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Investigation on impact damage evolution under fatigue load and shear-after-impact-fatigue (SAIF) behaviors of stiffened composite panels

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ABSTRACT

Experimental investigation on impact damage evolution under fatigue load and shear-after-impact-fatigue (SAIF) behaviors of stiffened composite panels were conducted in this paper. BVID was introduced into stiffened composite panels at a typical position. Damage areas, in-plane dimensions, dent depths were measured and relationships among the three parameters were determined by plane surface function. All fatigue specimens survived after 10^6 cycles at the given fatigue load level and the impact damage did not deteriorate or enlarge during fatigue loading. Then SAI (shear-after-impact) and SAIF experiments were conducted with a comparison of shear on virgin specimens. Buckling/failure modes of virgin, impact and impact-fatigue specimens were obtained. Buckling/failure load of impact specimens decreased by 17% and 16% compared to that of virgin specimens respectively. Additionally, fatigue load led a further 4.3% and 6.6% drop in buckling/failure load of impact-fatigue specimens compared to that of pure-impact specimens, although the impact damage area did not deteriorate or enlarge during and after the fatigue treatment.

1. Introduction

Composite materials, especially carbon fiber reinforced polymer, are widely used in aeronautic industry, which can make the aircraft lighter and better performances [1–3]. Composites are becoming more and more popular and tend to be an excellent replacement for traditional metallic materials because they exhibit higher strength to weight, stiffness to weight ratios, and good corrosion/fatigue resistance [4,5]. For example, only 12% of the total weight of Boeing 777 airplane is composite materials while the corresponding data increase to 52% for Boeing 787 airplane [5]. Stiffened composite panels, as typical thin-walled structures, are widely used in the structures of airplane, such as fuselages, horizontal/vertical tails, wings [6]. A famous conclusion about stiffened composite panels is that this type of structure can bear more loads instead of fracture when the exterior applied loads exceed the critical buckling load. Thus stiffened composite panels were considered as having post-buckling carrying ability. So buckling/post-buckling behaviors of stiffened composite panels are very important and attractive for the designers of aircraft. Many researches [7–10] and some famous projects such as COCOMAT [11] (Improved Material Exploitation at Safe Design of Composite Airframe Structures by Accurate Simulation of Collapse) and POSICOSS [12] (Improved Post-buckling Simulation for Design of Fiber Composite stiffened Fuselage Structures) have focused on these issues. The stiffened composite panels applied in airplane undergo different load cases during the service life as axial compression [13], in-plane shear [14] and other complex compression-shear coupling conditions [15]. Thus, in-plane shear load is one of the most common load cases for the stiffened composite panels in airplane structures. Lei [16] studied the buckling behaviors and failure modes of I-shaped stiffened composite panels under in-plane shear using digital fringe projection profilometry. The morphology of a discontinuous surface of stiffened composite panel was measured and monitored. The experimental results were also verified by the finite element analysis. Upendra [17] presented a parametric study on simply supported stiffened composite panels with blade stiffeners, which were subjected to in-plane shear loading. Effect of panel orthotropy ratio, stiffener depth, number of stiffener and stiffness ratio was investigated using finite element models. Upendra [18] also predicted buckling load of composite stiffened panels under in-plane shear load by artificial neural networks. A computationally efficient analysis tool was developed based on the finite element analysis, which can optimize buckling behaviors by training the key parameters.

Unfortunately, composites and composite structures are very susceptive to accidental low velocity impacts such as failing tools,

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runway debris and hailstones, which is a major limit for the application in airplane industry [19]. Barely visible impact damage (BVID) is one of the most common cases among the low velocity impact damage. One of the key issues for BVID is that delamination is mainly triggered by transverse crackling first. Thus Violeau [20,21] developed a calculation strategy for the simulation of a complete microscopic model, which could simulate complex problems with multiple cracks of composites. Ladevèze [22,23] built a bridge between the micro/mesomechanics of delamination for composites and developed the corresponding computational strategy. Previous work dealt mainly with the in-plane behaviors, the authors originally introduced extra out-of-plane behaviors, which was essential for the simulation of delamination. BVID in airplanes may lead to great decrease in the strength of composites and composite structures, which is harmful to the safety of airplanes. So it is important to learn the effect of BVID on the strength of aero stiffened composite panels. Li [24] investigated the effect of low velocity edge impact damage on the damage tolerance of wing relevant composite panels stiffened with both T-shaped and I-shaped stiffeners under compressive load case. Different types of damage and shapes were discovered by visual and ultrasonic C-scan. The experimental results demonstrated that local buckling, subsequent damage propagation and final fracture of the edge impacted stiffener were the triggers of the final failure of a stiffened composite panel. Feng [25] found the strength of stiffened composite panels decreased by different degrees according to the impact positions under axial compression. The negative influence of position B (corresponding point of smooth side to the edge of stiffeners bottom flange) is most serious.

