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STRESS ANALYSIS OF A SPLICE JOINT IN AN AIRCRAFT FUSELAGE WITH THE PREDICTION OF FATIGUE LIFE TO CRACK INITIATION

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ABSTRACT

The overall objective of this paper is to compare various properties of a splice joint in an aircraft fuselage with the prediction of fatigue life to crack ignition using composite material. The application of the composite material in the aircraft will help to increase the strength to weight ratio. The aluminium alloys are used in the splice joints. By replacing these alloys by composite material (Kevlar 49) the fatigue life of the fuselage can be increased. The fatigue life of the composite material is also determined.

Keywords: Aircraft, Composite, Fuselage, Fatigue, Materials.

INTRODUCTION

A complete airplane structure is manufactured from many parts. These parts are made from sheets, extruded sections, forgings, castings, tubes, or machined shapes, which must be joined together to form subassemblies. The subassemblies must then be joined together to form larger assemblies and then finally assembled into a completed airplane. Many parts of the completed airplane must be arranged so that they can be disassembled for shipping, inspection, repair, or replacement, and are usually joined by bolts or rivets. In order to facilitate the assembly and disassembly of the airplane.

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.

In 1891 a German Engineer named Otto Lilienthal studied aerodynamics and designed a glider that can fly a long distance. In 1891 itself Samuel Langley, realized that the power is required to help the man to fly. He built a model and named as "aerodrome". Octave Chanute was a successful engineer who designed several aircrafts around 1894. Then the Wright Brothers designed and had the first flight in 1903. From there the evolution of flying and aircrafts started.

MATERIALS

In the ancient time of manufacturing of aircrafts, the wood and fabric constructions are there due to the availability, low weight and prior manufacturing experience. The streamlining was not a primary consideration during that time, so the aircrafts are of low speed. For structural strength wires, braces, and other devices were used. The woods like 'spruce' are used, which are light and strong. The designers studied the different parts of the aircraft individually, the strength and wind resistance so as to increase the speed. Bracing wires were given a streamlined shape, and some manufacturers began to make laminated wood fuselages of monocoque construction (stresses carried by the skin) for greater strength, better streamlining, and lighter weight.

The metals are started using in aircraft as the aircrafts with wood and fabric were difficult to maintain and subject to rapid deterioration when left out in the elements. From that period various material evolution has been take place in the aircraft structure. By use of metals in aircraft enabled the reduction in weight and the easy of construction. High strength and thermal resistance like properties are added to the aircraft by the introduction of new materials and new alloys. Other than Aluminium, titanium, molybdenum, steel etc. are also used in the aircraft to achieve different properties. The corrosion and metal fatigue were two major problems raised by the use of metals. In 1950's the composite materials are introduced in the aircrafts (Boing 707) which helped to reduce the weight of the aircraft and to increase the strength of the parts. Then different types of composite materials are developed, which helped to increase different properties of the aircraft. In the latest airbus there was 53% was of composite material (Airbus A350XWB).

FATIGUE FAILURE

One of the major failure in the aircraft is fatigue failure. The fatigue failure is a failure that occurs due to the cyclic load on an object or due to cyclic stress or fluctuating loads. It is a phenomenon that occurs due to the initiation and propagation of the crack and it becomes unstable, cause the fatigue failure. There are mainly 3 stages included in fatigue failure.

- Crack initiation
- Crack propagation
- Fracture due to unstable crack growth

The above diagram shows the S-N curve to represent the low cycle fatigue and high cycle fatigue. The S-N curve is a graph drawn between the stress and the number of cycles of load to failure. The graph is drawn to represent the fatigue failure of an object. By plotting S-N curve the fatigue life can be predicted.

FINITE ELEMENT ANALYSIS

Analysis procedure in PATRAN is divided into two parts

- Pre-processing
- Post-processing

In Pre-processing, Geometry, Meshing, Adding Material Property, Assigning Boundary Conditions etc. are done. Where in post-processing the interpretation of results generated during the processing stage.

