Fatigue Analysis and Design Optimization of Aircraft’s Central Fuselage

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Abstract. The centre fuselage of an aircraft plays a very crucial role as most of the important parts such as the front fuselage, aft fuselage and the wings are connected to it. So any load applied on these parts will be transferred to the centre fuselage. Hence it was essential to study the centre fuselage briefly in order to attain high design safety. Fatigue is the process of repeated cyclic loading of a component, which leads to an early failure of the same. The sources of fatigue in an aircraft are the parts connected to it, such as the wings, and the differential pressure between the inside of the fuselage and the outside atmosphere as a result of cabin pressurization. The high pressure inside the fuselage tries to expand the fuselage, whereas the stringers and bulk head prevent it from happening. This change in pressure happens frequently and this results in the fatigue of the centre fuselage. The vibratory loads acting on the other parts are transferred to the centre fuselage, which are also major contributors to the fatigue. The centre fuselage of an aircraft was designed using Pro-E software. The standard aluminium alloy was selected for the material. The various loads acting on the centre fuselage were studied and added to the centre fuselage, along with a differential pressure in a cyclic manner.

1. Introduction

Components of aircraft are inevitably subjected to fluctuating stresses and most of the components fail by fatigue fracture. Majority of components in an aircraft fail due to fatigue, and this fatigue failure accounts to almost 55% of the total failures [1]. The fuselage is a primary load carrying structure of the aircraft and it is directly or indirectly connected to most of the other important parts of the aircrafts. The fuselage also assists in transferring loads from one part to another. The centre fuselage is connected to the front fuselage, aft fuselage and also the wings directly. Hence, it is an important structure where the effect of fatigue is considerable. Therefore, analysing the centre fuselage for fatigue is a very important aspect of aircraft structural design.
The other important contributing factor for fatigue is the cabin pressurization. The frequent changes in the difference between the pressure inside the fuselage and the outer environment also induces fatigue loads, which actually tries to expand the fuselage skin as a result of high internal pressure [2]. This was however countered by the presence of stringers and bulkheads. Considering these two factors, the fatigue analysis conducted would provide a deeper insight into the areas of higher stress concentration, structurally weak segments, overall stress distribution and life cycle of the components, which are very essential parameters for material and design consideration.

Aleksandar Grbovic and Bosko Rasuo [3] carried out experiment to study the crack growth in spar under cyclic loading and compared their results with the theoretical results. Mitchell MR, Landgraf RW, et al. [4] studied the effect of the fatigue on structural components and they explored various techniques to analyze and calculate the fatigue stresses induced in the components. McKeighan PC, Ranganathan N, et al. [5] studied the fatigue testing and analysis under variable amplitude loading conditions. In that work they used different materials and applied cyclic loading to analyse the failures. Similar studies are found in Payne, A. O [6], Christian Boller and Matthias Buderath[7] and Harris, C. E, Newman, J. C, jr [8]. Remya Varghese and Manu Jayakumar [9] has studies the cyclic loading on aluminium fuselage by conducting the stress analysis using MSC NASTRAN. Venkatesha, B, K, Prashanth, K, P and Deepak Kumar, T [10] has done an investigation on the growth rate of fatigue crack using FEA approach.

2. Fatigue
Fatigue can be described as alternating loading or in other words cyclic loading which leads to cyclic stresses and strains developed in the material. Under continuous cyclic loading, at a critical stage the material ultimately fails.

The failure starts with a microscope crack which was hardly difficult to find. This crack initiates at a region where the stress concentration was predominantly high such as the surface fuselage, key holes and joints. Once it was initiated, the stress in the localized area become greater and it starts propagating rapidly. This continues until the stress reaches the limiting value after which the failure happens in most cases without any prior warning. The fatigue failure happens in three stages

Crack initiation→Crack propagation→Failure.

Hence the aircraft flying at a very great height and very high speed has to be designed carefully to avoid unpredictable accidents due to fatigue loads which happen in a fraction of a second.

3. Methodology

3.1 Geometry modelling
The solid model of the central fuselage which bears maximum of loads experienced by an aircraft using Pro-E-Wildfire 5.0 software. The dimension of Airbus A330 was taken for the design of the fuselage. The design consists of skin, stringers and bulkheads everything sketched separately and then assembled together as shown in Figures 1, 2, 3 and 4.
3.2 Meshing
The mesh was generated with appropriate element size and smoothening level to get maximum accuracy in the result. A view of meshed model is shown in Figure 5.

The mesh generated had nearly 1,00,000 cells and around 2,75,000 nodes. The right and left end of the bulkheads were taken as fixed supports and the forces were applied at the top and bottom of the fuselage in outward direction.
3.3 Calculation of force

Differential pressure of 3 psi, 4 psi, 5 psi, 6 psi and 7 psi were converted into forces and analysed in different cases. The corresponding pressure values in SI units are 20684.3 N/m², 27579 N/m², 34473.8 N/m², 41368.5 N/m² and 48263.3 N/m² respectively.

\[ \sigma_{\text{hoop}} = \frac{Pd}{2t} \tag{1} \]

Where, \( P \) was the load in N/m², \( d \) was the diameter, and \( t \) was the thickness of fuselage skin. Substituting \( P \) from above, \( d = 5.64 \text{m} \), and \( t = 0.005 \text{m} \) we find the hoop stress values. The hoop stress can be defined as,

\[ \sigma_{\text{hoop}} = \frac{F}{A} \tag{2} \]

Where, \( F \) was the force value which has to be entered for analysis, and \( A \) was the area. \( A = (\text{Length of fuselage} \times \text{Thickness}) = (15 \times 0.005) = 0.075 \text{m}^2 \). The \( F \) values are 874945.89 N, 1166591.7 N, 1458241.74 N, 1749887.55 N and 2041537 N for 3 psi, 4 psi, 5 psi, 6 psi, and 7 psi respectively.

