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Study the effect of crack on aircraft fuselage skin panel under fatigue loading conditions

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***Abstract*— The fuselage is one of the main component in any aircraft and its function is to hold all parts together and carries passengers. This fuselage part experience a different loads like static, fatigue, dynamic, buckling during landing, flying and take-off conditions. Now a day's aircraft undergo different type of failure modes, due to improper design, pilot error, weather conditions etc. In the present work, study the effect of crack on fuselage skin panel under fatigue loading conditions. The result shows that fuselage skin panel with crack don't have life but uncracked fuselage have life under fatigue loading conditions. In fatigue analysis, the life, damage and safety factor for fuselage component under subjected conditions can be calculated.**

***Index Terms*— Crack, damage, fuselage, life, safety factor.**

I. INTRODUCTION

An aircraft is a machine that is able to fly by gaining support from the air and driven by jet engines or propellers. The main sections of an aircraft, the fuselage, tail and wing, determine its external shape. The load-bearing members of these main sections, those subjected to major forces, are called the airframe. Fuselage is based on French word fuseler, which means "to streamline". The fuselage, or body of the airplane, is a long hollow tube, which holds all the parts of an airplane together. The fuselage is hollow to reduce weight.

In order for an airplane to fly straight and level, the following relationships must be true [1]:

- Thrust = Drag
- Lift = Weight

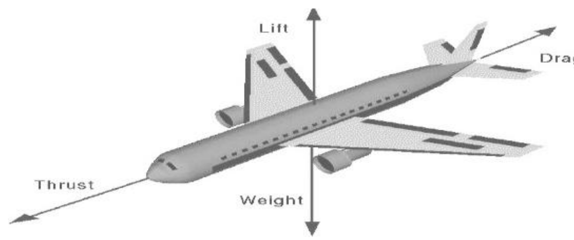

Fig. 1 The forces acting on aircraft

For analysis purpose Airbus A321 is used. It is a largest member of A320 family's. The Airbus A321 single-aisle medium range-airliner is the largest aircraft in the A320 range.

Fig. 2 Airbus A321

Airbus A321 Specifications [2]

Dimensions	
Length	44.5m
Wingspan	34.1m
Height	11.8m
Wing area	122.4m^2
Weight	
Maximum take-off weight	83000-93500kg
Maximum landing weight	73500-77800kg
Operating empty weight	48100kg
Maximum zero fuel weight	71500kg
Maximum payload	23400kg
Standard fuel capacity	23700-29680Litres
Performance	

Range with max payload	5000-5500km
Cruise speed	840km/h
Maximum speed	890km/h
Maximum operating altitude	11900m
Take-off field length	2180m
Landing field length	1580m
Engines	CFMI CFM56-5A/5B, 2*30000-33000 lb IAE V2500-A5, 2*30000-33000 lb
Fuel efficiency	18.2g/pass*km
Fuel flow rate	3200kg/h
Cabin Data	
Passengers	220(1-class)
Passengers	185(2-class)
Cabin width	3.7m

Many researchers have worked on designing this part through various techniques like finite element method, experimental method and analytical method. The researchers have carried out different analysis related to aircraft fuselage structure such as static, buckling, dynamic fracture, fatigue analysis etc., The static analysis can be made by different ways such that different conceptual designs that included as frames spacing was smaller compared to stringers spacing, frames spacing was larger compared to stringers spacing, frames and stringers spacing was approximately equal [3] and laminate constructions for stiffened fuselage panels in aircraft design [4]. The buckling analysis can be made by different ways such that post buckling response behavior of stiffened panels under compression [5] and post buckling response of stiffened panels under shear [6]. The dynamic fracture analysis can be made by different ways such that dynamic fracture analysis of aircraft fuselage with damage due to two kinds of blast loads [7], blast response of metal composite laminate fuselage structures with two material configurations such as aluminium and GLARE [8]. The researchers are also made analysis related to predicting the service durability of aerospace components [9], residual strength pressure tests analysis of stringer and frame stiffened aluminium fuselage panel with longitudinal cracks [10], weight comparison analysis between a composite fuselage and an aluminium alloy fuselage [11], impact of engine debris on fuselage skin panel [12], damage analysis of aircraft structure due to bird strike [13], damage prediction in airplane flap structure due to bird strike [14], and analysis of high energy impact on a sheet metal aircraft structures [15]. The fatigue analysis can be made by different ways such that damage tolerance analysis of aircraft reinforced panels [16], fatigue cracks at many rivet locations in the skin panel [17], and fatigue analysis for upper and lower folding beams on the rear fuselage [18]. The researchers have worked on aircraft fuselage analysis, but they gave less importance to fatigue analysis. Hence, the scope of this work reported in this paper is to study the effect of crack on aircraft fuselage skin panel under fatigue loading conditions.

