

Compression after impact behavior of asymmetrically tapered laminates: Experimental and numerical studies

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Abstract: This paper presents experimental and numerical studies on the compression after impact (CAI) behavior of composite tapered laminates. It introduces newly designed impact platforms and compression fixtures specifically tailored for the specimens. Drop-weight impacts are applied to the center of the specimens, and the resulting damage is briefly described. Compression tests are then conducted on both non-impacted and impacted specimens, with strain gauges used to monitor the strain distribution. Internal damage is detected using CT scanning and ultrasonic C-scan techniques. The numerical simulations are performed using ABAQUS/Explicit finite element analysis (FEA), incorporating an intra-laminar progressive damage model and an inter-laminar cohesive model, while additionally modeling resin pockets as elastomers. The simulation and experimental results indicate that before compression failure, impact damage in the thin section minimally affects the out-of-plane displacement, which is predominantly influenced by structural asymmetry. Stress concentration is observed at the junction between the thin and tapered sections in the compression test, while in the CAI test, stress concentration appears in the impact zone. The impact induces a notable shift in failure location and damage modes, resulting in decreased compressive strength, although the impact on stiffness remains minimal.

Keywords: tapered laminates, impact, compression, compression after impact, finite element modeling

1 Introduction

Carbon Fiber Reinforced Polymer (CFRP) composites are increasingly utilized in diverse engineering industries due to their high stiffness-to-weight ratio, excellent corrosion resistance, and superior fatigue properties [1]. To meet specific shape, weight efficiency, and aerodynamic requirements, composite structures often require thickness variation to create tapered structures. This structural design has been applied in various applications; for example, the composite hub of a helicopter rotor blade gradually thickens near the rotation center to increase stiffness and tapers out in the middle to enhance flexibility. Similarly, the composite fan blades of advanced civil aircraft

turbofan engines need thickness variations to reduce dead weight and meet performance requirements. Additionally, in ocean engineering, composite turbine blades gradually thin along the span to meet fluid dynamics requirements [2-4].

The common practice for achieving tapered composite laminates involves terminating plies at specific locations, known as drop-off [5, 6]. At these terminations, triangular resin pockets can form between adjacent plies, creating localized geometric and material discontinuities. Stress concentrations are known to develop at these drop-off terminations, significantly contributing to delamination initiation within resin pockets [5,7-9]. Moreover, in asymmetric tapered laminates, the discontinuity of the neutral axis results in a more complex stress distribution and displacement field [10]. While existing literature extensively examines delamination caused by drop-offs under tensile and cyclic loads and explores parameterized research for the rational or optimized design of tapered composite structures to resist damage [3, 5, 11-17], limited research has been conducted on the compression and CAI behavior of tapered composite laminates.

The behavior of asymmetrically tapered composites under compressive loading is intricate. The geometric taper introduces complexity in load distribution within the laminate. Adams and Finley [18, 19] conducted shear and compression tests on tapered laminates, emphasizing the significance of taper geometry in compression. Steeves et al. [20] performed axial compression tests on symmetrical tapered composite laminates, observing that micro-buckling is more pronounced at smaller tapers, while delamination dominates at larger tapers. Stress concentration at the termination of dropped plies has been identified as a factor in compressive failure. Wilson et al. [21, 22] used a fracture mechanics-based method to predict interlaminar stresses would cause delamination, noting that delamination onset is related to the termination ply thickness. Ganesan et al. [23] developed a nonlinear FEA, combining the maximum stress criterion with the progressive failure method, to analyze the effects of parameters such as taper, ply configuration, and fiber orientation on tapered laminates under compression. Gordon et al. [24] used the ply scarfing method to terminate plies and observed compression failure using high-speed cameras. They found that compared to laminates terminated using the conventional ply-drop method, the ply scarfing method suppressed delamination under compression. These studies collectively suggest that the failure of tapered composites under compression results from a combination of factors. Researching and predicting the compressive strength and failure mechanisms of asymmetric tapered laminates remains a challenging task.

Composite structures in aircraft are susceptible to impacts from foreign objects during production or in service, such as tools falling, debris on runways, and hail strikes. These impacts, even at low energy levels can leave barely visible indents on the surface of laminates but cause permanent damage inside, including fiber buckling, matrix cracking, and delamination. Such damage significantly reduces the strength of laminates, especially their compressive strength [25-26]. In aeronautics, this loss of strength due to impacts forces designers to consider damage tolerance and explore potential areas for material improvements. Therefore, understanding the mechanisms that lead to premature failure of the structure is crucial.

Numerous studies have investigated the CAI behaviors of composite flat laminates. The decrease in residual strength of laminates is primarily attributed to two factors. Firstly, delamination occurs during the impact process, resulting in the division of the laminates into multiple sub-laminates. These sub-laminates subsequently buckle and ultimately experience failure when subjected to compressive loads [27-30]. Secondly, cracks near the impact point cause local buckling and stress concentration, with fiber damage potentially propagating from the impact area, leading to compression failure of the laminate [31-33]. Podolak et al. [34] conducted a progressive analysis on thin-walled CFRP plates using a digital image correlation (DIC) system and visual inspection to explore the buckling behavior and damage propagation mechanisms during the CAI process. Tuo et al. [35] employed both numerical and experimental approaches to study the damage and failure mechanisms of composite laminates under low velocity impact (LVI) and CAI, establishing a relationship between CAI mechanical behaviors and multiple impacts. Taheri-Behrooz et al. [36] utilized a 3D failure model, combining Puck's nonlinear shear behavior model and a matrix damage model, to investigate the LVI and CAI behaviors of composites with quasi-isotropic stacking sequences. Shabani et al. [37] developed a high-fidelity FEA to predict the damage evolution and CAI strength of CFRP plates, comparing the CAI behaviors with and without support fixtures and emphasizing the influence of overall buckling behavior on residual strength.

Regarding the study of CAI on asymmetric tapered laminates, Abdulhamid et al. [38, 39] conducted experimental investigation on the impact and CAI behaviors of asymmetric tapered laminates. Their impact experiments revealed that material discontinuities arising from resin pockets had a lesser impact on damage mechanisms compared to the geometric structure. In the CAI experiments, they noted that compressive behavior was primarily influenced by the interaction between compression and bending, rather than the impact itself. Abdulhamid et al. [40] used a discrete ply model (DPM) to simulate impact and CAI of tapered laminates, incorporating material laws that account for matrix cracking, delamination, and fiber breakage. While the simulation results for impact closely matched the experimental results, the model underestimated the residual strength. In conclusion, there is limited research on the compression of asymmetrically tapered laminates containing impact damage.

This paper conducts experimental and numerical investigations into the impact, compression, and CAI behaviors of asymmetric tapered laminates. Due to the lack of compression and impact test standards for tapered specimens, custom compression fixtures and the impact platform are manufactured to establish reasonable boundary conditions. Strain gauges are used to monitor stress levels during compression, while ultrasonic C-scans and CT scans capture the internal damage morphology. High-fidelity modeling using Abaqus/Explicit FEA is developed. To address intralaminar damage, this model integrates the improved Hashin criterion and an energy-based progressive damage model. For interlaminar damage, it employs the cohesive zone model. The numerical results are compared with the experimental results to analyze the impact, compression, and CAI performance. The stress distribution in the tapered section of the laminate under compression and CAI tests is analyzed. In addition,

emphasis is placed on comparing and analyzing out-of-plane displacements, damage mechanisms, failure modes, and the compressive strength of compression and CAI tests.

2. Experimental

2.1 Specimens and tests boundary conditions

To investigate the compression and CAI behaviors in asymmetric tapered composite laminates, eight beam-type specimens with identical configurations are manufactured. The geometry and dimensions of the specimen are displayed in Fig. 1(a). The thin section is located at the mid-length, while the two thick sections are at both ends of the specimen. The thickness gradually increases from the thin section to the thick sections, forming tapered sections with an angle of approximately 5.5° . The ply angle table is shown in Fig. 1(b). Each specimen consists of 30 and 56 unidirectional carbon fiber epoxy plies in the thin and thick sections, respectively, with 26 plies dropped off in the tapered section. In the table, 'white' represents the continuous plies and 'green' represents the dropped plies. Fig. 1(c) illustrates the sequence of dropped plies and the location of the resin pockets in the upper half of the tapered section. The maximum number of continuous dropped plies is three and there are a total of 26 resin pockets in the tapered section.

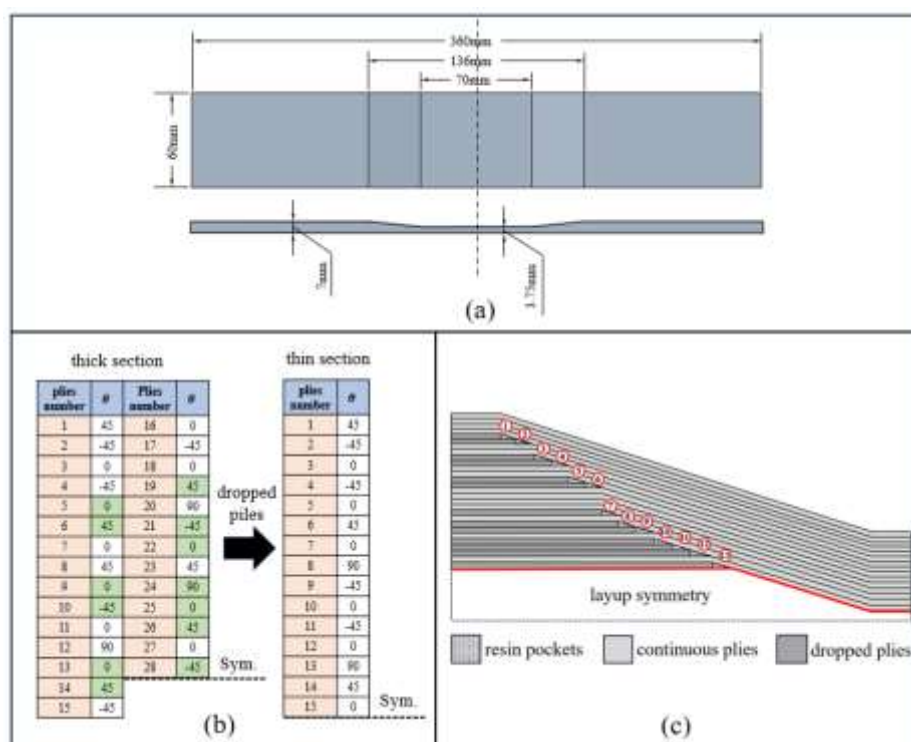


Fig. 1 Geometric dimensions and configuration of specimen: (a) specimen dimensions, (b) stacking sequence of the thin and thick sections, (c) schematic diagram of ply drop-off order and location.

The specimens are made from CCF300/QY9511, with each ply of prepreg having a nominal thickness of 0.125 mm. Prepregs of different lengths are hand-laid onto a flat tool plate, vacuum bagged and then autoclave-cured according to the manufacturer's recommendations (2 hours at 180°C , 7 bar pressure). After curing, the surfaces of the panels are blasted and degreased. The laminate panels are then carefully cut into individual specimens using a

CNC machine to dimensions of 60 mm in width and 360 mm in length. A microscope is used to inspect the specimens for any manufacturing defects, particularly in the resin pockets, to ensure the reliability of the experimental results.

Beam-type structures can experience premature failure from global buckling under axial compressive loads, especially in asymmetrical tapered laminates. The discontinuity of the neutral axis within the tapered region can induce compression-bending coupling under compressive loads. Therefore, it is crucial to fabricate an anti-buckling fixture for the asymmetrically tapered laminate following ASTM D6484 standards [41]. Fig.2 (a) illustrates the components of the fixture. Short grips A and B are identical, while long grip B, positioned on the tapered side, gradually increases in thickness to provide adequate lateral support. In contrast, long grip A is located on the smooth side and does not require special modifications. Two support plates prevent excessive deformation of the long grips post-specimen failure. The gray area denotes the clamping point on the MTS fatigue testing machine. Compressive load application to the specimen is facilitated through friction. Windows are incorporated in the middle of the long grip to accommodate strain gauges and wires.

Since the tapered laminates are not the standard-sized specimens, the drop-weight impact platform needs to be redesigned. Fig.2 (b) presents a schematic diagram of the platform, which is situated beneath the specimen, connects to the testing machine table and supports the specimen. It features a central opening measuring 50 mm×136 mm, allowing appropriate deformation of the specimen upon impact. Two clamp plates positioned above the specimen can secure it to the platform, preventing bounce during impact loading.

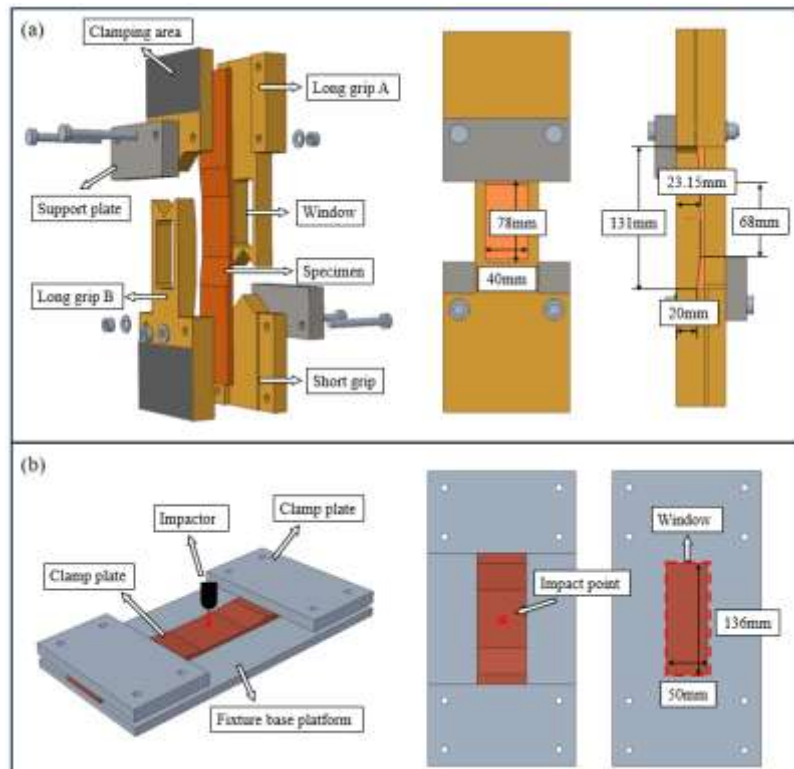


Fig. 2 Fixture of the compression and CAI tests, and drop weight impact platform: (a) the compositions and basic dimensions of the fixtures, (b) the compositions and basic dimensions of the platform.

2.2 Tests procedure

To monitor the stress levels and distribution of the specimens during compression and CAI tests, strain gauges are attached to the surface of the specimens. As shown in Fig.3, a total of 12 uniaxial strain gauges are placed on the thin section of the specimen. Gauges G1 to G6 are placed on the tapered side, while gauges G7 to G12 are placed on the smooth side. Back-to-back strain gauges, such as G1 and G7, G4 and G10, G2 and G8, are used to detect significant buckling of the laminate during the pre-testing process. Strain gauges on both sides, such as G1 and G4, G2 and G5, G3 and G6, are used to monitor whether the laminate is evenly loaded during the tests, allowing for adjustments to the size and assembly position of the clamp.

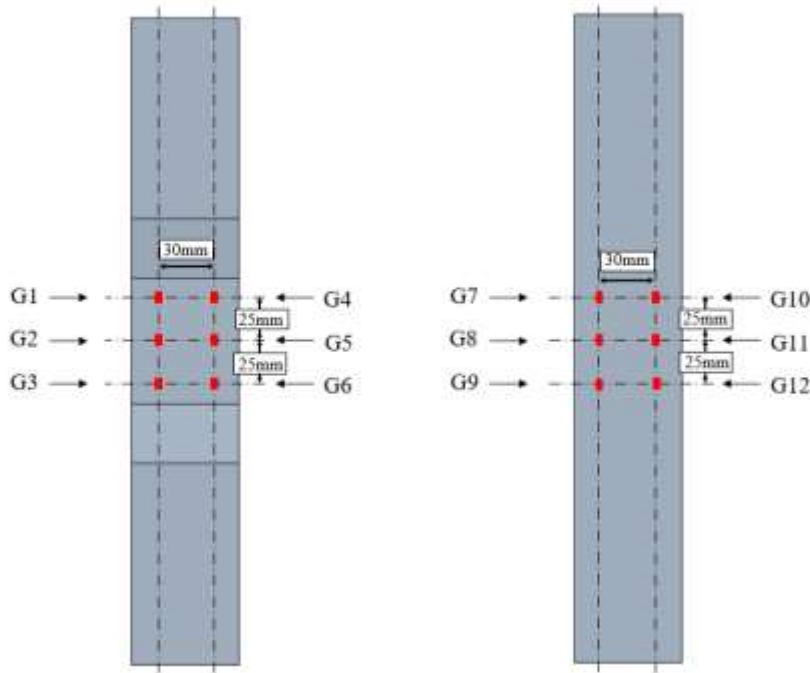


Fig. 3 Positions of the strain gauges on the specimen.

