Simulation of Wing Fuselage Attachment Using Fem

J. Precilla¹, Dr. P. Maniiarasan², A.T. Sam Rajan³

¹(Aeronautical Engineering, Nehru Institute of Engineering and Technology, Coimbatore, India, precilla1991@gmail.com)
²(Professor, Nehru Institute of Engineering and Technology, Coimbatore, India)
³(Mechanical Engineering, Sri Krishna College of Engineering and Technology, Coimbatore, India)

Abstract - In the last decades, it is notably true for the main structural elements such as wings, spars, fuselage bulkheads, sudden failure could lead to the catastrophic consequences. For the strict safety experiments are carried out to check the structural components before usage. In this ongoing project, it is to ensure that the static load carrying capability of the wing-fuselage attachment bracket. Rarely an aircraft will fail due to a static overload during service life. In this project, Stress analysis will be carried out for the geometry of the wing-fuselage attachment bracket. For the stress analysis calculation the finite element method is used. In the transport airframe an attempt will be made to foresee the fatigue life of wing-fuselage attachment bracket. Fatigue cracks will appear at the location of high tensile stress. Fatigue life calculation will be carried throughout the service loading condition using constant amplitude S-N data for various stress ratios and local stress.

Key words: Wing fuselage attachment bracket, Finite element method, bending moment.

I. INTRODUCTION

The important factor of designing an aircraft, it’s to finding the optimal proportion of the weight of the vehicle and payload. If a part in the aircraft fails, it does not necessarily result in failure of the whole structure. Fatigue is a phenomenon of cyclic loading on a structure; it depends on the magnitude and frequency of these loads in combination with the applied materials. Nf is defined as the number of stress cycles of a specified character that a specimen sustains before failure of a specified nature occurs; this phenomenon is called fatigue life. The Miner's rule states that there are k different stress magnitudes in a spectrum, Si (1 ≤ i ≤ k), each devote ni(Si) cycles, then if Ni(Si) is the number of cycles to failure of constant stress reversal Si, failure occurs,

\[ \sum N_i /N_f = C \]

where, C is assumed to be 1.

II. GEOMETRICAL CONFIGURATION

The wing fuselage attachment bracket considered for the study is shown in figure. It consists of a lug and a portion of the spar connected to several bolts. The lug consists of two pin holes with an integrated top flange and bottom flange which will be connected to the spar. The geometric dimensions of the lug attachment are shown in the figure. The three dimensional view of the lug is shown in the figure 2.1. The wing fuselage lug attachment geometry specification is shown below,

Geometry Specification

- Lug holes diameter = 40mm
- Lug thickness = 20mm

Spar I-section

- flange thickness = 8mm
- flange length = 480mm
- flange width = 150mm
- web thickness = 5mm
- web height = 234mm

III. MATERIAL SELECTION AND LUG DESIGN CONSIDERATION

The material considered for the spar and rivets of the structure is aluminum alloy-2024T351. The following properties,

1. Young’s Modulus, E = 7000N/mm²
2. Poisson’s Ratio, \( \mu = 0.3 \)
3. Ultimate Strength, \( \sigma_u = 485N/mm² \)

Design of wing fuselage lug attachment bracket:

The design of the wing fuselage lug attachment bracket is solved by using the strength of material based approach.

Strength of material based calculations is as follows: Basically there are three modes of failures in a lug joint.

Net-section failure:

Stress, \( \sigma_n = P/A \)

where, cross-sectional area=A= (W- dL) × tL

where, W= width of the lug,

\( d_L = \) lug hole diameter,

\( t_L = \) lug thickness

\( = (140-40) \times 20 = 2000 \text{ mm}^2 \)

So, \( \sigma_n = 1355/2000 = 0.6775 \text{ kg/mm}^2 \)

Bearing failure:

Bearing stress, \( \sigma_b = P/A_b \)
where, Bearing area = \( A_B = d_L \times t_L \)
where, \( d_L = \) lug hole diameter,
\( t_L = \) lug thickness
\( = 40 \times 20 = 800 \text{ mm}^2 \)
So, \( \sigma_b = \frac{1355}{800} = 1.69 \text{ kg/mm}^2 \)

Shearing Failure:
Shearing stress, \( \tau = \frac{P}{2A} \)
where, Sheared Lug area = \( A_S = L_L \times B_L \)
where, \( L_L = \) length of Sheared lug,
\( B_L = \) Breadth of the lug
\( = 66 \times 20 = 1320 \text{ mm}^2 \)
So, \( \sigma_b = \frac{1355}{2 \times 1320} = 0.51326 \text{ kg/mm}^2 \)

From the above calculations the stress value is found to be maximum in the case of bearing mode. As per the design consideration the maximum allowable stress in the lug is 40%. Therefore the above used configuration for the lug geometry is found to be satisfactory.

