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Fatigue Life Estimation of Fuselage Structure Due to Fluctuating Bending Loads

Mallikarjun[1], Vidyadhar Pujar[2]

[1]PG Scholar, The Oxford College of Engineering Bangalore, Karnataka, India
[2]Assistant Professor, The Oxford College of Engineering, Bangalore, Karnataka, India

Abstract: ***Aircraft structure is the most obvious example where structural efficiency results in light weight and high operating stresses. Airframe experiences variable loading in service. It is quite unlikely that the structure will fail due to a static overload. Aerodynamic load distribution on the airframe will make the fuselage structure to bend about wing axis. Bending of the fuselage structure introduces both tension and compression stress field in the structure. Linear static stress analysis is carried out for identification of the fatigue critical locations. The project includes the fatigue life estimation for the fuselage structure due to fluctuating loads on the fuselage. Stress analysis of the segment of the fuselage will be carried out by using finite element method. Local analysis will be carried out to capture the stress concentration factor and stress distribution near the high stress location. Miner's rule will be used for fatigue damage calculation with the help of the S-N diagram of the respective material used in the structure.***

Keywords: Airframe, Fuselage, Fatigue, Stress concentration, FEA, Miner's rule, Damage calculation

1. Introduction

The performance of the aircraft depends on the life span of the different components. There are number of cycles each and every component undergoes damage. Structure of an aircraft plays a major role in resisting different types of loads in different conditions. Usually different composite materials are used for building a light weight, stiff and resistive structure. The weight of structure also plays an important role in performance and life span of the aircraft. The major problem is to balance both structural material and weight which in turn increases the lifespan and performance of an aircraft. Normally aircraft structure damages due to the application of different cyclic loads at different segments of its journey. Due to continuous applications of loads the structures degrade. This degradation of structure due to application of cyclic loads is called fatigue analysis. Each and every component of an aircraft undergoes fatigue damage. Now our next step is to select the component which undergoes fatigue damage.

2. Material Specification

Selection of aircraft materials depends on any considerations, which can in general be categorized as cost and structural performance. The key material properties that are pertinent to maintenance cost and structural performance are:

- Density
- Young's modulus
- Ultimate and Yield strengths
- Fatigue strength
- Damage tolerance (fracture toughness and crack growth)
- Corrosion, etc.

Mechanical properties of the skin, stiffening members and rivets are required for finite element models. There is little information on the material properties of skin, stiffening members, and rivet material in the literature. Aluminium 2024-T3 is used for components fuselage and rivet. Table 3.1 describes few material properties.

Table 2.1: Material properties used for the analysis

Property	Aluminium 2024-T3
Density	2.77 Kg/cm3
Ultimate Tensile Strength	483 MPa
Tensile Yield Strength	362 MPa
Young's Modulus	70GPa
Poisson's Ratio	0.33
Fracture Toughness	72.37 MPa√m

3. Geometric Modelling

Fuselage is a part of aircraft structure having cylindrical shape. Basically the fuselage structure consists of circumferential member called bulkheads to maintain circumferential shape and it is taking hoop stress which is created due to internal pressurisation. It has one longitudinal member also known as longenors which take longitudinal stress and support to the skin. Bulkheads, longenors, tear strap and skin are connected by rivet connection.

The bulkheads has z cross section and total eleven bulkheads in the fuselage, the longenores has L cross section and total 28 longenores. As shown in below fig. 3.1.

