A FINITE ELEMENT ANALYSIS OF CRITICAL BUCKLING LOAD OF COMPOSITE PLATE AFTER LOW VELOCITY IMPACT

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ABSTRACT

Composite is a material formed from two or more materials that macroscopically alloy into one material. Nowadays, composite has been generally applied as lightweight structure of aircraft. This is due to the fact that composites have high strength-to-weight ratio. It means the composites have the capability to take on various loads, despite their lightweight property. Laminate composite is one type of composites that has been generally used in aircraft industries. This type of composite is susceptible to low-velocity impact induced damage. This type of damage can happen in manufacture, operation, or even in maintenance. Low-velocity impact could cause delamination. Delamination happens when the plies of laminated composites separate at the interface of the plies. This type of damage is categorized as barely visible damage, means that the damage could not be detected with visual inspection. Special method and tool would be needed to detect the damage. Delamination will decrease the strength of the laminated composite. Delamination can be predicted with numerical simulation analysis. With increasing capability of computer, it is possible to predict the delamination and buckling of laminated composite plate. This research presents the comparisons of buckling analysis results on laminated plate composite and damaged laminated plate composite. By the result of LVI simulation, it is shown that low velocity impact of 19.3 Joule causing 6398 mm² C-Scan delamination area inside the laminated composite. The delamination causes structural instability that will affect buckling resistance of the plate. The result of analysis shows that the existence of delamination inside laminate composite will lower its critical buckling load up to 90% of undamaged laminate's critical buckling load.

Keywords: *composite, laminate, delamination, buckling.*

1 Introduction

Composite has been widely used as material of lightweight structures in aerospace industries. Composite refers to the materials that consist of two or more different materials that alloy macroscopically. By macroscopically, it means the properties of each materials can still be seen and is distinguishable from one another. Generally, composite consists of fiber and matrix. Fiber acts as the reinforcement and the main strength provider of composites. Matrix acts as protection of the fibers and substance that glues the fibers in composite. Composite has been used as a substitute of metals in several parts in aircraft. It is due to composite's high strength to weight ratio which means the composite has strength that almost equal to metal's strength, with far lighter weight than metal's.

Like all lightweight structures, laminated composite structures are susceptible to buckling failure. Buckling is a form of structural instability that causes change on geometry of the structure. Buckling is usually caused by compression load but not limited to compression load only. In some cases, tension load and shear load can induce buckling phenomenon too.

Buckling can be affected by existence of imperfection in composite. One of the imperfection in composite is delamination which is caused by low velocity impact (LVI). LVI is defined as impact with speed of impactor at 1 to 10 m/s, and the response of structure is categorized as quasi-static (Richardson & Wisheart, 1996). LVI can happen at manufacture, operation, or maintenance of structure. The damage caused by LVI, in form of delamination, is categorized as barely visible damage. Delamination cannot be detected by visual inspection. It needs special method and tool to assess the damage. The existence of delamination lowers the strength of structure which includes buckling resistance and leads to easily-buckle structure.

Buckling on composite is still an interesting research topic. Many researches have been done to study buckling on composite. Juhász and Szekrényes (2015) has simulated buckling on composite with delamination and straight crack front. The damaged composite exposed to uniaxial compression load. The results shows that the delaminated plies buckles as plate, not as beam/column.

Prabhakaran et al. (2016) was analyzed the response of laminated composite exposed to tension, compression, and buckling load. The geometry of laminated composite was varied, and the simulations were done in

ANSYS 15.0. It was noted that the critical buckling load was reduced with increased length-to-thickness ratio of laminated composite.

Erdem et al. (2018) experimented and simulated buckling on laminated composite. The research had been done to analyze the behavior on pre-buckling and post-buckling of laminated composite with hole. It was noted that the stress concentration happened in hole edges, while the stress was small near free edges. The critical buckling load reduced with increased its hole diameter.

This research provides an analysis of buckling modes in undamaged laminate composite compared to modes in delaminated composite. The delamination is formed by LVI phenomenon, and the output of LVI will be used directly as input in buckling simulation. The composite that will be analyzed in this research is unidirectional composite HEXPLY AS4/8552, with ply stacking configuration $[-45^0/0^0/45^0/90^0]s$, exposed to LVI load with 19.3 Joule impact energy. The buckling modes will be compared one another, so the effect of LVI induced delamination on buckling failure in laminated composite can be concluded. The objectives of this research by simulation are to assess the damage in the form of delamination inside the laminated composite after low velocity impact, and determine the reduction of critical buckling load of laminate composite, compared to critical buckling load of undamaged composite, by the result of FEM simulation.

