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Study on the Fatigue Life Estimation of an Airframe Fuselage Window Cutout Panel

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Abstract: Major challenges of current aircraft industries are to overcome the increasing demand of lower weight, higher performance and stability and longer life of specimen at reduced cost. Many assembly processes in critical structures like military or commercial aircraft will be based on riveted joints. These joints are prone to crack initiation which may lead to multi-site damage, i.e. formation of simultaneous crack at all the rivet locations because of uniform stress, thus lot of effort is going to predict the life of structure under multi-site damage, in this project a part of fuselage connection is taken for multi-site damage analysis, and this connecting part is known as splice joint.

In this study, a splice joint which contains two splice panels, a doubler and stiffener made up of AL-2024 T3 is taken as the specimen. This assembly consists of 26 rivets arranged in zig-zag pattern. The load on the assembly (Splice joint) is observed to be cause of cabin internal pressurization of fuselage. The whole assembly is virtually modeled using MSC PATRAN software, followed by stress analysis in MSC NASTRAN.

The analysis is done for predicting the crack initiation sequence based on the stress values, for all these cracks, stress intensity factors are established and crack propagation for each crack is obtained based on the stress intensity factor. The change in Stress intensity factor vs. crack length behavior is studied and final failure of splice skin is analyzed. From final failure it is observed that splice skin fails because of net-section yielding criteria, even though the stress intensity factor is less than threshold stress intensity factor values.

Key Words: stress intensity factor, residual strength

I. INTRODUCTION

Cutouts in aircraft structures are very much vital in practical consideration. These cutouts are medium channel for mechanical and electrical systems to pass through and for hydraulic lines too. For a periodical or sudden inspection cutouts are very much important to serve as door or window. These cutouts are sometime fall in circle of fail due to various kinds of loads. The basic function of an aircraft structure is to form an aerodynamic shape and to protect the passengers and these can only be possible by thorough designing of successful structural window cut-out formation. Because aircraft structures main role is to transmit and resist the load which applied. So the Airframe structure is most important segment to come up with perfect structural model. With all these we can achieve a suitable fatigue life of an aircraft.

A. Material

To fight against crack initiation to achieve fatigue desugn, material selection is very much vital. And this way of designing structure will make our structure strength .In aircraft industry selecting a material for particular segment or overhaul is complicated comparing to others.

B. Importance of present study

The alarm of the present study is on the evaluation of the crack initiation and propagation with their lifetime evaluation too .Change in crack arrest or growth (increment or decrement) capability check of the bulkheads in the window cut-out panel having a large cut-out for passenger window. The exact passenger window cut-out segment in the fuselage is a critical segment from the fatigue crack initiation point of view. The maximum tensile stress location will be identified in the panel. MSC PATRAN and MSC NASTRAN were used for the static stress analysis. The catastrophic failure may happen in the airframe structure. When a fluctuating loads acts on the structure the load carrying capability declines gradually this is the general phenomenon of fatigue. In a metallic base material the fatigue rises in quick mode and it propagates it. In window cut-out panel a crack will be get initiated from highest tensile stress location for the analytical representation too. Then after various crack lengths will be measured and initiated for the further calculations like stress intensity factor, crack life, propagation cycles and all.

The initiated crack will be purely perpendicular to the loading direction. Here the stress intensity factors values will be playing an important role to identify the crack arresting capability of the bulkhead on either side of the crack, where here we consider the crack arresting capacity of a bulkhead in one side, with the material characters and geometry of a design we found the fracture toughness of the particular material with specific thickness. Then the residual strength of the structural member is calculated with the combination of a ultimate tensile load and the fracture toughness. And the founded values will be plotted for different values.

C. The Problem Is Accessed With Following Sequences

A window cutout panel associated with bulkheads and stiffeners, subjected to a pressurized load of 9psi by hoop stress calculation to determine the point of maximum tensile stress. This point will be considered for the crack initiation. The analysis is carried out for different crack lengths and the respective SIF is found using which fatigue life calculation and residual strength are obtained.

This project will investigate the fail-safe feature in a window cutout panel in the presence of a crack. This study will be carried out through a FEM and FEA approach. A segment of the fuselage will be considered for the stress analysis. Internal pressurization is one of the critical load cases considered in the fuselage design. Internal pressurization introduces both hoop stress and longitudinal stresses in the structure. Modeling will be done in MSC NASTRAN/PATRAN and stress intensity factor will be estimated using MVCCI method.

