A Study on Damage Tolerance Evaluation of the Vertical Tail with the Z stiffened panel of a Transport Aircraft

Channabasavaraj B Dharani[1], Charanraj A V[2], Rangaswamy M K [3], Devaraj R B[4]

[1] *Assistant professor (Mechanical) Bangalore Technological Institute*
[2,3,4] *Assistant professor (Mechanical) Bangalore College of Engineering and Technology*

***

Abstract– *Fuselage, wings, vertical tail and horizontal tail are important structural components of an airframe. Rudder is the control surface used in the vertical tail to control the yawing motion of the aircraft. Structural arrange and shape of the vertical tail are similar to that of the wing. A major difference could be absence of ribs and multiple spars (more than 2) in the vertical tail construction.*

Vertical tails have symmetrical airfoil cross sections. Therefore in the absence of rudder deflection there is no aerodynamic load acting on the fuselage. However significant side loads develop due to rudder deflection and this is the major design load for the vertical tail. For transport aircraft side gust load is also important from a design point of view.

The current study includes a stiffened panel of the vertical tail for evaluation of its damage tolerance capabilities through analytical approach. Loads representative of a small transport aircraft will be considered in this study. A stiffened panel which is the representative of the vertical tail structural features will be considered for the analysis.

Finite element analysis of the stiffened panel will be carried out to identify the location of maximum stress and the stress distribution on the stiffened panel. A crack will be initiated from the location of maximum stress in the stiffened panel. Side loads on the vertical tail will cause the tension stress field in the skin of the stiffened panel. Therefore the stiffened panel will be subjected to tensile stress field. Crack arrest capability of the stiffeners ahead of the crack in the skin will be evaluated analytically.

Modified virtual crack closure integral (MVCCI) method will be used for calculation of stress intensity factor at the crack tip. The stress intensity factor will be compared with the fracture toughness of the material at different crack increments.

I. INTRODUCTION

Fundamentally, an aircraft is a structure. When designing an aircraft, it's all about finding the optimal proportion of the weight of the vehicle and payload. It needs to be strong and stiff enough to withstand the exceptional circumstances in which it has to operate. Also, if a part fails, it doesn't necessarily result in failure of the whole aircraft.
Tail surfaces are used to both stabilize the aircraft and provide control moments needed for maneuver and trim. Because these surfaces add structural weight they are often sized to be as small as possible. The vertical stabilizer prevents side-to-side, or yawing, motion of the aircraft nose. The rudder is used to control the position of the nose of the aircraft. Side loads developed due to rudder deflection is the major design load for the vertical tail.

The major part of an aircraft structure consists of built-up panels of sheets and stringers, e.g. wing and fuselage skin panels; spar webs. Past experience has shown that, despite all precautions crack may arise in any of these structural elements. Crack will reduce the stiffness and load carrying capacity of the structure. Hence the possibility of cracking must be taken early in design stage i.e. the designer has to make his concept "Damage Tolerant".

Past experience has indicated that the time to initiation of cracks from most structural details such as sharp corners or holes is relatively short and that the majority of the life (i.e., 95%) is spent growing the resultant cracks to failure. To prevent catastrophic failure, one must evaluate the load carrying capacity that will exist in the potentially cracked structure throughout its expected service life. The load carrying capacity of a cracked structure is the residual strength of that structure and it is a function of material toughness, crack size, crack geometry and structural configuration.

II VERTICAL TAIL

Structural arrange and shape of the vertical tail are similar to that of the wing. A major difference could be absence of ribs and multiple spars (more than 2) in the vertical tail construction. Vertical tails have symmetrical airfoil cross sections. Therefore in the absence of rudder deflection there is no aerodynamic load acting on the fuselage. However significant side loads develop due to rudder deflection and this is the major design load for the vertical tail. For transport aircraft side gust load is also important from a design point of view.

CRUCIFORM TAIL (FIN MOUNTED TAIL)

The cruciform tail is arranged like a cross, horizontal stabilizer intersecting the vertical tail somewhere near the middle. The cruciform tail gives the benefit of clearing the aerodynamics of the tail away from the wake of the engine, while not requiring the same amount of strengthening of the vertical tail section in comparison with a T-tail design e.g. A-4 Sky hawk, Avro Canada CF-100 etc.

T-TAIL

A T-tail has the horizontal stabilizer mounted at the top of the vertical stabilizer .T-tails are often incorporated on configurations with fuselage mounted engines to keep the horizontal stabilizer away from the engine exhaust plume. The tail surfaces are mounted well out of the way of the rear fuselage, permitting this site to be used for the aircraft's engines. This is why the T-tail arrangement is also commonly found on airliners with rear-mounted engines e.g. Boeing 727, the Vickers VC10 etc.

