ANALYTICAL AND EXPERIMENTAL STUDIES OF DAMAGE IN COMPOSITE PLATES CONTAINING INITIAL OPENINGS

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ABSTRACT

This paper uses the method combining analytical calculation and tensile test to study the hole-edge stress and damage of the composite plate which contains initial openings. Using the conformal mapping method of complex-variable function, the principal stress and first damage load coefficient of each layer which satisfies the Tsai-Hill failure criterion are provided. Based on the ASTM D3039-76, the in-plane tensile properties of the carbon fiber reinforced polymer composites with high modulus are determined. The research shows that when laminates are subjected to the longitudinal direction tensile load, the 0°and ±45°plies is the main part which carries the load. And the maximum principal stress occurs in the range of 65°-115°and 245°-295°polar angles around hole. The mainly damage forms are matrix cracking and fiber breakage, and a few of delamination also happens at the same time. Matrix cracking damage occurs in the direction perpendicular to the tensile load and extends to the edges of specimens in the direction between 0°and 45°polar angle. Fiber breakage appears along the direction of the ±45°plies.

1. INTRODUCTION

In the study of damage tolerance of aircraft composite laminate structures, there are lots of damages which can affect the residual strength of the laminate, but researches often focus on two, including opening (hole, crack, etc.) or impact damage. The reason is that the opening strength and the residual strength of the impact damage are the key factors which can determine the design allowables for composite structures [1-3].

For the damage in equipment components, it can use through-holes or cracks with different shapes to simulate. And when the damage analysis is proceeding, further simplification, such as using through-holes, elliptical holes and regular shape crack to simulate damage, is used to model the corresponding model and compute the stress, deformation and bucking. These can be attributed to the related issues about composite perforated structures and the bearing ability of hole-edge. Machining

openings in composite laminates is inadvisable, mainly because cutting fiber will result in lower mechanical properties and low inter-laminar strength brings difficulties in restoring the strength of the opening areas. But it is inevitably in the actual structure, so it is necessary to study the damage failure laws of the composite structures with openings.

When it comes to the prediction of the mechanical properties of the laminates with openings, the research methods can be divided into hole-edge stress method, finite difference method, boundary element method and progressive damage analysis method [4-8]. Among them, the hole-edge stress method can establish the relationship between mechanical parameters and damage, and well predict the strength of the composite structure with openings. Currently, more scholars have used finite difference method, boundary element method and progressive damage analysis method to study the damage mechanism. They add damage judgement equations in numerical analysis and consider compressed failure modes [9-10]. At the same time, test method [11-12], as a good way to study and verify, is widely used in the strength and damage analyses of the composite specimens with initial damage.

In this paper, the hole-edge stress method based on complex-variable function is used to analyze the hole-edge stress field of the carbon fiber laminate containing various ply angles. And the Tsai-Hill failure criterion is used as the strength judgement formula. Then, based on ASTM D3039-76, a series of tensile tests is conducted to verify the simulation result. Combing the analytical calculations and tests, the tensile mechanical properties and the damage characteristics of the carbon fiber reinforced composites containing openings are studies.

2. ANALYTICAL FORMULATION

2.1 Curve mapping function of hole boundary

Corresponding to the equations describing the different boundaries of the initial openings, the form of the complex variables used by the hole-edge mapping function is:

$$z_j = \omega_j(\zeta) = \frac{R}{2}\left[a_j\left(\frac{1}{\zeta} + \sum_{k=1}^{N} m_k \zeta^k\right) + b_j\left(\zeta + \sum_{k=1}^{N} \frac{m_k}{\zeta^k}\right)\right] \tag{1}$$

Wherein $a_j = (1 + \mathrm{i}\mu_j)$, $b_j = (1 - \mathrm{i}\mu_j)$ and (j=1, 2); m_k is a coefficient determined by different hole shapes; z_j is the complex variables of the stress function in anisotropic problems and $z_j = x + \mu_j v$; μ_j is the root of the characteristic equation; ζ is the complex variables mapped to the unit circle; R is the conversion constant determined by the hole shape.