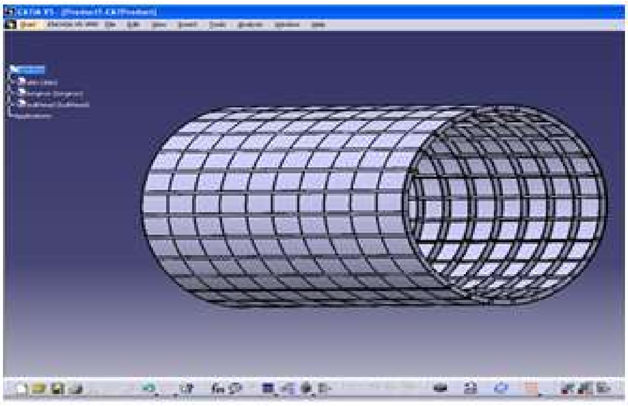

Figure 3.1: Fuselage Model

Dimensions

Length of the fuselage = 4500mm
Radius of the fuselage = 1000mm
Thickness of skin = 2mm

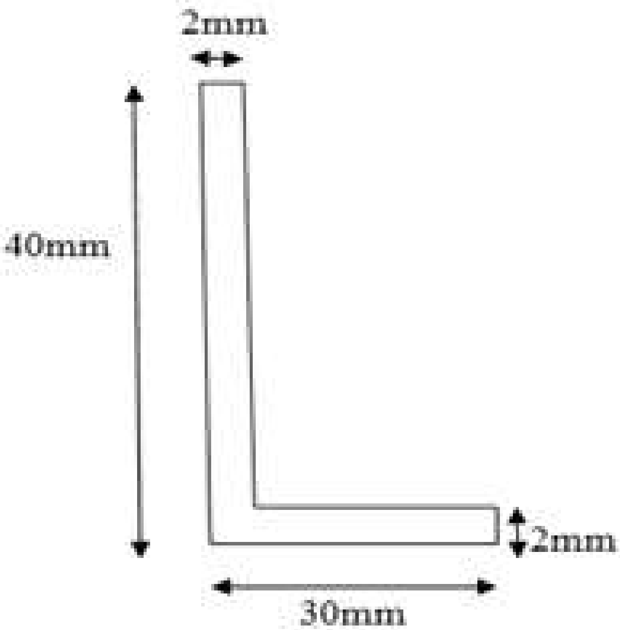

Figure 3.2: stringer

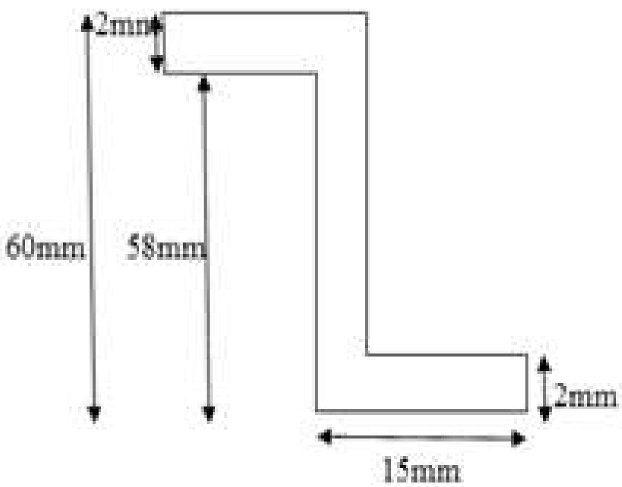

Figure 3.3: z- stringer

4. Linear Static Stress Analysis

4.1. Finite Elemene Model

The fuselage and its parts are meshed by one dimensional and two dimensional. Skin of the fuselage structure is meshed by shell elements (2 D elements) with unit aspect ratio. A bulkhead & longerons of the fuselage is meshed by one dimensional & two dimensional elements. Fine mesh is carried at stress concentration of frame to get accurate results. The inertia load is uniformly distributed on circumferential of bulkheads in transverse axial direction in compression .Both end of structure has bending moment and shear force, that bending moment and shear force calculated from tail end and rear end weight of the aircraft. The structure is fixed at nodes of the skin where the wings attached to the fuselage structure and that nodes of the skin are constrained in all six degree of freedom

