Proceedings of the 23rd National Heat and Mass Transfer Conference and
1st International ISHMT-ASTFE Heat and Mass Transfer Conference
IHMTC2015
17-20 December, 2015, Thiruvananthapuram, India

IHMTC2015-233

THERMAL MANAGEMENT OF LAUNCH VEHICLE INERTIAL NAVIAGATION PACKAGE FOR ISRO'S MARS MISSION

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ABSTRACT

Thermal management of inertial navigation package of launch vehicle was critical for Mars mission due to the longer flight duration of ~2600s as compared to all previous missions (~1200s). To ensure nominal system performance of the inertial sensors inside the package, their temperature levels have to be maintained within prescribed limits. Simulation of flight thermal environments experienced by the package till the end of operation is impossible to achieve through ground tests. Hence the approach adopted was to depend entirely on thermal modeling of the complex system and to predict its temperature within the acceptable accuracy levels of ±2^{o}C for the extended operating duration with a forwards bias. A thermal model of the package was developed considering the internal component level details. Model parameters were arrived at from experiments conducted under controlled environmental conditions. The maximum predicted temperature for Mars mission was 43.2^{o}C, against a measured value of 41.4^{o}C.

Keywords**:** Avionic package, Thermal management, Mars Mission.

INTRODUCTION

The inertial navigation package in launch vehicle (LV) provides information on vehicle position, velocity and attitude in a continuous manner to achieve the intended orbit. Accurate information from the package will help to reduce the orbital errors. The information is obtained using inertial sensors – gyroscopes and accelerometers, which are mounted to a rigid block called cluster housed inside the package **[1]**. To ensure nominal system performance of the inertial sensors inside the package, their temperatures have to be maintained within prescribed limits.

The package is mounted on honeycomb deck structure inside the heat shield (HS). Apart from its high power dissipation, the thermal environments experienced by the package includes convective cooling during prelaunch operations and cooling due to depressurization of compartment air in the ascent phase, solar load and albedo and radiation loss to space after heat shield separation **[2]**, as summarized in Tab.1. Thermal management adopted is to cool the package during pre launch operation using chilled dry air routed through flexible hoses. This limits the temperatures at lift off such that the internal and external thermal loads during flight will contain the package temperatures within its allowable limits. The challenge in Mars mission was to maintain the package temperatures within limits for a long duration of

~2600s which is more than double the normal flight duration of approximately 1200s. To decide on the adequacy of the pre launch cooling design, precise thermal modeling and quantification of all the external thermal environments were essential. As simulation of all these thermal environments experienced by the package till the end of operation is almost impossible to achieve through ground tests, the approach adopted was to depend entirely on thermal modeling of the complex system involving several components. Conventional lumped mass approaches, where the entire package is considered as a single node or avionic packages are modeled as cuboids **[2]**, were found to be inadequate. The challenge was to predict the package temperatures within an accuracy of ±2^{o}C for the entire operating duration.

TABLE 1: THERMAL ENVIRONMENTS

Thermal Environments	Pre-launch Phase	Flight Phase	
		Lift-off to HS Separation	HS Separation to spacecraft separation
Internal power dissipation	Present	Present	Present
Conduction to mounting deck	Present	Present	Present
Radiation to nearby structures	Present	Present	Present
Convection	Present	Present	Absent
Solar / Earth emitted radiation	Absent	Absent	Present
Radiation loss to space	Absent	Absent	Present

This paper brings out the complexity in the component level thermal modeling of the navigation package. Experiments performed under controlled environmental conditions which were used for arriving at the appropriate model parameters are described. Model was then integrated in the LV avionics bay model. The computed results using this integrated model are compared with previous flight measured data and validated. It was then used for predicting the package temperatures for Mars mission. The various external thermal environments experienced by the package are described and quantified. Comparison of the pre-flight computed temperature predictions using the developed model shows good agreement with flight measured data.

