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## STRENGTH DETERMINATION AND FAILURE PREDICTION OF COMPOSITE MATERIALS USED IN AIRCRAFT STRUCTURES

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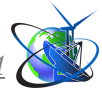
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**Abstract.** Composite materials are extensively used in modern aircraft structures due to their high strength-to-weight ratio, excellent fatigue resistance, and corrosion durability. Accurate determination of their mechanical strength is essential to ensure structural safety, reliability, and compliance with airworthiness certification requirements. Unlike metallic materials, composite materials exhibit anisotropic behavior and multiple interacting failure mechanisms, making strength prediction a complex task. This work presents a comprehensive investigation into the determination of strength of composite materials used in aircraft structures through experimental testing, analytical modeling, and numerical simulation. Classical Laminate Theory, commonly used failure criteria, and finite element-based progressive damage models are discussed. Representative stress-strain curves, fatigue behavior, failure envelopes, and statistical design allowables are included. The results demonstrate that an integrated multi-scale approach is essential for reliable strength prediction of aerospace composite structures.

**Keywords:** composite materials, aircraft structures, strength determination, failure criteria, CFRP

### Introduction

The aerospace industry has increasingly adopted composite materials in aircraft structures to achieve significant weight reduction while maintaining high mechanical performance. Modern commercial aircraft such as the Boeing 787 and Airbus A350 contain more than 50% composite materials by structural weight. This transition from traditional metallic materials to composites has introduced new challenges in structural analysis, testing, and certification. Unlike metals, composite materials are heterogeneous and anisotropic, meaning their mechanical properties depend strongly on fiber orientation, stacking sequence, and manufacturing quality [1-3]. Consequently, the determination of composite strength is more complex and requires a combination of experimental characterization, analytical modeling, and numerical simulation. This paper focuses on the determination of strength of fiber-reinforced polymer composites used in aircraft structures, emphasizing failure mechanisms, testing methodologies, and predictive models.[4].



## COMPOSITE MATERIALS IN AIRCRAFT STRUCTURES

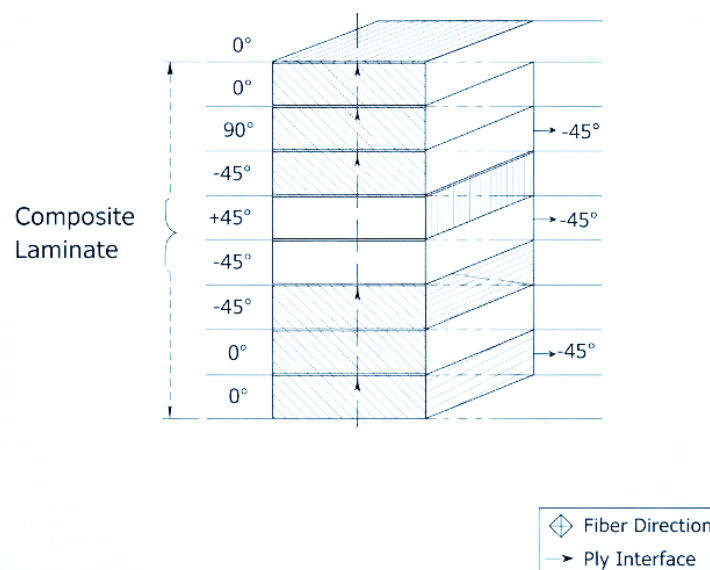
### Fiber-Reinforced Polymer Composite

Fiber-reinforced polymer (FRP) composites consist of high-strength fibers embedded within a polymer matrix. The fibers provide the primary load-carrying capability, while the matrix transfers load between fibers and protects them from environmental damage. The most common aerospace composites include carbon/epoxy, glass/epoxy, and aramid/epoxy systems.

Carbon fiber-reinforced polymers (CFRP) are widely used in primary aircraft structures due to their high stiffness, excellent fatigue resistance, and low density. Glass fiber composites are often used in secondary structures and radomes, while aramid fibers are preferred for impact-resistant components.

Polymer composites are increasingly utilized in the automobile and aerospace industries for weight reduction, improved performance, and environmental sustainability [4, 5, 6]. Fiber-reinforced plastics (FRP) play a crucial role in the automobile industry, offering weight reduction potential, high specific strength, and stiffness (Alberto, 2013) [7]. Polymer composites provide advantages over steel in terms of weight reduction, styling flexibility, tooling cost reduction, rust resistance, noise reduction, and higher damping properties [8, 9].

