Analysis of Bottom Skin of Stiffened Panel with a Cutout of a Transport Aircraft

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Abstract - *Aircraft are complex mechanical structures designed with high structural safety. But be it human errors or equipment failures, the transport aircraft faces severe damages in its service life. But the major causes of aircraft failures are structural damages with the initiation and subsequent propagation of fatigue cracks. The bottom skin of stiffened panel is always under stress during a flight. The stiffened panel undergoes fatigue load which initiates cracks and it propagates due to fluctuating service loads as the aircraft is in operation. The wings of an aircraft are used as fuel tanks and to access them without a cutout is difficult. This paper includes stress analysis of a circular cutout on the bottom skin on the stiffened panel of a transport aircraft's wing to identify the fatigue crack initiation location when having a circular cutout to access the fuel tank, which is carried out using NASTRAN and PATRAN solver.*

Key Words: **Cutouts, Stress analysis, Fatigue, Aircrafts**

1.INTRODUCTION

The airframe is one of the sophisticated structures. The single most important responsibility of the structural engineer is to ensure the safety of the airframe throughout its service life span. Let us say around 40 – 50 years of service life. Even to the last flight, the airframe should be safe to fly. The only way to ensure safety is to know the quantity of safety. The quantity of safety is given by either Factor of safety (FOS) or Margin of safety (MOS). The factor of safety is defined as the ratio of design ultimate load to design limit load. The margin of safety is the ratio of ultimate stress by maximum stress -1.

According to certification authorities, FOS should be 1.5 and also Design limit load has to be estimated from FAR (Federal Aviation Regulation) or any other certificating rule book then determine the Design ultimate load and then design the airframe. MOS of an airframe has to be maintained similarly to FOS. So from this one can get the single most important role of the engineer is to determine the failure load of the airframe. That is achieved by designing the airframe.

The airframe has to be designed for the following to ensure safety throughout its service life span:

- Static design
- Fatigue design
- Damage tolerance design

But failure due to fatigue is the most common reason for structural failure of an aircraft in service life span. The fatigue happens due to the fluctuating loads or cyclic loads by which its static strength gets deteriorated. This happens because of the repetitive flight operations. The parts of an aircraft become weak eventually and without proper attention and repairs, it begins to initiate the crack which internally intensifies the stress concentrations due to which the structure fails catastrophically. Poor designing, improper maintenance, or carelessness while constructing are the reasons which lead to structural damage causing failure due to fatigue, which is often fatal and frightening. Any element of the aircraft which has sharp edges, necked-down sections or sensitive areas like wing and fuselage skin panels, spar webs, stringers, stiffeners are prone to failure due to fatigue. But even after taking precautions and conducting inspections, cracks have arisen in these structural elements. The cracks lead to a reduction in stiffness and the total load carrying capacity of an aircraft.

1.1 Cutouts

Cutouts in any aircraft have never been approved by engineers because it adds weight to the overall design as well as increases the costs as deciding the shape and size of the cutouts is a difficult process. Cutouts are areas of stress concentration which is a problem in static and fatigue design which creates problems in strength, stiffness and other necessary aspects of an aircraft. Below are the reasons for having a cutout in an aircraft;

- Providing easy access to the fuel tanks at the bottom skin of the wing
- Providing landing and retracting gear opening at the bottom of the fuselage
- Providing accessibility for maintenance and assembly of the aircraft
- Having window cutouts in fuselage
- Having cutouts like hand holes for easy inspection

In this paper, the cutout is considered in the bottom skin of a stiffened panel of an aircraft where the stress acting is the tensile stress. The cutout is an auxiliary hole to provide easy access to the fuel tank and any discontinuity or flaw in that will lead to structural failure as it is a stress concentration area.

2. OBJECTIVE AND SCOPE

- The main objective of this paper is to study and evaluate the stress analysis of the cutout on the bottom skin for the wing structure of an aircraft.

- The cutout which is for providing easy access to the fuel tank will be used to check if there is any leakage in the fuel, cracks in the pipeline, etc.

3. METHODOLOGY

3.1 Specifications of the aircraft

To have the aerofoil data we have considered the aircraft "Cessna 172 Skyhawk". Cessna 172 Skyhawk is a four-seater (one for crew and three for passengers) American aircraft. It is a single-engine and high wing aircraft having a fixed-wing. Cessna 172 Skyhawk is the most successful aircraft in terms of its larger lifespan and popularity. It was first introduced by Cessna Textron Aviation in 1956 and it is still in production.

