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Effect of load amplitude change on the fatigue life of cracked Al plate repaired with composite patch

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ABSTRACT

In this study, the fatigue behavior of aluminum alloy 2024T3 v-notched specimens repaired with composite patch under block loading was analyzed experimentally. Two loading blocks were applied: increasing and decreasing at two stress ratio: $R = 0$ and $R = 0.1$. Failed samples were examined under scanning electron microscope at different magnifications to analyze their fractured surfaces. The obtained results show that under increasing blocks, the crack growth is accelerated for both repaired and unrepaired specimens. This is attributed to the increase of the loading amplitude in the second block. A retardation effect was observed for decreasing blocks loading in unrepaired specimens. However, this retardation effect is attenuated by the presence of the patch which lead to lower fatigue life for repaired specimens.

1. Introduction

Aged aircraft structure may contain fatigue cracks resulting from their long service. Fatigue loads, of constant and variable cyclic load amplitudes, are the major contributors to the service induced damages in the aerospace structures. Bonded composite patches have been successfully used to repair the damaged structures and it is structurally very efficient, can be applied rapidly and are cost effective [1–3]. The technology involves adhesively bonding patches of advanced fiber composite materials to repair damaged aircraft structures and to prevent stress corrosion cracking. Once repaired using bonded composite patch, there is no guarantee that the structure continue being subjected to the same loading amplitudes. Several studies of unrepaired cracks [4,5] show the significant effect of the load history on the fatigue life of metallic structures. Tensile and compressive overload cycles of near yield magnitude have been shown to accelerate crack growth [6,7] while crack retardation has been observed on long cracks following tensile overloads that were well below the yield strength [7]. Experiments composed of two constant amplitude loading blocks changing from a low to a higher stress level (*L*–*H* block) or vice versa (*H*–*L* block) are usually employed to study the block loading in materials. However the results obtained from these experiments are not consistent, showing a greater damaging effect due to the *L*–*H* blocks and retardation effect due to *H*–*L* blocks, depending on the material and loading parameters [4,8]. This is generally explained by the size of the plastic zone created at the crack tip as well as by the residual stresses and crack branching in other cases.

In the case of repaired cracks, the interaction of the load change with the stress bridging and redistribution, caused by the composite patch, may have a considerable impact on the crack repair.

Several works have been published on the effect of variable amplitude loading, cycle mix and the variation in mean stress on the adhesively-bonded joints [9–11], the effect of overloads and loading sequence on adhesively bonded double-lap joints [12–14].

Khan et al. [15] analyzed the fatigue life of Al 7075-T6 cracked specimens repaired with adhesively bonded composite patches for different load ratios and compared with that of unrepaired specimen under the same cyclic block loading for two different sequences (increasing and decreasing amplitude). They showed that the patch repair efficiency is not significant for increasing blocks of loading, whereas for decreasing blocks of loading, the improvement is relatively noticeable. The combination of the fiber bridging and the retardations effect leads to the significant improvement of the fatigue life for the repaired structures.

The repair performance of cracked aluminum plates using bonded composite patches can be evaluated by monitoring, among others, the repaired crack growth under different loads. It has been shown in previous studies [16–18] that the smaller the initial size of the repaired crack the higher is the efficiency of the patch repair.

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However, to the knowledge of the authors, there is no information in the literature on the behavior of a patch repair relating the load amplitude leading to the crack generation and that applied after the repair.

Behavior of cracks repaired with bonded composite patch under variable amplitude loading is more complex than under constant amplitude loading [19]. To understand the more realistic phenomenon which occur during the aircraft services such as the transient crack closure and the retardation effects associated with overloads, it is necessary to study the bonded composite repair under representative flight load spectra. The first case of such repair was executed on the wing skin of RAAF Mirage III aircraft [20], subjected to an operational flight load spectra and the other major case was composite repair on USAF C-141 aircraft [21]. The repair has been applied to over 150 wings in service and over 3 years of operational history has been observed during the investigation. Both cases have been reported to be successful. The finite element method was also carried out during the repair of RAAF Mirage III aircraft. It has been used to design several complex repair schemes, such as the repair of fatigue cracks in the lower skin of Mirage aircraft and cracks on the upper surface of the wing pivot fitting of FlllC aircraft in service with the Royal Australian Air Force (RAAF) [26]. Baker reported that many patch repairs of the Royal Australian Air Force have been in service for longer than 20 years without durability issues arising from environmental or fatigue damage [22].

