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Effect of impact damage positions on the buckling and post-buckling behaviors of stiffened composite panel

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ABSTRACT

Effect of impact damage positions on the buckling and post-buckling behaviors of stiffened composite panels under axial compression were investigated in this paper. Barely visible impact damage (BVID) was introduced to three different positions on the smooth sides at impact energy 50 J. Impact crater depths and damage areas were measured and relationships between the two parameters were affirmed. Compression after impact (CAI) experiments were conducted both on the damaged and undamaged specimens to achieve the effect of impact damage on buckling and post-buckling behaviors. The results show that only local buckling in skin bay occurs without global buckling appearing for both damaged and undamaged specimens, which contains four half waves. The buckling load of damaged and undamaged specimens varies little. However, the failure loads of damaged specimens decrease to different extent according to their impact positions, with a maximum decrease of 10% compared to the undamaged specimens. Failure modes of the damaged and undamaged specimens are similar and complex, which contain the debonding of skin to stiffeners, breaking of stiffeners as well as tearing and splitting of skin.

1. Introduction

Composite materials are widely used in aircraft because they show higher strength/weight and stiffness/weight ratios, thus making the aircraft lighter and better performances [1–4]. They tend to be an excellent replacement for traditional metallic materials such as aluminum alloy [5]. Composite thin wall structures, stiffened composite panels, not only have the advantages of composite materials but also exhibit the outstanding properties of thin wall structure [1,6]. Thus the very type structures are fully applied in fuselages, tail planes and wings of aircrafts. One common load case acted on the stiffened composite panels is compressive load, which may easily lead to the buckling of the stiffened panels [7]. However, it is well known that the stiffened composite panel can bear more loads instead of collapse when compressive load exceeds the buckling load. The failure load or collapse load of some specimens can be 4 times larger than the buckling load or even more [6,8,9]. This ability of stiffened panels is called the post-buckling carrying ability, which can be potentially used for saving weight. Nowadays the post-buckling design has been used for metal stiffened panels. However, the post-buckling design is not fully used to design the stiffened composite panels due to the lack of confidence on verification of post-buckling and collapse behavior [10]. So experimental methods tend to be most reliable and convincible. There have been projects like POSICOSS (Improved Post-buckling Simulation for Design of Fiber Composite stiffened Fuselage Structures) [11] and COCOMAT (Improved Material Exploitation at Safe Design of Composite Airframe Structures by Accurate Simulation of Collapse) [12] which has studied the issues. Degenhardt [13] showed the design process and analysis of stringer-stiffened panels and cylinders in projects COCOMAT and POSICOSS. The proposed numerical results had been successfully validated with experimental data and degradation must be taken into consideration to simulate the deep post-buckling region. The authors also investigated the buckling behavior as well as damage initiation and evolution of pre-damage CFRP stringer stiffened panels under cyclic loading [14,15]. Riccio and Raimondo [16,17] proposed a novel FE model considering both inter-laminar and intra-laminar damage to study on the debonding growth in stiffened composite panels under compression. The FE models could reach a very good agreement to the literature experimental data comparing to the FE model only considering inter-laminar damage. The matrix failure initiates firstly then fiber failure follows with the compressive load increasing. Moreover, many researchers also pay much attention to the buckling/postbuckling and stiffeners

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debonding behaviors of stiffened composite panels with various type stiffeners both in experimental and numerical methods [18–21].

However, composite materials are sensitive to impact damage because of the fragile property, which is an obvious disadvantage for the application in aircrafts. During the service life of aircraft, impact damage may be easily induced by various situations, such as impact of small and large objects, like hailstones, runway debris or falling tools. Thus, the impact damage may cause the decrease of structure strength, which is harmful to the safety of aircrafts. Barely visible impact damage (BVID) [22] is one of the most common cases in impact issues. Thus the issue about composite structures with BVID is one of the most interesting concerns for aircraft designers. Some researchers have focused on this issue. Riccio [23] studied the impact behavior of omega stiffened composite panels numerically by means of non-linear explicit FE analysis under different impact energy (15 J and 20 J). A good agreement could be obtained between the FE results and experimental ones, which indicated the model was very precise. Moreover, a sensitivity analysis of the FE model with different configurations were conducted to evaluate the degree of accuracy with particular attention to the effect of inter-laminar and intra-laminar failure combination. Muhammad [24] investigated the low velocity impact properties of carbon nanofibers integrated carbon fiber/epoxy hybrid composites. The effect of various impact energies on the fracture surface was analyzed. Fardin [25] proposed an efficient approach to determine the CAI strength of quasi-isotropic composite laminates. The prediction obtained by the approach could reach a good agreement to the experimental results. Maio [26] simulated the delamination damage induced by low velocity impact in a laminated composite using progressive damage model, which could reach a good agreement with experimental results. Raimondo [27] built a progressive failure model of composite laminates subject to low-velocity impact damage using finite element methods. The model was mesh-size-independent and had a good prediction to the damage area and the force history of impactor. Wang [28] conducted the experiment on the post-buckling behavior of stiffened composite panels with impact damage. The corresponding FE models were established using non-linear analysis, by which the results were in good agreement with experimental results. Faggiani [29] built a FE model to predict the low-velocity damage on stiffened composite panels based on continuum damage mechanics. The inter-laminar failure was induced into the models. The results had accurate predictions compared to the experimental data. Suh [30] studied the compressive behavior of stitched stiffened panel with a clearly visible stiffener impact damage. The effects of stitching on the compression behavior were affirmed by comparing to the unstitched specimens.

