PRATHAM

IIT BOMBAY STUDENT SATELLITE

Conceptual Design Report

Structural Subsystem

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1. Introduction

Every physical system that performs a certain function can be mechanistically divided into structure and mechanism. The structure is designed in such a manner that it maintains its integrity under all the conditions experienced by the satellite and the mechanism designed in order to achieve the required transfer or conversion of motion or energy. A satellite, being such a physical system itself, possesses a structure that has to withstand all the forces and moments that act on it during its lifetime and may also have mechanisms in areas which require certain controlled actions to be performed.

The design of the satellite structure is influenced by various factors, mainly classified as geometrical, structural and thermal. Constraints imposed on the total weight are stringent and require the weight to be minimised as far as possible to reduce launch costs. The geometrical constraints influencing satellite design include the achievement of certain ranges for the geometrical parameters of the shape, such as the location of the centre of mass, the principal axis vectors and the requirements of a diagonal moment of inertia tensor in the principal axes frame, which are mainly dictated by the attitude control requirements. Structural considerations are predominant during the lunch phase, wherein the satellite in conjunction with its launch interface system is subjected to various loads by the launch vehicle. These include static loads caused by the accelerations experienced and vibrational loads, as well as shocks at stage separations. The structure needs to able to withstand these loads and not undergo failure in any conceivable mode. In orbit, the satellite is primarily subject to thermal considerations, where gradients and transients of temperature set up in the body give rise to distorting thermal stresses. The same requirements as for the launch phase apply here, namely that the satellite should withstand these loads without significant deformations resulting in a loss of structural integrity. These criteria in turn influence other factors like the selection of materials to make the body and the configuration layout.

The primary mechanism to be incorporated in this satellite, as dictated by payload requirements, is one to deploy the monopole antennae to be used both for communication and for gathering experimental data. The constraints and requirements as imposed on this part of the system play a major role in the synthesis of this mechanism, following which analysis is performed to determine its kinematic and dynamic parameters of operation.

2. Overview of Structural Subsystem

2.1. Structural requirements for PRATHAM

The following section details the requirements imposed on the Structural Subsystem by the other subsystems of the satellite and the constraints under which the Structural Subsystem needs to design the satellite. It gives a broad look at the various tasks handled by the subsystem and their significance in the overall scheme of the satellite.

2.1.1. Weight budget and constraints

2.1.1.1. Introduction

Launch cost of any satellite is directly dependent on the weight to be launched. The heavier the satellite, higher the energy required to put it into orbit around the Earth and thus more is the fuel consumed by the launch vehicle, making the launch operation more expensive. To keep launch costs to a minimum, it is therefore of utmost importance to keep the total weight of the satellite to a minimum as far as possible.

2.1.1.2. Problem description

The maximum weight of the satellite has been fixed at 8.0kg based on various considerations. The weight budget is maintained by the System Engineer who ensures that the total weight does not exceed this maximum and that weights are pruned wherever possible. The Structures Subsystem adds inputs to the weight budget in commenting on the weight distribution between the subsystems. In particular, the structure of the satellite itself must not take up greater than 30% of the total weight of the satellite.

2.1.1.3. Significance

The weight budget is crucial in system engineering not only to help all subsystems keep their components as light as possible, but also to keep track of subsidiary components like connectors, cables and harnesses which do not appear explicitly in any inventories but add to the satellite weight. Total weight is thus monitored as well as its distribution among subsystems.

2.1.2. Structural issues in launch phase

2.1.2.1. Introduction

The satellite is carried to its orbit by a launch vehicle in a flight lasting about 17 minutes. During this period, the vehicle experiences high levels of acceleration, vibrations and shocks which are transmitted to the payloads attached to the flight decks of the vehicle. Thus, the satellite undergoes a structurally very demanding phase during the time of launch.

2.1.2.2. Problem description

The Structural Subsystem is required to design the satellite such that the satellite as a whole as well as all the components onboard survive the rigours of launch with no damage caused to any part. Stringent testing requirements are imposed by the launch vehicle with appropriate margin of safety, which are standards the satellite must necessarily conform to in order to be cleared for flight. Failure of even one of the flight tests leads to rejection of the satellite.

2.1.2.3. Significance
Compliance with the structural requirements laid down not only minimizes risk of damage to the satellite and its onboard components, it also ensures the flightworthiness of the assembly and ensures survival of the satellite long enough to be ejected into its Low Earth Orbit.

2.1.3. Structural issues in orbit

2.1.3.1. Introduction
The satellite in orbit experiences no body forces but goes through a cycle of temperature due to the harsh thermal environment at that altitude. Variations in temperature set up stresses of thermal origin due to expansion and contraction of the material which become an important design consideration for the structure of the satellite.

2.1.3.2. Problem description
Over the course of an orbit, the satellite passes from lighted to eclipsed regions and comes under solar radiation, albedo reflected from Earth and the Earth's own infra-red emission. It radiates heat away both to Earth and to cold space on the other sides. The temperatures experienced in orbit in relation to the base temperature at which the structure was fabricated on Earth determine the nature of stresses set up in the body. The cyclic nature of the stresses induces fatigue in the satellite which imposes constraints on the size of microscopic defects present in the structure and the lifetime of the satellite.

2.1.3.3. Significance
Thermal stress computations affect design parameters such as thickness of the satellite panels and clearance allowed for expansion to avoid distortion, as well as giving an estimate of how long the satellite is expected to survive in orbit under the given load cycle.

2.1.4. Configuration layout and geometrical parameters

2.1.4.1. Introduction
The configuration layout of the satellite describes the placement of components both internal and external to the satellite. The layout is primarily handled by the System Engineer with inputs from the Structural Subsystem.

2.1.4.2. Problem description
The task involved in configuration layout design is the optimal placement of all components onboard the satellite and efficient utilization of surface area and volume. Various constraints are imposed on the arrangement by other subsystems due to orientation and temperature requirements. The overall satellite also has certain geometrical constraints imposed by the ADCS Subsystem to ensure capture of the satellite after detumbling, which have to be met by appropriate spatial arrangement of the internal components.

2.1.4.3. Significance
Achieving a good configuration layout is paramount to the proper functioning of orientation and temperature sensitive components. It also reduces the complexity and additional weight of cables by reducing the use of long connectors and their harnesses. It is also a prerequisite

for the design of the thermal control of the satellite as well as mapping the internal magnetic field generated by the onboard power circuits and shielding the magnetometer.

2.1.5. Deployment mechanism

2.1.5.1. Introduction

PRATHAM has as its scientific goal the measurement of Total Electron Count of the ionosphere by phase difference method, to achieve which it carries two monopole antennae of specified dimensions, spatial locations and orientations. These monopoles are stowed for protection during the launch phase and need to be deployed once in orbit.

2.1.5.2. Problem description

Communication with the satellite and achievement of the scientific goal necessitates deployment of the two monopole antennae. It is crucial to meet the geometric and orientation requirements of the monopoles as specified by the Communication and Payload Subsystems. Redundancy must also be incorporated in the deployment mechanism to ensure that the satellite mission is successful.

2.1.5.3. Significance

Proper deployment of the monopoles is of utmost importance to the satellite as neither will the satellite be able to transmit data back to the ground station, nor will the scientific or social goals be achieved on improper or failed deployment and thus, it is crucial to the completion of the mission statement of PRATHAM and meeting the criteria for success.

3. Weight budget and constraints

3.1. Weight budget overview

The weight budget for the IITB student satellite PRATHAM is maintained by the System Engineer with inputs from the Structures Subsystem. The detailed description of the weight budget and its related issues can be found in the System Engineer's documentation. An overview of the subject is outlined here for the sake of completeness of the Structures Subsystem report.

The maximum permissible weight of the satellite has been fixed at 8.0kg. This is purely based on the limit imposed on the CubeSATs which allow a maximum weight of 1.0kg for a cube of side 100mm, carried over to PRATHAM which has approximately eight times the same volume as the CubeSATs. The weight budget is revised every two weeks as new components are added or old ones removed due to design changes.

3.2. Critical questions and grey areas

(a) The weights of components as accounted for in the present version of the weight budget are only estimations based on other satellite data and not precise values.
(b) Isogrid structures were considered as an option to reduce the weight of the structure which is currently around 25% of the total weight of the satellite. However, the theoretical complexities introduced as opposed to the (approximately) 25% increase in structural efficiency does not make the option favourable for our use.
(c) Weight of various components like the solar panel clips, antenna supporting substrate and thermal control components are yet to be looked at.
(d) Weight of connectors and cables seems unnecessarily high and merits a second look at type of connectors being used onboard.

4. Structural issues during launch phase

4.1. Overview of launch phase

The satellite is launched into Low Earth Orbit by the Polar Satellite Launch Vehicle having four stages and taking a total of seventeen minutes from lift-off to ejection. The PSLV C-9 Interface Control Document has been studied to get an idea of the interfaces involved and loading conditions experienced during launch. Launch loads experienced include static loads, vibration loads, acoustic loads and shocks and impose certain strict requirements on the structure of the satellite. The conditions for qualification of the satellite prior to flight must necessarily be met so that the satellite is allowed to be flown onboard the Launch Vehicle.

