PRATHAM

IIT BOMBAY STUDENT SATELLITE

Conceptual Design Report

Thermal Subsystem

By

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1. Introduction

The environment upto a height of around 500km above the Earth is strongly affected by the presence of the denser layers of the atmosphere and by interaction with the hydrosphere, and thus experience relatively mild conditions of temperature. However, above 500km, the atmosphere becomes too thin to moderate the ambient thermal conditions and consequently, the environment above such a height is prone to extremes of temperature, being directly exposed to solar radiation and deep space.

Spacecraft, which are primarily intended to perform scientific missions which require them to be positioned far away from the Earth's surface, spend their entire lifetime in the harsh environments of the upper atmosphere or interplanetary space. Being electromechanical systems, however, these spacecraft also have certain requirements of temperature ranges outside which their systems cannot function viably. Hence, every such spacecraft must incorporate a thermal subsystem to achieve an optimal thermal environment for its smooth functioning, through either active or passive means.

The requirements of thermal control and the challenges involved in realising these in case of nanosatellites in Low Earth Orbit (LEO) are unique. The IIT Bombay Student Satellite, PRATHAM, belongs to this class of satellites, having a mass of less than 10kg and orbiting in a LEO of approximate altitude 700km. The Thermals Subsystem of PRATHAM aims to maintain a cycle of temperature within the satellite which has a narrow range and lies around the normal terrestrial temperatures, minimize spatial gradients of temperature, dissipate excess heat generated by other subsystems and maintain sensitive components within their specified functional ranges of temperature.

2. Overview of thermal subsystem

2.1. Thermal requirements for PRATHAM

PRATHAM is a nanosatellite of dimensions 230mmX230mmX230mm and mass 7kg. The satellite body is machined from Aluminium 6061-T6 alloy and covered on five sides by solar panels mounted on a polymeric substrate and covered by a micron-thick layer of glass. It receives around 50W of incident power, converts 8W out of this to useful electrical power and dissipates 6W through the battery charger, transmitter and microprocessor circuits.

The satellite is proposed to be placed in a LEO of altitude 670km with an orbital period of 107minutes, wherein it will remain in the Earth's shadow region for 29% of its orbital period. The solar constant at this altitude is 1371 W/m^2.

Most electronic components used onboard the satellite are industrial-grade components having an operating range of – 45*C to +80*C with the battery having a more stringent requirement of +20*C to +60*C and solar panels requiring maintenance below +20*C.

2.1.1. The transient temperatures experienced in orbit

2.1.1.1. Introduction to the problem

The satellite experiences a cycle of varying temperature over the course of its orbit. The system is fully determinate as all relevant fields can be completely described and will eventually converge to a given cycle of temperature as a function of time, irrespective of the initial conditions at the instant of ejection into space.

2.1.1.2. Problem description

The aspects to be addressed with respect to the transient are threefold –

a. To compute the dependence of temperature of the satellite as a function of time (or position in the orbit) and derive the maximum, minimum and range of temperatures experienced

b. To find the time taken from the instant of ejection into space until the satellite temperature converges to the same function of time (or position in orbit) over all subsequent orbits

c. To identify the controllable design parameters affecting the maximum, minimum and range of temperatures experienced by the satellite and derive their respective effects

2.1.1.3. Significance of problem

The results obtained in this stage are carried over to structural analysis and to further stages of thermal design of the satellite. The temperatures obtained must be such that they fulfil the following two requirements –

a. The thermal stresses set up in the body must be within allowable limits and not cause significant distortions

b. The temperatures experienced must be conducive to the operation of all electronic components onboard the satellite

The design parameters must be appropriately modified within the constraints to achieve such a temperature cycle which satisfies both the above conditions and hotspots or coldspots within the satellite must be identified using these results.

2.1.2. The spatial gradients of temperature

2.1.2.1. Introduction to the problem

The satellite is positioned in space such that all faces of the cube do not receive equal irradiation. This distribution of irradiation as a function of satellite attitude impacts the ADCS sun sensor data interpretation, power generation by solar cells and spatial temperature gradients set up in the satellite body.

2.1.2.2. Problem description

The focal point of this segment of analysis is to determine the greatest possible difference of temperature between any two points on the body of the satellite at any instant of time.

2.1.2.3. Significance of problem

The results obtained from the analysis will be carried over to structural analysis to determine the extent of distortions experienced by the body of the satellite. Higher gradients can also lead to its being a significant factor in determining the internal configuration layout with different components being placed according to the local temperature optimal to them.

2.1.3. The internal thermal environment

2.1.3.1. Introduction to the problem

Each component of the satellite has an optimal range of temperature within which it gives the best performance. Components require different kinds of temperature control methods to maintain an optimal ambient environment. Excess heat generated in certain areas of the satellite need to removed and areas requiring higher temperatures need to be heated.

2.1.3.2. Problem description

The problem requires a stepwise analysis as follows –

a. To compile the operating ranges of the components within the satellite

b. To pinpoint the locations of heat dissipation and the amount of power lost instantaneously at each of these locations

c. To list the methods of passive control of satellite temperature and apply these accordingly

d. To study the active control of the battery box temperature and find the requirements of current in terms of magnitude and duty cycle

2.1.3.3. Significance of problem

The solution to this problem is essential for smooth functioning of all electronic subsystems of the satellite and the realization of the ultimate scientific goal intended for the mission.

2.2. Conceptual approach

This section gives an overview of the concepts utilized in solving the three problems defined and describes the method adopted in solving each of the questions.

2.2.1. The transient temperatures experienced in orbit

In this preliminary analysis, a lumped analysis of the satellite is undertaken which makes the assumption that the temperature of the satellite can be considered as uniform over the entire body and spatial gradients are absent. This assumption is later justified by comparing the resistances to heat transfer by conduction and radiation at the appropriate length scales.

All transfers of heat into the system from various sources and transfers without to various sinks are listed and the energy equation for the system is written, with the nett heat transfer being equated to the rate of change of internal energy of the body.

