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Failure design investigation of the role of crack stopper strap and bulk-head in wide fuselage aircraft

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ABSTRACT

The paper describes the static stress & damage tolerance calculations performed on a center body fuselage panel of a medium passenger airplane subjected to pressure loading. Stress calculations of each and every component, if performed manually will become herculean task for the stress engineer. Instead, FEA software can be used to perform these calculations immediately and effectively. For our purpose, we are using the commonly approved 'MSC NASTRAN' as a solver and MSC PATRAN for pre-processing & post-processing. During the analysis, primarily a fuselage panel with a longitudinal two bay crack is modeled and the stress values are obtained. The same model is then analyzed with a tear strap added to it. The residual strength of both of the models is obtained and the damage tolerance capability of the design is studied.

Keywords - Static stress, Stress concentration, Damage tolerance, Residual strength, Tear strap, MSC NASTRAN.

1. INTRODUCTION

Stiffened panels are the most generic structural elements in an airframe. The fuselage is a cylindrical shell, made up of stiffened panels. Currently large transport airplanes are being developed with "Large damage tolerance capability" as a design goal. An important concept in the design of the pressurized fuselage of large transport aircraft is the provision of crack stopper straps to arrest the fast fracturing of a crack.

In this paper the role of the crack stopper strap in the fail-safe design of the fuselage is investigated. As a first approximation a stiffened flat panel with a center longitudinal crack is considered. The strength of this cracked panel is investigated as a function of crack length in the absence of crack stopper straps. Crack stopper straps is then introduced at the locations of stiffeners perpendicular to the crack line and strength of the cracked flat panel is investigated as a function of crack length in the presence of crack stopper straps.

The failure criteria that is used in this study are

1. The skin crack will have a fast fracture when the maximum stress intensity factor becomes equal to the fracture toughness of the skin material at that thickness.

2. There is no rivet failure.

3. There is no failure of the stiffener normal to the crack line.

A panel strength diagram is derived from the stress analysis of this cracked stiffened panel. This diagram illustrates the strength of the skin and the stiffener as function of crack length.

2. MATERIAL PROPERTIES

The material considered for the structure is Aluminum Alloy – 2024-T351**,** with the following properties.

1. Young's Modulus, E = 70,000 N/mm^2
2. Poison's Ratio, μ = 0.3
3. Ultimate Tensile Strength, σ_u = 420 N/mm^2
4. Yield Stress, σ_y = 350 N/mm^2

Table 1 Composition of the material considered.

Composition	Wt. %	Composition	Wt. %
Al	90.7-94.7	Mn	0.3-0.9
Cr	max. 0.1	Si	max. 0.5
Cu	3.8-4.9	Ti	max. 0.15
Fe	max. 0.5	Zn	max. 0.25
Mg	5.2-5.8	Others	max. 0.15

3. STRESS ANALYSIS OF THE STIFFENED PANEL

The stiffened panel consists of

1. Skin

2. Bulkhead
3. Crack stopper strap (tear strap)
4. Longerons (stringer) and
5. Fasteners (rivets).

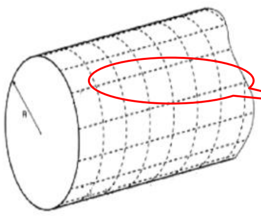
Fig. 1 Fuselage


Fig. 2 Detailed view of fuselage part

3.1 Geometric configuration of stiffened panel

Skin has the thickness of 1.5 mm. The skin houses rest of the components like Bulkheads, Longerons, Tear strap, which are assembled by riveting process. The rivets which are in rows holds the skin with longeron and the rivets which are in columns holds the tear strap and bulkhead, distance between the rows is 200mm and distance between the columns is 400mm, diameter of the rivet used is 5mm pitch of the rivet is 25mm.The fig 5.4 shows the CAD model of the skin with rivet holes.

