Evolution of Aircraft Stiffened Panels Using Contact Condition

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Abstract

Transport aircraft cruising at high altitudes require pressurization of the cabin for passenger comfort. The fuselage structure is then subjected to one pressurization cycle per flight. For the fuselage, this pressure cycling is the most critical loads that create fatigue cracks.

In order to preserve the structural integrity, throughout the economic service life of the aircraft, it should retain sufficient residual strength in the presence of fatigue cracks and also under discrete source damage.

"Two-bay crack arrest" implies that a skin crack, two bay long, will not lead to catastrophic failure of the airframe, under a specified design limit load.

In this project the 2-bay crack arrest capability evaluation through finite element analysis is carried out. A transport airplane fuselage structure is analyzed. Considering typical dimensions of a transport aircraft fuselage a finite element analysis is carried out.

A stiffened panel is considered for the FEM evaluation. Stiffened panel is a generic structural element which consists of structural elements like skin, bulkhead, longerons and rivets. A longitudinal crack simulating discrete source damage is simulated in the finite element model. Stress intensity factor calculations are carried out at different increments in crack lengths on the skin. Structural features are varied (tear strap, bulkhead, spacing etc.) to demonstrate

that a 2 bay crack arrest capability is built in by design. A residual strength diagram is plotted to demonstrate analytically the two bay crack arrest capability of the stiffened panel.

1.INTRODUCTION

1.1 Overview of Two Bay Crack Arrest

Transport aircraft cruising at high altitudes require pressurization of the cabin for passenger comfort. The fuselage structure is then subjected to one pressurization cycle per flight. For the fuselage, this pressure cycling is the most critical loads that create fatigue cracks.

In order to preserve the structural integrity, throughout the economic service life of the aircraft, it should retain sufficient residual strength in the presence of fatigue cracks and also under discrete source damage.

"Two-bay crack arrest" implies that a skin crack-two bay long-will not lead to catastrophic failure of the airframe, under a specified design limit load.

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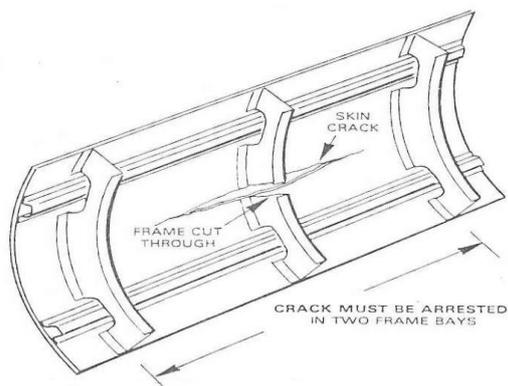**ISSN: 0975-5662,**   **Dec, 2015**   **www.ijrct.org**

Figure 1. Aircraft showing fatigue critical 1ocations

Transport aircraft certification rules also required that the fuselage structure must retain integrity in the presence of discrete source damage. A discrete source that can cause damage is a broken propeller hitting the fuselage side. The damage caused is modeled as if a bulkhead is broken and the skin over the bulkhead has developed a crack. The structural integrity requirements under such a discrete sources damage is that the fuselage structure must be capable of arresting a 2 bay crack.

During flight the crack created by the discrete source could become unstable and catastrophically fail but the bulkheads on either side of the broken bulkhead are capable of arresting this fast moving crack. That is the length of the crack can become equal to twice the bulkhead spacing and no more this requirement is called "Two Bay Crack arrest capability" It is now the aircraft industry standard for development of wide body passenger aircraft.
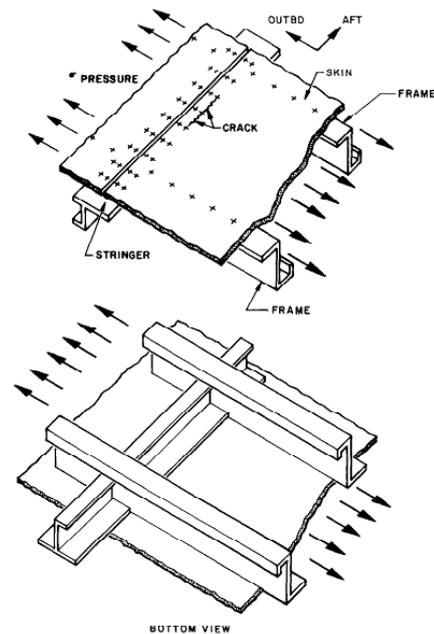

Figure 1. Fuselage longitudinal splice joints

This crack grows and crack arrest is possible provided the adjacent bulkheads and the skin to bulkhead rivets remain intact. If the rivets fail a two bay crack cannot get arrested and that is a design issue.

Therefore three things are required for a 2 bay crack to get arrested:-

i. $K < K_c$
ii. The rivets are intact.
iii. The bulkheads are intact.

1.2 Aerospace Design Philosophy

Aerospace Industry is in continuous pursuit to maximize the load carrying capacity of the structural component with a minimum weight. Emphasis here continues to be on the lightweight material, reduced margin of safety and accurate estimation of internal stress distribution in response to applied loading. This helps in achieving the objective of maximized ratio of load carrying capacity to structural weight.
Different design philosophies used for aerospace structural design are:

  i.   Safe Life approach
  ii.   Fail Safe approach
  iii.   Damage Tolerance approach

Safe-life means that the structure has been evaluated to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks.

Fail-safe means that the structure has been evaluated to assure that catastrophic failure is not probable after fatigue failure or obvious partial failure of a single, principal structural element.

Damage tolerance means the structure has been evaluated to ensure that serious fatigue, corrosion or accidental damage occurs within the operational life of the aircraft, the remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected.