The compression and CAI tests are conducted using the MTS 370.50 fatigue testing machine, shown as in Fig.4 (a), it has a maximum load of ± 500 kN, a static load error of $\pm 0.5\%$ FS and a dynamic load error of 1% FS. The TestStar II fatigue control system is used to control the displacement/force of the lower grip of the testing machine. The strain data during the test is collected using the DH3816 static strain testing system. Before starting the test, 12 strain gauges are attached to the surface of the specimen and the wires from the strain gauges are connected to the strain indicator. The specimen is then assembled in the fixture and secured in place using bolts and nuts. The fixture with the assembled specimen is clamped onto the fatigue testing machine. The computer controls the lower grip of the testing machine to apply an axial compression with a displacement of 3 mm at a loading rate of -0.5 mm/min. To prevent slippage between the specimen and the fixture, the pressure of the testing machine grips is adjusted to 3000 Psi. During the test, the testing machine records the load-displacement curves and the strain indicator measures strain every 5 seconds until the specimen is completely destroyed.

The INSTRON CEAST9350 drop weight impact testing machine is used for low-velocity impact testing, as

shown in Fig. 4(b), it has a steel hemispherical impactor with a diameter of 12.7 mm and a mass of 5.277 kg. For this study, the impact energy is set to 8 J, and the velocity of the impactor is calculated as 1.74 m/s. The center of the specimen is subjected to a low-speed impact following the ASTM D7136 standard [42]. After the impact test, the specimens are allowed to stand for 48 hours. Subsequently, the UltraPAC ultrasonic C-scanning equipment is used to detect internal delamination.

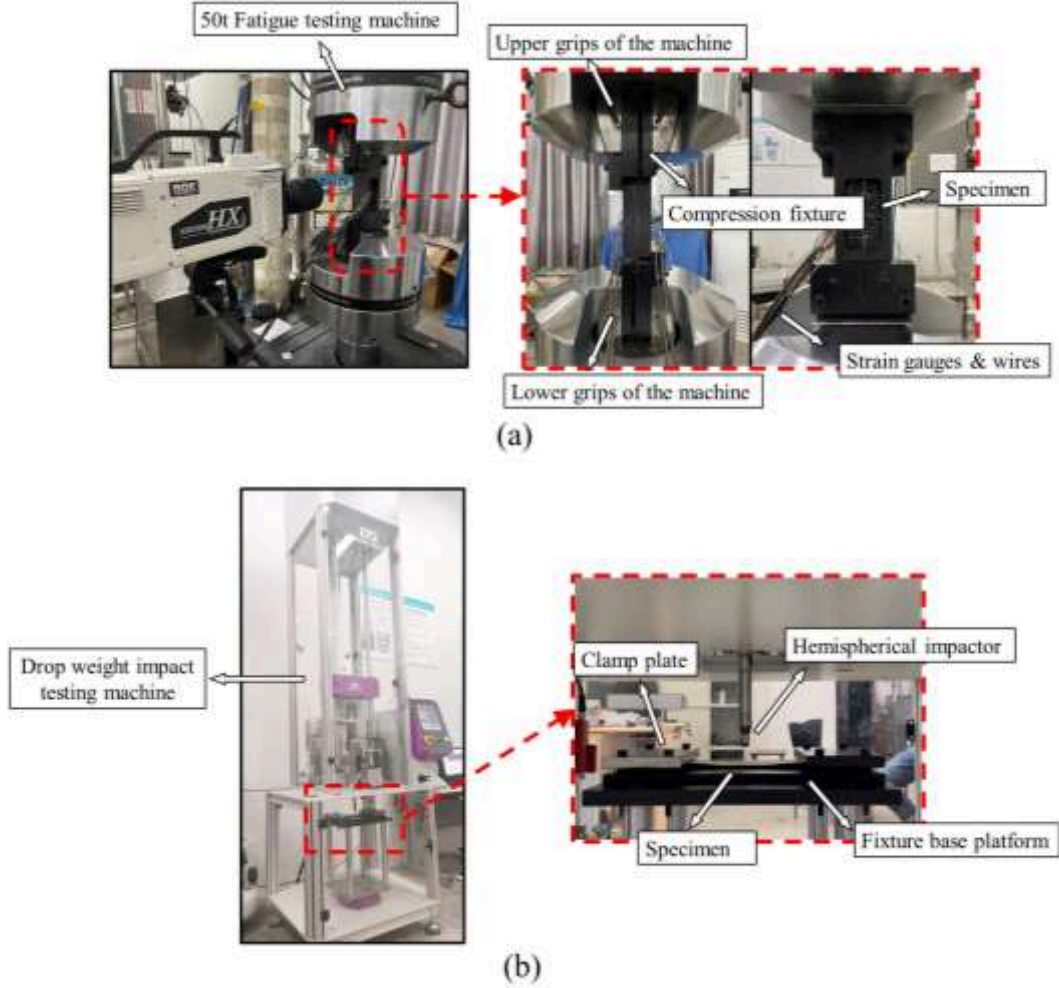


Fig. 4 Test setup: (a) compression and CAI tests, (b) Drop weight impact test.

3. Composite damage model

3.1 Intra-laminar damage

3.1.1 Stress-strain relationship

Since composite laminates accumulate damage and gradually degrade in performance under impact and compression loads, a continuous damage model has been introduced to describe the progressive failure behavior of composites. The initial constitutive relation is as follows:

$$\sigma_{ij} = C_{ijkl} \epsilon_{ij} \quad (1)$$

where σ_{ij} , C_{ijkl} , and ε_{kl} ($i, j, k, l = 1, 2, 3$) represent the engineering stress, stiffness matrix, and engineering strain components, respectively. The subscripts 1, 2, and 3 represent the fiber direction, in-plane transverse direction and out-of-plane direction. A single layer of composite is orthotropic. When the in-plane stress/strain of the layer reaches the damage criterion, the stiffness is reduced by adding damage variables to each component of the stiffness matrix. The symmetric damage stiffness matrix C_d is expressed as: Eqs. (2) and (3).

$$[C_d] = \frac{1}{\Delta} \begin{bmatrix} d_f E_{11}(1-d_m v_{23} v_{32}) & d_f d_m E_{11}(v_{21} + v_{23} v_{31}) & d_f E_{11}(v_{31} + d_m v_{21} v_{32}) & & & \\ & d_m E_{22}(1-d_f v_{13} v_{31}) & d_m E_{22}(v_{32} + d_f v_{12} v_{31}) & & & \\ & & E_{33}(1-d_f d_m v_{12} v_{21}) & & & \\ & & & \Delta d_f d_m G_{12} & & \\ & & & & \Delta d_f d_m G_{23} & \\ & & & & & \Delta d_f d_m G_{31} \end{bmatrix} \quad (2)$$

$$\begin{cases} d_f = (1-d_{ft})(1-d_{fc}) \\ d_m = (1-S_{mt}d_{mt})(1-S_{mc}d_{mc}) \\ \Delta = 1-d_f d_m v_{12} v_{21} - d_m v_{23} v_{32} - d_f v_{13} v_{31} - 2d_f d_m v_{12} v_{23} v_{31} \end{cases} \quad (3)$$

The variables d_f and d_m represent the damage variables for the fiber and matrix, respectively, determined by the damage variable d_l as indicated in Equation (3). The subscripts ' ft ', ' fc ', ' mt ', and ' mc ' describe the damage of fiber tension, fiber compression, matrix tension and matrix compression, respectively. The coefficients S_{mt} and S_{mc} are introduced to account for shear stiffness failure due to matrix damage. In this study, the value of S_{mt} and S_{mc} are set to 0.96 [43, 44].

3.1.2 Failure initiation criterion

Hashin's failure criterion [45] is one of the most well-known criteria for predicting the in-plane damage initiation in composite laminates. In this study, 3D Hashin's criterion is employed, which includes the normal stress component through the thickness for matrix failure. The four different failure modes are as follows:

Fiber tensile failure ($\sigma_{11} \geq 0$)

$$F_{ft} = \left(\frac{\sigma_{11}}{X_T} \right)^2 + \alpha \left(\frac{\sigma_{12}}{S_{12}} \right)^2 + \alpha \left(\frac{\sigma_{13}}{S_{13}} \right)^2 \geq 1 \quad (4)$$

Fiber compression failure ($\sigma_{11} < 0$)

$$F_{fc} = \left(\frac{\sigma_{11}}{X_C} \right)^2 \geq 1 \quad (5)$$

Matrix tensile failure ($\sigma_{22} + \sigma_{33} \geq 0$)

$$F_{mt} = \left(\frac{\sigma_{22} + \sigma_{33}}{Y_T} \right)^2 + \frac{1}{S_{23}^2} (\sigma_{23}^2 - \sigma_{22} \sigma_{33}) + \left(\frac{\sigma_{12}}{S_{12}} \right)^2 + \left(\frac{\sigma_{13}}{S_{13}} \right)^2 \geq 1 \quad (6)$$

Matrix compression failure ($\sigma_{22} + \sigma_{33} < 0$)

$$F_{mc} = \left(\frac{\sigma_{22} + \sigma_{33}}{2S_{23}} \right)^2 + \frac{\sigma_{22} + \sigma_{33}}{Y_C} \left[\left(\frac{Y_C}{2S_{23}} \right)^2 - 1 \right] + \frac{1}{S_{23}^2} (\sigma_{23}^2 - \sigma_{22}\sigma_{33}) \left(\frac{\sigma_{12}}{S_{12}} \right)^2 + \left(\frac{\sigma_{13}}{S_{13}} \right)^2 \geq 1 \quad (7)$$

In the above equations, σ_{ij} ($i, j=1, 2, 3$) is the equivalent stress tensor, X_T and X_C are the tensile and compressive strengths in the fiber direction; Y_T and Y_C are the tensile and compressive strengths in the transverse direction; S_{12} , S_{13} , and S_{23} are the shear strengths. α is the shear failure coefficient used to determine the contribution of shear stress to fiber tensile failure, in this study, it is set to 1.

3.1.3 Failure evolution

Once the stress/strain of the elements satisfy the failure initiation criteria, the damage evolution model needs to be defined. The linear model can effectively describe the damage process of composite. Before reaching the failure initiation criterion, the stress and strain exhibit a linear relationship, indicating that the element is undamaged. When the stress reaches the damage initiation criterion (peak stress), the corresponding strain is the damage initiation strain, indicating the onset of damage in the element. The strain corresponding to complete damage is the complete failure strain, which is calculated from the dissipated energy. The energy dissipation of the element can be expressed as:

$$G_I = \frac{1}{2} \sigma_{eq}^f \varepsilon_{eq}^f l_c \quad (8)$$

where G_I is the fracture energy density of failure mode I ; ε_{eq}^f and σ_{eq}^f are the equivalent failure strain and equivalent peak stress; l_c is the characteristic length, and it is the cube root of the element volume. This characteristic length maintains a constant energy release rate per unit crack area and makes the final result independent of the finite element mesh size. In this study, the method developed by Bažant and Oh [46] is used to avoid the material damage exhibiting localized characteristics.

The damage evolution law for each failure mode is expressed as:

$$d_I = \frac{\delta_{I,eq}^f (\delta_{I,eq} - \delta_{I,eq}^0)}{\delta_{I,eq}^f (\delta_{I,eq}^f - \delta_{I,eq}^0)} \quad (d_I \in [0, 1], I = ft, fc, mt, mc) \quad (9)$$

where $\delta_{I,eq}^0$ is the equivalent displacement at the final failure of each failure mode; $\delta_{I,eq}^f$ is the equivalent displacement at the initiation failure of each failure mode; they can be computed by following equations:

$$\delta_{I,eq}^f = \frac{2G_I}{\sigma_{eq}^0} \quad (10)$$

$$\delta_{I,eq}^0 = \frac{\delta_{I,eq}}{\sqrt{F_I}} \quad (11)$$

$$\sigma_{I,eq}^0 = \frac{\sigma_{I,eq}}{\sqrt{F_I}} \quad (12)$$

where $\delta_{I,eq}$ and $\sigma_{I,eq}$ are the equivalent displacement and equivalent stress for each failure mode; F_I is the value of the damage initiation criterion.

In this study, the 3D Hashin's matrix failure criterion is determined based on $\sigma_{22} + \sigma_{33}$, therefore considering the normal stress component through the thickness to calculate $\delta_{I,eq}$ and $\sigma_{I,eq}$. The formulas for equivalent displacement and equivalent stress are shown in the Table 1, where $\langle \cdot \rangle$ denotes the Macauley operator, it can be meaning is as follows:

$$\langle \omega \rangle = \frac{(\omega + |\omega|)}{2} \quad (13)$$

Table 1 Equivalent displacement and stress of each failure mode.

Failure modes	Equivalent displacement	Equivalent stress
Fiber tension	$\delta_{ft,eq} = l_c \sqrt{\langle \epsilon_{11} \rangle^2 + \epsilon_{12}^2 + \epsilon_{13}^2}$	$\sigma_{ft,eq} = \frac{l_c (\langle \sigma_{11} \rangle \langle \epsilon_{11} \rangle + \sigma_{12} \epsilon_{12} + \sigma_{13} \epsilon_{13})}{\delta_{ft,eq}}$
Fiber compression	$\delta_{fc,eq} = l_c \langle -\epsilon_{11} \rangle^2$	$\sigma_{fc,eq} = \frac{l_c \langle -\sigma_{11} \rangle \langle -\epsilon_{11} \rangle}{\delta_{fc,eq}}$
Fiber tension	$\delta_{mt,eq} = l_c \sqrt{\langle \epsilon_{22} \rangle^2 + \langle \epsilon_{33} \rangle^2 + \epsilon_{12}^2 + \epsilon_{23}^2 + \epsilon_{13}^2}$	$\sigma_{mt,eq} = \frac{l_c (\langle \sigma_{22} \rangle \langle \epsilon_{22} \rangle + \langle \sigma_{33} \rangle \langle \epsilon_{33} \rangle + \sigma_{12} \epsilon_{12} + \sigma_{23} \epsilon_{23} + \sigma_{13} \epsilon_{13})}{\delta_{mt,eq}}$
Fiber compression	$\delta_{mc,eq} = l_c \sqrt{\langle -\epsilon_{22} \rangle^2 + \langle -\epsilon_{33} \rangle^2 + \epsilon_{12}^2 + \epsilon_{23}^2 + \epsilon_{13}^2}$	$\sigma_{mc,eq} = \frac{l_c (\langle -\sigma_{22} \rangle \langle -\epsilon_{22} \rangle + \langle -\sigma_{33} \rangle \langle -\epsilon_{33} \rangle + \sigma_{12} \epsilon_{12} + \sigma_{23} \epsilon_{23} + \sigma_{13} \epsilon_{13})}{\delta_{mc,eq}}$

3.2 Inter-laminar damage

Delamination is a common failure mode of laminated composites under impact and compression, significantly reducing their bearing capacity. For complex structures like tapered laminate, using zero-thickness cohesive zone model for finite element modeling and analysis is more convenient. Therefore, in this study, a zero-thickness cohesive zone element based on the bilinear traction-separation relationship is employed to simulate interlaminar delamination in composites. The quadratic failure criterion and the Benzeggagh-Kenane (B-K) criterion [47] are used to predict the initiation and propagation of delamination, as shown in equations (14) and (15) respectively.

$$\left(\frac{\langle t_n \rangle}{t_n^0} \right)^2 + \left(\frac{t_s}{t_s^0} \right)^2 + \left(\frac{t_t}{t_t^0} \right)^2 = 1 \quad (14)$$

where t_n , t_s and t_t represent the normal traction stress and the two shear stresses; t_n^0 , t_s^0 and t_t^0 denote the interface normal strength and shear strength.

$$G_C = G_{IC} + (G_{IIC} - G_{IC}) \left(\frac{G_{Shear}}{G_T} \right)^\eta \quad (15)$$

where G_C , G_{IC} and G_{IIC} are the total, normal and shear critical fracture energy; G_{Shear} and G_T are the dissipated energy in the out-of-plane direction and the total dissipated energy in all three directions, respectively; η is the relevant material coefficient in the B-K law. In this study, it is set to 1.89 according to the literature [48].