II. GEOMETRY OF THE MODELS

In fatigue analysis, aircraft fuselage panel with and without crack of length of 0.2mm is considered to determine the effect of crack on life, damage and safety factor under fatigue loading conditions. The 3D model of uncracked and 0.2mm cracked fuselage component is as shown in figures 3 and 4 respectively.

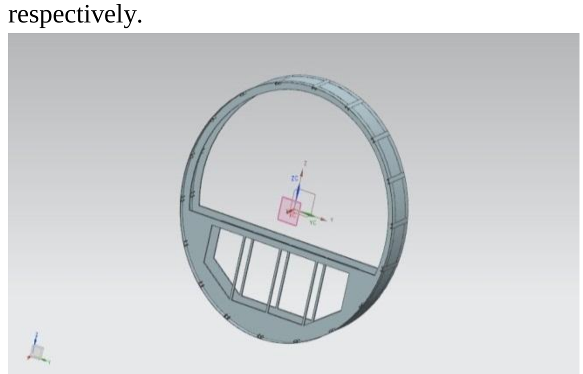

Fig. 3 Uncracked model

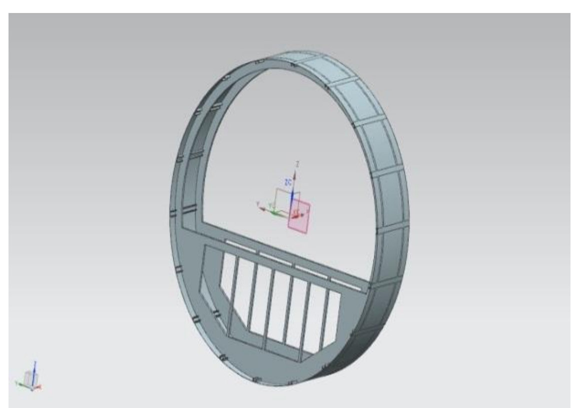

Fig. 4 0.2mm cracked model

III. MESHING OF THE MODELS

Meshing is the process of converting infinite degrees of freedom (DOF) to finite degrees of freedom. The component is tetra meshed with 4 noded tetrahedron elements due to complicated shape of the component using auto mesh generation feature in the software. The meshing process takes around 15 minutes of time in Intel core i3 processor, 4GB RAM equipped PC.