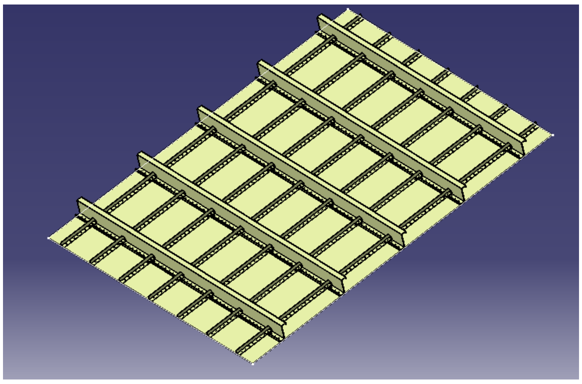
Fig.3 CAD model of the stiffened panel

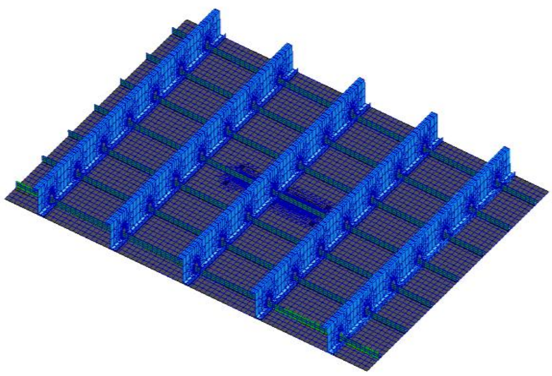
Fig. 4 Fully meshed stiffened panel

3.2 Finite element model of stiffened panel

Finite element meshing is carried out for all the components of the stiffened panel such that there is a node present at the point where riveting is to be done and fine meshing is done at the critical sections where stresses are expected to be more.

3.3 Loads and boundary conditions

A differential pressure of 9 psi (0.062066MPa) is considered for the current load case. Due to this internal pressurization of fuselage (passenger cabin) the hoop stress will be developed in the fuselage structure. The tensile loads at the edge of the panel corresponding to pressurization will be considered for the linear static analysis of the panel.

Hoop stress is given by:

$$\sigma_{hoop} = \frac{p*r}{t} \tag{1}$$

Where,

Cabin pressure (p) = 9 psi = 0.062066MPa
Radius of fuselage(r) = 1500 mm
Thickness of skin (t) = 1.5mm

After substitution of these values in (1) we will get

$$\sigma_{hoop} = 6.327 \text{ Kg/mm}^2$$

We know that:

$$\sigma_{hoop} = \frac{P}{A}$$

Above equation can be written as

$$P = \sigma_{hoop} *A \tag{2}$$

Uniformly distributed tensile load is applied on either side of the stiffened panel in the bulkhead direction which is the circumferential direction of the fuselage.

<u>Load on the skin</u>

Here

P_s = Load on skin
σ_{hoop} = 6.327 Kg/mm^2
A = Cross sectional area of skin in mm^2
i.e. Width *Thickness(2000*1.5)

Substituting these values in (2) we get,

P_s = 18981 kg

Uniformly distributed load on skin will be

$P_s = 18981/2000$
$= 9.4905$ Kg/mm

<u>Load on Tear strap</u>

Here
P_{ts} = Load on skin
$\sigma_{hoop} = 6.327$ Kg/mm^2
A = Cross sectional area of each Tear strap in mm^2
i.e. Width *Thickness(30*1.5)

Substituting these values in the (2) we get
P_{ts}=1423 Kg On each Tear strap

Uniformly distributed load on Tear strap will be
P_{ts} =9.4905Kg/mm

<u>Load on Bulkhead</u>

Here
P_b = load on Bulkhead in Kg
σ_{hoop} =6.327 Kg/mm^2

A = Cross sectional area of each Bulkhead in mm^2
i.e. (15+30+98+26)*2

Substituting the values in (2) we get
P_b =2303.028Kg on each BH

Uniformly distributed load on Bulkhead will be
P_b = 2303.028/182
= 12.654 Kg/mm on each BH

One side of all the edge nodes of stiffened panel is constrained with all six degree of freedom (i.e123456). The load is introduced at another edge of the stiffened panel. The uniformly distributed load in bulkhead direction is applied to both skin and bulkhead nodes. All the nodes are arrested in Z-direction, because the stiffened panel will not experience any non-uniform circumstantial displacement in the thickness direction (Z-direction).

The boundary conditions and load obtained from calculations are applied and the finite element analysis is performed.