INERTIAL NAVIGATIONAL PACKAGE

The navigational package, called ISU (Inertial Sensing Unit), houses the cluster containing gyros and accelerometers in a skewed configuration for fail safe operation **[1]**. It is mounted on vibration isolators inside the chassis which are good thermal insulators as well. Cluster and sensor temperatures are precisely controlled using heaters. ISU also contains the electronics needed to operate the sensors. In order to maintain the temperature of critical internal components within limits, it is essential to maintain the chassis temperature within 45^{o}C. This temperature constraint value has been arrived at based on various test results and is measured using an RTD pasted at the base of the package. Thus for Mars mission, the pre-launch cooling design should ensure that the chassis temperature should be within 45^{o}C till the end of operation. Figure.1 shows the ISU package mounted on honey comb deck in LV avionics bay along with other packages. Flexible silicone hose of one inch diameter used to supply chilled dry air to the package is also shown. The package is made of magnesium alloy.

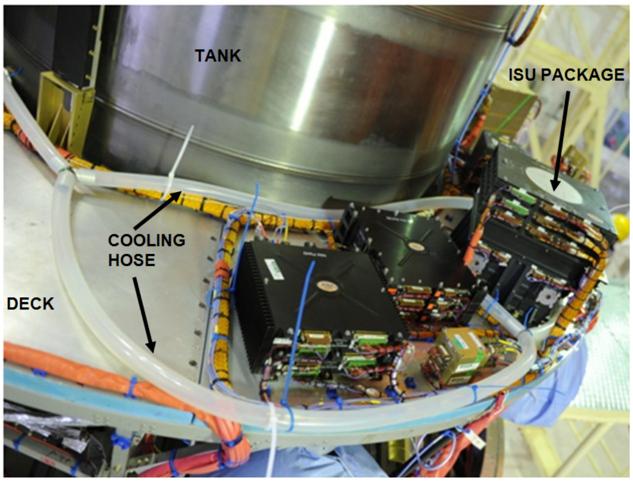

FIGURE 1. ISU PACKAGE IN LV AVIONICS BAY

THERMAL CHARACTERISATION TESTS

Simulating all the thermal environments experienced by the package from pre-launch phase till spacecraft (S/C) separation under laboratory conditions is nearly impossible especially due to the variation of solar load on different regions of package. Hence the thermal characterization tests were aimed at thermally mapping the package to understand the various heat flow paths and to generate sufficient data for thermal modeling. Thermal model parameters and other spatial boundary conditions were derived from these tests, details of which are described in subsequent sections. Similar approach has been adopted elsewhere also **[3]**. Figure.2 shows the test setup along with its schematic.

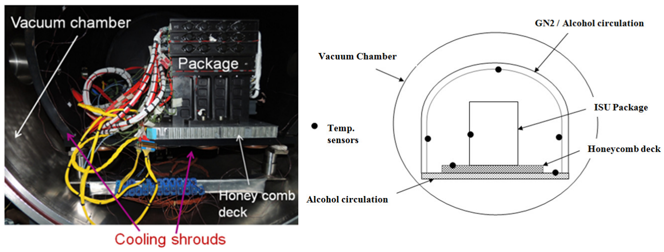

FIGURE 2. TEST SETUP AND ITS SCHEMATIC

ISU package is mounted on a 32mm thick aluminium honeycomb deck having 1mm skin thickness and 20Nm torque is applied for fastening. The deck thickness and torque values selected are based on the mounting scheme followed in LV avionics bay. Detailed temperature measurements are made at several locations on the package, shroud walls and on the honeycomb deck plate. As the package is working in vacuum conditions in space, the experiments are performed inside a vacuum chamber with vacuum level of 10^{-5} mbar to eliminate convection. The tests do not capture the thermal environments experienced by the package as mentioned in Tab.1. Effect of conduction to mounting deck and absence of convection after HS separation are only simulated as the objective is to generate data for thermal modeling.

TEST PROCEDURE

Package was powered ON for 4 hours and temperatures are allowed to stabilize. The chamber was then evacuated, shroud cooling was initiated and package temperatures were allowed to stabilize. In the absence of localized cooling to ISU package which is provided during prelaunch phase for safe functioning of the package, the base temperatures have to be controlled. Hence alcohol is circulated through tubes in the bottom shroud below honeycomb deck for precise temperature control. For cooling the cylindrical shroud, liquid nitrogen is intermittently circulated through the tubes in the shroud. Oscillations were seen in shroud temperatures during liquid nitrogen circulation. Once the deck temperature close to 30°C was achieved, cooling through the shrouds is stopped. The experiment was then continued for 3600s. Deck temperature value was adopted based on measurements in LV avionics bay in previous missions. Measured ISU package chassis temperatures at various points show a maximum variation of 15°C. The measured deck plate and shroud temperatures are shown in Fig.3.

