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Finite Element Modeling for Fracture Mechanics Analysis of Aircraft Fuselage Structure

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***Abstract*—Aircraft fuselage is a very complex structure with relatively low margin of safety, operating in highly dynamic environment. A small crack in the fuselage will lead to the catastrophic failure of aircraft. Hence fracture mechanics analysis and damage tolerance design methodology is generally accepted as an essential tool for predicting and validating fatigue crack growth in aircraft fuselage structure. Investigation of mixed mode fracture of aircraft fuselage with arbitrarily oriented through wall crack is the focus of the present study. Development of refined Finite Element Model and determination of Normalized stress intensity factor using ANSYS and special purpose post-processing subprogram 3MBSIF is presented in this paper. The methodology is validated using benchmark, a standard test problem with known target solution. Parametric study is carried out to quantify the effect of crack orientation and crack length on the stress intensity factors. Significant numerical results are presented and discussed.**

Keywords—Aircraft Fuselage; Stress Intensity Factor; Mixed Mode Fracture.

I. INTRODUCTION

Fracture is a failure mode due to unstable crack propagation resulting from applied stress. Fracture Mechanics provides a methodology for prediction, prevention, and control of Fracture in materials, components, and structures. A critical assessment of structural integrity (Stiffness, Strength, and Durability) is often based on fracture mechanics analysis. The aircraft fuselage skin carries cabin pressure and shear loads. Longitudinal stringers carry the longitudinal tension and compression loads. Circumferential frames maintain the fuselage shape and redistributes loads in to the skin. Bulkheads carry concentrated loads. The loading condition is so complex that a small crack in the fuselage will lead to catastrophic failure of aircraft. Hence fracture mechanics analysis and damage tolerance design methodology is generally accepted as an essential tool for predicting and validating fatigue crack growth in aircraft fuselage structure.

II. REVIEW OF RELATED RESEARCH

Many research studies have been carried out to analyze the fatigue crack growth in the aircraft fuselage structure. This section sum up the few research work done in the field of fracture mechanics analysis of aircraft fuselage structure over the past years.

S.M.O. Tavares et al. [1] carried out numerical analysis to determine stress intensity factors around longitudinal crack in an aircraft fuselage reinforced with stiffeners and frames subjected to internal pressure. The problem is solved using linear elastic fracture mechanics approach with geometric nonlinearity is taken in to account. Modified virtual crack closure technique is used to determine the stress intensity factor. Good agreement between numerical results and analytical results have found. It is shown that bulging of fuselage skin induces a considerable variation of the SIF along the thickness for large crack length.

Charles E. Harris et al. [2] has developed analysis methodology to predict the onset of widespread fatigue damage in lap joints of fuselage structure based on experimental database. The study is about assembling of extensive experimental database from very detailed teardown examination of fatigue cracks found in the rivet holes of several fuselage structural components. Based on this experimental data analysis methodology is developed, in which complicated aspects like residual stress due to riveted interference fit is taken in to account. These complicated aspects could be simplified without a significant loss in computational accuracy.

J.C. Newman, Jr. [3] has reviewed some of the advances made in stress analyses of cracked aircraft components, in the understanding of the fatigue and fatigue crack growth process, and prediction of residual strength of complex aircraft structures with widespread fatigue damage. The study is mainly based on the small crack behavior at open and riveted loaded holes and the development of small crack theory has led to the prediction of stress life behavior for components with stress concentrations under aircraft spectrum loading.

Anisur Rahman et al. [4] has carried out study on effect of bulging of aircraft fuselage on stress intensity factor. In the study both unstiffened and stiffened longitudinal lap joints fuselage panel subjected to internal pressure is considered. Modified crack closure integral method was used to calculate crack tip stress intensity factor. For short cracks, near constant response was obtained for the bulging factor as a function of the applied pressure and the presence of stiffeners slightly reduced the bulging factor for the shorter cracks. For longer cracks, the bulging factor varied nonlinearly as a applied pressure. The presence of the stiffeners significantly reduced the bulging factor, but not the level that bulging can be neglected.

III. PROBLEM STATEMENT

The focus of this study is on finite element modeling for computational fracture mechanics and its application to determine mixed mode Stress Intensity Factors and their variations along an arbitrarily oriented through wall crack front in an aircraft fuselage subjected to uniform internal pressure. Specifically aircraft fuselage skin of 3300mm diameter and 1.2mm thickness, reinforced with stiffeners and frames with an arbitrarily oriented surface crack subjected to internal pressure as shown in Fig 1. is considered for the present study. The fuselage is subjected to hoop stress, which is used as a reference stress ($\sigma=Pr/t$). This value is used to normalize the computed SIF's.

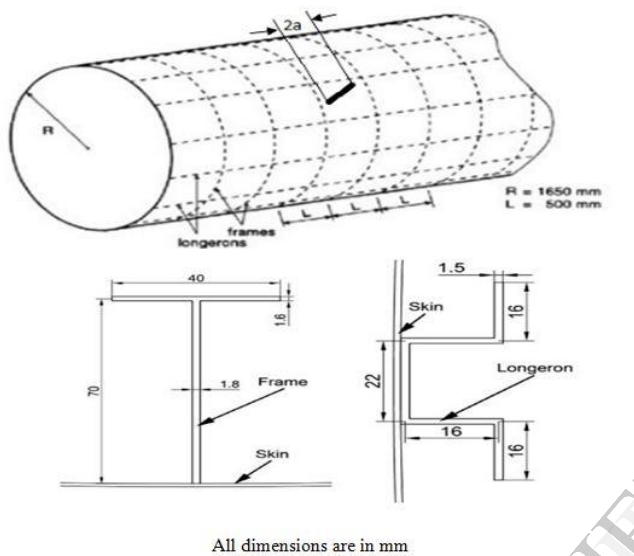

Fig1:Geometric Details of Aircraft Fuselage with Arbitrarily Oriented Crack

Table 1. Mechanical Properties of the Specimen

Young's Modulus, E	71000Mpa
Poisson's Ratio, υ	0.33
Tensile Yield Strength	280Mpa
Ultimate Tensile Strength	310Mpa

