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Fail Safe Design of an Aircraft Stiffened Panel by Stress Intensity Factor Determination Through Modified Virtual Crack Closure Integral Method

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Abstract - *The safe flying of an aircraft is of paramount importance during the design of various interconnected components. During service, the component parts are prone to crack initiation due to fluctuating loads and are going to grow over a period. Stiffened panel is one such component part which is prone to crack initiation. The panel must be able to withstand the loads even in the presence of crack and not fail catastrophically without giving any warning. A fail-safe design approach is employed for evaluating the design of the stiffened panel. The method involves the finite element analysis of the panel with crack to determine the stress intensity factor by modified virtual crack closure integral method. The stress intensity factor for three different skin thicknesses and varying crack lengths are determined and compared with the fracture toughness value of the material of the stiffened panel. The results show that the skin offers more resistance to crack propagation as the thickness increases. Also, the result shows that the stress intensity factor of the panel of thickness 2.2 mm goes beyond the fracture toughness value after it reaches a crack length of 635 mm and comes below the fracture toughness value as the crack length reaches 1016 mm indicating that the fail-safe design is achieved.*

Keywords — Aircraft Fuselage, Fail-safe Design Approach, Fluctuating Load, Fracture Toughness, MVCCI method, Stiffened Panel, Stress Intensity Factor.

1 INTRODUCTION

For the airplanes to fly free of danger, they have to satisfy damage tolerance requirements and follow airworthiness regulations. A structural component is said to be damage tolerant if it remains in operation after an initial damage is detected. Analysis of fatigue crack growth is the main focus of damage tolerance assessment. It involves determining how cracks propagate during service life.

Modern airplanes operate in a complex environment, loading conditions, human resource and economic requirements. The major components of the aircraft are designed to a satisfy a particular value of static and dynamic loading conditions, deformation and functional criterion. Service loads during the operation of an aircraft for design and verification of damage tolerance and durability are also very important. Fatigue and the resulting crack growth are a major challenge during the design of aircrafts. For the continued airworthiness of an aircraft during its entire economic service life, fatigue and damage tolerance design, analysis, testing and service experience correlation play a pivotal role.

The design of an aircraft considers finding an optimal proportions of payload and weight of the vehicle. It needs to be stiff and strong enough to fly under exceptional circumstances. Also, the aircraft has to fly even when one of the parts fail during the flight.

The skin is a load carrying member in the modern aircrafts. Folded sheet metals can carry compressive loads unlike the flat sheets that carry only tension. Stiffeners combined with a section of skin are analysed as thin walled structures, known by the name stringers.

In the current case, a part of stiffened panel from the fuselage segment is considered for the analysis and then subjected to tensile loading which is equal to the hoop stress developed in the fuselage. In case of the damage existing in the fuselage the damage should not exceed beyond the design limit and the structure should not undergo failure leading to catastrophic failure of the aircraft structure. So the design of structure to be made in terms of damage tolerance to avoid the failure of structure. In this context the damage existing in the skin can be tolerated by increasing the skin thickness of the stiffened panel.

The geometric model of the stiffened panel with fuselage segment is been created in CATIA modeling software and then imported into MSC.PATRAN for finite element modeling. The finite element model is solved using MSC.NASTRAN for solving stiffened panel subjected to the tensile loading with a center crack.

2 LITERATURE SURVEY

T. Swift[1] proposed new concepts on fatigue and damage tolerance capability of pressurized fuselage structure is extremely sensitive to stress level, geometrical design, and material choice. They have attempted to describe the development of fracture technology related to the design of pressurized fuselage structure capable of sustaining large, easily detectable damage.

Toor[2] has worked extensively on damage tolerant design approaches applied to aircraft structures. It was concluded that simple methods of fracture mechanics can be utilized for finding the degree of damage tolerance.

J.F.M. Wiggenraad and P.Arendsen[3] have investigated the importance of die design to be damage tolerant during different stages of damage.

N.K. Salgado, M.H. Aliabadi[4] investigated crack growth analysis in stiffened panels by finite element analysis technique. The stress intensity factors were found out from the analysis.

M. Adeel[5] has evaluated conventional and integrated stiffened panels for load bearing capacity and crack growth characteristics subjected to distributed tensile load. The crack growth characteristics foundout from finite element analysis for each type of panels are compared.

ShamsuzuhaHabeeb, K.S.Raju[6] have analysed a four stringer stiffened panel with a central crack for crack arrest and load bearing capabilities. It is found that the strength of the stiffened panel has reduced when compared to unstiffened panel.