The resulting differential equation is analytically solved using the Runge-Kutta method. The Fourier number for the system is computed and the appropriate time step chosen. By interpolating between successive instants of time the numerical dependence of temperature on time (or position in orbit) was computed. The procedure was extended to cover sufficient number of orbits until the thermal cycles obtained in successive orbits were coincident.

2.2.2. The spatial gradients of temperature

The computation of the thermal transient as outlined in Section 2.2.1 involved an assumption of absence of spatial gradients in the body of the satellite. The justification of this assumption was made analytically by computing an equivalent of the Biot number wherein the ratio of resistances to conduction and radiation was found and pronouncing it low enough to validate the assumption.

Further validation of this result was attempted both through a literature survey looking at thermal analysis results for a similar class of LEO nanosatellites and through a 3-D steady-state thermal analysis using ANSYS v10.0 at three sufficiently far apart points identified in the orbit, the points of maximum, minimum and median irradiation.

A more detailed analysis will be undertaken in the next stage of design with inputs from the ADCS subsystem regarding the attitude of the satellite as a function of orbit.

2.2.3. The internal thermal environment

The analysis of internal thermal environment was undertaken in association with the Power subsystem and the System Engineer.

Literature survey was performed to determine operating ranges for components and list methods of active and passive thermal control. The power budget of PRATHAM was the source for determination of points of dissipation of heat internal to the satellite.

A decision was made to apply an active control system with feedback for maintenance of optimal conditions for the battery with appropriate safety margins, as the battery is the most crucial system onboard the satellite. The appropriate heat equation was written for the battery box as in its intended configuration onboard the satellite and solved analytically to determine the requirement of power for heating. Literature survey was performed to choose temperature sensors for feedback and the type of heater to be embedded in the system.

2.3. Specific challenges

The challenges specific to thermal control of nanosatellites were studied in a literature survey and detailed below –

a. Nanosatellites have to rely almost entirely on passive thermal control systems as electrical power available is at a premium in these systems and cannot be diverted into active thermal control units

b. The surface of most nanosatellites (including PRATHAM) is almost entirely covered by solar panels to maximize generation of electrical power and hence thermal control on a large scale by altering surface absorptivity and emissivity parameters using paints and tapes is not a feasible option as the solar panels have fixed values of such parameters

c. Heaters can be embedded within nanosatellites but cooling is not possible as there is neither enough surface area to accommodate elements like refrigerators nor is it possible to mount pressurized pumped liquid cryogenic systems

d. The restricted weight budget limits the options available for thermal control elements

e. High level of integration of subsystems leads to restrictions on the possibility of thermal control of individual components

f. LEOs result in a relatively large number of thermal cycles

3. Transient temperatures in orbit

3.1. Overview of problem and significance

The aim of this analysis is to find out a ball park value of the range of temperatures experienced by the satellite over a cycle with period one orbit, the mean temperature around which these values oscillate and the time in which the satellite settles down to a fixed transient over the orbit. This is required in order to determine the thermal stresses set up in the satellite body, study the possibility of thermal buckling and determine how best to implement a thermal control system for the internal components.

3.2. Description

In this preliminary analysis, a lumped analysis of the satellite is undertaken which makes the assumption that the temperature of the satellite can be considered as uniform over the entire body and spatial gradients are absent. This assumption is later justified by comparing the resistances to heat transfer by conduction and radiation at the appropriate length scales.

All transfers of heat into the system from various sources and transfers without to various sinks are listed and the energy equation for the system is written, with the nett heat transfer being equated to the rate of change of internal energy of the body.

The resulting differential equation is analytically solved using the Runge-Kutta method. The Fourier number for the system is computed and the appropriate time step chosen. By interpolating between successive instants of time the numerical dependence of temperature on time (or position in orbit) was computed. The procedure was extended to cover sufficient number of orbits until the thermal cycles obtained in successive orbits were coincident.

3.3. Problem parameters

This section states the values of the various known parameters and mentions the unknown quantities to be found out.

3.3.1. Known quantities

The quantities already known to us are the dimensions of the body, solar cells and solar panles, their absorptivities and emissivities, the mass of the satellite, specific heat capacity of the body, the solar flux incident on the satellite as a function of time, the values of albedo and earth shine which are assumed to be constant.

3.3.1.1. Flux function

a. Solar flux incident on the sun facing side –

$$Flux = -Icos\alpha \qquad cos\alpha < 0$$
$$Flux = 0 \qquad cos\alpha \geq 0$$
$$\cos\alpha = 0.925\sin\left(\omega t + \left(3.3 * \frac{\pi}{180}\right)\right)$$
$$I = 1353\, W/m^2$$

b. Solar flux on the remaining two faces –

$$Flux = Isin(\alpha)\cos(\varphi) \qquad 24.09° < \omega t < 149.31°$$
$$Maximal\ Flux = Isin\left(\frac{\alpha}{\sqrt{2}}\right) \qquad \varphi = 45°, 24.09° < \omega t < 149.31°$$

$$Flux = 0 \qquad\qquad\qquad Otherwise$$

Here the eclipsed region is given by $24.09^0 < \omega t < 149.31^0$

c. Albedo on nadir surface –

$$Albedo = 135.3\ W/m^2 \qquad 24.09° < \omega t < 149.31°$$
$$Albedo = 0 \qquad\qquad\qquad Otherwise$$

d. Albedo on the other two faces not facing the sun –

$$Albedo = \frac{135.3}{\sqrt{2}}\ W/m^2 \qquad 24.09° < \omega t < 149.31°$$
$$Albedo = 0 \qquad\qquad\qquad Otherwise$$

e. Earth Shine –

$$Flux = 237\ W/m^2 \qquad Only\ on\ nadir\ surface$$

These Solar flux values are taken from the calculations of the Power Subsystem. For details of the calculations kindly refer the Power Systems documentation. The average value of albedo is also taken from the value considered in the same section. The factor of √2 in the equations for faces at an angle towards or away from the sun arises due to the view factor. Individual fluxes are applied to the problem and their superposition gives the overall flux function for the satellite.

