The effect of stiffness of bonded Boron/Epoxy (B/Ep) single-sided patches on the fatigue crack growth in an aluminum fuselage panel was studied. Through-the-thickness cracks were repaired with ‘stiff’ (S-BE) and ‘compliant’ (C-BE) B/Ep patches. Tests were performed with a full-scale Boeing 727 fuselage panel under simulated in-service flight load conditions. Fatigue crack growth and patch disbonds were monitored throughout the test using visual inspections, eddy current, flash thermography, and resonance ultrasonics. A full-field strain and displacement measurements throughout the patch and its surroundings were recorded using digital image correlation method. The strain fields show the effect of patch stiffness on load transfer throughout the patches. Fatigue crack growth rate in the C-BE patch was considerably higher than that in the S-BE patch.

1 Introduction

During the service life of an aircraft, the load capacity of the structure degrades primarily due to fatigue cracks and corrosion. The obvious solution to increase the life of the aircraft is to repair the damage and to restore the load path, weakened due to the presence of such cracks. These repairs are typically done using either mechanically fastened or adhesively bonded patches. The mechanically fastened repairs have many drawbacks such as removal of material from the substrate, load transfer only through the fasteners, and stress concentrations at the fastener holes. Overall, the mechanically fastened repairs do not restore the structure to its original strength. On the other hand, adhesively bonded repairs are aerodynamically more efficient and can restore the strength more effectively [1, 2]. The adhesively bonded repair requires minimal or no material removal and the load transfer from the substrate to the repair are more continuous as compared to mechanically fastened repairs [1, 2].

As a mathematical problem, adhesively bonded repair technology is a combination of a number of sub-problems, which includes load transfer from the parent material to the patch, calculating stress intensity factors at the crack tips, load attraction by bonded patches, effect of stresses due to the adhesive, effect of patch geometry, bending effect in the case of one-sided repair, residual thermal stresses, adhesive plasticity, mixed-mode loading, and many more. Over the past four decades extensive work had been done to address these and other challenges of bonded repairs. Several of the analytical solutions of bonded repairs can be found in [1-7]. In addition, many have developed finite element models to study bonded repairs. For example, four different approaches were proposed in [8-11] for flat plates, while few recent studies addressed repair of curved structures [12-15]. Experimental studies on repair of curved panels are limited [16]. In other words, the analytical, numerical, and experimental studies of adhesively bonded repairs focused primarily on flat plates subjected to in-plane loading. Yet, adhesive bonded technology has been successfully implemented to repair military aircrafts [2].

Clearly, there still remain many challenges before the adhesive bonded repair technology can be applied to commercial aircrafts. These challenges include a better understanding of the performance of bonded repairs under combined in-plane and lateral loading (pressure), developing reliable inspection methods to determine bond quality and durability, and establishing design and analysis tools for adhesively bonded repair of curved panels.

The Federal Aviation Administration (FAA) and The Boeing Company have teamed up in an effort to study the fatigue and residual strength performance of bonded repairs using boron/epoxy (B/Ep) and aluminum patches. The FAA Full-Scale Aircraft Structural Test Evaluation and Research (FASTER)
facility is being used to conduct structural tests of various damage/repair scenarios on several aluminum-made fuselage panels. The first panel test results revealed that properly designed and installed bonded repairs are durable under fatigue loading which is well above the design limit and can effectively contain large damage under severe static loads in excess of ultimate load requirements [17]. The second panel test results showed that the fatigue performance of the adhesively bonded repairs reduces as the repair quality degrades [18].

As a part of this second panel test program, data on the effect of patch/panel stiffness ratio on the fatigue crack growth in adhesively bonded repairs were obtained. Previous work, studying the effect of composite patch laminate configuration, has shown that for thin flat panels higher stiffness ratio provides slower crack growth [19] and that ply orientation affects strain energy at the crack tip [20]. To address realistic applications, the work discussed herein was conducted on a curved fuselage panel subjected to realistic flight load conditions. Two extreme stiffness ratios were considered, both having the same patch geometry, configurations, material, and similar installation procedures.