Specifications of the Cessna 172 Skyhawk aircraft are as follows:

- Aircraft model: Cessna 172 Skyhawk
- Engine model: Continental O-300
- Aerofoil: NACA 2414 (modified)
- Length of the aircraft: 8.28 m
- Chord length: 1502.7 mm
- Wingspan: 11 m
- Aspect Ratio: 7.32
- Empty weight: 767 kg
- Maximum take-off weight: 1111 kg
- Fuel capacity: 212 litres

3.2 Load Calculations

With the data of Cessna 172 Skyhawk aircraft, we can calculate the bending moment. A bending moment is a reaction taking place in a structural element when an external force is applied to that element, which causes it to bend. The maximum bending moment always takes place at the root of the spar where the wing and components of the fuselage are attached. The formula for bending moment is:

Bending moment (BM)= Force*Perpendicular distance

Let us consider Fig.1 a panel with a circular cutout for fuel access. As shown in the Fig1., the pressure (P) is applied on both sides of the panel.

The load acting on the panel and the total load can be calculated using the formulas:

Bending moment at cutout section is obtained using the BMD.

Tensile force to be applied on the panel edge= BM at Cutout section/ Aerofoil section depth

This load has to be distributed throughout the panel.

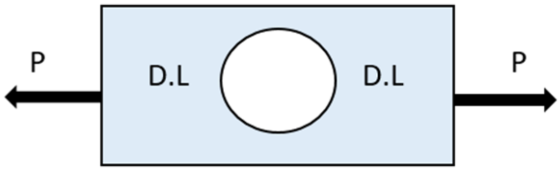

Fig-1: Image representation of panel with circular cutout

3.3 Material Specifications

The selection of aircraft materials depends on the factors such as structural performance and cost. Therefore, the material properties that are required are:

- Young's Modulus = 72 MPa
- Poisson's Ratio = 0.33
- Density = 27.7 KN/m^3
- Yield strength = 362 MPa
- Ultimate strength = 483 MPa

Aluminum alloys have a low density and excellent strength to weight ratios. Thus, the material considered for the stiffened panel is AL 2024-T3 with the above properties.

3.4 Load Calculations of Wing Box

Weight of the aircraft considered = 7000 kg

Design load factor = 3.2 g

Factor of safety considered in design of aircraft = 1.5

Thus, Total design load on the aircraft = 33600 kg-f

Total lift load on the aircraft is distributed as 80 % and 20 % on wing and fuselage respectively.

Total load acting on the wings = 26880 kg-f

The load acting on each wing = 13440 kg-f

Total span of each wing = 7500 mm

The resultant load is acting at a distance 3200 mm from the wing root.

The resultant load is at a distance of 1200 mm from the root end of the wing box considered for the analysis

The bending moment at the root end of the wing box = 13440 X 1200 = 16128×10^3 kg-mm

Load to be applied at the other end of the wing box

= $16128 \times 10^3/1400$ = 11520 kg-f

4. ANALYSIS OF THE PLATE WITH CIRCULAR CUTOUT

Aircraft's components in an earlier age were made up of wood, but now the aircraft is constructed from composite materials such as carbon fiber, aluminum alloys, etc. The aircraft stiffened panel is made up of several composite plates with stiffeners in longitudinal and/or transverse directions. These stiffeners are broadly classified as open type or closed type of boxes. A stiffened panel is designed to carry loads in the aircraft service periods with maintaining a level that is specified for damage tolerance. The stiffened panel is connected with stringers with joints and clamping mechanisms. It has a fixturing board and the skin.

Thus, after the calculations of the wing box and load calculations, a stiffened panel is designed with a circular cutout on the MSC Nastran and PATRAN software which is used for the stress analysis of the cutout on the panel.

4.1 MSC NASTRAN AND PATRAN

For performing the FEA analysis of the stiffened panel the software of MSC Nastran is used where the material properties, boundary conditions, load application, and meshing is performed in the MSC NASTRAN solver where the stress analysis of the stiffened panel is carried out and then the database file is imported in MSC PATRAN solver for reviewing the results.