Total Life(N_f)	=	Crack Initiation(N_i)	+	Crack Growth(N_p)
(S-N)		(σ-N)		(LEFM)

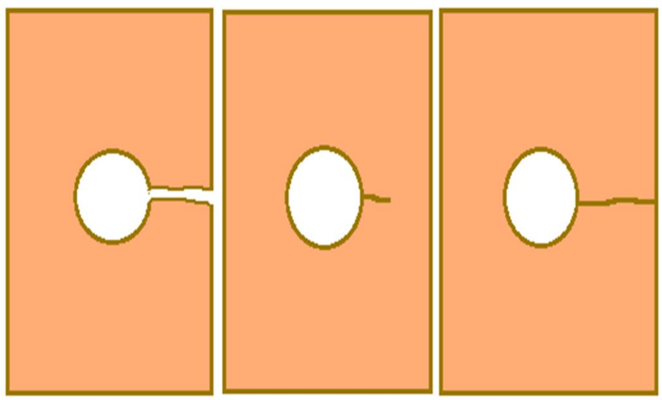

Figure 2.1 Schematic representation of fatigue failure

II. INTRODUCTION TO MSC/PATRAN

A finite element pre and post processor package MSC/PATRAN is a graphic based software package, primarily designed to aid in the development of finite element model (Pre-processing) and to aid the display and interpretation of analysis results (Post processing). MSC/PATRAN software is a mechanically computer aided engineering tool.

A. Three main Stages of fatigue failure

The process of fatigue failure involves three stages

1) Crack initiation
2) Crack propagation and
3) Catastrophic or over load failure after the crack reaches a critical size.

In the design of the airframe, the sources of damage to be considered are fatigue, corrosion and accidental. These damages should be serious enough to cause a loss of load carrying capacity. The entire operational life should be considered to assess the seriousness of these damages

The damaged structure should be able to carry reasonable loads. These damages must be detected during scheduled inspection and these should be repaired or replaced to restore the strength of an aircraft structure.

It is not just a wishful thinking that the airframe should be damage tolerant. It has to be built-into the structure at the design stage and rigorously maintained during the entire service life through NDI and repair/replace (maintenance policy).Aviation regulations require that most of the airframe structural components should be designed to be damage tolerant.

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B. Constant Amplitude Load Cycle

In this loads are varying constantly from maximum to minimum value. For example load on locomotive axle is of constant amplitude. The amplitude is predictable and is not erratic. The load varies as maximum and minimum or peaks and valleys. If peak or crest is represented as 'max' and the valley or trough is represented as 'min' then:

1) Load ratio or stress ratio (*R*) = $\sigma_{min/}\sigma_{max}$
2) Mean (*m*) = (max+min)/2
3) Range = max-min
4) Amplitude (*A*) = Range/2
5) 1 cycle= travel time from one peak to the next OR from one valley to the next.

C. MVCCI Method for SIF Evaluation

Modified Virtual Crack Closure Integral (MVCCI) method is used to determine stress intensity factor for different crack lengths in the window cutout panel. MVCCI method is based on the energy balance. In this technique, SIF is obtained for fracture mode from the equation.

$G=\frac{Ki2}{E}$ i=1, 2, 3….

Where Gi is the energy release rate for mode I, Ki is the stress intensity factor for mode i, E is the modulus of elasticity and β is 1 for plane stress condition. Calculation of the energy release rate is based on Irwin assumption that the energy released in the process of crack expansion is equal to work required to close the crack to its original state as the crack extends by a small amount Δa. Irwin computed this work as

$$W = \frac{1}{2}\int_0^{\Delta a} u(r)\sigma(r-\Delta a)dr$$

Where u is the relative displacement, r is the distance from the crack tip; Δa is the change in virtual crack length. Therefore, the strain energy release rate is

Therefore, the strain energy release rate is

$$G = \lim \Delta a \rightarrow 0 \frac{1}{2\Delta a}\int_0^{\Delta a} u(r)\sigma(r-\Delta a)dr$$

After simplification, modified strain energy release rate is

$$G = \frac{FX\Delta u}{2X\Delta aXt} \text{N/mm}$$

Where F is forces at the crack tip, Δu is crack opening displacement (COD), t is thickness of the skin and Δa is elemental edge length near the crack tip.