3.4 *Results obtained from the finite element analysis of the stiffened panel*

Pre-processing and post-processing is carried out by using MSC Patran software and Solved by using MSC Nastran (solver) software. The response of the stiffened panel in terms of displacements and stresses due to loads and boundary conditions are explained in the following sections.

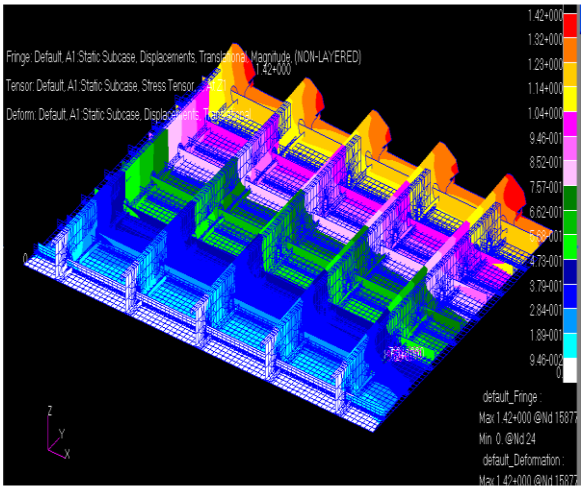

Fig. 5 Displacement contour of the stiffened panel

The Fig 5 shows the displacement contour of stiffened panel. Displacement contour increases from fixed end to loading end and it is shown by different colors fringes where white color showing minimum magnitude of displacement while red color showing maximum magnitude of displacement. The panel is constrained with all degrees of freedom at the bottom edge.

From the global analysis, stress contour on the skin, tear strap, longeron, bulkhead are obtained. It is clear that the maximum stress is at the rivet location where the rivets are used to fasten tear strap, bulkhead and longeron on skin. Invariably these maximum stress locations are the probable locations for crack initiation.

4. DAMAGE TOLERANCE EVALUATION OF THE STIFFENED PANEL

4.1 Virtual crack closure technique (VCCT) for determining stress intensity factor

There are different methods used in the numerical fracture mechanics to calculate stress intensity factors (SIF). The VCCT, originally proposed in 1977 by Rybicki and Kanninen, 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. The VCCT has a significant advantage over other methods; it has not yet been implemented into most of the large commercial general-purpose finite element codes. This technique can be applied as a post processing routine in conjunction with general-purpose finite element codes. The VCCT is based on the energy balance.

The detail calculation of the energy release rate is:

$$G = \frac{F\,u}{2\,\Delta c\,t} \quad (3)$$

Where

G = Strain energy release rate
F = Forces at the crack tip in kg or N
u = relative displacement
Δc = change in virtual crack length in mm
t = thickness of skin in mm

Then the SIF is calculated by FEM method by substituting Eq7.5 in below Eq7.6

$$K_I = \sqrt{GE} \ \ \text{MPa}\sqrt{m} \quad (4)$$

Where ,

K_I= stress intensity factor (SIF)
E =young's modulus = 7000Kg/mm^2
= 68670 Mpa
G = Strain energy release rate

Theoretically SIF value is calculated by:

$$K_I = \sigma_{hoop}\sqrt{\pi * a} * f(\alpha) \ \ \text{MPa}\sqrt{m}$$

(5) And , f (α)

$$= \frac{1+0.326(a/b)^2-0.5\frac{a}{b}}{\sqrt{1-\frac{a}{b}}} \quad (6)$$

Where

a = Crack length in mm
f (α) = Correction factor
b = Width of the plate (1500mm)

From the stress analysis of the stiffened panel it can be observed that a crack will get initiated from the maximum stress location Fine meshing is carried out near the crack to get accurate results. Other than the crack region coarse meshing is carried out. To get the mesh continuity from fine-mesh to coarse-mesh different quad and tria elements are used.

4.2 Validation of FEM approach for stress intensity factor (SIF) calculation

In this paper SIF (stress intensity factor) has been calculated by FEM (by using VCCT technique). SIF values are obtained analytically(FEM) by using (4) and (5) for un-stiffened panel having same dimension as skin in stiffened panel by applying boundary conditions which are discussed in section 3.3. SIF values are also obtained for stiffened panel using FEM.

