TRANSIENT HEAT TRANSFER ANALYSIS ON A SATELLITE IN AN ORBIT

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ABSTRACT

A parametric study has been carried out on emissivities of the surfaces, shell thicknesses and materials of a satellite structure in order to obtain useful information for satellite thermal engineering design. This is done through a three-dimensional finite element transient heat transfer model on a simple satellite structure using the commercial software ANSYS. The satellite is subjected to several heat loadings in an orbit of sun-tracking mode. Using actual empirically available data, a numerical model is developed and the temperature distribution in the satellite structure predicted. The numerical results of the thermal analysis are analyzed and discussed.

Keywords : *Transient heat transfer, satellite in orbit, thermal design*

INTRODUCTION

It is known that satellites perform better and last longer when their components remain within certain temperature limits. Satellite thermal control deals with the theory and practice by which these temperatures are produced. The process involves unique methods of analysis and test and often requires the use of some highly specialized hardware. Thus, the problem of thermal control must begin by establishing the temperature specifications under which the satellite is to exist during various stages of its life and to ensure that the specified values are not exceeded. This is asserted during development by thermal analysis, similarity studies and experimental tests. Additional confidence in the long-term operation is gained when the satellite and its separate components are shown to maintain their integrity and perform satisfactorily when subjected to short-term temperatures that go beyond those expected in the course of its mission.

The significance of spacecraft thermal control systems can be seen clearly in the study by Achutuni and Menzel [1] on the thermal design of satellites in an operational environment. The systems are mainly classified using either active or passive techniques. Active techniques are those that require power and can be commanded by the operator from mission control, including the use of electrically controlled louvers to regulate heat and variable conductance heat pipes. On the other hand, passive techniques do not require any power and rely solely upon engineering design features to minimize the heat load to the system. Examples include the optimal placement of radiators, thermal blankets for insulation, channeling of internal heat from the electronics and use of phase change devices such as heat pipes to absorb transient heat loads. Kwang *et al.* [2] conducted a study to predict the thermal distortion of the Korea Multi-Purpose Satellite (KOMPSAT) solar array during its orbital motion. The solar arrays were analyzed with composite face-sheets for the temperature distribution and thermal distortion with simulated low earth orbit environmental conditions. Temperature distribution and thermal distortion analyses were performed for different types of solar array materials for five different cases. Aluminium 2024-T3 was used for case 1, graphite-epoxy composite of four different thickness were used in cases 2 to 5. The composite test specimens were subjected to 80 thermal cycles in simulated solar space environments and results of temperature variations are plotted as shown in Figure 1.

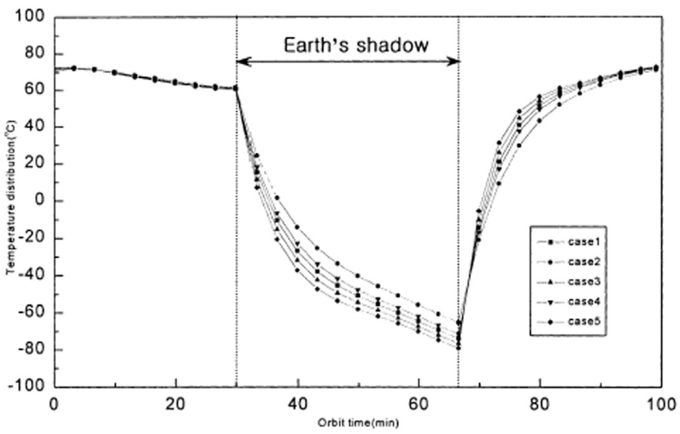

Figure 1: Temperature distribution for KOMPSAT's solar array panels [2]

It can be seen that the various materials and thicknesses in the entire five cases exhibit similar temperature distribution patterns under simulated testing. The trends of decrement in orbital temperatures are shown within the earth's shadows (eclipse) and steep increment of temperatures can be seen when the satellite is out of the eclipse in the sun-tracking mode.

Danetta [3] investigated the individual component temperature limit of Spartnik micro-satellite in the sun-tracking mode as well. The detailed temperature limits for its individual components of the micro-satellite are listed in Table 1, which shows that the absolute lower and the temperature limits of its structure using Al-6061-T4 are –100°C and 100°C respectively. This would in turn serve as a good temperature and material gauge for the present work.

