THERMAL BUCKLING BEHAVIOR OF COMPOSITE LAMINATES

Le-Chung Shiau[1], Shih-Yao Kuo[2] and Cheng-Yuan Chen[1]
[1]Department of Aeronautics & Astronautics, National Cheng Kung University, Tainan, Taiwan, ROC
[2]Department of Aviation Service Management, Aletheia University, Tainan, Taiwan, ROC

Introduction

The outstanding bending rigidity and low specific weight are two key factors that composite laminates have been often used in aerospace vehicles that need high strength-to-weight ratio. Laminated composite is employed to replace traditional metal for the skin panels of aircraft wings and fuselage in order to reduce the weight of flight vehicles. In the design of composite skins for aircraft wings, one of the important issues is buckling of the panels. High-speed aircraft structural panels are subjected not only to aerodynamic loading, but also to aerodynamic heating. The temperature rise may buckle the plate and exhaust the load-carrying capacity. Many researches on the thermal buckling for the plates made of conventional isotropic material and advanced composite material were reported [1-3]. However, relatively few works have been devoted to the study of the buckling mode [4-5]. In the present study, a 36 degree-of-freedom high precision high order triangular plate element with shear deformation is developed for thermal buckling analysis of composite laminated plates. The thermal buckling mode shapes of cross-ply and angle-ply laminates under various E_1/E_2 ratios, aspect ratios and the fiber angles are studied in detail.

Formulations

Consider a rectangular composite laminated plate with length a, width b, and total thickness h. The total transverse displacement of the plate can be assumed to be the sum of the displacement due to bending of the plate w_b and that due to shear deformation of the plate w_s. The equation of motion may be obtained by applying Hamilton's principle. For simplicity and efficiency, the two displacement functions for the bending and shear deformation are assumed to have the same form. A 36 degree-of-freedom high precision high order triangular plate element with shear deformation is developed for thermal buckling analysis of composite laminated plates. After assembling the element matrices and vectors and then applying suitable boundary conditions, the governing equation in matrix form for the thermal buckling analysis of the composite laminated plate is obtained as

$$\left([K]-\lambda\left[K_N^{\Delta T}\right]\right)\{Q\}=\{0\} \tag{1}$$

where $[K]$ is the stiffness matrix; $\left[K_N^{\Delta T}\right]$ is the geometrical stiffness matrix due to thermal effect; $\{Q\}$ is the global nodal degree of freedoms vector for the assembled structure. The thermal buckling analysis has been represented as the standard eigenvalue problem. The critical buckling temperature ΔT_{cr} may be found from the product of the lowest eigenvalue λ and the initial guessed value ΔT.

Table 1 Material properties

Property	AS4/3501-6	T300/5208
$E_1(Gpa)$	142	181
$E_2(Gpa)$	10.3	10.3
$G_{12}(Gpa)$	7.2	7.17
υ_{12}	0.27	0.28
α_1 ($10^{-6}/^{o}C$)	-0.9	0.02
α_2 ($10^{-6}/^{o}C$)	27	22.5

Numerical Results

Fig. 1 displays a map in E_1/E_2 and a/b space, which identifies the regions that various buckling modes of a rectangular $[(0/90)_2]_s$, $[(90/0)_2]_s$, $[0_2/90_2]_s$, $[90_2/0_2]_s$ plates may occur. The buckling mode will change from mode (1,1) to mode (n,1) where numbers in parenthesis are the numbers of half-waves in the x- and y- directions for a rectangular plate where n increased with the E_1/E_2 and a/b ratio. It is seen that the buckling mode will still be mode (1,1) if the E_1/E_2 ratio is less than 9, 5 and 3.8 for the $[(0/90)_2]_s$, $[(90/0)_2]_s$ and $[90_2/0_2]_s$ plates for very high a/b ratio. The thermal buckling mode shape is always mode (1,1) for $[0_2/90_2]_s$ plate. The E_1/E_2 ratios of AS4/3501-6 and T300/5208 laminates are 13.79 and 17.57. However, the α_2/α_1 ratios of the above two laminates are -30 and 1125. The high E_1/E_2 ratio and high α_2/α_1 ratio used in the present study will produce higher bending rigidity along the fiber direction and higher in-plane compressive force in a direction perpendicular to the fiber direction. The behaviors of the two laminates are also shown in Fig. 1. The aspect ratio that the buckling mode shift to mode (n,1) is lower for AS4/3501-6 laminate due to its bending stiffness is lower.

