see remarks in §7.1.2). This is why we chose to displaced the payload from the center only along one direction of the reference frame (see Figure 30).

Relating to the PCBs, their arrangement takes into account all the trade-offs described in § 5.4.2. After a lot of iterations, the following option is selected: a motherboard is used in order to connect the various electronic subsystems and to reduce the number of wires. Moreover the PCBs are stacked two by two giving a rigid structure. Finally, the both stacks are fixed on the faces of the satellite, allowing a large amount of free space for the payload subsystem and keeping an increased accessibility to the PCBs placed in the middle of satellite. Consequently, the advantages of all three trades précised in § 5.4.2 are used at a maximum.

6.2.1 Overall Architecture

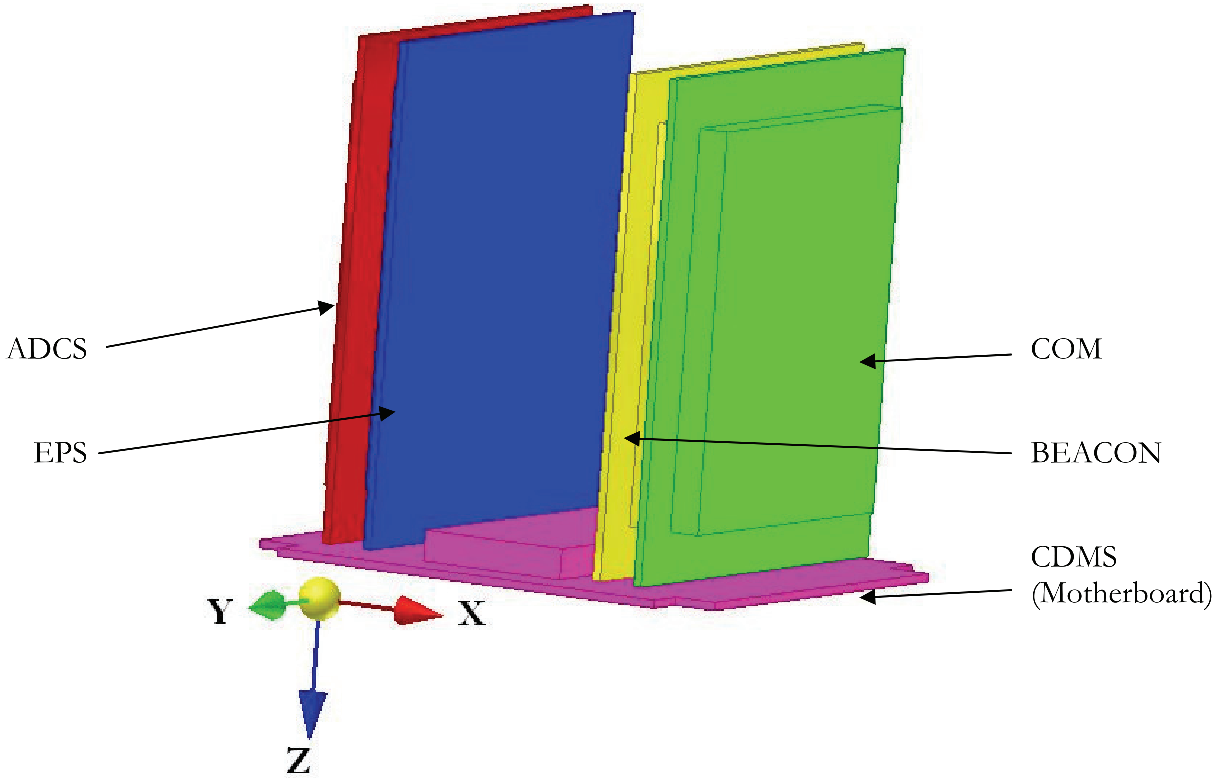

Figure 27 Arrangement of the main PCBs.

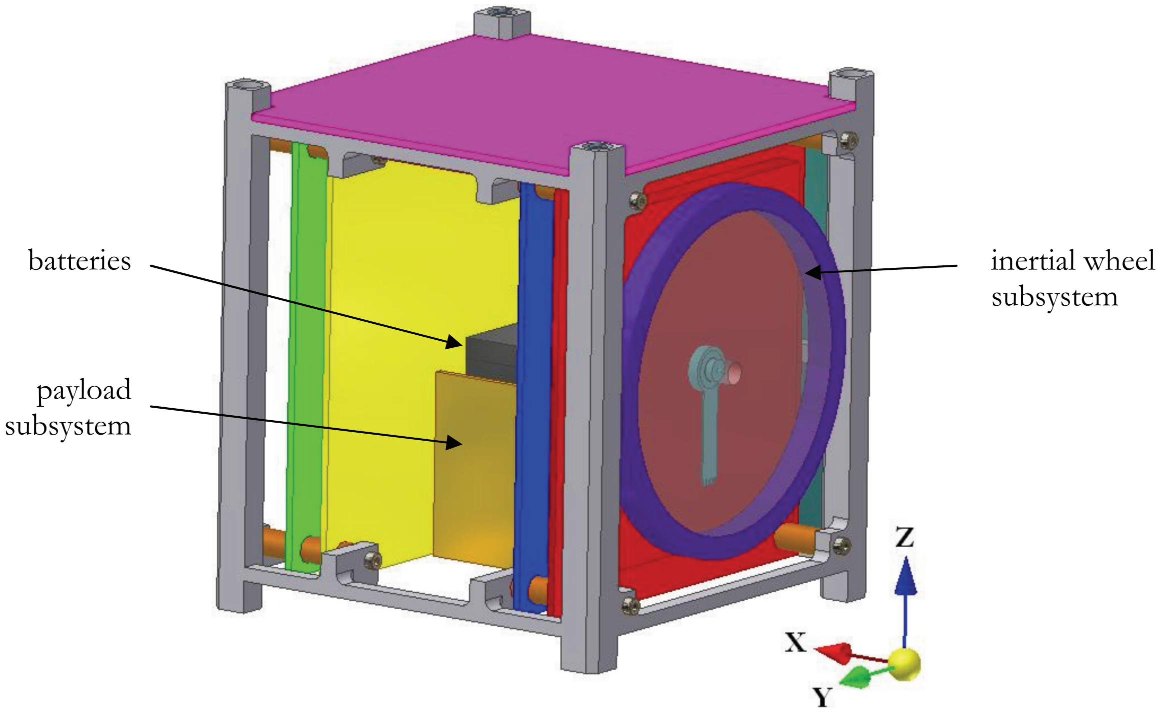

Figure 28 SwissCube internal layout.

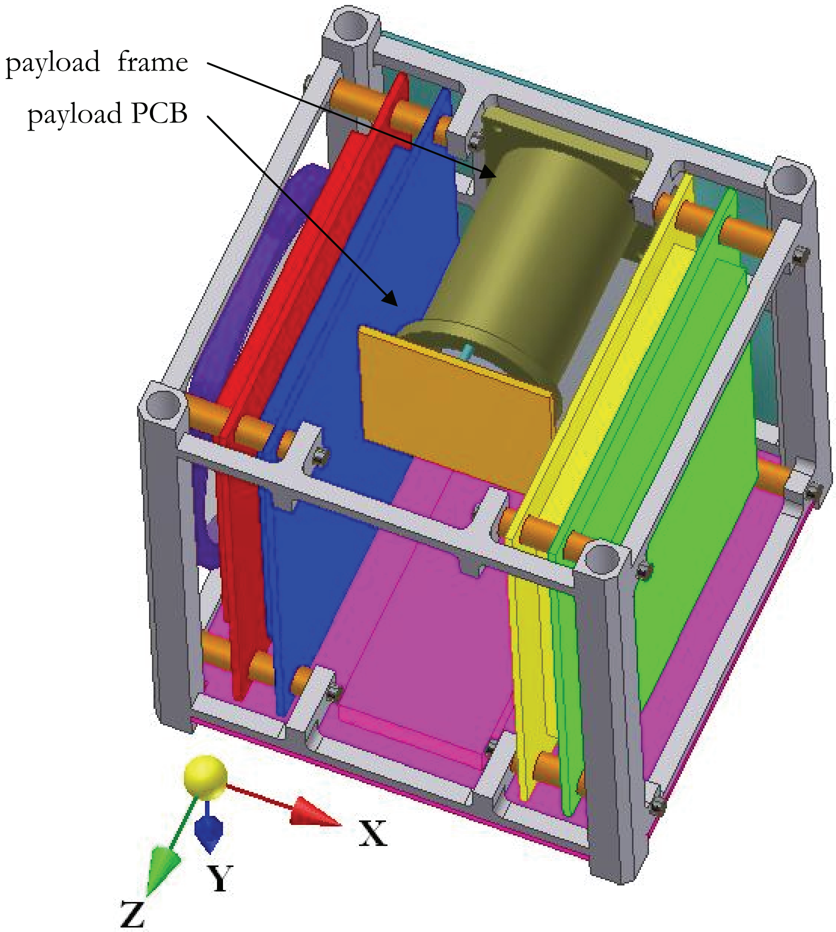

Figure 29 SwissCube alternate view of internal layout.

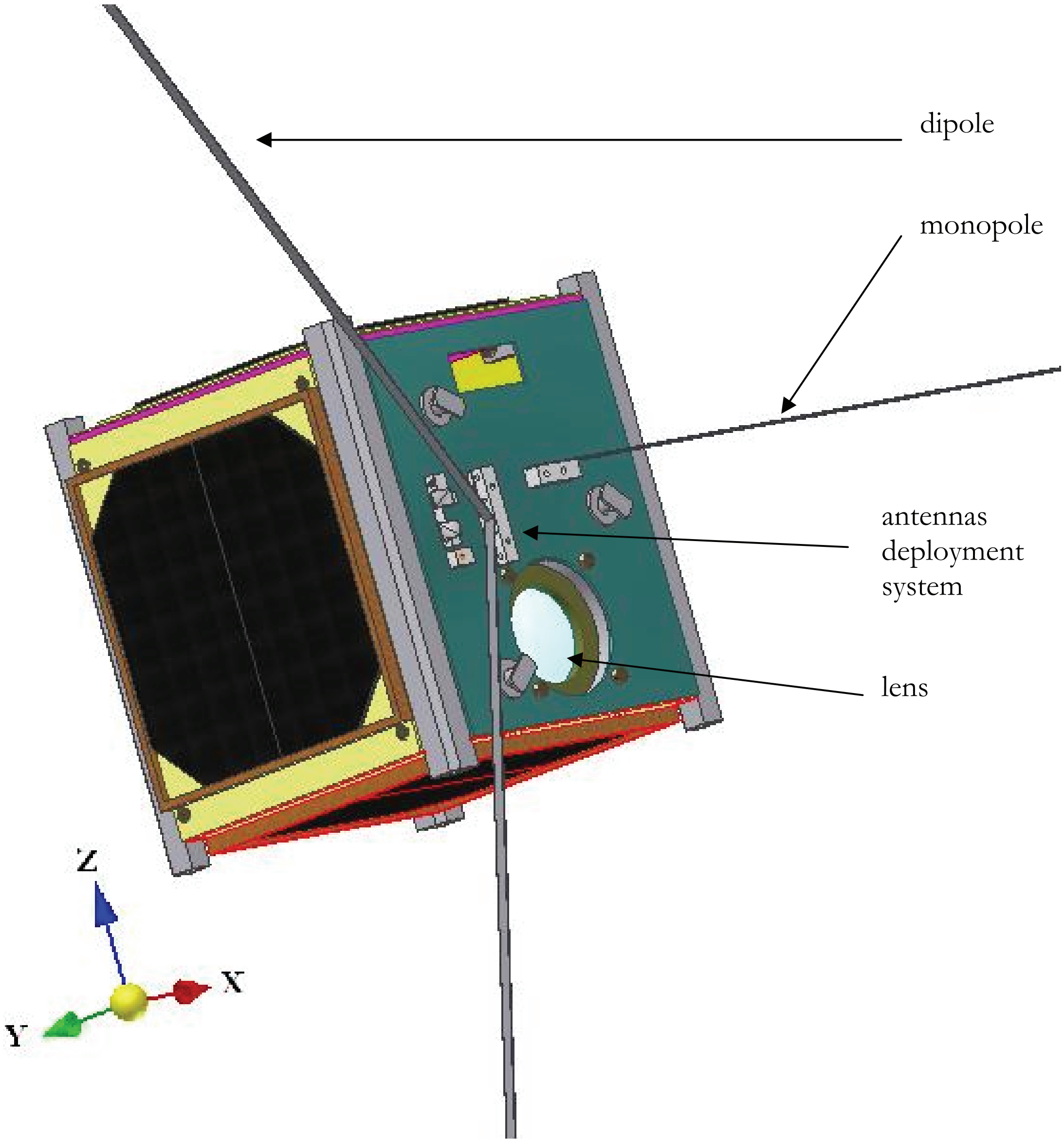

Figure 30 SwissCube external layout (antennas are cut).

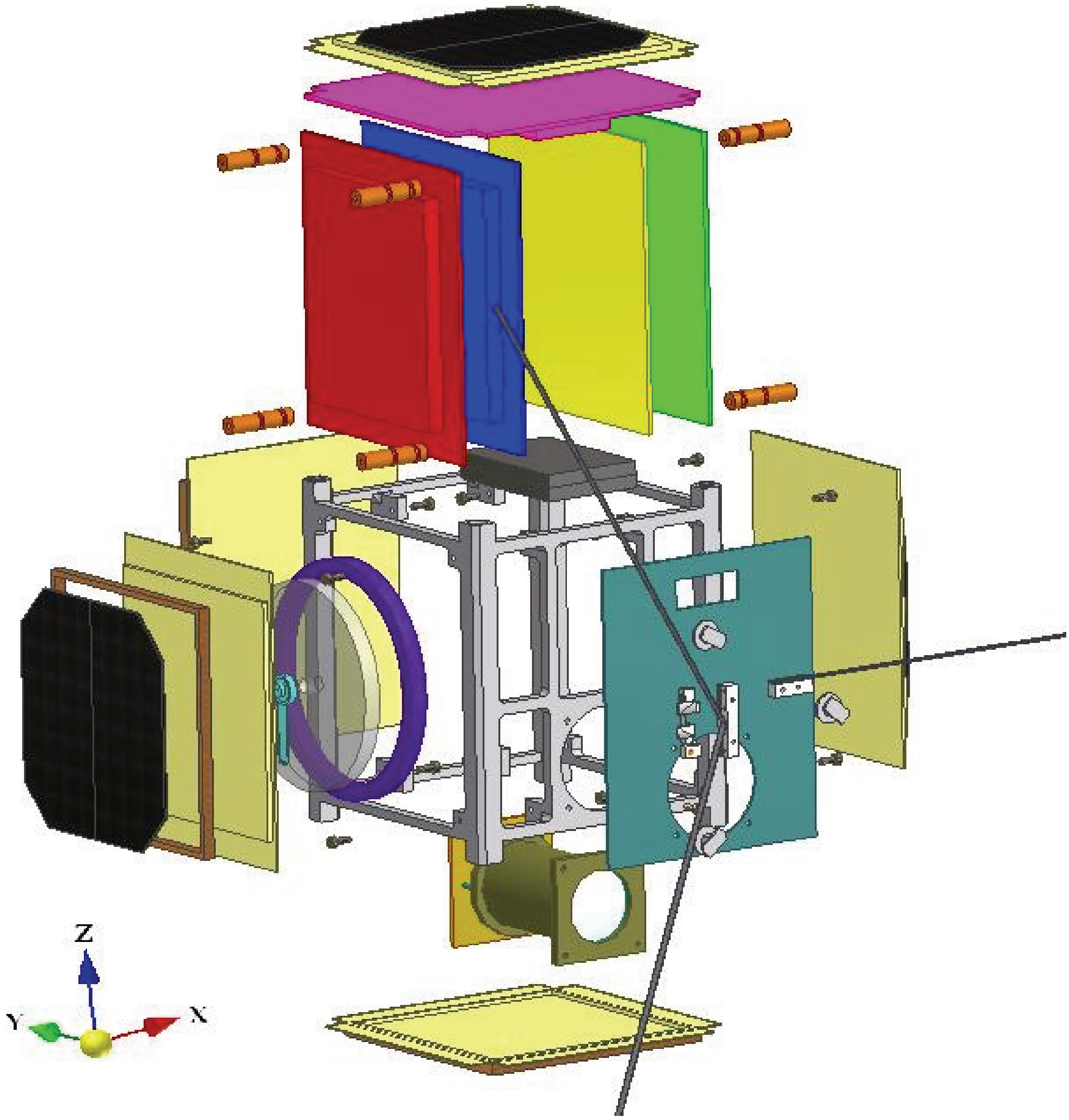

Figure 31 3D exploded view of the SwissCube.

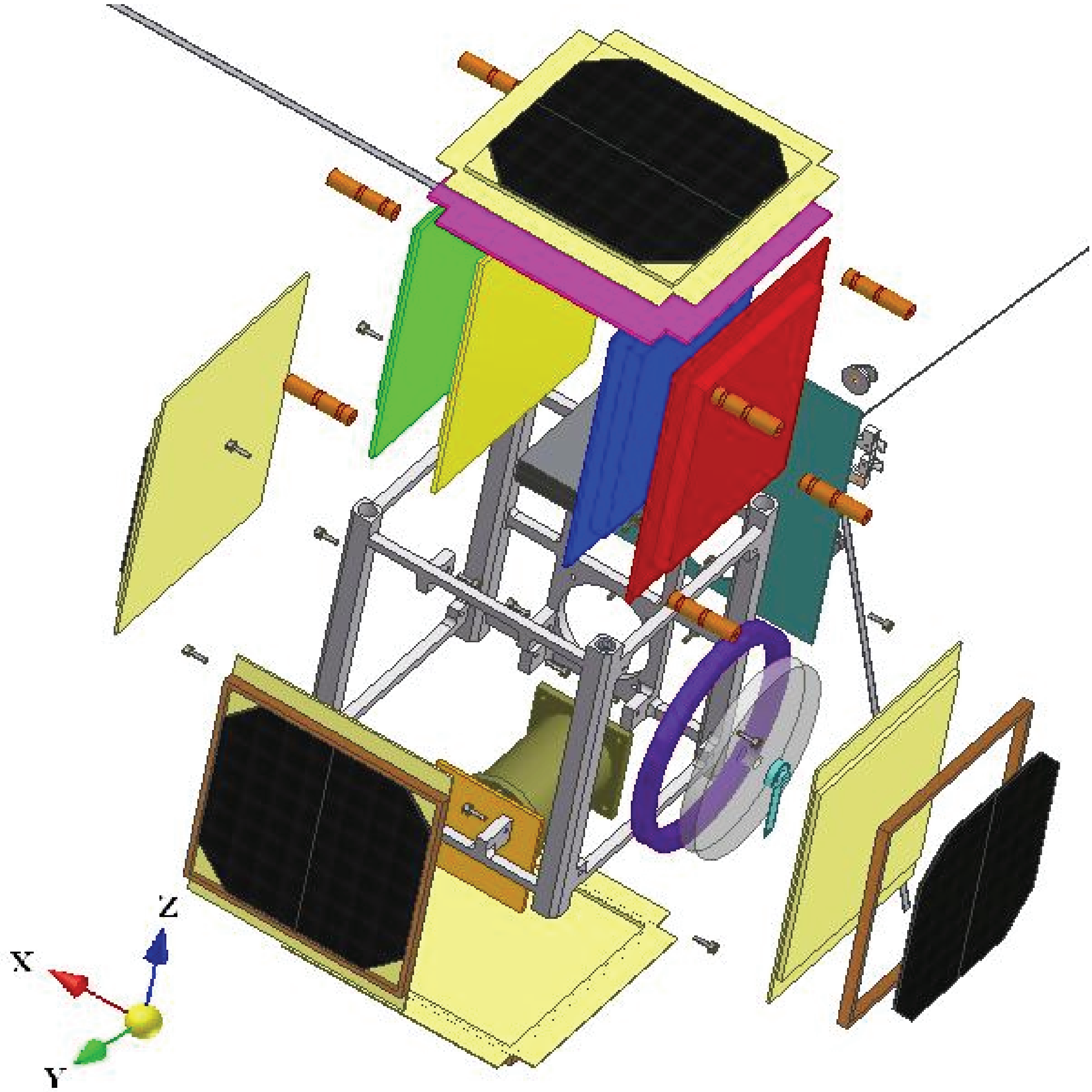

Figure 32 Alternate 3D exploded view of the SwissCube.

As the exact dimensions of the individual subsystems are not yet know (most are still in the prototype phase) it is not possible to produce a detailed established internal configuration design remains preliminary. However it has been possible to produce a preliminary design made using estimated masses and volumes. Each individual subsystem has been treated as a combination of parallelepiped shapes with an even mass distribution. This is an approximate assumption; design and analysis of the internal layout should be performed when more information on the size and masses of the individual subsystems is available.

6.2.2 Motherboard PCB

The motherboard PCB does not only serve as a connectic board, but has also the function of a proper electronic board for the CDMS subsystem. This has the advantage of having an essential subsystem for the data flow between the various subsystems in a central position of the data transfer. The placement of an electronic subsystem on the motherboard is due to additional free place in center because of the placement of the PCBs at the sides (see Figure 33). The estimated free place for electronic components is 40mm x 90mm (TBC).

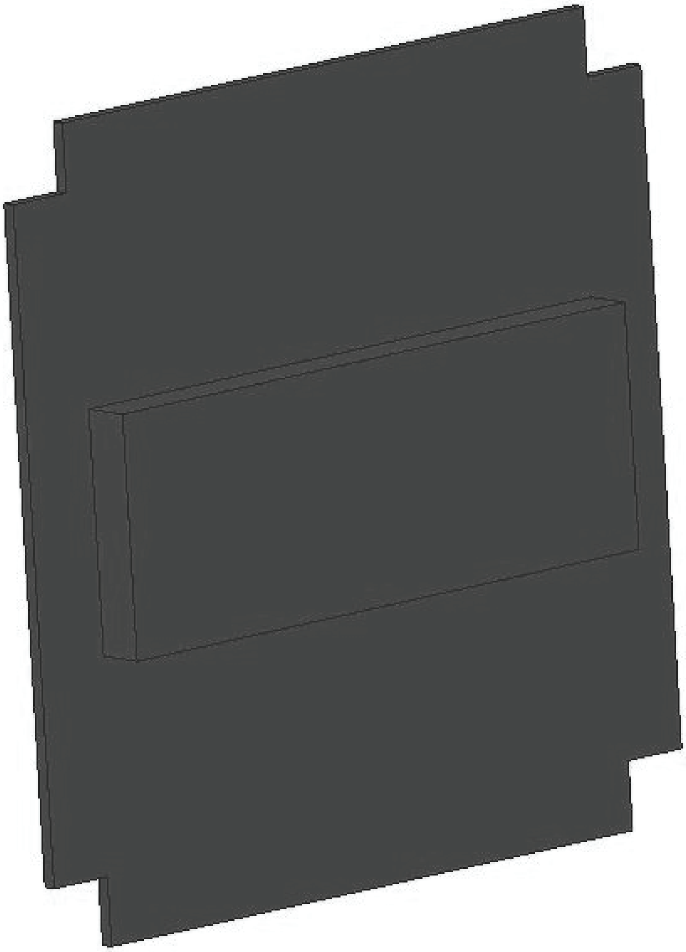

Figure 33 View of the motherboard.

6.2.3 PCBs

The main type of PCBs is of a rectangular shape with dimensions of 88 X 98 mm (see Figure 34). Additionally, there will be holes passing through the PCBs for the spacers establishing a thermal contact to the main frame. The PCBs will be fabricated from FR-4 material, which is a composite material (glass fiber reinforced epoxy). The standard thickness of a PCB being 1.6mm, the weight of a PCB of this surface will be 25g. The weight of the copper sheets can be estimated as being 3g (TBC) for a multilayer PCB.

The following subsystem will have their proper PCB: EPS, ADCS, BEACON and COM (see Figure 28). For connectical reasons the COM and BEACON as well as ADCS and EPS PCBs have to be next to each other. There are several advantages from this choice: First of all, the ADCS board (in red) can be placed next to the inertial wheel and will be equally in the corner formed by the three magnet torquers (TBC) in order to limit connection distances. The EPS subsystem will thus be placed closely to the center and so have short connection paths to the batteries as well as to the access port.

Additionally, the payload subsystem requires a small proper PCB which will be fixed directly behind the payload frame (see Figure 29). This PCB will only feature few components and thus the estimated dimensions where taken as 25mm x 25mm (TBC).

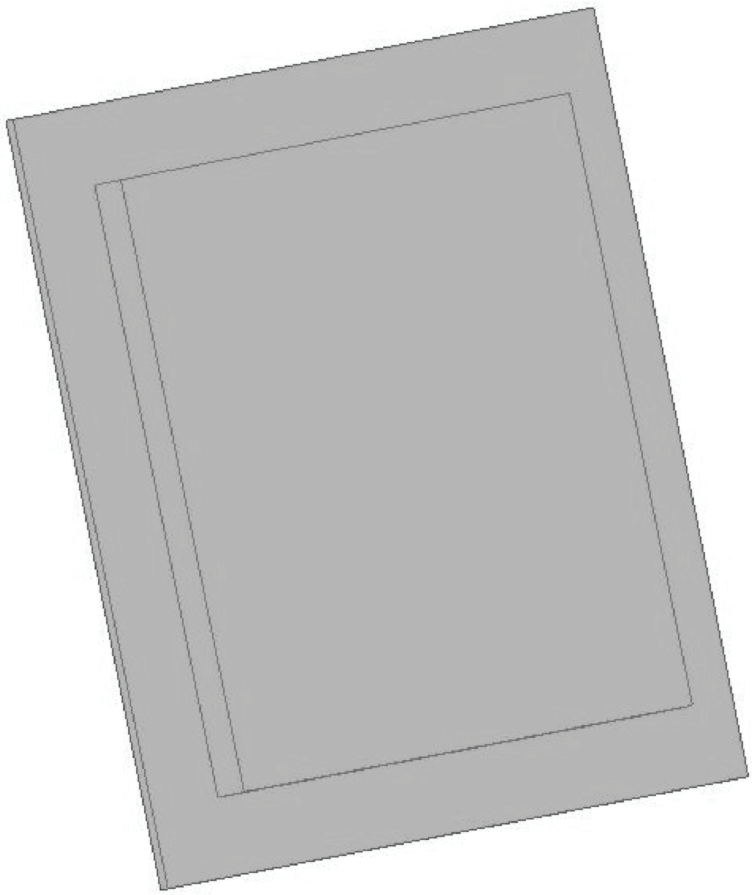

Figure 34 View of the main type of PCB.

6.2.4 Magnetic torquers and inertial wheels

The magnetic torquers have to cover the three different axis and so define a trihedron. Additionally, a inertial wheel is needed, since the magnetic torquers cannot create the torque in the direction parallel to the earth's magnetic field. So, in order to maintain global controllability an auxiliary actuator is needed to cover this uncontrollable axis (see ADCS report). This defines the relative position between the different actuators.

As mentioned previously the coils forming the magnetic torquers will be wound and glued around the protuberance of the side panels. For the inertial wheel, the fixation of the motor can either be done on the side panel or on the ADCS board itself (TBC)

6.2.5 Antennas

The antennas are rolled around three different points and will be detached using a burning wire mechanism. The deployment system is fixed on the same face used by the payload and the access port, thus keeping a maximum of five faces for the solar cells (Figure 30). There are two different antennas used; a dipole and a monopole. From a communication point of view, the antennas preferably have to be oriented in a way that the never directly points towards the ground station (see Mechanisms & Telecom reports).

6.2.6 Batteries

The batteries are placed close to the geometrical center of the satellite in order to optimize the inertial properties of the satellite. Connection is simple since they are situated close to the EPS board. Fixation will probably be made on the Y- face of the main frame as well as on the payload structure. There is also a possibility to fix the batteries directly to the EPS board (TBC).

6.2.7 Solar Cells

The solar cells are fixed on the side panels of the satellite in sets of two to maintain a sufficient voltage to supply the power system. The solar cells units are assumed to be 3018 mm^2 and are equipped with a protective glass cover (see Figure 35). The cells are assumed to be 10,5 mm thick (TBC), including glass. The cells are glued on five of the six sides of the satellite. The sixth side is occupied by the payload, the access port and also the antennas.

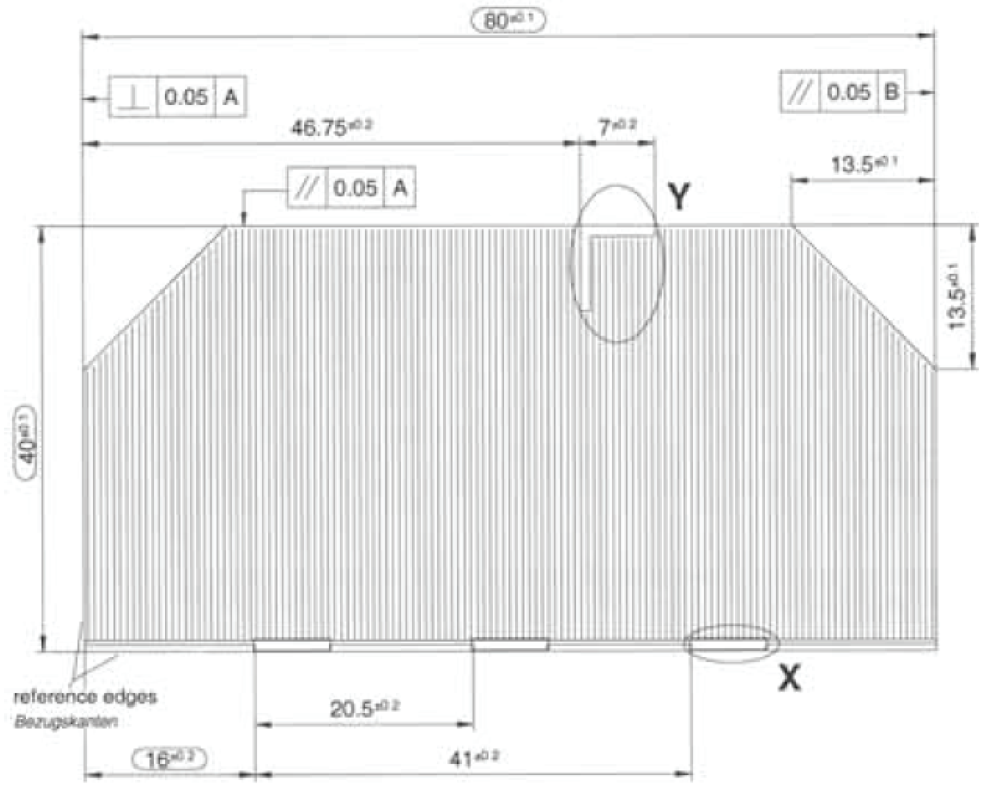

Figure 35 Drawing of the solar cell.

