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Recovered Transient Load Analysis for Payload Structural Systems

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The integration of the launch vehicle and the payload model and the subsequent computations to obtain payload load are expensive and time-consuming. These difficulties preclude the use of exact structural loads analysis in the early design stages where frequent design changes are occurring. Therefore, approximate methods have been employed during the early design/analysis process. A recovered transient analysis technique is proposed in the present study wherein the results of a previous launch vehicle/payload system can be used to obtain the necessary information of a new payload structure to be launched by an identical launch vehicle. The payload is supported in a statically determinate manner in the present study. The advantage of the proposed method is that the complete design/analysis process can be performed within the payload organization and achieve the same accuracy as that of the full-scale, multiorganization loads analysis. Also, the flight measured interface accelerations can be used as the forcing functions for more realistic representations of the dynamic environments. The method is applied to a real complex payload structure.

Introduction

FOR structures designed to withstand dynamic environments, the design loads are typically obtained from analyses of representative mathematical models under the relevant dynamic loads. Since the mathematical model is based on the design of the structures, this design/analysis approach is an iterative process. One cycle of the iteration consists of an update of the responses or load estimation, the associated structural design activity, the revision of the mathematical model, and a repeat of the response analysis. The entire process is expensive and time-consuming since the mathematical models are large and the delay between the availability of the model and the final dynamic response calculation is substantial.

The structural system for a space mission consists of a payload integrated into a launch vehicle. The launch vehicle/payload composite model is then subjected to appropriate external forcing functions for the response and load calculation. These external forcing functions are derived from the relevant dynamic environments representing the launch vehicle thrusts, staging events, aerodynamic loads, and others. The design/analysis process involves the integration of the payload model and launch vehicle model and the subsequent transient analysis. Considerable time and cost are required in the integration of the composite model and the response analysis since typically the payloads and the launch vehicle are developed and designed by separate organizations with their respective mathematical models which involve different computer codes, coordinate systems, units, and normalization procedures. In a previous experience, the Viking project, as many as 10 organizations were involved in generating the composite model and its integration and one design/analysis cycle required up to 6 months.[1]

To reduce the cost and schedule, several approximation techniques have been developed in the past. One method utilizes the fact that payload design changes during the design/analysis iteration are "small" such that a cost-effective perturbation procedure can be applied to update the response analysis due to design changes.[2] This method was further extended to estimate the modifications of launch vehicle/payload interface responses due to payload design changes.[3] The modified interface responses can then be applied to the base of the payload to calculate the payload responses and loads. This method is not only cost-effective but also can be implemented by the payload organization without interfacing with the launch vehicle organization. However, this method is developed based on the assumption that the payload is always much "smaller" than the launch vehicle. For shuttle launched payloads, the effects of payload dynamics can no longer be considered as "small" for future dynamic events such as the abort landing of a fully loaded shuttle orbiter. In these events, the interaction between the shuttle orbiter and its payload is critical to both the orbiter and payload loads.[4] Therefore, it seems that the launch vehicle/payload composite model integration and its transient analysis cannot be avoided. In other words, the payload organization must rely on the launch vehicle organization for the payload design loads. Since the existence of significant coupling between the shuttle orbiter and payloads, the launch vehicle organization usually will perform a dynamic analysis of the shuttle/payload system with a rigid payload model for the purpose of verifying the launch vehicle integrity. A method has been developed by which the launch vehicle/payload interface responses are modified such that the effect of payload elasticities are taken into consideration.[5] Then the payload responses and loads can be obtained by applying the modified interface responses to the base of the payload. The advantage is that the entire procedure can be implemented within the payload organization such that timely design/analysis iteration can be performed. However, the disadvantage is that only the analytical interface responses can be used since it is unlikely that a rigid payload will be flown. Therefore, the measured flight data cannot be directly applied in the design/analysis process.

The objective of the present investigation is to develop a method by which the interface accelerations of one launch vehicle/payload system, either analytically obtained or flight measured, can be used directly in another launch vehicle/payload system design procedure, thus, the name "Recovered Transient Analysis." This effort should be especially beneficial to the future shuttle payload design. The payload dynamic load can be obtained by performing a transient analysis of the payload model using the interface accelerations modified from another composite system which consists of an identical launch vehicle, the shuttle, and a

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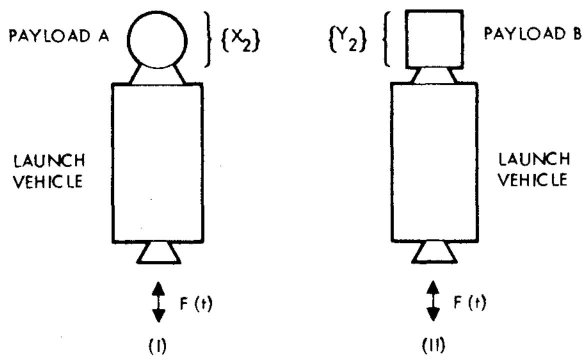

Fig. 1 Schematic of launch vehicle/payload composite systems.

different payload as the forcing functions. In principle, the proposed transient analysis recovers the interface accelerations for the unloaded launch vehicle and then modifies it to include the dynamic characteristics of the new payload. After a series of shuttle launches, the flight measured interface accelerations can be used to establish a payload forcing function data base for the subsequent payload design. The payload organization using the proposed loads analysis method can then perform payload loads analysis within the payload organization in a cost-effective and timely manner. The study considers that the payload is supported in a statically determinate manner.