2 Methodology
2.1. Simulation Data

Material properties for HEXPLY AS4/8552 was taken from NCAMP Wichita lab test. Damage evolution

properties was provided by Gonzalez et. al. (2012) in their research about low velocity impact on composite. The properties will be inputted as material for continuum shell element with Hashin-Rotem model. Properties for composite's model are shown in Table 2-1.

2.2. Model

There are two stages of simulation for this research. The first simulation is LVI model. LVI model designed with reference to ASTM D7136 for measuring damage resistance of composite exposed to impact damage.
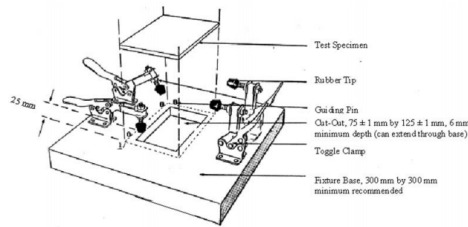

Figure 2-1: Impact damage testing on composite configuration

from ASTM D7136 (ASTM D7136, 2012)

The laminate composite model has 150 mm length and 100 mm width, with 5.8 mm thickness. The baseplate will follow the laminate's dimension, with 125x75 mm^2 rectangle shaped cutout in the middle. The baseplate, impactor, and rubber pins will be modeled as analytical rigid body. A 5 kilogram mass will be applied in the center of gravity of the impactor. The impact energy for this research is 19.3 Joule which translates to 2.779 m/s impactor speed.

The laminaes is modelled with continuum shells element, and the interfaces between the laminaes is modelled with cohesive element. There are 7 laminaes and 6 interfaces in the composite model. Mesh size for the model is 1.25 × 1.25 mm^2, considering the computation time and accuracy of simulation results. LVI model can be seen in Figure 2-2 and Figure 2-3.