2 Experimental Procedure

Testing was conducted using the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility located at the FAA William J. Hughes Technical Center. The FASTER fixture is capable of testing large curved panels representative of aircraft fuselage structures under realistic flight load conditions. Details of FASTER fixture can be found in [21].

2.1 Test Panel Description

The fuselage panel used in this study was from a retired Boeing 727-232 passenger service airplane, Fig. 1. The airplane was placed into service in 1974 and retired in 1998. Further information regarding the airplane can be found in [22].

The panel was removed from the crown of the airplane, having dimensions of 125 by 73 inches and a radius of 74 inches, Fig. 2. The substructure included six stringers (S), S-1R through 6R, in the longitudinal direction with 9.5-inch spacing and six frame stations (FS), FS-990 through 1090, in the hoop direction with 20-inch spacing. The Z-shaped frames, stringer clips, and hat-shaped stringers are made from 7075-T6 aluminum. A longitudinal lap joint was located along stringer S-4R, and a circumferential butt joint was located along FS-1010. The longitudinal lap joint consisted of two skin layers and a 0.02-inch-thick 2024-T3 aluminum doubler layer sandwiched between the two skin layers, which were connected together by three rows of rivets. The doubler was bonded to the outer skin, and there was a sealant between the doubler and the inner skin of the lap joint. The panel skin was 2024-T3 aluminum with a thickness varying from 0.040 to 0.080 inch at various bays. The region of constant skin thickness of 0.04 inches was chosen as the test section, Fig. 2.

2.2 Damage Scenarios and Patch Configurations

The damage scenario for the test was 2.8-in. long through-the-thickness notches, which were introduced at different mid-bay locations of the fuselage panel. These notches were pre-fatigued (at 75% of service load) to obtain a naturally growing crack at both ends. These cracks were repaired using different types of single-sided patches [18]. Since this paper focusses on the effect of B/Ep patch stiffness on the fatigue crack growth, only the details of these two patches are provided herein. Both patches were 8-in. x 3-in. octagonal shape, containing 5 plies with an approximate taper ratio of 20:1. The location and the details of both patches are shown in Fig. 3. The patch-to-panel stiffness ratio, S, is defined as:

\[
S = \frac{A_{11}}{Et} \tag{1}
\]

where E and t are the Young’s Modulus and thickness of the aluminum panel and

\[
A_{11} = \sum_{k=1}^{N} (Q_{11})_k t_k \tag{2}
\]

is the in-plane stiffness of the composite patch in the direction perpendicular to the crack. Thus, the ‘stiff’ B/Ep laminate patch (S-BE) has a layup sequence [+20°/-20°/0°/-20°/+20°], resulting in a patch-to-panel stiffness ratio \(S=1.16\), while the ‘compliant’ B/Ep laminate patch (C-BE) has a layup sequence [+110°/70°/90°/70°/110°], resulting in a patch-to-panel stiffness ratio \(S=0.13\). Details of both the patches are shown in Fig. 4.

2.3 Applied Loads

The test sequence for both repairs consisted of a baseline strain survey (to ensure, among other,
proper load introduction), fatigue pre-cracking, and fatigue cycling. The patches were installed, following specific process described in [23] after pre-cracking. The loads used for the fatigue test simulated service load (SL) conditions, including cabin pressurization (8.9 psi) and fuselage vertical bending, and were represented by an equivalent constant-amplitude cyclic loading. The magnitude of the applied loads used in the strain survey and pre-cracking was 75% of the SL conditions that was used in the fatigue test. For the baseline strain survey tests, quasi-static loadings were applied in ten increments up to the maximum loads listed in Table 1. For the pre-cracking and fatigue test, constant-amplitude loading was applied at a frequency of 0.033 Hz with an R ratio (minimum to maximum load) of 0.1. The details of the loads and repair types for each phase are provided in Table 1. The S-BE patch was subjected to 60,000 fatigue cycles while the C-BE patch was subjected to 20,000 fatigue cycles, as the crack growth for the latter repair was very rapid and non-linear.

