ISSN 2229-5518

Response Reduction at Payload Using Isolation System in a Typical Launch Vehicle

Lekshmi Priya S[1] ,Dr.T Sundararajan[2], Dr.K Balakrishna Rao[3]

Abstract- Launch vehicles present an efficiently viable method for placing satellites into orbit. A typical launch vehicle consist of propulsion modules either solid and/or liquid stages connected by interstages and payload module.Dynamic analysis is used to find out the response of the structure, mainly at the satellite interface due to various excitations encountered during flight. The basic objective of the work is to reduce the dynamic loads on the satellite by vibration isolation methods. In this work analysis is carried out by incorporating an isolation system at various locations and the results are compared to show the effectiveness of isolation system.

Index terms- launch vehicle, mode shape, natural frequency, frequency response, acceleration, vibration isolation.

I.INTRODUCTION

Large solid motors are found to produce vibrations due to thrust oscillations during their operation. If the frequency of these oscillations matches with the lateral/longitudinal modes of the structure, it would result in large responses at satellites. Generally isolation systems are designed as a solution to such problems. The present study is carried out to evaluate the requirement of isolation system and the its effectiveness.The general approach for dynamic solutions involving large systems is to develop a mathematical model describing the system's mass and stiffness to calculate modes of vibration. In aerospace industry due to complex and complicated systems a finite element model is created to estimate the response due to different excitations. This model should take into account the characteristics of the system design, the nature of the dynamic loading (type and frequency) and any interacting media (fluids, adjacent structures). Frequency response analysis is carried out to estimate response due to the applied forces. The first step in dynamic analyses is the free vibration which determines the structure's structural dynamic characteristics viz. natural frequencies and mode shapes. A launch vehicle structure can be idealized by beam-rod model, quarter-shell model,3-dimensional model or a combination of the above depending on the frequency requirements [4].

Due to large L/D ratio, a beam model is adequate for capturing the predominant responses which will be dominated by first few global modes. Solid motors experience sustained self-excited oscillations at the frequency of the first longitudinal acoustic mode of the chamber [3]. The frequency of this half-wave mode is determined by the length of the combustion chamber and the acoustic speed in the hot gas. Large segmented solid rocket motors will exhibit pressure oscillations with corresponding thrust oscillations. These oscillations will interact with the structural modes of the launch vehicle. In this paper different isolation schemes for reducing the vibration at the satellite due to solid motor pressure oscillation is attempted.

Finite Element package MSC/NASTRAN [5] was used for modeling and analysis of the structure. As part of the evaluation procedure, the launch vehicle is initially modeled without any isolation system and analyzed. Then required modifications are made to the system design by introducing isolators at various locations, and the responses at the satellite base are compared.

2.OBJECTIVE

A launch vehicle is subjected to various excitations during its mission. Excessive dynamic loads during its ascent can be detrimental to satellite .One of the options is to reduce the dynamic loads transmitted to the base structure at which payload is attached. In this paper analysis is carried out by incorporating an isolation system either between the strapon and the

[1]M Tech student,Civil Dept.,Manipal Institute Of Technoogy,Manipal,INDIA
[2] Head,SMSD,Vikram Sarabhai Space Centre,Trivandrum,INDIA.
[3]Proffessor,Civil Dept.,Manipal Institute Of Technology,Manipal,INDIA.
E-mail:[1]lekshmipriya085@gmail.com

core vehicle or at the satellite interface, thus protecting the satellite from the thrust oscillation of the strapons .Thus the main objective of the work is to find the response reduction at the payload interface after the introduction of isolators.

3. MODELLING

Launch vehicle is modeled using 3D beam element. The propellant tanks, solid propellant motor and inter-stages connecting them are modeled using equivalent area (A), bending moment of inertia (Ixx and Izz) and torsional moment of inertia (J) [5]. The structural mass is smeared over the appropriate sub-systems. The solid propellant mass is smeared along the motor case mass. For liquid propellant, slosh modeling is adopted by simulating the rigid mass with inertia and slosh masses at appropriate location from the tank bottom [2]. Longitudinal dynamics of the liquid propellants are represented using equivalent resonators. The spacecraft is simulated by its mass lumped at its centre of gravity.

Figure-[1] shows finite element model of a typical launch vehicle. The core to strapons are connected using equivalent beam elements. The connections are made in such a way that the thrust transfer is at the fore end attachment. Finite element models are generated incorporating stiffness corresponding to the isolator at core-strapon connection fore end as well as between spacecraft and payload adapter. Scalar spring element is used to model isolators. General purpose finite element software MSC/NASTRAN is used for dynamic analysis.

