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Damage Tolerance Evaluation for Wing Structure with Large Cutout

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Abstract: *Damage tolerance evaluation of a wing box with large cutout in the bottom skin is carried out in steps. Stress analysis is first carried out to identify the critical location for fatigue crack initiation. A local analysis is carried out to obtain more accurate stress value and the distribution of stress. Finite element method is adopted for stress analysis and damage tolerance evaluation. Damage tolerance evaluation includes the stress intensity factor calculations at crack tip. This is carried out by simulation of the crack in the finite element model. Stress intensity factor (SIF) at different crack lengths is calculated using Modified Virtual Crack Closure Integral (MVCCI) method.*

Keywords: *Damage Tolerance Evaluation, Finite Element Method, Wing box, SIF, MVCCI method.*

I. INTRODUCTION

Wings are the lift generating components in the airframe structure. Wings are also used as fuel tanks in the transport aircraft. Cutouts are provided in the bottom skin of the wing to permit entry into the airplane fuel tanks for inspection or component repair. Bottom skin is under tension stress field during flight. Cutouts in the bottom skin will act as stress risers due to stress concentration effect. The high tensile stress locations are the most probable locations of fatigue cracking in the structure. The damage tolerance design philosophy says that cracks are allowed in the structure, but the cracks should not lead to catastrophic failure of the structure. The damage-tolerance evaluation of structure is intended to ensure that should fatigue, corrosion, or accidental damage occur within the LOV (Limit of validity) of the airplane, the remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected [1].

In the damage tolerance design philosophy the safety is ensured by inspection. Identification of the critical locations in the structure is most important to ensure the safety of the structure throughout the service life of the structure. Aircraft designer needs to ensure the structural integrity of the airframe without compromising on the safety of the structure. This would be possible only by adopting the damage tolerance design principles. The current project includes the stress analysis of a wing box of a medium size transport aircraft having large cutout in the bottom skin to identify the critical location for fatigue crack initiation. The aircraft under consideration is a conceptual Light Transport Aircraft. A local analysis is followed to obtain more accurate stress value and the distribution of stress. Aluminium alloy 2024-T351 Material is used for the wing box. Finite element method is adopted for stress analysis of the structural components. MSC NASTRAN and MSC PATRAN FEM packages are used to carry out the analysis. The maximum stress location is found from the wing box FE model (Global analysis). Damage tolerance evaluation includes the stress intensity factor calculations at crack tip. This is carried out by simulation of the crack in the finite element model of the bottom skin of the wing (Local analysis). Stress intensity factor (SIF) at different crack lengths is calculated using Modified Virtual Crack Closure Integral (MVCCI) method. The SIF calculated at every crack length is compared with the fracture toughness of the material. Variation of SIF as a function of crack length is plotted.

II. LITERATURE SURVEY

Damage tolerance philosophy is a refinement of the fail-safe philosophy. It assumes that cracks will exist, caused either by processing or by fatigue, and uses fracture mechanics analyses and tests to determine whether such cracks will grow large enough to produce failures before they are detected by periodic inspection. Three key items are needed for successful damage-tolerant design: residual strength, fatigue crack growth behaviour, and crack detection involving non-destructive inspection. Of course, environmental conditions, load history, statistical aspects, and safety factors must be incorporated in this methodology.[2]

The recent Air Force requirement to apply linear elastic fracture mechanics approach in damage tolerance design of aircraft structures, warrants the critical review of various approaches. Pir M. Toor [3] has critically reviewed some damage tolerance design approaches and their application to aircraft structures. The paper consists of three main sections: The first section reviews the residual strength analysis methodology, assumptions and limitations of each method are discussed through a simple example. The second part surveys the various crack propagation laws, including linear and non-linear ranges and spectrum loading effects. In the third and last section, fracture mechanics methodology is applied to several types of built-up structural components under spectrum loading conditions. The comparison of test results and analysis of complex structures indicate that simple methods of fracture mechanics can be applied to find the damage tolerant strength and rate of crack growth.

The measurement of energy release rates using virtual crack extensions was made using finite element techniques by T.K. Hellen [4]. Finite element techniques were presented for the accurate determination of stress intensity factors for brittle materials, and for estimating the direction of crack propagation in multi-mode loading systems. The techniques utilized energy differences over small changes in crack length, and used in conjunction with high order elements and special crack tip elements gave very accurate results. Alternatively, to obtain results of engineering accuracy, much coarser finite element meshes can be used than previously, reducing computer costs and data manipulation time. The method also enhanced the feasibility of three-dimensional analysis, where traditional methods required such mesh refinements along crack profiles as to be prohibitive under the computer limitations in 1970‘s.

An efficient technique for evaluating stress intensity factors using the Finite Element approach is presented by E. F. Rybicki and M. F. Kanninen [5]. The method, based on the crack closure integral, can be used with a constant strain finite element stress analysis and a coarse grid. The technique also permits evaluation of both Mode I and Mode II stress intensity factors from the results of a single analysis. In their work example computations are performed for a double cantilever beam test specimen, a finite width strip with a central crack, and a pin loaded circular hole with radial cracks. Close agreement between numerical results given by this approach and reference solutions were found in all cases.

