

Fatigue Damage Estimation for a Fuselage Structure with Multiple Cut-outs under Cyclic Loading

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ABSTRACT: Discontinuities are common in airframe structure. These lead to the emergence of stress concentration regions. This turns deliberately pronounced when airframe structure experiences fluctuating loads during flight. These elevated tensile stress locations undergoing fluctuating loads may lead to fatigue cracks. In case of situations where these cracks are unnoticed they could lead to catastrophic failure of the structure. So it is highly pertinent to involve in the stress analysis of a segment of the fuselage with multiple cut-outs and fatigue damage calculation due to fluctuating pressurization loads. The fatigue damage calculation requires constant amplitude S-N data for various stress ratios and local stress history at the stress concentration. Fatigue life prediction methodologies through damage calculation are empirical in nature and therefore require test validation. Here CATIA software is used for the modelling, MSC PATRAN is used for meshing the fuselage panel and stress analysis is carried out using MSC NASTRAN.

KEYWORDS: Fatigue, fuselage panel, stress concentration, stress analysis.

I. INTRODUCTION

The aerospace industry, since it attained completeness as a separate industry is always subjected to numerous challenges. These unidentified problems which emerged long back turned more pronounced when much attention is laid upon the performance and design criteria. Several researches and experimentations are going on to tackle down and reduce the impact of these undesirable phenomena on aircrafts and other flying vehicles. It is always a matter of interest to revert attention to one of these undesirable damage or failure inducers. Fatigue is a usual and frequent mode of failure that occurs in a component subjected to alternate and repeating loads. In general, it is the weakening of a material caused by repeatedly applied loads. Also it is the progressive and localized structural damage that occurs when a material is subjected to cyclic loading. The nominal maximum stress values that cause such damage may be much less than the strength of the material typically quoted as much less than the strength of the material typically quoted as the ultimate tensile stress limit, or the yield stress limit. Fatigue occurs when a material is subjected to repeated loading and unloading. If the loads are above a certain threshold, microscopic cracks will begin to form and leads to failure [1]. This undesirable event occurs based on the stress involved and the number of load cycles [3]. The fatigue varies on its nature as high cycle, low cycle and very high cycle based on this [5]. A drastic change can be noticed in the failure life when any stress concentration factors are present in the structure [2],[9]. The elevated stress in these regions can be found using suitable finite element techniques and related softwares [4],[17].

The aircraft fuselage is composed of stressed skin, circumferential bulkheads and longitudinal stringers. The skin is connected to bulkheads and stringers mostly by rivets. Fuselage is associated with a considerable number of riveted joints and is subjected to differential internal pressurization. When the fuselage is frequently pressurized and depressurized during each takeoff and landing cycle of aircraft, the metal skin of fuselage expands and contracts resulting in metal fatigue [6],[11],[18]. Fatigue damage accumulates during every cycle of loading in the airframe structure during its operation. The requirement for weight reduction in airframe has led to increase in the use of

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aluminium castings [10]. It is primarily due to the high strength-to-weight ratio associated. When the weight considerations are being fulfilled by these light metals, on the other hand it fails under repeated pressurization loads [8]. This fatigue cracks developed thereby is mainly due to intense hoop stress. Cabin pressure results in radial growth of the skin and this radial growth is resisted by frames and stringers giving local bending along the fastener lines. Fuselage skin panels are curved and these panels are under biaxial tension loading due to cabin pressure. The service life of components is estimated based upon the damage estimation occurring on it with respect to the flying environment [7],[12].

II. GEOMETRIC CONFIGURATION

The geometric modelling of the fuselage with multiple cut-outs indicating bulkheads, stiffeners and rivets are carried out by using Catia. The cut-outs mentioned here refers to the openings for door and windows. The bulkheads serve to maintain the required external contour at the region and they provide rigidity and strength to the structure. For purposes of stabilizing the skin there are a multiplicity of stiffeners arranged orthogonally relative to one another. The stiffeners form a multiplicity of intersection regions, which in particular when using a differential form of construction represent weak points. Furthermore, as a result of their orthogonal arrangement relative to one another the stiffeners are not optimally orientated in terms of loading.. The Catia model representing the respective region is shown in figure 1. The view associated with all mentioned components are presented clearly with dimensions indicated. The tilt and front views are also given separately in figure.1.