Table 2-1: HEXPLY AS4/8552 Material Properties

|	HexplyAS4/8552	Value	Unit
Hashin Damage	Xt (Fiber tension strength)	2063.05	MPa
	Xc (Fiber compress strength)	1484.37	MPa
	Yt (Transverse tension str)	63.91	MPa
	Yc (Transverse compress str)	267.86	MPa
	Tau12 (In-plane shear str)	91.56	MPa
	Tau23 (Transverse shear str)	133.93	MPa
Damage Evolution	**Fracture toughness (Gc)**		
	GcLT (Longitudinal tensile)	81.5	N/mm
	GcLC (Longitudinal compress)	106.3	N/mm
	GcTT (Transversal tensile)	0.28	N/mm
	GcTC (Transversal compress)	0.79	N/mm
Elastic	E1 (Fiber direction stiffness)	131610	MPa
	E2 (Transverse fiber stiffness)	9238.98	MPa
	Niu12 (In-plane Poisson's ratio)	0.302	
	G12 (In-plane shear stiffness)	4826.33	MPa
	G23 (Transverse shear stiffness)	3548	MPa
	G13 (Longitudinal shear stiffness)	4826.33	MPa
Density		1.59E-09	tonne/mm^3
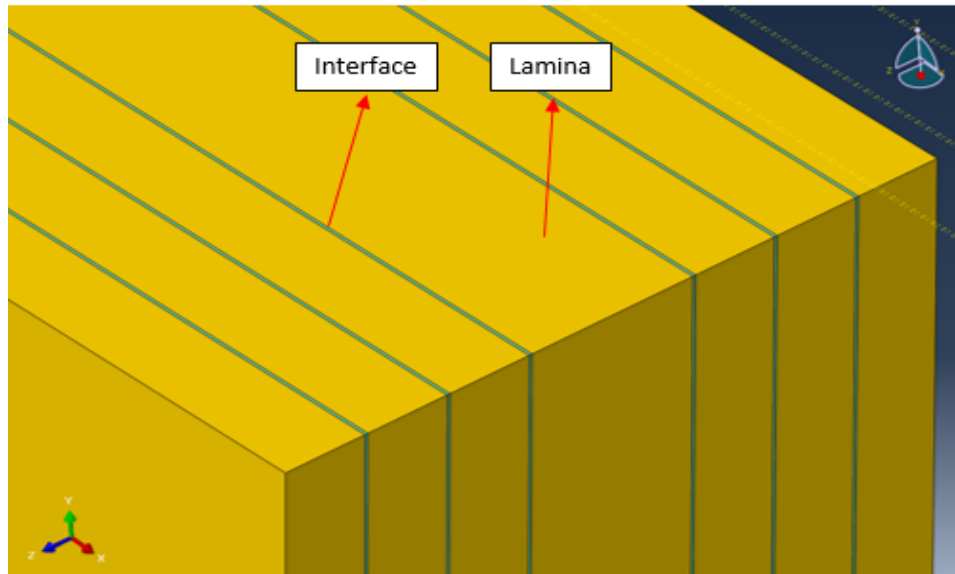
Figure 2-2: Laminaes and Interfaces in model shown by yellow elements
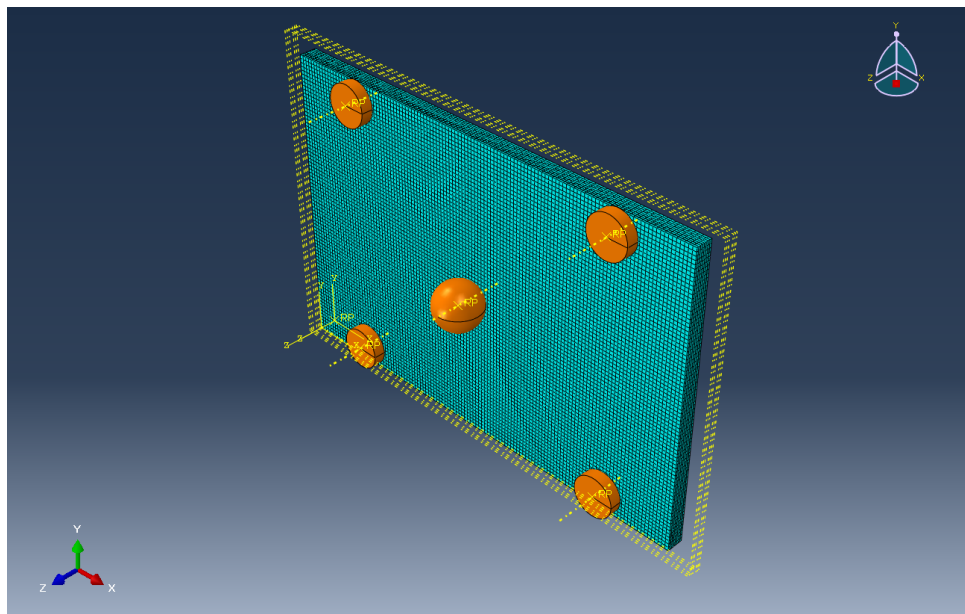and turquoise elements respectively

Figure 2-3: Model for low velocity impact on laminated composite

The laminaes will be modeled as continuum shell elements. It follows the Hashin-Rotem damage criterion as failure criterion for laminaes. The equations for Hashin-Rotem criterion is shown below,

Fiber Tension

$$(\sigma_{11} \geq 0) : F_f^t = \left(\frac{\sigma_{11}}{X^T}\right)^2 + \alpha \left(\frac{\tau_{12}}{S^L}\right)^2 \tag{2-1}$$

Fiber Compression

$$(\sigma_{11} < 0) : F_f^c = \left(\frac{\sigma_{11}}{X^C}\right)^2 \tag{2-2}$$

Matrix Tension

$$(\sigma_{22} \geq 0) : F_m^t = \left(\frac{\sigma_{22}}{Y^T}\right)^2 + \left(\frac{\tau_{12}}{S^L}\right)^2 \tag{2-3}$$

Matrix Compression

$$(\sigma_{22} < 0) : F_m^c = \left(\frac{\sigma_{22}}{2S^T}\right)^2 + \left[\left(\frac{Y^c}{2S^T}\right)^2 - 1\right]\frac{\sigma_{22}}{Y^c} + \left(\frac{\tau_{12}}{S^L}\right)^2 \quad (2\text{-}4)$$

In equations above X^T and X^c stands for tensile and compression strength of laminae in longitudinal direction. Y^T and Y^C stands for tensile and compression strength of laminae in transversal direction. S^L and S^T stands for longitudinal and transversal shear strength, while α represents shear stress coefficient and σ and τ as stress tensor. In this research, the value of α is zero, meaning it is assumed that the shear stress has no effect on fiber tension failure.