IV. AIM AND OBJECTIVE

Determination of mixed mode Stress Intensity Factors (K_I and K_{II}) and their variation with crack orientation for various values of crack length '2a' is the overall aim of this investigation. The following objectives have to be met in the sequel.

- Stress analysis of an aircraft fuselage under internal pressure to locate regions of crack initiation.
- Finite Element model development of the aircraft fuselage with arbitrarily oriented crack for computational fracture mechanics using ANSYS Workbench.
- Determination of mixed mode (Mode I and Mode II) Stress Intensity Factors (Membrane and Bending) using ANSYS and Special Purpose Post Processing Subprogram (3MBSIF).
- Validation of finite element model developed using benchmark.
- Parametric studies on shell structure to quantify the effect of crack length for arbitrarily oriented crack.
- Prediction of direction of crack growth and critical pressure at which crack growth occurs for axial and arbitrarily oriented crack.

V. FINITE ELEMENT MODEL DEVELOPMENT

The model is meshed suitably using SHELL281 element in ANSYS Workbench as shown in Fig 2. and Fig 3. The total elements and nodes used to discretize the aircraft fuselage are 51702 and 123845 respectively. The total of 72 singularity elements are generated around each crack tip, thus maintaining the singularity element angle 5^0 as shown in Fig 4. and Fig 5.

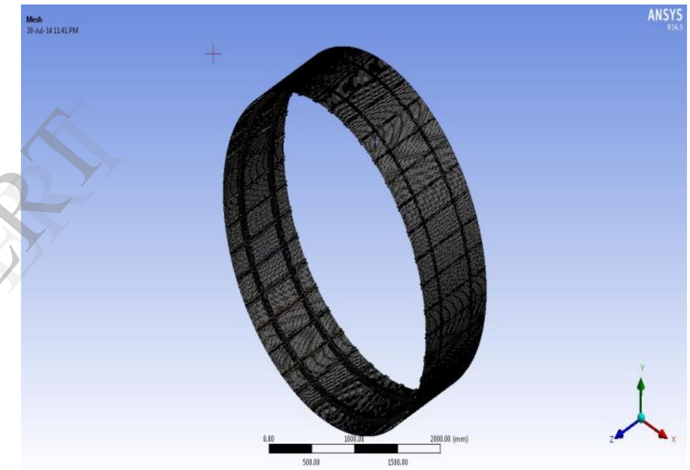

Fig 2: Finite Element Model of Aircraft Fuselage

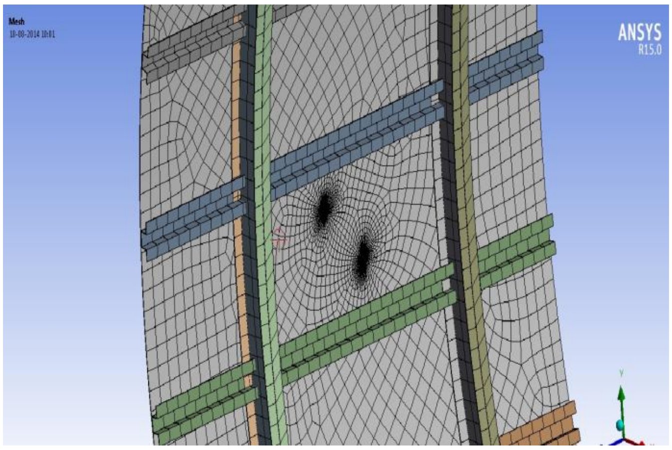

Fig 3: Enlarged View Finite Element Model of Skin, Stiffeners, and Frames with Crack

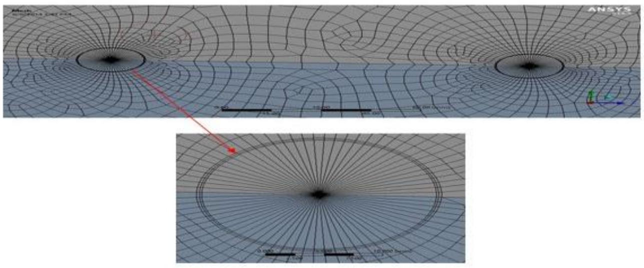

Fig 4: Finite Element Model of Aircraft Fuselage with an Axial Crack and Singular Elements around the Crack Tip (NS=72)

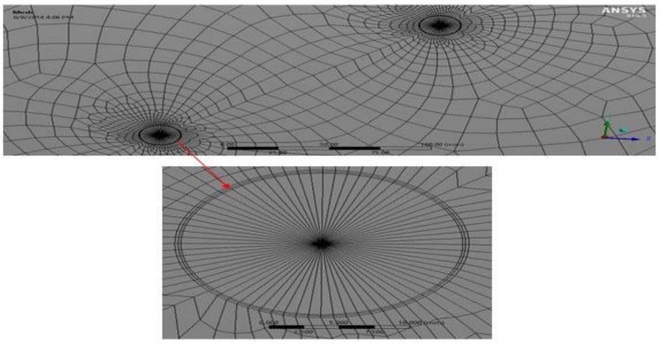

Fig 5: Finite Element Model of Aircraft Fuselage with an Arbitrarily Oriented Crack and Singular Elements around the Crack Tip (NS=72)

