

IMPROVING THE STRENGTH OF LAMINATED COMPOSITE PLATES WITH
A CUTOUT USING OPTIMAL LOCAL REINFORCEMENTS

by

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ABSTRACT

IMPROVING THE STRENGTH OF LAMINATED COMPOSITE PLATES WITH A CUTOUT USING OPTIMAL LOCAL REINFORCEMENTS

A composite structure is gravely weakened if a cutout is needed for functional purposes. Instead of uniformly increasing the thickness of the whole structure, locally reinforcing the region around the notches is a much more effective way of increasing the load-bearing capacity of a notched structure. The objective of this study is to find the optimum design of reinforcing layers placed around the hole in a composite laminated plate to maximize its ultimate strength with minimum use of reinforcing material. The ultimate failure load is determined using a progressive damage model based on the Puck failure criterion and the material property degradation method. Finite element (FE) models are developed to obtain the structural response of the plate under tension, compression, or pin loading.

The optimum local reinforcement design is found using a modified simulated annealing algorithm, which is a reliable stochastic global search algorithm. A multi-objective optimization scheme is adopted, in which the objective function includes both the ultimate strength and the reinforcement weight so that the ultimate strength can be maximized with minimum use of material. The optimization variables are chosen as the fiber-orientation angles and the size of the reinforcing layers. Optimizations are performed for different configurations of local reinforcements to determine the most effective way of placing the reinforcing layers. Significant improvements in the strength are obtained with minimal added mass through optimization. The findings of this study have significant implications for enhancing the design of notched composite structures.

ÖZET

ÇENTİKLİ KOMPOZİT YAPILARIN BÖLGESEL GÜÇLENDİRME OPTİMİZASYONU İLE MEKANİK ÖZELLİKLERİN İYİLEŞTİRİLMESİ

Kompozit yapılar, çeşitli fonksiyonel amaçlar için bölgesel delikler açıldığında yapısal olarak zayıflar. Tüm yapının kalınlığını eşit olarak artırmak yerine, kesitlerin etrafındaki bölgenin yerel olarak güçlendirilmesi, süreksizlik olan bir yapının yük taşıma kapasitesini artırmanın çok daha etkili bir yoldur. Bu çalışmanın amacı, bir kompozit plakada delik etrafına yerleştirilen güçlendirici katmanların optimum tasarımını bularak, en az güçlendirici malzeme kullanımı ile maksimum nihai dayanımı sağlamaktır. Nihai kırılma yükü, Puck kırılma kriterine ve malzeme özelliklerinin bozulma yöntemine dayalı bir ilerleyici hasar modeli kullanılarak belirlenmiştir. Bu yapıların farklı yük koşulları altındaki davranışların modellemek için sonlu eleman modelleri kullanılmıştır.

Optimum bölgesel güçlendirme tasarımını bulmak için, güvenilir bir stokastik küresel arama algoritması olan Simulated Annealing algoritması kullanılmıştır. Çoklu hedefli bir optimizasyon şeması benimsenir; burada hedef fonksiyonu, nihai dayanım ve güçlendirme ağırlığını içerir, böylece nihai dayanım malzeme kullanımını en aza indirerek maksimuma çıkarılabilir. Optimizasyon parametreleri; fiber-yönelim açıları ve güçlendirme katmanlarının boyutu olarak seçilmiştir. Farklı bölgesel güçlendirme optimizasyonlar gerçekleştirilerek, güçlendirme katmanlarını yerleştirmenin en etkili yolu belirlenmiştir. Optimizasyon yoluyla, minimal ek kütle ile önemli dayanım artışları sağlanmıştır. Bu çalışmanın bulguları, çentikli kompozit yapıların tasarımının iyileştirilmesi için önemli sonuçlar taşımaktadır.

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LIST OF SYMBOLS

C_{ij}	compliance matrix component
d	Diameter
d_i	Damage variables
E	Elastic moodulus
e	Edge distance
F	Force
G	Shear moodulus
G_{IC}	Mode one Fracture toughness
G_{IIC}	Mode two Fracture toughness
P_{ij}^k	Slope of fracture plane
S_L	Longitudinal Shear strength
S_T	Transverse shear strength
T_i	Temperature
U_i	Displacement in specified direction
w	Width distance
X_T	Tensile strengths in fiber directions
X_C	Compressive strengths in fiber directions
Y_T	Tensile strengths in transverse directions
Y_C	Compressive strengths in transverse directions
τ	Shear strength
α	The contribution of shear stress to the fiber tension mode
σ_{ij}	Stress components
Θ	Angle of ply
ν_{12}	Longitudinal Poisson's ratio
ν_{23}	Transverse Poisson's ratio
$\parallel$	Refers to Parallel
$\perp$	Refers to transverse

LIST OF ACRONYMS/ABBREVIATIONS

2D	Two Dimensional
3D	Three Dimensional
AAM	Active Appearance Model
AFP	Automated Fiber Placement
ASM	Active Shape Model
CDM	Continuum Damage Mechanics
CFRP	Carbon Fiber Reinforced Polymer
FC	Fiber Compression Failure
FEA	Finite Element Analysis
FEM	Finite Element Method
FF	Fiber Failure
FRP	Fiber Reinforced Polymer
FT	Fiber Tensile Failure
GA	Genetic Algorithm
GFRP	Glass Fiber Reinforced Polymer
IFF	Inter Fiber Failure
MC	matrix Compression Failure
MPDG	Material Property Degradation Method
MSA	Modified Simulated Annealing
MT	Matrix Tensile Failure
OHC	Open Hole Compression
OHT	Open Hole Tension
PDA	Progressive Damage Analysis
PFA	Progressive Failure Analysis
PFM	Progressive Failure Model
RVE	Representative Elementary Volume
SA	Simulated Annealing
SSB	Single Shear Bearing

