Stress Analysis of Fuselage Frame with Wing Attachment Beam and Fatigue Damage Estimation

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Abstract— **Airframe is a built-up structure. Fatigue cracks are bound to occur at maximum tensile stress locations. Fatigue cracks due to fluctuating loads are common problems experienced during service life of the aircraft. Cracks are allowed in the aircraft, but they should not lead to catastrophic failure of the structure. Fatigue cracking locations are identified through linear static stress analysis of the structure. Wings are the major components of the airframe. Spars in the wing carry most of the bending due to lift load during flight. Wings will almost behave like cantilever structures. Therefore, the maximum bending moment will be at the root. Wings are attached to the fuselage structure through attachment brackets. The bending moment and shear loads from the wing are transferred to the fuselage through the attachment brackets. The current project includes the linear static analysis of the bulkhead frames along with spar beam and fatigue damage estimation at the critical location due to fluctuating loads. Lug-holes and bolt-holes are likely to experience more stress due to high stress concentration. Stress analysis will be carried out finite element method. A local analysis will be carried out to capture high stress magnitude and stress distribution. Airframe experiences variable loading during flight conditions. A typical transport aircraft load spectrum will be used for fatigue damage calculation. In a metallic structure fatigue manifests itself in the form of a crack, which propagates. If the crack in a critical location goes unnoticed it could lead to a catastrophic failure of the airframe. Fatigue damage estimation will be carried out using constant amplitude S-N data for various stress ratios and local stress history at stress concentration.**

Keywords: Fatigue, Transport aircraft, Fuselage frame, Attachment bracket, Stress concentration, Fatigue-life, S-N data.

I. INTRODUCTION

Since ancient time transportation system played a vital role in the development and prosperity of any society. Nowadays airplanes are the most popular mode of transport when it is required to travel large distances. Aircrafts come in different sizes, wing and shape configurations. The reason behind aircraft structure is for ease of transportation of goods and people in civil as well as military areas. Earlier all the aircrafts were driven by pilots sitting in cockpit, but in this new era of modernization, the self remotely controlled, computerized controlled or automated aircrafts are available.

Most aircraft wings are classified into two types, one is fixed and other is rotary. Wings are powered with a forward propeller by thrust through a turbojet engine. The movement of aircraft generates lift to hold the plane in the air. To obtain this lift, airplanes are designed aerodynamically and they are pushed through the air. Mostly, four types of loads act on aircraft structure viz. thrust, drag, lift and weight.

Fatigue cracks, which occur during an aircraft's life due to fluctuating loads, are noticed at maximum tensile stress locations. Operation of an aircraft leads to generation and propagation of cracks, but this should not lead to catastrophic failure of the structure. For the prevention of this kind of failure, linear static stress analysis is performed.

Wings are the major components of an aircraft in which Spars carry most of the bending load due to the lift during flight. As wings behave as a cantilever structure, hence bending moment at the root is more than at the tip. In our project, linear static analysis of bulk head frames along with spar is performed for Robin DR 400 Dauphin and the fatigue damage estimation at the critical region due to fluctuating load is also carried out using constant amplitude S-N data for various stress ratios. In this model, rivets and cut out holes are the regions of critical stresses and hence they are analyzed using Finite Elements Method. Initially, global analysis is performed on the overall stresses in the wing assembly and the regions of maximum stresses are then analyzed locally to determine high stress magnitude and stress distribution at that region.