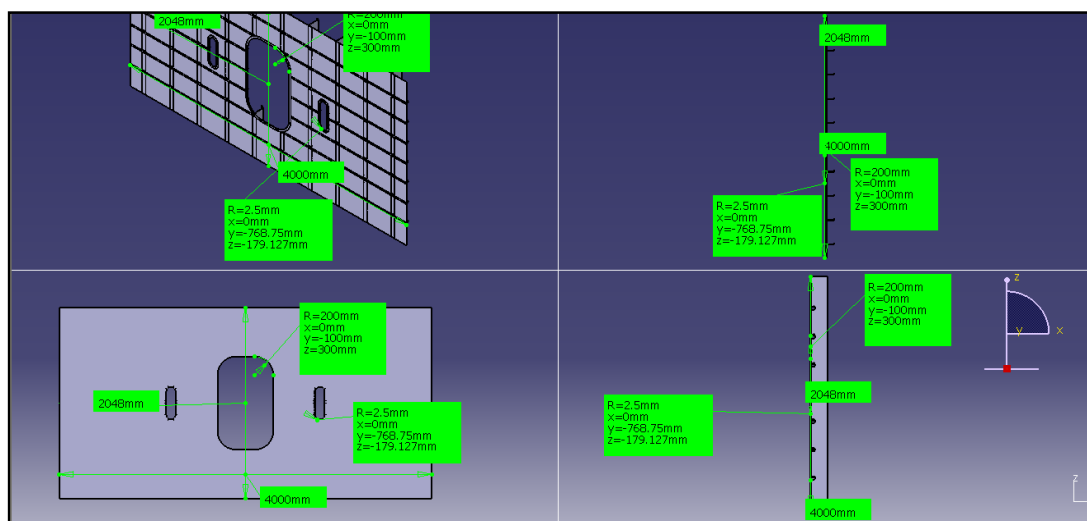


Figure 1 :CAD model of fuselage panel with multiple cut outs

III. MATERIAL SPECIFICATION

It should be really meticulous while selecting a material for a specific part of an aircraft. The selected material should mandatorily follow all the design guidelines and comply with the pertinent performance parameters. Weight and cost considerations coupled with the following features paved Aluminium 2024-T3 as the suitable material.

- Density
- Young's modulus
- Fatigue strength
- Ultimate and Yield strength
- Corrosion

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The material properties with the corresponding magnitudes are indicated in Table.1. The Aluminium with low density is selected as it is always a matter of choice for the designer due to its light weight and good tensile and yield strength. The material constants and Poisson’s ratio with comparatively moderate values enable the analysis of 2024-T3 alloy easier when comparing to the other alloys of aluminium.

Table. 1: Material properties of Aluminium 2024-T3

Property	Aluminium 2024-T3
Density	2.77 g/cm ³
Ultimate Tensile Strength	483Mpa
Tensile Yield Strength	362Mpa
Young’s Modulus	72Gpa
Poisson’s Ratio	0.33
Material Constant, C	5 x 10 ⁻¹¹
Material Constant, n	3

IV. MATHEMATICAL FORMULATION AND LOAD CALCULATION

The formulation involves representation of loads acting on the fuselage panel. This incorporates skin, bulkheads and stiffeners. When the panel associated with all these members subjected to pressurization at high altitudes, certain stresses act upon it. The stresses are highly pronounced over here when a differential pressure prevails. When altitude increases the need for pressurization turns more pertinent and this in turn makes the differential pressure between inside of cabin and outside more noticeable. In this type of conditions, hoop stress which acts tangential to the structure plays a vital role and is considered for calculation. Here from the International Standard Atmosphere [21], the pressure loads acting are obtained. Here calculation is presented for a differential pressure of 9 psi and for the other load cases the summary is tabulated in table.2

Diameter of given cylindrical fuselage = 2690mm

Thickness of skin = 2mm

Area of skin, $A_{skin} = 8000 \text{ mm}^2$

Area of bulkhead, $A_{bh} = 452 \text{ mm}^2$

Differential Pressure 9 psi = 0.0063kg/mm²

The hoop stress is given as, $\sigma_{hoop} = (P*d) / 2*t = 4.23675 \text{ kg/mm}^2$

This stress is being acted upon the skin and bulkhead. In order to represent the load acting on skin and bulkhead separately, the following approach is used.