Fuselage mounted tail unit (Conventional tail)

The vertical stabilizer is mounted exactly vertically on to the empennage (the rear fuselage). This is the most common vertical stabilizer configuration e.g. Airbus A380

Twin tail (H tail)

Two vertical stabilizers—often smaller on their own than a single conventional tail would be—are mounted at the outside of the aircraft's horizontal stabilizer. This arrangement is also known as an H-tail. It affords a degree of redundancy—if one tail is damaged, the other may remain functional. In some aircrafts are vertical surfaces mounted to the upper surface of the fixed stabilizer instead, at some distance inwards from the horizontal stabilizer's tips e.g. Mitsubishi G3M and Dornier Do 19

V-tail (Butterfly tail)

A V-tail has no distinct vertical or horizontal stabilizers. Rather, they are merged into control surfaces known as ruddervators which control both pitch and yaw. The V-tail has less wetted surface area, and thus produces less drag. Combining the pitch and yaw controls is difficult and requires a more complex control system. The V-tail arrangement also places greater stress on the rear fuselage when pitching and yawing. e.g. F-117 Nighthawk, Eclipse 400

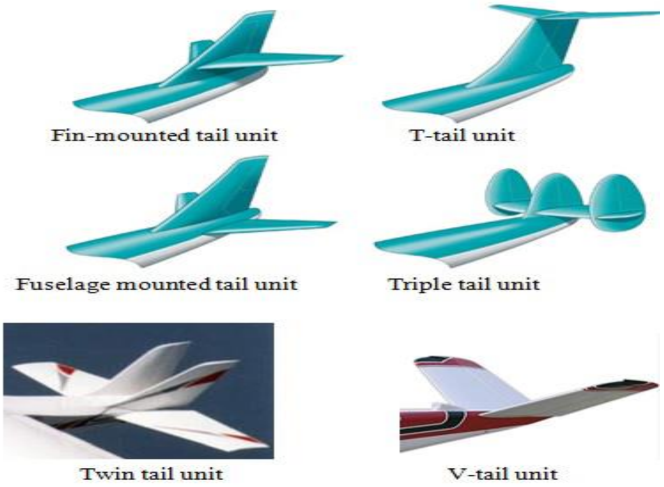


Figure 1 Types of Vertical Tail

SAFE LIFE DESIGN

This approach dates back to the mid-1800s, when the repetitive loading on mechanical structures intensified with the advent of the steam engine. Engineers and academics developed a curve relating the magnitude of the cyclic stress (S) to the logarithm of the number of cycles to failure (N). This curve, known as the S-N curve shown in the following fig.2, became the fundamental relation in safe life design.

In the safe life method, the S-N curve is used to design a component in such a way that it will not fail within a pre-determined number of cycles. For example, if a test specimen has not failed up to 107 cycles, it is assumed that the specimen would never fail before 107 cycles in the safe life design. Subsequently the component's durability is estimated, first by evaluating the highest operational stress on the component using hand calculations or finite element methods, and then comparing the component's highest operational stress to the stress scale on the test specimen's S-N curve. If the stress of the component is below the fatigue strength on the S-N curve, the component is said to be designed for infinite life. If the stress of the component is above the fatigue strength (e.g. stress S1 in the figure), the component is life limited (in figure at S1, the life is limited to between 105 and 106 cycles).

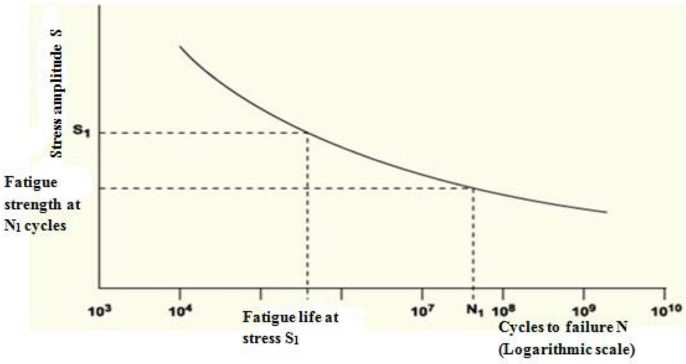


Figure 2 S N Curve

DAMAGE TOLERANCE ANALYSIS

In 1921, Griffith developed Fracture mechanics, a branch of physics that evolved to be applied to fatigue of metallic structures in the 1960s (Paris et al., 1961).Fracture mechanics quantifies the energy the crack has in a value called the stress intensity factor (SIF).This factor is a function of the applied load as well as the morphology of the crack.

Damage tolerance analysis assumes that fatigue cracks nucleate in a component during operational life, and that growth of these small cracks in fatigue will occur if sufficient energy exists in the system.

III VERTICAL TAIL STIFFENED PANEL

Vertical tail unit includes a fixed surface known as vertical stabilizer and a control surface (moving) known as rudder. The vertical stabilizer maintains the stability of the aircraft about its vertical axis. This is known as directional stability. Rudder is hinged to stabilizer and is used to control the yawing motion of the aircraft. This action is known as directional control. The shape structural arrangement of vertical tail is similar to that of wings. A major difference between wings and vertical tail could be the absence of ribs and multiple (more than 2) spars.

