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"Design and Analysis of Rib Cut Out in Wings or Fuselage of Aircraft"

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***Abstract* - Transport aircraft is a most complex structure. The aircraft fuselage shell is composed of highly stressed skin, circumferential frames, and longitudinal stringers . The skin is connected to the stringers and frame by rivets. Fuselage has more number of riveted joints and is subjected to a major loading of differential internal pressurization. When in the fuselage pressure high and low during each take-off and landing cycle of aircraft, the metal skin of fuselage undergoes expansion and contraction resulting in metal fatigue. Due to presence of so many number of rivet holes, the fuselage skin has a more number of high stress locations and these are locations of potential crack initiation. The large bodied transport aircraft are designed to sustain large fatigue cracks. This paper highlights importance of damage tolerance design of a fuselage structure of transport aircraft. In this paper, the stress intensity factor, quantifying the intensity of the stress field around a crack tip for a longitudinal crack under the pressurization load is studied. The objective of this paper is to find out crack initiation, crack growth, fast fracture and crack arrest features in the stiffened panel. The longitudinal crack is initiated from the rivet hole and stress intensity factor is calculated using modified virtual crack closure integral (MVCCI) method at each stage of crack propagation. In order to stop crack propagation which is capital importance of tear straps are used, which prevent the further crack propagation. In this paper the linear static stress analysis of stiffened panel of a fuselage is performed using MSC NASTRAN solver. The pre-processing of the model is done by using MSC PATRAN software.**

Keywords - Differential internal pressurization, damage tolerance, stiffened panel, longitudinal crack, stress intensity

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 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

Fatigue loads in a pressurized fuselage are mostly due to pressure cycles that occur with each takeoff or landing cycle during flight. The most common fatigue crack orientation in a pressurized fuselage is a longitudinal crack along the direction of maximum hoop stress. Damage tolerant designs use fracture mechanics data and analysis to predict crack growth rates and critical crack lengths [1]. 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. Cabin pressurization is the main source of loading causing longitudinal skin cracks. Two types of damage most frequently associated with the structural integrity of the fuselage are longitudinal cracks under high hoop stresses induced by cabin pressurization and circumferential cracks under stresses from vertical bending of the fuselage. The objective of paper was to present a systematic investigation of the damage tolerance design capability of typical aircraft fuselage structure for longitudinal cracks using linear elastic fracture mechanics [2]. Damage tolerant fuselage is supposed to sustain cracks safely until it is repaired or its economic service life has expired. Strength assessment of the structures is necessary for their in service inspection and repair. Damage tolerance analysis should provide information about the effect of cracks on the strength of the structure. Damage tolerance evaluation must include a determination of the probable locations and modes of the damage due to fatigue, or accidental damage. The aircraft must be capable of successfully completing a flight during which likely structural damage occur as a result of bird impact as specified in Federal Aviation Regulations 25.631 [3].

