

Stress Analysis and Fatigue Estimation of Centre Fuselage of an Aircraft

Manjunath M M¹, Vijeth G K², Rajiual Hoque³, Kannan P Mohanan⁴, Ashok Kumar A⁵, Shankar Gouda⁶

U G Student, Department of Mechanical Engineering, SCE, Bangalore, Chickballapur, India^{1,2,3,4}.

Asst. Professor, Department of Mechanical Engineering, SCE, Chickballapur, Karnataka, India⁵.

Asst. Professor, Department of Mechanical Engineering, SCE, Chickballapur, Karnataka, India⁶.

ABSTRACT: Aircraft designer needs to ensure the structural integrity of the airframe without compromising on the safety of the structure. This requires several stress analysis of different components, which represents the features of the airframe. Also these analysis predictions have to be substantiated by structural testing during the developmental phase. In current study a representative stiffened panel from a centre fuselage segment of an aircraft will be considered for the evaluation. The stress analysis of the stiffened panel will be carried out by using FEM approach. Aluminium alloy 2024-T351 material is used for the stiffened panel. Fuselage structure experiences the hoop tension and longitudinal tension because of the internal pressurization. If there is a crack in the unstiffened fuselage, under the flight condition it could lead to catastrophic failure of the structure. The fatigue crack will initiate normally from the highest tensile stress locations. Therefore rivet locations are the most probable locations for fatigue crack initiation. Miner's rule will be employed for the fatigue damage calculation and life estimation of the structure. S-N curve of the aluminium alloy 2024-T351 will be used for obtaining the number of cycles to failure at particular stress magnitudes.

KEYWORDS: Aircraft, Stiffened panel, internal pressurization, Finite element analysis, Fatigue crack, miners rule.

I. INTRODUCTION

Aircraft is the science involved with the study, design, and manufacture of flight-capable machines, or the techniques of operating aircraft, one of the significant parts in aeronautics is a branch of physical science called aerodynamics, which deals with the motion of air and the way it interacts with objects in motion, such as an aircraft. Main components of an aircraft are, Fuselage, Wing Empennage or Tail, Power Plant, Landing gear or under carriage.

Fuselage: The fuselage of an aircraft consists of sheet panels, stringers, and stiffeners held together by riveted lap joints. Although different joining techniques exist, the skin panels are typically fastened together with rivets. Numerous rivets are required to join the skin completely. In transport aircraft, the majority of the fuselage is cylindrical or near-cylindrical, with tapered nose and tail sections.

II. DESIGN CONSIDERATION

Introduction to Fatigue

Fatigue is the progressive and localized structural damage that occurs when a material is subjected to cyclic loading. The nominal maximum stress values are less than the ultimate tensile stress limit, and may be below the yield stress limit of the material.

These words are important perspectives on the phenomenon of fatigue:

- Process
- Progressive
- Localized
- Permanent structural change
- Fluctuating stresses and strains
- Point or points
- Cracks or complete fracture

International Journal of Innovative Research in Science, Engineering and Technology

(An ISO 3297: 2007 Certified Organization)

Vol. 4, Issue 5, May 2015

Characteristics of Fatigue

- In metals and alloys, the process starts with dislocation movements, eventually forming persistent slip bands that nucleate short cracks.
- Fatigue is a stochastic process, often showing considerable scatter even in controlled environments.

The S-N Curves

Tests on several specimens are conducted under identical conditions with varying levels of stress amplitude. The cyclic stress level of the first set of tests is some large percentage of the Ultimate Tensile Stress (UTS), which produces failure in a relatively small number of cycles. Subsequent tests are run at lower cyclic stress values until a level is found at which the samples will survive 10 million cycles without failure.

The results are plotted as an S-N diagram (see the below figure) usually on semilog or on log-log paper, depicting the life in number of cycles tested as a function of the stress amplitude. A typical plot is shown in the [Figure] below for two classes of materials.