6.2.8 Kill Switch

One or two kill switches are implemented in the design, positioned in the feet of the rails. Two switches are used for additional safety since the system will be turned on when one only of the switches is released. These switches should physically switch off/on all power in the satellite, so when stacked in the P-POD, no error should cause a malicious early deployment of booms and antennas, and in the same time this conserves power for the early stages of the space mission. The choice and implementation of the kill switch is not yet decided; this should be done in the next phase of the project.

6.2.9 Remove before Flight Pin

Along with the kill switches there is also a requirement for a "remove before flight" pin, to disable the satellite before and during integration with the deployer. Once the satellites are loaded into the deployer, the remove-before-flight pin is removed. This piece is also yet to be decided.

6.2.10 Separation springs

The top of the rails are to be equipped with separation springs to enable the deployment from the P-POD. The springs suggested in the Specification Document are "Stubby Spring Plungers" part number SSMD-50 supplied by M.J. Vail Inc. (The plunger design and function can be further studied on the Internet at http://mjvail.com/vlier/vlierpage15.htm.) This spring is represented in Figure 36.To accommodate the plungers, a threaded hole must be included in the top ends of the rails.

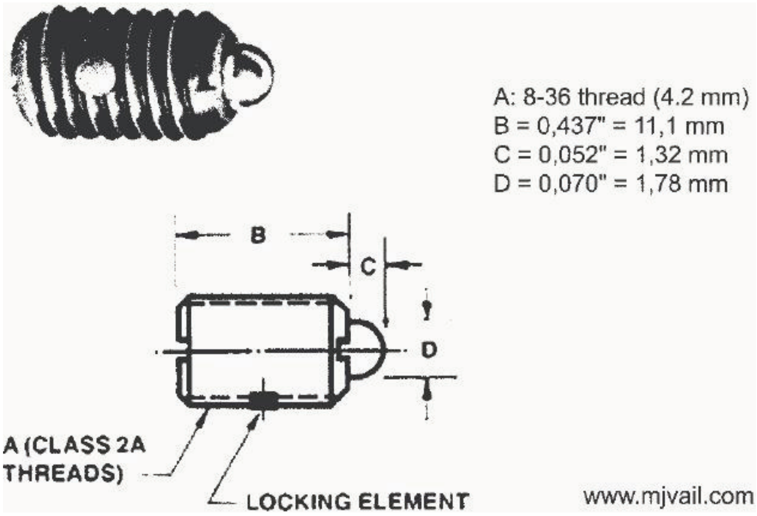

Figure 36 Suggested separation plunger design.

The plunger design and function can be further studied on the Internet at http://mjvail.com/vlier/vlierpage15.htm.

6.2.11 Spacecraft Harness

Thanks to the use of a motherboard, the number of necessary wires will be reduced to a minimum. There will however, remain some separate wires, in particular to connect the solar panels as well as the magnetic torquers. These points will be treated in more detail during the next semester.

6.3 Assembly Procedure

The assembly philosophy is as follows: the procedure assembly is separated into two major stages. On one hand the external structural elements are directly mounted on the main frame on the other hand the internal subsystems are assembled together and then fixed on the frame. This permits to combine the advantages of an inside/out architecture keeping an excellent accessibility to the electronic components and profit from the rigidity of a monoblock structure.

It is assumed that major assemblies of the panels are done in advance, meaning that the solar cells and the magnetic torquers are mounted on the panels. The assembly procedure is based on the allowed design space of each PCBs, which gives a relatively complicated assembly procedure, since the design space only allows small clearance during assembly. It is expected that that the actual assembly procedure will be less critical regarding clearance, since the real shape of the PCB's will not use the total design space available.

Here is a summary of the spacecraft assembly steps. During assembly the side panels and reaction wheel are first attached to the spacecraft (stages 1 to 3). Secondly the internal PCB's and the payload are inserted and connected through a motherboard (4-8). Connections are done either through the motherboard or the openings on Z- or Y- are used to connect any loose wires (magnetorquers, reaction wheel, etc.). The remaining side-panels are attached and connected (9-10).

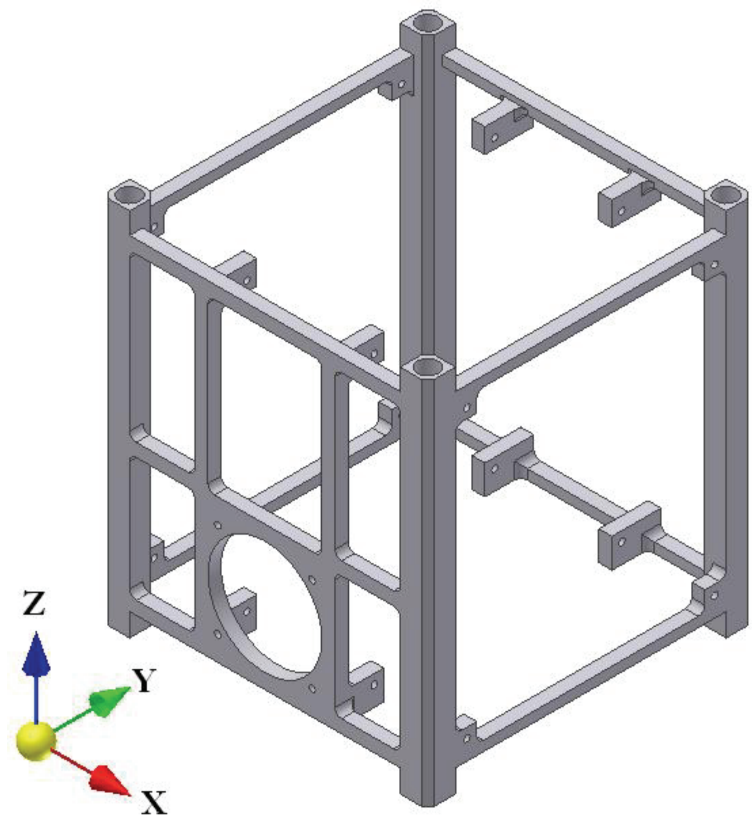

Figure 37 First stage of the assembly procedure.

The assembly procedure starts with the Monoblock (see Figure 37). Afterwards three side panels (X+, X-, Y+) are glued to the frame (see Figure 38).

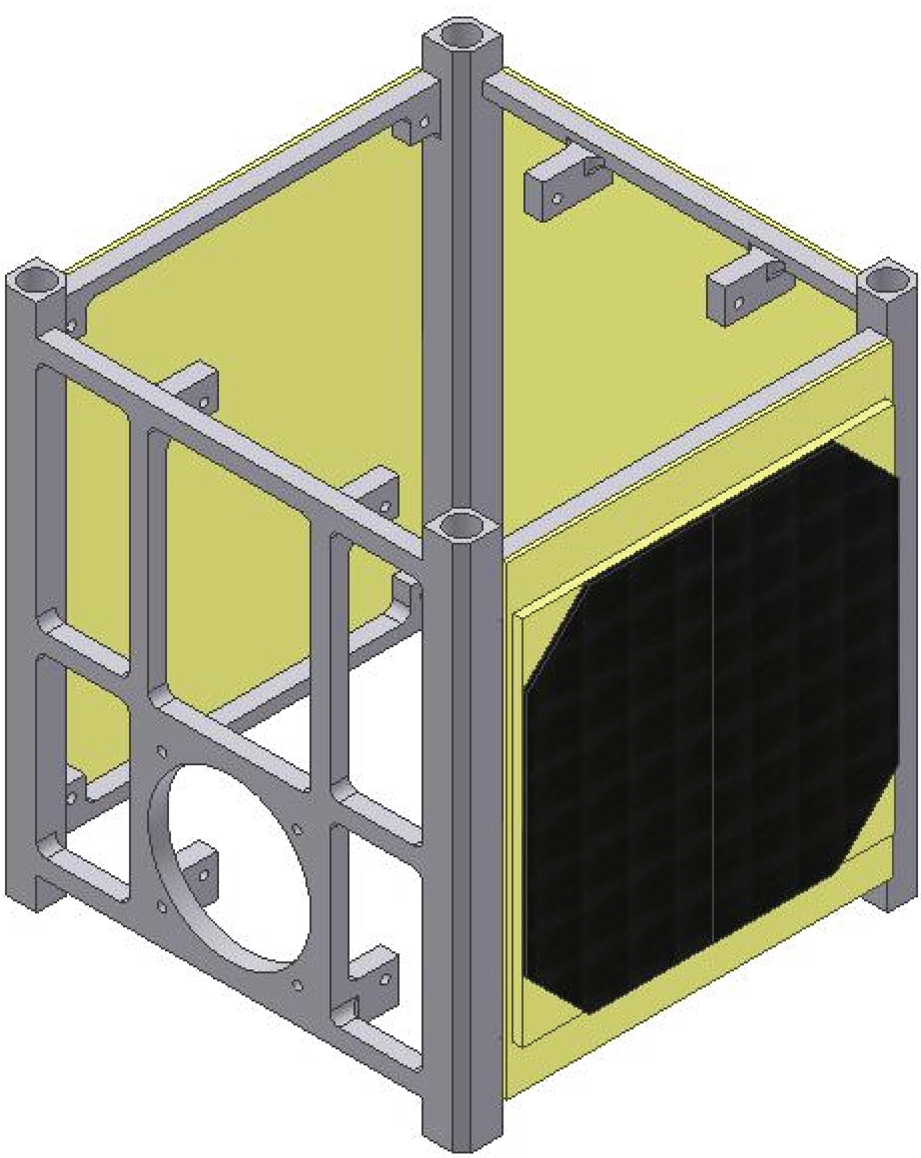

Figure 38 Second stage of the assembly procedure.

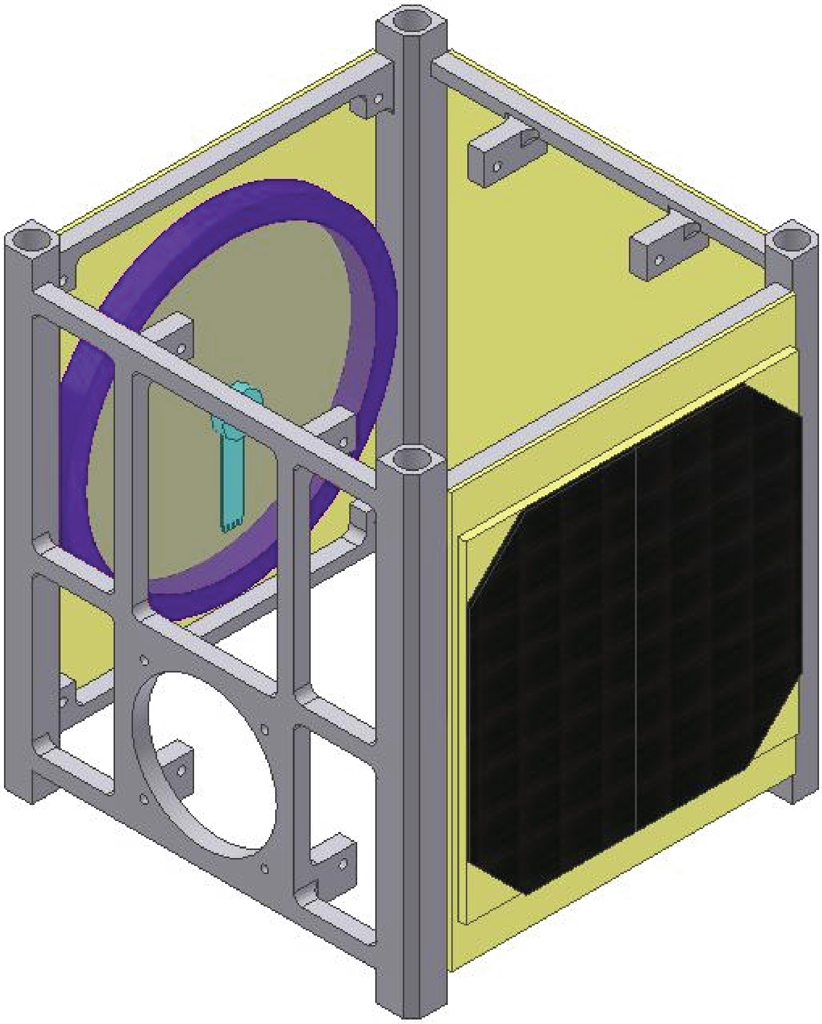

Figure 39 Third stage of the assembly procedure.

Then the inertial wheel subassembly is screwed to the internal X+ side of the frame (see Figure 39).

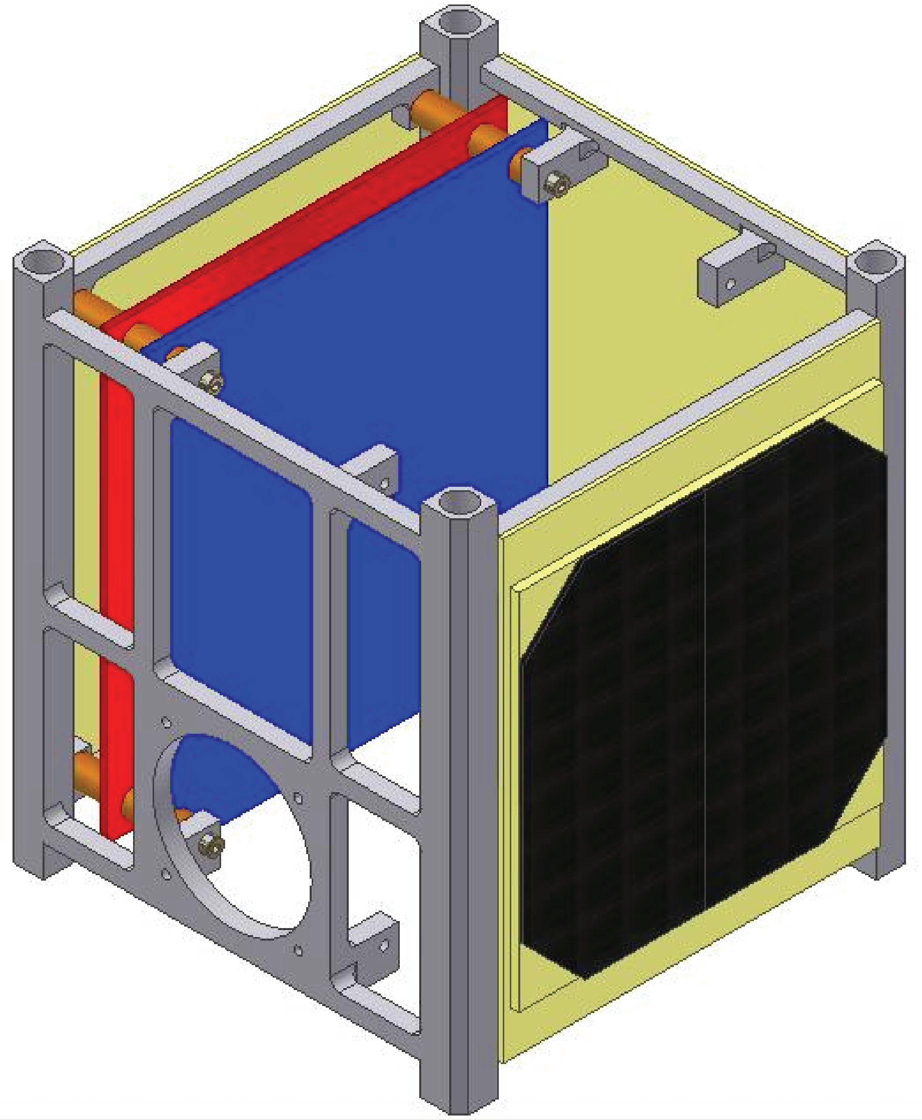

Figure 40 Fourth stage of the assembly procedure.

After that a first PCB's stack (ADCS in red and EPS in blue) is fixed inside the satellite frame on the X- side by M2 screws (see Figure 40). The inertial wheel is connected to the ADCS board and simultaneously the connections from the solar cells and magnetic torquers can be connected to the corresponding PCBs.

Then a second PCB's stack (COM in green and BEACON in yellow) is fixed the same way on the opposite side (X+) of the satellite frame (see Figure 41).

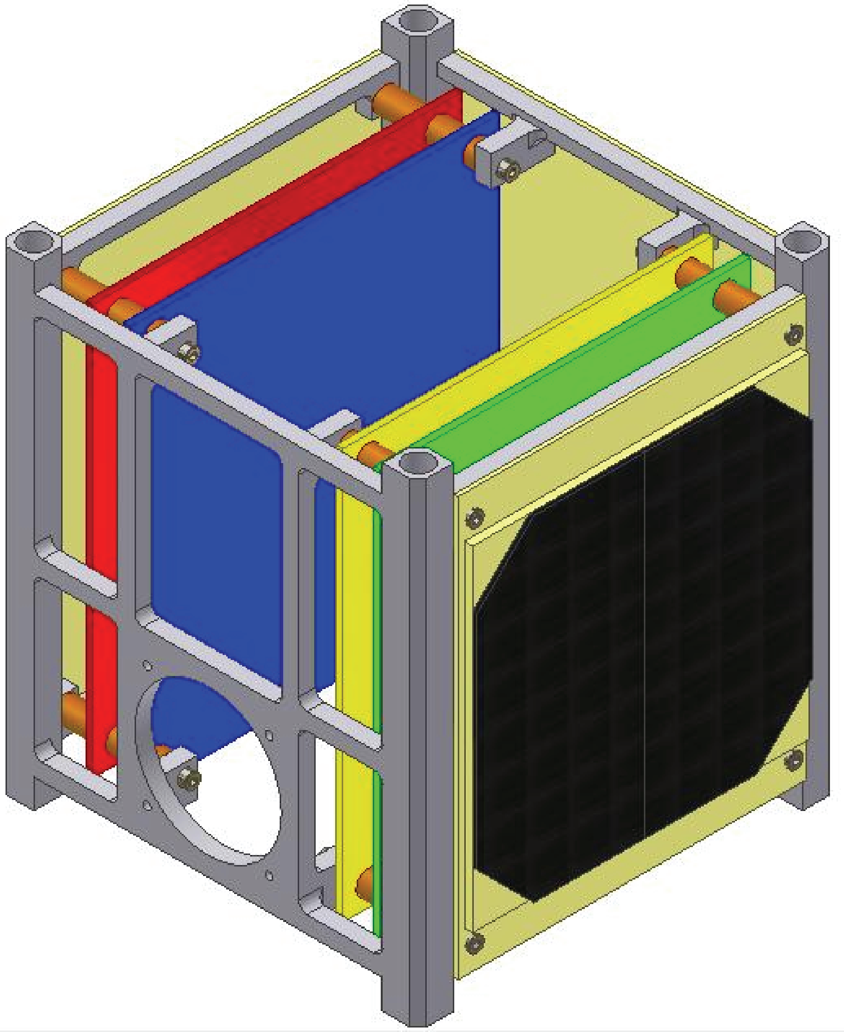

Figure 41 Fifth stage of the assembly procedure.

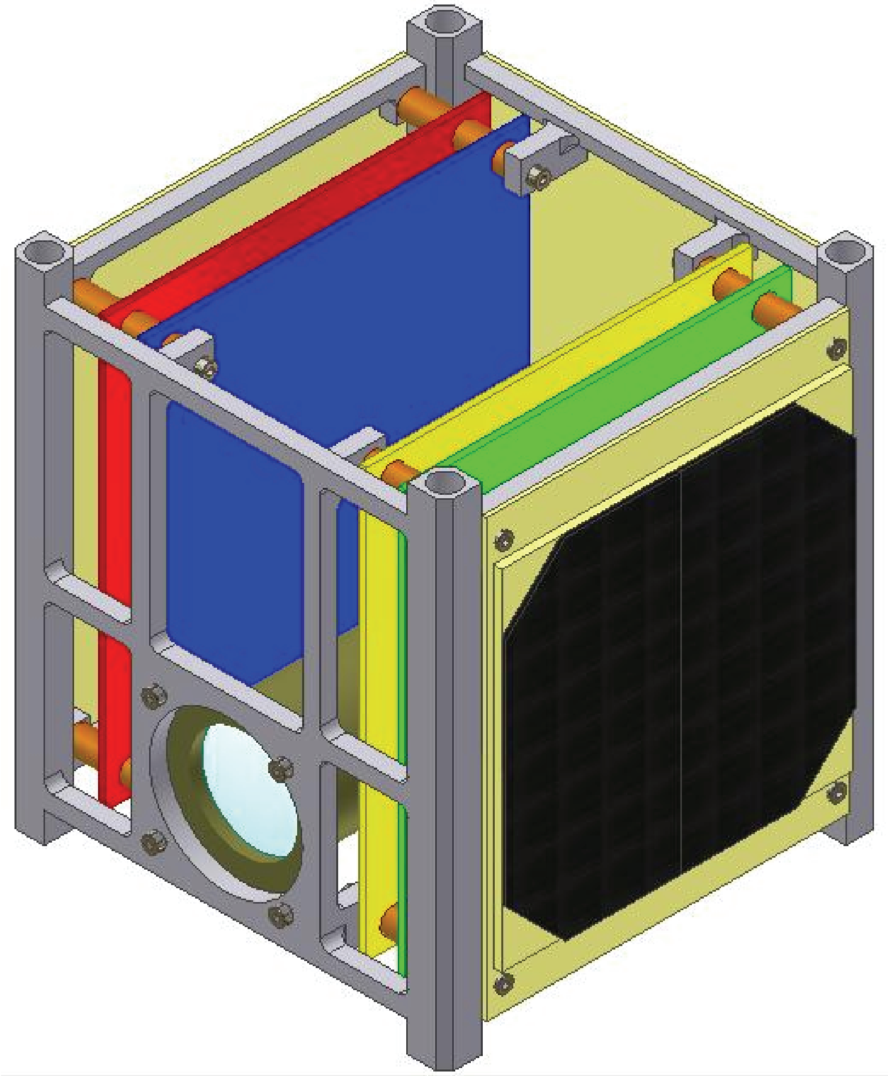

Figure 42 Sixth stage of the assembly procedure.

Afterwards the payload subsystem is introduced between the PCB's stacks and fixed on the Y- side of the frame by M2 screws (see Figure 42).

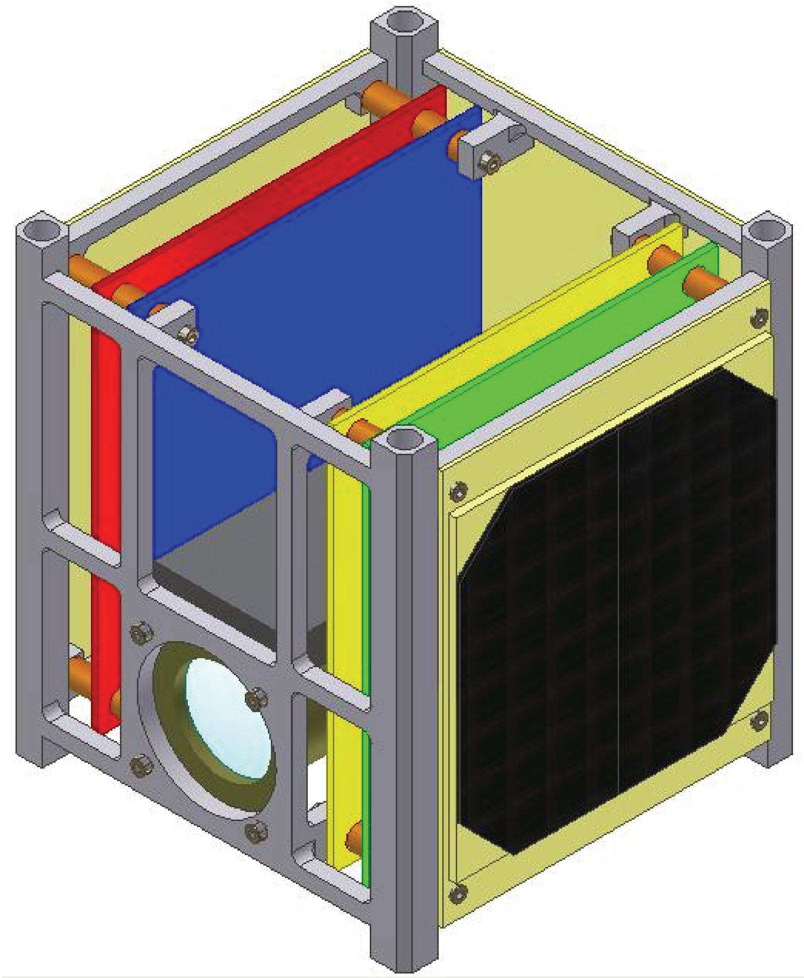

Figure 43 Seventh stage of the assembly procedure.

Batteries are inserted above the payload and probably fixed on the payload frame and at the same time on the Y- panel of the main frame (see Figure 43). The connections between batteries and EPS board can than be established.

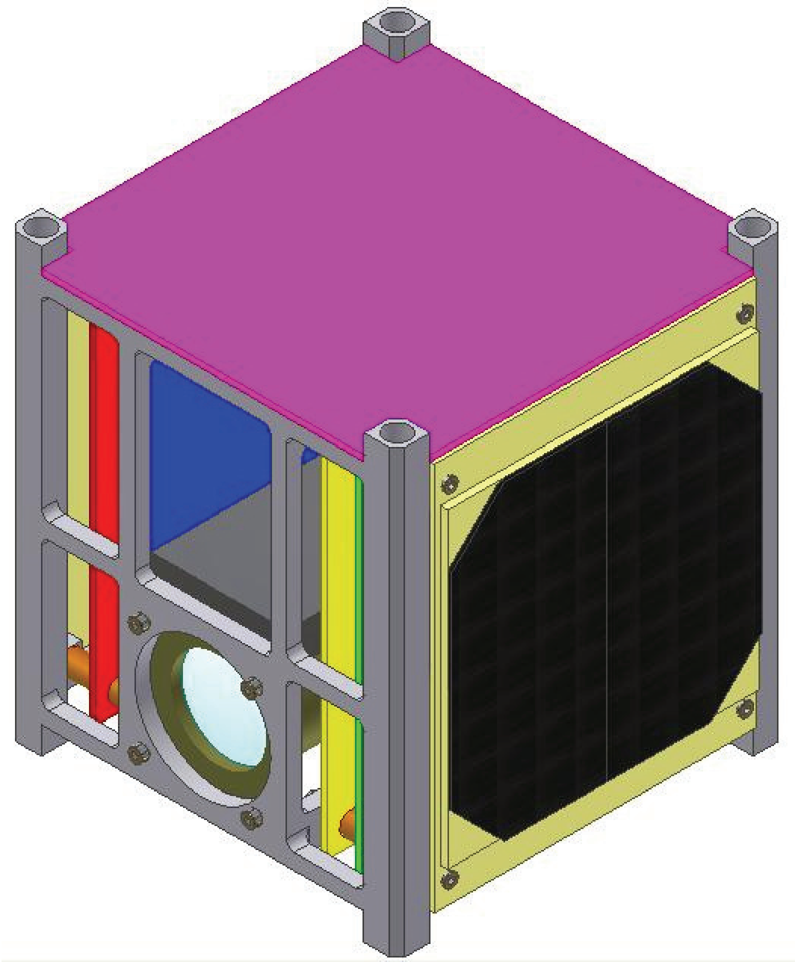

Figure 44 Eighth stage of the assembly procedure.

The motherboard (CDMS) is inserted from above and connections between all electronic subsystems are made (see Figure 44). The board is glued and also screwed on the main frame.

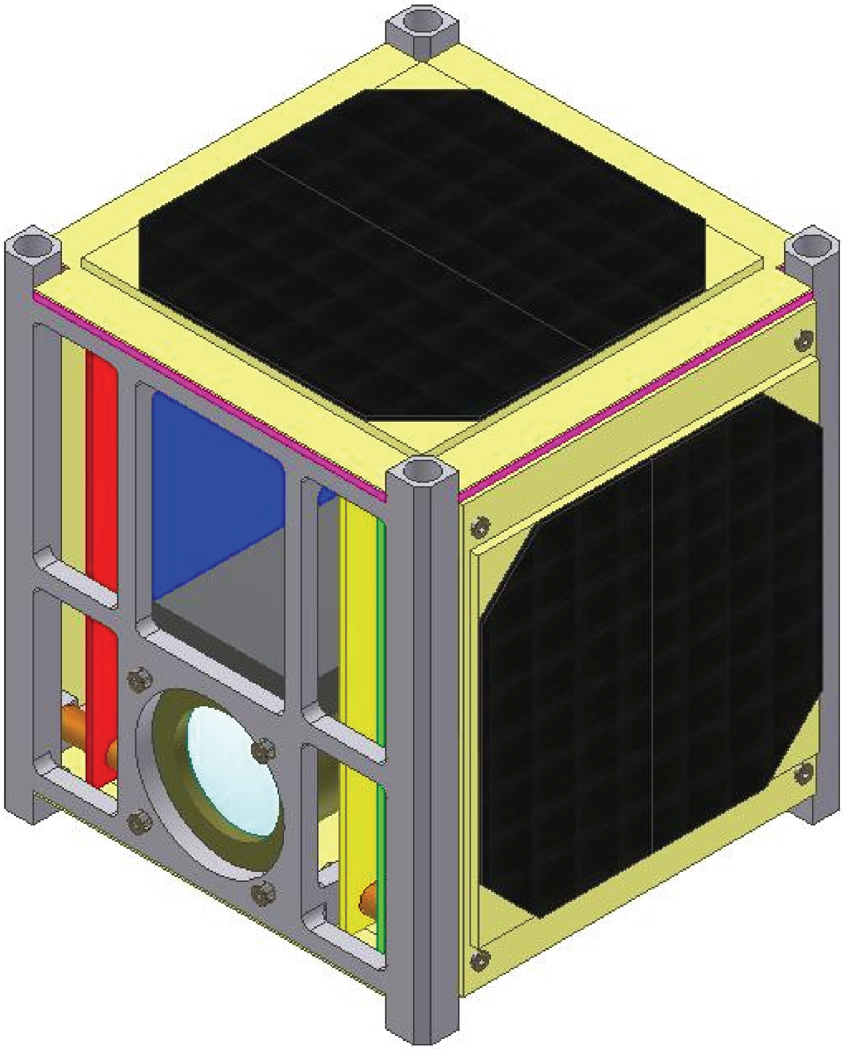

Figure 45 Ninth stage of the assembly procedure.