UD	Unidirectional
UMAT	User Material
WWFE	World Wide Failure Exercises
XFEM	Extended Finite Element Method



1. INTRODUCTION

The aviation and space sectors represent the highest point of technological progress and creativity, persistently aiming for improved performance, safety, and fuel efficiency in the design of aircrafts and spaceships. The fundamental aspect of this undertaking revolves around the widespread incorporation of composite materials, which signify a revolutionary change in the way the modern airplanes are built. These structural elements mostly contain cutouts that act as doorways, windows, access ports, or as a way to reduce weight. These structures are required to be damage tolerant, meaning they should continue to support design loads even in case damage occurs. Increasing the thickness all the way through the structure is an inefficient way of strengthening the structure, considering that this results in considerable material cost and increased weight. Instead, making the weaker regions stronger, for example, applying local reinforcements around the notches and in the regions where high stresses develop is a much more effective way of making the whole structure stronger. By locally reinforcing a structure, one may reduce local deformations and stress concentrations around the cutouts so as to prevent premature failures or local buckling. To do this, first, one must be able to estimate how a cutout affects the structural response of a composite part and how a reinforcement changes the stress distribution. The structural response can be predicted using a reliable structural analysis method. Secondly, one should be able to estimate how a reinforcement around a cutout affects the load-carrying capacity of a composite structure. This requires a reliable failure analysis method. Thirdly, one should be able to find effective reinforcement designs to improve the structural performance. This can be accomplished by employing a robust optimization algorithm, which can determine the optimum size, shape, and location of the reinforcements.

The failure behavior of glass or carbon-fiber-reinforced plastics is predominantly brittle. As a result, they are liable to catastrophic failure. Designing composite structures by just considering the first-ply failure load may lead to safe designs for normally expected loads. However, a catastrophic failure that may occur due to an unexpected

overload is not acceptable. A composite structure should maintain its structural integrity in case of damage. The structures should exhibit considerable damage tolerance and residual strength after initial damage. Relying solely on first-ply failure criteria may lead to unsafe designs. In order to develop an effective design, one should be able to determine the load-carrying capacity of the composite structure, in other words, to predict the damage progression in the composite material as the load is increased, ultimately leading to a breakdown of the structure's ability to sustain any load. For this purpose, progressive failure analysis (PFA) techniques are used.

1.1. Progressive Failure Analysis in Laminated Composite

The failure mechanisms in laminated composites are particularly complex, especially in structures with stress risers. If the fibers in a ply with a certain direction sustain damage, fibers aligned along a different orientation may continue to support the load. For this reason, the loads at which the first and last instances of ply failure are usually different. Understanding the failure behavior of laminated composite structures is crucial for developing safe designs. Employing progressive failure analysis aids designers in identifying the failure progression and determining the point at which the last ply fails in these constructions.

To perform Progressive Failure Analysis (PFA), one must carefully consider both the damage initiation and progression mechanisms while also providing a precise definition of all aspects that influence the failure progression.

The commonly used failure criteria for detecting the initiation of damage in composites are the maximum stress and maximum strain criteria [1], Tsai and Wu [2], Hoffman [3], Hashin [4], and Puck [5,6] criteria. Investigation of different failure criteria for identifying the onset of damage in fiber-reinforced polymer composites and damage progression methods has been a significant subject of interest in both academic and industrial spheres for many years. The worldwide failure exercises (WWFE) [7–10] assessed many failure criteria and damage evolution methodologies to establish a stan-

dard for the best-performing failure criterion. None of the criteria used in those exercises could effectively demonstrate how damage spread inside a single layer or across adjacent layers, nor could they accurately depict how variations in composite layups or geometry impacted the damage propagation in the composite laminates.

Although numerous damage models and criteria for composite failure exist, there is yet to be widely accepted model. In this thesis, the Puck failure criterion, which is considered to be one of the most reliable criteria, is used to predict the damage initiation, and the material property degradation is modeled by using proper degradation parameters, which significantly affect the damage progression and, consequently, the ultimate failure load of the structures.

1.2. Improving the Mechanical Behaviour of Laminated Composite

Although significant gain can be obtained in strength by locally reinforcing notched composite structures, quite a few studies in the literature considered this problem. Schlapfer and Kress [11] proposed a strength improvement method to determine the optimum local reinforcements placed around the hole of a composite laminate. In that method, sensitivities for layer thickness changes were calculated for each finite element. Local reinforcements were introduced by assigning thickness values to initially zero-thickness virtual layers based on the sensitivity information. In the optimum design, the loss in strength due to the hole was almost fully recovered with only 6% additional reinforcement material. Huang et al. [12] employed the finite element method (FEM) to optimize the fiber orientation of each element around a circular hole. The main goal of the study was to identify the most suitable fiber orientations that would maximize the load-carrying capacity of the structure based on the Tsai-Wu failure criterion. The optimization results showed the effectiveness of this approach in improving the strength of notched composite plates. Similar studies [13–17] were also conducted; however, the main drawback of this method is the difficulty in manufacturing a laminate composed of layers with variable fiber orientations. Matias et al. [18] used a genetic algorithm to reduce stress concentration in a metal plate by optimizing the shape, location, and fiber

orientations of a composite patch bonded onto the plate. One way of locally reinforcing a notched laminate is by using additional layers around the notch. Figure 1 shows some of the reinforcement configurations that may improve the load carrying capacity of plates with a hole. By determining the optimal shape and size of these additional layers and their placement, one can significantly improve the laminate strength with a slight increase in the material use.

