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Platforms Sciences Laboratory

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ABSTRACT

Currently an increasing number of RAAF F/A-18 aircraft are developing acoustic fatigue cracks on the rear fuselage. Acoustic fatigue is the result of high frequency lateral vibration of an aircraft panel as a result of time varying pressure waves caused by engine and/or aerodynamic effects. While the exact broadband power spectral density (PSD) spectrum is not available for this area of the the F/A-18, the PSD is known for the inlet nacelle area in which cracking has occurred. Cracking in the inlet nacelle proceeded cracking in the aft fuselage by a period of several years. As a result the inlet nacelle experiences a more severe spectrum and is the PSD adopted here. Viscoelastic damping materials are used in this repair in conjunction with the constrained layer technique to reduce the lateral vibration. Since viscoelastic materials have a limited temperature range it has been necessary to use two types of damping materials to cover the expected temperature range. A highly damped repair is proposed for the aft fuselage area and is shown to be significantly effective in reducing the crack growth rate.

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Executive Summary

Currently an increasing number of RAAF F/A-18 aircraft are developing acoustic fatigue cracks on the rear fuselage. Acoustic fatigue is the result of high frequency lateral vibration of an aircraft panel as a result of time varying pressure waves caused by engine and/or aerodynamic effects. While the exact broadband spectrum is not available for this area of the the F/A-18, the acoustic excitation spectrum is known for the inlet nacelle area in which cracking has occurred. Using this spectrum results in higher stress intensity factors in the aft fuselage than the inlet nacelle. Since cracking in the inlet nacelle proceeded cracking in the aft fuselage by a period years a conservative assumption is to use the spectrum from the nacelle inlet. An assessment of the thermal profile of the F/A-18 together with high engine power results in a temperature operating range of 0-140°C. Viscoleastic damping materials are used in the repair to damp out vibration of the panels and cover a limited temperature range. As a result it has been necessary to use two different damping materials to achieve the required performance. Validation of crack growth laws from limited acoustic fatigue data has enabled predictions of crack growth. Also the softening of the adhesive over the temperature operating range has been found to to increase the crack growth rate. Furthermore from closed form solutions it has been possible to predict the contribution to the crack growth rate as a result of thermal residual stresses. At low temperatures the resulting thermal stress intensity factors are comparable to those predicted as a result of acoustic fatigue, at high temperatures the reverse holds. However the overall prediction of the crack growth rate for a damped repair in comparison to a cracked un-repaired plate shows a reduction of crack growth by a factor of 500 up to 80°C and a factor of 70 at 140°C. Furthermore the components of this repair will not be damaged as a result of the extreme temperature range. As a result the highly damped repair proposed for this area has been shown to be very effective in reducing the crack growth rate.

Contents

Authors

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Mr. R.J.Callinan is a senior research scientist and graduated from RMIT (Aero. Eng.) in 1969 and from Monash University in 1971 (Civil. Eng.) and completed a M.Eng. Sc. in 1981 at Melbourne University. His work has been in the areas of finite element analysis, fracture mechanics and structural mechanics of composite and bonded repairs, military aircraft accident investigations. He has also been involved with design studies of low radar cross-section battlefield surveillance R.P.V.'s. In 1985 he was seconded to the USAF at Eglin AFB for 18 months, to carry out vulnerability studies on composite structures. More recently he has been involved in a specific program on validation of bonded repairs to RAAF aicracft, and bonded repairs subject to acoustic fatigue.

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Dr Galea graduated in 1980 with a Bachelor of Engineering (Mech) from the University of Queensland with first class honours and in 1983, he received a Masters of Engineering Science. He commenced employment with the Aeronautical Research Laboratory in 1983. In 1985 he commenced studies at the Institute of Sound and Vibration Research, University of Southampton, UK and received his Doctor of Philosophy from the University of Southampton in 1989. Dr Galea was appointed a Research Scientist in 1990 and Senior Research Scientist in 1992. Since 1990 Dr Galea has been working in the areas of composite structures, computational and experimental mechanics. He is currently managing the smart materials/structures for airframes task, at the Aeronautical and Maritime Research Laboratory. His other areas of research include bolted and bonded composite joints, repairs to acoustically-induced cracked structures and structural health monitoring.

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Mr. Sanderson has worked at AMRL since 1981. He has developed flight data reduction & analysis software for Mirage, F-111 & F/A-18 projects. Several of these programs have been implemented by NAE for part of their data reduction in the IFOSTP project. Since 1992, Mr. Sanderson has undertaken finite element analysis of composite and bonded structures for the F-111 and F/A-18 aircraft and provided training in finite element modelling and analysis.

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1. Introduction

Acoustic fatigue is due to a very high intensity excitation as a result of pressure waves caused by either engine/or aerodynamic effects. Acoustically-induced cracking has occurred on the external surface of the lower nacelle and aft fuselage skins on the F/A-18 as illustrated in Fig. 1. In the inlet nacelle region overall sound pressure levels (OASPL) greater than 170 dB have been measured in flight. In this case high sound pressure levels appear to be a result of an aerodynamic disturbance at the inlet lip [1]. In general typical cracks occur along a line of rivets or run parallel to the rivet line and may turn into the centre of the panel, as shown in Fig.1. Cracking generally occurs along the longer side of the panel where the bending stresses due to out of plane vibrations are a maximum. This report focuses on the aft fuselage, in which an increasing number of F/A-18's are exhibiting acoustic fatigue cracks.

The standard repair for such cracking is to incorporate intercostals or additional stiffeners in the panel. This has two effects; firstly to reduce the panels response, i.e. lower stress, for a given load and secondly it increases the resonant frequencies of the panel to frequencies well outside the recorded excitation frequencies.

In order to reduce the time and hence cost of repairing such cracked structures a bonded composite repair would be preferred. The benefits of such a repair are reflected in the estimated times required to carry out the various repairs. For the aft fuselage the time required to carry out the mechanical repair is 15-30 hours spread over 3 or 4 days. The mechanical repair also requires engine removal and installation which takes a three man team approximately 8 hours, followed by engine ground runs. In this case the mechanical repair consists of mechanically fastening a metallic doubler over the panel and is not a long term repair and eventually the replacement of a large area of skin will be required. The benefits of a bonded repair are that this repair would take 15 -25 hours spread over 2 days without having to remove the engine. Furthermore the bonded repair is expected to be a long term solution and the bonded repair does have the potential to reduce the repair time to 8 hours for reinforcement of uncracked structures to prevent cracking from occurring. It is expected that a highly damped repair will be a cost effective means of repairing cracked aircraft structures subjected to acoustic fatigue.

The work reported here will involve the estimation of the root mean square (r.m.s.) response of the stress intensity factor (K) in the cracked and cracked/repaired cases in order to understand the problems involved in patching cracks in such an environment. In particular the integrity of the repair will be considered over the expected operating temperature range. Also the adhesive shear and peel stresses bonding the boron epoxy to the aluminium are evaluated. Various configurations are considered including the techniques of constrained layer damping (CLD).

While the techniques of CLD are not new [2] it has taken a considerable time for aircraft applications. Highly damped repairs such as the durability patch have been recently proposed in [3, 4] and in [5, 6] a demonstrator has been applied to a lower centre fuselage skin panel (un-cracked) on an F-15 aircraft.

The major difference between the present work and the work previously presented in the study of the inlet nacelle cracking, [7, 8], is the influence of temperature on the viscoelastic response, residual stresses and softening of the adhesive.