2.4 Inspection Methods

Several Nondestructive inspection (NDI) methods were used to monitor and record damage formation and growth of the cracks and patch/skin disbonds. For disbond detection, a flash thermography system and a resonance ultrasonic technique were used. Visual inspections and eddy-current systems were used to monitor crack growth. Low frequency eddy current (LFEC) was used on the external surface of the patches and high frequency eddy current (HFEC) was used on the internal surface of the panel. Along with these inspection methods, the strain gages and Digital Image Correlation (DIC) method were used to monitor deformation and strain throughout the tests.

3 Results and Discussion

3.1 Post-Cure Residual Thermal Strains

The complete curing process of adhesive and primer took approximately 190 minutes. During the first 50 minutes, the temperature was ramped up to 250°F from room temperature, then held constant at 250°F for 90 minutes and finally ramped down in last 50 minutes. During the curing process of the repair installation, the strains in the vicinity of the patches were monitored. For this purpose, the strain gages were installed at the patch boundary 0.75-in. away from the edge of the patch both on the outer surface and inner surface of the fuselage skin as shown in Fig. 5. For both B/Ep patches, the residual thermal strains on the inner skin surface, along the patch boundary, are in tension while the corresponding outer skin strains are in compression. The residual thermal strains were much lower in C-BE patch as compared to the S-BE patch as shown in Fig. 6 for a representative strain gage. The figure shows that the S-BE patch causes higher bending at the patch boundary after curing as compared with the C-BE patch.

3.2 Mechanical Strains

3.2.1 Strain gage results

After patch installation the fuselage panel was subjected to fatigue loading as per the load levels listed in Table 1. During fatigue, at every 5,000 cycles, strain survey was conducted to obtain strain values at various locations. As mentioned earlier the strain survey loads were applied quasi-statically up to load levels of 75% of the maximum fatigue loading. The strain gages results for both patches revealed that the installation of the mid-bay patch caused eccentric loading of the mid-bay region. Fig. 7 and Fig. 8 show an example of the strain gage results, after 20,000 fatigue cycles, for both patches at their boundaries and at their center, respectively. Comparing the back-to-back strain gages for both patches, the result shows bending along the patch boundary with the outer surface in tension and inner surface in compression, Fig. 7. In comparison, the amount of bending for S-BE patch was larger than C-BE patch, Fig. 7b and Fig. 7c respectively.

Unlike the strains at the patch boundary, the strains at the center of the patch were complete contrast for both patches. The installation of the ‘stiff’ mid-bay patch caused inward deformation of the mid bay region, yielding high compressive strains at the patch center, Fig. 8b. For the ‘compliant’ mid-bay patch, the patch center had high tensile strains with strain values as high as 12,000 µε at 20,000 fatigue cycles, Fig. 8c.

In the case of the S-BE patch, the bending strains remained relatively constant throughout the fatigue test, as shown by strain gages SE-1 and SE-2 in Fig. 9. The results at the patch center (strain gage PE-1) show that the compressive strain decreased with fatigue cycles. However, these results might be misleading because the strain gages installed at the PE-1 location repeatedly disbonded during the test and repeatedly replaced during the test. A slight variation in the location of the new strain gage
resulted in inconsistent results because of the high compressive strain gradient at the patch center. Similar to the S-BE patch, the bending strains in the C-BE patch also remained relatively constant throughout the fatigue test, Fig. 9. The strain gage results at the patch center showed that the tensile strains increased with fatigue cycles. The high strains at the patch center (PE-1) indicate that some damage occurred. No effort was made to identify and characterize this damage.

3.2 Strain Fields in the B/Ep Patches

Representative results of the full-field hoop strain in the vicinity of S-BE and C-BE patches, measured via DIC at selected fatigue cycles are shown in Fig. 10 and Fig. 11, respectively. The magnitude of applied load used during DIC measurements was 75% of the maximum fatigue load. The patch boundary and the notch and pre-crack are indicated schematically in the figures. The white regions are areas where data could not be processed because of interference from strain gage wires.

