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Stability and Failure of the Edge-Closed Honeycomb Sandwich Panels with Face/Core Debonding

Bing Yan, Xinfeng Wang *, Song Pan, Mingbo Tong *, Jian Yu and Fan Liu

Department of Aircraft Design, Nanjing University of Aeronautics and Astronautics, Nanjing 210016, China; yanbing@nuaa.edu.cn (B.Y.); pansong123@nuaa.edu.cn (S.P.); ianyujian@nuaa.edu.cn (J.Y.); linj@nuaa.edu.cn (F.L.)

* Correspondence: xinfengw@nuaa.edu.cn (X.W.); tongw@nuaa.edu.cn (M.T.)

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Abstract: This study investigated the mechanical performance of the edge-closed honeycomb sandwich structure with face/core debonding under compressive load by experimental and numerical methods. Uniaxial compression tests of asymmetric sandwich structures with various debonding sizes between the carbon/epoxy face sheets and the honeycomb core were conducted. The experimental results showed that the failure of debonding specimens was mainly caused by the local buckling of face sheets at the debonding area. The failure zone of sandwich structures gradually translated from the edge-closed beveled area to the debonding area with the increase of debonding sizes correspondingly. Meanwhile, the stability and the load carrying capacity of sandwich structures were in a downtrend. Compared with the nondestructive specimens, the residual strength of two kinds of defective specimens decreased by 9.9% and 22.1%, respectively. The simulation of the developed numerical model based on the linear buckling theory and the continuum damage mechanics agreed well with the experimental data. The study has guide to assess on the damage tolerance of edge-closed honeycomb sandwich panels with face/core debonding.

Keywords: honeycomb sandwich; edge-closed; debonding; buckling; failure analysis

1. Introduction

Honeycomb sandwich composite is one of the most effective material structures applied to improve the bending stiffness and reduce the weight of the structure. In recent years, it has been widely used in aerospace aircraft structures such as the fuselage, the wings, the rudders and so on [1]. For instance, the proportions of honeycomb sandwich used in B-58 high-altitude bombers and F-111 fighters as high as 85% and 90%, respectively [2]. However, defects of honeycomb sandwich structures, such as cracks, debonding, core crush, and laminate delamination, etc., which generally caused by alternating loads, external impacts, lightning strike and environment during manufacturing and using, greatly affects the mechanical properties. Among them, face/core debonding is the most universal and dangerous defect in honeycomb sandwich structures which undermines the structural integrity seriously [3].

Studies on characterizing face/core debonding of sandwich structures have been conducted. In order to measure the mode I critical fracture toughness, the most common test method adopted is single cantilever beam (SCB) sandwich specimen which has been selected as the candidate of ASTM International test standard [4]. The SCB test method is easy to perform; Saseendran et al. [5] used this test method to measure mode I critical fracture toughness of various sandwich configurations, and concluded that the fracture toughness increased with the rise of core density, face-sheet thickness, and core cell size. Saseendran et al. [6] also adopted this method to examine the effect of face and core moduli and thickness on crack rotation in SCB sandwich specimen. For the measurement of

mode II critical fracture toughness, the Cracked Sandwich Beam (CSB) test [7] and End-Notched Flexure (ENF) test [8] are the common test methods. As for the test method under mix-mode condition, the Mixed Mode Bending (MMB) test is used [9,10], but the possible range of mode mixity phase angle in a MMB test is limited [11]. Therefore, Saseendran and Berggreen [11] adopted the Double Cantilever Beam–Uneven Bending Moments (DCB-UBM) test method to overcome the limitation of MMB test method.