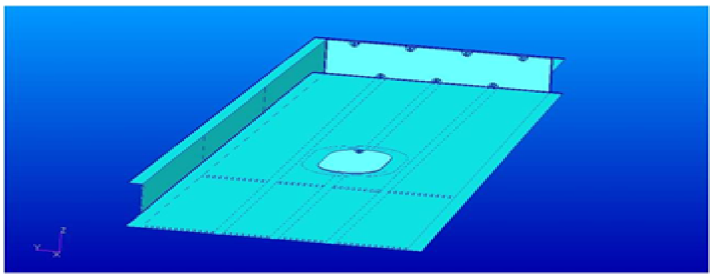
The methodologies for damage tolerant evaluation of stiffened panels under fatigue loading are presented by A Rama Chandra Murthy et al. [6]. The two major objectives of damage tolerant evaluation, namely, the remaining life prediction and residual strength evaluation of stiffened panels have been discussed in the paper. Concentric and eccentric stiffeners have been considered. MVCCI method is adopted for SIF calculations. From the studies, it has been observed that the predicted life is significantly higher with concentric and eccentric stiffener cases compared to the respective unstiffened cases. The percentage increase in life is relatively more in the case of concentric stiffener compared to that of eccentric stiffener case for the same stiffener size and moment of inertia.

Case studies on finite element based computation of strain energy release rate by modified crack closure integral were presented by R. Sethuraman and S.K. Maiti [7]. A modified crack closure integral method with square-root stress singularity elements is given for calculation of strain energy release rate for an in-plane extension of a crack. Case studies were presented to illustrate the improvement in accuracy. Results of case studies on a centre crack, an edge crack and a kinked crack were given to illustrate the effectiveness of the scheme.

III. ANALYSIS OF WING BOX

The wing box under consideration has a capsule shaped Fuel Access cutout in the bottom skin. The total span of the wing is 19330mm. The wings of the aircraft are attached at the bottom of the fuselage making it a Low-wing aircraft. Length of one whole wing is 9665mm. The wing box considered for analysis is 2388mm long and 2400mm away from the wing root. The wing box consists of 2 spars, 4 ribs, 3 stringers in the bottom skin and 4 stringers in the top skin.

Table1. Geometrical details of wing box

 **Fig. 3.1 Wing box (global model)**	Wing box length = 2882mm. Width of skin near at the root end= 1800mm. Width of skin near the free end = 1200mm. Thickness of bottom skin = 2.5mm. Cross-section of stringes- T ; Thickness= 1.5mm. Cross-section of ribs - I ; Thickness= 1.5mm. Cross-section of spar – C ; Thickness tapered – 2.75mm to 2mm from root end to free end. Rivets used dia- 5mm; Material used Al 2024 T351 alloy.

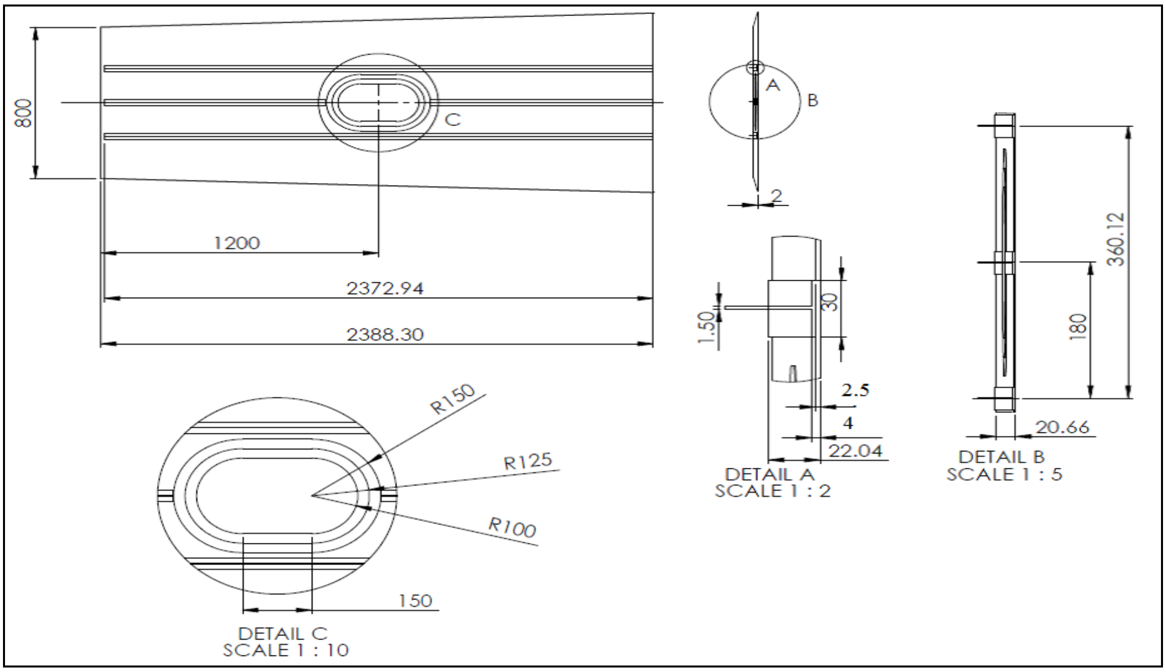


Fig.3.2 Bottom skin of wing box

3.1 Meshing

The CATIA model is imported and meshed in MSC PATRAN. The components other than rivets are meshed using 2D SHELL elements. The rivets are created using 1D BEAM elements. The quality of mesh is maintained so as to get the accurate value of stress. The meshed model of the wing box is shown in fig.3.3.

