Damage tolerance design of transport aircraft

H Vishnu¹, R Ajith Kumar¹,³, R Sujith¹, R Suraj¹ and K E Girish²

¹ Department of Mechanical Engineering, SCMS School of Engineering and Technology, Kerala, India
² Director, Bangalore Aircraft Industries Ltd., Bangalore, India

Abstract. In a transport aircraft, there are normally two spars to withstand bending loads. A large part of this bending moment is drawn from the main spar. The main spar in large transport wings is an integrally machined part that is physically connected to the skin and ribs. At some fastener openings, the mechanical fastening contributes to extreme tension concentration where a fatigue crack can originate. This crack will form first in the flange under service loading and then expand into the spar network. If not detected during service and repaired, this crack growth can lead to catastrophic failure. This investigation aims at examining the main spar's alternate structural design to make it damage tolerant. With one more intermediate flange and at a height of one-third from the bottom flange, the spar construction will be in two separate parts. The bottom flange and web can fail in the case of fatigue cracking, but the top flange, web, and intermediate flange will remain intact and can be designed to bear the requisite load limit design. An approach to finite element modelling and analysis is also used to analyze all forms of spar constriction and test the principle of harm tolerance nature.

1. Introduction
Damage tolerance is a feature of a system that refers to its ability to safely retain faults before they can be damaged by repair. The approach to engineering design to allow for risk tolerance is based on the premise that in any system there will be defects and these defects spread with use. In aerospace engineering, this technique is also used to handle the expansion of structural cracks by applying the concepts of fracture mechanics.

The analysis of risk tolerance of aerial structural components [1] provides descriptions of the types of damage that are likely to occur during production and damage caused in operation. Harm resistance is the ability to withstand fracture for a given period of time from the pre-existing cracks and is an integral feature of materials whose failure could result in loss of life or property. The second line of protection against premature fracture is then provided by the integration of structural configurations and materials that are resistant to sub-critical crack growth and final fracture. “Operational survivability” was introduced [4] in 1976 similar to previous work. It contained the principles of fail-safe and hazard tolerance. Aviation regulations for transport aircraft were adopted in 1994, where the definition of operating survivability was assigned as the primary one. In C-130 aircraft [5], due to the consequences of hard use, various cracks are occurred. Damage tolerance design greatly lessens the possibility of such cracks causing a serious accident or requiring excessive down time for emergency repair. Cracks such as these were contained sufficiently by damage tolerance deign to permit safe operation until the using command was able to ferry the aircraft to a repair base.
Under conditions of severe usage, damage can develop and propagate with unexpected rapidity. A damage tolerant structure, however, greatly increases the chances of damage being detected before it reaches critical proportions affecting safe operation. Cracking of a spar web is a typical wing damage mode that may be treated analytically. Damage tolerance is achieved by ascertaining that un-failed wing structure is capable of redistributing the damaged spar web load without causing propagation of the damage. Due to fatigue break, numerous catastrophic failures of aircraft systems occurred in the 20th century [2]. In 1948, a Martin-202 aircraft with 40 passengers crashed because of wing failure caused by a fatigue fracture in a 7075-T6 wing spar joint. In 1948, a Martin-202 aircraft with 40 passengers crashed because of wing failure caused by a fatigue fracture in a 7075-T6 wing spar joint.

2. Problem Definition
The primary aim is to examine the main spar’s alternative structural architecture to make it tolerant of damage. At a height of 1/3rd from the bottom flange, the spar construction would be in two different components with one or more intermediate flange. The bottom flange and web can fail in the case of fatigue cracking, but the top flange, web, and intermediate flange will remain intact and can be designed to bear the requisite load limit design. In order to analyze all forms of spar construction and verify the principle of damage tolerance architecture, a finite element modelling and analysis approach will be employed.

2.1. Geometrical configuration
The required part is modeled in CATIA and the same was imported into NASTRAN for the analysis. The 3D sketch of the model used for the analysis is shown in the figure 1.

![Figure 1. Isometric view of the spar](image)

The material used for the model is an aluminum alloy. The name of the material is AL 2024 T351 alloy. The geometrical specifications of the model is taken as L₁ = 500 mm, L₂ = 4000 mm, T₁ = 2.5 mm, T₂ = 5 mm, t₁ = 1.5 mm and t₂ = 3 mm.

3. Stress analysis of spar
The finite element mesh model consists of quad, tria and bar elements. The total number of elements is shown in table 1. Also the finite element analysis of spar is shown in figure 2.

![Figure 2. Finite element form of spar](image)
Table 1. Model summary of wing spar

| Parameter      | Value  |
|----------------|--------|
| Grid points    | 39265  |
| Bar elements   | 900    |
| Quad elements  | 10592  |
| Tria elements  | 56     |

3.1. Load calculations

The various steps used for calculating the load in spar in discussed in this section. The all up weight is 2500 kgf while the load factor is taken as 3. Therefore the total design load will be 7500 kgf. This load is distributed on the wings and fuselage as 80% and 20% respectively. Therefore the load on both the wings of aircraft will be 80% of the total design load (7500 kgf) and that will be equal to 6000 kgf. In this study, we are concerned with the single wing. Hence the load on a single wing will be half of the load in both wings and that will be equal to 3000 kgf. Further, the load on front spar will be 75% of load in each wing and it will be equal to 2250 kgf. This load on the spar is given as distributed load on the model as shown in figure 3. The load at tip must be found out as shown in figure 4. It was calculated by momentum balance and was found to be 802.875 kgf.