Compressive strength is seriously affected by face/core debonding of sandwich structures [12]. The comprehensive experimental study on compression failure of sandwich structures with face/core debonding was firstly carried out by Vadakke and Carlsson [13]. The results showed that failure of sandwich structures was mainly caused by buckling of debonding face sheet, and followed by rapid propagation of the debonding. Test results given by Aviles and Carlsson [14] showed that panels with square debonding failed at lower loads than those with circular debonding under in-plane compression, and the interface (core/resin) failure increased with the core density increased. Wenzhi et al. [15] studied the edgewise compressive strength of Carbon Fiber Reinforced Plastic (CFRP) sandwich structure with debonding area of different shapes and sizes, which showed that the size of the rectangular debonding defect had a slight effect on edgewise compressive strength, and the larger area of the elliptical debonding caused edgewise compressive strength to decrease by 18%. Nieh et al. [16] studied the effects of panel thickness and debonding length on the compressive strength and failure mechanism of composite sandwich panels through numerical simulation and experimental methods. The results showed that specimens of thicker sheets and smaller debonding area had higher buckling loads and failure strength. Pietrek and Horst [17] demonstrated that the failure load and debonding behavior of asymmetric sandwich structures with aluminum sheets were related to the size and depth of delamination by in-plane compressive tests. In terms of simulation, the buckling behavior of sandwich structures was developed by Frostig and Sokolinsky [18] based on high-order sandwich theory. The propagation behavior of interface debonding and core crack was predicted by El-Sayed and Sridharan [19] by a model with cohesive elements. Berggreen and Simonsen [20] developed a 3D model based on the finite element method to predict the residual strength of debonded sandwich panels. Based on finite element analysis and linear elastic fracture mechanics, Moslemian et al. [21] studied the initial buckling load and compressive strength of a delaminated sandwich panel with a circular core. Sayyidmousavi et al. [22] used a new 3D finite element modeling method to study the effects of face/core debonding size, shape, aspect ratio, fiber direction, and core stiffness on the buckling behavior of honeycomb panels under different boundary conditions.

The objects of the above studies are mostly for sandwich structures of the element level and constant thickness. However, the edges of the honeycomb sandwich structure are usually beveled and closed [23], which is easy to connect and transfer load. Castanie et al. [24] developed a new technology, which was well known for its asymmetric sandwich structure, and it was used to design lightweight structures. Yu [2] experimentally studied the effect of debonding defects in the beveled area on the failure process when edges-closed honeycomb sandwich structure was subject to compressive loads. The results showed that the debonding defect in the beveled zone had a small effect on the compressive strength of the structure, but had a large effect on the strain of the inner panel. Pan et al. [25] studied the influence of plate/core debonding defects on the buckling load and instability mode of edge-closed honeycomb panels by finite element method. Deng et al. [26] studied the stability and failure behavior of asymmetric sandwich structures under asymmetric compressive load through experiments and finite element analysis.

However, there was less research on the edge-closed honeycomb sandwich structure with debonding defects in the public literature. In this paper, experimental and simulation studies on the mechanical properties under axial compression load of the edge-closed honeycomb sandwich structure were developed. Two sizes of face/core debonding area were designed according to variation law of buckling load calculated by finite element analysis, and corresponding compression tests were designed. Then, the buckling load, ultimate load and failure mode of edge-closed honeycomb structures with

different debonding sizes under axial compressive load were obtained. The results of experiments were in good agreement with the results of the progressive failure model of face/core debonding composite sandwich structures established by combining cohesive zone model (CZM) technique and progressive failure analysis. The present study had guiding significance for the assessment on damage tolerance of edge-closed honeycomb sandwich panels with face/core debonding.

2. Experimental Procedures

2.1. Subsection

The specimens designed in this paper were divided into non-defective samples and samples with face/core debonding, which were consist of skin, honeycomb core and adhesive. The sheets were stacked by plain carbon fabric prepreg Cycom970/T300, provided by Cytec Industries Inc. (Paterson in United States). The thickness of prepreg was 0.216 mm and mechanical parameters are given in Table 1. Nomex honeycomb JY1-3.2–48 manufactured by Jiangsu Junyuan New Material Co., Ltd. (Nantong, China). was used in this study, and its equivalent mechanical properties [27] were given in Table 2. TS-JM-200 was used to adhere face-sheet onto core, and its mechanical properties are given in Table 3. All specimens were produced in autoclave.

Table 1. The mechanical parameters of plain carbon fabric prepreg.

E_1 (GPa)	E_2 (GPa)	G_{12} (GPa)	ν_{12}	$X_T = X_C$ (MPa)	$Y_T = Y_C$ (MPa)	S_{12} (MPa)
49	49	3.6	0.044	676.2	615.5	261.5

Table 2. Equivalent mechanical properties of Nomex honeycomb.