3.3.1.2. Dimensions

The satellite dimensions were yet to be fixed at the time of this analysis but here it has been considered based on inputs from the System Engineer and Structures Subsystem as being a cube of dimension 230mm. There are a total of 52 solar panels each of size 39.8mm X 69.5mm, 8 of which are placed on the zenith surface, and 12 on all the other faces excepting the nadir surface. Each solar cell is of area 26.6 cm^2. It is also assumed that the solar panels will occupy the whole of the surfaces on which the solar cells are placed with no exposed areas.

3.3.1.3. Absorptivities and emissivities of different areas

Absorptivity of solar cell	-	0.92
Emissivity of solar cell	-	0.83
Absorptivity of Aluminium	-	0.2
Emissivity of Aluminium	-	0.031
Absorptivity of Solar Panel	-	0.92
Emissivity of Solar Panel	-	0.84

All the values quoted have been obtained from the Power Subsystem.

3.3.1.4. Environment temperatures

The Earth is assumed to be a black body of 251 K.

The initial temperature of the satellite after it is launched into space is assumed as being 300 K. This is because it is noted that the launch interfaces developed by UTIAS for the PSLV C-9 launch have been qualified between 243K and 333K, the average of which values is around 288K, which is pushed up slightly to the higher side both due to ease of calculation and due to assumed heating effects produced on jettisoning of the heat shield. However it is further noted that the value of this initial temperature has little bearing on the results obtained.

3.3.1.5. Heat capacities
Specific Heat capacity of aluminium – 896 J/kg-K.
In the analysis the thermal mass and properties of solar panels and solar cells have been neglected.

3.3.1.6. Thermal conductivities
Thermal conductivity of aluminium – 167W/mK
The contact resistance between the aluminium body and the solar panels is assumed zero.

3.3.2 Unknown quantities
Using all the known values, we need to calculate the temperature as a function of time which will give us the maximum, minimum and average temperature of the satellite along with the time for it settling to a steady transient.

3.4. Assumptions made
It is assumed that the satellite is a lumped mass and all spatial temperature gradients are absent. Another assumption is that the attitude of the satellite is stabilized as soon as it is launched from the space vehicle into space. The actual attitude which will be adopted by the satellite in orbit is unspecified as yet and has been assumed to be that attitude which allows for maximum illumination. The albedo is assumed to be a constant at 10% of the solar flux over all areas on Earth, when it is actually a function of location which averages out to this.

3.5. Problem formulation and solution
This section describes the equations that are to be solved and the solutions implemented to solve these equations.

3.5.1 Heat Equation
The equation to be solved is:

$$F(T,t) = MC\frac{dT}{dt}$$

Mass of the aluminium body,	M	=	2kg
Specific heat capacity of aluminium,	C	=	896 J/kgK

3.5.2 Algorithm used
To solve the differential equation given in section 3.5.1, we have used the Runge-Kutta approximation of order 2.
Let the temperature at any instant t_i be T_i. Then by the Runge-Kutta approximation of order 2 the equation to be solved is simplified and temperature obtained iteratively using the following relations where 'h' is the time step –

$$T_{i+1} = T_i + \left(\frac{K_1 + K_2}{2}\right)$$

$$K_1 = h * F(T_i, t_i)$$

$$K_2 = h * (T_i + K_1, t_i + h)$$

3.5.3 Calculation of time step

$$Fo = \frac{2k}{\rho C l^2} \sim 35$$

$$\rho = 2700\frac{kg}{m^3}, C = 896\frac{J}{kgK}, k = 167\frac{W}{mK}, l \sim 0.002m$$

The Fourier number for the system is calculated and is of the order of 35. For the approximation to be close to the real situation the time step in the analysis should be less than 0.2 times the Fourier number, that is less than 7 seconds, and hence we have chosen the time step as 1 second.

3.5.4 Application in MATLAB

MATLAB was used to calculate the temperature transient. The flux function was defined for various cases using the flux function described in Section 3.3.1.1. After calculating the flux function we implement the Runge-Kutta approximation using the algorithm described in section 3.5.2. The code used for the calculation is given below.

```
function Flux = manas1(t,T)

alphasc=0.92;

esc=0.83;

alphaal=0.2;

eal=0.031;

alphasp=0.92;

esp=0.84;

omega=2*3.14/(60*107);

Areazsc=8*26.6*0.0001;

Areazsp=0.18*0.18-Areazsc;

Areassc=2*12*26.6*0.0001;

Areassp=2*0.18*0.23-Areassc;

Areansc=2*12*26.6*0.0001;

Areansp=2*0.18*0.23-Areansc;

Areanal=0.18*0.18;

Earthshine=237;

Albedo=135.3;
```

```
Mass=2;

Shc=896;

Sigma=5.67*0.00000001;

Sf=1353;

flag1=rem(omega*t,6.28);

    if(flag1>3.14)

        flag=round(omega*t/6.28)-1;

    else

        flag=round(omega*t/6.28);

    end

    costerm=0.925*sin(omega*t+3.3*3.14/180);

    sinterm=sqrt(abs(1-costerm*costerm));

    if(costerm>0)

        if (omega*t-6.28*flag>24.09*3.14/180) & (omega*t-6.28*flag< 149.31*3.14/180)

            Flux=Earthshine*Areanal*alphaal/(Mass*Shc)-
Sigma*(Areanal*eal+esc*(Areansc+Areazsc+Areassc)+esp*(Areansp+Areazsp+Areassp))*
T*T*T*T/(Mass*Shc);

        else
Flux=Earthshine*Areanal*alphaal/(Mass*Shc)+Albedo*((Areansp*alphasp+Areansc*alpha
sc)/1.414+Areanal*alphaal)/(Mass*Shc)+Sf*((Areassp*alphasp+Areassc*alphasc)*sinterm/
1.414)/(Mass*Shc)-
Sigma*(Areanal*eal+esc*(Areansc+Areazsc+Areassc)+esp*(Areansp+Areazsp+Areassp))*
T*T*T*T/(Mass*Shc);
        end

    else


Flux=Earthshine*Areanal*alphaal/(Mass*Shc)+Albedo*((Areansp*alphasp+Areansc*alpha
sc)/1.414+Areanal*alphaal)/(Mass*Shc)+Sf*((Areassp*alphasp+Areassc*alphasc)*sinterm/
1.414-(Areazsp*alphasp+Areazsc*alphasc)*costerm)/(Mass*Shc)-
Sigma*(Areanal*eal+esc*(Areansc+Areazsc+Areassc)+esp*(Areansp+Areazsp+Areassp))*
T*T*T*T/(Mass*Shc);

    End
```

