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A Study of Fatigue Failure of Composite Fuselage Structures

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Abstract;- *The use of composite materials in modern aircraft fuselage structures has significantly improved strength-to-weight ratio, corrosion resistance, and fuel efficiency. However, fatigue failure remains a crucial concern due to the cyclic loading conditions experienced during flight. This study investigates the fatigue behavior of composite fuselage structures, analyzing crack initiation, propagation, and failure mechanisms under repetitive loading conditions.*

An experimental setup is developed to simulate real-world aerodynamic and mechanical stresses on fuselage panels composed of carbon fiber-reinforced polymer (CFRP) and glass fiber-reinforced polymer (GFRP). Non-destructive testing (NDT) techniques, such as ultrasonic scanning and digital image correlation, are used to monitor micro crack formation and delamination. Additionally, finite element analysis (FEA) is conducted to predict stress distribution and fatigue life based on different loading scenarios.

The results indicate that composite fuselage structures exhibit superior fatigue resistance compared to traditional aluminum alloys, but localized stress concentrations at joints and fastener locations can lead to premature failure. The study provides design recommendations, material selection insights, and improved maintenance strategies to enhance the durability and reliability of composite fuselage structures in aerospace applications.

Keywords: *CATIA, FEA, Composites, Semimonocoque, aluminum, Finite Element, Fatigue, Safety Margins.*

I. INTRODUCTION

Aircraft manufacturers have been gradually increasing its reliance on composite materials. For example, Boeing 777 featured an all-composite empennage and composite floor beams. Nevertheless, the composite materials community is very much aware of the cost implications of introducing more composite materials. It was only when a technological breakthrough on the manufacturing side came about that it considered widespread use of such materials, for example, in the Boeing 787. Basically, this involves the same fiber-resin system as used in the Boeing 777 empennage but with radically different automated fiber-placement techniques. These techniques allow rapid and accurate positioning of fibers onto a mandrel that will initially create the stringers and then apply the fuselage skin to varying thicknesses, as desired. Each fuselage section is then autoclave cured and the mandrels are then disassembled and removed. The Boeing 787 fuselage is built in five main sections and composite materials that account for 50% of the aircraft's total structural weight. (Aircraft Technology Engineering Maintenance, 2005) Both Boeing and Airbus have recognized that they have the opportunity to increase the thickness of composite structures where there is a high probability of impact damage. Areas such as doors, door surrounds, wing tips, wing leading and trailing edges and wing-to-body fairings are all prone to ground vehicle impact damage and increasing the thickness of any composite structures in these areas should reduce the probability of significant damage.

The possibility of replacing damaged components at these locations still remains where the designs permit. (Aircraft Technology Engineering & Maintenance, 2005) Boeing intends to capitalize on in its 787 CFRP fuselage design as that it can work with larger pressure (from a cabin altitude of 8,000ft to a cabin altitude of 6,000ft) without adding substantial weight to the airframe structure. Furthermore, in view of the excellent corrosion resistance of advanced composites, Boeing is also contemplating the introduction of a cabin humidifier, also intended to make the flight experience a more pleasurable one. Finally, Boeing intends to make the windows on the 787 significantly larger than traditional windows. Airbus has claimed that it intends to do the same in each of these areas on it's a350 (Aircraft Technology Engineering & Maintenance, 2005; Wall, 2005). The one-piece, business jet fuselage, designed by Dassault Aviation in conjunction with BAE Systems, was manufactured using pre-impregnated carbon fiber slit tape and honeycomb core. Automated fiber placement enables manufacturability of a single-piece fuselage that can replace typical business jet structures made up of many individual components and thousands of fasteners (Leininger, 2005).The main scope of this paper is to present an analytical method of buckling analysis of laminated composite fuselage. This method was developed based on different references and it is demonstrated by the comparison between the analytical method results and the finite element analysis method results. A comparison between an aluminum fuselage and a composite fuselage is also presented showing the less weight advantage of the composite fuselage

II. FUSELAGE CONSTRUCTION

The proposed aircraft fuselage structure is a innovative fuselage concept. The whole fuselage is fabricated with Carbon fiber Reinforced Plastic (CFRP). The main advantages in this new design are: (1) very good integration; (2) faster fabrication and assembly (3) weight reduction (10-15%); (4) possibility of thickness variations; less waste of raw material; (6) higher passenger comfort level; (7) possibility of larger windows; (8) longer structural life (less sensitive to fatigue). There are also some disadvantages, although there are some possible solutions to overcome these disadvantages. The main disadvantages are: (1) electro-magnetic interference); (2) return of electrical current; (3) lightning protection ;(4) higher machinery investments; (4) higher certification costs.

