BONDED REPAIRS TO SURFACE FLAWS

R. Jones and R. J. Callinan

Structures Division, Aeronautical Research Laboratories, Defence Science and Technology Organisation, Melbourne, Australia

ABSTRACT

This paper describes two recently developed methods for the repair of surface flaws. Both involve the use of adhesive bonding and have been found to lead to dramatic increases in fatigue life.

KEYWORDS

Surface cracks, adhesive bonding, crack patching, finite element analysis.

INTRODUCTION

The Aeronautical Research Laboratories (ARL) in Australia has pioneered the use of adhesively bonded boron fibre reinforced plastic (BFRP) patches to repair cracks in aircraft components (Baker 1978). This procedure has been successfully used in several applications to RAAF aircraft, including the field repair of fatigue cracks in the lower wing skin of Mirage III aircraft (Jones 1982) and in the landing wheels of Macchi aircraft (Baker 1978). In each case, repairs were made by adhesively bonding a BFRP patch to the component with the fibres spanning the crack, the aim being to restrict the opening of the crack under load thereby reducing the stress intensity factors and thus preventing further crack growth.

The repair to the Macchi landing wheel was particularly interesting since it was, at that time, the first attempt to repair a surface flaw in a thick section. Since then, several other repairs have been designed for cracks in thick sections some of which are discussed by Jones et al (1982). In each case the section thickness has been 12. mm.

The present paper begins by presenting the reductions in the stress intensity factors, achieved by the use of BFRP patches, for surface flaws in a rectangular bar of aluminium 12 mm thick.

An extirely new scheme is then presented for repairs to cracked bolt holes. This procedure involves the use of a bonded insert. Numerical and experimental studies show this procedure to be particularly effective.

REPAIR OF SEMI-ELLIPTICAL SURFACE FLAWS

In recent years, a number of BFRP patches have been designed at ARL for surface flaws in thick sections, eg. the repairs to the Macchi and Mirage III main landing wheels and the repair to the console truss in F111 aircraft (see Table 1). In each case the cracked section was 12. mm thick.

TABLE 1 Current Applications of Crack-Patching

Cracking	Material	Component	Aircraft	Comments
Stress-corrosion	7075-T6	Wing plank*	Hercules	Over 300 repairs since 1975
Fatigue	Mg Alloy	Landing wheel*	Macchi	Life doubled, at least
Fatigue	2024-T3	Fin skin	Mirage	In service since 1978
Fatigue	AU4SG	Lower wing skin*	Mirage	Over 500 repairs since 1979
Fatigue	2024-T3	Upper wing skin	Nomad (fatigue test)	Over 105900 simulated flying hours
Fatigue	2024-T3	Door frame	Nomad (fatigue test)	Over 106619 simulated flying hours
Stress-corrosion	7075-T6	Console truss	F111	Recent repair
Lightning burn	2024-T3	Fuselage skin	Orion	Recent repair+

* Now RAAF standard procedures.

\+ CFRP patches and ambient temperature curing epoxy adhesive to simulate rapid repair of battle damage.

In the case of the Mirage and Macchi landing wheels the repairs are installed when the crack reaches a total length of 24 mm. In each case the cracks were found to be nearly semi-elliptical in shape with a surface length of 24 mm and a maximum depth of 6 mm. In order to study the effect on such a crack an initial investigation was undertaken on the repair of a similar semi-elliptical crack centrally located in a rectangular block of aluminium with dimensions as shown in Fig. 1 (in this figure only one quarter of the structure is shown). The block was subjected to a uniform uniaxial stress and the effect that various boron fibre patches had on the crack were calculated using a detailed three dimensional finite element analysis. Table 2 shows the calculated values of the stress intensity factors at point d, the point of deepest penetration, and s, the point at which the crack intersects the free surface. The fibre stresses σ_f are a maximum over the crack and

vary through the thickness of the patch. These values are also shown in Table 2 along with the peak adhesive stresses over the crack. In this study the adhesive was taken to be AF126, an epoxy nitrile, 0.1016 mm thick with a shear modulus of 0.7 GPa.