3.5.5. Excel spreadsheet

The analysis was independently performed on an Excel spreadsheet and the results found to match those obtained from the MATLAB code. Thus, this provided an internal check for the analysis within the team.

3.6. Results

The temperature varies from 242K to 301K giving a range of approximately 60K over an orbit. The time for stabilization of this cycle is about one or two orbits only, with average temperature being 276K. The variations in the Excel and MATLAB results are due to alteration of dimensions of the satellite.

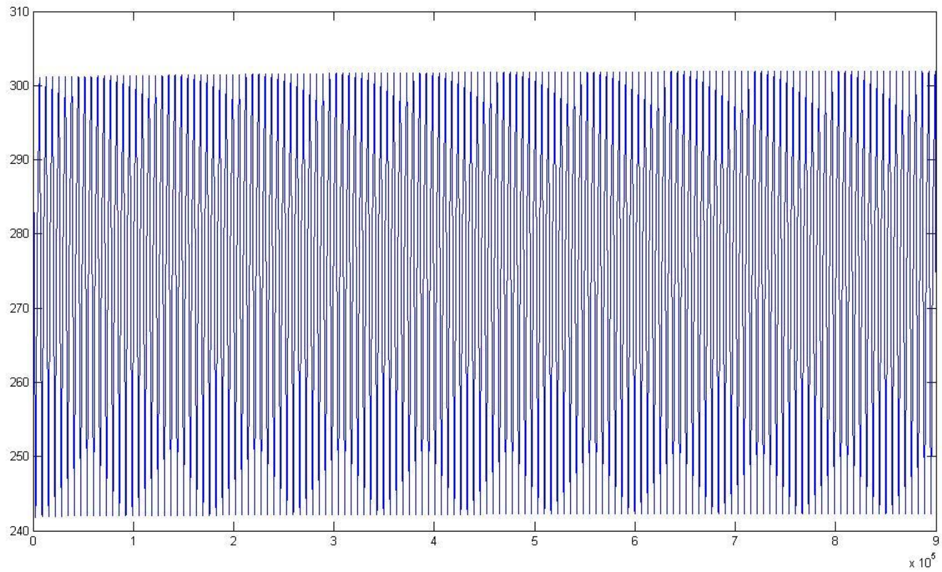


Fig.3.1. Graph obtained from MATLAB

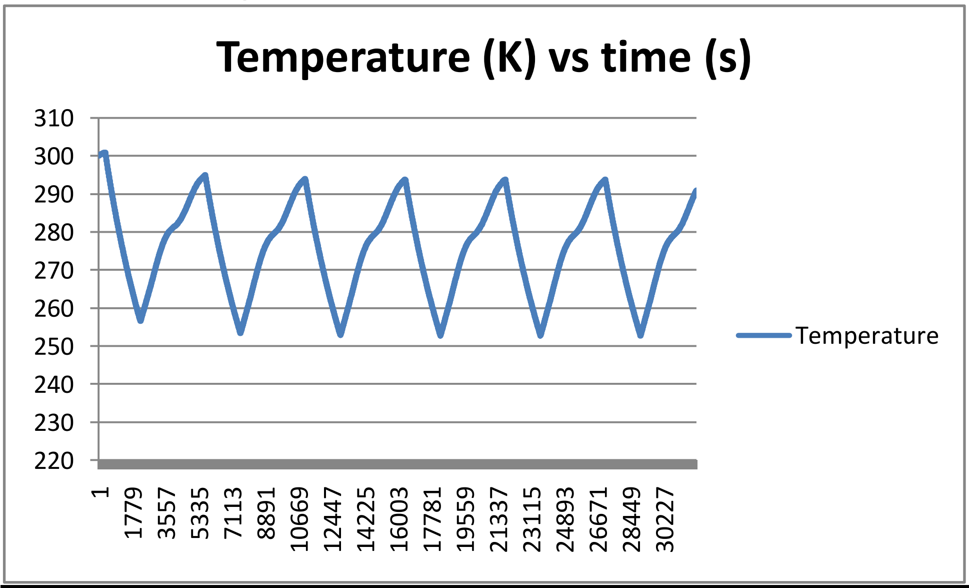


Fig.3.2. Graph obtained from MS Excel

3.7. Critical issues and grey areas

(a) Higher mass of entire satellite has to be considered
(b) Assumption of absence of spatial gradient has to be validated
(c) Contact resistance between solar panels and aluminium body has to be incorporated
(d) Resistance between aluminium panels cutting off thermal conduction paths must be considered in a more detailed model
(e) Heat absorbed by the satellite must be taken into account as well as the heat dissipation from internal components which appears as a generation term

3.7. Conclusion

The electronic components onboard are functional in the temperature range of 230K to 360K. The temperature range experienced by the satellite is within this operating range, but close to its lower limit. However, in this analysis, the area uncovered by solar panels has not been taken into account, which possesses a high a/e value of around 6 which will decrease the range of temperature. Also, the lower emissivity of aluminium and the higher thermal mass of the actual satellite will serve to increase the lower bound of temperature onboard. Thus, passive thermal control is facilitated.