Afterwards the top and bottom panels are glued onto the motherboard and the wires from the solar cells and magnetic torquers can be connected to the corresponding PCBs (see Figure 45).

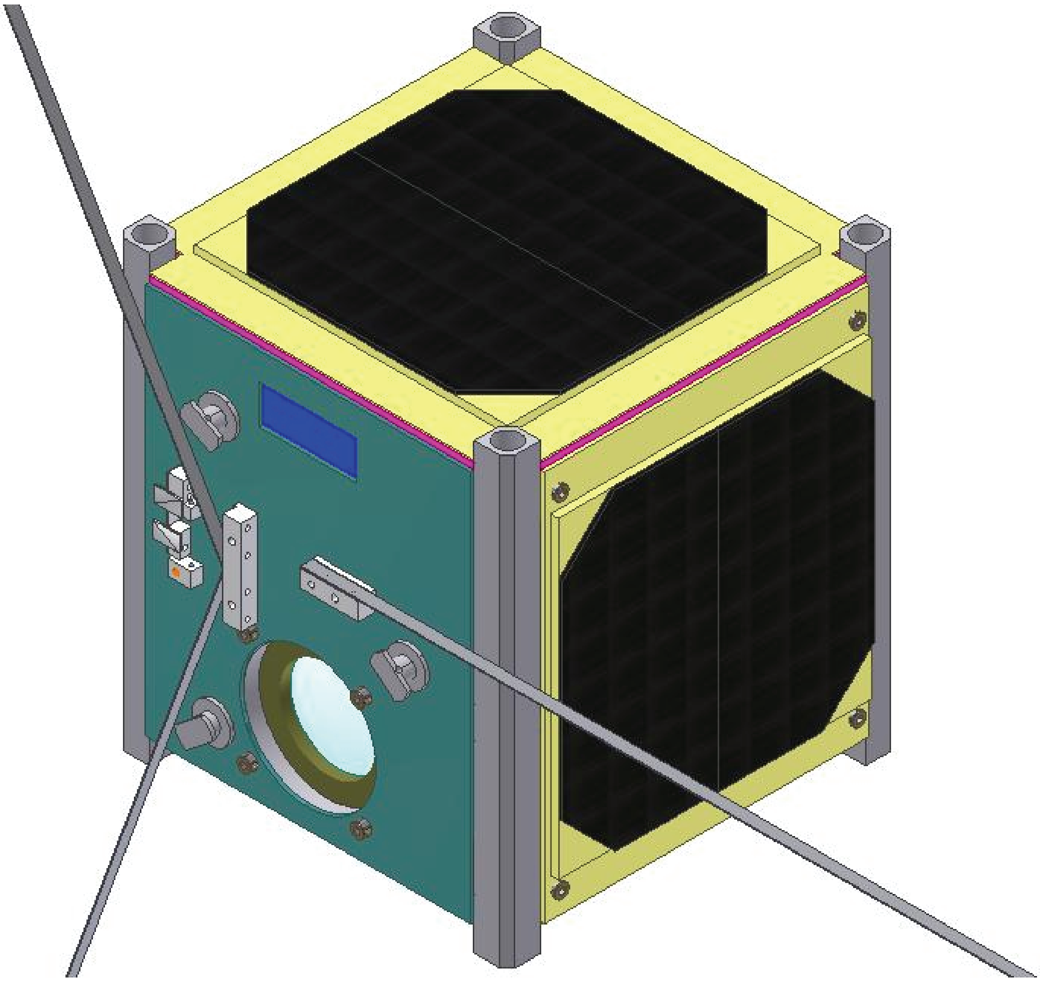

Figure 46 Tenth stage of the assembly procedure.

To finish the payload panel including the antennas deployment system is connected to the corresponding board and which complete the assembly procedure and closes the satellite (see Figure 46).

Due to the size and available space inside the satellite it is not possible to use a torque wrench for the mounting of the PCB's. Alternatively an Allen key and can be used.

To verify the indented assembly procedure a simple experiment can be done. Some replacement PCB's will be manufactured in correspondence with the designated design spaces for each PCB's. The boards will be placed inside the mock-up model of the frame according to the prescribed assembly procedure. The boards have to fit into the frame as intended.

6.4 Material Choices

For the main frame an aluminum alloy is used because the CubeSat standard specifies that the satellite must be constructed of a material with a similar thermal expansion coefficient to the materials used for the construction of the P-POD. The P-POD has been fabricated from the Aluminum alloy Al–7075–T73. We use the Aluminum alloy called Certal® (Al-7022-T651) with properties similar to Al–7075–T73 (see Appendix B - Material Properties). This alloy is not one of those recommended by ECCS Standards [15, 16], however its properties are considered good enough to use it as they are similar to Al-7075 and the CubeSat don't need a particular protection from corrosion.

Even if not adequate for the main structure of the satellite, the titanium will be used to make the structure of the payload because it has a coefficient of thermal expansion (CTE) similar to the glass which is used for the optics. Experience with titanium in space has met with good results [16], and its high strength will adequately protect the optics. The titanium alloy that will be used is not yet defined, but useful values for some alloys are found in Table 4 with the values of other metals used.

Table 4 Some metals properties

Material	Density [g/cm^3]	Tensile Strength [Mpa]		CTE, linear 20°C [µm/m-°C]	Thermal Conductivity [W/m-K]	Specific Heat Capacity [J/g-°C]
		Ultimate	Yield			
Certal®	2,76	540	460	23,6	120 - 150	-
Al-7075-T73	2,81	505	435	23,6	155	0,96
Germanium *	5,32	-	-	6,1	64	0,32
Titanium**	4,50	220	140	8,9	17	0,53
Timetal 35A (IMI 115) ** #	4,51	369	246	7,6	16	-
Ti 6Al 4V (IMI 318) ** #	4,42	924 - 1155	847 - 1078	7,9	6	-
Ti 4Al 4Mo - Si (IMI 550) ** #	4,60	1062 - 1210	970 - 1109	8,8	8	-

* Germanium (pur) is the material of the solar cells' substract

** Titanium (pur) or one of his alloys will be used for the payload's frame

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For the external panels of the CubeSat, a reinforced composite of carbon fibers in epoxy resin matrix will be probably used. Another possibility is the used of honeycombs between carbon panels (see Appendix B: Material Properties). More research is to do about these two alternatives. At the moment is not decided whether or not the panel where is the optic of the payload (payload panel) will also be manufactured from composites because it is not well defined if this material choice can generate a problem to fix the antenna deployment system. The choice is not yet final because we have yet to run a dynamic analysis of the deformations the panels will undergo, and also because we are still lacking information from the thermal subsystem team on the feasibility of carbon panels from a thermal perspective.

Table 5 Some useful properties of some reinforced composites

Material	Density [g/cm^3]	Tensile Strength [Mpa]		Tensile Modulus [Gpa]		CTE linear [µm/m-°C]		Thermal Conductivity [W/m-K]	Specific Heat Capacity [J/g-°C]
		at 0°	at 90°	at 0°	at 90°	Long.	Trans.		
Thornel® 1 *	1,4	110		15		14		6	1,2
Thornel® 2 **	1,4	103		13		11		10	1
Cycom® C69 #	1,63	930	33	195	8,2	-0,7	30	-	-
HexPly® 8552 UD ***	1,58	2207	81	141	10	-	-	-	-
HexPly® 8552 Woven ****	1,57	828	793	68	66	-	-	-	-

* Cytec Thornel®, Carbon Fiber (VCK or VCL) Carbon Cloth, Laminate Properties with Mil R-9299 Resin
** Cytec Thornel®, Graphite Cloth, Laminate Properties with Mil R-9299 Resin
*** Hexcel HexPly® 8552 UD Carbon Prepeg Epoxy Matrix, AS4 Fiber
**** Hexcel HexPly® 8552 Woven Carbon Prepeg Epoxy Matrix, AS4 Fiber
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The use of carbon fiber composite panels is motivated by the following advantages: a density lower than aluminum (approximately 40-50%, depending on the product), a good electromagnetic insulation compared to aluminum (the magnetic torquers must certainly be insulated for a correct operation), as well as a coefficient of thermal expansion that lies between that of the solar cells and the main aluminum frame (see Table 4).

The reasons which make us hesitate to choose a composite material for the panels are its low thermal conductivity as well as its lower capacity to absorb of radiations compared to aluminum (see Appendix B - Material Properties). The addition of aluminum foil (or another good conducting metal) on the faces interior and external of the panels might solve these two problems.

6.5 Fastening Preferences

For the various pieces of the structure like the main frame and the crossbars, screws will be used as fastening method. The main reason for this choice is that parts can be disassembled and reassembled numerous times. For the final assembly, screws have to be glued in order to prevent the risk of disassembly during launch due to vibrations.

Concerning the exterior panels, adhesives seem to be the best solution, because they provide a weak stress concentration on the interface and can be used to join different kinds of materials, like carbon reinforced composites and aluminum in our case.

Most often used structural adhesives are either two component epoxy pastes or adhesive films. Since in typically encountered applications for the SwissCube satellite only small surfaces are to be glued, the preferable solution will be two component epoxies rather than adhesive films. A list of possible structural adhesives as well as a possible thermal adhesive for bonding of the solar cells is given in Appendix C – Adhesives properties

All of these adhesives are either recommended for space use by the ECSS of NASA or specifically designed for space applications by the producers and thus present low outgassing coefficients. The selection of the optimal adhesive will be based on different parameters which are yet to be defined. First of all, thermal properties have to be suitable for an optimal heat transfer between the different structural parts and the solar cells and the adhesive has to be applicable in the temperature range yet to be defined by the thermal subsystem. Additionally, the cure temperature of the epoxy resin should be as low as possible, in order to allow fixing of the side plates without endangering any of the electronic components. A reasonable cure temperature is 80°C since the electronic components will have to be designed for these temperatures in order to pass thermal vacuum testing. In addition to this, the adhesives' physical properties such as the shear strength need to be as high as possible, in order to resist stresses induced by the CTE mismatch between the aluminum frame and the composite plates.

The preferred choice for the adhesive is ScotchWeld 2216 since it has a vast space experience and since its coefficient of thermal expansion in the defined temperature range compared to other epoxies. Additionally cure temperature is rather small (65°C). Unfortunately, the margin of security calculated is rater small (MOS = 59%, Appendix D - Static Analysis). This is certainly due to the conservative model but nevertheless thermal testing of the adhesive will be unavoidable. Another option to reduce thermal stress would be to reduce cure temperature at the cost of a higher curing time. Another solution could be silicon based glues with low elasticity and shear moduli. Nevertheless, special considerations for outgassing properties need to made for these kinds of glues.

7 BASELINE PROPERTIES

After having described our baseline, we are interested in the various examinations of this baseline, for example its inertial or physical properties as its static and dynamic analysis.

7.1 Physical and inertial Properties of the Baseline

A mass analysis is performed on the complete structural model using Autodesk Inventor and the information received from this (centre of mass location, total mass, moments of inertia etc.) is given below. Using this preliminary configuration, with the estimated masses and dimensions supplied by the other team members it is shown that the SwissCube will meet with the CubeSat specification with regard to the location of its centre of mass.

7.1.1 Total mass

The mass budget is based on preliminary subsystem mass estimates and is still missing information. The maximum mass of the SwissCube cannot exceed one kilogram. Thus it is critical that the approximate mass of the satellite is known and updated throughout its development to ensure it stays within the weight restriction. The current mass budget, as of June 2006, is given in Table 6. For the other subsystems, the detailed mass budget can be found in the System Engineering Report of Bastien Despont.

Table 6 Mass budget of the various faces and boards.

Harness**Boards / Faces**	**Mass [g]**
EPS board	146
ADCS board	60
CDMS board	94
Payload module	92
COM board	105
Beacon boarrd	50
General structure	195.45
face +x	21.8
face -x	40.1
face +y	37.8
face -y	13.6
face +z	37.8
face -z	19.5
Total	913.05

The mass given for the frame currently includes all structural parts as well as the motherboard, the connectors and the wirings. More modifications can be made to the panels, such as cutouts and holes, in order to lower their mass. The payload mass is a rough estimate given by the science team. Equally, the wiring for the entire satellite is just an estimate.

Currently, the SwissCube is within the 1 kg limit with a margin of 87 g. Though the CubeSat will most likely be heavier when finished, the 87 g provides a margin to work within.

7.1.2 Center of Mass and moments of Inertia

The center of mass and inertial properties are calculated from the center of reference frame, which is in the geometrical center of the cube (see Figure 47).

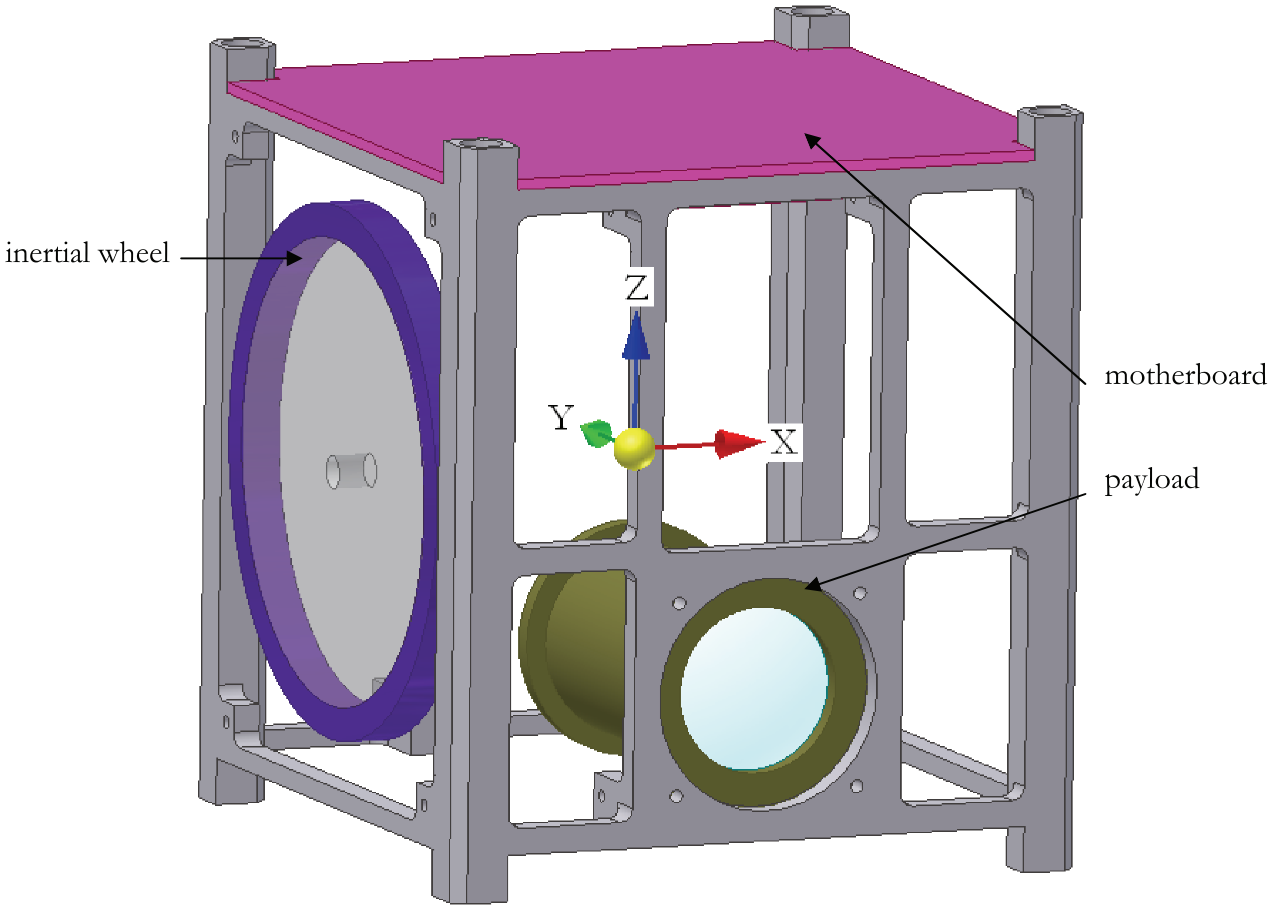


Figure 47 Reference frame

Table 7 and Table 8 give the various physical and inertial properties of the SwissCube. These values come from the CAD software, AutoDesk Inventor.

Table 7 General properties.

Mass	0.913 kg
Centre of mass	**(mm)**
X_c	-2.144
Y_c	-2.628
Z_c	1.477

Table 8 Inertial properties.

Physical moments of inertia	$(kg \cdot mm^2)$
I_{xx}	1814
I_{yy}	2520
I_{zz}	2328
Principal moments of inertia	$(kg \cdot mm^2)$
I_1	1805
I_2	2547
I_3	2310
Rotation XYZ/principal	(deg)
R_x	16.18
R_y	0.46
R_z	6.32

These numbers will change slightly when the final masses are known. They should be recalculated to ensure that the satellite remains compatible with the CubeSat requirements. As we can see, the centre of mass (C.o.M.) is largely within the specifications (The C.o.M. must be in 20 mm, in our case: 3.15 mm). Concerning the inertial values, the ADCS team is also satisfied, because the inertia of the payload axis is high (compared to the other axes) and thus certain stability or "inertia" for this axis is guaranteed. Moreover, the rotation of 16 degree around the X-axis (speed axis) is very important for the science subsystem because the payload must have a inclination between 17 and 27 degree around this axis to be able to take picture of the Nightglow (see Figure 48). If this rotation can be done thanks to the inertial characteristic of the satellite, this makes things easier for the ADCS team.

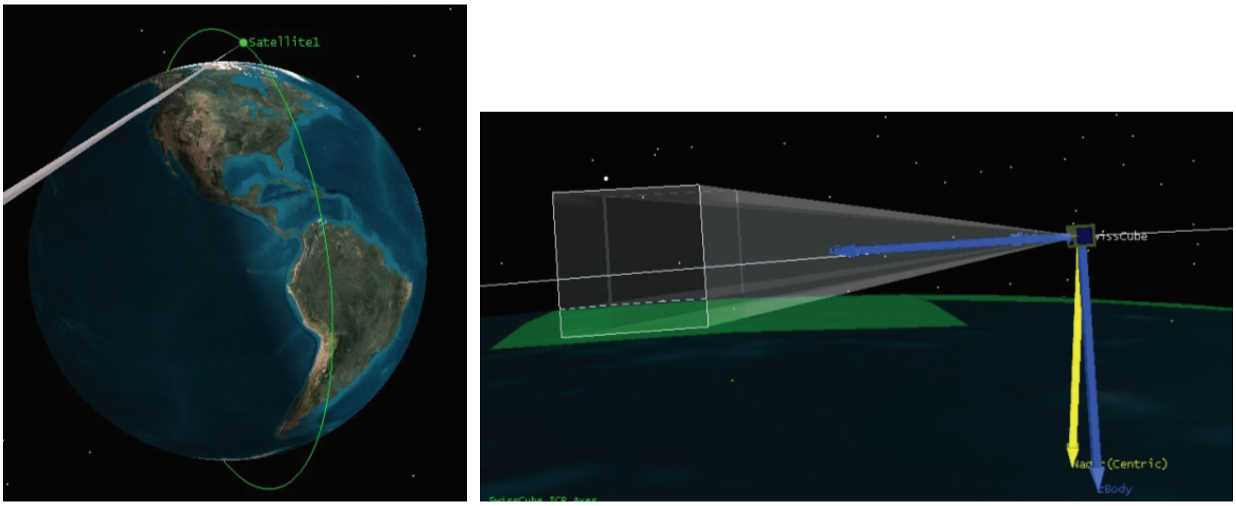


Figure 48 Orientation of the SwissCube.

7.2 Preliminary Static Analysis

The structure must be sufficiently rigid to withstand all static loads encountered during the manufacturing, transportation and operational life of the satellite. As a consequence the satellite should be designed to withstand the highest potential loads encountered during its lifespan. This is known as designing for the worst case. By ensuring that the satellite will not fail under worst case static loading conditions, it can be shown that the satellite will not fail under any static loads during its lifecycle.

7.2.1 Worst Case Load

It is perceived that the worst-case static loading will be experienced by the satellite during the launch sequence. For the worst case loading consider the arrangement of CubeSats with in the P-POD shown below.

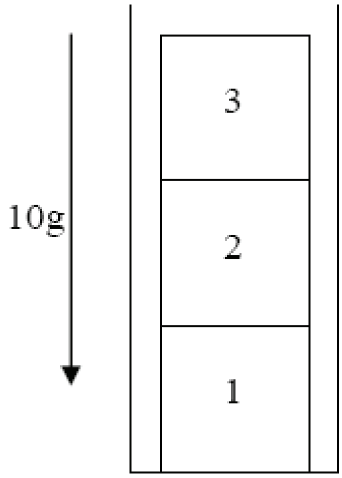


Figure 49 Layout of CubeSats in P-POD during launch.

For the worst case we shall consider the P-POD in a direction parallel to the direction of maximum acceleration during launch. As the deployment system holds three CubeSats the worst case will be experienced by the CubeSat in location 1 of Figure 49. During the launch sequence this CubeSat must maintain structural integrity while supporting not only its own weight but the weight of the two overlying picosatellites. Using the CubeSat design specifications the following assumptions can be made regarding the worst case static load of the SwissCube:

- The maximum acceleration will be equivalent to 7.5 g (Dnepr maximum acceleration)
- The mass of each of the three satellites is equal to 1kg

With a factor of safety of 1.25, the acceleration is 9.375 g, so it can be rounded to 10 g.

Therefore the worst-case loads on our satellite are simplified to those shown in Figure 50. The SwissCube will need to be able to tolerate a loading equivalent to an axial force of 196.2N with an acceleration of 10 g. This will be known as the worst case axial loading condition.

Now consider a case when the P-POD is aligned in a direction perpendicular to the direction of maximum acceleration. This is represented in Figure 51. In this case all three satellites will experience the same loading condition irrelevant of their location with in the P-POD. Assume that the P-POD does not transfer any forces into the CubeSats. Then the CubeSats will only be required to support their self weight in a 10 g. gravity field. This will be known as the worst case lateral loading condition.

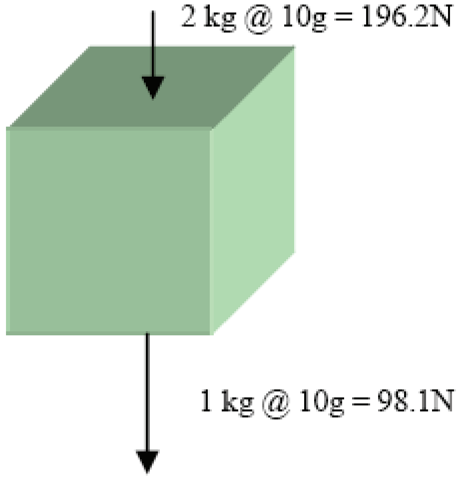


Figure 50 Worst case axial loading.

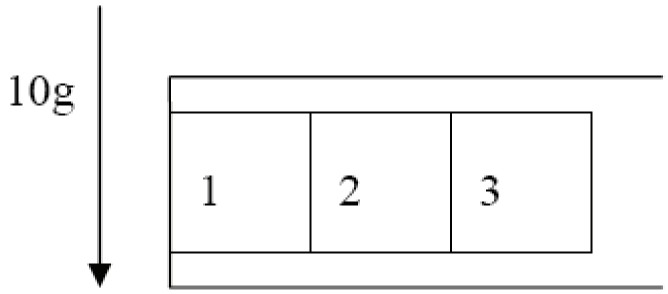


Figure 51 Layout of CubeSats in P-POD during launch.

7.3 Finite Element Analysis

The finite element method is a powerful mathematical tool used for the numerical solution of a wide range of engineering problems. In this case finite element analysis was used to estimate the deformations and stresses that the SwissCube will experience under a variety of different loads and freedom cases.

Finite Element Analysis (FEA) uses a complex system of points called nodes which make a grid called a mesh. This mesh represents the geometry of the structure and can be programmed to contain the material and structural properties which define how the structure will react to certain loading conditions.

Due to the complexity of the design and the abundance of components inside the CubeSat, it would be difficult and unnecessary to model everything inside of SwissCube. Instead, the design is simplified to represent only the basic structural components of the satellite that will be load bearing. Figure 52 shows the simplified version used in our analysis. The satellite is reduced to just five parts: the monoblock main frame, the eight spacers, the four rectangular PCBs, the motherboard, and the payload frame.

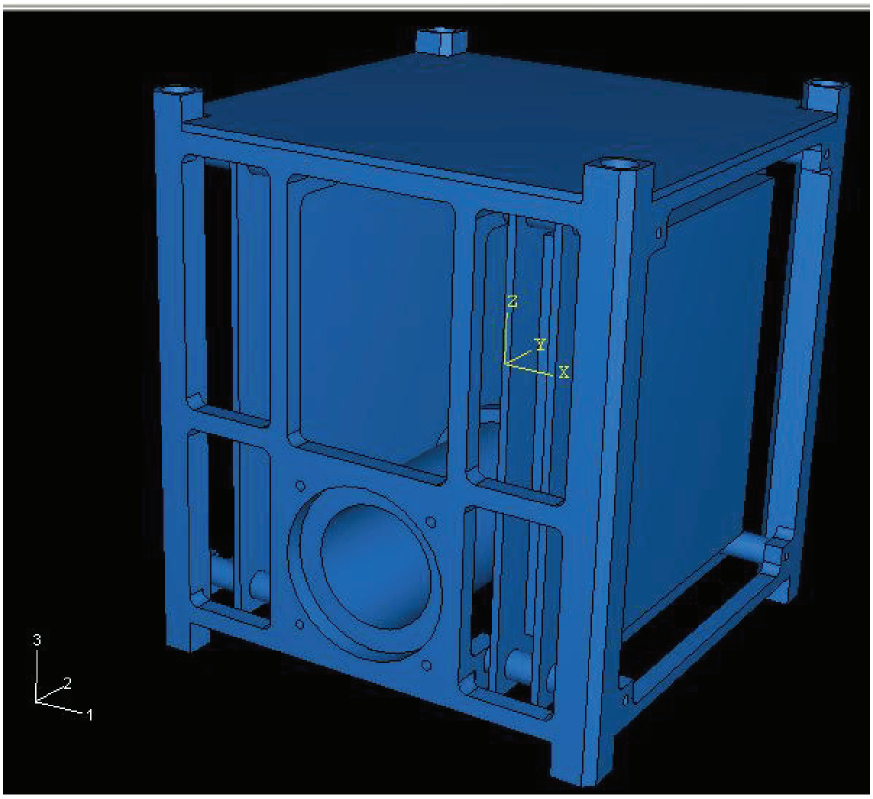


Figure 52 Simplified version used for FEA.

For simplicity, the joints between all the pieces are assumed to be tied so that there is no relative motion between them. In reality, the pieces will be bolted or glued together resulting in extremely rigid unions validating this assumption. The properties of the three different materials, aluminum Certal for the main frame, FR4 for the PCBs, titanium for the payload's frame, can be found in Appendix B - Material Properties).

Furthermore, it is still unknown whether the satellite will be launched horizontally, as illustrated in Figure 51, or vertically, as shown by Figure 49. This fact necessitated two separate finite element analyses. The force applied to each structure was a constant gravitational load of ten times the acceleration on the surface of the Earth and in the case of a vertical launch, an additional axial load of 196.2 N resulting from the two overlying picosatellites is present, so the load on each top of feet is one quarter, 49 N.

In the vertical case, the standard boundary conditions are that the four bottom feet are fixed in the z-direction and pressure loads from the two overlying CubeSats evenly distributed on the four top feet. In the horizontal case, the standard boundary condition is only that the two bottom rails are fixed in one direction. The models are run assuming a linear elastic response, and use NUMBER of TYPE elements

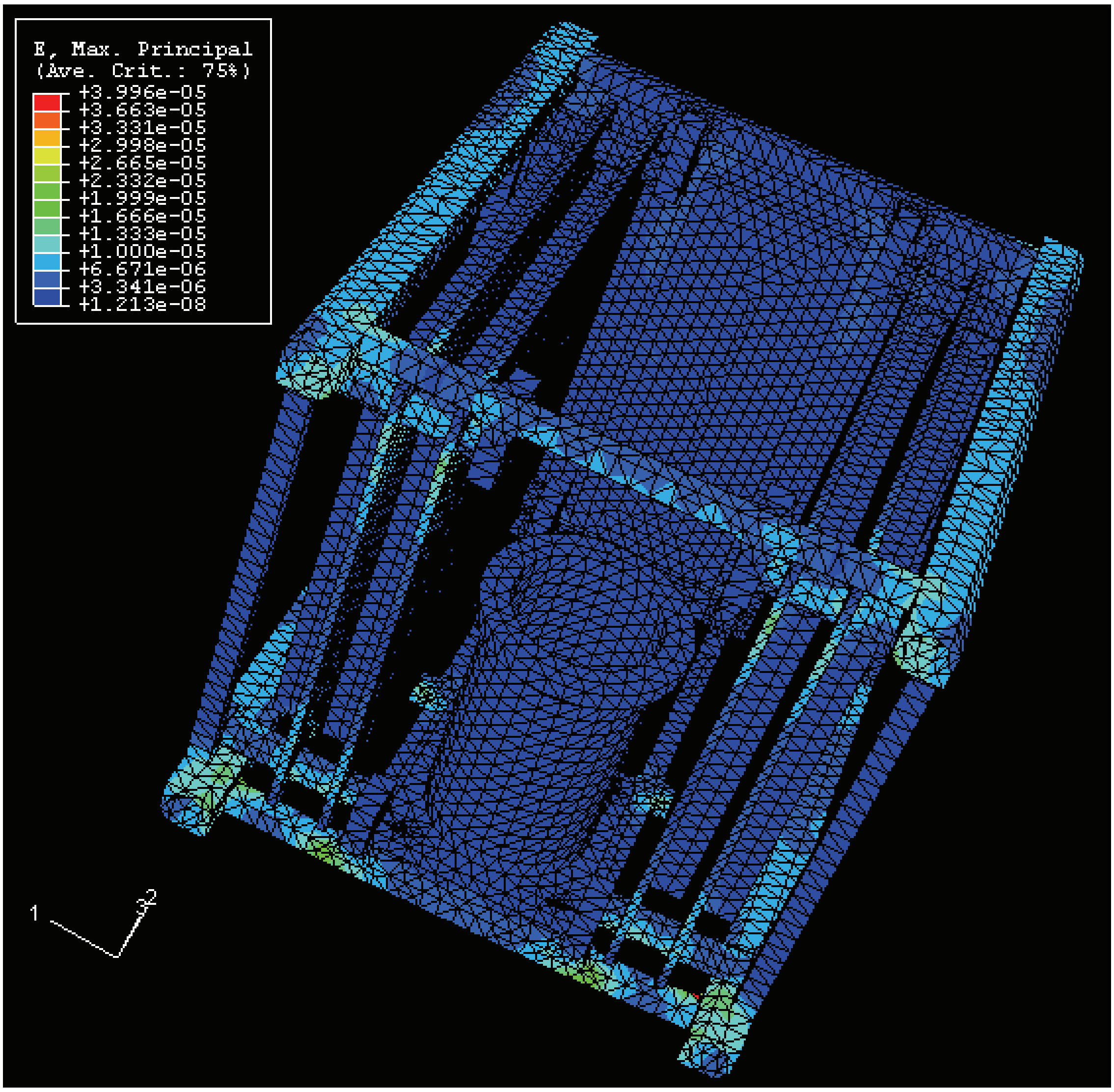


Figure 53 Strain in the vertical worst case.

