THERMAL AND MECHANICAL OPTIMISATION OF THE FIRST ISRAELI NANO-SATELLITE

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ABSTRACT

The mechanical and thermal design considerations of the InKlajn-1 Nano-Sat must take into consideration many aspects simultaneously. A balance between thermal requirements of the components and the small volume and mass available make it a challenging project. Optimisation of all mechanical parts and sub-systems was also paramount taking into account the budget and the boundaries imposed by the Cube-Sat standard.

Small modifications to the structure were made in order to accommodate components or solar panels, all unique to the Inklajn-1 design. The solar panel design was the largest sub system to be designed in-house; the design needed to be robust and reliable but also keeping within the volume boundaries of the P-Pod.

The accommodation of most of the bus systems is done within the size of a PC-104 circuit board thus allowing for a stacking sequence of each board above one another with standoffs keeping the desired space between them. However not all systems (mostly payload) are of the PC-104 size, some are off the shelf components with different shapes and sizes, and therefore the stacking sequence occupies only approximately half the length of the structure with the other components mounted in the other half.

In optimizing the thermal design an emphasis was placed on achieving thermal stability while reducing the need for active control. By utilizing comprehensive and iterative analysis methods a simple passive control scheme was devised. Waste of valuable system resources such as power, volume and mass was thus reduced. The placement of components along with strategically implementing MLI blankets and conduction pathways are key features of this optimal design.

KEYWORDS

CubeSat, PC-104, Solar panels, P-Pod, Nano-Sat,

INTRODUCTION

The first Israeli Nano-Satellite (named InKlajn-1 after the late Dr. Marcel Klajn, one of the key founders of the Israeli space program) was ignited in 2005 with the forming of INSA, the Israeli Nano-Satellite Association.
The satellite, weighing about 3.5Kgs, is being built by a group of experienced professionals from the local space industry working side by side with high school and university students.
This first Nano-Satellite will be used as a test bed for the bus sub-systems to be used on future projects, and will supply space heritage for local companies' products. The payload of InKlajn-1 includes an atomic clock, a GPS Receiver,, a high capacity battery, a corner reflector and an experiment for new coating for multi-layer insulators (MLI).
The miniature satellite design introduced numerous mechanical and thermal challenges due to size limitations and space environment conditions. An iterative process was required to ensure that the mechanical design enables suitable thermal conditions to each sub-system with minimal active thermal control (heaters).

The satellite configuration presented in this article is due to be launched in Q4 2009.

1. MECHANICAL DESIGN

The mechanical sub systems design started with a CubeSat Standard as defined by a group in Stanford, Cal-Poly and manufactured by Pumpkin Inc. A research project was initiated in order to review other CubeSat projects and identify available sub-systems.

The mechanical systems identified where:

- Modifications of the standard CubeSat structure needed for hardware accommodation
- Solar panels design and manufacture including deployment system
- Communication antennas deployment design
- Sub-system accommodation
- Hardware mounting provisions

1.1 Modifications of the standard CubeSat structure

As funds are limited in this project the outsourcing and purchasing was limited to the affordable and off-the-shelf components (excluding one interface electronic board designed to spec by Clyde Space of Glasgow). All other mechanical parts are being designed and manufactured in-house.

The standard skeleton structure consists of a body and two (Front and Rear) covers as seen in figure 1:

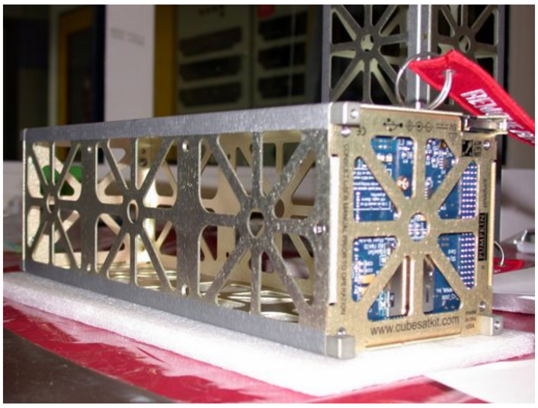

Figure 1- Rear cover with on board computer visible (FM430)

Certain areas are considered stay out zones such as the rails on the corners for deployment from the deploying canister (P-POD); however other areas can be drilled or changed according to the CubeSat design. In our case the front cover was changed to accommodate the GPS antenna and the retro reflector as seen in figure 2, the thermal consideration here was that the ceramic antenna and glass reflector are areas in which are not isolated from the hot or cold environments (thus exposing the satellite to open space) and therefore the cover will be wrapped with a thermal blanket from the inside.

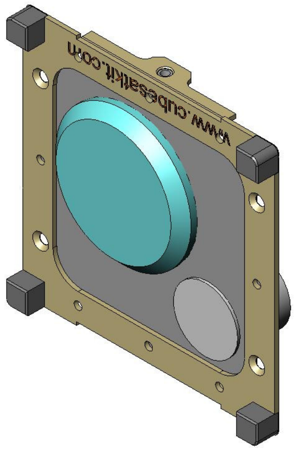

Figure 2-Modified Front Cover

Other minor changes were made to the body structure such as drilling and local filling for bracket and solar panel mounting. An emphasis was made on "As little changes as needed" regime so that the interface to the P-pod and the deployment was not endangered.

1.2 Solar Panels Design

The Solar panels are the major mechanical assembly of the Inklajn-1; the decision to design and manufacture in-house was made due to the high expense of outsourcing or purchasing, and due to the expertise present in IAI MBT MALAM division, to perform the task.

In order to obtain the maximal power the solar panels are deployed to a flower configuration, and the attitude control system points the panels towards the sun direction.