In recent article,[6] it was pointed out that current dynamic analysis methods are unresponsive to the design process because of the long computational times and their associated costs. Their use is virtually precluded in the early design stages where frequent design changes are occurring. The presently proposed method will not only simplify the analytical process thus reducing the cost but also will provide results in a timely manner due to minimal interfacing between the organizations. Therefore, the rigor and potential accuracy of systematic analytical procedure can be brought to bear on not only the design process in early stages but also guiding and supplementary qualification testings in the later stages of the project.

Approach

In the present investigation, a method to obtain the payload response of a launch vehicle/payload composite system from the results of another launch vehicle/payload composite system under the identical forcing function is developed. Figure 1 shows two composite systems with identical launch vehicle but different payloads. The governing equations for these two systems can be written in the finite-element formulation as

$$\begin{bmatrix} m_l & 0 \\ 0 & m_2 \end{bmatrix}\begin{Bmatrix} \ddot{x}_l \\ \ddot{x}_2 \end{Bmatrix}+\left(\begin{bmatrix} k_l & 0 \\ 0 & 0 \end{bmatrix}+\begin{bmatrix} k_{11} & k_{21} \\ k_{12} & k_{22} \end{bmatrix}\right)\times\begin{Bmatrix} x_l \\ x_2 \end{Bmatrix}=\begin{Bmatrix} F(t) \\ 0 \end{Bmatrix} \tag{1}$$

and

$$\begin{bmatrix} m_l & 0 \\ 0 & \bar{m}_2 \end{bmatrix}\begin{Bmatrix} \ddot{y}_l \\ \ddot{y}_2 \end{Bmatrix}+\left(\begin{bmatrix} k_l & 0 \\ 0 & 0 \end{bmatrix}+\begin{bmatrix} \bar{k}_{11} & \bar{k}_{21} \\ \bar{k}_{12} & \bar{k}_{22} \end{bmatrix}\right)\times\begin{Bmatrix} y_l \\ y_2 \end{Bmatrix}=\begin{Bmatrix} F(t) \\ 0 \end{Bmatrix} \tag{2}$$

where $\{x_l\}$, $\{y_l\}$ are launch vehicle degrees-of-freedom (DOF) of the composite systems (I) and (II), respectively; $\{x_2\}$, $\{y_2\}$ are payload DOF of the composite systems (I) and (II), respectively; $[m_l]$ is the mass matrix of the launch vehicle; $[k_l]$ is the stiffness matrix of the launch vehicle; $[m_2]$, $[\bar{m}_2]$ are the mass matrix of the payload A and payload B of systems (I) and (II), respectively; $[k_{11}]$, $[k_{12}]$, $[k_{21}]$, $[k_{22}]$ $[\bar{k}_{11}]$, $[\bar{k}_{12}]$, $[\bar{k}_{21}]$, $[\bar{k}_{22}]$ are the submatrices of the total payload stiffness; the matrix positioned into launch vehicle/payload interface DOF and payload DOF for systems (I) and (II), respectively; and $\{F(t)\}$ is the vector representing the external forcing functions acting on the launch vehicle DOF. Assuming a "partial" solution of Eq. (1) is available, the objective of the proposed method is to obtain the payload response $\{y\}$ of Eq. (2) by using the results of Eq. (1). The proposed procedure does not involve the launch vehicle model and the forcing functions acting on the launch vehicle; hence the entire process can be implemented by the payload organization alone.

Although damping is not included in Eqs. (1) and (2), it will be incorporated later in the form of modal damping. For the present study, it will be assumed at present that the payload is supported in a statically determinate manner such that

$$[k_{11}]=[\phi_R]^T[k_{22}][\phi_R] \quad [k_{21}]=-[\phi_R]^T[k_{22}]=[k_{12}]^T \tag{3}$$

where $[\phi_R]$ is the payload A rigid-body transformation matrix defined as the payload displacements due to unit displacement of the launch vehicle/payload interface DOF, $\{x_I\}$; and $\{x_I\}$ is the launch vehicle/payload interface DOF connecting payload to launch vehicle, a subset of the launch vehicle DOF $\{x_l\}$. Next, the motion of the payload will be decomposed into two parts, namely, the rigid-body motion and the elastic motion:

