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Experimental and numerical analysis for the post-buckling behavior of stiffened composite panels with impact damage

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ABSTRACT

The paper deals with post-buckling experimental investigation on stiffened composite panels with pre-damage. Two panels with four equally spaced stiffeners were manufactured and tested. The damage was implemented at the vulnerable stiffener edge. Attempts were made to obtain the buckling load, ultimate carrying capacity and other wanted structure properties. A numerical methodology for the post-buckling behavior analysis of pre-damaged panels was also presented. The load-shortening curves of numerical results were compared with that of experimental results, and it gave a good prediction up to the onset of buckling and the collapse load. Experimental strain analysis indicated that the outer sublaminate at the damage location was buckling before the global buckling occurred, and the post-failure views also showed that the multiple delaminations and the unstable buckling took place at the damage site firstly and transversely propagated. It was validated by the simulated damage patterns which showed that the compression matrix failure and the shear failure in the impact site led to the collapse of the panel.

1. Introduction

Composite materials have found increasing uses in aerospace industry due to their combination of excellent mechanical performances with light weight. Thin-walled composite panels with a certain number of shaped stiffeners are usually used in fuselages, tail planes and wings of the aircraft. In practice, the composite panels are often subjected to axial compression which would easily results in buckling of the panels. It is acknowledged that stiffened panels can have considerable post-buckling reserve strength, i.e., they can bear load much higher than the buckling load [1]. Considering the post-buckling behavior of the panels can improve the load carrying ability of the structures and achieve substantial weight savings. Therefore, it is important to study post-buckling properties. There have been project like POSICOSS (Improved Post-buckling Simulation for Design of Fiber Composite Stiffened Fuselage Structures) and COCOMAT (Improved Material Exploitation at Safe Design of Composite Airframe Structures by Accurate Simulation of Collapse) in Europe aiming to design fiber composite fuselage structures for post-buckling under ultimate load [2,3].

Many experimental studies have been performed on the post-buckling behavior of the stiffened composite panels under axial compression [1,4–9]. The buckling and collapse loads, and out-of plane deflection patterns were obtained for different type of stiffeners. Practically, composite are fragile and susceptible to damage from production process and in service which can never be totally avoided [10]. Any tiny damage could influences the behavior of the composite panels since they are mostly thin in the thickness direction. Therefore, it is important to study their behavior when the damage is presented in the structures. Some published literatures have presented the work about post-buckling behavior of the panels with pre-existing defects or pre-damage [9–13]. These pre-imperfect greatly affect the post-bulking behavior of the specimens.

Numerical analysis of composite panels post-buckling also has been carried out with the consideration of one or interaction of interlaminar crack growth, ply failure and skin-stringer debonding as a failure mode [5,10,14–19]. Among them, pre-damage was considered by replacing the adhesive of skin-stiffener by a centrally located full-width Teflon strip in reference [5,19], and by impact load caused free edge damage in reference [10]. The capability of analysis method in buckling and post-buckling of composite structures has been reviewed recently [20]. More work is needed to improve the numerical assessment of the post-buckling response

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of the composite structures, especially when the pre-damage is taken into the consideration.

In the present paper, both experiments and numerical analysis were performed into post-buckling of stiffened composite panel with pre-damage introduced by load impact. The pre-damaged panel was loaded in compression to collapse in the experiment. Numerical work was carried out with the nonlinear version finite element ABAQUS dynamic/explicit.

2. Experimental set up

The test specimens were flat panels made of carbon fiber reinforced composite and stiffened by four stringers with I-shaped cross section along the longitudinal direction, as shown in Fig. 1.

Two specimens were tested, which were numbered with panel A and B. The panels had a width of 600 mm and a length of about 870 mm. There are 60 mm lengths from each end at the longitudinal direction designed as fix and clamp section which are followed by 100 mm length of load transition sections. Two frames were connected to the skin and the stringers by means of clips in the section A-A. Therefore, the gauge length was 550 mm. The four stringers were equally spaced with the distance of 150 mm between each. The panels were made of CCF300 composite. The skin consisted of an 18-ply laminate with a stacking sequence of $[45_2/0_3/-45/90/0/90]_S$ for a total nominal thickness of 2.25 mm, while the stringers were composed of a 16-ply laminate with a lay-up consequence of $[0/45/-45/90/45/0_2/-45]_S$ for a total nominal thickness of 2.0 mm. The skin and the stringer are co-cured in an autoclave.