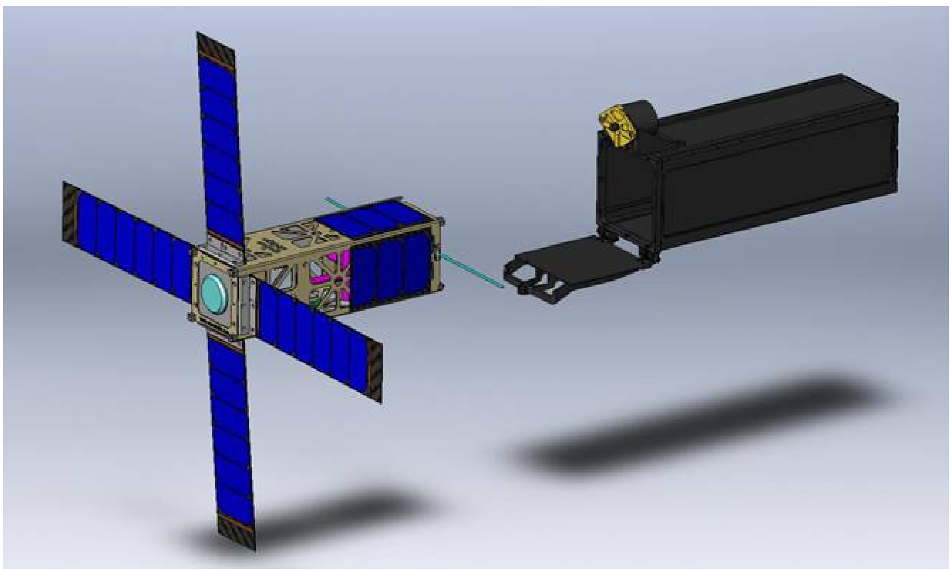

Figure 3- An illustration of the deployed solar panels after the satellite is ejected from the deployer

The design took into consideration all the restrictions apparent in the CubeSat standard, mainly concerning the insertion and deployment in and from the P-Pod and provisions so not to let excessive heat enter the structure via the hinges mounting points by using isolating washers.

The solar panels include the following main components:

- Deployment hinge with stopper
- Deployment hinge spring
- Graphite/Epoxy panel face sheet
- Solar cells
- Wiring
- Panel tip fixture for P-Pod deployment

The hinges where designed so not to protrude more than 6.5 mm above the structure and to be easily mounted on to the structure, the material chosen was to be compatible with the structure and all parts are from identical material in order to ensure even expansion or contraction due to temperature changes in the space environment (figure 4).
The deployment spring was calculated with the parameters of available accommodation in the hinge and the desired time of deployment (about 1.5 seconds). The material chosen for the springs was Stainless Steel 203 which gives better characteristics in lower temperatures than other music wires or stainless steels.
The importance of the spring quality and degradation over time is paramount as the solar panels do not have any locking device, and rely on the pre-loaded spring to maintain a face-to- face lock.

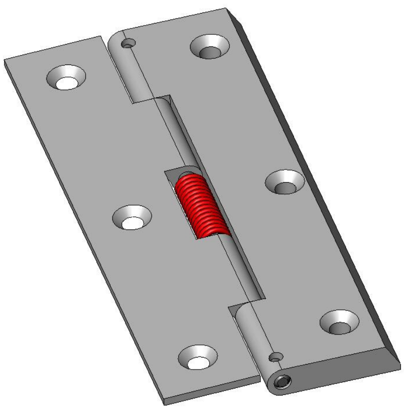

Figure 4- Solar panel hinges with deployment spring

The Graphite/Epoxy lay-up for the panel was chosen at a minimum thickness for a symmetrical and balanced laminate, the strength needed for the laminate is to ensure the solar cells do not exceed an allowed bending value during launch and deployment from the P-Pod.
A quasi-isotropic laminate lay-up is usually preferred for space applications due to its equal characters in all directions and it's near to zero coefficient of thermal expansion. The temperature gradients felt by the spacecraft can be of over 300°C in a very short time.

A perfect quasi-isotropic laminate was not used due the existence of a manufactured laminate left over from previous MBT projects. This laminate is symmetrical and balanced giving a near to zero coefficient of thermal expansion, and although not quasi-isotropic is suitable for the panel requirements.

The solar cells used are triple junction cells at full size (not cut to size) mounted to the panel with adhesive; the two end half size cells (fig 6) are an experiment by MAMAG, testing a new coating for MLI insulating covers. The experiment will measure the current generated by the cells as the atomic Oxygen particles of the space environment degrade the coating on the cells in time.
The wiring of the panels with thermistors and diodes are part of the electric power distribution sub-system (EPDS) and will not be discussed here. The mechanical aspect of the wiring is that all wires run along the panel on the rear side and enter the spacecraft at a designated point close to the rotating axis of the hinges.

The solar panels of the InKlajn-1 will not be tied to the satellite structure, the panel tips will be accommodated with a Teflon runner enabling the panel to run along the P-Pod inner walls during deployment and open as soon as the satellite is free from the pod.

As mentioned before, the solar panels do not have a deployment locking mechanism at the end of the rotation to final position; the panels will rely on the pre-loaded spring to rest the hinge face to face in the final position of 100 degrees (figure 5). Space proven lubricants will be used in the hinge against friction.
The fact that the panels do not stop at 90 degrees is due to the use of the solar panels as sun sensors for the purpose of attitude determination, the algorithm utilises this angle to determine the correct direction of the sun (due to the angle, each panel will give a different current) in respect to the deployed solar panels and thus calculate the correction needed.

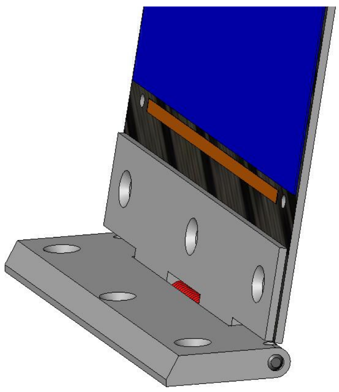

Figure 5-Solar panel end position

The manufacturing process of the panels is nearly complete, a deployment and end of travel shock test will be conducted at zero G conditions thus qualifying the design.
The final solar panel assembly can be seen in figures 6 and 7.

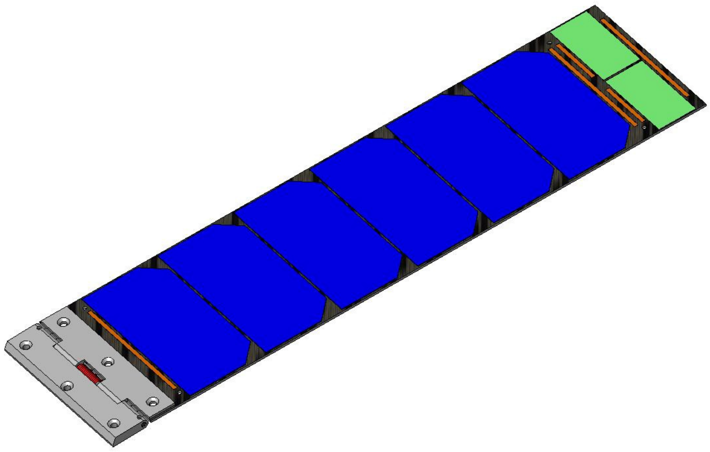

Figure 6- Full solar panel assembly top

Figure 7-Full solar panel assembly bottom

1.3 Communication antennas

The communication antennas definition and deployment design has not been finalised at this time due to alternating choice of transponders. In any case, the antennas will be of simple spring metal tape alloy (such as tape measure) and deployed in a coil fashion similar to the ISIS system (fig 8) or folded under the solar panels.
The antennas will have a length of approximately one- quarter wavelength.