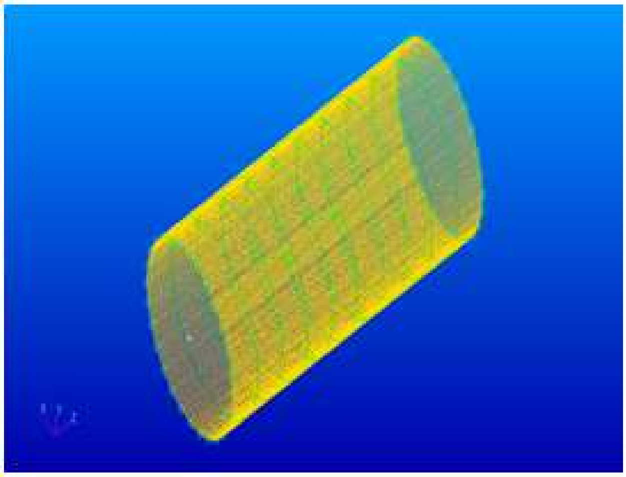

Figure 4.1: Finite Element Modelling of the fuselage

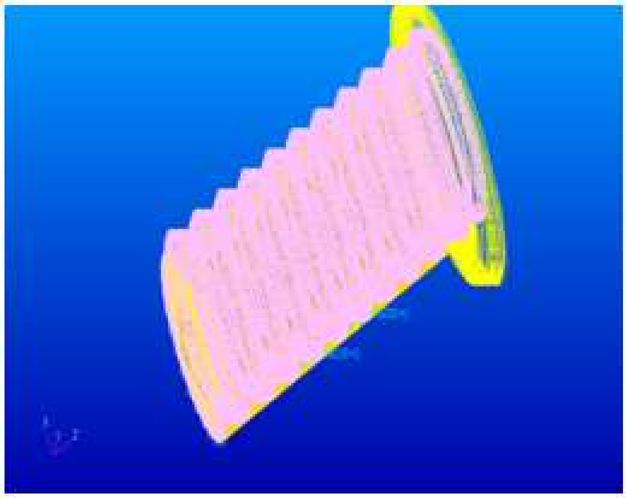

Figure 4.2: Loads and boundary conditions of the fuselage

4.2 Stresses and Deformation

The magnitude of maximum compressive stress is 30.4 kg/mm2 at fixed end on skin. The maximum stress locations are the probable locations where the crack initiation. Skin is the most critical stress locations for the crack initiation. The actual deformation in structure is how much distance transform of body from one position to another position it is different value along the length of structure. In this structure maximum deformation is at free end of structure and the magnitude of maximum deformation is mm.