III. PROBLEM DESCRIPTION

A window cutout panel is considered for the present study. Stiffener for a widow cutout panel is Z model and its placed both side of a cutout section.

Total width of the window cutout panel is 700mm and length is 900mm. And stiffener is placed in a 600mm space in between to other stiffener. Cutout section has L beading with flange and web.

A. Material Specifications

Al 2024-T3 material properties are considered for the present window cutout panel configuration.

In present scenario many components of an aircraft structure were modeled by Al2024-T3 material. Other configurations of Al-2024 series like T4 were also used for modeling. The key material properties are

1) Density
2) Young's modulus
3) Ultimate Tensile strength
4) Yield strengths
5) Fatigue strength
6) Damage tolerance (fracture toughness and crack growth)

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B. Materials Properties

Property	Aluminium 2024-T3
Density	2.77 g/cm3
Ultimate Tensile Strength	483 MPa
Tensile Yield Strength	362 MPa
Young's Modulus	72 GPa
Poisson's Ratio	0.33
Fracture Toughness	105 MPa√m

Table 4.1: Material properties used for the analysis

Mechanical properties of various components like skin, stiffening members and rivets are required for finite element models as an input in software. Aluminum 2024-T3 is used for components fuselage and rivet. Table 4.2 describes few material properties used for analysis

The main role of fuselage in aircraft

1) Load because of Passengers & Cargo
2) In the case of a combat military aircraft, it carries weapons
3) It is the interconnecting link in between the other main structural units – it links the wings to the tail unit and this conveys the wing loads intern to the wing roots; therefore it must be able to carry both the air loads and the weight of the tail unit.
4) Carries the major aircraft systems like avionics, radar, hydraulics, etc
5) Houses the undercarriage and bear the associated loads transmitted to it.
6) Internal cabin pressurization. In this study internal cabin pressurization is considered as load case. A internal pressure of 9 psi considered for the current case. Two types of stresses are developed in the fuselage structure due to cabin pressurization
7) Circumferential Stress (Hoop Stress)
8) Longitudinal Stress

Circumferential Stress (Hoop Stress):-

Hoop stresses are developed in circumferential direction which is equivalent to tensile stress due to equivalent tensile force in the same direction.

P= Differential cabin pressure = 9 Psi = 0.0620 kg/mm2

d = Diameter of the fuselage = 1946mm

t= Thickness of the window cutout plate = 2 mm

The hoop stress = p×d/2t

= 0.0620×1946/2x2

σ_h= 30.16 MPa

The hoop stress on stiffener = p×d/2t

= 0.0620×1946/2×1.5

σ_s = 40.21 MPa

As the skin was represented by 2-D finite element, the force per unit length of a window cutout panel replicates the force on window cutout panel due to hoop stress.

Force on skin due to hoop stress = (Hoop stress x cross sectional area)

= (30.16×900×2)

F= 54288N

C. Load and Boundary Condition

In the present window cutout panel, the internal pressure is considered. The circumferential hoop stresses and the longitudinal stresses developed in the fuselage cylindrical shell are equals to tensile stresses of the window cutout panel. The tensile load can be calculated by using the hoop stress developed in the window cutout panel and the corresponding cross sectional area. This tensile

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stress is uniformly distributed over the cross section. Uniformly distributed tensile load will be applied on the window cutout panel in transverse axial direction. Uniformly distributed load is applied on edges of skin and stiffeners in the loading direction. On the top of the panel edge nodes are constrained in three rotations and Y translations. At other end, all the edge nodes of window cutout panel are constrained in all six degree of freedom (three translations and three rotations).