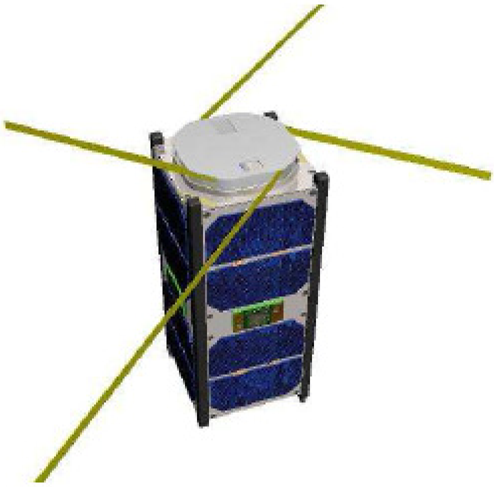

Figure 8-ISIS antenna system

1.4 Sub-system accommodation

The hardware mounting of the systems onto and into the CubeSat structure are apparent in two categories:

- Components on the PC-104 standard stack
- Components needing typical mounting provisions (bracket fixtures)

Approximately half of the volume in the 3U CubeSat structure is occupied by the spacecraft bus; all the bus components besides the magnetorquers are either mounted on or printed to the PC-104 board standard. These layers of boards are stacked together using standoffs and fit nicely to the CubeSat dimensions as part of the CubeSat standard.
Figure 9 shows the InKlajn-1 bus stack including the following layers (from bottom):

- FM-430 on board computer
- EPS-1
- EPS-2
- FM-430 on board computer
- Balancing weight layer
- Switch board (developed by INSA and Clyde Space Ltd)
- ISIS Transponder
- ALINCO transponder
- Magnet Torquers

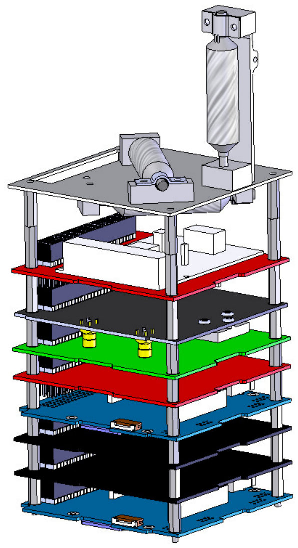

Figure 9- Inklajn-1 S/C Bus

1.5 Hardware mounting provisions

Other components are from other fields and are not space proven or suited for satellite installation, therefore the stack is ended at a certain point and the remaining space craft volume is accommodated with other components and mounting provisions.

These other provisions are in the form of base plates or brackets, all to undergo vibration, thermal and acoustic testing on spacecraft level (figure 10). The atomic clock and battery have high heat dissipation needs in relevance to other components in the spacecraft; therefore they are mounted on two large Aluminium base-plates performing as radiators. The heat from the components will be conducted through the base-plate to the structure and then radiated to space.

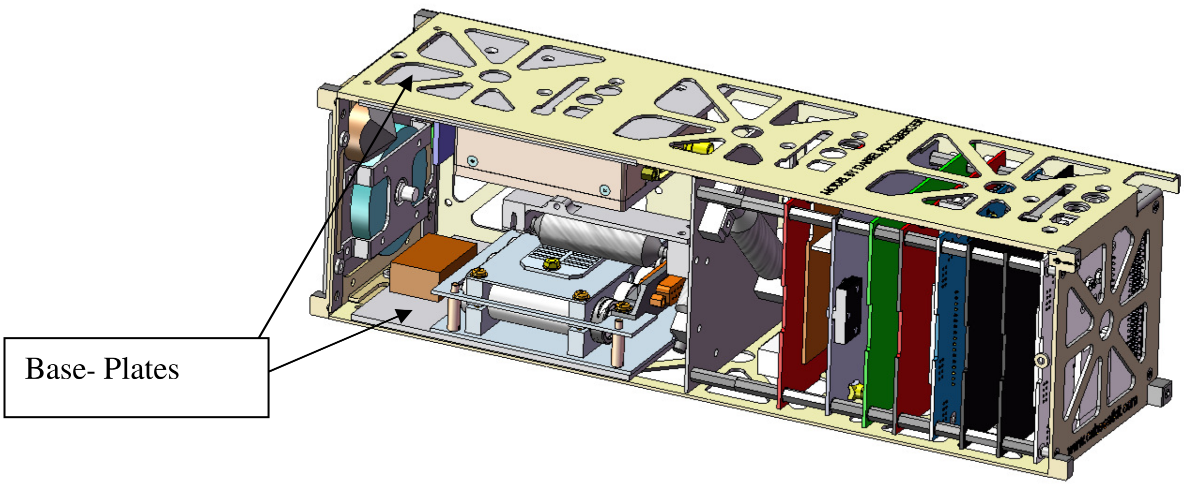

Figure 10- Components internal look

2. THERMAL OPTIMISATION

The thermal environment of any satellite in orbit can be thought of in terms of an energy balance between inputs and outputs to the system. The inputs are heat generated by the spacecraft onboard components, direct sunlight, sunlight reflected off the Earth (Albedo), and infrared radiation emitted by the Earth. The output is radiated heat from the satellite to space. McMordie [1] gives a didactical explanation of this problem. The inputs to the system are variable in both space and time and sometime are not completely predictable. The most notable variant in this energy balance is direct sunlight as the satellite goes in and out of eclipse. During illumination the satellite will be exposed to approximately 1400 W/m^2 from the sun compared to no radiation during eclipse. This has a drastic effect on the energy balance of the system and therefore requires close examination of the satellite's temperature variation throughout its orbit. The effect is only intensified when considering InKlajn-1 due to its low mass. The cold case of the satellite during eclipse may be just as limiting as the hot case when it is in full illumination.

The ratio between the periods of time spent in the sun to the amount of time spent in eclipse is dependant on the specific orbit in which InKlajn-1 is placed as well as the time of year and therefore affects the cold and hot cases. The specific orbit parameters were unknown during the majority of the design phase which brings forth the need for a robust, adaptable thermal design.

