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THERMALLY INDUCED DYNAMICS ON SPACECRAFT

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Abstract: Spacecraft structures are exposed to wide range of temperature variations in orbit. Due to the temperature variations there is a possibility of disturbance in the attitude and the spacecraft may vibrate. Few sensors and payloads in the spacecraft require stable platform, so it is essential to predict the disturbances at critical locations in spacecraft for the on-orbit temperatures. The responses are predicted using MSc. Nastran, commercial Finite Element Method software. A typical spacecraft is modeled to demonstrate the procedure. The spatial and temporal temperature variations are considered. Static and transient dynamic analysis is carried out. Quasi-static results are studied for the overall orbital periods and suitable period where transient analysis is mandatory are understood. Transient analysis is carried out at those periods, the responses, its frequency contents and forces are extracted and presented.

Keywords: Thermally induced vibration, Spacecraft Structures, MSc. Nastran

I. INTRODUCTION

The thermal environment in a typical spacecraft varies from +100oc to -100oc at different orbital positions of the spacecraft. The temperature varies suddenly when the spacecraft enters in to the eclipse and when the spacecraft comes out of the eclipse. The image taken by the spacecraft payloads or the sensors in spacecraft suffers accuracy during this eclipse transition. The reason for the loss in accuracy is due to the stability issues, which is due to the thermally induced dynamics. Thermally induced dynamics generally consists of quasi-static deformation and a dynamic response. The motions are categorized as thermal bending, thermo elastic shock, thermally induced vibrations and thermal flutter. When quasi static deformation is dominant the phenomena is called as thermal bending, the dynamic responses are from the thermally induced vibration and shock phenomenon. Thermal flutter phenomenon is the coupling between the structural vibration and thermal loading which make the spacecraft unstable. This occurs in very large spacecrafts or spacecrafts with lengthier solar booms kind. In this paper the coupling effect is not considered. Boley [1] studied the thermally induced vibration way back in 1972, introduced approximate analysis to predict the vibration in beams and plates. David and Thornton [2] studied the thermally induced vibration of a spin stabilized spacecraft, the study includes, uncoupled analysis where temperature distribution is independent of deflection and coupled analysis where the incident heat flux and temperature are dependant on the booms deflected position. In uncoupled analysis the steady state thermal response was determined and for the thermal response the structural response was determined. The coupled analysis assumes a deflected shape and determines governing equations for an approximate response. The stability of the coupled response was studied using Floquet theory. Sang Yang [3] analytically estimated the thermally induced dynamic response of the boom, which shows a quasi-static displacement and superimposed vibrations. The attitude angle disturbances have a pointing error (steady-state error) as well as vibrations. To attenuate the vibrations and constant attitude disturbances, an active control system with peizoceramic devices were used. Modified Positive position feedback (PPF) was used to control the disturbances. Tarun ghosh [4] had shown a simple method to obtain the rack accelerations in space station due to in-orbit thermal loading. The station will be used to conduct experiments, and there are specially built racks where accelerations no greater than micro-g's are desired. He estimated the temperature distributions, mapped the temperatures to structural Finite element model, estimated displacement and accelerations due to thermal loads, estimated mode shape and frequencies including geometric stiffening due to thermal induced displacement, the frequency, modes shapes and inertial forces were used to estimate the accelerations in rack. Takanori [5] mentioned that the structural disturbances have significant adverse effects on satellite attitude and pointing performance, as shown on the Advanced Land Observing Satellite (ALOS) with a large single-wing solar array paddle. To understand the phenomena and verify ALOS disturbances, a laboratory experiment was performed with scaled model of the paddle. The thermally induced dynamics in a vacuum chamber was simulated. Analytical models were also derived, verified with experimental and flight data. Mitsushige et el., [6] monitored the vibrations and the bending of the appendages caused due to sudden change in the thermal environment by installing onboard camera, the camera images shows significant disturbances on satellite's solar array paddle. The data shows both quasi-static deformation and rapid dynamic vibration in the penumbra. In this paper the deformations and responses on a typical spacecraft is predicted using Finite Element Methods. MSc. Nastran commercial finite element software is used for the analysis. For pre and post processing MSc. Patran is used.