Consider crack length, 2a=50

SIF calculation by Theoretical method

$$K_I = \sigma_{hoop}\sqrt{\pi * a} * f(\alpha) \ \ \text{(From (5))}$$

Where,

σ_{hoop} = 93.12 MPa
a =25 mm
f (α) = 1.00208 which is calculated using (6)

Substituting above values in (5).
SIF value will be

$$K_{I\ theoretical} = 26.1 \ \text{MPa}\sqrt{m}$$

SIF calculation by Analytical method (FEM)
Strain energy relies rate is calculated by (3) which is

$$G = \frac{F\,u}{2\,\Delta c\,t}$$

Where

F = 746.85N
u = 0.03217 mm
Δc = 0.78125 mm
t = 1.5 mm

Substitute all values in (3) then,
G = 10.25 MPa

Now Analytical SIF is calculated by (4) which is

$$K_{I\ fem} = \sqrt{GE}$$

Where

E = 7000 kg/mm^2 = 68670 MPa

Substituting G and E values in Eq7.6

$$K_{I\ fem} = 26.53 \ \ \text{MPa}\sqrt{m}$$

The above calculation is carried for different crack length considering fuselage internal pressurization of 13.5 psi (0.093111016MPa). Stress intensity factor values calculated by FEM (using VCCT technique) and stress intensity values calculated by theoretical method for un-stiffened panel are tabulated.

From the table 2 and the Fig 6 it is clear that SIF values obtained by using FEM (by using VCCT technique) for un-stiffened panel agrees with the SIF values calculated theoretically (5). Therefore FEM (by using VCCT method) for finding SIF value is valid.

Table 2 Comparison of analytical (FEM) SIF values with theoretical SIF value for un-stiffened panel

Crack length 2a in mm	C.F	SIF by Theoretical in MPa√m	SIF by FEA (Analytical) in MPa√m
50	1.0001	26.1	26.53
100	1.0005	36.92	37.59
150	1.0011	45.25	46.13
200	1.0021	52.30	53.39
250	1.0033	58.54	59.85
300	1.0048	64.23	65.75
350	1.0066	69.50	71.25
400	1.0087	74.46	76.45
450	1.0113	79.16	81.39
500	1.0140	83.68	86.16
550	1.0173	88.04	90.78
600	1.0208	92.28	95.28

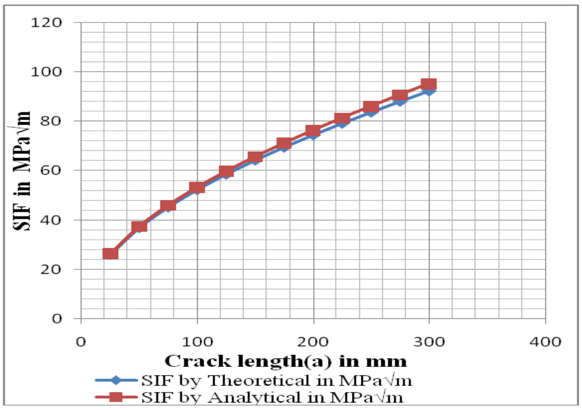

Fig. 6 Comparison of Theoretical SIF value with analytical SIF value

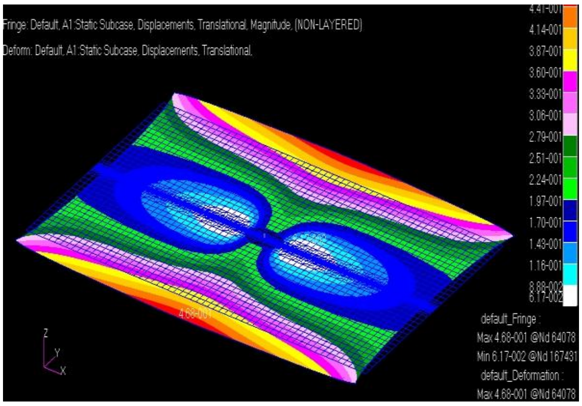

Fig.7 Displacement contour for un-stiffened panel with centre crack.