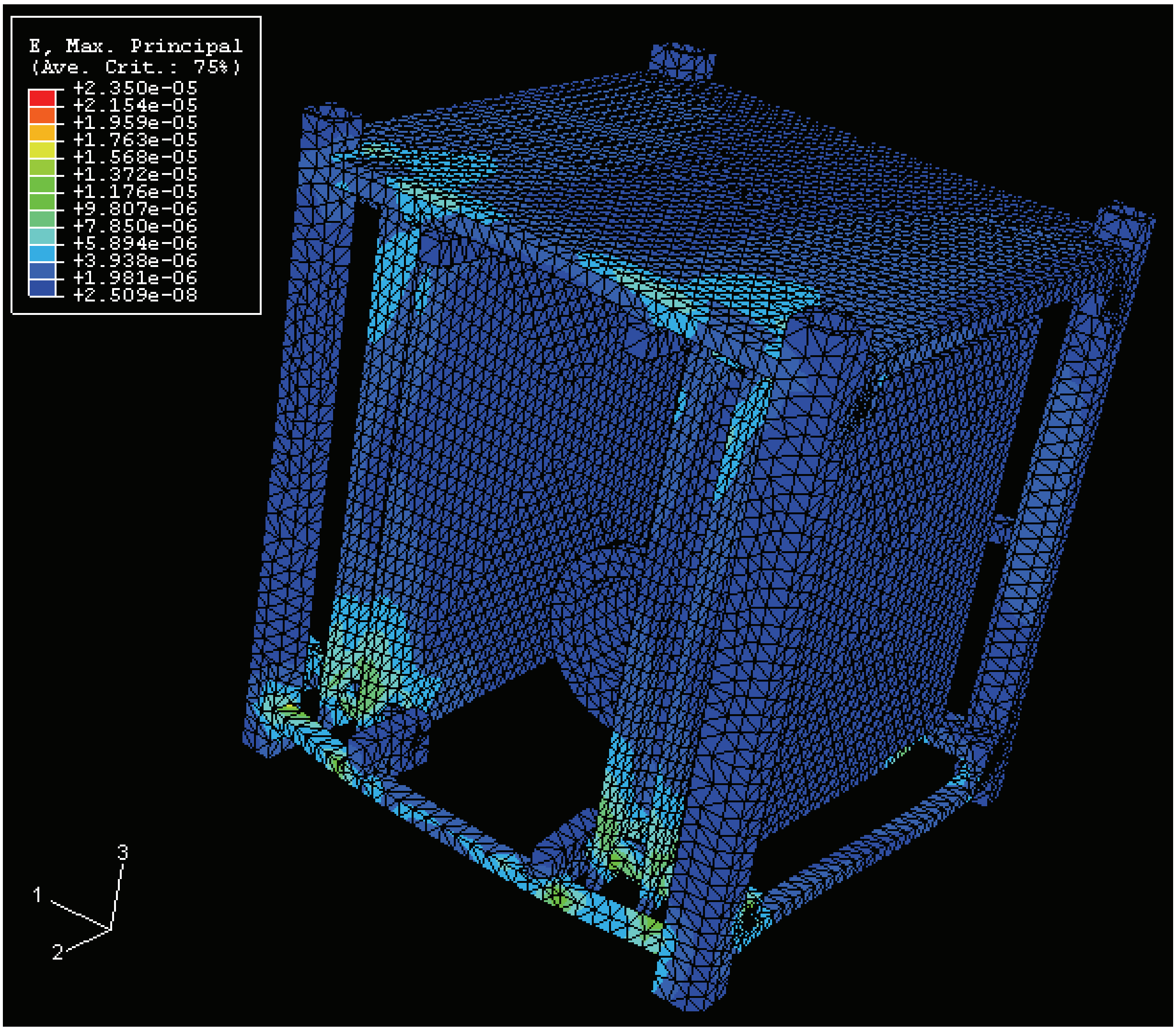


Figure 54 Strain in the horizontal worst case.

These scenarios probably overestimate the loads the spacecraft will experience during launch, because of the factor of safety. Nonetheless, the structure resisted without difficulties. We found that the maximum stresses in the structure in the both cases are well below the yield strength of the aluminum. For the horizontal case, the constraint computed by the Von Mises criteria is 1.9 MPa and 2.85 MPa in the vertical case (see Appendix E - Finite Elements Analysis). With a yield stress of 400 MPa for the aluminum and the formula for the margin of safety (MOS)[17]:

$$MOS = \frac{(allowable\ load)}{(applied\ load) \times FOS} - 1$$

Equation 1

The MOS in the horizontal case is 167 and 111 in the vertical case.This means that the satellite can support a load 167 and 111 times higher before entering the plastic region.

The maximum strains are $2.35 \cdot 10^{-5}$ mm and $3.99 \cdot 10^{-5}$ mm in the horizontal and vertical cases respectively. These minuscule deformations will have no impact on the structural integrity of the satellite.

Since this structure is basically constructed of thin beams, one final check is made to ensure that the reduced-section cross-bars and rails will not fail in buckling. Critical stresses of buckling for the rails and crossbars have been calculated as follows: σ_{cr} is equal to 716.4MPa and 137.5MPa for the rail and crossbar respectively. For the detailed calculations see Appendix D - Static Analysis. With a maximum stress of 1.3 MPa in the horizontal case for the rails and crossbars (see Appendix E - Finite Elements Analysis) and the formula for the MOS with a FOS of 1.25, the margins are 440 and 83 for the rails and crossbars in the horizontal case. For the vertical case, the maximum stresses are 1MPa and 2.85 MPa for the crossbars and rails respectively, so the MOS are 200 and 109 for the rails and crossbars in the vertical case.

Figure 53 and Figure 54 represent the deformation with various colors, but Abaqus allows a variety of other parameters to be displayed in a comparable format. For example, both stress and displacement can be similarly displayed. For these analyses, see Appendix D - Static Analysis. In the future, the finite element solver may be used to model various thermal loads.

7.4 Dynamic Analysis

It is important to verify that the structure will maintain structural integrity, and provide a suitable environment for the various subsystems under the dynamic loading conditions that are expected to be experienced during launch. These loading conditions can be found in.

The types of dynamic analysis that have been performed include:

- Harmonic vibration analysis
- Random vibration analysis
- Acoustic vibration analysis

The majority of the dynamic analysis will be performed using finite element analysis (FEA) Abaqus software. Unfortunately, only the static analysis could be performed this semester and the dynamic analyses are scheduled for the next semester.

7.5 Testing

To meet with the requirements of the CubeSat standard the completed SwissCube engineering model must be subjected to, and pass, both a vibration test and a thermal vacuum test.

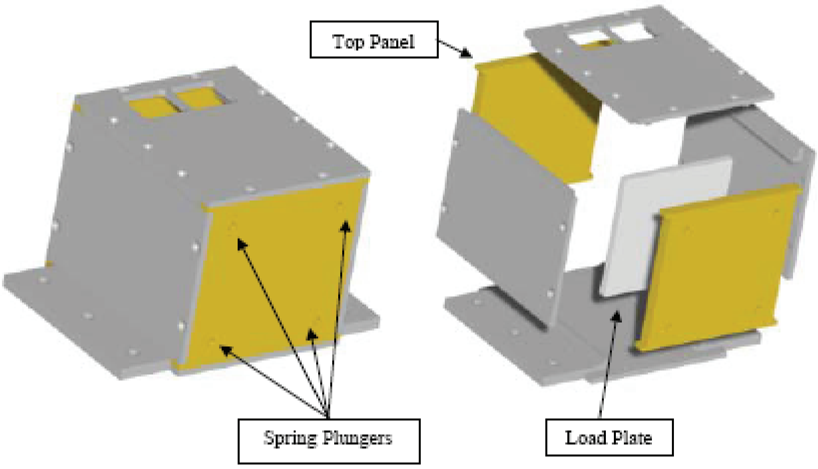


Figure 55 View of a Test POD.

The Test Pod [10] in Figure 55 is to be used by CubeSat developers as an environmental simulation of the P-Pod deployer, this will allow validation of the structural integrity of CubeSats under launch loads. The Test Pod interior is designed to simulate the environment inside the P-Pod deployer. The Test Pod allows CubeSat developers to test their satellites to the environment inside the P-POD deployer rather than designing to the launch vehicle loads.

Vibration testing will be done at California Polytechnic and will include a sine sweep test followed by a random vibration test on all three axes.

- The sine sweep test will range from 50-2000Hz for a period of three minutes. The sweep mode is logarithmic and will induce a maximum test level of $10G_{rms}$.
- The random vibration test will last for sixty seconds with a maximum test level of 10 G_{rms}.
- Procedures for conducting various vibration tests can be found at www.cubesat.calpoly.com

California Polytechnic will also run two complete thermal cycles. In the thermal vacuum test the integrated P-Pod will experience a high vacuum of $5x10^{-5}$ torr with a temperature range from -20 to 60 degrees Celsius [10]. This temperature range is subject to variation depending on the payloads of the CubeSats in the P-Pod. At each temperature extreme the CubeSats will soak for one hour, while allowing thirty minutes for ramping from one extreme to the other giving the thermal vacuum test a duration of six hours.

A complete test of all assembled structures will only be possible at a very late point of time in the development process. Therefore separate tests of the different structural elements must be carried out in earlier stages of the development, to prevent severe problems in the last phase before delivery.

The tests that are going to be performed by the structural subsystem are the following:

- Vibration testing
- Interface testing (adhesives)
- Solar cells mechanical testing

In addition to the vibration testing performed by CalPoly, vibration tests will be performed on the individual structural parts by the means of a shaker in order to determine the different modes of vibration.

Interface testing has to be achieved in order to determine the performance of the selected epoxy adhesive. This will include thermal cycling to temperatures below as well as above the expected temperature range to check resistance of the bond due to the CTE difference. Additionally, thermal cycling has to be performed to check fatigue performance of the bond.

Finally, tests of the mechanical resistance of the solar cells are going to be performed in order to determine their bending resistance. This will permit to determine maximal allowable bending / vibration amplitude of the composite panels during launch that will not endanger the solar cells.

8 CONCLUSION AND FUTURE WORK

Phase A in the design of the SwissCube structure and configuration consisted of establishing a baseline design of the structural elements as well as a definition of a baseline internal configuration. This involved the creation of a list of requirements important for the structural subsystem and a definition of the design approach. Determined by the launch vehicle constraints as well as the functional demands different structural trade-offs were established. Based on designs assumptions for the other subsystems, possibilities of internal configurations specific to each of the structural trade-offs were generated and their feasibility was examined. Finally, a selection of the baseline design was made, based on the optimization of the advantages resulting from the design principles of using the fewest possible parts as well as keeping maximal flexibility.

8.1 Recommendations for future study

During the design process it has become evident that there are certain areas that justify further consideration. Some aspects of the project that are worthy of further investigation are listed below.

- As yet, a completed internal layout of the satellite has not been established. When all other systems have arrived at a final design a final mass budget and internal layout configuration should be developed
- With more time it would have been valuable to conduct a more detailed structural analysis. In particular the response of the system to dynamic loading conditions is extremely important.
- The effect of the vibration of the faceplates on the operation and effectiveness of the solar cells should be established. This means ensuring that the solar cells will not be damaged during the launch of the satellite, and will operate efficiently while in orbit.
- Additionally, feasibility of the use of carbon fiber reinforced composites as side plates has to be further investigated in collaboration with the thermal subsystem.
- Mechanical interfaces have to be clearly defined for each part and a schemtic of the electrical connections through the motherboard needs to be established.
- Once the design and analysis of the structural subsystem has been completed (along with the other subsystems) the fabrication and testing of the engineering model should be undertaken. The testing will involve a thermal vacuum test as well as a vibrational test; both inside a test deployment pod.
- The design of the structural subsystem does not yet include the integration of the separation springs and deployment detection switches. As these components are both requirements of the CubeSat specifications it is important that adequate design solutions to these problems will be developed.

- Furthermore, an assembly protocol for the existing configuration needs to be established in more details and accessibility and maintainability of the different components have to be verified.

ACKNOWLEDGEMENTS

Although this project is the work of three people, it is also the fruit of valuable collaborations and interactions during the past few months. We would like to thank every person that has helped and assisted us, at the scientific, technical and human levels.

Thanks to Larissa Sorensen and all the Laboratory for Applied mechanics and reliability Analysis (LMAF-EPFL) of Pr. John Botsis, to Renato Krpoun from the Laboratory of Microsystems for Space Technologies (LMTS-EPFL) and Muriel Noca from the EPFL Space Center for sharing their knowledge on space technologies, to Lars Overgaard to all the information about the CubeSat of the Aalborg University, to M. Favre from Composite Design to his advice about the composites possibilities and properties, to Mr. Foletti and Mr Despont for their mental support.

Appreciation is also given to the machining workshop, in particular Mr. Jeanneret and his team, for the production of a prototype of the frame and for their useful remarks about the design and machineability of the various structural pieces.

REFERENCES

[1] Armen Toorian, *CubeSat as responsive Satellites*, California Polytechnic State University, 3rd Responsive Space Conference, 2005.

2] CubeSat Design Specification, Revision 9, Cal Poly, May 2005.

[3] Veronica Bashbush, *Characterization of the Internal and External Environments of the CubeSat P-POD and Test Pod - Master's Thesis*, California Polytechnic State University, January 2004.

[4] R. Krpoun, *CubeSat Project – Mission Description Document*, EPFL, May 2005.

[5] R. Krpoun, *Preliminary Functional Specifications*, EPFL, June 2005.

[6] M. Noca, *SwissCube Project Specifications*, EPFL, January 2006.

[7] Team Meeting, EPFL, 16 march 2006.

[8] Aidan Bettridge, *CASSat Structural Subsystem - Final Year Thesis*, University of Sydney, November 2004.

[9] DNEPR Safety Compliance Requirements, Version 1.0, Cal Poly, August 2004.

[10] CubeSat Test Pod User's Guide, Revision 4, Cal Poly, June 2005.

[11] VEGA User's Manual, Issue 3/Revision 0, March 2006.

[12] Puig-Suari J., Turner C., Ahlgren W.: Development of the Standard CubeSat Deployer and a CubeSat Class PicoSatellite, 2001.

[13] AIAA Aerospace Design Engineers Guide, Fifth Edition, Published by AIAA, 2003.

[14] http://spacecraft.ssl.umd.edu/akins_laws.html

[15] ECSS-Q-70-71A rev. 1(18 June 2004) QA selection of space materials

[16] ECSS-Q-70-36A Material selection for controlling stress-corrosion cracking (20 January 1998)

[17] ECSS-E-30 Part 3A Mechanical - Part 3 Mechanisms (25 April 2000)

[18] DNEPR User's Guide, Issue 2, November 2001

[19] C. H. Hsueh, *Thermal stresses in elastic multilayer systems*, Metals and Ceramics Division, Oak Ridge National Laboratory, Oak Ridge, TN 37831, 30 July 2002.

[20] http://www.3m.com/product/index.jhtml

Other non-referenced documents:

Caillibot, Grant, Kekez, Formation Flying Demonstration Missions Enabled by CanX Nanosatellite Technology, University of Toronto, 19th Annual AIAA/USU Conference on Small Satellites.

Canada's Smallest Satellite: The Canadian Advanced Nanospace eXperiment (CanX-1), University of Toronto

Chen-Joe Fong, Lessons Learned of NSPO's Picosatellite Mission: YamSat - 1A, 1B & 1C, National Space Program Office Taiwan, 16th Annual AIAA/USU Conference on Small Satellites, 2002.

CubeSat Project Critical Design Review, University of Tokyo, April 2001.

Hank Heidt, CubeSat:: A new Generation of Picosatellite for Education and Industry Low-Cost Space Experimentation, 14th Annual/USU Conference on Small Satellites, 2000.

Illinois Observing Nanosatelite (ION) final report, University of Illinois, May 2003.

Jacob H. Oehrig, TU Sat 1 An Innovative Low-Cost Communications Satellite, Taylor University, 15th Annual/USU Conference on Small Satellites.

Jake A. Schaffner, The Electronic System Design, Analysis, Integration, and Construction of the Cal Poly State University CP1 CubeSat, California Polytechnic State University, 16th AIAA/USU Conference on Small Satellites, 2002.

KUTESat Critical Design Review, Kansas University.

Matzen, Jespersen, Design specifications for the mechanical part of the DTUsat, Technical university of Denmark, January 2004.

QuakeSat Lessons Learned: Notes from the Development of a Triple CubeSat, 2004.

R. Krpoun, CubeSat Project – Management Plan, EPFL, May 2005.

R. Krpoun, Preliminary Interface Control Document (ICD), EPFL, September 2005.

The AAU-CubeSat student satellite project: Architectural overview and lessons learned, 2004.

The Design and Operation of The Canadian Advanced Nanospace eXperiment (CanX-1), University of Toronto.

UH CubeSat Critical Design Review, System Integration and Testing, University of Hawaii, May 2002

Zheng You, MEMSSat: a TestBed of MEMS Technologies for Space, Tsinghua University, 19th Annual AIAA/USU Conference on Small Satellites, 2005.

Internet links:

Aalborg University AAU CubeSat : www.cubesat.auc.dk

AMSAT - Launch Information : www.amsat.org/amsat-new/satellites/cubesats.php

ArianEspace : http://www.arianespace.com

California Polytechnic State University PolySat : www.polysat.org

Cornell University CubeSat : www.mae.cornell.edu/cubesat

CubeSat Community : http://cubesat.calpoly.edu

CubeSat Kit : www.cubesatkit.com

Dartmouth College DartSat : http://engineering.dartmouth.edu/~dartsat/

Iowa Sate University CySat : http://cosmos.ssol.iastate.edu/cysat/resources.html

Material Property Data : www.matweb.com

Montanna State University MEROPE : http://www.ssel.montana.edu/merope/

Norwegian Student Satellite Project NCUBE : http://www.ncube.no

SatnfordStanford University CubeSat : http://ssdl-delta.stanford.edu/narcissat

Stanford University QuakeSat : http://www.quakefinder.com/quakesat.htm

Taylor University TU SAT 1 : http://www.css.tayloru.edu/~physics/picosat/

Technical University of Denmark DTUsat : www.dtusat.dtu.dk

The Stensat Group : http://www.stensat.org

University of Hawaii CubeSat Project : http://www-ee.eng.hawaii.edu/~cubesat/

University of Illinois CubeSat UIUC : http://courses.ece.uiuc.edu/cubesat/

University of Nevada NevadaSat : http://www.unr.edu/NevadaSat/

University of New South Wales BlueSat : http://www.bluesat.unsw.edu.au/

University of Tokyo CubeSat : www.space.t.u-tokyo.ac.jp/cubesat/index-e.html

University of Toronto CanX , http://www.utias-sfl.net

http://www.3m.com/product/index.jhtml

www.hexcel.com/Products/Selector+Guides/

www.cytec.com/business/EngineeredMaterials/spacecomposites.shtm

APPENDICES

Appendix A - SwissCube Requirements and Specifications

The following documentation lists the specifications and requirements of the various aspects of the SwissCube picosatellite structural subsystem.

Physical

- External geometry of the SwissCube shall meet with the specifications detailed in Figure 56 of Appendix A - SwissCube Requirements and Specifications.

- The structural subsystem shall meet the physical requirements defined by the CUBESAT Design Specifications Document. Specifically,

    - The structural subsystem shall have overall dimensions of 10 cm x 10 cm x 10 cm.
    - The structural subsystem shall be built from materials with thermal expansion properties comparable to those of Aluminum alloys 7075-T73 and 6061-T6.
    - The structural subsystem of the SwissCube shall have four vertical rails with a 7 mm top overhang and 6.5 mm bottom overhang to maintain spacing between CubeSats in P-Pod
    - The edges of the rail feet shall be rounded.
    - The center of mass of the entire satellite shall be within 2 cm of the geometric center.
    - The structural subsystem shall not exceed a mass of 250g.
    - The surfaces of the rails, those in contact with the P-POD, shall be hard anodized.

- The SwissCube Structural Subsystem shall have a flight pin access area located on a side face with dimension limits shown in Figure 56 of Appendix A - SwissCube Requirements and Specifications.

- The structural subsystem shall have a data port (RJ45 jack) access area located on a side face with the dimension limits shown in Figure 56 of Appendix A - SwissCube Requirements and Specifications.

- 75% (85.125 cm) of flat rail surface area shall be available for rail contact within P-Pod

- 60% of the rail cross-sectional area shall be available for contact with neighboring CubeSats.

- No externally mounted components shall exceed 6.5 mm in height from exterior surface of the structural subsystem

- The satellite shall have an interface with separation springs on the top rail feet

- The satellite shall have a non-metal contact surface on the bottom surface of the rail feet.

Power and Command

- It is a requirement of the CubeSat specifications that the structural subsystems shall have at least one deployment detection switch (two is recommended) located on top of a rail, such that when depressed the switch remains flush with rail surface.

Loading Conditions

The following loading conditions can be found in.

- Static Loads
- Vibration Loads
- Shock Loads
- Pressure Loads

Environments

The structural subsystem of the SwissCube must be able to maintain integrity in the following environmental conditions:

- Natural Environments

- Ambient temperature up to 30 °C (TBC).
- Humidity of up to 80% (TBC).
- The satellite structural subsystem shall be constructed of materials that have minimal outgassing in the vacuum of space.
- To prevent pressurization loads, the structural subsystem will not have any sealed enclosures.

- Induced Environments

- The structural subsystem shall be able to withstand mechanical shocks, of TBD magnitude and TBD frequency, from possible accidents while handling or during transportation.
- The structural subsystem shall be able to withstand all vibrations encountered in transportation and handling and maintain structural integrity.
- The structural subsystem shall be able to withstand loading cases as defined by this document
- The structural subsystem must be able to withstand a 125% shock and vibration qualification test while in P-POD.
- The rate of pressure change inside the fairing will not exceed 0.35 $N/(cm^2 \cdot sec)$.

Structural Materials

- The main frame of the structural subsystem shall be constructed from "Certal" aluminum.
- The crossbars of the structural subsystem shall be constructed from "Certal" aluminum.
- Faceplates shall be constructed from composites (carbon fibers in an epoxy matrix).
- Structural members and faceplates shall be epoxied with 3M Scotch-Weld 2216 B/A Gray epoxy (TBC).
- Structural members shall be fastened with M2 screws and epoxy (TBC).

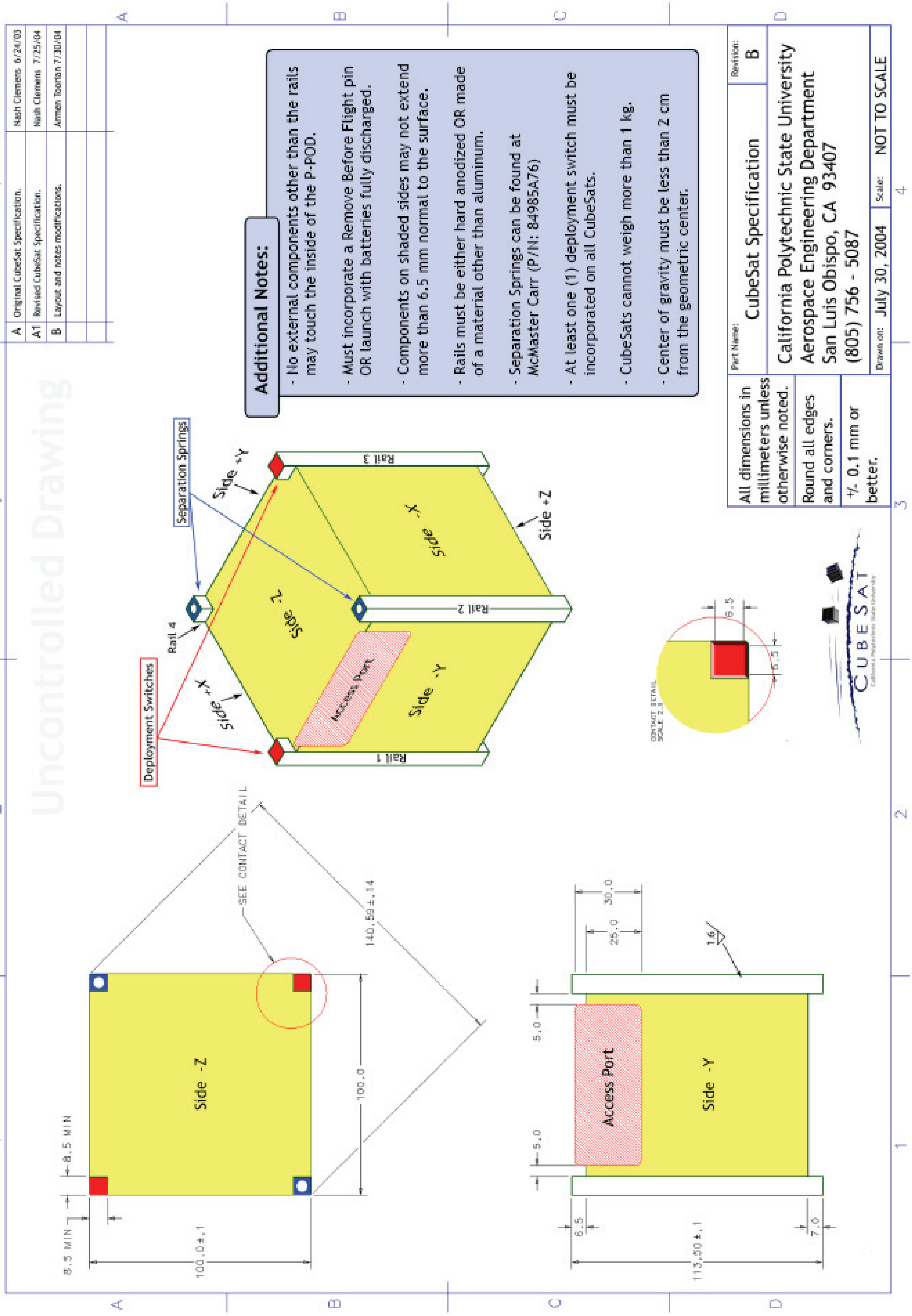


Figure 56 CubeSat Specification Drawing.

Appendix B - Material Properties

These data have been referenced from the website www.matweb.com – Material Property Data.

Al – 6061 – T6

Material Notes: Information provided by Alcoa, Starmet and the references. General 6061 characteristics and uses:

Excellent joining characteristics, good acceptance of applied coatings. Combines relatively high strength, good workability, and high resistance to corrosion; widely available. The T8 and t9 tempers offer better chipping characteristics over the T6 temper.

Applications:

Aircraft fittings, camera lens mounts, couplings, marines fittings and hardware, electrical fittings and connectors, decorative or misc. hardware, hinge pins, magneto parts, brake pistons hydraulic pistons, appliance fittings, valves and valve parts; bike frames.

Composition:

Component	Wt. %	Component	Wt. %	Component	Wt. %
Al	98	Fe	Max 0.7	Si	0.4 - 0.8
Cr	0.04 - 0.35	Mg	0.8 - 1.2	Ti	Max 0.15
Cu	0.15 - 0.4	Mn	Max 0.15	Zn	Max 0.25

Table 9 Physical properties of Al-6061-T6.

Physical Properties	Metric	English	Comments
Density	2.7 g/cc	0.0975 lb/in³	
Mechanical Properties			
Hardness, Brinell	95	95	500 kg load with 10 mm ball
Hardness, Knoop	120	120	Converted from Brinell Hardness Value
Hardness, Rockwell A	40	40	Converted from Brinell Hardness Value
Hardness, Rockwell B	60	60	Converted from Brinell Hardness Value
Hardness, Vickers	107	107	Converted from Brinell Hardness Value
Tensile Strength, Ultimate	310 MPa	45000 psi	
Tensile Strength, Yield	275 MPa	39900 psi	
Elongation at Break	12 %	12 %	In 5 cm; Sample 1.6 mm thick
Modulus of Elasticity	69 GPa	10000 ksi	Average of Tension and Compression. In Aluminum alloys, the compressive modulus is typically 2% greater than the tensile modulus
Notched Tensile Strength	324 MPa	47000 psi	2.5 cm width x 0.16 cm thick side-notched specimen, K_t = 17.
Ultimate Bearing Strength	607 MPa	88000 psi	Edge distance/pin diameter = 2.0
Bearing Yield Strength	386 MPa	56000 psi	Edge distance/pin diameter = 2.0
Poisson's Ratio	0.33	0.33	Estimated from trends in similar Al alloys.
Fatigue Strength	95 MPa	13800 psi	500,000,000 Cycles
Fracture Toughness	29 MPa-m½	26.4 ksi-in½	K_{IC}; TL orientation.
Machinability	50 %	50 %	0-100 Scale of Aluminum Alloys
Shear Modulus	26 GPa	3770 ksi	Estimated from similar Al alloys.
Shear Strength	205 MPa	29700 psi	
Electrical Properties			
Electrical Resistivity	4e-006 ohm-cm	e-006 ohm-cm	
Thermal Properties			
CTE, linear 20°C	23.6 µm/m-°C	13.1 µin/in-°F	20-100°C
CTE, linear 250°C	25.2 µm/m-°C	14 µin/in-°F	Estimated from trends in similar Al alloys. 20-300°C.
Heat Capacity	0.896 J/g-°C	214 BTU/lb-°F	
Thermal Conductivity	166.9 W/m-K	TU-in/hr-ft²-°F	
Melting Point	582 - 652 °C	080 - 1210 °F	
Solidus	582 °C	1080 °F	
Liquidus	652 °C	1210 °F	

Al-7075-T6

Material Notes:

General 7075 characteristics and uses (from Alcoa): Very high strength material used for highly stressed structural parts. The T7351 temper offers improved stress-corrosion cracking resistance.