Table 1: Temperature ranges for Spartnik's components [3]

System	*Equipment*	*Temperature Limits in Celsius*			
		Absolute Lower	Desired Lower	Desired Upper	Absolute Upper
Power	Battery Cell	-5	0	10	25
	Solar Array	-100	-10	25	100
	Regulator	0	0	25	25
Payload	Kodak CCD Camera	-20	0	40	60
	MMID	-40	-20	30	80
Communications	Receiver Board	-40	-20	30	90
	Antenna	-40	-20	30	90
ADAC	Horizon Sensor	-30	-20	30	50
	Bar Magnets	-40	-20	30	50
	Nutation Damper	-40	-20	30	50
	Hysteresis Rods	-100	-20	30	100
	Solar Paddles	-100	-20	30	100
Computer	Computer Board	-40	-20	30	85
Structures	Aluminum 6061 T4	-100	-30	30	100

Solar cells are well-known for displaying large temperature gradients over an orbit, typically ranging from +100°C to -100°C from sun-side to eclipse. Although there is no specific temperature requirement for the solar cells, there is some concern in the fact that solar cells have an approximate drop of 0.5% efficiency for each degree Celsius above 25°C. On the other hand, the solar cells gain a slight increase in efficiency for each degree Celsius below 25°C. In the case of Spartnik micro-satellite, which has limited solar cell area, care must be taken to keep the solar cells as cool, and therefore as efficient, as possible.

The effects of the temperature distribution with the variation of the emissivity of the satellite surface, the shell thickness and the material used for the satellite structure will be examined systematically. The range of temperatures obtained numerically will in turn provide a favorable thermal design specification for the satellite. Because the satellite's performance and durability are both enhanced by a more benign and stable temperature, the knowledge gained in the present work will be useful for reliable thermal engineering design of the satellite structure.

NUMERICAL MODEL

The heat conduction taking place in the satellite structure under transient analysis in three-dimension is governed by the following partial differential energy equation with no heat generation in the satellite structure:

$$\frac{\partial}{\partial_x}\left(k_x\frac{\partial T}{\partial_x}\right)+\frac{\partial}{\partial_y}\left(k_y\frac{\partial T}{\partial_y}\right)+\frac{\partial}{\partial_z}\left(k_z\frac{\partial T}{\partial_z}\right)= pc\frac{\partial T}{\partial_t} \qquad (1)$$

The Stefan-Boltzmann equation is used to determine the radiation heat transfer between surfaces:

$$q = \varepsilon\sigma\,(\,T_s^4 - T_\infty^4\,) \qquad (2)$$

To obtain the exact temperature from analytical considerations is generally not possible due to the complexity and nonlinear dependence of some of the parameters on temperature. Thus, the practical approach is to seek answers through numerical techniques [4]. A schematic of the numerical model setup is shown in Figure 2 with its input values tabulated in Table 2 and Table 3.

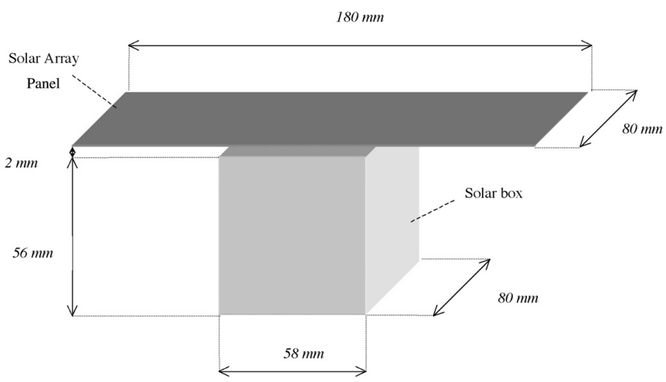

Figure 2: Dimensions of the satellite model

Table 2: Dimensions of the satellite model (cavity)

No	Component (mm)	Length (mm)	Breadth (mm)	Height (mm)	Thickness
1	Solar Array Panel (Cavity)	180	80	0	2
2	Solar box (Cavity)	58	80	56	2

