Design Of An Aircraft Wing Structure For Static Analysis And Fatigue Life Prediction

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Abstract

Wing structure consists of skin, ribs and spar sections. The spar carries flight loads and the weight of the wings while on the ground. Other structural and forming members such as ribs are attached to the spars, with stressed skin. The wings are the most important lift-producing part of the aircraft. The design of wings may vary according to the type of aircraft and its purpose. Experimental testing of wing structure is more expensive and time consuming process. In this project detailed design of trainer aircraft wing structure made by using CATIA V5 R20. Then stress analysis of the wing structure is carried out to compute the stresses at wing structure. The stresses are estimated by using the finite element approach with the help of ANSYS-12 to find out the safety factor of the structure. In a structure like airframe, a fatigue crack may appear at the location of high tensile stress. Life prediction requires a model for fatigue damage accumulation, constant amplitude S-N (stress life) data for various stress ratios and local stress history at the stress concentration. The response of the wing structure will be evaluated. In this study prediction of fatigue life for crack initiation will be carried out at maximum stress location.

Keywords- *Finite element analysis, Wing structure, Fatigue, Stress analysis, Life Prediction, Stress life.*

1. Introduction

In an aircraft wing structure ribs and spars are provided to support and give rigidity to the wing section. Although the major focus of structural design in the early development of aircraft was on strength, now structural designers also deal with fail-safety, fatigue, corrosion, maintenance and inspectability, and producability. Modern aircraft structures are designed using a semi-monocoque concept. A basic load-carrying shell reinforced by frames and longerons in the bodies, and a skin-stringer construction supported by spars and ribs in the surfaces. Proper stress levels, a very complex problem in highly redundant structures, are calculated using versatile computer matrix methods to solve for detailed internal loads. Modern finite element models of aircraft components include tens-of-thousands of degrees-of-freedom and are used to determine the required skin thicknesses to avoid excessive stress levels, deflections, strains, or buckling. The goals of detailed design are to reduce or eliminate stress concentrations, residual stresses, fretting corrosion, hidden undetectable cracks, or single failure causing component failure.

Fail-safe design is achieved through material selection, proper stress levels, and multiple load path structural arrangements which maintain high strength in the presence of a crack or damage. Analyses introduce cyclic loads from ground-air-ground cycle and from power spectral density descriptions of continuous turbulence. Component fatigue test results are fed into the program and the cumulative fatigue damage is calculated. Stress levels are adjusted to achieve required structural fatigue design life. Aircraft in flight experience concentrated shear stresses on their wings. Without adequate support, the wings would eventually fold up against the side of the plane. The spar carries flight loads and the weight of the wings while on the ground. Other structural and forming members such as ribs may be attached to the spar or spars, with stressed skin construction also sharing the loads where it is used.

Fatigue is a phenomenon associated with variable loading or more precisely to cyclic stressing or straining of a material. Just as we human beings get fatigue when a specific task is repeatedly performed, in a similar manner metallic components subjected to variable loading get fatigue, which leads to their premature failure under specific conditions. Fatigue cracks are most frequently initiated at sections in a structural member where changes in geometry, e.g., holes, notches or sudden changes in section, cause stress concentration.

2. Problem definition

In this study trainer aircraft wing structure with skin, spars and ribs is considered for the detailed analysis. The wing structure consists of 15 ribs and two spars with skin. Front spar having ‘I’ section and rear spar having ‘C’ section. Stress analysis of the whole wing section is carried out to compute the stresses at spars and ribs due to the applied pressure load.

The main objectives are:

- Global and local stress analysis of an aircraft wing structure to compute the stresses at spars and ribs due to Pressure force over the wing section with the help of ANSYS Mechanical-APDL

- Fatigue life prediction for crack initiation at spars and ribs region by Miner's Rule.

AA 2024-T351 is used in current wing structure due to high strength and fatigue resistance properties. The ultimate tensile strength of this material is 427 Mpa and yield strength is 324 Mpa.

3. Geometrical configuration

Wing structure modelled in CATIA-v5-R20 was been shown in figure 1. It consists of different components. The wing structure used here is having 15 in transverse direction and two spars in longitudinal direction.