Skin : Stress= Load/Area

$$\sigma_{hoop} = 4.23675 \text{ kg/mm}^2, A_{skin}=8000 \text{ mm}^2,$$

then Load, P = 33894 kg

Bulkhead : Stress= Load/Area

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$$\sigma_{hoop} = 4.23675 \text{ kg/mm}^2, A_{bh}=452 \text{ mm}^2,$$

then Load, P = 1915.011 kg

Total Load acting on 10 bulkheads = 1915.011 *10= 19150 kg

Table. 2 : Loads on skin and bulkhead for various differential pressures

Pressure (psi)	Pressure (kg/mm ²)	Hoop stress, $\sigma_{hoop} = (P*d) / 2*t$ (kg/mm ²)	Load acting on the Skin, $P = \sigma_{hoop} * A$ (kg)	Load acting on 10 bulkheads $P = \sigma_{hoop} * A$ (kg)
6	0.0042	2.8245	22596	12766.74
6.5	0.00455	3.0598	24479	13830.64
7	0.0049	3.295	26362	14894.53
7.5	0.00525	3.5306	28245	15958.43
8	0.0056	3.766	30128	17022.32
8.5	0.00597	4	32000	18080
9	0.0063	4.236	33894	19150.11

The Table.2 provides a clear illustration of variation of loads acting on the components, namely skin and bulkheads for different pressure conditions. It is found that the hoop stress increases based on rise of pressure. This points out to the influence of pressure on hoop stress. The loads on skin and bulkheads increases with the rise in hoop stress.

V. FINITE ELEMENT MODEL OF A FUSELAGE PANEL WITH MULTIPLE CUT-OUTS

The fuselage panel opening cut-out model is first prepared in the Catiamodeling software and then extracted into the software where finite element meshing and analysis is carried out.

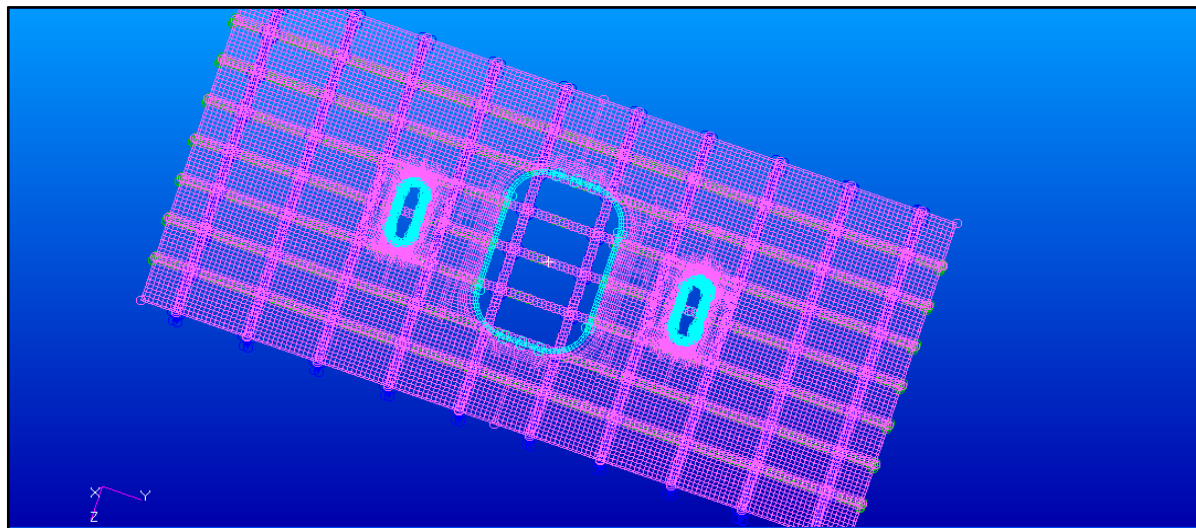


Figure 2 : Finite element meshing of fuselage panel with multiple cut-outs

The software used for analysis here is MSC NASTRAN. Finite element meshing is carried out for all the components of the panel. QUAD 4 and TRIA 3 elements are used for the purpose of meshing the structure. The representation is done with clear differentiation between cut-outs for door and windows.

Table 3 :Finite Element model summaries of the stiffened panel

Description	Type of elements	Number of elements	Aspect Ratio < 5
Skin	QUAD4, TRIA3	22718	4.23
Bulkhead	QUAD4, TRIA3	60387	4.9
Stiffener	QUAD4, TRIA3	4054	2.7
Rivet	BAR	1610	--
Cut-outs	QUAD4	31023	2.7

VI. LOADS AND BOUNDARY CONDITIONS

The aircraft fuselage cabin which is subjected to pressurization and due to the differential pressure prevailing between the inner space and outer region, certain loads and act upon the fuselage. The main and pertinent one is the hoop stress acting in the tangential direction. To facilitate the action of this load, suitable boundary conditions are to be given. Here it is done by fixing both the end of the fuselage panel. The loads and boundary conditions applied to the finite element model of the fuselage panel with cut-outs are shown below. The loads are given from both the sides of the stiffened panel. As hoop stress is considered, circumferential application of load is preferred.