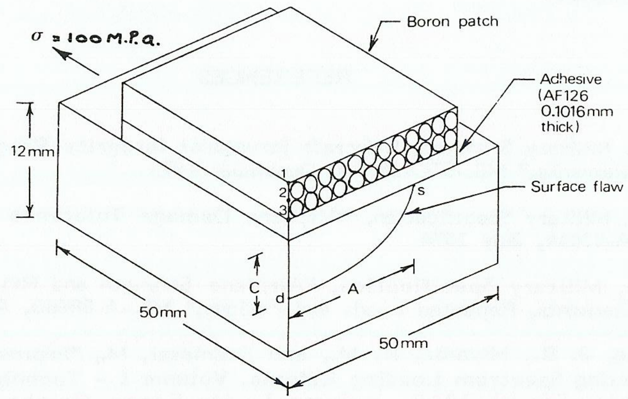

Fig. 1. Repair of surface flaw (1/4 structure modelled)

Table 3 shows the corresponding values of stress intensity factors, fibre stresses and adhesive stresses for the case when the surface flaw is semi circular, rather than semi-elliptical, with a surface length of 12 mm.

Using these Tables it is possible to design a repair for surface flaws up to 24 mm in surface length. The major design considerations are:

1. The maximum stress intensity factor should be as low as possible and preferably below the critical value for fatigue crack growth in the material.

2. The maximum adhesive stresses should be below the value of which fatigue damage accumulates in the adhesive. For AF126 which is an epoxy nitrile, this is 35 MPa.

3. The average stress through the thickness of the boron patch should not exceed 1000 MPa.

This failure rule is a modification of the average stress failure criterion which is commonly used for composite materials.

There will be circumstances in which it is not possible to satisfy all three of these design rules using the unidirection boron fibre patches which are considered here. In these cases the possibility of using a purpose designed unbalanced patch such as first described by Jones (1983) should be investigated.

TABLE 2 Semi-Elliptical Flaw A = 12 mm, C = 6 mm

Unpatched Values at d K_1 = 12.45 MPa $\sqrt{m}$, at s K_1 = 12.5 MPa $\sqrt{m}$

Number layers Boron	Stress Intensity factors K_1 at:		Fibre stress over crack σ_f/σ applied at points:			Adhesive Shear Stress over crack τ/σ applied
	d	s	1	2	3	
5	8.33	4.396	3.481	4.256	5.293	0.451
10	7.222	3.463	1.918	2.572	3.670	0.353
15	6.662	2.979	1.271	1.810	2.907	0.304
20	6.365	2.709	0.874	1.364	2.488	0.277
25	6.211	2.558	0.587	1.073	2.244	0.262

TABLE 3 Semi-Circular Flaw A = C = 6 mm

Unpatched Values at d K_1 = 9.06 MPa $\sqrt{m}$, at s K_1 = 11.38 MPa $\sqrt{m}$

Number layers Boron	Stress Intensity factors K_1 at:		Fibre stress over crack σ_f/σ applied at points:			Adhesive Shear Stress over crack σ/σ applied
	d	s	1	2	3	
5	6.882	5.412	3.333	3.966	4.811	0.402
10	6.232	4.552	2.113	2.672	3.615	0.337
15	5.842	4.035	1.541	2.005	2.972	0.300
20	5.605	3.712	1.153	1.576	2.582	0.276
25	5.465	3.509	0.852	1.276	2.337	0.261

At this stage it should be stressed that this work represents the first comprehensive analysis of the repair of surface flaws using BFRP batches.

REPAIR OF CRACKED HOLES

The fatigue life of cracked holes is of major interest to the aerospace industry and a great deal of effort has been spent on developing suitable repair schemes. It was in this context that discussions between the authors and Dr Baker, Materials Division ARL, led to the concept of first reaming out the crack, where possible, and then either bonding a steel bush into the hole, in the case of a loaded bolt hole, or in the case of a rivet hole, bonding the rivet into the hole, see Jones et al (1981).

In some instances edge distance considerations may not permit the complete removal of the crack. As a result it may be necessary to be able to inspect down the bore of the hole. Fortunately modern N.D.I. techniques are now capable of inspecting through a steel bush.

To study the feasibility of such a repair, a detailed finite element analysis was undertaken on a cracked lug. The geometry of the lug was as shown in Fig. 2. Two different crack lengths were considered, viz: 1.59 mm and 3.18 mm. The lug was taken to be of 7079-T6 aluminium alloy and was loaded via a rigid pin using the methods developed by Callinan (1977). The steel bush was bonded using an adhesive with a Youngs modulus of 2.5 x 10^3 MPa and a Poissons ratio of 0.32. The resultant finite elemtnt model of the repaired lug used 814 nodal points and utilized the special crack tip elements developed at ARL.