![Figure 3. Load distribution of spar](image)

![Figure 4. Load calculation at tip](image)

Table 2. Value of load along length

| Parameter | Load value along length |
|-----------|-------------------------|
| Length (mm) | L₀=0  | L₁=562.5  | L₂=1125  | L₃=1687.5  | L₄=2250  | L₅=2812.5  | L₆=3375  | L₇=3937.5  | L₈=4500  |
| Load (kgf) | P₀=438 | P₁=394   | P₂=328   | P₃=272    | P₄=227   | P₅=206    | P₆=175   | P₇=119    | P₈=92    |
4. Evaluation of damage tolerance

For the damage tolerance analysis of the spar, stress intensity factor is to be calculated. The stress intensity factor is calculated using Modified virtual crack closure integral method (MVCCI). In order to validate this method, a classical problem was chosen. A rectangular plate with a crack at the center was taken for the validation purpose. The schematic diagram of this plate is shown in figure 4.

![Schematic diagram of rectangular plate with crack at center](image)

Figure 5. Schematic diagram of rectangular plate with crack at center

The input data for the given problem are \( P = 2000 \text{ kg} \), \( 2h = 800 \text{ mm} \), \( 2b = 400 \text{ mm} \) and thickness = 2mm. The rectangular plate as shown in the above figure was modelled analysed in the Patran. From the data obtained from the analysis, using MVCCI method, stress intensity factor was calculated. These values are compared with experimental proved value. It was found that SIF values obtained through FEM method is same as the experimentally proved values. Hence this FEM method is validated and it can be used for SIF calculation of any geometry.

4.1. Procedure for the theoretical calculation of SIF

At right angles, in an infinite plane, to a uniform stress field \( \sigma \), the stress pressure factor for a through crack of length \( 2a \) is given by

\[
K_1 = \sigma \sqrt{\pi a}
\]  

(1)

If the crack is centrally positioned on a finite plate of \( 2b \) width and \( 2h \) height, the stress strength element has an estimated relationship given by

\[
K_1 = \sigma \sqrt{\pi a} \times \frac{1 - \frac{a}{2b}}{\sqrt{1 - \frac{a}{b}}} + 0.326 \left( \frac{a}{b} \right)^2
\]

(2)

where \( \sigma \) is remotely applied stress, \( a \) is half of the crack length and \( b \) is half of plate width

4.2. MVCCI method for calculating SIF

In this method, SIF is calculated with the help of data obtained from FEM method. As per this method, stress intensity factor, \( K \) is given by

\[
K = \sqrt{E \times G}
\]

(3)

where \( G \) is the strain energy release rate which is given by

\[
G = \frac{1}{2 \Delta a} \times \Delta \nu \times \frac{F}{t}
\]

(4)
where $E$ is the Young’s Modulus, $\Delta a$ is length of element near the crack tip, $\Delta \nu$ is opening node displacement, $F$ is the force acting at crack tip and $t$ is the thickness of the plate. The energy dissipated during fracturing per unit of newly generated fracturing surface area is the strain energy release rate (or energy release rate). For fracture mechanics, this quantity is important because the energy that must be given to a crack tip for it to expand must be balanced by the amount of energy dissipated due to the creation of new surfaces and other dissipative processes such as plasticity. Data needed for calculation of $K$ which is to be obtained from the FEM analysis is shown in the figure 5.

![Figure 6. Parameters required for SIF calculation](image)

### 5. Results and discussions

In the initial stage, the spar was analyzed and the maximum stress occurring region was found out. The stress values occurred at the corresponding regions is given in the table below. In this case, stress occurred due to tension is taken into consideration.

| Part         | Stress (kg/mm²) |
|--------------|-----------------|
| Top skin     | -25.8           |
| Top flange   | -31.4           |
| Bottom skin  | 7.33            |
| Bottom flange| 19.9            |

Table 3. Maximum shear stress values at each part

After finding out the maximum shear stress region, using MVCCI method, Stress Intensity Factor was calculated. The SIF values for each crack length was found out and tabulated in table 4.

| Parameter | Crack length and SIF values |
|-----------|-----------------------------|
| Crack length | 5 | 8 | 9.4 | 11 | 12 | 46 |
| SIF (K)    | 11.661 | 14.454 | 15.871 | 17.203 | 18.443 | 48.208 |

Table 4. Values of K for different crack lengths

The variation values of SIF along the crack length is shown in figure 7. It is clear from the graph that the SIF values are getting higher as the crack length is increased and hence the flange will get failed. After the complete failure of flange, the crack get initiated through the web and propagate throughout the spar which result in the failure of the aircraft structure.
In the further step, crack was simulated through the web and SIF was calculated for the different crack lengths. The variation of SIF for different crack lengths in web along crack length is plotted in figure 8. From the graphical plot it is clear that SIF value keep on increasing which finally results in the failure of the structure. So the design was further modified.

So in order to modify the model, an intermediate flange is introduced at a height one-third from the bottom flange and the same procedure was followed. Here also a graph is plotted to depict the variation of SIF along the crack length with intermediate flange in figure 9. It can be seen from the plot that the SIF value got reduced when the crack is approaching near the intermediate flange. So it is clear that by introducing an intermediate flange, the crack propagation can be arrested to an extent, hence the wing spar can be made as damage tolerant.
6. Conclusions
In this study the damage tolerance design of transport aircraft is studied. The stress analysis is carried out using finite element method (FEM). The stress on the bottom skin and bottom flange are obtained and it was found that stress on bottom flange is greater than that of bottom skin. After doing the finite element analysis, the Stress Intensity Factor (SIF) was calculated verified using MVCCI method. The damage tolerance calculation is carried out by finding out the SIF at different crack lengths. Modifications was done on the model to minimize the damage tolerance and it was found that the structure will fail in all cases until the use of intermediate flange. The stress analysis and SIF calculation has been carried out in the presence of intermediate flange. The results with the presence of intermediate flange show that SIF decreases when the crack get nearer to intermediate flange.

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