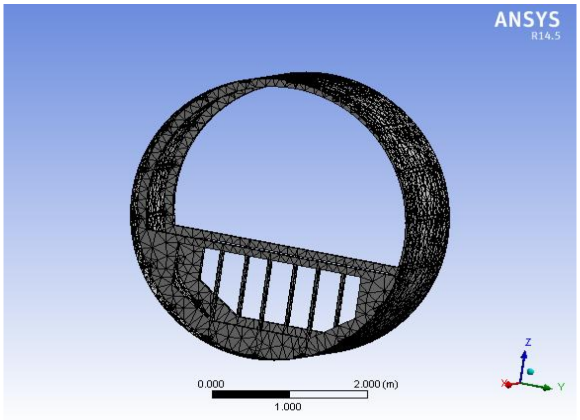

Fig. 5 Meshed uncracked model

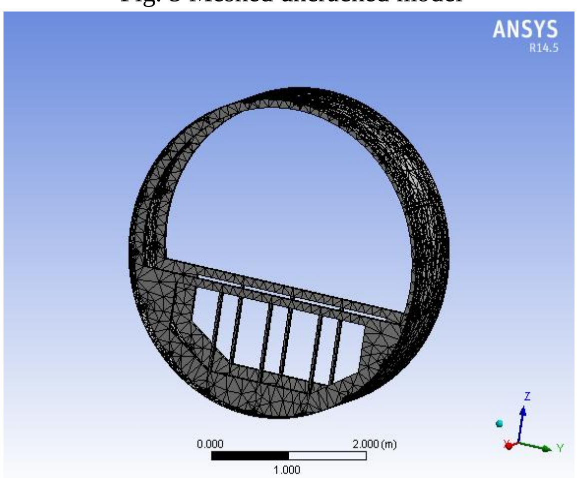

Fig. 6 Meshed0.2mm cracked model

A. Mesh details of the models

The mesh details of uncracked and 0.2mm cracked model are as shown in table 1.

Table 1 Mesh details of the models

Model Type	Type of Mesh	Type of Element	Number of elements	Number of nodes
Un cracked	Tetra	4 noded Tetrahedron	20645	41041
0.2mm Cracked	Tetra	4 noded Tetrahedron	20635	40811

B. Elements used:- 4 noded Tetrahedron element

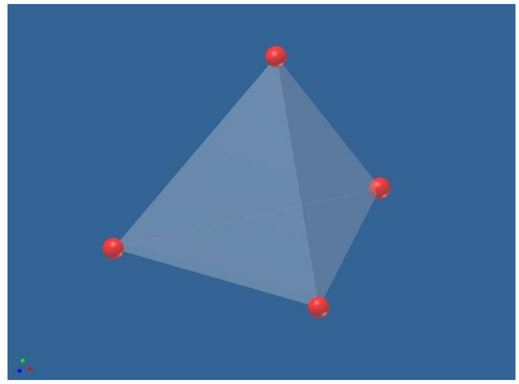

Fig. 7 4 noded Tetrahedron element

The 4 noded Tetrahedron element is a three dimensional element with 4 nodes at its corners. The elements is defined by four nodes having six degrees of freedom at each node; translations in the nodal x, y and z directions (UX, UY, UZ) and rotations about the nodal x, y and z directions (ROTX, ROTY, ROTZ).

IV. MATERIAL SELECTED

After the meshing process next step is to assign the material properties and its behaviour. Selection of materials in aircraft construction is rather complex and is based on trade off amongst conflicting requirement of high strength, low density and easy of fabrication or processing. The material used in various parts of vehicle structures generally are selected by different criteria. The material used in the fuselage structure is Aluminium alloy 2024-T351 and its composition as shown in table 2.