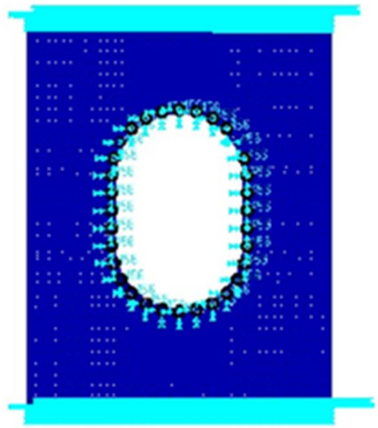

Figure: 4.1 Overall boundary conditions applied in a model

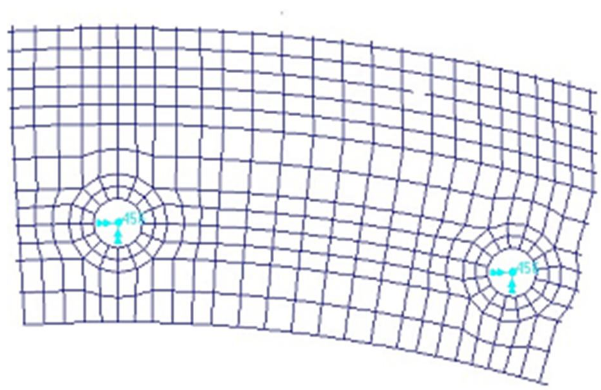

Figure: 4.3 Boundary condition for the rivets

IV. EVALUATION OF SIF BY MVCCI METHOD

Modified Virtual Crack Closure Integral (MVCCI) is a finite element based calculation for strain energy release rate. By knowing the strain energy release rate one can calculate the Stress Intensity Factor (SIF). Using this method, the SIF will be calculated as follows:

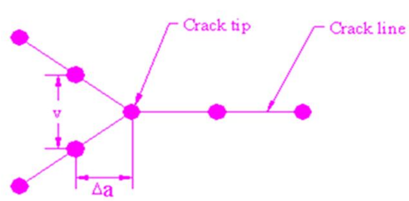

Figure: 4.2 MVCCI method illustration

Stress Intensity Factor (SIF) K = $\sqrt{}$ (G.E)

The strain energy release rate, "G" can be calculated by the formula:

G = $\frac{F \times \Delta u}{2 \times \Delta a \times t}$ N/mm

Where,

E is the Young's modulus

G = Strain energy release rate,
Δa = Edge length of elemental at crack tip,
Δv = differential displacement of crack opening node,
F = grid point force at crack tip,
t = thickness of plate at crack tip
Element edge length = Δa = 2.5mm
Thickness at the tip of the crack = t =2
Total force = F = 200 N
From equation 3.4 we have

$$G = \frac{F \times \Delta u}{2 \times \Delta a \times t}$$

$$= \frac{200 \times 0.00534}{2 \times 2.5 \times 2}$$

$$= 0.1034 \text{N/mm}$$

$$K = \sqrt{(G \times E)}$$

$$= 0.1034 \times 72$$

$$= 2.88 \text{ MPa}\sqrt{\text{m}}.$$

Therefore the Stress Intensity Factor (SIF) in the presence of 5mm crack is 2.88 MPa√m.
Similarly the SIF calculations for the incremental crack lengths can be calculated.

Figure:4.3 SIF calculation from FEA

10mm crack						
Displaced nodes	Displacement of displaced nodes T2 mm	Differential displacement Δu Mm	GP Force at node and elements		GP Force N	Total force F N
			Node	Element		
1010984	0.373	0.005345	433833	837136	98.82	200
1010988	0.369			837136	101.30	

Table 4.2: Material properties used for the analysis

A. *Residual Strength Estimation*

Amount of strength remaining in the panel after attaining a particular crack length should be estimated, For a crack length of 50mm

1) Residual strength of skin = $\sigma_{skin} = \frac{K_{IC} \times \sigma}{K_a}$

$$= \frac{103 \times 30.16}{3.128}$$

$$= 993 \text{MPa}$$

2) Residual strength in the stringer = $\sigma_{stiffener} = \frac{\sigma_{applied}}{\sigma_{stiffener}} \times \sigma_{ultimate}$

$$= \frac{20.48 \times 483}{33.4}$$

$$= 437 \text{MPa}$$

B. *Fatigue life Calculation*

Fatigue life calculation for an initial crack length of 10mm

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From Paris' law we have

$$\frac{da}{dN} = C\left(\Delta K\right)^m$$

Integration of above equation will gives the following simple expression for the number of cycles, N_{if}. From equation 3.9 we have

$$N_{if} = \frac{{A_f}^{1-m/2} - {A_i}^{1-m/2}}{C\left(F\Delta\sigma\sqrt{\Pi}\right)^m\left(1-\frac{m}{2}\right)}$$

Cycles needed for a crack to grow from its intial length A_i to A_f.