3.1 Input parameters for design

Root chord	:	2400 mm
Tip chord	:	700 mm
Semi Span length	:	5500 mm
Exposed Span	:	4750 mm
Airfoil (root)	:	NACA 64A215
(Tip)	:	NACA 64A210
Aircraft weight	:	14000 N
Lift Load	:	6g
Design Factor	:	1.5
Given Spar Position (in % of chord length)		
Front Spar	:	18-25
Rear Spar	:	62-70

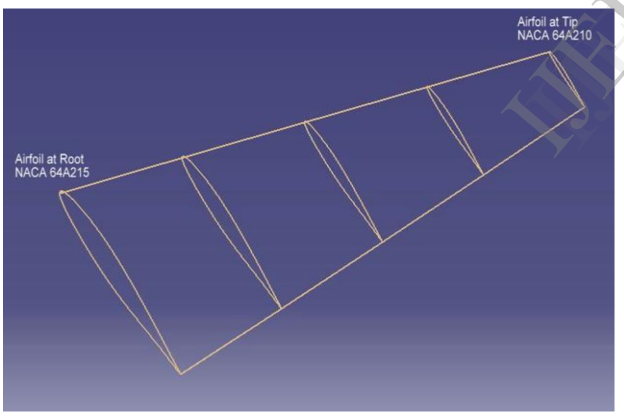


Figure 1 Airfoil

Front made up of I section and rear spar made up of C section. Skin section will cover these inner components.

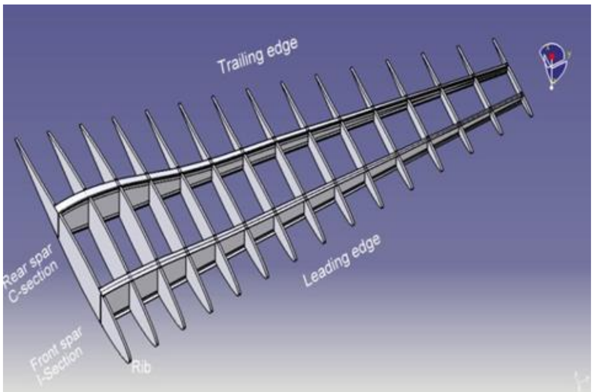


Figure 2 wing structure design using CATIA-v5 R20

Each part is modelled in CATIA and assembled to form wing structure

3.2 Loads acting over the wing structure

Lift load is considered as important criteria while designing an aircraft. Fuselage and wings are the two main regions where lift load acting in an aircraft. Here 80% of the lift load is acted on the wings (i.e., maximum lift load is acted on the wings) and remaining 20% in acted on the fuselage. Therefore in wings the maximum load is acted nearer to the wing roots.

Load calculation for the wing structure
Weight of the aircraft: 14000N
Design load factor: 6 "g"
Factor of safety: 1.5
Therefore, Total design load on the aircraft will be: 126000 N
As we mentioned earlier, total lift load on the aircraft is distributed as 80% and 20% on wing and fuselage respectively,
Hence total load acting on the wing = 100800 N
Therefore total load acting on the each wing = 50400 N
But we know the resultant load is acting at the distance 2138 mm (45% of from the wing root).
Bending stress at root section =89035 N
Bending moment at the root of the wing can be calculated as 189.64.07*10^6 Nmm

4. Finite Element Analysis

In this project ANSYS APDL (Advanced Parametric Design Language) software is used as the pre-processor and postprocessor. The pre-processing task includes building the geometric model by importing it from CATIA solid model of wing structure and extracting geometry, building the finite element model, giving these elements the correct material properties, setting the boundary conditions and loading conditions and finally, assembling these elements into a connected structure for analysis. Analysis is done in ANSYS solver phase. The analysis stage simply solves for the unknown degrees of freedom, as well as reactions and stresses. In the post processing stage, the results are evaluated and displayed. The accuracy of these results is postulated during this post processing task. The ANSYS APDL software together performs all 3 of the principle tasks of a finite element analysis.