Table 2 Composition of Aluminium alloy 2024-T351

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

Zn	Max 0.25
Others	Max 0.15

V. LOAD AND BOUNDARY CONDITIONS APPLIED

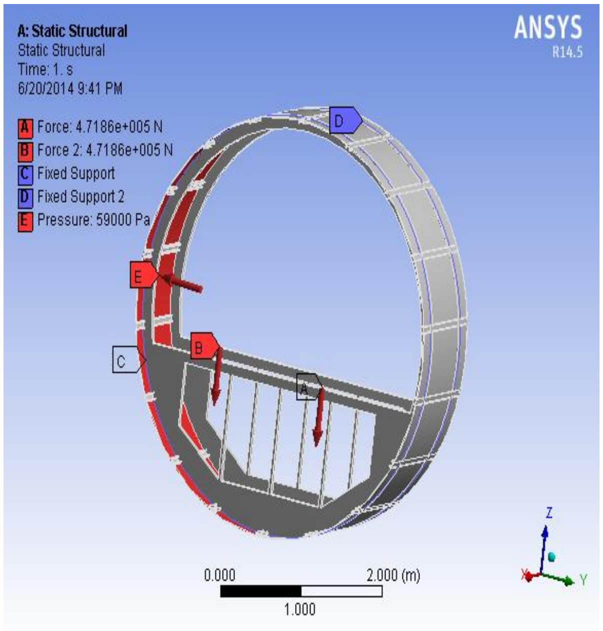

Fig. 8 Load and Boundary conditions applied for uncracked model

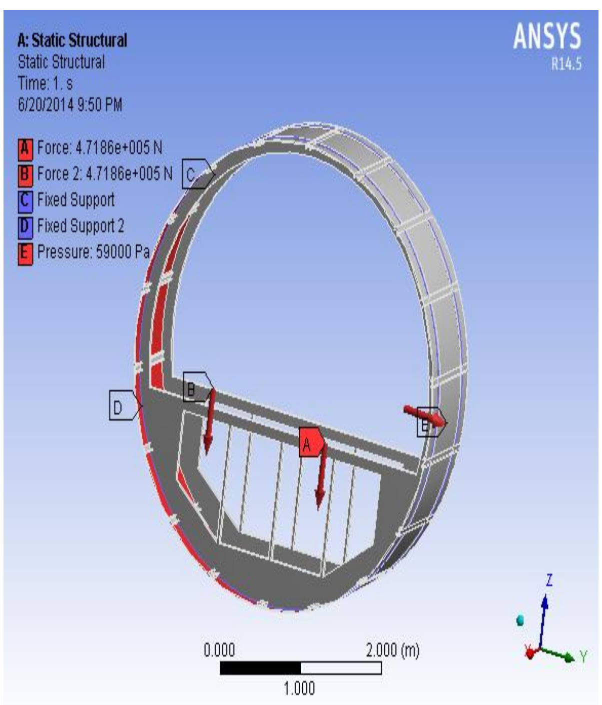

Fig. 9 Load and Boundary conditions applied for 0.2mm cracked model

After meshing the model, the next step in Finite element analysis is that all load and boundary conditions data are applied. The figures 8 and 9 represent load and boundary conditions applied for uncracked and 0.2mm cracked models respectively. This is done in ANSYS Workbench v14.5 software. In these figures, A and B represents operating weight of about 471861N which is applied on two roofs of a fuselage component, C and D represents fixed supports that is constraints applied on both end of frame with skin panel and E represents a cabin pressure of about 59KPa which is applied around a fuselage component.

After applying load and boundary conditions, the next step is to define the material properties. The material behavior considered is Isotropic elasticity. The properties of Aluminium alloy 2024-T351 is applied for fuselage structure. The following properties of material are inputted.

Young's Modulus, E=70,000 N/mm^2

Poisson's Ratio, μ = 0.3

Ultimate Tensile Strength, σu = 420 N/mm^2

Density = 2780 kg/m^3

The S-N curve for Aluminium alloy 2024-T351 is as shown in figure 10.

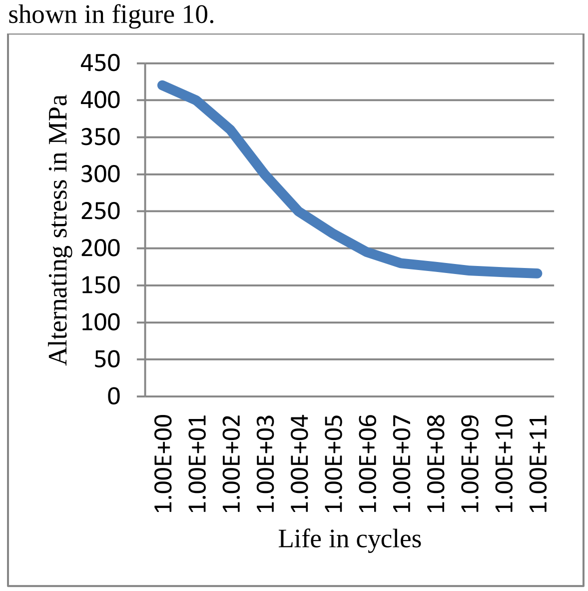

Fig. 10 S-N curve for Aluminium alloy 2024-T351

By using this S-N curve, the values of alternating stress & their corresponding number of cycles are extracted and then it is inputted in linear-semi log scale in the software. The next step in the analysis is deck preparation that means preparing final model for solving.