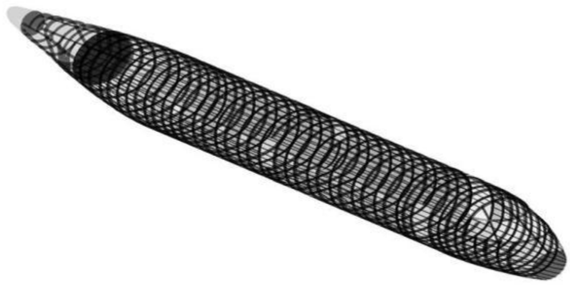


Fig 2.1 Fuselage Structural Layouts

The fuselage will be constructed in three parts along the longitudinal axis in order to facilitate the construction process and improve reparability. Each part of the fuselage will be manufactured by the FP (fibre Placement) process resulting in a single non-circular panel. All the stringers will be positioned in the mandrel of the ATL process and these stringers will be already fabricated and cured at this process stage. The result of the FP process will be the stringers mounted in the single non-circular panel skin. The next fabrication process is the panel skin cure. It is important to emphasize that the utilization of composite materials gives a possibility off thickness variation in almost all parts where they are used.

III. FATIGUE

Fatigue is a phenomenon associated with variable loading or more precisely to cyclic stressing or straining of a material. Just as we human beings get fatigue when a specific task is repeatedly performed, in a similar manner metallic components subjected to variable loading get fatigue, which leads to their premature failure under specific conditions.

In materials science, fatigue is the progressive and localized structural damage that occurs when a material is subjected to cyclic loading. The nominal maximum stress values are less than the ultimate tensile stress limit, and may be below the yield stress limit of the material. Fatigue occurs when a material is subjected to repeated loading and unloading. If the loads are above a certain threshold, microscopic cracks will begin to form at the stress concentrators such as the surface, persistent slip bands (PSBs), and grain interfa0ces. Eventually a crack will reach a critical size, and the structure will suddenly fracture.

3.1 characteristics of fatigue

In metal alloys, when there are no macroscopic or microscopic discontinuities, the process starts with dislocation movements, eventually forming persistent slip bands that nucleate short cracks.

Macroscopic and microscopic discontinuities as well as component design features which cause stress concentration (keyways, sharp changes of direction etc.) are the preferred location for starting the fatigue process. Fatigue is a stochastic process, often showing considerable scatter even in controlled environments. Fatigue is usually associated with tensile stresses but fatigue cracks have been reported due to compressive loads.

1.The greater the applied stress range, the shorter the life.

2.Fatigue life scatter tends to increase for longer fatigue lives.
- Damage is cumulative.
- Materials do not recover when rested.

Fatigue life is influenced by a variety of factors, such as temperature, surface finish, microstructure, presence of oxidizing or inert chemicals, residual stresses, contact etc., Some materials (e.g., some steel and titanium alloys) exhibit a theoretical fatigue limit below which continued loading does not lead to structural failure.

In recent years, researchers (see, for example, the work of Bathias, Murakami, and Stanzl-Tschegg) have found that failures occur below the theoretical fatigue limit at very high fatigue lives (10^9 to 10^{10}cycles). An ultrasonic resonance technique is used in these experiments with frequencies around 10–20 kHz.

3.2 FATIGUE STRENGTH FORMULATIONS

Fatigue strength experiments have been carried out over a wide range of stress variations in both tension and compression and a typical plot. Based on these results mainly, Gerber proposed a parabolic correlation and this is

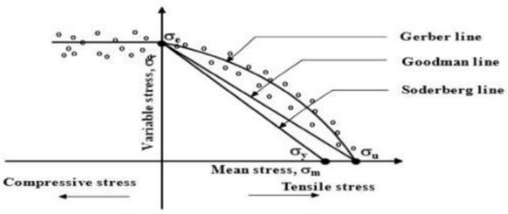
$$\left(\frac{\sigma_m}{\sigma_u}\right)^2 + \left(\frac{\sigma_v}{\sigma_e}\right) = 1 \quad \text{Gerber line}$$

given by Goodman approximated a linear variation and this is given by

$$\left(\frac{\sigma_m}{\sigma_u}\right) + \left(\frac{\sigma_v}{\sigma_e}\right) = 1 \quad \text{Goodman line}$$

