Damage tolerance evaluation of a stiffened panel with a passenger window cutout of a transport airframe fuselage structure

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***Abstract*- Aircraft is a complex mechanical structure and must be designed with a very high structural safety also with lightweight. 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. Normally fatigue cracks will initiate from stress concentration locations. Window cutout in the fuselage structure is a common functional requirement in the structure. Fatigue cracks will initiate from corners of window cutout since they are the stress concentration regions. To ensure safety of an aircraft during its entire economic service life, fatigue and damage tolerance analysis, testing and service experience correlation play a vital role. Current study includes the crack arrest capability evaluation of the bulkheads in the stiffened panel consisting of a large cutout for passenger window. The passenger window cutout region in the fuselage is a critical region 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. If the crack in a critical location goes unnoticed it could lead to a catastrophic failure of the airframe. Fatigue is a phenomenon by which the load carrying ability of a structure decreases when subjected to fluctuating loads. In a metallic structure fatigue manifests itself in the form of a crack, which propagates. A crack from the highest tensile stress location will be initiated in the analytical model of the stiffened panel. Panel will be analyzed for different crack lengths by growing the crack in the direction perpendicular to the loading direction. Stress intensity factor calculations will be carried out for different crack lengths. Crack arrest capability of the bulkheads on either sides of the window cutout will be evaluated analytically by comparing the stress intensity factor values with fracture toughness of the material at different crack lengths. The residual strength diagram will be plotted for both fuselage skin and bulkheads to demonstrate the crack arrest capability of the bulkheads through analytical approach.**

Keywords: Aircraft, Fuselage, Window cutout, stress analysis, FEM, Fatigue, Damage tolerance, Stress intensity factor, Crack arrest capability.

I.INTRODUCTION

Aircraft structure is the most obvious example where structural efficiency results in light weight and high operating stresses. An efficient structure must have three primary attributes: namely, the ability to perform its intended function, adequate service life and the capability of being produced at a reasonable cost. The major part of the aircraft structure consists of built-up panels of sheets and stringers, e.g. wing and fuselage skin panels, spar webs and stiffeners. Despite all precautions, cracks have arisen in many of these structural elements. These cracks reduce the stiffness and the total load-carrying capacity of the structure. The fuselage is the main structure in the aircraft that holds crew, passengers and cargo. An aircraft fuselage structure must be capable of withstanding many types of loads and stresses, and at the same time with low weight. The principal source of the stresses in this structure is the internal pressure in high altitude caused by difference of cabin pressurization and reduction of the outside pressure with increase in altitude, but the structure is subjected to other loads, as bending, torsion, thermal loads, etc. In this paper, the effect of internal pressure when the fuselage presents a crack is analyzed. The traditional aircraft fuselage is composed of the skin consisting of a cylindrical shell typically 2 mm thick, circular frames and axial stringers, and normally these components are manufactured with an aluminum alloy and are connected by rivets. The skin of fuselage is to carry cabin pressure and shear loads, longitudinal stringers to carry the longitudinal tension and compression loads due to bending, circumferential frames to maintain the fuselage shape and redistribute loads into the skin, and bulkheads to carry concentrated loads including those associated with pressurization of the fuselage.

II.LITERATURE REVIEW

Lots of research has been done in aviation field and some of them related to damage tolerance analysis are used as the reference in this work as follows:

Thomas Swift (1970)- Development of the Fail-safe Design Features of Aircraft Structures; Self-propagating crack arrested in low stress region at the crack tip near the Frames. Design included the capability to arrest a crack after a fast fracture has occurred. U.S. certification standards (14 CFR 25.271), crack arrest capability in the pressurized fuselage structure.

Hellene (1973) and Parks (1974) – Crack-propagation, using the stress intensity factor. Virtual crack extension (VCE) method was proposed which leads to increase the accuracy of SIF results. VCE requires only one complete analysis of a given structure to calculate SIF. Both the COD and VCE methods can be used to calculate SIF for all three fracture modes. The equivalent domain integral method which can be applied to both linear and nonlinear problems renders mode separation possible.

Newman, J. C. Jr (1974) – 2D, Non-linear FEA using an incremental theory of plasticity to predict crack closure and crack opening stresses, during the crack growth process under cyclic loading. A 2D finite element computer program has been developed, linear and nonlinear, for elastic-plastic material behavior in presence of crack under cyclic loading. Useful to study crack extension and crack closure behavior in a Centre cracked panel under constant amplitude and two level block loading. Calculated results were quantitatively consistent with experimental measurements. Therefore the finite element analysis performed gives further insight to mechanisms of fatigue crack growth during cyclic loading.

Rybicki and Kanninen (1977)– Proposed VCCT (virtual crack-closure technique). Accurate and efficient method for mixed mode energy release rate and relatively easy algorithm of application capability to calculate SIF for all three fracture modes. Stresses are higher in the vicinity of the crack tip, which are characterized by the parameter stress intensity factor.

Radon and Leevers (1985) - For a stationary crack in an elastic material, a compact tension specimen, C(T), will have higher constraint than a middle crack tension specimen, M(T). Thus, during the initial stages of loading, it is reasonable to expect that differences in crack initiation and growth may occur in the two crack configurations. However, during stable crack growth, large scale plastic deformation occurs and the initial elastic constraint differences are reduced.