For the S-BE patch, the DIC results at 0, 40,000, and 60,000 cycles indicate that the skin at the patch boundaries was in tension (shown by the red and yellow fringe patterns), Fig. 10, while the center of the patch along the notch was in compression (blue fringe pattern). The magnitude of the hoop strain in the vicinity of the patch, along three arbitrarily selected sections (at 0.75, 2.25, and 4.25-in. from the patch’s center), show that the hoop strain varied mostly in the patch’s center, i.e., at the top of the notch (section 1). The strain values ranged from 3,000 με in the skin outside the patch to -1,500 με in the center of the patch. The strain gradient in the vicinity of the notch was quite high: it varied from 0 to -1,500 με within ±0.5 in. off the notch centerline. The magnitude of the strain gradient reduced substantially in sections of the patch ahead of the fatigue crack (section 2) and more so outside the patch, in the skin (section 3). The strain also changed rapidly at the patch boundary, crossing from the skin into the patch (sections 1 and 2). These results indicate significant load attraction along the patch boundaries, yielding nominal stresses in the range of 21 ksi. Further, throughout the fatigue test (of 60,000 cycles), there were no changes in the strain field throughout the patch (within the experimental scatter), which would indicate that there was no load redistribution due to disbonding or crack growth.

Representative DIC results for C-BE repair at 0, 10,000 and 20,000 cycles are shown in Fig. 11. The skin at the patch boundary is under tension (as with S-BE patch), however the patch center was also in tension (unlike the S-BE patch), Fig. 10. The magnitude of the hoop strain in the vicinity of the patch, along three arbitrarily selected sections (at 0.75, 2.25, and 4.25-in. from the patch’s center), are also shown in the figure. The distance of the three sections from the patch center was kept same as the three sections on the S-BE patch. The strains along the sections 1 and 2 increased substantially with increasing fatigue cycles. Over a short distance of ±0.5 inch along this section, the strain ranged from 1,000 με to 12,000 με within 20,000 cycles. The strain gradient did reduce substantially in the region outside the patch, in the skin (section 3). These variation in strains along the patch center indicated load redistribution due to disbonds or fatigue crack growth. As it is discussed below, the load redistribution is attributed primarily to the latter.

3.3 Inspection Results

Flash thermography system and resonance ultrasonic system were used to monitor disbonds in both the S-BE and C-BE patches. Representative results, using flash thermography at the beginning and end of the fatigue test, are shown in Fig. 12 and Fig. 13 for S-BE patches, respectively. Similarly Fig. 14 and Fig. 15 show results for C-BE patch at the start and end of fatigue cycles, respectively. The images are captured at four different time slices, 0.5 sec., 1.5 sec., 2.5 sec. and 4.0 sec. for both patches. The thermography images detected the crack growth during the fatigue test. While thermography and resonance ultrasonic (not shown here) results for S-BE patch showed no indications of any disbonds, the results of C-BE patch showed a small disbond at the aft side of the notch. Monitoring throughout fatigue testing, there were no indications disbond growth from this disbond. The findings of the flash thermography system were confirmed by resonance ultrasonic inspections.

3.4 Crack Growth

The crack growth was measured during the fatigue test using visual and eddy-current inspections. Fig. 16 shows the half crack length growth during fatigue loading under the S-BE patch, on the aft and fwd sides of the notch. Crack growth was self-similar and co-linear (not shown here). Unlike the S-BE patch, the crack growth under C-BE patch was quite rapid, as shown in Fig. 17.
The comparison between the two patches shows that the fatigue crack growth rate of the S-BE patch is significantly slower than that of the C-BE patch, as expected. For the S-BE patch the fatigue crack extended from 1.529 inches to 2.387 inches in 60,000 cycles, at a constant rate of $1.43 \times 10^{-5}$ inch/cycle. In the C-BE patch the crack extended from 1.541 inches to 3.186 inches in just 20,000 fatigue cycles at an increasing rate, Fig. 18. Near 20,000 the crack growth rate was one order of magnitude higher, i.e., $1.4 \times 10^{-4}$ inch/cycle. In addition, a comparison of the measurements made during fatigue pre-cracking, before the patches were installed and during fatigue at SL conditions after the patches were installed, is shown in Fig. 19. The results show that the S-BE patch was quite effective in reducing the crack growth rate. On the other hand, the effectiveness of the C-BE patch reduced with fatigue cycles. At 10,000 cycles the crack growth rate was similar to the rate measured during fatigue pre-cracking, increasing to approximately double that rate at 20,000 cycles. It should be recalled that during fatigue pre-cracking the loads applied were just 75% of the actual loading simulating SL conditions.