Applications:

Aircraft fittings, gears and shafts, fuse parts, meter shafts and gears, missile parts, regulating valve parts, worm gears, keys, aircraft, aerospace and defense applications; bike frames, all terrain vehicle (ATV) sprockets.

Composition:

Component	Wt. %	Component	Wt. %	Component	Wt. %
Al	87.1 - 91.4	Mg	2.1 - 2.9	Si	Max 0.4
Cr	0.18 - 0.28	Mn	Max 0.3	Ti	Max 0.2
Cu	1.2 - 2	Other, each	Max 0.05	Zn	5.1 - 6.1
Fe	Max 0.5	Other, total	Max 0.15		

Table 10 Physical properties of Al-7075.

Physical Properties	Metric	English	Comments
Density	2.81 g/cc	).102 lb/in^3	Typical
Mechanical Properties			
Hardness, Brinell	150	150	AA; Typical; 500 g load; 10 mm ball
Hardness, Knoop	191	191	Converted from Brinell Hardness Value
Hardness, Rockwell A	53.5	53.5	Converted from Brinell Hardness Value
Hardness, Rockwell B	87	87	Converted from Brinell Hardness Value
Hardness, Vickers	175	175	Converted from Brinell Hardness Value
Ultimate Tensile Strength	572 MPa	83000 psi	Typical
Tensile Yield Strength	503 MPa	73000 psi	Typical
Elongation at Break	11 %	11 %	Typical; 1/16 in. (1.6 mm) Thickness
Elongation at Break	11 %	11 %	Typical; 1/2 in. (12.7 mm) Diameter
Modulus of Elasticity	71.7 GPa	10400 ksi	Average of tension and compression. Compression modulus is about 2% greater than tensile modulus.
Poisson's Ratio	0.33	0.33	
Fatigue Strength	159 MPa	23000 psi	500,000,000 cycles completely reversed stress; RR Moore machine/specimen
Fracture Toughness	20 MPa-m½	3.2 ksi-in½	K(IC) in S-L Direction
Fracture Toughness	25 MPa-m½	2.8 ksi-in½	K(IC) in T-L Direction
Fracture Toughness	29 MPa-m½	5.4 ksi-in½	K(IC) in L-T Direction
Machinability	70 %	70 %	0-100 Scale of Aluminum Alloys
Shear Modulus	26.9 GPa	3900 ksi	
Shear Strength	331 MPa	48000 psi	
Electrical Properties			
Electrical Resistivity	5.15e-006 ohm-cm	)6 ohm-cm	Typical at 68°F
Thermal Properties			
CTE, linear 68°F	23.6 µm/m-°C	.1 µin/in-°F	Average over 68-212°F range.
CTE, linear 250°C	25.2 µm/m-°C	4 µin/in-°F	Average over the range 20-300°C
Heat Capacity	0.96 J/g-°C	BTU/lb-°F	
Thermal Conductivity	130 W/m-K	-in/hr-ft^2-°F	Typical at 77°F
Melting Point	477 - 635 °C	) - 1175 °F	Typical range based on typical composition for wrought products 1/4 inch thickness or greater. Homogenization may raise eutectic melting temperature 20-40°F but usually does not eliminate eutectic melting.
Solidus	477 °C	890 °F	Typical
Liquidus	635 °C	1175 °F	Typical
Processing Properties			
Annealing Temperature	413 °C	775 °F	
Solution Temperature	466 - 482 °C	70 - 900 °F	
Aging Temperature	121 °C	250 °F	

Certal

These data come from the website http://www.aluplus.dk/pdf/CERTALeng.pdf

Technical Datasheet

CERTAL®

EN AW-7022 / AlZn5Mg3Cu

Edition September 2001

ALCAN ROLLED PRODUCTS

Alcan Aluminium Valais Ltd t +41 27 457 51 11
CH-3960 Sierre, Switzerland f +41 27 457 65 15

BRIEF DESCRIPTION

Certal® thick plates have been optimised to provide excellent **machinability, shape stability and high strength.** Certal® is therefore ideal for industrial tools. Applications include injection and blow-moulds for plastic bottles, plastic containers and shoes as well as heating plates, mechanical guides, tooling supports, jigs and fixtures.

PROCESSING METHODS

Weldability

• TIG/MIG	possible
filler alloy	AA 5183 AA 5356
• by resistance	good
Surface Treatments	
Anodizing	
• technical	good
• decorative	not adequate
Polishing	excellent
Hard Chroming	good
Chemical Nickel-Plating	good
Chemical texturing	well adapted
Machinability	excellent

AVAILABILITY

Certal® plates are delivered in temper T651 (quenched – stretched – artificially aged) in the following dimensions :

Thickness	Max. width
8.0 - 70 mm	2020 mm
71 - 90 mm	1820 mm
91- 120 mm	1520 mm
121- 140 mm	1020 mm

For thicknesses above 140 mm, the alloy Certal® SPC is recommended.

CHEMICAL COMPOSITION (weight %)

Si	Fe	Cu	Mn	Mg	Cr	Zn	Ti +Zr
max. 0.5	max. 0.5	0.5 1.0	0.1 0.40	2.6 3.7	0.1 0.3	4.3 5.2	max. 0.2

PHYSICAL PROPERTIES (nominal values)

Density	2.76 g/cm^3
Elastic Modulus	72000 MPa
Lin. thermal expansion coefficient (20°-100°C)	23.6 10^{-6} K^{-1}
Thermal conductivity (Temper T651)	120 - 150 W/mK
Electrical conductivity (Temper T651, 20°C)	18 - 22 MS/m

MECHANICAL STRENGTH

Min. tensile properties (Temper T651)[1)]

Thickness (over ... to)	Rm [MPa]	Rp0.2 [MPa]	A50 [%]
12.5 - 25 mm	540	460	8
25 - 50 mm	530	460	7
50 - 100 mm	500	420	6
100 - 140 mm	490	400	6

1) These guaranteed values are much higher than EN AW-7022 T651 values

Typical strength for various thicknesses

Thickness (over ... to)	Rm [MPa]	Rp0.2 [MPa]	A50 [%]	HB
8.0 - 25 mm	555	495	9	170
25 - 100 mm	550	495	8	165
100 - 140 mm	545	490	7	165

FR4 laminate

This data comes from the website: http://www.jjorly.com

FR4 laminate grades are produced by inserting continuous glass woven fabric impregnated with an epoxy resin binder while forming the sheet under high pressure. This material is used extensively in the electronics industry because its water absorption is extremely minimal. The FR4 is most commonly used in PCB (Printed Circuit Boards) applications. FR4 has excellent dielectric loss properties, and great electrical strength. It is also a fire retardant grade of G10. FR4 is also known as Garolite.

	UNITS	VALUES
MECHANICAL PROPERTIES		
BOND STRENGTH	LBS	2,500
COMPRESSIVE STRENGTH	PSI	60,000
FLEXURAL STRENGTH	PSI	55,000
SHEAR STRENGTH	PSI	19,000
TENSILE STRENGTH	PSI	40,000
IMPACT STRENGTH, IZOD (NOTCHED)	FT-LBS PER INCH OF NOTCH	7
SPECIFIC GRAVITY		1.82
FLEXURAL MODULUS OF ELASTICITY	PSI	2,700,000
ROCKWELL HARDNESS	M SCALE	M110
ELECTRICAL PROPERTIES		
DIELECTRIC CONSTANT	1 MEGACYCLE	5.2
DIELECTRIC STRENGTH	VOLTS PER MIL	400
DISSIPATION FACTOR	1 MEGACYCLE	0.025
ARC RESISTANCE	SECONDS	80
THERMAL PROPERTIES		
MAX CONSTANT OPERATING TEMPERATURE	°F	285
INSULATION RESISTANCE	Condition: 96 hrs., 90% relative humidity, 95°F megohms	200,000
WATER ABSORPTION	% 24 HRS	0.11
THERMAL CONDUCTIVITY	Calories/Sec./cm2/°C/cm	7 X 10-4
COEFFICIENT OF THERMAL EXPANSION	Cm/Cm°C	0.9

HexWeb Honeycombs

Introduction

Honeycomb is a lightweight core material for structural stiffening applications. This versatile material is widely used in the construction of aircraft components such as floors, interior panelling and helicopter rotor blade aerofoils. Other applications include railway carriage doors and ceiling panels, marine bulkheads and furniture. Honeycomb is also the ideal material for energy absorption (bumpers/fenders, lift shaft bases), for RF shielding and fluid and light directionalisation.

This guide has been compiled to assist with the selection of the best type of honeycomb for a particular application. More detailed information is included in the individual product data sheets.

METALLIC

	Product type	Strength	Stiffness	Dielectric Performance	Max Service Temp. °C (°F)	Thermal Conductivity/ Characteristics	Product Form	Density Range kg/m³ (lb/cf)	Recommended For Energy Absorption	Treatment Options	Environmental Resistance
ALUMINIUM	CR-PAA/CRIII 5052 Aerospace Grade Aluminium Honeycomb	High	Very High	Low Transmission	175 (350)	High	Hexagonal cell	16 to 192 (1 to 12)	Yes	CRIII Corrosion resistant coating that meets MIL-C-7438 Specifications	Good
							OX cell	42 to 169 (2.5 to 10.5)			
							Rigicell® (corrugated)	168 to 880 (10.5 to 55)	Yes		
							Flexcore	34 to 128 (2 to 8)		CR-PAA treated and primed	Excellent
							Double-Flex	44 to 77 (3 to 5)			
	CR-PAA/CRIII 5056 Aerospace Grade High Performance Aluminium Honeycomb	High	Very High	Low Transmission	175 (350)	High	Hexagonal cell	16 to 147 (1 to 9)	Yes	CRIII Corrosion resistant coating that meets MIL-C-7438 Specifications	Good
							Flexcore	34 to 128 (2 to 8)		CR-PAA treated and primed	Excellent
	ACG Commercial Grade Aluminium Honeycomb	High	Very High	Low Transmission	175 (350)	High	Hexagonal cell	29 to 114 (1.8 to 7)	Yes	CRIII Corrosion resistant coating that meets MIL-C-7438 Specifications (available in the US)	Good
										CRF Chromium free coating (available in Europe)	

NON-METALLIC

		Product type	Strength	Stiffness	Dielectric Performance	Max Service Temp. °C (°F)	Thermal Conductivity/ Characteristics	Product Form	Density Range kg/m³ (lb/cf)	Environmental Resistance	Cost
ARAMID	META - ARAMID PAPER	HRH-10 Aerospace Grade Aramid/Phenolic	High	Low	Good Transmission	175 (350)	Low	Hexagonal cell	24 to 144 (1.5 to 9)	Excellent	Moderate
								OX cell	29 to 72 (1.8 to 4.5)		
								Flexcore	40 to 88 (2.5 to 55)		
		HRH-78 Commercial Grade Aramid/Phenolic	High	Low	Good Transmission	175 (350)	Low	Hexagonal cell	32 to 144 (2 to 9)	Excellent	Moderate
								OX cell	29 to 72 (1.8 to 4.5)		
		HRH-310 Aramid/Polyimide	High	Low	Excellent Transmission	175 (350)	Low	Hexagonal cell	29 to 80 (1.8 to 5)	Excellent	High
	PARA - ARAMID	HRH-36 Kevlar* Paper/Phenolic	High	High	Good Transmission	175 (350)	Low	Hexagonal cell	24 to 96 (1.5 to 6)	Excellent	High
								OX cell	32 to 48 (2 to 3.0)		
								Flexcore	32 to 56 (2 to 3.5)		
		HRH-49 Woven Kevlar*/Epoxy	High	Low	Good Transmission	175 (350)	Very Low	Hexagonal cell	34 (2.1)	Excellent	High

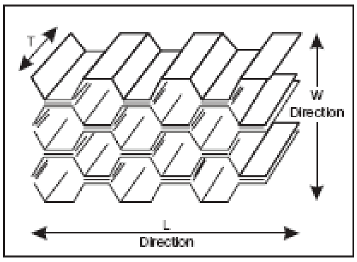
*Du Pont Trademark

NON-METALLIC

	Product type	Strength	Stiffness	Dielectric Performance	Max Service Temp. °C (°F)	Thermal Conductivity/ Characteristics	Product Form	Density Range kg/m³ (lb/cf)	Environmental Resistance	Cost
GLASS	HRP Fibre Glass/Phenolic	High	Moderate	Good Transmission	175 (350)	Low	Hexagonal cell	32 to 192 (2 to 12)	Excellent	Moderate
							OX cell	51 to 112 (3.2 to 7)		
							Flexcore	40 to 88 (2.5 to 5.5)		
	HFT Bias Weave Fibre Glass/Phenolic	High	High	Good Transmission	175 (350)	Low	Hexagonal cell	32 to 128 (2 to 8)	Excellent	High
							OX cell	96 (6.0)		
	HRH-327 Bias Weave Fibre Glass/Polyimide	High	High	Excellent Transmission	260 (500)	Low	Hexagonal cell	51 to 128 (3.2 to 8)	Excellent	Very High
CARBON	HFT-G Bias Weave Carbon/Phenolic	High	Very High	Low Transmission	175 (350)	Medium (High conductivity versions available)	Hexagonal cell + Reinforced	32 to 160 (2.0 to 10)	Excellent	Very High
	HFT-G-327 Bias Weave Carbon/Polyimide	High	Very High	Low Transmission	260 (500)	Medium	Hexagonal cell + Reinforced	32 to 240 (2 to 15)	Excellent	Very High

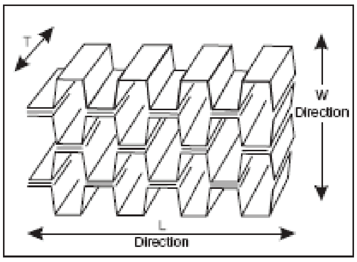
HexWeb® Honeycomb *Selector Guide*

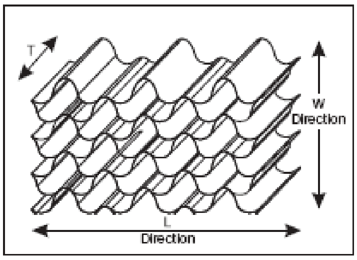
Honeycomb Configurations



Hexagonal Core
The standard hexagonal honeycomb is the basic and most common cellular honeycomb configuration, and is currently available in all metallic and non-metallic materials.

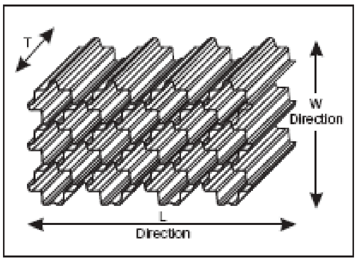
OX-Core®
The "OX" configuration is a hexagonal honeycomb that has been over-expanded in the "W" direction, providing a rectangular cell configuration that facilitates curving or forming in the "L" direction. The OX process increases "W" shear properties and decreases "L" shear properties when compared to hexagonal honeycomb.

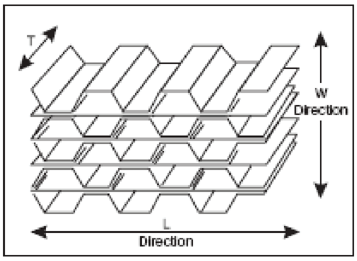




Flex-Core®
The Flex-Core cell configuration provides for exceptional formability in compound curvatures with reduced anticlastic curvature and without buckling the cell walls. Curvatures of very tight radii are easily formed. When formed into tight radii, Flex-Core provides higher shear strengths than comparable hexagonal core of equivalent density. Flex-Core is manufactured from aluminium HRH-10, HRH-36 and HRP substrates.

Double-Flex®
Double-Flex is a unique large cell Aluminium Flex-Core with excellent formability and high specific compression properties. Double-Flex is the most formable cell configuration.





Reinforced Hexagonal
Reinforced honeycomb has a sheet of substrate material placed along the nodes in the ribbon direction to increase the mechanical properties. The Reinforced Hexagonal configuration provides a heavy density honeycomb suitable for high load areas such as attachment points.

Other Configurations
The standard honeycomb configurations described above will meet almost all requirements. Hexcel can also design and fabricate special configuration honeycomb in response to specific needs.

Appendix C – Adhesives properties

Table 11 Different possible adhesives and their properties

Adhesive	Product type	Chemical comp.	Shear strength [Mpa]	Temp. range [°C]	CTE [µm/K]	Thermal cond. [W/mK]
Structural						
Araldite AV138/HV998	2-part paste	Epoxy	43 (tensile)	<120	67	0,35
Scotchweld 2216 (B/A)	2-part paste	Modified epoxy	21,3	50(-) à 80	45 - 182	0,39
Scotchweld 1614 (B/A)	2-part paste	Epoxy	21,1	55(-) à 80	-	-
Epo-tek U300	2-part paste	Epoxy	10,3	<200	43 (T<130°C)	-
DC 93500	2-part paste	Silicone	5,8 (tensile)	65(-) à 200	300	0,15
Redux 312	Adhesive film	Modified epoxy	43	55(-) à 120	-	-
Redux 340	Adhesive film	Modified epoxy	43	55(-) à 200	-	-
FM 73	Adhesive film	Epoxy	39,8	55(-) à 82	-	-
Thermal						
Epo-tek 930	2-part thermal	Epoxy (Boron nitride)	-	-	40	4,1

Adhesive	Resistivity [Ω m]	Cure Temp. [°C]	Outgassing (TML / RML / CVCM)[1]	Fabricant	ECSS recommended
Structural					
Araldite AV138/HV998	1,90E+13	65	0,84 / 0,57 / 0,02	www.vantico.com	yes
Scotchweld 2216 (B/A)	1,90E+10	70	1,42 / 0,75 / 0,01	www.3m.com	yes
Scotchweld 1614 (B/A)	-	65	-	www.3m.com	no
Epo-tek U300	1,00E+14	80-150	(passed NASA specs)	www.epotek.com	no
DC 93500	6,90E+13	RT	0,30 / 0,28 / 0,03	www.dowcorning.com	yes
Redux 312	-	120	1,10 / 0,40 / 0,05	www.hexcel.com	yes
Redux 340	-	175	-	www.hexcel.com	no
FM 73	-	121	(0,78 / - / 0,00)	www.cytec.com	no
Thermal					
Epo-tek 930	-	-	-	www.epotek.com	yes

[1] 24 hours at 125°C and 10^{-3}Pa (ECSS)

http://www.3m.com/product/index.jhtml

3M Scotch-Weld™ Epoxy Adhesive 2216 B/A

Technical Data **August, 2005**

Product Description

3M™ Scotch-Weld™ Epoxy Adhesive 2216 B/A is a flexible, two-part, room temperature curing epoxy with high peel and shear strength. Scotch-Weld epoxy adhesive 2216 B/A is identical to 3M™ Scotch-Weld™ Epoxy Adhesive EC-2216 B/A in chemical composition. Scotch-Weld epoxy adhesive EC-2216 B/A has been labeled, packaged, tested, and certified for aircraft and aerospace applications. Scotch-Weld adhesive 2216 B/A may be used for aircraft and aerospace applications if proper Certificates of Test have been issued and material meets all aircraft manufacturer's specification requirements.

Typical Uncured Physical Properties

Note: The following technical information and data should be considered representative or typical only and should not be used for specification purposes.

Product	3M™ Scotch-Weld™ Epoxy Adhesive					
	2216 B/A Gray		2216 B/A Tan NS		2216 B/A Translucent	
	Base	Accelerator	Base	Accelerator	Base	Accelerator
Color:	White	Gray	White	Tan	Translucent	Amber
Base:	Modified Epoxy	Modified Amine	Modified Epoxy	Modified Amine	Modified Epoxy	Modified Amine
Net Wt.: (lb/gal)	11.1-11.6	10.5-11.0	11.1-11.6	10.5-11.0	9.4-9.8	8.0-8.5
Viscosity: (cps) (Approx.) Brookfield RVF #7 sp. @ 20 rpm	75,000 - 150,000	40,000 - 80,000	75,000 - 150,000	550,000 - 900,000	11,000 - 15,000	5,000 - 9,000
Mix Ratio: (by weight)	5 parts	7 parts	5 parts	7 parts	1 part	1 part
Mix Ratio: (by volume)	2 parts	3 parts	2 parts	3 parts	1 part	1 part
Work Life: 100 g Mass @ 75°F (24°C)	90 minutes	90 minutes	120 minutes	120 minutes	120 minutes	120 minutes

Features

- Excellent for bonding many metals, woods, plastics, rubbers, and masonry products.
- Base and Accelerator are contrasting colors.
- Good retention of strength after environmental aging.
- Resistant to extreme shock, vibration, and flexing.
- Excellent for cryogenic bonding applications.
- The tan NS Adhesive is non-sag for greater bondline control.
- The translucent can be injected.
- Meets DOD-A-82720.

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Typical Cured Physical Properties

Product	3M™ Scotch-Weld™ Epoxy Adhesive		
	2216 Gray	**2216 Tan NS**	**2216 Translucent**
Color	Gray	Tan	Translucent
Shore D Hardness ASTM D 2240	50-65	65-70	35-50
Time to Handling Strength	8-12 hrs.	8-12 hrs.	12-16 hrs.

Typical Cured Electrical Properties

Product	3M™ Scotch-Weld™ Epoxy Adhesive	
	2216 Gray	**2216 Translucent**
Arc Resistance	130 seconds	
Dielectric Strength	408 volts/mil	630 volts/mil
Dielectric Constant @ 73°F (23°C)	5.51–Measured @ 1.00 KHz	6.3 @ 1 KHz
Dielectric Constant @ 140°F (60°C)	14.17–Measured @ 1.00 KHz	—
Dissipation Factor 73°F (23°C)	0.112 Measured @ 1.00 KHz	0.119 @ 1 KHz
Dissipation Factor 140°F (60°C)	0.422–Measured @ 1.00 KHz	—
Surface Resistivity @ 73°F (23°C)	5.5 x 10^{16} ohm–@ 500 volts DC	—
Volume Resistivity @ 73°F (23°C)	1.9 x 10^{12} ohm-cm–@ 500 volts DC	3.0 x 10^{12} ohm-cm @ 500 volts DC

Typical Cured Thermal Properties

Product	3M™ Scotch-Weld™ Epoxy Adhesive	
	2216 Gray	**2216 Translucent**
Thermal Conductivity	0.228 Btu-ft/ft² h °F	0.114 Btu-ft/ft² h °F
Coefficient of Thermal Expansion	102 x 10^{-6} in/in/°C between 0-40°C 134 x 10^{-6} in/in/°C between 40-80°C	81 x 10^{-6} in/in/°C between -50-0°C 207 x 10^{-6} in/in/°C between 60-150°C

Typical Cured Outgassing Properties

Outgassing Data
NASA 1124 Revision 4

	% TML	% CVCM	% Wtr
3M™ Scotch-Weld™ Epoxy Adhesive 2216 Gray	.77	.04	.23

Cured in air for 7 days @ 77°F (25°C).

Handling/Curing Information

Directions for Use

1. For high strength structural bonds, paint, oxide films, oils, dust, mold release agents and all other surface contaminants must be completely removed. However, the amount of surface preparation directly depends on the required bond strength and the environmental aging resistance desired by user. For suggested surface preparations of common substrates, see the following section on surface preparation.
2. These products consist of two parts. Mix thoroughly by weight or volume in the proportions specified on the product label and in the uncured properties section. Mix approximately 15 seconds after a uniform color is obtained.

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Handling/Curing Information *(continued)*

3. For maximum bond strength, apply product evenly to both surfaces to be joined.
4. Application to the substrates should be made within 90 minutes. Larger quantities and/or higher temperatures will reduce this working time.
5. Join the adhesive coated surfaces and allow to cure at 60°F (16°C) or above until firm. Heat, up to 200°F (93°C), will speed curing.
6. The following times and temperatures will result in a full cure:

Product	3M™ Scotch-Weld™ Epoxy Adhesive		
	2216 Gray	2216 Tan NS	2216 Translucent
Cure Temperature	Time	Time	Time
75°F (24°C)	7 days	7 days	30 days
150°F (66°C)	120 minutes	120 minutes	240 minutes
200°F (93°C)	30 minutes	30 minutes	60 minutes

7. Keep parts from moving until handling strength is reached. Contact pressure is necessary. Maximum shear strength is obtained with a 3-5 mil bond line. Maximum peel strength is obtained with a 17-25 mil bond line.
8. Excess uncured adhesive can be cleaned up with ketone type solvents.*

Adhesive Coverage: A 0.005 in. thick bondline will typically yield a coverage of 320 sq. ft/gallon

Application and Equipment Suggestions

These products may be applied by spatula, trowel or flow equipment.

Two-part mixing/proportioning/dispensing equipment is available for intermittent or production line use. These systems are ideal because of their variable shot size and flow rate characteristics and are adaptable to many applications.

Surface Preparation

For high strength structural bonds, paint, oxide films, oils, dust, mold release agents and all other surface contaminants must be completely removed. However, the amount of surface preparation directly depends on the required bond strength and the environmental aging resistance desired by user.

The following cleaning methods are suggested for common surfaces.

Steel or Aluminum (Mechanical Abrasion)

1. Wipe free of dust with oil-free solvent such as acetone or alcohol solvents.*
2. Sandblast or abrade using clean fine grit abrasives (180 grit or finer).
3. Wipe again with solvents to remove loose particles.
4. If a primer is used, it should be applied within 4 hours after surface preparation. If 3M™ Scotch-Weld™ Structural Adhesive Primer EC-1945 B/A is used, apply a thin coating (0.0005") on the metal surfaces to be bonded, air dry for 10 minutes, then cure for 30 minutes at 180°F (82°C) prior to bonding.

*When using solvents, extinguish all ignition sources, including pilot lights, and follow the manufacturer's precautions and directions for use. Use solvents in accordance with local regulations.

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Surface Preparation *(continued)*

Aluminum (Chemical Etch)

Aluminum alloys may be chemically cleaned and etched as per ASTM D 2651. This procedure states to:

1. Alkaline Degrease – Oakite 164 solution (9-11 oz/gal of water) at 190°F ± 10°F (88°C ± 5°C) for 10-20 minutes. Rinse immediately in large quantities of cold running water.
2. **Optimized FPL Etch Solution (1 liter):**

Material	**Amount**
Distilled Water	700 ml plus balance of liter (see below)
Sodium Dichromate	28 to 67.3 grams
Sulfuric Acid	287.9 to 310.0 grams
Aluminum Chips	1.5 grams/liter of mixed solution

   To prepare 1 liter of this solution, dissolve sodium dichromate in 700 ml of distilled water. Add sulfuric acid and mix well. Add additional distilled water to fill to 1 liter. Heat mixed solution to 66 to 71°C (150 to 160°F). Dissolve 1.5 grams of 2024 bare aluminum chips per liter of mixed solution. Gentle agitation will help aluminum dissolve in about 24 hours.

   To etch aluminum panels, place them in FPL etch solution heated to 66 to 71°C (150 to 160°F). Panels should soak for 12 to 15 minutes.

3. Rinse: Rinse panels in clear running tap water.
4. Dry: Air dry 15 minutes; force dry 10 minutes (minimum) at 140°F (60°C) maximum.
5. If primer is to be used, it should be applied within 4 hours after surface preparation.

Plastics/Rubber

1. Wipe with isopropyl alcohol.*
2. Abrade using fine grit abrasives (180 grit or finer).
3. Wipe with isopropyl alcohol.*

Glass

1. Solvent wipe surface using acetone or MEK.*
2. Apply a thin coating (0.0001 in. or less) of 3M™ Scotch-Weld™ Structural Adhesive Primer EC-3901 to the glass surfaces to be bonded and allow the primer to dry a minimum of 30 minutes @ 75°F (24°C) before bonding.

*When using solvents, extinguish all ignition sources, including pilot lights, and follow the manufacturer's precautions and directions for use. Use solvents in accordance with local regulations.

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Typical Adhesive Performance Characteristics

A. Typical Shear Properties on Etched Aluminum
ASTM D 1002
Cure: 2 hours @ 150 ± 5°F (66°C ± 2°C), 2 psi pressure

Test Temperature	Overlap Shear (psi) 3M™ Scotch-Weld™ Epoxy Adhesive 2216 B/A Gray Adhesive	2216 B/A Tan NS Adhesive	2216 B/A Trans. Adhesive
-423°F (-253°C)	2440	—	—
-320°F (-196°C)	2740	—	—
-100°F (-73°C)	3000	—	—
-67°F (-53°C)	3000	2000	3000
75°F (24°C)	3200	2500	1700
180°F (82°C)	400	400	140

Test Temperature	Shear Modulus (Torsion Pendulum Method)
-148°F (-100°C)	398,000 psi (2745 MPa)
-76°F (-60°C)	318,855 psi (2199 MPa)
-40°F (-40°C)	282,315 psi (1947 MPa)
32°F (0°C)	218,805 psi (1500 MPa)
75°F (24°C)	49,580 psi (342 MPa)

B. Typical T-Peel Strength
ASTM D 1876

Test Temperature	T-Peel Strength (piw) @ 75°F (24°C) 3M™ Scotch-Weld™ Epoxy Adhesive 2216 B/A Gray Adhesive	2216 B/A Tan NS Adhesive	2216 B/A Trans. Adhesive
75°F (24°C)	25	25	25

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Typical Adhesive Performance Characteristics *(continued)*

C. Overlap Shear Strength After Environmental Aging-Etched Aluminum

Environment	Time	Overlap Shear (psi) 75°F (24°C) 3M™ Scotch-Weld™ Epoxy Adhesive 2216 B/A Gray Adhesive	2216 B/A Tan NS Adhesive	2216 B/A Trans. Adhesive
100% Relative Humidity @ 120°F (49°C)	14 days 30 days 90 days	2950 psi 1985 psi 1505 psi	3400 psi 2650 psi	1390 psi
*Salt Spray @ 75°F (24°C)	14 days 30 days 60 days	2300 psi 500 psi 300 psi	3900 psi 3300 psi	1260 psi
Tap Water @ 75°F (24°C)	14 days 30 days 90 days	3120 psi 2942 psi 2075 psi	3250 psi 2700 psi	1950 psi
Air @ 160°F (71°C)	35 days	4650 psi	4425 psi	
Air @ 300°F (149°C)	40 days	4930 psi	4450 psi	3500 psi
Anti-icing Fluid @ 75°F (24°C)	7 days	3300 psi	3050 psi	2500 psi
Hydraulic Oil @ 75°F (24°C)	30 days	2500 psi	3500 psi	2500 psi
JP-4 Fuel	30 days	2500 psi	2750 psi	2500 psi
Hydrocarbon Fluid	7 days	3300 psi	3100 psi	3000 psi

*Substrate corrosion resulted in adhesive failure.