Richard D. Young, Marshall Rouse(1999)- Experimental test for the residual strength of stringers and stiffened panel with cracks is done. Two types of damage are considered: a longitudinal crack midway between stringers and a longitudinal crack adjacent to a stringer and along a row of fasteners in a lap joint that has multiple-site damage (MSD). After the analyses it is concluded that the presence of MSD affects crack growth stability and reduces the residual strength of stiffened fuselage shells with long cracks. **Shamsuzuha Habeeb and K.S. Raju (2009)-** Crack Arrest Capabilities in Adhesively Bonded Skin and Stiffener is studied. Fracture analyses were conducted on stiffened panels with CTOD fracture criteria. SIF was calculated for both the stiffened panel for various crack lengths at same load. Fracture occurs when the SIF reaches a critical value and SIF decreases as the crack grows at stiffener, increasing the load carrying capability of the stiffened panel.

Karthik N (2013)- Analysis of the Fuselage Structure for Multi-Site Damage is done. Two competing mechanism of failure are studied; fracture and net section yielding (plastic collapse). The mode of failure depends on any of the above two occurs at a lower load. The SIF calculations are carried out by using MVCCI method. After comparing Fracture toughness from SIF for particular aluminum properties and from SIF for different crack length, the results reveals that the fuselage panel is not failed by Fracture criteria. The two Validations of results depicts that the mode of failure of fuselage panel is by Net section yielding criteria.

Madhura B. M. (2014) - Damage tolerance analysis of a wing with presence of large landing gear cutout through stress analysis. Cracks initiated from the location of the maximum tensile stress (at the rivet hole) and evaluation is done. Analytically calculated SIF of plate in the presence of crack is compared with the theoretical SIF. Hence the damage tolerance can be done, by using stiffeners in the wing, so the crack gets arrested when it reaches to stiffeners and stress reducers.

III.METHODOLOGIES

The finite element method is a numerical technique for solving engineering problems. It is most powerful analysis tool used to solve simple to complicated problems.

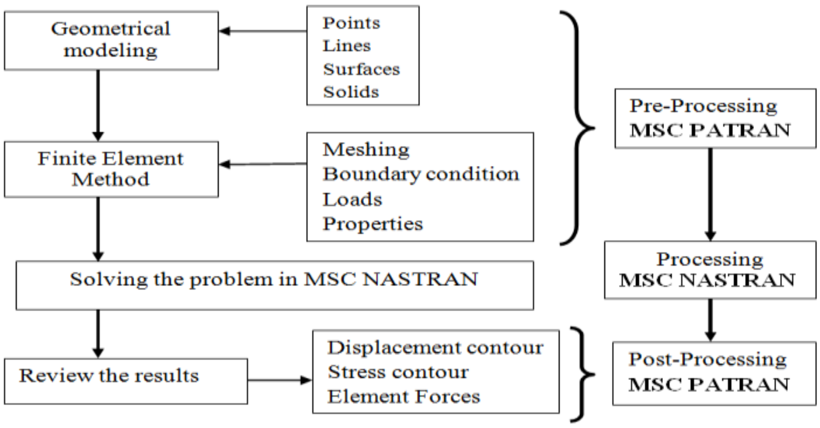

Fig.1: Steps involved in Finite Element Analysis

The pre-processing stage involves the preparation of nodal co-ordinates & its connectivity, meshing the model, load & boundary conditions and material information for finite element models carried in MSC-PATRAN described in Fig.1. The processing stage involves stiffness generation, modification and solution of equations resulting in the evaluation of nodal variables, run in MSC-NASTRAN. The post-processing stage deals with the presentation of results, typically the deformed configurations, elemental stresses and forces etc.

IV. GEOMETRICAL CONFIGURATION

Fuselage has cylindrical panel of radius 1275 mm, length 2500 mm and thickness of skin is 2 mm. It is represented with two window cut out on both sides of the fuselage to do global stress analysis. The fuselage structure has 4 bulkheads and 12 stringers. Bulkheads are provided to support the skin and to prevent the bursting of fuselage in radial direction. The stringers are provided to support the skin in longitudinal direction and to prevent the bending of skin of fuselage structure. A typical fuselage structure with window cut out, bulkheads and stringers is shown below:

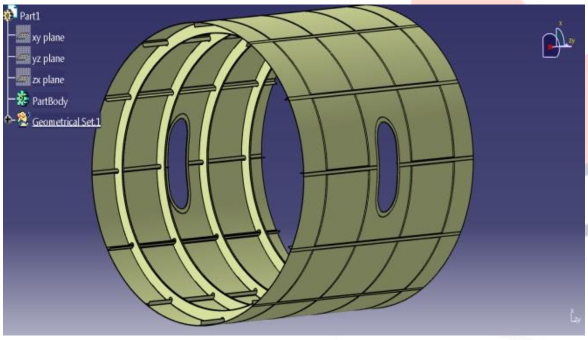

Fig.2: Detailed fuselage CAD model

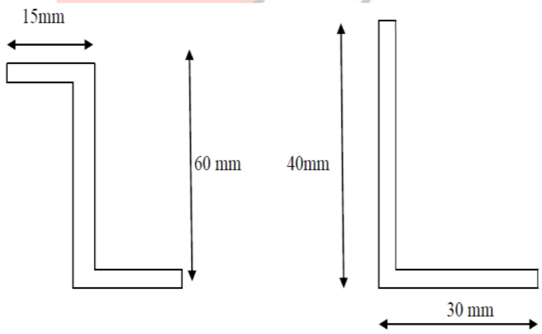

Fig.3: Cross-Section of stringers and bulkheads

V. MATERIAL SPECIFICATION

Selection of aircraft materials depends on any considerations, which can in general be categorized as cost and structural performance. The key material properties those are pertinent to maintenance cost and structural performance are-

- Density
- Young’s modulus
- Ultimate and yield strength
- Fatigue strength
- Damage tolerance (fracture toughness and crack growth)
- Corrosion, etc.