4.2. Description of qualification criteria and tests

4.2.1. Criteria for qualification

The criteria for qualification of satellite for flight aboard the launch vehicle is passing of various tests for static loading, sinusoidal vibrations, random vibrations and testing for shock and impact. The load factor for the satellite is specified by the Interface Control Document to be 1.25, that is, the ultimate load that can be withstood by the satellite is upto 1.25 times the actual qualification levels specified.

There will be a total of three models of the satellite which will be made by the team. First of these is the lab model which is made to understand the concepts involved in satellite at the end of the Preliminary Design phase ending in September 2008. Then due to constraints of time involved, only the qualification model and flight model will be made after the detailed design phase, which ends in December 2008 The final fabrication and testing phase is expected to end in April 2009. If time and budget permit, the team will also fabricate a spare model along with qualification and flight models.

4.2.2. Static loading

Static loading occurs on the satellite during launch as a result of the accelerations experienced during flight. The static loads that are used for testing to verify the design to ensure that structure meets the safety margins are as listed in Table 4.1.

Table 4.1. Static loading levels during launch

Longitudinal	+7g / -2.5g
Lateral	+6g / -6g

These figures take into account the levels experienced during vibration testing. Lateral loads are considered to act simultaneously with longitudinal loads. Earth’s gravity is also included in the above levels. All loads apply at the centre of gravity of the satellite as body forces.

4.2.3. Sinusoidal vibrations

The levels defined for qualification in the sinusoidal vibration sweep test are as given in Table 4.2. The satellite is tested in conjunction with its launch interface on the shaker table.

Table 4.2. Sine Sweep test levels for qualification

	Frequency range	Qualification level	Acceptance level
Longitudinal axis	5-10 10-100	10mm (0 to peak) 3.75g	8mm (0 to peak) 2.5g
Lateral axis	5-8 8-100	10mm (0 to peak) 2.25g	8mm (0 to peak) 1.5g
Sweep rate		2oct/min	4oct/min

These levels are defined at the interface of the satellite with the deployer. The test is to be carried out along all three axes of the satellite, on the flight model.

4.2.4. Random and acoustic vibrations

The conditions for testing for random and acoustic vibrations are the same as that of sinusoidal vibration testing. The test levels are as given in Table 4.3.

Table 4.3. Random and acoustic vibration test levels

Frequency (Hz)	Qualification	Acceptance
	PSD (g^2/Hz)	PSD (g^2/Hz)
20	0.002	0.001
110	0.002	0.001
250	0.034	0.015
1000	0.034	0.015
2000	0.009	0.004
g RMS	6.7	4.47
duration	2min/axis	1min/axis

4.2.5. Stiffness requirements

The requirements for stiffness of the satellite are such that there should be no component onboard the satellite which is free to vibrate at a natural frequency below the specified limits. This implies that all extended structures must thus comply with the stiffness requirements, as well as the structure as a whole.

The frequency requirements for the flight model of the satellite are as given in Table 4.4.

Table 4.4. Stiffness requirements for flight model

Fundamental frequency in longitudinal mode	>90Hz
Fundamental frequency in lateral mode	>45Hz

The frequency must be evaluated by both mathematical modelling and by testing. Constraints to be applied are under the base fixed condition (the satellite is fixed at the bottom face of bottom deck). The boundary condition in the mathematical model must be of that used in the final flight configuration.

4.3. Design issues related to launch

4.3.1. Material for satellite body

Aluminium 6061-T6 alloy has been chosen as the material for the satellite body. It is ductile, lightweight, easily machinable, can have complex structures milled out of whole ingots, easily procurable, cheap, sufficiently stiff and is also characterized for space applications, thus making it a very attractive option. Due to its favourable properties and space heritage, aluminium 6061-T6 alloy has been decided upon as the material of choice and hitherto there has been no cause, either due to weight constraints or structural requirements, to revise this decision.

4.3.2. Thickness of satellite panels

The thickness of the panels making up the satellite body were arbitrarily given an initial thickness of 3mm. On performing simulations for static loading with thinner 2mm panels, results were found to be still satisfactory with weight reduction being significant. Hence 2mm was adopted as the new thickness for the panels. Buckling analysis shows this value to be insufficient, but it has hitherto not been altered as the analysis results are in doubt.

4.3.3. Number of joints in body

The stiffness requirements of the structure impose the limit on the number of joints permissible in it. Currently, the structure is proposed to be fabricated from six panels of aluminium. However, a harmonic analysis to determine if this satisfies stiffness requirements is yet to be performed.

4.3.4. Method of fastening

It is proposed to use helicoils of SS304 alloy for fastenings on the structure due to their space heritage. The diameter of these is yet to be fixed.

4.4. Analyses performed

For simulation of the model in order to evaluate design parameters of the structure, static, buckling, modal, harmonic and random vibration analyses are required to be undertaken over the course of the design phase.

In the present conceptual design phase, the focus was mainly on familiarization with ANSYS v10.0, which was chosen as the simulation platform. The analyses performed were mainly to validate the design that was proposed. The analyses that will be performed in future phases will be to extract various design parameters that will be crucial to the safety of the satellite.

The parameters to be extracted from the analyses are –
(a) Thickness of the outer panel
(b) Position and strength of joints
(c) Shape and size of the various components

The major goal of performing an analysis in the present phase of design was to learn to do the analysis rather than obtain a significant result as none of the students have had any prior experience with Finite Element Methods. Analysis was performed on a few test cases to learn the methodology. The results obtained here do not reflect the results for actual satellite.

4.4.1. Static analysis of panel

1. Aim of analysis –
The aim of the analysis is to find out the stress that is developed in the plate when given static load is applied.

2. Type of analysis –
Static

3. Problem specification –
A plate of given dimensions as used for making the bottom panel of the satellite is subjected to static loading as experienced during the launch phase.

4. Geometry –
Square plate of side 230mm and thickness 2mm.

5. Material properties –
Al6061T6 with Ex=70GPa, PR=0.3, Density=2700Kg/m^3

6. Element type used –
SOLID95 with all degrees of freedom including translational and rotational at all nodes

7. Real constants applied –
Thickness of plate 2mm

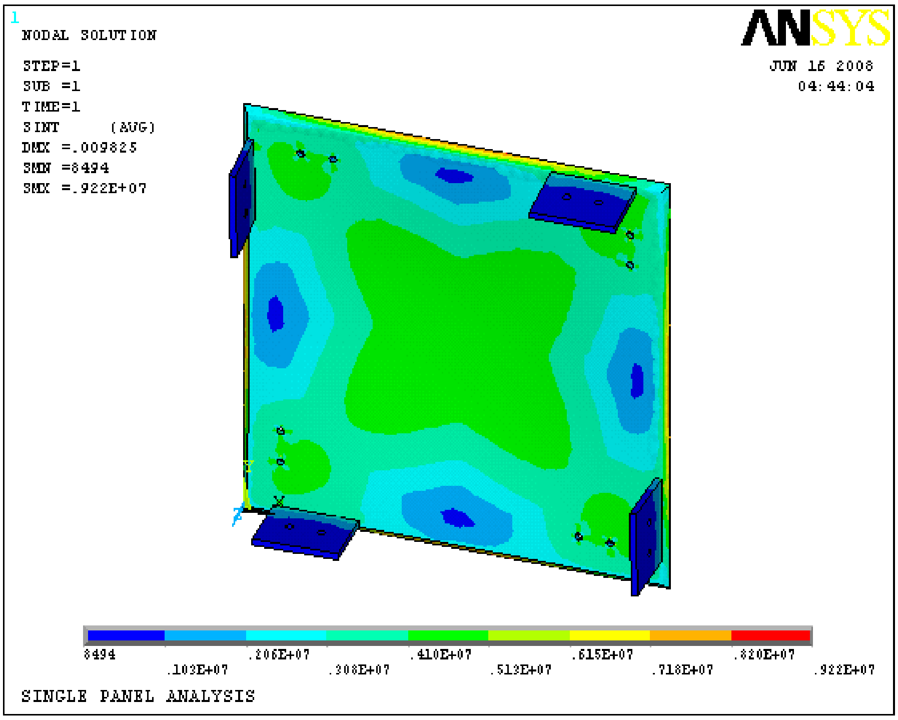
8. Constraints applied at each point –
All edges constrained for movement in all directions.

9. Loads applied –
The static loads as specified in Table 4.1 derived from the PSLV C-9 Interface Control Document, applied normal to the plate.

10. Frequency / Time step –
NIL

11. Transcript of results –
Maximum vonMises stress 9.2MPa

12. Screenshots of contour plots obtained –



13. Screenshots of graphs obtained –
NIL

14. Interpretation of results –
The maximum stress obtained in the analysis is far less than the yield strength. Hence the panel will not fail under static loads even in such a worst case analysis.

15. Conclusions and implications for our satellite –
The major design driving factor will not be static loads but will be vibrational loads. We need not worry about the static loading and can consider reducing the thickness of the side if other factors allow it.