D. Heat Aging of 3M™ Scotch-Weld™ Epoxy Adhesive 2216 B/A Gray
(Cured for 7 days @ 75°F [24°C])

Overlap Shear (psi)	Time aged @ 300°F (149°C)			
Test Temperature	0 days	12 days	40 days	51 days
-67°F (-53°C)	2200	3310	3120	2860
75°F (24°C)	3100	5150	4930	4740
180°F (82°C)	500	1000	760	1120
350°F (177°C)	420	440	560	—

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Typical Adhesive Performance Characteristics (*continued*)

E. Overlap Shear Strength on Abraded Metals, Plastics, and Rubbers.

Overlap shear strengths were measured on 1" x 1/2" overlap specimens. These bonds were made individually using 1" by 4" pieces of substrate (Tested per ASTM D 1002).

The thickness of the substrates were: cold rolled, galvanized and stainless steel – 0.056-0.062", copper – 0.032", brass – 0.036", rubbers – 0.125", plastics – 0.125". All surfaces were prepared by solvent wiping/abrading/ solvent wiping.

The jaw separation rate used for testing was 0.1 in/min for metals, 2 in/min for plastics, and 20 in/min for rubbers.

	Overlap Shear (psi) @75°F (24°C)	
	3M™ Scotch-Weld™ Epoxy Adhesive	
Substrate	**2216 B/A Gray Adhesive**	**2216 B/A Tan NS Adhesive**
Aluminum/Aluminum	1850	2350
Cold Rolled Steel/Cold Rolled Steel	1700	3100
Stainless Steel/Stainless Steel	1900	
Galvanized Steel/Galvanized Steel	1800	
Copper/Copper	1050	
Brass/Brass	850	
Styrene Butadiene Rubber/Steel	200*	
Neoprene Rubber/Steel	220*	
ABS/ABS Plastic	990*	1140*
PVC/PVC, Rigid	940*	
Polycarbonate/Polycarbonate	1170*	1730*
Acrylic/Acrylic	1100*	1110*
Fiber Reinforced Polyester/ Reinforced Polyester	1660*	1650*
Polyphenylene Oxide/PPO	610	610
PC/ABS Alloy / PC/ABS Alloy	1290	1290

*The substrate failed during the test.

Storage

Store products at 60-80°F (16-27°C) for maximum storage life.

Shelf Life

When stored at the recommended temperatures in the original, unopened containers, the shelf life is two years from date of shipment from 3M.

Scotch-Weld™
Epoxy Adhesive
2216 B/A

Precautionary Information

Refer to Product Label and Material Safety Data Sheet for Health and Safety Information before using this product.

Product Use

All statements, technical information and recommendations contained in this document are based upon tests or experience that 3M believes are reliable. However, many factors beyond 3M's control can affect the use and performance of a 3M product in a particular application, including the conditions under which the product is used and the time and environmental conditions in which the product is expected to perform. Since these factors are uniquely within the user's knowledge and control, it is essential that the user evaluate the 3M product to determine whether it is fit for a particular purpose and suitable for the user's method of application.

Warranty and Limited Remedy

Unless stated otherwise in 3M's product literature, packaging inserts or product packaging for individual products, 3M warrants that each 3M product meets the applicable specifications at the time 3M ships the product. Individual products may have additional or different warranties as stated on product literature, package inserts or product packages. 3M MAKES NO OTHER WARRANTIES, EXPRESS OR IMPLIED, INCLUDING, BUT NOT LIMITED TO, ANY IMPLIED WARRANTY OF MERCHANTABILITY OR FITNESS FOR A PARTICULAR PURPOSE OR ANY IMPLIED WARRANTY ARISING OUT OF A COURSE OF DEALING, CUSTOM OR USAGE OF TRADE. User is responsible for determining whether the 3M product is fit for a particular purpose and suitable for user's application. If the 3M product is defective within the warranty period, your exclusive remedy and 3M's and seller's sole obligation will be, at 3M's option, to replace the product or refund the purchase price.

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Except where prohibited by law, 3M and seller will not be liable for any loss or damage arising from the 3M product, whether direct, indirect, special, incidental or consequential, regardless of the legal theory asserted, including warranty, contract, negligence or strict liability.

ISO 9001:2000

This Industrial Adhesives and Tapes Division product was manufactured under a 3M quality system registered to ISO 9001:2000 standards.

3M

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Appendix D - Static Analysis

Shear Stress on the adhesive

The adhesive between the composite panels and the aluminum frame will be subject to shear strain due to temperature differences. A preliminary analysis of the shear stress is given below:

Assumptions:

- ScotchWeld 2216 (B/A) is used as adhesive
- The coefficient of thermal expansion of the composite panels is $\alpha_{comp} = 15\ \mu m\ m^{-1}\ K^{-1}$ (worst case assumption)
- The thickness of the adhesive layer is t=0,25mm
- Minimum temperature is 251K (thermal subsystem)
- The thermal mismatch is zero at the cure temperature of 65°C

The worst case stress will be encountered at the minimum temperature of 251K, which presents the maximum temperature difference. Using the data from Appendix C – Adhesives properties for the adhesives shear modulus (G) and ultimate shear stress (τ_{max}) as well as the coefficient of thermal expansion for the aluminum frame (Appendix B - Material Properties):

- $\alpha_{alu} = 23{,}6\ \mu m\ m^{-1}\ K^{-1}$
- $\alpha_{epoxy} = 45\ \mu m\ m^{-1}\ K^{-1}$
- G = 1,947 GPa (@ -53°C)
- τ_{max} = 20,69 MPa (3000 psi) (@ -40°C)
- L = 100mm

Difference in coefficients of thermal expansion is: $\Delta CTE = 10{,}6\ \mu m\ m^{-1}\ K^{-1}$

Difference in temperatures: ΔT = 251- 338= -87K

Based on the article on multilayer thermal stresses [19], the following considerations can be made. Considering a two phase system of the aluminum substrate and the epoxy adhesive, the epoxy will retract more upon temperature decrease than the aluminum. Since the film is slim and the elasticity modulus of the epoxy is strongly weaker, the aluminum will impose its deformation on the adhesive through the interface (Figure 57). This will induce a additional shear stress in the adhesive.

The deformations are given by:

$$\varepsilon_{alu} = \alpha_{alu}\,\Delta T \qquad \textbf{Equation 2}$$

$$\varepsilon_{epoxy} = \alpha_{epoxy}\,\Delta T \qquad \textbf{Equation 3}$$

The imposed deformation requires that $\varepsilon_{epoxy}' = \varepsilon_{alu}$

The resulting shear stress is thus:

$$\tau_{epoxy} = G_{epoxy}(\alpha_{alu} - \alpha_{epoxy})\Delta T = 1{,}75MPa \qquad \textbf{Equation 4}$$

This result is coherent with the result for a two layer system found by Hsueh.

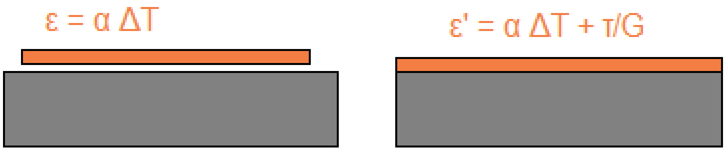

Figure 57 Deformation of the adhesive layer in the mechanically free and bonded case.

Considering the aluminum and the composite both have elasticity modules higher than the epoxy adhesive, we will consider the case where the stresses induced by both substrates will add up. This hypothesis is largely underestimating since the strain will decrease at a certain distance from the interface and thus the two contributions will not add up on the whole distance.

Consequently, the following results where obtained:

τ_{epoxy} = 3,62 MPa (alu) + 5,08 MPa (comp) = 8,7 MPa

Using a security factor of FOS = 1,5 as defined by ECSS standards the margin of security (MOS) is given by:

$$MOS = \frac{\tau_{max}}{\tau \cdot FOS} - 1 = 0{,}59 \qquad \textbf{Equation 5}$$

This margin of security is very small and thus probably other adhesives have to be considered as well. Unfortunately the various Epoxies have a big CTE.

Buckling analysis

In mechanical structures, in addition to the stable elastic deformations, for high stresses there is a risk that the deformation of the structure becomes mechanically unstable. This phenomenon is called buckling.

The differential equation for the buckling of a column is given by Euler formula:

$$\frac{\partial^2 \delta}{\partial z^2} = -\frac{P_{cr}}{EI}\delta \qquad \textbf{Equation 6}$$

where δ is the lateral displacement and z is the coordinate along the length of the column. E is the columns young modulus and P_{cr} the critical load and I the moment of inertia in the plane perpendicular to z. This permits the solution:

$$\delta = C_1 \cos\left(z\sqrt{\frac{P_{cr}}{EI}}\right) + C_2 \cos\left(z\sqrt{\frac{P_{cr}}{EI}}\right) \qquad \textbf{Equation 7}$$

with boundary conditions given by the restraints of the column. For the simply supported case that we are considering the boundary conditions are δ = 0 at z = 0 and z = L. Thus C_1 = 0 and for a non trivial solution:

$$\frac{P_{cr}L^2}{EI} = k^2\pi^2 \qquad \textbf{Equation 8}$$

where k is a positive integer.

This can be re-arranged to give the critical buckling load of a column:

$$P_{cr} = \frac{k^2\pi^2 EI}{L^2} \qquad \textbf{Equation 9}$$

The only mode of buckling observed in practice is the first mode (n=1), occurring at the lowest loads. The critical buckling stress is given by the following expression:

$$\sigma_{cr} = \frac{P_{cr}}{A} = \frac{\pi^2 EI}{A{L_e}^2} \qquad \textbf{Equation 10}$$

L_e is the effective length of the column. The effective length is the length of a simply supported column that would have the same critical load as that of a column of length L but with different boundary conditions. In this case we are going to use a factor of K=1 for the expression L = K L_e which is also a worst case assumption.

Values used:

E = 72 GPa

Rail:	a = 8.5mm	d = 6,5mm	A= 39mm^2	L_e = 94mm
Crossbar:	b = 3mm,	h = 4mm,	A= 12mm^2	L_e = 83mm

(the chamfers on the rails are neglected)

The inertia moment for the rails can be calculated by (REF):

$$I = \frac{a^4}{12} - \frac{\pi r^4}{4} = 347{,}4mm^4$$ **Equation 11**

where r is the radius of the hole and a is the side length

The inertia moment of a rectangular section (crossbar) can be calculated by :

$$I = \frac{bh^3}{12} = 16mm^4$$ **Equation 12**

where b and h are the length and width of the section

Using formula 10 we obtain the following critical stresses:

Rail: σ_{cr} = **716.4MPa**

Crossbar: σ_{cr} = **137.5MPa**

Appendix E - Finite Elements Analysis

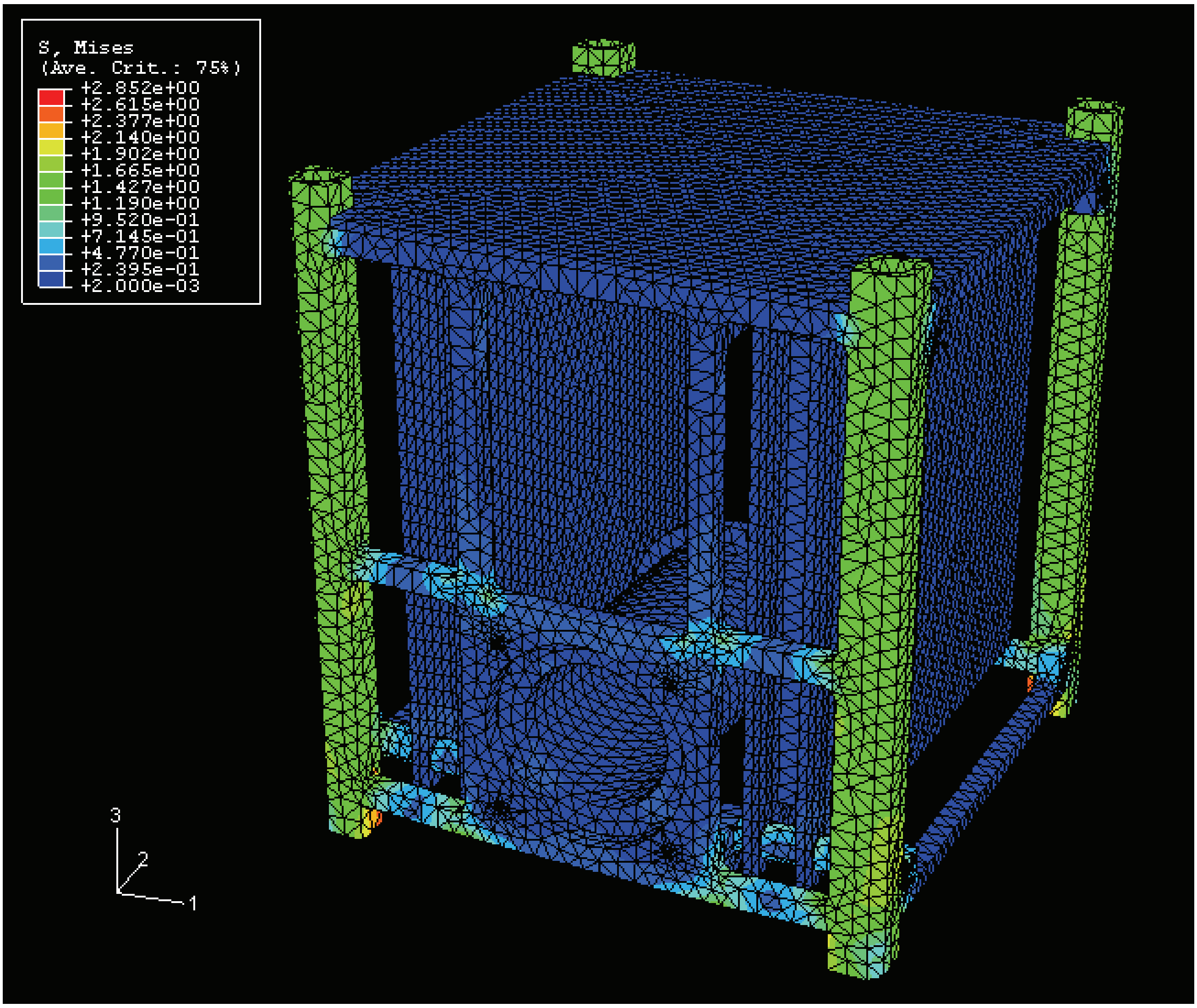

Figure 58 Von Mises in the vertical worst case.

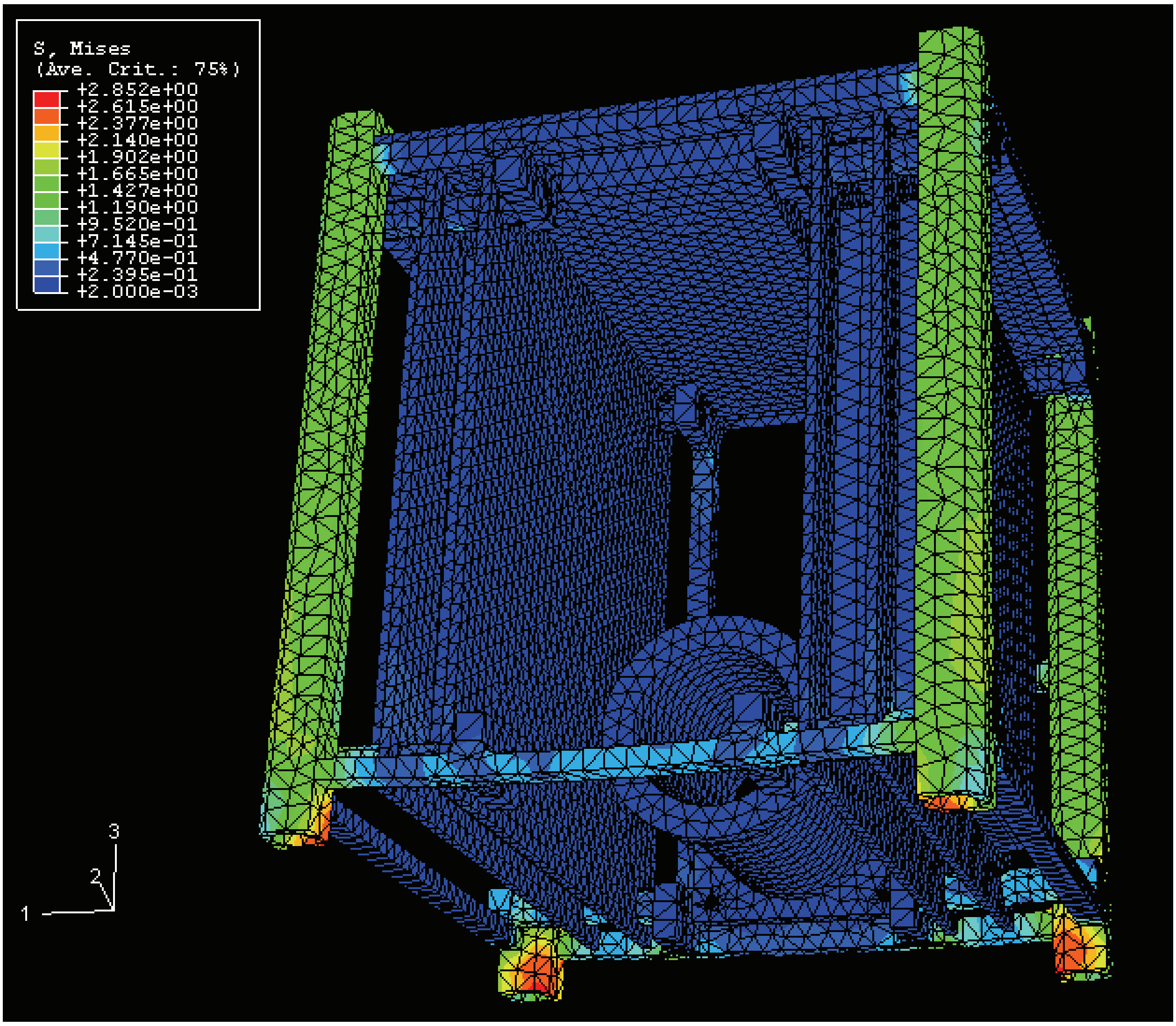

Figure 59 Von Mises in the vertical worst case.

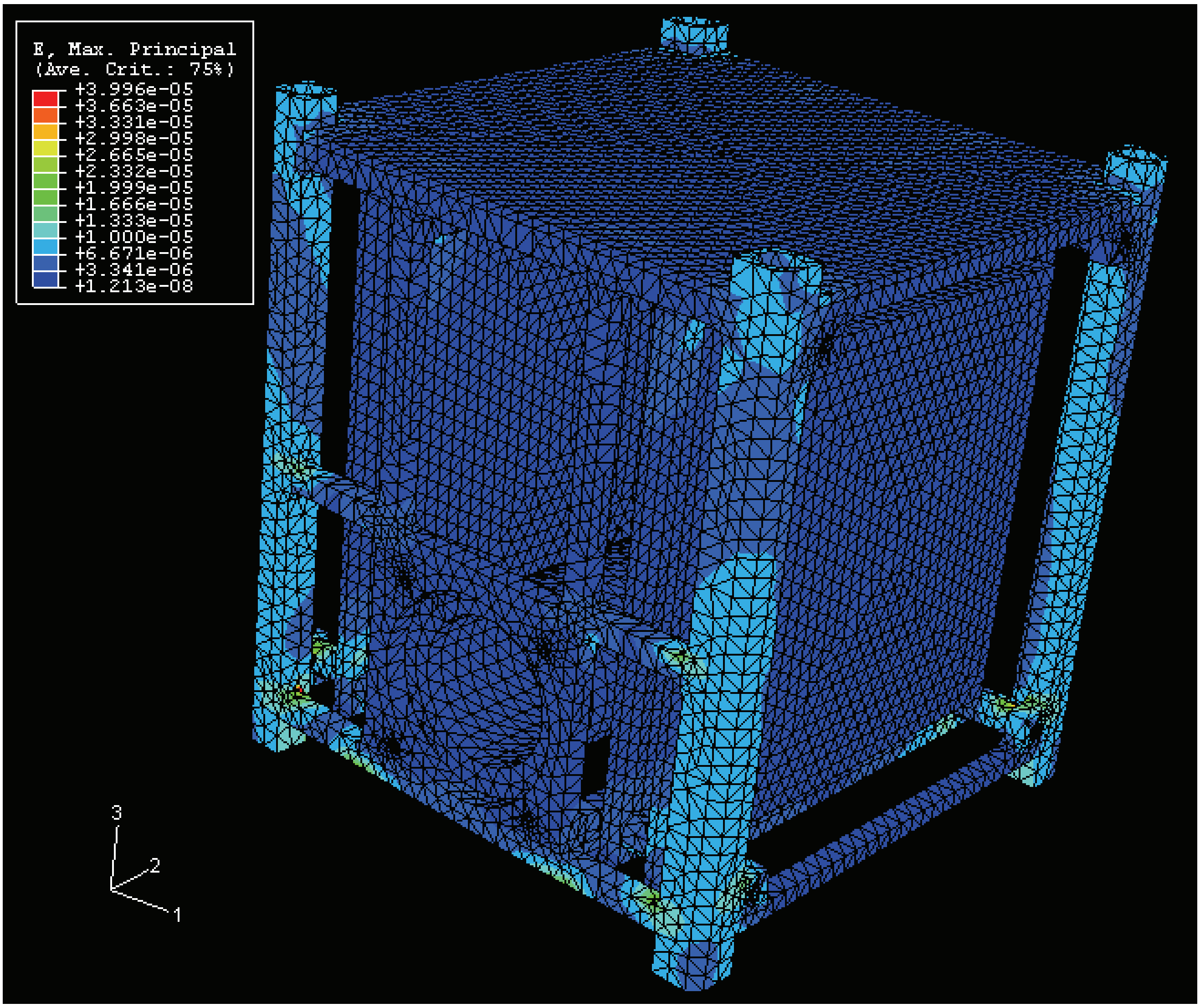

Figure 60 Strain in the vertical worst case.

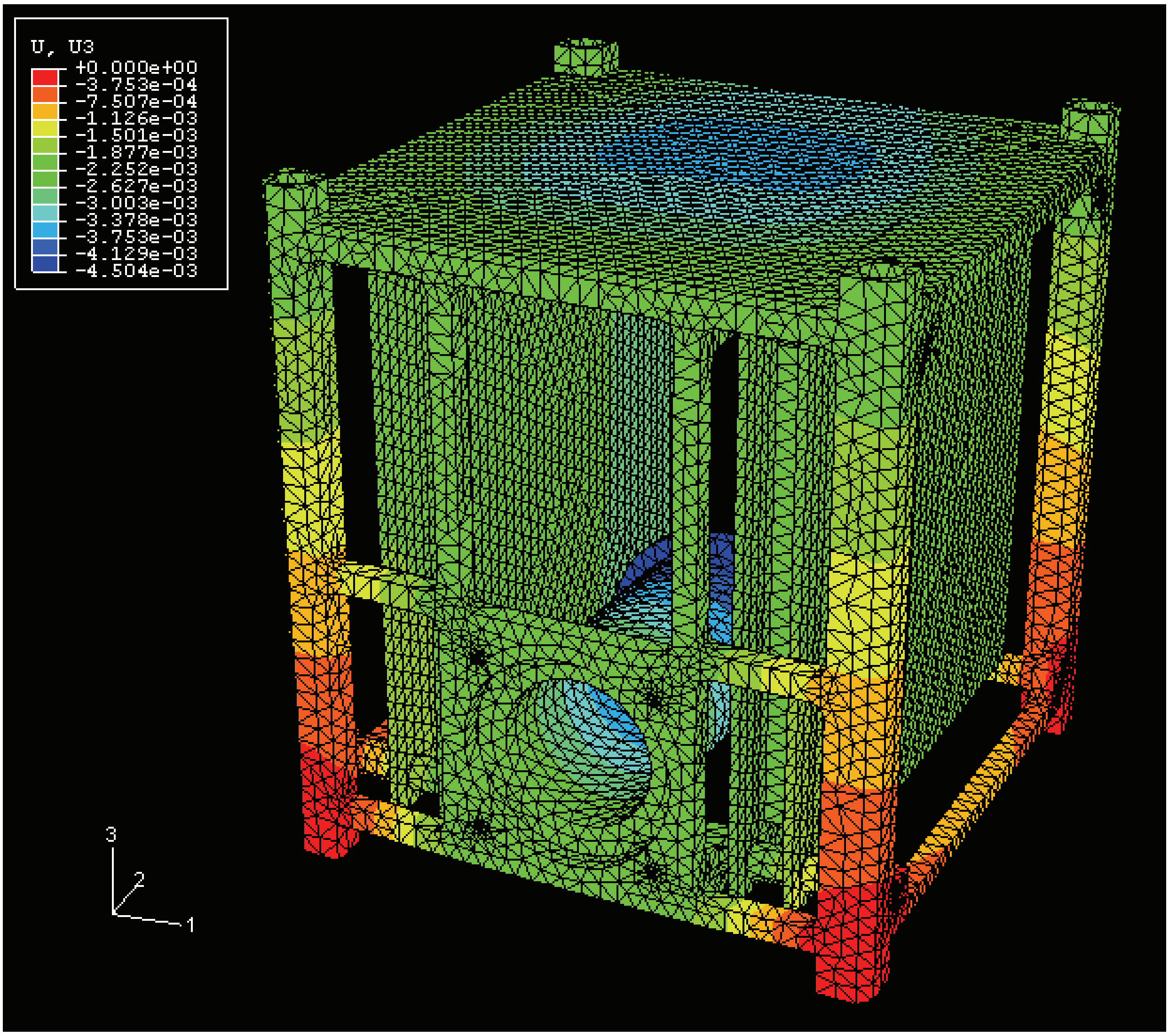

Figure 61 Displacement in axis 3 in the vertical worst case.

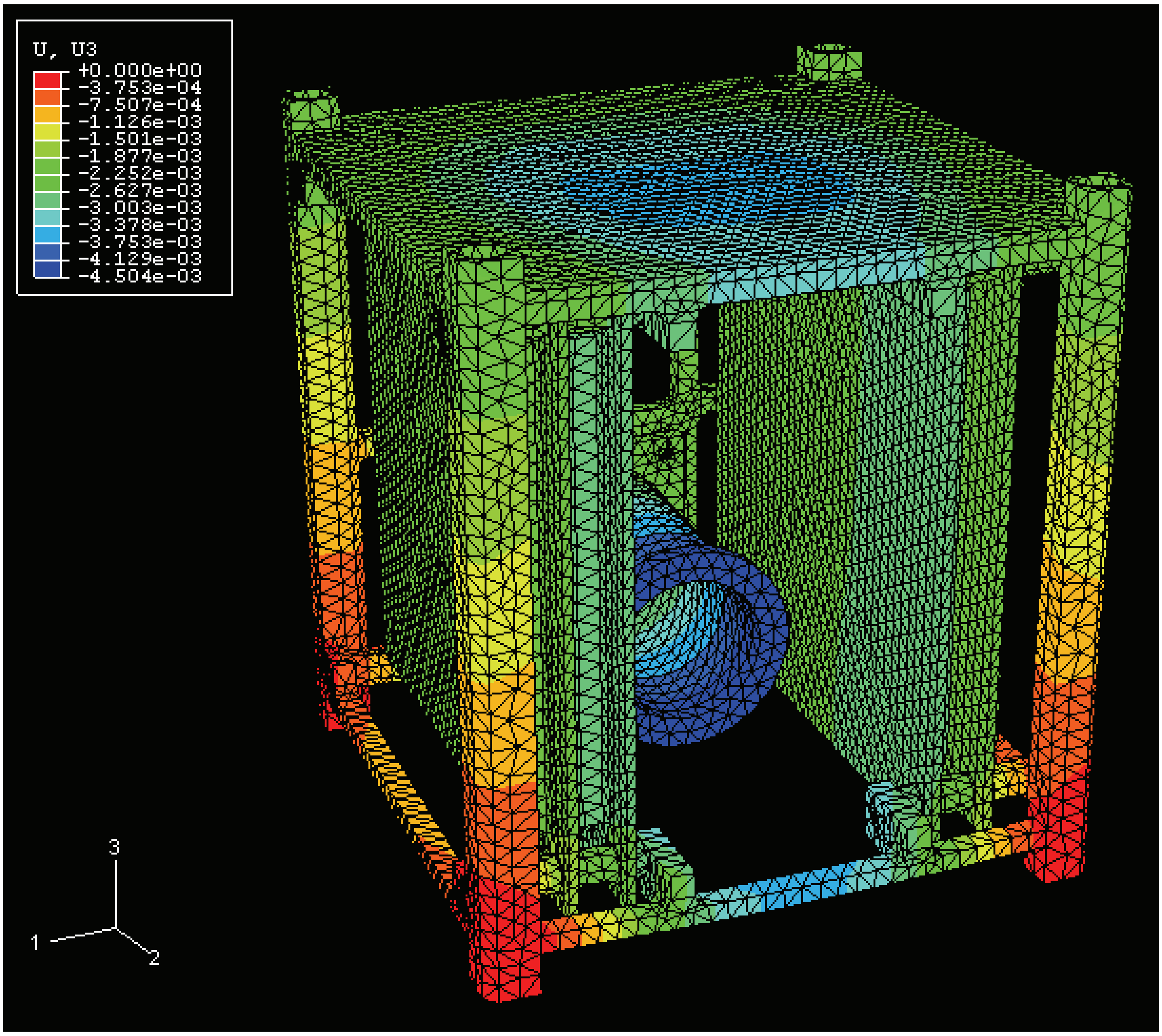

Figure 62 Displacement in axis 3 in the vertical worst case.