4. Finite element model

The finite element software ABAQUS/Explicit is used to establish the LVI, compression and CAI models. The intra-laminar damage model is implemented using a VUMAT subroutine; the inter-laminar damage is configured through the ABAQUS user interface. Table 2 lists the basic mechanical properties and fracture toughness of unidirectional laminate; Table 3 lists the interface properties.

Table 2 Material properties of CCF300/QY9511 [49,50].

Parameter	Value
Modulus (GPa)	$E_{11} = 136$; $E_{22} = E_{33} = 9.81$; $G_{12} = G_{13} = 5.5$; $G_{23} = 4.2$
Poisson's ratio	$\mu_{12} = \mu_{13} = 0.303$; $\mu_{23} = 0.3$
Strength (MPa)	$X_T = 2035$; $X_C = 1376$; $Y_T = 69$; $Y_C = 225$; $S_{12} = S_{13} = 134.7$; $S_{23} = 110$
Fracture toughness (N/mm)	$G_{ft} = 120$; $G_{fc} = 30$; $G_{mt} = 0.21$; $G_{mc} = 1.123$

Table 3 Cohesive properties of interface [51].

Parameter	Value
Elastic stiffness (N/mm ²)	$E_n = E_s = E_t = 10^6$
Maximum traction (MPa)	$t_n^0 = 50$; $t_s^0 = t_t^0 = 90$
Fracture toughness (N/mm)	$G_{IC} = 1.2$; $G_{IIC} = G_{IIIC} = 2.1$
B-K coefficient	$\eta = 1.89$

4.1 Boundary conditions and tapered laminate modeling

Solid elements are used to model each ply of the tapered laminate and the thickness of each ply is 0.125 mm. For the tapered region modeling, as shown in the Fig.5 (a), homogeneous elastic elements are used to represent the resin-rich triangular region, which the mechanical properties are equivalent to the resin pockets. Zero-thickness cohesive zone elements are inserted between the adjacent layers. This method can conveniently and effectively simulate delamination in tapered region.

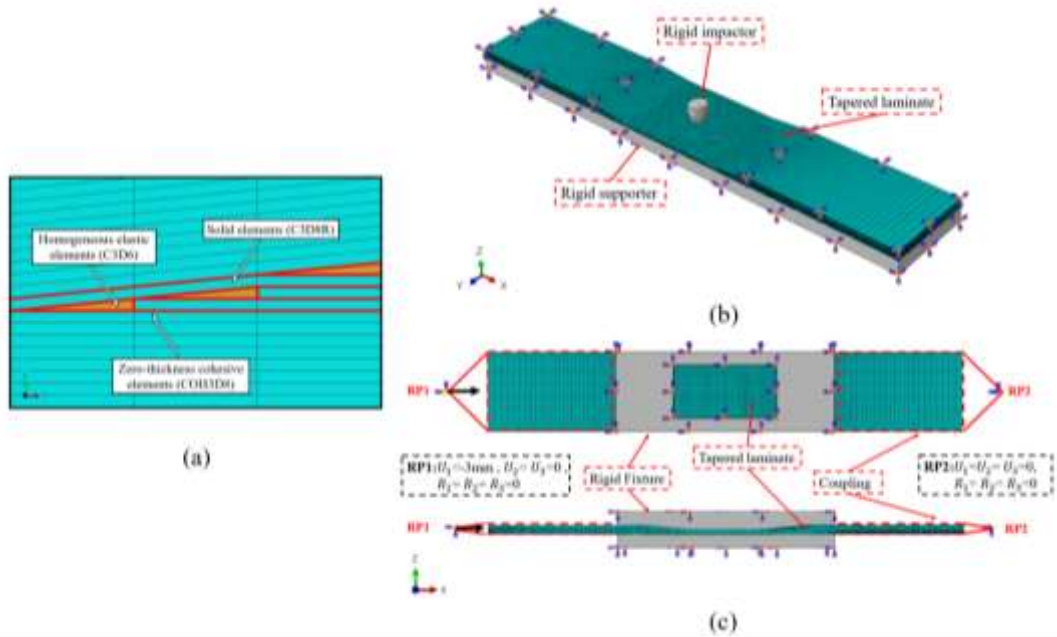


Fig. 5 Finite element models and boundary conditions: (a) tapered region modeling, (b) impact boundary condition, (c) compression and CAI boundary conditions.

The boundary condition of the impact model is shown in Fig.5 (b). The same rigid supporter as the platform of

experiment is established and the upper surface of the thick sections of laminate is fixed. The impactor is a hemispherical rigid body with a diameter of 12.7 mm. The mass of the impactor is set to 5.27 kg and its velocity in the z-direction is set to -1.74 mm/s. For the contact between the impactor and the laminate, as well as the contact between adjacent layers, the penalty method with a friction coefficient of 0.3 is used to describe the tangential behavior, while hard contact method is employed to describe the normal behavior.

Fig.5 (c) shows the boundary conditions of the compression and CAI models, where a rigid fixture identical to the experiment is placed on the thin and tapered sections. There is a 0.1 mm gap between the fixture and the thin section, as well as between the fixture and the tapered sections. The upper and lower surfaces of the thick section of the laminate are coupled with RP1 and RP2, respectively. RP1 is subjected to a 3mm compression displacement in the x-axis direction and the remaining degrees of freedom constrained, while RP2 is fixed.

4.2 Element types and mesh density

The eight-node solid elements with reduced integration (C3D8R) and the 6-node linear triangular prism elements (C3D6) are used to single ply and resin pockets. The enhanced stiffness hourglass control method is implemented to avoid fake deformation. Bilinear cohesive elements (COH3D8) with zero-thickness are used to simulate delamination. The discrete rigid body elements (R3D4) are assigned to the impactor, supporter and fixtures. The meshed model consists of 424,500 C3D8R elements, 197,500 COH3D8 elements and 2,600 C3D6 elements. To ensure the accuracy of the model, the mesh element size is set to 1.2 mm×1.2 mm in the thin and tapered sections. Meanwhile, to improve computational efficiency, the mesh size gradually increases in the thick section.

5 Results and discussion

5.1 Impact behavior

This section presents the impact response and damage from both simulation and experiment, serving as an introduction to the CAI analysis discussed in the subsequent paragraphs. Fig.6 (a) depicts the impact force-time curve. In the experimental data, the impact force peaks at 5.72 kN before sharply dropping and exhibiting significant oscillations over time, indicating matrix cracking and delamination damage. In contrast, the simulation shows a slight drop in impact force at 3.54 kN, signifying the onset of matrix cracking and delamination. The simulation predicts the maximum impact force of 5.23 kN, which is 6.9% lower than the experimental value. Fig.6 (b) illustrates the force-displacement curve. The experimental data shows a maximum displacement of 2.43 mm, while the model predicts a slightly higher displacement of 2.53 mm. Fig.6 (c) displays the absorbed energy-time curve. The absorbed energy from the experiment measures 5.36 J, whereas the simulation predicts 2.58 J. The dissipation of energy is influenced by factors like elastic deformation, friction, intralaminar, and interlaminar damage, making accurate prediction challenging.

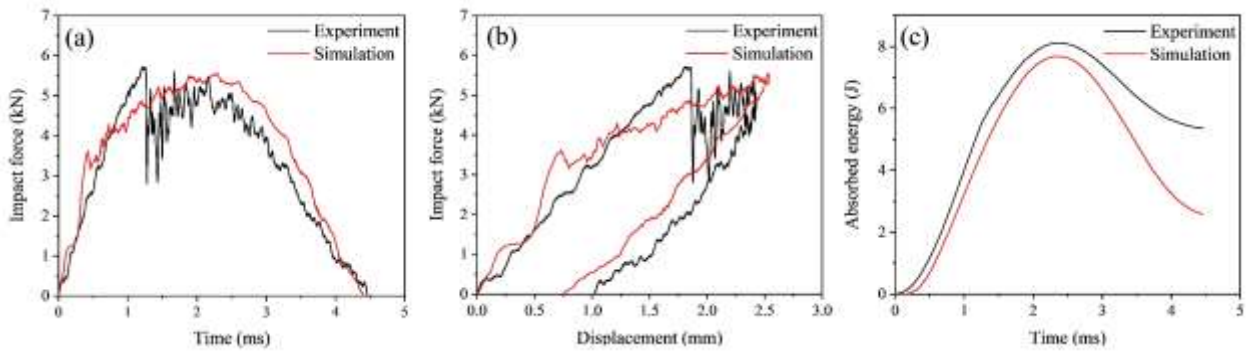


Fig. 6 Impact response of experiment and simulation: (a) force-time curve, (b) force-displacement curve, (c) absorbed energy-time curve.

Low-energy impacts can cause Barely Visible Impact Damage (BVID) on the surface of the laminate. Therefore, CT scanning technology is used to obtain the impact damage at the longitudinal section and compare it with the simulation results. Fig.7 (a) and Fig.7 (b) show the matrix tensile damage and compressive damage from the model. The matrix tensile damage is primarily distributed in the lower half of the laminate, while the matrix compressive damage is located in the upper half. Fig.7 (c) shows the CT scan image, revealing that delamination is mainly situated on both sides of the impact point. Matrix cracking predominantly occurs in the lower half of the laminate because this region experiences in-plane tensile stress, whereas the upper half experiences compressive stress.

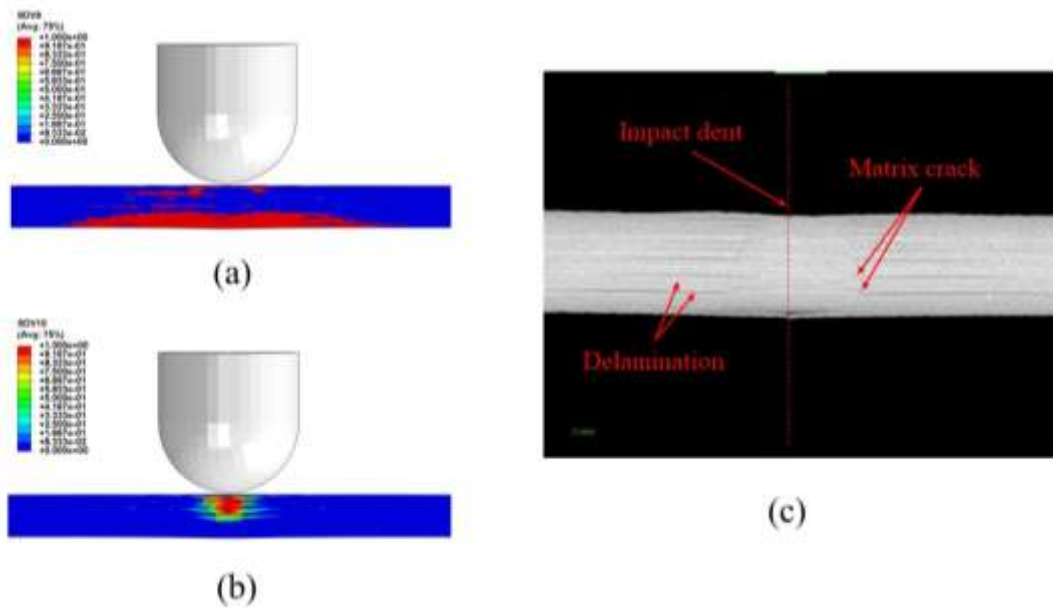


Fig. 7 Impact damage at the longitudinal-section of the experiment and simulation: (a) matrix tension damage of simulation, (b) matrix compressive damage of simulation, (c) CT scan image of the specimen.

To understand the matrix damage in plies with different orientations, four representative plies with orientations of 0° , 45° , -45° , and 90° are selected for analysis. As shown in Fig.8 (a), the matrix tension damage is located on both sides of the impact point and extends along the fiber direction. The damage is not entirely symmetrical about the neutral axis due to the shear stress between adjacent plies with different orientations.

Several typical CT scan images of delamination are shown in Fig.8 (b). The delamination between plies with $0^\circ/90^\circ$, $45^\circ/-45^\circ$, $-45^\circ/90^\circ$, and $0^\circ/45^\circ$ orientations appears as twin triangles. This occurs because, under the impact load, matrix cracks initiate and propagate along the fiber direction. The fibers separate from the resin and slide within each layer perpendicular to the fiber direction, leading to the formation of inter-laminar shear stress regions between adjacent continuous layers. The zones of shear stress, delineated by the separated strips of the two adjacent plies, form triangular shapes that increase in size from the impacted side to the non-impacted side. The matrix cracking and delamination predicted by the model are consistent with the CT scan images.

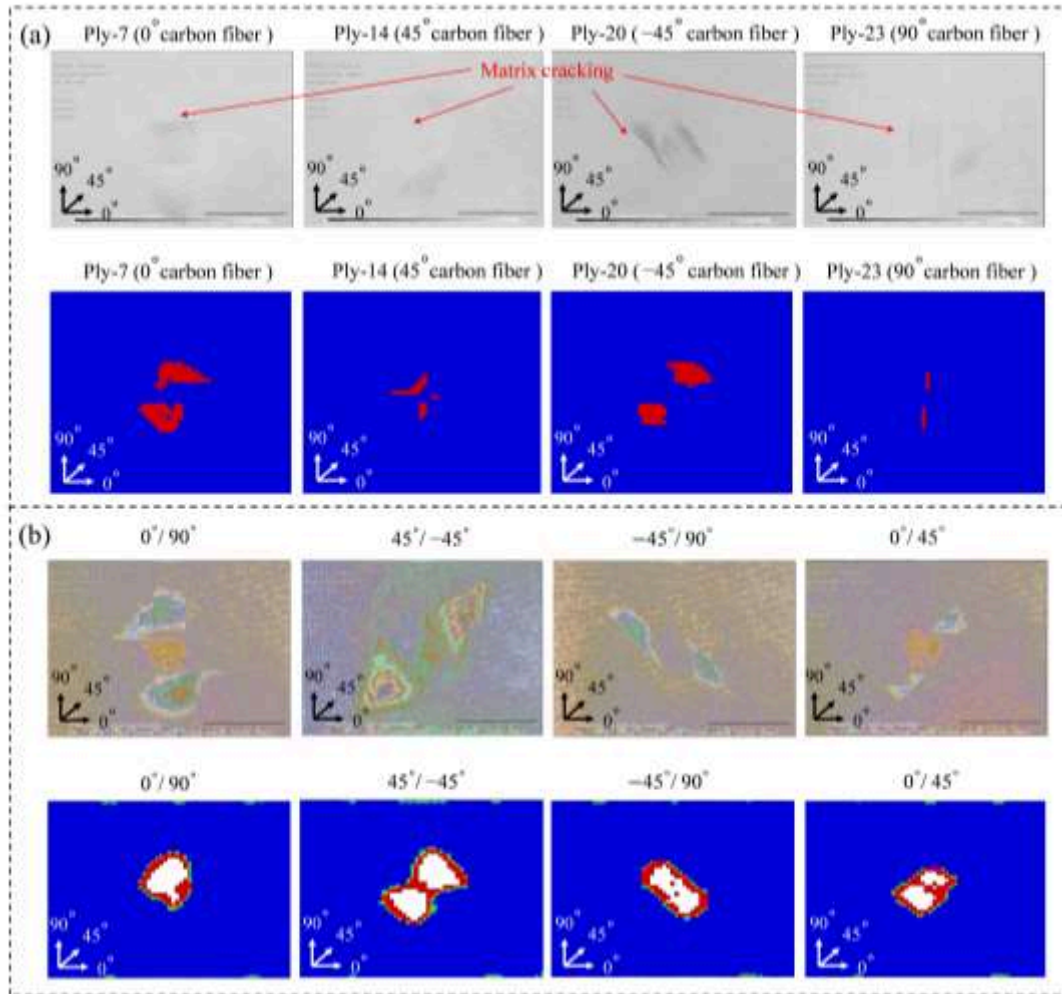


Fig. 8 Matrix cracking and delamination of typical plies from CT scan in the experiment and the cohesive in the simulation: (a) matrix cracking, (b) delamination.