4.4.2. Static analysis of jointless box

1. Aim of analysis –
The aim of the analysis is to find out the stress that is developed in the structure when a static load as described in Table 4.1 derived from the PSLV C-9 Interface Control Document is applied

2. Type of analysis –
Static

3. Problem specification –
To find out the stress levels that are developed in the structure when launch static loading is applied

4. Geometry –
A cubical box made up of 6 square plates, each of side length 230mm and thickness 2mm

5. Material properties –
Al6061T6 with Ex=70GPa, PR=0.3, Density=2700Kg/m^3

6. Element type used –
SHELL63 having all degrees of freedom including translational and rotational at all nodes

7. Real constants applied –
Thickness of shell 2mm

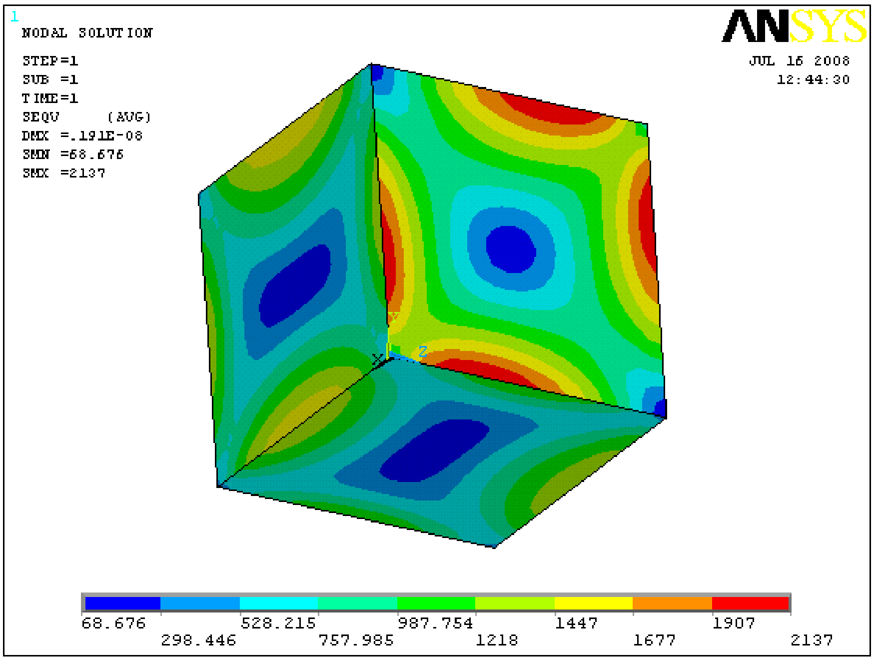
8. Constraints applied at each point –
All edges constrained for movement in all directions

9. Loads applied –
The launch static loads as specified in Table 4.1

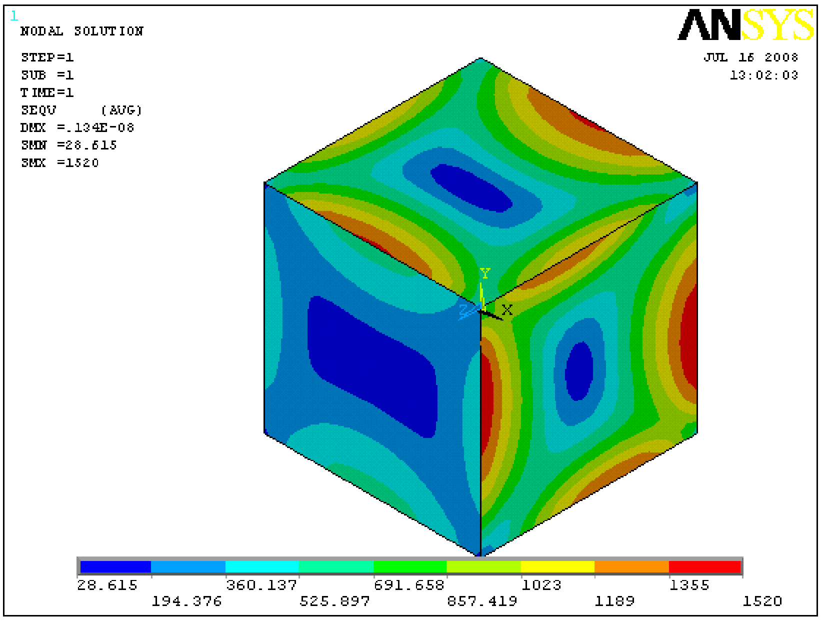
10. Frequency / Time step –
NIL

11. Transcript of results –
Maximum vonMises stress experienced is 2137 MPa.

12. Screenshots of contour plots obtained –
(a) Both loads in positive direction – vonMises stress –



Both loads in negative direction – vonMises stress –



13. Screenshots of graphs obtained –
NIL

14. Interpretation of results –
The maximum stress obtained in the analysis is far less than the ultimate tensile strength. Hence the structure will not fail due to static loads.

15. Conclusions and implications for our satellite –
The major design driving factor will not be static loads but vibrational loads. It is not required to worry much about the static loading and reducing the thickness of the side to reduce weight can be considered if allowed by other factors.

4.4.3. Modal analysis of circuit board

1. Aim of analysis –
The aim of the analysis is to find out the fundamental frequencies of the printed circuit boards.

2. Type of analysis –
Modal

3. Problem specification –
To find out the fundamental frequencies of printed circuit boards made of epoxy glass but assumed isotropic instead of orthotropic

4. Geometry –
Rectangular plate of dimensions 100mm X 70mm and thickness 2mm

5. Material properties –
Epoxy glass with Ex=32GPa, PR=0.2, Density=2100Kg/m^3

6. Element type used –
SHELL63 with all degrees of freedom including translational and rotational at all nodes

7. Real constants applied –
Thickness of shell 2mm

8. Constraints applied at each point –
All corners constrained for movement in all directions

9. Loads applied –
NIL

10. Frequency / Time step / Number of modes extracted –
5 modes extracted

11. Transcript of results –

SET	FREQUENCY
1	323.36
2	612.32
3	621.32
4	903.47
5	1509.4

12. Screenshots of contour plots obtained –
NONE

13. Screenshots of graphs obtained –
NONE

14. Interpretation of results –
The PCB has a natural frequency that is more than the minimum frequency specified by the ICD. The frequency can further be increased by having a bound central point. The natural frequency is not affected greatly by the thickness of the PCB. Increasing the number of bound points increases the frequency of the PCB.

15. Conclusions and implications for our satellite –
The natural frequency of the PCB according to the initial analysis being well above specified limits. The PCB design driving factor will thus be electrical considerations. Structural considerations are just to validate the design proposed by electrical parts. It might be required to increase the bound points or increase damping to provide the necessary minimum frequency.

4.4.4. Modal analysis of jointless box

1. Aim of analysis –
The aim of the analysis is to find out the fundamental frequencies of the outer aluminium box of the satellite.

2. Type of analysis –
Modal

3. Problem specification –
To find out the fundamental frequencies of the outer box of satellite which is an aluminium shell and has no joints, but is modelled as a solid structure

4. Geometry –
A cubical box made up of 6 square plates, each of side length 230mm and thickness 2mm

5. Material properties –

Al6061T6 with Ex=70GPa, PR=0.3, Density=2700Kg/m^3

6. Element type used –
SHELL63 having all degrees of freedom including translational and rotational at all nodes

7. Real constants applied –
Thickness of shell 2mm

8. Constraints applied at each point –
All edges are constrained for all direction movement. This constraint is strictly not correct but has been assumed as being valid for this analysis.

9. Loads applied –
The static loads as specified in Table 4.1

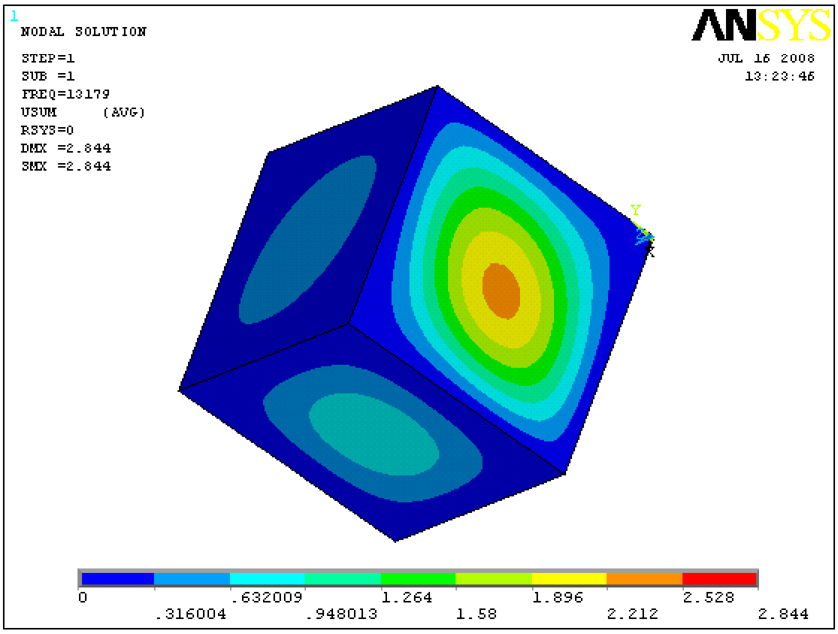
10. Frequency / Time step / Number of modes extracted –
5 modes extracted