The crack propagation stage is studied by using stress intensity factor. There are different methods used in the numerical fracture mechanics to calculate stress intensity factors (SIF). The crack opening displacement (COD) method and the force method were popular in early applications of FE to fracture analysis. The virtual crack extension (VCE) methods proposed by Parks [5] and Hellen [4] lead to increased accuracy of stress intensity factor results. The virtual crack extension method 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. However, additional complex numerical procedures have to be applied to get results. The equivalent domain integral method which can be applied to both linear and nonlinear problems renders mode separation possible [6]. The VCCT, originally proposed in 1977 by Rybicki and Kanninen [7], is a very attractive SIF extraction technique because of its good accuracy, a relatively easy algorithm of application capability to calculate SIF for all three fracture modes. Andrzej Leski [8], the implementation of the Virtual Crack Closure Technique in engineering FE calculations. SIF was fundamental quantity that governs the stress field near the crack tip. It depends on the geometrical configuration, crack size and the loading conditions of the body. The stresses are higher in the vicinity of the crack tip, which are characterized by the parameter stress intensity factor. Sethuraman.R and S.K.Maiti [9] have given the mathematical formulae for calculating the stress intensity factor using finite element software tool called as modified virtual crack closure integral technique for mode I. Thomas Swift [10] focused on Development of the Fail-safe Design Features of Aircraft Structures. A self propagating crack was arrested in a region of low stress ahead of the crack tip by providing adequate circumferential and longitudinal stiffening. The crack tip stress was reduced as the load is redistributed into the stiffener. The capability to vary the degree of damaged in stiffened structure. The stringer does not play much role in arresting longitudinal cracks in the fuselage structures. But frames were more effective in arresting longitudinal cracks. The radial tension stress due to pressure varies across a longitudinal skin bay and reaches a maximum value midway between frames. Transfer of load from the skin into the doublers causes a high attachment bearing stress which, when combined with the radial tension stress, may cause a fatigue crack in a longitudinal direction. Bending due to transfer of some of the pressure loading into the frame increases the axial tension stress in the stringer flanges locally, causing fatigue cracks in the skin which propagate into the two adjacent skin bays. The design included the capability of arrest a crack after a fast fracture has occurred. For the pressurized fuselage structure in transport aircrafts, the crack arrest capability must be demonstrated in order for the design to comply with U.S. certification standards (14 CFR 25.271). Accepted practice for demonstrating compliance with the standard is a full-scale ground test, in which an explosive charge drives a blade into the fuselage, creating a longitudinal skin crack approximately one frame bay in length. The blade is positioned to create the crack midway between stringers and centered on a frame, which is also cut by the penetrating blade shown in Fig.1. This configuration maximizes the local flexibility of the failing structure, and thus provides a conservative test of the ability of the surrounding structure to arrest a large fracture initiated by foreign object damage [11].

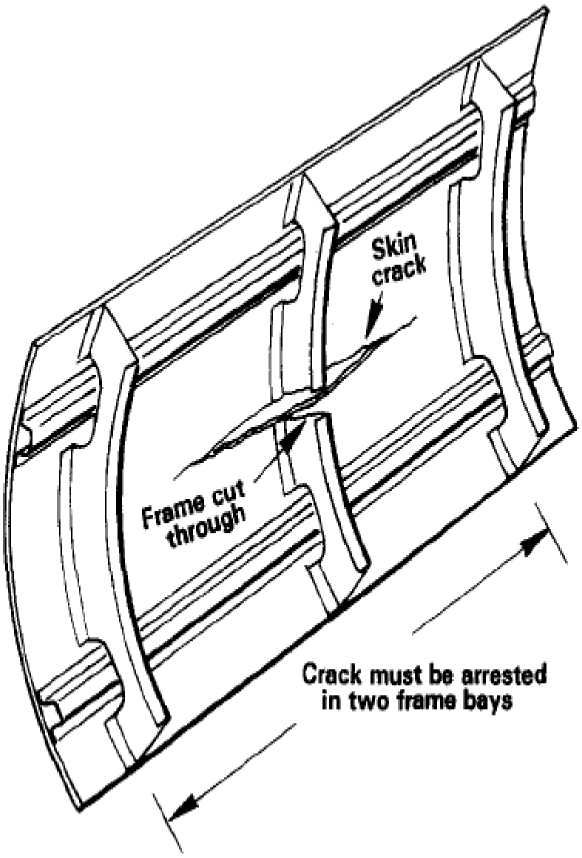

Fig.1 : Demonstration of fuselage crack arrest capability

Shamsuzuha Habeeb and K.S. Raju [12] worked on Crack Arrest Capabilities in Adhesively Bonded Skin and Stiffener. The crack arrest capabilities and the load bearing characteristic of a stiffened panel subjected to uniform remote displacement field. Stringers were usually joined to the skin using rivets. Fracture analyses were conducted on stiffened panels with crack tip opening displacement fracture criteria. A linear elastic static stress analysis was performed and the stress intensity factor was calculated for both the stiffened panel for various crack lengths keeping the same loading condition. Fracture occurs when the stress intensity factor reaches a critical value. Also observed, the stress intensity factor decreases as the crack grows at stiffener, increasing the load carrying capability of the stiffened panel.