2. Cracking in aft fuselage

The chemically milled steps for panels in the aft fuselage are shown dotted in Fig. 1. Cracks occur in the chemically milled step region, and are generally parallel to and midway along the longest side. The aft fuselage section shown in Fig. 1 contains a composite of all known crack locations and occurs between fuselage stations Y557.500 and Y598.000. The conventional repair involves removal of the crack and use of a metallic doubler attached by mechanical fasteners. To carry out this type of repair the engine needs to be removed.

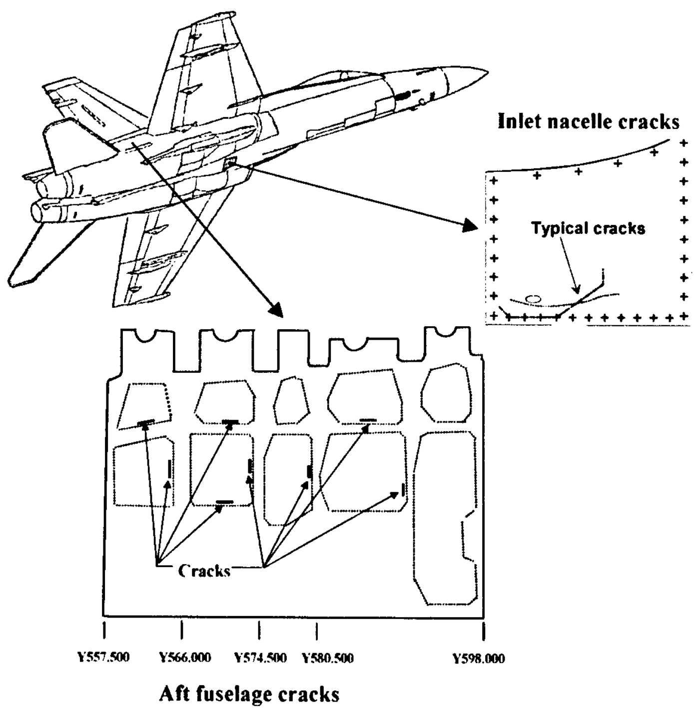

Figure 1. Acoustic fatigue cracking on the F/A-18

3. Sound pressure levels

Since the cracking on the inlet nacelle preceded that in the aft fuselage then it is conservative to assume the same power spectral density (PSD), as measured at the nacelle as shown in Fig. 2. The overall sound pressure (OASPL) measured at the nacelle was 172db, which is considerably greater than the design value of 160 dB for the airframe, [1]. Premature cracking in the aft fuselage suggests that an additional aerodynamic disturbance is responsible for the higher than expected sound pressure level (SPL). The possibility exists that the flap may be responsible for this. Fig. 3a and 3b show the flap deflection for landing and in-flight conditions. Note that the in-flight position corresponds to the location of the metallic repair. In the case of the B52, [9], it was found that cracking in the aft fuselage skin was a result of the aerodynamic disturbance caused by the deflected flap position during take-off. However aerodynamic disturbances can also be caused by high angle of attack manoeuvres resulting in separation of airflow over the flap.

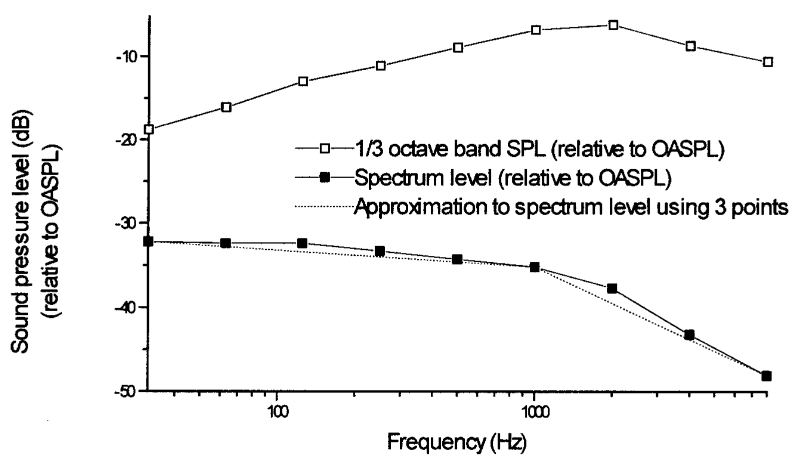

Figure 2. Spectrum and one-third octave band levels of sound pressure over the external nacelle inlet, (where OASPL=170 db), [1]

Table 1 Power Spectral Density for inlet nacelle

Frequency (Hz)	**Three points for curve (dB)**	**Pressure spectrum level (dB)**	$S_l(f)$ **(MPa)^2/Hz**
31.5	-32.2	140	4.0 x10^{-8}
1000.	-35.2	137	2.005x10^{-8}
8000.	-48.1	124.1	1.028x10^{-9}

The relationship between the SPL and the r.m.s. fluctuating pressure (p) is given in [10] as:

$$p_{rms} = 10^{(SPL/20-4.69897)} \qquad (1)$$

and PSD of acoustic pressure, i.e. PSD of the excitation, at any given frequency is given by:

$$PSD = p_{rms}{}^{2} = 10^{(SPL/10-9.3979)} \qquad (2)$$

The spectrum curve in Fig. 2 has been approximated with the three points listed in Table 1.

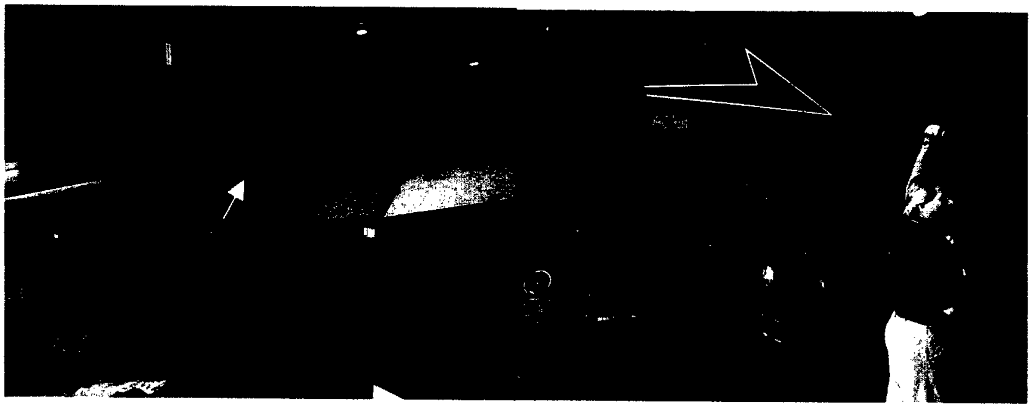

Figure 3a (Left) Metallic repair of acoustic fatigue cracking (white arrow). Also the flap is fully deflected position for landing.
Figure 3b (Right) Flap fully retracted showing the in-flight position

4. Random response analysis

The random response analysis capability of the NASTRAN program has been used to solve this problem [11]. This involves a solution in the frequency domain after the transfer function, $H(\omega)$, is generated. Together with the PSD of the excitation, $S_I(\omega)$, the PSD of the response, $S_J(\omega)$, is determined:

$$S_J(\omega) = |H(\omega)|^2 S_I(\omega) \qquad (3)$$

This analysis allows the statistical properties of the system to be evaluated. Random vibrations considered here involve all frequencies at any one instant in time. After calculating the PSD, the root mean square (r.m.s.) of the response can be determined by computing the square root of the PSD area:

$$j_{rms} = \sqrt{\frac{1}{2\pi}\int_0^\infty S_J(\omega)\, d\omega} \qquad (4)$$

A similar application of finite element techniques to undertake a PSD analysis to acoustic fatigue problems has been reviewed by Climent and Casalengua [10]. The response of interest are the displacements near the crack tip from which stress intensity factors are computed. For the adhesive the quantities of concern are shear and peel stresses. These quantities are considered in section 11.