In the previous studies, the first-ply failure approach was used to determine the load that caused initial damage, which means the first-ply-failure load was maximized with no regard to the ultimate failure load. In this study, the optimum designs are found that have the maximum ultimate strength. The failure load causing initial damage at locally high-stressed regions around a cutout is usually a small fraction of the ultimate failure load of the composite structure. The experimental studies on the failure behavior of open-hole tension specimens, like references [19–21], show that the overall stiffness of the specimens degrades only after considerable accumulation of damage. This means that, small-scale local damage can be tolerated under working loads for notched composite structures. The design of notched composite structures should be based on the full knowledge of damage tolerance and the final strength. For this reason, maximum-strength design of notched composites should better be based on the final failure strength, not the first-ply failure strength.

The previous studies introducing design optimization methods to maximize the strength of notched composite laminates by locally reinforcing them used the variable-thickness or variable fiber orientation approach. Although promising results were obtained in those studies, their application requires automated tow-placement machines, which impedes widespread industrial application of those methods. In this study, considering the manufacturability of the optimum designs, additional rectangular layers around the cutout are used as reinforcing elements.

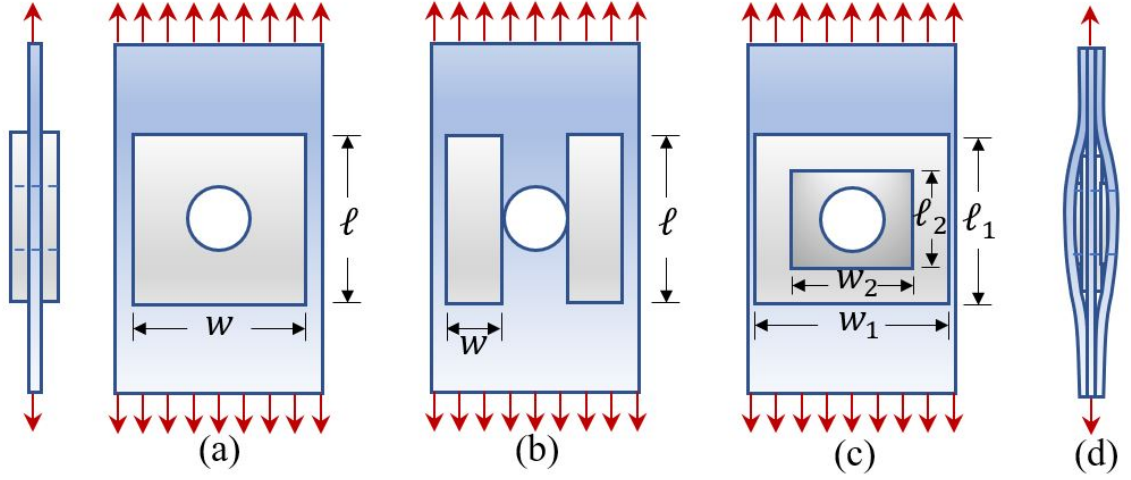


Figure 1.1. Local reinforcement configurations. a) One rectangular patch at each side b) Two rectangular patches at each side c) Rectangular patches with plydrop d) Side view of blended patches configuration

1.3. Aims and Objectives

The primary objective of this thesis is to determine the optimal local reinforcement designs that maximize the ultimate failure load of composite laminates containing a cutout. The optimization involves determining the optimal fiber orientations and the size of the reinforcements. Different reinforcement configurations are optimized to determine the most effective configuration. Additionally, this thesis aims to develop a progressive damage model capable of accurately predicting the progressive failure behavior of notched laminated composites and the final failure load.

The thesis aims to achieve the following specific objectives using both theoretical and numerical modeling approaches:

- Develop a methodology to carry out progressive failure analysis on various configurations of laminated composites with open holes subjected to uniaxial tension and validate the progressive damage model using experimental results. This analysis aims to predict the ultimate failure load accurately by using a reliable failure criterion and proper material degradation parameters.

- Evaluate the accuracy of the progressive failure model for composite laminates with holes under compression loading conditions.
- Determine the bearing strength of pin-loaded composite laminates using the progressive failure model.
- Optimize local reinforcements to maximize the strength of laminated composites with cutouts under uniaxial tension loads and pin loads.

The main contributions of this PhD thesis are summarized in the following:

- Development of a progressive damage model that accurately predicts the ultimate strength of composite laminates containing a hole under tension and compression loads and pin loading for various stacking sequences.
- Development of a method to find the optimal local reinforcement designs to maximize the final strength of notched laminated composite structures.

2. THEORETICAL BACKGROUND

This introduction explores the designing of laminated composite materials in many applications, including their failure mechanisms, optimization methods, and their essential role in reinforcing structural with stress concentration in the aviation and space industry.