III. METHODOLOGY

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.

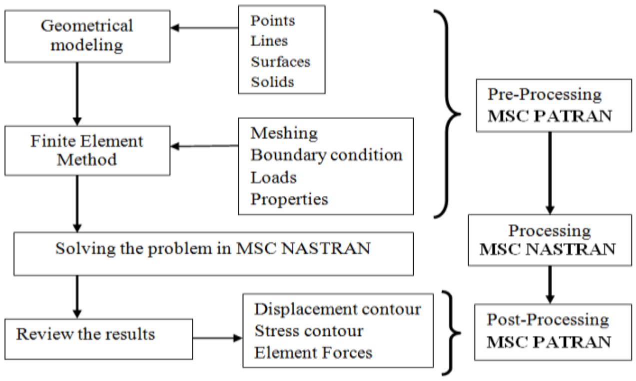

Fig. 2 : Steps involved in Finite Element Analysis.

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.2. 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 2000 mm, length 2500 mm and thickness of skin is 2 mm. It is the sectional cut out of the fuselage to do global stress analysis. The stiffened panel represents a most generic in fuselage structure. The stiffened panel dimensions are 2500 mm in the longitudinal direction and 1750 mm in transverse direction. The thickness of the stiffened panel skin is 2 mm. The stiffened panel has five frames (four bays) with 500 mm spacing and seven stringers (six bays) with 250 mm spacing. The frame has Z & L cross section with 753 mm2 of cross sectional area and stringer has Z cross section with 177 mm2 of cross sectional area. The thicknesses of all flanges of the stiffened panel are 3 mm. The frames and stringers are attached to the skin by row of rivet, 5 mm diameter placed at a pitch of 25 mm shown in Fig. 3 and corresponding cross sections shown in Fig.4. Geometric modeling is carried out using CATIA V5 software as shown in Fig4.