II. TYPICAL SPACECRAFT

A typical spacecraft is used for this study. The spacecraft structure consists of three subsystems, primary load carrying member, decks for mounting equipments/ payloads and appendages. The appendages shall be solar panels to generate power in orbit, antennas to send and receive signals. The appendage sizes will be larger and the thickness of appendage is generally less to reduce the weight of the spacecraft. The appendages will be deployed in orbit and the deployed frequencies will be very low compared to the elastic modes of the spacecraft.

III. STRUCTURAL FINITE ELEMENT MODEL

The spacecraft structural finite element model is shown in Fig. 1.

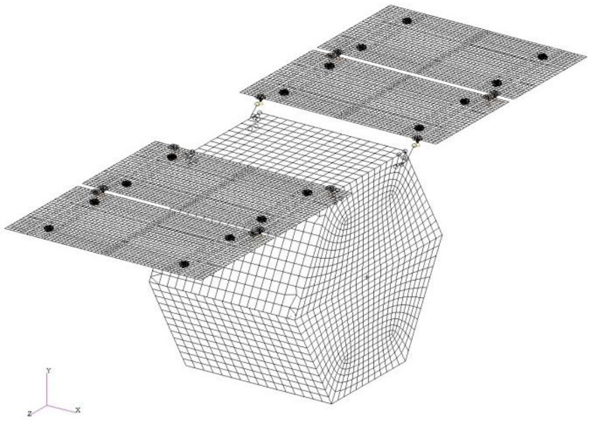

Fig.1 Structural Finite element model of a typical spacecraft

The Finite element model is generated using MSc. Patran software. One dimensional element and two dimensional elements are used as per requirements. The preprocessor checks like aspect ratio checks, warp angle checks, taper checks for all the elements are done. The models are verified for the boundaries and normal. Rigid elements are used to simulate the connections. The appendage in the considered typical spacecraft is solar panels. The solar panels are modeled, the inter solar panel hinges and solar panel to deck connections are also simulated. Two solar panels with hinges make solar arrays. The assembled finite element model is checked for its modeling correctness. The model possesses only six rigid body modes. The rigid body modes strain energies are zero. The stiffness matrix is checked for correctness during each stage of formulation in Nastran. The generated model is verified by applying unit enforced displacements and rotation. Unit gravity load is applied in the model and the forces in the support are verified with theoretical prediction. For verification the thermal coefficient of expansion of all the materials in the model is changed to uniform value, a uniform temperature load is applied in all elements/nodes, the initial temperature is also set to uniform value, linear static analysis is carried out with boundary condition as explained in future section. The model uniformly expands in all three directions. The model passed all the required checks and the same can be used for structural analysis.

IV. MODES OF SPACECRAFT

The spacecraft in orbit will have deployed appendages and there will not be any boundary conditions. The modes of the typical spacecraft are estimated using MSc. Nastran. Initial ten modes are tabulated in Table 1. First six modes are rigid body modes. Seventh to tenth modes are solar array modes.

Table 1 Modes of typical spacecraft

Mode No.	Frequency (Hz)	Description
1 to 6	0	Rigid body modes
7	6.5	Solar arrays cantilever mode with respect to deck – moving in phase and out of phase
8	7.3	
9	10.1	Bending modes of array – moving in phase and out of phase.
10	10.1	

The seventh and eighth modes are solar arrays cantilever mode with respect to deck. In seventh mode both side array move in phase and in eighth mode both moves out of phase. Similar behaviour is seen in 9th and 10th mode also.

V. TEMPERATURE PREDICTION AND MAPPING

Temperatures are predicted for the typical spacecraft using Finite Difference Method. Separate thermal model was generated and the transient/steady state thermal analysis was carried out. Here it’s assumed that, for the typical spacecraft considered, the predicted temperatures will not change even if the quasi-static displacements are considered. The thermal model generation and analysis details are out of scope of this paper. The thermal analysis results needs to be mapped to the structural finite element model. The sub systems are sub divided as zones in both the thermal and structural model. Zone to zone mapping is carried out for accurate mapping. The mapping procedures mentioned in [7] are followed. The mapping procedure is carried out using IDEAS TMG module. The mapped temperatures in structural model compares well with the predicted thermal model results. The transient thermal analysis in thermal model is carried out for one full orbit. The temperatures are predicted for every 5o of orbital position. The mapped temperatures are available for every 80 seconds. The observed orbital temperature between these periods was found to have linear variation, so the temperatures can be interpolated for every 10 seconds from the 80 seconds mapped temperature data. A suitable interpolation program to read the mapped temperature data and interpolate the data for every 10 seconds for the corresponding elements/nodes was written in C language. The program output will be in Nastran bulk data format.

