In today’s structural design, fatigue and damage tolerance analysis have become most important and challenging task for the designers because of failure of structure due to different type of damages. Some of these damages have caused a loss of entire structure i.e. Whole Aircraft Itself.

This paper presents details of, types of damages which are expected to occur during manufacturing and in service induced damages. Modeling of different type of damages like Cracks, Dents, Nicks, Gouges, and Scratches, Estimation of Strength, Life using Fatigue Analysis and Crack Growth using Lefm approach of components with defects is very essential for their acceptability for the desired function.

In this presentation, we share our experiences in carrying out the Allowable Damage Limit (ADL) analysis of metallic material of a transport aircraft. Metallic materials are widely used for many aircraft primary and secondary structural components. However, metallic materials are susceptible to damage, which can be induced by service loads and accidental impacts. The main focus of the present work is to establish the analysis procedure to validate the component’s strength, life, and residual strength and inspection intervals with defined defects in it.
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The team has an average aerospace experience of 5 years and is led by Kiran Rao. This team has successfully executed projects on allowable damage limit for a commercial aircraft program under the guidance of S A Hakeem and K Badari Narayana.
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Introduction

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. For this reason the aircraft structural integrity programs are put in place to provide an integrated and systematic approach to aircraft structural engineering and maintenance.

Two major approaches were developed in the past, namely, the safe-life and the fail-safe design concepts. The safe-life approach correlating the time to failure of the specimen with the applied loads characteristics to predict the time to failure of real components using Minor’s rule approach. The other is fail-safe concept, in which linear elastic fracture mechanics approach (LEFM) are used to predict the crack stability, crack growth and hence the minimal time between the two inspections to avoid a crack reaching critical size. The later concept called the damage tolerance, whose function is to assess the effect of cracks in the structure. The analysis of damage tolerance behavior plays an important role in the structural integrity program. Fig. 1 shows the typical procedure for damage tolerance analysis used in the present work.

This paper presents details of, types of damages, modeling different type of damages, estimation of their life using fatigue analysis and crack growth analysis using LEFM approach.

Damage Tolerance and Allowable Design Limit Analysis

2.1 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 'sub-critical' 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.

2.2 Allowable Damage Limit (ADL)
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.
2.3 Types of failures

Any structure will fail due to one or a combination of failures such as elastic & inelastic deformations, buckling, fatigue, creep, corrosion & fracture. This paper describes the procedure to carry out the damage tolerance analysis with initial cracks.

![Fig. 1 Damage Tolerance Analysis Process](image-url)
Types of damages

Damages may occur due to manufacturing and assembling errors includes gouges, nicks, burrs, scratches & dents. Material flaws include porosity, constituent particles, inclusions, forging or casting defects & improper thermal/mechanical treatment of basic alloy. Service induced damage includes cracks, damage due to foreign object includes bird strike effects etc. Some typical damages seen in aero-structural components are shown in Fig. 2.

![Fig. 2 Typical damages in aero-structural components](image)

3.1 Simulation of damages

All regions of a component must be able to withstand damages like dents, cracks, scratches and gouges, worst case will be cracks. Fig 3 and 4 show the simulation of dents and cracks (center crack).

![Global model with loads and BCs](image)

![Local model with loads & BCs](image)

![Stress (σyx) plot – with dent](image)

![Fig. 3 Simulation of Dent](image)
Different Fatigue Design Methodologies

The interest here is to determine the influence of the preexistent cracks on the structural component. In the following sections different fatigue design concepts are briefly discussed.

a) Infinite Life Design
In this design concept component stresses are kept below the endurance limit of the material. The endurance limit is the maximum constant amplitude stress that can be applied to a component without causing a fatigue failure.

Although the endurance limit concept was useful for solving the fatigue problems originally, this approach has many serious limitations. The endurance limit is extremely sensitive to the condition of the test specimen. Notches, small scratches, nicks, dings serve as stress concentrations that quickly cause localized fatigue cracking and greatly reduce the endurance limit. Since, it is impractical to design high-performance structures for infinite lives by the endurance limit methodology. It is now accepted that most structures will have a finite fatigue life.

b) Safe-Life Design
The safe-life approach treats fatigue as a crack nucleation process and does not explicitly consider the possibility for crack growth. The safe-life approach led to several inadequate aircraft designs in the 1960s; however, its current use is discouraged, in the main aircraft structures, except, landing gear and in some helicopter components.

