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Fatigue Analysis of A Panel Consisting Of Window Cutout and Frames in the Fuselage of A Transport Airframe

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Abstract - ***Aircraft is a complex mechanical structure with a very high structural safety. Aircraft will rarely fail due to a static overload during its service life. As the aircraft continues its operation, fatigue cracks initiate and propagate due to fluctuating service loads. To ensure airworthiness of an aircraft during its entire economic service life, fatigue and damage tolerance design, analysis, testing and service experience correlation play a vital role. This paper deals with the problem of stress analysis of a fuselage panel. The panel consists of a window cutout and stiffeners on either side of the cutout. The maximum tensile stress location will be identified in the panel. MSC PATRAN and MSC NASTRAN will be used for the static stress analysis.***

In a structure like airframe, a fatigue crack will appear at the location of high tensile stress. Further these locations are invariably the sites of high stress concentration. Therefore, the first step in the fatigue design of an airframe is the identification of high tensile stress. This is facilitated by a local refined FEA. This is followed by an estimation of the local stress at the highest stress concentrator. In the second phase of this paper the problem of prediction of life to fatigue crack initiation under the constant amplitude service loads will be addressed.

Key Words:

1. Introduction

Aircraft is a complex mechanical structure with a very high structural safety. The major aircraft structures are Wings, fuselage, and empennage. The primary flight control surfaces, located on the wings and empennage, are ailerons, elevators, and rudder. These parts are connected by seams, called joints. All joints constructed using rivets, bolts, or special fasteners are lap joints. Fasteners cannot be used on joints in which the materials to be joined do not overlap - for example, butt, tee and edge joints. A fayed edge is a type of lap joint made when two metal surfaces are butted up against one another in such a way as to overlap.

The largest of the aircraft structural components, there are two types of metal aircraft fuselages: Full monocoque and semimonocoque. The full monocoque fuselage has fewer internal parts and a more highly stressed skin than the semimonocoque fuselage, which uses internal bracing to obtain its strength.

The full monocoque fuselage is generally used on smaller aircraft, because the stressed skin eliminates the need for stringers, former rings, and other types of internal bracing, thus lightening the aircraft structure.

The semimonocoque fuselage derives its strength from the following internal parts: Bulkheads, longerons, keel beams, drag struts, body supports, former rings, and stringers.

1.1 Bulkheads

A bulkhead is a structural partition, usually located in the fuselage, which normally runs perpendicular to the keel beam or longerons. A few examples of bulkhead locations are where the wing spars connect into the fuselage, where the cabin pressurization domes are secured to the fuselage structure, and at cockpit passenger or cargo entry doors.

1.2 Longerons and Keel Beams

Longerons and keel beams perform the same function in an aircraft fuselage. They both carry the bulk of the load traveling fore and aft. The keel beam and longerons, the strongest sections of the airframe, tie its weight to other aircraft parts, such as power plants, fuel cells, and the landing gears.

1.3 Drag Struts and Other Fittings

Drag struts and body support fittings are other primary structural members. Drag struts are used on large jet aircraft to tie the wing to the fuselage center section. Body support fittings are used to support the structures which make up bulkhead or floor truss sections.

Former rings and fuselage stringers are not primary structural members. Former rings are used to give shape to the fuselage. Fuselage stringers running fore and aft are used to tie in the bulkheads and former rings.

1.4 Cutouts

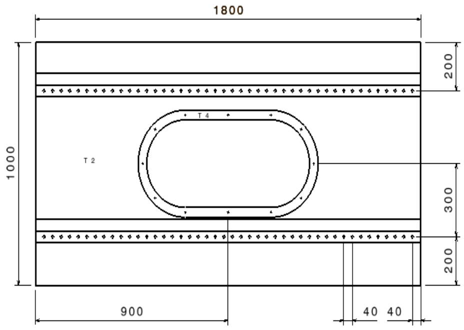
In aircraft structure, Cutouts are inevitable in structures due to practical consideration. Cutouts are commonly found as access ports for mechanical and electrical systems. Cutouts are also needed to provide access for hydraulic lines. For damage inspection. In addition, the designers often need to incorporate cutouts or openings in a structure to serve as doors and windows. Those structural panels with cutout are subjected to various kinds of loads and could fail if overloaded. Therefore, the stress variations and failure characteristics etc of those structural panels with cutouts must be fully understood to obtain knowledge for structural design.