Fig. 9 shows the surface damage from an 8 J impact. On the impacted side near the impact dent, there is only one matrix crack parallel to the 45° fiber. On the non-impacted side, multiple 45° plies splitting cracks are evident, with longer and more pronounced cracks. The impact leads to the exceeding of the matrix tensile strength, initiating the formation of cracks. Fig.10 (a) and (b) show the delamination area and shapes from the ultrasonic C-scans and the cohesive from simulation. There is a good correlation between them, and both exhibit damage profiles that are approximately circular. The delamination area obtained from the ultrasonic C-scans is 302.53mm^2 , while the model

predicts a delamination projection area of 331.02 mm². This means that the delamination projection area from simulation is approximately 9.42% larger than the experiments.

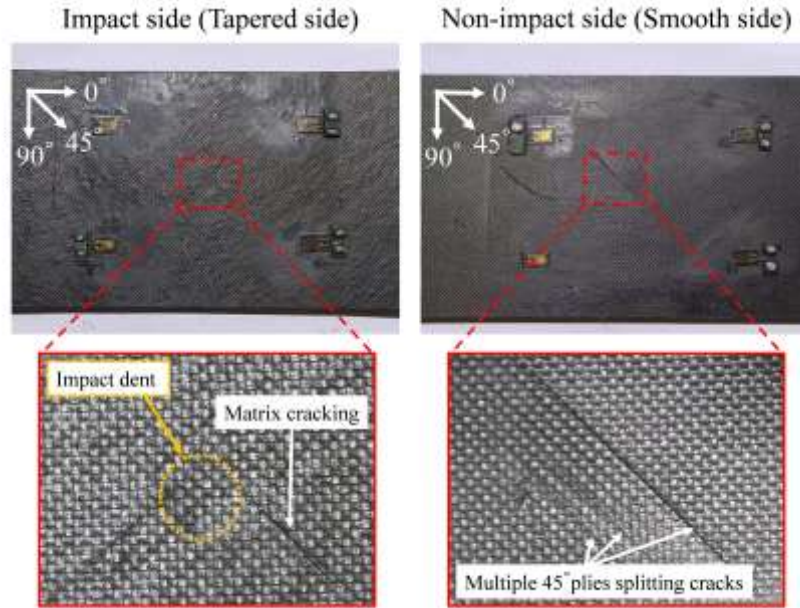


Fig. 9 Visual inspection of composite asymmetric tapered laminate after LVI.

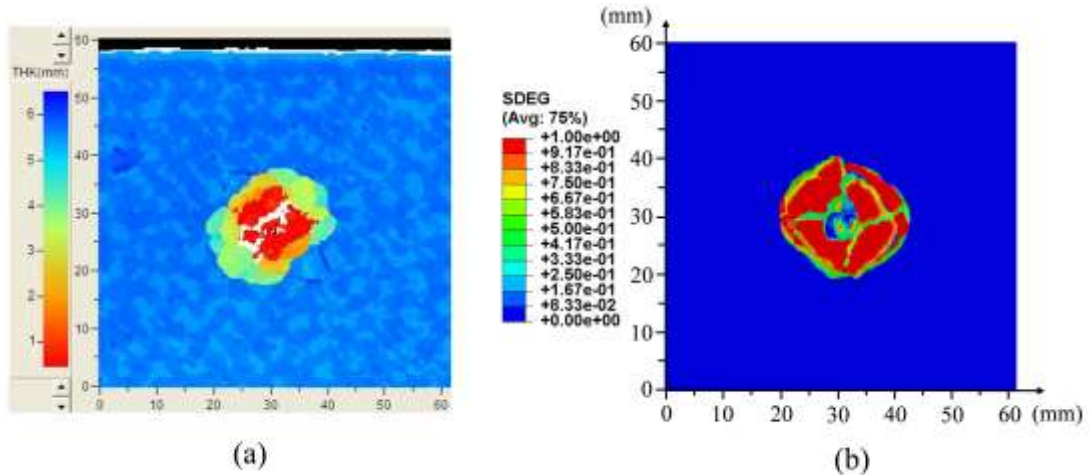


Fig. 10 Comparison of delamination in LVI test through experiment and simulation: (a) delamination profile from ultrasonic c-scan, (b) the stacking of each layer of damage cohesive elements.

5.2 Compression and CAI behavior

5.2.1 Strain and out-of-plane displacement field

Fig. 11(a) and Fig. 11(b) present strain-load curves for compression derived from experiment and simulation. On the tapered side, strains near the junction of the thin and tapered sections (e.g. #1, #3, #4, and #6) increase more rapidly compared to those in the middle of the thin section, such as #2 and #5. Conversely, on the smooth side, the situation is exactly the opposite. Fig. 11(c) and Fig. 11(d) show the CAI strain-load curves from both experiment and simulation. Aside from the differing strain levels observed at specimen failure, the trend of strain variation with load at different locations is consistent with what is observed during compression test. This suggests that LVI

damage minimally alters strain distribution. Furthermore, these curves do not bifurcate as the load increases, indicating that the laminate does not buckle but exhibits small out-of-plane displacements.

Fig. 11(e) and Fig. 11(f) show the compression and CAI strain histograms from both experiments and simulations, the strains are extracted under a compression load of 120 kN and a CAI load of 60 kN. Generally, the strain values from the simulations are higher than those from the experiments. However, the observed pattern remains consistent with previous analyses: the strain values at locations #1, #3, #4, and #6 are higher than those at #2 and #5; similarly, the strain values at locations #7, #9, #10, and #12 are lower than those at #8 and #11.

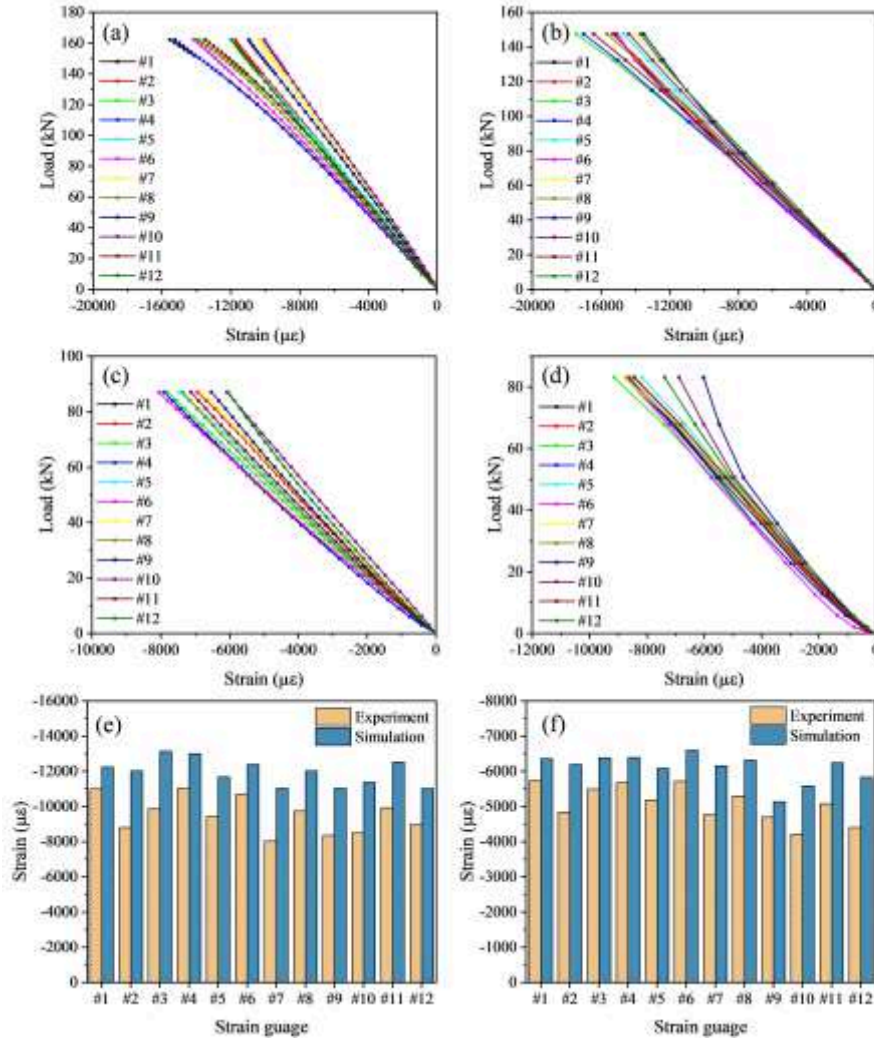


Fig. 11 Strain-load curves from compression and CAI tests, and strain histograms at certain loads: (a) experimental strain-load curves for compression, (b) simulation strain-load curves for compression, (c) experimental strain-load curves for CAI, (d) simulation strain-load curves for CAI. (e) strain histograms for compression, (f) strain histograms for CAI.

The strain analysis of the thin section alone is insufficient to capture the complete distribution of out-of-plane displacement. Fig. 12 presents the out-of-plane displacement maps for both compression and CAI from simulations under different loads. The trends in out-of-plane displacement are fundamentally same for both compression and CAI: two bulges emerge in the thin section near the tapered region, accompanied by a convex inward protrusion at the tapered section. However, a notable difference during the CAI is that LVI induces local deformation in the

center of the thin section. Nonetheless, this effect has a minimal impact on the overall out-of-plane displacement distribution.

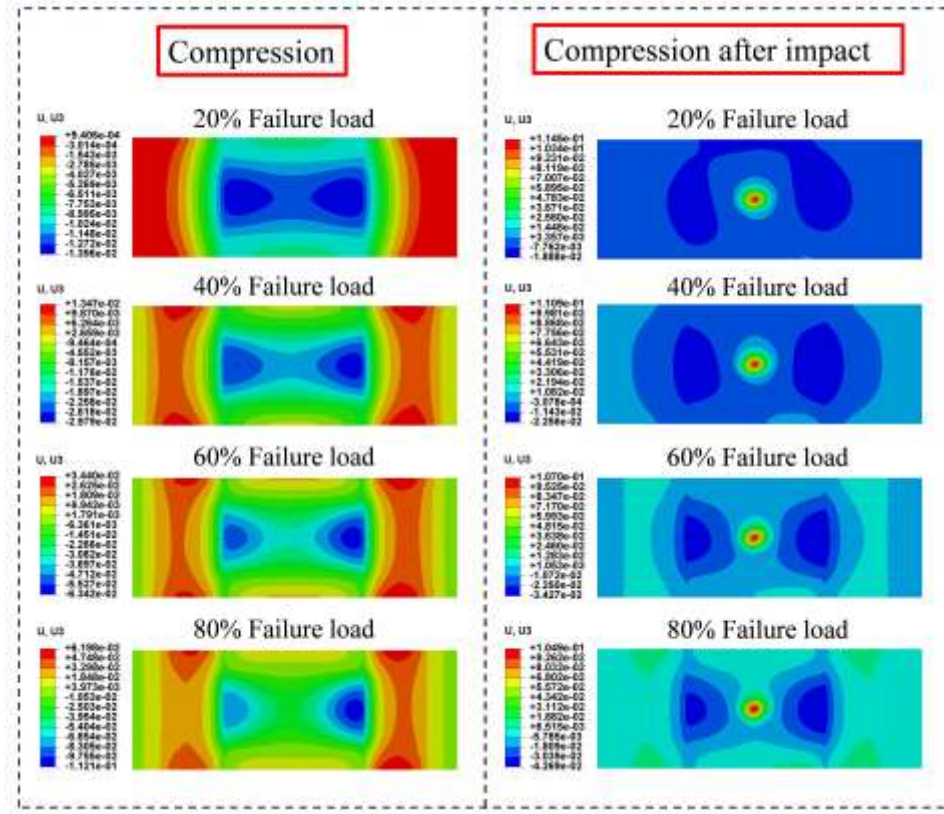


Fig. 12 Contour plots of out-of-plane displacement from compression and CAI models under various loads.

To provide a more intuitive understanding of the out-of-plane displacement process in the tapered laminate, Fig 13(a) presents the out-of-plane displacement values extracted along the central cross-section of the model under various compression loads. Fig 13(b) illustrates the out-of-plane displacement curves during compression. These curves are predominantly positive on both sides within the tapered region, peaking in the middle, which indicates an out-of-plane displacement toward the tapered direction. As the curve approaches the tapered position near the thin section, it transitions from positive to negative, reaching its minimum at the junction of the thin and tapered sections. This behavior suggests a maximum out-of-plane displacement directed toward the smooth side. In the center of the thin section, the curve gradually increases, while the trend of out-of-plane displacement begins to decline. Fig 13(c) depicts the out-of-plane displacement curves for the CAI scenario. Outside the impact point, the out-of-plane displacement trend mirrors that of the compression. However, at the impact point, the out-of-plane displacement is positive and reaches its maximum value.

The analysis of strain and out-of-plane displacement reveals that the LVI damage at the center of the thin section has minimal effect on its compressive out-of-plane displacement field. Instead, structural asymmetry is the dominant factor. Under compressive load, this asymmetry induces an out-of-plane bending moment at the junction between the thin section and the tapered section of the laminate. As illustrated in the Fig. 13(d), this bending moment couples with the compressive load to act on the laminate, resulting in out-of-plane displacement.

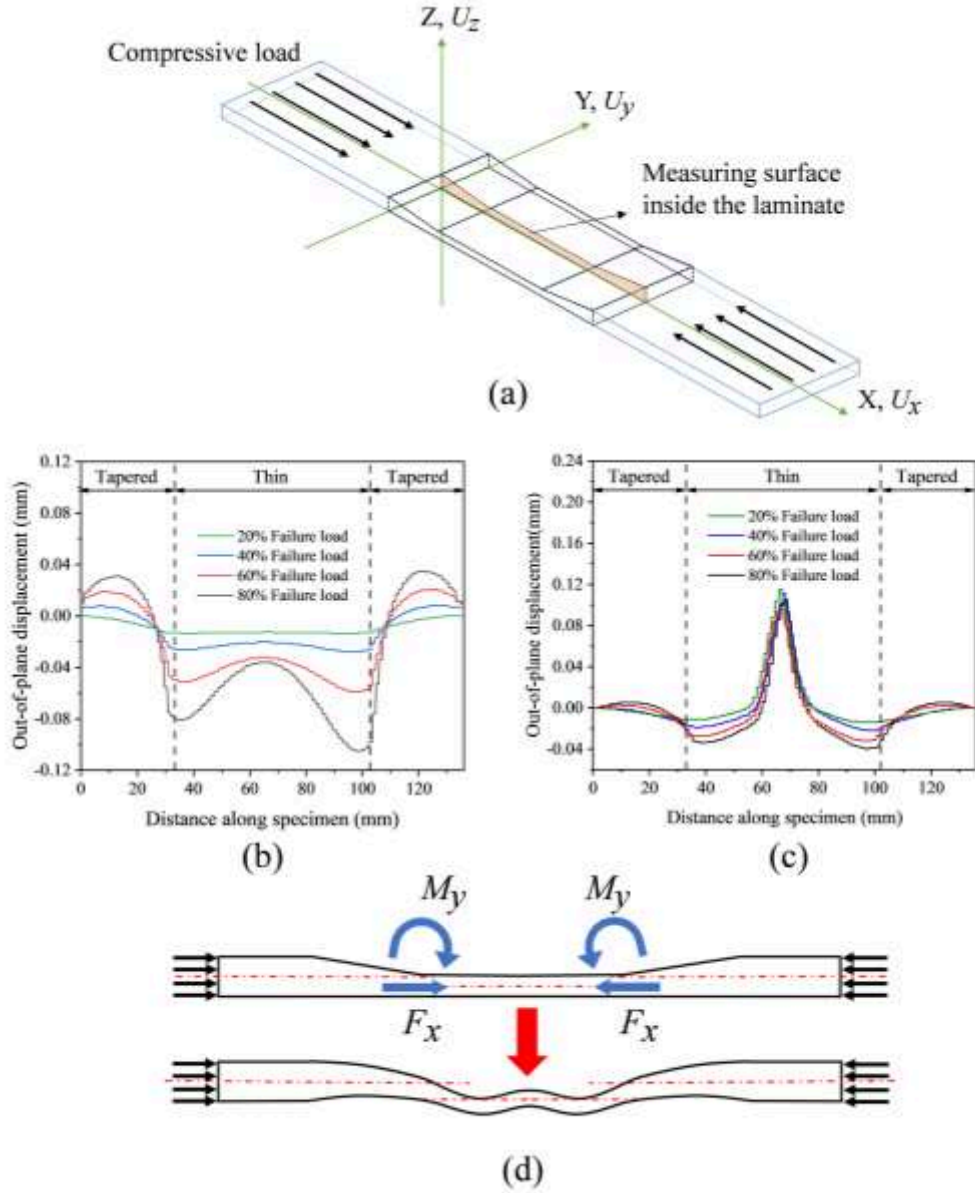


Fig. 13 Evolution of out-of-plane displacements in compression and CAI tests through simulations, along with the underlying mechanism: (a) specimen in global coordinates showing central cross-line location (b) Out-of-plane displacements curves of compression, (c) out-of-plane displacements curves of CAI, (d) schematic diagram of the coupled bending moment and compression load acting on the tapered laminate.

