DAMAGED ALUMINUM PANEL REINFORCED BY COMPOSITE PATCHES UNDER SHEAR LOADS- A NUMERICAL AND EXPERIMENTAL INVESTIGATION

Abstract

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Aging metallic aircraft, manufactured in the early 70's, are still being operated around the world, subjected to new types of damage, appearing in new locations on the aircraft. In past years, the use of composites has increased for aluminum aircraft components. Researches substantiated these repairs to be effective from durability point of view. Many studies evaluated the effect of plies application sequence, orientation sequence, effect of stress amplification factor and more. However, all studies focused on simple tensile loading. Forward and center sections of aircraft fuselage contain a skin, which plays a major role in load transferring, usually due to torsion originated by fuselage bending. This research aims to study the effect of the above mentioned parameters on composite repair patch for a panel, subjected to shear loads. The research is based on a new approach of finite element modeling, consisting of a combination of 2D and 3D elements. The damage was simulated by a central located hole with various diameter sizes. It was found that the smaller the diameter, the smaller the area of the damage in the aluminum plate. The patches bonded on the damaged aluminum panel were made of laminated composite carbon fiber, with various layups. Special attention was taken to include the influence of the adhesive on the overall behavior of the patch attached to the damaged plate. Accordingly, the finite element model was parametrically analyzed for strength and buckling, as a function of patch size, overlaps, sequences and bonding influence. These analyses were then substantiated by a dedicated set of experiments – representative elements (see Figs. 1a-c). To enable a correct

comparison between the actual tested panels fixed in a shear type frame, the frame, including its bolts were modeled using a finite element code. Based on the test results, the accuracy of the finite element model was evaluated and the method of repair patch modeling was validated (See Figs. 2-4). The following conclusions can be drawn:

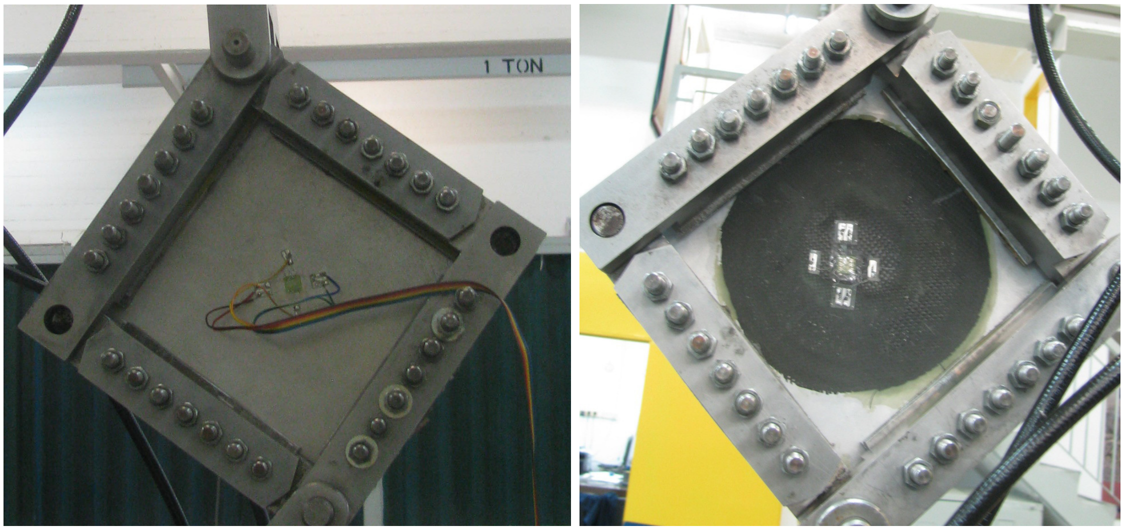
a. Metallic panels under shear loads must be verified for both analysis – stress and buckling. As for buckling depends directly on geometry and less on strength properties, this issue must be verified. Nevertheless, from the performed analysis, it may be concluded that designing a composite repair patch for an actual aircraft panel (heat treated, having higher strength properties as appose to annealed-O treatment), shall cover the design requirements for buckling. In our analysis, verified by an experiment, we encountered the buckling phenomenon due to usage of an untreated Aluminum.