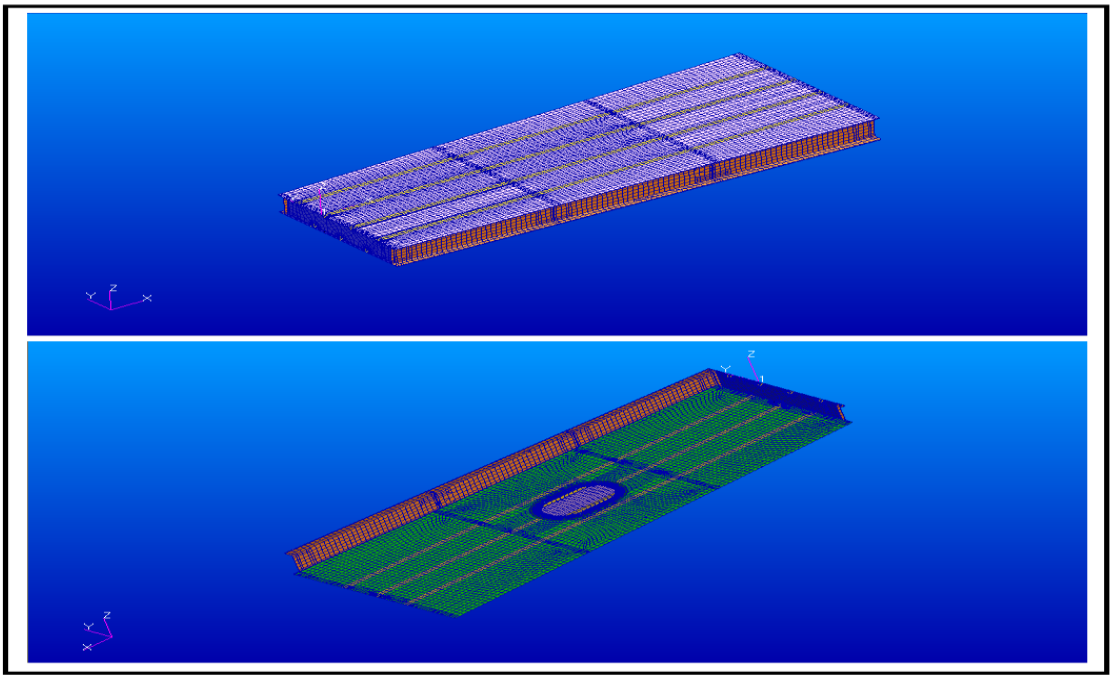


Fig.3.3 Meshed model of the wing box (global model)

3.2 Boundary Conditions (Wing box)

The wing can be treated as a cantilever beam with a tip load. The wing box root end can be considered as fixed end and the wing box tip end can be considered as free end. All six degrees-of-freedom of the free nodes at the wing box root end are constrained making it the fixed end. The aircraft is designed for 3g conditions to withstand loads thrice its weight. For 3g condition the aircraft is designed to withstand 10000kgf or 98100N. 80% of the lift load is taken by the wings. Hence one wing takes 40% of the lift load. In this case

40% of lift load is 40% of 10000, i.e, 4000kg or 39240N.The tip load is deduced using aerodynamic data called span factor and load factor. The tip load for the wing box is found to be 2384.109kgf or 23388.109N. This tip load is uniformly distributed over the perimeter of the wing box. The perimeter of wing box at the tip being 1730mm, the resultant tip load is converted to equivalent 'edge loads' (for application of load in PATRAN) by dividing the resultant tip load by the perimeter of the tip of wing box. The resulting edge load that is applied throughout the perimeter of the free end of wing box in PATRAN is 1.378kg/mm or 13.519N/mm.