The interfaces of laminaes will be modelled as cohesive zone model. Cohesive zone model virtually models the traction-separation behavior of cohesive elements. There are two modes of separation in cohesive zone model, the peeling mode and sliding mode.
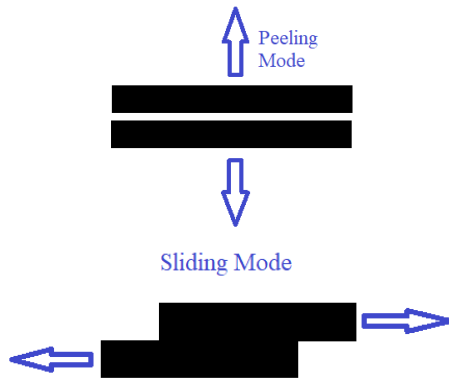

Figure 2-4: Two modes of separation in CZM

The CZM models the relation of traction and separation (Shi, 2008). The expressions for this model are,

Traction-Separation

$$\{t\} = [k]\{\varepsilon\} \quad (2\text{-}5)$$

Softening / Stiffness Reduction

$$k_i^{fail} = k_i(1 - d_i) \quad (2\text{-}6)$$

From the Eq. (2-5), ε stands for strain vectors of separation, t stands for traction vectors, and k represents stiffness matrix of cohesive zone. Before failing, the relation between traction and separation can be considered as linear relation. After the cohesive zone fails, softening will happen in cohesive zone, as shown in Eq. (2-6). The softening will causing reduction of cohesive material stiffness. This softening depends on the value of fracture energy, damage evolution factor, and damage variable *d*. if the damage factor *d* has reached 1, the cohesive zone will lost its stiffness, and can be considered as material failure. By those equations, the damage in cohesive zone can be modeled in simulation.

CZM modelling is used in this research, considering the capability of this model to capture the delamination at the interfaces of laminaes. By calculating the fiber and matrix stresses in lamina with Hashin-Rotem damage criterion and its relation to the traction-separation effect in cohesive layers, delamination profile can be predicted. This method assumes only the interlaminar shear stress and interlaminar normal stress that takes account on delamination process (Seno, 2016).

The delamination result of LVI is used as an input for buckling simulation. The simulation is performed to assess the delaminated composite behavior which is exposed to buckling load. The compression force of 1 Newton is applied to the plate. Here, the simply supported boundary condition applies to the plate edges. The model for buckling simulation is shown in Figure 2-5.

The outputs of buckling simulation of the laminate composite are the eigenvalues and the corresponding mode shape of buckling. The mode shape shows the possibility of deformation in

structure. The eigenvalues determine the value of critical buckling load. The equation to determine the value of critical buckling load is shown below,

Critical buckling load

$$P_{cr} = F \ x \ \lambda$$

(2-7)