Mechanical properties of the skin, stiffening members and rivets are required for finite element models. There is little information on the material properties of skin, stiffening members and rivet material in the literature. Aluminum 2024 is used for components fuselage and rivet. Table 1 describes few material properties used for analysis.

Properties	Al2024-T3
Density	2770 kg/m3
Ultimate tensile strength	483 MPa
Tensile yield strength	362MPa
Young's modulus	72000MPa
Poisson ratio	0.33

Table 1 material properties used for analysis

VI. Stress analysis

The fuselage is provided with two capsule shaped window cutout and 4 bulkheads and 12 stringers. The boundary conditions applied to structure would be in the form of constraints provided to prevent the translations and rotation in X, Y and Z directions are shown below:

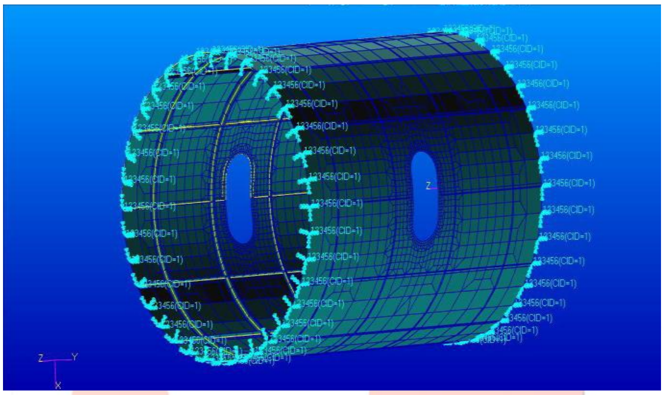

Fig.4: Boundary conditions to fuselage model

6psi differential pressure is applied inside the fuselage structure
As;
6 psi = 6 x 7 x 10-4 kg/mm2
= 42 x 10-4 kg/mm2
= 0.0042 kg/mm2

The hoop stresses are developed in the fuselage structure by applied internal pressurization. The hoop stress (σc) can be related with internal pressure in a thin-walled pressure vessel:
Hoop stress;
(σc) = pd/2t
= 2.6754 kg/mm2

The maximum stress on skin is near the cutout at the rivet location. The tensile stress is uniformly varying from fixed end to loading end. The magnitude of maximum tensile stress is 7.45 kg/mm^2 shown in Fig.5 at rivet location. The maximum stress locations are the probable locations for crack initiation. Skin is the most critical stress locations for the crack initiation. Generally longitudinal crack is initiated from rivet hole.

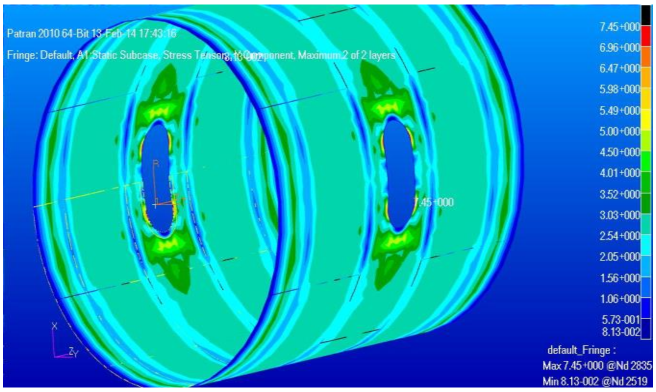

Fig.5 maximum stress near cutout

VII. Local Analysis of Stiffened Panel

The stiffened panel represents a most generic in fuselage structure. The stiffened panel dimensions are 2000 mm in the longitudinal direction and 1200 mm in transverse direction. The thickness of the stiffened panel skin is 2 mm. The stiffened panel has two stringers of L-section and two stringers of Z-section along the longitudinal direction of plate, they are attached to the skin by row of rivets, 3 mm diameter placed at a pitch of 40 mm as shown in figure 6.

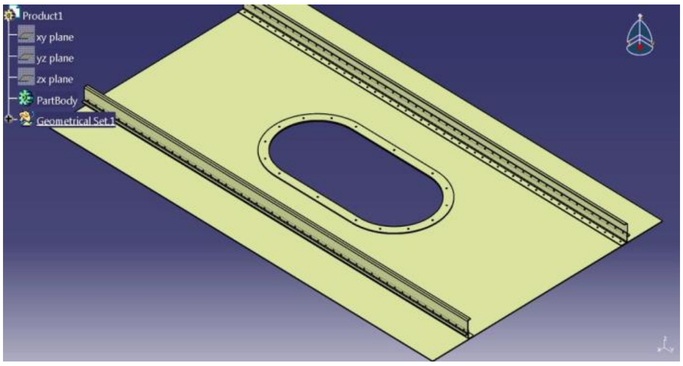

Fig.6: CAD model of stiffened panel

The components of the stiffened panel are meshed by four and three node shell elements. Skin of the stiffened panel is meshed by shell elements with aspect ratio maximum up to 5. L section and Z section stringers of the stiffened panel are meshed by 4 noded shell elements. Fine mesh is carried at the cutout portion and rivet holes on the cutout of skin to get accurate results. The rivets are placed on the skin to hold the stringers. Riveting is carried out by selecting the node on the skin and the corresponding node on the other component (stringer). Rivets are stimulated by using beam elements indicated in yellow color shown in Figure 8. Aspect ratio should be less than 5 in all components of the stiffened panel. Meshing is checked for any duplicate nodes and elements.