c) Damage Tolerance Design
In order to overcome this shortcoming of the safe-life approach, damage tolerant design methods were developed that assume the structure contains initial cracks. The initial crack usually based on the inspection limits. There are two general approaches, with variations, that may be followed to guarantee that the structure does not fail in service, they are;
i. Slow Crack Growth: The slow crack growth design criteria select component material and sets stress levels so that the assumed preexistent crack will not grow to failure during service and is the normal approach for single load path structure. For increased safety, the allowed service life usually obtained by dividing the total crack growth period by a factor of 2. The component would have to be inspected at this time before continued operation would be permitted.

ii. Fail-Safe Design: This design concept assumes the possibility of multiple load paths and/or crack arrest features in the structure so that a single component failure does not lead to immediate loss of the entire structure. The load carried by the broken member is immediately picked up by adjacent structure and total fracture is avoided. It is essential; however, that the original failure be detected and promptly repaired, because the extra load they carry will shorten the fatigue lives of the remaining components.

4.1 Fatigue life estimation using Fracture of mechanics approach

In linear elastic fracture mechanics (LEFM), most formulas are obtained using either ‘plane stress’ or ‘plane strain’, associated with the three basic modes of loading[1], namely, opening, sliding and tearing. In the present analysis, LEFM approach is assumed where in Stress Intensity Factors (SIF) $K_I$ controls the crack growth. Fig 5 depicts a typical fracture and fatigue behavior of a cracked body.

![Fracture and Fatigue behavior of a cracked body](image)

The SIF, $K$ is related to applied load, crack size and geometry of the component. In equation form it is given by:

$$K = \sigma_{\text{max}} F \pi a$$

Where, $\sigma_{\text{max}}$ is applied stress, $a$ is crack size and $F$ is dimensionless factor (these factors are well documented for a number of crack configurations [2-4]. $K$ can be $K_I$, $K_{II}$ or $K_{III}$ depending upon the type of crack tip deformation.)
4.2 PARIS MODEL
A linear relationship between $\log \left( \frac{da}{dN} \right)$ vs. $\log \Delta K$ is observed between these two asymptotes, is used for fatigue crack growth predictions by Paris [5] equation in the equation
$$\frac{da}{dN} = C(\Delta K)^n$$
Where $C$ & $n$ are material / Paris constants.

Typical Examples used to Illustrate ADL and Damage tolerance Concepts

MSC Nastran is used for addressing all the ADL and damage tolerance problems. The von-Mises stresses are used to establish the margin of safety (MOS) for static strength, fatigue and crack growth loading environments given with these examples.

Configuration: A panel of dimension 21” X 21” is considered for analysis with a hole of diameter 3.0” to simulate the effect of damages like dents, cracks, scratches and gouges, critical being the crack, thickness of the panel is 0.02”. Damage is simulated by removing the material at the center of the plate. Fig 6 shows the dimensions, FE model of the panel.

Material properties: The material considered is Ti 6242, with Elastic Modulus, $E$ (psi) = 1.482E+07 and Poisson’s Ratio = 0.32.

Material allowable: $F_{tu}$@ RT = 143 ksi at RT, $F_{ty}$@ RT= 136 ksi at RT. Temperature reduction factor at 800 F = 0.72 for $F_{tu}$ and 0.6 for $F_{ty}$

5.1 Comparison of Stress Distribution to validate the model
The stress estimated (Fig 6) for the present problem is validated against available reference solutions [7], these calculations are given below;

Total load applied in X-direction, $P = 2304.51$ lb
Area of cross-section, far from hole, $A = 21 \times 0.02 \Rightarrow 0.42 \text{ in}^2$
Applied stress, $\sigma = 2304.51 / 0.42 \Rightarrow 5486.92 \text{ psi}$
Maximum stress from FE analysis, $\sigma_{\text{max}} = 17774 \text{ psi}$
Stress Concentration Factor, SCF, $K_{tg}$ (from FE results) = $\sigma_{\text{max}} / \sigma$
$K_{tg} = 17774 / 5486.92 \Rightarrow 3.24$
Stress Concentration Factor, SCF, $K_{tg}$ [7] = 3.09
There about 4.85 % difference, this makes reasonably good comparison and this mesh density is considered for further analysis.

