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Analysis of Partial Fractures on Aircraft Composites Wing

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In the stability test of an aircraft composites wing, partial fractures are found in the rivet joint region of the wing skin panel. Besides visual examination, other experimental techniques used for investigation are: crack morphology and fracture characteristics by environmental scanning electron microscopy (ESEM), metallographic observation of cracks and composition analysis of fiber surface by x-ray fluorescence spectrometry (XFS). The results are obtained through the analyses of damage morphology, structure stress and load. Fracture areas of the panel, in which notch effect was formed around the rivet, fractured under alternate compressive load. The wing skin panel fractured at the rivets under compression load. For the inconsistent deformation in the compression process, the damage mode of local areas is shear fracture. The primary cause of the panel fracture is insufficient design strength. However, interface pollution leads to structural strength decline, which induces the fractures occurred.

1. Introduction

The use of composite materials for structural applications has many advantages, which include higher specific strength and stiffness, better durability, etc. Modern aircraft design incorporates carbon fiber composite materials into primary load-bearing structures such as the fuselage, wings, wing-box, and empennage (Burns, et al. 2010). When composite laminate structures bear the effects of compression, shear, torsion and bending load, the most common failure mode is the lose of bearing stability, also known as buckling (Chinese Institute of Aeronautics. 2002). In order to ensure use safety of the structure, stability analysis and corresponding strength check need to be conducted. In the stability test, multiple areas in the composite wing skin panels of an aircraft fractured which is shown in Figure 1 and Figure 2. Materials of the composite structure are carbon-fiber CCF300 and Bismaleimide resin QY8911, for which autoclave forming is used. The wing skin panels consist of skin, strengthen stringer and rivets, as shown in Figure 1 and Figure 2. Skin and stringer are strengthened by rivets after co-curing. In the paper, the fracture

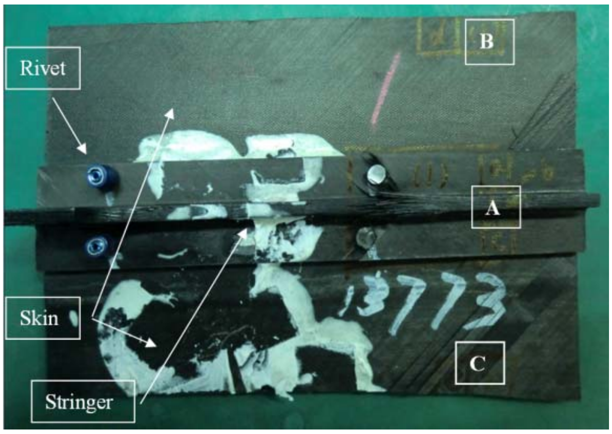

Figure 1: Partial fracture in the wing skin panel a

characteristics of wing skin panel are obtained by the observations and analyses of damage morphology. Combining with the analyses of the interface constituent, load and structural stress, causes of partial fracture of the wing skin panel are concluded.

Figure 2: Partial fracture in the wing skin panel b

2. Experimental procedure and results

2.1 Observation of macro-morphology

Damage panel has fractured at the right rivets, which stringer fibers are broken completely, the main crack penetrates the panel.

Figure 3: Surface morphology of fracture

It can be seen clearly that there are some cracking traces on the skin bottom surface, as is shown in Figure 3. In the both ends of the fracture, stringer dislocates with an angle on the fracture both sides, and there are interlaminar cracks in stringer near the interface of stringer and skin. In the front-end, cracks propagate from the right to the left, but it is just contrary to that of the rear-end. In the area, the stringer bottom arched after buckling around the fracture (see Figure 4). Skin seriously delaminates, and the plies of skin completely crack along the interlamination, but there are some longitudinal cracks across

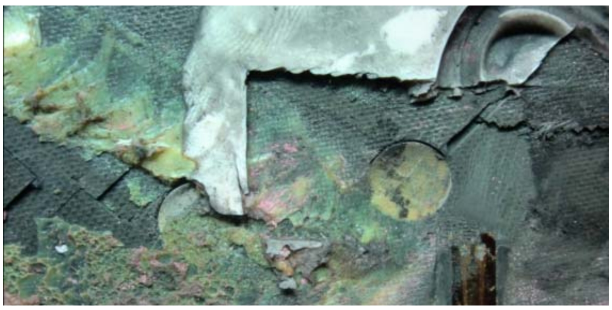

Figure 4: Stringer fracture morphology

2.2 Metallographic observation of damage region

Some specimens were cut from the damage area for metallographic observation, as is shown in Figure 1 (Region A, B and C). In region A, skin plies delaminate, and there are also some longitudinal cracks. However, few delaminations and cracks can be found in the stringer bottom near co-curing interface (see

Figure 5). Damage morphologies of skin by the both ends of stringer are slightly different. There are some longitudinal cracks in the front-end of stringer, which connect with the interlaminar cracks. The nearest co-curing interface ply has cracked, and the cross crack is characterized by ragged fracture. The longitudinal cracks penetrate one or two plies and connect with the interlaminar cracks. The cracks opening points to the co-curing interface (see Figure 6). There are interlaminar cracks in the skin plies nearby the stringer in region B and region C. However, in region B the fractures fragmentize and more longitudinal cracks exist.