### Laminate Architecture



**Figure 1 - Schematic of a composite laminate showing multiple plies with different fiber orientations.**



Aircraft composite structures are typically manufactured as laminated plates composed of multiple plies with different fiber orientations. The stacking sequence is carefully designed to meet strength, stiffness, and damage tolerance requirements. Typical ply orientations include  $0^\circ$ ,  $\pm 45^\circ$ , and  $90^\circ$  relative to the principal loading direction. Fig.1

## MECHANICAL STRENGTH AND FAILURE MODES

### Strength Parameters

The strength of composite materials is characterized by several parameters:

- Longitudinal tensile and compressive strength
- Transverse tensile and compressive strength
- In-plane shear strength
- Interlaminar shear and tensile strength

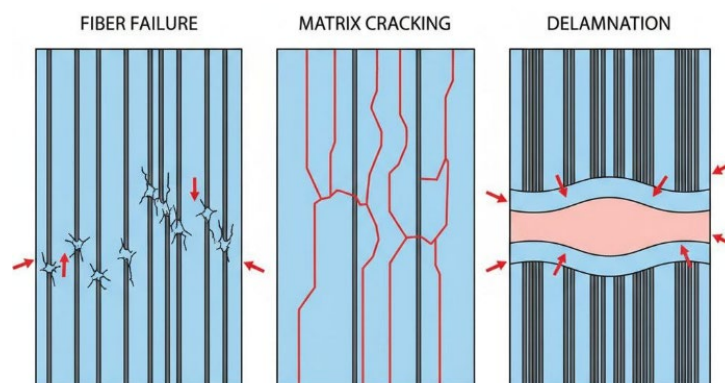
Each strength parameter corresponds to different loading conditions and failure mechanisms relevant to aircraft structures.

### Failure Mechanisms

Composite materials exhibit multiple failure modes, including:

- Fiber fracture under tensile or compressive loading
- Matrix cracking under transverse tension or shear
- Fiber–matrix debonding
- Delamination between plies

Delamination is particularly critical in aircraft structures because it may propagate under cyclic loading and significantly reduce residual strength. Fig.2



**Figure 2 - Common failure modes in composite laminates: fiber failure, matrix cracking, and delamination.**



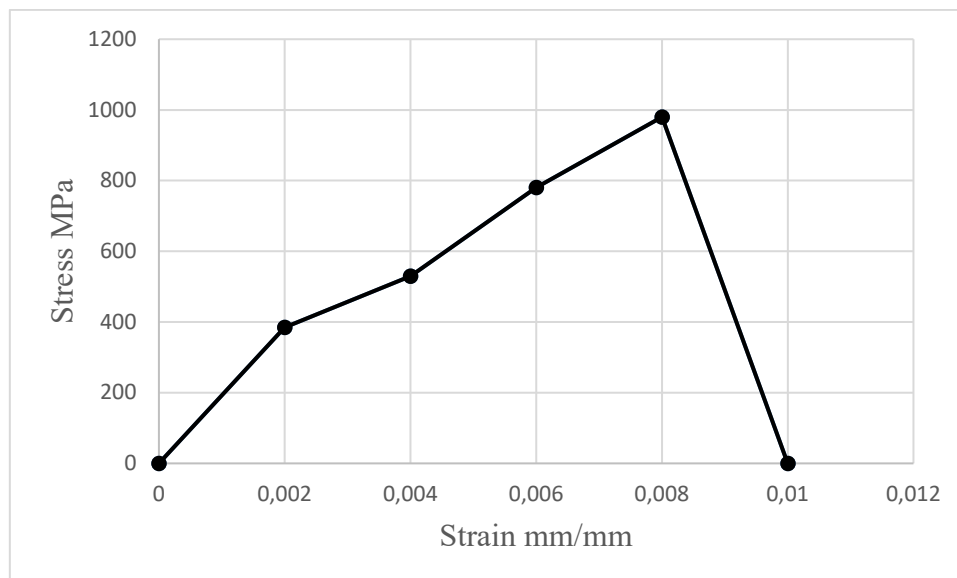
## EXPERIMENTAL DETERMINATION OF STRENGTH

### Coupon-Level Testing

Experimental determination of composite strength begins with coupon-level testing conducted in accordance with standardized procedures such as ASTM D3039 for tensile testing and ASTM D6641 for compression testing. These tests provide fundamental material properties required for design and analysis.

