FEM Analysis of Practical Airframe Stiffened Panel

MALLIKARJUN N GUJAMAGADI[1], MANGALAPPA M NAIK[2]

[1,2] Senior Grade Lecturer, Department of Mechanical Engineering, B.V.V.S Polytechnic(Autonomous), Bagalkot, Karnataka, India.

Abstract: A stiffened panel is a generic representative structural element of an airframe structure. A stiffened panel is therefore considered for the analysis. It is almost impossible to completely represent the structure with all the structural details in the finite element modeling. Nevertheless it is important to represent the necessary structural features in the finite element model to get the correct response of the structure for a given loading condition. This is where an understanding of the structural response of the multiple load paths, multi layered, and discretely connected airframe structure helps. It will be shown that the conventional method of representing the structure in FEM will lead to inappropriate stress distribution and incorrect identification of critical locations. Appropriate FEM will be used to represent the details of the stiffened panel.

Index Terms – Structural analysis, FEM, Stiffened panel.

I INTRODUCTION TO AIRCRAFT STRUCTURE:

Aircrafts are generally built-up from the basic components of wings, fuselage, tail units and control surfaces. Each component has one or more specific functions and must be designed to ensure that it can carry out these functions safely. Any small failure of any of these components may lead to a catastrophic disaster causing huge destruction of lives and property. When designing an aircraft, it's all about ending the optimal proportion of the weight of the vehicle and payload. It needs to be strong and stiff enough to withstand the exceptional circumstances in which it has to operate. Durability is an important factor. Also, if a part fails, it doesn't necessarily result in failure of the whole aircraft. It is still possible for the aircraft to glide over to a safe landing place only if the aerodynamic shape is retained-structural integrity is achieved.

The basic functions of an aircraft's structure are to transmit and resist the applied loads; to provide an aerodynamic shape and to protect passengers, payload systems, etc., from the environmental conditions encountered in flight. These requirements, in most aircraft, result in thin shell structures where the outer surface or skin of the shell is usually supported by longitudinal stiffening members and transverse frames to enable it to resist bending, compressive and torsional loads without buckling. Such structures are known as semi-monocoque, while thin shells which rely entirely on their skins for their capacity to resist loads are referred to as monocoque.

The load-bearing members of these main sections, those subjected to major forces, are called the airframe. The airframe is what remains if all equipment and systems are stripped away. In most modern aircrafts, the skin plays an important role in carrying loads. Sheet metals can usually only support tension. But if the sheet is folded, it suddenly does have the ability to carry compressive loads. Stiffeners are used for that. A section of skin, combined with Stiffeners, called stringers, is termed a thin-walled structure.

MAJOR AIRCRAFT COMPONENTS:

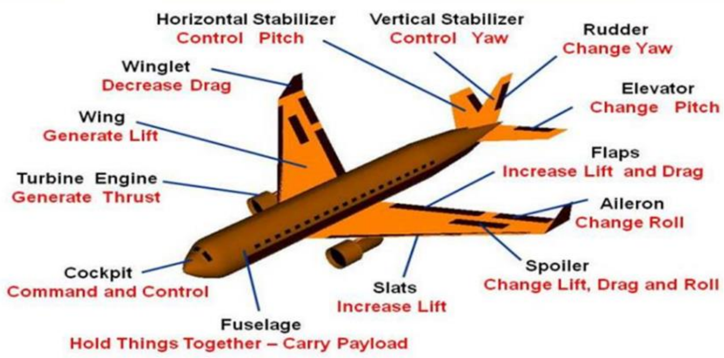

Fig.1.1 Airplane parts and its function

Fuselage:

The main body structure is the fuselage to which all other components are attached. The fuselage contains the cockpit or flight deck, passenger compartment and cargo compartment. While wings produce most of the lift, the fuselage also produces a little lift. A bulky fuselage can also produce a lot of drag. For this reason, a fuselage is streamlined to decrease the drag. We usually think of a streamlined car as being sleek and compact - it does not present a bulky obstacle to the oncoming wind. A streamlined fuselage has the same attributes. It has a sharp or rounded nose with sleek, tapered body so that the air can flow smoothly around it.