Soderberg proposed a linear variation based on tensile yield strength σY and this is given by

$$\left(\frac{\sigma_m}{\sigma_y}\right) + \left(\frac{\sigma_v}{\sigma_e}\right) = 1 \quad \text{Soderberg line}$$



Graph 1.2 Fatigue Strength

IV. MATERIAL SPECIFICATION

Selection of aircraft materials depends on any Considerations, which can in general be categorized as cost and structural performance. The key material properties that are pertinent to maintenance cost and structural performance are

- Density
- Young’s modulus
- Ultimate and Yield strengths
- Fatigue strength
- Damage tolerance (fracture toughness and crack growth)
- Corrosion, etc.

4.1 .CARBON FIBER REINFORCED POLYMER

The Carbon-fiber-reinforced polymer, carbon-fiber reinforced plastic or carbon-fiber reinforced thermoplastic (CFRP, CRP, CFRTP or often simply carbon fiber, or even carbon), is an extremely strong and light fiber reinforced polymer which contains carbon fibers. The binding polymer is often a thermoset resin such as epoxy, but other thermoset or thermoplastic polymers, such as polyester, vinyl ester or nylon, are sometimes used. The composite may contain other fibers, such as aramid e.g. Kevlar, Twaron, aluminum, or glass fibers, as well as carbon fiber. The properties of the final CFRP product can also be affected by the type of additives introduced to the binding matrix (the resin) . The most frequent additive is silica, but other additives such as rubber and carbon nanotubes can be used. CFRPs are commonly used in the transportation industry; normally in cars, boats and trains, and in sporting goods industry for manufacture of bicycles, bicycle components, golfing equipment and fishing

Table 4.1 mechanical property of carbon composite

PROPERTY	**VALUE**	**UNIT**
Coefficient of thermal expansion -longitudinal	2.1	10^6K^{-1}
Coefficient of thermal expansion –transverse	2.1	10^6K^{-1}
Compressive strength – longitudinal	570	Mpa
Compressive strength – transverse	570	Mpa
Density	1.6	g cm^{-3}
Shear modulus –in plane	5	Gpa
Shear modulus – in plane	90	Mpa
Ultimate compressive strain – longitudinal	0.8	%
Ultimate compressive strain – transverse	0.8	%
Ultimate shear strain – in plane	1.8	%
Ultimate tensile strain – longitudinal	0.85	%
Ultimate tensile strain – transverse	0.85	%
Young modulus – longitudinal	70	Gpa
Young modulus – transverse	70	Gpa

4.2 APPLICATION OF CARBON EPOXY

The Boeing 787 Dreamliner, 50%. Specialist aircraft designer and manufacturer Scaled Composites have made extensive use of CFRP throughout their design range including the first private spacecraft Spaceship One. CFRP is widely used in micro air vehicles (MAVs) because of its high strength to weight ratio. In the MAVSTAR Project, CFRP reduces the weight of the MAV significantly and the high stiffness of the CFRP blades overcome the problem of collision between blades under strong wind. Concrete is a very robust material, much more robust than cement, and will not compress or shatter even under quite a large compressive force.

However, concrete cannot survive tensile loading (i.e. if stretched it will quickly break apart). Therefore to give concrete the ability to resist being stretched, steel bars, which can resist high stretching forces, are often added to concrete to form reinforced concrete. Shape memory polymer composites are high-performance composites, formulated using fibre or fabric reinforcement and shape memory polymer resin as the matrix. Since a shape memory polymer resin is used as the matrix, these composites have the ability to be easily manipulated into various configurations when they are heated above their activation temperatures and will exhibit high strength and stiffness at lower temperatures. They can also be reheated and reshaped repeatedly without losing their material properties. These composites are ideal for applications such as lightweight, rigid, deployable structures; rapid manufacturing; and dynamic reinforcement . Composites can also use metal fibres reinforcing other metals, as in metal matrix composite (MMC) or ceramic matrix (CMC), which includes bone (hydroxyapatite reinforced with collagen fibres), cermet (ceramic and metal) and concrete. Ceramic matrix composites are built primarily for fracture toughness, not for strength. Organic matrix/ceramic aggregate composites include asphalt concrete, mastic asphalt, mastic roller hybrid, dental composite, syntactic foam and mother of pearl.