2.2 Stress of laminate

The stress functions of the composite laminate containing an opening can be written as:

$$\phi_1(z_j) = \phi^0(z_j) + \phi^*(z_j) \tag{2a}$$

$$\psi_1(z_j) = \psi^0(z_j) + \psi^*(z_j) \tag{2b}$$

Wherein j=1, 2; $\varphi_1(z_j)$, $\psi_1(z_j)$ are two analytic complex-variable functions which are used to simplify the calculation; φ^0, ψ^0 are corresponding to the stress functions describing the composite

structures without holes and φ^*, ψ^* are the stress functions describing the composite structures with holes.

When the external force is given, the internal forces (N_x, N_y and N_{xy}) of the plate containing holes can be got by the following formulas:

$$N_x = 2\,\mathrm{Re}\left[\mu_1^2\phi'(z_1) + \mu_2^2\psi'(z_2)\right] \tag{3a}$$

$$N_y = 2\,\mathrm{Re}\left[\phi'(z_1) + \psi'(z_2)\right] \tag{3b}$$

$$N_{xy} = -2\,\mathrm{Re}\left[\mu_1\phi'(z_1) + \mu_2\psi'(z_2)\right] \tag{3c}$$

Formula (3) shows that if the boundary functions ($\varphi_1(z_j)$, $\psi_1(z_j)$) which meet the function (1) can be obtained, the stress/internal force distribution of the anisotropic plate containing holes also can be calculated.

2.3 Hole-edge stress of each layer

Average strain is derived by the average stress as below:

$$\bar{\varepsilon} = A^{-1}\bar{\sigma} \tag{4}$$

Furthermore, each layer stress can be expressed as:

$$\sigma^{(k)} = \left[\bar{Q}\right]^{(k)}\bar{\varepsilon} \tag{5}$$

2.4 Failure criterion

Using Tsai-Hill strength theory, strength of each layer is checked. The strength theory formula is as follow:

$$\frac{\sigma_1^2}{X_t X_c} - \frac{\sigma_1\sigma_2}{X_t X_c} + \frac{\sigma_2^2}{Y_t Y_c} + \frac{X_c - X_t}{X_t X_c}\sigma_1 + \frac{Y_c - Y_t}{Y_t Y_c}\sigma_2 + \frac{\tau_{12}^2}{S^2} = 1 \tag{6}$$

Where, X_t is the longitudinal tensile strength (along the first main direction of materials), X_c is the longitudinal compressive strength (along the first main direction of materials), Y_t is the transverse tensile strength (along the second main direction of materials), Y_c is the compressive tensile strength (along the second main direction of materials), S is shear strength (along 1-2 plane). σ_1 is the stress along the main direction of each ply and σ_2 is the stress which is perpendicular to the main direction of fibers; τ_{12} is the shear stress (along 1-2 plane).

Substituting the stress (along the main direction of fibers) in each ply to the failure criterion (6), each ply's failure load can be obtained and the destruction can also be predicted.