For greater strength, especially in the thinner air foil sections typical of trailing edges, a honeycomb-type construction is used. Some larger carrier-type aircraft have vertical stabilizers that are carrier-type vertical stabilizers that are folded hydraulically to aid aircraft movement aboard aircraft carriers.

The stiffened panel is the elementary part of most of the airframe structures with intermediate and higher loading intensity. The typical vertical tail stiffened panel consists of longitudinal stiffeners fastened to the skin by utilizing many rivets. The primary function of the stiffener is to transfer loads acting on the skin on to the ribs and spars. Since vertical tail unit does not include multiple ribs and spars, the role of the stiffeners in the vertical tail is considered to be very important.

GEOMETRICAL CONFIGURATIONS OF STIFFENED PANEL

A part of the vertical stabilizer is considered in the present study. As discussed earlier, the stiffened panel includes skin and longitudinal stiffeners fastened to skin by utilizing rivets. The fig 3 shows the draft sheet of the vertical tail stiffened panel. Modelling of the stiffened panel is done in CATI A V5 R20 software. Geometrical configurations of the individual components of stiffened panel are as shown in fig.3 All the dimensions are in mm.

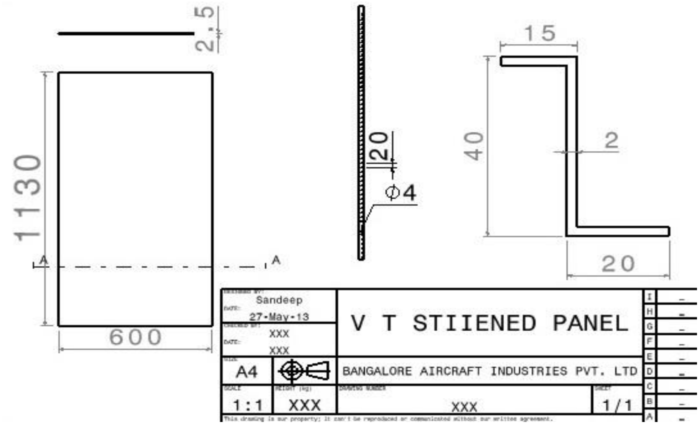


Figure 3 Geometrical Configurations of Stiffened Panel

The stiffened panel considered for the present study has the following dimensions. Panel height = 1130mm, panel width = 600mm, skin thickness = 2.5mm, stiffener spacing = 150mm. The part of the panel between two adjacent stiffeners is called as a bay. Three Z stiffeners are considered in the panel so that crack can be initiated at mid stiffener and crack arrest capabilities of stiffener ahead of the crack in the skin can be evaluated between two bays. Figure 4 shows the part model of the vertical tail stiffened panel.

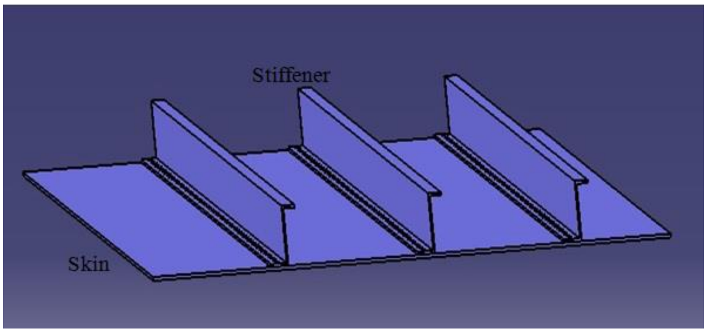


Figure 4 Stiffened Panel Part Model

Geometrical configurations of stiffener

Stiffener consists of top flange, web and the bottom flange. Bottom flange is fastened to the skin by using rivets. The fig.5 shows CAD model and geometrical configurations of stiffener.

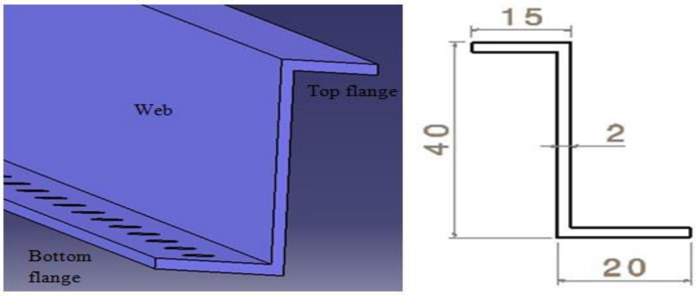


Figure 5 CAD Model of Stiffener

The stiffener has following dimension. Top flange = 15mm, web = 38mm, bottom flange =20mm, stiffener thickness = 2mm, rivet pitch = 20mm, rivet hole diameter = 4mm.

WHY Z STIFFENER?

Various configurations of the stiffener are used in aircraft structures such as Z, C, Hat, I, T etc. Among the various configurations available, Z stiffeners are used in the present study. The configurations such as Hat, I, T need two rows of rivets to be attached to the skin while Z and C need single row of rivets. Since, rivet locations act the stress concentration regions, to avoid stress concentration regions in the panel Hat, I, T configurations are not used.

