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Experimental study on delamination growth of stiffened composite panels in compression after impact

Tian Ouyang[a], Wei Sun[b], Zhidong Guan[a,*], Riming Tan[a], Zengshan Li[a]

[a] *School of Aeronautic Science and Engineering, Beihang University, Beijing 100191, China*
[b] *Unmanned Aerial Vehicle Technology Institute the Third Academy of CASIC, Beijing 100074, China*

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ABSTRACT

The purpose of this paper was to investigate the failure behaviors of composite stiffened panels with skin low-velocity impact (LVI) damage under axial compression. Barely visible impact damage (BVID) was introduced into the L-shaped and T-shaped stiffened panels within the skin between stiffeners. Compression after impact (CAI) tests were performed and delamination growth of specimens was observed directly. The results show that there are two types of delamination growth processes, which are unstable and stable respectively. Due to the restriction effect of the stiffener, the delamination growth is easier to initiate in the outer sublaminates on the smooth side of the skin compared to the ones on the stiffener side. In addition, according to the thermal-deply result, adjacent impact delaminations can be connected by matrix cracks, which results in a larger area of the sublaminate. It is revealed that buckling of the sublaminates and subsequent transverse delamination propagation in large area are the triggers leading to the final failure of the stiffened panel.

1. Introduction

Carbon fiber reinforced composites are widely used in the fields of aeronautics and astronautics due to their excellent strength/weight and stiffness/weight ratio. In aviation industry, thin walled composite structures like stiffened panels are extensively applied in many parts of the aircraft, such as fuselage, wings and horizontal/vertical tails. However, composites are vulnerable to impact damage because of their low out-of-plane strength. LVI by foreign objects during the maintenance and service may cause internal damage even though there is no obvious damage or only barely visible damage on the surface of the composite structures [1,2]. This internal damage can result in significant reduction of the residual compression strength and stiffness [3,4], and thus poses a serious threat to the flight safety.

Numerous studies have been conducted on LVI and CAI of composite materials. Experimental and numerical researches on the permanent dent and delamination behaviors of composite laminates subjected to LVI were performed in Refs. [5–10]. Compression behaviors of composite laminates after impact with the consideration of impact-induced cracks, delamination, global/local buckling and ply failure were analyzed in Refs. [11–14]. As for composite stiffened panels, LVI response and CAI behaviors will be different according to the impact sites due to the existence of stiffeners. Faggiani et al. [15] numerically predicted LVI damage in the skin bay of a stiffened panel with a detailed finite model based on continuum damage mechanics. Greenhalgh et al. [16] investigated the compression behavior of a stiffened panel with skin impact damage. The post failure SEM results indicated that delamination growth caused the final failure of the specimens. In terms of impact on the skin over the stiffener, a numerical-experimental study on an omega-stiffened composite panel subjected to LVI was presented by Riccio et al. [17]. Sun et al. [18,19] studied the LVI damage and CAI behaviors of T-shaped stiffened panels through experimental and numerical methods. They found that stiffener damage occurred first with the skin dent lower than BVID and the damaged stiffener had a significant influence on the buckling and failure of the stiffened panel. To analyze the effect of flange edge impact, compression after impact and fatigue tests were carried out by Feng et al. [20]. According to the results, impact damage and the following fatigue load had no obvious influence on the buckling load and modes. Wang et al. [21] also studied the compression behavior of stiffened panels subjected to flange impact. Experimental and numerical results suggested that multiple delaminations and unstable buckling took place at the damage site firstly and propagated transversely. Masood et al. [22] focused on the influence of stiffener configuration on the post-buckled response and collapse of composite panels with flange impact damage. The experiment results showed that stiffener configurations had a significant effect on the axial stiffness and failure load, while the overall behavior of the panel was not changed by the impact damage. Besides,

* Corresponding author.
E-mail address: d5062010@163.com (Z. Guan).

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impact on the free edge of a stiffener as a result of falling tools during maintenance was also paid attention to by many scholars. Ostre et al. [23] investigated the edge impact damage on a T-stiffener. The mechanical response and damage types of the edge impact were well elaborated by their proposed numerical model. Li and Chen [24–26] studied the effect of edge impact on stiffened panels both experimentally and numerically. Through the CAI tests, they found that damage propagated from the impacted stiffener, and final fracture of the impacted stiffener determined the ultimate load of the stiffened panel.

Taking into account the composition of a stiffened panel, the skin bay area is relatively larger than stiffener area, which leads to a greater probability of the skin bay being impacted. As a result of skin bay impact, the damage induced in is mainly delamination. However, according to the authors' knowledge, although many scholars have conducted studies on delamination growth of composite structures under compression [27–32], the damage propagation process captured in the experiment is generally caused by prefabricated delamination and impact delamination growth process is seldom directly observed. A detailed investigation on impact delamination growth process is necessary so that people can have a better understanding on the compression failure mechanism of the stiffened panel with skin bay impact damage. Thus, in this paper, compression behaviors of L-shaped and T-shaped stiffened composite panels with skin bay impact damage were studied. By analyzing the post-impact delamination distribution of the skin and observing the delamination growth process under compression, failure mechanism of the impacted stiffened panels was discussed.