VI. ANALYSIS AND DISCUSSIONS

As explained earlier the thermally induced disturbances will have quasi-static displacements and transient vibration. Those two disturbances are predicted by performing linear static analysis and transient dynamic analysis using direct approach. Other way of predicting the disturbances is performing transient dynamic analysis for overall orbital period, the time and space consumption for such way of prediction is not practically possible. So the first way of prediction is followed. The Finite Element Model of the typical spacecraft is used for the analysis, the load for the linear static analysis is the mapped temperature, and the initial temperature is room temperature. The spacecraft will be on free-free condition for the on orbit analysis, but a boundary condition should be provided to avoid the singularity. The boundary condition should remove the singularity and it should allow the spacecraft to freely expand for temperature load and there should not be any reaction forces. This simulated boundary condition is called kinematic boundary condition [7]. The base of the primary load carrying member is used for simulating this boundary condition. Four point support option is chosen, a parametric study is performed and a suitable boundary condition is chosen as shown in Fig 2 satisfying the earlier mentioned conditions.

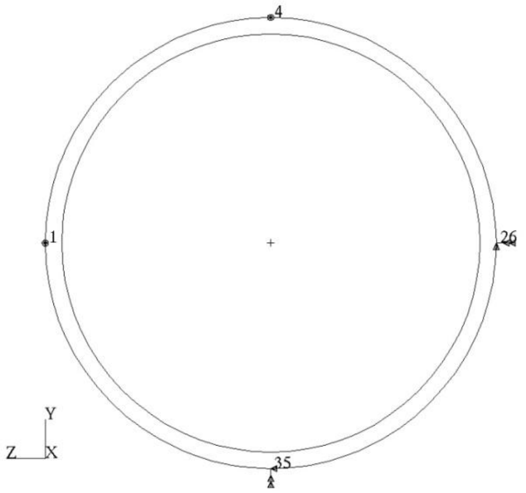

Fig.2 Kinematic Boundary condition

Linear static analysis is performed at every instant of mapped temperature. Few restitution points are chosen in solar panel, decks and primary load carrying member. The restitution points are shown in Fig. 3 and mentioned in Table 2.

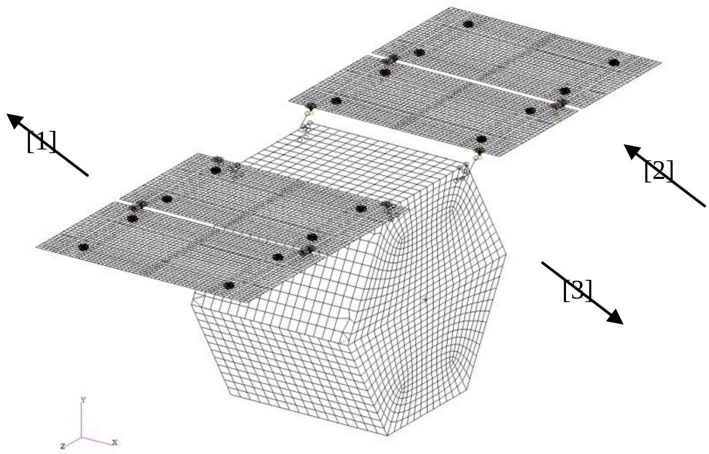

Fig. 3 Restitution points

The out of plane displacements at the restitution points are extracted and the displacement pattern over the period of two orbits are shown in Fig. 4 to Fig. 7. One orbital period is 5880 seconds.