4. Spatial gradient in orbit

4.1. Validity of lumped analysis

4.1.1. Computation of resistance to conduction

$$Rcon = \frac{\mathrm{l}}{\mathrm{k} * \mathrm{A}} = 0.026$$

Thermal conductivity of aluminium, k = 167 W/mK
Length scale (body diagonal), l = 398mm

4.1.2. Computation of resistance to radiation

$$Rrad = \frac{1}{\sigma * \varepsilon * \mathrm{A} * (\mathrm{T}^3 + \mathrm{T}^2\mathrm{T}_0 + \mathrm{T}\mathrm{T}_0^2 + \mathrm{T}_0^3)} = 7.7$$

4.1.3. Justification of lumped approximation

$$\frac{Rcon}{Rrad} = \frac{\mathrm{l} * \sigma * \varepsilon * (\mathrm{T}^3 + \mathrm{T}^2\mathrm{T}_0 + \mathrm{T}\mathrm{T}_0^2 + \mathrm{T}_0^3)}{\mathrm{k}} = 0.003$$

$$\frac{\mathrm{Rcon}}{\mathrm{Rrad}} \ll 0.1$$

Then lumped analysis is said to be reasonably accurate.

4.2. Steady state finite element analysis

4.2.1. Aim of the analysis

The analysis has been performed on a representative case wherein a snapshot of the satellite at maximum temperature instant 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 has 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.

4.2.2. Description of problem

The satellite is considered to be a cuboid of height 230mm and square base length of 180mm made of aluminium 6061-T6 alloy. Fluxes due to incoming solar radiation, albedo as reflected from earth and terrestrial radiation in infra-red are considered to be incident on the satellite, with the satellite radiating to Earth, a black body of 251K, from the nadir surface and to deep space, a black body at 3K, from the other five surfaces. The satellite is thus modelled in ANSYS v10.0 and subjected to a steady state thermal analysis to find the temperature gradient in such a case.

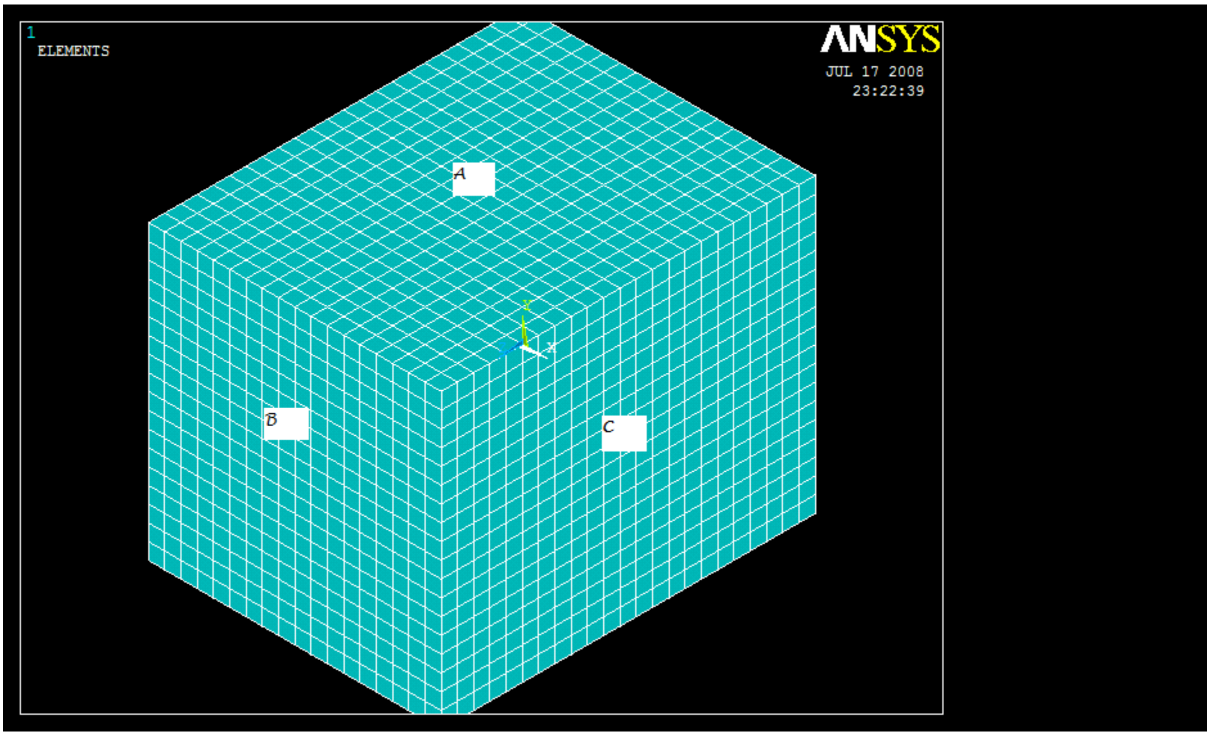
4.2.3. Preprocessing parameters

4.2.3.1. Type of analysis

Thermal analysis – Steady state analysis with radiation

4.2.3.2. Geometry

Cuboidal shell of outer dimensions 180mm X 180mm X 230mm and shell thickness 2mm, with six faces denoted by A, B, C, D, E, F



4.2.3.3. Material properties

a. Material – Aluminium 6061 T-6 alloy

b. Density – 2700 kg/m^3

c. Heat capacity – 860 J/kgK

4.2.3.4. Element type used

SHELL157

4.2.3.5. Real constants applied

Thickness (uniform) is 2 mm

4.2.3.6. Constraints applied at each point

A constant temperature of 300 K is given to nadir surface

4.2.3.7. Loads and fields applied

Flux incident on body

a.	Surface E (nadir)	:	460 (albedo)
b.	Surface B, F	:	162.5 (factor of albedo)
c.	Surface A, C	:	550 (solar)
d.	Surface D (Zenith)	:	0