E_1 (MPa)	E_2 (MPa)	E_3 (MPa)	G_{12} (MPa)	G_{13} (MPa)	G_{23} (MPa)	ν_{12}	$\nu_{13} = \nu_{23}$
0.1953	0.1953	136	0.073	40	23	0.94	0.0003
X_{TT} (MPa)		S_{LT} (MPa)			S_{WT} (MPa)		
1.93		1.26			0.62		

Table 3. Mechanical properties of adhesive.

E (GPa)	G (GPa)	ν	t_n (MPa)	$t_s = t_t$ (MPa)	G_n (N·mm ⁻¹)	$G_s = G_t$ (N·mm ⁻¹)
3.2	1.23	0.3	58	65	0.744	3.816

The basic configuration of the two type of test specimens is shown in Figure 1. The flat panel of specimens was the outer skin, and the convex panel was the inner skin. The square area between the inner and outer skins was the honeycomb core. The design bases [28] of specimens are summarized in Table 4. The stacking sequence of sheets in honeycomb area from outside to inside was [(45/0)₂/45/C₁₀/45/0/45], and that of edge-closed area from outside to inside was [(45/0)₂/45/(0/45)_s/45/0/45]. More layers and reinforced plates were laid in the closed area to prevent the edge-closed area from destroy in advance.

Table 4. The design basis of specimens.

Design Parameters	Reference
The minimum number of skin layers is 2	Prevent moisture penetration and meet the minimum structural strength requirements
The number of skin layers reaches 5	Avoid damage caused by low energy impact
The empirical height value of Non-metal core is about 12.7 mm	Provides appropriate shear strength and stiffness
±45 ply	Enhanced in-plane shear
0/90 ply	Strengthening structural bending
Beveling angle is generally 20~30°	Minimize the bonding pressure on the inclined surface

Figure 1. The configuration of specimens.

Compared with non-defective specimens, specimens with debonding defects form debonding areas by prefabricating a polytetrafluoroethylene film with a diameter D between the outer skin and the honeycomb core. For selecting the representative debonding size D , ABAQUS was used to analyze the influence of eight different debonding sizes between 10 mm and 80 mm on the buckling load of honeycomb structure, as shown in Figure 2. According to the change law of the first-order buckling load of the structure with the debonding size, the debonding diameter with a large load change was finally selected, 40 mm and 80 mm respectively.

Figure 2. The influence of debonding sizes on buckling load.

The test matrix in this paper is shown in Table 5. For research purposes, all specimens were numbered as CDX-Y. C represented the compressive load, DX represented the debonding size, and Y represented the serial number of various samples. For example, the number CD0-1 represents the first non-defective compressive sample.

Table 5. Test matrix.

Type	Defective Diameter (mm)	Number of Specimens
Non-defective specimens	0	3
Debonding specimens	40	3
	80	3

2.2. Test Procedure

Back-to-back resistance strain gages (gage numbering was BE 120-3BC which was provided by Zhonghang Electronic Measuring Instruments Co., Ltd.) were arranged and grouped on both sides of the compressive specimens. Only longitudinal strain was required in result processing, so the longitudinal strain gages are drawn in Figure 3. For specimens with defects, all gages were divided into four group. (1) Group one: the resistance strain gages stuck at the center of the beveled area near the edge-closed area longitudinally (numbered 8 and 14 on inner skin, 1 and 7 on outer skin);

(2) group two: the resistance strain gages set on the intersection points of the horizontal and vertical 1/4 lines of the honeycomb area (numbered 2, 3, 5 and 6 on inner skin, 9, 10, 12 and 13 on outer skin); (3) group three: the resistance strain gages at the center of the specimens (numbered 4 and 11 on inner and outer skin respectively); (4) group four: the resistance strain gages arranged on the intersection of the debonding boundary and the centerline of the honeycomb zone (numbered 15 and 17 on inner skin, 16 and 18 on outer skin). The fourth group of strain gages were not arranged in non-defective compressive specimens.

Figure 3. Strain arrangement on both sides of specimens.