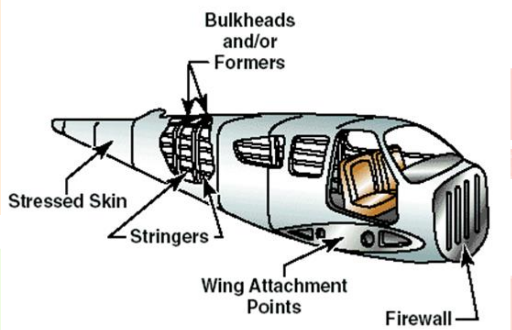

Fig 1.2 air craft fuselage

Unlike the wing, which is subjected to large distributed air loads, the fuselage is subjected to relatively small air loads. The primary loads on the fuselage include large concentrated forces from wing reactions, landing gear reactions and pay loads. For airplanes carrying passengers, the fuselage must also withstand internal pressures. Because of internal pressures, the fuselage often has an efficient circular cross-section. The fuselage structure is a semi-monocoque construction consisting of a thin shell stiffened by longitudinal axial elements (stringers and Longerons) supported by many traverse frames are rings (Bulkheads) along the length. The fuselage skin carries the shear stresses produced by torques and transverse forces. It also bears the hoop stresses produced by internal pressures. The stringers carry bending moments and axial forces. They also stabilize the thin fuselage skin.

Wings:

The wings are airfoils attached to each side of the fuselage and are the main lifting surfaces that support the airplane in flight. Wings vary in design depending upon the aircraft type and its purpose. Most airplanes are designed so that the outer tips of the wings are higher than where the wings are attached to the fuselage. This upward angle is called the dihedral and helps keep the airplane from rolling unexpectedly during flight. Wings also carry the fuel for the airplane. The wing is a framework made up of spars, ribs, skin and (possibly) stringers. This is the main lifting surface.

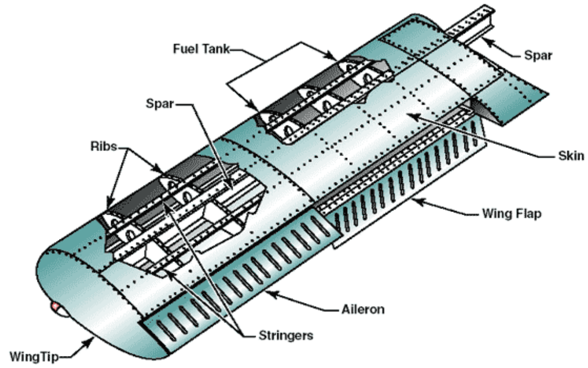

Fig.1.3 wing and its components

AIRCRAFT MATERIALS:

Metallic Materials

The most common metals used in aircraft construction are aluminum, magnesium, titanium, steel, and their alloys. Traditional metallic materials used in aircraft structures are Aluminum, Titanium and steel alloys. In the past three decades applications of advanced fiber composites have rapidly gained momentum. To date, some modern military jet fighters already contain composite materials up to 50% of their structural weight. Selection of aircraft materials depends on any considerations, which can in general be categorized as cost and structural performance. Cost includes initial material cost, manufacturing cost and maintenance cost. The key material properties that are pertinent to maintenance cost and structural performance are

- Density (weight)
- Stiffness (young's modulus)
- Strength (ultimate and yield strengths)
- Durability (fatigue)
- Damage tolerance (fracture toughness and crack growth)
- Corrosion

Seldom is a single material able to deliver all desired properties in all components of the aircraft structure. A combination of various materials is often necessary. Table 1.1 lists the basic mechanical properties of some metallic aircraft structural materials.

Table 1.1 Materials & properties

Material	Properties		
	E GPa(msi)	γ	σ_t MPa(ksi)
Aluminum			
2024-T3	72(10.5)	0.33	449(65)
7075-T6	71(10.3)	0.33	538(78)

σ_t = Tensile ultimate strength, σ_y = Tensile Yield strength

II.LITERATURE REVIEW

Erdogan et al [1] studied on fatigue and fracture of cylindrical shells containing a circumferential crack, here the cylindrical shells containing a circumferential crack subjected to axial tension and stress from the vertical bending of fuselage are considered. and Longitudinal crack growth occurs mainly due to hoop stresses developed due to internal pressurization, He concluded that the longerons does not play much role in arresting longitudinal cracks in the fuselage structures. But Bulkheads are more effective in arresting longitudinal cracks.