$$\{x_2\}=[\phi_R]\{x_I\}+\{x_e\} \tag{4}$$

The first term on the right-hand side of Eq. (4) is the rigid-body motion. The second term, $\{x_e\}$, is the elastic motion, or relative motion with reference to the interface. It should be noted that only the elastic motion $\{x_e\}$ will generate internal loads in the structure. Using Eqs. (3) and (4), Eq. (1) can be transformed into the following form:

$$\begin{bmatrix} m_l+m_{rr} & \phi_R^T m_2 \\ m_2\phi_R & m_2 \end{bmatrix}\begin{Bmatrix} \ddot{x}_l \\ \ddot{x}_e \end{Bmatrix}+\begin{bmatrix} k_l & 0 \\ 0 & k_{22} \end{bmatrix}\begin{Bmatrix} x_l \\ x_e \end{Bmatrix}=\begin{Bmatrix} F(t) \\ 0 \end{Bmatrix} \tag{5}$$

where

$$[m_{rr}]=[\phi_R]^T[m_2][\phi_R] \tag{6}$$

denoted as rigid-body mass of payload A.

Similarly, Eq. (2) can be transformed into the following form:

$$\begin{bmatrix} m_l+\bar{m}_{rr} & \bar{\phi}_R^T \bar{m}_2 \\ \bar{m}_2\bar{\phi}_R & \bar{m}_2 \end{bmatrix}\begin{Bmatrix} \ddot{y}_l \\ \ddot{y}_e \end{Bmatrix}+\begin{bmatrix} k_l & 0 \\ 0 & \bar{k}_{22} \end{bmatrix}\begin{Bmatrix} y_l \\ y_e \end{Bmatrix}=\begin{Bmatrix} F(t) \\ 0 \end{Bmatrix} \tag{7a}$$

where $[\bar{\phi}_R]$ is the payload B rigid-body transformation matrix defined similarly as $[\phi_R]$ for payload A.

$$[\bar{m}_{rr}] = [\bar{\phi}_R]^T[\bar{m}_2][\bar{\phi}_R] \tag{7b}$$

denoted as rigid-body mass of payload B. Also the motion of the payload B is decomposed into two parts similar to that of payload A, Eq. (4), as

$$\{y_2\} = [\bar{\phi}_R]\{y_I\} + \{y_e\} \tag{8}$$

where $\{y_I\}$ is subset of $\{y_l\}$ defined as the launch vehicle/payload interface DOF similar to that of $\{x_I\}$.

Since the objective is to utilize the results of the composite system (I) to solve for the response of the composite system (II), the solution of Eq. (5) which is the governing equation for the system (I) will be examined.

Let

$$\begin{Bmatrix} x_l \\ x_e \end{Bmatrix} = \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} \{q(t)\} \tag{9}$$

where $[\phi]$ and $[\phi_e]$ are the eigenvectors of Eq. (5) for the launch vehicle DOF and payload elastic DOF, respectively. $\{q(t)\}$ is the generalized coordinate vector. The eigenvectors satisfy the following orthogonality conditions:

$$\begin{bmatrix} \phi \\ \phi_e \end{bmatrix}^T \begin{bmatrix} m_l + m_{rr} & \phi_R^T m_2 \\ m_2\phi_R & m_2 \end{bmatrix} \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} = \begin{bmatrix} \diagdown & & \\ & I & \\ & & \diagdown \end{bmatrix}$$

$$\begin{bmatrix} \phi \\ \phi_e \end{bmatrix}^T \begin{bmatrix} k_l & 0 \\ 0 & k_{22} \end{bmatrix} \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} = \begin{bmatrix} \diagdown & & \\ & \omega^2 & \\ & & \diagdown \end{bmatrix} \tag{10}$$

It should be noted that $[m_{rr}]$ and $[\bar{m}_{rr}]$ are not the same size as $[m_l]$. However, $[m_{rr}]$ and $[\bar{m}_{rr}]$ only multiply $\{\ddot{x}_I\}$ and $\{\ddot{y}_I\}$, respectively, and since $\{x_I\}$ and $\{y_I\}$ are the subset of $\{x_l\}$ and $\{y_l\}$, respectively, the additions such as $[m_l]\{\ddot{x}_l\} + [m_{rr}]\{\ddot{x}_I\}$ and $[m_l]\{\ddot{y}_l\} + [\bar{m}_{rr}]\{\ddot{y}_I\}$ are valid. $[\diagdown\omega^2\diagdown]$ is the composite system (I) eigenvalue matrix. From Eq. (4), the complete eigenvectors for the payload A can be obtained as

$$[\phi_S] = [\phi_R][\phi_I] + [\phi_e] \tag{11}$$

where $[\phi_I]$ is the eigenvector for the interface DOF and is a subset of the eigenvectors for the launch vehicle DOF $[\phi]$.