Table 2 Restitution points

No.	Description
1	Solar panel corner
2	Solar panel center
3	In Deck
4	In primary load carrying member

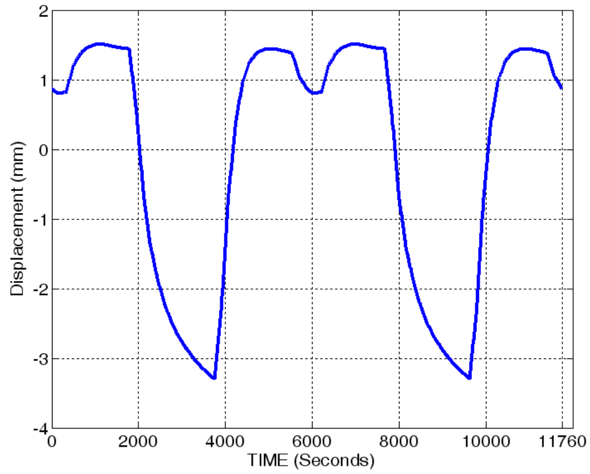

Fig. 4 Out of plane displacement in solar panel corner

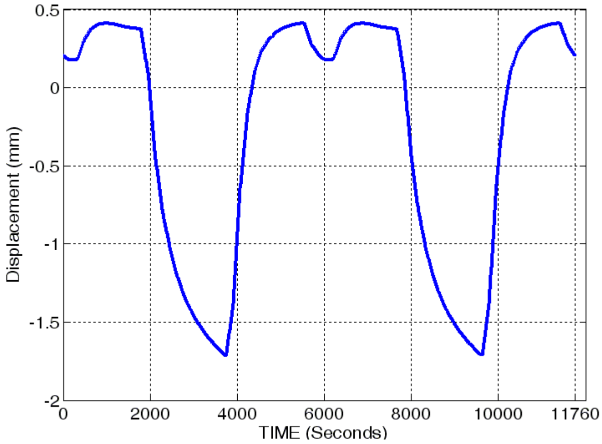

Fig. 5 Out of plane displacement in Solar panel centre

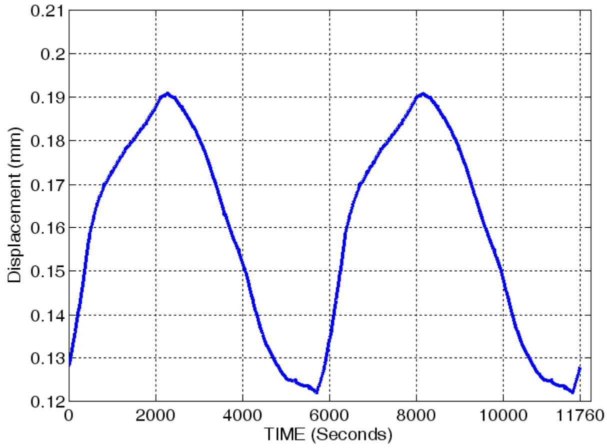

Fig. 6 Out of plane displacement in deck

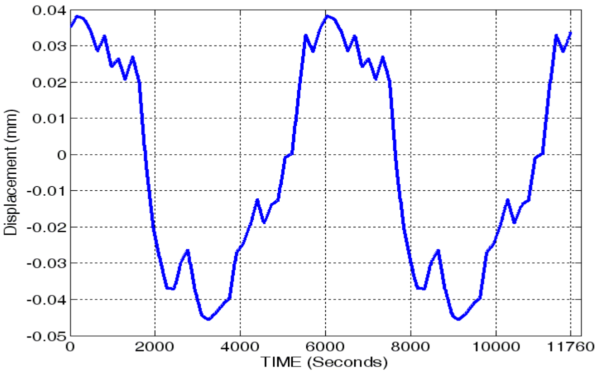

Fig. 7 Out of plane displacement in primary load carrying member

The quasi-static displacement in solar panel corners varies from 1.5 mm to -3.2 mm over a period of orbit. The displacement in solar panel centre varies from 0.4 mm to -1.7 mm. The spacecraft enters into eclipse period at about 1790 seconds, both the above mentioned restitution points have sudden change in displacement during this eclipse entry. The spacecraft comes out of eclipse period at about 3750 seconds; during this transition also there is a sudden change in displacements in solar panels. The out of plane displacement in deck location shows 60μm variation. The quasi-static displacement does not change by big magnitude during eclipse transition. The out of plane displacement in primary structure is 100 times lesser than the displacements in solar panel edge. The displacement variation is in the order of 80 μm and there is small change in displacement during eclipse transition. From the above results it's understood that the thermally induced vibration will occur mainly during the orbital eclipse transition. So a transient dynamic analysis is carried out in MSc. Nastran. There are two approaches in transient dynamic analysis, one is direct approach and other is modal based approach. The direct approach will use the full matrices while solving the equilibrium equation, in modal approach the matrices will be reduced with few modes. Modal approach will be less time consuming. In present analysis the temperature loads are defined and it has both static and temporal variation, full matrices are required for loading function calculations, so the analysis can be performed only using direct approach though it takes more time. Transient analysis is carried out at eclipse entry, the initial temperature is the temperature mapped in the elements/nodes before entering the eclipse and the mapped temperature after entering eclipse are mentioned as loads. In this load definition the spatial variation of temperature on the spacecraft and temporal variation of temperature is also mentioned. The analysis is performed with a time interval of 1E-3 seconds. From experiments it is understood that during such micro vibration the structural damping will be about 0.5% and the value is mentioned as damping. The analysis is performed for about 2500 steps or till it reaches the equilibrium condition. The out of plane displacement responses are extracted at the points as mentioned in Table 2. The displacement results are added to the quasi static results and it's shown in Fig. 8 to Fig. 11.