2.1. Composite Materials

Carbon-fiber-reinforced polymers (CFRPs) and other advanced fiber-based composites significantly transformed the aviation and space industry due to their exceptional characteristics. These materials possess a lightweight quality, a high ratio of strength-to-weight, remarkable stiffness, resistance to fatigue and corrosion, and customizable mechanical properties. These characteristics make them the first option for aircraft and spacecraft designers and manufacturers who want to reduce weight, increase fuel efficiency, and boost overall performance. These materials have become popular because they can endure various climatic conditions and the loads that develop during airplane and spaceship operations. The continuous desire to design fuel-efficient and high-performing airplanes has instigated a fundamental change in aviation design. The exponential expansion of composite materials can be attributed to the quest for improved weight-to-power ratios. This has led to their transition from being just auxiliary components to becoming fundamental structural elements. Significant examples include the Boeing 787 Dreamliner, which incorporates 50% composite materials, representing a significant change from its predecessor, the Boeing 747, which only used 1% composite materials. Furthermore, military aircraft, exemplified by the F35, have used composite materials, with 40% of the aircraft's structure consisting of these sophisticated materials, highlighting their crucial significance in modern aerospace engineering [22].

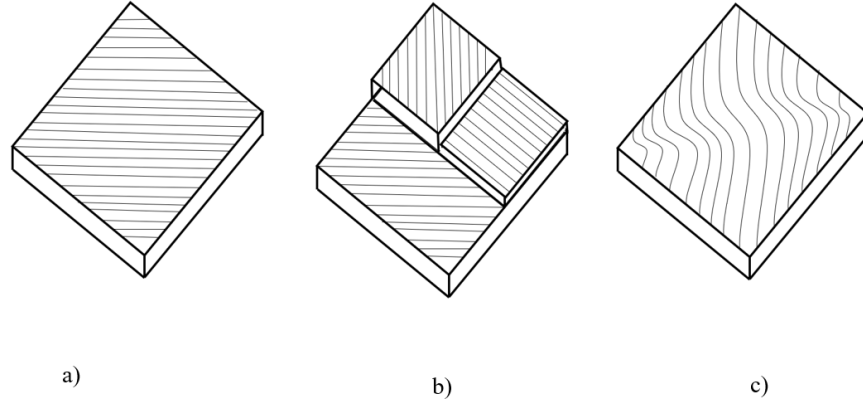


Figure 2.1. Different composite laminates configurations. a) Laminated composite b) Variable thickness c) Fiber steered

Composite laminates, which are essential components of aeronautical structures, can be classified into two types: constant stiffness laminates and variable stiffness laminates, which are shown in Figure 2.1 [23, 24]. Each type has unique engineering possibilities. Constant stiffness laminates retain a consistent arrangement of fibers at the same angles across layers, which guarantees uniform stiffness qualities. Variable stiffness laminates, on the other hand, allow for customized stiffness across the structure by incorporating spatial variations in fiber angles or by combining different stacking for different regions in the structure which is known as blended laminates.

2.2. Stress Concentration

One reason for the necessity of locally optimizing structures is the presence of stress risers within the structures. Aircrafts and space structures have numerous cut-outs and holes, which adversely affect its performance due to the stress concentrations in these areas. For instance, in the lower left wing skin of an F-16 airplane, fatigue cracks are initiated at fasteners located near the vent holes. A few cracks expanded to the second row of fasteners. Cracks that go beyond the vent tube flange would create a direct route to the wing fuel tank, leading to a fuel leak. When this happened, the lower wing skin was considered to have reached the end of its service life [25].

The composite crew module utilized in space missions includes significant openings, which provide diverse functionalities and points of access within the structure, A figure illustrating this structure can be found in [26].

The fuselage of transport cargo planes contains strategically placed giant cutouts that are specifically built to facilitate efficient cargo access. These cutouts include enormous doors that enable easy loading and unloading [27].

The designer must carefully analyze how these holes may impact the strength of the laminate. The decrease in hole diameter has a disproportionate impact on the tension and compression strength of composite laminates, which is widely recognized. The magnitude of this impact is greater than what is anticipated by the basic net section stress, but it is not as significant as what is projected by the stress concentration factor (K_t) for brittle failure. Therefore, the intensities of OHT and OHC are often determined through experimental measurements. These are included in the first level of testing within building-block test programs. An estimated breakdown of failure modes for the fuselage and wing of an airplane is reported in [28]. Figure 2.2 illustrates that OHT, OHC, and bearing failure modes collectively contribute to about 40% of failures in aircraft structures. However, the specific percentage may vary dramatically across different applications.

A conventional approach for building a composite structure involves initially determining the dimensions of the joints, followed by determining the dimensions of the rest of the structure, taking into account the potential failure modes in composite airframe structures. This is done because the implementation of ply-drop regulations and other manufacturing limitations sometimes results in the joint becoming the primary determinant of the design. Alternatively, with the complete determination of strength in structures with stress concentration and the application of optimization techniques, it is possible to build composite structures with stress risers that exhibit superior performance.

