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AN ENGINEERING ANALYSIS METHOD FOR POST-BUCKLING OF COMPOSITE STIFFENED PANELS

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Abstract

The paper presents a project simplified theory and a method of calculating post-buckling loading capacity of composite stiffened panels based on the basic mechanics characters of composite materials, combining FEM theory and engineering experience. Composite J-stiffened panels are taken as an example to illustrate thinking and computational formula of post-buckling analysis from the perspective of engineering. The validity and effectiveness of the method are demonstrated with the experiment. It is shown the analysis method of stability loading capability can solve the geometric non-linear problem of composite stiffened panel structure load capability and keep high accuracy in the aero-structure design.

1 Introduction

Composite materials can be used by combining stiff panel and fibers embedded a matrix. The fibers carry load, whereas the matrix provides protection and support the fibers and allows picking the load among adjacent fibers. Composite stiffened panels are widely used in aircraft structure such as aircraft wing, fuselages, tails and helicopter skin because of their high efficiency in terms of stiffness strength the skin and to get good weight ratios. Generally, composite stiffened panels are subjected to buckling under compression loads.

The use of stiffened panels made by composite materials, which offer considerable high strength-to-weight and stiffness-to-weight ratios, it can bring a further substantial reduction in structural weight. However, until recently the use of composite materials appears largely limited to panels and to sandwiches structures designed to work only in the buckling field, due to the complexity of the post buckling phenomenon. In any case, the design of stiffened composite panels, able to overcome the buckling load and to work in the post-buckling field, represents one of the major challenges for the aircraft design. Nowadays, a deep knowledge of structure stability analysis allows composite stiffened panels able to take considerable loads in the post-buckling field during their life, in that way reducing significant the weight of the structures.

More recently, with the advent of new composite materials, offering very high stiffness and strength to weight ratios, more and more effort is devoted to replace the classical aluminum stiffened panels with entirely built in composite material. Since composite laminates are anisotropic materials obtained by stacking together piles of prepreg with fibers oriented differently, the post-buckling loading capacity analysis seems more complex than that of aluminum alloy.

For simplified structure analysis, project simplified method[2] which concludes the simple theoretical calculation formula and experience correction coefficient according to the experimental data is often used compared to fully theoretical analysis. It is simple and rapid to estimate the stability and loading capacity of stiffened panels especially for structure design. However, as seen from the literature[3-8], a considerable amount of work had been concentrated on the linear or nonlinear of buckling analysis in finite element software such as NASTRAN, ABAQUS and MARC. Jeff[9] conducted a non-linear analysis to solve for the geometric non-linearity of the post-buckling structure. A non-linear static analysis in MSC.Nastran is used to predict the buckling load precisely and follow the deformation path of the structure up to its ultimate loads. Bisgani and Lanzi[10] performed the post-buckling analysis of composite panels using dynamical analysis approach. They compared the RIKS method and explicit method of ABAQUS and also conducted structural optimization. Baranski and Biggers[11] analyzed the post-buckling response of composite shell structures by implementing the progressive damage of the structure. ABAQUS and the modified Hashin's failure criterion were applied. York and Williams[12] proposed an approximate method to predict the buckling load stiffened structures and compared the efficiency of the suggested module with the exact method. Se-Hee Oh[13] used a commercial code ABAQUS to suggest efficient post-buckling analysis techniques for the composite stiffened shell structure. To predict the failure load of a structure, various progressive failure analysis schemes were implemented in the post-buckling analysis. To reduce the computational time, accelerated of the accelerating module was proposed and the efficiency of the accelerating module was compared with the original analysis module. Giavotto[14] develops a general theory for isotropic shells of arbitrary shape and load conditions, including also the effect of temperature distribution. A single variational principle, that comprehends both equilibrium and compatibility equations, is derived. Stiffened isotropic panels are considered by Byklum and Amdahl[15], who develop a computational model for local post-buckling analysis. The work is formulated using large deflection plate theory and energy principles. The mode takes into account the plate-stiffeners interaction so that the stiffener instabilities can be captured. Analytical formulations of post-buckling loading capacity for engineering have not been developed.

Theoretical analysis of post-buckling behavior of stiffened panels is non-linear. The post-buckling range con be extended from the initial buckling load up to the ultimate load. The initial non-linearity is due to additional in plane strains and stresses caused by the out-of-plane displacements. Additional geometrical non-linearity may arise during large out-of-plane displacements and rotations in the post-buckling range.