H. Vlieger [2] proposed a method that relates the crack resistance of a stiffened panel to that of an un-stiffened sheet. Vlieger takes full account of sheet-stringer interaction in the cracked region. Vlieger bring into account the stiffener failure criterion , crack arrest criterion and a brief about residual strength theory.

H. Vlieger [3] studied the effect of crack elements in a built-up structure. He characterized that crack built-up structures has the ability to transfer load from the cracked to the intact element, thus reliving the most critical part of the structure. The interaction of intact and cracked element will be essential for residual strength and crack propagation behavior of the built-up structures as a whole.

Thomas P. Rich et al [4] derived general equation for the prediction of structural failure of two dimensional cracked components in which the geometrical features of the component affect the stress intensity factor of the crack. The general equation is used to construct a new fracture diagram for a uniformly stressed sheet containing a crack which is constrained by two stiffening elements fastened to the sheet.

Pir M. Toor [5] focuses its attention on designing a fail-safe fuselage structure considering circumferential cracks under stresses from vertical bending of the fuselage. The analysis of these types of cracks is complex, first due to the complex structural configuration

III.INTRODUCTION TO STIFFENED PANEL:

Stiffened panels are the most generic structural elements in an airframe. The fuselage is a cylindrical shell made up of stiffened panel but for the analysis, small part of fuselage is taken, which is rectangular stiffened panel and relevant loads and boundary conditions are applied and analyzed. The stiffened panel consists of

- Skin
- Bulkhead
- Longerons (stringer)and
- Fasteners (rivets).

The geometric dimensions of the stiffened panel are discussed in session 2.2, which is modeled by using CATIA V5 software and analyzed by using FEA tool MSC NASTRAN, MSC PATRAN. The material will be taken as 2024-T3 sheet aluminum alloy.

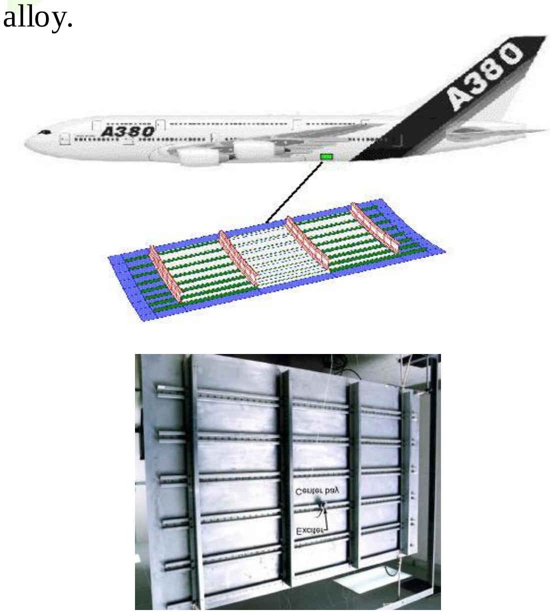

Fig 1.4 CAD model of individual component of the stiffened panel

GEOMETRIC CONFIGURATION OF THE STIFFENED PANEL:

Geometric modeling is carried out cad software .

Skin

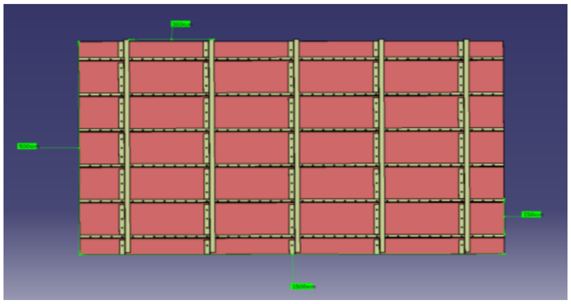

Fig 1.5 Detailed view of skin

The above fig 1.5 shows the skin dimensions. Skin has the thickness of 1.5 mm. The skin houses rest of the components like Bulkheads, Longerons, Tear strap, which are assembled by riveting process, From fig 1.5 , it is clear that the rivets which are in rows holds the skin with longeron and the rivets which are in columns holds the tear strap and bulkhead, distance between the rows is 150mm and distance between the columns is 300mm, diameter of the rivet used is 5mm pitch of the rivet is 37.5mm.The fig 1.6 shows the CAD model of the skin with rivet holes.