11. Transcript of results –

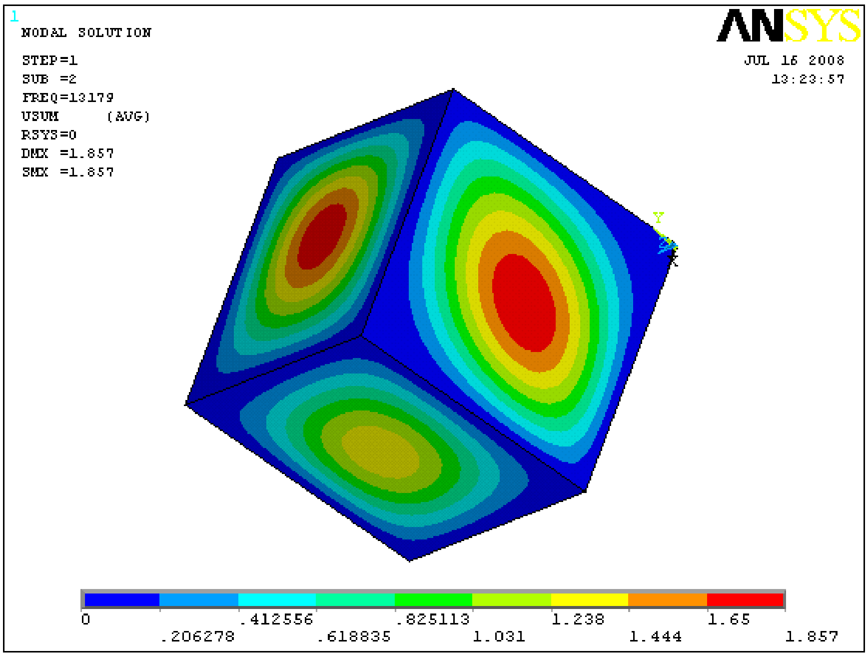
SET	FREQUENCY
1	13179.0
2	13179.0
3	13179.0
4	13179.0
5	13179.0

12. Screenshots of contour plots obtained –

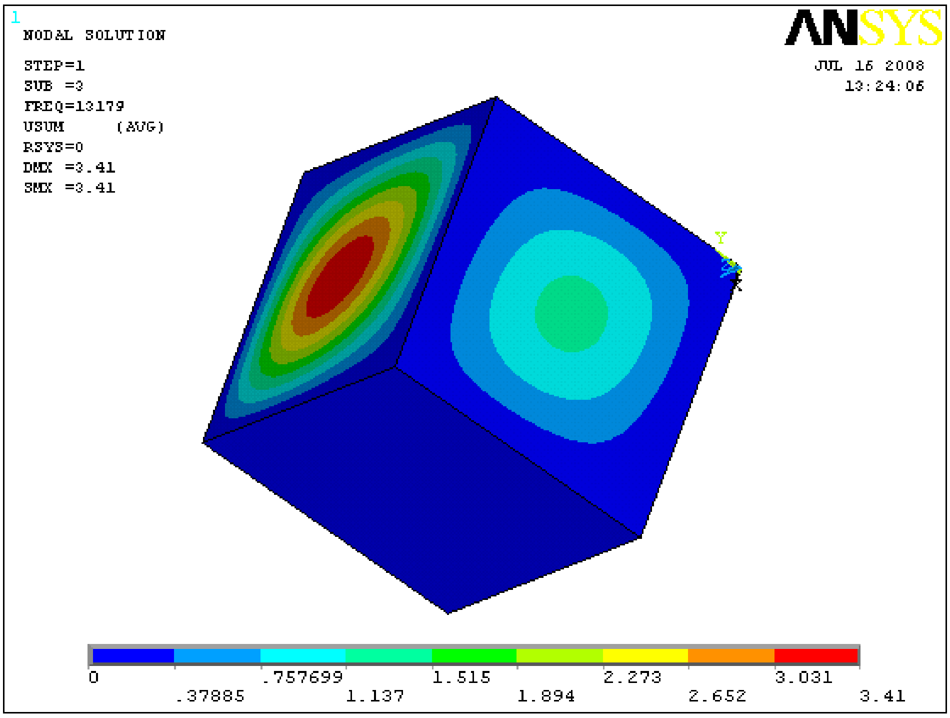
(a) First mode shape – Displacement vector –



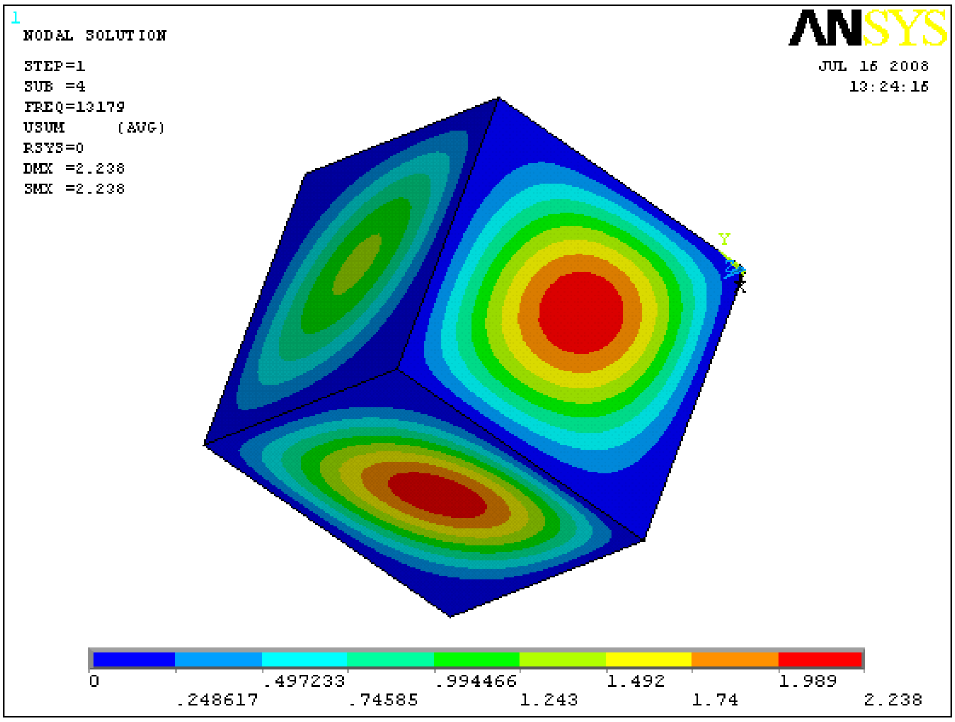
(b) Second mode shape – Displacement vector –



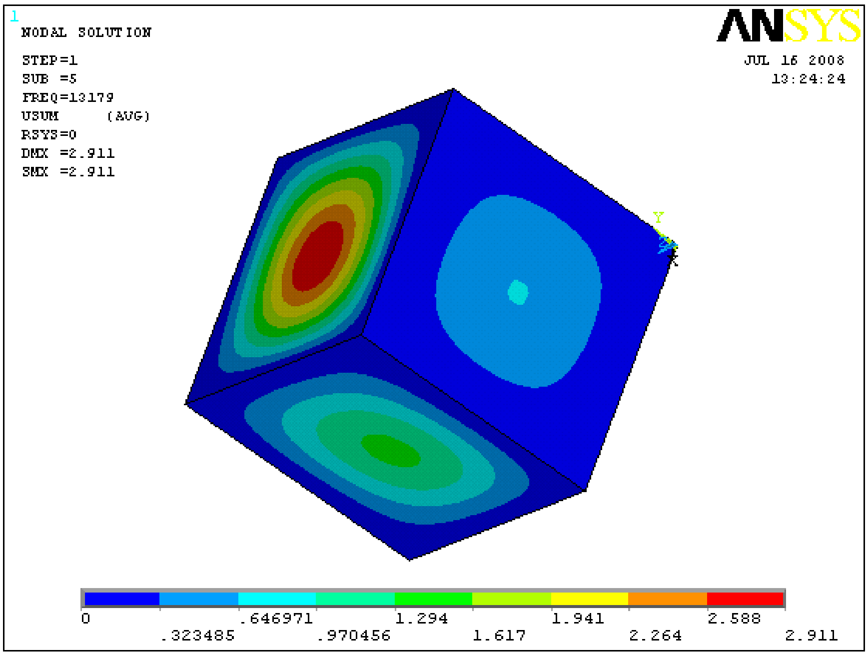
(c) Third mode shape – Displacement vector –



(d) Fourth mode shape – Displacement vector –



(e) Fifth mode shape – Displacement vector –



13. Screenshots of graphs obtained –
NONE

14. Interpretation of results –
The natural frequency obtained from the analysis is far greater than that expected. There is some mistake either in modelling or the constraints applied.

15. Conclusions and implications for our satellite –
We cannot draw any conclusions as the analysis performed might possibly be giving the wrong results.