As far as the authors know, the effect of impact damage positions on the buckling and post-buckling behaviors of stiffened composite panels under axial compression has been concerned little. So the issue was studied in this paper. BVID was introduced to the stiffened panels on three different positions. Then the compression experiments were conducted both on the stiffened panels with BVID and the undamaged ones to achieve the effect of impact damage.

2. Configurations of stiffened composite panel

Specimens are made of UD carbon fiber/epoxy resin BA9916-II/HF10A-3K prepreg with thickness 0.125 mm, and plain woven materials BA9916-II/HFW220TA with thickness 0.23 mm, which are both manufactured by Beijing Institute of Aeronautics Materials of China. All the material properties parameters are shown in Table 1. Ply sequence of the panels and stiffeners are shown in Table 2. In term of the materials, the 45* plies (surface ply of skin, shown in Table 2) are plain woven BA9916-II/HFW220TA materials and other plies are UD BA9916-II/HF10A-3K materials.

Table 1
Material property parameters of BA9916-II/HF10A-3K and BA9916-II/HFW220TA.

E_{11}/MPa	E_{22}/MPa	E_{33}/MPa	G_{12}/MPa
Material property parameters of BA9916-II/HF10A-3K			
124,000	10,000	10,000	4510
G_{13}/MPa	G_{23}/MPa	υ_{12}	υ_{13}
4510	3260	0.16	0.16
υ_{23}	X_t/MPa	Y_t/MPa	X_c/MPa
0.20	1448	55	1172
Y_c/MPa	S_{12}/MPa	S_{13}/MPa	S_{23}/MPa
172	90	161	161
E_{11}/MPa	E_{22}/MPa	E_{33}/MPa	G_{12}/MPa
Material property parameters of BA9916-II/HFW220TA			
55,000	52,000	56,000	4140
G_{13}/MPa	G_{23}/MPa	υ_{12}	υ_{13}
4140	3760	0.28	0.28
υ_{23}	X_t/MPa	Y_t/MPa	X_c/MPa
0.30	600	540	631
Y_c/MPa	S_{12}/MPa	S_{13}/MPa	S_{23}/MPa
584	60	60	116

Table 2
Ply sequence of composite stiffened panel.

Region	Ply sequence
Panel	[45*/45/0/0/0/-45/90/0/90]s
Stiffener	[0/45/-45/90/45/0/0/-45]s

Nominal dimensions of the stiffened composite panel are 860 mm (length) * 650 mm (width) shown in Fig. 1. The two ends are potted, leaving a gauge length 750 mm. Each stiffened composite panel consists four I-shape stiffeners with the interval 150 mm. Detailed dimensions of stiffeners are shown in Fig. 2.

3. Experiments description

3.1. Impact experiments

To achieve BVID, the low-velocity impact experiments are conducted using drop weight machine which has crosshead brakes to prevent the multiple impacts of impactor. The impactor has a mass of 6 kg with a hemispherical nose diameter of 8 mm. It is guided through a smooth column and impacts on the target positions. Impact experiment setups are shown in Fig. 3. The height of impactor can be adjusted in order to obtain various impact energies and velocities. The energy is calculated using the potential energy equation $E = mgh$, where 'E' is the impact energy, 'm' is the mass of the impactor, 'g' is the acceleration due to gravity, and 'h' is the drop height. The impact energy is set as 50 J to achieve BVID. Then the depths of impact craters are measured. Finally an

Fig. 1. Specimen.