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Effects of connective parts on stability of folded stringers stiffened composite panel

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ABSTRACT

In this paper, in order to select a better material between aluminum alloy and composite for the connective parts which is on a folded omega stringers stiffened composite panel under an axial compression, numerical investigation is conducted. The axial compression is divided into two stages which are buckling and post-buckling. By using software ABAQUS, 9 models were analyzed respectively. By comparing among the 9 models, firstly, composite joints have the highest collapse load with the same T stringer or without T stringer and next are aluminum alloy joints. Secondly, aluminum alloy T stringer has the highest buckling load and collapse load than composite one with the same joints or without joints and the composite T stringer the second. Thirdly, the joints and the T stringer are compulsory, because either the joints or the T stringer or both of them can enhance buckling load and collapse load remarkably. An important question for future studies is that the failure mechanisms occurring at the connective parts including their surrounding areas and choosing reasonable failure criterion for the material of these sections have to be researched by numerical methods and experimental tests.

1. Introduction

Carbon fiber-reinforced polymers (CFRP) have been widely used in the aerospace engineering as a result of their high specific strength and stiffness amongst other properties. Large composite structures with complex shapes are usually manufactured by connective parts due to difficulties of integrated manufacture, cost and transportation. So the selection of felicitous connection is very important. Adhesive bonding not only results in a uniform stress distribution in joint structures but also allows the bonding of different materials and assembly of composite structures. Therefore, as Butler [2], Taylor and Owens [24] suggested bonded composite structures, which enable considerable reductions on conventional mechanical fastener and part counts, would be widely used in primary structures of next generation aircraft and other engineering structures. Cocuring is a kind of the adhesive methods. Since its adhesive has the same material properties as the resin of the composite adherend, the analysis and design of the cocured joints for composite structures become simple comparing to other joints which use additional adhesives by Kim et al. [12] analyzing. Also, its strength is higher than other joints which use additional adhesives. Therefore, cocured joints have been increasingly used in composite structures.

Nowadays, post-buckling is becoming an important way to further reduce weight and en- hance bearing capacity of the structures. Degenhardt et al. [8] proposed that in the aircraft industry, using post-buckling design can significantly reduce the cost by decrease of structural weight. It is required to design structures that can withstand higher loads even after they have buckled. By introducing post-buckling in a structure, the load bearing capability can be increased. Take the metallic flying machine as an example, the use of "post-buckling" structures that were designed to resist loads significantly higher than buckling loads had led to highly efficient structures which has been reported in much of literature. Present time, post-buckling analysis is becoming an important research field. Paulo et al. [20] observed that both factors on the ultimate load of integrally stiffened panels when subject to longitudinal compression have a high sensitivity by different shapes and magnitudes of the initial geometrical imperfections. Through the lateral buckling analysis of cold-formed thin walled beams subjected to pure bending moments, SudhirSastry et al. [22] drew a conclusion that considering the material and manufacturing costs, beams with rounded cross section are efficient in resisting the buckling loads. Yan et al. [26] proposed

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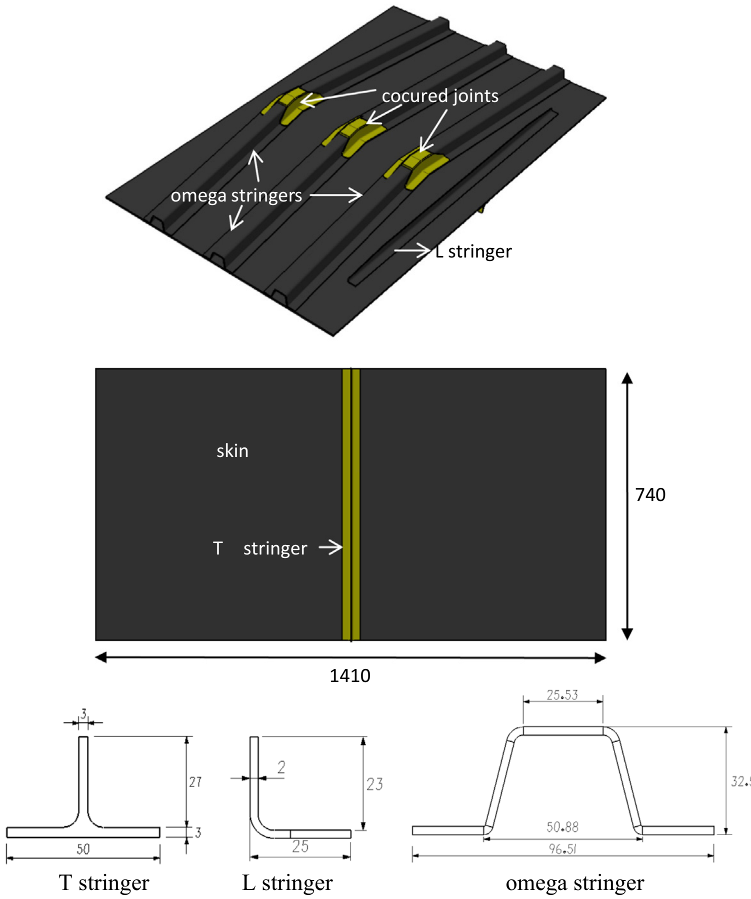