The work here presented aims at predicting the failure loads of composite stiffened panels under compression. Carbon Fiber Reinforced Composite is anisotropic and the needs considering the post-buckling field suggest the use of geometric non-linear analysis. To overcome these difficulties a calculation method of post-buckling loading capacity of composite stiffened panels in engineering is developed. The principle and equations are presented and discussed in the following,.

2 Calculation principles

The primary or pre-buckling path is linear for an ideal structure not subjected to buckling. Here the structure experiences large deflections, but is still able to take considerable strength. The structure finally collapses at a limit point, which gives the ultimate strength.

Linear buckling analysis can be used to predict the buckling load and mode shape through eigenvalue solutions. If the structure contains a localized defect or damage, then local buckling exists, and it will couple with global buckling of the structure in the post-buckling range. In this case, linear buckling analysis can be inaccurate results since non-physical overlapping of delaminates is occurred. Furthermore, beyond buckling, the load-deflection curve is no longer linear due to the large out-of-plane displacements of the structure. The analysis is not able to characterize the post-buckling behavior of the panel, but can provide good estimations of the buckling loads and basic local buckling mode shapes. This is useful to estimate the appropriate load level of the structure. The curvature of the panel also has a considerable effect on the buckling and ultimate strengths. The curved panel is more likely to buckle as sine wave. Furthermore, the buckling stress for a curved panel in shear and compression is higher than the buckling stress for a flat panel with corresponding dimensions.

The classic buckling loads formula of composite laminated structures is given by[2]

$$N_{cr} = \frac{\pi^2 \sqrt{D_{11} D_{22}}}{b^2}\left[K - 2\left(1 - \frac{D_{12} + 2D_{66}}{\sqrt{D_{11} D_{22}}}\right)\right] \qquad (1)$$

Where b is the width of composite laminated structures; K is buckling coefficient; D_{ij} (i, j=1, 2, 6) are bending stiffness coefficients.

Failure loads N_f of a composite laminated structures is given by the following equation from theoretic analysis and experiment data, the basic consideration is that the metal stiffener panel failure physical model is used in the composite material one with relative modifications:

$$\begin{cases} N_f = N_{cr} \times \alpha \times \left(\dfrac{N_e}{N_{cr}}\right)^n & N_{cr} < N_f \\ N_{cr} = N_f & N_f < N_{cr} < N_e \\ N_{cr} = N_f = N_e & N_e < N_f < N_{cr} \end{cases} \qquad (2)$$

Where Ne is border loads, Ne=0.8Np; Np represents ultimate failure loads which are obtained by establishing Mindlin two-dimensional finite element model with 8-noded isoparametric element[16] and searching for the relationship of compression loads and strain under a series of continuous loads; Parameters α and n equal to 0.822 and 0.575 respectively according to reference[1].

The relationship of N_e, N_f and N_{cr} is shown in Fig. 1.

3 Analysis methods

Composite stiffened panels consist of skin and stiffeners. It is therefore taken as hypothesis, which is then described in the following way: (1) Composite stiffened panels should be undamaged without imperfection. (2) Stiffeners support skin simply. (3) Composite stiffened panels have similar damage mechanism to the metal stiffened panels.

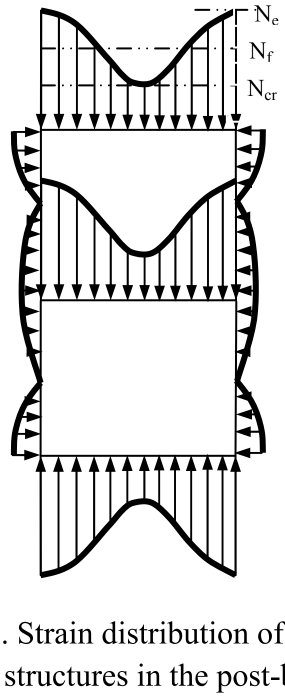


Fig.1. Strain distribution of composite laminated structures in the post-buckling load

Based on the above assumptions, the loading capacity of the skin and of the stiffeners is considered respectively.

3.1 The loading capacity of stiffeners

$\overline{N}_{crst}$ given by Eq. (1) is identified as buckling loads of stiffeners and $\overline{N}_{fst}$ regarded as failure loads of stiffeners is obtained by Eq. (2).