4.2.3.8. Frequency / Time step

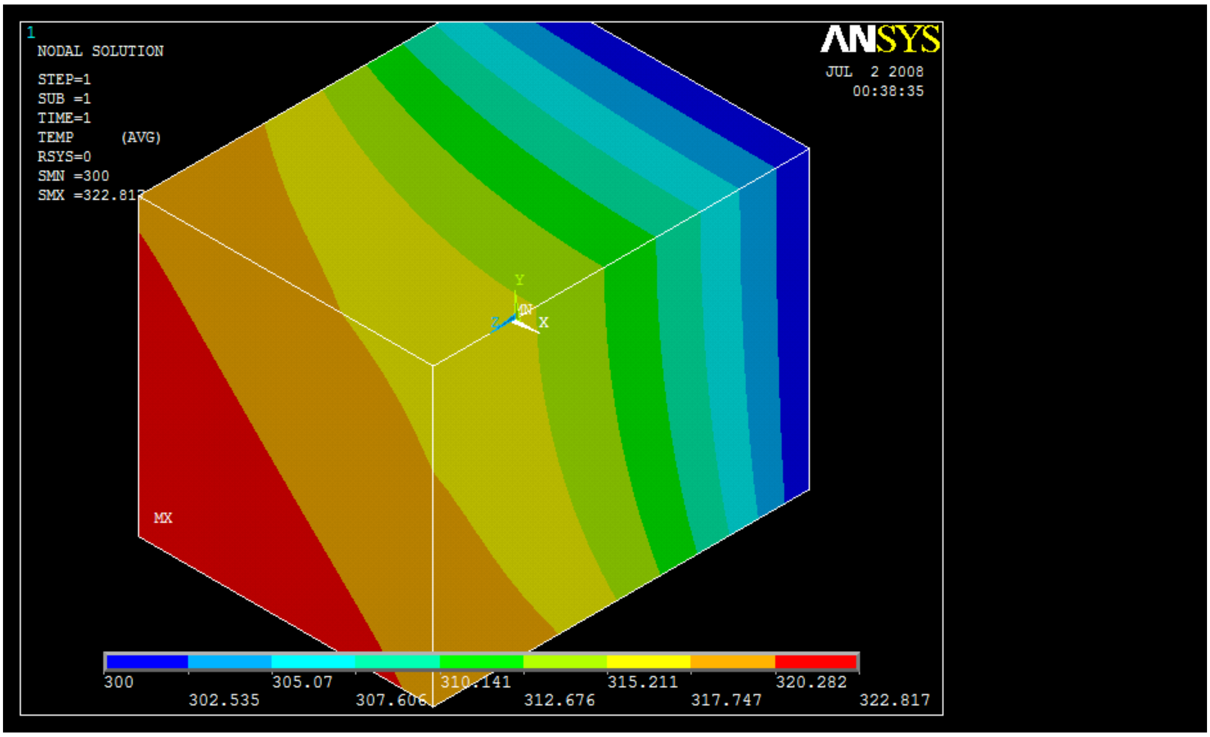
NONE

4.2.4. Solution and postprocessing

4.2.4.1. Transcript of results

Gradient of temperature observed is 23K.

4.2.4.2. Screenshots of contour plots obtained



4.2.4.3. Interpretation of results

Temperature ranges from 300K to 322.817K The surface opposite to nadir has the maximum temperature region. Range of 22K is observed.

4.2.5. Shortcomings, assumptions and grey areas

(a) Absorptivity is assumed to be equal to emissivity. This is not the case in the actual model and makes a lot of difference.

(b) Solar panels are in thermal contact with the aluminium panels and add to the thermal mass which has not been accounted for.

(c) Internal thermal mass has not been accounted for which reduces the range of temperature.

(d) It is doubtful whether application of constant temperature of 300K to nadir surface is a correct boundary condition.

(e) Also, radiation from satellite to Earth and to deep space has not been accounted for. This may probably make this analysis erroneous.

4.2.6. Conclusion

This range of 22K is small enough so as not to greatly impact thermal stresses and cause major distortions. However, it is a significant fraction of the range experienced in an orbit and appears to invalidate lumped analysis in the previous stage. Hence, points (a), (d) and (e) in Section 4.2.5 must necessarily be looked at and this analysis corrected in the next stage to get a more realistic result.

5. Internal thermal environment

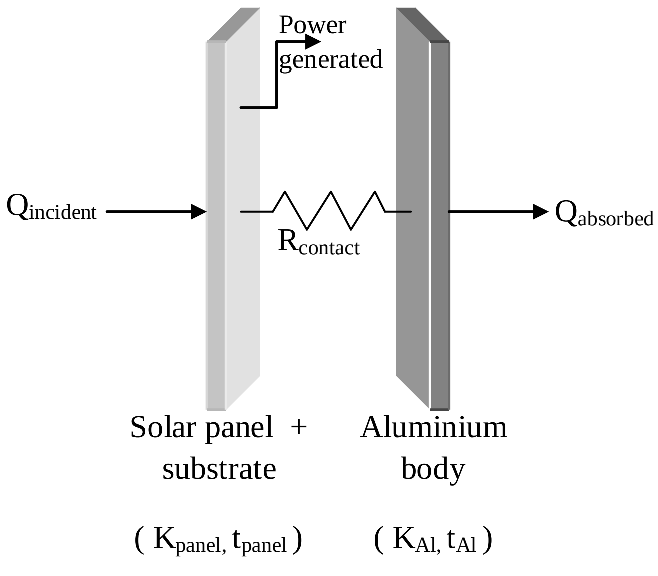
5.1. Operating ranges of components

5.1.1. Electronic components onboard satellite

The electronic components used onboard the satellite are designed to operate between the temperatures of -45*C and +85*C according to standards specified for industrial grade electronics. A margin of safety of twenty degrees is taken based on inputs received from engineers at ISRO and TIFR, thus fixing the range of operating temperature of components onboard the satellite as -25*C to +65*C. Thus, care is to be taken that at all times, the temperature at the location where the component is present does not fall outside these limits.