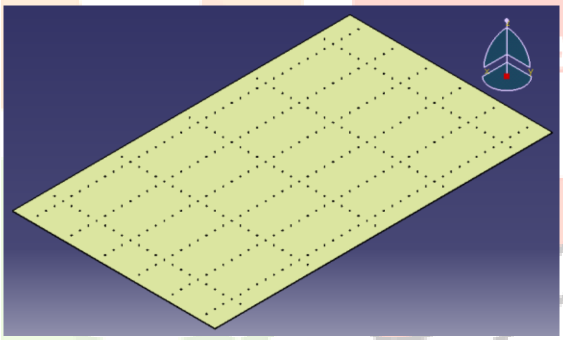

Fig 1.6 Shows the CAD model of the skin with rivet hole

Complete Finite element model of the stiffened panel

Finite element meshing is carried out for all the components of the stiffened panel such that there is a node present at the point where riveting is to be done and fine meshing is done at the critical sections where stresses are expected to be more.

The complite Finite element model of the stiffened panel as show in figure 1.7

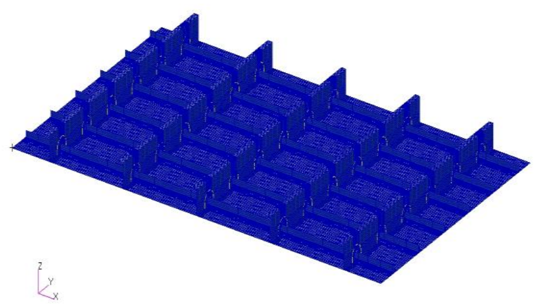

Fig 1.7 Finite element model of the stiffened panel

IV. RESULT AND DISCUSSION:

LOADS AND BOUNDARY CONDITIONS:

A differential pressure of 9 psi (0.062066MPa) is considered for the current case. Due to this internal pressurization of fuselage (passenger cabin) the hoop stress will be developed in the fuselage structure. The tensile loads at the edge of the panel corresponding to pressurization will be considered for the linear static analysis of the panel.

Hoop stress is given by

$$\sigma_{hoop} = \frac{p*r}{t}$$

Where

Cabin pressure (p)=9 psi=0.062066MPa

Radius of curvature of fuselage(r) = 1250mm

Thickness of skin (t) = 1.5mm

After substitution of these values in we will get

$$\sigma_{hoop} = 5.27 \text{ Kg/mm}^2$$

We know that

$$\sigma_{hoop} = \frac{P}{A}$$

Above equation can be written as

$$P = \sigma_{hoop} * A$$

1) Uniformly distributed tensile load is applied on either side of the stiffened panel in Y axial direction

<u>Load on the skin</u>

Here

P_s=Load on skin

σ_{hoop} =5.27Kg/mm^2

A=Cross sectional area of skin in mm^2

i.e. Width *Thickness(1500*1.5)

Substituting these values in the eq we get

$$P_s = 11857.5\text{kg}$$

Uniformly distributed load on skin will be

$$P_s = 11857.5/1500$$
$$= 7.870 \text{ Kg/m}$$

<u>Load on Bulkhead</u>

Here

P_b =load on Bulkhead in Kg

σ_{hoop} =5.27 Kg/mm^2

A =Cross sectional area of each Bulkhead in mm^2

i.e. (18+67+20)*1.5 =157.5 mm^2

Substituting the values in eq we get

$$P_b = 830.02\text{Kg on each BH}$$

Uniformly distributed load on Bulkhead will be

$$P_b = 830.02/105$$

$$= 7.870 \text{ Kg/mm on each BH}$$

Generally results obtained from the finite element analysis of the stiffened panel:

Pre-processing and post-processing is carried out by using MSC Patran software and Solved by using MSC Nastran (solver) software. The response of the stiffened panel in terms of displacements and stresses due to loads and boundary conditions described in the previous sections are explained in the following sections.