Where P_{cr} represents the critical buckling load, F as the applied load in simulation, and λ as the eigenvalue from the simulation result.
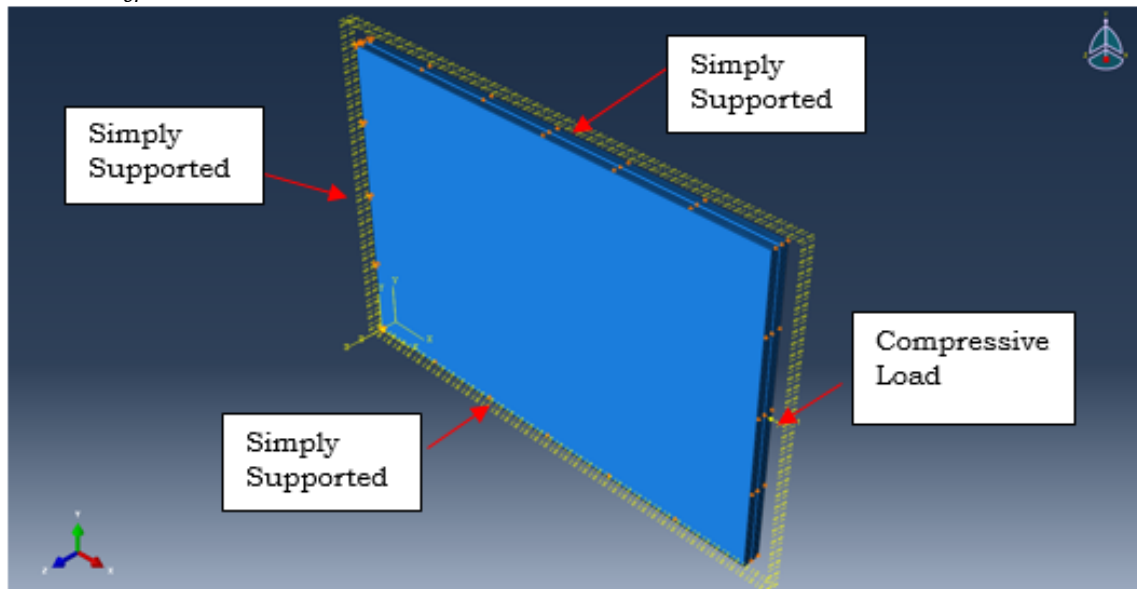

Figure 2-5: Model for buckling simulation

3   Result and Analysis

3.1 LVI Simulation Results

To validate the result of low velocity impact simulation, the energy parameters will be observed. The graph for energy parameter can be seen in Figure 3-1. The total energy ETOTAL must be relatively constant in every time step. The artificial energies such as ALLAE-Artificial Energy parameter, and ALLVD-Viscous Dissipation parameter must be smaller compared to ETOTAL. This means that in every time step, the artificial energies created by the elements in the model are tolerable. In this simulation, the ETOTAL has a constant value over time and the artificial energy parameters are small compared to total energy, as seen in graph. The result of LVI simulation can be considered valid for the given model and material property.

The result of LVI simulation is delamination profile of the plate after

impact. By the result of simulation, the delamination can be shown per interface or as C-Scan type display. The delamination profile in C-Scan type is shown in Figure 3-2. This research focuses only in the effect of delamination to buckling resistance reduction, so the damage in the laminae will not be shown.

The red area indicates the value of damage variable *d* has reached 1.0 and the interface has lost its stiffness, as stated in Eq. (2-6). This means the red area indicates the delaminated interface. Greenish boundary indicates the scalar damage is still below 1.0, and the material has experienced softening, but not yet fail. Blue area means the interface is still intact, and no delamination occurred in this area.

The damage of laminae is governed by the Hashin-Rotem criterion (Eq. (2-1)~(2-4)). It is noted that the value of stress tensors in a laminae is different

from other laminae, depends on its orientation. This causing a difference of deformation tendencies in a laminae with another, so the damage shape in every laminae is different from the others, and follows its fiber orientation.

There is a tendency for delamination in every interface to follow a peanut-like shape. This is because the delamination shape of the interfaces follows the damage in laminae below (Abrate, 1998). As stated before, every laminae has its damage shape, and the difference in deformation tendency will induce interlaminar stress between laminaes. The crack propagation as the effect of impact and interlaminar stresses will

Delamination	Value	Unit
C-Scan	6398	mm^2
Total	8194	mm^2

induce the delamination in interface, and it will follow the same stress path with laminae below, creating a peanut-like shape. The size of area depends on the value of interlaminar shear stress, interlaminar normal stress, and the difference in ply orientation.

From the delamination output, total area and C-Scan area of delamination can be known. The delamination area from the result of LVI simulation is shown in Table 3-1.