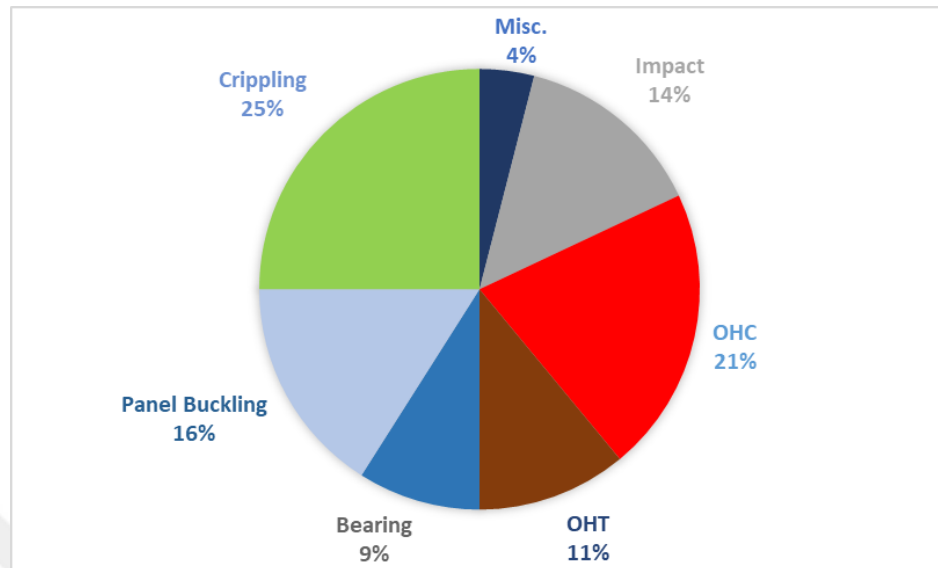


Figure 2.2. Composite airframe failure modes.

2.3. Design of Joints

It is obvious that joints need to be considered as a crucial step in the design process. In almost all hardware designs, a structural joint is an essential component. A joint in a composite structure may be composed entirely or in part of composite materials. Mechanical fastening or adhesive bonding are two possible connecting techniques. While bonding and material fastening share the goal of facilitating smooth load transfer between joint elements, the mechanisms involved in load transfer are significantly different. When fastening mechanically, load is transferred by shearing (and, less ideally, bending) the fasteners on the faces of the holes that pass through the joint members [29].

It is essential to have an in-depth understanding of the probable failure processes while designing a composite joint. Commonly employed in composite structures, mechanically fastened joints are prone to failure due to many fundamental causes, such as net-tension failure, shear-out failure, and bearing failure. Sometimes, there are situations where a combination of failure modes occur simultaneously, each with unique

characteristics and impacts on the joint. Typical failure modes in composite joints are [29]:

- Tension failure
- Shear Out failure
- Bearing failure
- Cleavage-tension failure
- pull-through failure
- bolt failure

CFRP laminates have been utilized in many aerospace, Formula One cars, and sports goods structures. Nearly half a century ago, researchers initiated investigations into the mechanical properties, fracture behavior, and response to stress concentrators of these materials. The analysis of damage mechanisms in CFRP laminates under different loading conditions, such as tension, compression, impact, and bending, has been extensively studied through numerical and experimental research. Improving the design of composite laminates will be facilitated by a more profound understanding of the damage mechanisms.

An essential component of studying the microscale mechanics of damage and failure involves observing and understanding these mechanisms visually. This is essential because experiments require substantial costs. Due to significant advancements in computer power, basic models have transformed into complex simulations like finite element applications for progressive failure analysis. This chapter provides a comprehensive examination of several methodologies employed in predicting the fracture behavior of composite laminates. The complicated fracture behaviors of fiber-reinforced composites are influenced by various aspects, including the composite laminate's lay-up and the type of stress concentration, due to the complicated structure of these materials. The selection of the lay-up significantly affects the level of anisotropy in laminates.

Damage refers to the development of cavities or cracks at many scales, including microscopic, mesoscopic, and macroscopic levels. These damages leads to the degradation of the material's mechanical properties [30]. The fundamental issue in the mathematical modeling of damage is the various scales involved, from damage initiation to propagation [31]. The initial failure originates at a microscopic level and subsequently grows to cause damage at a macroscopic level. Although it is desirable to determine failure at microscopic level but analyzing occurrences of failures at the micro-level is complex; therefore, attention should be redirected to the meso-level which determines the failure in ply level [32]. The propagation or accumulation of damage under increasing loads is the usual cause of catastrophic failure in composite structures, meaning that the load at which the structure fails is barely matches the load at which the first ply fails [33].

Damage concerns have been addressed through the proposal of analytical models, mostly grounded in theories like micromechanics, continuum damage mechanics, and fracture mechanics. Various length scales can be used to address damage and failure processes in composite lamina and laminates, such as matrix cracking, interface debonding, fiber breaking, and interlaminar delamination. Advantages and limitations of these models were reported by [34]. The typical approach for micromechanical scale is using RVE models (unit cell) and for mesoscale scale using XFEM and cohesive law or continuum damage mechanics.

2.4. Failure Mechanisms in Composites

A thorough characterization of the failure mechanisms is essential for conducting an accurate analysis of progressive damage in composites. Several failure mechanisms exist in FRP layered materials, as opposed to isotropic ones. These failures can occur either within a single layer (intralaminar failure) or between multiple layers (interlaminar failure) [35]. Damage in FRP composites occurs via distinct pathways because of the material's highly diverse phases, which include brittle fibers and a flexible matrix. The processes that can occur include cracking in the matrix, micro-buckling

of the fibers, debonding between the fibers and the matrix, delamination, and fiber breaking [36].

The damage typically begins with matrix cracks that run perpendicular to the main load direction; these cracks can then expand and combine into delamination under greater loads because the matrix is far weaker than the fibers. Fiber-matrix (interface) debonding can happen when stresses are concentrated in one area. Some fibers break as the load increases. When the laminate's remaining load-carrying capacity reaches its limit, the load is redistributed until the laminate fails catastrophically [32]. The structure's load-carrying performance is drastically reduced as a result of the combination between these several damage mechanisms, which impact the laminate's local mechanical properties [36]. Mechanical properties of the these layers and stacking sequence of the laminated composite and loading direction determines the occurrence of failure modes.