The INSTRON static hydraulic testing system was used as the loading equipment for this compressive test. The maximum applying force of this machine was up to 1500 kN. The JM3813 data acquisition system was used to measure strain of specimens. The size of the loading ends of specimens exceeded the width of loading heads of testing machine, so it was necessary to design a suitable fixture to support the ends of the specimens. The structure of the fixture is shown in Figure 4. The U-shaped loading ends of fixture were connected to the loading end and the fixed end of the specimen by bolts. In order to avoid the direct contact between the bolts and specimen, which could cause the surface of sample to be damaged, cushion blocks were added between the U-shaped loading ends and the sample. The columns on both sides were used to prevent specimens from being destabilized during loading and causing specimens to fail in advance. The displacement control method was used during test, and the loading rate was set to 0.5 mm/min. The strain was measured every 5 kN until the specimen was failed. Finally, load-displacement curve was output, and the maximum load value was recorded.

Figure 4. Fixture of tests.

3. Test Results and Discussion

3.1. Strain Distribution

As the quantity and variety of specimens were too large, only strain measurement results of typical specimens of which experimental phenomena were easy and clear to be observed were selected for analysis in this section, and the trends of the strain-load curves of other samples were basically consistent. As shown in Figures 5–7, the strain values of the inner and outer skin remained the same only at the initial stage of loading, this was attributed to the beveled area closed to the edge of the specimen which led to the initial loading eccentricity. Thereafter, the strain values of the back-to-back resistance strain gages were different, which indicates that the structure was subjected to bending load due to the existence of eccentric loads. When the slope of the strain-load curves showed a reverse trend (vertical red dotted line in the figure), it indicated that the structure was buckled.

Figure 5. Typical strain-load curves of the third non-defective compressive sample (CD0-3).

Figure 6. Typical strain-load curves of the third compressive sample with defective diameter 40 mm (CD40-3).

Figure 7. Typical strain-load curves of the third compressive sample with defective diameter 80 mm (CD80-3).

The strain-load curves of the non-defective specimens (CD0-3) are shown in Figure 5. When the load reached 65 kN, the slope of strain-load curves of outside skins in the beveled zone turned to a positive value (such as 8 in Figure 5). Subsequently, the slope of the curves at other positions also showed a reverse trend (such as 3, 4 in Figure 5). The beveled area was the weak area of the edge-closed honeycomb structure, so the beveled area was generally damaged before other locations, which indicated that the structure had been damaged before the overall buckling occurred.

The strain-load curve of the test piece (CD40-3) with debonding defects is shown in Figure 6. When the load reached 60 kN, the slope of the strain-load curves of inner sheets in the debonding area turned to a positive value (such as 4, 17). The slope of the curve of the inner sheets around the honeycomb area changed from negative value to positive one (such as 3, 5) when the load continued to increase to 65 kN. Finally, the curve slope of the outer sheets in the beveled area showed a reverse trend (such as 14) when the load was 75 kN. Firstly, the sandwich panels with a defective size of 40 mm locally buckled in the debonding area, and then local debonding of the panel and the honeycomb core occurred in the beveled area near the loading end.

The strain-load curves of the test specimens (CD80-3) with debonding defects are shown in Figure 7. When the load reached 35 kN, the curve slope of the inner sheets in the center of the debonding area showed a reverse trend (such as 4). When the load continued to 50 kN, the slope of the curve of the outside sheets in the beveled area changed to positive value (such as 8). Meanwhile, the strain curves in the debonding area changed randomly (such as 15, 17). Therefore, the sandwich structure with a defect size of 80 mm firstly suffered from local buckling in the defect area at a lower load, and subsequently damage occurred in the debonding area and beveled area.

The strain-load curves of gages 4 and gages 11 in the center of the inner and outer sheets of the different type specimens are selected to draw in Figure 8, and are contrasted to discuss. For edge-closed honeycomb sandwich panels, the phenomenon of strain-load curves' slope changing from negative to

positive occurred on the inner sheets (gages 4), therefore, the inner sheets with beveled section was more prone to buckling or damage. In addition, the larger the debonding size was, the more likely the curve slope was to have inflection points of positive and negative value under low load.

Figure 8. Typical strain-load curves of specimens with different defective.

