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Design and analysis of adhesively bonded thick composite patch repair of corrosion grind-out and cracks on 2024 T3 clad aluminum aging aircraft structures

A. Chukwujekwu Okafor *, Hari Bhogapurapu

Structural Health Monitoring and NDE Laboratory, Department of Mechanical and Aerospace Engineering, University of Missouri-Rolla, 1870 Miner Circle, Rolla MO 65409-0050, USA

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Abstract

Many military and commercial aging aircrafts flying beyond their design life may experience severe crack and corrosion damage, and thus lead to catastrophic failures. In this paper, the design, fabrication and analysis of adhesively bonded thick composite patch repair of circular corrosion grind-out and a crack propagating on the periphery of the corrosion grind-out on thick 2024 T3 clad aluminum aircraft panel is presented. Thick orthogonal composite patch configurations of 7–25 plies were designed separately for crack and corrosion grind-out using CRAS. Using the principles of superimposition a single patch was designed to repair both the crack and corrosion grind-out. Finite element analysis (FEA) was performed on the test specimen subjected to uniaxial tensile loading. Stress distribution and displacements were obtained and analyzed. Dog-bone shaped tensile test panels were fabricated with damage and repaired with boron/epoxy patch of 11 plies. The patched and unpatched panels were subjected to tensile tests. The experimental and the FEA results show that the maximum skin stress decreases significantly and shifted away from damaged area after the application of composite patch. The load carrying capacity of patched specimen significantly increased over that for unpatched specimen.

Keywords: Crack and corrosion grind-out; Thick composite patch repair; Aging aircraft structure; Superimposition; Stress distribution; Finite element analysis

1. Introduction

Aging aircraft problem is one of the serious challenges to commercial and military aircraft operators. The high acquisition costs associated with the purchase of modern aircrafts, coupled with the increased budget cuts in the acquisition of new aircraft fleets have resulted in the utilization of aircrafts beyond their original design life. Corrosion and fatigue are the major factors that contribute to the aging of aircraft. During its service life, an aircraft is subjected to severe structural and aerodynamic loads which may result from repeated landings and take-off, fatigue, ground handling, bird strikes and environmental degradation such as stress corrosion, which cause damage or weakening of the structure. A repair or reinforcement of the structure to restore the structural efficiency and thus assure the continued airworthiness of the aircraft has become an important issue in recent years. Advanced composite materials have many advantages like high specific strength and stiffness, lightweight, resistance to corrosion, directional dependence of the material properties, ability to be formed to conform complex shapes and contours and to meet variable stiffness requirement.

The technique of repairing cracked metallic aircraft structures using high strength advanced composite materials is commonly known as "crack patching". The composite reinforcement, also known as a patch, can be attached to the damaged or weakened structure either by mechanical fastener or adhesive bonding. The use of adhesively bonded composite patches [1] has several advantages over

* Corresponding author. Tel.: +1 573 341 4695; fax: +1 573 341 6899.
E-mail address: okafor@umr.edu (A.C. Okafor).

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mechanically fastened repair methods, which include reduced installation cost, increased strength and fatigue life, reduced repair down time, elimination of unnecessary fastener holes in an already weakened structure and stress concentrations at fasteners, corrosion resistance, high stiffness and lightweight [1].

This paper presents the design, finite element modeling and analysis technique for single sided thick composite patch repair of corrosion grind-outs and crack on the periphery of 2024-T3 aluminum aircraft panels [2–5]. The objectives of this research are (1) to develop a procedure for designing thick composite patch (the number of plies more than 10, preferably greater than 20) for the repair of circular corrosion grind-out and crack at the periphery of the circular grind-out, (2) to study the complicated stress distribution on the damaged aluminum specimen repaired with symmetric octagonal patch under axial tensile fatigue loading, (3) to experimentally verify the durability of patched and unpatched panels to validate the FEA results.