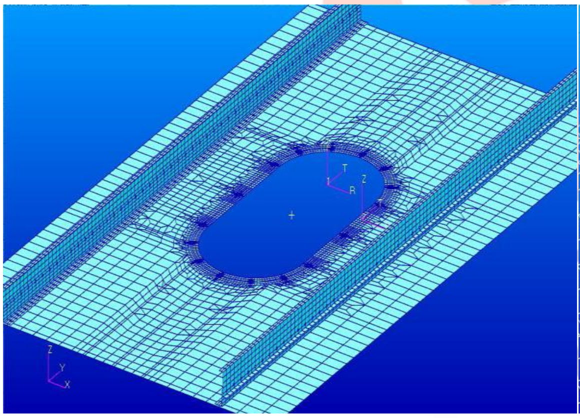

Fig.7: Finite element model of stiffened panel

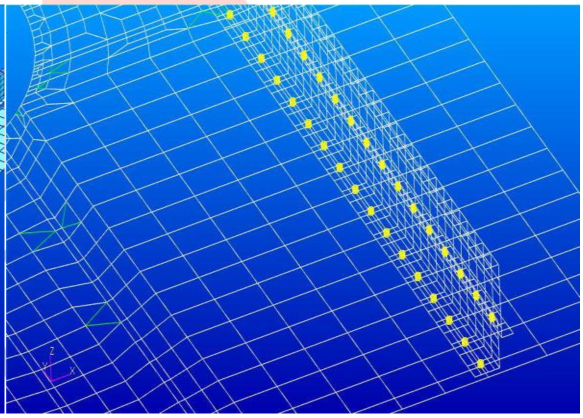

Fig.8: stiffened panel with rivets on stringers and skin

VIII. Load and Boundary Conditions

For the stiffened panel analysis, the cabin pressure is acting tensile in nature. The radial hoop stresses developed in the fuselage cylindrical shell are equals to tensile stresses of the stiffened panel. The hoop stress developed in the model and corresponding cross sectional area gives the tensile load. This tensile stress is uniformly distributed over the cross section. Uniformly distributed tensile load is applied on the stiffened panel in transverse axial direction. Uniformly distributed load is applied on edges of skin and stringers in the transverse direction. At other end, all the edge nodes of stiffened panel are constrained in all six degree of freedom (three translations and three rotations) as shown in Figure 9.

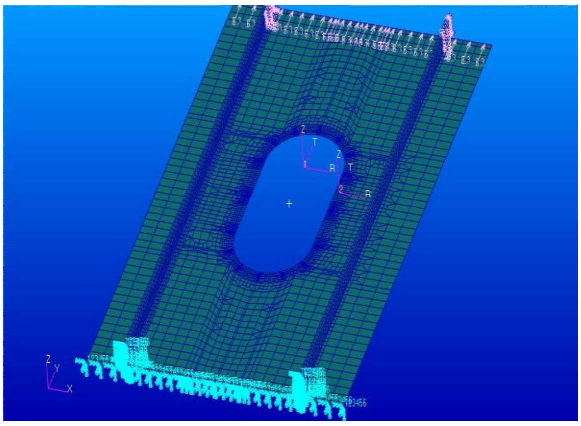

Fig.9: loads and boundary condition over stiffened panel

Uniform distributed load of 5.35Kg/mm^2 is applied throughout the edge of the stiffened panel and its components.

IX. STRESS CONTOUR OF STIFFENED PANEL

The maximum stress on skin is at the rivet location in the cutout where the rivets are used to fasten the window frame on the skin. The tensile stress is uniformly varying from fixed end to loading end. The magnitude of maximum stress is 9.39 kg/mm^2 shown in Figure 10 at rivet location. The maximum stress locations are the probable locations for crack initiation. Skin is the most critical stress locations for the crack initiation. Generally longitudinal crack is initiated from rivet hole.