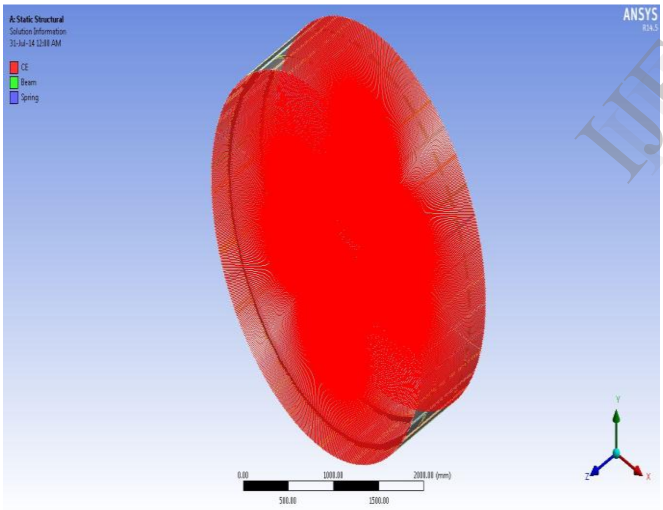

Fig 6: Boundary Condition

Rigid link elements are used as shown in Fig 5. to constrain the aircraft fuselage. These rigid link elements enforce kinematic relationships between the displacements at two or more nodes in the analysis. The rigid link element is defined by specifying the master node and the slave node.

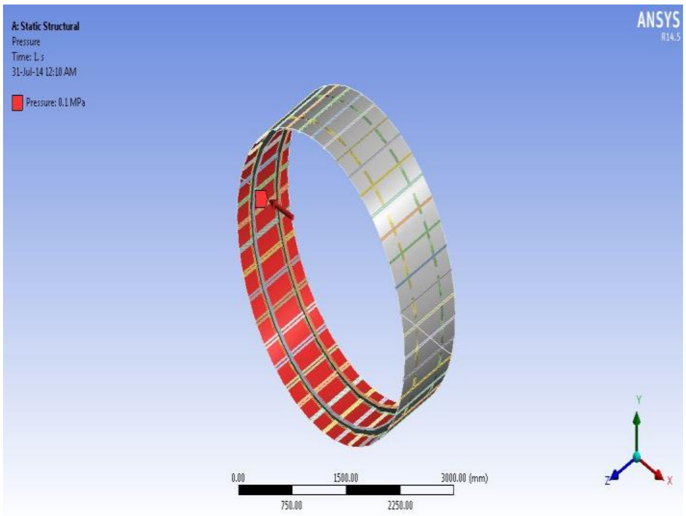

Fig 7: Load Application

The aircraft fuselage with axial and arbitrarily oriented crack is subjected to internal pressure of 0.1Mpa, as shown in Fig 7.

VI. FINITE ELEMENT MODEL VALIDATION

A benchmark is a standard test problem with known target solution in the form of formulae/graphs/tables. These are used to validate finite element models developed using ANSYS and stress intensity factors calculated using ANSYS (KCalc) and 3MBSIF.

Table 2. Normalized Mode-I SIF for ϴ=0^0 (Plane Stress Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS	Target
	K_{Io}^M	K_{Io}^B	$K_{Io} = K_{Io}^M + K_{Io}^B$	K_{Io}	K_{Io}
0.4	1.055	0.0003	1.0553	0.932	0.957
0.6	1.127	0.0013	1.1283	1.035	1.105
0.8	1.234	0.0016	1.2356	1.161	1.268
1	1.347	0.0021	1.3491	1.293	1.405
1.2	1.474	0.0024	1.4764	1.417	1.526
1.4	1.635	0.0035	1.6385	1.556	1.626

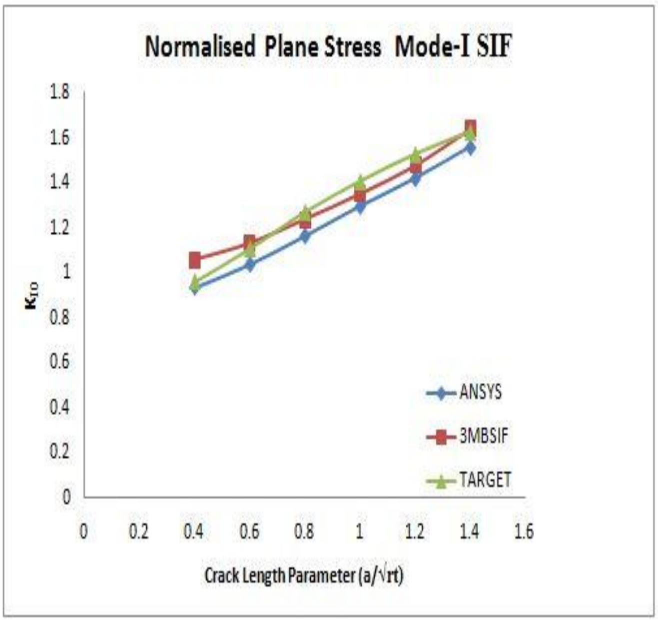

Fig 8: Normalized Mode-I Plane Stress SIF for ϴ=0^0 vs Crack Length Parameter

Table 3. Normalized Mode-I SIF for ϴ=0^0 (Plane Strain Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS
	K_{Io}^M	K_{Io}^B	$K_{Io} = K_{Io}^M + K_{Io}^B$	K_{Io}
0.4	1.1336	0.0004	1.134	1.046
0.6	1.2505	0.0015	1.252	1.161
0.8	1.3853	0.0018	1.3871	1.303
1	1.5027	0.0024	1.5051	1.451
1.2	1.6474	0.0027	1.6501	1.59
1.4	1.7872	0.0039	1.7911	1.747