4.4.5. Buckling analysis of panel

1. Aim of analysis –
The aim of the analysis is to find out the loads at which the plate of given properties buckles.

2. Type of analysis –
Eigenbuckling

3. Problem specification –
To find out the loading conditions at which the plate modelled after the side panel of the satellite undergoes buckling

4. Geometry –
Square plate of side length 230mm and thickness 2mm

5. Material properties –
Al6061T6 with Ex=70GPa, PR=0.3, Density=2700Kg/m^3

6. Element type used –
SHELL63 with all degrees of freedom including translational and rotational at all nodes

7. Real constants applied –
Thickness of shell 2mm

8. Constraints applied at each point –
All edges constrained for Z direction movement. Vertical sides constrained for Y direction movement. Horizontal sides constrained for X directional movement.

9. Loads applied –
Unit pressure applied on horizontal sides. It is advisable to apply unit pressure as ANSYS gives the results as a factor of the load applied.

10. Frequency / Time step / Number of modes extracted –
5 modes extracted

11. Transcript of results –

SET	VALUE	LOAD STEP	SUBSTEP	CUMULATIVE
1	51898.00	1	1	1
2	67506.00	1	2	2
3	0.11812E+06	1	3	3
4	0.18790E+06	1	4	4
5	0.20832E+06	1	5	5

12. Screenshots of contour plots obtained –
NONE

13. Screenshots of graphs obtained –
NONE

14. Interpretation of results –
The plate will buckle if the pressure on the horizontal sides (pressure that initiates the buckling) exceeds value of 51898.00 Pa. Although we cannot say anything about actual satellite scenario, this buckling pressure is on lower side and translates to about roughly about 23 N force which is far less than the expected value of around 100N.

15. Conclusions and implications for our satellite –
We might have to increase the thickness of the side of our satellite. Some more simulations are required that take into account effects of other joints and other attachments on the sides.

4.4.6. Vibration analysis of jointless box

This analysis was initially planned in order to find out the mode shapes and stresses of the jointless aluminium box under the conditions of the sinusoidal sweep test and to perform a simulation of the random vibration levels as outlined in Table 4.2 and Table 4.3 and derived from the PSLV C-9 Interface Control Document. However time was insufficient to perform this analysis. Also, conceptual doubts were encountered in this analysis which inhibited further progress along these lines.

4.5. Critical questions and grey areas

4.5.1. Questions regarding loading data

(a) In the PSLV C-9 Interface Control Document, when specifying the static loading condition there is an ambiguous statement that the static loads include the vibration load levels as well. We are in doubt as to what exactly this means and if it means that static loads are in fact lesser than those stated, in which case if the value given is obtained by adding both.
(b) The vibration analysis loads are given in terms of 'g'. The static loads will not give major vibration response at frequencies other than at their natural frequency. The dynamic response is due to the transients that are superposed on the static loads. Here these are not given. Hence we are in doubt as to what loading conditions to do harmonic vibration analysis on.
(c) It is to be found out how the vibration and static tests are performed and if they are performed separately or integrated with the satellite deployer. This is because if they are done on assembly with the deployer, both will have to be modelled together and then simulated. This will include simulating the deployer with its springs as joining elements. The other option is that the simulations can be done considering the satellite only and some checks can be performed to ascertain that loads in assembly will be less than those without.
(d) It is yet to be considered how the presence of solar panels on the surface will affect the simulation results and how to affix these solar panels.

4.5.2. Difficulties faced in simulation

(a) The team is inexperienced in the application of FEM analyses and use of such softwares and hence lacks a thorough knowledge of types of elements present in ANSYS and their application. Thus some results are erroneous due to deficiency of knowledge in using the right elements.
(b) Modeling in ANSYS is tedious and takes a lot of time. However the models drawn in 3-D CAD applications like Solidworks cannot be transferred to ANSYS easily through the IGES format, and generally give better resulta with the Parasolids format. Modelling remains one of the major problems faced by the team. Alternatives proposed are to make simplistic models in ANSYS or to make detailed models in other applications like Pro/Engineer.
(c) Validation of results is required until the team acquires sufficient proficiency and theoretical knowledge of the subject and can have a higher degree of certainty regarding the correctness of results obtained.
(d) Interpretation of data obtained is often not easy and the team requires help to interpret the results obtained correctly.
(e) Presently all modal and harmonic analyses have been tried with undamped systems. A value for damping has to be specified, otherwise displacement will be infinite at the natural frequency.
(f) Modeling and analyzing a structure with joints is an area yet to be looked into.

5. Structural issues in orbit

5.1. Overview of orbit conditions

5.1.1. Description of orbit and overall flux received

The satellite is assumed to be placed in a 10.30 am, 98° polar Sun synchronous orbit at an altitude of 670 km. The time t = 0 is when the satellite is closest to the North Pole (N.P.) and ω is the angular speed of the satellite. The duration of the orbit is assumed to be 107 minutes.

At any time, out of each pair of opposite faces, only one face receives sunlight. Thus, for the three pairs of faces, the intensity of sunlight on the sunlit face is

$$I_1 = -S\cos\alpha$$
$$I_2 = S\left|\sin\alpha\cos\varphi\right|$$
$$I_3 = S\left|\sin\alpha\sin\varphi\right|$$

where S = 1353 W/m^2 is the solar constant at that altitude and cos α = 0.925sin(ωt+3.3^O).

Here, φ is a variable angle and the satellite can rotated such that maximum amount of sunlight is incident on the solar panels. This condition is satisfied at φ = 45°. For this condition, at maximum irradiation,

$$I_1 = -S\cos\alpha$$
$$I_2 = S\frac{\sin\alpha}{\sqrt{2}}$$
$$I_3 = S\frac{\sin\alpha}{\sqrt{2}}$$

The eclipsed region of the satellite's orbit is given by 24.09^O < ωt < 149.31^O.

5.1.2. Spatial variation of flux

Solar flux incident on the sun facing side is -I*cos(α) whenever cos(α) is negative, where I=1353 W/m^2 and cos(α)=0.925*sin(ωt+3.3*3.14/180); else it is zero (1)

Solar flux on the remaining two faces=I*sin(alpha)/1.414 whenever the satellite is not in the eclipsed region; else it is zero where the eclipsed region is given by 24.09^0 < ωt < 149.31^0 (2)

Albedo on nadir surface=135.3 W/m^2 whenever it is not in the eclipsed region; else it is zero (3)

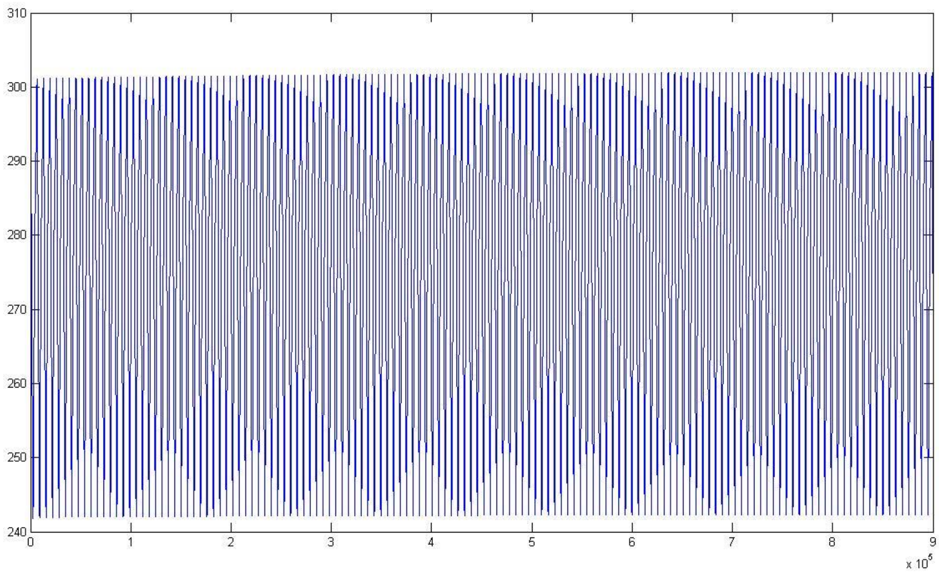
Albedo on the other two faces not facing the sun= 135.3/1.414 W/m^2 whenever it is not in the eclipsed region; else it is zero (approximately 10% of insolation) (4)

Earthshine=237 W/m^2 which is only incident on the nadir surface (5)

5.1.3. Temporal variations of temperature

The heat equation for the satellite was solved by Runge Kutta second order method using both an Excel spreadsheet and a MATLAB code, as outlined in Section 3 of the Thermal

Subsystem report. The temperature varies from 242 K- 301 K which is a range of about 60 K. The mean temperature is 270 K. The satellite settles into a steady cycle of temporal variation after one orbit only.



5.1.4. Spatial variations of temperature

Spatial variations of temperature are assumed to be small owing to the small size, relatively high density and high conductivity of the satellite. The ratio of conduction resistance to radiation resistance of the satellite is found to be 0.003 which implies low thermal gradients across the satellite body. It is attempted to validate this assumption in the analysis described.

5.2. Description of structural considerations in orbit

It is evident from the temperature variation with time that the satellite will experience mainly compressive stress. Since the variation is around 60K, these stresses become significant and fatigue should be considered in the lifetime estimation of the satellite.

