# Estimation of Fatigue Crack Growth in Patched Cracked Panels

RAMESH CHANDRA* and R. SUNDER
*Structures Division, National Aeronautical Laboratory, Bangalore 560017, India*
**Currently Research Associate, Department of Aerospace Engineering, University of Maryland, College Park, Maryland 20742, USA*

ABSTRACT

The fatigue and fracture performance of a cracked panel can be significantly improved by providing high stiffness composite patches as reinforcements. Efficient design of reinforcement to a cracked panel needs a methodology to estimate fatigue crack growth rate and life of panel subjected to a real life flight spectrum loading condition. This paper attempts to contribute in this direction. Estimation of fatigue crack growth in flight simulation loading involves appropriate representation of load spectrum and selection of suitable crack-growth and load interaction models. In the present study load spectrum is represented by Markov matrix and rain-flow cycle counting is used. Crack closure based crack-growth model is selected. Fatigue crack growth rates in center-cracked panels are estimated. Stress intensity factor (SIF) - crack length relation including the thermal mismatch is obtained by employing analytical technique. Fatigue crack growth rate versus SIF range relation, a material property is obtained from authors' earlier investigation. The comparison of estimated crack growth rates with the experimental ones is found to be good. Residual stress in panel arising due to high temperature curing adhesive system influences the fatigue crack growth considerably.

KEYWORDS

Fatigue Crack Growth; Patched Cracked Panels

NOTATION

| | |
|---|---|
| a | Semi crack length |
| CA | Constant amplitude loading |
| da/dN | Crack growth rate in CA loading |
| E | Young's modulus of panel |
| FS | Flight simulation loading |
| $K_p$ | SIF in patched panel due to mechanical loading |
| $K_T$ | SIF in patched panel due to thermal loading |
| $K_{pt}$ | $K_p + K_T$ |
| $K_u$ | SIF in unpatched panel |

| | |
|---|---|
| $K_{max}$ | $\sigma_{max}\sqrt{\pi a}\, y(a)$ |
| $K_{min}$ | $\sigma_{min}\sqrt{\pi a}\, y(a)$ |
| $K_c$ | Fracture toughness |
| N | cycles in CA loading |
| R | $\sigma_{min}/\sigma_{max}$ |
| w | Semi width of panel |
| y(a) | Finite width correction factor |
| $\alpha_s$ | Coefficient of linear thermal expansion of panel |
| $\alpha_p$ | Coefficient of linear thermal expansion of patch |
| $\Delta T$ | Temperature difference of curing temperature and ambient temperature |
| $\Delta K$ | $K_{max} - K_{min}$ |
| $\Delta K_{th}$ | Threshold SIF range |
| $\Delta K_{eff}$ | $(\sigma_{max} - \sigma_o)\sqrt{\pi a}\, y(a)$ |
| $\Delta K_o$ | $(\Delta K_{eff}/\Delta K)\Delta K_{th}$ |
| $\sigma_o$ | Crack opening stress |
| $\sigma^o$ | Applied stress |

## INTRODUCTION

The 'Fail Safe' concept is widely used in the design of aircraft structural components. The design of such 'damage tolerant' structures is governed by detailed specifications(N.N.,1974, 1977), which are intended to ensure that a specified fatigue crack will not grow to critical proportions in a period between successive routine inspections.

Fatigue cracked components are often repaired in service. Standard repair schemes normally involve strengthening the component by connecting a reinforcing member by means of bolts or rivets and thereby reducing the crack tip stress intensity factors and enhancing the fatigue lives.

Recent technological advances in fiber reinforced composite materials and adhesive bonding have led to the development of efficient repair schemes using these. In such repairs, (a) the load transfer between component and reinforcement is affected with minimal stress concentration effects, (b) the properties and geometry of the reinforcement can be tailored to suit the particular application and (c) composite repair patch does not add significantly to the weight of the component. Boron-epoxy, boron-aluminium and carbon-epoxy have been used in some repair schemes for aircraft structures (Baker, 1978). A general research program on development of methodology for design of reinforcement in cracked plates is currently being pursued at National Aeronautical Laboratory, India (Chandra et al., 1984, 1985, 1986, 1987, 1988, 1989a, 1989b)

The various steps involved in the design of repairs with composite patches to achieve substantial enhancement of fatigue life of cracked panels are:

1. Determining base line data on fatigue crack growth for the specified panel parameters (material and thickness), establishing a crack growth law in terms of growth rate versus stress intensity factor (SIF) range.
2. Determining the fatigue load spectrum for the component to be repaired and selection of a suitable 'load-interaction' model for crack growth.