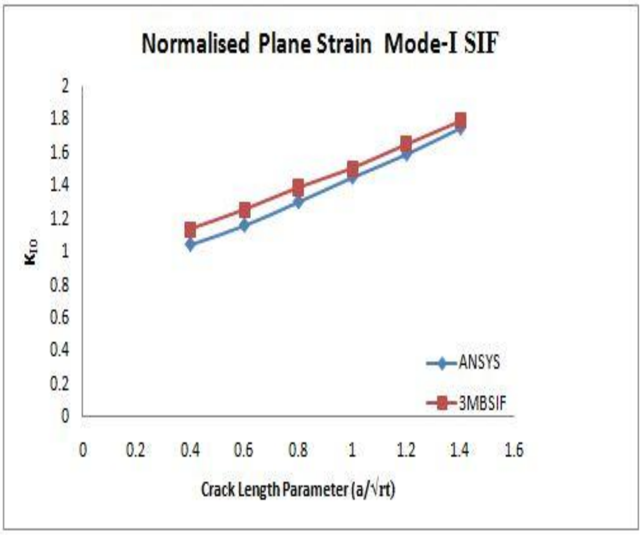

Fig 9: Normalized Mode-I Plane Strain SIF for ϴ=0^0 vs Crack Length Parameter

Observation: It can be observed that Normalized Mode-I stress intensity factor obtained from ANSYS and 3MBSIF for plane stress assumptions are very closely matches with the target solution.

a. Contour Plots

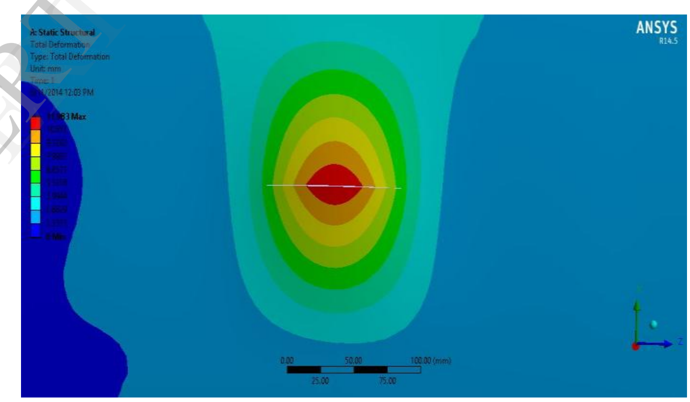

Fig 10: Total Displacement (ϴ=0^0)

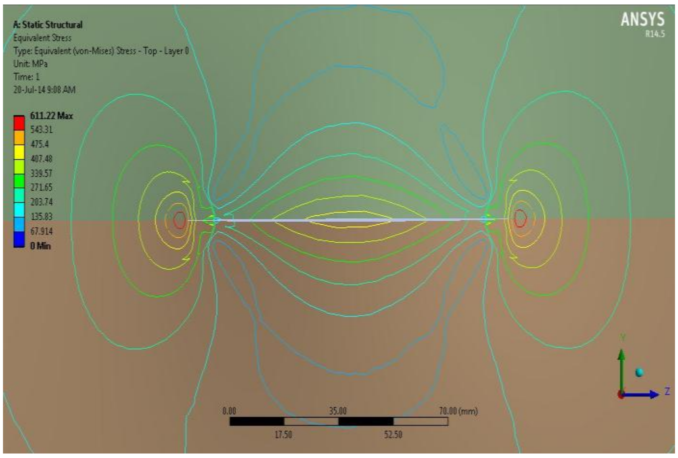

Fig 11: von-Mises Stress: Top Layer (ϴ=0^0)

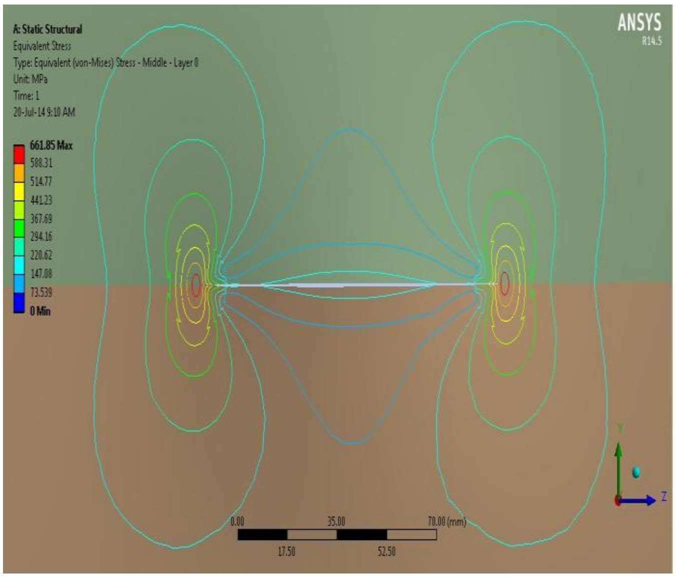

Fig 12: von-Mises Stress: Middle Layer ($\Theta=0^0$)