5.1.2. Solar panels on surface

The solar panels, being silicon based, experience a reduction in efficiency of conversion of solar energy to electrical energy with increase in temperature. Efficiency of solar cells drops below 70*C. Thus, for generating sufficient power to meet the requirements onboard the satellite and to maintain the efficiency of the solar cells as high as possible, the temperature of the solar panels has to be maintained around 30*C. As the solar panels are in thermal contact with the aluminium panels comprising the body, the solar panels achieve their equilibrium temperature in conjunction with the body as a whole according to the following thermal circuit, and this temperature is determined as in Section 3 of this document.



5.1.3. Batteries and battery box

The satellite uses Lithium Ion batteries provided by SAFT Co, France, for storing energy and powering the satellite in the eclipse region. The batteries have a stringent operating temperature range outside which the voltage of the battery drops drastically. This range is specified in the battery datasheet as +20*C to +60*C, and taking margin of safety as 10*C on either side, the battery box is to be maintained between +30*C and +50*C at all times.

5.2. Thermal hotspots

With inputs from the Power subsystem, the following areas within the satellite were identified as thermal hotspots where dissipation of energy in the form of heat is significant, along with the heat being dissipated at each location –

Sl No	Location	Power	Duty cycle
1	Solar panels	~ 40 W	71%
2	Beacon with transceiver circuit	1.6 W	100%
3	Monopole antenna	1.6 W	3.26%
4	Onboard Computer hardware	1 W	100%
5	Battery charger diode	0.8 W	71%
6	Magnetorquer coils (per coil)	0.972 W	3%
	TOTAL (Excluding solar panels)	**5.972 W**	**(At maximum)**

5.3. Passive thermal control methods

The following were the methods of passive and semipassive thermal control studied given together with their proposed areas of application on PRATHAM –

Sl No	Element	Description	Pros	Cons	Application
1	Radiators	Surfaces radiating to cold space or Earth	Easy to implement, zero cost	Constrained by surface area requirements for solar cells	Nadir surface heat dissipation to Earth
2	Heat pipes	Sealed tubes containing wick which circulates a working fluid	Readily available, effective, cheap, reliable	Possibility of outgassing from faulty specimens	Removal of heat from hotspots
3	Louvers	Series of blades with changing angles to control rate of radiation	Closed loop control possible	High complexity	N/A
4	Thermal switches	Sensors opening or closing thermal conduction paths	Closed loop control possible	High complexity	N/A
5	Coatings	Paints, metallized layers, polymers	Easy to implement, cheap	Degradation in space environment, low exposed area for use	N/A
6	Multilayer insulation	Metallized polymer layers alternating with metal foil and porous polymer	Space qualified, reliable, highly effective	Expensive	Maintenance of temperature around components
7	Fluid pumped loops	Vapour Compression refrigeration systems	Control possible across large spatial spread	High complexity, pressurized system management	N/A