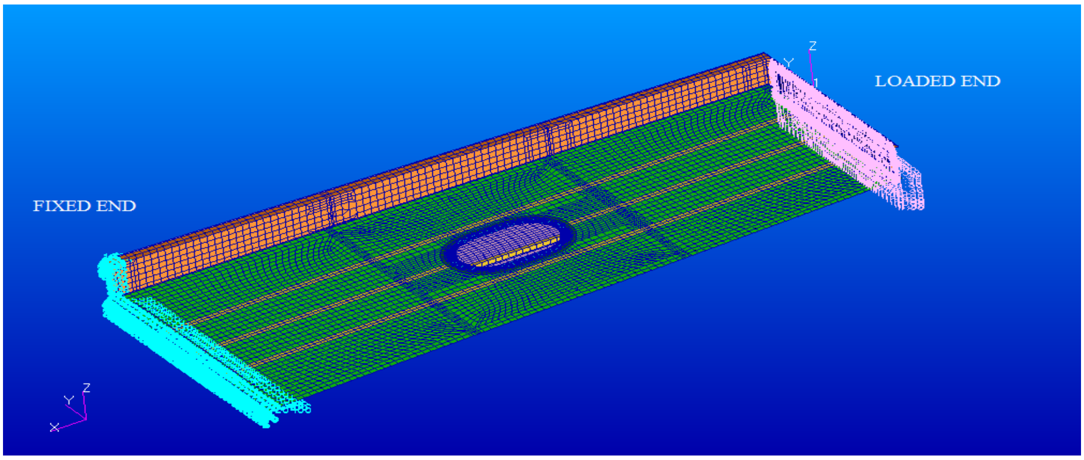


Fig.3.4 Boundary conditions of wing box (global model)

3.3 Maximum Stress in wing box

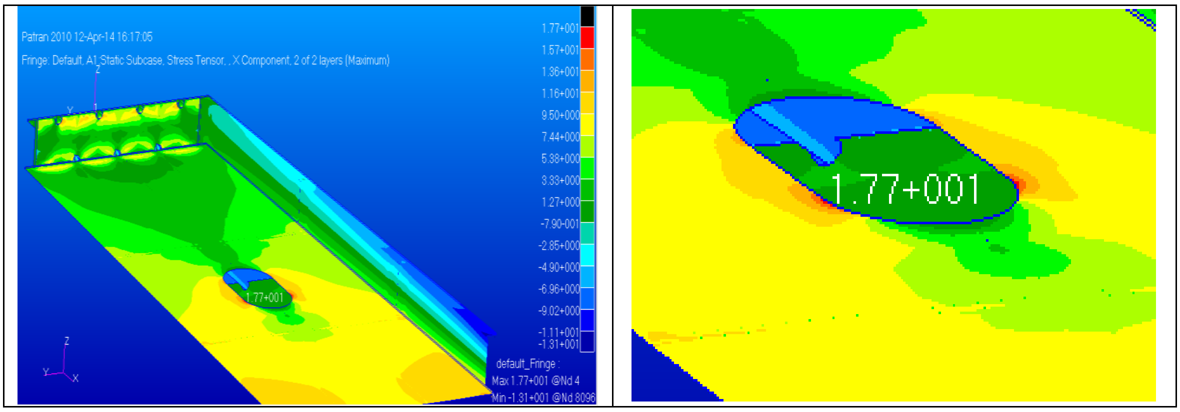


Fig.3.5 Maximum stress in wing box (global model)

The location of maximum stress in the wing box is marked in red fringes in figure. The stress is maximum in the bottom skin near the cutout. The value of stress is 17.7kg/mm^2 or 173.637N/mm^2.