Methodology of finding SIF values for un-stiffened panel using FEM was extended to get SIF values for stiffened panel. The Fig 7 shows the displacement counter of un-stiffened panel. Orientation of crack is in longitudinal direction and crack widens due to loading in transverse direction. Where red fringes shows the maximum displacement which is at the centre of the panel

4.3 Evaluation of the effect of tear strap (crack stopper strap) for crack arrest capability

As a result of the investigations into the accidents in the 1950's, aircraft manufacturers began to incorporate into their fuselage designs features which would increase the ability of the aircraft to sustain damage caused by fatigue cracking; i.e., a damage tolerant design philosophy. A reinforced doubler on the inside of the fuselage skin, termed tear strap, crack stopper strap, or fail-safe strap, is commonly employed.

Tear straps are simply strips of material attached circumferentially to the skin of the fuselage. These tear straps are made up of aluminum alloy. For the analysis aluminum 2024-T3 is considered.

4.3.1 Stress intensity factor (SIF) approach

For the evaluation of effect of tear strap (crack stopper strap) for crack arrest capability, as a first approximation a stiffened flat panel with a center longitudinal crack is considered. The SIF value of this cracked panel is investigated as a function of crack length in the absence of crack stopper straps. Crack stopper straps are then introduced at the locations of stiffeners perpendicular to the crack line. The SIF Values of the cracked flat panel is investigated as a function of crack length in the presence of crack stopper straps. In the parametric study the thickness is varied. The SIF values obtained for stiffened panel without tear strap and stiffened panel with tear strap are compared with the critical stress intensity factor K_{Ic} (Fracture toughness of the material)

If SIF (K) at the crack tip approaches or exceeds an upper limit of stress intensity factor (K_{Ic}), then the crack will zip through leading to catastrophic failure of the structure. When the crack stress intensity factor due to remote loading reduces below the fracture toughness of the material then a crack will get arrested.

4.3.2 Residual strength prediction

The residual strength is the remaining strength in the structure under given geometry and loading condition. Residual strength is one of the main characteristics of survivability. The computational results obtained were used to determine the complete residual strength diagram of panel configuration considered.

Residual strength equations:

For skin,

$$\sigma_{residual} = \frac{K_{Ic}}{k_I} * \sigma_{remote} \quad (7)$$

Where,

K_{Ic}= Fracture toughness = 98.9MPa√m

K_I= Stress Intensity Factor

σ_{remote} = 62.07 MPa

For bulkhead

$$\sigma_{residual} = \frac{\sigma_{ultimate}}{\sigma_{bulkhead}} * \sigma_{remote} \quad (8)$$

Where,

$\sigma_{ultimate}$ = 420MPa

$\sigma_{bulkhead}$ = maximum stress in the bulkhead MPa σ_{remote} = 62.07 MPa

For Rivets

$$\zeta_{residual} = \frac{\zeta_{ultimate}}{\sigma_{rivits}} * \sigma_{remote} \quad (9)$$

Where,

$\zeta_{ultimate}$ = 210MPa

σ_{rivits} = maximum stress in the rivets MPa

σ_{remote} = 62.07 MPa

By using above (7), (8) and (9) residual strength values are calculated for stiffened panel without considering tear strap, then stiffened panel with tear strap.

5. RESULTS AND DISUSSION

The crack arrest capability of bulkheads in the stiffened panel with and without the presence of tear strap is studied. The SIF value for different crack length is calculated by analytical (FEM) method initially, for stiffened panel without tear strap and then considering the tear strap.

5.1 Stress Intensity Factor plots

Table 3 shows the SIF value for different crack length initially only for stiffened panel without tear strap and then considering stiffened panel with tear strap of 1.5mm thickness. Fracture toughness for aluminum alloy 2024 T3 is KIc=98.9MPa√m.

Table 3 Comparison of K_I FEA values of stiffened panel for given loads and boundary conditions

Crank length (a) in mm	**K_I FEA without Tear strap in MPa√m**	**K_I FEA with Tear strap in MPa√m**
100	41.53	31.00
200	50.80	51.54
300	58.46	58.68
400	65.74	65.28
500	71.43	70.67
600	76.77	75.54
700	80.28	77.57
800	76.15	69.96

In the structure the maximum stress value obtained at the two bay bulkhead rivets region, which is 8.58kg/mm^2.This is the obtained value only for pressure load condition .But in the actual condition of an aircraft while it is operating at higher altitudes it will experience both pressure load and aerodynamic loads.