5.2.1. Cyclic stresses and fatigue

Using the mechanical and thermal properties of Aluminum 6061-T6 alloy, the maximum compressive stress is found to be around 97MPa, considering 300 K as the base temperature at which the structure is fabricated. Since the range is almost completely below this reference temperature, the thermal stress cycle will be mainly compressive with a mean stress value of about 50MPa.

5.2.2. Thermal distortions and buckling

The body of the satellite is made up of Aluminium 6061-T6 alloy having linear Coefficient of Thermal Expansion (CTE) 23 X $10^{-6}/^{0}C$ and Young's modulus 70GPa. The volumetric CTE is thus 69 X $10^{-6}/^{0}C$ and the stress produced per degree variation in temperature is thus calculated as being $E\alpha = 4.83\ MPa/^{0}C$, mainly compressive. This stress, depending on the magnitude of temperature variations, can lead to distortion of the structure and buckling.

5.2.3. Satellite lifetime expectancy

The required duration of data collection by the satellite for the Total Electron Count experiment onboard is a minimum of four months for proper validation of the data. It is thus proposed to design the satellite for a minimum lifetime of atleast one year so that its required targets are achieved. Given a time period of 107 minutes for a single orbit, the satellite makes about 5000 orbits in a year, with each orbit being one thermal cycle of stress. The cyclic nature of the stress imposes the limitation of fatigue life on the structure. However, typical short-cycle fatigue lifetimes are of the order of 10^5 cycles and thus, the satellite is not expected to fail due to fatigue during its design lifetime.

5.3. Analyses performed

5.3.1. Thermal stress analyses

Thermal stress analysis is a two-part simulation where after obtaining the temperature data on all nodes a coupled analysis is performed by using the temperature data as the load input for the structural analysis. The thermal and structural environments are written separately, temperature is first solved for from flux inputs and then the environment switched to structural analysis to find out the thermal stresses produced.

5.3.2. Specifics of thermal stress analysis

5.3.2.1. Aim of the analysis

The analysis has been performed on a representative case wherein a snapshot of the satellite at a certain point in orbit has been considered for steady-state analysis. Given all the fluxes incident on and emitted from the satellite on different faces and an assumed reference temperature, the satellite had been subjected to analysis with the aim of finding the gradient in temperature across the body and the difference in temperature between the locations of the two extrema. This data is then used to find thermal stress contours on the satellite

5.3.2.2. Description of problem

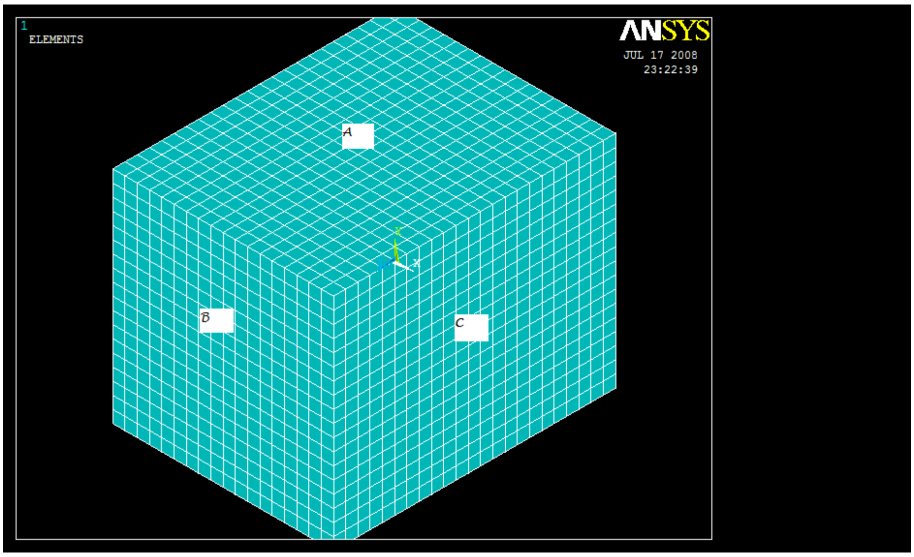
The satellite is assumed to be a cube of side length 230mm made of aluminium 6061-T6 alloy. The satellite was thus modelled in ANSYS v10.0 and subjected to a steady state coupled analysis to find the thermal stress in such a case.

5.3.2.3. Type of analysis

Coupled analysis – thermal and structural

5.3.2.4. Geometry

Cubical shell of outer dimensions 230mm X 230mm X 230mm and shell thickness 2mm, with six faces denoted by A, B, C, A’, B’, C’.



5.3.2.5. Material properties

(a) Material – Aluminium 6061 T-6 alloy

(b) Thermal expansion coeff- 23.1 µm·m^{-1}·K^{-1}

(c). Heat capacity – 896 J/kgK

5.3.2.6. Element type used

SHELL 41

5.3.2.7. Real constants applied

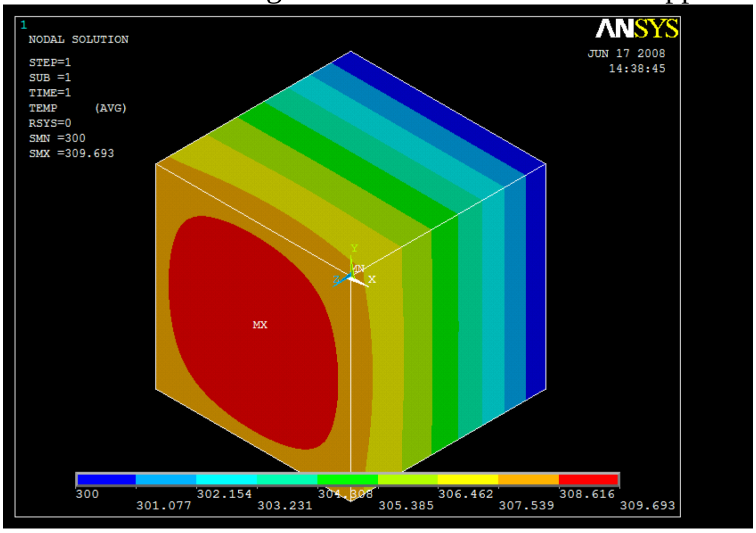
Thickness (uniform) is 2 mm

5.3.2.8. Constraints applied at each point

The cube has zero displacement on the entire nadir surface

5.3.2.9. Loads and fields applied

The temperature contour shown in the figure below serves as the load applied.



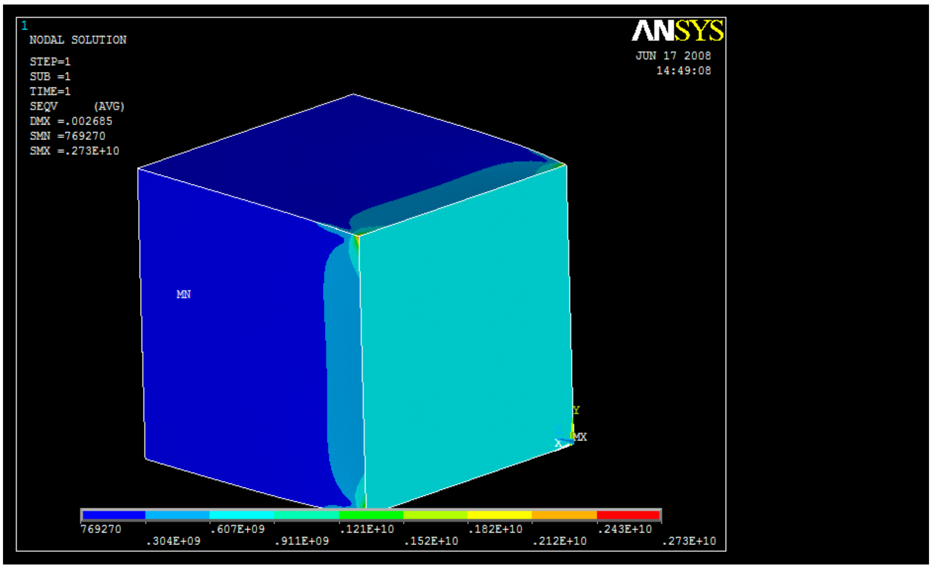
5.3.2.10. Frequency / Time step

NONE

5.3.2.11. Transcript of results

Maximum vonMises stress is 2730MPa.

5.3.2.12. Screenshots of contour plots obtained



5.3.2.13. Interpretation of results

The stress levels are far too high with gradient of stress being extremely steep. The stress levels also do not correspond with the values expected from the temperature gradient theoretically. Errors are thus apparent in the analysis which are to be identified and corrected. However the purpose of learning thermal stress analysis was served.

5.3.2.14. Shortcomings, assumptions and grey areas

(a) Absorptivity is assumed to be equal to emissivity in the thermal analysis which is not true.
(b) Constraining of body is incorrect.
(c) Further checking and help is required for thermal analyses.