3.4 Local Model

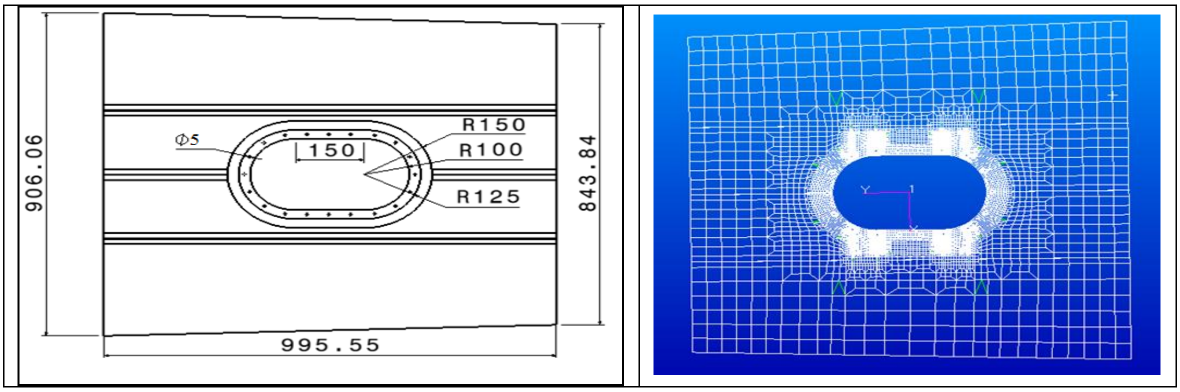


Fig.3.6 Geometry of local model; FE Model of local model

3.5 Boundary Conditions (Local model)

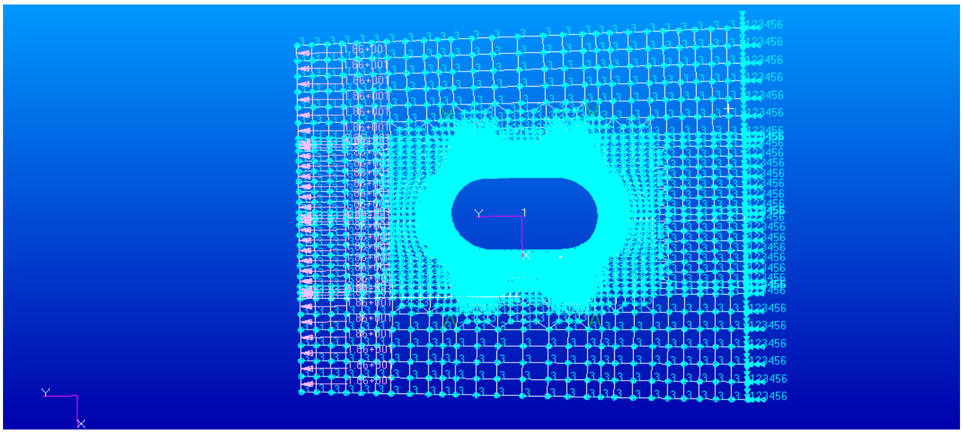


Fig.3.7 Boundary conditions of local model

The boundary conditions for the local model which is a segment of the bottom skin of the wing box can be deduced from the cantilever beam analogy. It is a well-known fact that when a cantilever beam is subjected to a tip load in the upward direction, the fibers in the bottom of the beam, i.e, the fibers below the neutral axis is under tension. The bottom most fibers experience the maximum tensile stresses whereas the top most fibers are under compression. The wing box can be considered as a cantilever beam with an upward load at the free end. Thus the bottom skin is under tension and the top skin is under compression. Hence the boundary condition for the local model can be considered as fixed at one end (near wing root) and tensile load acting on the other end (wingtip end). Also the translation in ‘z’ direction is constrained in order to ensure the application of pure tensile load. The loads can be summarized as follows:
Edge load on skin = 18.606 kg/mm = 182.52N/mm
Edge load on stringer’s flange = 18.606 + 9.997 = 28.603 kg/mm= 280.60 N/mm
Edge load on stringer’s web = 9.997 kg/mm = 98.07N/mm

3.6 Maximum stress in local model

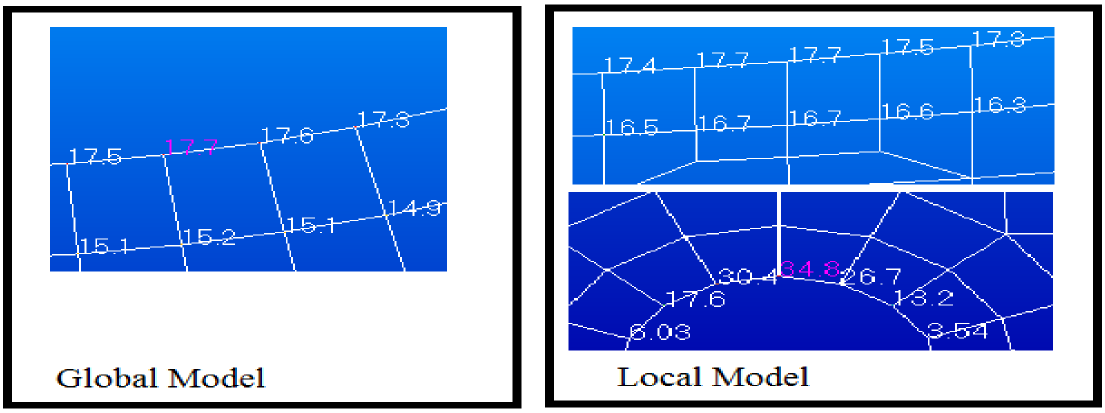


Fig.3.8 Comparison of stress values from global and local model

The figure 3.8 shows the Maximum stress induced in the global model (17.7 kg/mm2) with a different colour in the global model box. The stress is replicated in the local model as we can see in the local model box at the same location in capsule cutout (top picture in local model box). The local model box in the figure also shows the maximum stress induced in the rivet hole. Thus the maximum stress in the local model is 34.8 kg/mm2 i.e, 341.388 N/mm2 (MPa). The location of this maximum stress is in a rivet hole near to the fixed end. Precisely at the rivet hole near the point, where the curvature of the semi-circle portion of the capsule cutout changes to zero. In other words, the point where the semi-circular curvature of the capsule cutout starts becoming a straight line. This is location where the initiation of a fatigue crack is most imminent.

3.7 MVCCI

Modified Virtual Crack Closure Integral (MVCCI) is a finite element based computation of strain energy release rate. By knowing the strain energy release rate we can calculate the Stress Intensity Factor (SIF). Using the MVCCI method, the SIF is calculated as follows:

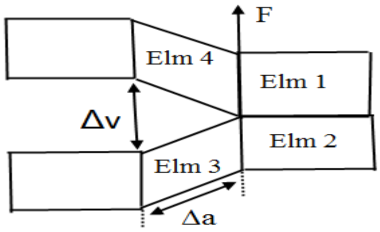


Fig.3.9 Nomenclature for MVCCI method

Stress Intensity Factor (SIF) K = $\sqrt{G \times E}$

Where,

G is the strain energy release rate, E is the Young's modulus

The strain energy release rate, 'G' is calculated by the formula:

