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DAMAGE TOLERANT& FATIGUE LIFE INVESTIGATION FOR AN ALUMINIUM MATERIAL IN AERO STRUCTURAL COMPONENT

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ABSTRACT

In today's structural design, fatigue and damage tolerance analysis have become the most important and challenging task for the designers because of failure of structures due to different types of damages and cracks. Some of these damages have caused a loss of entire structure or the whole aircraft Itself. Any structure will fail due to one or a combination of failure such as elastic and linear deformations, bucking, fatigue, creep, corrosion and fracture.This work presents the details of fatigue crack analysis performed in the critical region of an aero structural component .From the fatigue crack analysis, maximum life was estimated, based on both analytical and finite element analysis techniques.

INTRODUCTION

Damage tolerance is a safety requirement and equates to the failsafe design approach employed for aero structural critical components by the rotary wing industry since the early 60's. Damage tolerance is the ability of the structure to resist failure due to the presence of defects, cracks, or other damage for a time period sufficient to enable their detection.

The fail-safe and damage tolerance design philosophies each have the common objective of providing structural integrity at a reasonable level of assurance for all safety-of-flight structures that is structure whose failure could cause direct loss of the aircraft. Reference 5 points out an important distinction between fail safety and damage tolerance. Fail safety as it was defined prior to the evolution of damage tolerance is based upon the premise that one cannot be certain that cracks (or damage) will not initiate at some time during the aircraft life and these cracks must be detected before the strength drops below a certain level. Damage tolerance, on the other hand, assumes the existence of initial flaws in the structure and the structure is designed to retain adequate residual strength until damage is detected and corrective actions taken.

In designing to this philosophy, the criteria must address both the static residual strength and damage propagation for the structure under consideration. Today's economic reality dictates that a fleet must be operated beyond intended design life. The estimate material properties must be evaluated as precisely as possible within all the environmental conditions to avoid failures.

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1.3 **Damage tolerance and allowable design limit analysis**

Damage tolerance

Damage tolerance is ability it resists fracture from the preexistent cracks for a given period of time and is an essential attribute of components whose failure could result in catastrophic loss of life or property. The damage tolerance addresses two points concerning an initially cracked structure. First, it is desired to determine fracture load for a specified crack size. Second, it is necessary to predict the length of time required for a 'subcritical' crack to grow to the size that causes fracture at given load. It is assumed that the crack can extend in a sub-critical manner by fatigue and/or stress corrosion cracking.

Allowable damage limit

Allowable Damage Limit is the damage that allows an operator to continue operating an aircraft without any repair. Damage permitted by these data must have no significant effect on strength and fatigue life of the structure, which must still be capable of fulfilling its design function.

METHODOLOGY

In this work, fracture mechanics approach is adopted, an analytical approach as a preliminary step investigation fatigue life and crack growth rate is estimated in a structure with damage.

LITERATURE REVIEW

Harold K. Reddick studied that the safe-life and damage-tolerant design approaches apply to both metallic and fibrous composite helicopter structures. However, to maintain the theme of this workshop, this presentation will emphasize the application of these design approaches to fibrous composite structures. In this paper helicopter flight-critical structures, although limited primarily to rotor blades, have been successfully designed for damage tolerance. This has been accomplished through redundancy, use of damage-tolerant materials and sizing to low operating stress levels. Design and demonstration have been based on laboratory testing.

EXPERIMENTAL DETAILS

OBJECTIVE

PROBLEM DEFINITION

Damages may occur due to manufacturing and assembling errors includes gouges, nicks, burrs, scratches and dents. Material flaws include porosity, constituent particles, inclusions, forging or casting defects & improper thermal/mechanical treatment of basic alloy. Service induced damage includes cracks, This type of damages increases in any structure it will fail. Damage tolerant design procedure was used to overcome this above problem.

OBJECTIVE OF THIS WORK

The major objectives of this work are

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- To develop a model to study the propagation of cracks in aircraft structure
- To validate the model developed using experiments.
- The model developed, can be used for investigate various types of cracks under different operating conditions and design components based on damage tolerant design concept.

