Failure Analysis of Circular Composite Patch Adhesively Bonded on Damaged Carbon Fiber Reinforced Polymer Panel

V. S. Bhise\textsuperscript{a}, S. V. Nimje\textsuperscript{b}\textsuperscript{*} and S. S. Siddhant\textsuperscript{c}

\textsuperscript{a} Asst. Professor, Sinhgad Institute of Technology & Science, Pune, India
\textsuperscript{b} Asst. Professor, Defense Institute of Advanced Technology (Deemed University), Pune, India
\textsuperscript{c} PG Student, Defense Institute of Advanced Technology (Deemed University), Pune, India

1. INTRODUCTION & OBJECTIVE

The use of fiber reinforced plastic composites in the transport industry (aerospace, automotive and marine) has risen significantly in recent years especially in aircrafts like Airbus A350 XWB and Boeing 787 Dreamliner due to their high specific strength/stiffness, corrosion resistance and improved fatigue resistance etc. During the service life of this aircrafts, the composite structures are prone to damage due to repeated loadings and take off, bird strike accidental impact and environmental conditions. Generally, the damage in composite materials is in the form matrix cracking, fiber breakage, debonding and delaminations. These damages in composite structures reduce their structural strength and decrease their design life. Often the cost of aircraft composite structures is very high, it is not possible to replace the damaged complex structure and hence damaged portion needs to be repaired to extend the life of an aircraft. These repairs can possibly achieve either by using mechanical fasteners or adhesively bonded patches. Adhesively bonded patch repair having some advantages over mechanical fasteners like no stress concentration and larger area is available for load transfer hence in most of the practical applications adhesively bonded patch repair is preferred [1,2].

In the present research work, three-dimensional finite element analyses have been carried out to evaluate stress field in the adhesively bonded patch repaired structure. Subsequently, the onset of failures over the bond line interfacial surfaces has been predicted by Tsai and Wu [3] coupled failure criterion through computation of failure indices 'e'. The performance of both bonded repair techniques applied to CFRP laminates was assessed under uniaxial tensile loading.

2. SPECIMEN GEOMETRY AND FINITE ELEMENT MODELLING

The geometry of the damaged panel and repaired panel is shown in Figure 1. The composite panel and patch is made up of Gr/E (T300/934) material. The stacking sequence of panel is [45/-45/0/90]. The panel has a circular hole of radius 5 mm at the center for simulating the damaged area. The patch is bonded to the panel using epoxy adhesive. The panel is subjected to uniaxial in-plane tensile load of 15 kN. The Material properties of CFRP panel, patch and adhesive are taken from Ref. [2]. The finite element model of the considered geometry is shown in Figure 2. The commercially available finite element package ANSYS 14.5 is used in the current study. The panel, adhesive and patch are modeled with 20-noded solid 186 brick element. In the thickness direction, the panel is meshed with eight elements, adhesive with two elements and patch with four elements. The layer angles are defined by assigning appropriate element coordinate system to both panel and patch elements. Orthotropic material properties are assigned for the panel and patch elements, isotropic material properties are assigned for adhesive and a tensile load of 15 kN is being applied.

\textsuperscript{*} Further author information: (Send correspondence to S.S.A)
V. S. Bhise.: E-mail: vsbhise_sits@sindhgad.edu, Telephone: +91 (020) 66831761
S. V. Nimje.: E-mail: sunil.nimje@diat.ac.in, Telephone: +4020-2430 4204, Address: Department of Mechanical Engineering, Defense Institute of Advanced Technology (Deemed University), Pune, India.
3. RESULTS & HIGHLIGHTS OF IMPOINTANT POINTS

Figure 3 shows the stress variation in panel, patch and adhesive layer for the repaired configuration at a load of 15 kN. From figure 3(a), it can be observed that at two critical locations (A, B) high stress levels are present in the panel. Zone A is the transverse edge of the hole and zone B is the longitudinal edge of the patch. By closely observing in figure 3(b), it is found that zone C (overlap edge) is one of the most critical location on the adhesive layer from which damage could initiate in the form of patch debonding. Figure 3(c) reveals that $\sigma_{xx}$ stress component in the patch is higher at the hole edge (Zone D) [3]. These locations are considered for computation of failure indices ‘e’.

REFERENCES