3. Selection of materials for patch and adhesive.
4. Initial design of patch: material parameters (fiber orientation, stacking sequence and ply thicknesses), geometry and location with respect to crack to reduce SIF to acceptable levels. Weight constraints and feasibility of location and geometry are significant considerations.
5. Computation of stress intensity factor for proposed patched configuration.
6. Prediction of crack growth life of patched component using SIF obtained and crack growth law established earlier.
7. Iteration of steps 3 to 6 till desired fatigue life is achieved.

It is readily seen that the two critical needs in this context are (i) simple technique for estimating SIF and (ii) good modeling of fatigue crack growth in cracked panels repaired by bonded composite patches.

Efficacy of the repair patch depends, inter alia, on the characteristics of the adhesive system. The required mechanical properties of the adhesive are normally met by a high temperature curing system. Although, there are attempts to develop room temperature curing adhesives with higher strength (Patrick, 1973), as of now no room temperature curing adhesive suitable for repair patch work is available to us. Hence, one is forced to employ high temperature adhesives.

A high temperature adhesive system causes a residual stress field in the panel due to thermal mismatch arising from the differing thermal expansion coefficients of panel and patch materials. This residual stress plays an important role in repair patch design. Baker et al (1979) studied such residual stresses. These are very strong function of boundary mechanical constraint of the panel. In the real life situation of a wing panel, the constraint provided by the ribs and stiffeners can reduce the residual stresses considerably. However, even the reduced value of residual stress can influence fatigue crack growth as it is very sensitive to SIF.

The main objective of this paper is to establish a prediction methodology for fatigue crack growth in patched cracked plates. This is conveniently illustrated for center-crack plates with symmetric patches. The SIF versus crack length relation due to the combination of mechanical and thermal loads is obtained by employing the analytical technique (Chandra et al., 1985). The material property, fatigue crack growth rate versus SIF range, as obtained in reference (Chandra and Sunder, 1984) is used. Results from some tests on center-crack plates with symmetric patches described in reference (Chandra, 1986) are used to assess the prediction methodology. A good correlation between predicted and experimental crack growth rates is established.

## SELECTION OF CRACK GROWTH AND LOAD INTERACTION MODELS

The fatigue crack growth rate is a function principally of SIF range, threshold SIF, stress ratio, fracture toughness and crack opening stress. In view of the relatively high growth rates encountered in our tests (Chandra, 1986) ($10^{-5}$ < da/dN < $10^{-1}$ mm/cycle) we propose the relation

$$\frac{da}{dN} = C(\Delta K_{eff})^m[1-(K_{max}/K_c)^2]^{-1} \qquad (1)$$

which includes the closure effect but neglects the threshold effect. BS L72, the material used in our experiments is close to 2024 T3. And $(\Delta K)_{eff}$ for 2024T3 as given by Elber (1970) is:

$$(\Delta K)_{eff} = (0.5 + 0.4R)\,\Delta K \quad (2)$$

As the stress ratio used in our experiments is zero the eq. (2) becomes:

$$(\Delta K)_{eff} = 0.5\,\Delta K \quad (3)$$

It is important to consider load interaction effects for prediction of crack growth rates under flight simulation loading conditions. There are many load interaction models available in the literature. To assess the suitability of some of these, a round-robin study was performed, Chang (1981). Sunder et al. (1984a 1984b) compared his method to the ones reported in this study and found that his method gave better prediction ratios. Hence, we use Sunder's method for this work.