5. Finite element model

In order to study the cracking mechanisms of the cracked and repaired/cracked aft skin cases, a simplified model has been developed in which the skin is considered to be a flat rectangle. Most of the skin panels in which cracking occurs have a chemical milled step in which the panel is reduced in thickness to 0.91mm. While the width and height dimensions of each panel vary, the F.E. model has representative dimensions as shown in Fig. 4. Also shown in Fig. 4 the local uni-directional patch extends partially across the panel, but since the crack lies near the boundary the repair will continue across the adjacent panel as shown in the dotted outline. In order to accurately represent the bending of the panel, 2 layers 20 noded brick elements have been used through the thickness. The F.E. mesh is shown in Fig. 5. The local boron/epoxy repair [O_3] has been applied to make up for material lost as a result of the crack. The highly damped section of the repair covers the entire panel including the local boron patch. This section consists of two viscoelastic layers with two boron/epoxy [0/90] constraining layers as shown in Fig. 6. The viscoelastic layers and adhesive have been modelled using 3D elements as are the constraining layers. The total structure consists of 7000, 20 noded brick, elements and 29000 nodes.

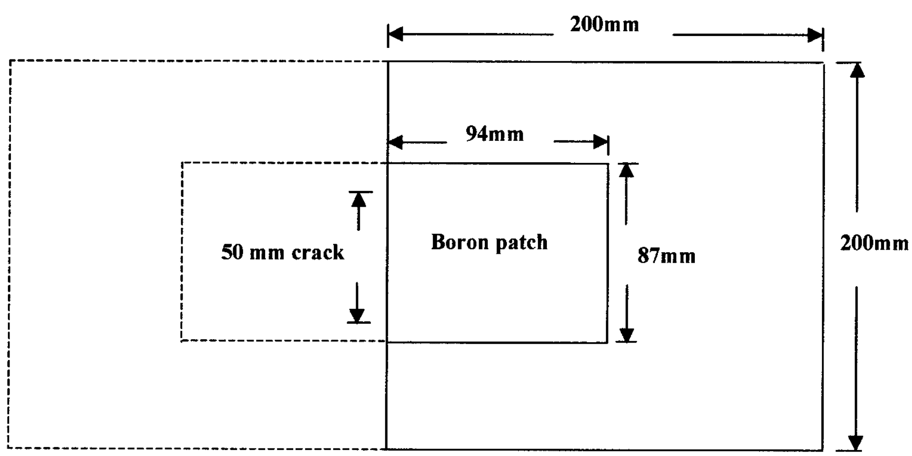

Figure 4. Dimensions of repair, plan view

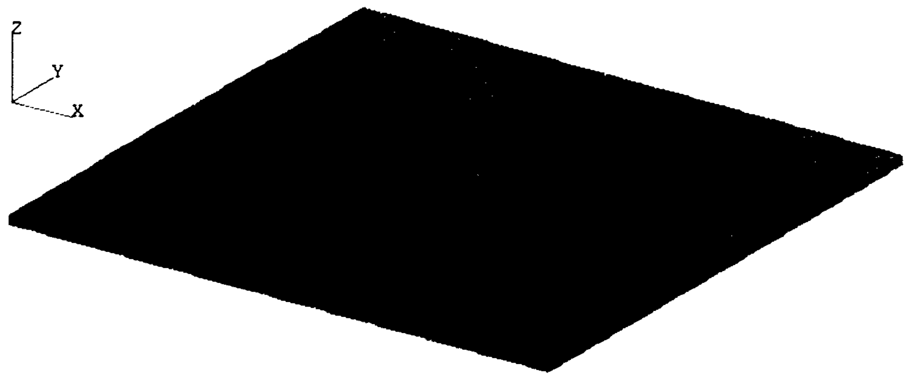

Figure 5. Finite element model of multi-layer highly damped repair, containing a crack of length 50mm

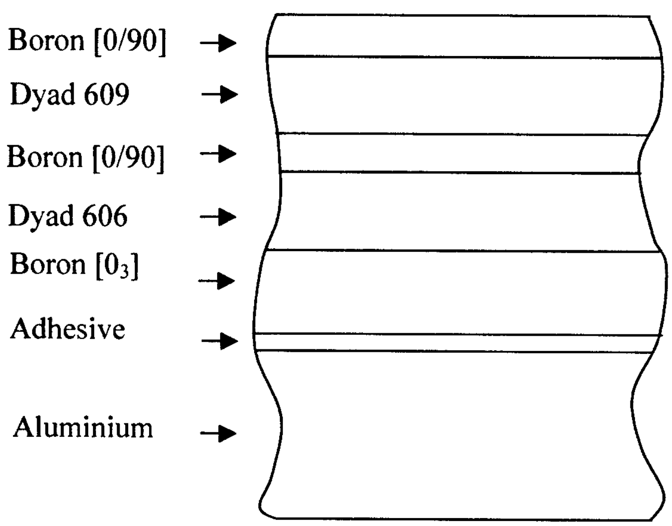

Figure 6. Components of the highly damped repair

6. Computation of Stress Intensity

Since the skin thickness is approximately *1 mm*, the condition of plane stress is assumed. The computation of the stress intensity factor will be determined directly from the crack opening displacements (COD) near the crack tip using displacements from the PSD analysis. The r.m.s. crack tip stress intensity factors for mode *I* are derived from the standard asymptotic relation:

$$K_{rms} = \frac{EU_{rms}}{4}\sqrt{\frac{2\pi}{l}} \qquad (5)$$

where K_{rms} is the root-mean-square stress intensity factor evaluated at the desired through thickness location

E is Young's modulus

U_{rms} is the root-mean-square of the crack opening displacement at distance l from the crack tip (perpendicular to the plane of the crack) and at the desired through thickness location.

For the plate two elements will be used through the thickness while one element is used through the thickness for the adhesive and patch. The size of the crack tip element to be used will be selected such that $l << 1/k$, where $k = (\sigma_b / K_{\max})^2 \approx 1.0mm$, where σ_b is the far field bending stress, and a typical value of K/σ_b will be taken as 0.033, [12].

7. Modes and frequencies

A modal analysis has been carried out using NASTRAN program, [11], to determine the mode shapes and frequencies. The first four mode shapes are shown below in Fig. 7. The PSD of the response is plotted in Fig. 8 From this curve it is clear that mode 1 provides the greatest contribution to the response at a frequency of 203Hz. The mean square contribution of the other modes is more than one order of magnitude smaller than that for mode 1.