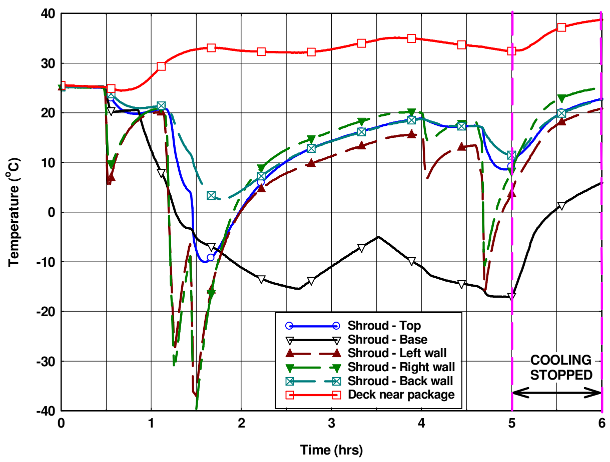

FIGURE 3. MEASURED DECK PLATE AND SHROUD TEMPERATURES

THERMAL MODEL OF ISU PACKAGE AND EXPERIMENTAL SETUP

The detailed FE model of the package was developed using commercial software NX9.0 **[4]** as shown in Fig.4. Magnesium alloy properties are considered for the chassis. In order to simplify the model and to optimize the mesh, details such as small chamfers, harnesses and screws have not been modeled. However to simulate the thermal mass of chassis, the density values were adjusted to simulate the measured mass. Tetrahedral elements are used for meshing the package and the element size varies from 2mm to 20mm to capture the geometric features. Total stabilized power dissipation of the package is around 160W and may vary as and when the active control is present. Power dissipation values of components are applied at appropriate locations in chassis. Evaluation of thermal properties of electronic boards is extremely difficult **[3]** and hence is not modeled. Their power dissipation is directly applied at the respective locations in the chassis assuming that the power is uniformly dissipated in the cards and is entirely transferred to chassis. This approach will help to predict the chassis temperature with a forward bias, as the thermal mass of electronic cards is not accounted.

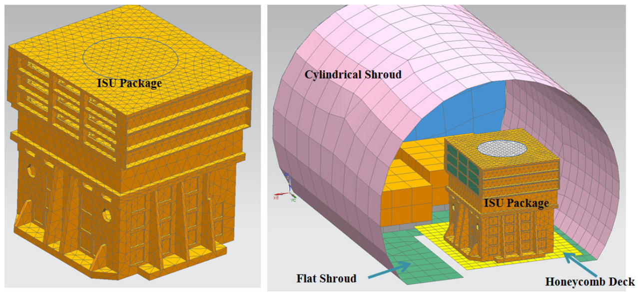

FIGURE 4. FE MODEL OF ISU PACKAGE AND THERMAL MODEL OF TEST SETUP

It was observed during the tests that in regions where active heaters are mounted and temperatures are maintained at predefined set values, such as inertial sensors and cluster, the temperatures were stable as long as the chassis temperature was maintained below 45°C. Hence the model uses the measured temperatures as boundary conditions in these inner regions. As the objective is to predict the chassis temperature accurately, this assumption is justified. Even for a few degree changes in the cluster temperature the chassis temperature will not be affected as the temperature levels of inner components are below 60°C and the dominant mode of heat transfer between cluster and chassis is by radiation.

The thermal model of the test setup used for validation is also shown in Fig.4. Other avionic packages have been modeled as cuboids and their location, mass, power dissipation and surface finish have been accounted.

The detailed FE model of the ISU package has been incorporated inside the shroud. Radiation exchange with the shroud and nearby packages is simulated. Thermal response of shroud walls and honeycomb base plate are not simulated and their measured temperatures are used as boundary condition. ISU package is bolted to deck plate along its perimeter. Contact conductance between the package and deck plate is an important parameter in heat transfer mechanism. For perimeter bolt pattern and for honeycomb mounting panel, contact conductance value of $115 W/m^2K$ which corresponds to bare interface have been used in the thermal model **[5]**.