F.Carta, A.Pirondi[7] have studied on the effect of bonded reinforcements between skin and stiffener to determine the crack propagation rate in the skin. The results obtained from finite element analysis are in close proximity to experimental results.

3 MATERIALS AND METHODOLOGY:

The following steps are adopted in evaluating the stress intensity factor of a cracked stiffened panel.

1. Creation of Geometric model of the stiffened panel including various component parts.
2. Estimation of loads on the stiffened panel.
3. Finite element analysis of stiffened panel.
4. Evaluation of stress intensity factor by MVCCI method for various thickness of stiffened panel.

3.1 GEOMETRIC CONFIGURATION OF THE STIFFENED PANEL

Stiffened panel as shown in figure 3.1 is the main component of the fuselage to which all parts are connected, so that it must be able to resist bending moments, torsional loads and cabin pressurization.

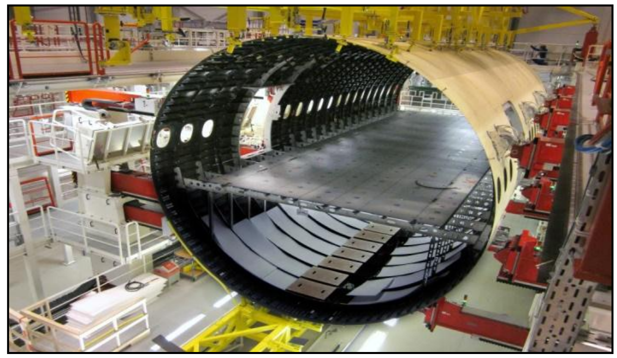

Figure 3.1: Fuselage of an aircraft

A segment of the stiffened panel considered for the finite element analysis is as shown in figure 3.2

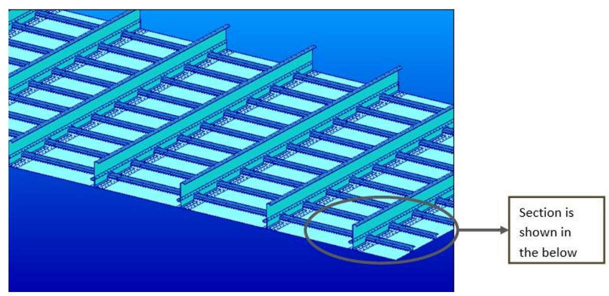

Figure 3.2: Geometric model of stiffened panel

The component parts of the stiffened panel are indicated in figure 3.3.

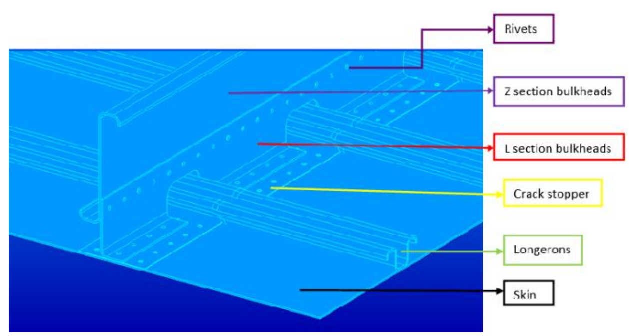

Figure 3.3: Enlarge view of stiffened panel

The dimensions and material of each of the components are tabulated in table 3.1. All dimensions are in mm.

Table 3.1: Dimensions of stiffened panel components

Name of the Parts	Dimensions in mm	Thickness in mm	Material	Number
Bottom Skin	1905×3556	1.8	Al	1
Bulkhead L stringer- Bottom Flange Web Flange	1905×23.6 1905×56.0	1.8 1.8	Al	7
Bulkhead Z Frames Top Flange Bottom Flange Web	1905×9.525 1905×24.02 1905×114.3	1.8 1.8 1.8	Al	7
Longerons Side Flange Top Flange Web Bottom Flange Web Top Flange Side Flange	3556×8.381 3556×16.09 3556×25.40 3556×22.22 3556×25.4 3556×16.09 3556×8.381	1.8 1.8 1.8 1.8 1.8 1.8 1.8	Al	9
Crack stopper	1905×76.12	1.8	Al	7
Rivets	Diameter	4.4	Al	4830

Stiffened panel are the generic part of the fuselage section. The fuselage is an integration of many number of stiffened panel. Geometric model considered for the analysis is same as the actual stiffened panel of the fuselage. For the analysis purpose the curved panel is assumed to be a straight panel and the meshing is carried out. The geometric configuration of various components of the stiffened panel are as shown from figures 3.4 to 3.9.