CRACK GOROWTH INVESTIGATION

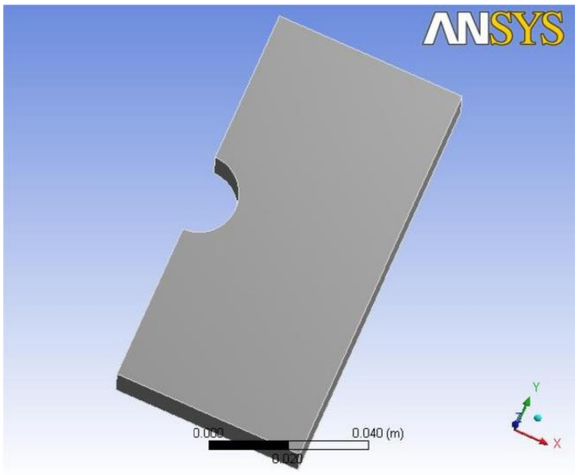
Analytical approach

A panel of dimension 100 X 100 mm is considered for analysis with a hole of diameter 10mm to simulate the effect of damages like dents, cracks, scratches and gouges, critical being the crack, thickness of the panel is 20mm . Damage is simulated by removing the material at the center of the plate.

Crack growth analysis using fracture mechanics approach [10]
For the estimation of fatigue crack growth life the panel is assumed to contain either center or edge crack as Based on panel size, thickness, and maximum stress observed in the location and using simple Paris type crack growth law. The number of elapsed cycles for the defect to grow from the initial value to a value where the stress intensity factor (SIF) reaches to fracture toughness of the material or the crack size for which the net section stress reaches the material yield. Knowing the allowable number of cycles based on 'C' checks the life is estimated. A routine based on Paris crack growth law for life estimation is given (for Center Crack).

FEA of crack growth approach

The half crack panel of 100× 50 mm cross section having aluminium alloy property is considered for the FEA, once meshing is done, constraints are given to the panel, static force 1000N is applied for the analysis for the applied boundary condition, the stress state of the panel is extracted, which has maximum stress, once the stress state are extracted , fatigue analysis performed for the panel to access its fatigue life.



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TEST PROCEDURE

Fatigue experiments were carried out on a 100 kN capacity Instron 8801 servo hydraulic machine. The experimental setup used is shown in the Fig.. All tests were performed at room temperature using a sinusoidal waveform at a loading frequency of 20 Hz. The SENT specimens, both unrepaired and repaired, were loaded cyclically at two different stress/load ratios of R = 0 and R = 0.1. The specimens were pre-cracked to different lengths before repairing with composite patch. The pre-cracking was accomplished by cycling the specimens. This pre-cracking process was accomplished to ensure that the effect of the machined starter notch is removed. All samples were bonded under same pressure and temperature conditions to ensure the adhesive quality, constancy and thickness. Three samples at least were tested for each condition to ensure the result constancy and accuracy. For each test, the number of cycles with respect to crack length was recorded using high-speed camera for later processing. Then, the data obtained was processed to produce the fatigue behavior of the samples as displayed in the results section. For the given panel using the experimental procedure, the crack growth are obtained. initial crack size of the panel is 5mm finally it reached crack is 17.5mm . it needs for 25e8 cycles to reach the final stage.

RESULTS AND DISCUSSION

Comparison

Finally the fatigue crack analysis and maximum life is estimated,this work was done by both analytical and finite element analysis technique. Both result are finely matched, small variation between analytical and FEA method. This result is show in below

Table

S.no	TYPE OF ANALYSIS	MAX STRESS (Pa)	FATIGUE LIFE
01	ANALYTICAL	1.5e6	22.35e8
02	FEA	1.5368e6	25e8

CONCLUSIONS

The analytical and software analysis of the damage structure has been performed .the effects of the maximum & fatigue life is analyzed. Discuss the importance of damage fatigue and damage tolerance analysis procedure for any damage structure in aero structural

SCOPE OF FUTURE WORK

- **The fatigue analysis results can be experimentally validated**
- To analysis the composite structure & find the result

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