Table 3-1: Delamination Area

C-Scan area refers to area of projected delamination, meanwhile the total area refers to summation of delaminated area of every interfaces. In experiment of LVI, C-Scan area is preferable for output because of its extraction method. The total area will be more difficult to extract and need a special method and tool to extract. From the result of simulation, largest delamination profile was found in interfaces near neutral axis. The shear stress is the largest near the neutral axis. This stress plays a big role in delamination process. Large value of shear stress near neutral axis, plus interlaminar normal stress from impact will create large-sized delamination in the interfaces near neutral axis.
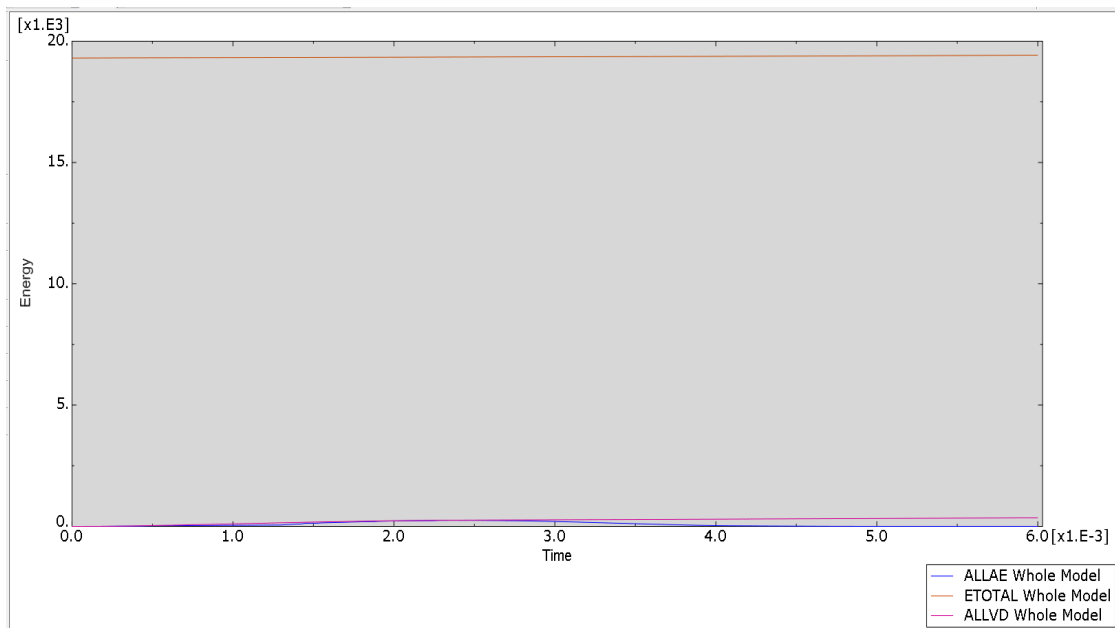