2. LITERATURE SURVEY

[1] T.Swift had discussed and suggested that manufacturers of future transport aircraft to retain the large damage-tolerance capability designed into the first wide-bodied aircraft and to modify their methodology to establish inspection thresholds for those structures incapable of sustaining large obviously detectable damage. In the current economic environment there appears to be a general trend to lower the level of safety built into the original wide-bodied aircraft to reduce assembly costs and weight. Most of the wide-bodied aircraft are designed to sustain a two-bay skin crack with a broken central stiffener at limit load. Most of the large transport aircraft manufacturers establish the threshold for detailed inspection of principal structural elements through a fatigue evaluations process without considerations of initial manufacturing flaws. The majority of large transport aircraft developed in the USA have circumferential crack- stopper straps attached directly to the fuselage skins to guard against explosive decompression failure in the event of undetected fatigue damage or discrete source damage. This was thought necessary after the Comet disasters in 1954. There is a current trend to eliminate these crack stoppers for future designs and depend only on shear clips for crack arrest capability to save on assembly costs.

[2] H. VLIEGER, worked on The Residual Strength Characteristics of Stiffened Panels Containing Fatigue Cracks. In his paper, he discussed about heavy structural members, where plane strain conditions prevail, linear fracture mechanics can be used for predicting residual strength. Aircraft structures consist largely of sheet structures with plane stress conditions where linear fracture mechanics do not seem to apply. Yet it is in the aircraft main structure that large fatigue cracks can develop and that has to be designed fail-safe. The present paper describes a method to predict the residual strength of a cracked sheet structure. Contrary to an un-stiffened sheet, the sheet structure contains stiffening elements that can act as crack stoppers. This crack arresting action and its consequences for the residual strength are considered in the analysis. The paper proposes a method that relates the crack resistance of a stiffened panel to that of an un-stiffened sheet. It takes full account of sheet-stringer interaction in the cracked region. A criterion for crack arrest is put forward. Ultimate panel failure after crack arrest is initiated either by subsequent unstable crack growth or by stiffener failure. Critical

load conditions for both failure modes are presented. In case crack arrest does not occur, the residual strength of the un-stiffened panel constitutes a safe lower bound. Computational results of the interacting rivet forces by both analytical and numerical (finite element) methods are presented. From these the load concentration in the stiffener and the reduction of the stress intensity at the crack tip can be determined. This enables the complete residual strength characteristics to be predicted. The results of residual strength tests on bonded and riveted panels with symmetric strip stiffeners or eccentric Z-stringers fully substantiate the method proposed for residual strength calculations.

[3] PIR M. TOOR, worked On Damage Tolerance Design of Fuselage Structure Longitudinal Cracks. In his paper, various analytical and empirical approaches used in evaluating the damage tolerance capability of the fuselage structure are critically evaluated and compared. A model which accounts for the influence of frames, straps and curvature is developed. This model is then used in an example problem having typical military cargo aircraft fuselage structural elements.

(1) Basic fracture mechanics concepts along with damage tolerance design philosophy are discussed to ensure the safety of an existing aircraft structure.

(2) Parameters such as load transfer and special boundary conditions which affect crack growth characteristics are discussed briefly.

(3) Fracture mechanics methodology is developing at a very rapid rate. Therefore, an analyst should be aware of the current state-of-the-art.

[4] M. Adeel worked on Study on Damage Tolerance Behavior of Integrally Stiffened Panel &Conventional Stiffened Panel. In his paper, damage tolerance behavior of integrally and conventional stiffened panel is investigated based on the fracture mechanics and finite element analysis. The load bearing capability and crack growth characteristic of both types of the stiffened panels having same configuration subjected to distributed tensile load is examined in this paper. A fourteen-stringer stiffened panel is analyzed for a central skin crack propagating towards the adjacent stringers. Stress intensity factors and fatigue crack propagation rates of both types of the stiffened panels are then compared. The analysis results show that integral stiffening causes higher stress intensity factor than conventional stiffened panel as the crack tip passes through the stringer and the integrally stiffened panel has less load bearing capability than the riveted stiffened panel.

3.METHODOLOGY
3.1 IntroductionTo Fracture Mechanics

Fracture mechanics is based on the assumption that all engineering materials contain cracks from which failure starts. The estimation of the remaining life of machine or structural components requires knowledge of the redistribution of stresses caused by the introduction of cracks in conjunction with a crack growth condition. Cracks lead to high stresses near the crack tip; this point should receive particular attention since it is here that further crack growth takes place. Loading of a cracked body is usually accompanied by inelastic deformation and other nonlinear effects near the crack tip, except for ideally brittle materials. There are, however, situations where the extent of inelastic deformation and the nonlinear effects are very small compared to the crack size and any other

characteristic length of the body. In such cases the linear theory is adequate to address the problem of stress distribution in the cracked body. Stress intensity factor governs the linear elastic stress field near the crack tip.

There are two alternate approaches to fracture analysis:

i.   The energy criterion
ii. The stress intensity approach

3.2 The Energy Criteria

According to this approach the crack extension (i.e. fracture) occurs when the energy available for crack growth is sufficient to overcome the resistance of the material. The material resistance may include the surface energy, plastic work or any other type of energy dissipation associated with a propagating crack. This approach was first proposed by Griffith in1920. A crack can form (or an existing crack may grow) if during the process to attain equilibrium causes the total energy to decrease or remain constant. Thus the critical conditions for fracture can be defined as the point where the crack growth occurs under equilibrium conditions, with no net change in total energy.

$$E = U^e + U^p \qquad (3.1)$$

Where   E is Internal Energy
        U^e is Elastic strain Energy
        U^p is Plastic Work

The Griffith energy balance for an increment increase in the crack area, dA, under equilibrium conditions can be expressed as the following relation

$$\frac{dE}{dA} = \frac{d\pi}{dA} + \frac{dW_s}{dA} = 0 \qquad (3.2)$$

Where E is the total energy, π is the total potential energy supplied by the internal strain energy and external forces, and W_s is the work required to create new surfaces.