Pinho et al. [37] characterized fiber-reinforced laminated composite failure modes as Fiber Tensile failure (FT), Fiber Compression Failure (FC), Matrix Tensile Failure (MT), Matrix compression Failure (MC), and delamination. When FT occurs due the fibers significant strength and stiffness a substantial quantity of energy is rapidly release generally resulting in a catastrophic breakdown. The resin shear behavior and flaws, such as the initial fiber misalignment angle and voids, have a significant impact on FC mode. Usually, kinking bands are seen at a smaller scale and occur due to fiber micro-buckling, matrix shear failure, or fiber failure. The fracture surface that occurs as a result of MT mode is usually perpendicular to the direction of the applied load and this fracture surfaces often exhibit fiber splitting. MC mode is observed to happen at an angle relative to the loading direction, indicating that the failure process is mostly due to shear forces. Delamination refers to the separation between two adjacent plies in a laminate, and a composite structure's mechanical properties, especially its bending stiffness, are significantly reduced after it has been delaminated [32, 37].

2.5. Failure Criteria of FRP Composite Materials

The failure mechanisms of composite materials are complex and diverse, impacted by factors such as geometric dimensions, material qualities, stress conditions, and manufacturing flaws [38]. Hence, the establishment and utilization of failure criteria are of great significance in the analysis, design, and assurance of the safety of composite structures subjected to various and intricate loading scenarios [39]. Despite the existence of multiple failure criteria suggested throughout the years, persistent endeavors are being made to bring novel criteria or alter the ones already in place.

Three categories can be used to classify the failure criteria as follows [40]:

- The limit criteria involve predicting the failure load and failure mode by independently comparing the stresses or strains of the lamina with their corresponding strengths. No interaction between these factors is taken into account. Illustrative examples include maximum strain and maximum stress.
- One of the interactive criteria is to use a single quadratic or higher-order polynomial equation that includes all the stress or strain components to estimate the failure load. By comparing stress or strain ratios, the failure mode can be indirectly determined. Hill, Tsai-Hill, Tsai-Wu, Hoffman, and Yamada-Sun are other examples.
- The matrix failure is differentiated from the fiber failure by the discrete mode criteria. Equations may be subject to the influence of one or more stress components, wherein normal stresses and shear stresses may coexist. In situations involving numerous stress components, the failure mode is ascertained through the comparison of stress or strain ratios, employing a methodology akin to that utilized in interactive criteria. Included among these are Hashin and Puck. It is assumed that damage initiation has taken place when the failure index reaches a value of one.

2.6. Continuum Damage Mechanics

Several Continuum Damage Mechanics (CDM) models have been developed to predict the material responses and damage behaviors of composite materials, providing an efficient method to depict damage processes with restricted experimental data. These models incorporate internal state variables to distinguish between different types of damage, such as cracking in the fibers, matrix, and interfaces. CDM originally suggested by Kachanov [41] as a method to investigate the failure of metals under creep rupture, the use of Continuum Damage Mechanics (CDM) has demonstrated great usefulness in performing progressive failure assessments in composite laminates [38]. This method offers an essential understanding of how damage begins and spreads in various loading scenarios, assuming that the problem size is far greater than the size of any defects or micro-cracks.

2.7. Composite Design Optimization

Design optimization aims to identify the most effective structure that supports the applied loads and satisfies the requirements from a set of feasible designs. As shown in Figure 5.4 Composite airframe failure modes for metallic structures corner doublers are used to reinforce the door large cutout on the airframe structure and in Figure 2 the web with stiffener with a hole is reinforced by a Donut doubler in which the diameter of the hole is less than the half of the beam depth and the ring doubler width ratio to the hole diameter could change between 0.35 and 0.5 [27].

At Airbus [42], structural optimization is divided into two distinct categories: global structural optimization and local structural optimization. These two approaches fulfill certain objectives and requirements. When detailed composite part sizing is necessary, local structural optimization is employed. The weight of the A380 construction was reduced by utilizing local structural optimizer software to optimize numerous stiffening panels.

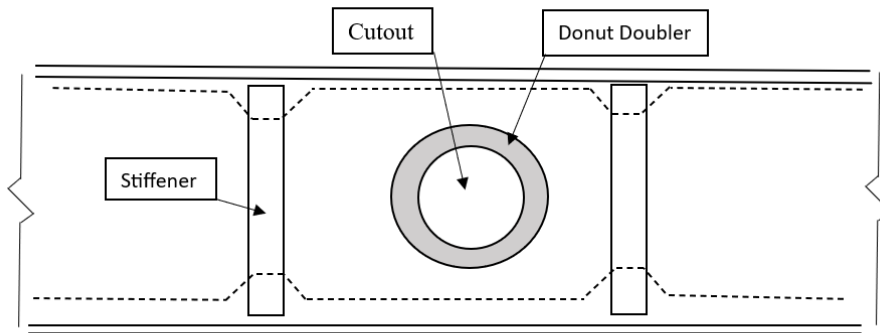


Figure 2.3. Doubler stiffened the web of aircraft with holes.