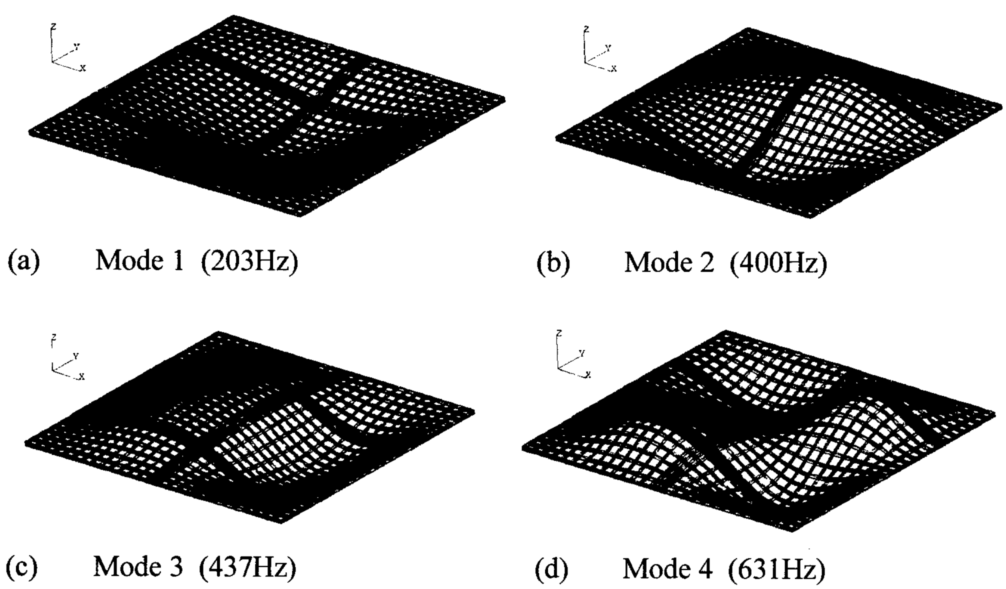

Figure 7. Modes and frequencies for repaired plate

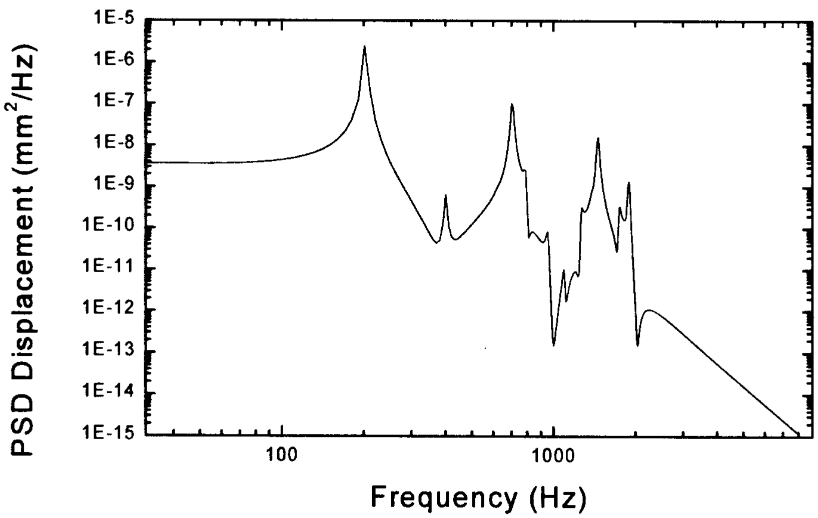

Figure 8. PSD of the response (i.e. mean square of the displacement near the crack tip)

8. Acoustic fatigue crack growth data

In this study it has been necessary to show that standard crack growth laws can be applied to structures subject to high frequency out-of-plane bending excitation. While acoustic fatigue data is very limited a comparison has been made with L73 data (U.K. spec.) and a standard crack growth law. The L73 (clad) alloys, [13], shown in Fig. 9, is equivalent to 2014-T6. The crack growth law for 2014-T6 is given by [14] as:

$$\frac{da}{dn} = \frac{C_f (\Delta K)^m}{(1-R)K_f - \Delta K} \quad \text{(mm/cycle)} \qquad (6)$$

where $C_f = 1.0x10^{-5}$, $m = 2.87$, $K_f = 59.9$, $R = \sigma_{\min} / \sigma_{\max}$ and here ΔK is assumed to be equal to ΔK_{rms}

The crack growth law is usually determined from testing at a frequency of typically 4-10 Hz. Also equ(6) is only valid for a range of 6-50 MPa $m^{1/2}$. A comparison between the crack growth law prediction and the 2014-T6 data is shown in Fig. 10. The experimental data corresponds to a sinusoidal data at 175Hz and shows surprisingly good agreement. The random data has not been used since the slopes of the curves show significant disagreement, with similar strain levels.

To determine results for 7075-T6 consider equ(6) with the following coefficients: $C_f = 1.37x10^{-5}$, $m = 3.02$, $K_f = 63.9$ which have been determined experimentally.

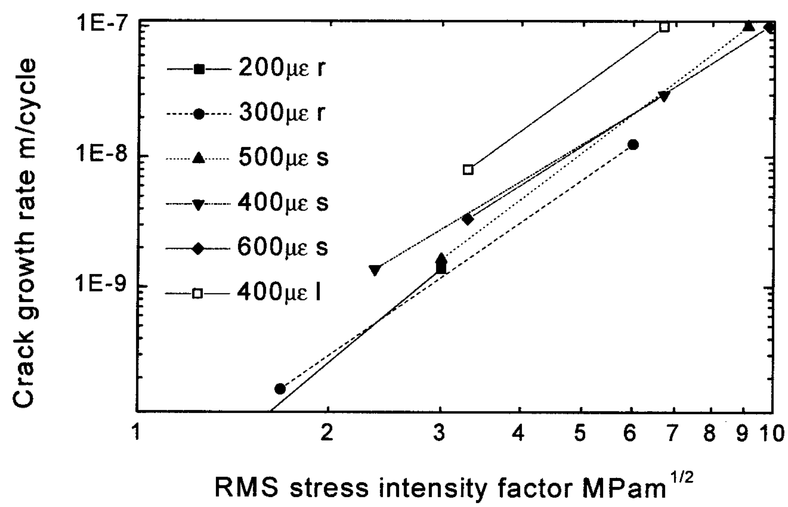

Figure 9. Acoustic fatigue crack growth data for clad 2014-T6, ($\mu\varepsilon$ denotes microstrain, r *denotes random,* s *denotes sinusoidal and* l *denotes low reversal rate).*

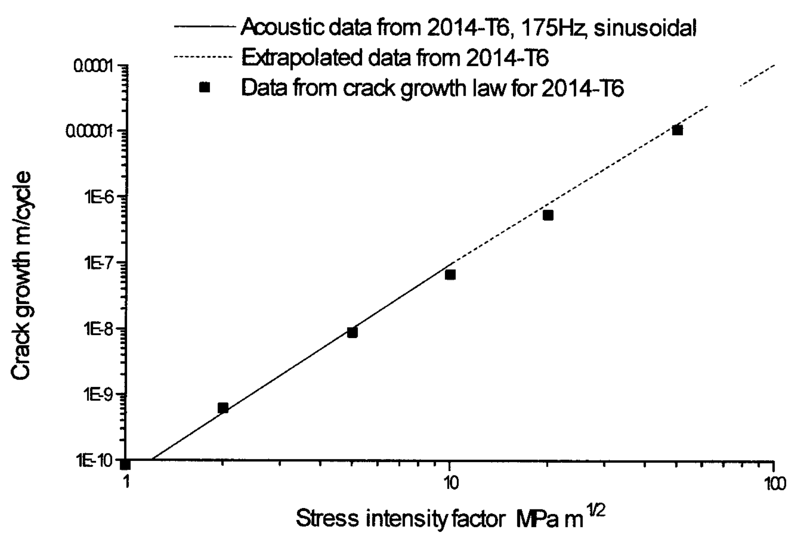

Figure 10. Crack growth data and crack growth law

9. Residual thermal stresses

Thermal stresses arise due to the different coefficients of thermal expansion of the materials involved and may be significant at the extremes of the operating temperatures. Furthermore it is known that the residual thermal stresses due to the bonding process may be significant. Typically the bonding process results in curing at a temperature of approximately 80-120^{O}C. In the case of a boron/epoxy repair to an aluminium plate there is a large difference in the coefficients of thermal expansion. As a result the residual stress intensity for a repair involving boron/epoxy is comparable to the stress intensity for the out-of-plane vibration. At high operating temperatures this effect is alleviated. However the shear modulus of the adhesive, bonding the boron/epoxy to the plate, softens and increases the stress intensity factor. As a first approximation the residual stress in the plate beneath the repair is given by [15] as:

$$\sigma_O = \frac{E_1 s}{1+s}(\alpha_1 - \alpha_2)|\Delta T| \qquad (7)$$

where:

α_2 is the coefficient of thermal expansion of the boron/epoxy,

α_1 is the effective coefficient of thermal expansion of the plate,

E_O is Young's modulus of the reinforcement,

E_1 is Young's modulus of the plate

ΔT is the change from cure to operating temperature

In this case $E_1 = 71000.$ MPa , $E_0 = 207000.$MPa,

$$\alpha_1 = \frac{\alpha}{(1+\nu)} = 15.3x10^{-6} /^O C \text{ and } \alpha_2 = 4.2x10^{-6} /^O C$$

The relative stiffness of a repair is given by: $s = E_O t_o / (E_1 t_1)$

where :

t_1 is the thickness of the plate

t_o is the thickness of the reinforcement

The relative stiffness of a normal repair (s) usually is a value of 1.0. However in the construction of a highly damped repair a boron patch is firstly applied over the cracked section of the aluminium panel. The damping treatment is then applied to the complete panel (and adjacent panel). Since the basic boron repair involves [O_3] plies while the two constraining layers will have[0/90] plies then the value of $s \approx 2$. Knowing the variation of K / σ_O allows the determination of K using equ(7):

$$K = (\frac{K}{\sigma_O})\frac{E_1 s}{(1+s)}(\alpha_1 - \alpha_2)|\Delta T| \qquad (8)$$

A value of $K / \sigma_O = .033$ has been assumed, [12].

10. Damping

The loss factor for a viscoelastic material is the measure of the damping. Constrained layer damping involves the dissipation of energy by shear of a viscoelastic material as a result of a constraining layer (in our case [0/90] boron). Viscoelastic materials are suitable over a limited temperature range. In the case considered here it has been necessary to use two different viscoelastic materials to cover the expected temperature operating range. Two commercially available structural damping materials have been evaluated, such as Soundcoat and Isodamp. From this testing the Soundcoat Dyad 606 and 609 [16], have been chosen as being the most suitable materials that are available for this application. The manufactures data for Dyad 606 and 609 are shown in Figs. 11 and 12. A combined plot for Dyad 606 and 609 is shown in Fig. 13 for a frequency of 200Hz and gives a guide for the expected performance using a combination of Dyad 606 and 609. The F.E. runs were done for the case of Dyad 609 on the outer side rather than the Dyad 606. It is normal practice to put the material that performs better at the higher temperatures on the inner side. In our case we did the opposite which means we could have achieved better results at higher temperatures.

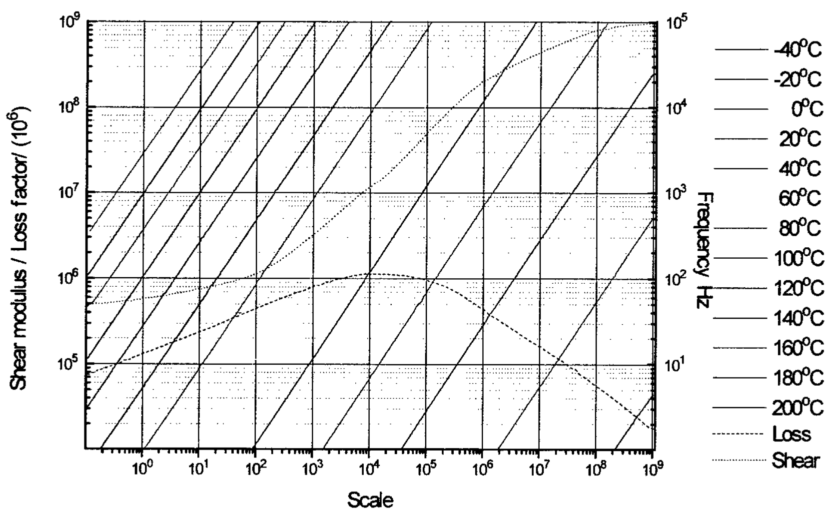

Figure 11. Dyad 606 material data. (To obtain loss factor divide shear modulus scale by 10^6)

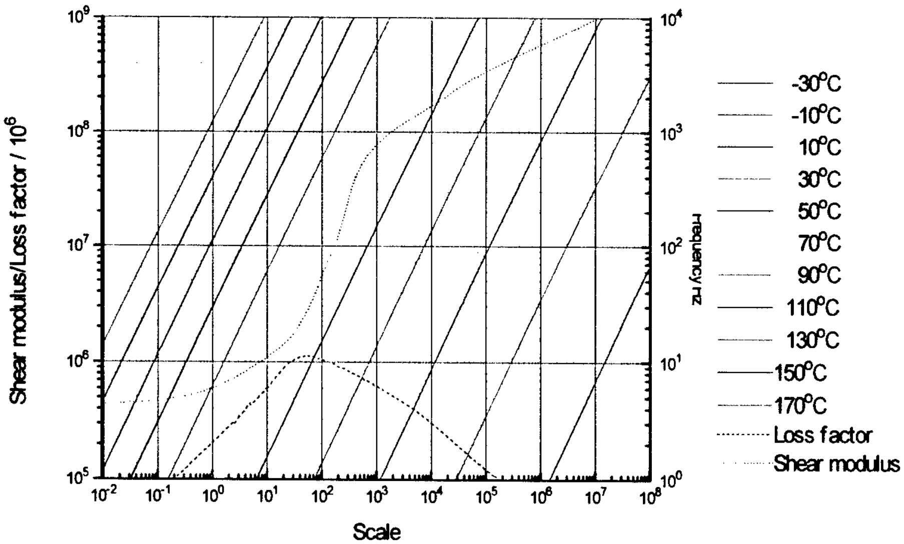

Figure 12. Dyad 609 material data.(To obtain loss factor divide shear modulus scale by 10^6)

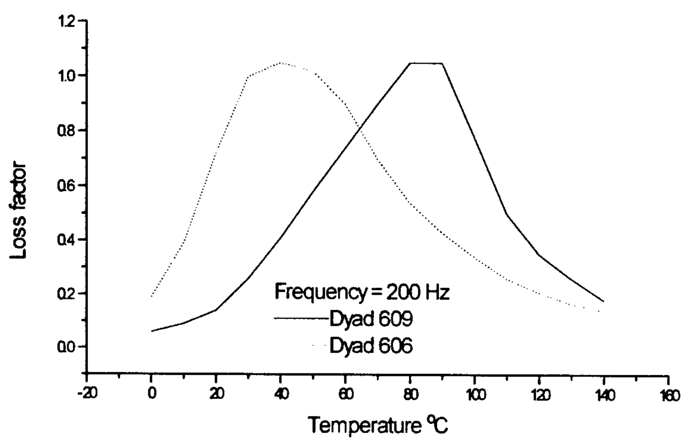

Figure 13. Comparison of Dyad 606 and 609 over the entire temperature range.

11. Adhesive data

Stress strain data is available for FM300, [17], and is shown in Fig. 14 as a function of temperature. The service temperature for this adhesive is from -55^{O}C to 150^{O}C. Only two data points exist for each temperature case, as a result values for shear modulus have been extrapolated. Using dry data the shear modulus is given by: $G = 1281 - 7.419T$ (MPa).