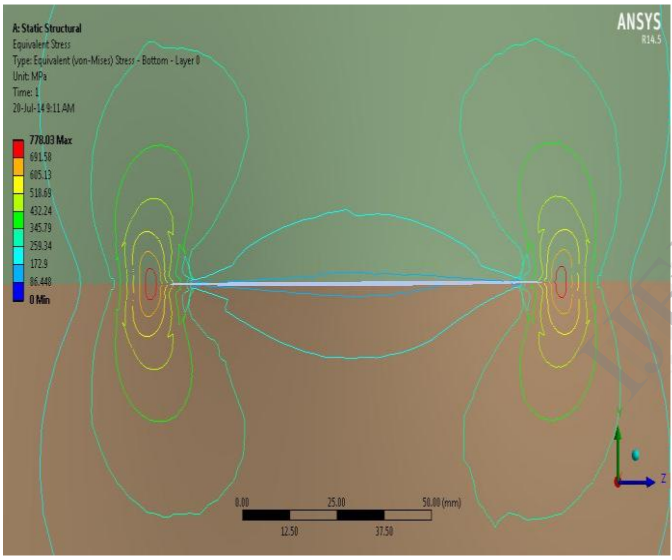

Fig 13: von-Mises Stress: Bottom Layer ($\Theta=0^0$)

VII. CASE STUDY

The structural integrity of the aircraft fuselage with arbitrarily oriented crack is studied with the help of Finite Element Method by varying the crack length parameter and maintaining only 45^0 crack angle.

a. Stress Analysis

The stress analysis of the aircraft fuselage without cracks is carried out to determine the critical location of crack initiation. The ends of the fuselage are coupled with rigid links and all degrees of freedom constrained. The fuselage is pressurized with an applied internal pressure of 0.1MPa. The von-Mises plot of the fuselage is shown in Fig 14. The deep red region in the figure is the critical location for initiation and growth of the crack. Also the bulging of fuselage skin is observed in between the stiffeners and the frame as shown in Fig 15.

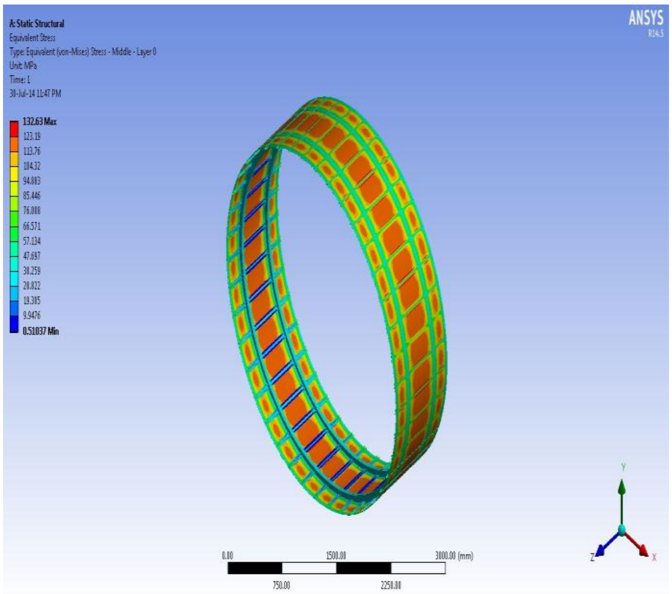

Fig 14: von-Mises Stress Contour Plot (Aircraft Fuselage with out Crack)

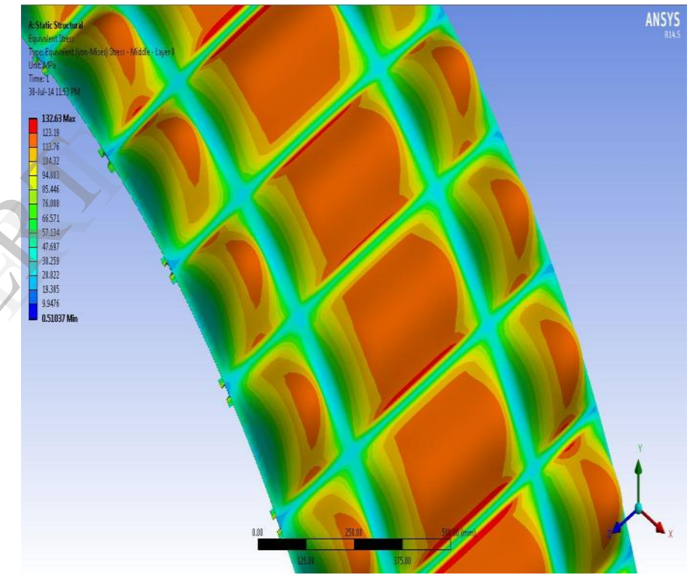

Fig 15: Deformed Shape Details, Displacement Scale Factor 27×

b. Case Study

The aircraft fuselage with arbitrarily oriented crack of differed length is analyzed. The parameter 'a/√rt' is introduced, which governs the crack length.

Where,

a= Half Crack Length (mm)

r= Radius of Fuselage Skin (mm)

t= Thickness of Skin (mm)

The crack length parameter is varied from 0.4 to 1.4. The input parameters are shown in table 4. The stress intensity factors for varying crack lengths are shown in Fig 16 to 19. The Mixed Mode SIFs at crack tip is obtained from Finite Element method using ANSYS and 3MBSIF program. The

SIFs are evaluated for plane stress and plane strain assumptions.