IV LOADS ON THE WINGS FUSELAGE LUG ATTACHMENT BRACKET

The maximum bending moment occurs at the root of the spar where wing and fuselage components will be attached to each other. The load experienced by the lug attachment is obtained as follows:

- Weight of the Aircraft structure = 1200 kg = 11772 N
- Load factor = 3.0 g
- Factor of safety = 1.5
- Total load = 1200*4.5 = 5,400 kg = 52974 N
- Lift load acting on the wing = 80% of total load = 5,400*0.8 = 4,320 kg = 423792 N
- The load acting on each wing = 4320/2 = 2160 kg = 211896 N
- Total number of spars used in the wing = 2
- The percentage of load sharing between front & rear spar Front spar 55%
  Rear spar 45%
- Therefore the load acting on the front spar = 2160*.55 = 1,188 kg = 1165428 N
- Span of the wing = 2300 mm
- Location of resultant load from the root of the wing = 850 mm
- Bending moment at the root = 1188*850 = 1009800 kg-mm
- Load to be applied at the end of the spar length considered for the analysis = 1009800/745 = 1355 kg = 13292.55 N

V FINITE ELEMENT ANALYSIS OF WING FUSELAGE LUG ATTACHMENT

Meshing is a discretization technique. To analyze the structural domain, the domain can be sub divided into smaller sub domain and each sub domain is called mesh. This process of dividing sub division is called meshing. For meshing the component Msc PATRAN is used and the generation and connectivity of the mesh over different part of a component is stated. Finite element model of the wing fuselage Lug attachment is as shown below figure. Meshing is implemented by using CQUAD4 and CTRIA3 elements. Triangular elements are used for the transformation between the finer mesh to coarser mesh. Fine meshing is accomplished in the locations where there is stress concentration. Coarser mesh is accomplished at rest of the regions in the structure.
A maximum stress of 33.8 kg/mm² is observed at the midpoint of the hole section. The maximum stress value obtained from the analysis is used as the input for the fatigue calculations.

VI. FATIGUE CALCULATION FOR WING FUSELAGE LUG ATTACHMENT

From the stress analysis of the lug joint, the maximum tensile stress location is identified. Specially the aircraft wing experiences variable spectrum loading during flight. An flight load is considered for the fatigue analysis of the lug joint. With the equation of the Miner’s rule the fatigue calculation is concluded. For the fatigue life calculation the variable spectrum loading is abridged as block loading. Each block consists of analogous load cycles abridged to 100 flights. Each block consists of 300 cycles. The aircraft which is considered for the ongoing work is designed for 30,000 flights. Damage calculation is accomplished for the complete service life of the aircraft.

Table 6: The Range of g, the Actual Number of Cycles in that Range and the Fatigue Cycle.

<table>
<thead>
<tr>
<th>Load Magnitude Range</th>
<th>Nf</th>
</tr>
</thead>
<tbody>
<tr>
<td>CYCLES</td>
<td>Nf</td>
</tr>
<tr>
<td>75,000</td>
<td>10⁴</td>
</tr>
<tr>
<td>65,000</td>
<td>10⁷</td>
</tr>
<tr>
<td>43,000</td>
<td>10⁷</td>
</tr>
<tr>
<td>27,500</td>
<td>10⁴</td>
</tr>
<tr>
<td>500</td>
<td>10⁴</td>
</tr>
<tr>
<td>300</td>
<td>10⁷</td>
</tr>
<tr>
<td>GAG (Ground Air Ground) cycle of 300 cycles</td>
<td>10⁷</td>
</tr>
</tbody>
</table>

The ‘g’ represents as acceleration due to gravity. The load corresponding to 1g is equivalent to the weight of the aircraft.

From Miner’s equation,

\[ \sum \frac{n_i}{N_i} = 1 \]  \hspace{1cm} (eq 6.1)

Where

- \( n_i \) = Applied number of cycles
- \( N_i \) = number of cycles to failure

Calculation of the damage accumulated using miner’s formula for all ranges from 0.5g to 4.5g

\[ \begin{align*}
D_1 &= \frac{n_1}{N_1} = 7.5 \times 10^{-4} \\
D_2 &= \frac{n_2}{N_2} = 6.5 \times 10^{-4} \\
D_3 &= \frac{n_3}{N_3} = 2.75 \times 10^{-4} \\
D_4 &= \frac{n_4}{N_4} = 2.75 \times 10^{-4} \\
D_5 &= \frac{n_5}{N_5} = 5 \times 10^{-5} \\
D_6 &= \frac{n_6}{N_6} = 3 \times 10^{-5} \\
D_7 &= \frac{n_7}{N_7} = 3 \times 10^{-5}
\end{align*} \]

From Miner’s rule…………..

Total damage accumulated for all load case is given by

\[ D_a = 2.116 \times 10^{-2} \]

The fatigue calculation the variable spectrum loading is simplified as block loading. Each block consists of load cycles corresponding to 100 flights.

Design service life = 30,000 flights
Total no of blocks = 100 blocks
Each block = 300 flights
So the total damage for 100 blocks = 2.116*10²*100 =0.2116

Total damage accumulated is 0.2116, which is less than 1. Therefore a crack will not get initiated from the location of maximum stress in the lug for the given load spectrum.

VII. FUTURE ENHANCEMENTS AND RESULT

Stress analysis of the wing fuselage lug attachment bracket is carried out and maximum tensile stress is identified as one of the lug-holes. From the stress analysis the maximum tensile stress observed in the lug is 33.8 kg/mm² and the maximum displacement 2.67 mm is observed at the free end of the spar beam. The fatigue life calculation is carried out for an estimation of life to crack initiation. From the fatigue life calculations the maximum damage fraction is 0.2116. The value of damage fraction is much less than 1. Hence the crack will not get initiated for the given load spectrum.
REFERENCES