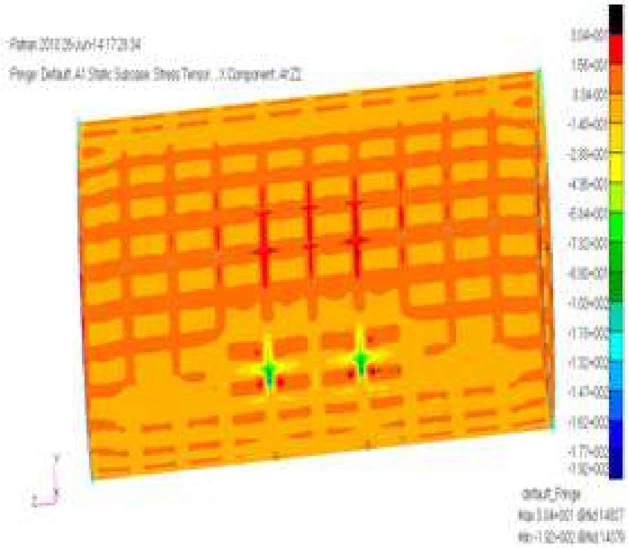

Figure 4.3: Stress Counter

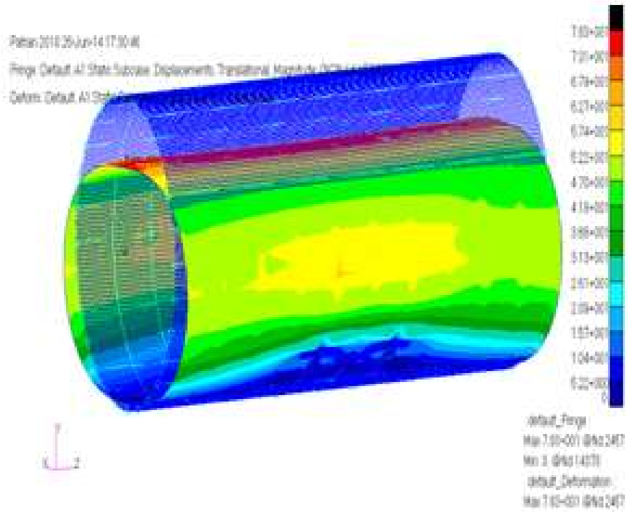

Figure 4.4: Deformation

5. Local Analysis of Stiffened Panel

5.1 Finite Element Model

The stiffened panel has taken from the fuselage structure where the maximum stress in the fuselage structure occurred from the global analysis. The stiffened panel taken from between two bulkheads and centre of two longenors. The longenors are connected to the skin of the fuselage with rivets. One end of stiffened panel structure nodes are fixed and another end apply the axial load for both skin and longenors. The structure is fixed at nodes of the skin and longenors are constrained in all six degree of freedom (three translations and three rotations), and also the stiffened panel constrained in z-direction.

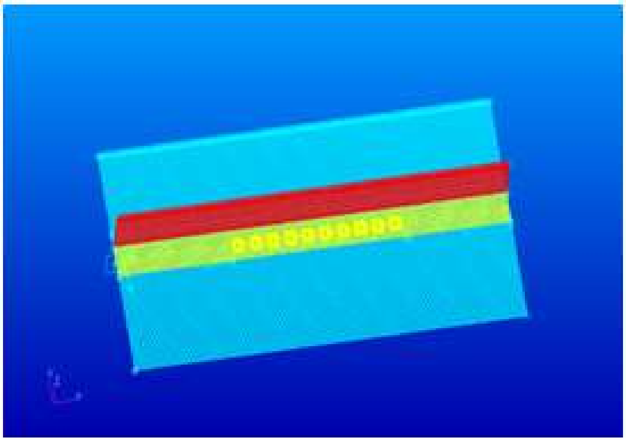

Figure 5.1: Finite Element Modelling of stiffened panel

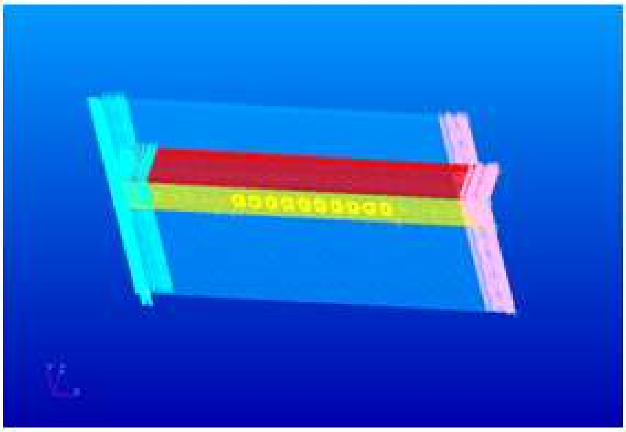

Figure 5.2: Load and boundary condition of stiffened panel

5.2 Stresses and Deformation

The maximum stress developed near the rivet holes of both skin and longenores and nominal stress all over the stiffened panel. The stress near the hole is three times of nominal stress. In figure the red colour shows the maximum stress. At the rivet hole the localization stresses because the area reduces and also stress concentration