### Stress–Strain Behavior

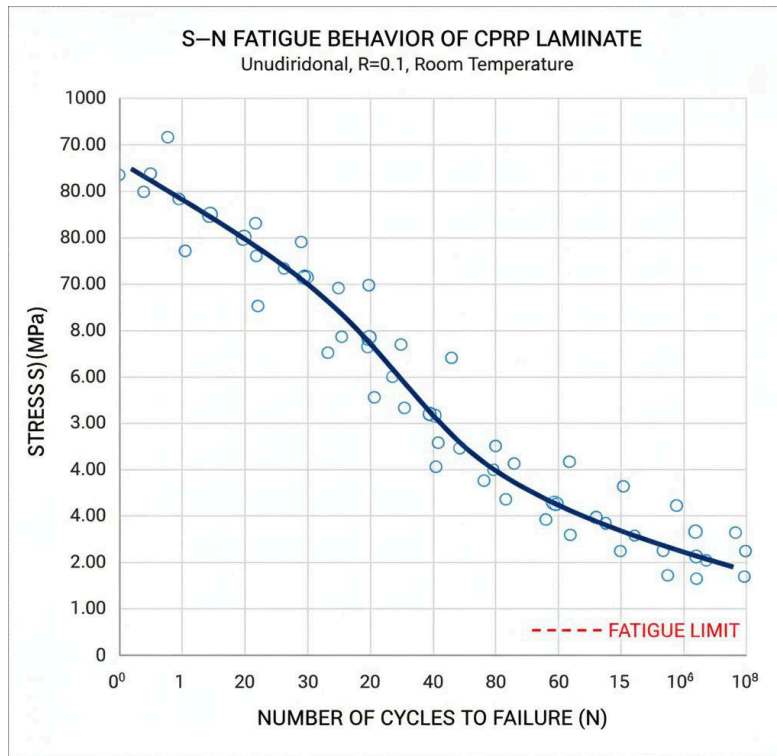
Composite materials typically exhibit linear elastic behavior up to failure when loaded along the fiber direction. Matrix-dominated loading, however, may show nonlinear behavior prior to failure. Fig.3



**Figure 3 - Typical tensile stress–strain curve of a carbon/epoxy composite laminate.**

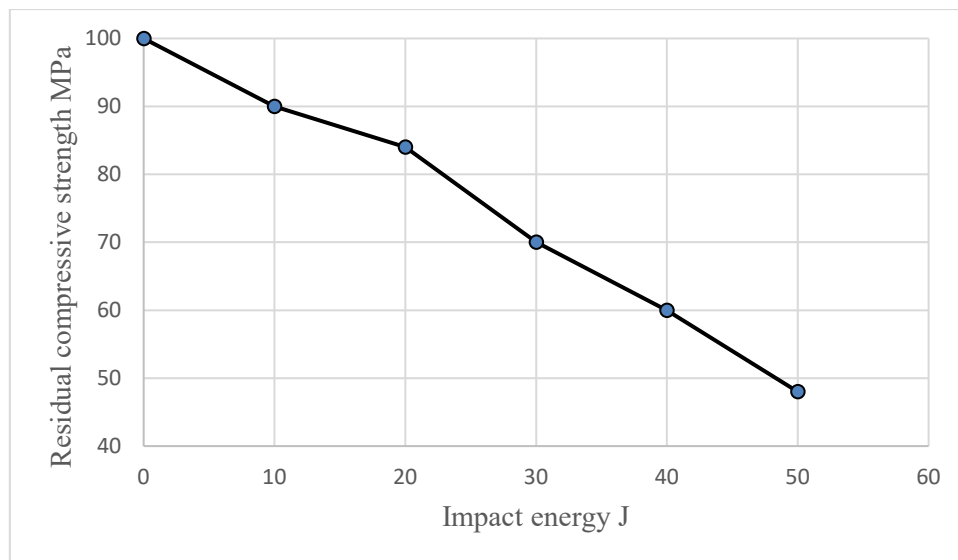
### Fatigue and Damage Tolerance

Aircraft structures are subjected to millions of loading cycles during service. Fatigue testing is therefore essential for determining long-term strength and durability. Impact damage tolerance is evaluated using compression-after-impact (CAI) tests, which assess residual strength after low-velocity impacts. Fig.4.



**Figure 4 - S–N fatigue curve for a CFRP laminate.**

Impact damage tolerance is evaluated using compression-after-impact (CAI) tests, which assess residual strength after low-velocity impacts. Fig.5.