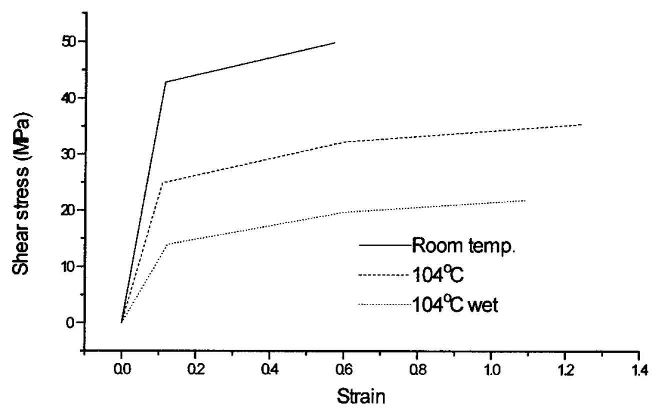

Figure 14. Stress strain diagram for FM300K for a temperature range shown. (plot1,FM300K, demonstrator)

12. Thermal environment for F/A-18

Results of a thermal profile of the F/A-18 have been obtained from [18]. These flight tests were performed to establish the thermal profile at the edges of the flight envelope in order to provide design information for bonded repairs. The operating extremes are shown in Table 2 for the aft fuselage in the area in which cracking has occurred. The most relevant conditions are those at high dynamic pressures which can lead to maximum temperatures of 141^{O}C. This would be the extreme limit for the highly damped repair. Temperatures for the coldest conditions correspond to minimum power and would not be expected to contribute to acoustic fatigue damage. Hence without additional data, the highly damped repair will be expected to perform over a temperature of 0 - 141^{O}C. Since temperatures corresponding to the hottest day of the year only occurs several times each year the ISA condition is more likely with a temperature range of 78 - 103^{O}C.

Table 2 Thermal conditions for F/A-18 aft fuselage

Condition	Altitude	Mach no.	Temperature °C
ISA	SL	1.1	78
ISA	FL350	2.0	103
Coldest day	FL500	0.74	-63
Hotest day	SL	1.1	114
Hotest day	FL350	2.0	141

13. Theoretical results

In Fig. 15 a plot is shown of the variation in the ratio of stress intensity factor, for the cracked unpatched plate (K_{up}) to the stress intensity of the repaired plate (K_p), with increasing operating temperature for a patch incorporating both 606 and 609 (viscoelastic material) VEM and two layers of Dyad 609 VEM. The stress intensity factor (SIF) was calculated using the F.E. model described in section 5 and the analysis includes the thermal residual stress contribution and softening effect of the adhesive. A significant reduction of K is shown over the complete temperature range. Results using two layers of 609 are also shown in Fig. 15.

Fig. 16 shows the crack growth rate for 7075-T6 cracked plate incorporating a patch with two layers of Dyad 609 over the temperature range of interest. In this figure, equ(6) is used to obtain the crack growth rates, with relevant coefficients, including both thermal residual stresses estimated from a closed form solution and adhesive softening. The crack growth rate is relatively low for temperatures of 80°C, beyond this the crack growth increases significantly. In comparison Fig. 17 shows that the crack growth rate for the cracked unrepaired panel compared to the repaired panel is approximately 2700 time greater for temperatures around 40°C. Even at 140°C the crack growth rate of the cracked unrepaired panel is 70 times then that for the repaired panel.

The adhesive shear and peel stresses in the adhesive layer of the boron repair the are shown in Fig. 18 for the room temperature case. At this particular SPL the stresses are relatively low and no yielding will occur under peel or shear.

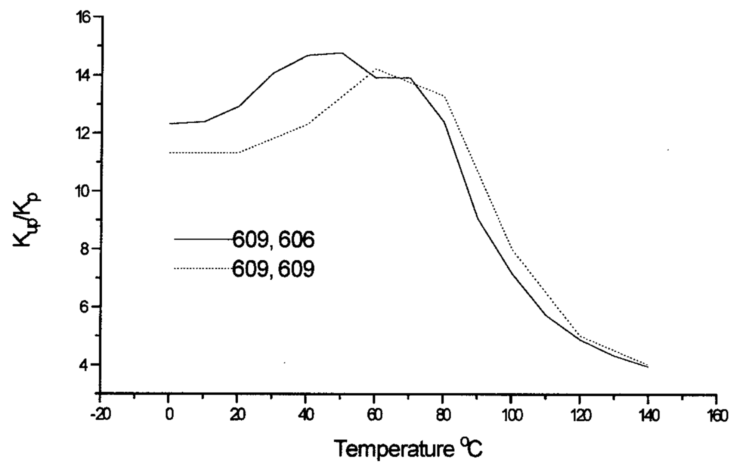

Figure 15. Plot of K_{up}/K_p versus temperature for configurations using 609 and 606.

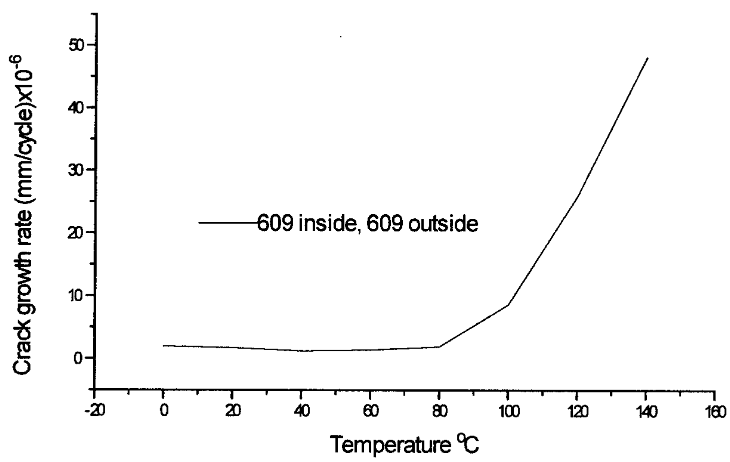

Figure 16 Crack growth rate for highly damped repair.

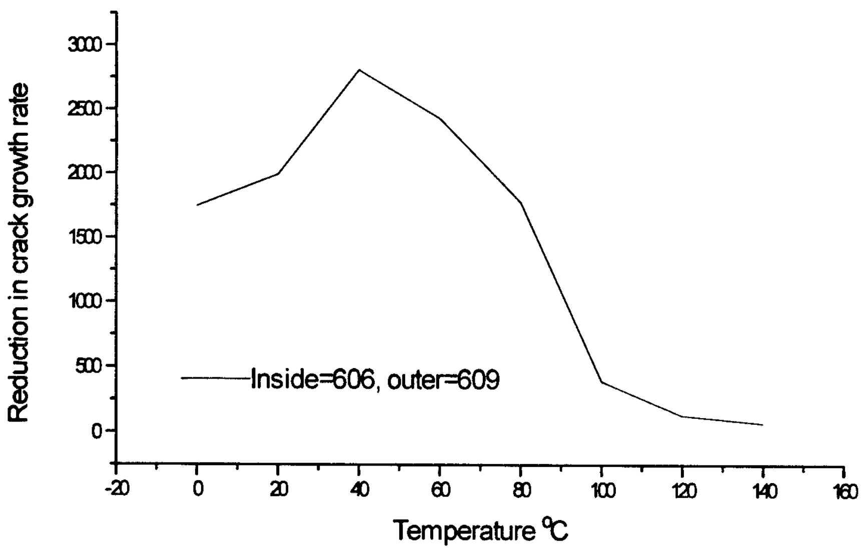

Figure 17. Reduction in crack growth rate for highly damped repair compared to cracked un-repaired plate.