4. FREQUENCY RESPONSE ANALYSIS

A frequency-domain model is a set of input-output transfer functions with respect to the independent variable frequency (ω). Frequency domain methods are most efficient for random vibrations and periodic loadings. As the amplitude and the frequency of the excitation are varied, the response also changes. In this manner, the response of the system over a range of excitation frequencies is determined. When subjected to dynamic forces, a structure's total response is the sum of the responses of its modes of vibration [6].

The physical displacements are expressed in terms of modal coordinates $\eta(t)$ as

$$q_1 = \phi\eta(t) \qquad (1)$$

The equation of motion under constraints given in equation [2]

$$[\overline{M}]\{\ddot{q}_1\} + [\overline{C}]\{\dot{q}_1\} + [\overline{K}]\{\dot{q}_1\} = \{\overline{Q}\} \qquad (2)$$

Eliminating $\mathbf{q_1}$ from Eq.(2) and pre-multiplying

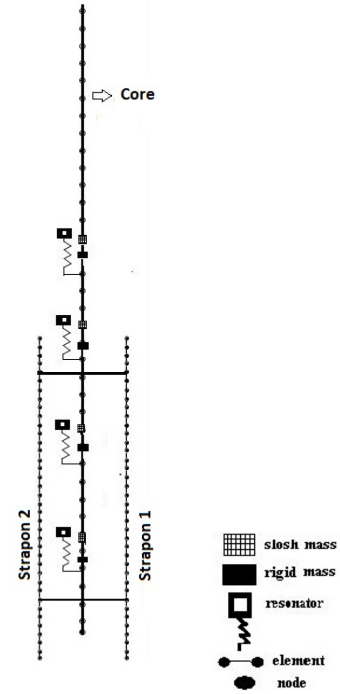


Figure [1]: Finite Element model of a typical launch vehicle

by the transpose of the modal matrix Φ, decoupled equations will be of the form:

$$\overline{M}\ddot{\eta} + \overline{C}\dot{\eta} + \overline{K}\eta = \{\overline{Q}\} \qquad (3)$$

Where,

$$\overline{M} = \phi^T M\phi, \overline{C} = \phi^T C\phi, \overline{K} = \phi^T K\phi, \quad \overline{Q} = \phi^T Q$$

Both $\overline{M}$ and $\overline{K}$ are diagonal matrices. If columns of Φ are mass normalized, then

$$\overline{M} = I, \overline{K} = \wedge I$$

where $\wedge = \begin{bmatrix} \omega_1^2 & \cdots & 0 \\ \vdots & \ddots & \vdots \\ 0 & \cdots & \omega_n^2 \end{bmatrix}$

Equation (2) reduces to

$$\ddot{\eta} + \overline{C}\dot{\eta} + \wedge\eta = Q$$

The steady state solution is obtained by assuming that the response is harmonic with frequency ω. The derivation is same as that of single-degree of freedom system, where $e^{i\omega t}$ is factored to obtain the following relations.

$$[\wedge - \omega^2 I + i\omega\overline{C}]\eta = Q$$

$$\eta = [\wedge - \omega^2 I + i\omega\overline{C}]^{-1}Q$$

$$q_1 = \phi[\wedge - \omega^2 I + i\omega\overline{C}]^{-1}$$

u(iω) = H(iω) F(iω)

Where, H(iω) = $\phi[\wedge - \omega^2 I + i\omega\overline{C}]^{-1}\phi^T$

The transfer function is

$$H_{jk}(i\omega) = \sum_{r=1}^{n} \frac{\varphi_{jr}\varphi_{kr}}{\omega_r^2 - \omega^2 + i2\gamma_r\omega_r\omega}$$

In rationalized form, $H_{jk}(i\omega)$

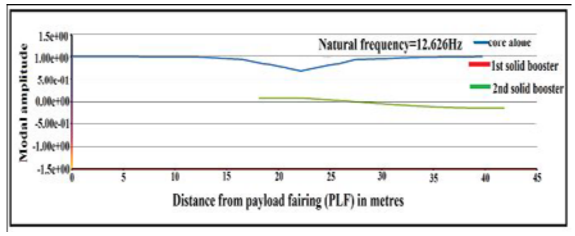
$$\sum_{r=1}^{n} \frac{\varphi_{jr}\varphi_{kr}(\omega_r^2 - \omega^2)}{(\omega_r^2 - \omega^2)^2 + (2\gamma_r\omega_r\omega)^2} - i\sum_{r=1}^{n} \frac{\varphi_{jr}\varphi_{kr}(2\gamma_r\omega_r\omega)}{(\omega_r^2 - \omega^2)^2 + (2\gamma_r\omega_r\omega^2)}$$