Table 3: Input values used for analysis [5]

No	Material	Element Type	Real Constants Type	Shell Thickness (mm)	Material Properties			
					Thermal conductivity, k(W/m-k)	Density, ρ (kg/m^3)	Heat capacity, c (J/Kg-K)	Emissivity, ε
1		Shell 57	1	2	190	2770	963	1
2	Al_6061-T6	Matrix 50	2	2	190	2770	963	1
3		Surface 152	3	2	190	2770	963	1

Heating sources are present throughout the life of a satellite, but because operational time in orbit is by far the longest, most of thermal engineering is expanded on ensuring long-term temperature stability in space. Parametric studies were done on a range of possible orbits to determine thermal environmental loading. Three major loads were considered: direct solar radiation, Earth-reflected solar radiation, and Earth-generated radiation.

Satellite heating in Earth orbit is considered by most thermal analysts to be caused only by equipment dissipation and radiation received from the sun and the Earth. Effects from the other heavenly bodies, elementary particle bombardment, friction with low-density matter, and actual space background (≈ 2.7 versus 0 K for deep space) are usually assumed negligible when determining the temperature.

Most measurements above the Earth's atmosphere indicate a solar constant between 1365 and 1373 W/m^2 at mean distance from the sun. The spectral distribution is about 75 ultraviolet (uV) in the 0.31 – 0.40 μm, Solar Infra Red (IR) has shorter wavelengths than the IR emitted at normal satellite temperatures, thus one can take advantage of this difference and condition a surface to have simultaneously a high reflectivity in the solar spectrum and high emissivity in IR. The property connected with this idea is solar absorptivity as, which is the fraction of unhindered solar energy that is absorbed by a surface [4]; that is:

$$S^a\,(unhindered) = a^s\, S \cos\,\theta \tag{3}$$

where S^a (W/m^2) is absorbed solar energy when the solar vector (magnitude S) impinges at angle θ off the surface normal.

Albedo, A (from the Latin albus, for whiteness) is heating from sunlight reflected off Earth. It can be estimated in those regions with some accuracy as function of the sun's elevation and the satellite's orbital parameters. It is usually considered to be in same spectrum as solar radiation and often quoted as a fraction of the solar constant [4]; that is:

$$A = f\,S \tag{4}$$

with f known as the Albedo factor.

Emitted radiation from the planet Earth is considered diffuse and equivalent in intensity and wavelength to that from a black surface at about –20 °C, which gives a nominal value of Earth flux E equals 236 W/m^2. A tolerance of + 38 W/m^2 (16%) is sometimes imposed in thermal analysis [4].

Some common low Earth orbits are schematically shown in Figure 3. The vectors on the cube represent various sides of the satellite in the different orbits. Z_S and Z_E represent the Celestial North Pole and sun's centre respectively while i, is the inclination (angle) from the Earth's equator to the satellite's orbit. However, the present satellite research work in NTU is currently employing the sun-oriented polar orbits with vector 2 parallel to the solar vector (as shown in b, where vector 2 of the cube is always pointing towards the sun) for its satellite's orbit. Hence, the numerical would mainly focus on the satellite's orbital path in the sun-tracking mode where its solar array panel would be pointing towards the sun.

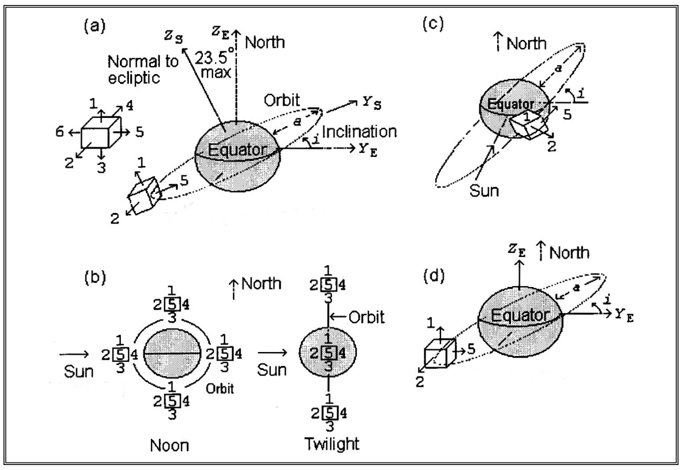

Figure 3: Typical low Earth orbits and surface orientations: (a) sun-oriented inclined orbit with vector 2 parallel to solar vector; (b) sun-oriented polar orbits with vector 2 parallel to solar vector; (c) Earth-oriented, vector 2 passes through Earth centre; and (d) star pointer, vector 1 points toward celestial north [4]