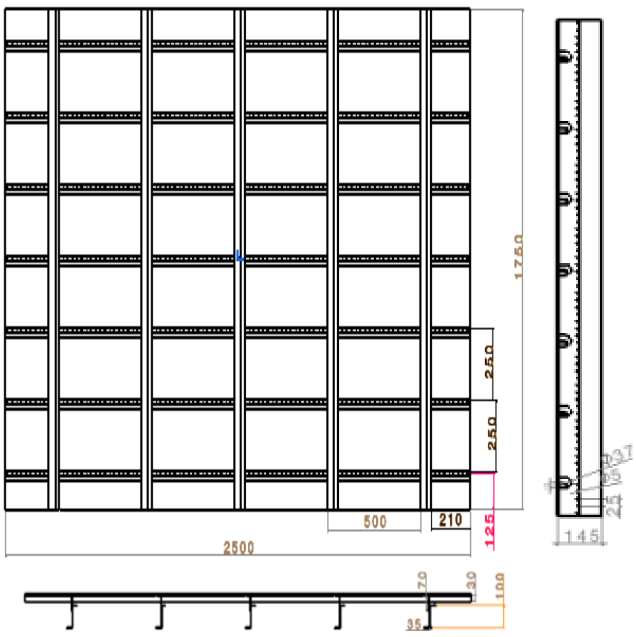

Fig. 3 : Detailed drafting view of the stiffened panel

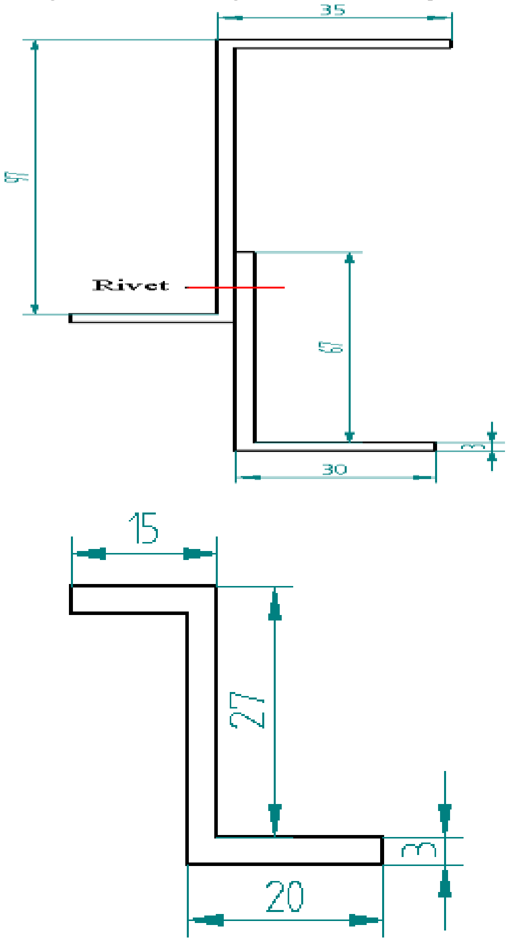

Fig 4 : Cross sections of frame and stringer

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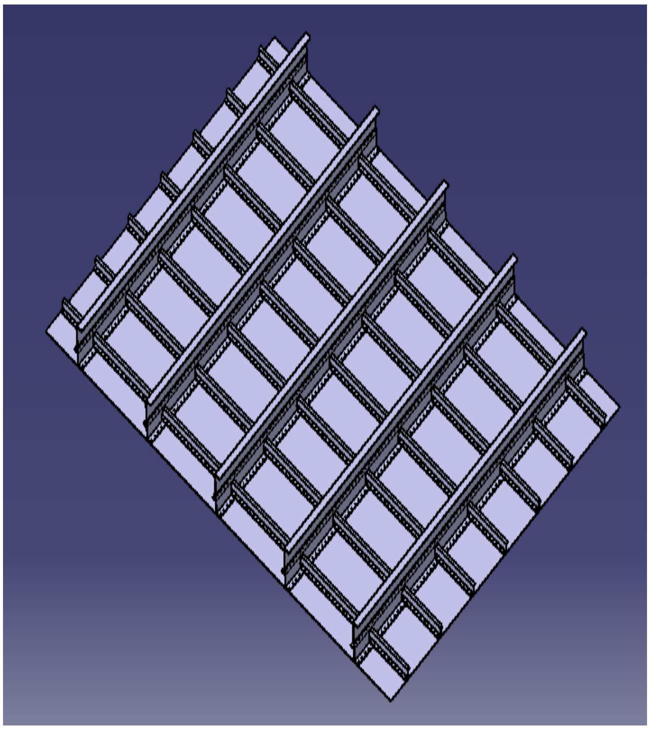

Fig. 5 : CAD model of the stiffened panel

V. MATERIALS USED FOR ANALYSIS

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

Young's modulus

Ultimate and Yield strengths

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-T3 and 2117-T4 is used for components fuselage and rivet respectively. Table 1 describes few material properties used for analysis.

Property	**Aluminium 2024-T3**	**Aluminium 2117-T4**
Density	2.77 g/cm3	2.77 g/cm3
Ultimate Tensile Strength	483 MPa	490 MPa
Tensile Yield Strength	362 MPa	350 MPa
Young's Modulus	72 GPa	71.7 GPa
Poisson's Ratio	0.33	0.33
Fracture Toughness	72.37 MPa√m	76.54MPa√m

VI. STRESS ANALYSIS OF THE STIFFENED PANEL

It involves in pre-processing stage, processing stage and post processing stage. Pre-processing stage involves details of mesh, load & boundary conditions. Pre-processing and post-processing stage is carried in MSC PATRAN.