A_i = initial crack length = 5mm

A_f = final crack length = 10mm

C = material constant = 3xE-8

F = correction factor = 1.01

$\Delta\sigma$ = applied stress = 30.16 MPa

M = material constant = 3.2

$$N_{if} = \frac{10^{1-3.2/2} - 5^{1-3.2/2}}{3\times E-8\left(1\times 20.5\sqrt{\Pi}\right)^{3.2}\left(1-\frac{3.2}{2}\right)}$$

N_{if}=11913cycles

Upon substituting the above values for the equation given above, number of life cycles required to grow a crack from 5mm to 10mm is 11913 cycles.

V. RESULTS AND DISCUSSION

Linear static stress analysis of the window cutout panel is carried out as a global analysis and local analysis takes place near the rivet holes. Fracture mechanics approach is carried out to get required output from local configuration. Fatigue life initiation and propagation are predicted.

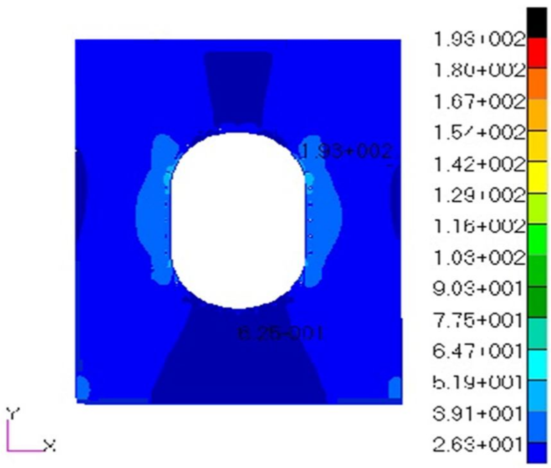

Figure:7.1 Stress contour for the window cutout panel

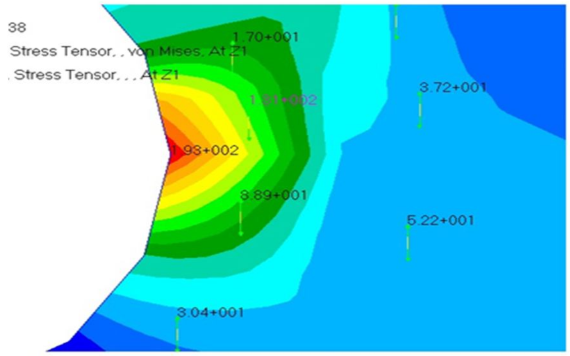

Figure:7.2 Max tensile stress nearby rivet hole

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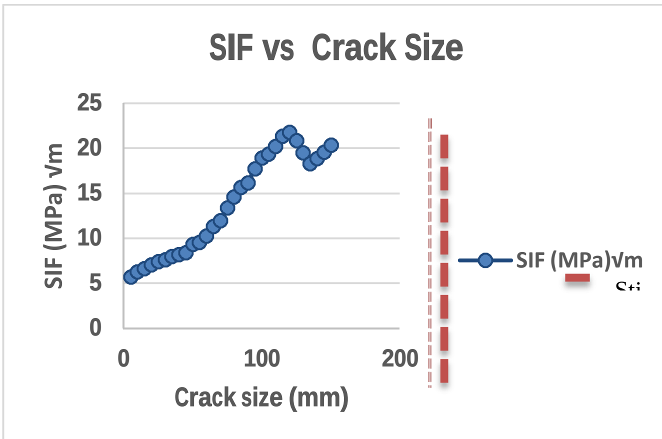
VI. STRESS ON THE PLATE

Once the loads and boundary conditions are meticulously imparted on the model, it is subjected to stress analysis. The main objective are to find out the magnitude of maximum stress and the stress concentration regions, here the load applied is 9Psi. Stress analysis of fuselage structure shows the maximum stress locations for a load case of 30.16 (9Psi). The contour plots of specific result over the model. The counter shows that maximum stress at holes provided for beading in the window cutout region which is greater than other part of the fuselage and its value is 193 MPa. Which is less than the yield stress of the aluminum 2024 T351 alloy i.e. 345 MPa as shown before.