Among Z and C configurations, the Z configuration has higher moment of inertia than C, for the same dimensions. And, also riveting the stiffener to the skin is much easier with Z configuration when compared to C configuration.

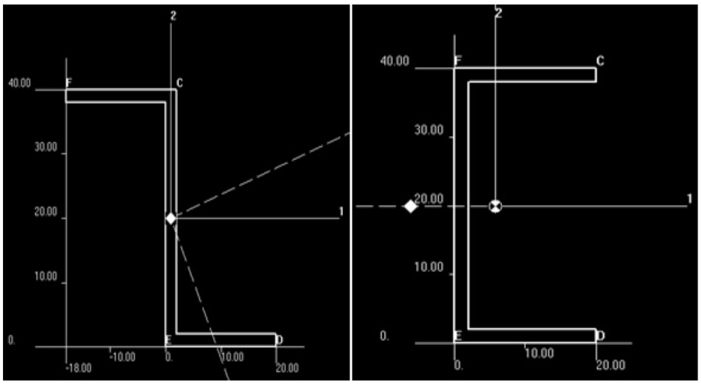


Figure 6 Comparison of MOI of Z and C Configurations

Moment of inertia of Z configuration are I1= 36682.67mm4, I2=9170.667mm4 while that of C configuration are I1=36682.67 mm4, I2=5760.14 mm4. Thus it is clear that MOI of Z configuration is higher than C configuration. Hence Z stiffeners are consider in the present study

IV FINITE ELEMENT MODEL AND STRESS ANALYSIS OF STIFFENED PANEL

Finite element meshing of the vertical tail stiffened panel is done in such a manner that there is a node at each rivet location in the stiffened panel. Stiffened panel is meshed with uniform coarse mesh because there are no critical regions (cut outs) in the stiffened panel. Finite element meshing of the stiffened panel is done to carry out the stress analysis of the panel.

FINITE ELEMENT MODEL OF STIFFENED PANEL

Fig.6 shows finite element meshing of the stiffened panel. 2D shell QUAD4 elements are used for both skin and stiffener. 1D BEAM elements are used for rivets which fastens the stiffener to the skin.

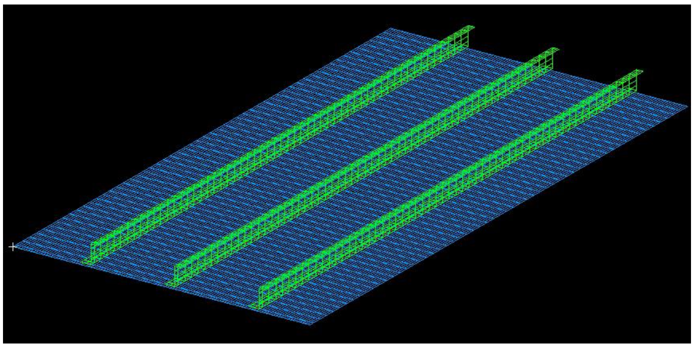


Figure 6 Finite element mesh of the stiffened panel

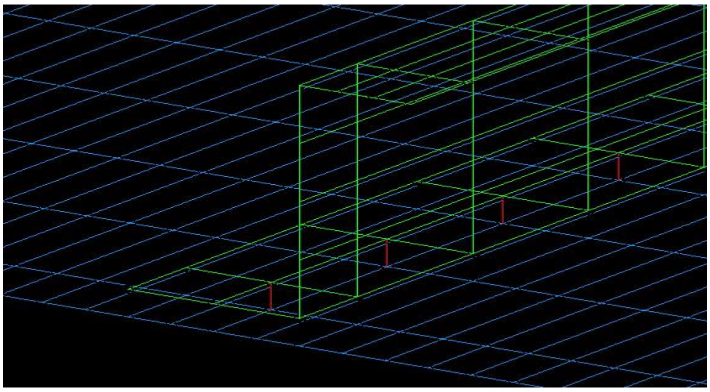


Figure 7 close up view of the stiffened panel

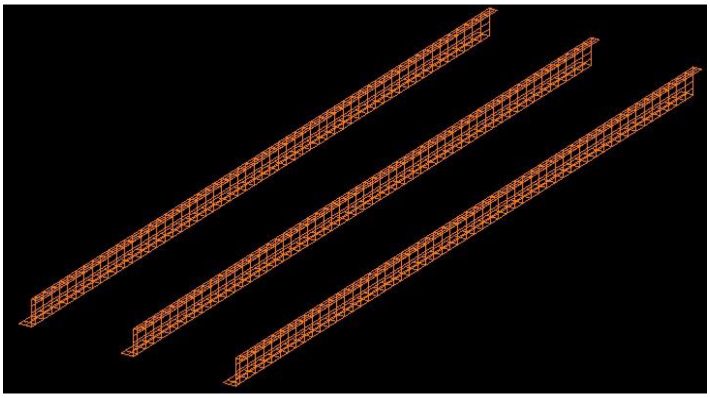


Figure 7Finite element mesh of stiffener

In stress analysis, the stiffened panel is subjected to loads and boundary conditions. The panel is subjected to yielding load of the material to obtain the stress distribution and displacements in the loading direction. Analysis is carried out using MSC NASTRAN solver.