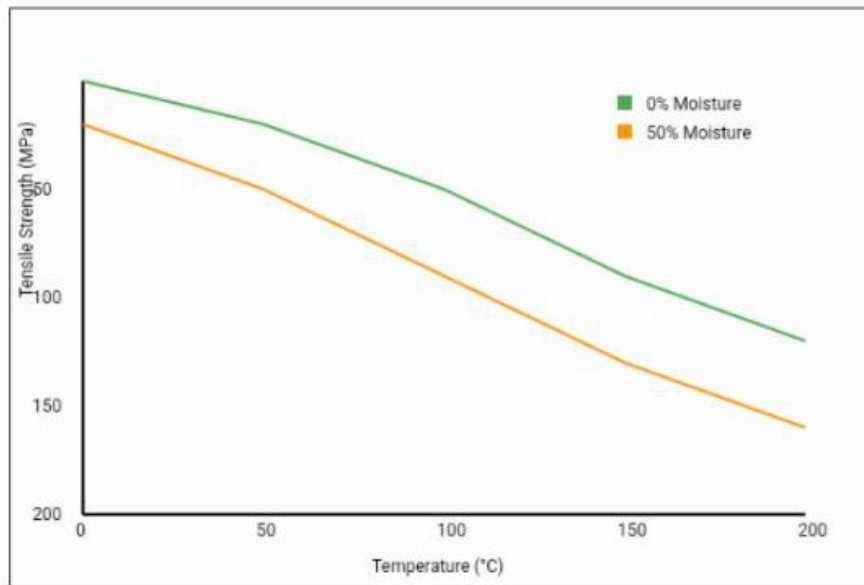
**Figure 5 - Residual compressive strength as a function of impact energy.**

### ENVIRONMENTAL EFFECTS

Composite materials used in aircraft are exposed to varying temperature, humidity, and ultraviolet radiation. Moisture absorption and elevated temperatures can



degrade matrix properties and reduce strength. Hygrothermal aging tests are performed to quantify these effects. Fig.6.



**Figure 6 - Effect of temperature and moisture on composite tensile strength.**

## ANALYTICAL STRENGTH PREDICTION

### Classical Laminate Theory

The stress–strain relationship for an orthotropic lamina in its principal material coordinates is given by:

$$\sigma_1 = Q_{11}\epsilon_1 + Q_{12}\epsilon_2$$

$$\sigma_2 = Q_{12}\epsilon_1 + Q_{22}\epsilon_2$$

$$\tau_{12} = Q_{66}\gamma_{12}$$

where  $Q_{11}$ ,  $Q_{22}$ ,  $Q_{12}$ , and  $Q_{66}$  are the reduced stiffness coefficients derived from the elastic properties of the lamina. Fig.7.

## FAILURE CRITERIA

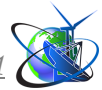
### Maximum Stress Criterion

Failure is predicted when any stress component reaches its corresponding strength limit:

$$|\sigma_1| / X \geq 1$$

$$|\sigma_2| / Y \geq 1$$

$$|\tau_{12}| / S \geq 1$$



### Tsai–Hill Criterion

The Tsai–Hill criterion is expressed as:

$$(\sigma_1 / X)^2 + (\sigma_2 / Y)^2 - (\sigma_1\sigma_2 / X^2) + (\tau_{12} / S)^2 = 1$$

### Tsai–Wu Criterion

The Tsai–Wu criterion accounts for interaction between stress components:

$$F_1\sigma_1 + F_2\sigma_2 + F_{11}\sigma_1^2 + F_{22}\sigma_2^2 + 2F_{12}\sigma_1\sigma_2 + F_{66}\tau_{12}^2 = 1$$

### Hashin Failure Criterion

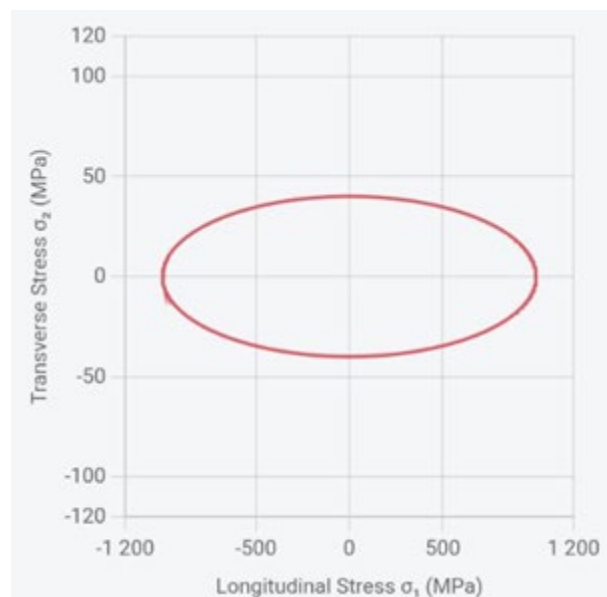
The Hashin criterion distinguishes between fiber and matrix failure modes.