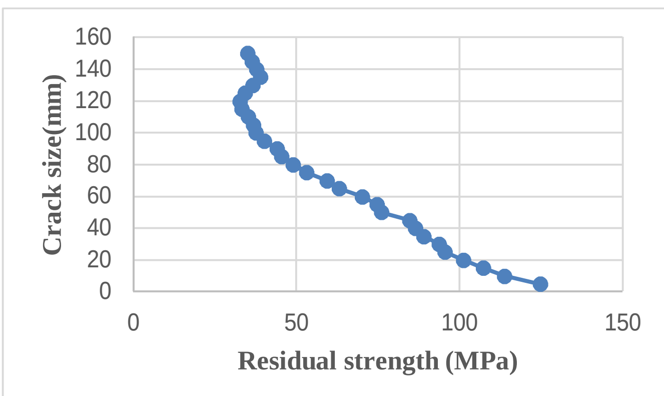
The maximum stress at rivet hole cut outs and found magnitude of maximum tensile stress is 193MPa and element stress as far field stress is 131MPa concerning to the maximum tensile location element. This stresses are uniform in all the part of cut outs. The maximum tensile stress locations are the probable locations for crack initiation. Above shows close up view of stress contour at rivet hole cut out.

A. *SIF Evaluations of Various Crack Lengths*

Stress analysis will be carried out for the window cutout panel In case of a window cutout panel the maximum stress distribution location was found in the skin at the riveted location. The maximum stress obtained in the structure, material used for the structure, ultimate strength of the material. If maximum stress is less than the tensile yield strength of the material. As per stress-strain diagram of respective material, structure will not fail for applied load.



Graph: 8.1 Crack size Vs SIF

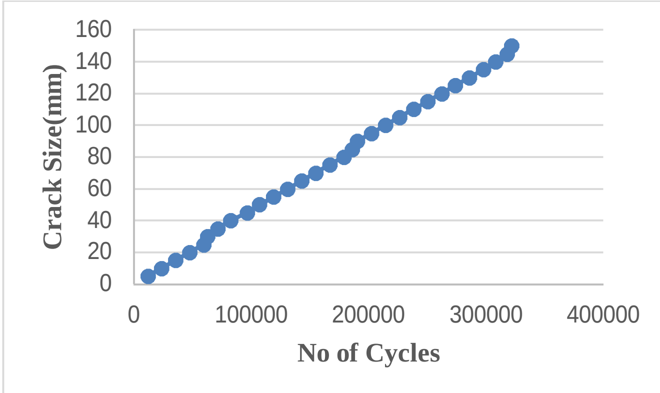


Graph: 8.2 Skin Residual Strength Vs Crack size

B. *Life Cycles Calculations Of A Window Cutout Panel*

Number of life cycles required to attain a particular crack length were calculated and are tabulated. Graph of number of cycles required to attain that crack length was plotted for a window cutout panel. In case of a window cutout panel, once it crosses the stiffener there is a small increment in the number of cycles required to attain the further crack lengths. In case of a window cutout panel with tear strap once it reaches near the stiffener position there is an increment in the life of a panel and it is clear from the graph

For the evaluation of no of cycles for a window cutout panel python programming language was used to evaluate easily. So the crack size from 5mm to 150mm there are 30 segments of cracks, after sectoring the range of a crack size then the SIF values was also done in the same way, so it will be easy for evaluating the cycles and plotting the graph.



Graph: 8.3 No of Cycles Vs Crack Sizes

VII. CONCLUSION

By using MSC-Nastran and MSC-Patran the stress analysis was carried. Nearer to the fuselage window corner location the maximum stress was observed. To come up with the gradient stress field nearer to the cutout, finer mesh was done by screening a thorough literature survey. At the cutout region 193MPa maximum stress was seen. And this maximum stress is obtained in the presence crack in that cutout location. Stress intensity factor was calculated by using MVCCI method. For various crack length in incremental wise SIF was calculated. According to crack length function the SIF too increases. In stiffener location SIF decreases with the crack length increase. With the above sequential the load get transferred from skin to stiffeners. So crack growth is getting arrested. And also catastrophic failure check was done by indulging SIF value and fracture toughness for each crack in the specimen area. By calculating residual strength for both skin and stiffener region it decreases as crack length propagates and vice versa at stiffener region. With this the fetch result is the stiffener raise its role in long crack on the skin region.

VIII. FUTURE SCOPE

A. Same work can be carried out for 3D model
B. More structural members can be introduced in the panel for the analysis
C. Alteration can be done by varying different material properties

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