3.2. Fracture Load

Owing to the initial gaps in the fixture, specimens and loading heads of testing machine, there was a rigid body displacement at the beginning of the test, which resulted in “unreal” initial non-linearity in the load-displacement curves of all the test samples in Figure 9. Table 6 shows the ultimate loads and their average values for all test samples. It can be seen from the Table 6 that the data had little discreteness (the coefficient of variation of the results are 11.25%, 1.39%, 5.46% respectively), indicating that the manufacturing process of different specimens had good consistency. Comparing the results of three types tests, the ultimate loads of all specimen, especially that of CD80, decreased significantly with the increase of the debonding sizes.

Figure 9. The load-displacement curves of all types of specimens.

Table 6. The ultimate loads of various specimens.

Number	Ultimate Loads (kN)			Mean Value (kN)	CV
	1	2	3		
CD0	97.6	98.3	92.3	96.0	11.25%
CD40	86.5	85.4	87.6	86.5	1.39%
CD80	74.4	76.9	72.9	74.7	5.46%

3.3. Ultimate Failure Modes

The failure mode of the specimen CD0-3 is shown in Figure 10. Due to the additional bending moment caused by the eccentric load, the structure of the CD0-3 bulged out along the inner sheet. The honeycomb core in the beveled area was crushed and sheared, which caused the sheets to lose support and form a large area of depression. Meanwhile, fiber crushing and delamination occurred in the reinforced area. Finally, the structure failed and could not continue to carry.

Figure 10. Compressive failure mode of CD0-3. (a) Failure mode of inner skin; (b) Failure mode of outer skin.

The failure mode of the specimen CD40-3 is shown in Figure 11. The honeycomb core between debonding area and beveled area was crushed and sheared, so the inner and outer skins lost support and a large area bulged out. Meanwhile, fiber crushing and delamination occurred on the both sides of laminated area. As a result, the structure could not continue to carry.

Figure 11. Compressive failure mode of CD40-3. (a) Failure mode of inner skin; (b) Failure mode of outer skin.

The failure mode of the CD80-3 specimen is shown in Figure 12. The honeycomb core in the defect area and the beveled area was crushed and sheared due to the local buckling of the specimens. The inner skin of the defect area was dented and the outer skin bulged. Damage continued to accumulate and the structure could not continue to carry. Unlike the failure modes of CD0 and CD40 specimens, no fiber breakage and delamination were found in the inner and outer skins on both sides of the lamination zone during the entire test.

Figure 12. Compressive failure mode of CD80-3. (a) Failure mode of inner skin; (b) Failure mode of outer skin.

4. Progressive Damage Model

4.1. Failure Theory of Honeycomb Sandwich Panels

Progressive Failure Analysis (PFA) is a computational method based on the assumption that damaged materials can continue to bear the load according to their degraded properties [29]. It is widely used in strength prediction and damage propagation analysis of composite structures. For sandwich structures under compression, damage may initiate in face-sheets, in the core and at the interface between sheets and core. Therefore, the progressive failure model of face/core debonding composite sandwich structures was established to predict the strength and observe the failure process. The process of progressive failure analysis is shown in Figure 13.

Figure 13. The progressive failure process for composite honeycomb panels.

4.1.1. Intra-Laminar Progressive Failure

The material properties of outside the plane could be ignored because of the relatively thin thickness of composite face-sheet, and the structure was considered to be in a state of plane stress. Therefore, the two-dimensional hashin criterion [30,31] was used to simulate the panel failure in this paper. Four distinct failure modes were modeled, including fiber compressive failure (FCF), matrix compressive failure (MCF), fiber tensile failure (FTF) and matrix tensile failure (MTF). Considering the brevity of this paper, the abbreviations of four failure modes were used to introduce failure process in following sections.

4.1.2. Debonding

The cohesive zone model (CZM) technique based on damage mechanics and fracture mechanics has an irreplaceable advantage in predicting the initiation and expansion of cracks. This technique was originally proposed by Dugdale [32] and Barenblatt [33] to deal with the fracture process area at crack tips. Heretofore, CZM technique has been widely used to simulate the inter-lamina of composite laminates [34] and the adhesive of sandwich composites [5]. The conclusions in literature [5] show that the CZM approach is robust, and is able to predict debonding phenomenon of sandwich structure accurately.

