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Damping effect on supersonic panel flutter of composite plate with viscoelastic mid-layer

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ABSTRACT

The panel flutter of composite plate with viscoelastic mid-layer in supersonic airflow is investigated in this study. The hysteric damping model is used to describe the viscoelastic property of the mid-layer material and the first piston theory to model the aerodynamic forces. Hamilton's principle is employed to derive the partial differential equations governing the vibrations of the laminated composite plate. By Galerkin method the governing partial differential equations are truncated into a set of ordinary differential equations. The critical dynamic pressure for panel flutter has been studied by considering the eigenvalue problem of the set of ODEs. It was concluded that the introduction of aerodynamic damping postpones the threshold of flutter, while the viscoelastic damping of the soft mid-layer presents a phenomenon of dual effect. When the viscoelastic damping value is low, the internal material damping shows detrimental effect to the flutter depression. If the viscoelastic damping is increased further, the flutter resistance of the composite plate will be enhanced. The convergence of the Galerkin method is also discussed in this study. The influences of some parameters to the convergence have been investigated in details.

1. Introduction

Panel flutter is a self-excited oscillation which is known to cause fatigue damage of flight vehicles. At critical dynamic pressure, the panel motion becomes unstable. This vibration occurs due to energy transfer from the gas flow to the panel. The amplitude of the flutter oscillation grows with time until it is restrained due to the in-plate tensile stress induced by the geometric nonlinearities. Although this type of self-excited oscillation involves nonlinear terms, the linear theory can determine the flutter boundaries.

Since 1960s, the nonlinear aeroelastic behavior of panels has been studied by several investigators. The Galerkin method is one of the useful approaches to solve the panel flutter problem. Bolotin [1] has studied the nonlinear problem by the use of Galerkin's method in spatial variables to determine the properties of the limit cycle oscillation. Galerkin's method and then numerical integration was employed by Dowell [2,3], Ventres and Dowell [4] to study nonlinear panel flutter. Their results showed that the aerodynamic damping, in-plane forces, and static pressure differentials have significant effect on the behavior of the plate. Finite element method is another powerful tool to study the panel flutter problem [5,6].

In recent years, composite materials are being increasingly used in the exposed skin of flight vehicles because of their high stiffness and strength-to-weight ratio. Birman and Librescu [7] analyzed the supersonic flutter of shear deformable laminated composite flat panels. Shiau and Lu [8] studied the nonlinear flutter behavior of two-dimensional simply supported symmetric composite laminated plates using a direct numerical integration method and analyzed the effects of anisotropic preperties and elastic modulus ratio. Dixon and Mei [9] applied Finite element method to study the nonlinear flutter of thin laminates, critical flutter values and limit-cycle amplitudes at various lamination orientations and numbers of layers were investigated. Singha and Mandal [10] used 16-noded isoparametric shell element and discussed the flutter of laminated composite plates and cylindrical panels. The meshless method was applied to the panel flutter problem by Zhao et al. [11]. Kouchakzadeh et al. [12] employed Galerkin method to investigate the panel flutter of a general laminated composite plate. The effects of in-plane force, fiber orientations and aerodynamic damping on the nonlinear aeroelastic behavior of the plate have been discussed by numerical simulations. For smart laminated plates with electrorheological fluid core and orthotropic faces, Jafar and Jalil [13] discussed the effects of parameters such as electric field

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strength, electrorheological fluid type and fiber angle of orthotropic faces on the critical aerodynamic pressure. Flutter suppression is also one of the main objectives of the aeroelastic analysis. Recently, researches regarding the active vibration control of structures by means of smart material have been intensifying. Song and Li [14] studied the active aeroelastic flutter characteristics and vibration control of the supersonic composite laminated plates with the piezoelectric acuator/sensor pairs. They found that the aeroelastic flutter properties of the laminated plate can be improved using the proportional feedback control strategy while the negative velocity feedback control which provide an active damping can destablilize the flutter stability. Li [15] investigated the active aeroelstic flutter properties of supersonic plates using the piezoelectric material. The numerical results demonstrated that the active stiffness and active mass had prominent effects on the flutter characteristics, with the increase of the feedback control gains, the active aeroelastic properties for the lower order modes were gradually improved.

Damping refers to the extraction of mechanical energy from a vibrating system usually by conversion into heat. Damping serves to control the steady state resonant response and to attenuate traveling waves in the structure. Passive damping especially the use of surface damping treatments in the automotive, commercial airplane has only been in recent years. Rao [16] described some of recent industrial applications of passive damping using viscoelastic materials and pointed out the symmetric configuration in which the base and the constraining layers have the same thickness and stiffness is by far the most effective design since it maximizes the shear deformation in the core layer. For sandwich plates with viscoelastic Damping, Wang [17] updated assumed modes in order to reduce the number of modes and maintain the accuracy of predictions for frequency, mode shapes and frequency response. Karim and Chen [18] investigated the effects of anchored constrained viscoelastic layers on the flexural response of simply supported Euler beams or plate strips under base excitations, they found the two-end anchored configuration is most effective in vibration suppression.