ROBIN DR 400 DAUPHIN (4-seater aircraft)

Fig. 1. ROBIN DR 400 DAUPHIN (4-seater aircraft)

Specifications[7]:

Wing Span:	8 m
Half Wing Span:	4 m
Chord Length:	0.7m
Velocity:	56.5 m/s
Density:	1.225 kg/m^3
Service Sealing:	3600m
Aerofoil Name:	"NACA 23014"
Wing Aspect Ratio:	5.6
Fuselage Maximum Width:	1.10 m
Maximum Take off Weight:	1004 kg

A. *Load Case*

Lift force is the main force acting on the wing which allows the aircraft to fly. Spars experience most of the bending. While taking off the aircraft, wing's root experience more force than tip and hence bending moment occurs more at the root of the Spar. One end of the attachment beam is fixed since it is connected to fuselage. The top and the bottom lug holes of fuselage and wing attachments are constrained with all six degrees of freedom. Force is applied in upward direction at the other end. At the time of take-off and landing, the displacement of the wings is assumed to be higher because of forces imposed by air in order to move in upward direction with greater speed.

II. MATERIAL PROPERTIES

In the present work, three types of materials are used for analysis

A. *Aluminium Alloy 2024-T3 [6]*

In the current project, both 'I' section and Spar are made of aluminium alloy because of its strength and light weight; and this material is also taken for the landing gear wells beam. 2024-T3 aluminium sheet is used in aircraft skin due to its excellent fatigue resistance and shiny finish. It is used in application requiring higher strength and lower weight. It can be welded only through friction welding and has minimum machinability. Aluminium alloy 2024-T3 has a density of 2.78 g/cc, electrical conductivity of 30% IACS, Young's modulus of 73 GPa or 7000 kg/mm^2, Poisson's ratio of 0.3. Its melting point begins at 500^0 C (932^0 F). 2024 aluminium alloys have composition of 3.8-4.9% copper. 0.3-0.9% manganese, 1.2-1.8% magnesium and less than a half per cent of zinc, silicon, chromium, nickel, bismuth and lead. It can elongate only for 10-15%.This type of aluminium alloy loses its strength at high temperature (200-250^0 C) and have good corrosion resistance property. Corrosion results in oxide layer on the skin that forms as a result of reactions with the atmosphere. This type of alloy can be formed into any shape by performing rolling, stamping, drawing, spinning, hammering, roll-forming and forging. Many operations can be done to this type of alloy such as boring, turning, milling etc. Aluminium alloy does not need protective coating as it is already shiny finished, but however often it is anodized to improve colour and strength of the aircraft.

B. *Alloy Steel AISI 4340 [6]*

Steel alloys are designed by AISI 4340. It composes different kinds of steels having composition exceeding the limitation of Si, Cr, Ni, Mo, Mn, C, B and Va. Steel alloy has high toughness and strength when heat treatment is performed. Low alloy steels contain chromium, molybdenum and nickel. In our project this steel is used in 'C' section, Lug member, fork member and pins. Their main uses are in the aircraft landing gear, power transmission gears and shafts and other important structural parts of aircraft. Its Young's modulus is 20000 kg/mm^2 or 200 GPa and Poission's ratio of 0.3.

C. *Titanium (Ti6Al 4V alloy) [6]*

This type of alloy is commonly used in aircraft industries. It has a mainly chemical composition of 6% aluminium, 4% vanadium, 0.2% oxygen, 0.25% iron. Titanium alloy are made stronger as compared to aluminium and steel because of their higher toughness, rigidity, corrosion resistance, better stiffness and thermal properties. When heat treatment is performed, it gives excellent combination of strength, welding and fabric ability. Titanium alloy can sustain heat up to 4000^0 C (7500 F), due to this temperature limit, it is used in aerospace industry, marine industry, offshore and power generation industries. This type of alloy is used to make blades, rings, discs, hand tools, airframes, fastener components, sports equipment, aircraft structural components etc. Titanium alloy has the Young's modulus of 110 GPa or 11000 kg/mm^2 and Poisson's ratio of 0.3.