The numerical thermal model is the working tool in the development of a satellite thermal control system. It is used to predict temperatures on a large scale, with most structures and other components interacting with one another and with the surrounding environment. The thermal model is generally considered a deliverable item and therefore, a standard format should be prepared to make the model easy to interpret, run and modify. In this case, the commercial developed software ANSYS 5.6 is used to evaluate the satellite's temperature distribution.

RESULTS AND DISCUSSIONS

Upon the application of assumed heat fluxes, the satellite has its temperature distribution as shown in Figure 4, where temperatures are in degrees Celsius. The application of various simulated loadings, coupled with the designed solar space conditions reveal the temperature distribution pattern of the satellite. It can be seen from Figure 4 that the temperature is fairly evenly distributed right from the centre of the solar array panel to the solar box.

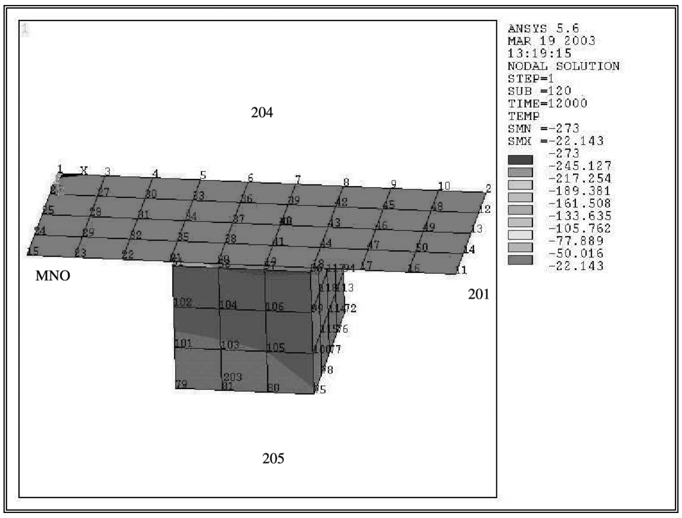

Figure 4: Temperature distribution of actual model of satellite of thickness 2mm

The initial objective is to ensure that the determined temperature of the overall satellite structure is within the general specified temperatures of orbital structures, thus the main point of interest for this project is the eventual temperature distribution along the solar array panel and the solar box. For the analysis of the experimental modelling, three selected nodal temperatures are examined. Nodes 37, 67 and 86 are selected because the maximum and the minimum temperatures are typically found within these nodes. The actual positions of the three stated nodes are shown in Figure 5.

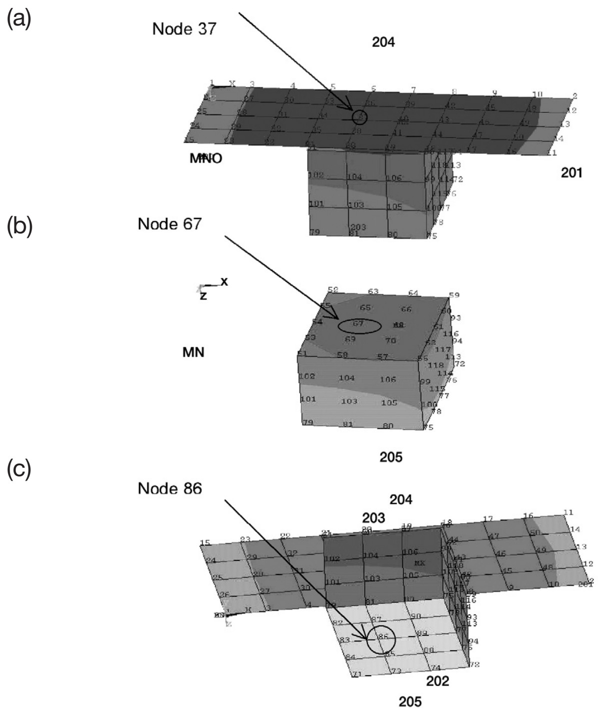

Figure 5: Contour plots of nodal temperature distribution on (a) Node 37 on solar array panel; (b) Node 67 on solar box; (c) Node 86 on bottommost of solar box