4.1 Analysis of wing structure

The geometric model of the wing structure done in CATIA V5 software package is imported to ANSYS APDL for pre-processing. Each part is extracted by its points and lines to get geometrical accuracy to the model. In our wing structure we do have fifteen ribs including two spars. Each spar consists of two flanges, two sides and a web. This has to be meshed separately creating different groups by using ANSYS-12 shown in figure 3

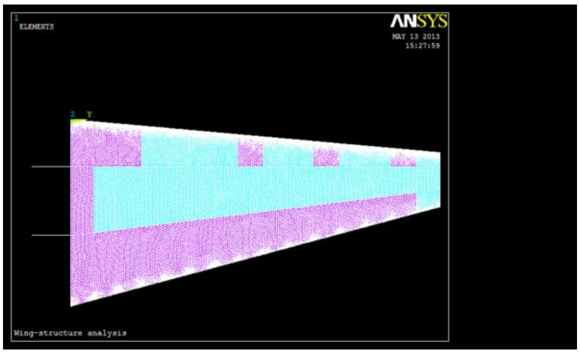


Figure 3 Meshed wing section

Special care is to be taken for meshing the region in the web. Finite element properties are provided to two spars used here are I and C-section structures. While meshing top skin as well as bottom skin of the wing structure it is taken care that mesh seeds are provided for the all the positions for the later simplicity. All the elemental and material properties (Aluminium alloy AA 2024-T351) are provided for analysis. Figure 3 shows the whole finite element mesh generation.

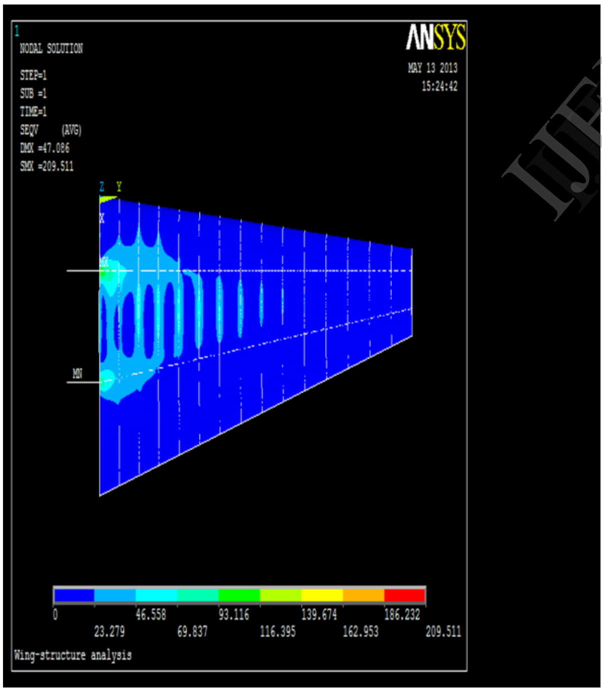


Figure 4 Wing analysis using Ansys-12

The stress distribution for the given loads has been observed and that reveals the stress is distributed uniformly but maximum stresses are developed nearer to root of wing section which is shown in figure 4. The magnitude of maximum principal stress developed here is 209.511 N/mm2

The structure is safe because the stress magnitude which was obtained from the analysis is less than the yield strength of the structural material.

Factor of safety= Yield Strength /Normal Working Load Factor of safety of wing structure is 1.54, which greater than the design factor of wing. Once wing structure is safe from linear static analysis, next step is the fatigue life prediction of the wing structure.

5. Fatigue Life Calculations

Normally aircraft wing experiences variable spectrum loading during the flight. A typical transport aircraft flight load spectrum is considered for the fatigue analysis of the wing structure. Calculation of fatigue life is carried out by using Miner‘s Rule. For the fatigue calculation the variable spectrum loading is simplified as block loading. Each block consists of load cycles corresponding to 100 flights. Damage calculation is carried out for the complete service life of the aircraft. The load factor "g" is defined as the ratio of the lift of an aircraft to its weight and represents a global measure of the load to which the structure of the aircraft is subjected. As we know the maximum stress value obtained from the analysis is corresponding to 6 g condition. Therefore the stress value corresponding to 1 g condition is obtained as 35.07 N/mm2 Correction factors for fatigue life calculations of wing structure is considered. Hence maximum stress with correction factor is calculated (Jaap, 2004).