5.RESULTS AND DISCUSSIONS

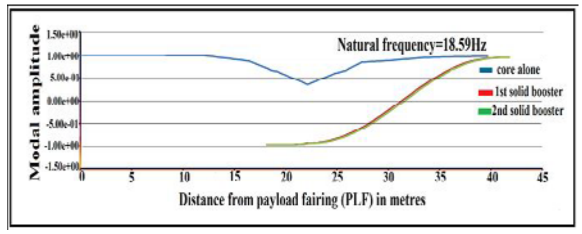
Finite Element package was used for modeling of the structure .Once the launch vehicle is modeled, normal modal analysis was carried which gave frequencies and associated mode shapes of the structure. Table [1] gives the natural frequencies of the model without isolators at core to strapon joint as well as at space craft interface. Since the aim is reduction of the axial response, the first few longitudinal modes are given in Table-[1]. Frequencies 12.62 Hz and 18.59 Hz are the dominant modes up to 30Hz.

TABLE [1]
AXIAL FREQUENCIES

Natural Frequency(Hz)	**Remarks of the mode shapes**
12.63	Strapon balancing core
18.60	Strapon dominated by 1st axial mode
35.67	Strapon dominated by 2nd axial mode
52.85	Strapon dominated by 3rd axial mode
68.97	Core axial and strapon 4th axial mode



Fig[2] : Longitudinal mode shape for natural frequency of 12.62Hz



Fig[3] : Longitudinal mode shape for natural frequency of 18.59Hz

Figure [2-3] shows the longitudinal mode shape of the core and solid boosters corresponding to 12.63Hz and 18.59 Hz without isolators, respectively. The modes are solid strapon dominated axial modes. Usually the large segmented solid rocket motors the pressure oscillation frequency will be less than 30Hz. Frequency response analysis was carried out to find out the response at the satellite base due to the thrust oscillations of the two solid boosters. Force is applied at the head end of solid strapons.Three different cases where considered to find out the response reduction.

- Without isolators
- With isolator at satellite interface
- With isolator at core strapon interface

The excitation force of 1N (unit force) was applied from 0.1Hz to 100Hz.Frequency response analysis was done to find out the response (acceleration) due to unit force at solid strapon head end. The response at the location corresponding to the satellite interface was the main area of interest. The variations in the acceleration of the three different cases, at the above referred location are shown in Figures [4-6].