By substituting Eq. (9) into Eq. (5) and using the orthogonality conditions as expressed in Eq. (10), one obtains the governing equations for the generalized coordinates $q(t)$.

$$\{\ddot{q}\} + [\diagdown 2\rho\omega \diagdown]\{\dot{q}\} + [\diagdown\omega^2\diagdown]\{q\} = [\phi]^T\{F(t)\} \tag{12}$$

Note that the modal damping, $[\diagdown 2\rho\omega\diagdown]$, has been included.

One other quantity, $[m_{er}]$, representing the inertia coupling effects between the launch vehicle and the payload is often the output of the analysis. It is defined as follows:

$$[m_{er}] = [\phi_S]^T[m_2][\phi_R] \tag{13}$$

In summary, the following results from the composite system (I) will be available to the payload organization of the composite system (II):

1) The system eigenvalues $[\diagdown\omega^2\diagdown]$.
2) The system eigenvectors at interface DOF $[\phi_I]$.
3) Rigid-body mass and inertia coupling matrices, $[m_{rr}]$ and $[m_{er}]$.
4) Time history of the generalized coordinates $\{q(t)\}$ or $\{\ddot{q}(t)\}$.

With this information, the responses of system (I) represented by Eq. (7) will be sought. First, the external forcing functions $\{F(t)\}$ must be eliminated by combining Eqs. (5) and (7a).

$$\begin{bmatrix} m_l + \bar{m}_{rr} & \bar{\phi}_R^T\bar{m}_2 \\ \bar{m}_2\bar{\phi}_R & \bar{m}_2 \end{bmatrix} \begin{Bmatrix} \ddot{z}_l \\ \ddot{y}_e \end{Bmatrix} + \begin{bmatrix} k_l & 0 \\ 0 & \bar{k}_{22} \end{bmatrix} \begin{Bmatrix} z_l \\ y_e \end{Bmatrix}$$

$$= \begin{bmatrix} [m_{er}]^T - [\bar{m}_{rr}][\phi_I] \\ -[\bar{m}_2][\bar{\phi}_R][\phi_I] \end{bmatrix} \{\ddot{q}(t)\} \tag{14}$$

where

$$\{z_l\} = \{y_l\} - \{x_l\}, [\bar{\phi}_R]\{x_I\} = [\phi_R]\{x_I\} \tag{15}$$

In Eq. (14), the external forcing function $\{F(t)\}$ has been replaced by the modal acceleration $\{\ddot{q}(t)\}$ which will be available to the payload organization. However, to solve for Eq. (14), the eigenvalues and eigenvectors of the system (I) are required. From Eq. (14), it appears that the launch vehicle mass and stiffness matrices, $[m_l]$ and $[k_l]$, are needed which means the launch vehicle model is needed. Since it is an a priori assumption that the launch vehicle modal will not be available, a method will be devised by which the required eigendata can be extracted without the use of the detailed launch vehicle model, only the generalized or modal launch vehicle model. Basically, the method will first obtain the eigendata for an unloaded launch vehicle by removing payload A from the composite system (I). Next, the unloaded launch vehicle will be coupled with payload B to obtain the eigendata for the composite system (II). For reasons of matrix size compatibility, a system consists of an unloaded launch vehicle and a cantilevered payload A will be studied.

$$\begin{bmatrix} m_l & 0 \\ 0 & m_2 \end{bmatrix} \begin{Bmatrix} \ddot{x}_0 \\ \ddot{x}_2 \end{Bmatrix} + \begin{bmatrix} k_l & 0 \\ 0 & k_{22} \end{bmatrix} \begin{Bmatrix} x_0 \\ x_2 \end{Bmatrix} = 0 \tag{16}$$

It is obvious that payload A is not coupled with the launch vehicle; therefore the results from Eq. (16) includes the eigendata for an unloaded launch vehicle and a cantilevered payload A. Let

$$\begin{Bmatrix} x_0 \\ x_2 \end{Bmatrix} = \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} \{u\} \tag{17}$$

Substituting Eq. (17) into Eq. (16) and premultiplying the transpose of the transformation matrix, one obtains

$$\begin{bmatrix} \phi \\ \phi_e \end{bmatrix}^T \left(\begin{bmatrix} m_l + m_{rr} & \phi_R^T m_2 \\ m_2\phi_R & m_2 \end{bmatrix} - \begin{bmatrix} m_{rr} & \phi_R^T m_2 \\ m_2\phi_R & 0 \end{bmatrix} \right)$$

$$\times \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} \{\ddot{u}\} + \begin{bmatrix} \phi \\ \phi_e \end{bmatrix}^T \begin{bmatrix} k_l & 0 \\ 0 & k_{22} \end{bmatrix} \begin{bmatrix} \phi \\ \phi_e \end{bmatrix} \{u\} = 0 \tag{18}$$