2. Experiments

2.1. Specimens

As shown in Fig. 1, two types of specimens were tested and both of them are flat panels made of carbon fiber reinforced composite stiffened by two stringers with either L shape or T shape. The length and width of the L-shaped stiffened panel are 230 mm and 188.5 mm respectively, and the skin thickness is 4.2 mm. Besides, the spacing between the stiffeners is 115.5 mm. For T-shaped stiffened panels, the above mentioned parameters are 230 mm, 150 mm, 3 mm respectively and the distance between stiffeners is 77 mm. In order to uniformly transfer the compression load and avoid crush of the load ends, both ends of the specimens are reinforced by epoxy resin blocks with steel frame on the periphery. Regardless of the reinforced blocks, the effective lengths of both types of specimens are 180 mm.

Fig. 1a shows the detail dimensions of the L stiffener. The thickness of the L stiffener is 2.64 mm at the flange and gradually increases to 3.96 mm at the web by inserting 11 plies from the rounded corner. On the left side of the web, there is a 0.48 mm thick L shape component supporting the stiffener. Fig. 1b exhibits the detail dimensions of the T stiffener. It consists of two 1.56 mm thick L stiffeners and a transition layer with the thickness of 0.48 mm under the flange. All the specimens were made of T300/QY8911 unidirectional prepreg and material parameters are shown in Table 1.

Table 1
Material parameters of the T300/QY8911 prepreg.

T300/QY8911 prepreg		
E_1	Longitudinal Young's modulus	135 GPa
E_2	Transverse Young's modulus	8.8 GPa
G_{12}	In-plane shear modulus	4.47 GPa
ν_{12}	Poisson's ratio	0.33
X_t	Longitudinal tensile strength	1548 MPa
X_c	Longitudinal compressive strength	1226 MPa
Y_t	Transverse tensile strength	55.5 MPa
Y_c	Transverse compressive strength	110.5 MPa
S	In-plane shear strength	89.9 MPa
t	Ply thickness	0.12 mm

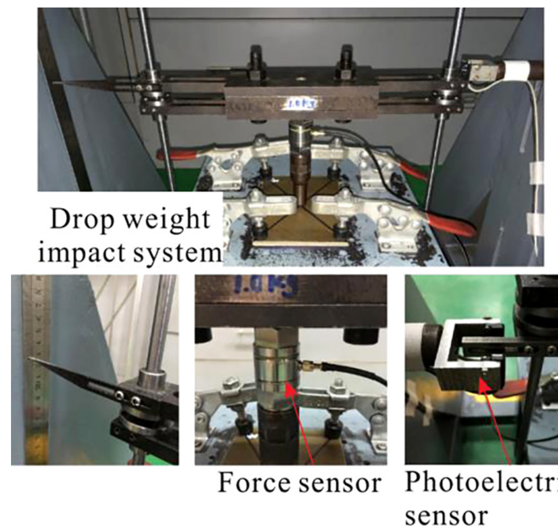

Fig. 2. Drop weight impact system.

Altogether 3 L-shaped stiffened panels (labeled as L1–L3) and 3 T-shaped stiffened panels (labeled as T1–T3) were involved in the experiments. Stacking sequences of the specimens are presented as follow:

L-stiffened panel: $[45/-45/0/-45/0/45/0/45/0/45/90/45/-45/0/45/-45/90/\bar{0}]_s$ for the skin, $[45/-45/0/-45/45/0/-45/0/45/0/90]_s$ for the stiffener flange, $[45/-45/0/-45/45/0/45/0/-45/0/-45/0/45/0/90/0/\bar{0}]_s$ for the stiffener web, and $[45/-45/-45/45]$ for the L shape component next to the stiffener.

T-stiffened panel: $[45/-45/0/-45/45/0/-45/45/90/45/-45/45/0]_s$ for the skin, $[45/-45/0/0/-45/45/0/0/90/0/0/45/-45]$ for the stiffener flange and web, and $[45/-45/0/0]$ for the transition layer under the flange.

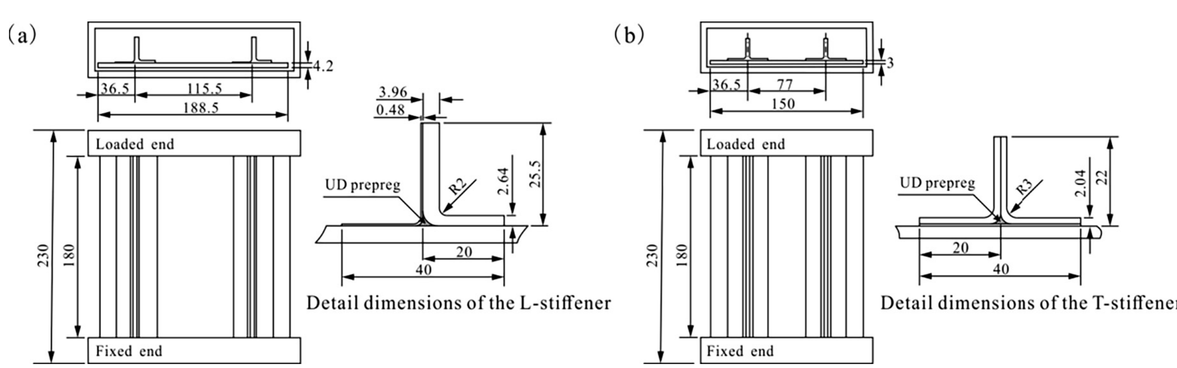

Fig. 1. Configuration of the composite stiffened panel specimens (Unit: mm).