III. FINITE ELEMENT MESH, LOAD AND BOUNDARY CONIDITION

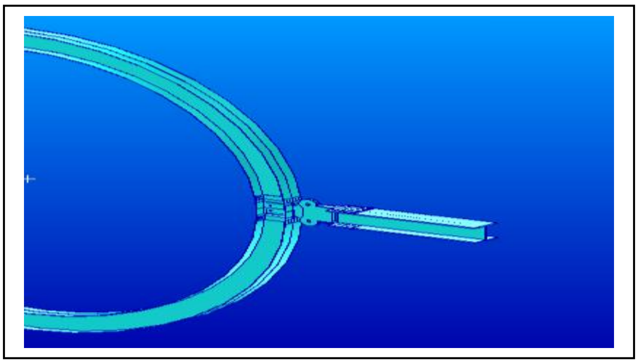


Fig. 2. Catia model imported into Patran

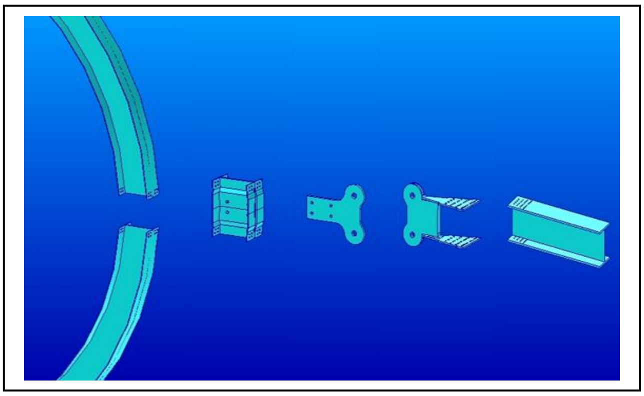


Fig. 3. Exploded view of aircraft wing structure

In this project, a 3D model of aircraft wing designed in Catia is imported into Nastran/Patran software and the user accordingly sets the required system of units to measure distance in mm. Then the 3D model is converted into 2D model using mid curve extraction. Then meshing is performed on each part using different operations such that each part should be in good aspect ratio. Various elements used for meshing are of quad or tri shape. After meshing equivalence, boundaries, duplicates and normal are checked for correctness. Then pins and rivets are created in different groups for ease of operation. After this material properties are applied to each group as given in table[1].

TABLE I. MATERIAL ASSIGNED FOR WING PARTS

Parts	Material
'I' section and Spar	Aluminium Alloy 2024-T3
'C' section, Lug, Fork and Pins	Steel Alloy AISI 4340
Rivets	Titanium (Ti6Al-4V Alloy)

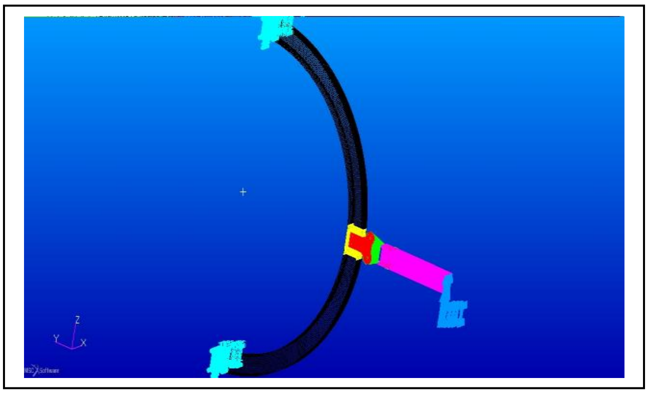


Fig. 4. Meshing, Loading and Boundary Conditions

After applying material properties, load and boundary conditions are applied. Here ends of the 'I' section are fixed and load, which is lift load of the aircraft, is applied on Spar as shown in the figure 4.3. To find lift load for the aircraft, coefficient of lift is determined with the help of XFLR5 V6 software in which the required aerofoil specifications of NACA-23014 is fed.