In determining the control scheme, every component's acceptable temperature ranges were examined. The satellite bus components include the structure, On-Board Computer subsystem (OBC), Command and Telemetry subsystem, Power subsystem, Attitude Control System (ACS), and Thermal Control System. The payload on InKlajn-1 includes an atomic clock, a GPS receiver and a battery pack. The vast majority of components were selected such that they withstand temperature fluctuation between -10°C and +50°C.

There are a number of methods to control satellite temperature. Some of them are passive such as radiators, phase change materials, MLI blankets, heat pipes and thermal paints, and some are active (i.e. require use of electrical energy) such as louvers and heaters. The design chosen for InKlajn-1 must also consider the program constraints of cost, mass, volume, and power. These constraints are present in all satellite programs, but they are especially tight in small spacecraft such as InKlajn-1.

The approach has been to minimize the use of program resources and come up with a design that most efficiently meets the TCS (Thermal Control System) requirement.

Due to the above considerations it is possible to significantly narrow the design possibilities for the TCS. The intent is to design the TCS such that energy consumption is minimized, i.e. the passive approach. In some cases, the use of heaters may be necessary.

In general, thermal designs are more successful if they are made to be independent from other sub-systems. The spacecraft design may require significant changes along the development cycle, which increases the desire for an easily adaptive and lean thermal design. Use of simplified conduction pathways and choices regarding MLI blankets will be used to achieve

the goal of transporting heat away from the at-risk components and reaching desired temperatures throughout the satellite.
Thermal paints are a relatively inexpensive way to facilitate heat transfer within InKlajn-1; they may be used on the solar panel rear side and the inside planes of the structure.

2.1 Analytical Thermal Analysis

In the thermal analysis process, it is beneficial to first get a rough idea of expected temperature ranges. For this purpose an analytical calculation was performed based on the energy balance between the heat absorbed in the form of radiation, the heat generated by the components that make up the satellite and the heat released by it. This is a steady state analysis that will not show temperature changes with time. The results of this analysis are qualitative only and do not change the need for a rigorous dynamic temperature calculation.

The internal heat generation and satellite orbit assumptions used for this analysis are as follows.

Orbit Definition and satellite attitude

The orbit used in this analysis was a sun synchronous orbit at an altitude of 650km. The orbit dynamics play a significant role in determining temperatures on the satellite as they determine view factors and length of exposure to direct sunlight, Albedo radiation, and earthshine. Heat input from these sources is determined by multiplying the area facing the source by its radiation intensity and absorptivity. The satellite's Z-axis points towards the sun to maximize solar energy production as shown in figure 11

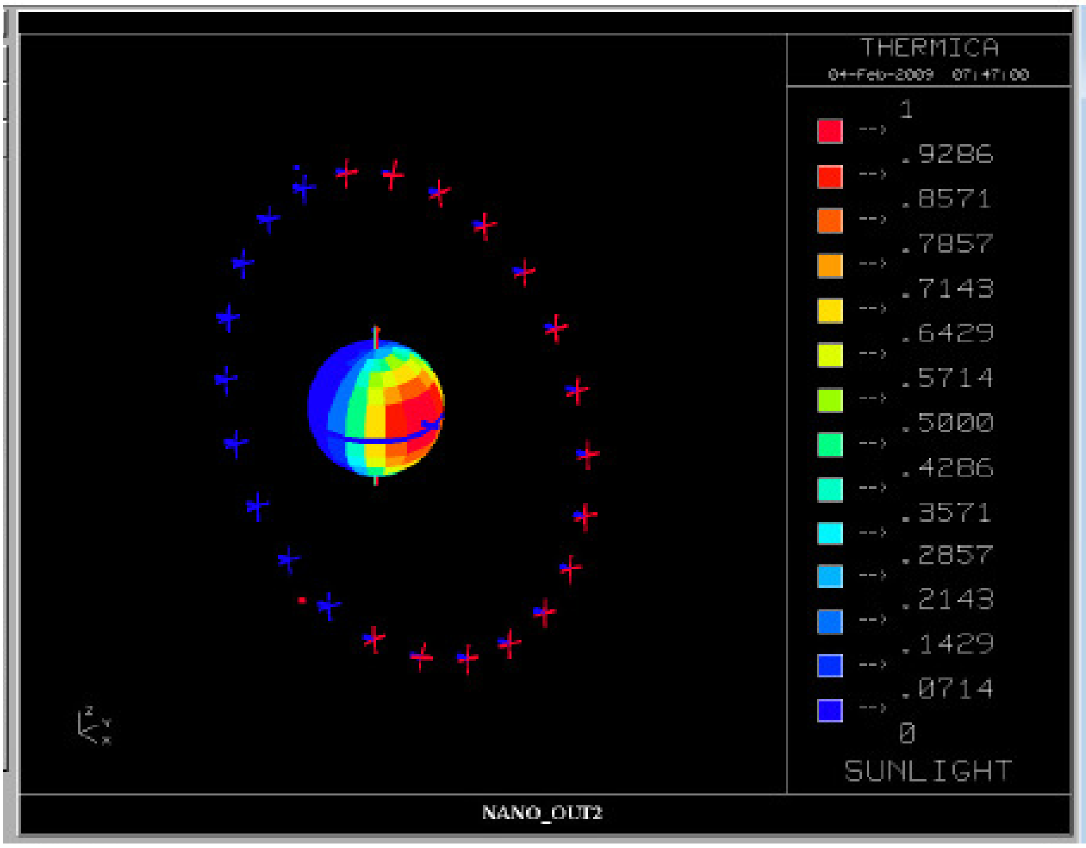

Figure 11 – Orbit definition

Heat Generation

Heat generation influences the temperature profile of the components in the satellite. The heat generated by each electronic board was determined and used as an input to the analysis. Generated heat on InKlajn can be seen below in figure 12