To enhance the efficiency of aerospace structures, Multi-level optimization is being performed. Multi-level optimization is the process of decomposing a complicated optimization problem into smaller, more feasible sub-problems at several hierarchical levels [24]. This methodology enables a systematic and incremental resolution of problems, frequently dividing the process of optimization into system-level considerations and component-level specifics. It streamlines the overall optimization process by systematically solving minor problems in a step-by-step manner and eventually incorporating the solutions into a hierarchical structure.

For the purpose of optimizing the structure at the component level, the utilization of multi-objective optimization can effectively address the task of maximizing numerous objectives that may be in conflict with each other. Multi-objective optimization seeks to identify a collection of solutions, referred to as Pareto-optimal solutions, that offer the most favorable compromises between many objectives that frequently conflict with one another. A review paper on design optimization of variable stiffness composite is reported by Ghiasi et al. [24]. Some of the optimization methods that are widely used in composite optimization are direct search methods like Genetic Algorithm, Simulated Annealing, and Evolutionary programming, which are considered stochastic methods [24]. These methods are renowned for their versatility, scalability, and proficiency in addressing intricate and multidimensional optimization issues. They provide adaptable solutions in multiple domains thanks to their ability to explore a wide range of solution spaces and their efficiency in identifying nearly optimum answers, but they have their

own limitations. Simulated Annealing (SA) operates by exploring the neighborhood of a single point, often leading to a higher probability of getting trapped in local minima. To mitigate this limitation of SA, a Modified Simulated Annealing (MSA) is employed in this study. In order to achieve the globally optimal design, it is crucial to widen the search within the solution space. Instead of making minor alterations to the current configuration to generate a nearby solution, allowing substantial variations in the current configurations enables a more extensive exploration across the design space.

The formulation of the composite material design optimization issue aims to achieve the most optimal material design that exhibits superior performance while adhering to the stated constraints. The primary aim of the method is generally to minimize a selected objective function. In recent decades, researchers have made efforts to enhance the structural performance of composite constructions by considering different objective functions [43, 44]. These include minimizing stress [45, 46], maximizing strength and stiffness [47–49], minimizing natural frequency [50], minimizing deflection [51, 52], and maximizing buckling load [53–58] [49–54]. Minimizing the quantity of material utilized in composite constructions is necessary as a result of the exorbitant production expenses and the need for lightweight characteristics in the buildings. To achieve this purpose, it is often beneficial to optimize the parameters of the laminated composite, such as the orientations of the fibers, the thickness of each ply, and the sequence in which they are stacked. Ghiasi et al. [59] presented a comprehensive study of optimization techniques for constant stiffness laminated composites, which encompass gradient-based approaches, direct search and heuristic methods, specialized techniques, and hybrid methods.

Composite optimization design spaces are often non-convex [60] or multimodal [61], which makes gradient-based optimization more difficult due to noisy gradient evaluations. Consequently, when optimizing the layup of composite materials, heuristic methods such as GA and SA are frequently chosen.

For constant stiffness laminated composite fiber orientation, layer thickness, and stacking sequence are used as variables in single and multi-objective optimization conducted by many researchers using GA [62–67] and Simulated Annealing [68–71] and a recently comprehensive review article between GA and Evolutionary computation for composite optimization is reported by Wang and Sobey [72].

2.8. Blending Rules in Laminated Composite

Discontinuity in the layers in patch design procedures results in a loss of integrity and raises the difficulty and cost of manufacturing. Optimizing the thickness, stacking order, and ply orientation of the layers can achieve stiffness variation between panels; nevertheless, variations in thickness, unpaired ply orientation between plies, and stacking order typically result in ply-drop between panels. Different optimization techniques were utilized to identify the best solutions for blending rules that satisfy the continuity of the plies and reduce the consequences of ply drops.

For variable thickness variable stiffness designs, extra continuity restrictions must be utilized in addition to the manufacturing constraints for constant thickness variable stiffness laminates. It is rare for there to be an even distribution of load in several real-world design applications. Ply drops may be present in certain places of the panel that are subjected to modest loads, but the designer must avoid overdesigning the structure. Variable thickness and continuity requirements must be taken into consideration when using an optimization technique to customize the design of particular regions with ply drops. Blending of composite laminates refers to maintaining continuity between the various spots in the composite panel.

Composite structural optimization encompasses several goals. Each aspect of performance, including strength, stiffness, buckling load, strain, and natural frequency, can be separately maximized or analyzed together in a multidisciplinary approach. Additionally, weight and cost can also be minimized. One way to find the optimum design is using the multi-objective optimization method. In this study, multi-objective

optimization is used to maximise the stiffness of the structure by adding minimum weight.



3. LITERATURE REVIEW

3.1. Introduction

The objective of this chapter is to present the literature review pertaining to the subject matter addressed in this study: Design and optimization of CFRP composite with stress concentration. Three individual sections will be involved, including experimental investigation, numerical simulations and surrogate-model based optimization.

The present chapter commences by examining our existing understanding of the detrimental effects and failure mechanisms in notched fiber-reinforced laminates. Below, a comprehensive array of in-plane loading techniques is examined. The significance of stress concentration is especially crucial due to the presence of notches and holes in real-world applications of laminates, arising from mechanical joining techniques. The last section of this chapter presents an overview of the existing literature in this field. The objective of this chapter is not to offer a comprehensive inventory of all the elements defined before. Instead, its structure is designed to pinpoint the deficiencies in our current understanding of the topic and to justify our chosen method for addressing the issue at hand.