4.2 FEM (FINITE ELEMENT METHOD)

The finite element method is a numerical technique used for solving problems that are described by partial differential equations that arise in mathematical modelling. The problems solved in FEM are related to structural analysis, fluid flow, heat transfer. In FEM the differential equations are solved using boundary value conditions and meshing.

There are several steps while solving MSC Nastran and PATRAN, which are as follows:

1. Pre-Processing in MSC PATRAN
    - Geometric modelling
    - Finite element modelling
    - Material definition
    - Property definition
    - Deck preparation
2. Solving in MSC NASTRAN
3. Post Processing in MSC PATRAN
    - Reviewing the results

4.3 PROCEDURE

The stiffened panel of an aircraft has a rectangular shape, so a rectangle of 800mm×400mm is drawn which is considered as the stiffened panel of the aircraft, and two circles of 50mm and 60mm radius respectively are the circular cut-outs on the rectangular panel. After providing the measurements to the rectangular panel and the circular cut-out, meshing is carried out. Meshing in MSC NASTRAN and PATRAN solver is carried out using the mirror meshing technique; as the rectangular plate has four quadrants, we mesh one quadrant and apply the same to all the three other quadrants including the circular cut-out.

The load is applied to the plate calculated with the Aerofoil data of "Cessna Aircraft 172 Skyhawk" which is 1035.17 kg/mm which is distributed along each part of the stiffened panel and the circular cutout. And after applying loads and boundary conditions on the model, the stress analysis of the panel with the circular cutout is performed in the MSC NASTRAN.

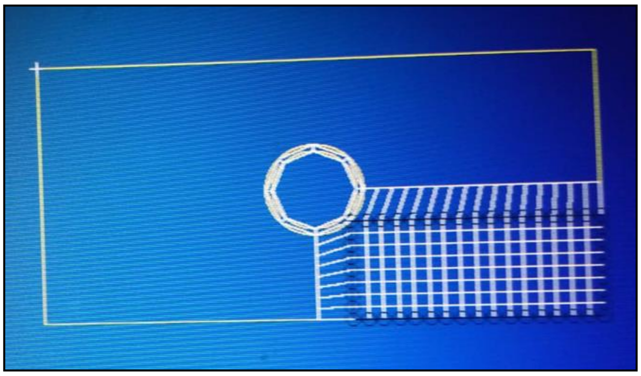

Fig-2: Meshing of one quadrant on the plate

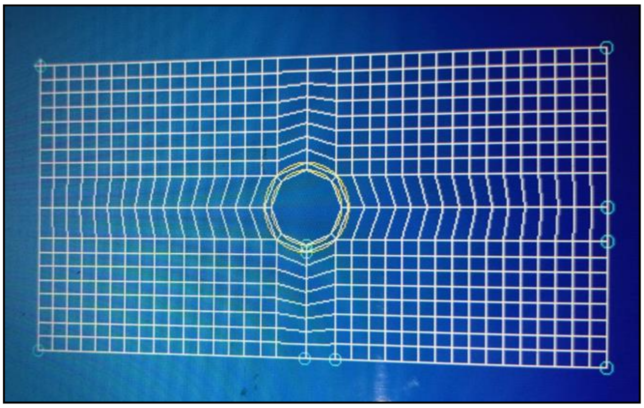

Fig-3: Mirror meshed stiffened panel and cut-out

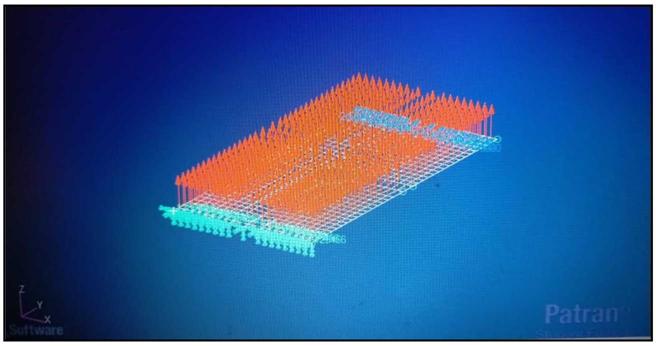

Fig-4: Loads applied to the stiffened panel

5. RESULTS

- The result file produced by the MSC NASTRAN solver is imported in MSC PATRAN to review the results. The result is carried out in form of stress contour over the whole structure of the plate with circular cut-out and deformation of parts.
- The maximum stress if found near the cut-out region. The fine mesh carried out helps to capture gradient stress distribution.
- The maximum stress found near the fuel access cutout is of 31.6 kg/mm^2