The average buckling stress of stiffeners results in

$$\overline{\sigma}_{crst} = \frac{\overline{N}_{crst}}{t_{st}} \tag{3}$$

Where t_{st} is the thickness of the panel's stiffener.

In the same way, the average failure stress of stiffeners results in

$$\overline{\sigma}_{fst} = \frac{\overline{N}_{fst}}{t_{st}} \tag{4}$$

3.2 The loading capacity of skin

The three different failure models [1] which are named the short column failure, the long column failure, the medium-long column failure are discussed on the basis of L/ρ(where L is the length of the panels; ρ is least radius of gyration) , as shown in Fig. 2. The medium-long column failure is the main failure model in aero structure including the composite material stiffened panels.

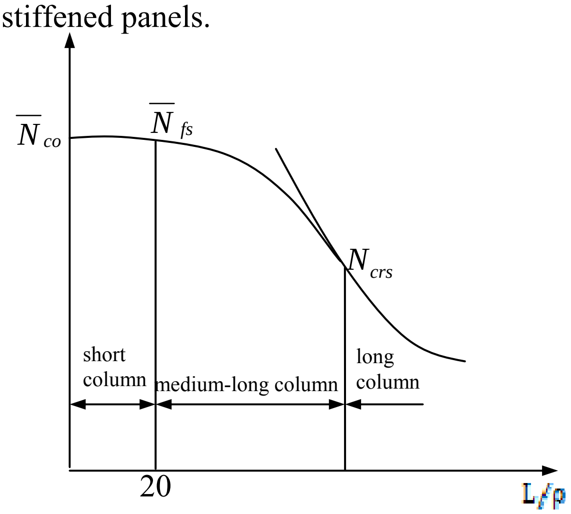


Fig.2. Failure models of skin

3.2.1 The buckling loads of skin

The following expression is adopted to define the buckling loads of skin:

$$N_{crs} = \frac{\pi^2\sqrt{D_{11}D_{22}}}{b_s^{\,2}}[K_c - 2(1 - \frac{D_{12}+2D_{66}}{\sqrt{D_{11}D_{22}}})] \tag{5}$$

Where N_{crs} is selected as the buckling loads of skin; b_s is the skin distances between two contiguous stiffeners; K_c is obtained in the curve of skin buckling coefficient[1].

3.2.2 The short column failure model loads of skin

$\overline{N}_{fs}$ is identified as the short column failure loads of skin based on Eq.(2), as shown in the following equation:

$$\begin{cases} \overline{N}_{fs} = \overline{N}_{crs} \times 0.822 \times (\frac{N_{es}}{\overline{N}_{crs}})^{0.575} \\ N_{es} = \max(0.8N_{ps}, 0.8\overline{\sigma}_{fst} \times t_s) \end{cases} \tag{6}$$

Where N_{es} represents skin failure ultimate loads and is derived by the Mindlin finite element method; $\overline{\sigma}_{fst}$ is the average failure stress of stiffeners(see Eq.(4)); t_s is the thickness of skin.

3.2.3 The long column failure model loads of skin

The long column failure model loads N_{el} are also called Euler stability loads and its calculation model equation becomes

$$N_{el} = \frac{\pi^2\overline{E}_{st}t_s}{(L/\rho)^2} \tag{7}$$

Where $\overline{E}_{st}$ is equivalent modulus of elasticity of stiffeners; ρ is least radius of gyration written in the form

$$\rho = \sqrt{I/A} \tag{8}$$

Where I is moment of inertia to centroidal axis of panels; A is the area of stiffened panel.

3.2.4 The medium-long column failure model loads of skin

Considering failure occurs in the medium-long region for the most of composite stiffened, therefore medium-long column failure loads $\overline{N}_{co}$

is derived in the following equation:

$$\begin{cases} \bar{N}_{co}=\bar{N}_{fs}=\bar{N}_{crs}=\dfrac{0.8N_{ps}+0.8\bar{\sigma}_{fst}\times t_s}{2} & \bar{N}_{crs}>\bar{N}_{fs}>\bar{N}_{co} \\ \bar{N}_{co}=\bar{N}_{crs}=\bar{N}_{fs} & \bar{N}_{crs}>\bar{N}_{es}>\bar{N}_{fs} \\ \dfrac{\bar{N}_{co}}{\bar{N}_{fs}}=1-(1-\dfrac{\bar{N}_{crs}}{\bar{N}_{fs}})\dfrac{\bar{N}_{crs}}{N_{el}}(\dfrac{N_{20}^{1/2}-N_{el}^{1/2}}{N_{20}^{1/2}-\bar{N}_{crs}^{1/2}})^2 & \bar{N}_{es}>\bar{N}_{fs}>\bar{N}_{crs} \end{cases} \quad (9)$$