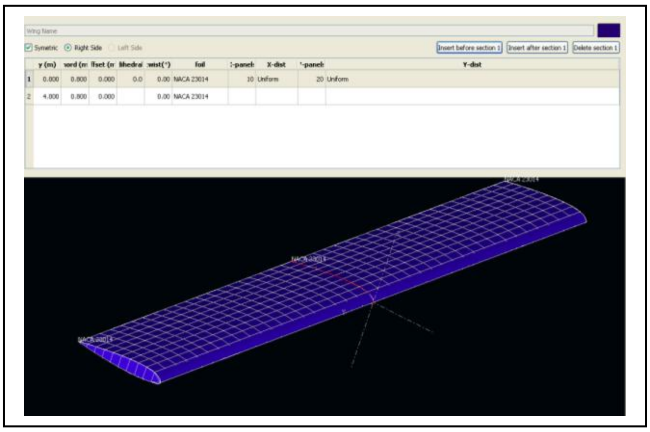


Fig. 5. Coefficient of Lift using XFLR5

From this we get the coefficient of lift for each span from which load for each span can be determined. Here we used 3g condition as a factor of safety for aircraft. To find the total lift load at each span of wing, we use the following formula

$$L = \frac{1}{2} C_l V^2 \rho S \tag{1}$$

Where, Cl = Coefficient of lift
V = Velocity
ρ = Density
S = Wing Area

These calculations give the loads on each span and the summation of all these loads gives total load on the aircraft wing. The Spar dimension is 1m and comparing it with the front and rear Spars of the Robin aircraft it is found that the given Spar is rear Spar of the Robin aircraft. In general, the front Spar and rear Spar are designed to take 55% and 45% of the lift load respectively. Now this total lift load is applied at the root of the Spar edges and then we move to the analysis process. Nastran is used as the solver and the results of this are imported into Patran to access the results in the form of von Mises Stresses in y-components at positions z1 and z2. Now maximum stresses among z1 and z2 are taken for local analysis considerations.

IV. LOCAL ANALYSIS

From the Global Analysis, we can show that the maximum tensile stress is in Lug and 'C' section. Therefore, to find the accurate stress location we are going with Local Analysis procedure, so that by taking only that part which is subjected to more critical stresses and again re-meshing it with finer mesh near the critical holes using modified quad elements, more accurate results are obtained. This is done by applying equivalent loads and boundary conditions to only Lug and 'C' section and repeating the same procedure as mentioned in the previous section to get the results. These results are further processed using analytical calculations to predict the fatigue life of the aircraft structure.

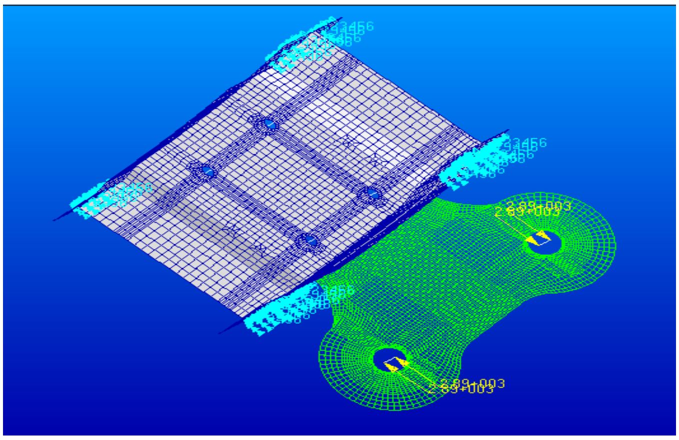


Fig. 6. Local Analysis Meshing and Boundary Conditions

V. FATIGUE DAMAGE ESTIMATION

As a body is subjected to cyclic loading, cracks are initiated mostly near the surface and at maximum stress regions. With the further application of load cycles these cracks propagate in the form of striations. The propagation continues and various cracks coalesce to grow in the whole body until fracture occurs.

In this project we are dealing with Crack Initiation, which is occurring at the maximum stress location. The maximum stress region is obtained by Global Analysis and more accurate magnitude of stress at that location is obtained by Local Analysis. Fatigue life of the aircraft structure is calculated using the maximum stress values from Local Analysis and S-N data for variable amplitude fatigue loading using Miner's rule.