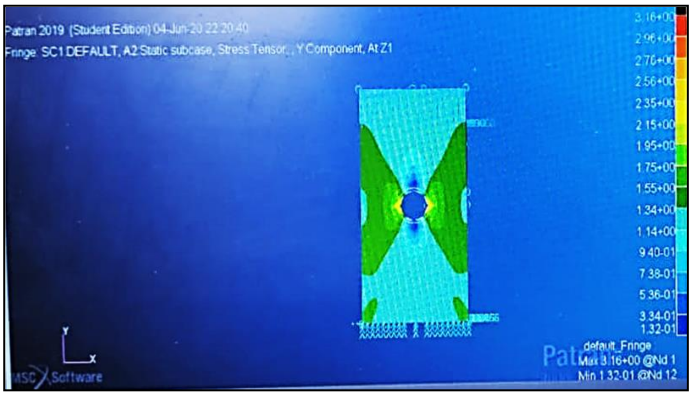

Fig-5: Stress contours on the stiffened panel with cut-out

After reviewing the results in MSC PATRAN, to obtain a mesh independent value for maximum stress several iterations are carried out and the graph is plotted using the bending moment data.

The iterations, bending moment data, graph is shown in Fig. 6,7,8 respectively.

0.1	6.2	0	588.79	0	0									
0.2	6.15	0.612	584.79	612	357891.5	0								
0.3	6.1	1.224	579.29	1224	709051	354525.5	0							
0.4	6	1.836	569.8	1836	1046153	697435.2	348717.6	0						
0.5	5.8	2.448	550.807	2448	1348376	1011282	674187.8	337093.9	0					
0.6	5.6	3.06	531.816	3060	1627357	1301886	976414.2	650942.8	325471.4	0				
0.7	5.5	3.672	522.317	3672	1917948	1598290	1278632	958974	639316	319658	0			
0.8	5	4.284	474.4	4284	2032330	1741997	1451664	1161331	870998.4	580665.6	290332.8	0		
0.9	4	4.896	379.6	4896	1858522	1626206	1393891	1161576	929260.8	696945.6	464630.4	232315.2	0	
0.98	2.3	5.508	218.426	5508	1203090	1069414	935737	802060.3	668383.6	534706.8	401030.1	267353.4	133676.7	0
	52.65		5000.036		12100717	9401035	7059244	5071978	3433430	2131976	1155993	499668.6	133676.7	0
	23.881		1111	5000										

Fig-6: Iterations to obtain mesh independent value for maximum stress

0	12100717
612	9401035
1124	7059244
1836	5071978
2448	3433430
3060	2131976
3672	1155993
4284	499669
4896	133676
5508	0

Fig-7: Bending moment iterations

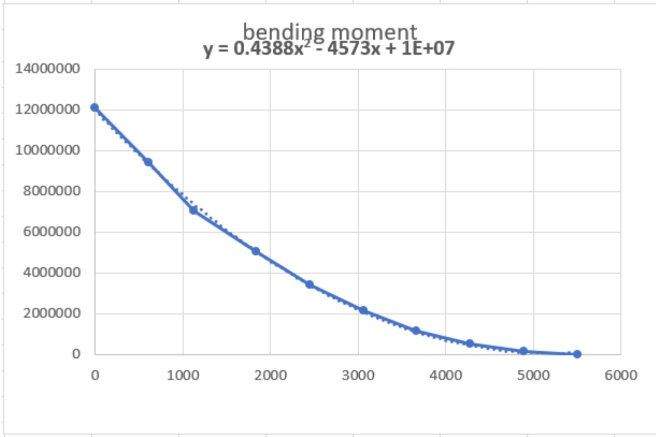

Fig-8: Bending moment graph

6. CONCLUSIONS

- The FEA analysis has been run on the software of MSC NASTRAN and PATRAN. The stress analysis of the circular cut-out on the stiffened panel for fuel access is carried out using the FEA approach.
- The maximum stress location is identified near the cut-out area where the load is distributed.
- The fatigue crack usually initiates from the location of maximum tensile stress in the structure and if these stresses are neglected they can lead to catastrophic failure and can also result in loss of life.

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