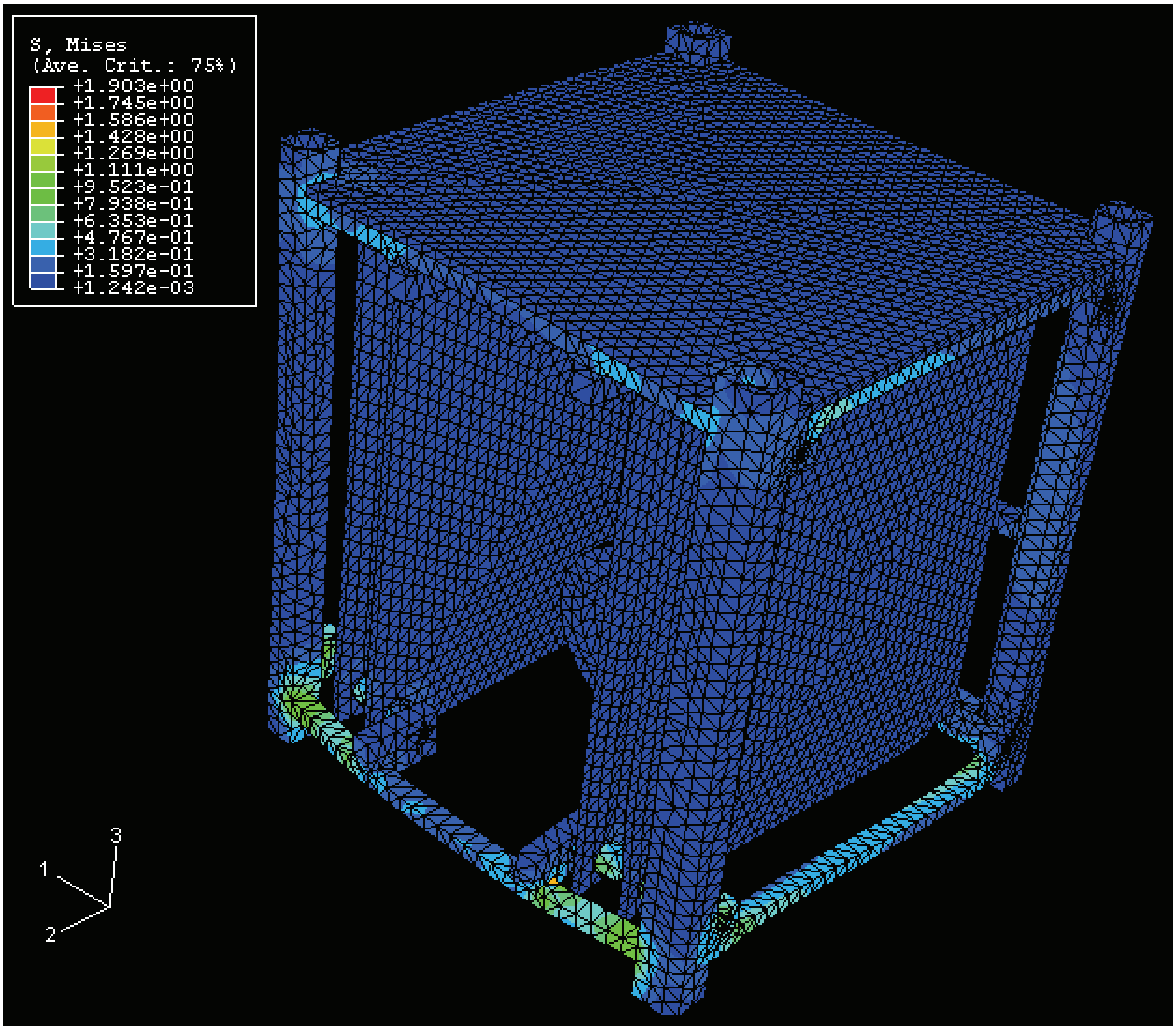

Figure 63 Von Mises in the horizontal worst case.

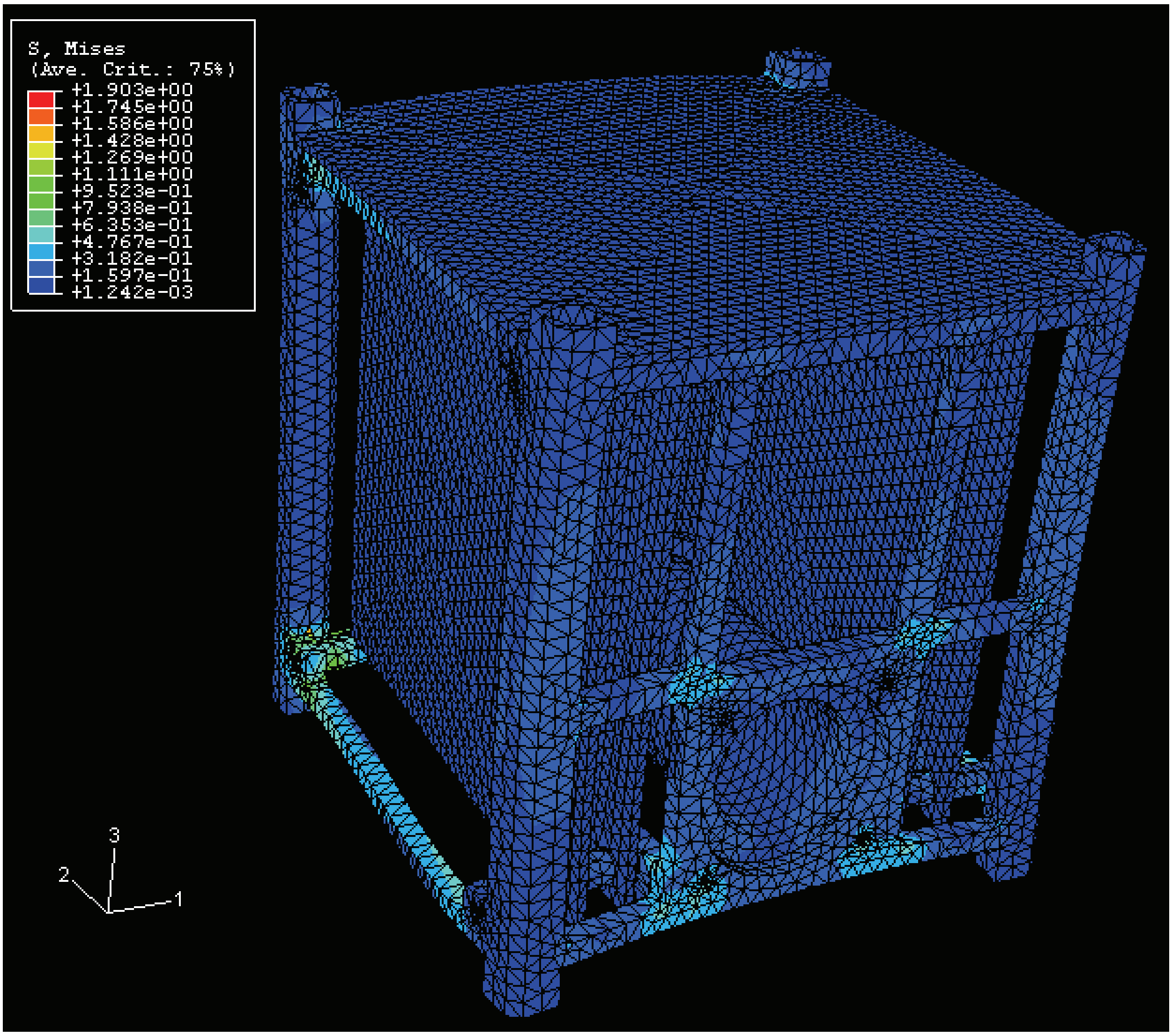

Figure 64 Von Mises in the horizontal worst case.

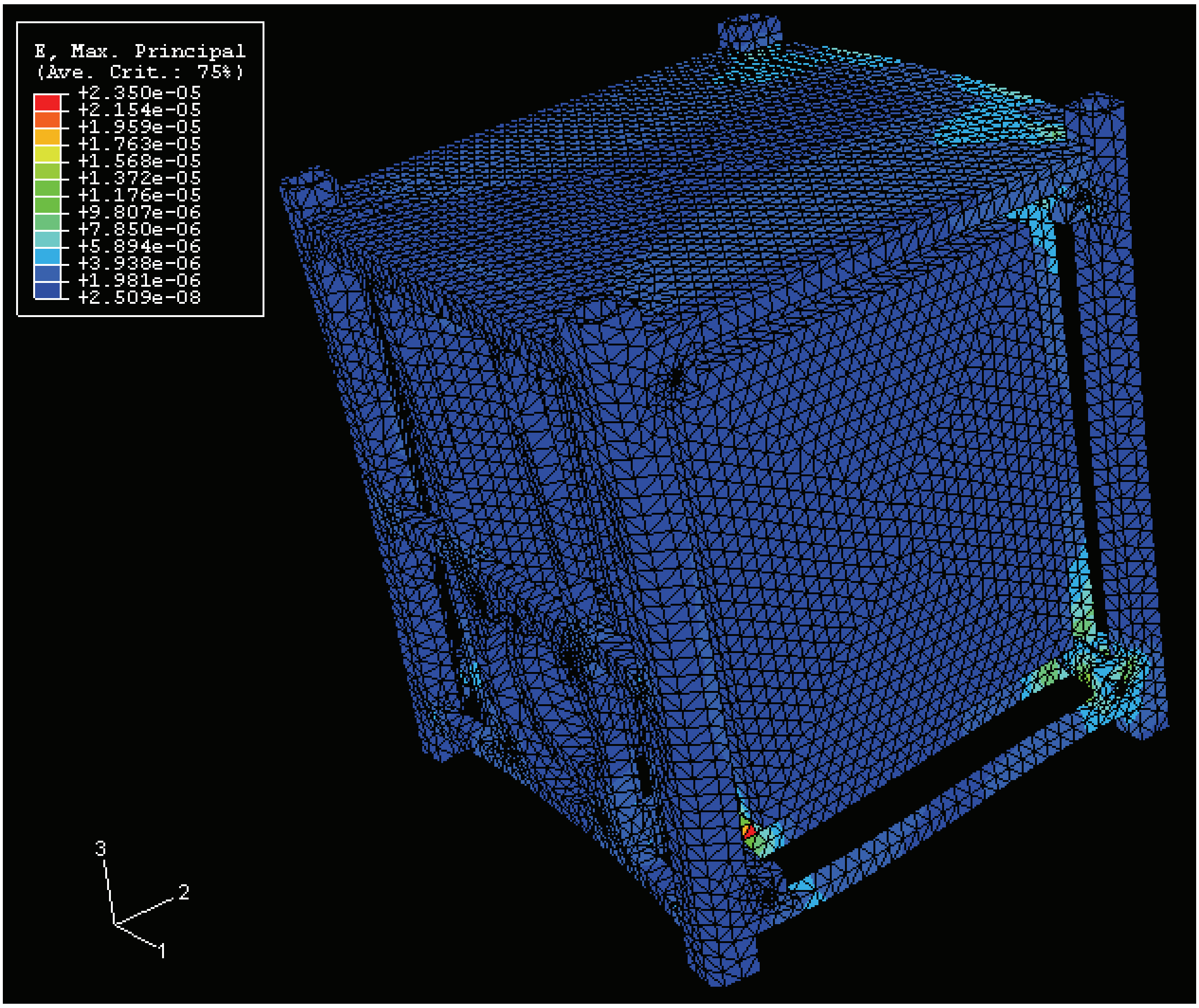

Figure 65 Strain in the horizontal worst case.

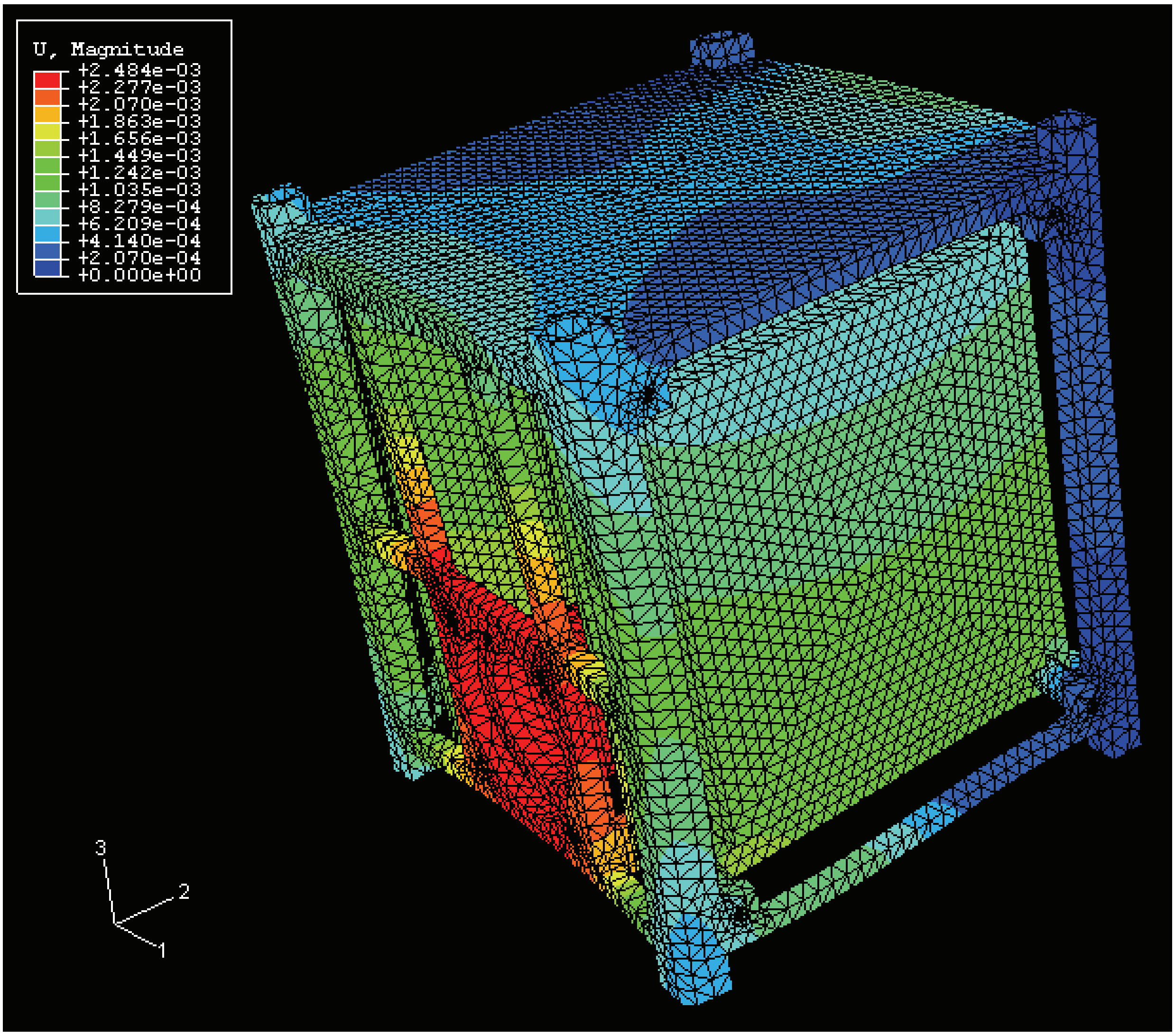

Figure 66 Displacement in the horizontal worst case.

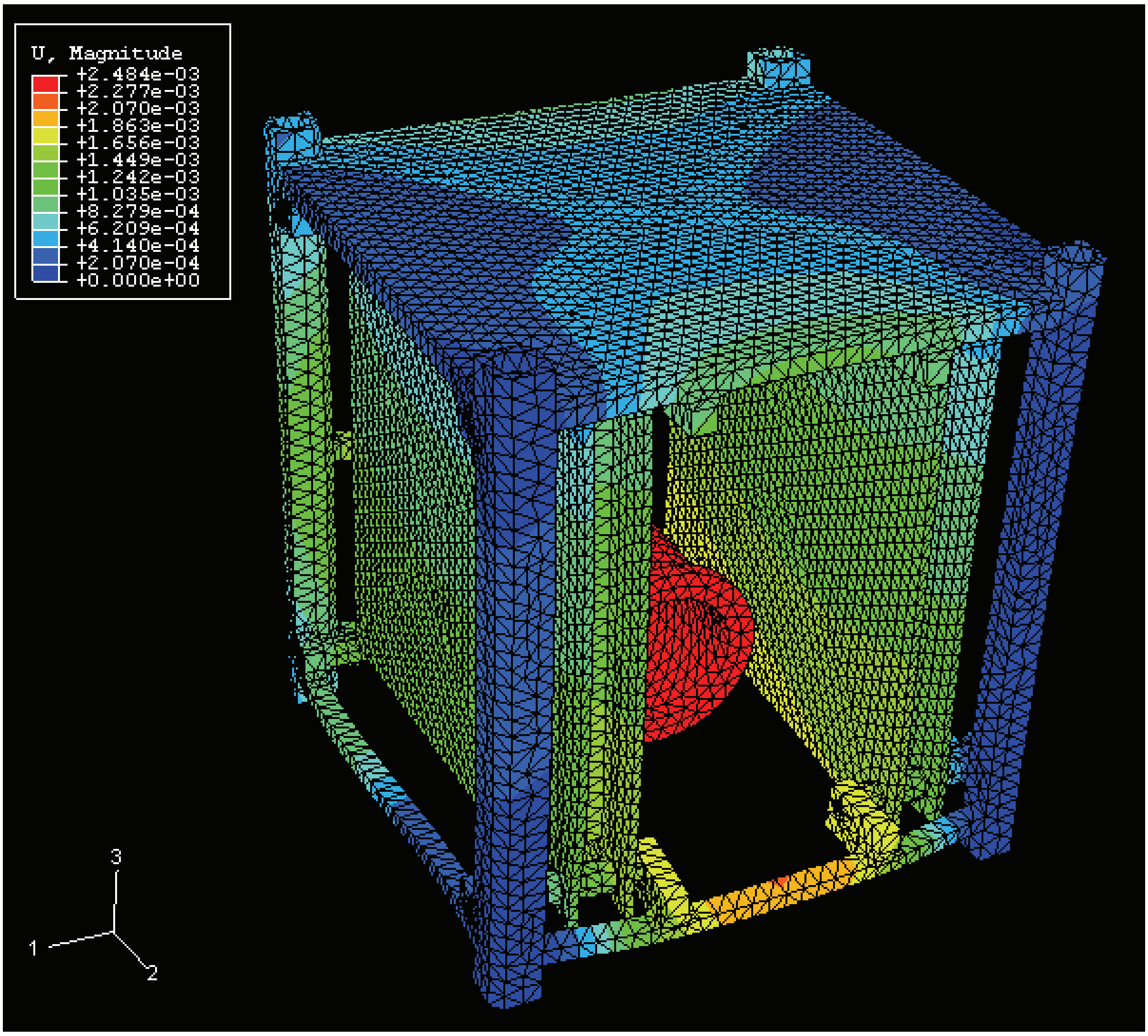

Figure 67 Displacement in the horizontal worst case.

Appendix F – Launchers properties

VEGA

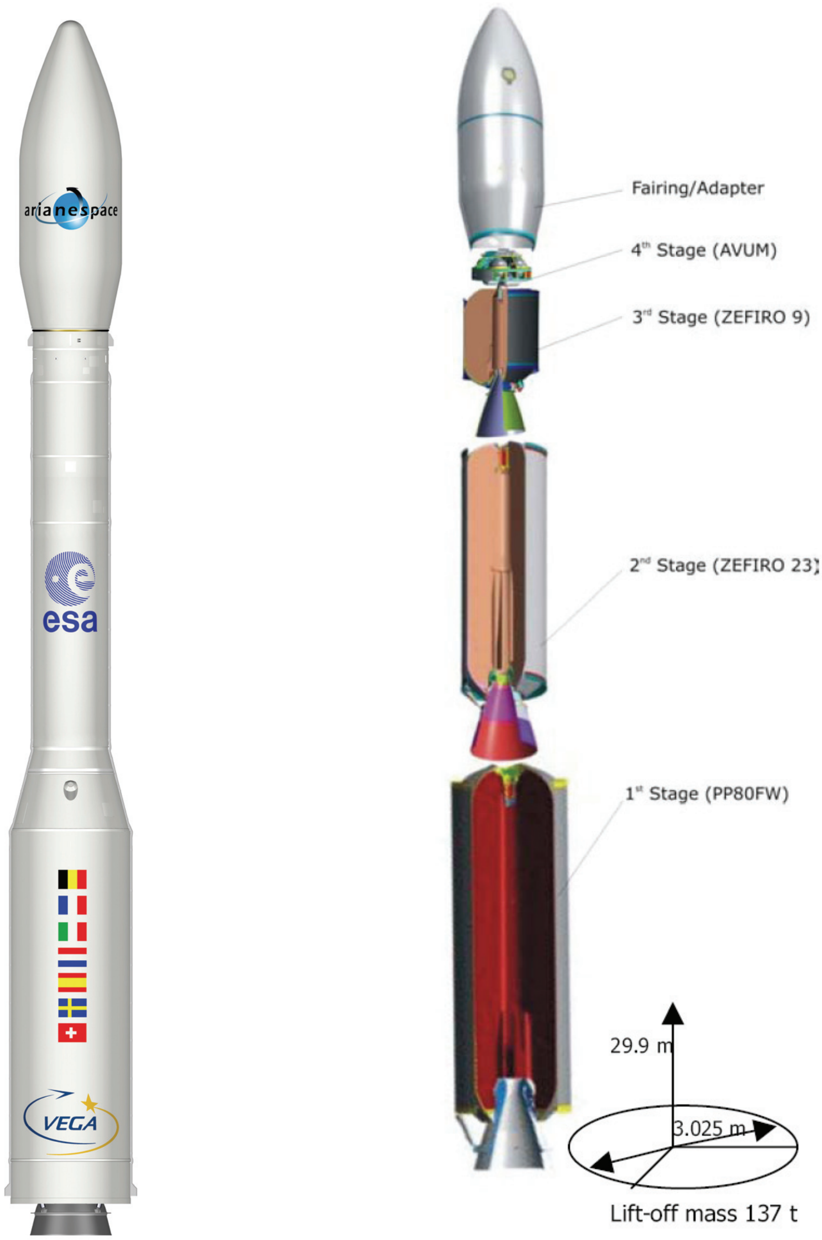

Figure 68 Views of the VEGA launcher.

Note: Power Spectral Densities (PSD) at high and low levels are not available.

The next pages come from the document VEGA User's Manual [11].

PAYLOAD FAIRING	
Fairing	
Diameter:	2.600 m
Length:	7.880 m
Mass:	490 kg
Structure:	Two halves - Sandwich panels CFRP sheets and aluminum honeycomb core
Acoustic protection:	Thick foam sheets covered by fabric
Separation	Vertical separations by means of leak-proof pyrotechnical expanding tubes and horizontal separation by a clamp band

PAYLOAD ADAPTERS	
Off-the-shelf devices:	Clampband, Ø937 (60 kg);

DUAL CARRYING STRUCTURE	
Off-the-shelf devices:	Under development

MINI SATELLITE CARRYING STRUCTURE	
Off-the-shelf devices:	ASAP Plate type (TBD kg);

AVUM UPPER STAGE	
Size:	2.18-m diameter × 2.04-m height
Dry mass:	418 kg (TBC)
Propellant:	367-kg/183-kg of N_2O_4/UDMH
Subsystems:	
Structure:	Carbon-epoxy cylindrical case with 4 aluminum alloy propellant tanks and supporting frame
Propulsion	RD-869 - 1 chamber
- Thrust	2.45 kN - Vac
- Isp	315,5 s - Vac
- Feed system	regulated pressure-fed, 87l (3,72 kg) GHe tank MEOP 310 bar
- Burn time/ restart	Up to 667 s / up to 5 controlled or depletion burn
Attitude Control	
- pitch, yaw	Main engine 9 deg gimbaled nozzle or four 50-N GN_2 thrusters
- roll	Two 50-N GN_2 thrusters
- propellant	GN_2; 87l (26 kg) GN_2 tank MEOP 6 / 36 bar
Avionics	Inertial 3-axis platform, on-board computer, TM & RF systems, Power

.	1st STAGE	2nd STAGE (CORE)	3rd STAGE
Size:	3.00-m diameter × 11.20-m length	1.90-m diameter × 8.39-m length	1.90-m diameter × 4.12-m length
Gross mass:	95 796 kg	25 751 kg	10 948 kg
Propellant:	88 365-kg of HTPB 1912 solid	23 906-kg of HTPB 1912 solid	10 115-kg of HTPB 1912 solid
Subsystems:			
Structure	Carbon-epoxy filament wound monolithic motor case protected by EPDM	Carbon-epoxy filament wound monolithic motor case protected by EPDM	Carbon-epoxy filament wound monolithic motor case protected by EPDM
Propulsion	P80FW Solid Rocket Motor (SRM)	ZEFIRO 23 Solid Rocket Motor	ZEFIRO 9 Solid Rocket Motor
- Thrust	2261 kN – SL	1196 kN – SL	225 kN - Vac (TBC)
- Isp	280 s – Vac	289 s – Vac	295 s – Vac (TBC)
- Burn time	106,8 s	71,7 s	109.6 s
Attitude Control	Gimbaled 6.5 deg nozzle with electro actuator	Gimbaled 7 deg nozzle with electro actuator	Gimbaled 6 deg nozzle with electro actuator
Avionics		Actuators I/O electronics, power	Actuators I/O electronics, power
Interstage/Equipment bay:	**0/1 interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: Actuators I/O electronics, power		
	1/2 interstage: Structure: conical aluminum shell/inner stiffeners Housing: TVC local control equipment; Safety/Destruction subsystem	**2/3 interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: TVC local control equipment; Safety/Destruction subsystem	**3/AVUM interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: TVC control equipment; Safety/Destruction subsystem, power distribution, RF and telemetry subsystems
Stage separation:	Linear Cutting Charge/Retro rocket thrusters	Linear Cutting Charge/Retro rocket thrusters	Clamp-band/ springs

Figure 69 VEGA Characteristics.

Spacecraft design and Verification requirements — Chapter 4

4. - Spacecraft design and verification requirements

4.1. Introduction

The design and dimensioning data that shall be taken into account by any Customer intending to launch a spacecraft compatible with the Vega launch vehicle are detailed in this chapter.

4.2. Design requirements

4.2.1. Safety Requirements

The Customer is required to design the spacecraft in conformity with the CSG Safety Regulations.

4.2.2. Selection of spacecraft materials

The spacecraft materials must satisfy the following outgassing criteria:

- Total Mass Loss (TML) ≤ 1 %;
- Collected Volatile Condensable Material (CVCM) ≤ 0.1 %.

measured in accordance with the procedure "ECSS-Q-70-02A".

4.2.3. Spacecraft Properties

The following is applicable for a single launch configuration. In case of multiple launch configuration, the specific requirement applicable to the each spacecraft will be defined by Arianespace.

4.2.3.1. Payload mass and CoG limits

Spacecraft (payload) mass and position of CoG shall comply with a limitation for static moment applied on the spacecraft to adapter interface.

These limits are presented in Figure 4.1. If outside the limits, please contact Arianespace.

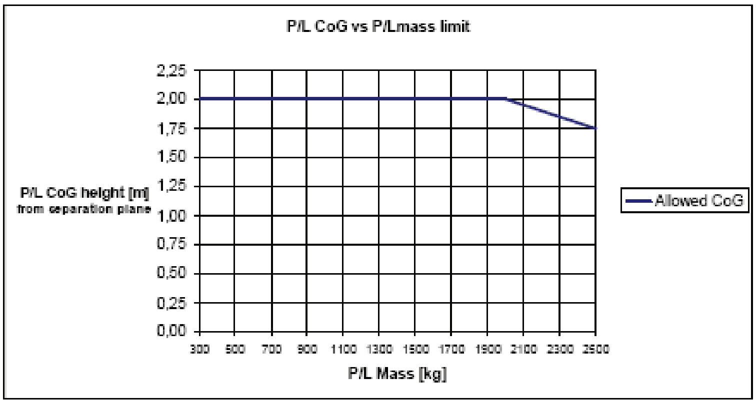

Figure 4.1 – Limits of payload CoG position versus payload mass.

4.2.3.2. Static unbalance

In the LV axis system O, X, Y, Z, the spacecraft CoG must located within a distance:

- less than 15 mm from the longitudinal axis OX for the spin-up stabilisation and
- less than 30 mm for 3-axis stabilization.

Should Spacecraft be outside this domain, please contact Arianespace.

4.2.3.3. Dynamic unbalance

There is no predefined requirement for spacecraft dynamic balancing with respect to ensuring proper operation of the LV. However, these data have a direct effect on spacecraft separation.

To ensure the separation conditions in spin-up mode described in the Chapter 2, the maximum spacecraft dynamic unbalance ε corresponding to the angle between the spacecraft longitudinal geometrical axis and the principal roll inertia axis shall be: $\varepsilon \leq 1$ degree.

Should Spacecraft be outside this figure, please contact Arianespace.

4.2.3.4. Frequency Requirements

To prevent dynamic coupling with fundamental modes of the LV, the spacecraft should be designed with a structural stiffness which ensures that the following requirements are fulfilled. In that case the design limit load factors given in next paragraph are applicable.

The cantilevered fundamental mode frequencies of a spacecraft hard-mounted at the interface with an off-the shelf adapter must be:

In lateral axis:

$\geq$ 15 Hz for spacecraft mass $\leq$ 2500 kg

In longitudinal axis:

20 Hz $\leq$ F $\leq$ 45 Hz for spacecraft mass $\leq$ 2500 kg

The cumulated effective mass associated to the longitudinal modes within the above frequency range must exceed 60% of the total mass.

In case of concern with the above values, please contact Arianespace.

4.2.4. Dimensioning Loads

4.2.4.1. The design load factors

The design and dimensioning of the spacecraft primary structure and/or evaluation of compatibility of existing spacecraft with Vega launch vehicle shall be based on the design load factors.

The design load factors are represented by the quasi-static g-loads that are the more severe combinations of dynamic and steady-state accelerations that can be encountered at any instant of the mission (ground and flight operations).

The QSL reflects the line load (sometimes named mechanical fluxes, Φ) at the interface between the spacecraft and the adapter.

The flight limit of the QSL for a spacecraft launched on Vega and complying with the previously described frequency requirements and with the static moment limitation are given in the Table 4.1.