In fatigue analysis, Stress life approach is used to calculate life of fuselage component under stated testing conditions. For mean stress correction theory, the Goodman criterion is used. The ANSYS Workbench v14.5 software is calculating life to infinite number of cycles from $1*10^9$ cycles. The required output parameters like Von-Mises stress, displacement, life, damage and safety factor are clearly defined. During solving process, software take around 20 minutes time in a Pentium dual core processor, 2GB RAM equipped PC. The solving time can be minimized by using high configured computer.

For infinite number of cycles, the Goodman equation can be written as follows.

$$\sigma m/\sigma u + \sigma a/\sigma en = 1$$

Where, σm = Mean stress in MPa

σu = Ultimate stress in MPa

σa = Alternating stress in MPa

σen = Endurance strength in MPa

In fatigue analysis, load applied is completely reversed that is stress ratio becomes -1. It can be written as follows.

$$\text{Stress Ratio } R = \sigma min/\sigma max = -1$$

$$\text{Mean stress } \sigma m = (\sigma max + \sigma min)/2$$

$$\sigma m = 0$$

The graphical representation of completely reversed loading condition, Goodman criterion is as shown in figure 11.

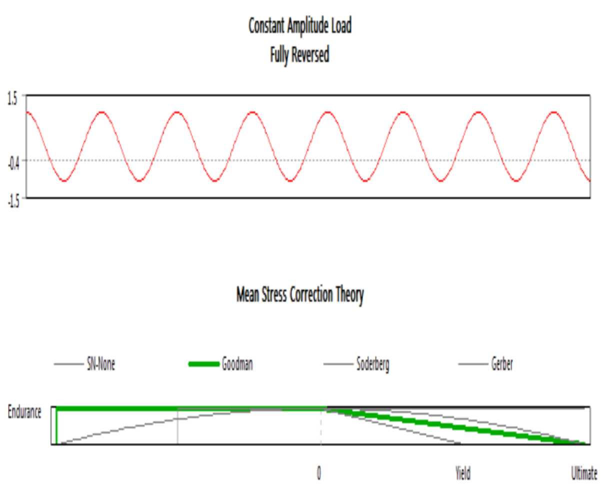

Fig. 11 Completely reversed loading condition and Goodman criterion

VI. RESULTS AND DISCUSSIONS

The Von-Mises stress, Deflection, Life, Damage and Safety factor plots for uncracked and 0.2mm cracked model are shown in figures from 12 to 21 through JPEG file format.