5.2.2 Stress distribution

This section employs the FE model to analyze the stress distribution for compression and CAI. Fig. 14(a) depicts a schematic diagram of the section used to derive stress in the laminate without LVI damage, positioned centrally across the width from the thin section to the junction of the tapered and thick sections. Fig. 14(b) illustrates the extraction of σ_{11} from all 0° plies, as well as σ_{13} and σ_{33} from partial interface layers. Similarly, Fig. 14(c) shows the schematic for stress analysis in the laminate with LVI damage, while Fig. 14(d) details the extraction process for σ_{11} from all 0° plies, as well as σ_{13} and σ_{33} from the partial interface layers.

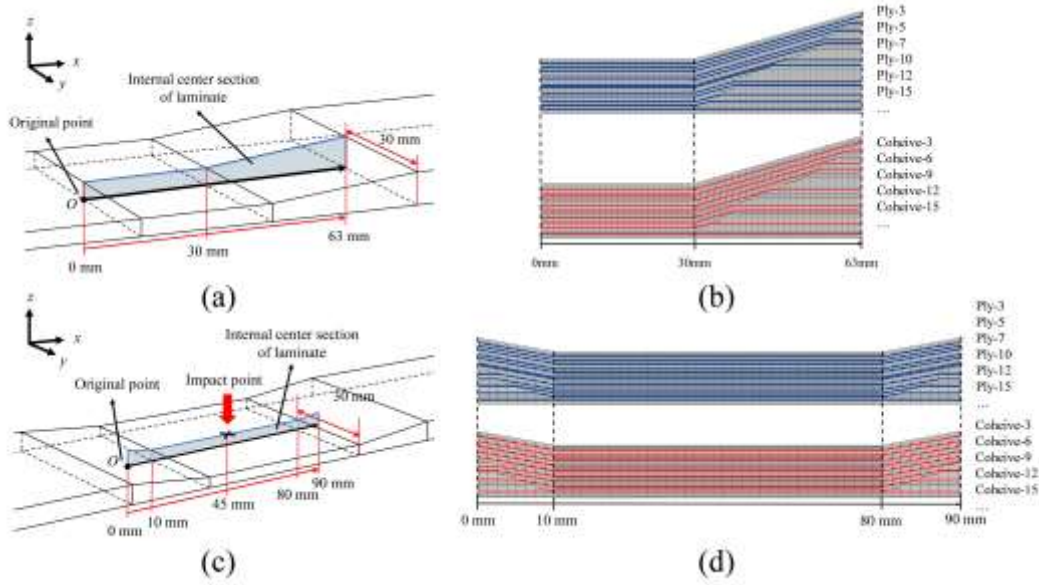


Fig. 14 Schematic diagram of the central cross-section of stress distribution along the x -axis, and the plies and interface layers from which stress is extracted: (a) Internal center section of compressive model, (b) All 0° plies and partial interface layers from compression model, (c) Internal center section of CAI model. (d) All 0° plies and partial interface layers from CAI model.

Fig. 15(a) presents the sectional stress distribution of the compression model under a compressive load of 140 kN. As the number of plies increases, the stiffness at that location also increases, resulting in a reduction in the stress concentration of σ_{11} , particularly for the 0° ply. Significant stress concentrations are observed in inter-laminar shear stress σ_{13} at the junction between the thin and tapered section, with some concentration also noticeable in inter-laminar stress σ_{33} , though less pronounced. For the laminate with LVI damage, as shown in Fig. 15(b), under a compressive load of 80 kN, stress σ_{11} shows a concentration at the impact point. This may be due to delamination caused by the impact, leading to plies buckling under the compressive load. The stress concentrations of inter-laminar shear stress σ_{13} and inter-laminar stress σ_{33} at the impact point are not as pronounced.

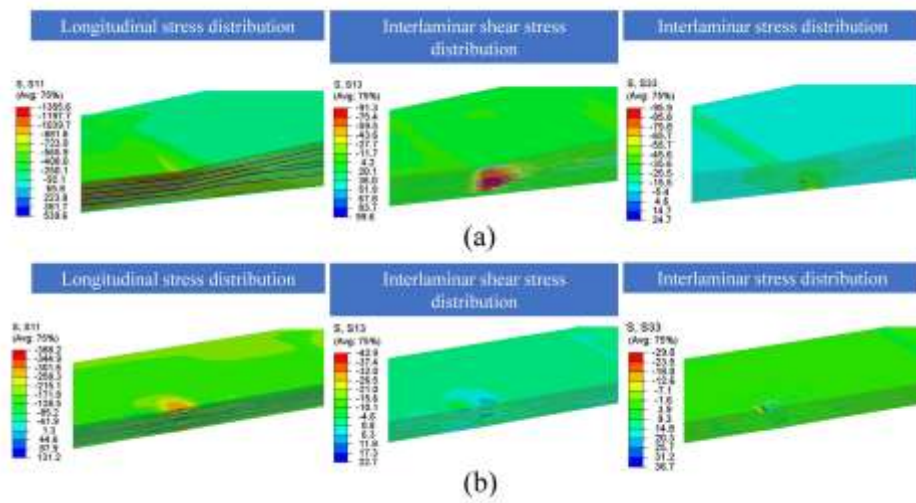


Fig. 15 Stress distribution of the central section from compression and CAI simulations: (a) distribution of longitudinal stress, interlaminar shear stress, and interlaminar stress from compression test, (b) distribution of longitudinal stress, interlaminar shear stress, and interlaminar stress distributions from CAI test.

Fig. 16 presents longitudinal stress distribution curves along the x -axis for all 0° plies under a 140 kN compressive load. These curves are derived from compression simulation using the element centroids in solid elements. A comparison of plies at different positions reveals that stress concentrations at the junction between thin and tapered sections are more pronounced closer to the tapered side, whereas plies near the smooth side exhibit minimal stress concentration at junction. This suggests that structural asymmetry in tapered sections generates higher axial compressive stress due to bending moments. Sub-layers near the tapered side experience greater stress than those in other positions. Local stress peaks also occur near dropped plies like Ply #7, Ply #21, and Ply #24, where these plies transfer axial loads to adjacent plies, causing localized stress concentration. However, these concentrations are significantly smaller than those induced by structural changes.

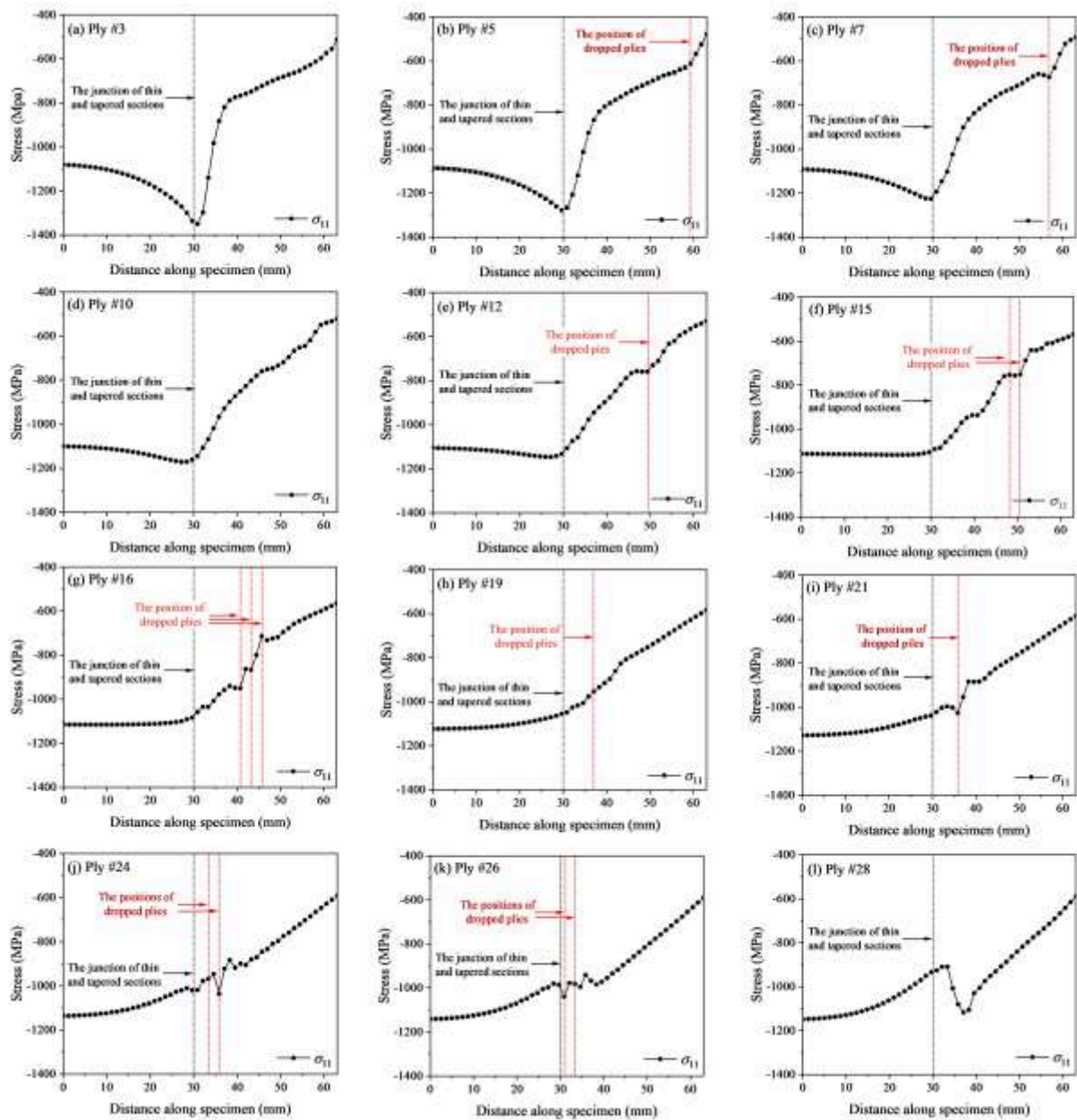


Fig. 16 Stress σ_{11} distribution curves along the x -axis for all 0° plies under a compressive load of 140 kN from the compression simulation.

Fig. 17 presents the variation of interlaminar shear stress and interlaminar tensile stress along the length of the laminate, based on data extracted from element centroids in the interface layers of cohesive elements. Compared to interlaminar tensile stress, interlaminar shear stress exhibits higher levels in thin and tapered sections, with notable peaks at their junctions. This significantly contributes to delamination initiation. Moreover, localized stress peaks in interlaminar shear stress are observed at positions corresponding to dropped plies, such as Cohesive #6, Cohesive #9, and Cohesive #15. Consistent with previous longitudinal stress analyses, stress peaks caused by dropped plies are smaller than those induced by structural variations.

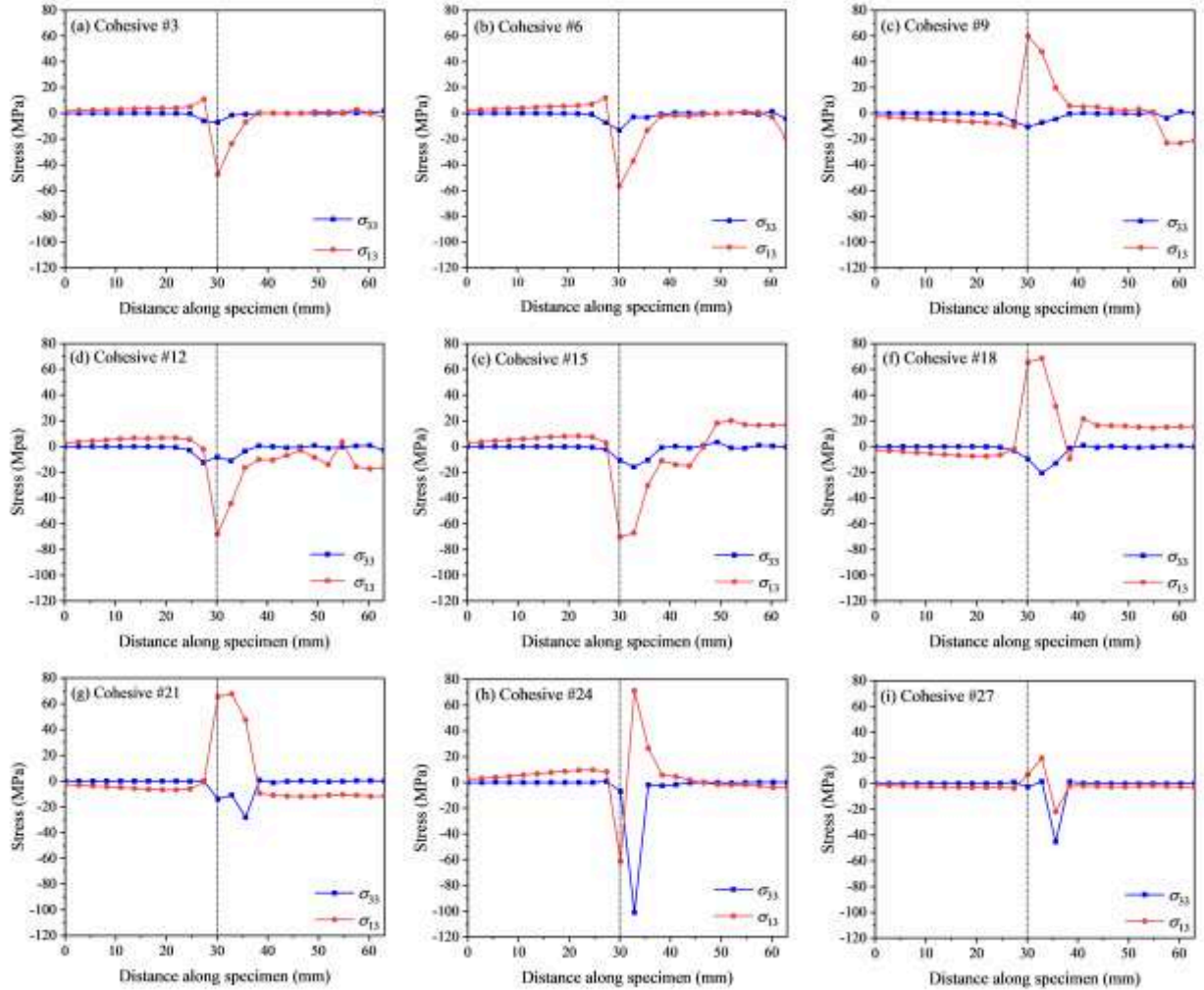


Fig. 17 Stress σ_{13} and σ_{33} distribution curves along the x-axis for partial interface layers under a compressive load of 140 kN in the compression simulation.