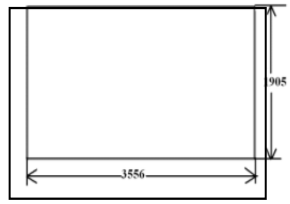

Figure 3.4: Dimensions of skin

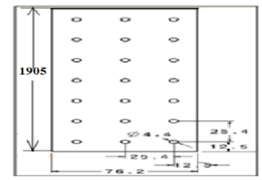

Figure 3.5: Dimensions of crack stopper

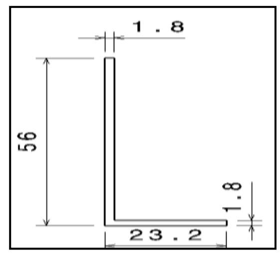

Figure 3.6: Dimensions of L bulkhead

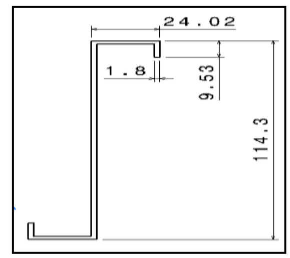

Figure 3.7: Dimensions of Z bulkhead

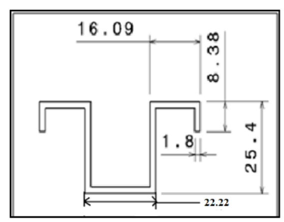

Figure 3.8: Dimensions of Longerons

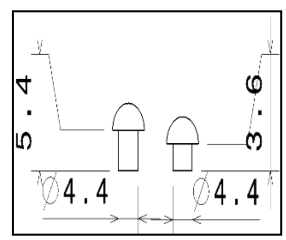

Figure 3.9: Dimensions of rivets

3.2 ESTIMATION OF LOADS IN THE STIFFENED PANEL:

The fuselage of an aircraft structure is subjected to an internal pressurization in the range of 0-10 psi. Aircraft structure is generally cylindrical in nature and is subjected to hoop stress. As the stiffened is modeled as straight panel the hoop stresses are converted into corresponding tensile stresses acting on the components.

Hoop stress is given by

$$\sigma_{hoop} = \frac{p_i \times r_i}{B} \qquad (3.1)$$

The hoop stress is calculated by assuming the following values.

Cabin pressure, pi = 6 psi

Radius of the fuselage, ri = 1500 mm

Thickness of skin, B = 1.8 mm

$\boldsymbol{\sigma_{hoop} = 34.335\ Mpa}$

$$\sigma_{hoop} = \sigma_{tensile} = \frac{F}{A} \quad (3.2)$$

$$F = \sigma_{hoop} \times A \quad (3.3)$$

Force per unit length i.e. the load acting along the length is calculated by

$$q = \frac{F}{L} = \sigma_{hoop} \times t \quad (3.4)$$

Force per unit length acting on the skin, stopper and bulkhead are calculated by the following equations:

$$q_s = \sigma_{hoop} \times t_s \quad (3.5)$$

$$\boldsymbol{q_s = 61.803\ N/mm}$$

$$q_c = \sigma_{hoop} \times t_c \quad (3.6)$$

$$\boldsymbol{q_c = 21.803\ N/mm}$$

$$q_b = \sigma_{hoop} \times t_b \quad (3.7)$$

$$\boldsymbol{q_b = 61.803\ N/mm}$$

3.3 DAMAGE TOLERANCE EVALUATION OF STIFFENED PANEL

All the components of the stiffened panel are made of Aluminium 2024 T3 material. The properties of the Aluminium 2024 T3 are tabulated in table 3.2.

Table 3.2: Material properties of Aluminium 2024 T3

SN	Property	Value
1	Young's modulus	73 GPa
2	Poisson's ratio	0.3
3	Ultimate tensile strength	483 MPa
4	Ultimate shear strength	283 MPa
5	Fracture toughness	$37 MPa\sqrt{m}$

The finite element analysis of the stiffened panel is carried out using NASTRAN solver. Meshing is done on the components using quadrilateral and triangular shell elements. The crack is introduced at the center of the skin by disconnecting the common nodes of the elements on the crack front. The region near the crack tip is fine meshed with an edge length of 0.8 mm to avoid the stress singularity. The rest of the portion is meshed with coarse mesh.