Using the orthogonality relationship of Eq. (10) and the relationship in Eq. (11), Eq. (18) can be reduced into the form of

$$([\diagdown I \diagdown] + [\phi_I]^T[m_{rr}][\phi_I] - [m_{er}][\phi_I]$$

$$- [\phi_I]^T[m_{er}]^T)\{\ddot{u}\} + [\diagdown\omega^2\diagdown]\{u\} = 0 \tag{19}$$

All the matrices in Eq. (19) are available to the payload organization and they are in much simpler form than the launch vehicle mass and stiffness matrices. The eigenvalues and eigenvectors for Eq. (19) will be denoted as $[\diagdown\omega_0^2\diagdown]$ and $[\psi]$, respectively. From Eq. (17), the eigenvectors for unloaded launch vehicle interface DOF will be

$$[\phi_{0I}] = [\phi_I][\psi] \tag{20}$$

As mentioned before, the eigenvalues and the eigenvectors obtained from Eq. (19) contain the effect of the cantilevered payload A. It must be identified and removed from the results. One way is to use the orthogonality condition to identify the payload modes. Let

$$[\phi_{0S}] = [\phi_S][\psi] \tag{21}$$

From Eqs. (17) and (21), it is clear that the cantilevered payload modes are among the modes $[\phi_{0S}]$. Since the payload modes are orthogonal modes with respect to the mass matrix, the following multiplication will ideally produce unit diagonal terms and zero off-diagonal terms for those payload modes:

$$[M_g] = [\phi_{0S}]^T[m_2][\phi_{0S}] \tag{22}$$

Although truncation errors will contaminate the unities and zeros, the errors should be small if sufficient modes are taken. Thus, the matrix $[M_g]$ can be used to sort out the payload modes.

After the eigendata of the unloaded launch vehicle are obtained, the eigendata for the new composite model, Eq. (14), consisting of the identical launch vehicle and a new payload B can be obtained. Let

$$\begin{Bmatrix} z_I \\ y_e \end{Bmatrix} = \begin{bmatrix} \phi_0 & 0 \\ 0 & \bar{\phi}_2 \end{bmatrix} \begin{Bmatrix} u_I \\ u_2 \end{Bmatrix} \tag{23}$$

where $[\phi_0]$ and $[\bar{\phi}_2]$ are the eigenvectors of the unloaded launch vehicle and the cantilevered payload B, respectively. [Note: the matrix $[\phi_{0I}]$ is available from Eq. (20)]. Therefore, they satisfy the following orthogonality condition:

$$[\phi_0]^T[m_I][\phi_0] = [\diagdown I \diagdown] \qquad [\phi_0]^T[k_I][\phi_0] = [\diagdown \omega_0^2 \diagdown] \tag{24}$$

$$[\phi_2]^T[\bar{m}_2][\phi_2] = [\diagdown I \diagdown] \qquad [\phi_2]^T[\bar{k}_{22}][\phi_2] = [\diagdown \bar{\omega}_2^2 \diagdown] \tag{25}$$

where $[\diagdown \omega_0^2 \diagdown]$ and $[\diagdown \bar{\omega}_2^2 \diagdown]$ are the eigenvalues of the unloaded launch vehicle and the cantilevered payload B, respectively. Using the transformation in Eq. (23), the eigenproblem for the new composite model, Eq. (14), can be rewritten as

$$\left([\diagdown I \diagdown] + \begin{bmatrix} \phi_{0I}^T \bar{m}_{rr} \phi_{0I} & \phi_{0I}^T \bar{m}_{er} \\ \bar{m}_{er} \phi_{0I} & 0 \end{bmatrix} \right) \begin{Bmatrix} \ddot{u}_I \\ \ddot{u}_2 \end{Bmatrix} + \begin{bmatrix} \omega_0^2 & \\ & \bar{\omega}_2^2 \end{bmatrix} \begin{Bmatrix} u_I \\ u_2 \end{Bmatrix} \tag{26}$$

where

$$[\bar{m}_{er}] = [\bar{\phi}_2]^T[\bar{m}_2][\bar{\phi}_R] \tag{27}$$

The elements in the coefficient matrices of Eq. (26) are typically available either from the unloaded launch vehicle eigendata such as the $[\phi_{0I}]$ and $[\diagdown \omega_0^2 \diagdown]$ or from the cantilevered new payload eigendata such as the $[\bar{m}_{rr}]$, $[\bar{m}_{er}]$, and $[\diagdown \bar{\omega}_2^2 \diagdown]$. Therefore, the eigenvectors and eigenvalues can be calculated by the payload organization. They satisfy the following orthogonality conditions:

$$\begin{bmatrix} \bar{\psi}_I \\ \bar{\psi}_p \end{bmatrix}^T \left([\diagdown I \diagdown] + \begin{bmatrix} \phi_{0I}^T \bar{m}_{rr} \phi_{0I} & \phi_{0I}^T \bar{m}_{er}^T \\ m_{er} \phi_{0I} & 0 \end{bmatrix} \right) \times \begin{bmatrix} \bar{\psi}_I \\ \bar{\psi}_p \end{bmatrix} = [\diagdown I \diagdown] \tag{28}$$