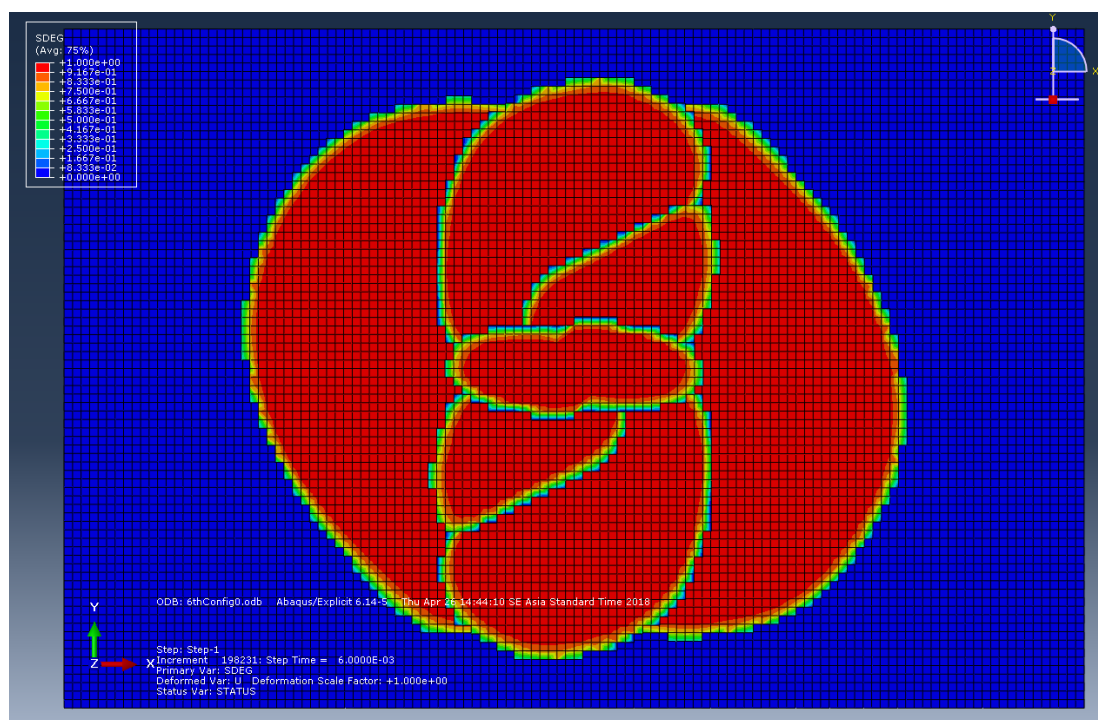Figure 3-1: Energy history graph

Figure 3-2: C-Scan of delamination

3.2 Buckling Simulation Results

The result from LVI will become input for buckling simulation. There are two simulations for buckling analysis. The first one is buckling on undamaged plate, and the second is buckling on delaminated plate, by the result of LVI simulation. As stated before, the preload for this simulation is compression load of 1 Newton.

The outputs of the simulation are eigenvalues and eigenvectors of plate.

The eigenvalues can be used to determine the value of critical buckling load in every mode by multiplying the eigenvalues with preload, as stated in Eq. (2-7). The positive eigenvalues and critical buckling load for the first three modes and their corresponding critical buckling load can be seen in Table 3-2. The graph showing the difference of P_{cr} visually for every mode is shown in Figure 3-3.

Table 3-2: Eigenvalues and critical buckling

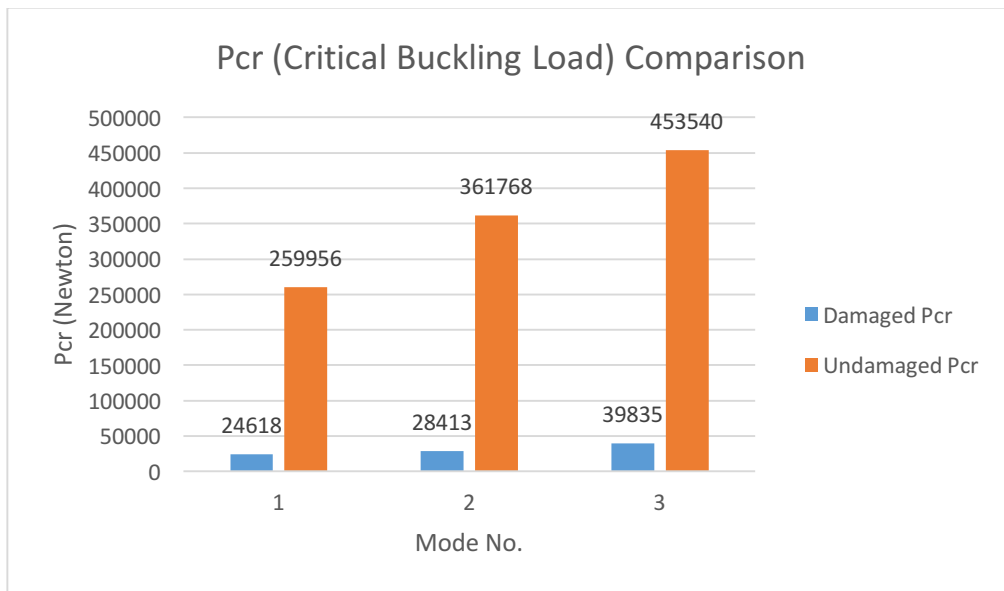| Mode	Delaminated		Undamaged	
	Eigenvalue	Critical Buckling Load	Eigenvalue	Critical Buckling Load
1	24618	C: 24618 N	259956	C: 259956 N
2	28413	C: 28413 N	361768	C: 361768 N
3	39835	C: 39835 N	453540	C: 453540 N

*C refers to compression load

Figure 3-3: Critical buckling load comparison graph

Table 3-3: Isometric and top view of plate buckling profiles

Mode	Undamaged	Damaged
1		
2		
3		

From the result of simulation, we can see that the critical buckling load is reduced in delaminated composite compared to undamaged composite. The corresponding critical buckling of mode 1 for delaminated and undamaged model are 24,618 N and 259,956 N, for both are compressive load. There is more than 90% reduction on critical buckling load in delaminated composite mode 1, compared to critical buckling of undamaged composite mode 1.