2. Problem definition and methodology

A stiffened panel which represents the structural features near the passenger window cut out is considered for the analysis. The basic configuration of the stiffened panel consists of a window cutout and stiffeners on either side of the cutout, as shown in Figure. The stiffened panel has variable thickness. The dimensions of the panel are shown in figure. In practical the stiffened panel will be a curved panel. For the current study it is idealized as a flat panel. The panel has the following structural elements and different salient features

- Skin
- Bulkhead
- Rivets
- Variable thickness
- Large cutout
- Rivet holes around the cutout

The skin is a thin sheet of material stiffened using bulkheads. The cutout surrounding area has higher thickness. The bulkheads are riveted to the skin. Fuselage cabin experiences hoop stresses due to the pressurization. These hoop stresses are considered as uniform tensile stresses at the edges of the panel. The maximum tensile stress magnitude and its location are identified through the stress analysis of the stiffened panel.



2.1 Aircraft cabin pressurization

Aircraft are flown at high altitudes for two reasons. First, an aircraft flown at high altitude consumes less fuel for a given airspeed than it does for the same speed at a lower altitude because the aircraft is more efficient at a high altitude. Second, bad weather and turbulence may be avoided by flying in relatively smooth air above the storms. Many modern aircraft are being designed to operate at high altitudes, taking advantage of that environment. In order to fly at higher altitudes, the aircraft must be pressurized. It is important for pilots who fly these aircraft to be familiar with the basic operating principles.

In a typical pressurization system, the cabin, flight compartment, and baggage compartments are incorporated into a sealed unit capable of containing air under a pressure higher than outside atmospheric pressure. On aircraft powered by turbine engines, bleed air from the engine compressor section is used to pressurize the cabin. Superchargers may be used on older model turbine-powered aircraft to pump air into the sealed fuselage. Piston-powered aircraft may use air supplied from each engine turbocharger through a sonic venturi (flow limiter). Air is released from the fuselage by a device called an outflow valve. By regulating the air exit, the outflow valve allows for a constant inflow of air to the pressurized area.

2.2 Material

Selection of materials in aircraft construction is rather complex and is based on trade off amongst conflicting requirement of high strength , low density and ease of fabrication or processing. The material used in various parts of vechile structures generally are selected by different criteria. The material used in the fuselage structure is aluminium alloy 2024-T351 and it has the following properties.

Young's Modulus, E = 70,000 N/mm^2

Poison's Ratio, μ = 0.3

2.3 Finite element model

An isometric view of the 3-D CATIA model of the stiffened panel is shown in figure The CATIA model is imported to the finite element analysis pre-processor software to prepare the the finite element model of the panel. The geometry extraction is carried out which will be used as input to the finite element mesh generation.The mesh generated for the stiffened panel is shown in figure . The panel consists of a large cutout, tiny holes around the cutout, variable thickness and bulkheads on either sides of the cutout. All these features are considered for the development of the finite element model. The mesh generated near the tiny hole around the cutout is shown in figure .

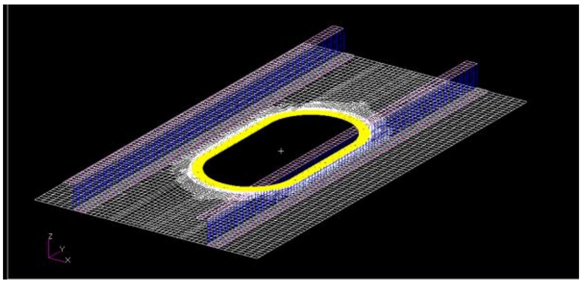


Fig-1 : discretization of model

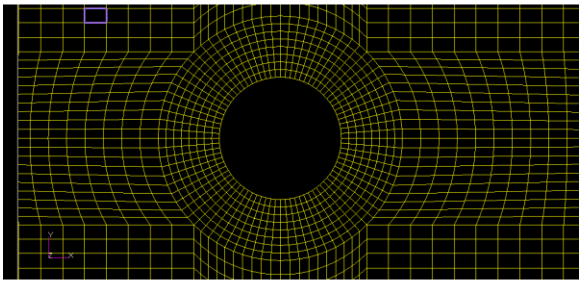


Fig-2: mesh around a Rivet hole

2.4 Load and boundary condition

The cabin pressurization is considered as one of the critical load cases for the fuselage in the design of the airframe. Due to pressurization in the fuselage will experience the hoop stress and longitudinal stresses. The window cutouts in the fuselage are in the circumferential direction. Therefore the hoop stress is more critical than the longitudinal stress in the case of the stiffened panel with window cutout. The internal pressurization will be in the radial direction, but the hoop stresses developed will be in the circuferential direction. The segment of the fuselage which curved in practical is idealized as a flat panel in the current study. The load acting on the edge of the panel will be equal to the load due to hoop stress at that section. Essentially the load acting on the panel will be tension-tension loading. For the finite element analysis of the stiffened panel one edge of the panel is constrained with all degrees of freedom. The load is introduced at the opposite edge to that of the constrained edge.