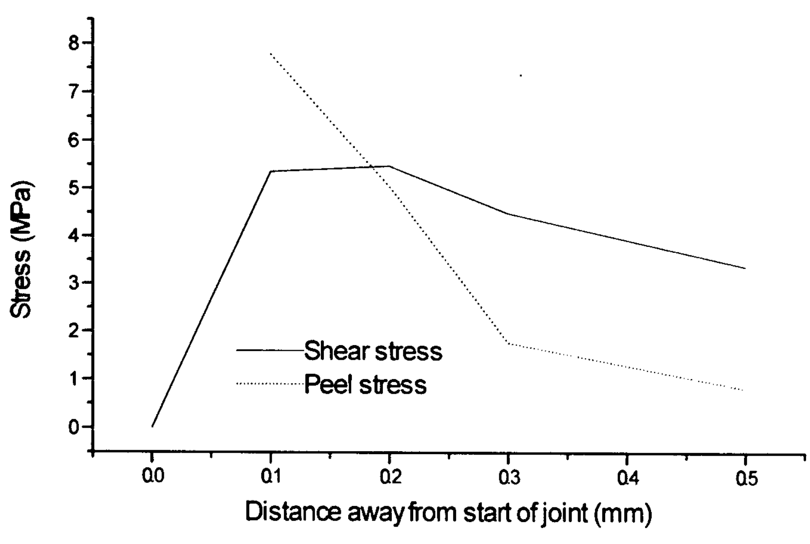

Figure 18. Adhesive stresses bonding boron to aluminium at room temperature.

14. Experimental work

Limited experimental work has been carried out in [19,20,21] in support of the analytical work. While differences exist between the aft fuselage geometry and specimens, similar experimental trends are expected. This work covers testing butterfly specimens which involve the flexing or bending motion shown in Fig. 19, provided by a shaker, in order to simulate the acoustic excitation. The geometry of the specimen is shown in Fig. 20. These specimens consist of bare aluminium specimens with boron/epoxy patches and Dyad 609 and 606 damping layers with $[0,90]_s$ constraining layers. The highly damped repair was applied using room temperature curing adhesive Hysol EA9320. All panels were pre-cracked with a crack length of 25mm. In each case the maximum force applied to the specimen by the shaker was 90N. This forcing level achieved a nominal *rms* strain of about 1000 $\mu\varepsilon$ ahead of the crack. Comparative results are shown in Fig. 21. In the case of bare aluminium specimens approximately 100,000 cycles were obtained before the resonant frequency of the specimen dropped by 15%. A 15% drop in resonant frequency was used as a criteria to indicate extensive crack growth. For boron/epoxy repairs approximately 1,000,000 cycles were obtained before the resonant frequency of the specimen dropped by 15%. In the case of the highly damped repair the first test with a shaker force of 90N gave no reduction in resonant frequency at 10,000,000 cycles, as shown by the dotted line. A second run was made with the shaker force increased to 180N. The results for this run are shown in Fig. 21. At approximately 10,000,000 cycles a 15% reduction of resonant frequency has occurred. The highly damped repair tested is slightly different to in that analysed above in section 5 in that the constraining layers of $[0/90]_S$ are used in the experimental test rather than the [0/90] plies in the aft fuselage F.E. model. However the experimental results indicate that highly damped repairs are very effective in reducing the crack growth rate.

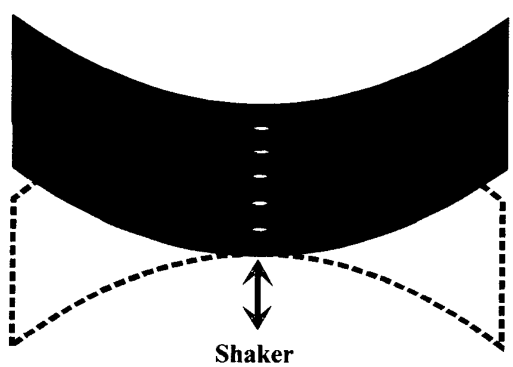

Figure 19. Flexing motion for butterfly specimen provided by a shaker.

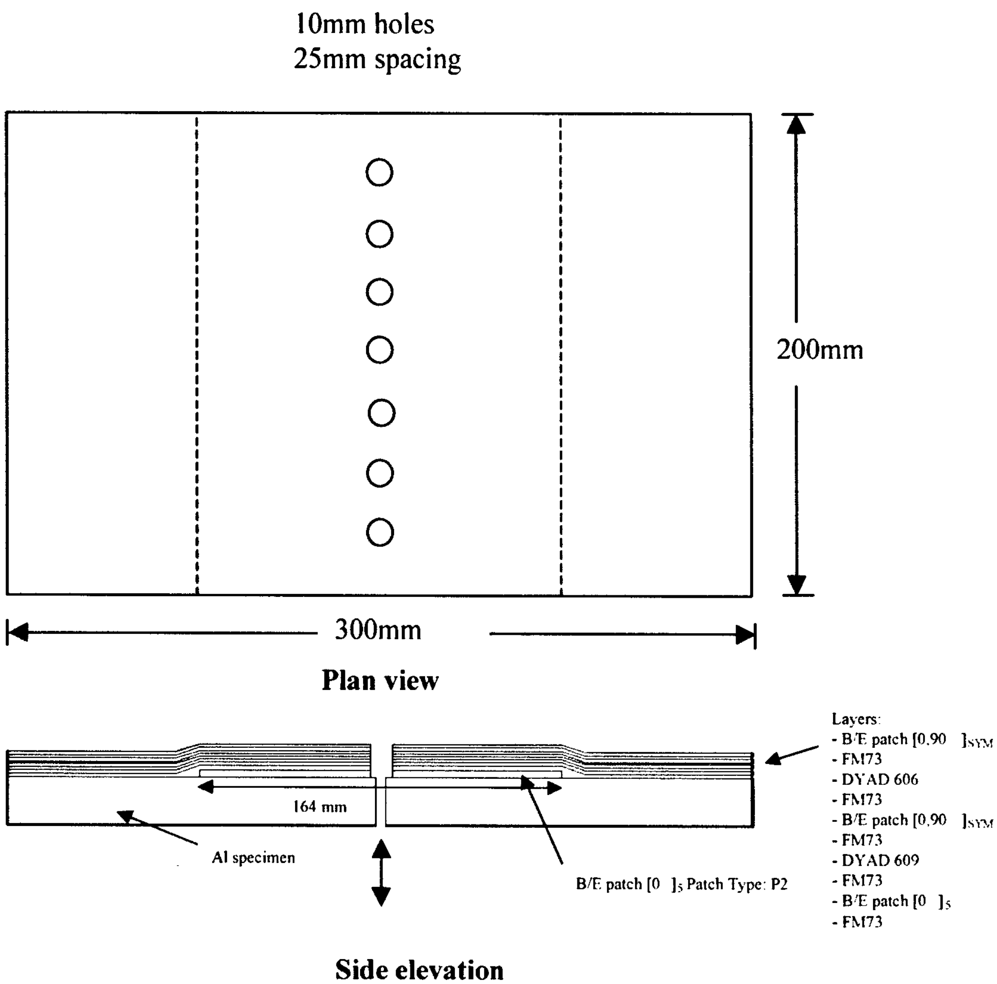

Figure 20. Highly damped repair configuration, from ref(19)