Miner's rule

$$D = \sum \frac{n_i}{N_{fi}} = \frac{\text{Actual Cycle Count at Stress Level}}{\text{Cycle count at the time of failure under an axial loading}} \quad (2)$$

The maximum stress is obtained for both Lug and 'C' section at 2.5g condition.

The formulae used for the fatigue calculations are given below

σ_{max} = Maximum Stress in the Cycle

σ_{min} = Minimum Stress in the Cycle

σ_m = Mean Stress $= \frac{\sigma_{max}+\sigma_{min}}{2}$

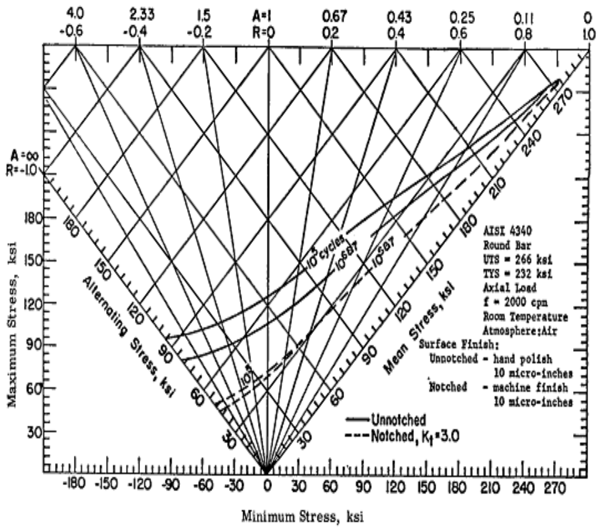
σ_a = Alturnating Stress Amplitude $= \frac{\sigma_{max}-\sigma_{min}}{2}$

$\Delta\sigma$ = Range of Stress = $\sigma_{max} - \sigma_{min}$ (3)

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

A = Amplitude ratio $= \frac{\sigma_a}{\sigma_m}$

Using above equations, for given stress amplitude, maximum stress and stress ratio, N_{fi} is obtained from Graph 1.



Graph 1. Typical constant-life fatigue diagram for heat-treated AISI 4340 alloy steel (bar), Ftu = 260 ksi [5]

Damage is then calculated using Nfi obtain from the Graph 1. using the following equation

$$\text{Damage, } D = \frac{Ni'}{Nfi} \quad (4)$$

VI. Result

A. Global Analysis

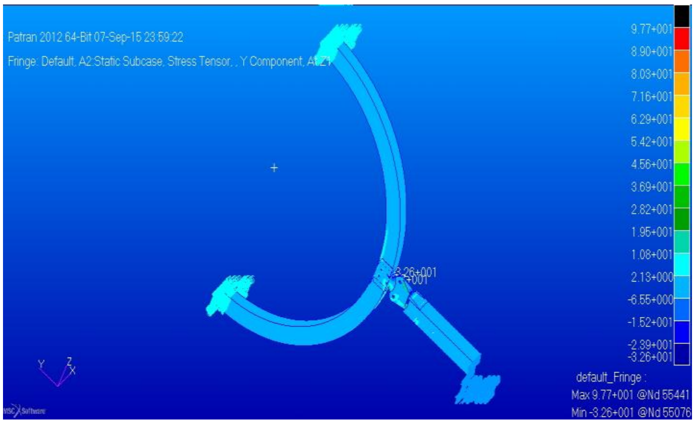


Fig. 7. MTS in y-direction for the whole assembly

The results in y-direction for z1 and z2 positions are given in the table below

TABLE II. MTS for various parts at Z1 and Z2 positions

Sl. N	Part	MTS in z_1 position kg/mm^2	MTS in z_2 position kg/mm^2
1	'I' section	8.04	12.2
2	'C' section	87.0	87.0
3	Lug	97.7	97.7
4	Fork	44.8	33.5
5	Spar	8.81	3.11

Global Analysis is done to obtain the maximum bending moment on the wing for 3g condition, which is found to be 976.0963729 kg-m.