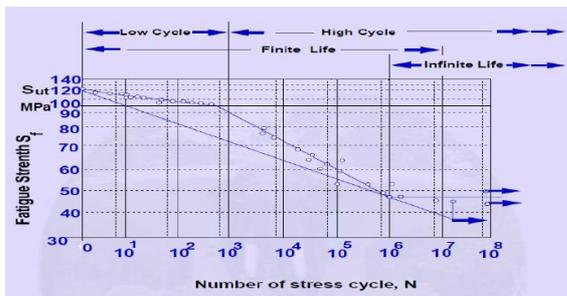


Fig 1.1 (a): S-N curve for two classes of materials

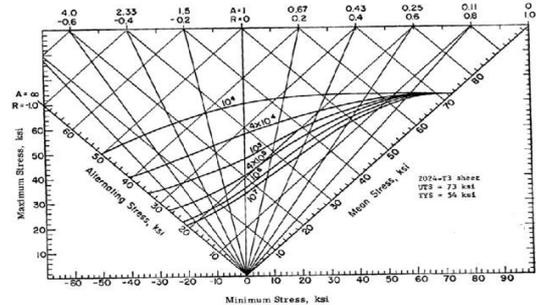


Fig 1.1 (b): S-N curve for Aluminium alloy 2024 T351

Several useful terms and symbols; these include:

$$\sigma_m = \text{Mean stress} = \frac{\sigma_{max} + \sigma_{min}}{2}$$

$$\sigma_{Max} = \text{Maximum stress in the cycle}$$

$$\sigma_{Min} = \text{Minimum stress in the cycle}$$

$$\sigma_a = \text{Alternating stress amplitude} = \frac{\sigma_{max} - \sigma_{min}}{2}$$

$$q\Delta\sigma = \text{Range of stress} = \sigma_{max} - \sigma_{min}$$

$$R = \text{Stress ratio} = \frac{\sigma_{min}}{\sigma_{max}}$$

$$A = \text{Amplitude ratio} = \left(\frac{\sigma_a}{\sigma_m} \right) = \frac{1-R}{1+R}$$

III. MATERIAL SPECIFICATION

Material for centrefuselage: Aluminium 2024-T351 [1].

1. Young's Modulus, $E = 73.1 \text{ Gpa}$
2. Poison's Ratio, $\mu = 0.33$
3. Ultimate Strength, $\sigma_u = 470 \text{ Mpa}$
4. Yield strength, $\sigma_y = 280 \text{ Mpa}$
5. Fatigue strength, $\sigma_f = 138 \text{ Mpa}$

IV. FINITE ELEMENT ANALYSIS

Stress analysis of fuselage segment

Finite Element Model of the fuselage segment Skin

Fig. 4.1 shows the mesh was carefully generated such that there is a node present at the point where riveting used to be simulated. The fig shows the rivets that are placed on the skin to hold the bulkheads, Longerons.

International Journal of Innovative Research in Science, Engineering and Technology

(An ISO 3297: 2007 Certified Organization)

Vol. 4, Issue 5, May 2015

Table 4.1: Elements detail of Skin

Sl. No	Types of element	Count
1	No of CQUAD element	12844
2	No of Nodes	13090
3	TOTAL no of element	12844

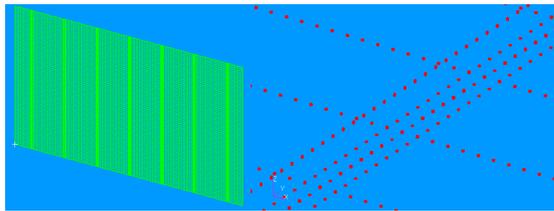


Fig. 4.1: Meshed skin with beam elements as rivet

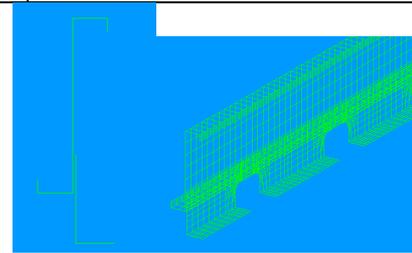


Fig 4.2: 2D Bulkhead

Fig 4.2 shows the bulkhead, has Z and L cross-section. The bulkheads are placed on top of the skin and riveted onto the skin. These bulkheads run in circumferential direction. There are stringer cut-out (mouse holes) provided in the bulkhead so that the stringers run continuously from one end to the other in longitudinal direction.