Where N_{20} is reported in the flowing expression and

$$N_{20}=\frac{\pi^2 E_{st}t_s}{400} \quad (10)$$

Also, the buckling and the failure stress of skin are given by respectively

$$\begin{cases} \bar{\sigma}_{crs}=\dfrac{\bar{N}_{crs}}{t_s} \\ \bar{\sigma}_{co}=\dfrac{\bar{N}_{co}}{t_s} \end{cases} \quad (11)$$

3.3 The failure loads of composite stiffened panels

The aim of this paper is to predict the failure loads of composite stiffened panels. Consequently the expression of the failure loads P_{co} is taken equal to

$$P_{co}=\bar{\sigma}_{co}\times t_s\times L+\bar{\sigma}_{sft}\times n\times A_{st} \quad (12)$$

Where w represents the width of panels; n is the total numbers of stiffeners; A_{st} is the area of a stiffener.

3.4 The stiffeners design analysis

To investigate whether the stiffeners design is reasonable, some conditions are taken into account:

$$\begin{cases} \bar{\sigma}_{crst}>\bar{\sigma}_{crs} \\ \bar{\sigma}_{fst}>\bar{\sigma}_{co} \end{cases} \quad (13)$$

It means both the buckling and post-buckling of stiffeners occur later than that of skin. It is possible to use for the calculating failure loads of composite stiffened panels under compression loads in engineering.

4 Case study

As an example, a composite material T-stiffened panel is used for the post-buckling loading capacity analysis in engineering application which is shown in Fig. 3. The skin of the T-stiffened panels made of carbon fiber reinforced plastic(CFRP) prepreg has length of 308mm, width of 300mm and thickness of 2.76mm with 4 equally spaced stiffeners. Table 1 gives nominal geometrical data and material properties for material CCF300/QY8911 which was used throughout are reported in Table 2. Table 3 contains laminate set-up of the panels.

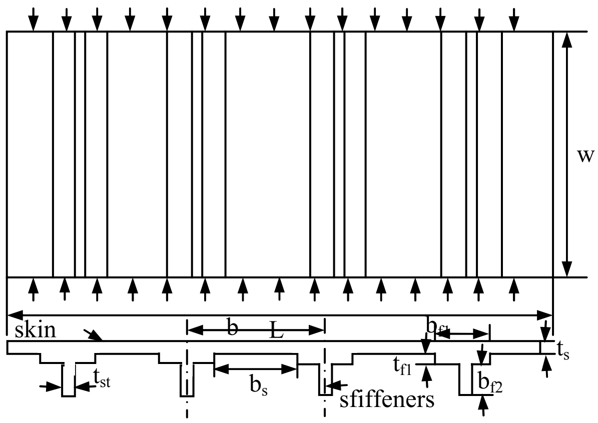


Fig. 3 Composite T-stiffened panels under compression

Table 1. Nominal geometrical data of composite T-stiffened panels

L/mm	w/mm	t_s/mm	b/mm	b_s/mm
308	300	2.76	77	34
b_{f1}/mm	t_{f1}/mm	b_{f2}/mm	t_{st}/mm	
40	1.8	18.2	2.76	