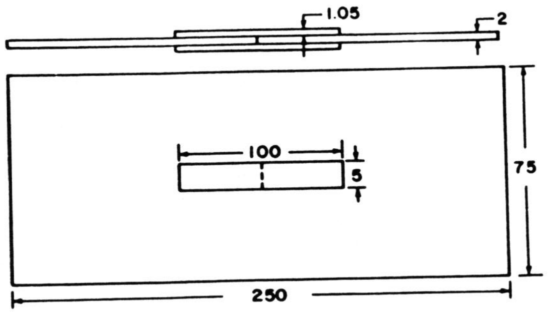


Fig. 1 Geometry of center-cracked panel (CCP).

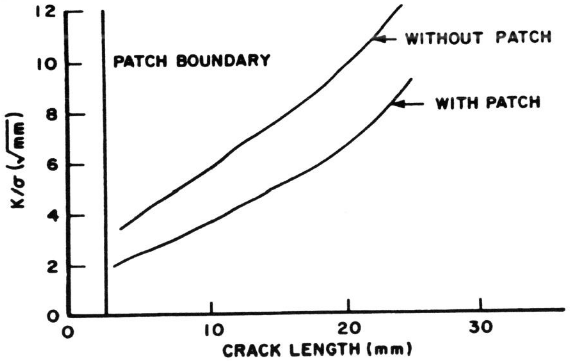


Fig. 2 SIF - crack length curve for CCP due to mechanical loading.



ESTIMATION OF TOTAL SIF

Figure 1 shows the patched center-crack plate for which crack growth predictions are made. SIF's for this configuration for various crack lengths are obtained by employing the analytical technique (Chandra et al. 1985). Various geometrical and material parameters for this configuration are:

Aluminium alloy cracked sheet:

| | | | |
|---|---|---|---|
| Young's modulus | = 69 KN/mm$^2$ , | Shear Modulus | = 26 KN/mm$^2$ |
| Poisson's ratio | = 0.32 , | Thickness | = 2 mm |
| Crack length | = 3 to 25 mm | | |

Boron-epoxy patch:

| | | | |
|---|---|---|---|
| Young's modulus | = 208 KN/mm$^2$ , | Shear Modulus | = 4.94 KN/mm$^2$ |
| Thickness | = 2.1 mm , | Length | = 100 mm |
| Width | = 5 mm, Distance between the center lines of the two patches = 2.5 mm | | |

Redux film adhesive:

Shear Modulus = 0.965 KN/mm$^2$, Thickness = 0.05 mm
$\Delta\alpha(\alpha_s - \alpha_p) = 18\times10^{-6}$/deg, $\Delta T$ = 100 deg

This patch is idealized as two patches abutting each other at the center line of the sheet. Such an idealization yields good estimate of SIF (Chandra, 1986). Figure 2 shows the variation of SIF due to mechanical loading with respect to crack length. This SIF vs. crack length relation is represented as polynomial function through curve fitting technique as:

$$K_p/\sigma\sqrt{a} = f_m(a/w) \quad (4)$$

where

$$f_m(a/w) = 1.26 - 5.31(a/w) + 56.25(a/w)^2 - 203.08(a/w)^3 + 295.00(a/w)^4$$

Similarly the variation of SIF due to residual stress with crack length for center crack configuration is obtained (Chandra, 1986). This is represented in the functional form through curve fitting as:

$$K_T = E\,\Delta\alpha\,\Delta T\,\sqrt{a}\; f_T(a/w) \quad (5)$$

Total SIF in patched cracked panel is obtained by adding relations (4) and (5).

$$K_{pt} = \Delta\sigma\sqrt{a}\; f_m(a/w) + E_s\,\Delta\alpha\,\Delta T\,\sqrt{a}\; f_T(a/w) \quad (6)$$

It is noted from the above relation that the reduction of SIF due to patch is offset by tensile residual stress due to thermal mismatch. As SIF due to thermal mismatch is independent of applied stress range, the net reduction in SIF due to patch will depend upon applied stress. It is to be noted from this equation that the influence of residual stress on SIF is low at high stress levels.