3. TENSILE TEST

3.1 Test method

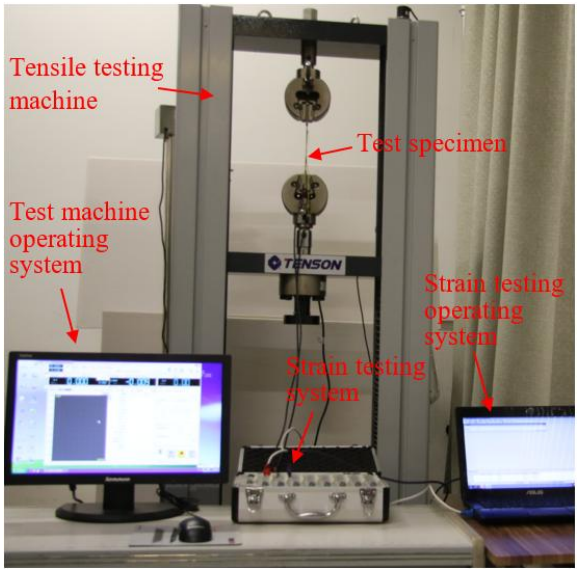


Figure 1. Tensile test system

The tensile strength test and damage test of the composite specimens in this paper are conducted on Tenson micro-controlled universal testing machine. Based on ASTM D3039-76, the loading rate is 1-3 mm/min when the in-plane tensile properties and the stress-strain curve of the carbon fiber reinforced polymer composites with high modulus are determined and when the tensile strength is determined, the loading rate is 1-6 mm/min. The end condition of the test is that the fracture percentage reaches 5%, and after the stopping, it does not return to the initial state. The strain test system is set to timing acquisition and the time interval is 1s. The initial load applied for the specimen is about 5% of the damage load and then it continuously increases until the specimen damages. During the test, the maximum load and damage forms are recorded, and the load-deformation is drawn.

3.2 Test specimens.

The test specimens are carbon fiber reinforced resin composites produced by Guangwei Carbon Fiber Ltd. The fiber type is HS-Carbon and the resin type is 7901. The layer thickness of the laminate is 0.13 mm and the size of the specimens is 230×25×2 mm^3, shown in Figure 2. The ply angle of the laminate is $[0_3/(\pm45)_3/90_3]_s$.

In order to prevent the specimens from occurring clamping damage, aluminum plates (50 × 15 × 2 mm^3), as reinforcing sheets, are bonded on the ends of the specimens, and the adhesive type is MA380. All the specific parameters are referenced from the relevant content in ASTM D3039-76.

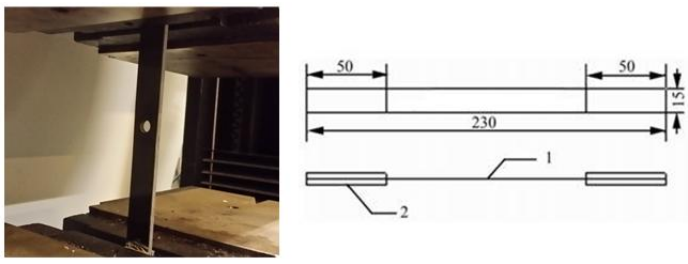
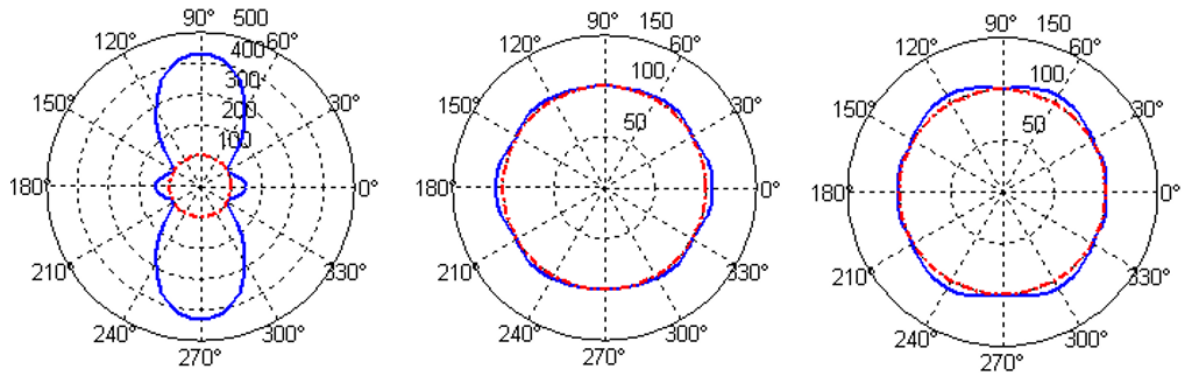


Figure 2. Clamping of the specimens

The reinforcing sheet can be made of cross-ply glass reinforced laminates (2-3 mm in thickness) or aluminum plate (1-3 mm in thickness). The adhesive used should ensure that the reinforcing sheet does not fall off during the test and the curing temperature should be low. When the treatment of the sample surface is processing, no fiber damage is allowed. The test has ten groups and each group has five specimens.