Material properties of the stiffened panel

Property	Value
Modulus of elasticity, E	70.3-73.1 GPa
Ultimate tensile strength,	483 MPa
Tensile yield strength,	345 MPa
Hardness	120 HB
Shear strength,	283 MPa
Elongation	18 %
Poisson's Ratio,	0.33
Shear modules	28 GPa
Fracture toughness	80 MPa

Stress contour of the stiffened panel

The maximum tensile stress acting in the panel is of the magnitude 702.39MPa (71.6 kg/mm2). And minimum tensile stress is having magnitude of 264.87MPa (27 kg/mm2). The fringe shows the Y component (loading direction) of the stress distribution in the stiffened panel.

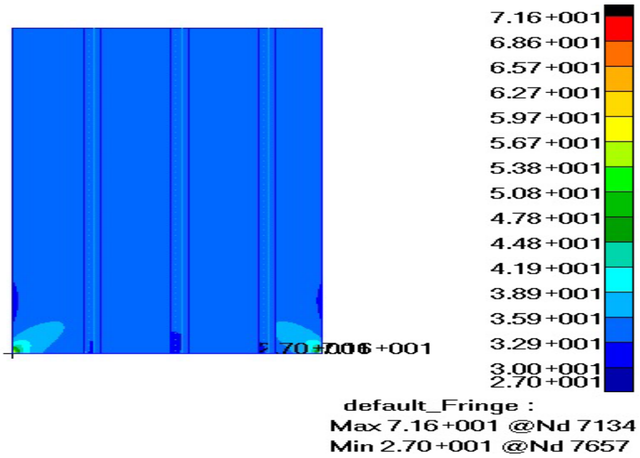


Fig 8 stress counter

DISPLACEMENT CONTOUR OF STIFFENED PANEL

Fig.9 shows the displacement contour of the stiffened panel. The displacement is only along the height of the panel.

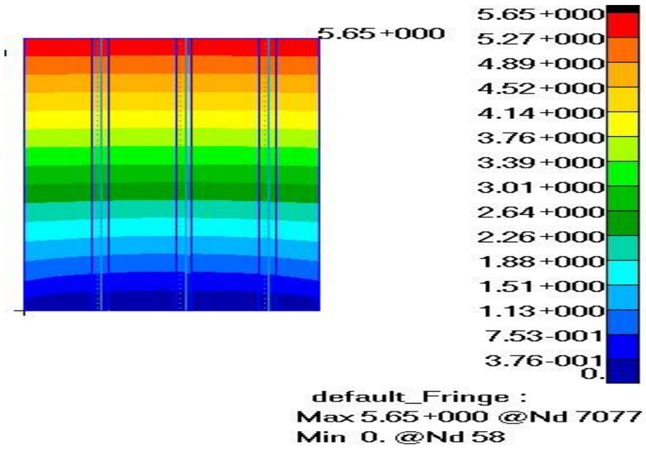
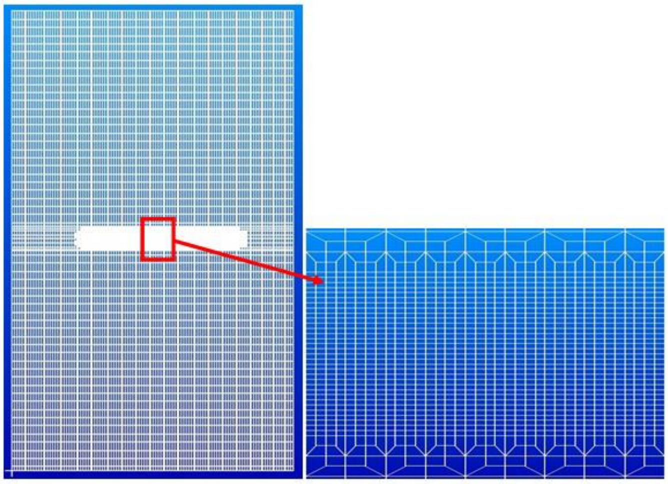


Figure 9 Displacement contour of stiffened panel

V DAMAGE TOLERANCE ANALYSIS OF STIFFENED PANEL

FEM of stiffened panel for damage tolerance analysis

For damage tolerance analysis of the stiffened panel, the meshing in the region where crack is to be generated needs to be changed. Fine meshing is to be carried out in the stiffened panel near the crack to obtain crack propagation results accurately. FEM of stiffened panel is shown in the following figure. Fine meshing is carried out in the region between three stiffeners. The skin is meshed by four noded shell elements shown in Fig.9. For mesh continuity from fine mesh to coarse mesh different four noded shell elements are used. The elemental edge length 1.25 mm is maintained at crack region.



a. Stiffened panel b. Zoom in view of fine mesh

Figure 10 Finite element model of stiffened panel

STRESS CONTOURS FOR CRACK LENGTH, 2A=10MM

Stress distribution and displacement in the stiffened panel for a crack of length 10mm in the skin are as shown in the fig.10. The crack is initiated at location of the middle stiffener. Orientation of crack is in longitudinal direction and crack widens due to loading in transverse direction. The stresses at crack tip are maximum and the magnitude of maximum stress is found to be 307.053MPa (31.3 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.