b. Studying parametrically the effect of plies application sequence, based on analysis and verified in the experiment, it was found out that application of a ply in 0^O (along load application) decreases the stress amplification factor on the damage edge, enabling allegedly a later failure, comparing to a patch, applied with in 45^O to the applied load direction. Corresponding to the previous statement, the experiment demonstrated a later buckling (higher load of buckling) for the patch applied in the load direction (0^O). Thus, 0^O plies should be used when adding extra repair plies to increase the capability of load carrying.

c. Along the present study a parametric investigation was performed, based on a finite element model, designed by a new approach of connecting shell elements with 3D elements. Based on the performed tests on 4 representing specimens, the finite element model was substantiated in all sampling points of buckling load and strain measurement locations. It was found that the finite element model is always conservative by a maximum of 10%, that apart from only one measurement of strain of one of the composite patches. In that patch, a lack in smoothness may be noticed due to insufficiently proper sealing of the damage.

d. As a result of the analysis validation, it may be concluded that a wet-layup patch repair effectively restores the loss of strength.

(a)



(b) (c)

Fig. 1 (a) – The test set-up, (b) the aluminum panel in the shear frame, (c) the aluminum panel with a side patch

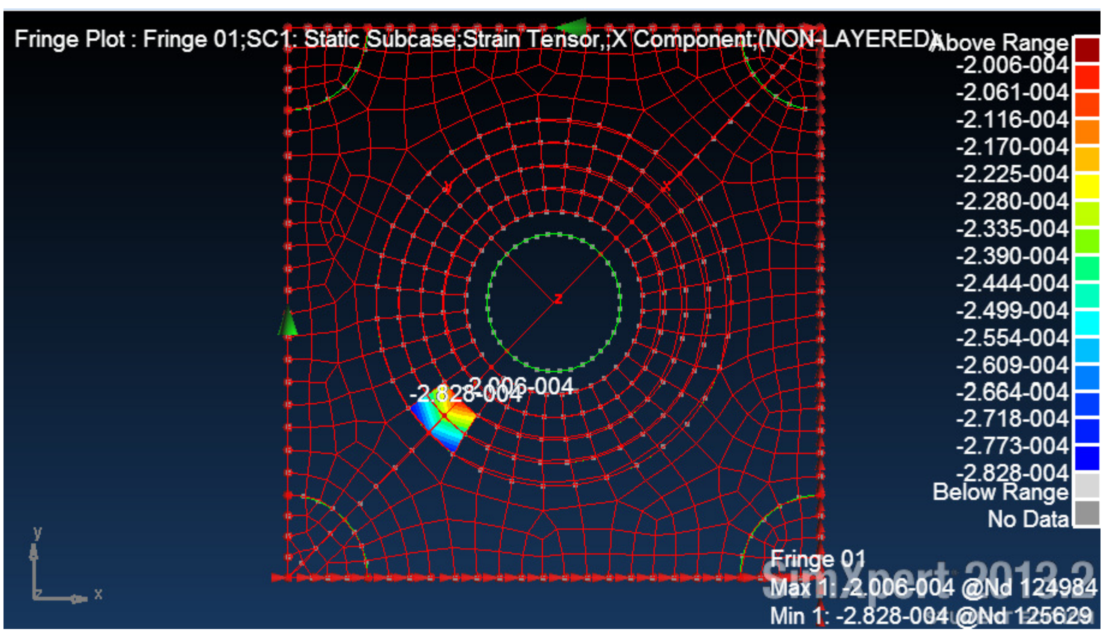


Fig. 2 The FE model of an aluminum plate with a central hole simulating damage

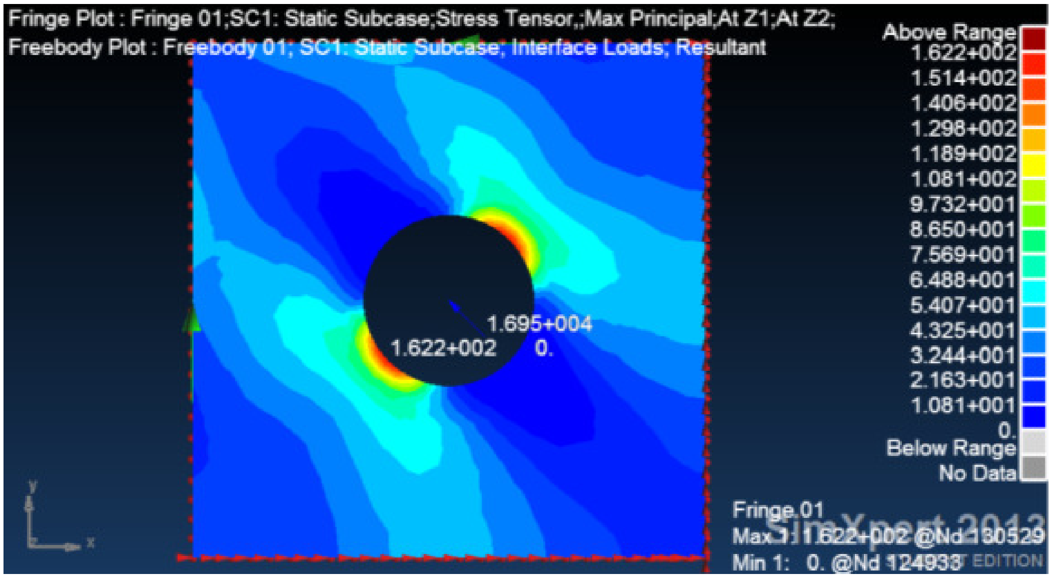


Fig. 3 The FE model of an aluminum plate with a central hole simulating damage in shear – Maximal principle stresses

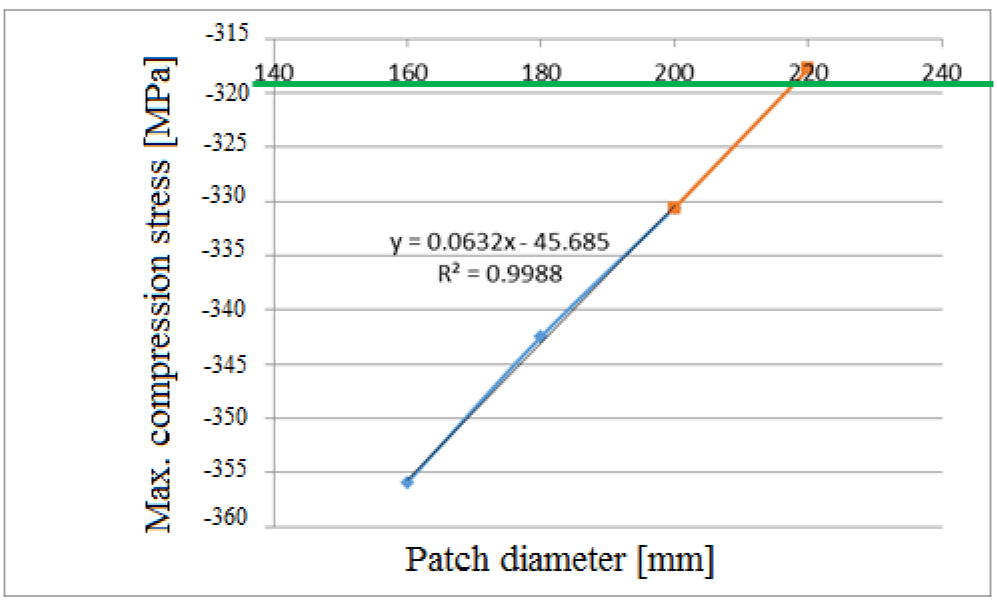


Fig. 4– Max compression stress in the composite repair patch vs. patch size – The larger the patch, the smaller is the stress in it.