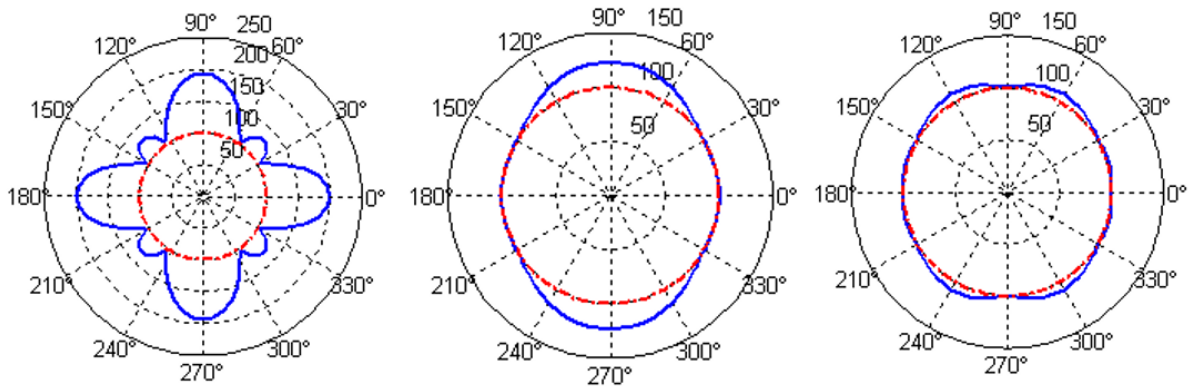
4. ANALYSIS OF RESULTS

4.1 The stress distribution around holes



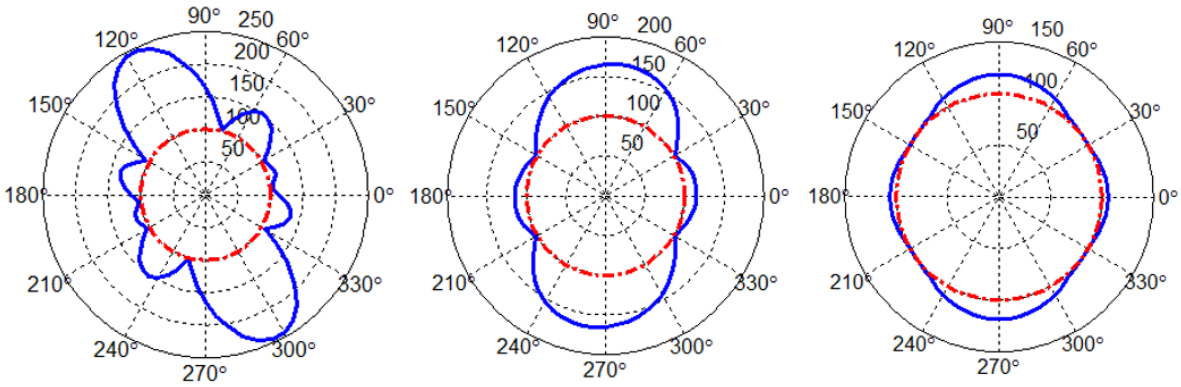
(a) σ_1 (b) σ_2 (c) τ_{12}

Figure 3. Principal stress distribution along θ of 0° layers with round hole



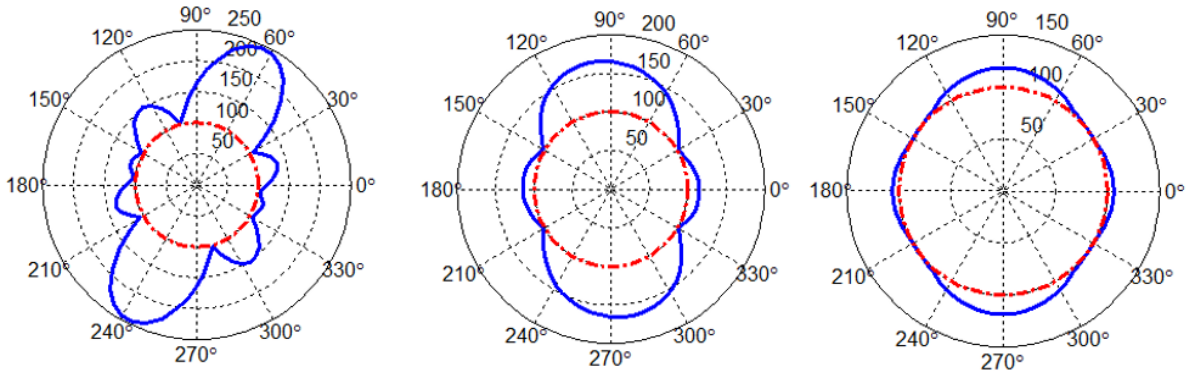
(a) σ_1 (b) σ_2 (c) τ_{12}

Figure 4. Principal stress distribution along θ of 90° layers with round hole



(a) σ_1 (b) σ_2 (c) τ_{12}

Figure 5. Principal stress distribution along θ of 45° layers with round hole



(a) σ_1 (b) σ_2 (c) τ_{12}

Figure 6. Principal stress distribution along θ of -45 °layers with round hole

According to the parameters of the laminate, the stress distributions of σ_1, σ_2 and τ_{12} around holes are obtained and shown in Figure 3-6.

When the polar angle θ of the tensile load (p) is 0 º, in the x-axis direction, σ_1 is the maximum of all layers in the laminate which is made of single-layers with same material properties and different ply angles. The principle stress (σ_1) distribution curves (Fig. 3(a), 4(a), 5(a) and 6(a)) show that the maximum stress value is in the direction of 90 ºpolar angle of the 0 ºlayer and it is followed by the 45 º and -45 º layer, respectively, in the direction of 115 º, 295 º, 65 º and 245 º polar angles. In the σ_2 distribution curves (Fig. 3(b), 4(b), 5(b) and 6(b)), the values in the 45 º and -45 º layer is the largest and