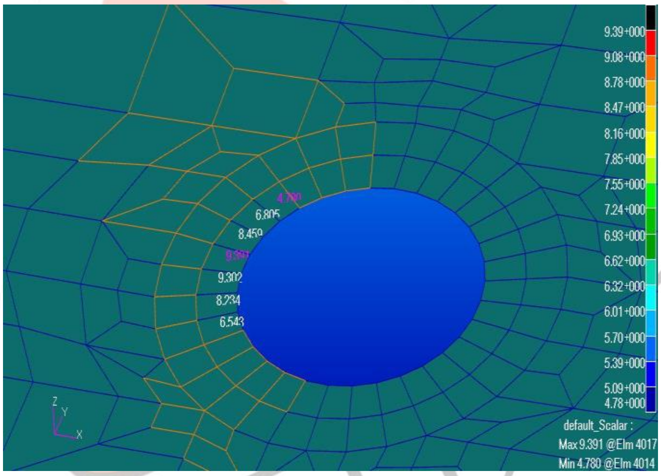

Fig.10: maximum stress location

X. CRACK INITIATION IN STIFFENED PANEL

From the stress analysis of the stiffened panel, cracks are initiated from the maximum tensile stress location. There are two structural elements at the rivet location near the high stress location. Even though maximum stresses are found on rivet hole at window cutout on the skin, cracks are initiated in perpendicular to the loading direction. Rivet hole is the most critical stress locations for crack initiation. So, the maximum tensile stress location on stiffened panel is at rivet hole on the cutout. Crack initiation period is studied by using stress concentration factor. Longitudinal crack is initiated from rivet location, which is perpendicular to loading direction. The crack is propagating as a function of number of fatigue cycles due internal pressurization of fuselage. The first approximation of the stiffened panel with a crack of 10mm crack length from both sides of rivet hole (maximum tensile stress location) is considered and crack length increases in longitudinal direction with number of fatigue cycles. Crack initiation period is studied by using stress concentration factor, which does not play much important role.

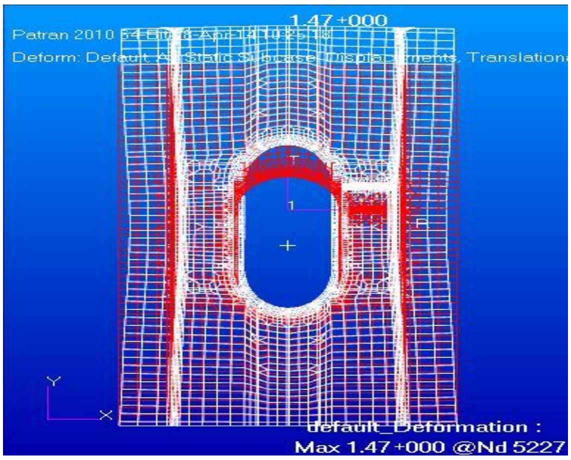

Fig.11: 10mm crack at rivet hole in stiffened panel

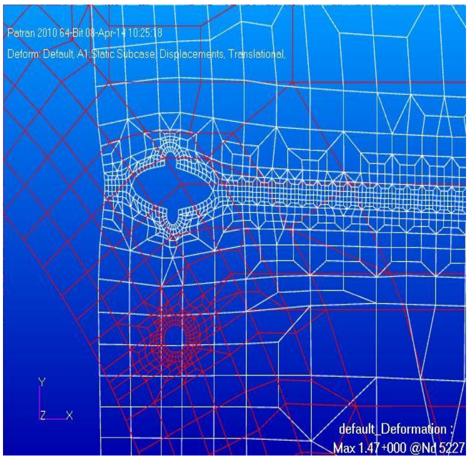

Fig.12: close up of crack at rivet hole

XI.CRACK PROPAGATION

Crack propagation stage is studied by using stress intensity factor approach. The stress intensity factor plays major role in crack growth period, which is determined by using modified virtual crack closure integral (MVCCI) method. Cracks propagate due to sharpness of crack tip. The skin is meshed by four and three node shell elements. Fine meshing is carried out near the crack up to crack length of 180 mm to get crack propagation results. For mesh continuity from fine mesh to coarse mesh different four node and three node shell elements are used. The elemental edge length 1 mm is maintained at crack region. Finer meshing is done in the region of maximum stress, up to the stiffener because this is the expected region of the crack initiation and propagation as shown in the figure below.

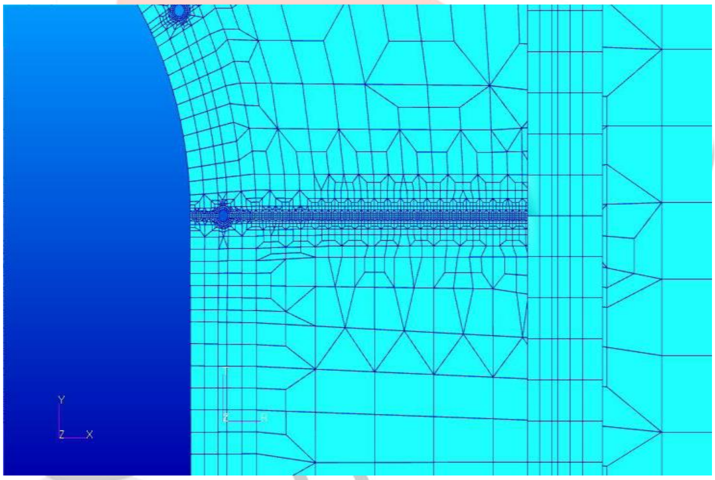

Fig.13 Fine to coarse mesh elements for crack propagation

XII.MVCCI METHOD FOR SIF CALCULATION

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

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

Where, G is the strain energy release rate and can be calculated by;

$$G = \frac{1}{2\Delta a} \times \Delta V \times \frac{F}{t}$$

And,
F is the force at the crack tip in Kg,
Δa is the elemental edge length near the crack tip in mm,
T is the thickness in mm,
ΔV is the crack opening displacement.
The stress intensity factor value at the crack tip can be calculated as follows:

- Force at the crack tip is calculated by means of adding two elemental forces above the crack tip.

- Crack opening displacement is calculated by means of subtracting the two elemental displacement values at the crack tip.