3.3 Stress Intensity Approach

This approach was first proposed by Irwin. According to this approach, when certain cracked configurations are subjected to external forces, the stress field in a linearly elastic body near the crack tip is given by
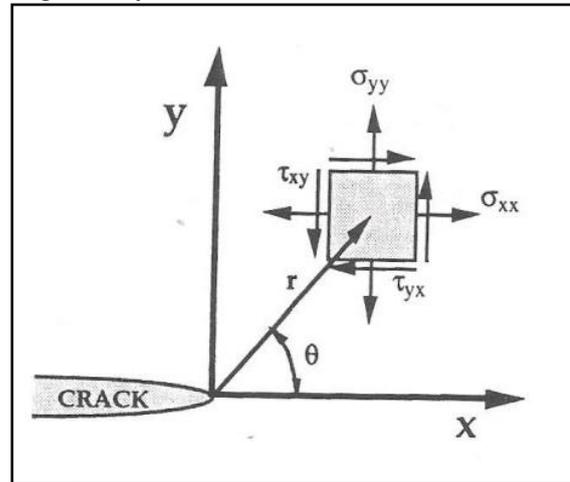

Figure 3 Crack tip stress

$$\sigma_{ij} = \left(\frac{K}{\sqrt{r}}\right) f_{ij}(\theta) + \sum_{m=0}^{\infty} A_m r^{\frac{m}{2}} g_{ij}^{(m)}(\theta)$$
$$(3.3)$$

Where σ_{ij} is the stress tensor, r and θ are defined in Figure 3.1. K is a constant, and f_{ij} is a dimensionless function of θ. The higher order terms depend on the geometry and the solution for any given configuration contains a leading term that is proportional to $1/\sqrt{r}$. Thus the stress near the crack tip varies with $1/\sqrt{r}$. Describes a stress singularity, since the stress is asymptotic to r = 0.
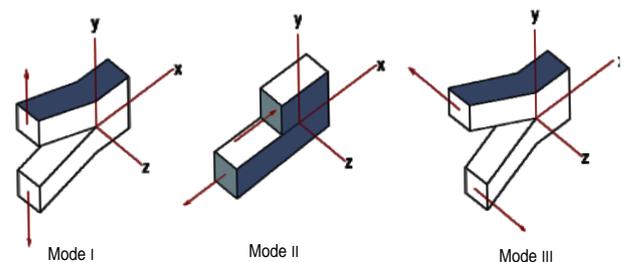
3.4 Modes of Fracture

Figure 4. Different modes of fracture

$$\lim_{r \to 0} \sigma_{ij}^{(III)} = \frac{K_{III}}{\sqrt{2\pi r}} f_{ij}^{(1)}(\theta)$$

Mode-I or Tensile Mode

In this mode, the crack faces separate in a direction normal to the plane of the crack and the corresponding displacement of the crack walls are symmetric with respect to the x-z and the x-y planes.

Mode-II or Shearing Mode

It is an in-plane sliding mode in which the crack faces are mutually sheared in a direction normal to the crack front. Here the displacements of the crack walls are symmetric with respect to the x-y plane and Anti symmetric with respect to the x-z plane.

Mode-III or Tearing Mode

The displacements of the crack walls in this case are Anti symmetric with respect to both the planes (x-y, x-z). Hence this mode is also called anti-plane shear or tearing mode. Since mode II or III fractures require higher loads, estimations based on mode I would be on a conservative side.

Stress Intensity Factor

Each mode of loading produces the $1/\sqrt{r}$ singularity at the crack tip, but the proportionality constant, k, and f_{ij} depend on the mode. This constant, k, is replaced by the stress intensity factor, K, where $K = k\sqrt{2\pi}$. For mode I, II, III the stress intensity factor is written as K_I, K_{II} and K_{III} respectively.

Thus the stress fields ahead of a crack tip in an isotropic linear material can be written as for Modes I, II and III respectively.

$$\lim_{r \to 0} \sigma_{ij}^{(I)} = \frac{K_I}{\sqrt{2\pi r}} f_{ij}^{(1)}(\theta)$$

$$\lim_{r \to 0} \sigma_{ij}^{(II)} = \frac{K_{II}}{\sqrt{2\pi r}} f_{ij}^{(1)}(\theta)$$

3.5 Concept of Crack arrest

The crack in a structure advances if the energy release rate 'G' is greater than the resistance of material 'R' (i.e. $K_I=K_{IC}$) .Crack gets arrested when 'G' becomes smaller than 'R'. Crack arrest in a structure can be achieved by increasing the material resistance 'R' by providing strip of materials of higher toughness in the suitable location. As the crack penetrates the strip, more resistance is experienced by the progressive crack giving rise to a situation where G <R i.e. energy released by the crack is less than the resistance of the material.

3.6 Crack arresting techniques
1.   In plane crack arrest
2.   Out of plane crack arrest

1.In plane crack arrest

It generally involves stiffening members or strips are being welded as an integral load carrying component in conjunction with the primary structure for arresting the progressive crack. The material used for the crack arresting strip has a higher level of notch toughness than the base material to which it is welded generally used in ship hull structures.

2.Out of plane crack arrest

In this, Stiffening members or strips are used which are either riveted or welded integrally so that it has better advantages than the in plane type and also used widely in many applications and is generally used in aircraft applications.

4.STIFFENED PANEL MODELING

4.1 Introduction

Transport-fuselage shells are designed to support internal pressure and mechanical flight loads which result in local panel loads that consist of various levels of longitudinal tension, circumferential tension, and shear stresses.