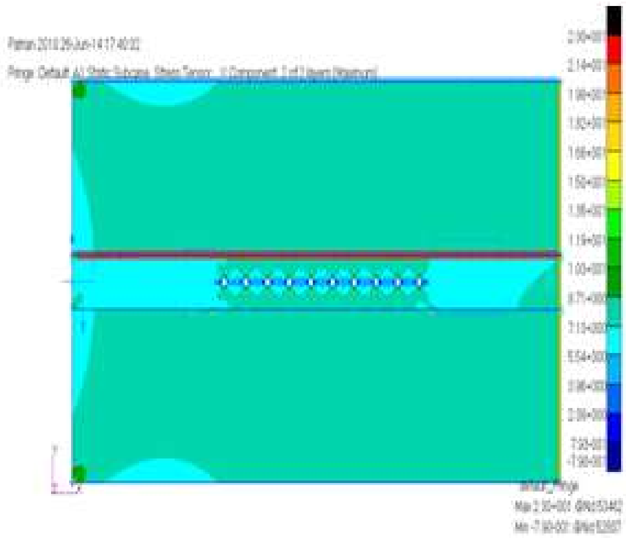

Figure 5.3: Stress distribution in stiffened panel

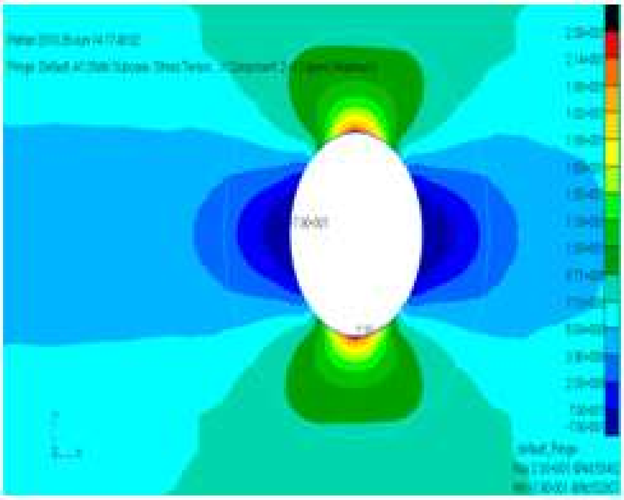

Figure 5.4: Close-up view Stress distribution in stiffened Panel

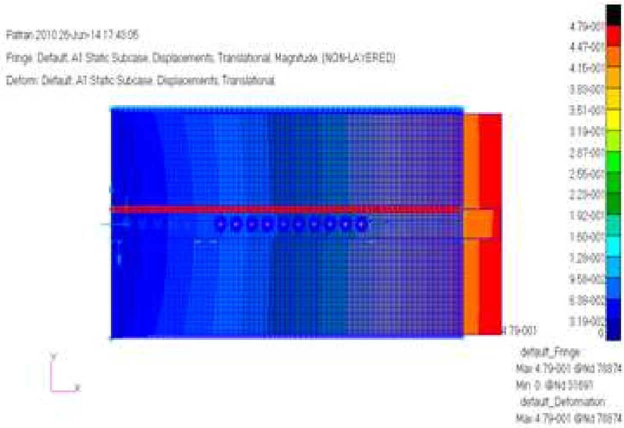

Figure 5.5: Deformation of stiffened panel

6. Results and Discussion

6.1 Calculation of Stress and deformation

1. Stress calculation:

Load on the skin = 3315.2 kg
Load on the longenores = 1036 kg
Cross section are = w × t mm2
Cross section area of skin = 224 × 2 = 448 mm2
Cross section are of longenores = (40×2) + (30×2) = 140 mm2
Total load on stiffened panel = 3315.2 + 1036 = 4351.2 kg
Total area of the stiffened panel = 448 + 140 = 588 mm2

Stress on the stiffened panel = $\frac{\text{Load}}{\text{Area}}$

$$\sigma = \frac{L}{A}$$

$$\sigma = \frac{4351.2}{588}$$

σ = 7.4 kg/mm2

The nominal distributed over the stiffened panel is 7.4 kg/mm2 except near the rivet hole. At the rivet hole the stress is maximum and three times of the nominal stress is 20 kg/mm2.