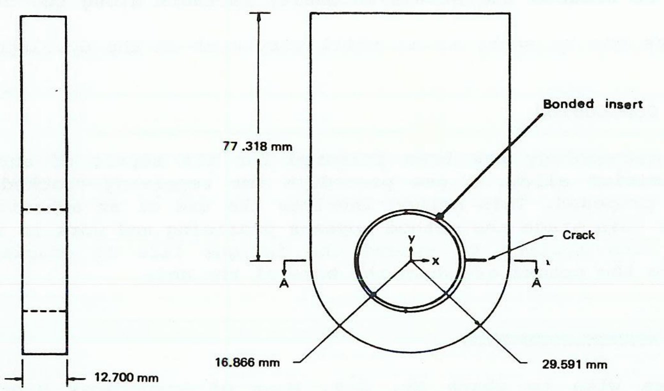

Fig. 2. Dimensions of lug

The effect that this repair has on the stress intensity factor K_I is shown in Table 4. Here K_{ip}/K_{iu} is the ratio of the stress intensity factors before and after the sleeve was inserted whilst $\tau A/P$ and $\sigma A/P$ are the non dimensionalized values of the peak shear stress and the maximum compressive stresses developed in the bond. Here A is the internal surface area of the hole and P is the total load applied to the lug.

TABLE 4

Crack length	Adhesive	Sleeve thickness	$\tau A/P$	$\sigma A/P$	K_{ip}/K_{iu}
1.59 mm	0.1 mm	1.0 mm	0.47	2.29	0.50
3.18 mm	0.1 mm	1.0 mm	0.59	2.29	0.46
1.59 mm	0.3 mm	3 mm	0.68	1.31	0.36

We thus see that significant reductions in the stress intensity factors are obtained by the use of a bonded sleeve.

Following the success of this initial study, a series of laboratory tests was undertaken (Mann et al (1984)). These tests compared, amongst other things, repairs to a 3.3 mm rivet hole. In one set of tests the hole was reamed to 4.0 mm removing any cracks present and a rivet was then bonded into the hole using a paste adhesive. In another series of tests an uncracked hole was expanded by cold working to 4 mm, and the rivet was inserted but not bonded. The average life in the first series of tests, with the bonded rivet, was 2.19 times greater than for the second series of tests.

As a result of these experimental and numerical studies into bonded sleeves it appears that this repair procedure may be a valuable addition to the methods currently available for the repair of cracked holes. The main advantages of this method are:

1. It reduces the fretting at the hole;
2. It reduces the stress intensity factaors along the crack front.

However, as can be seen, it is still very much in the development stage.

CONCLUSION

A design methodology has been prsented for the repair of surface flaws in an aluminium alloy. A new procedure for repairing cracked holes has also been proposed. This method involves the use of an adhesively bonded sleeve. At this stage the method appears promising and work is in progress to assess its ability to extend the fatigue life of cracked fastener holes where the cracks are down the bore of the hole.

ACKNOWLEDGEMENTS

The authors wish to thank Mr. J.Y. Mann of Structures Division, ARL, for access to his test results on the fatigue life of the bonded rivet hole specimens.

REFERENCES

Baker, A.A. (1978). Work on advanced fibre composites at the Aeronautical Research Laboratories, Australia. Composites, 9, 11-16.
Callinan, R.J. (1977). Residual strength of a cracked lug. Structures Note 422, Aeronautical Research Laboratories, Australia.
Jones, R. (1983). Neutral axis offset effects due to crack patching. J. Composite Structures, 2, 163-174.
Jones, R., and R.J. Callinan (1981). New Thoughts on stopping cracks which emanate from holes. Int. J. Fractaure 17, R53-R55.
Jones, R., M. Davis, R.J. Callinan, and G.D. Mallinson (1982). Crack patching: analysis and design. J. Struct. Mech., 10, 177-190.
Mann, J.Y., R.A. Pell, R. Jones, and M. Heller (1984). The use of adhesive bonded rivets to lessen the reductions in fatigue life caused by rivet holes. Structures Report 399, Aeronautical Research Laboratories, Australia.