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Typical metallic fuselage structure consists of built-up stiffened panels with a thin skin attached to longitudinal stringers and circumferential frames. Failure initiation and propagation in the built-up structure may involve crack initiation in the skin or stiffening elements, or fatigue or strength failure of the fastener elements connecting the skin to the stiffening elements. The structural integrity of a built-up stiffened panel structure subjected to internal pressure can be studied analytically with a structural analysis capability.

The stiffened panel is of 500 mm in width and 400 mm in height and there are five bulkheads each spaced at 133.7mm and 2 Longerons each spaced at a distance of 202mm as shown in the figure Figureand there are totally nine stringers along the longitudinal direction with spacing of 140 mm.

Table 1: Skin Geometric Details

SKIN GEOMETRIC DETAILS		
Skin width	400	mm
Skin length	500	Mm
Skin thickness	2.00	Mm
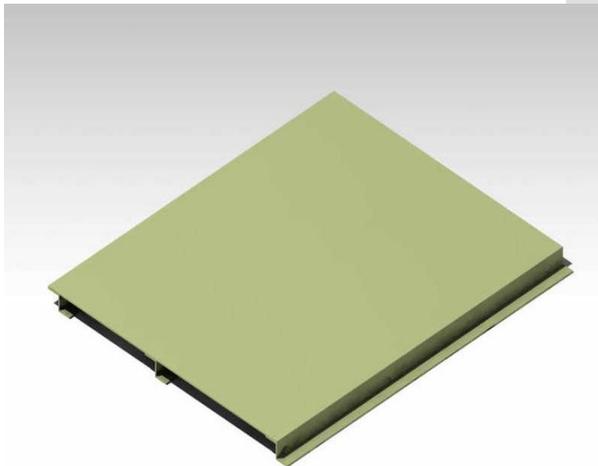
Figure 5. Design of stiffened panel

Table 2: Skin Geometric Details

SKIN GEOMETRIC DETAILS		
Skin width	400	Mm
Skin length	500	Mm
Skin thickness	2.00	Mm
Riveted hole	4	Mm
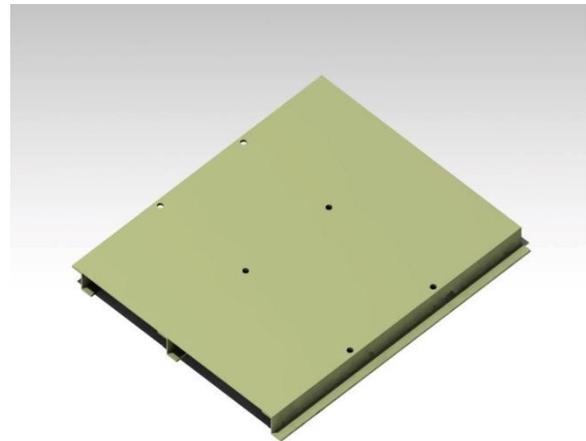
Figure 6. Design of stiffened panel with riveted hole

4.2 Structural details

An aircraft is a typical multi-load-path structure having various internal structural details. A typical fuselage consists of bulkheads, Longerons, skin, and rivets, shear clips, tear straps, etc,

Here as first part of whole damage tolerance evaluation exercise; the fuselage curved panel has been idealized as a flat panel having same diameter and equal number of bulkheads, Longerons and other internal structural details. The fuselage was modeled having 2000 mm diameter and 2.0 mm skin thickness. This fuselage was subjected to internal pressurization of 6.5 psi. Hoops stress of 33.61 N/mm^{2}was developed. The longitudinal stress was half of hoops stresses.

5. Analysis of Stiffened Panel
5.1 Finet element Software Package

The analysis of stiffened panels been attempted using

- NX 9.0
- ALTAIR HYPERMESH
- ANSYS

5.2 Meshing the Model

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Meshing is done in ALTAIR HYPERMESH by using various meshing options. Good quality of the mesh can be achieved from making use of options like element density, type, biasing, smoothing, shaping, sizing, etc. Once elements are created, care should be taken to organize them to respective collectors, as the elements behave according to the property given to the collector's card image.

• Importing the IGES file Altair Hyper mesh
• Geometry cleanup.
• Extraction the mid surfaces.
• Assigning first order elements.
• & element size.
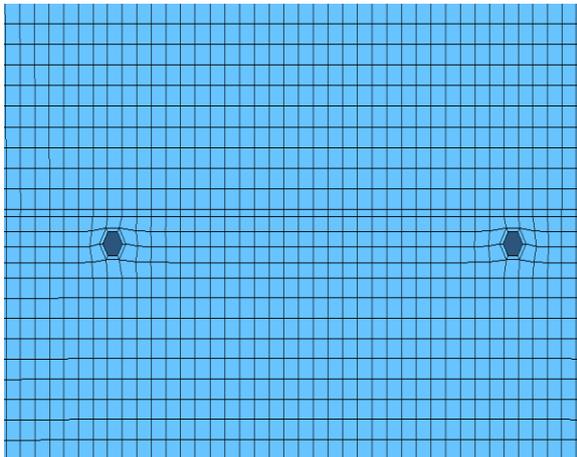
Figure 7. Meshing

5.3 Element Quality Criteria

Once required mesh pattern is got ALTAIR HYPERMESH, it is necessary to check the quality of mesh generated, this can be done using quality checks available in. The elements can be checked for war page, aspect ratio, Taper, min & max. Quad angle etc .

For 1-D elements:
• Check for free 1-d.
• Check for rigid loops.
• Dependency Connectivity
• Duplicates.

For 2-D elements
• Elements are checked for quality parameters like warpage, aspect ratio, skew and Jacobin.
• Check the maximum and minimum interior angles of all elements.
• Checking for shell normal.