4 Conclusion

Full-scale fatigue test was performed using a retired Boeing 727 fuselage panel with bonded repairs subjected to realistic flight load conditions. The effect of stiffness of the single-sided bonded Boron/Epoxy (B/Ep) composite patches on the fatigue crack growth in the aluminum fuselage skin was studied. Two extreme scenarios were considered, one using a ‘stiff’ B/Ep patch and the other a ‘compliant’ B/Ep patch. Both patches experienced bending at the boundary during the curing process and during fatigue loading, with the ‘stiff’ B/Ep patch experiencing larger bending than ‘compliant’ B/Ep patch. Further, the fatigue loading caused inward deformation of the mid bay region causing high compressive strains at the center of ‘stiff’ B/Ep patch. Due its low stiffness, there was little if any inward deformation at the center of the ‘compliant’ boron/epoxy patch. Instead, the patch center experienced tensile strains, which increased with the increasing fatigue cycles. As expected, the fatigue crack growth for the ‘compliant’ patch was rapid and progressed non-linearly, while for the ‘stiff’ B/Ep patch, crack progression was slow and constant. The results show that the ‘stiff’ B/Ep patch could significantly reduce the crack growth rate: In this study the difference was one order of magnitude at 20,000 cycles.

Acknowledgments

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References

Table 1. Summary of applied loads.

<table>
<thead>
<tr>
<th>Description (Test Phase, applied load, number of cycles)</th>
<th>Load Phase</th>
<th>Maximum Load</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Pressure (psi)</td>
</tr>
<tr>
<td>Strain survey, 75% of SL</td>
<td>Quasi-static</td>
<td>6.7</td>
</tr>
<tr>
<td>Fatigue precracking, 75% of SL</td>
<td>Cyclic, R = 0.1</td>
<td>6.7</td>
</tr>
<tr>
<td>Fatigue, SL</td>
<td>Cyclic, R = 0.1</td>
<td>8.9</td>
</tr>
</tbody>
</table>
Fig. 7. Comparison of bending at the patch boundary after 20,000 fatigue cycles for both patches, showing larger bending effect in the S-BE patch (locations of strain gages are indicated in the inserted schematic).

Fig. 8. Strains at the center of the patches showing high compressive strain in the S-BE patch and high tensile strains in the C-BE patch (locations of strain gages are indicated in the inserted schematic).
Fig. 9. Strains at the vicinity of the patches throughout fatigue (locations of strain gages are indicated in the inserted schematic).

Fig. 10. Strain field in the vicinity of S-BE patch (measured via DIC along three sections), showing that the skin at the patch boundary was under tension and center region under compression.

Fig. 11. Strain field in the vicinity of C-BE patch, (measured via DIC along three sections), showing that both the skin at the patch boundary and the center of the patch were under tension.
Fig. 12. Images of S-BE repair patch captured using Flash Thermography system at 0 cycles at different time slices.

Fig. 13. Images of S-BE repair patch captured using Flash Thermography system at 60,000 cycles at different time slices.

Fig. 14. Images of C-BE repair patch captured using Flash Thermography system at 0 cycles at different time slices.

Fig. 15. Images of C-BE repair patch captured using Flash Thermography system at 20,000 cycles at different time slices.

Fig. 16. Fwd and aft half crack length for S-BE patch showing slow crack growth.

Fig. 17. Fwd and aft half crack length for C-BE B/Ep patch showing rapid crack growth.
Fig. 18. Comparison of fatigue crack growth for S-BE and C-BE patches

Fig. 19. Comparison of fatigue crack growth for S-BE and C-BE patches