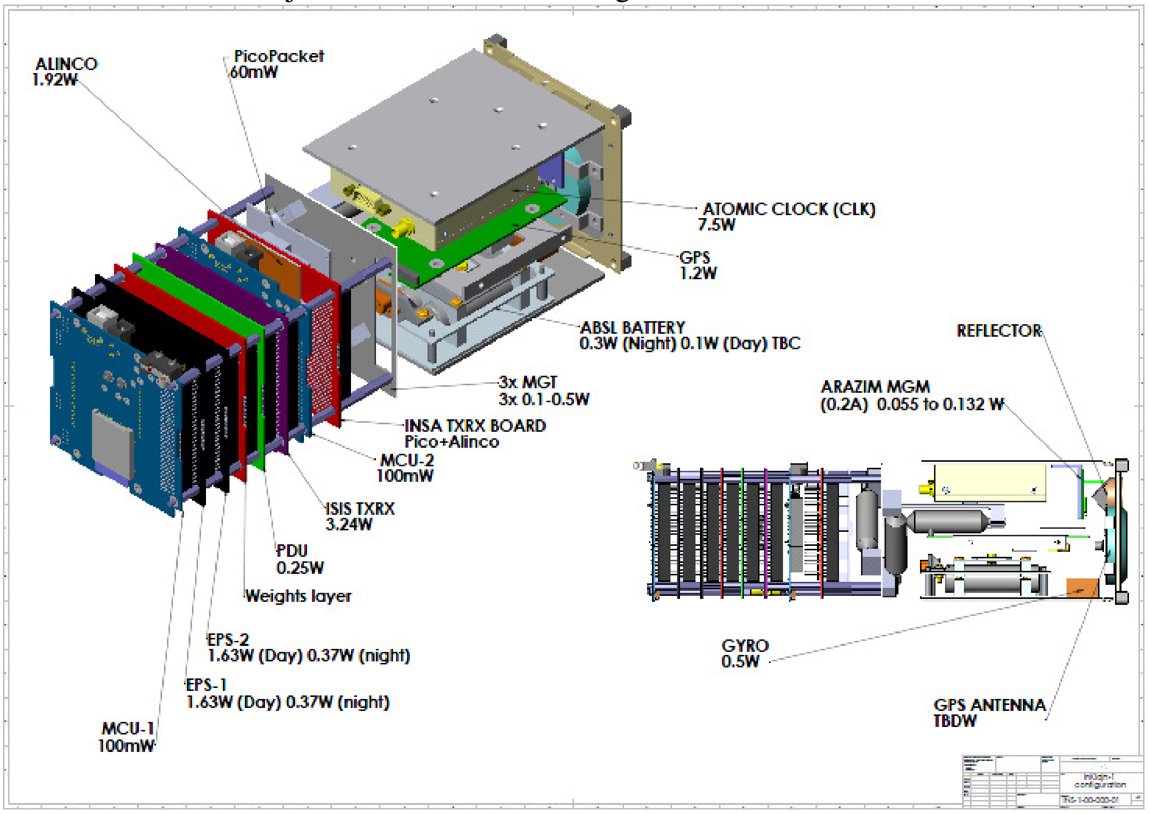

Figure 12: Heat generated in various InKlajn components

Based on the info presented in figure 12 two cases were defined in terms of heat generation; a hot and cold case with 20W and 11W of heat generation respectively. The cold case is one where the atomic clock and GPS unit are turned off.

With these assumptions the energy balance is defined by equation 1.

$$q_{absorbed} - q_{emitted} + q_{power-generated} = 0 \qquad (1)$$

The power absorbed for the hot case can be broken down as follows in equation 2,

$$q_{absorbed} = G_{DS} A_{DS} \alpha + G_{DS} a A_E \alpha \left(\frac{R_E{}^2}{(H + R_E)^2} \right) + G_{IR} A_E \varepsilon \left(\frac{R_E{}^2}{(H + R_E)^2} \right) \qquad (2)$$

With the three terms in equation 2 referring to heat absorbed from direct sunlight, albedo, and Earth IR respectively. Terms in equation 2 are defined in table 1.

Table 1: Term definition

Term	Definition	Term	Definition
G_{DS}	Solar irradiance near the Earth, between 1300-1400 W/m^2	R_E	Radius of the Earth - 6378km
A_{DS}	Area facing the sun [m^2]	H	Height of the satellite above the Earth surface
α	Absorptivity	G_{IR}	IR irradiance [W/m^2] at Earth surface
a	Albedo, generally around 35%	A_E	Area facing the Earth
ε	Emissivity	σ	Stephen Boltzmann Constant 5.67x10^{-8} W/(m^2-k^4)

The power emitted is found using Stephan-Boltzmann law as shown in equation 3,

$$q_{emitted} = A_{out}\varepsilon\sigma T^4 \tag{3}$$

Substituting equation 3 and equation 2 into equation 1 and solving for the temperature results in equation 4.

$$T_{HOT} = \left(\frac{G_{DS}A_{DS}\alpha + G_{DS}aA_E\alpha\left(\frac{R_E^{\,2}}{(H+R_E)^2}\right) + G_{IR}A_E\varepsilon\left(\frac{R_E^{\,2}}{(H+R_E)^2}\right) + q_{power-generated}}{A_{out}\varepsilon\sigma} \right)^{1/4} \tag{4}$$

All the terms remain in the hot case condition while in the cold case the albedo and direct sunlight go to zero as shown in equation 5.

$$T_{COLD} = \left(\frac{G_{IR}A_E\varepsilon\left(\frac{R_E^{\,2}}{(H+R_E)^2}\right) + q_{power-generated}}{A_{out}\varepsilon\sigma} \right)^{1/4} \tag{5}$$

Notice that the cold case temperature is independent of absorptivity.

Utilizing values relating to InKlajn-1 and to the cases defined above and allowing emissivity and absorptivity to vary from 0.2 to 1, the worst case hot temperatures are found and plotted in figure 13.

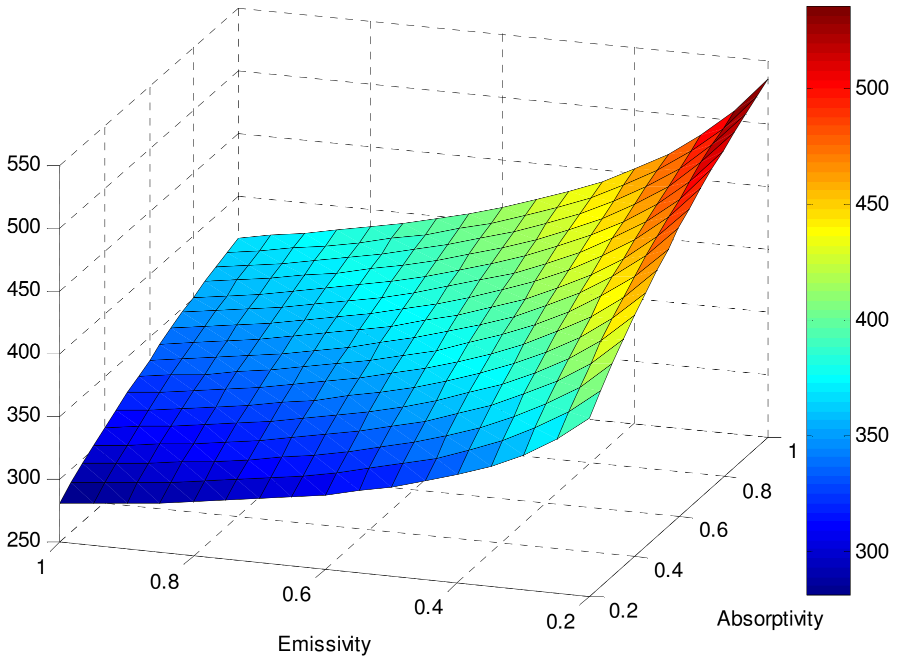

Figure 11-Worst case hot- steady state temperature

The worst case cold temperature is shown in figure 14.