5.4. Conclusion

The values obtained for thermal stress far exceed any reasonable limits and do not tally with any theoretical estimations. The error in the analysis needs to be corrected. Appropriate boundary conditions need to be identified and applied. Also, the option of other applications for thermal analysis can be explored. Hitherto, analytical estimates have proved to be a better base to build upon rather than simulation data. Analytical estimates place stress magnitudes at around 100MPa which is significant but reasonable. Fatigue lifetime however will not be a major limiting factor on the design lifetime of the satellite. The phenomena of flutter and snap have hitherto not been studied, and their effects on deployed structures of the satellite will have to be looked into in the future. Thus, a rudimentary study of thermal stress has been undertaken. However, future design phases will have to incorporate much better analyses than the one performed in the conceptual phase.

6. Configuration layout and geometric parameters

6.1. Overview of configuration layout

The internal and external configuration layout of the satellite has been designed by the System Engineer with inputs from the Structures Subsystem. The detailed description of the configuration layout can be found in the System Engineer's documentation. An overview of the subject is outlined here for the sake of completeness of the Structures Subsystem report.

6.1.1. Dimensions of satellite

The dimensions of the satellite have been chosen primarily based on concerns of external configuration layout and inertia tensor of the satellite. The arrangement of twelve solar panels on four sides and eight on the fifth side, each side having rows of four panels each, with panels measuring 39.8mm by 69.5mm, has been concluded as being ideal for power generation eliminating the requirement for maximum power point tracking. To incorporate this solar panel design along with sun sensors and guide rails for deployment, the overall shape and dimensions of the satellite have been chosen as a cube of length 230mm.

6.1.2. Internal architecture of satellite

A choice was made between mounting circuit boards on a rack within the satellite and fastening them flush with the walls of the body, that is, rack mounting or panel mounting. A House of Quality chart was prepared independently by Structures Subsystem and System Engineer to evaluate the advantages and disadvantages of both architectures. Both charts showed the panel mounting architecture to be superior to the rack mounting architecture, primarily due to reasons of better mechanical support, more thermal control options and easier grounding. A detailed description of the internal and external architecture of the satellite is given in the System Engineering report.

6.1.3. Geometric parameters of design

The geometric parameters of the design refer to the orientations of the principal axes and the inertia tensor of the satellite. For simplicity of the control law, it is preferred that the principal axes coincide to best possible extent with the geometric body axes of the satellite. Requirements of control stability and capture after detumbling impose the following constraints on the principal moments of inertia –

$$I_y > I_r, I_p$$

$$I_y < (I_r + I_p)$$

A cube has two of its principal moments of inertia, say I_r and I_p, being of same value, hence by making the third principal moment of inertia I_y such that $I_r < I_y < 2I_r$ the constraints can be met most easily. Thus, a cubical geometry is chosen for the satellite.

6.2. Critical questions and grey areas

(a) The circuit boards must be dimensioned and supported such that they meet the structural and stiffness criteria required of the satellite.

(b) It is not yet known how solar panels will be affixed to the body of the satellite.

(c) An idea to be considered is the deployment of the magnetometer in a position between the two monopole antennae embedded in the dielectric substrate between them.

(d) It has not been easy to satisfy the required moment of inertia constraints while modeling the satellite and further work is required in this regard.

(e) Location of interfaces for launch such as the RBF pin and deployment switches are not yet identified and restrict further work on internal configuration.

7. Deployment mechanism

7.1. Requirements on deployment mechanism

The scientific goal of the IITB student satellite PRATHAM is the measurement of Total Electron Count of the ionosphere through phase difference method. This involves transmission and reception of linearly polarized radio waves at 400MHz and 433MHz. The satellite thus incorporates two monopole antennae onboard with stringent dimensional and positional constraints to achieve best scientific results.

The two antennae are primarily annealed copper rods of cylindrical cross-section. The lengths of the two antennae are 187.4mm and 173.1mm with dimensional tolerances of upto 1mm. The diameters of the antennae are respectively 2.4mm and 2.2mm with more stringent dimensional tolerances of 0.1mm to achieve maximum purity of linear polarization of radio waves. The requirement of in-plane spatial separation of around 200mm coupled with requirements of planarity with geometric tolerance 0.15mm and parallelism with in-plane geometric tolerance 0.15mm leads to the proposal of having a thin dielectric substrate with the antennae on either side of the dielectric plate. The entire structure is to be deployed at an angle of 13degrees to 17degrees above the plane coincident with the zenith surface of the satellite with the monopoles pointing parallel to the negative pitch axis.

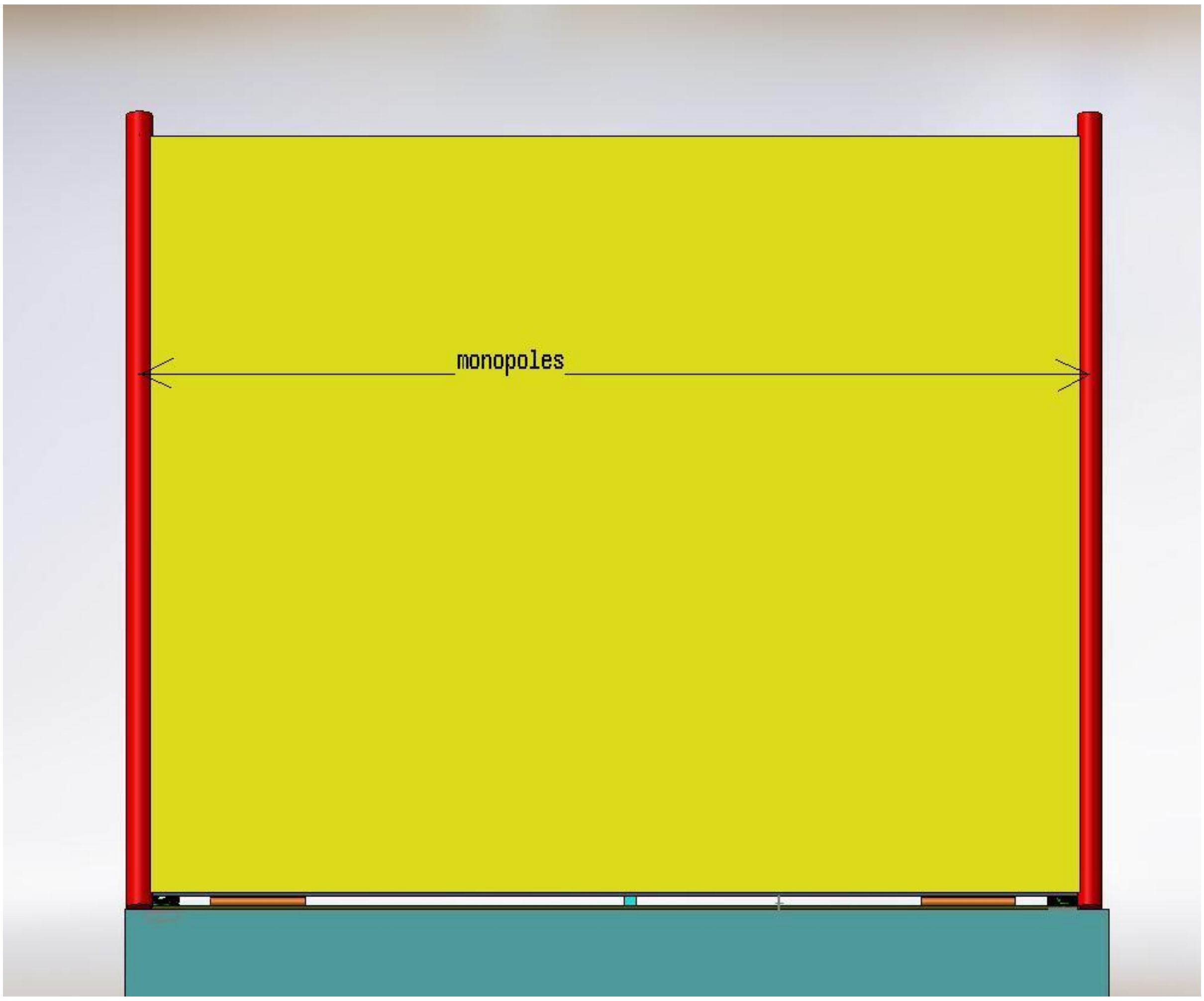
7.2. Justification for rejection of pre-deployed structure

The launch vehicle interface developed by University of Toronto – Institute of Aerospace Studies (UTIAS) for its Generic Nanosatellite Bus (GNB) has the provision of launching a mechanism in the deployed state. It was studied whether it would be possible for PRATHAM to take advantage of this provision with regard to the monopoles unit. An analytical calculation was performed to find the natural frequency of vibration of the monopole, which resulted in a value of 56Hz, which is below the minimum permissible frequency of 90Hz. Thus, the pre-deployed state cannot be used and it was decided to employ a deployment mechanism. However, this analysis has not been performed for the unit with the dielectric sheet as a whole, but only for an isolated monopole, and hence results are far more conservative than the actual case.

7.3. Structural elements of deployment mechanism

7.3.1. Dielectric substrate

The two monopoles are connected with a dielectric substrate of nominal thickness 2mm. The material for the substrate is yet to be chosen.



7.3.2. Hinge

There are a pair of hinges which attach the body of the satellite. These are of SS304 alloy and are attached to both body and dielectric substrate by helicoils, also of SS304.