6.1 Finite element modeling

The components of the stiffened panel are meshed by four noded shell elements. Skin of the stiffened panel is meshed by shell elements with unit aspect ratio. Frame of the stiffened panel is meshed by 4 noded & 3 noded shell elements. Fine mesh is carried at mouse hole of frame to get accurate results. Three noded shell elements are used for the sake of continuity from fine mesh region to the coarse mesh region. The rivets are placed on the skin to hold the frames and stringers. Riveting is carried out by selecting the node on the skin and the corresponding node on the other component. Rivets are stimulated by using beam elements indicated in yellow color shown in Fig.6 & Fig.7. Aspect ratio should be less than 5 in all components of the stiffened panel. Meshing is checked for any duplicate nodes and elements. Table 2 gives number of elements and element type in the stiffened panel.

Table 2 FE model summaries of the stiffened panel

Skin	QUAD4	66810	1.00
Product description	**Type of elements**	**No. of elements**	**Aspect ratio < 5**
Frame	QUAD4,TRIA3 , A3	38424	2.12
Stringer	QUARD	42190	1.01
Rivet	BEAM	1330	--

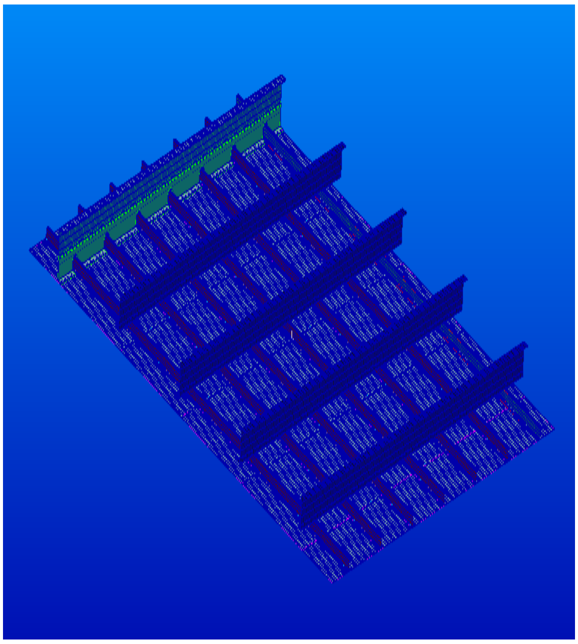

Fig 6 : Finite Element Modeling of the stiffened panel

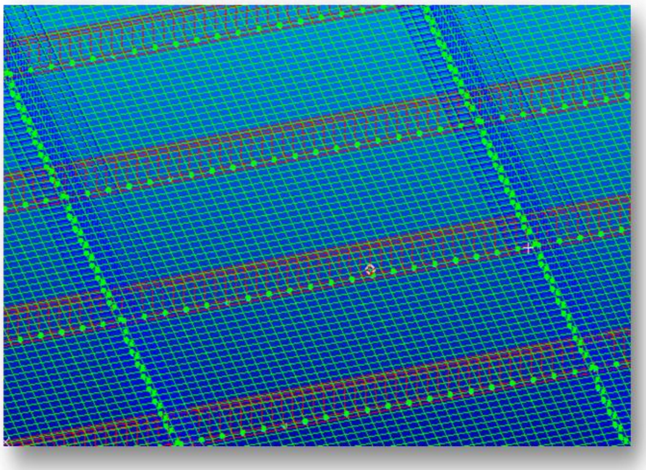

Fig 7 : Close up view of the stiffened panel with rivets

6.2 Load and boundary condition

A differential internal pressure of 6 psi is considered for the current case. The hoop stresses are developed in the fuselage structure by applied internal pressurization. The hoop stress (σh) can be related with internal pressure in a thin-walled pressure vessel: The hoop stress, $\sigma h = p*r/2$ (1)
Where r is the radius of fuselage shell (2000 mm) and t is the thickness of skin (2 mm). After applying these values the hoop stresses are 4.2 kg/mm2.
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 shown in Fig. 8. Uniformly distributed load is applied on edges of skin and frame in the transverse direction. But, the stringers are passing in longitudinal direction in the stiffened panel. Since it is uniaxial tensile loading, the stringers are not subjected to loading. At other end, all the edge nodes of stiffened panel are constrained in all six degree of freedom (three translations and three rotations) shown in Fig.9.