Table 4.2: Elements detail of Bulkhead

Sl. No	Types of element	Count
1	No of CQUAD element	39004
2	No of Tria element	378
3	TOTAL no of Nodes	27720
4	TOTAL no of element	66724

Stiffener

The Longerons has hat cross-section. These are riveted to skin in longitudinal x-direction. The meshing is maintained such that there is a node present at the point where a rivet is used to fasten the longerons onto the skin. The Finite elements mesh for stiffener shown in fig 4.4(a). Fig 4.4(b) shows the Meshed components of the skin panels are assembled and equivalence is given, after that boundary verification is done in order to check the connectivity to each component

Table 4.3: Elements detail of Stiffener (Longeron)

Sl. No	Types of element	Count
1	No of CQUAD element	90719
2	TOTAL no of Nodes	123350
3	TOTAL no of element	90719

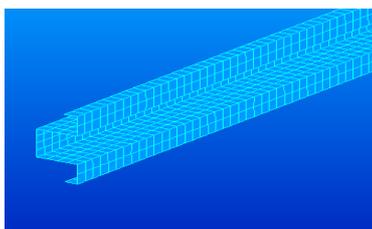


Fig 4.3(a): 2D Stiffener

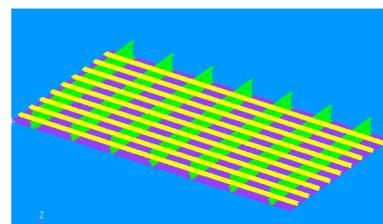


Fig 4.3(b): Meshed component assembled

Quality verification for the FE model

The accuracy of the stress analysis result depends on the quality of the mesh generated and the quality of the elements used to represent the geometry of the components. Following are the element quality parameters obtained from the mesh generated for the segment of the fuselage being analyzed.

International Journal of Innovative Research in Science, Engineering and Technology

(An ISO 3297: 2007 Certified Organization)

Vol. 4, Issue 5, May 2015

CQUAD

Aspect = max 3.0
Warp = max 0.0

CTRIA

Aspect = max 1.732
Skew = min 44.999

Skew angle = min 89.97
Taper = max 0.000591

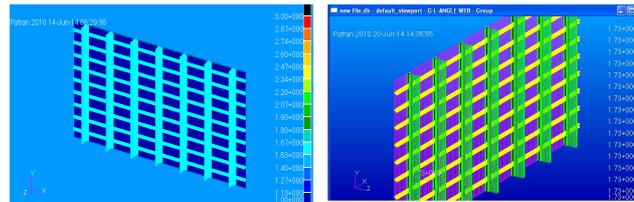


Fig 4.4: Element aspect ratio

Load cases and boundary conditions on fuselage segment

A differential pressure of 6 psi (0.0007 kg/mm²) is considered for the current case. Due to this internal pressurization of fuselage (passenger cabin) the hoop stress will be developed in the fuselage structure. The tensile loads at the edge of the panel corresponding to pressurization will be considered for the linear static analysis of the segment.

Hoop stress is given by,

$$\sigma_h = \frac{Pd}{2t}$$

Where,

Cabin pressure (p) =6 psi=0.0042 kg/mm²

Diameter of fuselage (d) = 2800 mm

Thickness of skin (t) = 1.8 mm

After substitution of these values in Equation we will get,

$$\sigma_{hoop} = 3.27 \text{ kg/mm}^2$$

By using equation (4.1) Hoop stress is calculated, loads acting on skin, Tear strap and Bulkhead can be calculated using equation is shown as follows.