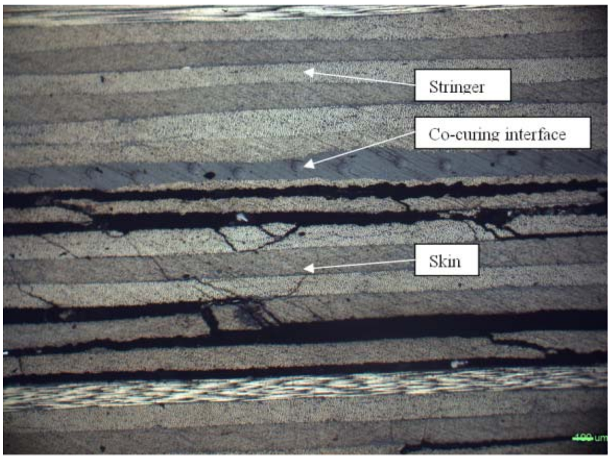

Figure 5: Region A metallographic morphology

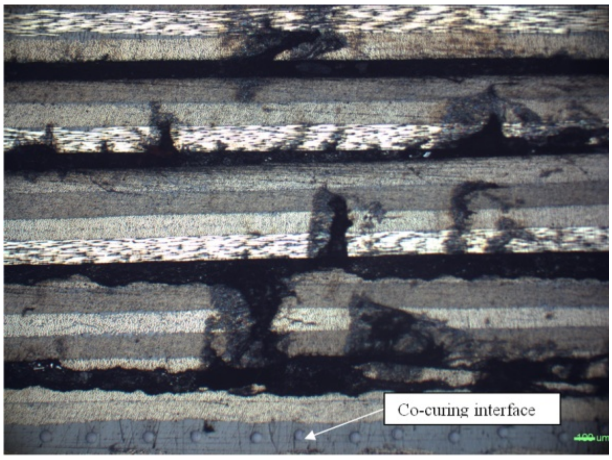

Figure 6: Longitudinal crack opening direction

2.3 ESEM observation of fracture morphology

Specimens separately cut from region A, region B and region C were observed under ESEM. Each fracture is formed by interlaminar cracks, and the fractures are flat. There are some traces of matrix cracking and fiber pull-out break on the fracture, and longitudinal cracks which perpendicular to fiber direction can be found in local areas. On the fracture, fibers are bared smooth and matrix remains smooth indentation. Several fibers broke with a positive fracture. Uneven fractures and fibers irregular breakage can be found in the skin nearby co-curing interface (See Figure 7,Figure 8,Figure 9).