$$\begin{bmatrix} \bar{\psi}_I \\ \bar{\psi}_p \end{bmatrix}^T \begin{bmatrix} \omega_0^2 & \\ & \omega_2^2 \end{bmatrix} \begin{bmatrix} \bar{\psi}_I \\ \bar{\psi}_p \end{bmatrix} = [\diagdown \bar{\omega}^2 \diagdown]$$

where $[\bar{\psi}_I]$, $[\bar{\psi}_p]$ are the eigenvectors and $[\diagdown \bar{\omega}_2 \diagdown]$ are the eigenvalues. The eigenvectors of the launch vehicle/payload interface DOF will be as follows:

$$[\bar{\phi}_I] = [\phi_{0I}][\bar{\psi}_I] \tag{29}$$

The responses of the composite model, Eq. (14), will be solved by using the following transformation:

$$\begin{Bmatrix} z_I \\ y_I \end{Bmatrix} = \begin{bmatrix} \phi_0 & 0 \\ 0 & \phi_2 \end{bmatrix} \begin{bmatrix} \bar{\psi}_I \\ \bar{\psi}_p \end{bmatrix} \{\bar{q}(t)\} \tag{30}$$

Then the uncoupled modal equations can be obtained as

$$\{\ddot{\bar{q}}\} + [\diagdown 2\bar{\rho}\bar{\omega} \diagdown]\{\dot{\bar{q}}\} + [\diagdown \bar{\omega}^2 \diagdown]\{\bar{q}\} = [G]\{\ddot{q}(t)\} \tag{31}$$

where

$$[G] = \left[[\bar{\phi}_I]^T[m_{er}]^T - [\bar{m}_{rr}][\phi_T] - [\bar{\psi}_p]^T[\bar{m}_{er}][\phi_I] \right] \tag{32}$$

Since the quantities on the right-hand side of Eq. (31) are available to the payload organization, the modal response $q(t)$ can be obtained from the LV organization in a timely manner. It should be noted that model damping $[\diagdown 2\bar{\rho}\bar{\omega} \diagdown]$ has been included in Eq. (31).

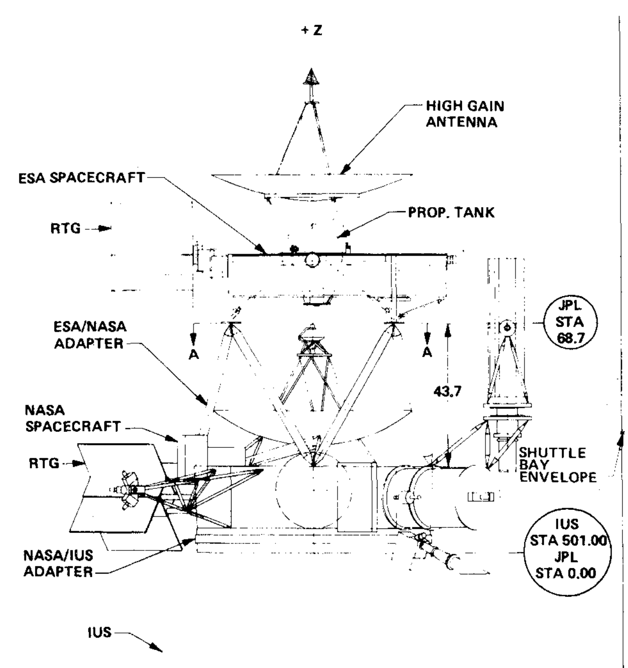

Fig. 2 ISPM spacecraft.

The physical accelerations of each payload DOF can be obtained by using Eqs. (8), (15), and (30) as follows:

$$
\begin{aligned}
\{\ddot{y}_2\} &= [\bar{\phi}_R]\{\ddot{y}_I\} + \{\ddot{y}_e\} \\
&= [\bar{\phi}_R]\left(\{\ddot{z}_I\} + \{\ddot{x}_I\}\right) + \{\ddot{y}_e\} \\
&= [\bar{\phi}_R]\left([\bar{\phi}_{0I}][\bar{\psi}_I]\{\ddot{\bar{q}}(t)\} + \{\ddot{x}_I\}\right) + [\phi_2][\bar{\psi}_p]\{\ddot{\bar{q}}(t)\} \\
&= \left([\bar{\phi}_R][\phi_{0I}][\bar{\psi}_I] + [\phi_2][\bar{\psi}_p]\right)\{\ddot{\bar{q}}(t)\} + \{\ddot{x}_I\} \qquad (33)
\end{aligned}
$$

The structural member loads, $\{p\}$, can be obtained by the following operations:

$$\{p\} = [S]\{y_e\} = [S][\bar{\phi}_2][\bar{\psi}_p]\{\bar{q}(t)\} \qquad (34)$$

where $[S]$ is the loads coefficient matrix which relates the elastic deformation to the member loads. Typically, in a payload dynamic analysis, the product $[S][\bar{\phi}_2]$, rather than $[S]$ is obtained.