Displacement contour of the stiffened panel

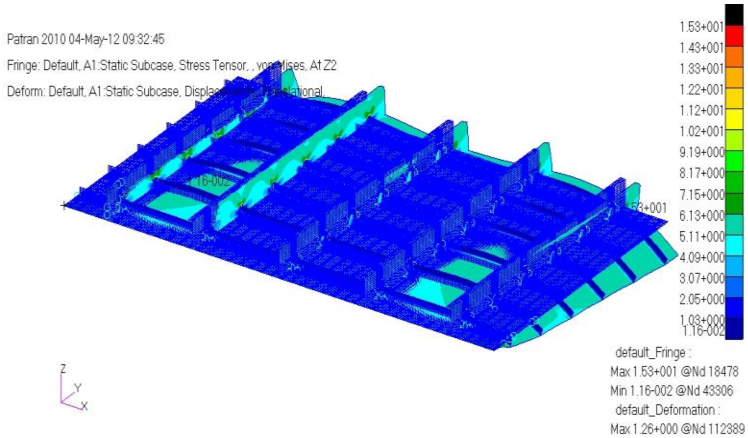

Fig 1.8 Displacement contour of the stiffened panel

The Fig 1.8 shows the displacement contour of stiffened panel. Displacement contour increases from fixed end to loading end and it is shown by different colors fringes where white color showing minimum magnitude of displacement while red color showing maximum magnitude of displacement. The panel is constrained in the loading direction at the mid-section. One can observe a symmetrical displacement contours from the mid-section in the above figure.

Stress contour of the stiffened panel

Skin

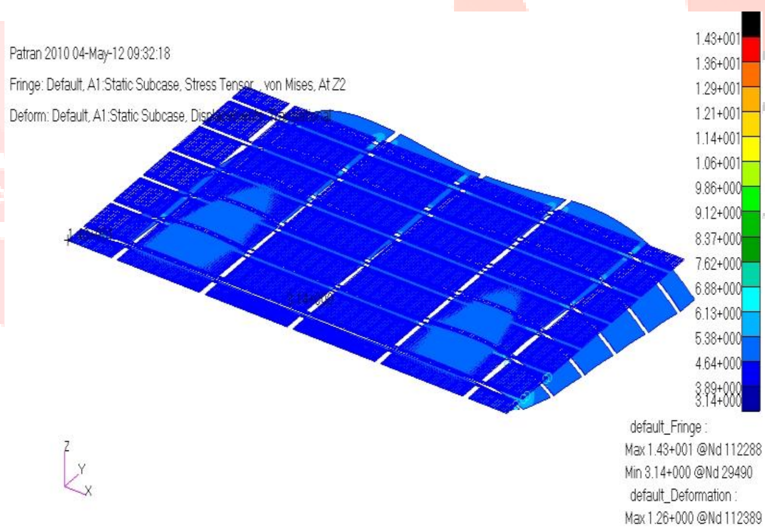

Fig 1.9 Stress contour for skin

Fig 1.9 shows the stress contour on the skin from global analysis results. It is clear that the maximum stress on skin is at the rivet location where the rivets are used to fasten the tear strap, bulkheads, longerons and skin. The magnitude of maximum tensile stress is 3.15e1kg/mm^2 in the loading direction can be observed from the Fig 1.9. The maximum stress locations are the probable locations for crack initiation. Invariably these locations will be at rivet locations in the skin. Representation of layered structure is important in identifying critical stress locations, integral representation will miss lead as for as critical locations are concerned.

V.CONCLUSION:

- A stiffened panel which is genetic structural element of the fuselage structure is evaluated analytically for its crack arrest capability.
- The internal pressure is one of the main loads that the fuselage needs to hold. In the current project also pressurization load case is considered for the analysis.
- Finite element analysis (FEA) approach is used for structural analysis of the stiffened panel.

VI. REFERENCE

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