Fiber tension:

$$(\sigma_1 / X_t)^2 + (\tau_{12} / S)^2 = 1$$

Matrix compression:

$$(\sigma_2 / Y_c)^2 + (\tau_{12} / S)^2 = 1$$



**Figure 7 - Representative failure envelope for a composite lamina.**

## NUMERICAL SIMULATION AND FINITE ELEMENT ANALYSIS

Finite Element Analysis (FEA) is widely used to predict composite strength at the structural level. Progressive damage models incorporate failure initiation criteria and damage evolution laws to simulate stiffness degradation and ultimate failure. Analytical-numerical and numerical methods are used to calculate the strength of fiber composite structures.



The finite element method [15] is one of the most popular approaches for numerical modeling of composite materials. It allows dividing a complex geometry into finite elements, each of which is calculated separately, taking into account the material properties and the effects of loads. This method is suitable for calculating large structures, such as aircraft parts. Software systems based on it, such as ANSYS [16], PATRAN (NASTRAN) [35], SOLIDWORKS [36], and others, are widely used in calculating the stress state of complex models in the aerospace industry.

The boundary element method [17] allows solving complex problems taking into account the boundary conditions and the distribution of stresses and strains in the composite. This method is often used to analyze local zones (e.g., at the fiber-matrix interface) and predict local failures.

An analytical and numerical method for studying the stress-strain state of fiber composites is the generalized Fourier method [18]. This method has been used to solve problems for a reinforced layer [19], for a layer with one [20] or two [21] cylindrical tubes, and for a layer with cylindrical inhomogeneity subjected to periodic loading [22-24].

### **Damage Evolution**

Material stiffness degradation is modeled as:

$$E = (1 - d)E_0$$

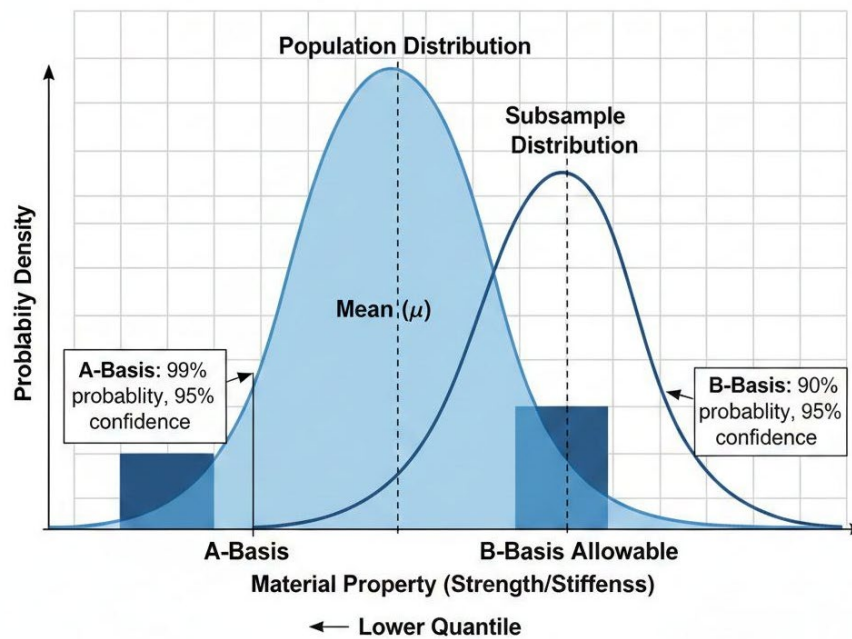
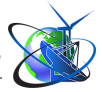
where  $d$  is the damage variable ranging from 0 (undamaged) to 1 (fully damaged).

### **Delamination Modeling**

Delamination is modeled using cohesive zone elements based on traction-separation laws. Failure occurs when the critical energy release rate is exceeded.

### **CERTIFICATION AND DESIGN ALLOWABLES**

Aircraft certification follows a building-block approach, progressing from material coupons to full-scale structural tests. Design allowables such as A-basis and B-basis values are statistically derived from experimental data to ensure conservative and reliable designs. Fig.8.



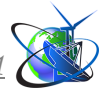
**Figure 8 - Statistical distribution used for determination of composite design allowables.**

### Summary

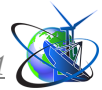
The determination of strength of composite materials used in aircraft structures requires a comprehensive approach combining experimental testing, analytical modeling, and numerical simulation. Due to anisotropy and multiple failure mechanisms, composites demand more sophisticated strength prediction methods than metallic materials. The methodologies presented in this work provide a reliable framework for composite strength assessment and aircraft structural design.

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