they are respectively in direction of the 85 ºand 295 º, 65 ºand 245 ºpolar angles. The values of τ_{12} are all small and less than 120MPa. Thus, when the laminate is subjected the load (p) in the x-axis direction, the 0 º, 45 º and -45 º layers are the main part to bear the load and the maximum value of σ_1 appears in the range of 65 º-115 ºand 245 º-295 ºpolar angle around the hole.

4.2 The initial failure load around holes

Taking p/t as the undetermined variable, the hole-edge stresses, σ_1, σ_2 and τ_{12} are substituted into the formula (7). Then, p/t can be calculated and namely, it is the first damage load coefficient of the

ply. Comparing the coefficient of each ply, the ply, which has the minimum one, is the first one to appear damage. The first damage load coefficient of each ply is shown in Figure 7.

In Figure 7, the hole-edge area, whose coefficient is smaller, is easier to appear damage. During the range of 0 º180 ºpolar angles, the minimum damage coefficient occurs in the range of 65 ºto 115 ºpolar angle. And the coefficient of the 45 º and -45 º layers is the smallest and they are prone to occurs damage. Then it is followed by the 90 ºlayer and the 0 ºlayer is the maximum. The damage coefficient of each layer is almost on the symmetry of the 90 ºpolar angle direction (the y-axis direction) and only the ones of the 45 ºand -45 ºlayer have a slightly deviation.