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

Where,

G = Strain energy release rate, Δa = elemental edge length at crack tip, Δv = differential displacement of opening node, F = grid point force at crack tip, t = thickness of plate at crack tip

3.8 SIF calculations

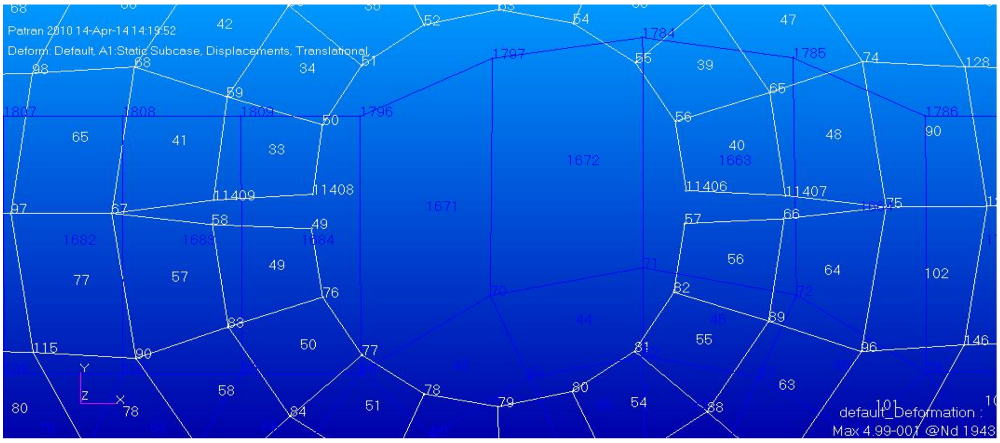


Fig.3.10 A simulated 5mm crack near the rivet hole

Table 2 FE data for SIF calculation for 5mm crack

5 mm Crack						
Displaced nodes	Displacement of displaced nodes T2 mm	Differential Displacement Δv mm	GP Force at node and Elements		GP Force kg	Total Force F kg
			node	Element		
66	0.42808260	0.02452010	75	48	53.57285	142.94
11407	0.45260270			90	89.37	

Element edge length is maintained at 1.25mm and the thickness at the tip of this crack is 3.5mm
F = 142.94 kg
SIF calculations in S.I. units:
F = 142.94 × 9.81 = 1402.241 N

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

$$G = \frac{1}{2 \times 1.25 \times 10^{-3}} \times 0.025 \times 10^{-3} \times \frac{1402.241}{3.5 \times 10^{-3}}$$

$$G = 3926.27\ \mathrm{N/m}$$

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

The Young's modulus of Aluminium 2024- T351 is 72.4 GPa (72.4×10^{9} N/m^2)

$$K = \sqrt{3926.27 \times 72 \times 10^{9}} = 16.87\ \mathrm{MPa}\sqrt{\mathrm{m}}$$

Therefore the Stress Intensity Factor (SIF) in the presence of 5mm crack is $16.87\ \mathrm{MPa}\sqrt{\mathrm{m}}$
Similarly the SIF calculations of the incremental crack lengths can be calculated.

IV. Results And Discussions

SIF for all lengths of crack is tabulated below:

Table 3 SIF for all lengths of cracks

Crack length mm	Thickness 't' mm	Elemental Edge Length 'Δa' mm	Differential Displacement 'Δv' mm	Grid Point Force Total 'F' N	Strain Energy Release Rate 'G' J/m^2	Stress Intensity Factor 'K' $\mathrm{MPa}\sqrt{\mathrm{m}}$
5	3.5	1.25	0.02452010	1402.24	3929.49	16.87
10	3.5	1.25	0.03079650	1737.15	6114.07	21.04
15	3.5	1.25	0.03802010	2110.92	9172.27	25.77
20	3.5	1.25	0.06213050	3153.33	22390.63	40.26
25	3.0	1.25	0.06652280	3082.99	27345.22	44.49
30	3.0	1.25	0.06912730	3199.63	29490.90	46.21
35	3.0	1.25	0.07167760	3315.98	31690.87	47.90
40	3.0	1.25	0.07431560	3435.46	34041.10	49.64
45	3.0	1.25	0.07943630	3399.95	36010.59	51.06
50	2.5	1.25	0.08055130	3102.71	39988.37	53.81
55	2.5	1.25	0.07874510	3029.13	38164.66	52.57
56.25	2.5	1.25	0.07788460	2992.93	37296.50	51.96
57.50	2.5	1.25	0.07641850	2922.40	35732.07	50.86
58.75	2.5	1.25	0.07316050	2780.06	32542.49	48.54
60	4.0	1.25	0.07058100	3922.33	27684.20	44.77

Fig. 4.1 Variation of SIF as a function of crack length

Table 4 Comparison of SIF with Fracture Toughness

Crack length mm	Thickness 't' mm	Fracture Toughness 'K_{IC}' $\mathrm{MPa}\sqrt{\mathrm{m}}$	Stress Intensity Factor 'K' $\mathrm{MPa}\sqrt{\mathrm{m}}$
5	3.5	95	16.87
10	3.5	95	21.04
15	3.5	95	25.77
20	3.5	95	40.26

25	3.0	97	44.49
30	3.0	97	46.21
35	3.0	97	47.90
40	3.0	97	49.64
45	3.0	97	51.06
50	2.5	98	53.81
55	2.5	98	52.57
56.25	2.5	98	51.96
57.50	2.5	98	50.86
58.75	2.5	98	48.54
60	4.0	93	44.77

It can be noted from the table that the SIF at any crack length does not exceed the fracture toughness. Thus the structure is safe. However maintenance must be carried out once the crack tip reaches the stringer (stiffener).This is to avoid the damage to stringer.