Table 4: Input Parameters

Input Parameters			
Crack Length Parameter '$a/\sqrt{rt}$'	Half Crack Length 'a' (mm)	Crack Angle 'α' (degrees)	Applied Internal Pressure 'P' (MPa)
0.4	17.798	45	0.1
0.6	26.698	45	0.1
0.8	35.597	45	0.1
1	44.497	45	0.1
1.2	53.396	45	0.1
1.4	62.296	45	0.1

Table 5: Normalized Mode-I SIF for Ө=45^0 (Plane Stress Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS
	K_{Io}^M	K_{Io}^B	$K_{Io} = K_{Io}^M + K_{Io}^B$	K_{Io}
0.4	0.8687	0.000376	0.869	0.8299
0.6	0.9456	0.000788	0.946	0.8957
0.8	1.0027	0.00137	1.0041	0.9657
1	1.0403	0.00212	1.0424	1.0181
1.2	1.0526	0.00277	1.0554	1.0307
1.4	0.9285	0.00361	0.9321	0.9221

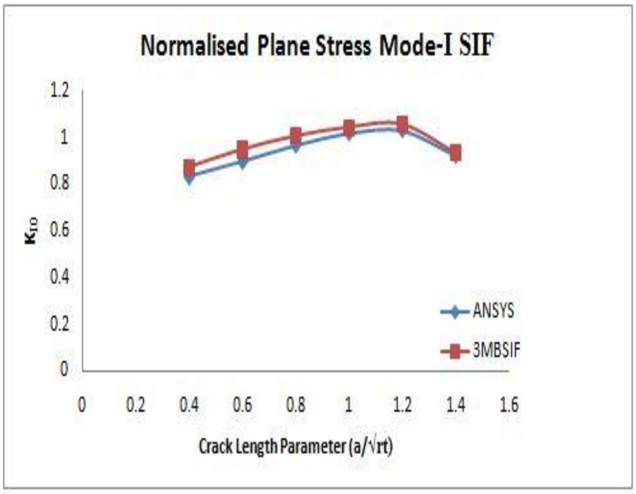

Fig 16: Normalized Mode-I Plane Stress SIF for Ө=45^0 vs Crack Length Parameter

Table 6: Normalized Mode-I SIF for Ө=45^0 (Plane Strain Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS
	K_{Io}^M	K_{Io}^B	$K_{Io} = K_{Io}^M + K_{Io}^B$	K_{Io}
0.4	0.9753	0.0005	0.9758	0.9314
0.6	1.0273	0.0009	1.0282	1.0052
0.8	1.1116	0.0019	1.1135	1.0838
1	1.1665	0.0024	1.1689	1.1425
1.2	1.1754	0.0028	1.1782	1.1567
1.4	1.0352	0.0038	1.039	1.0348

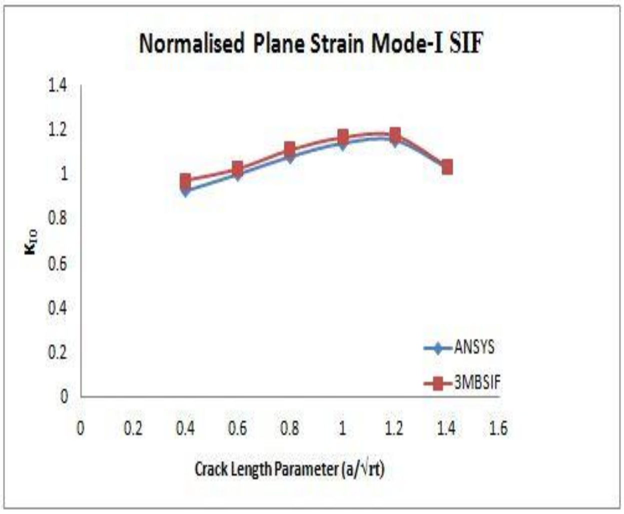

Fig 17: Normalized Mode-I Plane Strain SIF for Ө=45^0 vs Crack Length Parameter

Table 7: Normalized Mode-II SIF for ϴ=45^0 (Plane Stress Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS
	K_{IIo}^M	K_{IIo}^B	$K_{IIo}=K_{IIo}^M+K_{IIo}^B$	K_{IIo}
0.4	0.1962	0.00017	0.1964	0.1776
0.6	0.3156	0.0002	0.3158	0.2887
0.8	0.3421	0.0005	0.3426	0.3281
1	0.3905	0.00087	0.3914	0.3637
1.2	0.4144	0.0011	0.4155	0.3893
1.4	0.4286	0.0015	0.4301	0.4108

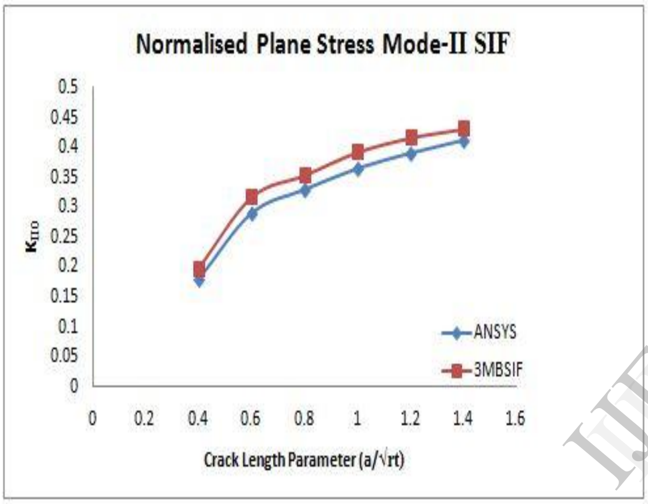

Fig 18: Normalized Mode-II Plane Stress SIF for ϴ=45^0 vs Crack Length Parameter

Table 8: Normalized Mode-II SIF for ϴ=45^0 (Plane Strain Assumption)