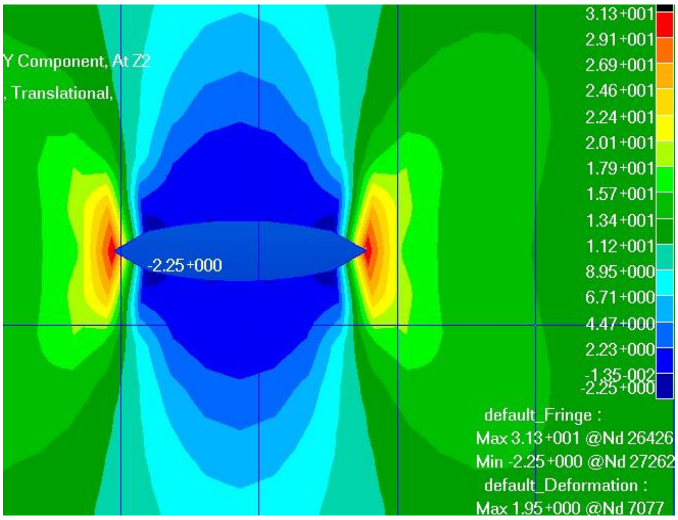


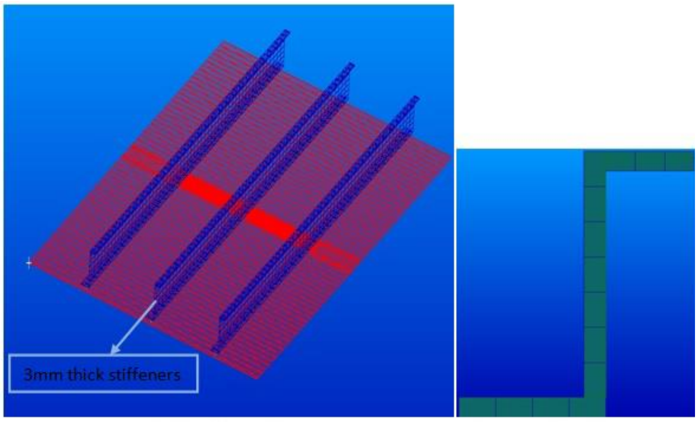
Figure 11 Close up view of stress contour for skin at crack tip

MODIFICATION

When the crack is nearer to the stiffeners, a reduction in the stress intensity factor is observed as a result residual strength of the increases near the stiffeners.

The decrease in the SIF is results in the crack arrest. But the reduction in the SIF is by very small magnitude. So to study the effect of thickness of stiffener on the SIF, damage tolerance analysis of the stiffened panel is again carried out using stiffeners of thickness 3mm. Fig.12 shows the finite element model of the panel in which 3mm thick stiffeners are riveted to the skin.

With this panel, again the damage tolerance is carried out to determine the SIF for different crack lengths and to study the variation in the residual strength of the skin with propagation of the crack.



a. Stiffened panel b. 3mm thick stiffener

Figure 12 FE Model of stiffened panel with 3mm thick stiffeners

VI RESULTS AND DICUSSION

For Linear Static Analysis of stiffened panel, the finite Element model of the panel with loads and boundary conditions applied is solved in MSC Nastran. MSC is used for pre-processing and post-processing. The solver gives output as per our output requests for the FEM model of panel. For static analysis stress and displacement output are requested. Stresses in the elements and force at nodes can be observed by using grid point stresses and elemental stresses.

1.Concentration of additive can be increased up to 15% and evaluating the performance emission parameters.

2.Exhaust gas recirculation system can be used for reducing NOx emission.

3.Preheating of biodiesel blends by exhaust gas to get better atomization.

The results can be seen for the entire model or scooped to a part or specific region and contour plot shows the stresses acting for the given loads. In staticanalysis, the panel is subjected to allowable load taking the stress concentration factor due to rivet hole in to consideration. Maximum principal stress contour for the stiffened panel is shown in the following fig.13

The maximum principal stress acting in the panel is 247.21MPa (25.2 kg/mm^2)and is acting at extreme bottom corners. Due to the eccentric loading between the skin and the stiffener the panel tends to undergo in plane bending. Minimum principal stress acting on the panel is 91.23MPa (9.3 kg/mm^2).