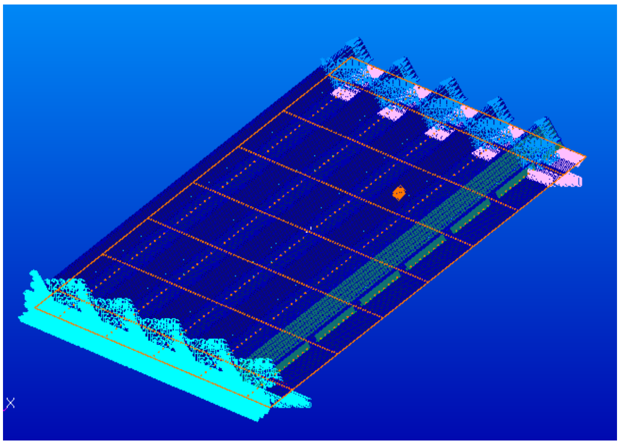

Fig. 8 : Loads and boundary conditions of the stiffened panel

Stress contour for skin and frame

The maximum stress on skin is at the rivet location where the rivets are used to fasten the frames and stringer on the skin. The tensile stress is uniformly varying from fixed end to loading end. The magnitude of maximum tensile stress is 8.19 kg/mm2 shown in Fig.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.

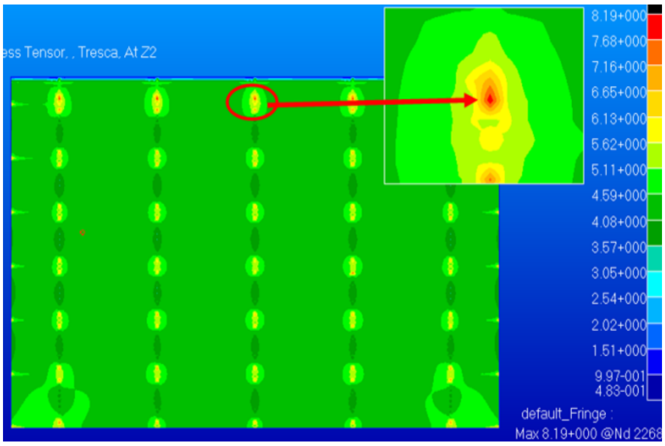

Fig 9 : Stress contour for skin

The maximum stresses are induces at mouse hole cut outs and found magnitude of maximum tensile stress is 13.4 kg/mm2 shown in Fig. 11. This stresses are uniform in all the stringer cut outs. The maximum tensile stress locations are the probable locations for crack initiation. Fig. 12 shows close up view of stress contour at mouse hole cut out