• Check for free edges.
• Check for connectivity.
• Check for duplicates.
• Eliminating the duplicates.
• Maintaining the continuity of the elements.
• Eliminating the free edges and rotating of elements.
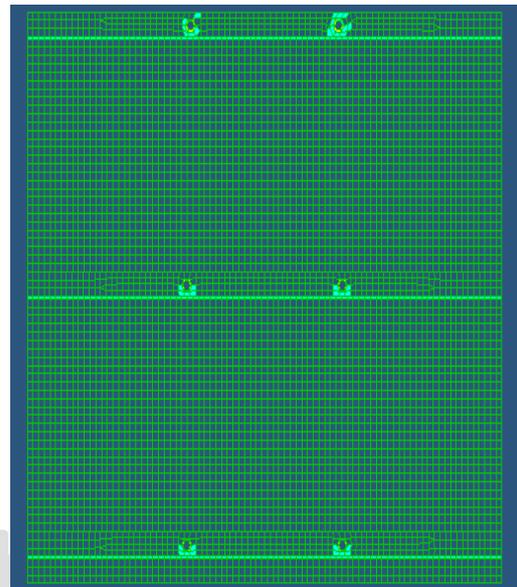
Figure 8. Quality mesh

Table 5.1 FE model details

Nodes	2719
CQUAD	2587
CTRIA	7
Total Elements	2594

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Table 5.2 Element quality details

#	Criterion	Failure criteria	% failed
1	min size	0.05	0
2	max length	1	0
3	aspect ratio	5	0
4	warpage	5	0
5	max angle quad	150	0
6	min angle quad	40	0
7	max angle tria	120	0
8	min angle tria	30	0.04
9	skew	40	0
10	jacobian	0.6	0.04
11	chordal dev	1	0
12	% of trias	15	0.27

- 2D mesh
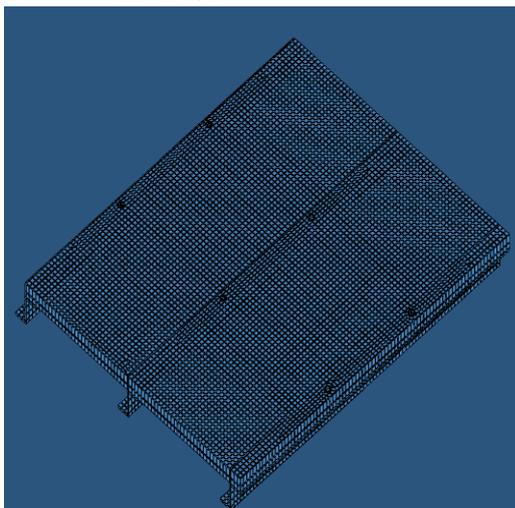
Figure 9. Ideal mesh

5.4 Rivets

- Modeling of the rivet according to the user requirements.
- Required dimensions and 1D element.
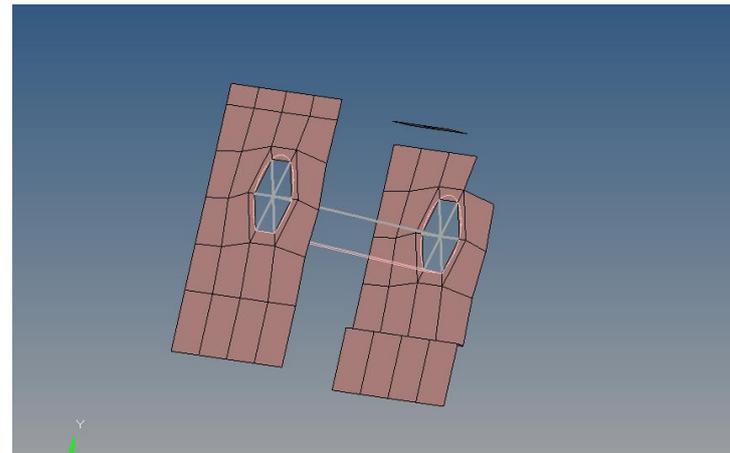
Figure 10.1-D elements

5.5 Contact condition

- In contact condition is applied between the component of the skin elements and stringers elements.

- By Appling these boundary condition maximum SIF increases with the increasing of load, but as the crack reaches close to bulkhead, the bulkhead will share the maximum load, the value of SIF drops and it never reaches Critical SIF equal to K_{1C}, resulting in higher residual strength in the skin leading to crack arrest.

Table 5.3 FE contact details

Component	Target	Friction
Component 1	Target 1	1
Component 1	Target 2	1
Component 1	Target 2	1

- Skin is indicated as component 1

- Longerons are indicated as target.

6. RESULTS &DISCUSIONS

6.1 Static and modal analysis for the stiffened panel at thickness 0.002m.

Analysis of the discretized model shown in Figure 5.3 and 5.4 was done with the help of the software Altair Hypermesh. The analyzed results were displayed via ANSYS. The results obtained from the analysis are discussed in the following section of our report.

The stress contour of the stiffened panel with varying thickness shows the

maximum stress and minimum stress at varying location with load condition.

- P =8896N,
- $E = 70E^9$ N/m^2
- u= 0.33

Modal analysis of the stiffened panel at thickness 0.002m.