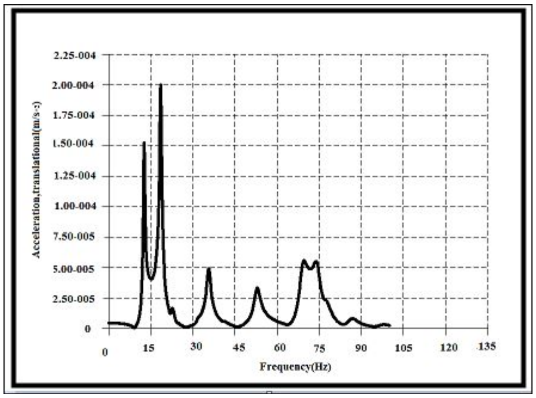


Fig [4]: Response curve of model without isolators

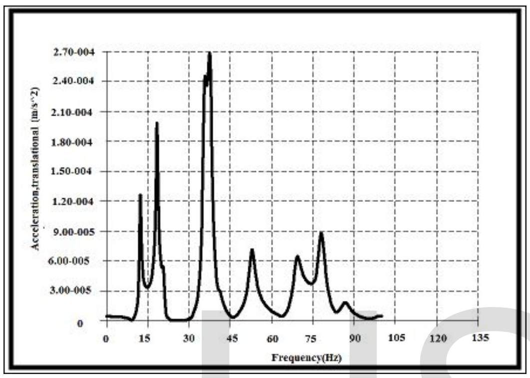


Fig [5]: Response curve of model with isolators at payload interface

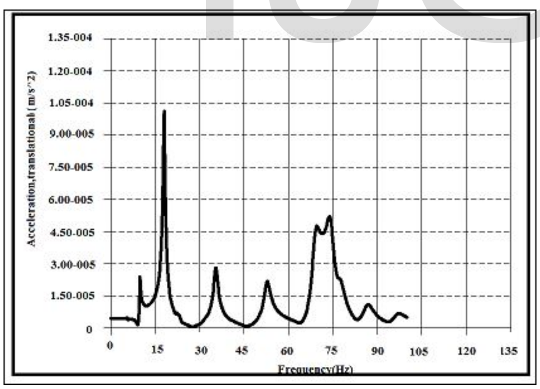


Fig [6]: Response curve of model with isolators at core strapon interface

The Figures [4-6] clearly shows the effect of isolation on the acceleration at the satellite interface. It is evident that the response is least possible in case when the isolators were introduced in between core strapon than from the other two cases. At frequency of 12.63Hz the axial response got reduced by 19.33%, when isolators at satellite interface. While it got reduced by 86%, when the isolators where at the core strapon interface. A maximum axial acceleration of 2.00e-04m/s² occurred at a frequency 18.60Hz due to the application of unit force (1N)excitation at thrust location without isolators .It got reduced by almost 50% when the isolators where installed at core strapon interface. Tables [2-3] shows the percentage reduction in axial acceleration for the dominant modes up to 100Hz after the introduction of isolators at satellite interface as well at core strapon interface respectively.

TABLE[2]
AXIAL ACCELERATION FOR UNIT FORCE (1N) AT THRUST LOCATION WITH ISOLATOR AT SATELLITE INTERFACE

Without Isolators		Isolator at Satellite Interface		% Reduction in Axial Acceleration
Axial frequency (Hz)	Translational Acceleration (m/s^2)	Axial frequency (Hz)	Translational Acceleration (m/s^2)	
12.63	1.50e-04	12.31	1.21e-04	19.3%
18.60	2.00e-04	18.40	1.95e-04	2.5%
35.67	5.00e-05	35.50	2.65e-04	Response increased
52.85	3.50e-05	52.87	7.00e-05	Response increased
68.97	5.50e-05	69.06	6.20e-05	Response increased

TABLE [3]
AXIAL ACCELERATION FOR UNIT FORCE (1N) AT THRUST LOCATION WITH ISOLATOR AT CORE STRAPON INTERFACE

Without Isolators		Isolator at Core Strapon Interface		% Reduction in Axial Acceleration
Axial frequency (Hz)	Translational Acceleration (m/s^2)	Axial frequency (Hz)	Translational Acceleration (m/s^2)	
12.63	1.50e-04	9.90	2.10e-05	86%
18.60	2.00e-04	18.08	1.00e-04	50%
35.67	5.00e-05	35.51	2.80e-05	44%
52.85	3.50e-05	52.75	2.20e-05	37%
68.97	5.50e-05	71.74	5.00e-05	9%

Significant reduction in the responses at the satellite base is observed in case when the isolators where installed at core strapon interface.

6.CONCLUSION

There is a need to reduce dynamic loads on launch vehicle so that spacecraft and their instruments can be designed with more concentration on orbital performance rather than launch survival. A softer ride to orbit will allow more sensitive equipment to be included in missions, reduce risk of equipment or component breakdown, and possibly allow the mass of the spacecraft bus to be reduced. From the different cases analyzed, it is clear that the vibration isolation systems between core and strapon performed very well to reduce structure vibration levels transmitted to the satellite. The isolation system was designed specifically to reduce the effects of solid motor resonant burn in the 10 Hz to 30 Hz frequency range, which it did very well, when installed at core strapon interface.

REFERENCES

[1] G. B. Paddock,''*Dynamic Stability of Space Vehicles Volume I – Lateral Vibration Modes*", NASA CR-935, November 1967.

[2] R. Gieseke, R. Schuett and D. Lukens, "*Dynamic Stability of Space Vehicles Volume III – Torsional Vibration* Modes", NASA CR-937, November 1967.

[3] Dynamic Environmental Criteria, *NASA Technical Handbook*, NASA-HDBK-7005, March 2001

[4] P. J. Grimes, L. D. McTigue, G. F. Riley,and D. I. Tildelt, " *Advancements In Structural Dynamic Technology Resulting From Saturn V Programs"*,NASA CR-1540,June 1970

[5] MSC/NASTRAN Quick Reference Guide, The MacNeal Schwendlar Corporation, USA.

[6]S.R.Damodaraswamy and S.Kavitha,"*Structural Dynamics And Aseismic Design*",PHI Learning Private Limited ,DelhI-2013