$$P = \sigma_h * A$$

Where,

P-Load

A- Cross section area

Load acting on skin

Form equation

$$3.27 = \frac{P}{3556 * 1.8}$$

P= 20930.61 Kg

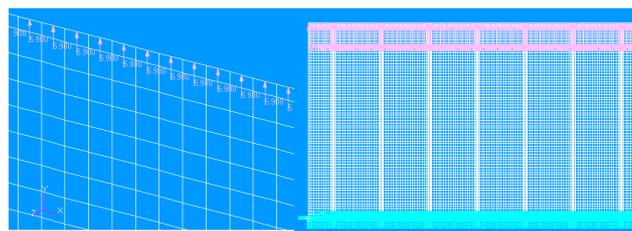


Fig 4.5: uniformly distributed load acting on skin

Uniformly distributed load on skin will be

$$P_s = \frac{20930.61}{3556}$$

P_s = 5.885 kg/mm

Load acting on Bulk head

Length of Z-section

$$Z = (8.625 * 2 + 22.23 * 2 + 112.5)$$

Z= 174.21mm

Length of L-section

$$L = 56.85 + 24.5$$

L= 81.35mm

Therefore total length of bulkhead= (Z+L) =255.56

Form equation

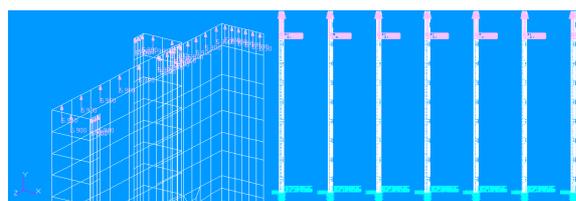


Fig 4.6: uniformly distributed load acting on bulkhead

$$3.27 = \frac{P}{255.56 \times 1.8}$$

Pb= 1504.22 Kg (load acting on each bulkhead)

Total load acting on bulkhead= 10529 kg

Uniformly distributed load on each bulkhead is

$$Pb=1504.22/255.56$$

$$Pb=5.885 \text{ kg/mm}$$

Displacement Contour of the fuselage segment

The Fig.4.7 shows the displacement contour of fuselage segment, Displacement can be obtained from fixing at one end and applying load at other end by constraining stringer ,skin, and bulkheads in z-direction, as shown by different colours fringes showing minimum magnitude of displacement while red colour showing maximum magnitude of displacement as 1.42 mm.

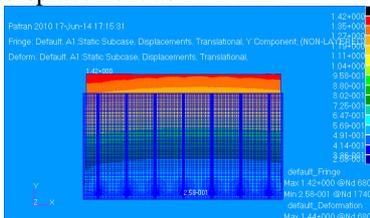


Fig 4.7: Displacement Contour of the fuselage segment

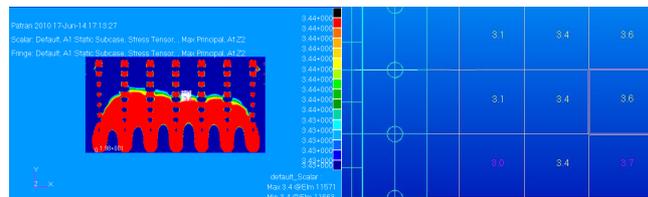


Fig 4.8: Stress Contour of the fuselage segment

Stress Contour of the fuselage segment

Fig.4.8 shows the stress contour on the skin from fuselage segment analysis results. It is clear that the maximum stress on skin is at the rivet location where the rivets are used to fasten the bulkheads, longerons, tear strap and skin. The magnitude of maximum tensile stress is 3.4kg/mm².The maximum tensile stress locations are at rivet in the skin. The fatigue cracks will initiate normally from the highest tensile stress locations therefore rivet locations are the most probable locations for fatigue crack initiation. The panel will be analysed with central crack bulkhead broken condition is discussed.

V. RESULTS AND DISCUSSIONS

Load calculation

A differential pressure of 6 psi (1psi=0.0007 kg/mm²) is considered for the current case. Due to internal pressurization of fuselage (passenger cabin) the hoop stress will be developed in the fuselage structure. The tensile loads at the edge of the panel corresponding to pressurization will be considered for the linear static analysis of the panel.