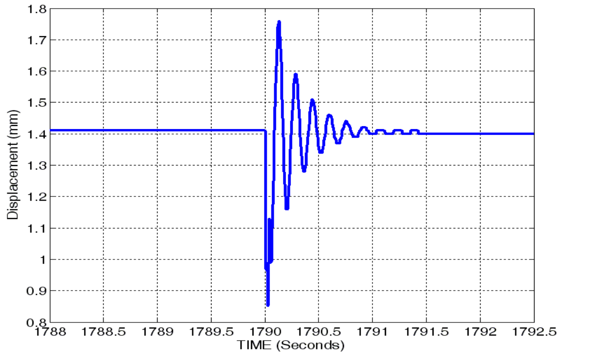

Fig. 8 Response at solar panel corner during orbital eclipse entry

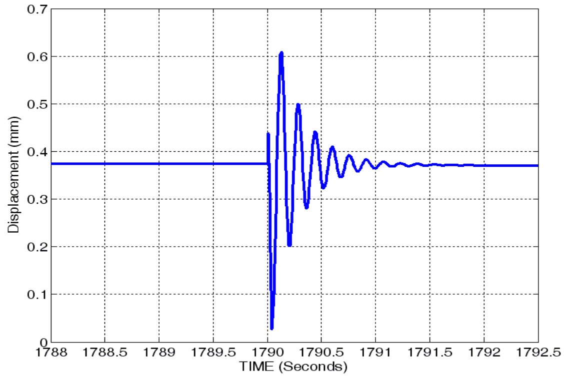

Fig. 9 Response at solar panel centre during orbital eclipse entry

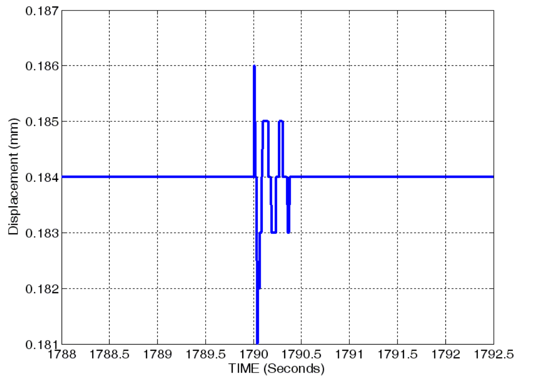

Fig. 10 Response at deck during orbital eclipse entry

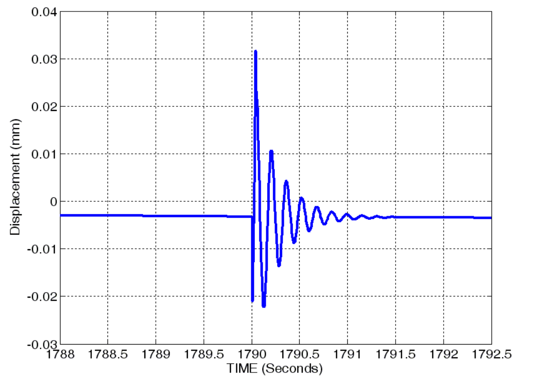

Fig. 11 Response at primary load carrying member during orbital eclipse entry

The response plot at solar panel corner shows about 1mm relative displacement and the response attenuates in 1.5 seconds after eclipse entry. At solar panel centre the relative displacement is about 0.6 mm. At deck it is about 5 µm and in primary load carrying member its 50 µm. The transient analysis is also performed during orbital eclipse exit, where the initial temperature is the mapped temperature at elements/nodes of spacecraft before exit and the consecutive temperature sets are mentioned as loads. Fig. 12 to Fig. 15 shows the responses during orbital eclipse exit.