Comparison of the measured and computed bottom chassis sensor temperature, where the constraint of $45^{o}C$ is specified, is shown in Fig.5. Computed results compares with the measured data within $1^{o}C$. Maximum deviation between measured and computed values is $2.8^{o}C$, at all locations in the chassis where the measurements were made. Thus the thermal model is able to capture the various heat flow paths in the package. The internal boundary conditions and the contact conductance used for the ISU package FE model are also appropriate. The validated package FE model is now incorporated in the LV avionic bay model to predict the temperatures in flight, by incorporating external boundary conditions as in flight.

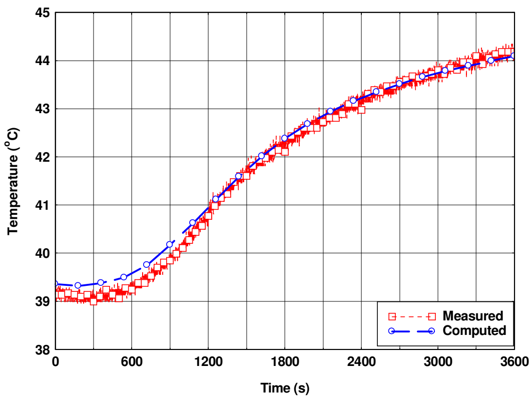

FIGURE 5. MEASURED AND COMPUTED BOTTOM CHASSIS SENSOR TEMPERATURE IN TEST

LV AVIONICS BAY THERMAL MODEL

The LV avionics bay model, developed using commercial software NX9.0 **[4]**, is shown in Fig.6. ISU package FE model, with the test derived boundary conditions is incorporated. The nearby packages and structures were also modeled to account conduction and for their blocking effects on radiative heat transfer. Nearby packages are modeled as cuboids as done for the test case. Element size varies from 50mm for the nearby packages to 100mm for tank structure. Total model comprises of 47,204 elements and 18,321 nodes. Implicit scheme with 1s time step and convergence criteria of 1e-8 is used for analysis.

THERMAL ENVIROMENTS

During pre launch operations, the ISU package is cooled with chilled dry air at around $15^{o}C$. Approximately 100kg/hr air was supplied to the package through silicone flexible hose of 1 inch diameter for earlier missions. For Mars mission, based on the pre launch cooling analysis, 150kg/hr air was recommended. The realized flow rate was 154.8kg/hr at $15.4^{o}C$. Flow rate was derived by measuring the flow velocity at the exit of the cooling hose using a portable vane type anemometer. Temperature was measured using a digital thermometer. In order to distribute this flow rate around the package, the hose exit was sealed using a tape and circular holes were made in the hose at a number of points along the length. The flow rate was evenly distributed around the package to ensure uniform cooling. Conventional correlation including Martins correlation for jet impingement **[6]** was thus not valid to evaluate the convective heat transfer coefficient at the package side walls. An appropriate value of convective heat transfer coefficient was derived by simulating the launch rehearsal conditions using the developed thermal model such that the measured temperatures at lift-off could be reproduced.

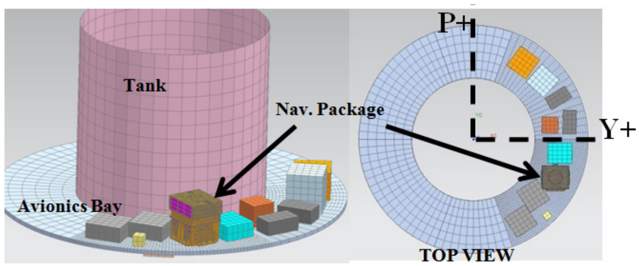

FIGURE 6. LV AVIONICS BAY THERMAL MODEL

The package is subjected to convective cooling during depressurization of air inside HS during ascent phase of the LV for ~100s. The measured ambient temperature variation inside HS for previous flights is shown in Fig.7. The measured ambient temperature variation during ascent phase differs from flight to flight. This is because the aerodynamic heating on the HS differs as the trajectory changes. However the variation in the ambient temperature due to these factors is within $5^{o}C$. Thus for Mars mission, in order to compute the natural convection from the package, ambient temperature at each instant of time is computed as arithmetic average of previous flight measured values as shown in Fig.7. Natural convection from the package to this temperature is assumed during ascent phase. The effect of falling pressure during depressurization of air is not considered in the computation of natural convection. This is an approximation adopted in the thermal model. Beyond 100s as the pressure levels are benign, convection is neglected. The inside surface of HS is covered with thick blanket insulation. Its surface temperature is expected to be same as the ambient temperature and thus radiation from the package to the same ambient temperature is assumed during ascent phase. The suitability of these assumptions will be verified during validation which is described in the next section.