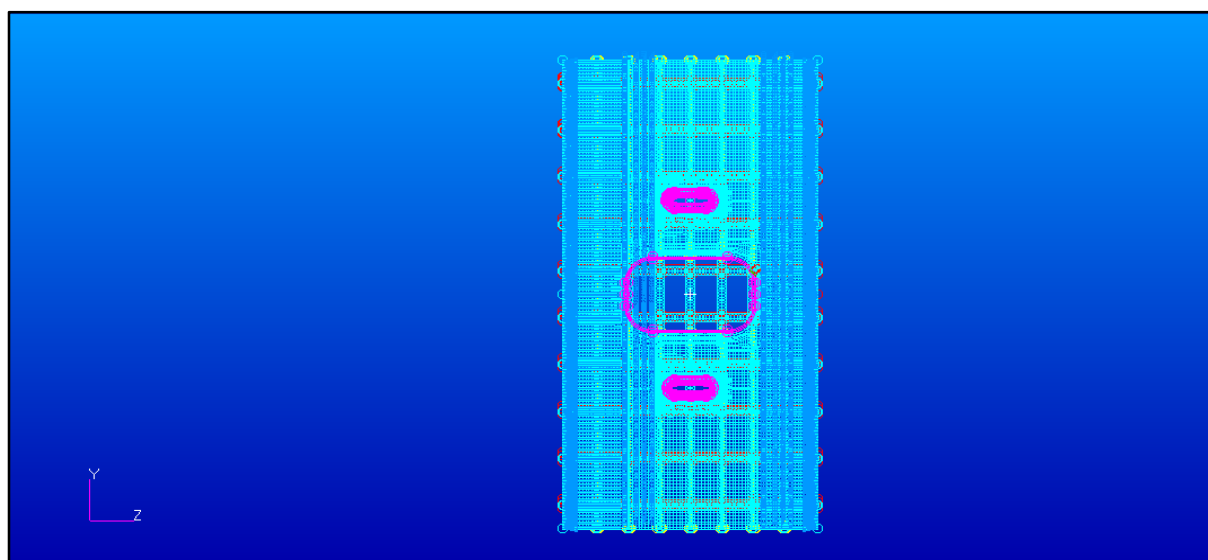


Figure 3: Loads and Boundary conditions applied to the stiffened panel

VII. STRESS ANALYSIS AND DISPLACEMENTS

Once the loads and boundary conditions are meticulously imparted on the model, it is subjected to stress analysis. The main objectives are to find out the magnitude of maximum stress and the stress concentration regions. Here loads are given from 6psi to 9 psi and the maximum magnitude obtained is 24 kg/mm² corresponding to 9 psi. Also the location of stress concentration region remained same and it is found around the bulkhead cut-out region. Here the stress analysis results corresponding to the maximum load of 9 psi is shown. When the stress concentrated region is examined specifically, the stress magnitude will be 24.236 kg/mm².