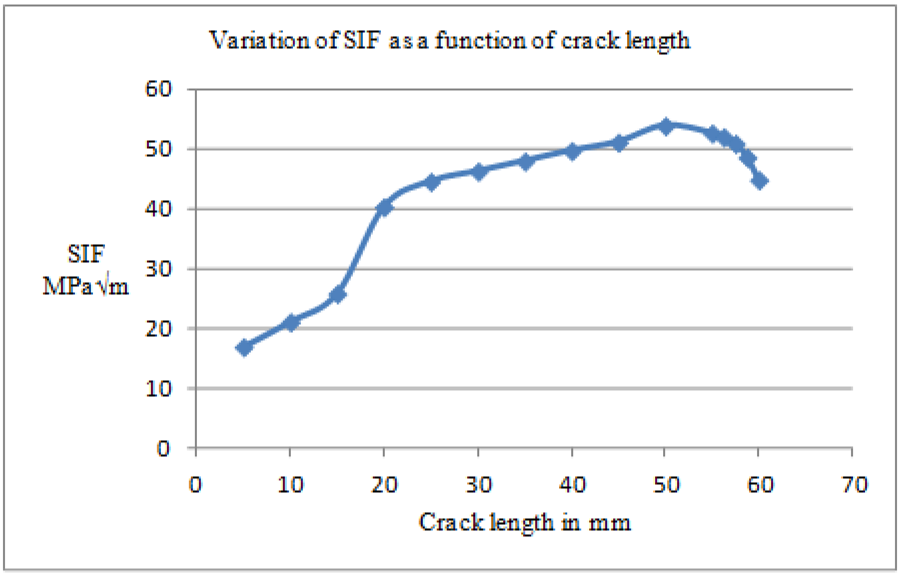


Fig 4.2 Variation of SIF as a function of crack length

The variation of SIF (Stress Intensity Factor) as a function of crack length is shown in figure. From that curvature of the graph it is evident that the SIF initially increases with the increasing length of the crack. The highest SIF value observed is 53.81 MPa√m corresponding to a crack length of 50mm. When the crack extends beyond 50mm the SIF starts decreasing. This is because of the presence of stiffener (stringer) in the crack path. The stringer in the bottom skin is placed 60mm away from the cutout. Hence when the crack extends beyond 50mm, the crack tip is getting closer to the stiffener (stringer). The displacement at crack tip reduces near the stiffener, which results in decrease in strain energy release rate. Thus the SIF starts decreasing as the crack tip approaches the stringer.

Calculation of Residual strength:

The residual strength capability is defined as the amount of static strength available at any time during the service exposure period considering that damage is initially present and grows as a function of service exposure time. Residual strength is the load or force that a damaged structure or material can still carry without failing. Residual strength is calculated as follows:

$$\sigma_{residual} = \frac{K_{IC}}{K} \times \sigma_{remote}$$

Where,

$\sigma_{residual}$= Residual Strength

σ_{remote} = Remotely applied Tensile stress

K_{IC} = Fracture Toughness

K = Stress Intensity Factor

$$\sigma_{remote} = \frac{15704.3}{844 \times 2.5} = 7.44\ \mathrm{kg/mm^2}$$

$$\sigma_{remote} = 72.99\ \mathrm{N/mm^2}$$

For 5mm crack:

$$\sigma_{residual} = \frac{95}{16.87} \times 72.99 = 411.03\mathrm{N/mm^2}$$

Similarly for other crack lengths residual strength is calculated and plotted in fig 4.3

It is evident from the figure that the residual strength decreases with increase in crack length. The residual strength slightly increases near the stiffener. Moreover it interrupts the decreasing trend of the residual strength. If the stringers are not provided, the crack propagates rapidly, the residual strength keeps decreasing and results in catastrophic failure of the wing. Thus the use of stingers is an effective means of increasing the residual strength of damaged panels.