$\frac{a}{\sqrt{rt}}$	3MBSIF			ANSYS
	K_{IIo}^M	K_{IIo}^B	$K_{IIo}=K_{IIo}^M+K_{IIo}^B$	K_{IIo}
0.4	0.2512	0.0002	0.2514	0.1993
0.6	0.3795	0.0003	0.3798	0.3239
0.8	0.4097	0.0006	0.4103	0.3682
1	0.4292	0.0009	0.4301	0.4082
1.2	0.4511	0.0013	0.4524	0.4369
1.4	0.4739	0.0017	0.4756	0.4609

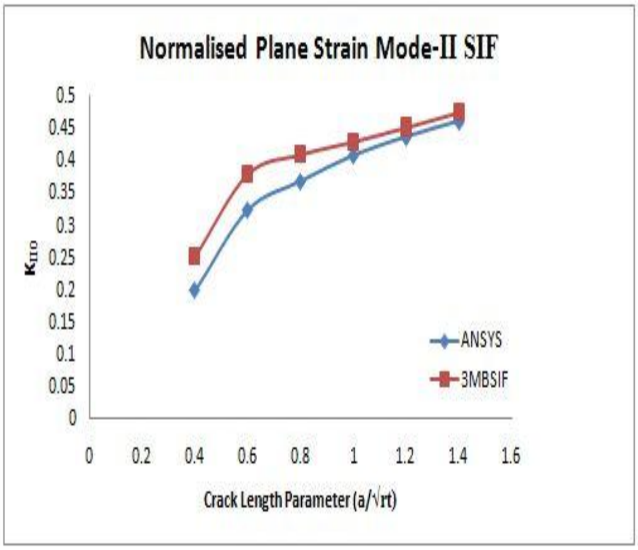

Fig 19: Normalized Mode-II Plane Strain SIF for ϴ=45^0 vs Crack Length Parameter

c. Contour Plot

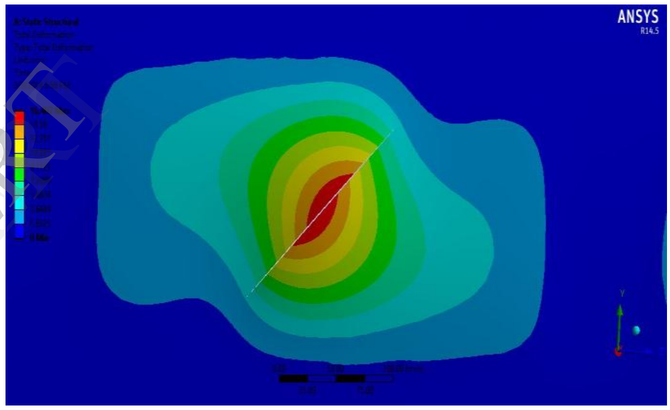

Fig 20: Total Displacement (ϴ=45^0)

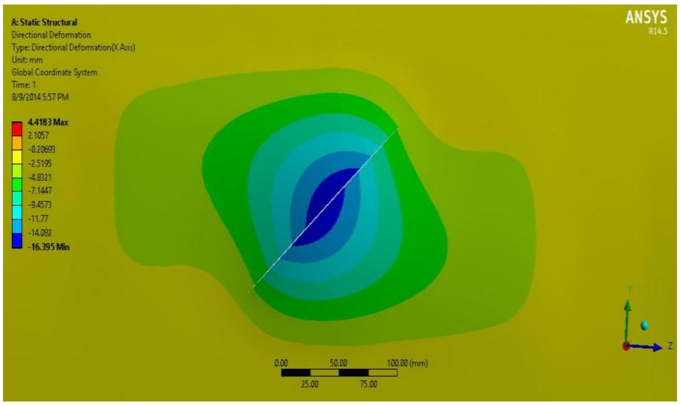

Fig 21: Displacement in x-direction (ϴ=45^0)

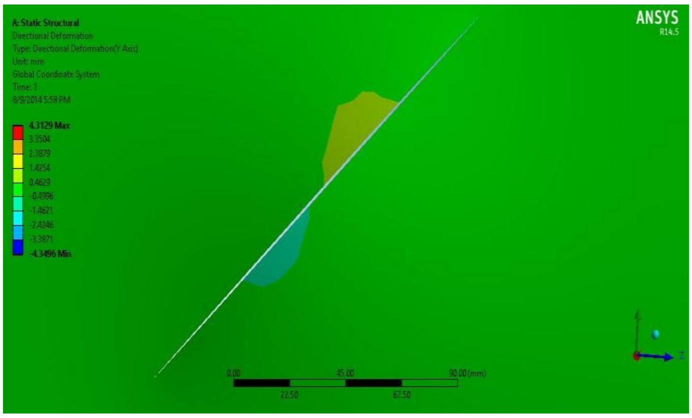

Fig 22: Displacement in y-direction ($\Theta=45^0$)