V. GEOMETRIC MODELLING

Fuselage is a part of aircraft structure having cylindrical shape. Basically the fuselage structure consists of circumferential member called bulkheads to maintain circumferential shape and it is taking hoop stress which is created due to internal pressurisation. It has one longitudinal member also known as longenors which take longitudinal stress and support to the skin. Bulkheads, longenors, tear strap and skin are connected by rivet connection**.** The bulkheads has z cross section and total elevenbulkheads in the fuselage, the longenores has I cross section and total 36longenores.

Dimensions

Length of the fuselage = 4500mm

Radius of the fuselage = 1000mm

Thickness of skin = 2mm

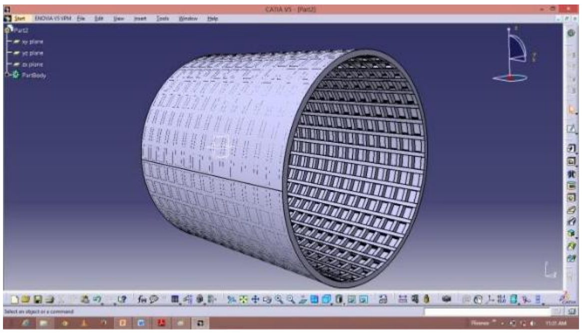


Fig 5.1 Fuselage Model

VI. STATIC STRUCTURAL ANALYSIS

6.1 Finite Elemene Model

The finite element method is a numerical technique for solving a range of physical problems. Often being the first choice for detailed structural analysis, finite element analysis discretizes the distribution of a variable through a complex geometry by dividing the region into small element of simple geometries. The elements are interconnected mathematically at the nodes ensuring that the boundary of each element is compatible with its neighbor whilst satisfying the global boundary conditions. All physical problems are broken down into series of matrix equations, where the governing equations of the system take a specific form for the type of problem to be solved. Finite element analysis, therefore, breaks down a complex problem into series of coupled equations in matrix form, which are normally solved using general purpose solvers.

6.2 Stresses and Deformation

The maximum stress developed near the rivet holes of both skin and long enores and nominal stress all over the stiffened panel. The stress near the hole is three times of nominal stress. In figure the red colour shows the maximum stress. At the rivet hole the localization stresses because the area reduces and also stress concentration

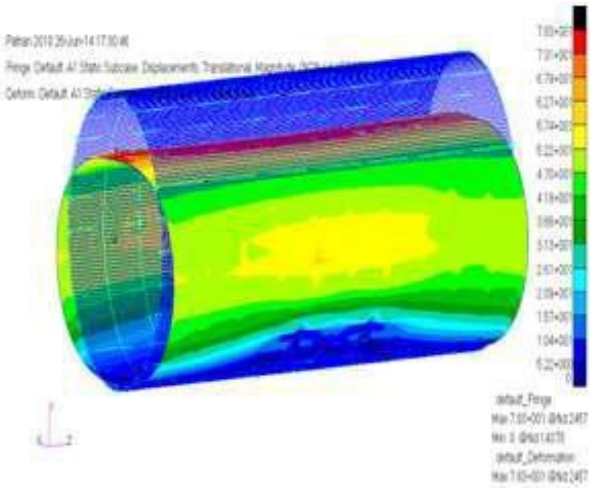


Fig6.1 Deformation

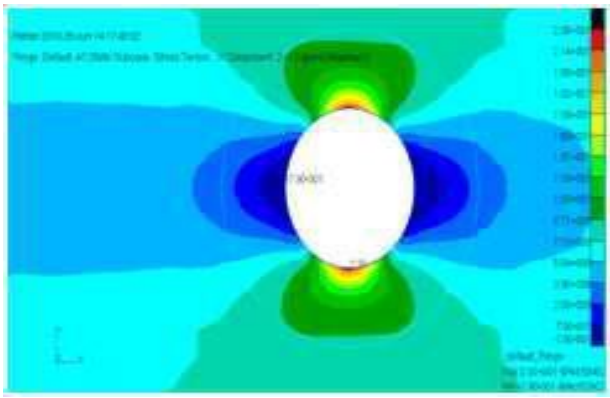


Fig Close-up view Stress distribution in stiffened Panel

VII. RESULTS AND DISCUSSION

The aircraft fuselage section carrying the different types of loads. In this we are considering the three main loads. We are applying the three loads to aluminium alloy and the same load is applied to the carbon epoxy. Then that loading results are taken and it is discussed. That loads are given below.