5.2 Static strength estimation
For this problem, the stress resultant forces $N_x$, $N_y$ and $N_{xy}$ are estimated from the global solution and converted as appropriate nodal forces based on the panel dimension. Fig. 6 shows the loads and BCs for the panel with resulting von-Mises stress distribution.
Interlaminar Stress Calculation in Laminates Using an 8 Noded Brick Element Based on Mixed Finite Element Formulation

Fig. 6 Geometry, applied loads and stress distribution

Strength estimation: Using the maximum stress (von-Mises) obtained from the analysis, static margin of safety (MOS) obtained for 8000F as 1.48 and 1.95 for limit and ultimate loads respectively.

5.3 Fatigue life estimation
This allows us to calculate the damage for limit load using the maximum von-Mises stress estimated for configuration shown in Fig 6. The estimated number of cycles is very large. Fig 7 shows the typical S-N data for the given material indicating the current stress level and the expected life. From the below fig. it is seen that there no damage produced for the given limit load.

Fig. 7 S-N data for Ti-6242 Material

5.4 Damage Tolerance Design using fracture mechanics approach
Material constants and allowable: The material constants and allowable (nearest values from NASGROW 3.0 [8] are chosen) for Ti-6242 are taken as Yield Strength = 138 ksi, fracture toughness, $K_{IC} = 50$ ksi $\sqrt{\text{in}}$ and the Paris constants $C = 2.52\times10^{-9}$ (in/cycle) (ksi$\sqrt{\text{in}}$)$^{-n}$ and $n = 3.01$.

For the estimation of fatigue crack growth life the panel is assumed to contain either center or edge crack as shown in Fig 8. 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 below (for Center Crack)
Plate Width, \( W = 21.0" \), Initial Crack Size, \( a = 3.0" \) and thickness of the panel, \( t = 0.02" \)
Crack Configuration= through crack
Max Stress, \( \sigma_{\text{max}} = 10.572 \) ksi, Min Stress, \( \sigma_{\text{min}} = 0 \) ksi
Half crack size, \( a = \frac{a}{2} = 1.5" \), half Plate width, \( b = \frac{w}{2} = 10.5" \)
Applied Stress, \( \sigma_{\text{app}} = \sigma_{\text{max}} - \sigma_{\text{min}} = 10.572 \) ksi
Final half crack size based on Net Section Stress, \( a = b \left( 1 - \frac{\sigma_{\text{max}}}{\sigma_y} \right) = 9.70" \)

5.5 Fatigue Crack Growth Estimation for Center Crack

\[
a_i = a_{i-1} + \Delta a
\]

\[
F_i = \left[ 1.0 + 0.128 \left( \frac{a_i}{b} \right) - 0.285 \left( \frac{a_i}{b} \right)^2 + 1.523 \left( \frac{a_i}{b} \right)^3 \right]
\]

\[
\Delta K_i = F_i \sigma_{\text{app}} \sqrt{\pi . a_i}
\]

\[
\left( \frac{da}{\Delta N} \right)_i = C \left( \Delta K_i \right)^{\frac{2}{3}}
\]

\[
\left( \frac{da}{\Delta N} \right)_{\text{avg}} = \left[ \left( \frac{da}{\Delta N} \right)_i + \left( \frac{da}{\Delta N} \right)_{i+1} \right] / 2
\]

\[
(dN)_i = \frac{\Delta a}{\left( \frac{da}{\Delta N} \right)_{\text{avg}}}
\]

\[
N_i = \sum_{i=0}^{i} dN_i
\]

Where, \( a_i \) – Half crack size at \( i \)th iteration, \( \Delta a \) – Step size for crack growth, 
\( F_i \) – Shape Function, \( \Delta K \) - fracture toughness, \( \frac{da}{dN} \) – rate of crack growth, \( (dN)_i \) - no. of cycles for crack to grow \( \Delta a \) and \( N_i \) - No. of cycles for crack to grow from \( a_0 \) to \( a_i \).
For the given configuration using the above procedure, the crack growth both for CCT and edge crack configurations are obtained, the same is given in Fig 9 from the number of cycles and the allowable cycles for the panel, the MOS and the total number of inspection intervals are selected.

**Conclusion**

This paper discusses the importance of fatigue and damage tolerance analysis procedures, which are dominant factors in the aircraft structural design process as well as the following in service life. These concepts are illustrated with typical examples.

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**About Aerospace Practice**
TCS has been executing projects for Goodrich since August 2003 in the areas of Aero Structures, Landing Gear, Wheels & Brakes and Hoist & Winch etc. The Goodrich team currently has 25 industry bread design and Analysis Engineers. Goodrich team works closely with Aerospace Practice on Niche areas.

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