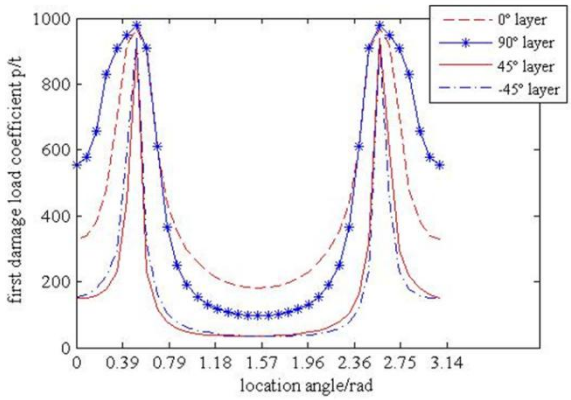


Figure 7. First damage load coefficient comparison of different laminate plate

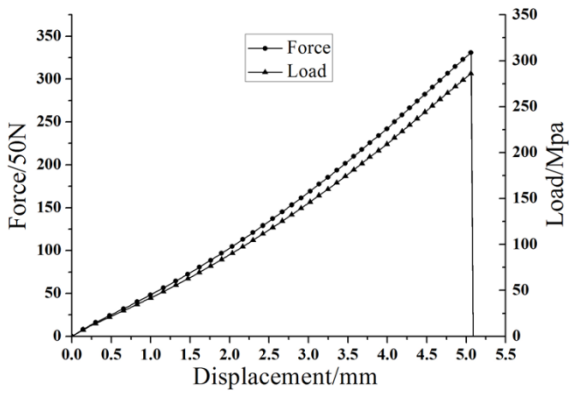


Figure 8. Tensile load vs. displacement curve

In the tensile test, the *F-δ* curve can reflect the changes of material stiffness. From Figure 8, it shows that the curve keeps straight up in the initial phase. When the displacement reaches around 0.78 mm, the curve slopes change. It shows that the internal damage at this point occurs and the mechanical properties of the specimens have changed. The test result is correspondence with analytical result of initial damage.

4.3 Damage form

The failure forms and the X-ray picture of the specimens made of carbon fiber reinforced resin composites are shown in Figure 9. The forms are mainly matrix cracking and fiber breakage, and

delamination also occurs with the damage happens. The matrix cracking occurs along the x-axis direction and gradually extend to the edge of the plate width. The fiber breakage appears along the direction of the ±45° plies. In the X-ray picture, the matrix cracking starts from the x-axis direction around the opening. And the fracture doesn't extend along the longitudinal direction, but in the range of 0°-45°polar angle. It shows great agreement with the theoretical calculations.

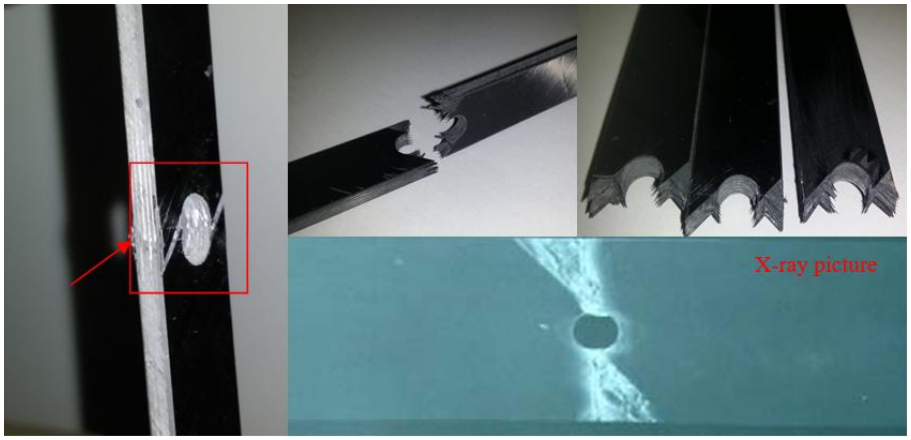


Figure 9. The failure picture and the X-ray picture

5. CONCLUSIONS

In this paper, the complex-variable function is used to analyze the hole-edge stress field of the carbon fiber laminate containing various ply angles. And the Tsai-Hill failure criterion is used as the strength judgement formula. Then, based on ASTM D3039-76, a series of tensile tests is conducted to verify the simulation result. Combing the analytical calculations and tests, the tensile mechanical properties and the damage characteristics of the carbon fiber reinforced composites containing openings are studies.

Using analytical calculation and experimental testing, it found that when laminates are subjected to the longitudinal direction tensile load, the 0° and ±45° plies is the main part which carries the load and the maximum principal stress occurs in the range of 65°-115°and 245°-295°polar angles around holes. The mainly damage forms are matrix cracking and fiber breakage, and a few of delamination also happens at the same time. Matrix cracking damage occurs in the direction perpendicular to the tensile load and extends to the edges of specimens in the direction between 0° and 45° polar angle. Fiber breakage appears along the direction of the ±45° plies.

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