## RESULTS AND DISCUSSION

Crack growth rate vs. SIF range relation obtained from tests on plain edge crack specimens (Chandra and Sunder, 1984) is expressed in mathematical form as:

$$\frac{da}{dN} = C \frac{(\Delta K_{eff})^m}{1-(K_{max}/K_c)^2} \qquad (7)$$

where

$C = 1.624.10^{-2}$, $m = 3.052$, $K_C = 2530\ N/mm^{3/2}$, $\Delta K_{eff} = 0.500\ \Delta K$

Using SIF-Crack length relation (equation (6)) and crack growth rates - SIF range relation (equation(7)) fatigue crack growth rates for patched and unpatched center-cracked plates under constant amplitude loading conditions are predicted. Whereas fatigue crack growth rate prediction under flight simulation loading was based on Sunder's model.

Figures 3-4 show the influence of patch on predicted crack growth rates in center crack plates under CA loading. It is to be noted that patch reduces the crack growth rates considerably. Figures 5-6 indicate the influence of patch on predicted crack growth rate in center crack plates under FS loading. It may be seen from these figures that in the case of FS loading also, the observations made in case of CA loading in respect of influence of patch on crack growth rate hold good.

Figures 3 and 4 show the correlation of predicted and experimental crack growth rates in patched and unpatched plates for 80 and 100 $N/mm^2$ respectively for constant amplitude loading. Figure 6 shows the similar correlation for unpatched plate for 25 $N/mm^2$ for flight simulation loading. Correlation is noted to be good. Thus, the prediction technique presented in this paper is validated. Also, it is emphasized that SIF due to residual stress is quite important and must be considered in prediction of fatigue crack growth rates in patched cracked plates, adhesive bonded using high temperature curing system.

## CONCLUSION

In this paper, a prediction technique for fatigue crack growth in unpatched and patched plates subjected to flight simulation loading has been presented. This techniques was applied to center-crack plates with and without symmetric patches. Good correlation between predicted and experimental crack growth rates was found. Also it is shown that SIF due to residual stresses plays a significant role in prediction technique of crack growth for patched cracked plates adhesively bonded with high temperature curing system and must be considered.

## REFERENCES


Baker, A.A. (1978). A summary of work on application of advanced fibre composites at Aeronautical Research Laboratories, Australia. Composite 9(1), 11-16.





Baker, A.A. et al. (1979). Repair of fatigue-cracked aircraft structures with advanced fibre composites: residual stress and thermal fatigue studies. Proc. 10th ICAF, 4.3/1-4.3/30.

Chandra, R. and R. Sunder (1984). Fatigue crack growth in patched plates under constant amplitude and flight simulation loadings. Advances in Fracture Research, Proc. Sixth Int'l. Conf. on Fracture, 5, New Delhi, India. 3523-3532.

Chandra, R., M.V.V. Murthy, T.S. Ramamurthy, and A.K. Rao (1985). Analytical estimation of stress intensity factors in patched cracked plates. Engg. Fracture Mechanics, 21(3), 479-494.

Chandra,R. (1986). A study on cracked plates with composite patch reinforcements. Ph.D. Dissertation, Department of Aerospace Engg., Indian Institute of Science.

Chandra, R. and K. Guruprasad (1987). Numerical estimation of stress intensity factors in patched cracked plates. Engg. Fracture Mech. 27(5), 559-569.

Chandra, R. K. Guruprasad and K. Dwarkanath (1988). Energy release rates in patched cracked plates. To appear in Engg. Fracture Mechanics.

Chandra, R. (1989a). Experimental determination of stress intensity factors in patched cracked plates. To appear in Engg. Fracture Mechanics.

Chandra, R. and A. Subramanian (1989b). Stress intensity factors in plates with a partially patched central crack. To appear in Experimental Mechanics.

Chang, J.B. (1981). Round-robbin crack growth predictions on centre-cracked tension specimens under random spectrum loading. Methods and Models for Predicting Fatigue Crack Growth under Random Loading, ASTM STP-748, 3-40, ASTM Philidelphia.

Elber, W. (1970). Fatigue crack closure under cyclic tension. Engg. Fracture Mechanics, 2, 37-45.