As the problem is defined for the pressure load the obtained value is not the design limit load. By using strength of material approach we can obtain the design limit load. From the strength of materials approach the nominal stress obtained will be three times less than the maximum stress obtained.

The maximum obtained stress

= 3 * 8.58 = 25.74 kg/mm^2

We know that aluminum maximum yield strength is 35 kg/mm^2. The structure is normally designed in such a way that the maximum stress developed at DLL will be equal to the yield strength of the material. By using incremental ratio, which is ratio of aluminum yield strength to the obtained maximum stress.

$$\sigma_{incremental} = \frac{\sigma_{yeild}}{\sigma_{obtained}} = \mathbf{1.35}$$

so we can increase the stress to 1.35 times the originally obtained values. Those values are calculated and tabulated below.

Table 4 Comparison of K_I FEA values with tear strap and without tear strap of stiffened panel for actual loads and boundary conditions

Crank length (a) in mm	K_I FEA without Tear strap in MPa√m	K_I FEA with Tear strap in MPa√m
100	56.06	82.35
200	68.58	69.57
300	78.92	79.21
400	88.74	88.12
500	96.43	95.40
600	103.63	101.97
700	108.37	104.71
800	102.80	94.44

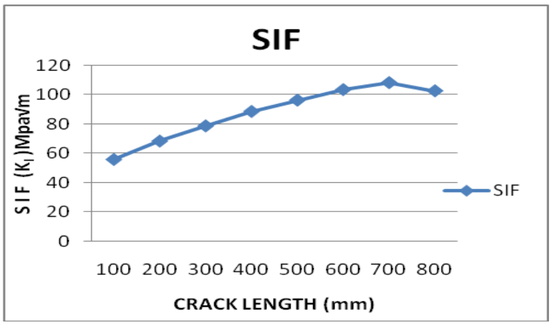

Fig. 8 S.I.F as a function of crack length without tear strap

From the results it is clear that the SIF value increases as crack length increases, as the crack approaches near to the stiffening member (bulkhead and tear strap) SIF decreases because near the stiffener region the load gets transferred from skin to the stiffener. Therefore the SIF in the skin reduces. When the crack is propagated beyond the bulkhead position, there will be an increase in SIF because the load shared by the skin increases gradually. The increasing trend in the curve is observed as the crack moves away from the bulkhead position. so by using the tear strap we can control the crack growth rate(k_I) within the two-bay structure.

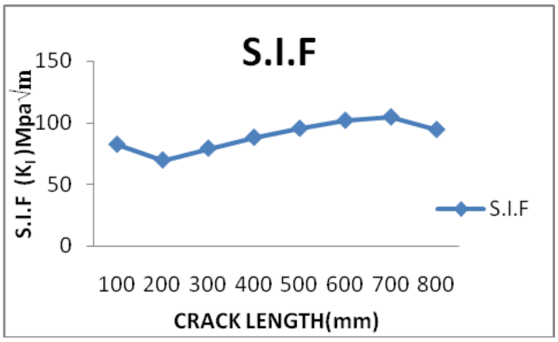

Fig. 9 S.I.F as a function of crack length without tear strap

5.2 *Residual strength plots*

Residual strength diagram for the stiffened panel indicates the remaining strength in all the structural members of the panel. A panel strength diagram is derived from the stress analysis of centre cracked stiffened panel. This diagram illustrates the variation of strength of the skin and the stiffener as a function of crack length.

The residual strength of the bulkhead is relatively more than the residual strength of the skin. The residual strength of the bulkhead keeps reducing with the increase in crack length. Similarly the residual strength of the skin also reduces as a function of crack length. But when the crack tip approaches the bulkhead, the residual strength of the skin increases because the SIF in the skin reduces. This response is an indicator of the crack arrest near the bulkhead. If the residual strength of the bulkhead reduces below the residual strength of the skin then the bulkhead will not be effective in arresting the crack. The Residual strength in skin and residual strength in bulkhead and residual strength of rivets are calculated by using (7) (8) and (9).

From the plot of residual strength v/s crack length it is clear that initially, residual strength in the bulkhead is more. As the crack length increases the residual strength in bulkhead decreases. Initially, in skin residual strength is less then bulkhead residual strength.