With the use of shell thickness of 2 mm, emissivity at 1 and material of Aluminium 6061-T6, three nodal temperatures are plotted with a simulation of 2 orbits in the sun tracking mode, using an Excel program, as shown in Figure 6. The figure shows the trends of increasing temperatures at about 60 degrees into its orbit where the satellite encountered solar heat flux and the decreasing temperatures at about 300 degrees, where it enters into eclipse again.

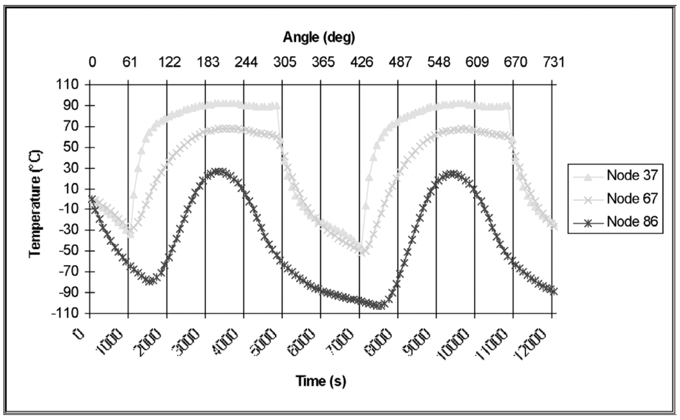

Figure 6: Temperature distribution of actual model of satellite of thickness 2mm

To find out the effects of thickness on the maximum nodal temperature, modelling of different thickness is conducted and a summary is tabulated to show the shell thickness against the maximum nodal temperature. It can be seen from Table 4 that there is a significant drop in temperature, and values graphically displayed in Figure 7. However, a design thickness of 2 mm is chosen, as an increase in thickness of the structure would mean an overall increase in weight of the satellite as a whole.

Table 4: Different shell's thickness against the maximum nodal temperature

No	Component	Material (mm)	Shell thickness	T_{max}(˚C) decrease (%)	Percentage
Varying shell's thickness (with constant material)					
1	Satellite's structure	Al_6061-T6	2	93	-
2			4	87	6.5
3			6	84	9.7
4			8	82	11.8
5			10	80	14.0

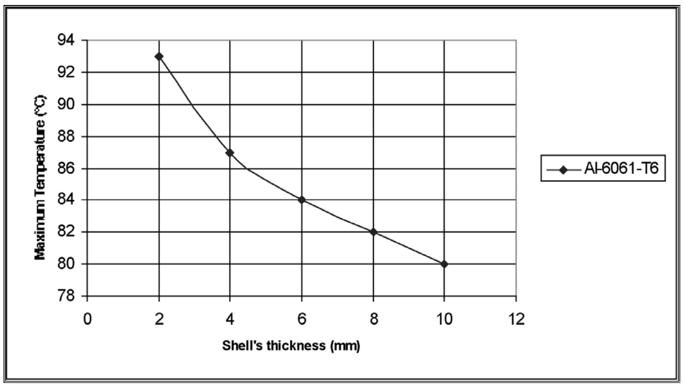

Figure 7: Plot of the maximum temperature versus the shell's thickness for Al-6061

To examine the effect of emissivities on the maximum nodal temperature, values are again tabulated in Table 5. From Figure 8, it can be seen that with a decrease in emissivity there is an increase in the maximum individual nodal temperature. The plots reveal that a decrement of $\varepsilon = 0.1$ would mean an increment of about 10% in its maximum nodal temperature. Since emissivity is defined as the ratio of energy emission by the non-black surface to emission if it were black at the same temperature, the results clearly unveil that a lower emissivity will generally mean a lower dissipation of energy.