Fig. 1. Geometry of the stiffened composite panel.

an energy method, and with it, the buckling of the stiffeners in the press bend forming of aluminum alloy integral panels with high-stiffener can be predicted reasonably.

However, large numbers of experts and scholars including Orifici et al. [18] presented the application of post-buckling design to composite structures has been limited, as present analysis tools are not capable of accurately representing the damage mechanisms that lead to structural failure of composites in compression. At present many scholars resort to the finite element method, for example, SudhirSastry et al. [23], Huang et al. [11]. Loughlan and Hussain [15] using Nastran solver, Perret et al. [21] using ABAQUS solver, et al. all got satisfactory results for general engineering purposes. Krueger and Minguet [13] adopted a global–local approach and applied to composite stiffened panels by using the B-K criterion at the skin–stiffener interface and on 3D local models, and the results were in good agreement with results obtained from full solid models. Nevertheless, considering these aspects of analyzing and modeling independently each phenomenon, accuracy and efficiency, in the present study the post-buckling analysis and the debonding are separated. Bertolini et al. [3] in an post-buckling mode, did not analyze the post-buckling, but focused on the debonding phenomenon analysis, and the results was close to the test. Perret et al. [21] analyzed the global behavior of a composite stiffened panel in buckling, and did not to consider an adhesive layer at the skin–stiffener interface. Later, the numerical results were validated by the experimental results (Perret et al. [21]). Through experiments approach, some scholars validate that stiffened composite panels did not debond at the skin and stringers interface by reasonable connection. Kong et al. [14] reported that

Table 1
In-plane failure criteria and property reduction.

Failure type	Criterion	Property reduced
Fiber, tension	$\left(\frac{\sigma_{11}^2}{X_T^2}\right)^{\frac{1}{2}} \geqslant 1$	$E_{11},\ E_{22},\ \nu_{12}$ $G_{12},\ G_{23},\ G_{31}$
Fiber, compression	$\left(\frac{\sigma_{11}^2}{X_C^2}\right)^{\frac{1}{2}} \geqslant 1$	
Matrix, tension	$\left(\frac{\sigma_{22}^2}{Y_T^2}+\frac{\tau_{12}^2}{S_{12}^2}\right)^{\frac{1}{2}} \geqslant 1$	$E_{22},\ \nu_{12}$
Matrix, compression	$\left(\frac{\sigma_{22}}{Y_C}\left(\frac{Y_C^2}{4S_{23}^2}-1\right)+\frac{\sigma_{22}^2}{4S_{23}^2}+\frac{\sigma_{12}^2}{4S_{12}^2}\right)^{\frac{1}{2}} \geqslant 1$	
Fiber–matrix shear, Tension	$\left(\frac{\sigma_{12}^2}{S_{12}^2}\right)^{\frac{1}{2}} \geqslant 1$	$\nu_{12},\ G_{12},\ G_{31}$
Fiber–matrix shear, Compression	$\left(\frac{\sigma_{11}^2}{X_C^2}+\frac{\sigma_{12}^2}{S_{12}^2}\right)^{\frac{1}{2}} \geqslant 1$	

Table 2
Lay-up and thickness of some parts on the stiffened composite panel.

Lay-up and thickness	Value
Skin thickness	4.5 mm
Skin lay-up	$[\pm 45/0_2/45/0_2/-45/0/45/90/-45/0_2/45/90/-45/0]_S$
Omega stringer thickness	2.5 mm
Omega stringer lay-up	$[\pm 45/0_3/45/90/-45/0_2]_S$
L stringer thickness	2.5 mm
L stringer lay-up	$[\pm 45/0_3/45/90/-45/0_2]_S$
Thickness of single laminar	0.125 mm