Table 3.3 is showing the result of corresponding buckling shape of every

mode. It is observed that the buckling shape for undamaged composite is more visible than the damaged one. In first glance, it looks like there is no deformation in the damaged composite. But, the deformation is actually happening inside the composite, especially in the delaminated interfaces.

The example of the buckling profile in interface can be seen in Figure 3-4. The comparison of deformation in interface between damaged and undamaged composite can be seen in Figure 3-5.
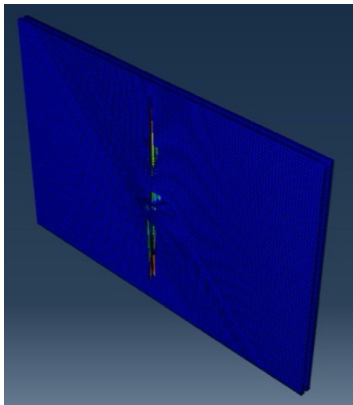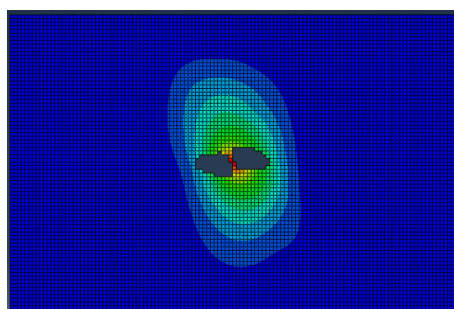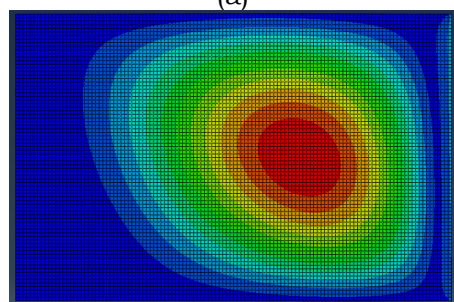

Figure 3-4: Buckling profile in delaminated interfaces for mode 1

(a)

(b)

Figure 3-5: Buckling profile in interfaces for mode 1 ((a) damaged composite and (b) undamaged composite)

In undamaged composite, the buckling shape of interface will follow the shape of whole plate. It is because there is no imperfection inside the composite, and so the laminaes and interfaces deform as one plate. While in damaged composite, the existence of imperfection makes the interfaces deform on their own, thus the stability of plate is reduced. The delamination was seen to cause instability in every interfaces. Even the slightest deformation can make the composite unstable. This causes significant reduction of critical buckling load in damaged composite.

The existence of imperfection such as delamination or void inside composite can be a source of structural instability that will lead to easily deformed structure when exposed to a certain type of load. The imperfection induces inhomogeneity inside the composite, so the interfaces and laminaes tends to deform on its. Meanwhile, the result of buckling simulation on undamaged composite shows high critical buckling load result, because of the lack of imperfection inside the structure. The unstability in undamaged composite with boundary condition similar to this research simulation will happen only when exposed to high compressive load.

4   Conclusions

This research shows that the LVI phenomenon and buckling on composite can be simulated and analyzed with finite element method. From the results of simulation and analysis, we can conclude that, LVI phenomenon can cause damage inside the laminated composite in form of delamination. In

this research, the total deformation inside composite is 8,194 mm², while the C-Scan delamination shows 6,398 mm² result area. The critical buckling load will significantly reduced with the existence of imperfection inside laminated composite - in this research it is delamination. In this research, 6398 mm² C-Scan delamination area can cause reduction of critical buckling load up to 90% of undamaged composite critical buckling value. It is because the delamination will causing structural unstability inside the composite. The instability, in form of deformation, is observed in the interfaces of delaminated composite. The undamaged composite shows high value result for critical buckling load, while the damaged composite will have its buckling resistance reduced.

Acknowledgements

Contributorship Statements

RAR developed the model for simulation, analyzed the result, and write the manuscript. MGS and HS acted as supervisor, method designer, and tools provider.

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