Table 5: Different shell's thickness against the maximum nodal temperature

No	Component	Material	Thermal conductivity k (W/m-K)	Density,ρ (kg/m^3)	Heat capacity, C_p (J/kg-K)	ε	T_{max} (˚C)	Percentage increase (%)
Varying Emissivity (with constant material and thickness at 2mm)								
1	Satelite's structure	Al_6061-T6	190	2770	963	1	93	-
2						0.9	102	9.7
3						0.8	111	19.4
4						0.7	120	29.0
5						0.6	130	39.8

In another summary, Table 6 is tabulated for reference on the different materials used against the maximum nodal temperature. From Figure 9, it can be seen that there are no specific correlation between the different materials used against the maximum nodal temperature obtained. However, it can be seen that the different grades of Aluminium used exhibit very similar range of temperatures.

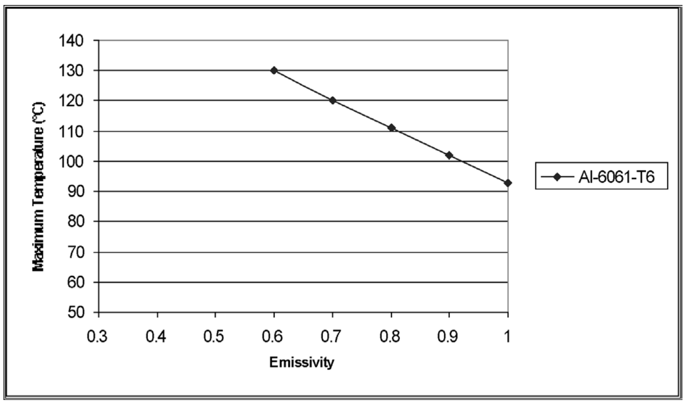

Figure 8: Plot of the maximum temperature versus the shell's thickness for Al-6061

Table 6: Different materials against the maximum nodal temperature

No	Component	Material	Thermal conductivity k (W/m-K)	Density, ρ (kg/m^3)	Heat capacity, C_p (J/kg-K)	T_{max} (°C)	Change in percentage (%)
1	Satelite's structure	Al_6061-T6	156	2700	963	93	-
2		Al 2024-T3	190	2770	963	92	-1.1
3		Al_7079-T6	160	2820	880	95	2.2
4		Pure copper	393	8900	385	90	-3.2
5		Titanium	7.44	4850	544	110	18.3

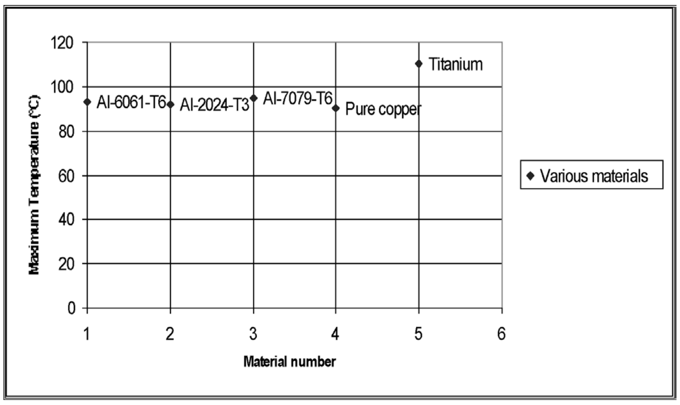

Figure 9: Plot of the maximum temperature profile for various materials

CONCLUSION

The effects of thickness of the satellite structure, emissivities and various material properties have been examined using the finite element ANSYS commercial software. The numerical results from the three-dimensional thermal analysis compared reasonably well with those results found in the open literature. The predicted temperature distributions of the satellite are slightly higher, compared to the open literature values due to the difference in the configurations of structural thickness and materials used. The effects of materials, emissivities and shell thicknesses on the temperature distribution of the satellite in a sun-tracking mode are significant.

NOMENCLATURE

c heat capacity J/kg K
r density, kg/m^3
k thermal conductivity, W/m K
q heat flux, W/m^2
ε emissivity
σ Stefan-Boltzmann constant, 5.67×10^{-8} $W/m^2 K^4$
T temperature, K
T_∞ sink temperature, K

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