Raizenne et al. [23] conducted series of fatigue tests on repaired aircraft panels using a "clipped" FALSTAFF spectrum in which the negative loading has been removed. Poole [25] reported the beneficial effect of single side patch repair on thin aluminum sheets and suggested that the effect of variable amplitude loading spectra on patch debonding should be studied in terms of patch efficiency. He also suggested that there is a clear requirement for a model to predict debonding, and patch efficiency under a wide range of loading spectra. Baker [27] conducted fatigue test on wing skin of an Australian Defense Force F-111C aircraft repaired with an adhesively bonded boron/epoxy fiber composite patch. During these tests, the patch was successful in preventing growth of the crack for around a further 9000 simulated flying hours.

In the work done by Walker and Rose [28], safety critical bonded composite repair to the outer lower wing skin of an F-111 aircraft has been fully validated and substantiated. The first wing to be repaired exceeded 665.9 h of actual operational usage and a further 8074.4 h in a full-scale wing fatigue test [28].

In this study, we aim to evaluate experimentally the effect of the load amplitude change prior–post the repair on its efficiency. Two blocks of constants maximum loading set-ups are considered in this study: Low–High (7–12 kN) and High–Low (12–7 kN). One of the two loading blocks is responsible to creating and propagating a crack, from a notched specimen, to a length of 3 mm representing the initial detected crack size to be repaired. After repair, the specimen is subjected to the second block where the maximum load is changed and the crack growth is monitored until failure. Similar tests are conducted but without patch for the purpose of comparison. Previously published results [29] are used to highlight the material effect on the performance of the composite patch repair through the two different load levels.

2. Experimental setup

2.1. Chemical characterization of the material

The material used in this study was in the form of thin plates. Few samples of the material was taken and chemically analyzed on SEM equipped with Energy Dispersive X-ray analyzer (EDS) to confirm the composition and properties of Al 2024 tempered at T3. The chemical composition of the aluminum alloy (Al 2024-T3) is given in Table 1.

2.2. Specimen details

In this study, we used Al2024-T3 single edged notched tension (SENT) specimens (see Fig. 1) of dimensions 150 × 50 × 2 mm. An initial v-notch of 6 mm depth and 60° angle was created, in accordance to ASTM E647 standards [30], in the center of each specimen by milling, as shown in Fig. 1. The specimens were then pre-cracked under fatigue loading to different initial crack lengths. The presence of the v-notch facilitates the mode I propagation of the crack. Once the desired crack length is reached, the sample was unloaded and taken for surface preparation followed by bonding the composite patch.

2.3. Patch preparation

Composite patches are made using 8 plies of unidirectional carbon/epoxy pre-pregs. The dimension of each ply was 250 × 250 mm. The pre-pregs were sandwiched in between two

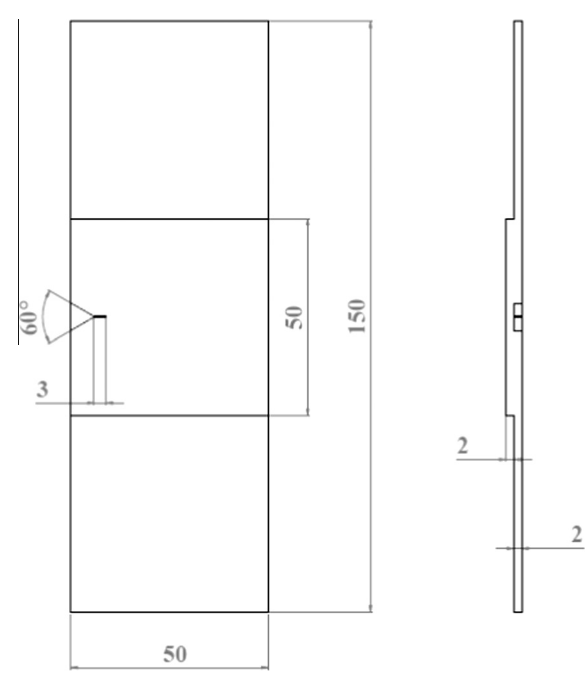

Fig. 1. Specimen details.

Table 1
Chemical composition of the aluminum alloy 2024-T3 (in wt.%).

Atom	Al	Cr	Cu	Fe	Mg	Mn	Si	Ti	Zn	Other
Mass content	90.7–94.7	Max 0.1	3.8–4.9	Max 0.5	1.2–1.8	0.3–0.9	Max 0.5	Max 0.15	Max 0.25	Max 0.15