Fig. 18 presents longitudinal stress distribution curves along the x-axis for all 0° plies from CAI simulation under an 80 kN compressive load. Apart from the impact-damaged area, the longitudinal stress distribution in the non-impacted area is similar to that obtained from previous compression simulation. This indicates that localized impact damage does not significantly alter the longitudinal stress distribution in the non-impacted areas. In the impact-damaged area, stress exhibits significant fluctuations with multiple peaks and valleys, some of which in certain plies (e.g., Ply #10, Ply #12, Ply #15, and Ply #16) even exceed the stress peaks at the junctions between

thin and tapered sections. This suggests that LVI damage can create new, more significant longitudinal stress concentrations in the sub-laminates. These stress concentrations result from delamination caused by impact, leading to the buckling of sub-layers due to the loss of out-of-plane constraints.

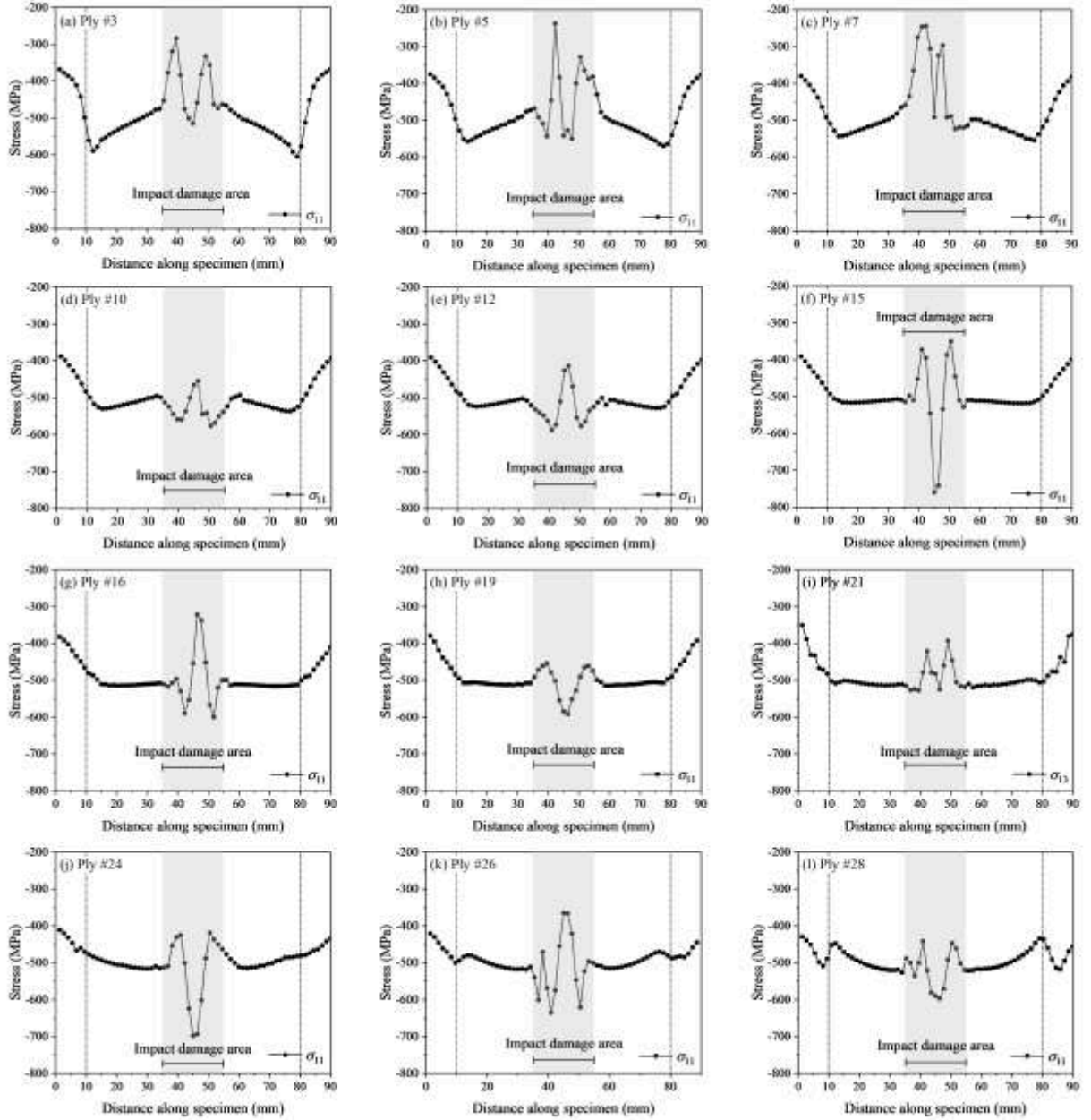


Fig. 18 Stress σ_{11} distribution curves along the x -axis for all 0° plies under a compressive load of 80 kN in the CAI simulation.

Fig. 19 depicts the variation of inter-laminar shear stress and inter-laminar stress along the length of the laminate from CAI simulation under the 80 kN compression load. Similar to the conclusions drawn from previous compression simulation, inter-laminar shear stress continues to dominate at the junction between the thin and tapered sections even in the case of impact damage. Due to the complete failure and deletion of cohesive elements caused by LVI, the inter-laminar shear stress and normal stress in some units drop to zero. However, it is noteworthy that the interface layers close to the impact damage area still exhibit localized stress peaks.

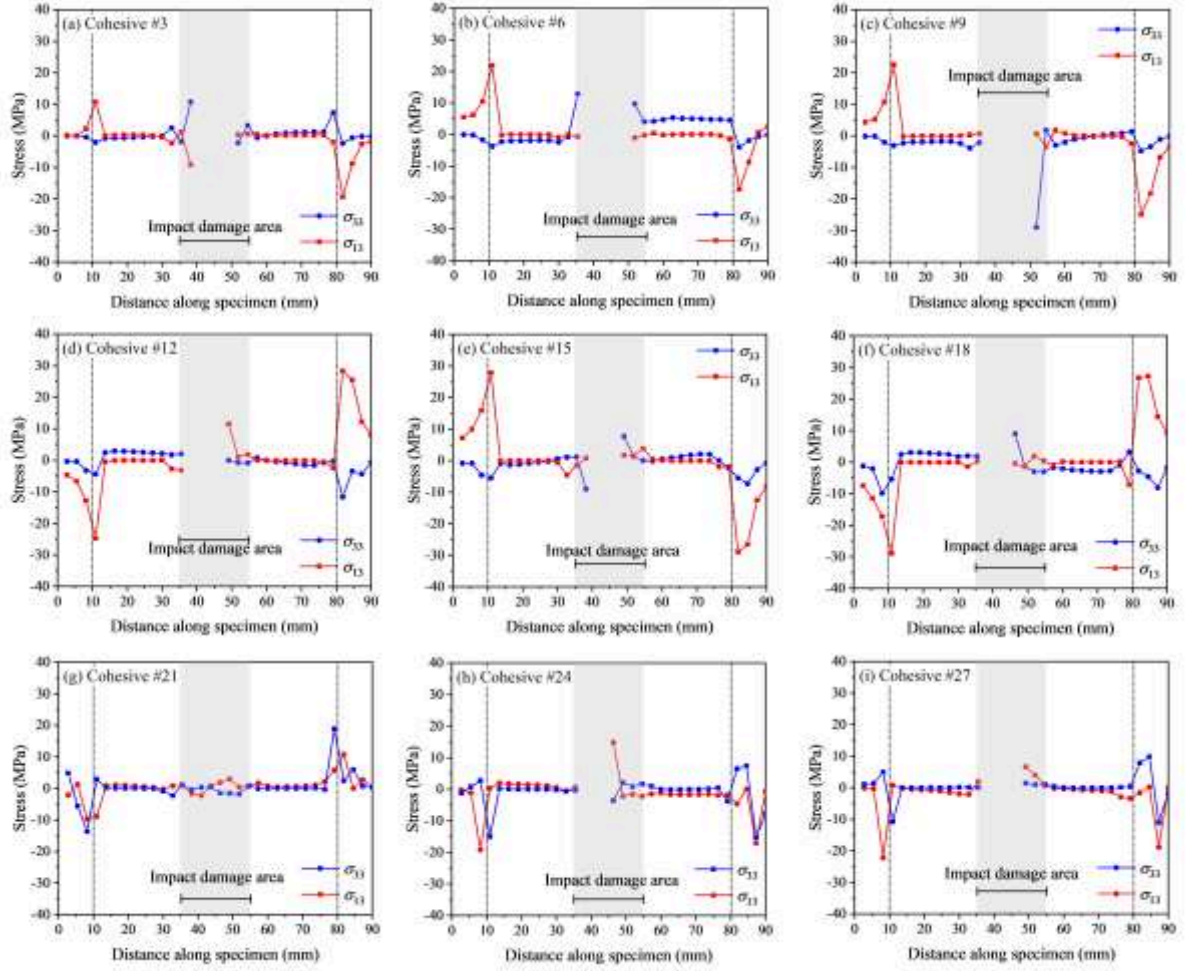


Fig. 19 Stress σ_{13} and σ_{33} distribution curves along the x -axis for partial interface layers under a compressive load of 80 kN in the CAI simulation.

5.2.3 Failure mechanism

The visual inspection and microscopy aim to reveal the potential damage mechanisms and identify the origins in the mechanical behaviors of the laminates during compression and CAI tests. Furthermore, by comparing the damage between compression and CAI, it provides a better understanding of how LVI damage in the thin section affects and alters the damage mechanisms and failure behaviors.

As illustrated in Fig. 20, the visual and microscopic inspections reveal the failure mechanisms of the intact tapered laminate under compressive load. In the lateral side-view microscopy, the damage primarily manifests as delamination and sub-laminates fractures. Prominent delamination occurs at the tapered section, where an inter-laminar crack initiates at the resin pockets at the junction between the thin and tapered section, then propagates along the inter-laminar path towards the thick section. Above the cracking resin pockets, the sub-laminates buckle due to the loss of inter-laminar constraint, leading to fractures. The fractured sub-laminates undergo subduction under compressive load, leading to an increase in the length of inter-laminar cracks. Additionally, the interface layer above the sub-laminates exhibits local cracks under the influence of inter-laminar shear stress. Due to the

fracture of the sub-laminates in the middle, the interaction between plies causes the surface sub-laminates to buckle and fracture. The difference in fracture locations between the surface and middle sub-laminates further explains the assumed sequence of damage occurrence.

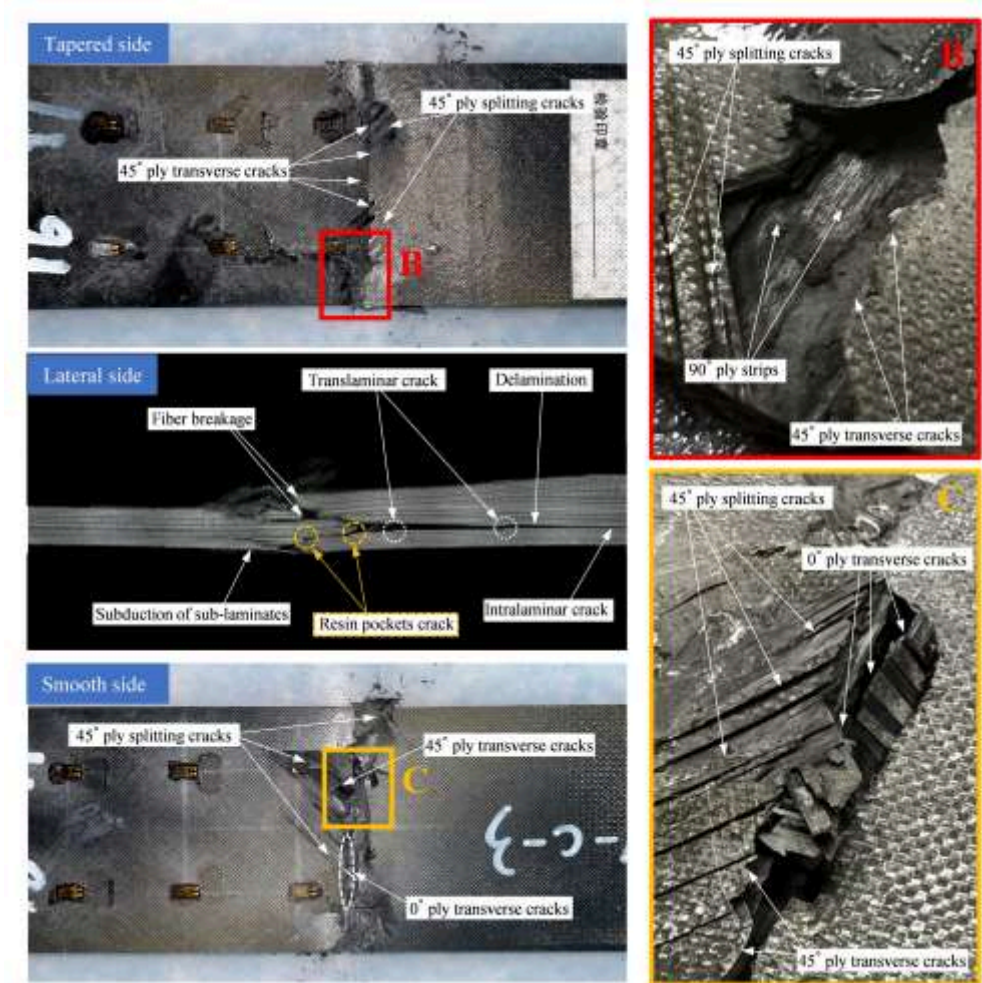


Fig. 20 Damage inspection of intact tapered laminate under compression test (macroscopic visual inspection and microscopic inspection).

Several instances of translaminar cracking observed within the delamination cracks indicate that the 0° plies are constrained by adjacent 90° plies. Although they do not bear the primary load, the 90° plies reduce the likelihood of buckling in the fibers aligned along the compression axis. This interaction between the 0° plies and 90° plies induces bending stress in the 90° plies, leading to translaminar cracks traversing the matrix.

Macroscopically, the failure of the surface 45° ply on the tapered side is primarily characterized by transverse cracks and splitting cracks. These cracks are located at the junction and extend across the entire width of the laminate. Analysis of local damage micrographs at the failure site (Fig. 20, B-red) shows that fiber fracture is the predominant form of damage in 45° strips, while matrix cracking occurs only in some of the 45° strips. These two types of damage alternate, producing a serrated damage morphology. Inside the fracture site, the 90° ply exhibits matrix cracks along the fiber direction, likely caused by transverse tensile stresses or inter-laminar shear stresses.

Similar patterns of damage are also noted on the smooth side of the laminate, where the 45° plies distinctly show significant transverse cracks that cut across the ply and splitting cracks that ran along it. The associated damage micrograph (Fig. 20, C-yellow) further reveals that the 45° strips are notably more deformed in a bending fashion at the site of the fracture, and the transverse cracks seem to be more cleanly defined; they appear less jagged compared to other observed fractures. Deeper within the laminate structure, the internal 0° plies are found to bear significant fractures and have undergone severe bending deformations as a result of the compressive load. This severe warping at the site of rupture renders the cracks especially conspicuous.

The microscopic damage inspection of the CAI test on the tapered laminate, viewed laterally as depicted in the Fig. 21, reveals damage patterns akin to those observed in the intact laminate under compression. The damage predominantly features delamination and fractures in the sub-laminates, with delamination occurring at only one or two inter-laminar interfaces and minimal buckling of the surface sub-laminates. This suggests that the support of fixture effectively limited the out-of-plane deformation of the sub-laminates to a certain extent. Wedge-shaped bands are observed at the extremities of the sub-laminates adjacent to the upper and lower surfaces of the fracture site, indicative of shear stress. The sub-laminates in the middle undergo delamination, leading to an increase in the length of inter-laminar cracks and consequently enlarging the delamination area.



Fig. 21 Damage inspection of the tapered laminate under CAI test (macroscopic visual inspection and microscopic inspection).

In the macroscopic inspection on the impact side, the 45° plies exhibit splitting and transverse cracks located on both sides of the impact point, extending to the edges. Local microscopic image (Fig. 21, B-red) reveals that these cracks primarily consist of fiber fractures and matrix cracking. However, compared to the intact laminate under compression, these cracks are smaller and exhibit minimal bending. On the non-impact side, as illustrated in Fig. 21 (Non-impact side), the damage significantly differs from that of the intact laminate. Near the impact area, there are almost no transverse cracks in the 45° strips, while significant transverse cracks appear at the edges (where supported by fixtures). Splitting cracks along the 45° direction nearly span the entire width of the laminate, a result of matrix cracking in the 45° plies caused by LVI on the non-impact side, with cracks propagating under compression due to shear stress. These multiple 45° strips, lacking constraint and without out-of-plane support from fixtures, can freely undergo out-of-plane buckling (Fig. 21, C-yellow). These strips do not have sufficient space to exceed the fiber failure strain, hence the fibers do not fracture.