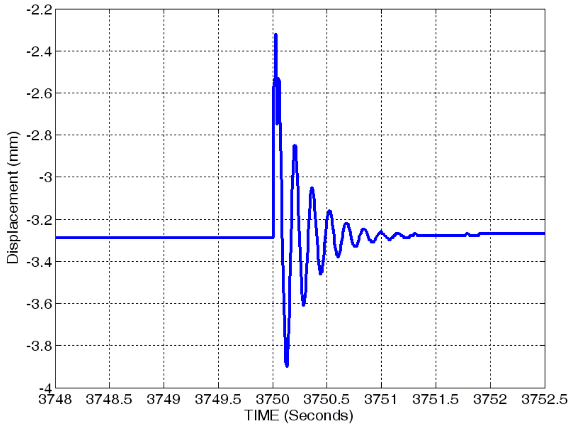

Fig. 12 Response at solar panel corner during orbital eclipse exit

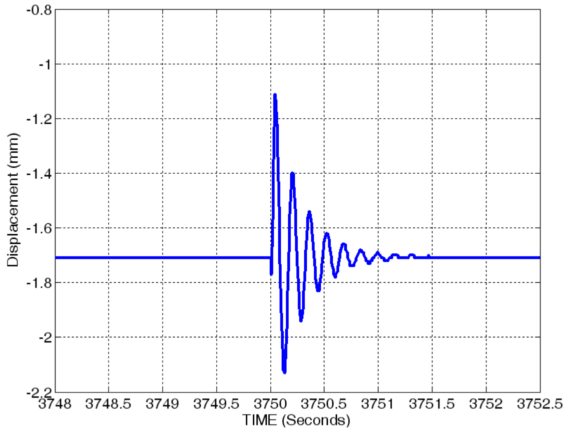

Fig. 13 Response at solar panel center during orbital eclipse exit

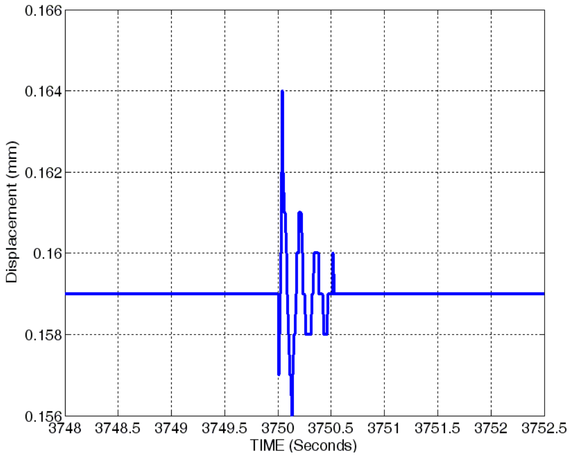

Fig. 14 Response at deck during orbital eclipse exit

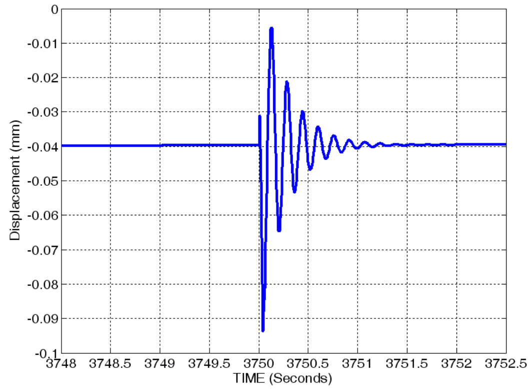

Fig. 15 Response at primary load carrying member during orbital eclipse exit

The response plot at solar panel corner shows about 2 mm relative displacement and the response attenuates in 1.5 seconds after eclipse entry. At solar panel centre the relative displacement is about 1 mm. At deck it is about 8 µm and in primary load carrying member its 90 µm. All the responses settle to the quasi static displacement after 1.5 seconds of this transition. The temperature change in orbit will be continuous, but the prediction is done at every 10 seconds as mentioned in section 5. To understand whether the discrete temperature loads create the vibration, which will not occur in orbit, a transient analysis is performed for other mapped temperature sets with previous temperature sets as initial. The responses are very small compared to the responses seen during orbital transition