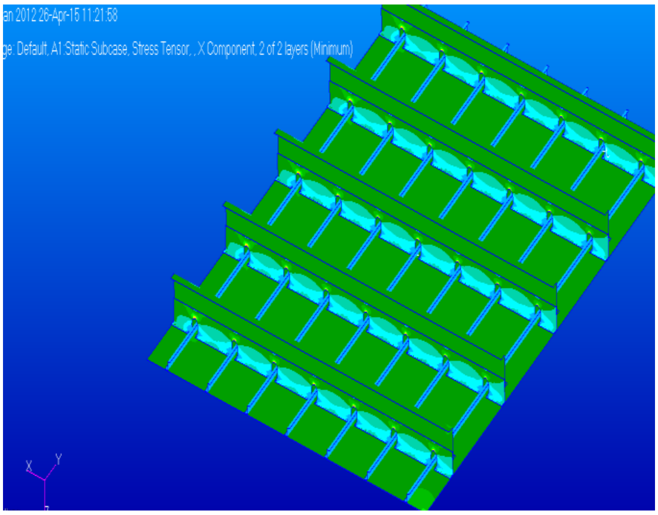

Fig 10 : Stress contour for frame

VII.CRACK INIATION IN THE 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 mouse hole cut outs, cracks are initiated in perpendicular to the loading direction. Skin is the most critical stress locations for crack initiation. So, the maximum tensile stress location on stiffened panel is at skin near the rivet hole. Crack iniation period is studied by using stress concentration factor. Once crack is initiating from rivet location and it linkup with next rivet location, then it become lead crack and finally it leading to catarostrophic failure. Longitudinal crack is initiating from rivet location, which is perpendicular to loading direction. The crack is propagating as a function of number of fatigue cycles due internal pressurization. The first approximation of the stiffened panel with a center longitudinal crack is considered for varying crack length in the skin shown in Fig. 13. Crack iniation period is studied by using stress concentration factor, which does not play much important role.

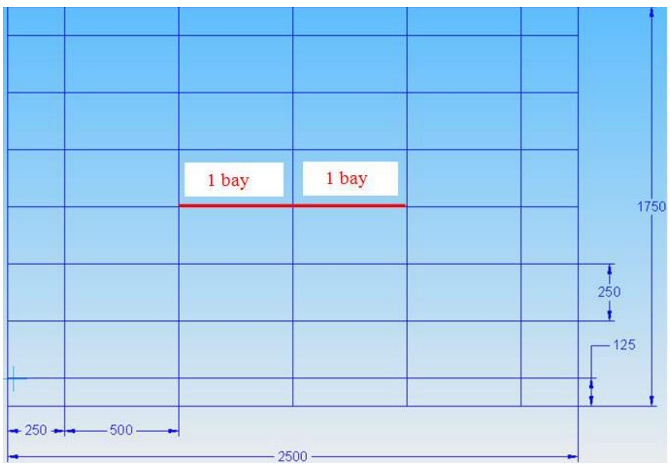

Fig. 11 : Crack iniation in the stiffened panel

VIII. CRACK PROPAGATION IN STIFFENED PANEL

The 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 noded shell elements shown in Fig. 14. Fine meshing is carried out near the crack upto crack length of 1000 mm to get crack propagation results. For mesh continuity from fine mesh to coarse mesh different four noded and three noded shell elements are used. The elemental edge length 1.5625 mm is maintained at crack region.

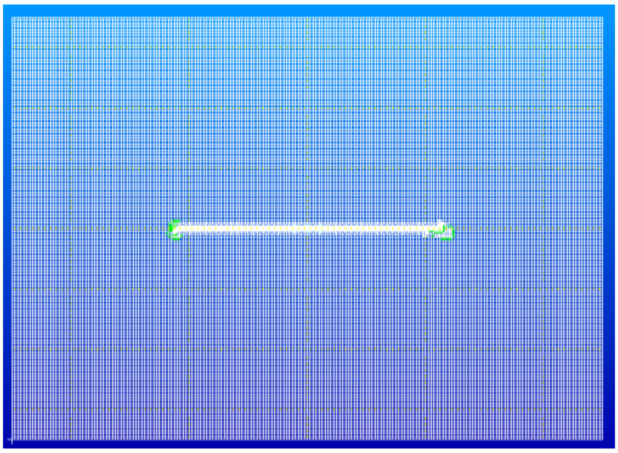

Fig.12 : Finite Element Model of the stiffened panel skin near the crack

MVCCI method for calculation of SIF

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.

$$G_i = \frac{k_i^2}{E}\beta \quad (i=1,2,3)$$

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

$$G = \lim_{\Delta a\to 0}\frac{W}{\Delta a} = \lim_{\Delta a\to 0}\frac{1}{2\Delta a}\int_0^{\Delta a} u(r)\sigma(r-\Delta a)dr,$$