A linear static stress analysis is performed for the stiffened panel for various crack lengths keeping the same loading condition. Fig.13 shows the stress contour for the stiffened panel skin crack. Orientation of crack is in longitudinal direction and crack widens due to loading in transverse direction. The stresses at crack tip are maximum with magnitude 44 kg/mm^2. Energy is stored in material as it is elastically deformed. This energy is released when the crack propagates. This energy helps to creation of new fracture surfaces.

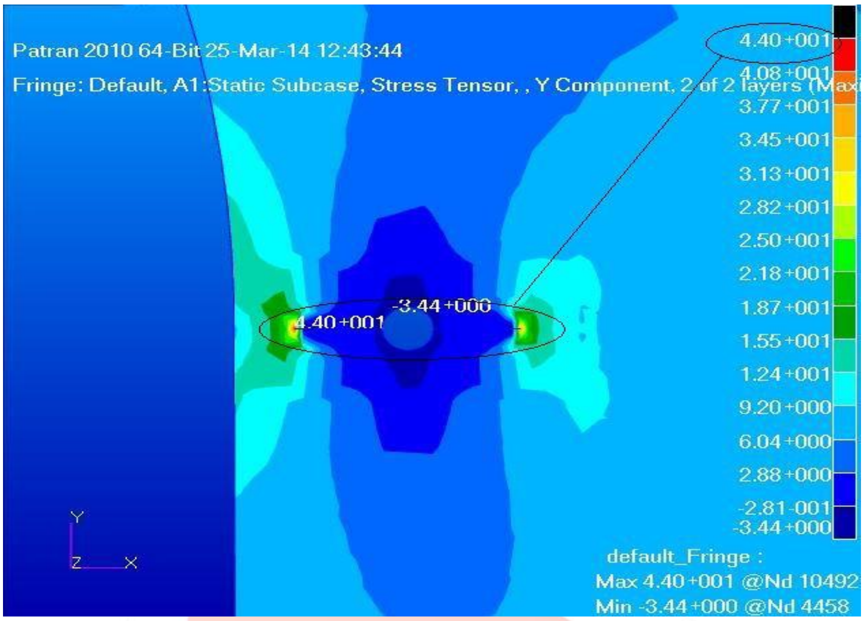

Fig.14: stress contour near crack tip

From MSC Nastran solver,
For example,
Crack length of 10 mm,

- Crack opening displacement (COD), $\Delta V = 0.0202$ mm,
- Forces at the crack tip opening displacement, F= 167.71 Kg ,
- Element edge length at the crack tip, $\Delta a = 1$mm,
- Thickness of the skin and cutout, t_{skin}= 2 mm and t_{cutout} = 5 mm,

Strain energy release rate, G=0.338774 Kg/mm,
Stress intensity factor,
KI FEA =48.6972 Kg/mm$^2\sqrt{mm}$,
Considering Correction factor of 0.3162,
so, stress intensity factor,
K_{IFEA}= 48.6972 X 0.3162 = 15.09614 MPa$\sqrt{m}$.
The above calculations are carried out for different crack length ranging from 10 mm to 180 mm considering a known load. The stress intensity factor value is calculated by using MVCCI method for the stiffened panel. The stress intensity factor is tabulated in steps of 10 mm crack length shown in Table below:

Table 1: SIF values for stiffened panel

ΔC (mm)	ΔV (mm)	F(Kg)	t (mm)	G(Kg/mm)	K (MPa √m)
10	0.0202	167.71	5	0.338774	15.09614
20	0.0676	209	5	1.41284	30.82884
30	0.062358	194.1	2	3.025922	45.11693
40	0.0621	193.149	2	2.998638	44.91307
50	0.06295	1958.122	2	3.070732	45.44977
60	0.0641	196.9	2	3.155323	46.07153
70	0.066	200.8	2	3.25798	46.81499
80	0.066	204.064	2	3.367056	47.59221
90	0.06691	206.72	2	3.457909	48.23002
100	0.06798	209.9	2	3.567251	48.98662
110	0.06885	212.6	2	3.659378	49.61515

120	0.06975	215.41	2	3.756212	50.26732
130	0.07038	217.27	2	3.822866	50.71136
140	0.070912	218.92	2	3.882077	51.10258
150	0.07117	219.71	2	3.90919	51.28072
160	0.07125	220.11	2	3.920709	51.35622
170	0.07082	218.53	2	3.869074	51.01692
180	0.0692	213.32	2	3.690436	49.82526

Now, from the given table it is easy to conclude that, the stress intensity factor increases as the crack grows. The stress intensity factor is 15.09614MPa√m at crack length of 10 mm and increases to 49.82526MPa√m at crack length of 180 mm. The maximum stress intensity factor is 51.35622MPa√m at crack length of 160 mm, which is less than fracture toughness of the material.

Condition for crack propagation,

$$\mathbf{K_{I\,FEA} \geq K_{IC}}$$

Here, $K_{I\,FEA}$ is the stress intensity factor for AL-2024 stiffened panel and K_{IC} is the fracture toughness of the AL-2024 material.
Fracture toughness of Al for 2mm thickness is 105.97MPa√m and for 5 mm thickness 98.97MPa√m.