Intra-laminar cracking may be visible while analyzing the buckled 45° ply strips (Fig. 21, C-yellow). The visible textures beneath the buckled fibers suggest that only part of the layer has yielded. Such behavior in the ply is not feasible without the presence of cracks between the fibers, which are constrained to expand solely in the horizontal direction. The position of the ply residues may also contain fiber imprints left due to fiber relaxation.

To gain deeper insights into the internal damage morphology of the tapered laminate during compression and CAI tests, CT scanning technology is employed to inspect the damaged areas. As depicted in the Fig. 22(a), the cross-section CT scans are positioned centrally along the width of the laminate, while the longitudinal-sections are specifically located within the compression and CAI damage zones.

In the longitudinal-section CT images of the compression test, as show in Fig. 22(b), extensive fiber twisting and breakage are evident at the damaged sites. Sub-laminates near the surface in thin section show more pronounced buckling compared to tapered section, likely due to the decreased stiffness of the taper. The CT scan cross-sections reveal extensive delamination at fracture sites, more pronounced than what is observed in previous lateral microscopic inspections. The central area exhibits greater delamination due to the lack of fixture support in the z -direction, contrasting with the edges. Additionally, at fixture-supported boundaries, there are instances of sub-laminate buckling and breakage. This is attributed to out-of-plane buckling of central sub-laminates during compression damage, while constrained boundaries experience deformation and eventual fracture due to out-of-plane shear stresses. This phenomenon is also observable in the cross-sectional CT results of the CAI tests.

In comparison to compression, as show in Fig. 22(c), longitudinal-section CT image of CAI reveals a more intricate pattern of sub-laminate subduction. Fibers and matrix are crushed by kinking or crack migration within the matrix, leading to shearing and bending that divide the intact laminate into multiple single layers or sub-laminates. As sub-laminates delaminate, these bands propagate through each layer, increasing interlayer crack lengths and consequently expanding delamination zones. This type of damage formation is documented in the literature as part of the “splitting mode” [52, 53] or “delamination fracture” [54] in thick composites.

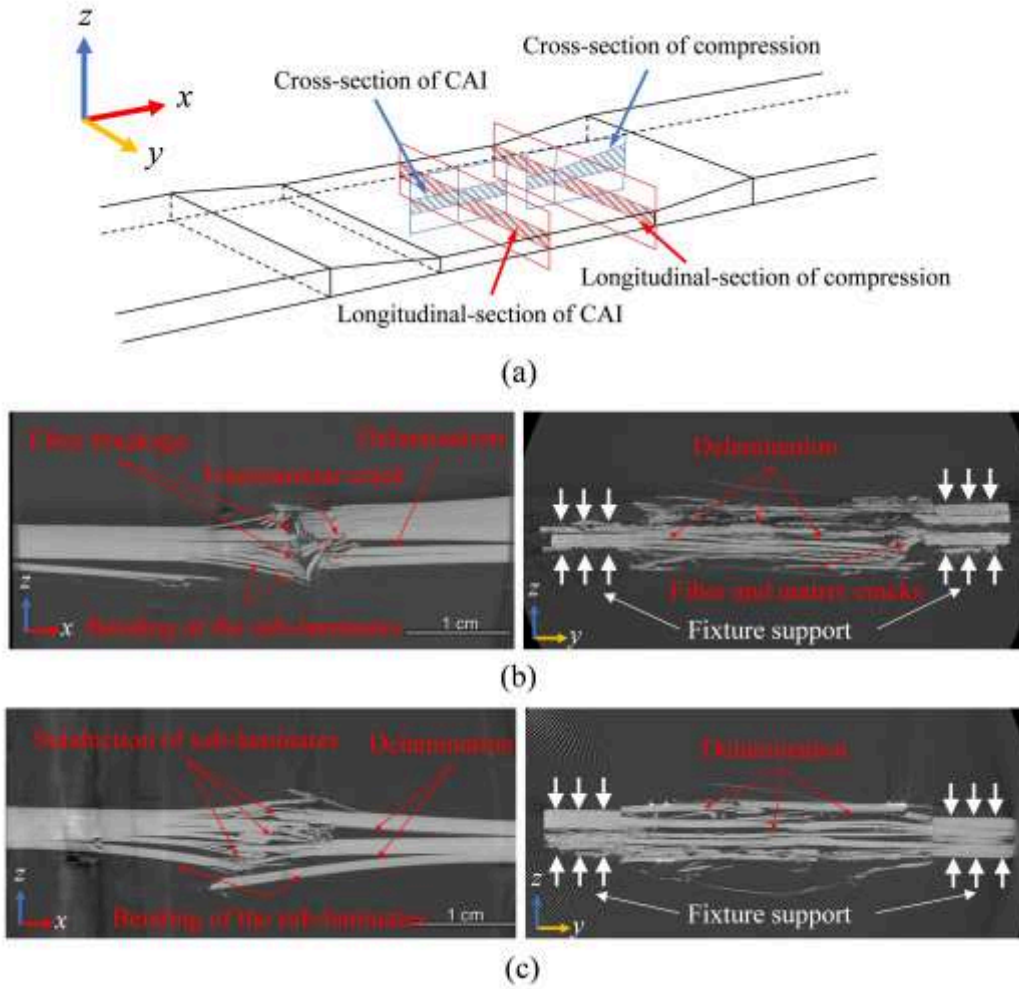


Fig. 22 Internal damage from CT scans of the specimens under compression and CAI tests: (a) schematic diagrams of the scanned longitudinal-section and cross-section, (b) CT scan images of compression failure, (c) CT scan images of the CAI failure.

In Fig. 23, fiber compressive failure, matrix cracking, and delamination in the FE models for compression and CAI are visualized using the transparent method. In the compression results, at Point "1", before laminate failure, no fiber or matrix failure is observed within the laminate, with only minor delamination initiation at the junction between the thin and tapered section. At Point "2", upon reaching failure, fiber compressive damage, matrix cracking, and delamination appear at the junction and rapidly expand to the edges of the laminate. At Point "4", the damage reaches its maximum extent, and the laminate is completely destroyed. In the CAI simulation results, from Point "1" to Point "2", the LVI damage within the laminate does not change significantly. From Point "2" to Point "4", all damage propagates simultaneously from the impact zone along the transverse direction to the side boundaries. Although the FE model predicts that compression strength and CAI strength are primarily related to fiber failure, the influence of delamination and matrix damage on fiber failure and strength is still significant. Specifically for CAI, the initial damage from LVI, such as matrix cracking and delamination, causes the sub-laminates to buckle outwards, which dictates the failure location and damage mode in CAI, leading to a reduction in strength.

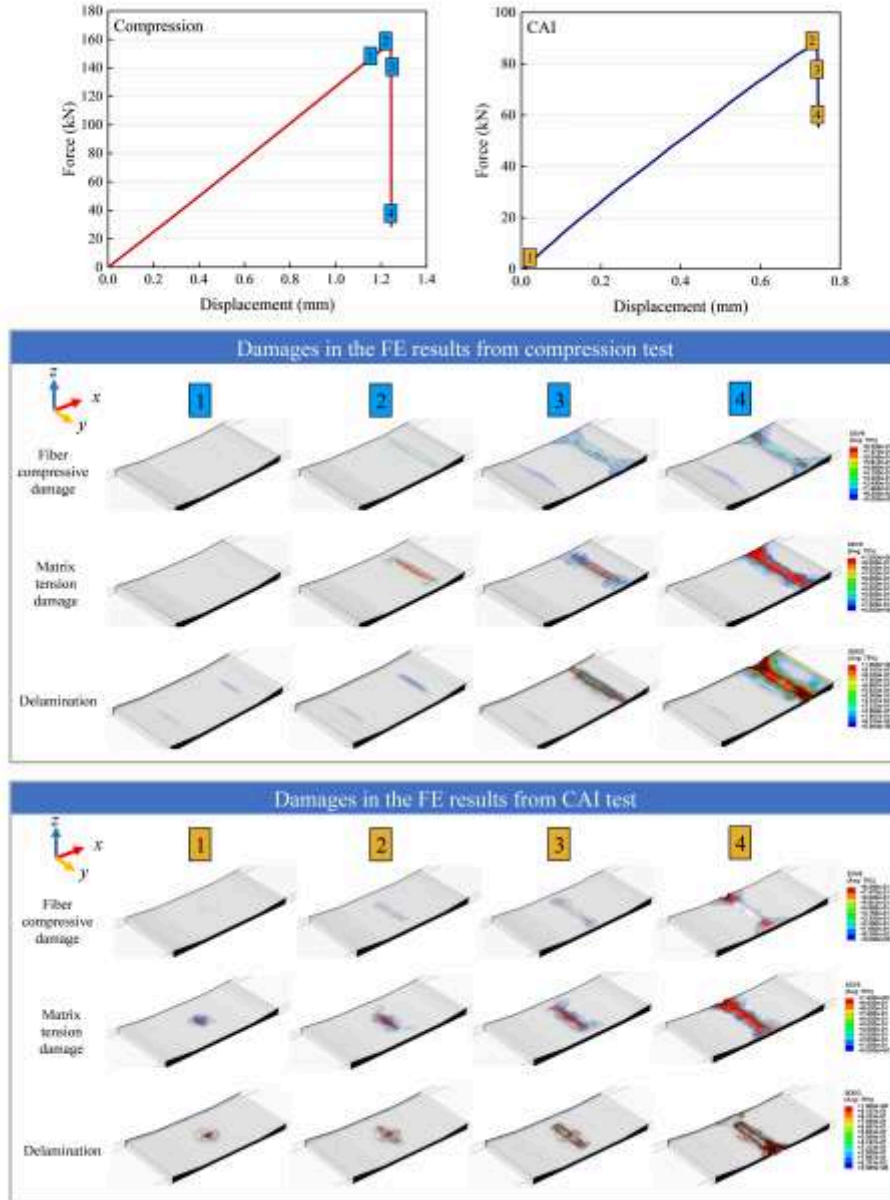


Fig. 23 Damage in the FE models from compression and CAI tests, and the correlation between the damage and the four points in the force-displacement curves.

As depicted in Fig. 24, FE models are used to analyze the out-of-plane displacements and delamination across different transverse locations under compression and CAI at four distinct points. Preliminary analyses revealed that prior to laminate failure, the out-of-plane displacement is predominantly influenced by the structural asymmetry. However, as the failure load is reached, notable changes in out-of-plane displacement are observed, attributed to the buckling of sub-laminates caused by delamination. During compression, delamination is initiated at the junction between the thin and tapered sections, leading to sub-laminates buckling towards the smooth side due to the structural asymmetry. In the case of CAI, delamination spread from the impact-damaged area to both sides, causing sub-laminates near the upper surface (impact side) to displace upwards and those near the lower surface (non-impact side) to displace downwards, culminating in an "S" shaped deformation of the laminate. As the compressive

displacement increased, the deformation evolves from an "S" shape into a bulging form, both upwards and downwards. Although the supports of fixture prevent the buckling of sub-laminates from extending to the edges, delamination reaches the edges upon laminate complete failure.

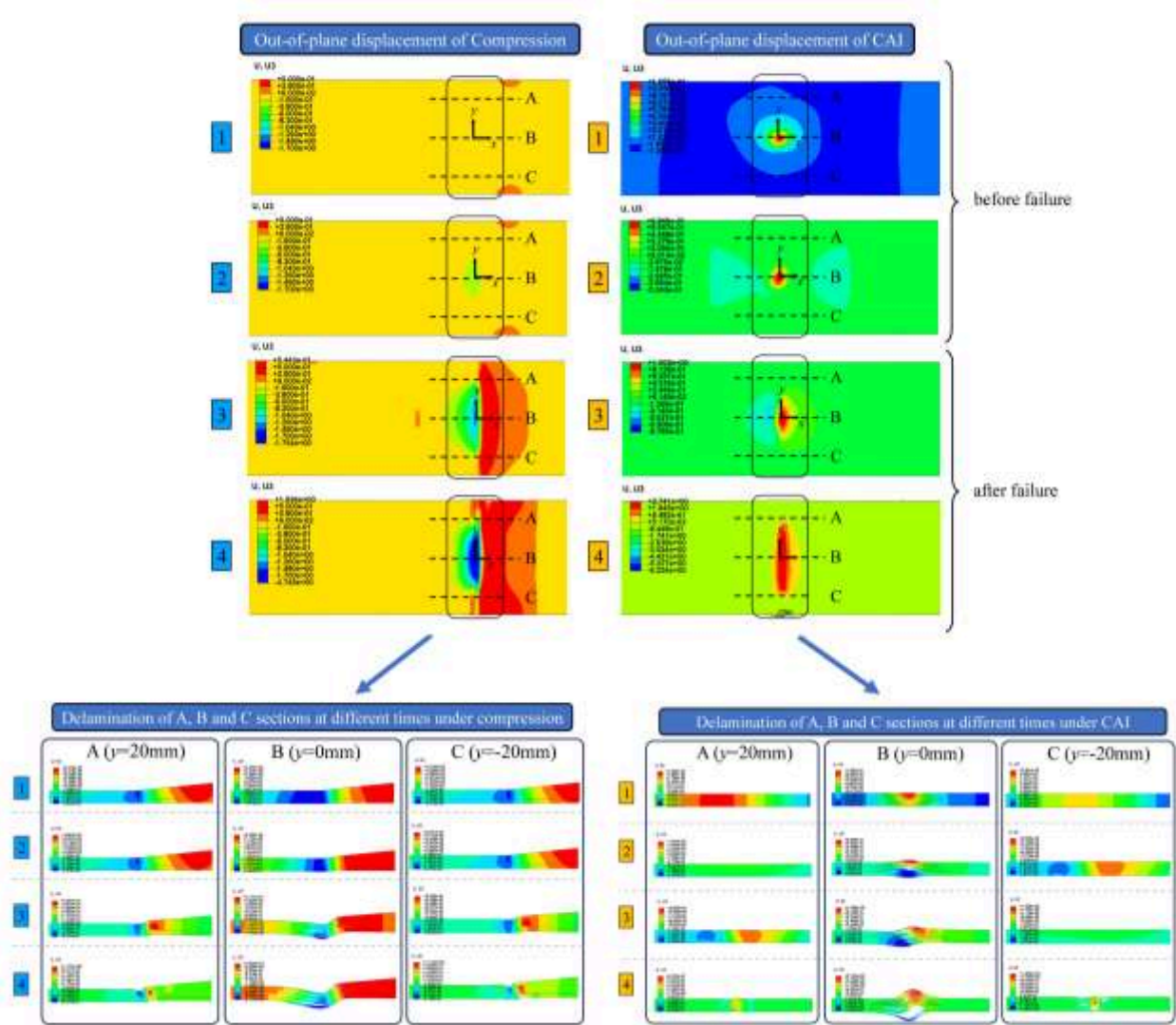


Fig. 24 Out-of-plane displacement fields at four points in the load-displacement curves, along with the delamination and sub-laminates buckling in different longitudinal-sections for compression and CAI tests in the FE model.

The delamination of compression and CAI obtained from Ultrasonic C-scan in experiments and cohesive elements in simulations are shown in Fig. 25(a) and Fig. 25(b). For compression, delamination irregularly occurs at the junctions of the thin and tapered sections across the laminate width. The delamination area of compression detected by Ultrasonic C-scan is 2554.2 mm², whereas the simulation predicts 2080.4 mm², about 18.5% lower than the experimental results. Regarding CAI, delamination is located in the middle of the thin section. The delamination area of CAI detected by Ultrasonic C-scan is 2357.6 mm², while the simulation predicts 1814.8 mm², approximately 23.0% lower than the experimental results. Although the values from simulations are lower than experiments, they effectively capture the location and distribution of delamination.