Several cohesive laws exist characterizing the relationship between displacement and traction of the crack, among them, the bilinear law is widely used, and it can take into account both the precision and efficiency of calculation [35]. Therefore, bilinear constitutive model is adopted in this paper to model the adhesive.

Damage onset occurs when the following quadratic stress failure criterion is satisfied [36]

$$\left(\frac{\langle\sigma_n\rangle}{N}\right)^2 + \left(\frac{\tau_s}{S}\right)^2 + \left(\frac{\tau_t}{T}\right)^2 = 1 \quad (1)$$

where $\langle \rangle$ is the Macaulay bracket, σ_n , τ_s and τ_t represent the normal and two shear tractions, N , S and T are the normal and two shear strengths.

A fracture energy-based linear softening law is adopted to address the damage evolution of the adhesive, which gives

$$\int \sigma d\delta = G_C \quad (2)$$

where σ and δ are the effective traction and displacements at interfaces. G_c is the mixed-mode fracture energy release rate which determined by Benzeggagh and Kenane criterion [37]. As shown by Equation (3).

$$G_C = G_{IC} + (G_{IIC} - G_{IC}) \left(\frac{G_{II}}{G_I + G_{II}} \right)^\eta \quad (3)$$

where G_{IC} and G_{IIC} are the critical fracture energy release rate of mode I and mode II. η is an interaction parameter that describe the effect of mixed-mode ratio $G_{II}/(G_I + G_{II})$ of total fracture toughness G_c .

4.1.3. Honeycomb Core Progressive Failure

The honeycomb core is equivalent to an orthotropic structure by using the sandwich board theory, and the Besant criterion [38] is adopted as the damage criterion of the honeycomb core. Its expression is as follows:

$$F_{core}^2 = \left(\frac{\sigma_{TT}}{X_{TT}} \right)^n + \left(\frac{\tau_{LT}}{S_{LT}} \right)^n + \left(\frac{\tau_{WT}}{S_{WT}} \right)^n \quad (4)$$

where, σ_{TT} , σ_{LT} , σ_{WT} are the normal stress of the honeycomb core in the T direction and the shear stress in the L_T and W_T direction respectively. X_{TT} , X_{LT} , X_{WT} are the compressive strength and shear strength of the honeycomb core in the corresponding direction respectively. The power index $n = 2$. When the damage variable $F_{core} = 1$, it is considered that the honeycomb core is damaged.

The direct degradation method can efficiently and accurately simulate the evolution of the honeycomb core at various stages. The stiffness reduction method of the honeycomb core in the User-defined Material Mechanical Behavior (UMAT) subroutine is as follows:

$$\begin{aligned} E'_{11} &= \lambda_{11}E_{11}, E'_{22} = \lambda_{22}E_{22}, E'_{33} = \lambda_{33}E_{33} \\ G'_{12} &= \lambda_{12}G_{12}, G'_{13} = \lambda_{13}G_{13}, G'_{23} = \lambda_{23}G_{23} \end{aligned} \quad (5)$$

The superscript in the formula represents the stiffness after reduction. $\lambda_{ij}(i, j = 1, 2, 3)$ is the reduction factor in each direction, and the factors [39] in different damaged status are given in Table 7.

Table 7. Equivalent stiffness reduction factor of honeycomb.

Damaged Status	λ_{11}	λ_{22}	λ_{33}	λ_{12}	λ_{13}	λ_{23}
Intact	1	1	1	1	1	1
Damaged	0.4	0.4	0.45	0.4	0.4	0.4
Fold	0.2	0.2	0.2	0.2	0.2	0.2