- Force
- Pressure

In this project we are analyzing the four solution for the aircraft fuselage section by applying load and pressure. In which each analysis should be carrying a pressure force. The name of the four analyses is given below. Equivalent stress and Equivalent strain,

Total deformation and directional deformation

Type	Total Deformation	Directional Deformation	Equivalent (von-Mises) Elastic	Equivalent (von-Mises) Stress

			Strain	
Minimum	0. m	-4.1866e-004 m	1.0127e-007 m/m	7088.9 Pa
Maximum	1.4831e-003 m	4.4636e-004 m	6.4485e-002 m/m	4.5139e+009 Pa
Type	Life	Damage	Safety Factor	
Design Life		1.e+10 cycles		
Minimum	0. cycles		1.9096e-002	
Maximum	1e+10	1.e+032		

Table 7.1Result For Carbon Composite

7.1 Calculation of Stress and deformation

1. Stress calculation:

Load on the skin = 3315.2 kg
Load on the longerons = 1036 kg

Cross section are = w × t mm^2

Cross section area of skin = 224 × 2 = 448 mm^2 Cross section are of longenores = (40×2) + (30×2) = 140mm^2

Total load on stiffened panel = 3315.2 + 1036 = 4351.2 kg Total area of the stiffened panel = 448 + 140 = 588 mm^2

σ = 7.4 kg/mm^2

The nominal distributed over the stiffened panel is 7.4kg/mm2 except near the rivet hole. At the rivet hole the stress is maximum and three times of the nominal stress is20 kg/mm^2.

VIII. FATIGUE LIFE ESTIMATION

8.1 S –N Curve

From typical constant life diagram for un-notched fatigue behaviour of carbon epoxy composite High-Master diagram is shown in below figure. The reference test condition R=0 used for obtain fatigue properties. For this condition smin=0 is called ‘pulsating tension’ under constant amplitude loading or Zero to tension loading. The numbers of cycles to failure from graph. Table shows the alternating stress level below which the material has an infinite life. For most engineering purposes, infinite is taken to be 1 million cycles. According to Palmgren-miner’s rule the stress amplitude is linearly proportional to the ratio of number of operation cycles to the number of cycles to failure from the graph gives the damage accumulated.

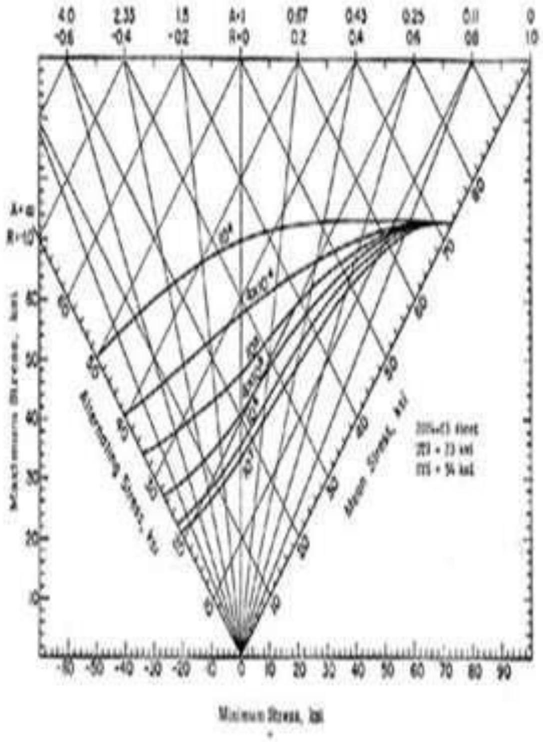


Figure 8.1: S-N curve

IX. CONCLUSION

On considering the fuselage of any aircraft, fatigue life is the main criteria. In our project we had found the fatigue life of an aluminum fuselage and carbon epoxy and from that we had concluded that the fuselage made up of carbon epoxy shows higher fatigue life than that of the aluminum fuselage. Under applied boundary condition, the carbon epoxy has maximum fatigue life (1e+10 Mpa) which is greater than that of the aluminum fuselage's maximum fatigue life (1e+09 Mpa). And the carbon epoxy has maximum damage rate (1e+32 Mpa) which is less than that of the aluminum fuselage's maximum damage rate (1e+36 Mpa). From this we have concluded that the Carbon epoxy fuselage is better than that of the aluminum fuselage Fatigue life estimated of the fuselage structure considering he maximum stress of the stiffened panel with the help ofS-N curve and Miner's rule. The damage accumulated of the Fuselage structure is 0.0044 from this it is observed that, the remaining life of the structure is 0.9966.

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