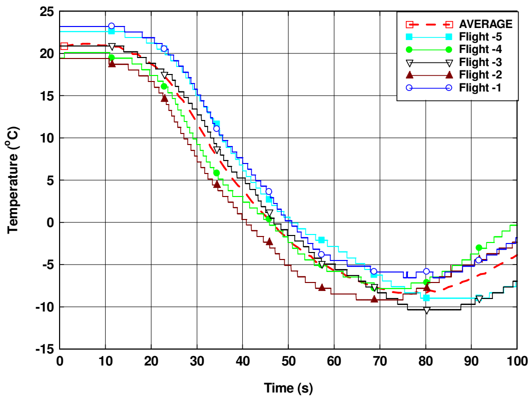

FIGURE 7. MEASURED HS AMBIENT TEMPERATURE AFTER LIFT OFF

The external thermal loads on the package after HS separation include direct solar radiation, albedo, Earth emitted radiation and radiation loss to deep space. These are computed using NX9.0 software based on the trajectory, date and time of launch. Sun and planet vectors computed using in house software is provided as inputs. Variation of total absorbed power on ISU package for Mars mission, which includes solar, albedo and Earth emitted radiation, is compared with previous flights in Fig.8. Spatial variation of external load on the package has been considered. The time integrated heat load values for these missions are now obtained from the area under the curves shown in Fig.8 and are summarized in Tab.2. Even though the total mission duration is more than double that of regular missions, the time integrated heat load received by the package, is comparable with some of the earlier missions and is not the maximum. This is because, as per the trajectory for Mars mission, the LV goes into shadow of Earth after ~1200s. Hence solar and albedo loads are absent after this time thereby making the thermal environments benign.

TABLE 2: ABSORBED TIME INEGRATED EXTERNAL HEAT LOAD ON ISU PACKAGE

Mission	Mission duration (s)	Absorbed heat load (W)
Flight -1	1288.6	87.1
Flight -2	1064.0	35.5
Flight -3	1060.5	47.2
Flight -4	1216.8	31.5
Flight -5	1075.0	132.7
Mars Mission	2653.0	36.3

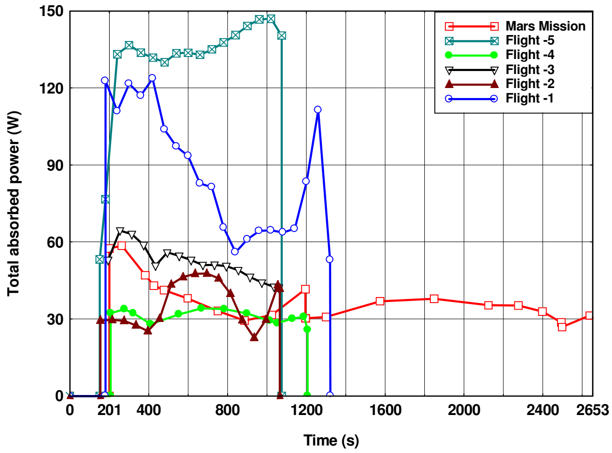

FIGURE 8. COMPUTED ABSORBED EXTERNAL HEAT LOAD ON PACKAGE

RESULTS

LV avionics bay thermal model is used to compute the package temperatures in flight considering all the relevant thermal environments. In order to validate this model, computed temperatures are compared with flight measured values for earlier missions. A typical comparison of computed temperature using the thermal model with a previous flight measurement (Flight-5) is shown in Fig.9. In this particular mission, the package was exposed to solar radiation after HS separation at 179.42s after lift-off till ~1075s and hence the steep rise in temperature is observed after 179.42s. After 1075s, due to stage reorientation, the package does not receive direct solar radiation and hence the change in slope of the temperature curve is observed. It can be seen that the thermal model is able to capture the effect of transient external thermal environments experienced in flight. Computed temperature values are within the desired accuracy of ±2°C.