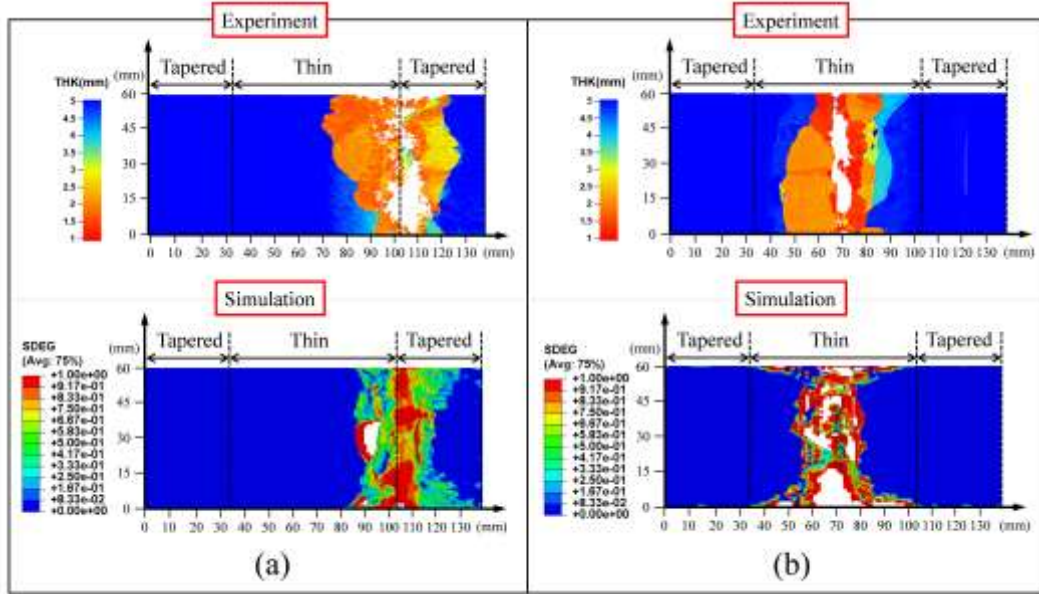


Fig. 25 Delamination of compression and CAI from Ultrasonic C-scan in experiment and cohesive elements in simulations: (a) compression, (b) CAI.

5.2.4 Compressive strength and residual compressive strength after impact

Fig. 26(a) and Fig. 26(b) present the load-displacement curves of compression and CAI from both experiments and simulations. The curves from the four compression tests exhibit high consistency, shown in Fig. 26(a), exhibit high consistency, indicating the repeatability and reliability of the experimental results. Before failure, there are no significant fluctuations in the curve, which signifies the absence of buckling or damage in the specimen. Upon laminate failure, there is a sudden drop, signifying abrupt damage propagation. In contrast to compression results, the CAI curves from the four tests exhibit greater dispersion. This is attributed to the initial damage introduced by impact, such as matrix cracking and delamination, which displays a certain degree of randomness. Such randomness can influence mechanical behavior and crack propagation during CAI, resulting in variations in laminate stiffness and compressive strength.

Fig. 26(c) shows histograms of compression strength and CAI strength from experiments and simulations. The average compression strength in experiments is 722.01 MPa and the CAI strength is 394.53 MPa. Due to impact, the compressive strength decreased by 45.36%. The simulated compressive strength and CAI strength are 698.62 MPa and 394.53 MPa, which are 3.24% and 9.01% lower than the experimental values. There is a strong correlation between numerical simulations and experiments, validating the accuracy of FE models in predicting compression and CAI strength of asymmetric tapered laminates.

Table 4 presents the original data on failure loads and stiffness from the experiments, along with the corresponding values predicted by the model. In the experiment, the impact results in an average decrease of 45.36% in the failure load and a reduction of approximately 2.54% in average stiffness. In the simulation, the impact led to a decrease of about 43.53% in failure load and a 7.93% reduction in stiffness. This indicates that LVI damage located in the thin section significantly reduces the compressive strength of the tapered laminate, while

having only a minimal impact on its stiffness.

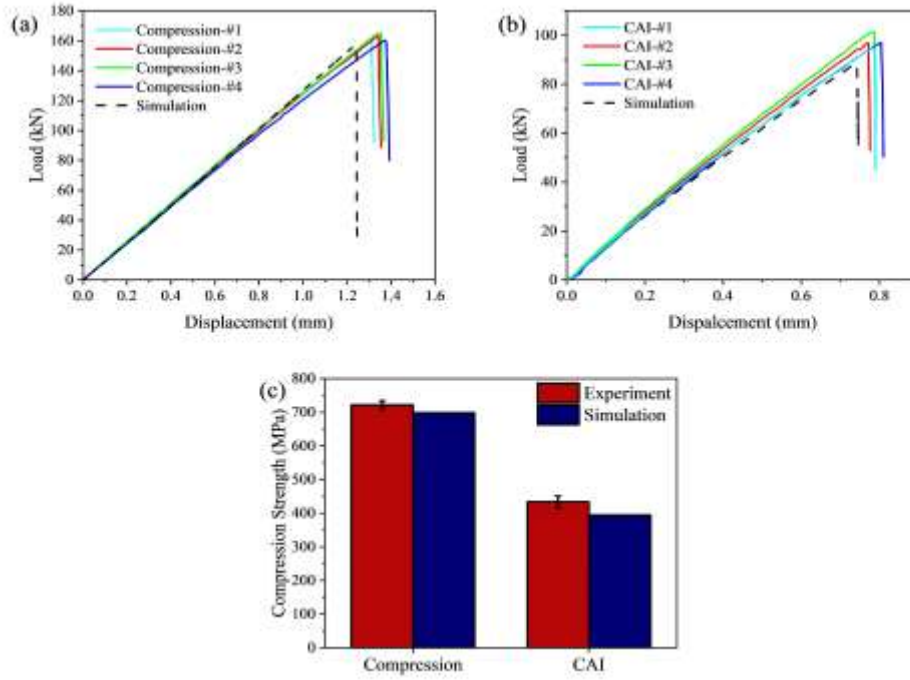


Fig. 26 Load-displacement curves and strength of compression and CAI from experiments and simulations: (a) load-displacement curves of compression, (b) load-displacement curves of CAI, (c) compression and CAI strength.

Table 4 Compression and CAI failure loads and stiffness from experiments and simulations

Compression test				CAI test		
	The labels of specimens	Failure load (kN)	Stiffness (N/m)	The labels of specimens	Failure load (kN)	Stiffness (N/m)
Experiments	#1	161.51	1.2376×10^6	#1	94.81	1.1897×10^6
	#2	163.10	1.2393×10^6	#2	96.79	1.2072×10^6
	#3	164.89	1.2513×10^6	#3	101.56	1.2270×10^6
	#4	160.28	1.2084×10^6	#4	97.07	1.1874×10^6
	Average	162.45	1.2342×10^6	Average	97.56	1.2028×10^6
	Coefficient of variation	1.06%	1.28%	Coefficient of variation	2.14%	1.45%
Simulations	—	157.19	1.2670×10^6	—	88.77	1.1665×10^6
Difference	—	3.24%	2.66%	—	9.01%	3.02%

Note: Stiffness = $\frac{P_2 - P_1}{\delta_2 - \delta_1}$ ($P_2 = 90\%P_{\max}$, $P_1 = 10\%P_{\max}$; $\delta_2 = 90\%\delta_{\max}$, $\delta_1 = 10\%\delta_{\max}$)

5.2.5 Numerical analysis of impacts and CAI at different locations and energies

Using numerical analysis methods, impacts with different energy levels are conducted at two distinct positions, and the corresponding CAI is calculated. Fig.27 illustrates the two impact positions: Impact Position A, as previously described, is located at the center of the thin section of the laminate, while Impact Position B is situated at the junction between the thin section and the tapered section.

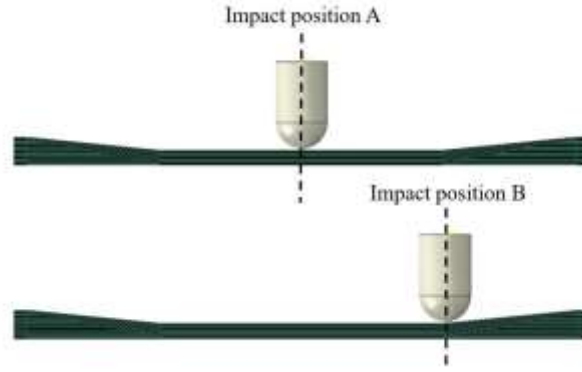


Fig. 27 Schematic diagram of two impact positions A and B.

As shown in the Fig.28, the impact response curves for Impact Position A and Impact Position B are presented for impact energies of 4J, 8J, and 12J. With the increase in impact energy, both the maximum impact force and maximum displacement at different impact positions exhibit an increasing trend. Notably, at the same impact energy, the maximum impact force at Position B is higher, while the impact displacement is smaller. This is attributed to the increased stiffness resulting from the additional plies in the tapered section of the laminate, which alters the impact response.

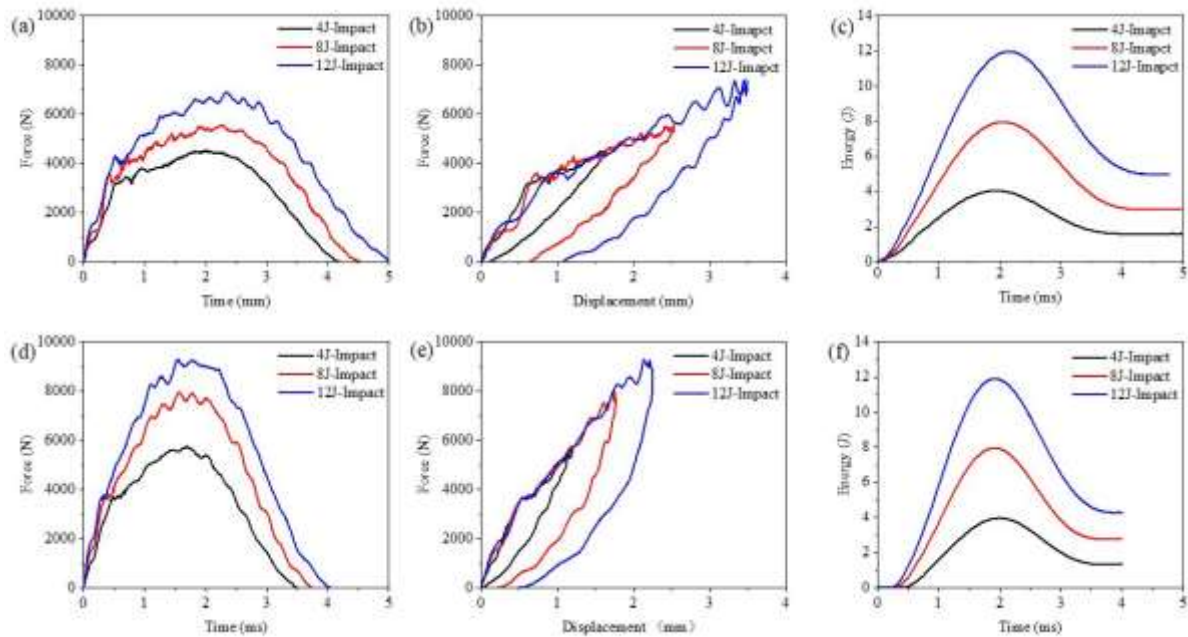


Fig. 28 Impact response at two different positions from the simulation: (a) force-time curve of position A, (b) force-displacement curve of position A, (c) absorbed energy-time curve of position A, (d) force-time curve of position B, (e) force-displacement curve of position B, (f) absorbed energy-time curve of position B.

Fig. 29 shows the load-displacement curves for CAI. It can be observed that when the impact position is at the center of the thin section, the failure load is significantly affected by the impact energy. Specifically, the CAI failure load decreases by 23.46%, 39.45%, and 56.91% for impact energies of 4 J, 8 J, and 12 J, respectively. In contrast, when the impact position is at the junction between the thin section and the tapered section, even a lower impact energy of 4 J results in a 35.05% reduction in the CAI failure load. For impact energies of 8 J and 12 J, the

CAI failure load decreases by 39.96% and 44.65%, respectively. This may be due to the stress concentration at the junction under compressive loads, making it more sensitive to impact damage, while the magnitude of the impact energy has a lesser effect.

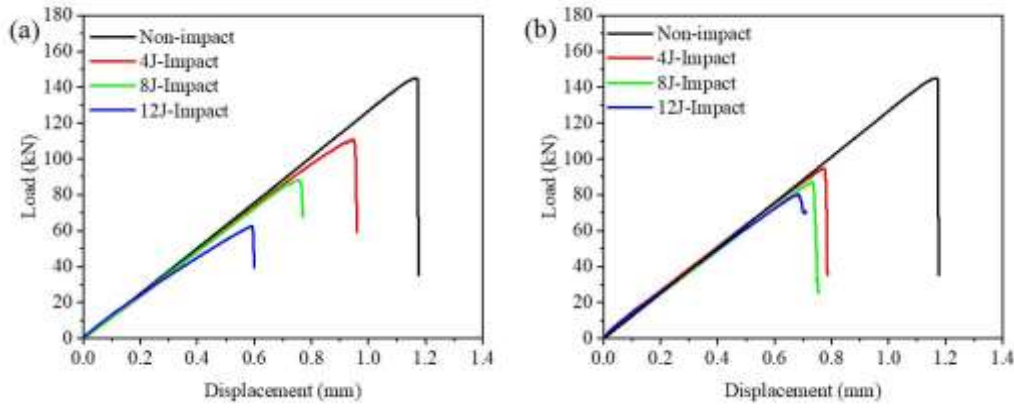


Fig. 29 Load-displacement curves of CAI from simulations: (a) impact position A, (b) impact position B.

6 Conclusion

The study investigates the mechanical behavior and damage mechanisms of asymmetric tapered composite laminates under LVI, compression, and CAI through experimental and simulation methods. The focus is on comparing laminate performance under compression and CAI. Strain gauges measure strain at different positions, while ultrasonic C-scan and CT scanning assess internal damage. Explicit FEA with continuous damage models and cohesive elements predicts laminate strength and damage, which are compared to experimental results. The main results are as follows:

- For the compression test, the structural asymmetry of the laminate causes unique out-of-plane displacement due to the coupling of bending moment and axial load. In the CAI test, prior to failure, the out-of-plane displacement is minimally affected by the 8J LVI damage and is primarily dominated by structural asymmetry.
- For the compression test, axial stress concentrates at the junction between the thin and tapered sections. Resin pockets cause localized stress peaks, but their impact is less significant than structural features. Interlaminar shear and normal stresses also peak at the junction, with resin pockets near the junction exhibiting particularly high stress peaks. For the CAI test, delamination, matrix cracking, and sub-laminate buckling caused by LVI result in axial stress peaks in the impact-damaged area, with some layers experiencing higher stress peaks than at the junction. Interlaminar shear and normal stresses also peak in the impact area. CAI failure is mainly driven by delamination and matrix cracking caused by LVI, with sub-laminate buckling and fracture under compressive load leading to reduced strength.
- From the failure results obtained through experiments and simulations: For the compression test, the sub-laminates at the fracture sites are almost entirely broken. Due to bending and shear stress, the ends of the broken sub-laminates are primarily blunt and form wedge-shaped bands. Delamination is minimal but remains

a failure mode. For the CAI test, surface 45° plies splitting cracks caused by LVI extend under compressive loads, causing 45° strips to buckle out-of-plane. CT images show clear wedge-shaped bands at the ends of the broken sub-laminates, which spread out the layers when entering delamination.

- LVI damage at the center of the thin section significantly alters the compression failure location, shifting it from the junction of the thin and tapered sections to the sides of the impact point. The impact reduces compressive strength from 722.01 MPa to 394.53 MPa (approximately 45.36%) but has a minimal effect on stiffness, which decreases by only about 2.54%. This underscores the importance of considering the damage tolerance of asymmetric tapered laminates.

The out-of-plane displacement, stress distribution, and failure mechanisms of asymmetric tapered laminates under compressive loads are significantly influenced by their structural characteristics. LVI damage in the thin section notably alters the mechanical behavior compared to intact laminates. The FE model established in this study effectively captures the failure modes of compression and CAI, predicting the corresponding compressive strength. This provides valuable insights for the future application of asymmetric tapered laminates.

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