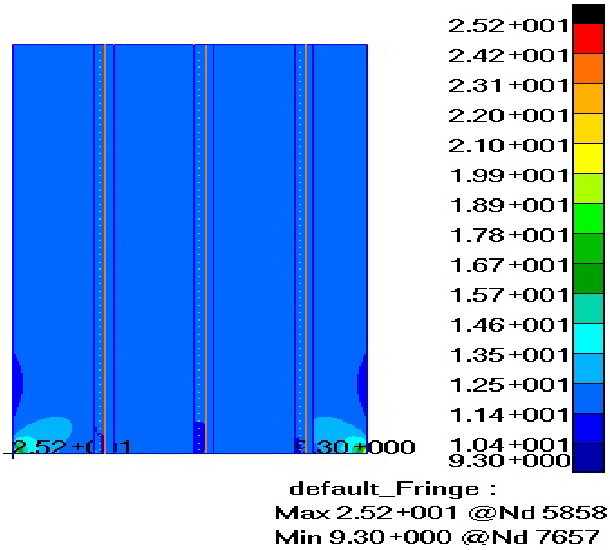


Figure 13 Maximum principal stress contour

DAMAGE TOLERANCE ANALYSIS OF STIFFENED PANEL

Damage tolerance analysis of stiffened panel is carried out to determine the variations in the stress intensity factor for different increments in the crack length. So that the crack arrest capabilities of the stiffeners can be evaluated analytically. Crack is initiated below the mid stiffener so that it can be extended to the remaining two stiffeners on the either side. The 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.

STUDY OF CRACK PROPAGATION IN THE STIFFENED PANEL

SIF as a function of crack length different crack lengths are plotted shown in fig.14. It is observed that, SIF increases gradually with increase in the crack length.

When the crack approaches near to the stiffener, the value of SIF decreases.

The forces in the elements near crack tip deceases when the crack is nearer to the stiffeners. It is found that, the value of SIF is 14.43MPa at crack length of 10 mm and increases to 71.80MPa as crack approaches to 300mm and then decreases to 71.363MPa at stiffener location.

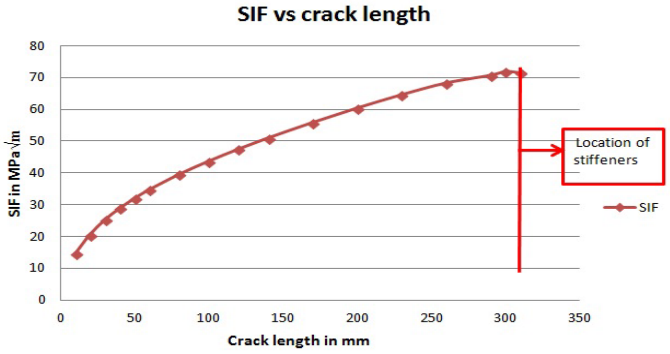


Figure 14 Variation of SIF as a function of crack length

ROLE OF STIFFENERS TO ARREST CRACK PROPAGATION

Residual strength of skin and stiffener are plotted as a function of crack length in fig.9.3.Residual strength of both skin and stiffener gradually decreases with the increase in crack length. For crack length 10 mm residual strength of skin is found to be 655.45MPa. The minimum residual strength of skin is found to be 131.71MPa when the SIF is highest for crack length 300 mm.This decrease in the residual strength of skin is due to the increase in the SIF with the increase in the crack length. This continues till the crack approaches stiffeners.

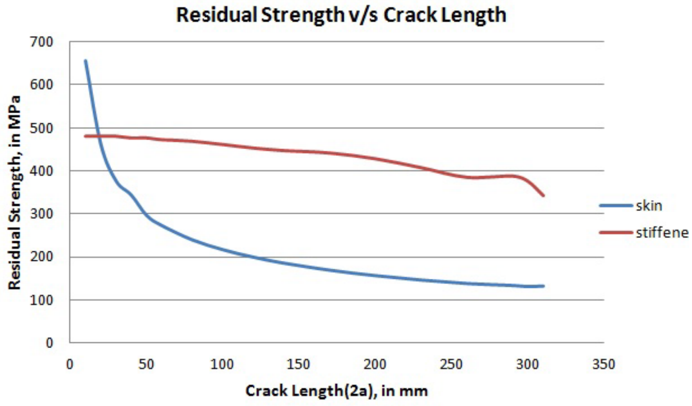


Figure 15 Residual strength of skin and stiffener as a function of crack length

VARIATIONS IN RESIDUAL STRENGTH WITH CRACK LENGTH

Residual strength of skin with 2mm and 3mm thick stiffeners are plotted as a function of crack length in fig.16