It is well known that the service life of both civil and military airplanes can be as long as 25 years or even more. So fatigue performances of the composites and composite structures applied in airplane structures are also a major concern for the designers and manufacturers of airplane. As far as the authors known, most current researches on the stiffened composite panels with pre-impact damage focus on axial compressive load conditions. Research about the stiffened composite panels with pre-impact damage under in-plane shear load is limited. Particularly, the research about the effect of fatigue load on the stiffened composite panels with pre-impact damage under in-plane shear load is more limited. So this issue was investigated in this paper. Firstly, BVID was introduced into the stiffened composite panels. Secondly the fatigue experiments were conducted on the pre-damage stiffened composite panels. Finally, shear experiments were conducted on the survival stiffened composite panels, with comparisons of shear experiments on virgin specimens and pure-impact specimens.

2. Configurations of stiffened composite panel

The stiffened composite panel in this paper is applied in vertical tail of one typical airplane, which was made of two types of composite materials. They are unidirectional carbon fiber/epoxy resin BA9916-II/HF10A-3K prepreg with thickness of 0.125 mm, and plain woven BA9916-II/HFW220TA with thickness of 0.23 mm. The thickness of panels and stiffeners are 2.46 mm and 2 mm respectively. All the specimens were manufactured by Beijing Institute of Aeronautics Materials of China. Material property parameters and ply sequences of specimen are listed in Tables 1 and 2 respectively. The surface plies of panels (the plies with symbol $*$) are plane woven BA9916-II/HFW220TA materials and other plies are UD BA9916-II/HF10A-3K materials.

Each specimen consists of four I-shaped stiffeners, which are equally spaced of 153 mm (Fig. 1). Geometric dimensions of specimen are 610 mm $*$ 610 mm. There are extra 90 mm length extensions in four sides, which have been reinforced to be as clamp parts.

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

Material property parameters of BA9916-II/HF10A-3K			
E_{11}/MPa	E_{22}/MPa	G_{12}/MPa	υ_{12}
124,000	10,000	4510	0.16
S_{12}/MPa	X^C/MPa	Y_c/MPa	
90	1172	172	
Material property parameters of BA9916-II/HFW220TA			
E_{11}/MPa	E_{22}/MPa	G_{12}/MPa	υ_{12}
55,000	52,000	4140	0.28
S_{12}/MPa	X^C/MPa	Y_c/MPa	
90	631	584	

Table 2
Ply sequences of specimens.

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

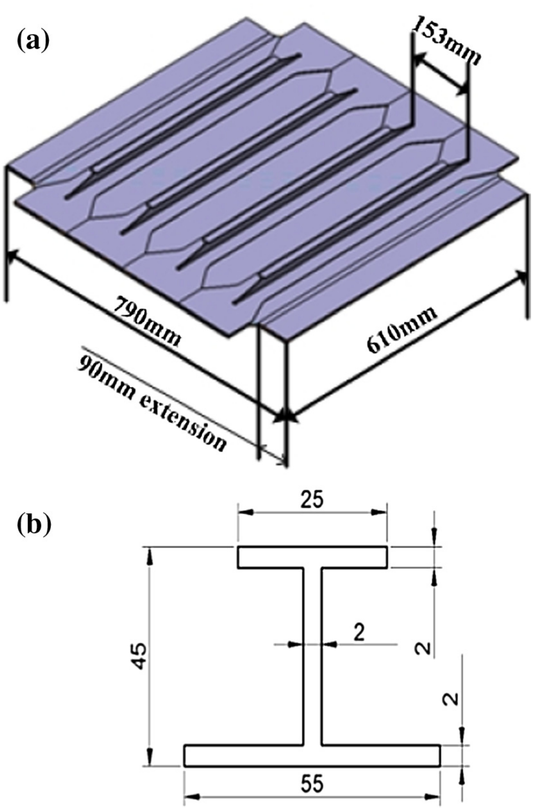

Fig. 1. Specimen and dimensions: (a) specimen; (b) detailed dimensions of stiffener.

3. Experiments

Total 9 specimens were involved in the experiments, which could be equally divided into three groups (labeled as group V, I and IF respectively). The three groups of specimens had different experimental procedures.

For group V, in-plane shear experiments were conducted on the three virgin specimens. For group I, firstly BVID was introduced into the specimens. Secondly SAI experiments were conducted on the impact damage specimens. For group IF, firstly BVID was introduced into the specimens. Secondly, fatigue experiments were conducted on the impact damage specimens. Thirdly, SAIF