A maximum displacement is obtained at different from the analysis. Because of the presence of the stiffeners, the panel will tend to bend. In the process it has a out of plane displacement which is shown in below figures at different thickness.
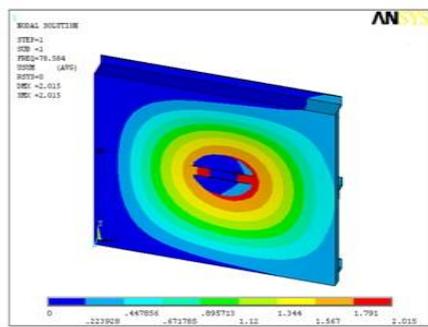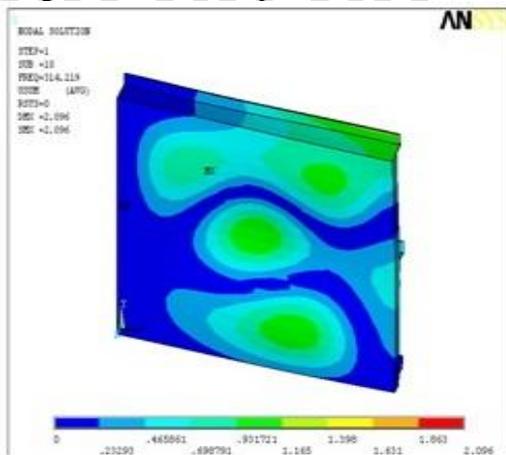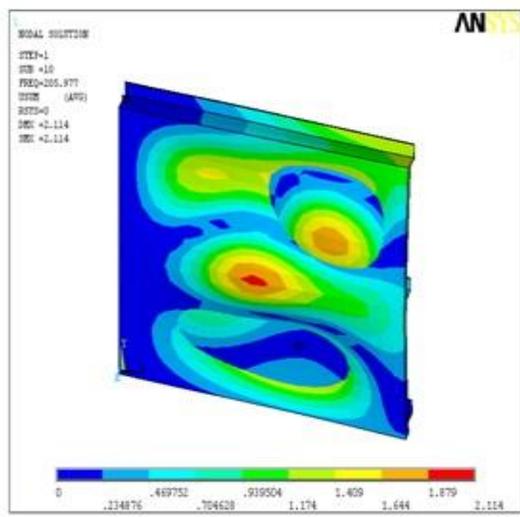

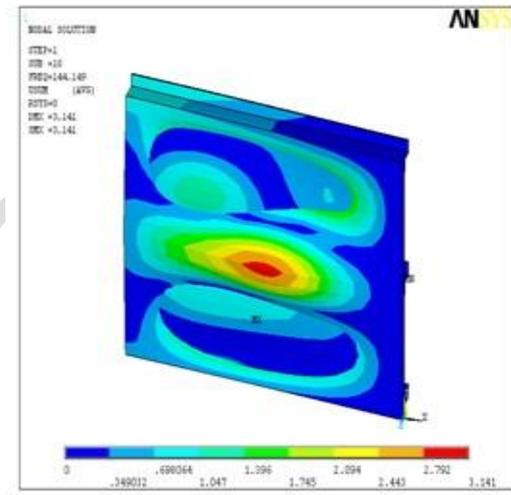Figure 9. Vector Displacement at mode 1

Figure 10.Vector Displacement at mode 10

6.2 Static and modal analysis for the stiffened panel at thickness 0.0016m.

Modal analysis of the stiffened panel at thickness 0.0016m.
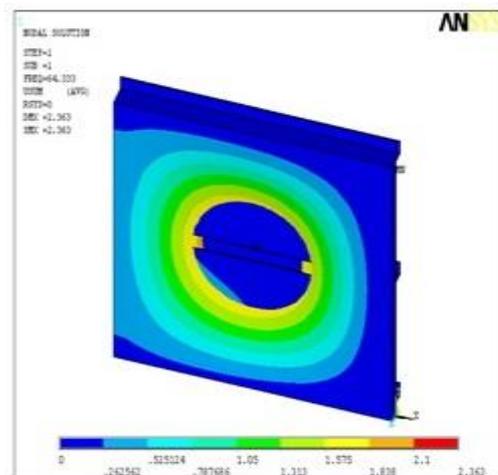
Figure 11. Vector Displacement mode 1
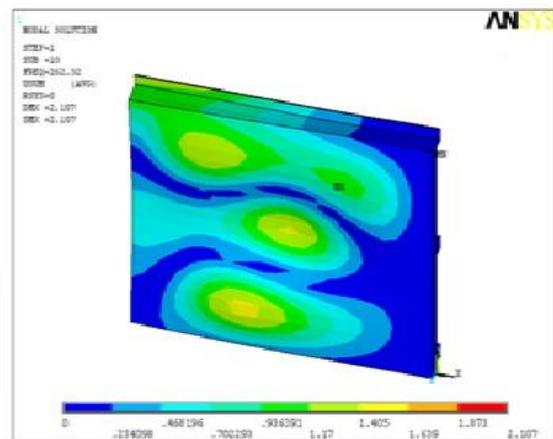
Figure 12. Vector Displacement at mode 10

6.3 Static and modal analysis for the stiffened panel at thickness 0.0012m.

Modal analysis of the stiffened panel at thickness 0.0012m.