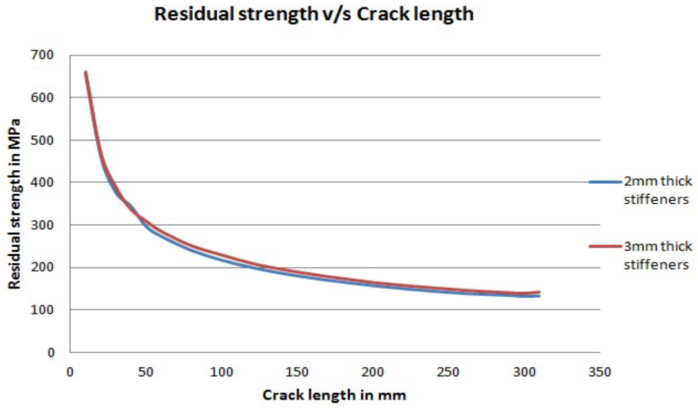


Figure 16 Variation in Residual strength of skin with 2mm and 3mm thick stiffeners

VII CONCLUDING REMARKS AND FUTURE SCOPE OFWORK

Stress analysis of the stiffened panel was carried out and maximum tensile stress was identified. Center longitudinal crack was initiated from rivet location of skin. Crack propagation was estimated by using stress intensity factor approach. Analytical Stress intensity factor calculations were carried out for various incremental cracks from 10 mm to 310 mm using 2mm and 3mm thick stiffeners. With 2mm thick stiffeners, the maximum value of stress intensity factor 71.8MPa$\sqrt{M}$ is observed at crack length of 300 mm. The value of stress intensity factor 71.36MPa$\sqrt{M}$ is observed at stiffener location. With 3mm thick stiffeners, the maximum stress intensity factor of 68.14MPa$\sqrt{M}$.

The accuracy of Modified virtual crack closure integral (MVCCI) method to determine SIF at crack tip is verified by employing the method to study the crack propagation in a rectangular plate. The results obtained using MVCCI method is compared with theoretical values based on LEFM approach.

From damage tolerance of the stiffened panel it can be concluded that:

1. The residual strength of the stiffened panel containing crack can be pre-dicted with reasonable accuracy using COD fracture criteria and finite element methods, provided the skin fracture toughness and stiffener ultimate strength values are known.

2. The riveted stiffened panel of practical design can be dimensioned such that, below a certain stress level, crack can be arrested between the two stiffeners.

3. By increasing the stiffener thickness, SIF at crack tip can be reduced by reducing the forces in the elements near crack tip. And thus, the residual strength of the skin can be improved.

FUTURE SCOPE OF THE WORK :

- To carry out the analysis by applying biaxial loading condition

- To carry out the damage tolerance analysis of the stiffened panel for different crack configurations

- To analyze the propagation of crack in the stiffened panel for different load spectrums of transport aircraft.

REFERENCES

[1] Dhruba J. Ghose, Natale S. Nappi and Christpher J. Weirnicki, "Residual strength of damaged marine structures", Ship structures committe, Report

SR-1341, Pages 39-48,September 1994.

[2] J. B. Ransom, E. H. Glaessgen, I. S. Raju and C. E.Harris, "Recent advances in Durability and Damage tolerance methodology", American Institute

of Aeronautics and Astronautics, at NASA Langley Research Centre (LaRC).

[3] P. M. S. T. de Castro, S. M. O. Tavares, V. Richter-Trummer, P. F. P. de Matos, P. M. G. P. Moreira and L. F. M. da Silva, "DAMAGE TOLERANCE

OF AIRCRAFT PANELS" Vol 18, Pg 35-46, 2010.

[4] A. Rama Chandra Murthy, G. S. Palani and Nagesh R. Iyer,"Damage tolerant evaluation of cracked stiffened panels under fatigue loading", Indian Academy of Sciences, Vol. 37, Part 1, pp. 171–186, February 2012.

[5] L. Wang, W.T. Chow, H. Kawai and S N Atluri, "Residual Strength of Aging Aircraft with Multiple Site Damage/Multiple Element Damage", AIAA Journal, Vol 36, No 5, May 1998.

[6] Prof. Ing. Antonin Pistek, CSc., Ing. Miroslav Pesak, " Optimization of Stiffened Panel with the help of Mathematical Programming Experimental Verification, Brno University of Technology, Institute of Aerospace Engineering, Brno, 616 69, Czech Republic, 26th International Congress of the Aeronautical Sciences, 2008.

[7] F. Carta and A. Pirondi, "Damage Tolerance Analysis of Aircraft Reinforced Panels", Industrial Engineering Department, University of Parma, v.le G.P. Usberti 181/A, 43124 Parma- Italy, 2011.

[8] Amardeep M. Bhiwapure, H. V. Lakshminarayana, K. K. Brahma, "Assessment of Cracked Stiffened Panels using Fracture Mechanics, Vol. VI, No.2, September 2007.

[9] Dr. Robert Tryon, "Structural Design Concepts: Overview of safe life and damage tolerance",Chief Technology Officer at VEXTEC,2013

[10] Evgeny Barkanov, "Introduction to finite element method",Institute of materials and structures, Faculty of Civil Engineering, Riga Technological University,2001

[11] Venkatesha B. K., Suresh B. S. and Girish K. E., "Analytical Evaluation ofFatigue Crack Arrest Capability in Fuselage of Large Transport Aircraft", Vol 1, Issue 1, 2012.