6.2 Fatigue Life Estimation

6.2.1 S –N Curve

From typical constant life diagram for un-notched fatigue behaviour of 2024- T3 Aluminium alloy High-Master diagram is shown in below figure. The reference test condition R=0 used for obtain fatigue properties. For this condition smin=0 is called 'pulsating tension' under constant amplitude loading or Zero to tension loading. The numbers of cycles to failure from graph.

Table shows the alternating stress level below which the material has an infinite life. For most engineering purposes, infinite is taken to be 1 million cycles. According to Palmgren-miner's rule the stress amplitude is linearly proportional to the ratio of number of operation cycles to the number of cycles to failure from the graph gives the damage accumulated.

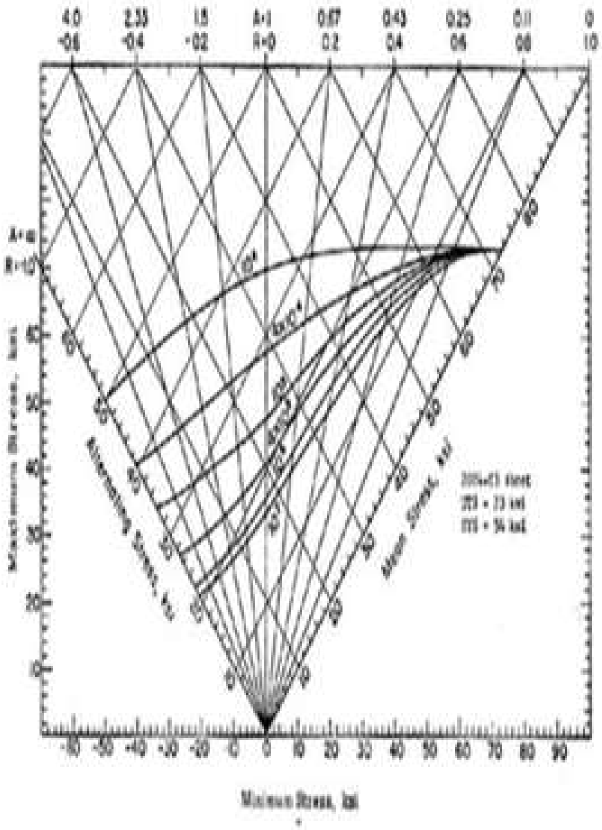

Figure 6.1: S-N curve

Table 6.1: Values of Damage Accumulated

Serial No.	Different G Condition	Alternative Stress (ksi)	Ratio R	No. cycle Induced Ni	No. cycles to failure Nf	Damage accumulated Di
1	0.5G to 0.75G	0.85	0.666	15000	10^7	0.0015
2	0.75G To 1G	0.8335	0.75	11000	10^7	0.0011
3	1G To 1.25G	0.8333	0.8	10000	10^7	0.001
4	1.25G To 1.5G	0.8335	0.833	8000	10^7	0.0008
5	0 To 1.75G	5.8335	0	20	10^7	0.000006
6	0 To 2G	6.667	0	1	10^7	0.0000001
7	-0.5G To 1.5G	6.665	-0.33	100	10^7	0.00001

The total damage accumulated is Di= 0.0044< 1 which is less than unity, therefore the crack not initiate

7. Conclusion

Linear static Stress analysis of the Fuselage structure was carried out and maximum stress was identified on the skin. Around the maximum stress location, we have taken one cut-out of the fuselage called stiffened panel. Local analysis of the stiffened panel was carried out by applying average tensile load on skin and maximum tensile load on longenore. The maximum stress found around rivet holes of both skin and longenore is 20kg/mm2.

Fatigue life estimated of the fuselage structure considering the maximum stress of the stiffened panel with the help of S-N curve and Miner's rule. The damage accumulated of the Fuselage structure is 0.0044 from this it is observed that, the remaining life of the structure is 0.9966. The fuselage structure lost 44121 fatigue cycle and remaining life is 9993406cycles.

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