This indicates that fuselage is safe when only internal pressure acting on fuselage. But in reality, the fuselage of the aircraft structure is subjected to different kinds of loads that include aerodynamic loads, landing loads, taxing loads, pressurization and reaction loads. Due to all this loads, the value of stress intensity factor may exceed fracture toughness of the material. Then crack rapidly propagates and finally leads to catastrophic failure of the fuselage structure. This rapid failure is reduced by using damage tolerance concept. This design philosophy recommends use of bulkheads and stringers. Crack arrest in a fuselage structure is achieved by increasing the material resistance to crack propagation by providing strip of materials of higher toughness in the suitable location.

Residual stress also plays an important role in preventing the crack propagation in structure, thus the residual stresses are needed to be calculated for each crack length in skin.

Basically residual stress is given by the equation:

σ_{res} = (fracture toughness X σ_{remote})/SIF

Table 2: residual stress for different crack length

ΔC (mm)	σ_{remote}	σ_{res}
10	4.459	29.23261
20	4.459	14.31475
30	4.459	10.4337
40	4.459	10.48106
50	4.459	10.35729
60	4.459	10.21752
70	4.459	10.05525
80	4.459	9.891044
90	4.459	9.760241
100	4.459	9.60949
110	4.459	9.48466
120	4.459	9.634665
130	4.459	9.282666
140	4.459	9.21166
150	4.459	9.179603
160	4.459	9.166107
170	4.459	9.227069
180	4.459	9.447751

XIII. CRACK ARREST CAPABILITY

The use of bulkheads and stingers is an effective mean of increasing the residual strength of damaged panels. An unstable fast fracture can be confined to a local area by providing an area of low stress ahead of the crack tip. The crack tip stress is reduced, as a large part of the redistributed load is transferred into the bulkheads. Bulk heads are provided in such a way that they oppose the leading crack. They are provided on the skin in perpendicular direction to the crack propagation direction. In addition, the bulkheads can also be used as bending material to increase the frame stiffness and static strength. The midway bulkheads are installed by riveting, a crack is just as likely to start at a rivet hole and propagate both ways. The possibility of starting a crack at a midway would be reduced if the stiffeners were bonded to the skin.

- *Design and location of bulkheads-*

Bulkheads are principally designed for arresting longitudinal skin cracks. Bulkheads are designed with thickness 2 mm and 30 X 40 and 60 X 15 mm dimensions. The material used for bulkheads is Aluminum 2024-T3. Bulkheads are meshed by four node shell elements with unity aspect ratio.

- *Residual stress analysis in bulkheads-*

  Residual stress for bulkheads can easily be calculated by using the following equation-

  $\sigma_{res} = (\sigma_{UTS} \text{ X } \sigma_{remote})/\sigma_{max}$

Where,
Ultimate tensile strength of Al 2024 = 45 Kg/mm2
σmax = Maximum stress on bulkhead
σrem = Remote stress applied on whole structure i.e. 4.459 Kg/mm2.
Maximum stress on bulkheads for 10 mm crack length is given in the figure 14.

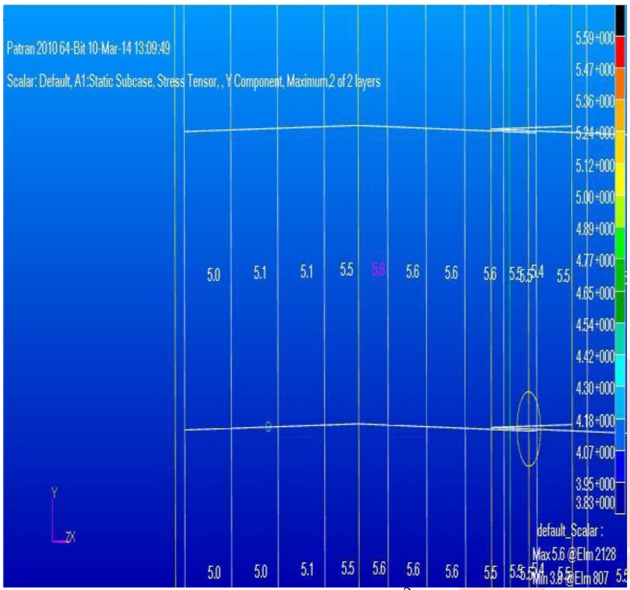

Fig.15 maximum stress (5.6Kg/mm^2) in bulkheads

Table 3 residual stresses for bulkheads

ΔC (mm)	σ_{max}	σ_{res}
10	5.6	35.83
20	5.88	34.125
30	6.01	33.3868
40	6.15	32.62
50	6.38	31.45
60	6.64	30.219
70	6.81	29.464
80	6.95	28.871
90	7.26	27.63
100	7.61	26.36
110	8.01	25.0505
120	8.47	23.69008
130	9.01	22.27026
140	9.64	20.814
150	10.4	19.293
160	11.2	17.915
170	12.3	16.313
180	13.7	14.64635

XIV. RESULTS AND DISCUSSION

a) Study of Crack propagation in stiffened panel-

SIF vs different crack lengths are plotted shown in Fig15. It is observed that, SIF increases gradually with increase in the crack length. When the crack reaches nearer to the stiffeners, the value of SIF decreased. It is found that, the value of SIF 15.096 MPa√m at crack length of 10 mm and increases to 51.35 MPa√m as crack approaches to 160 mm and then decreases to 49.825MPa√m at stiffener location. This plot indicates the stiffener is able to arrest the further crack propagation.