The differential pressure to be applied inside the fuselage will be in the range of 6 psi to 9 psi depending on the altitude at which the aircraft will be flown. There are two cases considered in the linear static analysis. One at 6 psi and the other at 9 psi. since it is linear static analysis the response of the structure also will be linear. The tension load applied on the edge of the stiffened panel is calculated as below.

<u>Load Case-1</u>

Pressure ⇛ 6psi = 0.00422 kg/mm^2

Hoop stress = p*r/t

Where

The radius of fuselage 'r' = 3000mm

Thickness of the skin't' = 2mm

Hoop stress = 0.00422*3000/2

= 6.33kg/mm^2

The stress developed at the edge of the panel is same in the structural elements skin, tear strap and bulkhead. This is because of the displacement compatibility between these three members and the material used for these members is the same.

3. Result and discussion

The cabin pressurization is considered as one of the critical load cases for the fuselage in the design of the airframe. Due to pressurization in the fuselage will experience the hoop stress and longitudinal stresses. The window cutouts in the fuselage are in the circumferential direction. Therefore the hoop stress is more critical than the longitudinal stress in the case of the stiffened panel with window cutout.

There are two cases considered in the linear static analysis. One at 6 psi and the other at 9 psi. and also there several iterations are carried out to get the desired degree of accuracy.

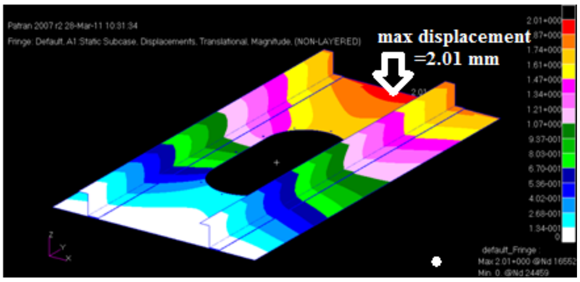


Fig-3: Displacement Plot for load case 1

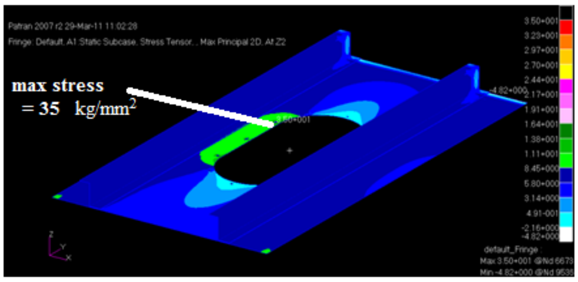


Fig-4: Stress Plot for load case 1

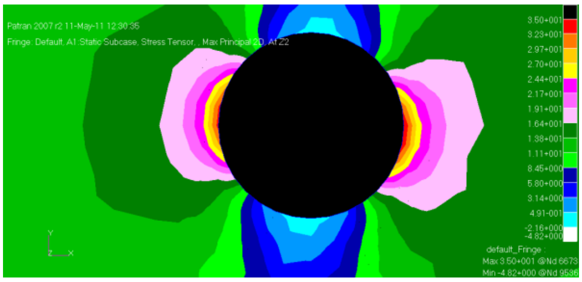


Fig-5: Stress contour around a rivet hole for load case

Table -1 shows convergence results of different mesh divisions are considered in the rivet hole.

Number of elements around hole	Max principal stresses in Kg/mm²	Displacement in Mm
32 elements	35.0	2.01
64 elements	36.2	2.01
80 elements	36.6	1.98

Load Case-2

In the second case the 9 psi pressure load considered in the static analysis. The load in the panel axial direction.