4.2. Finite Element Model

The progressive failure model of face/core debonding composite sandwich structures is modelled in the well-known finite element (FE) software ABAQUS supported by Dassault SIMULIA (Dassault Systèmes, Vélizy-Villacoublay, France). An UMAT subroutine contained the damage initiate and propagation criterion of honeycomb core is coded to simulate the progressive failure of Honeycomb core in ABAQUS/Standard platform. Figure 14 illustrates the grid discretization and boundary conditions of the edge closed honeycomb panel with debonding defects. The 0° layering direction of composite material follows the red arrows direction in the Figure 14 (A) of the cross-sectional view A-A. The continuous shell element SC8R (green elements) was used to simulate each layer of the inner and outer skins, and the total number of elements is 7810. The three-dimensional solid element C3D8R (white elements) was used to simulate the honeycomb core, and the total number of elements is 16426. In order to accurately express the face/core debond, the cohesive element COH3D8 (red elements) was used to simulate the adhesive connecting the skin with the honeycomb core. The thickness of adhesive was set to 0.02 mm, and the total number of such elements was 7382. The failure methods in Section 4.1 were used to judge the failure of the three materials. To ensure the continuity of the displacement of the element nodes, the skin, the adhesive and the honeycomb core were connected by common nodes. No elements were established in the center of the specimen ((b) in A-A section view) which represented the debonding area between carbon panel and honeycomb, and general contact was generally established to prevent the inner and outer skins from embedding each other. The bottom of the honeycomb structure was fixed, and constraints in the z direction were set in the remaining three sides. Based on the buckle module of the ABAQUS/Standard solver, the concentrated force in the Y-direction was applied to the top of the honeycomb panel to analyze the structural buckling. At the same time, keywords in the input file were modified to output the node information of all modes. Based on the Static General module, the concentrated load was replaced by the displacement load, and the keywords of the input file were modified again. The node information obtained from the buckling analysis, as an initial defect, was introduced to the post-buckling analysis model.

Figure 14. The grid discretization and boundary conditions of the debonding model.

5. Numerical Results and Discussion

5.1. Effect of Imperfections on Stability

According to the method described by Tafreshi and Oswald [40], the instability modes of contained global buckling, mixed buckling and local buckling. As shown in Figure 15, the out-of-plane displacement of the red region was larger than that of the blue region. When D was 0, the instability mode of the defect-free model was global buckling. When D is 40 mm or 80 mm, the local buckling of the panel occurred in the debonding area. The larger the proportion of the structure occupied by the defect was, the more obvious the local instability mode of the structure was. It is worth noting that the small damages or defects occurring in specimens during the molding, manufacturing, and transportation processes may lead to asymmetry in the structure during the test process, and these factors were not considered in the linear buckling analysis.

Figure 15. Buckling modes of different honeycomb panels. (a) the first-order and second-order buckling modes of non-defective specimens; (b) the first-order and second-order buckling modes of specimens with defective diameter 40 mm; (c) the first-order and second-order buckling modes of specimens with defective diameter 80 mm.

As given in Table 8, the experimental results of the first-order buckling load of the two types of defective honeycomb panels were basically consistent with the simulated values. However, the simulated value of the defect-free honeycomb panel was higher than its failure load, this is because the material failure was not considered during the linear buckling analysis. The sloped area was just a weak link of the edge closed structure, so before the global buckling of the structure occurred, the sloped area was damaged, which resulted in losing bearing capacity in advance. Overall, further nonlinear post-buckling analysis is required in simulation in order to accurately assess the final bearing capacity of the structure.

Table 8. Buckling loads of various honeycomb panels.

Number	Mean Value of Experimental Results (kN)	Simulated Value (kN)	Error/%
CD0	65	108	66.2
CD40	60	68	13.3
CD80	35	32	−8.6

5.2. Effect of Imperfections on Stability

The particularity of the material molding process made the structure itself have certain initial defects. After the buckling deformation, the structure showed obvious nonlinearity. Therefore, the nonlinear analysis of sandwich panels needed to introduce initial geometric defects to accurately obtain the final failure load of the structure. Considering that the first-order buckling mode and buckling load obtained by simulation were close to the test results, the node displacement of first-order buckling was introduced into the post-buckling model of the honeycomb panel. As given in Table 9, the ultimate loads of the honeycomb panel obtained by the numerical analysis method in this paper were in good agreement with the experimental values, and the maximum relative error did not exceed 10%. Compared with CD0, the carrying capacities of the CD40 and CD80 structures decreased by 9.9% and 22.1%, respectively.

Table 9. Fracture loads of various honeycomb panels.