Hoop stress is given by,

$$\sigma_h = \frac{Pd}{2t}$$

Where, Cabin pressure (p) =6 psi=0.0042 kg/mm²

Diameter of fuselage (d) = 2800 mm

Thickness of skin (t) = 1.8 mm

After substitution of these values in Equation (4.1) we will get,

$$\sigma_{hoop} = 3.27 \text{ kg/mm}^2$$

Loads acting on each component of the fuselage segments are as follows:

Sl no	Description	Load(Kg)
1	Skin	20930.61
2	Tear strap	158.22
3	Bulkhead	1504.22

International Journal of Innovative Research in Science, Engineering and Technology

(An ISO 3297: 2007 Certified Organization)

Vol. 4, Issue 5, May 2015

Fatigue Life Calculation Results

Table 5.1: Number of cycles to failure and damage accumulated from miner's formula

Range of 'g'	No of cycle	Damage Accumulated
0.5 to 0.75g	15000.00	1.50E-03
0.75 to 1g	11000.00	1.10E-03
1 to 1.25g	10000.00	1.00E-03
1.25 to 1.5g	8000.00	8.00E-04
0 to 1.75g	20.00	2.00E-06
0 to 2g	1.00	1.00E-07
-0.5g to 1.5g	100.00	1.00E-05

From results of fatigue analysis for different pressure cycles got the damage fraction is less than unity i.e. $1.5E-03+1.10E-03+1E-03+8E-04+2E-06+1E-07+1E-05=0.000441$. According to Palmgren-Miner linear damage rule when the damage fraction is less than unity the material is safe, often satisfactorily for failure is predicted. The damage at which failure is expected to occurs when the damage fraction is equal to 1

VI. CONCLUSION

- In current study a representative fuselage segment from a transport airframe is considered for the evaluation.
- Aluminum alloy 2024-T351 material is used for the fuselage segment.
- Fuselage structure experiences the hoop tension and longitudinal tension because of the internal pressurization,

REFERENCES

- [1] Uwe Zerbst, Markus Heinemann, Claudio Dalle Donne, Dirk Steglich. "Fracture and damage mechanics modelling of thin-walled structures" – An overview Engineering Fracture Mechanics, Volume 76, Issue 1, Pages 5-43.
- [2] E. Wyart, D. Coulon, T. Pardoën, J.F. Remacle, F. Lani , "Application of the sub structured finite element/extended finite element method (S-FE/XFE) to the analysis of crack in aircraft thin walled structure" Engineering Fracture Mechanics, Volume 76, Issue 1, Pages 44-58.
- [3] Pir M Toor, "On Damage Tolerance Design of Fuselage Structure (Longitudinal Crack)", engineering fracture mechanic, Vol 24. No 6, pp915-927.1986.
- [4] Pir M Toor, "On Damage Tolerance Design of Fuselage Structure (Circumferential Crack)", engineering fracture mechanic, Vol 26. No 5, pp771-782, 1987.

BIOGRAPHY



Mr. Manjunath M M persuing his B.E in Mechanical Engineering under VTU, Belgaum from Sha-Shib College of Engineering, Chikkaballapur.



Mr. Rajiual Hoque persuing his B.E in Mechanical Engineering under VTU, Belgaum from Sha-Shib College of Engineering, Chikkaballapur.



Mr. Kannan P Mohanan persuing his B.E in Mechanical Engineering under VTU, Belgaum from Sha-Shib College of Engineering, Chikkaballapur.

International Journal of Innovative Research in Science, Engineering and Technology

(An ISO 3297: 2007 Certified Organization)

Vol. 4, Issue 5, May 2015



Mr. Vijeth G K persuing his B.E in Mechanical Engineering under VTU, Belgaum from Sha-Shib College of Engineering, Chikkaballapur.



Mr. Ashok Kumar Currently working as Asst. Professor in Sha-Shib College of Engineering, Chikkaballapur. He did his M.tech in Machine Design from Nagarjuna College of Engineering & Technology, Bangalore. He has published several papers in various International Journals.



Mr. Shankar Gouda Currently working as Asst. Professor in Sha-Shib College of Engineering, Chikkaballapur. He did his M.tech in Product Design & Manufacturing from EPCET, Bangalore. He has published several papers in various International Journals.