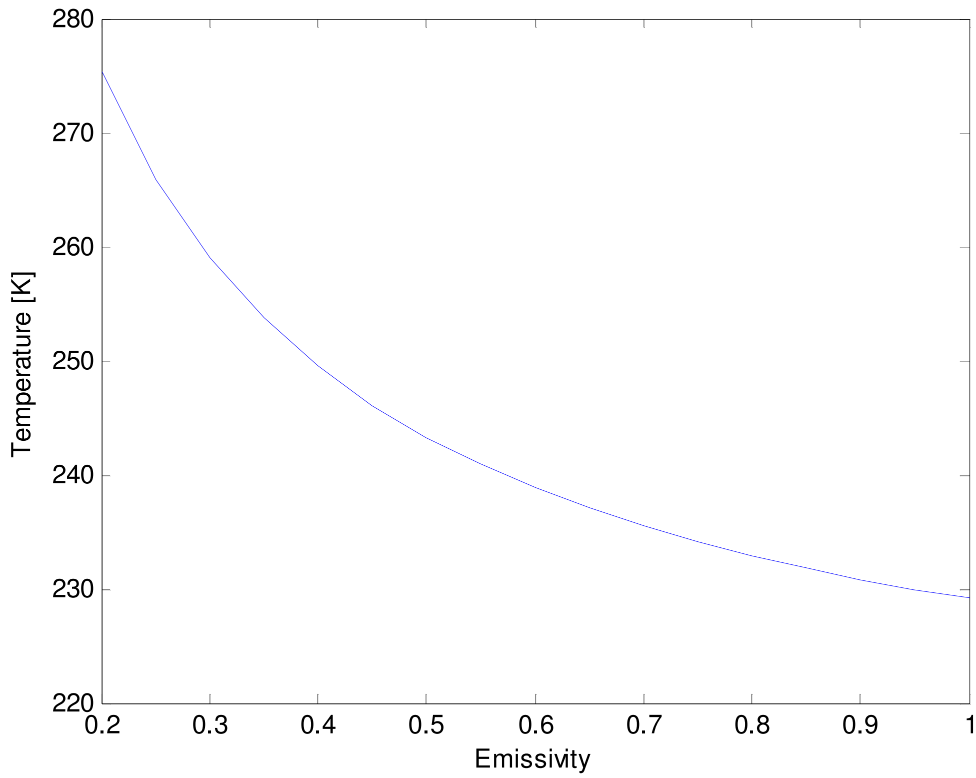

Figure 12-Worst case cold- temperatures vs. emissivity

The worst case cold and worst case hot temperatures can be seen as effective upper and lower bounds for InKlajn-1; however the actual temperature will likely never approach these. One reason is that the satellite will never have enough time to reach steady state. At the rate of the emitted energy of InKlajn-1 the satellite will only reach a temperature of 270K by the end of eclipse if it had started the eclipse period at 300K. In other words, the mass of the satellite does not allow for rapid enough temperature changes and so the satellite will never reach these upper and lower bounds.
Another important distinction is that temperature will not be constant throughout the satellite. The solar panels which have a large exposed surface area and little mass will vary greatly in temperature while rest of the satellite will not.

2.2 Numerical Thermal Analysis

In order to get a more detailed understanding of local dynamic temperatures in the satellite a CAD-based thermal model was created. The model was used in optimizing radiators, heaters, and the overall mechanical design.

Pre-processing

A numerical thermal analysis was done using Thermica, a graphical pre- and post- processor for the SINDA/G finite difference analysis thermal analyzer. The first step in conducting the thermal analysis is building the physical model of InKlajn-1 in Thermica. As shown in figure 10 a detailed CAD model of the satellite is readily available and has been instrumental in creating the physical model. However, due to the computational expense of the thermal analysis, many simplifications were done in the thermal model geometry. Many of the intricacies of the structural design are thermally inconsequential, and so were left out. Instead of modelling fasteners and small components, features were merged into bigger elements. Much thought was put into modelling the thermal contacts between elements as accurately as possible such that the simplifications made in creating the physical model do not adversely affect the accuracy of the thermal model. Regardless, creating a model with a rougher resolution runs the risk of averaging out temperature extremes in elements that are finer than the resolution of the model. Such is the case with electronic boards which are modelled by four elements and may hide hot spots that exist on certain components. Consequently, larger margins of safety on the results were used where this phenomenon was deemed significant. In total, about 150 elements were used to model InKlajn-1 in Thermica.

Thermal Property Assignment

After representative elements of the satellite were created each element was assigned a thermal mass which is the product of the element's mass and specific heat capacity. The model resolution necessitated a non-trivial approach to assigning this property since each element has undergone some simplification. A simple example is the structural wall of the satellite. In reality the wall consists of a complex grid whereas in the thermal model it appears as a solid plate. Since the mass of the wall is known as well as its material composition, adjusting the thermal mass of the elements was fairly straightforward. On the other hand properties of elements that model the electronic boards were much more difficult to approximate. The boards and the components on them consist of various materials at varying ratios from board to board. In these cases, the overall mass of the board was used in conjunction with approximations of the material composition of each board. The material properties were then averaged, weighted by their prevalence on the board.

Thermal Contact Approximation

Approximating the thermal contacts between all the elements presented a unique challenge. Extensive analysis of the mechanical CAD model was used to determine the contact value that should be used between contacting elements. Conduction contacts were calculated as shown in equation 6.

$$G = \frac{kA}{dx} \tag{6}$$

Where k is the thermal conductivity [W/(m-K)], A is the cross sectional area of the contact [m^2], and dx is the effective distance between the elements [m]. Contact conductance was calculated according to equation 7.

$$G = hA \tag{7}$$

Where h is the heat transfer coefficient [W/(m^2-K)]. Multiplying the value of G by the temperature difference of the contacting elements results in the heat transfer rate used to solve for the elements' temperature with time.