N.N. (1974). Airplane damage tolerance requirement. MIL-specs.,83444, USAF.

N.N. (1977). Damage tolerance and fatigue evaluation of structure. FAA, 14 FCR Part 25, Para. 25.571.

Newman, J.C., Jr. (1981). A crack-closure model for predicting fatigue crack growth under aircraft spectrum loading. Methods and Models for Predicting Fatigue Crack Growth under Random Loading, ASTM STP 748, 53-84, ASTM Phildelphia.

Partrick, R.L.(1973). Treatise on adhesion and adhesives. Marcel Dekker, Inc. New York.

Sunder, R. (1984a) Automated fatigue crack propagation testing. Advances in Fracture Research, Sixth Int. Conf. on Fracture, 5, New Delhi, 3211-3220.

Sunder, R. (1984b). Crack growth under random loading-fatigue cycle analysis for testing and life predictions. Proc. Int. Symp. on Strength at Sonic and Ultrasonic Frequencies, Kiev, Russia.

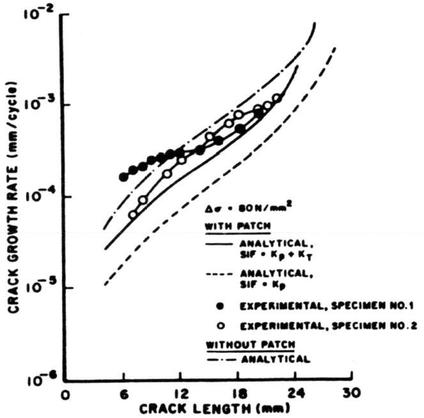


Fig. 3 Crack growth rates under CA loading, $\Delta\sigma$ = 80 N/mm$^2$.

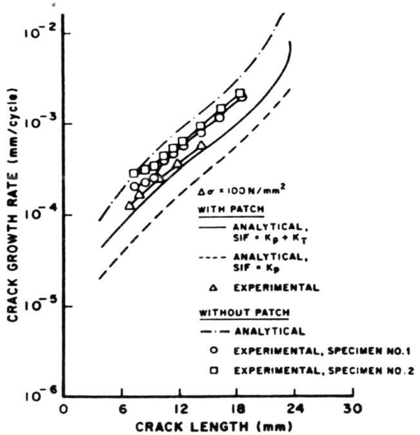


Fig. 4 Crack growth rates under CA loading, $\Delta\sigma$ = 100 N/mm$^2$.



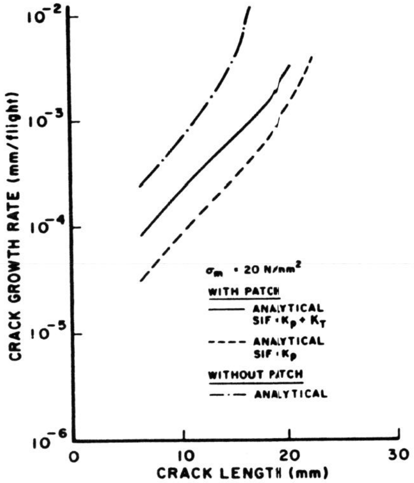


Fig. 5 Crack growth rates under FS loading, $\sigma_m$ = 20 N/mm$^2$.

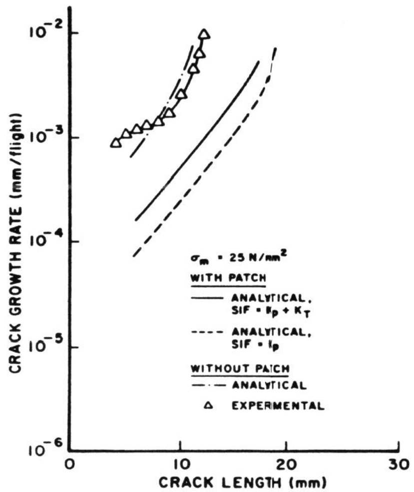


Fig. 6. Crack growth rates under FS loading, $\sigma_m$ = 25 N/mm$^2$.