The damping effects are also one of the most important and interesting aspects of the theory of stability for elastic nonconservative systems (including panel flutter). Bolotin and Zhinzher [19] applied a method of expansion in fractional powers of parameters into the stability problem of a cantilever bar subjected at the free end to a tangential force and a dead load. They found that for real laws of damping a considerable part of quasistability region belongs to the instability region. Lottati and Kornecki [20] analyzed the effect of an elastic foundation and various damping forces on the stability of a fluid-conveying pipe. Results indicated that internal damping destabilizes the system in the post-divergence region for the pipe with both ends fixed. The coupling effect between external and internal damping for panel flutter were investigated by Lottati [21], Higuchi and Dowell [22] studied the effect of structural damping on flutter of Plates with a follower force. The calculation showed that small damping may destabilize the nonconsevative system and the destabilizing effect depends on the slenderness ratio. Bismarck-Nasr and Bones [23] discussed the destabilizing and stabilizing effect of hysteric type structural damping in panel flutter analysis. Shin et al. [24] performed supersonic flutter analysis of cylindrical composite panels with structural damping treatments using the finite element method based on the zig-zag layerwise shell theory. They compared the aeroelastic stabilities of an original base panel with various damping characteristics of unconstrained layer, constrained layer, and symmetrically co-cured sandwich laminates.

In this study, the Galerkin method coupled with the eigenvalue problem solving technique is used to investigate supersonic panel flutter characteristics for composite sandwich plate with viscoelastic mid-layer. The effects of aerodynamic damping and structural viscoelastic damping on the panel flutter are discussed in detail. The convergence of Galerkin method applied to panel flutter of composite plate with soft core is demonstrated.

2. Governing equations of the composite sandwich panel

We consider a rectangular plate on simple supports within supersonic flow. The plate is assumed to consist of two laminated face plates and a layer of soft viscoelastic core. The geometry and the corresponding coordinate system are presented in Fig. 1. It shows that the origin of the coordinate system xyz is placed at the center of the core, i.e., $z = 0$ in the core mid-plane. In this study, the first-order transverse shear deformation will be considered for the face layers of the composited sandwich plate. The detailed investigation of such theory may refer to the recent monograph by Carlsson and Kardomateas [25] and the book by Reddy [26]. The analysis is based on the following assumptions:

(i) The face sheets are thin compared to the core, i.e., $f_t, f_b \ll c$, and in a state of plane stress ($\sigma_z = \tau_{xz} = \tau_{yz} = 0$).
(ii) The in-plane stresses, σ_x, σ_y, and τ_{xy}, in the core are negligible.
(iii) In-plane displacements, u and v, are uniform through the thickness of the face sheets and assume their mid-plane values.
(iv) The out-of-plane displacement w is independent of the z coordinate, i.e., the thickness strain, $\varepsilon_z = \partial w / \partial z = 0$.
(v) The in-plane displacements in the core, u and v, are linear in the thickness coordinate z.

Firstly, we study the top and bottom face layer. Based on first-order shear deformation plate theory and the assumptions (i), (ii) and (iii), the deformation-strain relations for top and bottom face layers may be expressed respectively as

$$\begin{Bmatrix} \varepsilon_{xx}^t \\ \varepsilon_{yy}^t \\ \gamma_{xy}^t \end{Bmatrix} = \begin{Bmatrix} \frac{\partial u_0}{\partial x} \\ \frac{\partial v_0}{\partial y} \\ \frac{\partial u_0}{\partial y} + \frac{\partial v_0}{\partial x} \end{Bmatrix} + \left(c + \frac{f_t}{2}\right) \begin{Bmatrix} \frac{\partial \varphi_x}{\partial x} \\ \frac{\partial \varphi_y}{\partial y} \\ \frac{\partial \varphi_y}{\partial x} + \frac{\partial \varphi_x}{\partial y} \end{Bmatrix} - \left(z + c + \frac{f_t}{2}\right) \begin{Bmatrix} \frac{\partial^2 w}{\partial x^2} \\ \frac{\partial^2 w}{\partial y^2} \\ 2\frac{\partial^2 w}{\partial x \partial y} \end{Bmatrix} \tag{1a}$$

$$\begin{Bmatrix} \varepsilon_{xx}^b \\ \varepsilon_{yy}^b \\ \gamma_{xy}^b \end{Bmatrix} = \begin{Bmatrix} \frac{\partial u_0}{\partial x} \\ \frac{\partial v_0}{\partial y} \\ \frac{\partial u_0}{\partial y} + \frac{\partial v_0}{\partial x} \end{Bmatrix} + \left(c - \frac{f_b}{2}\right) \begin{Bmatrix} \frac{\partial \varphi_x}{\partial x} \\ \frac{\partial \varphi_y}{\partial y} \\ \frac{\partial \varphi_y}{\partial x} + \frac{\partial \varphi_x}{\partial y} \end{Bmatrix} + \left(z - c - \frac{f_b}{2}\right) \begin{Bmatrix} \frac{\partial^2 w}{\partial x^2} \\ \frac{\partial^2 w}{\partial y^2} \\ 2\frac{\partial^2 w}{\partial x \partial y} \end{Bmatrix} \tag{1b}$$

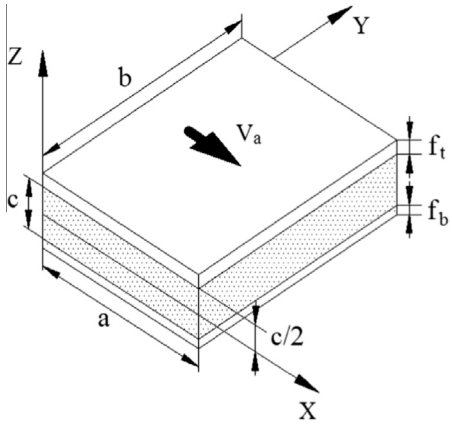

Fig. 1. Sandwich panel geometry.