Radiation contacts were calculated by multiplying the emissivity of the elements' by their radiating area. View factors between all elements are automatically calculated by Thermica using a Monte-Carlo-based ray tracing radiation program.

Following the completion of the pre-processing steps described above, the temperature of all elements of InKlajn was solved for using SINDA/G.

Initial Results and optimization

Since the solar panels located on the satellite's side walls act as radiators, the initial impression was that these radiators would excessively cool the electronic boards and therefore must be insulated from internal components. This was achieved by implementing thermal blankets between the panels and the interior of the satellite. However, since the view factors between all the boards and the solar panels are low (up to 0.1) the panels acted as poor radiators. Temperatures on board elements were too high and risked exceeding operating limits. Consequently, insulating blankets behind the panels were removed and additional, more effective radiators were placed in the top and bottom walls of the satellite. Exterior emissivity values after these changes were made can be seen in figure 15

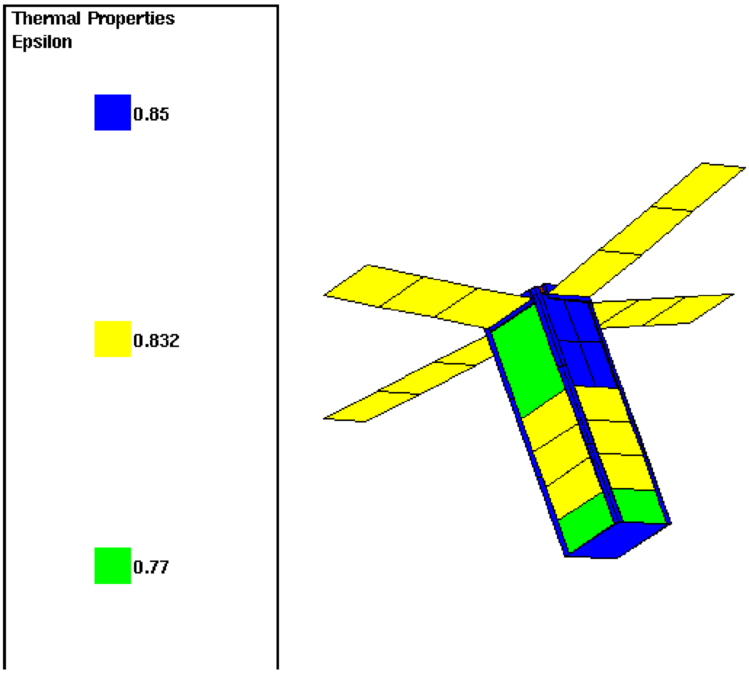

Figure 15: Exterior emissivity values

Figure 15 shows solar panels in yellow, thermal blankets in green and radiators in blue.
To further reduce the prevalence of localized hot-spots on electronic boards and to facilitate heat transfer out of the boards it was decided to coat them with a high emissivity layer (Solitan coating). The coating will also penetrate between the components and the board and enhance heat transfer to the board. The layer will make the element-averaged temperature assumption more valid while the high emissivity will allow more heat to transfer from the boards to the satellite's radiators.
The new goal for this configuration was for board temperatures to remain between -10°C and +50°C throughout the orbit.

Hot Case Results

Resulting board temperatures in the hot case are shown in figure 16.

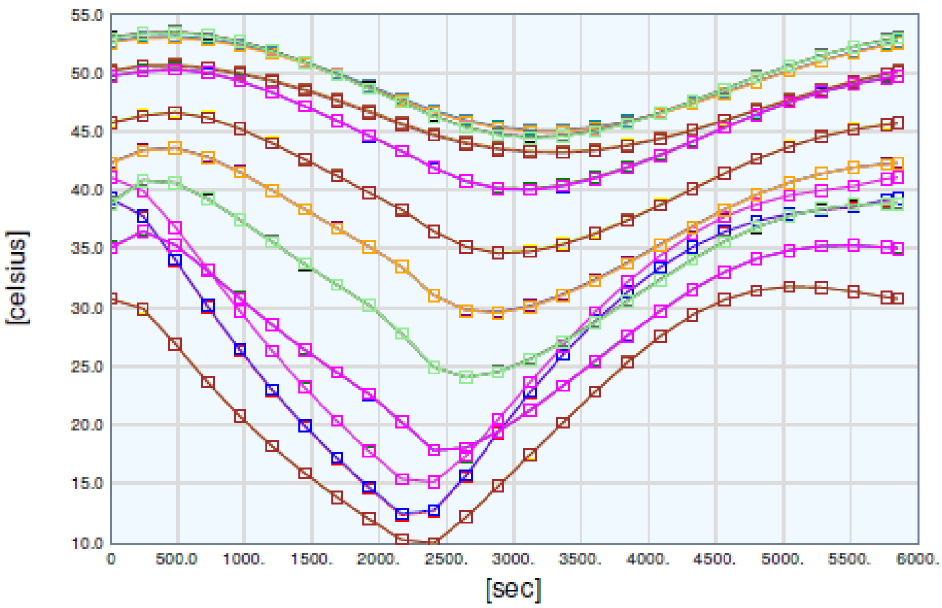

Figure 16: Hot case board temperatures

The results show a mild deviation from desired temperature on the hot side for certain boards that are less capable of transferring heat to their environment. As expected, boards further away from the bottom wall became increasingly hotter. This is a result of the physical connection between the board stack and the satellite exterior located at the bottom wall. Through this conductive contact heat is transferred to the exterior panels and subsequently to the space environment.
Exterior temperatures for the hot case are shown in figure 17:

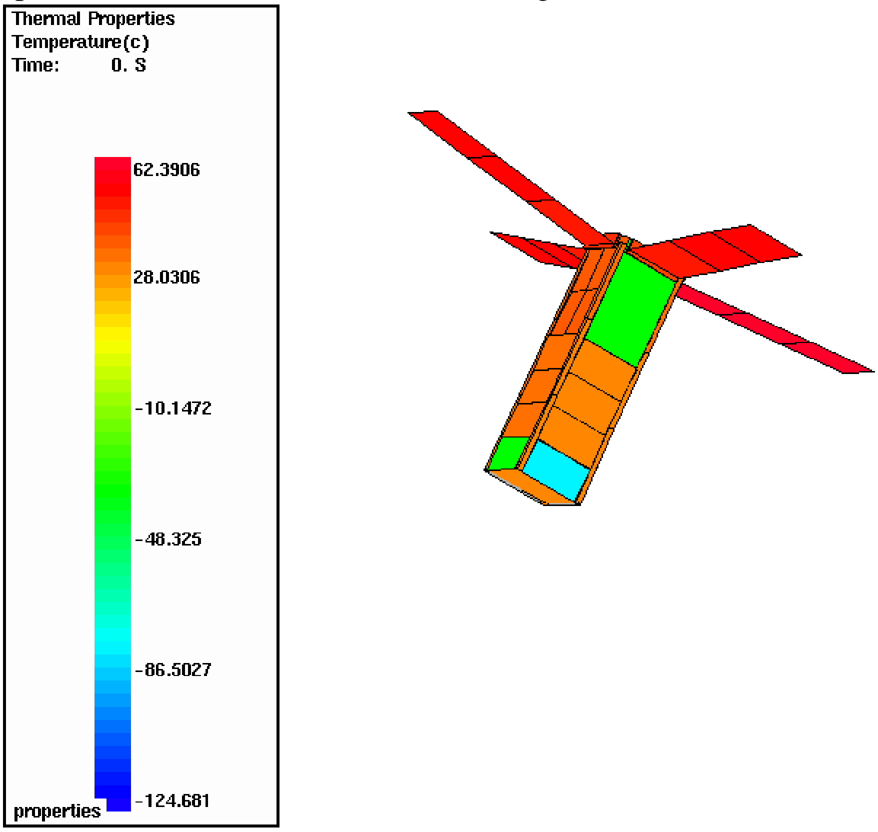

Figure 17: Exterior hot-case temperatures

Exterior temperatures are shown to vary greatly between elements. The largest variation results from low thermal contact between the blankets and the rest of the satellite structure. While the temperatures encountered on the exterior may seem rather extreme, the temperature of the internal layer of the MLI will be much closer to the internal temperature of the spacecraft.

Cold Case Results

Board temperatures in the cold case are shown in figure 18:

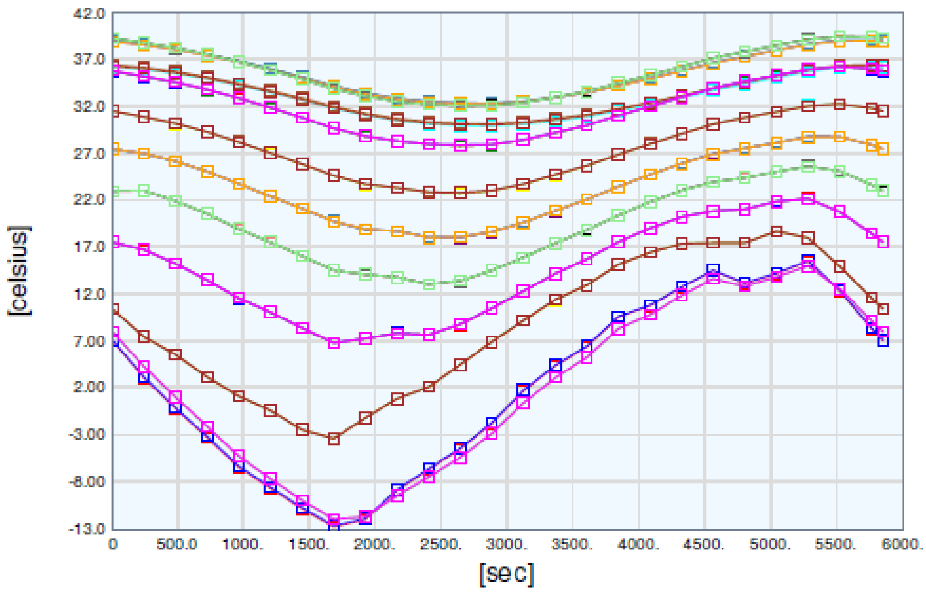

Figure 18: Cold case board temperatures

These temperatures are fairly mild and are within desired limits except for a brief deviation on the cold end for one of the boards. This is not expected to risk board integrity. Based on these results no heaters will be used on the electronic boards. If the concept of operation changes such that more boards will be off in a nominal mode provisions such as thermal blankets will be used so as to not exceed temperature limits.
Exterior temperatures in the cold case did not significantly vary from the hot case due to the limited thermal contact between the boards and the rest of the structure.

The results of the detailed component-level analysis were used to achieve an optimal mechanical and thermal design. The optimal design takes in both the thermal and mechanical requirements of the satellite and comes up with a solution that meets those requirements while minimizing the cost on system resources. By increasing heat transfer between hot and cold components the design minimizes the need for heaters that use up valuable power, as well as reduces the risk associated with overheating components.

Risk Mitigation

Results of the thermal model have been used to mitigate temperature-related failure risks. The mechanical redesign has been shown to be an effective way to reduce predicted temperatures on critical components. However, due to the limitations in the model and proximity of predicted temperatures to the desired temperature envelope further steps have been taken to reduce risk.
To address the main deficiency of the model in averaging out hot spot IR imagery will be used to identify local hot spots and the relative temperatures of all components on the electronic boards. If needed, these hot spots will be handled on a case by case basis; for example by creating a copper conduction pathway between the critical components and the radiators.
The satellite will also be tested in a vacuum chamber in order to verify and calibrate the thermal model to within an accuracy of ±10°C.

CONCLUSION

The Inklajn-1 satellite has introduced many challenges; these arose due to the use of new technologies (MEMS) for space applications, tight accommodation issues and thermal control complexity.
The thermal challenges resulted in a close look at all the mechanical materials and an assessment of the quality of conduction paths throughout the spacecraft. Techniques of thermal spreading (such as Solitan coating) and radiating paints must be used in order to moderate the temperatures in the spacecraft.
The payload bay of the design initially included two large radiators (one on the atomic clock side and one on the battery side), this mechanical design turned out to serve the thermal needs precisely and was kept after the thermal analysis was completed.
The notion that the side solar panels may act as radiators and cool the satellite to a freezing point turned out to be incorrect due to the low view factor between the radiators and the electronic boards.
The design is close to configuration freezing in which the components design and accommodation will be finalized and the thermal control design determined.
The selection of components and their ability to coexist will be examined in the integration process of the satellite; other issues will be tested in the verification and testing program including vibration testing, thermal vacuum testing and IR imaging.

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