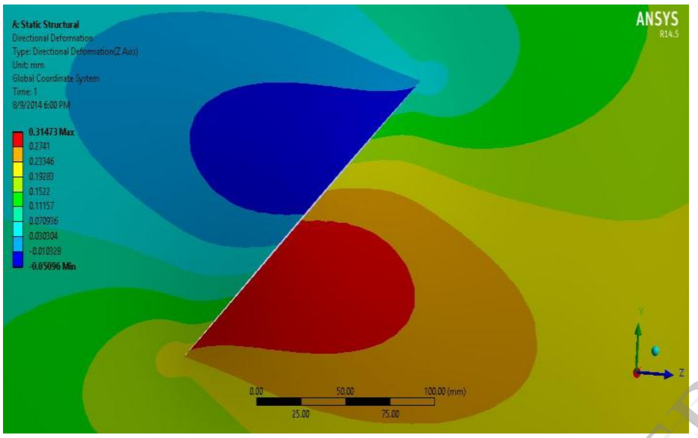

Fig 23: Displacement in z-direction ($\Theta=45^0$)

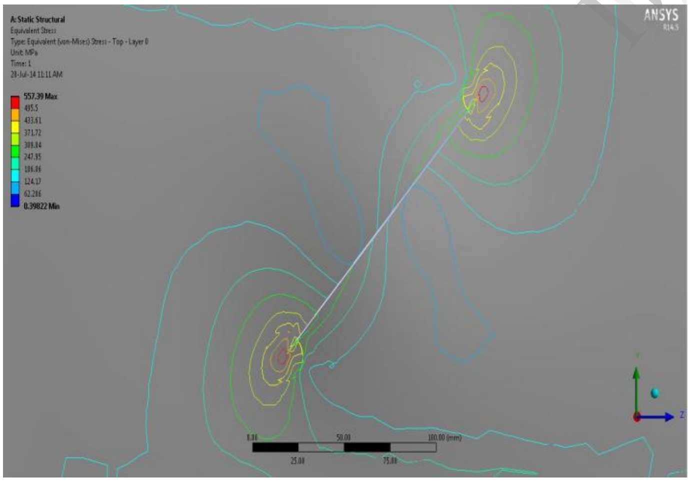

Fig 24: von-Mises Stress: Top Layer ($\Theta=45^0$)

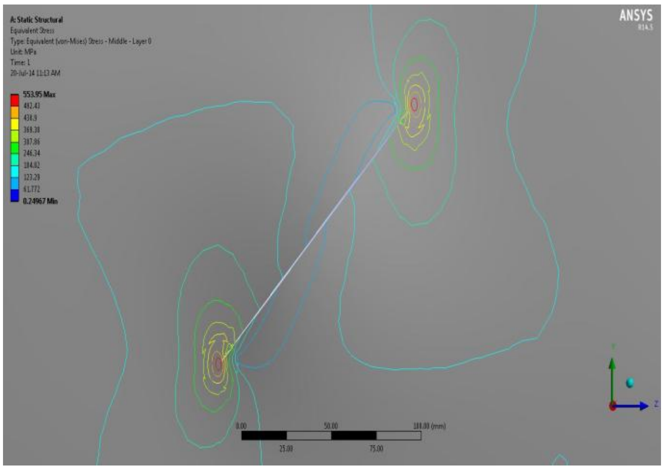

Fig 25: von-Mises Stress: Middle Layer ($\Theta=45^0$)

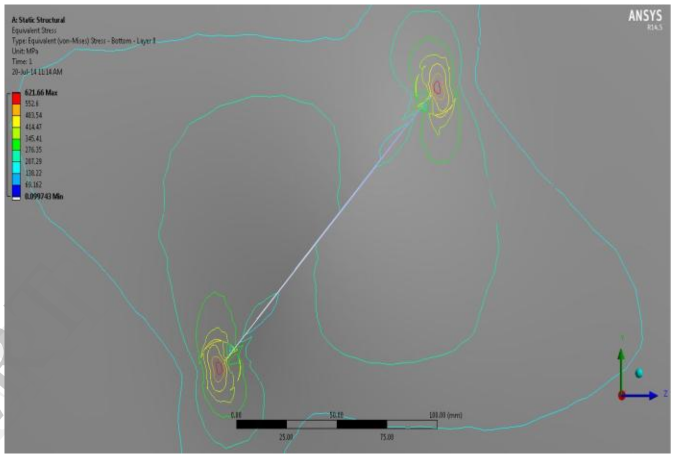

Fig 26: von-Mises Stress: Bottom Layer ($\Theta=45^0$)

VIII. CONCLUSION

In fatigue and fracture mechanics analysis of aircraft fuselage structure, accurate determination of crack tip stress intensity factors for arbitrarily located and oriented crack of various length is essential. The problem is so complex that analytical methods and experimental techniques are not applicable. Finite element modeling using ANSYS Workbench is demonstrated in this study to be a viable and versatile approach. Finite element modeling of the complex shell structure featuring integration of the skin, discrete stringer and complex ring frames is successfully accomplished using ANSYS Workbench. Local modeling around each crack tip using a refined mesh of singular elements is straight forward Determination of mixed mode stress intensity factors denoted by K_I, K_{II} using KCacl command is an added feature.

In the present study a special purpose post-processing subprogram called 3MBSIF was also used to compute the mode-I and mode-II components of membrane and bending stress intensity factors denoted by K_I^M, K_{II}^M, K_I^B, K_{II}^B. It is gratifying to note a close correlation between the two post processing approaches. However it is 3MBSIF that can locate the surface on which fracture initiates.

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