Table 1 ISPM finite-element model

	NASA S/C	ESA S/C	Total
Number of grid points	112	122	234
Number of static degrees of freedom	400	430	830
Number of finite elements	298	221	519
Number of dynamic degrees of freedom	150	87	237
Number of vibration modes of interest	51	84	135
Number of modes retained—loads analysis			40
Frequency			143 Hz

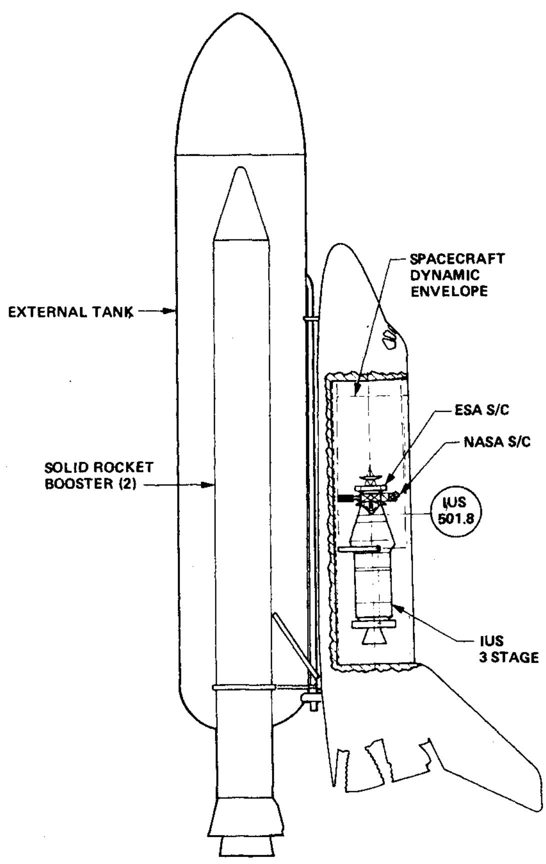

Fig. 3 STS/IUS with ISPM payload.

Sample Problem

The proposed payload transient analysis technique will be demonstrated on a realistic complex structural system, namely, the International Solar Polar Mission (ISPM) spacecraft. The ISPM consists of two separate spacecrafts, one sponsored by NASA and the other by the European Space Agency (ESA). Figure 2 shows the schematic of the ISPM spacecraft with the major structural components identified. Table 1 contains the descriptions of the mathematical model.

The two spacecrafts were designed to be launched in tandem from a single shuttle/Inertial Upper Stage (IUS) to conduct scientific explorations in the sun's polar orbit, approximately 90 deg from the ecliptic plane. The space shuttle launch vehicle configuration consists of the orbiter, external oxygen and hydrogen tank (ET), and two solid rocket boosters (SRBs). For interplanetary trajectories, an IUS is included for additional propulsion for the payloads to reach distant planets. Figure 3 shows the ISPM/shuttle/IUS composite system in the liftoff configuration. The load conditions on which the shuttle structural system is designed are numerous, such as the liftoff, high boost acceleration, SRB burn, SRB staging, orbiter main engine burn, orbiter main engine cutoff, external tank jettison, space operation, entry and descent, TAEM (terminal area energy management), landing approach, and various abort conditions. However, for the payload structure design it was found that the liftoff and abort landing events are of importance. In the sample problem only the liftoff environment will be considered.

Piror to the ISPM project, another planetary spacecraft, Galileo, has been designed to be launched by the identical shuttle/IUS launch vehicle system. The shuttle/IUS/Galileo composite model has been analyzed for the dynamic environments representing the liftoff and abort landing. Results of the analysis will be used as described previously for the analysis of the shuttle/IUS/ISPM system. The first composite system, shuttle/IUS/Galileo, is represented by 150 normal modes at the interface DOF and their corresponding frequencies, i.e., $[\phi]$ and $[\diagdown\omega^2\diagdown]$. Also, the modal response