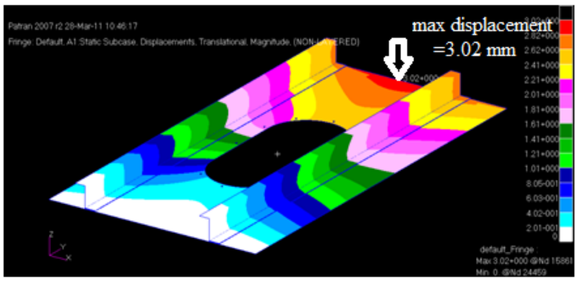


Fig-6: Displacement Plot for load case 2

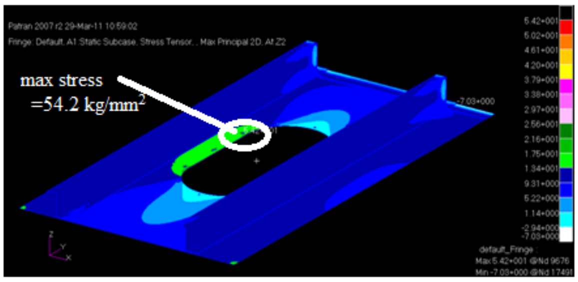


Fig-7: Stress Plot for load case 2

Figure shows displacement and stress plot for the maximum pressure load case condition. Maximum displacement corresponding to 9 psi loading condition is 3.02 mm and occurs on the load applied region of the stiffened panel. The dark red colour contour on the surface of the stiffened panel indicates that the maximum displacement location. Stress plot clearly indicates that rivet hole is taking maximum stress is 54.2 kg/mm² in the cutout region. So we have design to reduce stress level to acceptable limit i.e. yield strength of the material is 35 kg/mm². The following steps are carried to get the stress acceptable limit.

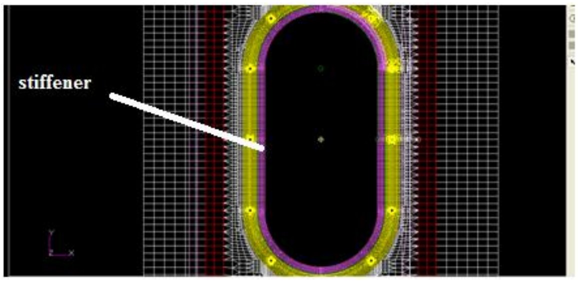
Step-1 increasing thickness in the cutout region

Table -2; variable thickness result

Cutout thickness	Max principal stress in Kg/mm²	Displacement in Mm
t=2.5	51.5	3.02
t=3	49.1	2.97
t=3.5	47.0	2.95

Table shows the Still stress magnitude is higher than acceptable limit (35 kg/mm²). If we further increases the thickness in the cutout region of the stiffened panel the weight of the panel increases. Weight of the structure also important consideration in the design process.

Step-2 Adding flange stiffener inside of the cutout.



Figure; Stiffener inside of the cutout

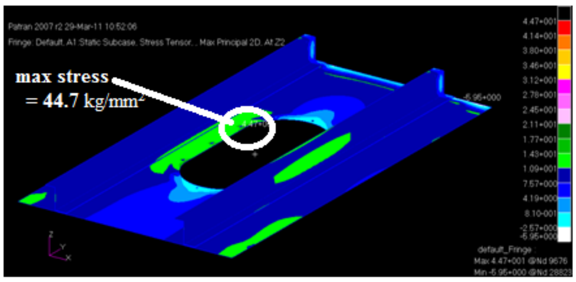
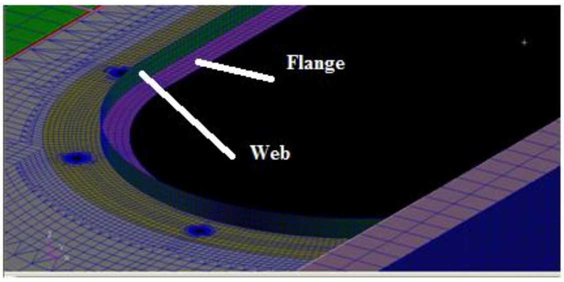


Fig-4: Stress Plot

The maximum stress is 44.7 kg/mm² and occurs in rivet hole near cutout in the stiffened panel. In this step still stress is higher than the acceptable limit.

Step-3 adding L type stiffener to the cutout

In this step both flange and web type stiffener added inside the cutout in the stiffened panel as shown in figure.