and the values can be neglected. The transient responses at the restitution points were summarized, the frequency contents in the time domain response should be known for further studies, so FFT is taken to the responses. Since response during orbital eclipse exit is slightly higher than eclipse entry the FFT plot during eclipse exit is shown in Fig. 16 to 19.

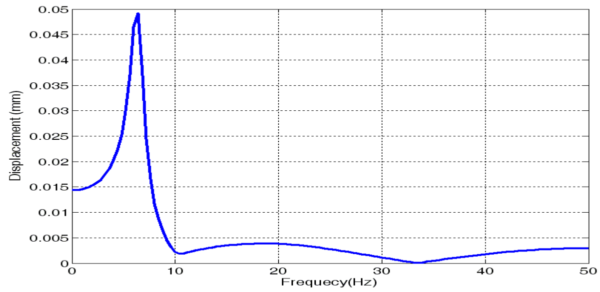

Fig. 16 Frequency content of the transient response – solar panel corner

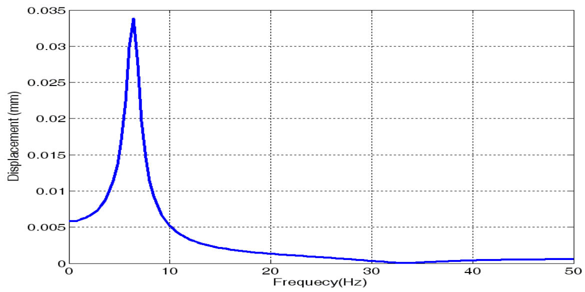

Fig. 17 Frequency content of the transient response – solar panel center

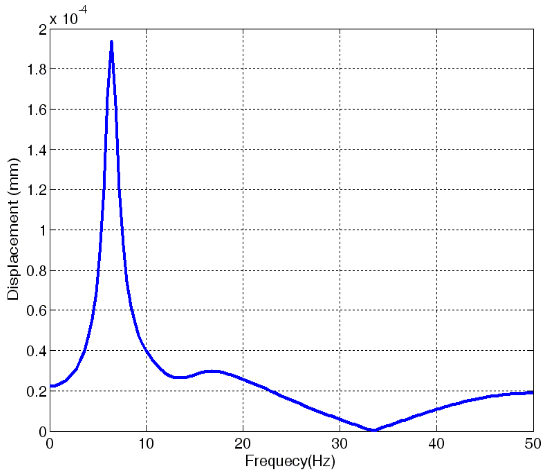

Fig. 18 Frequency content of the transient response – in deck

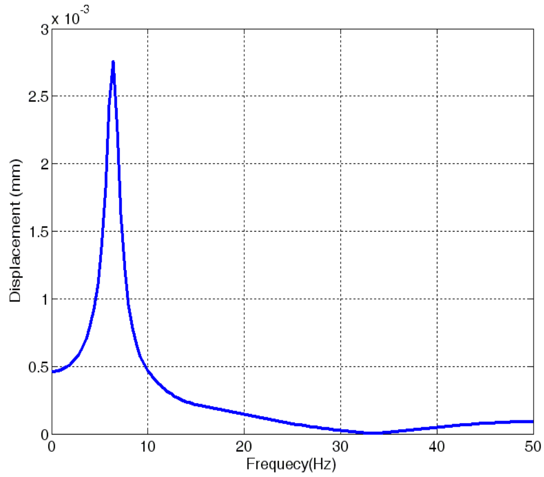

Fig. 19 Frequency content of the transient response – primary load carrying member

From the frequency plots it is clearly visible that the solar array modes are getting excited due to this sudden temperature differences during orbital eclipse transition periods. The forces in the solar array deck to deck interfaces are extracted during orbital eclipse exit. The solar panel to deck interface locations are shown in Fig. 20. All the three component forces are shown in Fig. 21 to Fig. 23.