A vertical drop-weight impact machine was used to introduce the impact damage before the compression test, which can control the impact energy by adjusting the impactor mass and drop height. The panels were impacted underneath the stiffener edge with the energy of approximately 50 J to simulate the incident impact during the manufacturing and operating. The impact site was selected in the second stiffener edge. The impact damage was detected by ultrasonic technique.

The compression experiment has been conducted on a hydraulic test machine called FTS-5000 with a 5000 kN load sensor accurate at 0.5% on the test range. Fig. 2 shows the setting of the panel in the test. It intends to reproduce the CC-SS boundary conditions (clamped at bottom and at the top and simply-supported along the sides). The displacement of the panel was recorded as the movement of the machine's crosshead. Many uniaxial strain gauges bonded back-to-back were arranged at the surface of the skin and the stringers. The details of strain gauges are shown in Fig. 3. Before the experiment, the trial test would be carried out to make sure the panels were uniformly loaded. A quasi-static axial compression up to about 66% of the linear buckling load was performed at the beginning in order to settle non-linearity due to fiber misalignments.

3. Experimental results

For impact over the stiffener edge, a crater could be found in the face of panels. The damage conditions measured by the C-scan are shown in Fig. 4, i.e. 41.5×79 mm and 45.6×115 mm, respectively for the two panels. It is interesting that the shape of the damage areas looks like 'peanut' with the major axis in the longitudinal direction, which was also observed by Wiggenraad and Zhang [21].

Panel A was tested first. A weak noise was heard when load increased to 325.2 kN, but no visual damage can be observed. The cracking sound burst out again until the loading was rising up to 499.7 kN. Eventually, collapse occurred at load value of 512.1 kN accompanied by a very loud noise, along with the

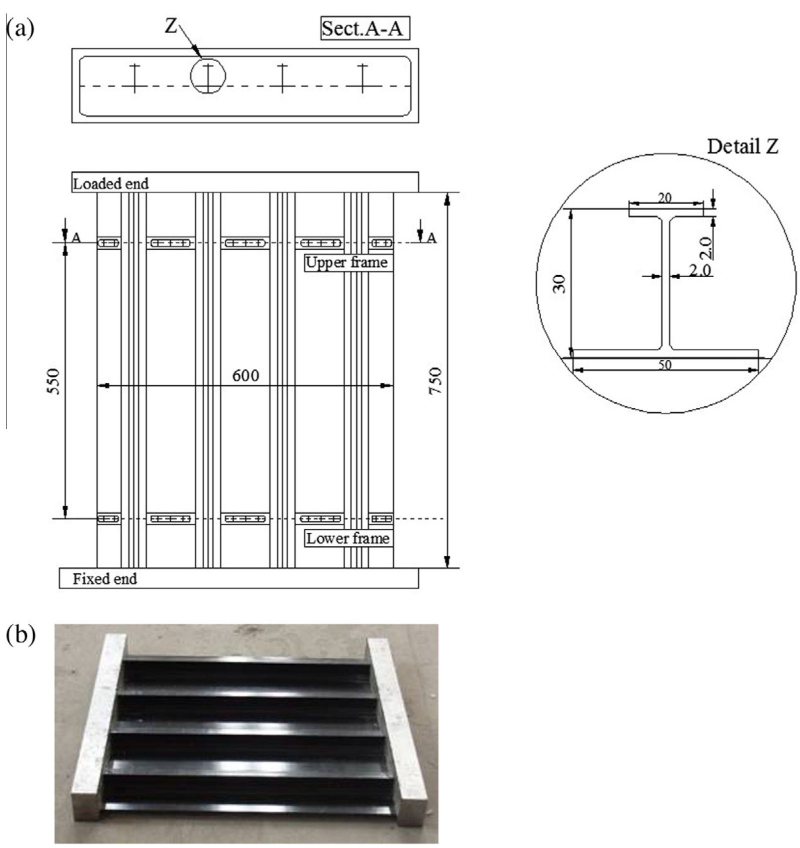

Fig. 1. Configuration of the stiffener specimen (all dimensions in mm).