Fig. 2 displays a map in θ and a/b space, which identifies the regions that various buckling modes of a rectangular $[(\pm\theta)_2]_s$ plate may occur. Once the fiber angle is less than 45°, the plate will have higher bending rigidity in the *x*-direction than in the *y*-direction but will be subjected to lower in-plane compressive force in the *x*-direction than in the *y*-direction. The bending rigidity and in-plane compressive force do not change with the length or width of the plate but the bending stiffness of the plate does, which is an important factor in the determination of the type of the buckling mode that the plate may have. For instance, for a $[(\pm\theta)_2]_s$ plate with a/b = 1 and θ = 0°, the critical buckling mode is mode (1,2) due to higher bending stiffness and lower in-plane compressive force in the *x*-direction and vice versa in the *y*-direction. If the square plate is changed to a rectangular plate with a/b = 2, the bending stiffness of the plate in the *x*-direction will be reduced and the difference in bending stiffnesses in the *x*- and *y*-direction is also reduced, thus the buckling mode will change from mode (1,2) to mode (1,1). On the contrary, for a $[(\pm\theta)_2]_s$ plate with θ = 90°, the difference in bending stiffnesses in the *x*- and *y*-direction will be enlarged with the aspect ratio. Hence, the buckling mode will change from mode (2,1) for a square plate to mode (*n*,1) for a rectangular plate where *n* increased with the aspect ratio. The behaviors of AS4/3501-6 and T300/5208 laminates are similar. The E_1/E_2 ratio of AS4/3501-6 is less than the one of T300/5208.The fiber angle that the buckling mode shift to mode (n,1) is lower for AS4/3501-6 laminates due to its bending stiffness is lower.

Conclusion

A triangular plate element with shear deformation is developed for thermal buckling analysis of composite laminated plates. The mode shapes of cross-ply laminates under various E_1/E_2 ratios and aspect ratios are presented. The mode shapes of angle-ply laminates under various aspect ratios and the fiber angles are also presented. The high E_1/E_2 ratio and high α_2/α_1 ratio of AS4/3501-6 and T300/5208 laminates produce higher bending rigidity along the fiber direction and higher in-plane compressive force in a direction perpendicular to the fiber direction. Therefore, the higher thermal buckling mode shapes are formed.

References

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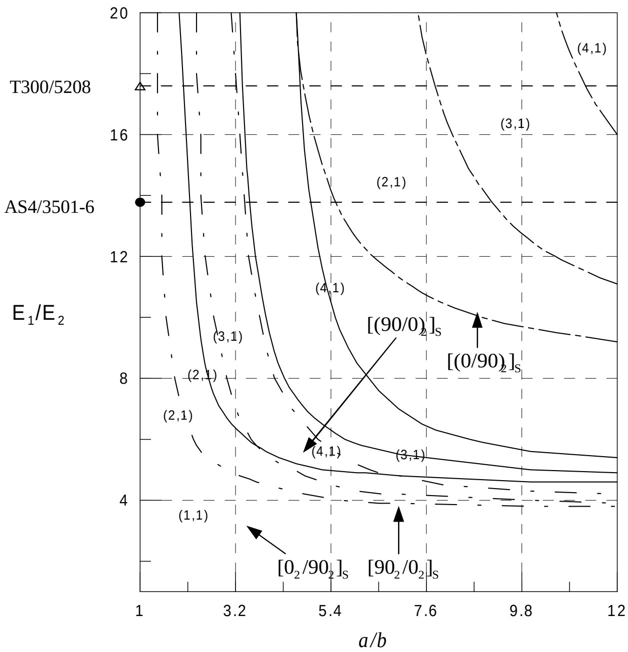

Fig.1 Boundary of thermal buckling mode for simply supported cross-ply laminates.

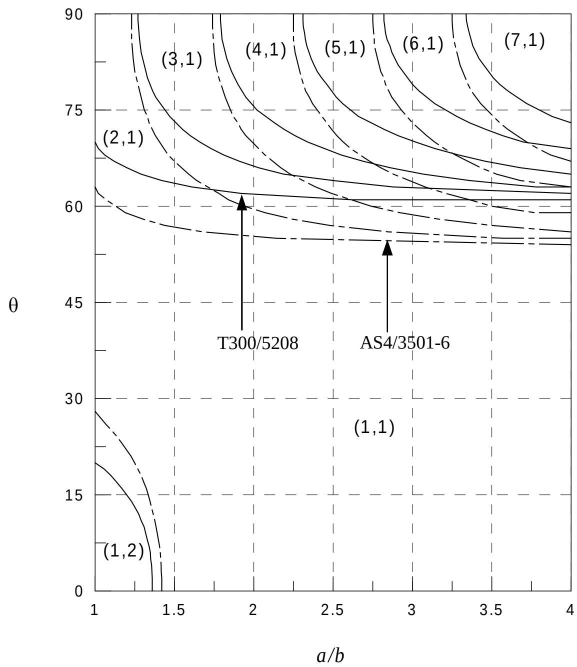

Fig. 2 Boundary of thermal buckling mode for simply supported $[(\pm\theta)_2]_s$ angle-ply laminates.