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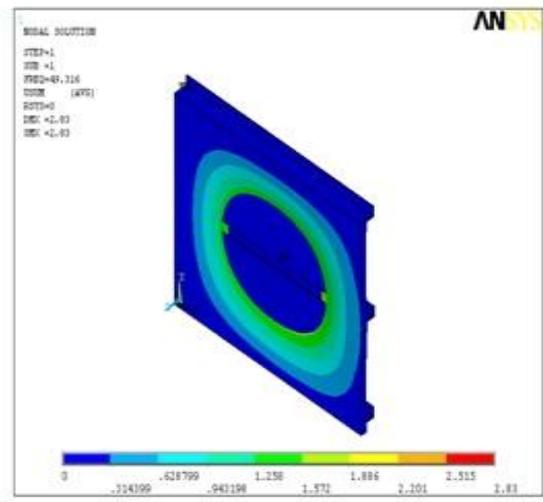
Figure 13. Vector Displacement at mode 1
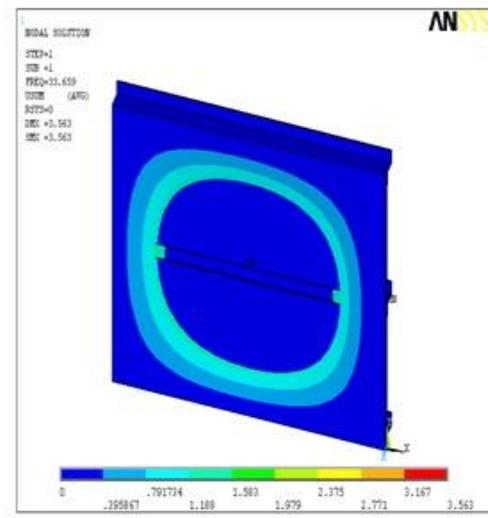
Figure 15. Vector Displacement at mode 1
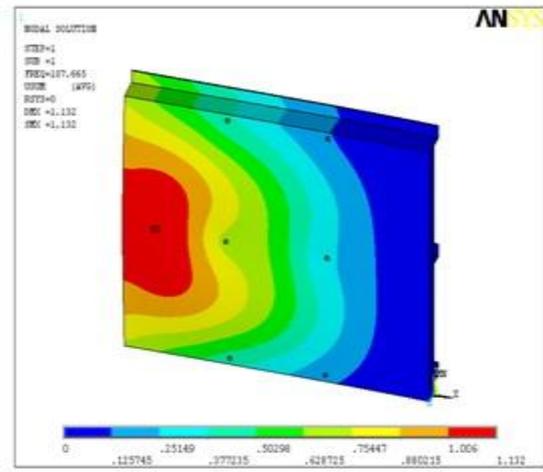

Figure 14. Vector Displacement at mode 10

Figure 16. Vector Displacement at mode 10

6.4 Static and modal analysis for the stiffened panel at thickness 0.0008m.

Modal analysis of the stiffened panel at thickness 0.0008m.

6.5 Static and Modal analysis of the stiffened panel with rivet at thickness0.002m.

In this analysis we are considering stiffened panels with riveted holes, a maximum displacement is obtained at different for different thickness from the analysis. Because of the presence of the stiffeners, the panel will tend to bend. In the process it has a out of plane displacement which is shown in below figures at different thickness at applied boundary conditions.

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- P =8896N,
- E = 70e^9 N/m^2
- u= 0.334

Modal analysis of the stiffened panel at thickness 0.002m for riveted holes.
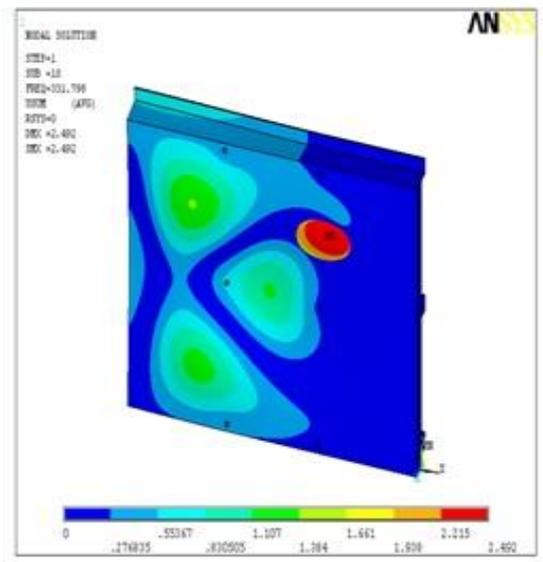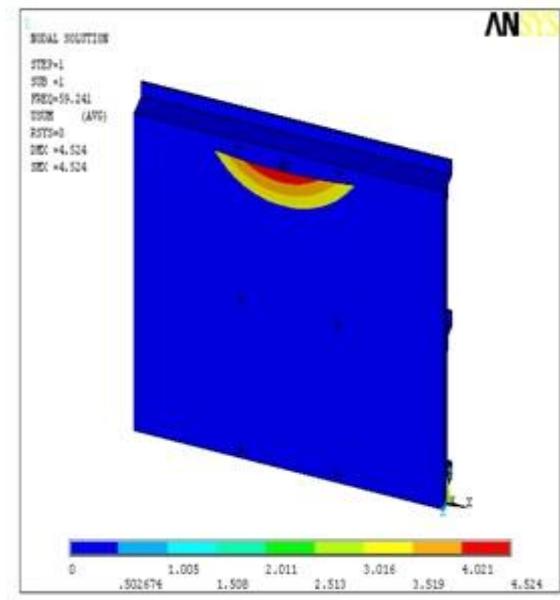

Figure 17. Vector Displacement at mode 1

Figure 18. Vector Displacement at mode 10

6.6 Static and Modal analysis of the stiffened panel with rivet at thickness0.0016m.

Modal analysis of the stiffened panel at thickness 0.0016m for riveted holes.
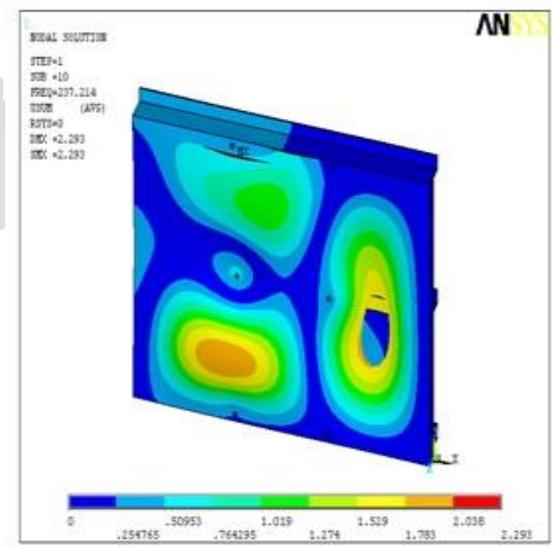