7.3.3. Spring

The mechanism incorporates a spring between the substrate and the body in compressed state to assist the deployment process. The spring constant and material are yet to be determined. At present, the spring is taken to be helical, but the option of tape springs and torsion springs integrated with the hinges are also being considered.

7.3.4. Locking disc

This consists of a half-disc having a hole at a given position corresponding to the angle of deployment of the system.

7.3.5. Locking peg

This is a spring loaded peg attached to the substrate edge inside the satellite body. It rests against the locking disc with spring compressed in the stowed position and passes through the hole in the locking disc when the spring relaxes after deployment.

7.3.6. Nylon tether

A thin nylon cord is strung along the unanchored edge of the substrate to keep it stowed during launch.

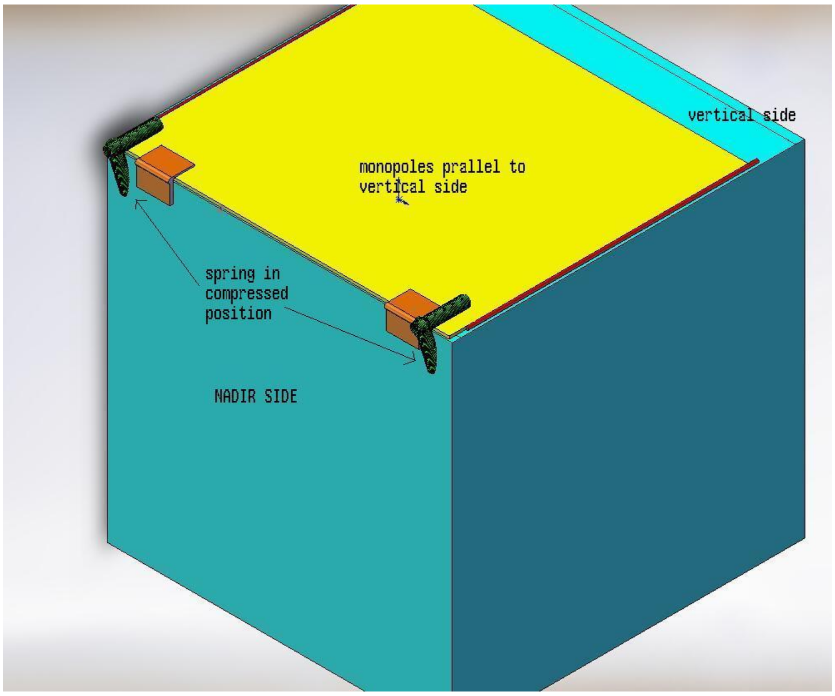
7.3.7. Nichrome heater

The nylon cord has a thin nichrome wire entwined along its length which acts as a heater and heats up upon the passage of power through it on reception of the appropriate signal form the onboard electronics.

7.4. Working of deployment mechanism

7.4.1. Stowing of monopoles during launch

The two monopoles with their separating dielectric plate are held parallel and adjacent to the outer surface of one of the vertical sides of the satellite and tethered in this position using a nylon wire. In this position the springs are compressed. The nylon wire anchors the substrate in place and prevents deployment before the specified signal is received.

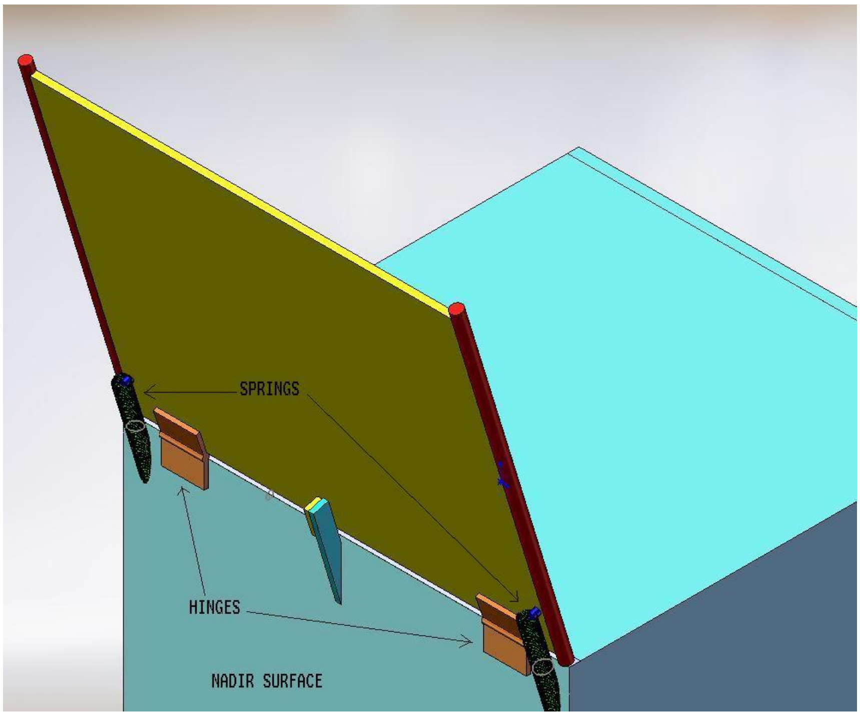


BEFORE DEPLOYMENT

7.4.2. Release of monopoles

To release the system from its stowed position, a signal is transmitted from the onboard circuit to power the nichrome heater. The nylon wire is melted on powering the heater and the substrate is released from stowed position. The spring expands to its natural length causing the plate to deploy and at the appropriate angle, locking occurs.

AFTER DEPLOYMENT



7.4.3. Locking of monopoles into position

As the monopoles are deployed, the locking peg on the inside traverses the arc of the locking disc. At the required angle, when the hole on the locking disc is reached, the spring of the locking peg is no longer under compression and expands to its free length causing the peg to slot into the hole preventing further movement of the monopole-substrate assembly. Thus, the single rotational degree of freedom of the assembly is completely constrained along both forward and backward directions.

7.5. Effects produced on deployment

7.5.1. Increased Moment of Inertia

The deployed system is of low mass but being of large dimensions comparable to the satellite results in an increase in Moment of Inertia of the satellite after deployment. The moment of inertia is increased along all three body axes, but the maximum increase is about the central Earth-pointing axis. The control law of the satellite thus has to be modified to incorporate this change. This change in moment of inertia also impacts the decision on whether to deploy the monopoles before or after the completion of the detumbling phase of the satellite.

7.5.2. Jerk produced

On deploying and locking, the satellite experiences a reaction at the line of the pivot about which the mechanism deploys. This translates into an angular velocity and acceleration imparted to the satellite. This has to be damped by the actuators of the attitude control subsystem in order to maintain the stabilized attitude of the satellite.

7.5.2.1. Energy method for theoretical calculation of jerk

The only source of energy which gets converted into rotational energy of the satellite after the deployment is the energy stored in the springs. The maximum value of angular velocity can be obtained by equating the spring potential energy and the rotational kinetic energy after deployment.

$$W = R\theta\sqrt{k/I}$$

k – Spring constant of springs (10 N/m)

I – Moment of inertia about Earth-pointing axis.(0.658 Kg-m-m)

Ө – Angle moved by antenna (1.86 radians)

R – Radius of cross section of spring (2 mm)

7.5.2.2. Experimental methods

The energy of the system conserved when it just hits the locking mechanism in an inelastic collision can be calculated practically by allowing it to hit a ballistic pendulum at the same point. The deflection of the pendulum is measured and then by equating the difference between potential energy gained by the pendulum and the kinetic energy of the mechanism to the rotational kinetic energy of the satellite, the angular velocity can be obtained.

7.5.2.3. Results

The theoretical maximum angular velocity of the system was calculated as being 0.83 degree/s. The actual value will be lower due to the inelastic collision at the point of locking. Results from the experiment are yet to be documented.

7.6. Critical questions and grey areas

(a) The most important issue yet to be addressed is the incorporation of single-fail redundancy measures in the deployment mechanism, which is so far unsupported by any such redundancy measures.
(b) The monopoles dissipate around 0.6W of heat which must be radiated to cold space without unduly increasing the temperature of the monopoles or the substrate.
(c) The effect of thermal distortions on the planarity and parallelism of the arrangement must be inquired into. Currently, distortion due to thermal expansions and contractions over a diurnal range causes a worst case expansion of around 300 microns at the metal base of the monopoles which translates into a worst case deviation from parallelism of about 120 microns, which is close to the error limit allowable.
(d) The design parameters of the dielectric sheet have to be computed or chosen.
(e) Spring constant of the springs and tension in the nylon wire have to be determined.
(f) A more refined calculation is required for the effects produced on deployment.

8. Conclusion

The structural problem of the design of the IITB student satellite PRATHAM has been outlined along the five lines of weight budgeting, launch structural considerations, orbit structural considerations, configuration layout and deployment mechanism. Preliminary studies have been undertaken in each of these areas and design concepts established. Critical issues that need further investigation or verification have been identified and grey areas where there exists ambiguity in the problem definition or interpretation of results have been pointed out. The results stated in this document are to be verified and further plans for the preliminary and detailed design phases are to be made on the basis of inputs received from this design phase.