After simplification, modified strain energy energy release rate

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

Where F is forces at the crack tip, u is crack opening displacement (COD), t is thickness of the skin and a is elemental edge length near the crack tip. 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 panels for various crack lengths keeping the same loading condition. Fig. 15 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 and found its magnitude is 30.2 kg/mm2. 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
From MSC NASTRAN solver,

For example, crack length of 100 mm (2a=100 mm, a=50 mm)
Crack opening displacement (COD), Δu= 0.029099 mm

Forces at the crack tip opening displacement, F= 889.2991 N

Elemental edge length at the crack tip, a= 1.5625 mm Thickness of the skin, t = 2 mm Strain energy release rate, G=4.1408 N/mm SIF for mode I loading, KI FEA=16.86 MPa√m

The above calculation is carried for different crack length 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 50 mm crack length shown in Table 3.

Table 3 Stress intensity factor values for the stiffened panel

Crack length 2a in Mm	**COD Δu in mm**	**Forces at crack tip, F in N**	**SIF, $K_{I\,FEA}$ in MPa√m**
50	0.019		
100	0.03	90.12	17.06
150	0.034	104.05	19.51
200	0.035	116.80	20.98
250	0.041	126.96	23.67
300	0.045	137.58	25.81
350	0.045	139.55	26.00
400	0.048	154.17	28.22
450	0.055	162.17	30.98
500	0.056	169.81	31.99
550	0.057	176.81	32.93
600	0.058	183.70	33.86
650	0.062	189.93	35.60
700	0.063	195.65	36.42
750	0.064	200.48	37.16
800	0.065	207.48	38.10
850	0.066	207.47	38.39
900	0.067	209.75	38.89
950	0.069	211.73	39.65
970	0.066	205.69	38.22
100	0.068	208.43	39.06

IX.STUDY OF CRACK PROPAGATION IN THE STIFFENED PANEL

SIF vs different crack lengths are plotted shown in Fig. 18. It is observed that, SIF increases gradually with increase in the crack length. When the crack reaches nearer to the frame, the value of SIF keeps decreasing. It found that, the value of SIF 13.10 MPa√m at crack length of 50 mm and increases to 39.67 MPa√m as crack approaches to 950 mm and then decreases to 39.67 MPa√m at frame location. This plot indicates the frame is able to arrest the further crack propagation.

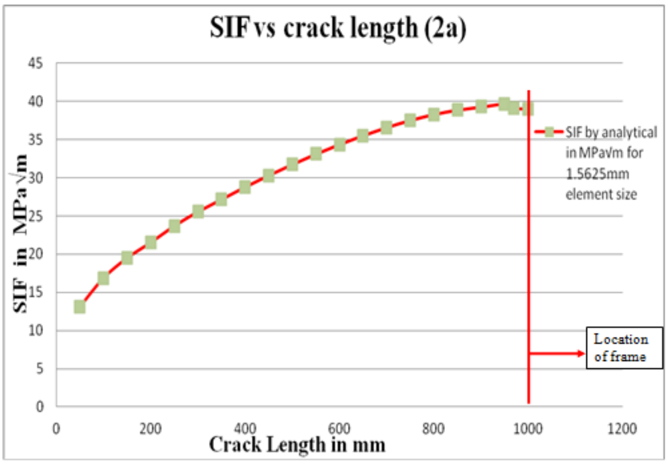

Fig. 13 : Variation of SIF as a function of crack length

X.CONCLUSIONS

Stress analysis of the stiffened panel was carried out and maximum tensile stress was identified at the skin hole. Center longitudinal crack was initiated from rivet hole location of skin. Fatigue crack propagation was estimated by using stress intensity factor approach. Stress intensity factor calculations were carried out for various incremental cracks from 50 mm to 1000 mm. The high value of stress intensity factor 39.65 MPa√m was observed at crack length of 950 mm. The value of stress intensity factor 39.06 MPa√m was observed at frame location. These values are calculated without introducing tear strap. Further work in this paper is have to change material for stiffened panel and calculate SIF and compare SIF after introducing tear strap. also in this paper changing thickness of tear strap for calculating SIF.

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