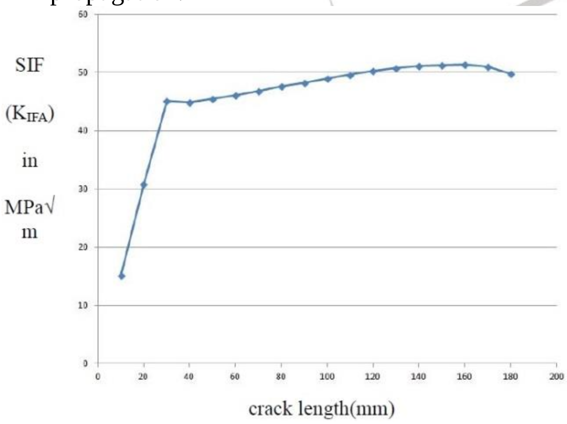

Figure 16: SIF v/s crack length

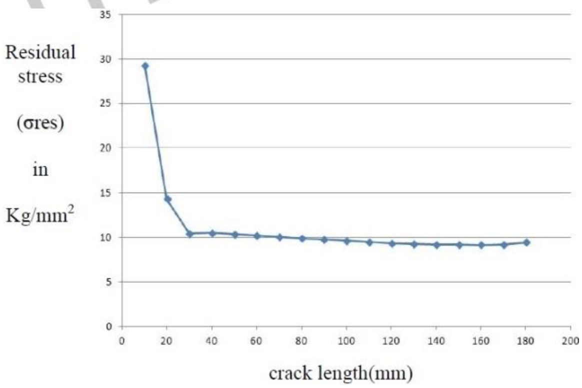

Figure 17: Residual stress v/s crack length

b) Role of stiffener to arrest crack propagation-

Residual strength for bulkheads are found by using analytical calculations and then plotted with crack length as shown below:

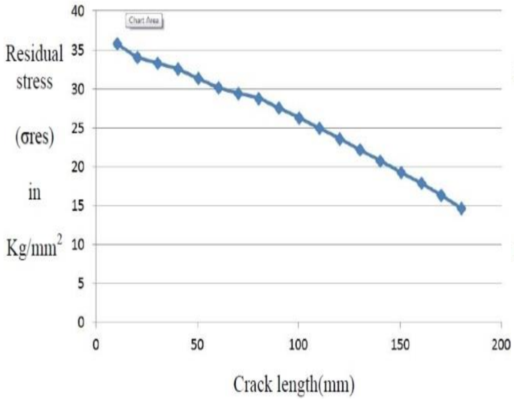

Fig.18: Residual stress v/s crack length

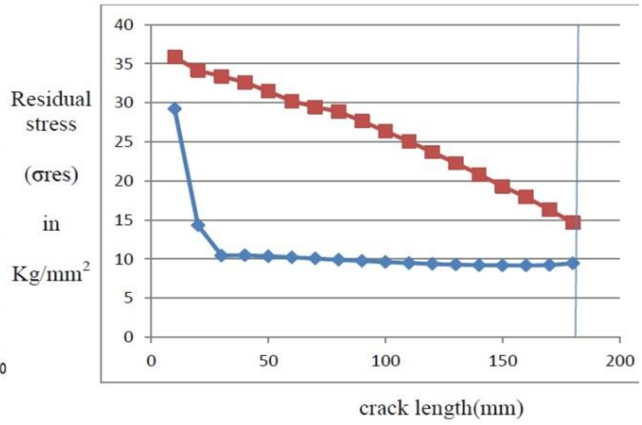

Fig.19: comparison b/w residual strength of bulkheads and skin

c) Comparison –
The structure is safe only when the residual stress of bulkheads is more than the residual stress of skin/cutout material. To prevent the catastrophic failure of fuselage structure the bulkheads are needed to be used. The comparison is made by using the plot between residual stress and crack length as shown in figure 19.

XV.CONCLUSION

The stress analysis using the analysis software MSC-Nastran and MSC-Patran was successfully completed. Maximum stress was observed near the window corner location. A detailed iterative analysis was carried out by converting courser mesh into finer mesh to capture the gradient stress field near the cut out. Maximum stress observed near the cutout region was 7.54 kg/mm2.

The stress analysis of stiffened panel was carried out in presence of crack. Stress intensity factor calculations were carried out using MVCCI method. Stress intensity factor was calculated for different incremental crack lengths. It was observed that the SIF increases as a function of crack length. Near the stiffener location it is seen that the SIF reduces with the increasing crack length. This indicates that the load get transferred from skin to stiffeners which may help in arresting a growing crack. The SIF at every incremental crack length was compared with the fracture toughness of the material to check whether the structure will lead to catastrophic failure or not.

Residual strength calculations were carried out for both skin and bulkheads at different crack length and plotted as a function of crack length. It was observed that the residual strength of skin reduces as the crack length increases. But near to the stiffeners it was observed that the residual strength picks up with increasing crack length. At any given crack length the residual strength of bulkheads was much greater than that of the skin. This indicates that the stiffeners will be intact in presence of long crack on the skin.

XVI.ACKNOWLEDGEMENT

I wish to thank **Prof. Lakshmi Prasad,** SMBS, VIT University, Vellore for guiding and encouraging me throughout the work. I would like to extend many thanks to Program Manager (CAD/CAM) **Dr. R. Vasudevan,** Professor, SMBS, VIT-University, vellore for supporting during course. Also sincere thanks to **K. E. Girish,** Director, Bangalore aircraft industries limited**,** Bangalore for giving all available facilities and technology inputs during work.

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