Initially the structure is analyzed for a skin thickness of 1.8 mm and a central crack of 2a = 25.4 mm. One end of the stiffened panel which is parallel to the crack is fully constrained and the other end is subjected to loads as shown in figure 3.1

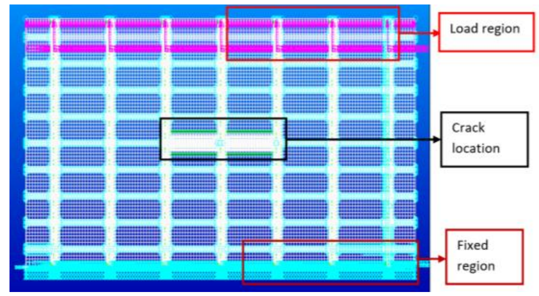

Figure 3.10: Meshed panel with loads and boundary conditions

Assuming that the maximum damage occurs to the panel when the bulk heads are broken i.e. as the crack passes through the bulkheads. The broken bulkheads are modeled by deleting the elements in that region. The deformation plot of the above model is shown in figure 3.11. The plot indicates that there is a deformation in Z direction. In order to avoid bending of the stiffened panel, the deformation in Z direction needs to be constrained at all the nodes and the model is analysed by introducing the new boundary conditions as shown in figure 3.12.

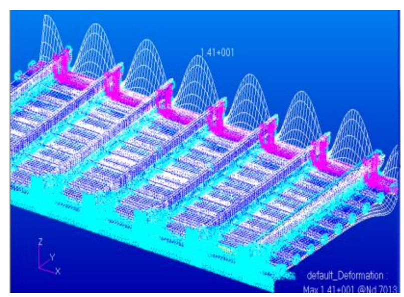

Figure 3.11: Deformation plot of stiffened panel without constraining in Z direction

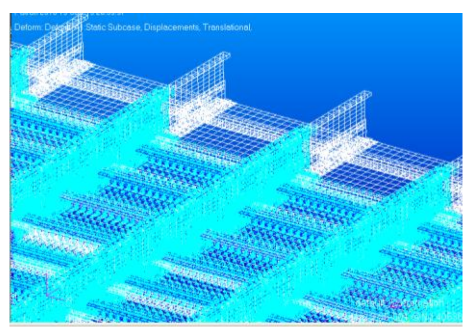

Figure 3.12: Deformation plot of stiffened panel with constraint in Z direction

3.4 SIF calculation by MVCCI method for skin thickness of 1.8 mm

After performing the FE analysis, the SIF of the loaded panel is calculated by modified virtual crack closure method as below.

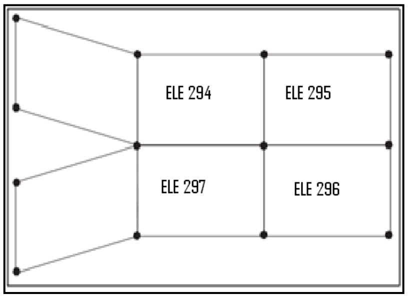

Figure 3.13: Cracked region before crack tip opening

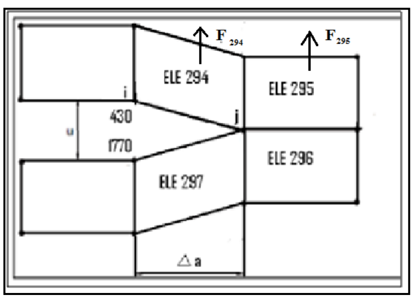

Figure 3.14: Cracked region after crack tip opening

Figure 3.13 and 3.14 shows the crack tip configuration before and after virtual crack closure. These configurations are used for calculating energy release rate (G) at the crack tip as follows.

Energy release rate is given by,

$$G = \frac{F\,u}{2\,B\,\Delta a} \qquad (3.5)$$

Where,

G = Strain energy release rate in N/m

F = Forces in N.

Fj = Elemental forces at the tip of crack in N.

u = Relative displacement in mm.

B = Thickness of plate in mm.

Δa = Change in virtual crack length in mm.

Consider a crack length of 25.4 mm

Table 3.3: Displacement vector for stiffened panel

Node number	Distance between (ui)nodes in mm
84081	0.478857
101162	0.461792

The relative displacement(u) between the nodes (84081, 101162) is (refer figure 3.14)

u = u84081 – u101162 = 0.478857- 0.461792 = 0.01706 mm

Table 3.4: Grid point force balance for stiffened panel.

Element number	Element force (Fj) in N
80285	238.99
85576	245.83

The total forces (F) at the crack tip is (refer figure 3.14)

F = F80285 + F85576 = 484.82N

Substituting all values in Equation 3.5 gives

G= 2872.73 N/m

SIF is calculated by Equation

$$\text{KI fem} = \sqrt{GE} \qquad (3.6)$$

KI fem=14.48 MPa$\sqrt{m}$

The same analysis is carried out on the stiffened panel by varying the crack length from 25.4 mm to 1016 mm with increments of 50.8 mm, which is equal to the distance between adjacent rivets along the Longerons.