Figure 19. Vector Displacement at mode 1

Figure 20. Vector Displacement at mode 10

6.7 Static and Modal analysis of the stiffened panel with rivet at thickness0.0016m.

Modal analysis of the stiffened panel at thickness 0.0012m for riveted holes.
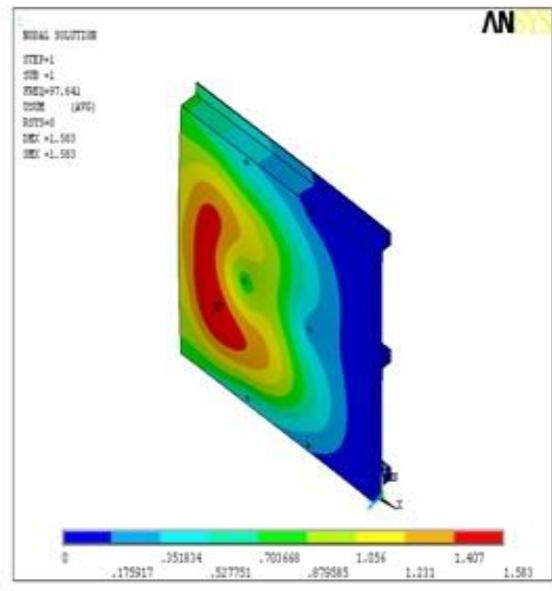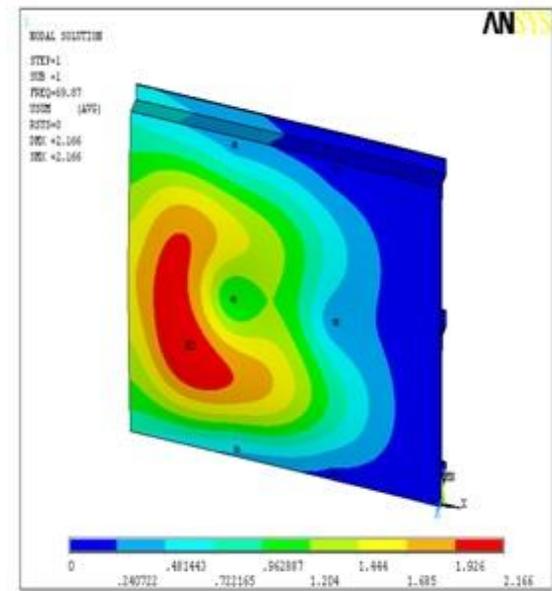
Figure 21. Vector Displacement at mode 1
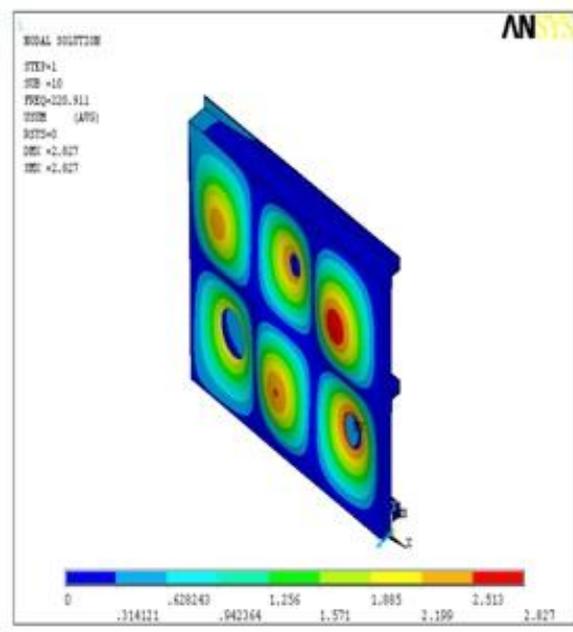

Figure 23. Vector Displacement at mode 1
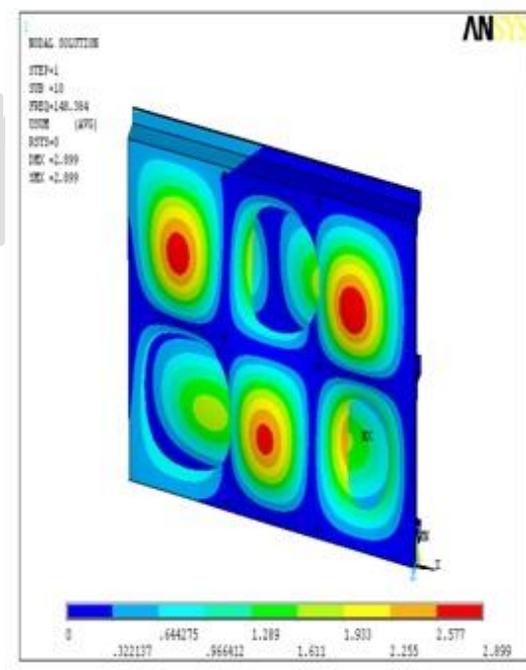

Figure 22. Vector Displacement at mode 10

Figure 24. Vector Displacement at mode 10

6.8 Static and Modal analysis of the stiffened panel with rivet at thickness0.0008m.

Modal analysis of the stiffened panel at thickness 0.0008m for riveted holes.

7. Conclusion

In this project i had done the contact analysis between skin and stiffened panel. I give the contact condition between the skin and stiffened panel nodes. So that

when we are apply load on the skin the stiffened panel also deform and vice versa.

The natural frequency of stiffened panel thickness of 0.8mm giving higher frequency

The natural frequency of stiffened panel with riveted hole thickness of 0.8mm giving higher frequency.

The thickness of the stiffened panel 1st mode is very low compare to stiffened panel thickness 0.8mm, 1.6mm, 2mm.

The thickness of the stiffened panel with riveted hole is the 1st mode is very low compare to stiffened panel thickness 0.8mm, 1.6mm, 2mm.

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