Figure; T type stiffener inside of the cutout

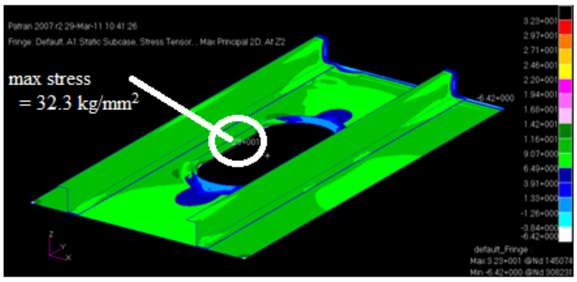


Fig-4: Stress Plot

Table-3 shows convergence results of different mesh divisions are considered in the rivet hole.

Number of elements around hole	Max principal stresses in Kg/mm²	Displacement in mm
32 elements	31.1	2.65
64 elements	31.9	2.68
80 elements	32.3	2.65

The maximum stress is 32.3 kg/mm² occurs on the rivet hole near cutout region in the stiffened panel. The maximum stress is acceptable limit when compared to the design material Strength.

4. Prediction of Fatigue Life Analysis

From the stress analysis of the stiffened panel the maximum tensile stress location is identified. A fatigue crack will always initiate from the location of maximum tensile stress. From the stress analysis it is found that such a location is at one of the rivet hole. A typical flight load spectrum is considered for the fatigue analysis of the stiffened panel. Calculation of fatigue life to crack initiation is carried out by using Miner`s Rule.

The various correction factors are considered in the calculation of fatigue cycles, they are

For surface roughness (e_{sr}) – 0.8

For type of loading (e_l) – 1

For reliability design (e_r) -0.897

Stress correction factor can be calculated as follows

= maximum stress/correction factor coefficients

= 21.6 / (0.8*1*0.897)

= 30.10kg/mm^2

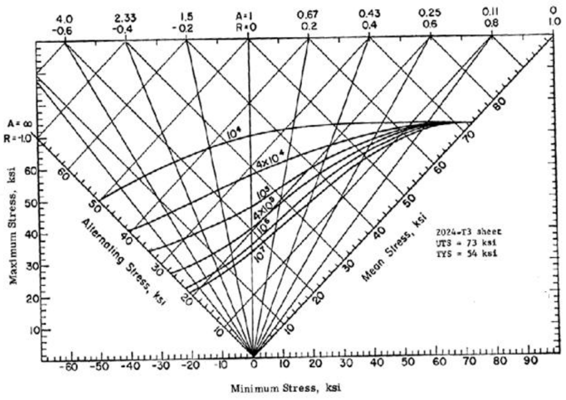


Figure : Typical constant life diagram for unnotched fatigue behavior of 2024-T3 Aluminum alloy.

Using the maximum stress and value of Rin the S-N curve given in **Figure** The fatigue cycle for various stress levels are found out.

From Miner's equation,

$\sum n_i/N_f = C$

Where n_i= number of actual cycle

N_f= number of fatigue cycles to failure

Pressure in psi	Actual no of cycles	Fatigue cycles from graph	Damage accumulated from miner's formula
6	14000	4*10^5	0.035
6.5	7500	3.25*10^5	0.023
7	4500	2.5*10^5	0.018
7.5	1800	85*10^3	0.021
8	1200	70*10^3	0.017
8.5	500	32,500	0.0153
9	500	25,000	0.02

Damage accumulated from miner's formula

Total damage accumulated for all load case is given by

$D=d_1+d_2+d_3+d_4+d_5+d_6+d_7$

D= 0.035+0.023+0.018+0.021+0.017+0.0153+0.02

D=0.1493

Considering the scatter factor 3 times to the total damage accumulated for load case is 0.457. According from miner's rule the total damage accumulated for all load case is less than 1 therefore failure crack will not initiated in that specified location so aircraft structure doesn't fails.

5. Conclusions

1. The structural analysis of the stiffened panel with window cutout is carried out using FEA approach.
2. Internal pressurization of the fuselage which is one of the critical load cases is used to calculate the remotely applied load on the stiffened panel.
3. The maximum stress location is identified at one of the rivet holes.
4. A mesh refinement is carried out at the rivet hole location to get the correct magnitude of the stress.
5. Few iterations are carried out for different load cases to improve the design of the structure near the window cut out location
6. S-N data for the fatigue life calculation is referred from Bruhn's airframe design and analysis book.

7. Fatigue life to crack initiation is calculated using Miner's rule.
8. The damage accumulated because of the fatigue loads experienced by the panel are well within the critical damage.

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