Table 2. Material properties of composite T-stiffened panels

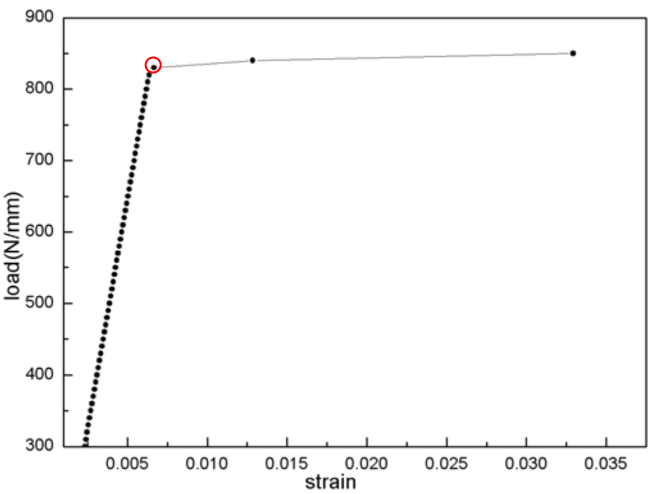
Material	Ply thickness mm	Modulus of elasticity E_{11}/GPa	E_{22}/GPa	G_{12}/GPa	ν_{12}
T700/BA9916	0.13	136.2	7.43	5.15	0.312
	Mechanical property/MPa X_T	X_C	Y_T	Y_C	S
T700/BA9916	2470	1062	57.4	225	89

Table 3. Laminate set-up of composite T-stiffened panels

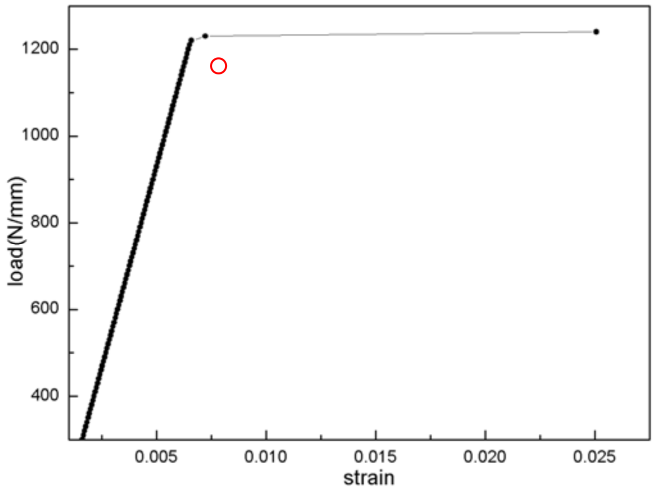
Part	Material	Thickness/mm	Number
Skin	T700/BA9916	2.76	23
Stiffeners	T700/BA9916	2.76	23
	Laminate set-up		
Skin	:45/0/∓45/0/∓45/90/±45/0/∓45/90/±45/0/±45/0/∓45]		
Stiffeners	[±45/-45/0$_2$/∓45/0/90/0/±45/0/∓45/0/90/0/-45/0$_2$/±45/0$_2$/∓45]		

4.1 Edge ultimate failure loads N_{es}

The finite element models of skin and stiffeners which investigates the edge ultimate failure load N_{es} Eq.(6) used in the post-buckling $\overline{N}_{fs}$ analysis, adopting the Mindlin theory are established respectively. In particular, non-linear finite element analysis can be carried out under a sequence of compression loads. Thus, the edge load vs. strain curves of skin and stiffeners are obtained by non-linear analysis, as shown in Fig. 4. It makes easy to evaluate the ultimate failure loads of skin and stiffeners where loads don't keep linear relationship with strain (see red circle in Fig.4 (a) (b)). Therefore, the values of the ultimate loads of skin and stiffeners equal to 830N/mm and 1230N/mm respectively.



(a) skin



(b) stiffeners

Fig. 4 The edge load vs. strain curves of skin and stiffeners

4.2 Numerical results

Numerical results obtained by deduced equations in engineering are presented in Table 4 and discussed in this paragraph, as compared with experiment results.

The experimental value of failure loads is 399KN: it turns out numerical results appear to be in good agreement with experiment results and this calculation method of post-buckling loading capacity in engineering can be used for accurate prediction of the failure loads in the future. On the other hand, the condition which both the buckling stress and the failure stress of stiffeners are larger than those of skin illustrates the stiffeners design is reasonable.

Table 4.Comparison between numerical results and experiment results

	$\overline{\sigma}_{crst}$ MPa	$\overline{\sigma}_{fst}$ MPa	$\overline{\sigma}_{crst}$ MPa	$\overline{\sigma}_{co}$ MPa	P_{co} KN
Numerical results	356.5	356.5	263	263	398
Experi-ment results					399

5 Conclusion

This paper illustrates the analysis theory and method for the post-buckling loading capacity in engineering. The post-buckling capacity analysis of composite T-stiffened panels performed is close to and in good agreement with the experimental results in terms of the failure loads. The results show that the use of this analysis methods appear to particularly profitable to predict the failure loads of composite stiffened panels.

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