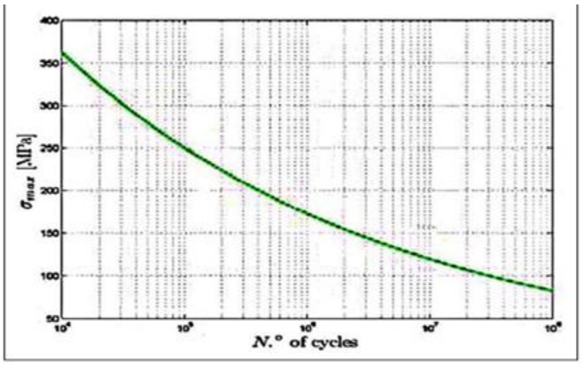
- Surface roughness correction factor = 0.8
- Type of loading = 1
- Correction factor for reliability in design =0.897

Maximum stresses with correction factor for all the other conditions are calculated and shown in Table 1.

g' Condition	**Maximum stress N/mm^2**	**Maximum stress with correction factor N/mm^2**
1	35.07	48.87
2	70.14	97.74
3	105.21	146.61
4	155.33	216.45
5	175.35	244.35
6	210.42	293.22

Table 1: Stress Values at Various "g" Conditions with Correction Factors

When the alternating or maximum stress is plotted versus the number of cycles to failure (fatigue life) for a given material, the curve is known as S-N curve (Michael, 1988). Using the maximum stresses value at different g conditions, corresponding number of cycles to failure is obtained from S-N curve of Aluminium 2024 T351 as shown in Figure 9 (Serrano et al., 2010).
Figure 4 Typical S-N diagram for fatigue behavior of Aluminium alloy 2024 T351



The simplest and most practical technique for predicting fatigue performance is the Palmgren-Miner hypothesis. The hypothesis contends that fatigue damage incurred at a given stress level is proportional to the number of cycles applied at that stress level divided by the total number of cycles required to cause failure at the same level. If the repeated loads are continued at the same level unit failure occurs, the cycles ratio will be equal to one.

From Miner's equation (Jadav et al., 2012), $\Sigma ni/Nf = C$

Where, ni= Applied number of cycles

Nf = number of cycles to failure

Table 2 shows damage D, accumulated on each range of load condition.

Range of "g"	Average "g"-values	Applied No. of cycles Ni	No. of cycles to failure N_f	Damage accumulated (n_i/ N_f)
1 "g" to 2"g"	48.089	48000	$1*10^8$	$4.8*10^{-4}$
2 "g" to 3"g"	80.15	33000	$8*10^6$	$4.1*10^{-3}$
3 "g" to 4"g"	112.214	26000	$9*10^5$	0.028
4 "g" to 5"g"	144.272	20000	$2*10^5$	0.1
5 "g" to 6"g"	176.333	10	$7*10^4$	$1.428*10^{-4}$

Table 2: Damage Accumulated from Miner's Formula

Total damage accumulated for all load case is given by

$Da= D1+ D2+ D3+ D4+ D5+ D6+D7= 0.327$

Total damage accumulated is 0.327, which is less than 1. Therefore a crack will not get initiated from the location of maximum stress in the wing structure for given load spectrum. Hence total damage is 0.327 for 1 block of loading or for 100 flights. One flight is considered 10 flying hours which eventually means 100 flights as 1000 flying hours. For damage to become critical (D= 1), the number of blocks required is 3.058 blocks or 3058 hours. Hence it is advised to meet the wing structure components maintenance atleast by this required time.

6. Conclusion

Stress analysis of the wing structure is carried out and maximum stress is identified at wing root which is found out to be lower than yield strength of the material. Normally the fatigue crack initiates in a structure where there is maximum tensile stress is located. The fatigue calculation is carried out for the prediction of the structural life of wing structure. Since the damage accumulated is less than the critical damage in the wing structure is safe from fatigue considerations. Life of the particular region in wing structure is predicted to become critical and found out to be 3058 flying hours or 3.058 blocks, hence advised to conduct the maintenance without fail during this period. Fatigue crack growth analysis can be carried out in the other parts of the wing structure. In the future work damage tolerance evaluation and structural testing of the wing structure can be carried out for the complete validation of all theoretical calculations. As well as wing structure optimization can also be carried out to meet the appropriate factor of safety of wing section.

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