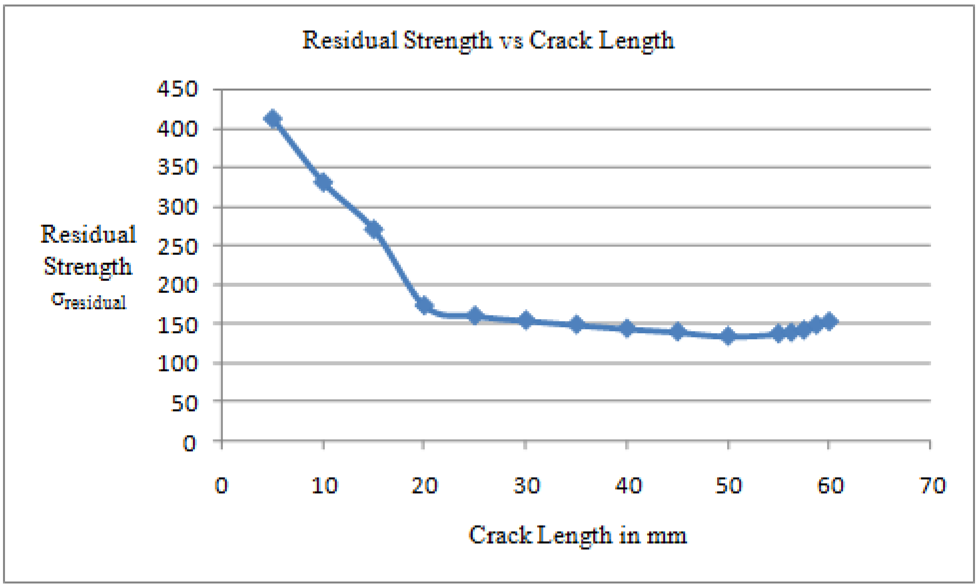


Fig 4.3 Variation of residual strength as a function of crack length

The effectiveness of MVCCI method is illustrated by comparing the SIFs obtained from MVCCI method with the SIFs obtained from the conventional (analytical) method for a plate with centre crack. A plate of length 600mm and breadth 400mm is considered for MVCCI method. Its Young's modulus is considered to be 70GPa. It is subjected to a tensile load of 9810N. The thickness of plate is 2mm. By conventional approach the value of SIF for a various crack lengths is calculated and compared with values from MVCCI method as follows:

Table 5 Comparison of SIF from analytical method with SIF from FE method

Crack Length mm	Stress Intensity Factor (Analytical) 'K_I'	Stress Intensity Factor (FEA) MVCCI 'K'
20	2.17	2.14
30	2.66	2.65
40	3.08	3.08
50	3.46	3.47
60	3.80	3.82

V. CONCLUSIONS

In this work, the damage tolerance evaluation of wing box of a medium size transport aircraft having a large cutout in the bottom skin has been carried out. Finite element method was adopted for stress analysis of the structural components. MSC NASTRAN and MSC PATRAN FEM packages were used to carry out the analysis. The damage tolerance evaluation was carried out in steps. The stress analysis of the entire wing box under the 3g lift load condition was first carried out to identify the critical location for fatigue crack initiation. The critical location was found to be in the bottom skin of the wing. A local analysis was followed to obtain more accurate stress value and the distribution of stress. The maximum stress from the global model was simulated in the local model and at the same location as in the global model. The anomalies in the simulation could be neglected since the maximum stress and its location from global model was replicated in the local model. Hence it was considered sufficient that, only the portion where maximum stress from global model was replicated be considered for analysis. The rest was neglected using the "Plot-Erase" option.

The maximum stress in the local model was found near a rivet hole, close to the fixed end of the wing box. The stress in the rivet hole was found to be 341.388 N/mm^2 (MPa). Since it is lesser than the tensile yield strength (The tensile yield strength of Aluminium 2024- T351 it is 345 MPa) the wing box can be considered safe for static strength under 3g loading conditions. The damage tolerance evaluation includes stress intensity factor and residual strength calculations for various crack lengths. The cracks of incremental length were simulated in the local model. The SIF was calculated for each crack length using Modified Virtual Crack Closure Integral (MVCCI) method which uses FE data like nodal displacements and grid point forces. The SIFs so calculated was compared with fracture toughness.

The SIF initially increased with the increasing length of the crack. When the crack extended beyond 50mm the SIF started decreasing. This is because of the presence of stiffener (stringer) in the crack path. The stringer in the bottom skin is placed 60mm away from the cutout. Hence when the crack extends beyond 50mm, the crack tip is getting closer to the stiffener (stringer). The displacement at crack tip reduces near the stiffener, which results in decrease in strain energy release rate. Thus the SIF starts decreasing as the crack tip approaches the stringer. For all lengths of crack the SIF remained below the fracture toughness. Thus the structure is safe. However maintenance must be carried out once the crack tip extends nearer to the stringer (for 58.75mm). This is to avoid the damage to stringer.

The residual strength calculations revealed that the residual strength decreases with increase in crack length. The residual strength slightly increases near the stiffener. Moreover it interrupts the decreasing trend of the residual strength. If the stringers are not provided, the crack may propagate rapidly, thus the residual strength keeps decreasing and results in catastrophic failure of the wing. Thus the use of stingers is an effective means of increasing the residual strength of damaged panels. Also it is shown that for a rectangular plate there is a close agreement between the SIF values from analytical method and the SIF values from MVCCI (Finite Element) method. Thus MVCCI method is effective for determination of Stress Intensity Factors of cracked structures. Hence it can be concluded that the results obtained from the MVCCI method for the wing box is in close agreement with the actual SIF.

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