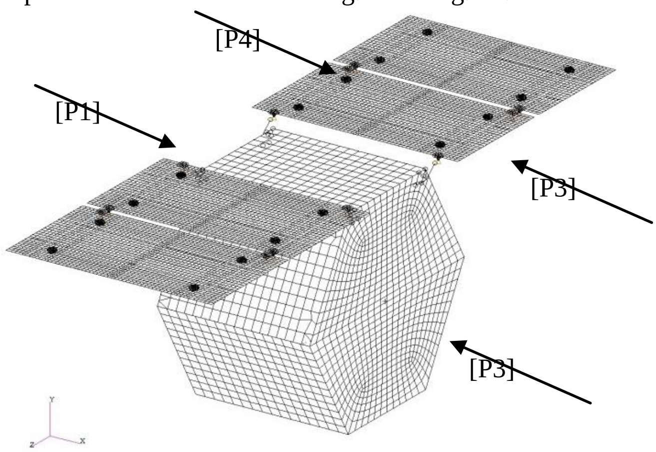

Fig. 20 Solar panel to deck interfaces

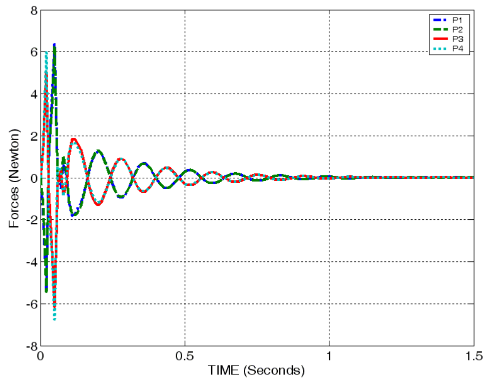

Fig. 21 Forces (Fx) in solar panel to deck interface during orbital eclipse exit.

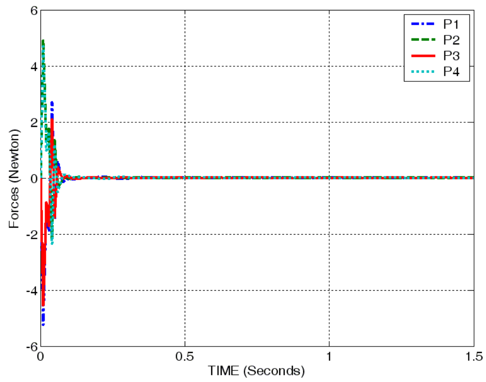

Fig. 22 Forces (Fy) in solar panel to deck interface during orbital eclipse exit.

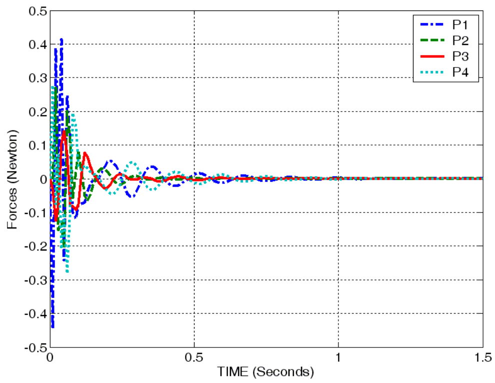

Fig. 23 Forces (F_z) in solar panel to deck interface during orbital eclipse exit.

The forces in the interfaces are about 6N and the considerable forces exist for one second after the eclipse transition. These forcing functions can be used as input to the spacecraft main body for further studies like vibration control.

VII. CONCLUSION

The procedure to estimate the thermally induced responses on the typical spacecraft is demonstrated. The appendages show significant displacements than the decks and primary load carrying members. Most of the payloads or sensors will be placed on the decks or primary load carrying members, the disturbances are sufficient to make sensitive equipments to loose its accuracy during orbital eclipse transition. If the appendage dimensions are longer, the appendage frequencies will be much lesser than this spacecraft and the time taken to attenuate will also be higher. The same procedure can be extended to any other spacecrafts and a suitable control methods shall be demonstrated when the disturbance and attenuation times are higher.

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