2. Design of patch

A combination of shear, axial, normal, peel and cleavage loading make up the primary loads acting on adhesive bonds. A good repair must be effective in minimizing these effects. Patch design must address the following issues.

2.1. Patch thickness

It has been found that the patch is most effective when the patch stiffness ($E_p t_p$) is equal to skin (aluminum panel) stiffness ($E_s t_s$). Where E_p and E_s are the modulus of elasticity of patch material, and skin material, respectively; t_p and t_s are the thickness of composite patch and skin (aluminum) material. The ratio of patch stiffness to skin stiffness is called the Stiffness Ratio (SR) and has an ideal value of 1. The recommended stiffness ratio (SR) ranges from 1.0 to 1.6.

2.2. Adhesive

Adhesive properties influence the strength of the joint since the load is transferred from the skin to patch via the adhesive. The thickness of adhesive layer is an important issue in designing a patch. Thin adhesive layers are shown to perform better than thick layers because with increase in adhesive thickness the patch gets softer. But thickness provides good durability to the entire patch, as thick patch adhesive layer attracts lesser strains. Stiffer adhesives perform better but if the adhesive is very stiff, there is a danger for the patch reaching the yield point earlier than usual.

2.3. Peel stress

Due to the shift in neutral axis with a single sided repair, unwanted secondary bending moments are introduced. These bending moments create high peel forces at the ends of repair. Tapering the repair ends and increasing the overlap length takes care of this problem. Peel (through the thickness) stresses have been shown to increase with increase in patch thickness and decrease with increase in adhesive thickness. The maximum peel stresses induced in the adhesive and adjacent adhered (skin) at the end of the overlap is given by

$$\sigma_{\text{peel}} = t_{\max}\left[\frac{3E_z(1-v_{zx}^2)t_r}{E_x t_A}\right]^{1/4} \tag{1}$$

where $t_{\max}$ is the maximum shear stress in the adhesive, E_z is adhesive elastic modulus in the z-direction, v_{zx} is the Poisson ratio in the zx plane, t_r is the thickness of the repair patch, E_x is the adhesive elastic modulus in the x-direction and t_A is the thickness of the adhesive.

2.4. Stress intensity factor

Reduction in the stress intensity factor (SIF) is the key issues in designing a patch. There is a considerable decrease of SIF just by applying one layer of patch. The rate of decrease of stress intensity is monotonic with the increase in the patch thickness. An increase in skin thickness causes an increase in the SIF; an increase of the fiber stresses near the crack and an increase of adhesive stresses near a crack, but has little effect in the adhesive stresses at the edges of patch. The damage tolerance and durability of repaired structure can be assessed by analyzing the SIF and crack growth rate and establishing a relationship between the two. The fatigue crack growth rate relationship (the Paris law) of unpatched panel is given as

$$\frac{\mathrm{d}a}{\mathrm{d}N} = C(\Delta K)^n \tag{2}$$

where $\mathrm{d}a/\mathrm{d}N$ is the crack growth rate, a is the half crack length, N is the number of cycles, C and n are material constants, and ΔK is the stress intensity factor range.

2.5. Strain in the composite patch and adhesive layer

The maximum strain in the fibers $\varepsilon_{\text{f-max}}$ must be kept below the breaking strain of the fibers. For boron laminate, the limiting strain is taken as $\varepsilon_{\text{f-max}} = 0.005$ so that we require that the patch stress satisfy

$$\sigma_f(\sim \varepsilon_f E_b) = 0.005E_b$$

where σ_f is the stress in the patch fibers, ε_f is the strain in the patch fibers and E_b is the Young's modulus of the composite in the direction of the fibers, which in the present patch is equal to 195 GPa (2.82E3 ksi). If we denote the stress concentration in the fibers as C, this then tells us that the patch thickness must be such that

$$\sigma_f = C\sigma \leqslant 1.04\ \text{GPa}\ (150.84\ \text{ksi})$$

where σ is the applied stress. Hence, knowing the applied stress determines the maximum permissible value of C