Number	Mean Value of Experimental Results (kN)	Simulated Value (kN)	Error/%
CD0	96.0	90.7	−5.5
CD40	86.5	82.2	−4.9
CD80	74.7	69.3	−7.2

5.3. Effect of Imperfections on Honeycomb Failure

As shown in Figure 16, the red area represents failed elements, and the blue represents non-failed elements. In the model CD0, the additional bending moment generated by the asymmetric load caused the structure to buckle along the inner skin. Observed from the sides of inner skin or outer skin, the honeycomb damage of the sloped area that was the weak link appeared first. As the buckling displacement of the structure increased, the damage expanded from the periphery to the center of the structure. Compared to the inner skin, the outer skin was closer to the loading plane of the edge closed structure, which resulted in more serious damage.

As shown in Figure 17, the sloped area and the debonding area were the weak areas of CD40 model. Because the diameter of the debonding area was still small, the honeycomb damage in the defect area did not appear before that of the sloped area. Observed from the sides of inner skin or outer skin, the honeycomb in the sloped area bore carrying more stress relatively, so damage appeared there first. As the degree of buckling increased, the damage expanded toward the center of the structure.

Figure 16. Honeycomb damage propagation in the CD0 Model. (a) the damage propagation of inner skin; (b) the damage propagation of outer skin.

Figure 17. Honeycomb damage propagation in the CD40 Model. (a) the damage propagation of inner skin; (b) the damage propagation of outer skin.

Owing to the large debonding diameter in the model CD80, local buckling appeared obviously in the debonding area of the skin, as shown in Figure 18. Observed from the inner skin, the honeycomb at the edge of the defect area carried more stress relatively, so damage appeared there first. As the degree of buckling increased, the damage extends to the sloped area in the vertical loading direction.

Observed from the outer skin, the honeycomb of the inclined area appeared damaged first, and then the damage expanded along the vertical loading direction.

Figure 18. Honeycomb damage propagation in the CD80 Model. (a) the damage propagation of inner skin; (b) the damage propagation of outer skin.

5.4. Effect of Imperfections on Skin Failure

As shown in Figures 19–21, the clouds of damage modes and damage states of the skins were obtained by different angles and various types of models. In the model CD0 and CD40, the honeycomb in bevel area was subject to more stress under the compressive load, which resulted in its damage at first, but at this moment, the composite skin was not damaged due to its high strength. As the degree of buckling increased, the damage area of the honeycomb core gradually expanded. After the support of the honeycomb core was lost between inner and outer skins, the bending stiffness of the structure suddenly decreased, and the local stress and strain were too large, which caused the damage of fibers and matrix. As the damage of the honeycomb core and the composite skin expanded, the structure lost its bearing capacity. Because the skin of the model CD80 did not fail during loading, there was no failure cloud.

Figure 19. Failure modes of inner skin in the CD0 Model. (a) Matrix compressive failure (MCF); (b) Fiber compressive failure (FCF); (c) Matrix tensile failure (MTF); (d) Fiber tensile failure (FTF).

Figure 20. Failure modes of outer skin in the CD0 Model. (a) Matrix compressive failure (MCF); (b) Fiber compressive failure (FCF); (c) Fiber tensile failure (FTF); (d) Matrix tensile failure (MTF).

Figure 21. Failure modes of outer skin in the CD40 Model. (a) Fiber compressive failure (FCF); (b) Matrix compressive failure (MCF).

6. Conclusions

Finite element analysis and experimental verification are applied in this paper to study the failure behavior of edge-closed honeycomb panels with non-defect and debonding defects. Significant conclusions from the study are three-fold:

- (1) For specimens with face/core debond, the buckling load, ultimate load and failure mode obtained by finite element analysis are in accordance with the test results, which verifies the reliability of the finite element model and failure theory of honeycomb panels.
- (2) Edge-closed honeycomb panels with non-defect usually occur panel damage or honeycomb damage in the sloped area and closed area. However, for the ones with face/core debond (e.g., CD40), the debonding area becomes the main damaged area. When the debonding diameter is larger (e.g., CD80), the serious local buckling causes that the structure loses its bearing capacity without obvious damage to the panel.
- (3) With the increase of the defective size, the stability and bearing capacity of the structure show a downward trend. It can be seen that the interface debonding defect will seriously affect the mechanical properties of edge-closed composite honeycomb panels, so appropriate repair methods should be adopted in time to repair the defective structure.

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