B. Local Analysis

The MTS in local analysis for Lug and 'C' section are 122 kg/mm2 and 42.5 kg/mm^2 respectively.

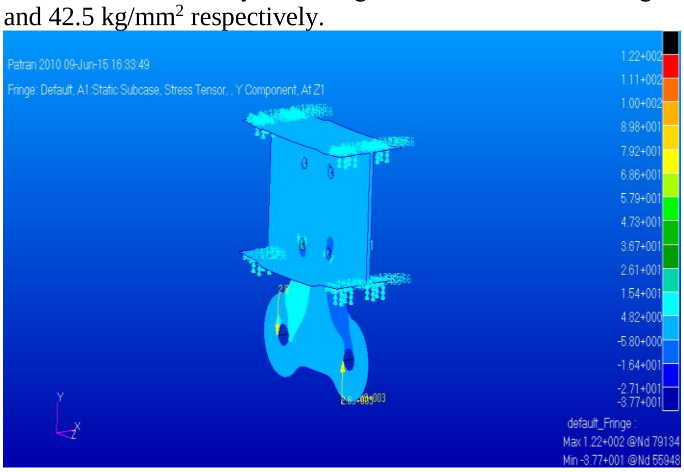


Fig. 8. MTS in Local Analysis of C-section

C. Fatigu Analysis

This data is provided by the designer which is collected from existing aircraft during the flight. Here 'g' is the ratio of lift of the aircraft to its weight. It is a measure of load that an aircraft experiences.

TABLE III. GIVEN FATIGUE DATA [7]

Sl. No.	RANGE OF "g"	Cycles(Ni)	Scatter factor	Cycles(Ni')
1	0.50g to 0.75g	40,000	5	200000
2	0.75g to 1.00g	55,000	5	275000
3	1.00g to 1.25g	38,000	5	190000
4	1.25g to 1.50g	25,000	5	125000
5	0 to 1.75g	500	5	2500
6	0 to 2g	300	5	1500
7	0 to 2.5g	250	5	1250

TABLE IV. FATIGUE CALCULATION FOR LUG

stress min (Ksi)	Stress max (Ksi)	Stress Amplitude	Stress Ratio	Nfi	D
28.6207308	42.9318	7.1555346	0.666655738	Infi	0
42.9318	57.2428692	7.1555346	0.749993853	infi	0
57.2428692	71.5525308	7.1548308	0.800011803	infi	0
71.5525308	85.8636	7.1555346	0.833327869	infi	0
0	100.1746692	50.0873346	0	infi	0
0	114.4843308	57.2421654	0	1000000	0.0015
0	143.1064692	71.5532346	0	10000	0.125
					0.1265

TABLE IV. FATIGUE CALCULATION FOR 'C' SECTION

Stress Minimum (Ksi)	Stress Maximum (Ksi)	Stress Amplitude	Stress Ratio	Nfi	D
9.970502346	14.95575352	2.492625587	0.666666667	Infi	0
14.95575352	19.94100469	2.492625585	0.75	Infi	0
19.94100469	24.92625587	2.49262559	0.8	Infi	0
24.92625587	29.91150704	2.492625585	0.833333333	Infi	0
0	34.89675821	17.44837911	0	Infi	0
0	39.88200938	19.94100469	0	Infi	0
0	49.85251173	24.92625587	0	Infi	0
					0

CONCLUSION

Global finite elements analysis is performed on Robin aircraft to locate the region of MTS in y-direction for Wing assembly. Total lift load is calculated for the Wing structure. This load is then applied to the finite elements model. After the analysis it is found that MTS is occurring at Lug member. Local analysis is then performed for this region to get accurate results. The results of Local analysis are then used for the determination of Fatigue Damage calculations to predict the Fatigue Life of the given aircraft.

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