STRESS ANALYSIS THROUGH FEA

The stress analysis of the aircraft fuselage is done by 3 stages. A segment of the fuselage is considered for the finite element analysis. The splice joint of the fuselage is simulated in the model. After the global analysis is done and the result is taken the flat plate analysis is to be carried out. The flat panel is considered from the selected area of a fuselage. A local stress analysis is carried out to capture the maximum stress and stress distribution around the rivet hole. The geometrical specifications and loading conditions are taken from the flat panel analysis. This problem is similar to plate with a hole, but the loading is at both plate edge and the rivet hole. For global analysis the the aircraft fuselage is selected with,

Length, L=3000mm
Radius, R=1000mm

After meshing the fuselage structure in to various elements the load and boundary conditions have to be applied on the fuselage structure.

To find the stress in the fuselage we have to apply the force to the fuselage. Here we are analyzing the fuselage with an internal pressure of **6.35psi**. By applying the properties of the material **Kevlar 49**, the result for stress contour of fuselage, at Max principal stress is determined.

After global analysis flat plate analysis is done to determine the stress concentration at the splice joint in the joint area in the fuselage.

In the flat plate analysis the analysis is done after applying the properties of Kevlar 49. The boundary conditions play a crucial role in capturing the true response of the structure. Here X-direction is allowed for load to act at both ends, where tension-tension load acting at either end.

Fuselage is subjected to an internal pressure of **6.35 psi**. (0.04374 N/mm²). This load is converted in to appropriate hoop stress as follows.

$$\sigma = \frac{p \cdot r}{t} = \frac{0.04374 \cdot 1000}{1.8} = 24.30 \text{ N/mm}^2$$

$$\sigma = \frac{P}{A}$$

$$24.30 = \frac{P}{160 \approx 1.8}$$

Pressure, $P = 7000 \text{ N}$

A maximum principal stress of 40.3 MPa can be observed from the stress contour.

After both global and flat plate analysis the local analysis has to be done to determine the maximum stress region when the load is applied. The local analysis is done after selecting the region having the riveted joints at splice joint. Selecting one rivet hole region and after applying the load the maximum stress location can be easily determined.

Dimensions of plate with a hole.

- Length, $L=40\text{mm}$.
- Width, $W=10\text{mm}$.
- Radius, $R=2.5\text{mm}$

In this case where at one end all degrees of freedom is constrained and at other end P1 is applied, at the riveted hole region P2 is applied.

P1 is the free body load acting on the skin panel due to the tensile loading

P2 is the free body load acting on the rivet due to the tensile Loading

- $P1=455\text{N}$
- $P2=353\text{N}$

The above figure shows the local analysis that done at the riveted region in the fuselage and the maximum stress location is determined.

FATIGUE LIFE PREDICTION

Stress contour of plate with a hole for fatigue life prediction. From FEA,

$$\sigma_{\max} = 236 \text{ N/mm}^2$$

From Kevlar 49 (Material),

- UTS (Min) - 3600MPa
- UTS (Max) - 3620MPa
- UTS (Avg) - 3610MPa

Therefore,

$$\frac{\sigma_{\max}}{\sigma_{\text{uts}}} = \frac{236}{3610} = 0.06$$

From S-N curve, fatigue cycle to failure for 0.06 is 80,800 cycles.

CONCLUSION

The allowable stress for the material under consideration is 350 N/mm² and from the analysis, we have found that the maximum induced stress in the structure is 42.2 N/mm². This means that the structure is well within the safe limit for the load case considered. Experiments are carried out by using flat panel then it is co-related with fuselage. In this regard experimental cost will come down, when panel tested instead of entire fuselage and even time saving. As the maximum stress location is found at the rivet region. There should be detailed inspection of the structure at this critical locations i.e. rivet region. The maintenance personnel or service personnel will be instructed to check at the critical location first. The maintenance personnel should also be capable to replace the cracked rivet with a new one and

strengthen the region such that there shall not be any disaster during the operation of the aircraft. The prediction method of fatigue life helps the maintenance personnel by giving smaller number of inspections.

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