Fig. 5.1. Louvres

http://orbitaldebris.jsc.nasa.gov/photo gallery/gallarypage/solarmax.jpg

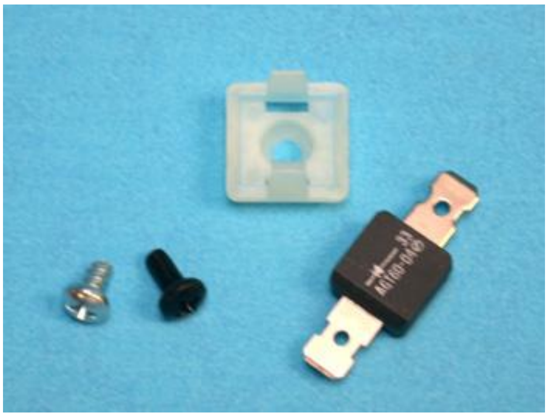


Fig. 5.2. Thermal switches and diodes

http://www.mbelectronics.com/images%5Citems%5C78806418361%20diode.jpg

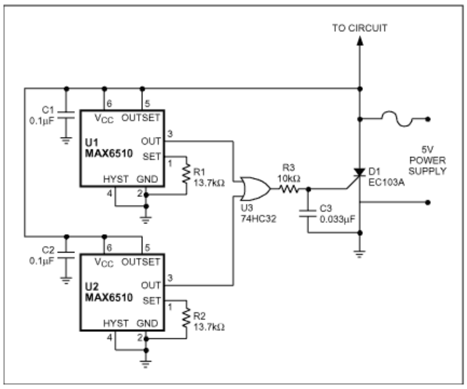


Fig. 5.3. Thermal diode circuit

http://media.maxim-ic.com /images/appnotes/2010/2010Fig01.gif

Fig. 5.4. Multi-layer Insulation (MLI) sheets

en.wikipedia.org/wiki/Multi-layer_insulation

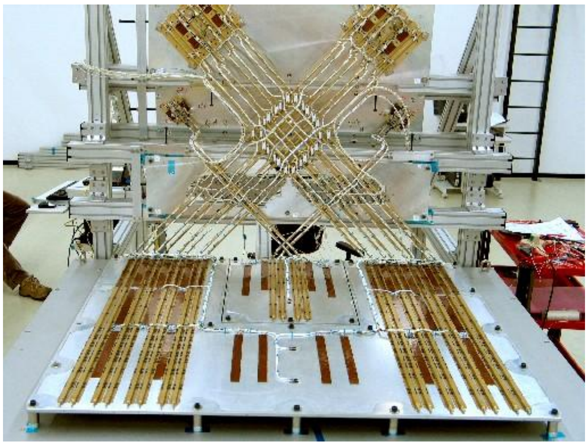


Fig. 5.5. Heat pipe architecture in ALADIN
www.dutchspace.nl

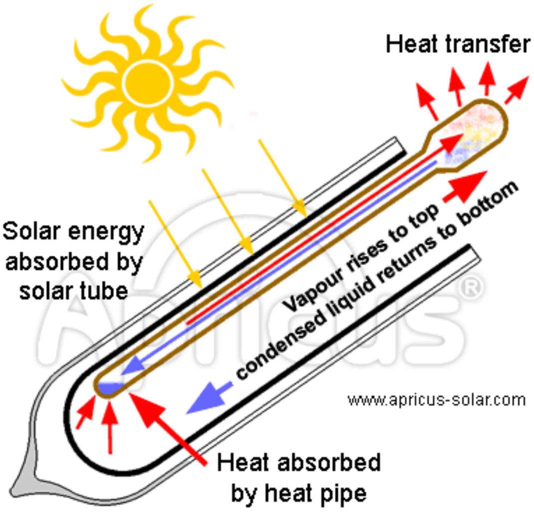


Fig. 5.6. Heat pipe schematic
http://www.apricus.com/image/Products/heat-pipe-basics.gif

5.4. Active thermal control of battery box

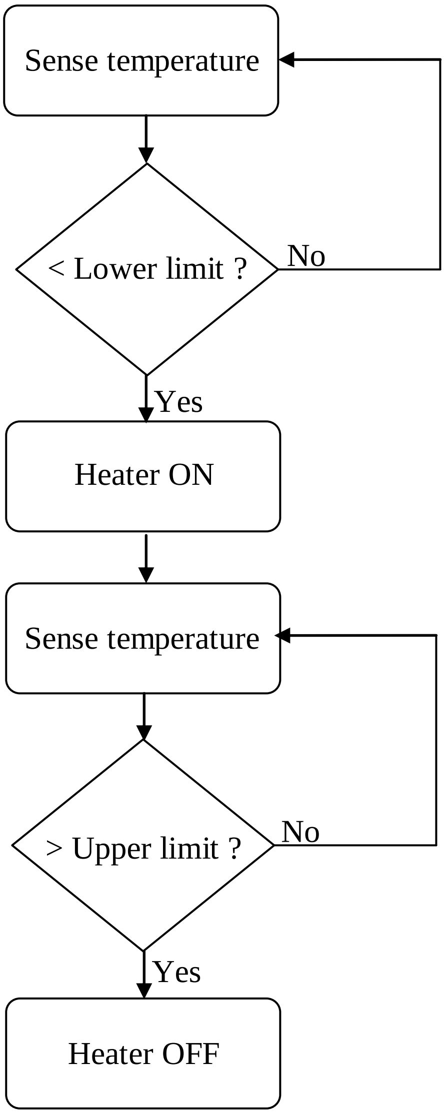
5.4.1. Introduction

The battery box positioning is dependent on the spatial variation of temperature across the body and will be decided subsequent to the detailed thermal modelling of the satellite. It is proposed to include an active thermal control system with temperature sensors and an integrated heater. To reduce the heat loss from the battery box and thus minimize the duty cycle for power consumed by the heater, it is proposed to cover the battery box with multilayer insulation which can also accommodate expansion and contraction of the batteries.

5.4.2. Proposed thermal control system

The active thermal control system proposed to maintain the battery box within optimal temperature ranges is a closed loop system incorporating a heater and temperature sensors. The sensors will be placed on the box to report the temperature to the housekeeping processor. The battery box is insulated to minimize losses and thus reduce the duty cycle required for the heater. When temperature of the battery box falls below the lower limit, the processor sends a signal to power on the heater and raise the temperature. When temperature of the battery box reaches the upper limit, the heater is switched off by a signal from the processor. The battery box cools slowly by dissipating heat through losses until the lower limit is reached again.

The following flowchart depicts the actions taken by the thermal control system –



A list of sensors and heater types has been studies for characteristics, advantages and disadvantages. However, a decision on which to use is yet to be reached.

5.4.2.1. Temperature sensors

Sl No	Reference No	Description	Pros	Cons
1	S614PDY12T	Encased probe, stainless steel,2 or 3 PTFE leads	Easy to implement, very low cost	Length is 5cm
2	S245PD12	Ceramic glass body with silver leads	Very small size and cheap too.	None as of now
3	S270PD12	Ceramic body with platinum leads	Very high precision values	Costly
4	S665PDY40A	Kapton substrate with elastomer cover	Can be integrated very easily with the heater	Cost

5	S651PDY24A	Kapton with foil backing	Can be integrated very easily with the heater	A bit costly
6	TC40JT36A	Patch style thermocouple	Easy to bond to substrate,can be fixed to battery easily	None as of now

5.4.2.2. Heaters

Sl No	Type	Description	Pros	Cons
1	Kapton	Ring shaped or rectangular strip;flexible	Easy to implement, easy to bond to substrate	None as of now
2	Silicone rubber	Discs of rigid rubber pads	Readily available, cheap	Not flexible, application more industrial
3	Wire wound rubber	Flexible	Readily available	Not been used in space applications a lot
4	Mica	Rigid disc shape.	Closed loop control possible	Possibility of outgassing from faulty specimens
5	Thermal clear transparent	Transparent rectangular plate like structure	Satisfies all reqirements	Degradation in space environment, more use in industry applications
6	AP (all polymide)	Description same as kapton	A more advanced variety of Kapton	Expensive; extensive datasheets not available from the net; have to request companies.

6. Conclusion

The thermal problem involved in the design of the IITB student satellite has been studied under three major thrust areas, the temporal variation of temperatures, the spatial gradient of temperatures and the control of the internal thermal environment. Concepts involved in thermal design have been studied, preliminary analyses performed, some results noted and critical issues raised. Questions for further investigation have been posed and weak areas identified. With inputs from the review of the conceptual design phase, it is planned to proceed with the preliminary design and detailed design stages and refine, redo or elaborate on analyses as required.