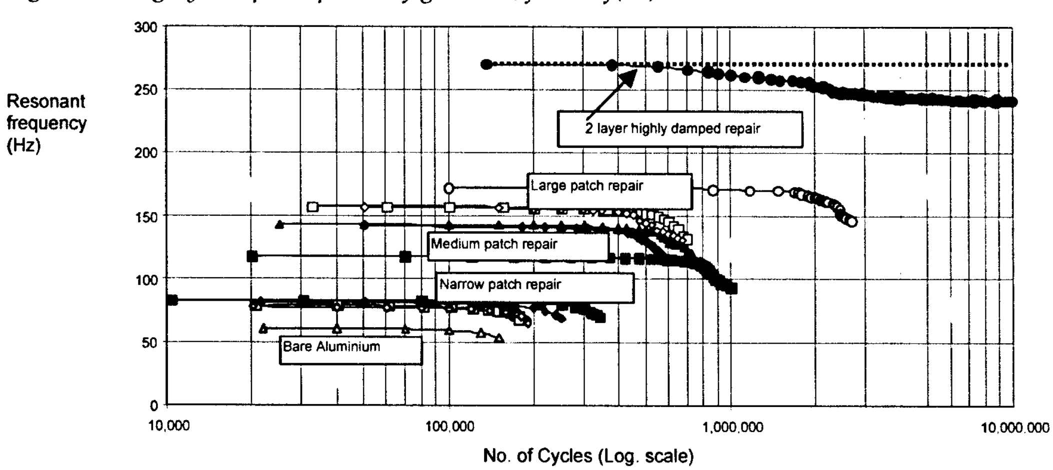

Figure 21. Comparative test lives for various specimens, from ref(22)

15. Conclusions

While sufficient data is not available to precisely define the acoustic/thermo environment, operating temperatures for extremes of the design envelope have been identified. Using two different damping materials, Soundcoat Dyad 606 and 609, enable the repair to be effective over a wide temperature range. Results show that the highly damped repair is effective in reducing the stress intensity factor in intense excitation conditions. Crack growth laws which have been validated for limited acoustic fatigue data are used to predict crack growth rates for cracks for the un-repaired and repaired cases. The F.E. results indicated that compared to unpatched panels, the application of the proposed highly damped repair would reduce the crack growth rates by a factor of 2700, at temperatures around 40°C, and 70, for temperatures around 140°C. The damping materials, boron and FM300 adhesive will survive temperatures up to 140°C.

16. References

1. A/B/C/D Aircraft Lower Nacelle Skin Acoustic and Strain Measurements and Sonic Fatigue Analysis. *MCDonnell Douglas Corporaton* 94B0044. Mar,1994.

2. J. Soovere and M.L. Drake Aerospace Structures Technology Damping Design Guide - Volume II: Design Guide, AFWAL-TR-84-3089, 1985.

3. L.Rogers, I.R.Searle, R.Ikegami, R.W.Gordon and D.Conley. Durability patch: application of passive damping to high cycle fatigue cracking on aircraft. *SPIE Smart Structures and Materials -Passive Damping and Isolation Conference.* San Diego, Ca. Paper no 3045-28, p214-223, Mar. 1997.

4. L.Rogers, J.Maly, I.R.Searle, R.Ikegami, W.Owen, D.Smith, R.W.Gordon and D.Conley. Durability patch: repair and life extension of high cycle fatigue damage on secondary structure of aging aircraft. *The First Joint DoD/FAA/NASA Conference on Aging Aircraft.* Ogden, Ut, 8-10 July, p595-623, 1997.

5. S.Liguore, R.Perez and K.Walters. Damped composite bonded repairs for acoustic fatigue. *3rd AIAA/CEAS Aeroacoustic Conference*, Atlanta, GA May 12-14, p774-783, 1997.

6. S.L.Liguore, K.Hunter, R.Perez, and T.H.Beier. Flight Test Evaluation of Damped Composite Repairs For Sonic Fatigue. *Sonic Fatigue Session of the 40th AIAA/ASME/ASCE/AHS SDM Conference* St. Louis, MO April 12-15, p1498-1508, 1999.

7. R.J.Callinan, S.C.Galea and S.Sanderson. Finite Element Analysis of Bonded Repairs to Edge Cracks in Panels Subjected to Acoustic Excitation. *Composite structures*, p649-660, 1997.

8. R.J.Callinan, W.K.Chiu and S.C.Galea. Optimization of a Composite Repair to Cracked Panels Subjected to Acoustic Excitation. Paper A98-31631 21st *International Congress on Aeronautical Science.* 13-18 Sept. 1998, Melbourne, Australia.

9. E.Haugse, P.S.Johson, D.L.Smith, L.Rogers and J.Ryan Durability Pactch and Damage Dosimeter: A Portable Battery Powered Data Acquisition Computer and Durability Patch Design Process. *The Third Joint FAA/DoD/NASA Conference on Aging Aircraft.* p105-116, Sept 20-23, 1999. Albuquerque, New Mexico.

10. H. Climent and J. Casalengua. Application of a PSD technique to acoustic fatigue Stress Calculations in Complex Structures. *Symposium on ' Impact of Acoustic Loads on Aircraft Structures'*, Lillehammer, Norway,p12-1,12-11, May 1994.

11. Blakely MSC/NASTRAN, basic dynamics analysis, user's guide. *The MacNeal-Schwendler Corporation,* Dec,1993.

12. R.J.Callinan, C.H.Wang, S.C.Galea, S.Sanderson and L.R.F.Rose Analytical Solution for the Stress Intensity Factor in Bonded Repairs to Panels Subjected to Acoustic Excitation. *Presented to Joint FAA/DoD/NASA Conference on Aging aircraft.* 20-23 Sept. 1999.

13. K.P. Byrne Strains Affecting the Growth Rate of Edge Cracks in Acoustically Excited Panels. *Institute Sound and Vibration Research Technical. Report. 59,* Nov, 1972.

14. L.Schwarmann. Material Data of High-Strength Aluminium Alloys for Durability Evaluation of Structures. *Aluminium-Verlag,* Dusseldorf; 1998.

15. M.Davis. *RAAF Engineering Standard C5033.* Composite Materials and Adhesive bonded Repairs.

16. Anon. *Soundcoat, manufactures data sheets for Dyad 606 and 609.*

17. Anon. FM300 Film adhesive. *Manufactures data. Cytec Fiberite Inc.*

18. S.Barlow Determination of F/A-18 Thermal Profile. *Formal Report Task* 00176. AR-009-066. July 1985.

19. D.Bohret. Acoustic fatigue testing of bare aluminium panels with boron/epoxy patch repairs. *Aerostructures document* ER-FA18-54-APM100, 19 June 2000.

20. D.Bohret. Acoustic fatigue testing of bare aluminium panels with highly damped repairs. *Aerostructures report* ER-FA18-54-APM101, 29June 2000.

21. D.Bohret. Acoustic fatigue testing. *Aerostructures report* ER-FA18-54-APM086, 19 June 2000.

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19. ABSTRACT

Currently an increasing number of RAAF F/A-18 aircraft are developing acoustic fatigue cracks on the rear fuselage. Acoustic fatigue is the result of high frequency lateral vibration of an aircraft panel as a result of time varying pressure waves caused by engine and/or aerodynamic effects. While the exact broadband power spectral density (PSD) spectrum is not available for this area of the the F/A-18, the PSD is known for the inlet nacelle area in which cracking has occurred. Cracking in the inlet nacelle proceeded cracking in the aft fuselage by a period of several years. As a result the inlet nacelle experiences a more severe spectrum and is the PSD adopted here. Viscoelastic damping materials are used in this repair in conjunction with the constrained layer technique to reduce the lateral vibration. Since viscoelastic materials have a limited temperature range it has been necessary to use two types of damping materials to cover the expected temperature range. A highly damped repair is proposed for the aft fuselage area and is shown to be significantly effective in reducing the crack growth rate.