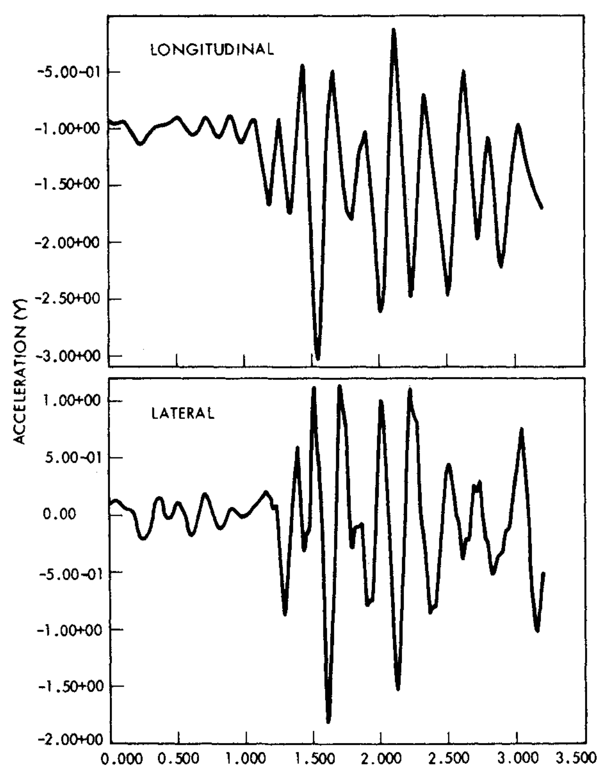

Fig. 4 IUS/Galileo interface accelerations.

Table 2 Member loads for NASA/ESA spacecraft adapter truss

Member	Miniload analysis (N)	Recovered transient analysis, (N)
701	4,278	2,862
702	11,955	7,968
703	14,557	10,943
704	9,634	7,692
705	4,726	3,759
706	12,094	9,430
707	14,284	12,197
708	8,688	6,891

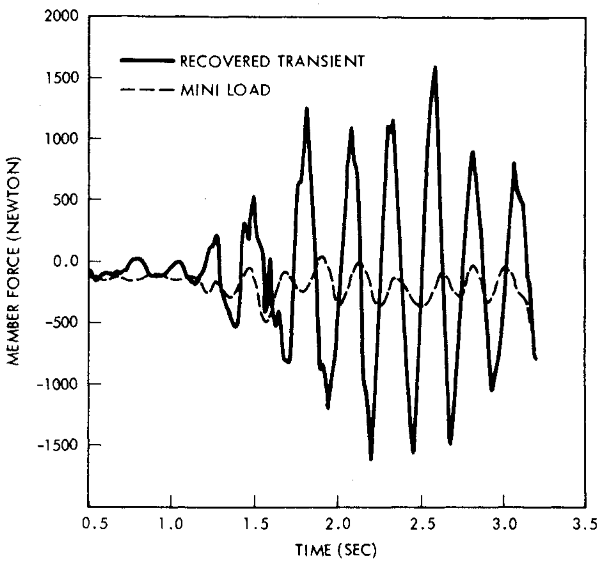

Fig. 5 Load time history for RCS thruster outrigger.

time histories $\{q(t)\}$ are available. The cantilevered ISPM model is represented by 40 normal modes and their corresponding frequencies, i.e., $[\phi_2]$ and $[\diagdown\bar{\omega}_2^2\diagdown]$. A constant modal damping coefficient $C/Cc=0.01$ is assumed for all the modes. Resulting member loads of selected components from the recovered transient analysis will be compared to those from the miniloads analysis in which the payload will be subject to a base motion obtained from IUS/Galileo interface accelerations.

Figure 4 shows the IUS/Galileo interface acceleration for the liftoff condition. These accelerations have been used as forcing functions for the miniloads analysis. Since the ISPM spacecraft is in the same class as that of the Galileo spacecraft, the approximate dynamic environment for the ISPM can be estimated from Fig. 4. In this case, approximately a $3g$ peak acceleration will be experienced at the base of the ISPM spacecraft in longitudinal direction and $2g$ peak acceleration in the lateral direction. Table 2 shows the loads obtained by the recovered transient method for the NASA/ESA spacecraft adapter trusses. The comparison of these loads to those obained by the miniloads analysis shows that good agreement has been achieved. Since the spacecraft adapter trusses are the structural members connecting the two spacecraft, the loads are mainly derived from the low-frequency motion of the interface. Therefore, it is not surprising that the miniloads analysis predicts the loads accurately. However, for the structural members dominated by the high-frequency local motion, the discrepancies between the results of these two analyses can be substantial. Figure 5 shows the time histories of the loads obtained by a miniloads analysis and recovered transient analysis for a RCS thruster outrigger. These loads are an order of magnitude different from each other.

Concluding Remarks

One of the objectives of the space transportation system (STS) is to deliver the payload inexpensively and timely. Logically the payload design should also be inexpensive and carried out in a timely manner. In the present report, such a structural design methodology has been presented. The reduction in cost and schedule resulted from the fact that the analytical process is greatly simplified and the interface between organizations is minimized. However, the biggest advantage is that the vigorous systematic analytical procedure can be brought into the early design process such that major design changes in the later stage can be avoided.

The questions of accuracy or error due to truncation of modes to the launch vehicle and payload and the treatment of damping have not been addressed in the present study. Their importance warrants further studies to complete the presently proposed methodology.

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