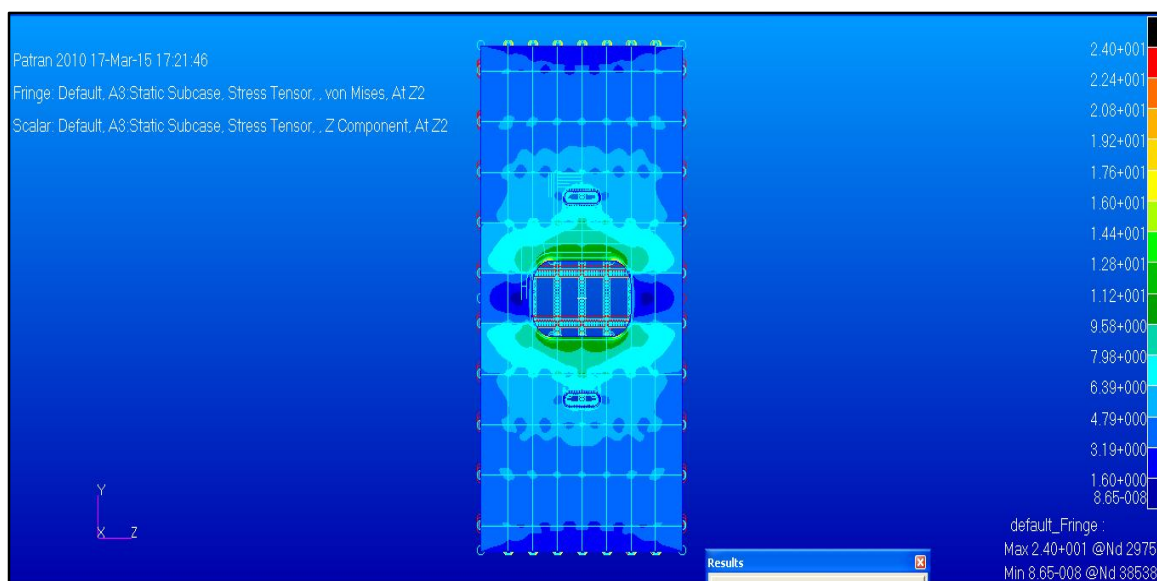


Figure 4: Stress analysis of the stiffened panel with multiple cut-outs

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The above figure illustrates the stress analysis of the selected structure and also indicates the location of maximum stress i.e. near the bulkhead cut-out. This can be referred as a stress concentration region, where stress accumulates as a result of discontinuities in the structure. The probability of crack occurrence is very high in this area.

VIII. DAMAGE ESTIMATION USING PRINCIPLES OF FATIGUE

The fatigue problem relating to metals has been investigated for more than a century. The failure occurs depending upon the total number of cycles and the stress involved. Based on this criteria the fatigue can be broadly categorized into low cycle ,high cycle and very high cycle fatigue. Also there are theories which state the probability of damage. Palmgren Miner’s rule states that where there are k different stress magnitudes in a spectrum, S_i ($1 \leq i \leq k$), each contributing $n_i(S_i)$ cycles, then if $N_i(S_i)$ is the number of cycles to failure of a constant stress reversal S_i , failure occurs when:

$$\sum_{i=1}^k \frac{n_i}{N_i} = C$$

C is experimentally found to be between 0.7 and 2.2. Usually for design purposes, C is assumed to be 1. For the case considered here, the different stress magnitudes are given in Table.2 and k=7. The calculation for 9 psi is shown below and the other cases are illustrated in Table. 4. It is obtained that the maximum stress value near the bulkhead cut-out is 24.236kg/mm² or 34.622 ksi. To improve the accuracy of the calculations, correction factors are to be included. This takes into account the irregularities and surface discontinuities. To calculate the value of stress amplitude (σ_{amp}), we make use of the maximum stress and correction factors.

Stress Concentration Factor : 1

Design Reliability : 0.897

Surface Roughness : 0.8

Then maximum stress incorporated with correction factors = 34.622/(0.897*0.8) = 48.2469 ksi. The stress amplitude (σ_{amp}) is given by, (σ_{amp}) = ($\sigma_{max} - \sigma_{min}$)/2 = 24.1235 ksi. We have n_i = 850 cycles (for 9 psi). Then using constant-life diagram [22], the value of N_i is obtained as 90000 cycles.

Table 5: Representation of stress amplitude calculation and corresponding life cycles

Differential Pressure (psi)	Max stress near bulkhead (kg/mm ²)	Max stress incorporating correction factor (ksi)	σ_{amp}	n_i	N_i	n_i / N_i
6	16.159	32.048	16.024	26000	10^7	0.00260
6.5	17.500	34.838	17.419	10500	$9 * 10^5$	0.01150
7	18.800	37.416	18.708	8200	10^6	0.00820
7.5	20.100	40.014	20.006	2450	10^6	0.00245
8	21.465	42.726	21.363	1050	$8 * 10^5$	0.00133
8.5	22.799	45.387	22.694	950	$11 * 10^5$	0.00084
9	24.236	48.247	24.124	850	$9 * 10^4$	0.00944

Using Table.5, the maximum stress near bulkheads can be obtained. Also the magnitudes of stress amplitude is found out. In order to check the condition for Miner’s rule the n_i / N_i ratio is to be analysed. That can be obtained directly from Table.5

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Then, $\sum (n_i / N_i) = 0.03637$

According to Miner's rule the design is safe if $\sum (n_i / N_i) < 1$. The condition is valid here and the material is safer to operate in the conditions selected for study.

IX. CONCLUSIONS

- The study conducted here based on the fuselage panel with multiple cut-outs enabled to locate the region where stress is concentrated. This is found near the bulkhead cut-outs and the magnitude of maximum stress is 24.236 kg/mm².
- The damage estimation is done hereafter to investigate the structure is safe under the prevailing load conditions and cycles. It has been validated that the selected fuselage panel can accommodate the impact of load spectrum considered here.
- Also a weight optimization is done so as to analyze the load bearing capacity of a thinner panel. This has been validated by reducing the panel thickness from 2mm to 1mm keeping all other criterias and parameters same.

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