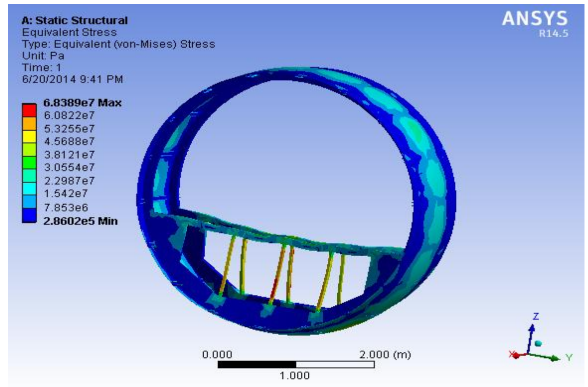

Fig. 12 Von-Mises stress plot for uncracked model

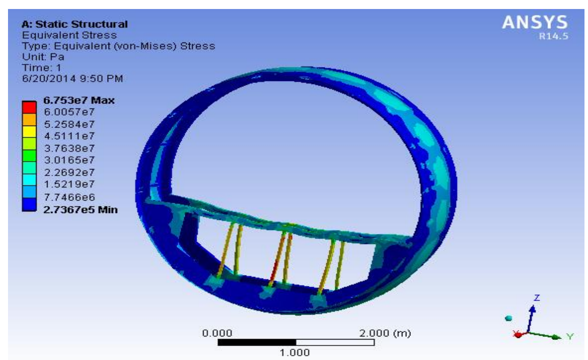

Fig. 13 Von-Mises stress plot for cracked model

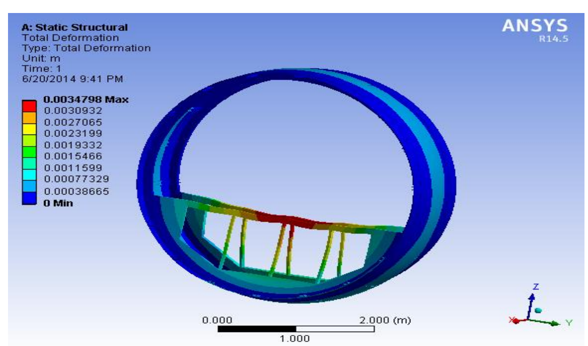

Fig. 14 Deflection plot for uncracked model

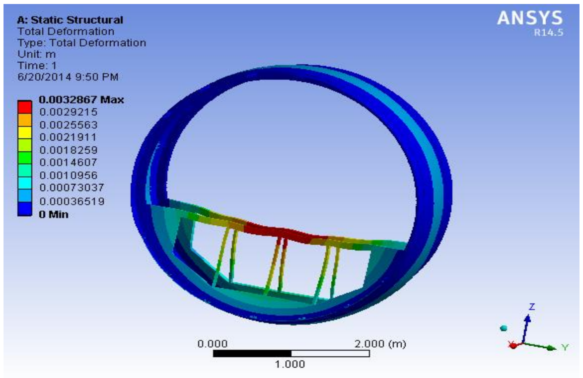

Fig. 15 Deflection plot for cracked model

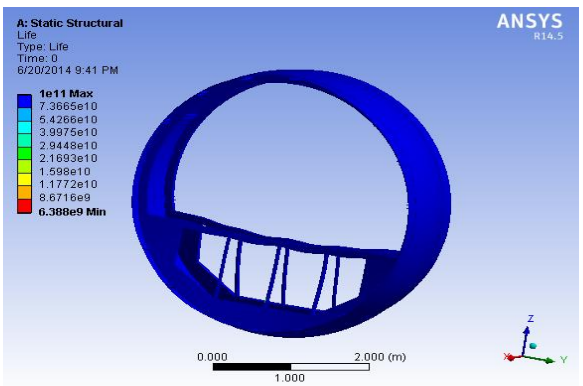

Fig. 16 Life plot for uncracked model

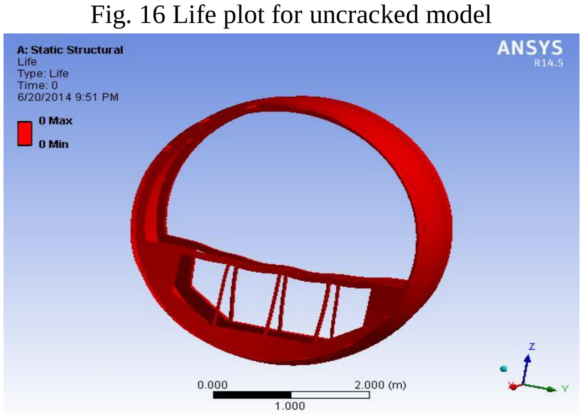

Fig. 17 Life plot for cracked model

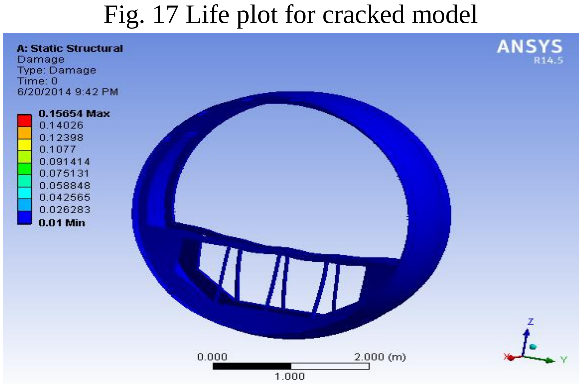

Fig. 18 Damage plot for uncracked model

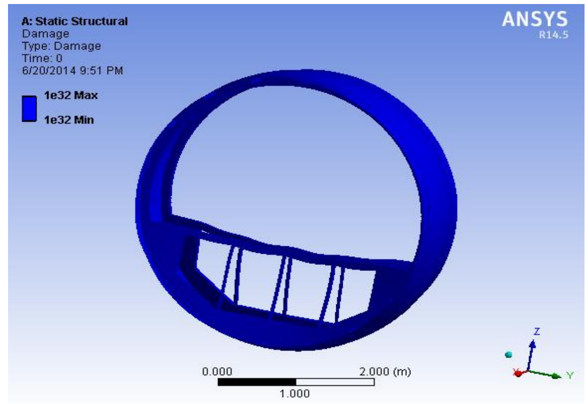

Fig. 19 Damage plot for cracked model

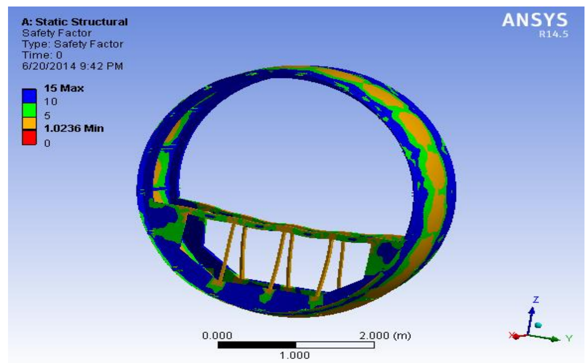

Fig. 20 Safety factor plot for uncracked model

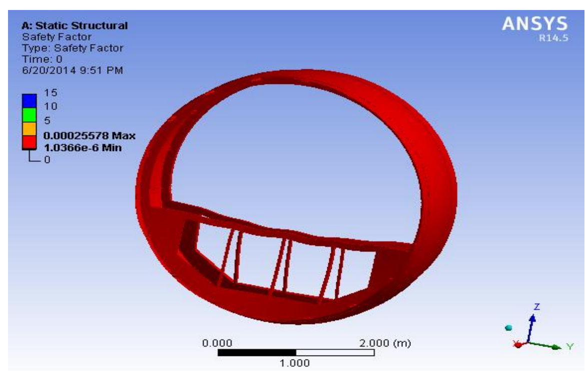

Fig. 21 Safety factor plot for cracked model

The results are tabulated as follows.

Table 3 Results for different models

Model Type	Safety factor	Von-Mises stress in MPa	Deflection in mm	Life in Cycles	Damage
Uncracked	1.023	68.38	3.47	$6.38*10^{9}$	0.1565
0.2mm cracked	$1.0366*10^{-6}$	67.53	3.28	0	$1*10^{32}$

VII. CONCLUSION

From results of fatigue analysis, uncracked mod elhave able to withstand load cycles hence it have life under stated test conditions, but cracked model don't able to withstand load cycles hence there is no life under stated test conditions. The uncracked model have life upto $6.38*10^9$ cycles, damage value is 0.1565 and safety factor value is 1.023, therefore uncracked model is safe for stated test conditions under fatigue loading. But in 0.2mm cracked model don't have life due to crack, damage is very high that is infinite due to crack and hence it have very less safety factor value, therefore 0.2mm cracked model is fail for stated test conditions under fatigue loading.

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