Table 4.1 – Flight Limit levels - Quasi Static Loads

Load Event		QSL (g) (+ = tension; – = compression)					
		Longitudinal			Lateral		
		Static	Dynamic	Total	Static	Dynamic	Total
1	Lift-off phase	– 1.5	± 2.0	min – 3.5 max+ 0.5	-	-	± 0.9
2	Flight with maximum dynamic pressure (Qmax)	– 2.5	± 0.5	min – 3.0 max – 2.0	-	-	± 0.9
3	First-stage flight with maximal acceleration	– 4.5	± 0.5	min – 5.0 max – 4.0	-	-	± 0.5
4	Third stage maximal acceleration	- 4.5 See note 1	± 0.2	min – 4.7 max – 4.3	-	-	± 0.2
6	Stages ignition	-	- 5.0 + 3.0	min – 5.0 max+ 3.0	-	-	± 0.2

Note 1:

This value depends on the payload mass according to the following law:

$$QSL\ (g) = 5.8 - M\ (kg) / 1000$$

Note 2:

- The factors apply on payload CoG,
- The minus signs indicate compression along the longitudinal axis and the plus signs tension,
- Lateral loads may act in any direction simultaneously with longitudinal loads,
- The gravity load is included,

4.2.4.2. Line loads peaking

The geometrical discontinuities and differences in the local stiffness of the LV (stiffeners, holes,…) and the non-uniform transmission of the launch vehicle thrust at the spacecraft/adapter interface may produce local variations of the uniform line loads distribution.

Peaking loads induced by the Launch Vehicle:

The integral of the variations along the circumference is zero and line loads derived from the QSL are not affected, but for the correct dimensioning of the lower part of the spacecraft this excess shall be taken into account, and has to be added uniformly at the S/C-adapter interface to the launch vehicle mechanical fluxes obtained for the various flight events.

Such local over line loads are specific of the adapter design.

For the Ø 937 adapter a value lower than 10% over the line loads seen by the spacecraft is assumed, with a minimum value of 5 N/mm.

Peaking loads induced by spacecraft:

The maximum value of the picking load induced by the spacecraft is allowed in local areas to be up to 10% over the dimensioning flux seen by adapter under limit load conditions. An adapter mathematical model can be provided to assess these values.

4.2.4.3. Handling loads during ground operations

During the encapsulation phase, the spacecraft is lifted and handled with its adapter, for this reason the spacecraft must be capable of supporting an additional mass of 100 kg. The crane characteristics, velocity and acceleration are defined in the EPCU User's Manual.

4.2.4.4. Dynamic loads

The secondary structures and flexible elements (e.g., solar panels, antennas, and propellant tanks) must be designed to withstand the dynamic environment described in Chapter 3 and must take into account the safety factors defined in paragraph 4.3.2.

4.3. Spacecraft compatibility verification requirements

4.3.1. Verification Logic

The spacecraft authority shall demonstrate that the spacecraft structure and equipments are capable of withstanding the maximum expected launch vehicle ground and flight environments.

The spacecraft compatibility must be proven by means of adequate tests. The verification logic with respect to the satellite development program approach is shown in Table 4.2.

Table 4.2 – Spacecraft verification logic for structural tests

S/C development approach	Model	Static	Sine vibration	Acoustic	Shock
With Structural Test Model (STM)	STM	Qual test	Qual test	Qual test	Clamp-band release test **
	FM1	By heritage *	Protoflight test	Protoflight test	Clamp-band release test **
	Subsequent FM's	By heritage *	Acceptance test (optional)	Acceptance test	Clamp-band release test
With ProtoFlight Model	PFM = FM1	Qual test or by heritage *	Protoflight test	Protofligt test	Clamp-band release test **
	Subsequent FM's	By heritage *	Acceptance test (optional)	Acceptance test	Clamp-band release test
Recurrent S/C	FM	By heritage *	Acceptance test (optional)	Acceptance test	Clamp-band release test

Note:

* If qualification is claimed "by heritage", the representativeness of the structural test model (STM) with respect to the actual flight unit must be demonstrated.

** The clamp-band release test with STM should be realized with proper and detailed instrumentation

The mechanical environmental test plan for spacecraft qualification and acceptance shall comply with the requirements presented hereafter and shall be reviewed by Arianespace prior to implementation of the first test.

Also, it is suggested, that Customers will implement tests to verify the susceptibility of the spacecraft to the thermal and electromagnetic environment and will tune, by these way, the corresponding spacecraft models used for the mission analysis.

4.3.2. Safety factors

Spacecraft qualification and acceptance test levels are determined by increasing the design load factors (the flight limit levels) — which are presented in Chapter 3 and Chapter 4 — by the safety factors given in Table 4.3. The spacecraft must have positive margins of safety for yield and ultimate loads.

Table 4.3 - Test Factors, rate and duration

SC tests	Qualification		Protoflight		Acceptance	
	Factors	Duration/ Rate	Factors	Duration/ Rate	Factors	Duration/ Rate
Static (QSL)	1.25 ultimate 1.1 yield	N/A	1.25 ultimate 1.1 yield	N/A	N/A	N/A
Sine vibrations	1.25	2 oct/min	1.25	4 oct/min	1.0	4 oct/min
Acoustics	1.41 (or +3 dB)	120 s	1.41 (or +3 dB)	60 s	1.0	60 s
Shock	N/A	2 releases	N/A	2 releases	N/A	1 release

4.3.3. Spacecraft compatibility tests

4.3.3.1. Static tests

Static load tests (in the case of a STM approach) are performed by the customer to confirm the design integrity of the primary structural elements of the spacecraft platform. Test loads are based on worst-case conditions — i.e., on events that induce the maximum mechanical fluxes into the main structure, derived from the table of maximum QSLs and taking into account the additional line loads peaking.

The qualification factors given previously shall be considered.

4.3.3.2. Sinusoidal vibration tests

The objective of the sine vibration tests is to verify the spacecraft secondary structure dimensioning under the flight limit loads multiplied by the appropriate safety factors.

The spacecraft qualification test consists of one sweep through the specified frequency range and along each axis.

Flight limit amplitudes are specified in Chapter 3 and are applied successively on each axis. The tolerance on sine amplitude applied during the test is ± 10%.

A notching procedure may be agreed on the basis of the latest coupled loads analysis (CLA) available at the time of the tests to prevent excessive loading of the spacecraft structure. However, it must not jeopardize the tests objective to demonstrate positive margins of safety with respect to the flight loads.

Table 4.4 – Sinusoidal vibration tests levels

Direction	Longitudinal		Lateral		Sweep rate (octave/min)
Frequency Band (Hz)	5 – 45	45 – 100	5 – 25	25 – 100	
Qualification levels (g)	1.0	1.25	1.0	0.62	2
Acceptance levels (g)	0.8	1.0	0.8	0.5	4

4.3.3.3. Acoustic vibration tests

Acoustic testing is accomplished in a reverberant chamber applying the flight limit spectrum provided in Chapter 3 and increased by the appropriate safety factors. The volume of the chamber with respect to that of the spacecraft shall be sufficient so that the applied acoustic field is diffuse. The test measurements shall be performed at a minimum distance of 1 m from spacecraft.

Table 4.5 – Acoustic vibration test levels

Octave Center Frequency (Hz)	Qualification level	Acceptance level	Test Tolerance
	(reference: 0 dB = 2 x 10^{-5} Pa)		
31.5	127	124	-2, +4
63	132	129	-1, +3
125	138	135	-1, +3
250	135	132	-1, +3
500	134	131	-1, +3
1000	123	120	-1, +3
2000	103	100	-1, +3
OASPL (20 – 2828 Hz)	141.5	138.5	-1, +3
Test duration (s)	120	60	

These values do not take into account any fill factor correction, in case of concern please contact Arianespace.

4.3.3.4. Shock compatibility verification

The verification of the spacecraft's ability to withstand the separation shock generated by the LV shall be based on one of the two following methods:

Qualification by release test.

For qualification, one clamp-band release test is conducted with the tension of the belt set as close as possible to its maximum value.

This test can be performed on the STM, on the PFM, or on the first flight model provided that the spacecraft structure close to the interface as well as the equipment locations and associated supports are equivalent to those of the flight model.

Acceptance test

The acceptance test consists of performing one clamp-band release under nominal conditions (nominal tension of the band, etc.). This single release test is usually performed at the end of the mechanical fit-check (see Chapter 5). The flight type adapter with the associated separation systems and consumable items can be provided in support of these shock tests.

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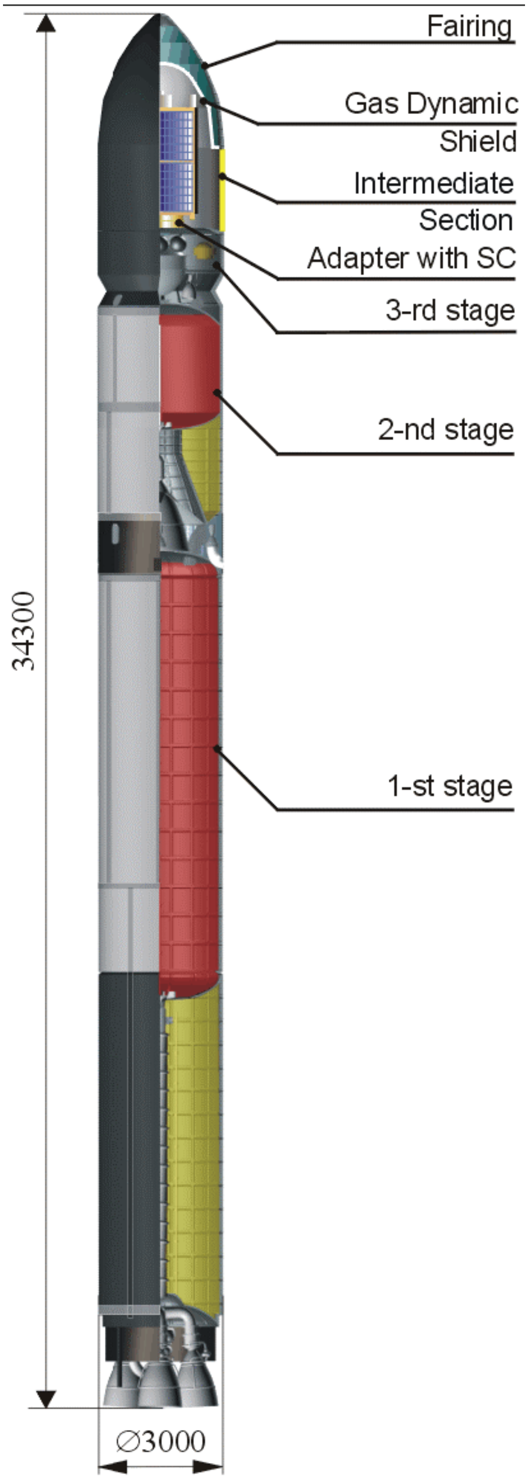

Figure 70 Dnepr-1 General View

Table 12 Dnepr-1 Main Characteristics

Liftoff mass (with the spacecraft mass of 2000 kg), kg	
1st stage	208900
2nd stage	47380
3rd stage	6266
Thrust in vacuum, tons	
1st stage	461.2
2nd stage	77.5
3rd stage (primary mode/throttled back operation mode)	1.9/0.8
Propellant components for all stages	
Oxidizer	Amyl
Fuel	Heptyl
Effective propellant capacity, kg	
1st stage	147900
2nd stage	36740
3rd stage	1910
Flight reliability	0.97
SC injection accuracy (Orbit altitude H_{cir}=300 km)	
for orbit altitude , km	±4.0
period of revolution, sec.	± 3.0
for inclination, degrees	±0.04
for ascending node right ascension, degrees	±0.05
Orbit Inclination, degrees	50.5°; 64.5°; 87.3°; 98°

The next pages come from the document DNEPR User's Guide [18].

9. Spacecraft Environments

9.1 Stiffness Criteria (Frequency Requirements)

The spacecraft should be designed with a structural stiffness, which ensures that the values of fundamental frequency of the spacecraft, hard mounted at the separation plane, are not less than:

- 20 Hz in the longitudinal axis; and
- 10 Hz in the lateral axis.

If it is not possible to comply with the above requirements, SDB Yuzhnoye will carry out an additional analysis of the LV dynamic characteristics and loads that will take into account the spacecraft fundamental frequencies.

9.2 Quasi-static and Dynamic Loads

Tables 9.2-1 and 9.2-2 contain quasi-static and dynamic components of accelerations that act on the SC/LV interface during the ground handling, launch and in-flight.

Spacecraft dimensioning and testing must take into account safety factors, which are defined by the spacecraft authority, but should be no less than the values given below:

- 2.0 for ground handling;
- 1.5 during launch while LV is moving inside the TLC;
- 1.3 during launch after the LV exits from the TLC;
- 1.3 during the LV flight.

The spacecraft should remain operable after the effect of the above accelerations.

9.3 Vibration Loads

Described below are vibrations acting on the Spacecraft attachment points during the LV flight. Two types of vibrations are as follows:

- Harmonic oscillations; and
- Random vibrations.

The harmonic oscillations are characterized by the amplitude of vibro-accelerations and frequency. The parameters of harmonic oscillations are given in Tables 9.3-1 and 9.3-2.

The random vibrations are characterized by spectral density of vibro-accelerations and the duration of influence. The random vibration parameters are given in Table 9.3-3.

The random vibrations are spatial with approximately equal intensity of vibro-accelerations in each of the three randomly selected mutually perpendicular directions.

The values of amplitude and spectral densities are given in the extreme octave points. The change of these values within the limits of each octave is linear in the logarithm frequency scale.

Table 9.2-1 Accelerations at SC/LV Interface during Transportation

Load Source	Acceleration		
	Longitudinal (X)	Lateral (y)	Lateral (z)
SHM Transportation	±0.4	-1.0±0.7	±0.5

Table 9.2-2 Maximum Quasi-static and Dynamic Accelerations at SC/LV Interface

Load Source	Acceleration	
	Longitudinal (X)	Lateral (y, z)
LV movement inside TLC	2.5±0.7	±0.3
After LV exit from TLC	±1.0	±0.8
1st stage burn:		
Maximum dynamic head	3.0±0.5	0.5±0.5
Maximum longitudinal acceleration	7.5±0.5	0.1±0.5
2nd stage burn – maximum longitudinal acceleration	7.8±0.5	0.2
3rd stage burn	-0.3...-0.5	0.25

Notes to Tables 9.2-1 and 9.2-2:

- Lateral accelerations may act in any direction, simultaneously with longitudinal ones;
- The above values are inclusive of gravity force component;
- Dynamic accelerations are preceded by "±" symbol;
- The above values are correct for the spacecraft complying with the fundamental frequency requirements contained in paragraph 9.1.

Table 9.3-1 Amplitude of Harmonic Oscillations at SC/LV Interface. Longitudinal Axis (X)

Frequency sub-band, Hz	5-10	10-15	15-20
Amplitude, g	0.5	0.6	0.5
Duration, sec.	10	30	60

Table 9.3-2 Amplitude of Harmonic Oscillations at SC/LV Interface. Lateral Axes (Y, Z)

Frequency sub-band, Hz	2-5	5-10	10-15
Amplitude, g	0.2-0.5	0.5	0.5-1.0
Duration, sec.	100	100	100

Table 9.3-3 Spectral Density of Vibro-accelerations at SC/LV Interface

Frequency sub-band, Hz	Load Source	
	Liftoff, LV flight segment where M=1, q_{max}	1st stage burn (except for LV flight segment where M=1, q_{max}), 2nd stage burn, 3rd stage burn
	Spectral Density, g^2/Hz	
20-40	0.007	0.007
40-80	0.007	0.007
80-160	0.007-0.022	0.007
160-320	0.022-0.035	0.007-0.009
320-640	0.035	0.009
640-1280	0.035-0.017	0.009-0.0045
1280-2000	0.017-0.005	0.0045
Root Mean Square Value, σ, g	6.5	3.6
Duration, sec.	35	831

9.4 Shock Loads

Shock loads are wide-band, fading processes and are characterized by the shock spectrum and the duration of action.

The activation of the separation pyro-devices is a source of the vibro-pulse loads at the spacecraft attachment points (the duration of shock process is up to 0.1 sec). The shock spectrum values are given in Table 9.4-1. They are accurate for the Q=10 and for each of the three randomly selected mutually perpendicular directions. The change of the shock spectrum values versus frequency within each sub-band is linear (in the logarithm frequency scale and shock spectrum values).

9.5 Acoustic Loads

The sources of acoustic loads are:

- 1st stage motor burn;
- frame surface pressure fluctuations in the turbulent boundary layer.

The acoustic loads are characterized by the duration of action, integral level of the sound pressure within the frequency band of 20-8,000 Hz, and the levels of sound pressure within the octave frequency band with the mean geometric frequencies of 31.5; 63; 125;...; 2,000; 4,000; 8,000 Hz.

Table 9.4-1 Shock Spectrum at Spacecraft Attachment Points

Load Source	Frequency subband, Hz							Number of shock impacts
	30-50	50-100	100-200	200-500	500-1000	1000-2000	2000-5000	
	Shock Spectrum Values, g							
Separation of fairing, 3rd stage and neighboring spacecraft	5-10	10-25	25-100	100-350	350-1000	1000	1000	*
Separation of SC	5-10	10-25	25-100	100-350	350-1000	1000	1000-3000	1

Note: * - number of shock impacts is contingent on a number of spacecraft installed in the SHM.

Table 9.5-1 Acoustic Loads

Mean Geometric Frequency of Octave Frequency band, Hz	Level of Sound Pressure, dB
31.5	125
63	132
125	135
250	134
500	132
1000	129
2000	126
4000	121
8000	115
Integral Level of Sound Pressure, dB	140
Duration, sec.	35

9.6 Temperature and Humidity Conditions and Thermal Effect on Spacecraft

During operations with the spacecraft at SC processing facility, the air temperature around the spacecraft is maintained within 21 - 27^{0}C, with relative humidity of not more than 60%.

During SC/SHM integration at AITB, the air temperature is maintained within 5 - 35^{0}C, with relative humidity of not more than 80%.

When transporting the SHM to SHM processing facility and to the launch silo, the temperature inside the Transporter-Erector is within 10-25^{0}C with relative humidity of no more than 80%.

During operations with the SHM at SHM processing facility, the air temperature around the spacecraft is maintained within 5 - 35^{0}C, with relative humidity of not more than 80%.

When loading the Space Head Module into the launch silo and mating it with the LV, the SHM is affected by the temperature within 0-45^{0}C during the time period of no more than 30 minutes and with the temperature within 5 - 35^{0}C during the time period of no more than 5.5 hours, with the relative humidity being no more than 80%.

When the SHM is inside the silo, the temperature inside the silo is within the range of 5 - 25^{0}C with the possible short-term increase of up to 35^{0}C and relative humidity is of no more than 80%, and the temperature around the spacecraft is within the range of 5 - 30^{0}C with the relative humidity being no more than 70%.

Spacecraft heat emission while on the LV inside the silo and in-flight were not taken into account.

Thermal flux acting on the spacecraft from the inner surface of gas-dynamic shield will not exceed 1,000 Wt/m^2.

9.7 Pressure Underneath LV Fairing

Pressure change inside the fairing envelope during the ascent phase is given in Figure 9.7-1.

The maximum rate of in-flight pressure change inside the fairing envelope does not exceed 0.035 kgf/(cm^2 per sec.), except for transonic phase of flight where a short term (2-3 seconds) increase up to 0.035 kgf/(cm^2 per sec.) is possible.

Data contained in this section may be specified for each specific mission.

9.8 Gas-dynamic Effect on Spacecraft

Following separation from the Space Head Module the spacecraft encounters a short term impact (several seconds) of the 3rd stage motor plume.

All combustion products (composed of: N_2 – 28%, H_2 – 27%, H_2O – 21%, CO_2 – 18%, CO – 6%) are in gaseous state; solid or liquid phases are not present.

Parameters of the 3rd stage motor plume affecting the spacecraft are given in Table 9.8-1.

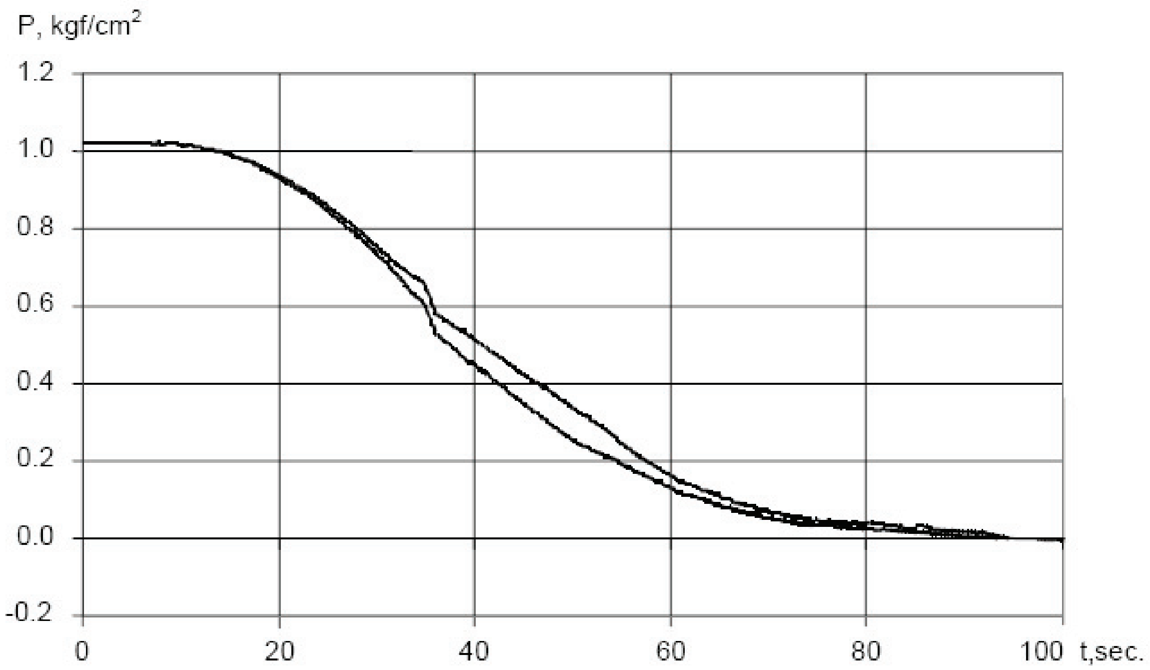

Figure 9.7-1 Pressure Change Rate inside Payload Envelope

9.10 Spacecraft Tests Required to Meet Dnepr LV Launch Services Requirements

The customer shall demonstrate that the spacecraft meets the requirements detailed in the entire section 9 of this User's Guide, by means of analyses and ground tests.

For spacecraft qualification and acceptance, sinusoidal, shock and random tests are mandatory.

A test plan established by the spacecraft authority describing the tests, which are executed on the spacecraft, shall be provided to SDB Yuzhnoye.

After completion of the tests, the test results report shall be submitted to SDB Yuzhnoye.

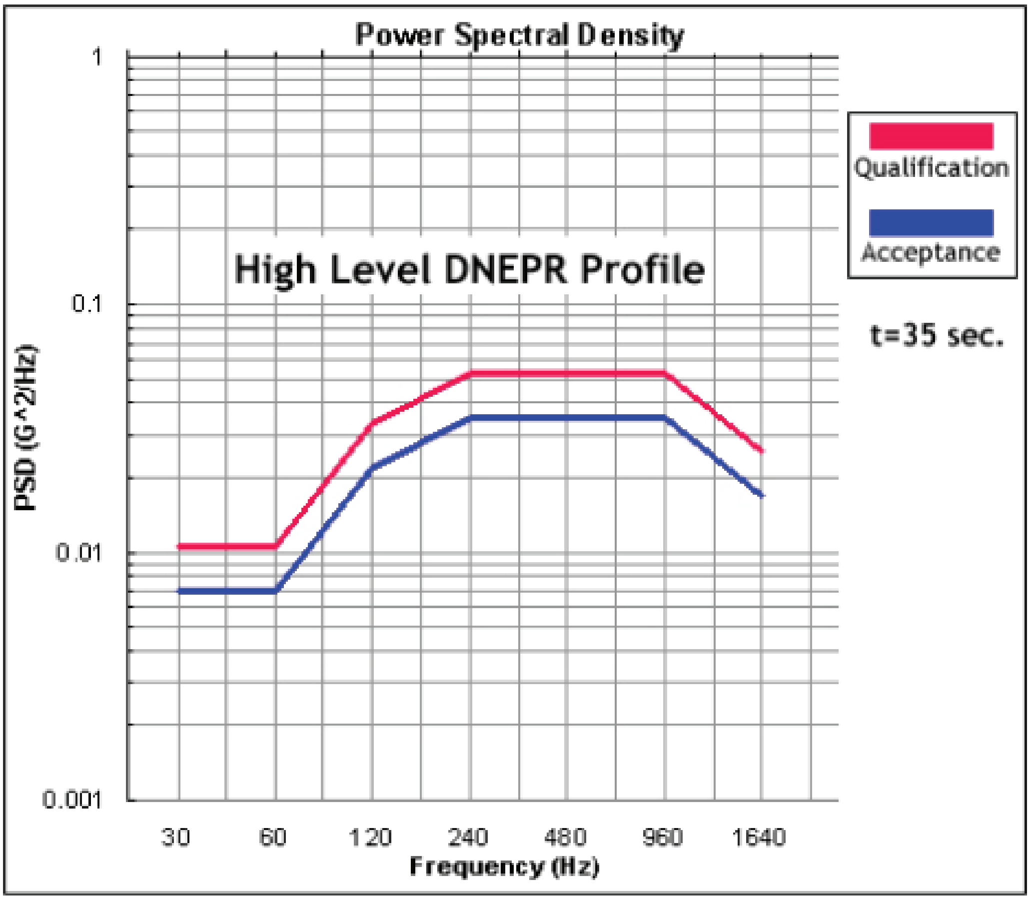

Figure 71 PSD for the High Level DNEPR Profile

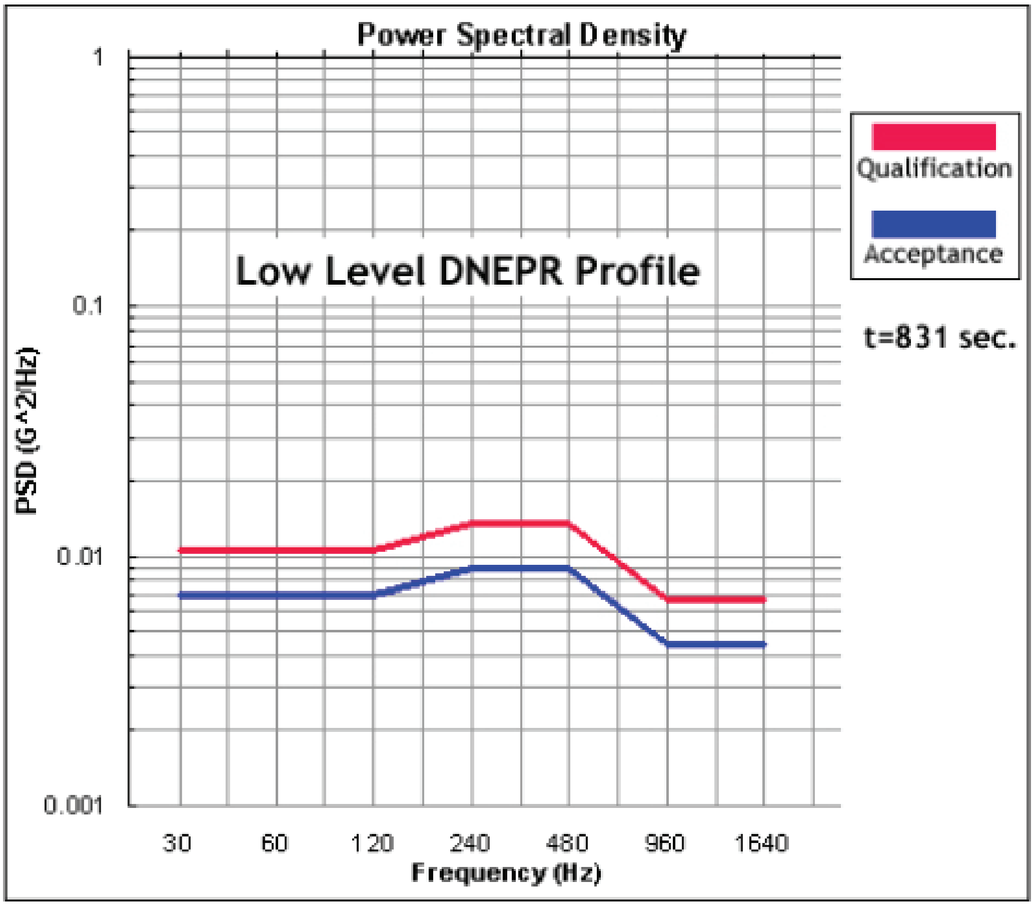

Figure 72 PSD for the Low Level DNEPR Profile

Appendix G - Existing CubeSats and their main structural properties

Institution	Project	Status	Structural Configuration and Layout	Materials	% of total mass
CalPoly	CP1	Launch with DNEPR-1, June 2006	All components are fastened to the structural frame	Al	43
CalPoly	CP2	Launch with DNEPR-1, June 2006	2 Part modular structure t allow easy access to internal components	Al	N/A
Cornell	ICE-Cube 1	Launch with DNEPR-1, June 2006	Frame structure with braces and bolts used to connect PCBs and payload	Al-7075 for load bearing members Al-6061 for non load bearing	31
Washington University	UW CubeSat	N/A	N/A	Al-7075 and Al-6061	21
University of Toronto	CanX-1	Launched June 2003, non-operational	Shelf (stack) style layout	Al-7075 and Al-6061-T6	37
University of Tokyo	CubeSat XI-IV	Launched June 2003	N/A	Al-7075	N/A
Iowa State University	CySat	Design and fabrication	Piecewise machined aluminum construction	Aluminum	N/A
Stanford	NarcisSat	N/A	6 part machined aluminum structure bolted together	Aluminum	N/A
University of Hawaii	Mea Huaka (Voyager)	Launch with DNEPR-1, June 2006	CubeSat Kit	CubeSat Kit	35
Technical University of Denmark	DTUsat	Launched June 2003, no contact	Monolithic cube machined out of solid piece of aluminum. Circuit boards were placed along inside walls with the battery and payload in the centre.	Al-7075	N/A
Montana State University	MEROPE	Launch with DNEPR-1, June 2006	Machined from aluminum with PCBs fastened to structural walls.	Al-7075 and Al-6060-T6	28
Aalborg University	AAU CubeSat	Launched June 2003, non-operational	Same frame as DTUsat, PCBs placed along walls with camera payload in centre	Aluminum	N/A
Dartmouth College	Dartsat	N/A	Does not use any screws, all components epoxied	N/A	N/A
Taylor University	TUSAT1	N/A	Aluminum frame with PCB board shell. It is a double CubeSat	AL-6061-T6	20