Now the entire analysis is performed on the stiffened panel by varying the skin thickness to 2.0 mm and 2.2 mm and the results are obtained (SIF).

4 RESULTS

4.1 SIF for skin thickness of 1.8 mm

The FE analysis is carried out on the stiffened panel by varying the crack length from 25.4 mm to 1016 mm with increments of 50.8 mm, which is equal to the distance between adjacent rivets along the Longerons. The SIF for all the cases is calculated by MVCCI method and are tabulated in table 4.1.

Table 4.1: Stress intensity factor (K_{IFEM}) of stiffened panel for skin thickness 1.8 mm

Crack length (2a) in mm	Relative displacement (U) in mm	Forces at the crack tip (F)in N	Strain energy release rate (G) in N/m	KI(FEA) in MPa√m
25.40	0.0170	484.8	2872	14.48
76.20	0.0280	783.05	7625	23.10
127.0	0.0321	894.0	9988	26.44
177.8	0.0346	961.8	1156	28.45
228.6	0.0369	1024.1	1313	30.96
279.4	0.0390	1084.3	1471	32.08
330.2	0.0411	1141.2	1629	33.77
381.0	0.0430	1195.7	1789	35.39
431.8	0.0449	1247.1	1949	36.93
482.6	0.0467	1297.6	2107	38.14
533.4	0.0484	1345.0	2264	39.81
584.2	0.0501	1389.1	2463	41.52
635.0	0.0516	1431.2	2564	42.37
685.8	0.0529	1469.3	2702	43.49
736.6	0.0541	1501.9	2824	44.46
787.4	0.0551	1528.0	2924	45.24
838.2	0.0557	1544.5	2987	45.73
889.0	0.0557	1545.7	2992	45.77
939.8	0.0541	1518.0	2890	44.98
990.6	0.0510	1412.1	2503	41.86
1016	0.0430	1511.4	2261	39.78

4.2 SIF for skin thickness of 1.8 mm, 2.0 mm and 2.2 mm.

Now the entire analysis is performed on the stiffened panel for different skin thickness of 2.0 mm and 2.2 mm. The variation of the SIF with the crack length for different values of skin thickness are represented in graphical form as shown in figure 4.1.

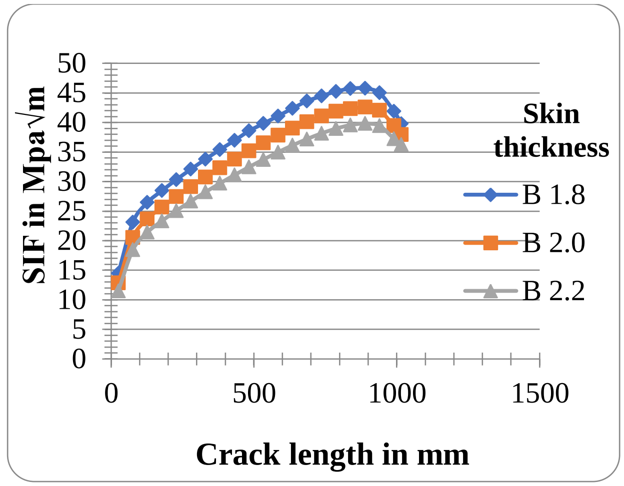

Figure 4.1: Stress intensity factor (kI) v/s crack length for different skin thickness

5 DISCUSSIONS

Figure 4.1 shows the variation of stress intensity factor of the stiffened panel skin component with respect to the crack propagation. Figure 4.1 indicates that, the panel with a skin thickness of 2.2 mm has the least value of SIF (36.22MPa$\sqrt{m}$) which is lesser than the fracture toughness of the skin material. This result indicates that a stiffened panel with a skin of thickness of 2.2 mm is going to fail as the crack length reaches beyond 635 mm. Further, the crack can propagate to 1016 mm and the SIF is brought down to 36.22 MPa$\sqrt{m}$ indicating that the failsafe design is achieved. The reason for decrease in SIF beyond a crack length of 635 mm is due to the transfer of load from skin to other parts of stiffened panel. Also, the results show that with the increase in the skin thickness the skin offers more resistance to the crack propagation.

6 CONCLUSION

Modified virtual crack closure integral method adopted for damage tolerant design of a stiffened panel gives accurate results. Thus the method can also be applied for other structures where damage tolerance design approach can be followed.

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