4. Results and discussion

After all the pre-processing work being completed, the static analysis in ANSYS workbench for total deformation, equivalent stress and life cycles are done for all the five cases and two materials i.e., aluminium and structural steel and the results are compared. Two sample ANSYS result post processing visualization are shown in Figures 6 and 7.
Figure 6. Comparison of deformation, stress and life cycles of aluminium and structural steel for 3 psi pressure differential.

Figure 7. Comparison of deformation, stress and life cycles of aluminium and structural steel for 4 psi pressure differential.
It was observed that the patterns in which the stress was distributed and the deformation happened are the same for various differential pressures and also for different materials. However, the magnitude of the deformation, equivalent stress and no. of cycles are different in every case.

The magnitude of maximum stress obtained from the analysis has been verified with the numerical solution considering the no. of cycles as the parameter. The results obtained in both the cases were close in magnitude, hence confirming the correctness of the design and analysis. However, the variation in the maximum stress values between the analysis and mathematical equation can be attributed to the variation of constants in the equation and the inadequate meshing in the design process. The quality of meshing can also be found here by comparing the analytical solution with the software solution. The fatigue properties of aluminium alloy and Structural steel are summarised in Table 1 and Table 2.

| Pressure (psi) | Load (N)   | Max. Deformation (mm) | Max. Equivalent Stress (MPa) | Min. Life cycle |
|---------------|------------|-----------------------|-----------------------------|-----------------|
| 3             | 874946     | 14.0                  | 104.7                       | 990000          |
| 4             | 1166592    | 18.7                  | 139.6                       | 733000          |
| 5             | 1458242    | 23.4                  | 174.5                       | 128000          |
| 6             | 1749888    | 28.1                  | 209.5                       | 29347           |
| 7             | 2041538    | 32.8                  | 244.4                       | 4544            |

| Pressure (psi) | Load (N)   | Max. Deformation (mm) | Max. Equivalent Stress (MPa) | Min. Life cycle |
|---------------|------------|-----------------------|-----------------------------|-----------------|
| 3             | 874946     | 5.0                   | 104.0                       | 358000          |
| 4             | 1166592    | 6.6                   | 138.6                       | 98747           |
| 5             | 1458242    | 8.3                   | 173.2                       | 47408           |
| 6             | 1749888    | 9.9                   | 207.9                       | 22761           |
| 7             | 2041538    | 11.6                  | 242.5                       | 13244           |

A similar procedure follows for the 5, 6 and 7 Psi loading conditions and results are presented in the Table 1 and Table 2. It was inferred that as the load applied increases, the maximum deformation and maximum equivalent stress increase, and the life cycle decrease. Such behaviour was in agreement with the S-N curve which also states the same.

4.1 Validation of solution
The solution obtained from ANSYS workbench solution was compared with the analytical solution as obtained from the below mentioned Basquin equation.

$$\sigma_s = \sigma_f^* (2N_f)^b$$  (3)
Where $\sigma_f$ was the fracture strength, $N_f$ was the no. of cycles to failure, and $b$ was Basquin’s exponent (-0.05 to -0.12). The value of $\sigma_f$ can be obtained from the material properties and $N_f$ can be obtained from the analysis results for each case. By substituting these two values in the Basquin equation, we get the maximum stress value $\sigma_a$. This value can be verified with the maximum stress value as obtained from the ANSYS analysis. Table 3 and Table 4 presents comparison between the analytical and ANSYS solutions.

| Load (psi) | Min. Life Cycle | Max. Stress (Analytical) (MPa) | Max. Stress (ANSYS) (MPa) |
|-----------|-----------------|--------------------------------|---------------------------|
| 7         | 4545            | 242.4                          | 244.4                     |
| 6         | 29347           | 206.9                          | 209.5                     |
| 5         | 128000          | 182.5                          | 174.5                     |
| 4         | 733000          | 157.4                          | 139.6                     |
| 3         | 990000          | 126.1                          | 104.7                     |

Table 3 Comparison of Fatigue analysis solution for Aluminium material.

| Load (psi) | Min. Life Cycle | Max. Stress (Analytical) (MPa) | Max. Stress (ANSYS) (MPa) |
|-----------|-----------------|--------------------------------|---------------------------|
| 7         | 13244           | 235.6                          | 242.5                     |
| 6         | 22761           | 220.8                          | 207.9                     |
| 5         | 47408           | 202.2                          | 173.2                     |
| 4         | 98747           | 185.2                          | 138.6                     |
| 3         | 358000          | 158.7                          | 103.9                     |

Table 4 Comparison of Fatigue analysis solution for Structural steel material.

5. Conclusion
This project showed the various fatigue prone regions of a centre fuselage, thereby identifying the areas of concern to increase material toughness and strength for avoiding fatigue failure. It was found that Structural steel has better fatigue properties when compared to Aluminium alloy. Structural steel has a severe weight penalty when compared to Aluminium alloy, therefore, its usage to specific parts which demand high strength will also imply penalty on overall aircraft weight. This work indicates that aluminium material has more advantageous strength to weight ratio. Hence, Aluminium material can be preferred for air-craft fuselage applications over steel in design of parts with high fatigue strength.

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