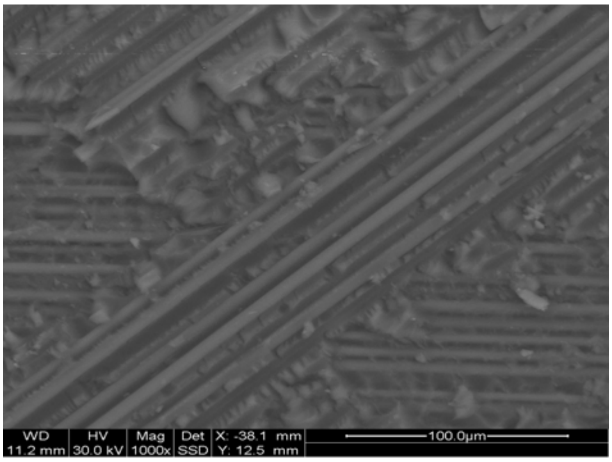

Figure 7: Fracture morphologies by ESEM a

Figure 8: Fracture morphologies by ESEM b

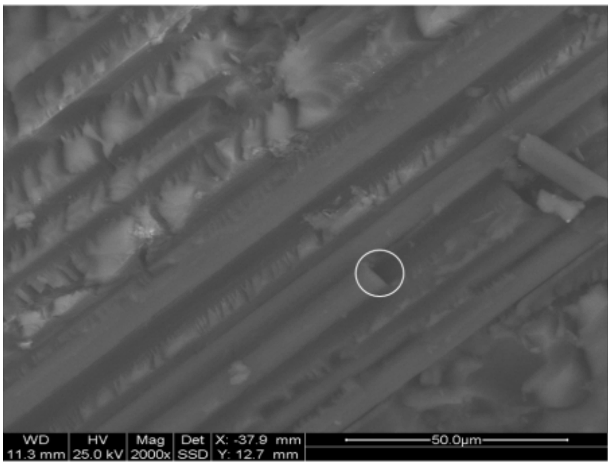

Figure 9: Fracture morphologies by ESEM c

2.4 XPS component analysis
Specimens from damaged area, undamaged area, Processing Coupon and different wings are conducted composition analysis to the fiber surface under XRF-1800 Fluorescence Spectrometer. Results are shown in Table 1, Table 2 and Table 3 (except C and H).

Table 1: Results of constituents of fiber surface in different areas (wt%)

Category	Ni	Cu	Fe	Si	S	Cr	Al	Mg	Mn
Damaged area	64.68	22.76	8.36	1.97	1.58	0.51	0.06	0.05	0.04
Undamaged area	64.51	24.44	6.56	2.17	1.82	0.30	0.07	0.13	

Table 2: Results of constituents of fiber surface of Processing Control Coupon (wt%)

Category	Ni	Cu	Fe	Si	S	Cr	Al	Mg	Mn
Processing Control Coupon	63.61	23.34	8.86	2.07	1.48	0.48	0.06	0.06	0.05

Table 3: Results of constituents of fiber surface of different wings (wt%)

Category	Ni	Cu	Fe	Si	S	Cr	Al	Mg	Mn
Wing 1#	66.18	23.71	7.35	0.67	1.48	0.41	0.06	0.08	0.02
Wing 2#	65.91	24.42	6.56	0.74	1.82	0.30	0.07	0.13	0.05

3. Analysis and discussion

3.1 Fracture characteristics and properties

Wing panel fractured at the right rivet, and main crack penetrated the panel. Both sides of the fracture formed a certain dislocation. It can be found on the fractures that there are some smooth bare fibers and traces of fibers pull-out. Skin nearby the fracture delaminates, and interlaminar cracks propagate along plies interface. However, as the distance becomes far from co-curing interface, delaminations decline. In local areas, fibers broke with a positive fracture. There are lots of longitudinal cracks which are perpendicular to fiber direction, and their opening points to co-curing interface. Therefore, through the analyses of opening and path of the longitudinal cracks, it can be concluded that skin bend toward the side of stringer under compressive load, and skin surface subjects to tensile stress. There is a crush ply on the skin bottom, but the upper surface ply breaks. Moreover, the fracture morphologies on the both sides of the rivet are obviously different. Thus, the compressive load is varied, whose direction is similar with the cracks. Therefore, under alternate compressive load, the panel buckled repeatedly at the centre of the rivet, and finally fractured for fatigue.

3.2 Analysis of failure causes

Local areas of the composites wing fractured in the stability test, thus the reasons are the excessive load or its own insufficient strength. However, the test conditions are adopted according to the standard, and no abnormal load appears in the test. Thus, the primary cause of the wing skin panel fracture is probably the insufficient strength. There are many factors that influence the structure strength of the panel. Firstly, the design strength was not enough. Secondly, the insufficient strength is caused by the factors in the process such as prepreg pollution. Ordinary, inappropriate process will lead to the difference in micro-morphology (Zhang et al. 2003). However, the results show there are no abnormal morphologies on the fracture, such as large area debonding. Only in a few areas could be found that there are some smooth matrix traces after fibers pull-out and smooth bare fibers of on the fracture. In ideal condition, matrix fragmentize with a fish scale pattern along fibers, especially inside a ply, and there should notbe the smooth fibers and matrix indentation (Zhang et al. 2003). According to the results of XPS test, Si content of the fiber surface with the same furnace is higher than that of others. It is possible that the interface of fiber and matrix has been polluted in the preparation of prepreg or other process steps. Therefore, the pollution will lead to weak bond in the areas (Baldan. 2004). With the change of load, these areas will gradually grow into crack source, accordingly it causes the strength descent. These factors result in structure resistance descent, so the factors should be taken into consideration in design and be avoided. Therefore, the primary cause is the insufficient design strength of the wing skin panel, but the interface pollution leads to the strength decline and accelerate cracks initiation.

4. Conclusion

Under the effect of alternate compressive load, the wing panel formed notch effect at the rivets and fractured for fatigue. The composites wing was damaged in the surrounding areas of the rivets under compression load, which is characterized by compression fracture. For the inconsistent deformation in the compression process, the damage mode of local areas is shear fracture. The primary cause of wing skin panel fracture is that the structural design strength is not enough. However, interface pollution leads to the structure strength descend and induce the fractures to be happened.

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