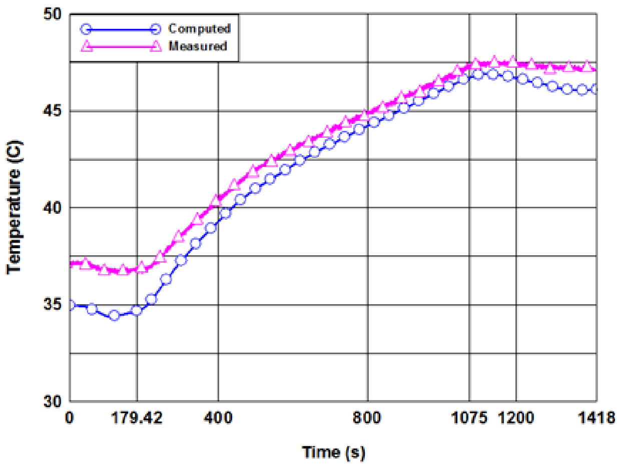

FIGURE 9. COMPUTED AND MEASURED ISU PACKAGE TEMPERATURES FOR PREVIOUS MISSION (FLIGHT -5)

Computed temperatures are lower at lift off due to uncertainties in the mass flow rate and temperature of the chilled air reaching the package during prelaunch conditions on the day of launch. Deviations between the computed and measured temperatures for other flights are also similar, thus validating the thermal model.

Accurate measurements of coolant flow parameters were performed during pre launch operations for Mars mission to compute the package temperatures with a positive bias. Pre flight computed results using the thermal model showed a maximum temperature of 43.2°C till the end of operation using the implemented cooling design, which was within the specified constraint of 45°C. The maximum measured package temperature till the end of operation was 41.4°C. Computed results using the thermal model are compared with the flight measured data in Fig.10. Thus the thermal model was able to predict the temperature levels expected in flight till the end of operation. The prelaunch cooling design which was arrived from pre flight computations was also adequate.

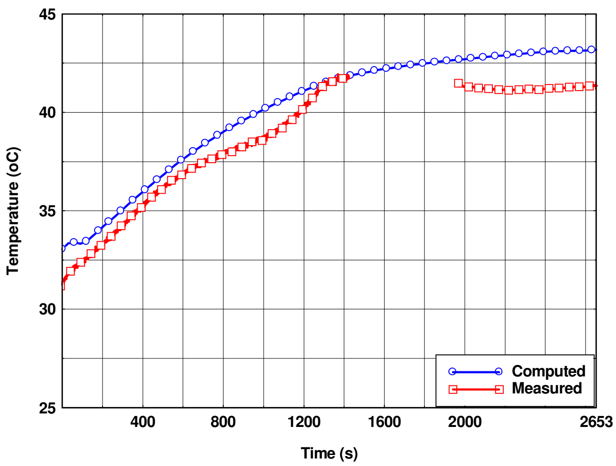

FIGURE 10. PRE FLIGHT COMPUTED AND MEASURED TEMPERATURES IN MARS MISSION

CONCLUSIONS

Thermal management of inertial navigational package was critical for Mars mission due to the longer flight duration as compared to all previous flights. As simulation of all the relevant thermal environments in flight through lab tests was not possible, the pre launch phase cooling design for the package was arrived based on thermal modeling. A detailed thermal model of the package was developed considering the component level details inside. The model parameters were arrived from experiments carried out under controlled environmental conditions. Computed temperatures using the FE model of the package found to be within 2.8°C of the measured values at all locations in the chassis where measurements were made.

The package FE model was then integrated in the LV avionics bay model to predict the temperatures in flight. All the relevant thermal environments experienced in flight is modeled. Results computed using this model for earlier flights showed satisfactory results and were within the desired accuracy of ±2°C. The model was also able to capture the effect of transient external thermal environments experienced in flight. Pre launch cooling design for mars mission was arrived based on the model computed temperatures. The maximum measured temperature of the package for Mars mission was 41.4°C against a computed value of 43.2°C thus proving the adequacy of the developed thermal model.

ACKNOWLEDGMENTS

Authors' acknowledge the support provided by IISU and MISD teams for the technical inputs and help provided for testing of the package.

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