5.2.1 Stiffened Panel Strength Diagram

A combined plot of Residual strength of all the panel structural elements is known as Stiffened Panel strength diagram. Here the residual strengths of Cracked skin panel, the rivet and the outer bulkheads are shown. In order to demonstrate a two bay crack arrest, one needs to show that none of the three break when the crack becomes 2-bay long which is clear from the above diagram.

Also it is evident that if the failure occurs the skin will fail first, followed by the stringer and finally the rivets. If the rivets fail first, then the entire panel will fail without indication, resulting in catastrophically failure.

It is observed from the panel strength diagram that the presently analyzed 2 bay long cracked stiffened panel can carry a remote stress of 83.78 MPa.

Case1: without tear strap

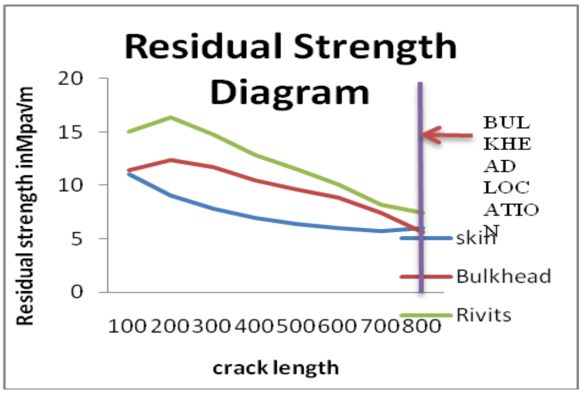

Fig. 10 Stiffened panel residual strength diagram without tear strap

Case 2 : For with tear strap

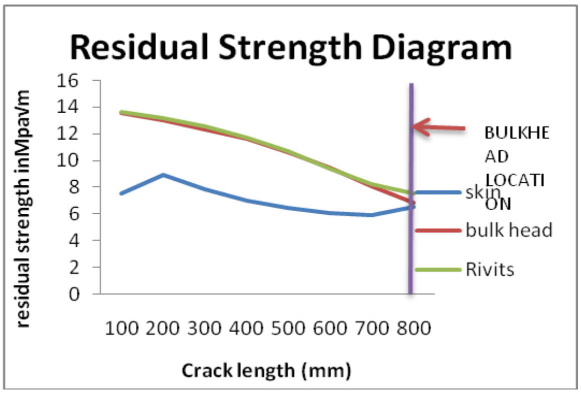

Fig. 11 Stiffened panel residual strength diagram with tear strap

6. CONCLUSION

- Damage tolerance philosophy is widely used in the aircraft structural design to reduce the weight of the structure.
- A stiffened panel which is generic structural element of the fuselage structure is evaluated analytically for its crack arrest capability.
- The internal pressure is one of the main loads that the fuselage needs to hold. In the current project also pressurization load case is considered for the analysis.
- Finite element analysis (FEA) approach is used for structural analysis of the stiffened panel.
- Stress analysis is carried out to identify the maximum tensile stress location in the stiffened panel. The magnitude of maximum tensile stress in loading direction is 23.3 Kg/mm^2 (228.59 MPa) which is in the bulkhead at the stringer cut-out. The maximum stress locations are the probable locations for crack initiation. Invariably these locations will be at stringer cut-out locations in the bulkhead.
- There are other possibilities of crack initiation at different locations in the stiffened panel due to discrete source of damage. It may be due to bird hit, foreign object hit. For the analysis centre cracked stiffened panel with central broken bulkhead and tear strap is considered which is due to discrete source of damage.
- Modified virtual crack closure technique (VCCT) along with FEA analysis results are used for calculation of stress intensity factor (SIF).
- The effect of tear strap in arresting two-bay crack is studied.
- From the residual strength calculation it indicates that without Tear strap (crack stopper straps) a two bay crack will not get arrested in the stiffened panel.
- Tear straps (crack stopper straps) with thickness 1.5 mm shows that a two bay crack is arrested it the stiffened panel.
- These results were obtained for the rivet pitch of 25mm in the bulkheads by varying the pitch of the rivet may alter the crack arrest capability of the stiffened panel.

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