3.2. Continuum Damage Model

Matzenmiller et al. [73] created a constitutive model based on CDM to investigate the effective elastic characteristics and elastic-brittle behavior of composite materials. Their methodology consisted of utilizing a uniform continuum in the constitutive theory, which integrated the treatment of anisotropic damage and elasticity. This model incorporated damage variables to phenomenologically depict defect states and their consequences on the reduction of stiffness. The growth rates of damage were determined using power functions of internal state variables, which were obtained by multiplying local damage parameters with scalar growth functions.

Maimí et al. [74, 75] introduced a new constitutive model that has been verified to accurately forecast the initiation and spread of damage in composite laminates under plane stress conditions. Four different types of damage were taken into account, including fiber tensile fracture, fiber compressive buckling, matrix tensile failure, and compressive cracking. These damages were orientated at certain angles with respect to the direction of the ply thickness. Various failure mechanisms under varied directional stresses were represented using scalar damage variables.

In a different study, Talreja [76] presented a continuum damage model characterized by internal damage variables depicted as tensorial quantities. This model specifically accounted for matrix cracking and delamination as distinct damage modes. It assumed these modes did not interact but were considered separately, accounting for each damage mode one at a time before superimposing their effects.

3.3. Failure Criteria of FRP Composite Materials

Validating the predictions of damage models and criteria against experimental findings is a crucial factor in their advancement. A comprehensive assessment of existing failure criteria for fiber-reinforced composites was conducted through World wide failure exercises I [7, 8, 77], World Wide Failure exercises II [9, 78], and World wide failure exercises III [10]. It also should be highlighted that all of the criteria reviewed in the WWFE-I cases were in-plane failure criteria. For example, in these cases, Soden et al. [77] concluded that various theories can accurately characterize the physical behavior of unidirectional FRPs under biaxial stress at the lamina level, based on the validation of existing data from experiments. They discovered that almost all of the theories were able to provide precise forecasts of strength and strain in the specific laminates at low levels of strain prior to initial failure. However, none of these theories were capable of handling the significant nonlinear deformation that occurred in some experiments. Furthermore, the Puck and Cuntze theories, which are considered the most precise, were only able to predict the ultimate fracture strength of multidirectional laminates with an accuracy of within 50% variation in over 85% of the test

instances, when compared to the experimental results.

Our primary focus is the assessment of these failure criteria within the context of CFRP composites featuring cutouts under uniaxial stress. Specifically, our interest lies in evaluating the performance of these criteria against a particular test case, namely Test Case 12 of WWFE-III [10]. In this test case, the effect of different hole sizes on the strength of laminated composite plate is investigated by different researchers but for this case, some studies found that by increasing the hole size the strength of the specimens decreases and some other studies find that the strength of the structure decreases and some studies reported no strength difference when the hole size increases. There is a clear disagreement on whether the strength is influenced by the hole diameter. Consequently, some of these models rely on too simplistic assumptions that lack a solid physical basis [10].

When comparing the material properties of IM7/8552 carbon/epoxy in WWFE-III [79] with the stated values for this material in NCAMP [80], there are discrepancies in the input values used for analysis. These inconsistencies could potentially result in inaccuracies when compared with experimental data. It is worth mentioning that different parameters that are used in the progressive analysis like material properties and damage degradation parameters as well as the loading step in the analysis have major effects on the predicted failure strength.

3.4. Progressive Failure Analysis - OHT

Tan and Seng [19] developed a progressive failure model to examine the behavior of laminated composites with open hole subjected to in-plane tension loads. The model utilizes simplified damaged ply constitutive relations to replicate the presence of damaged layers within a lamina. This model takes into consideration environmental factors, such as thermal residual stresses. A parametric study was undertaken to investigate the impact of finite element mesh size and damage degradation parameters on the ultimate failure load of specimens. Sengodan et al. [81] also created a model

that consider the environmental effects on the laminated plate with an open hole. Tan et al. [82] employ their progressive failure model to determine the compressive stress of a laminated plate containing an open hole.

Chang et al. [83] introduced a damage model for notched laminates that experience tensile loads for laminates with different lay-up configurations. The analysis made use of the 2D planar stress assumption, with loads steadily increasing in small stages. It was assumed that the stress-strain interactions followed a linear pattern. The Yamada failure criteria is employed to ascertain the initiation of failure. The generated results were compared with experimental results for various stacking sequences, and it was found that some of the results did not align well with the experimental data.

Shokrieh and Lessard [84, 85] introduced and validated a model that is capable of modeling the fatigue characteristics of laminated composites subjected to various loading situations, with or without the presence of stress concentrations. This model has the capability to ascertain the remaining strength, remaining stiffness, and fatigue lifespan of composite laminates with any given geometry and stacking sequence under complex fatigue loading conditions. Hashin's criteria were used to establish the failure modes.

Hallet and Wisnom [86] introduced a novel method for modeling notched composite materials by utilizing interface elements to represent the damage inside and between the layers. The purpose of developing this method was to create a model for delamination in composites. It has the capability of predicting the onset and spread of delamination, but it needs prior knowledge of the likely locations where failure may occur. In another investigation conducted by Hallet et al. [87], the researchers examined the impact of stacking sequences on laminated composites with an open hole. The investigation utilized IM7/8552 carbon fiber/epoxy. Experimental findings indicate that delamination in quasi-isotropic laminates typically occurs when the structures are subjected to the maximum failure load.

Ridha et al. [88] proposed a progressive failure model to predict the effect of specimen size and laminate lay-up on the open-hole tension (OHT) strength of orthotropic composite laminates. The maximum stress failure criterion is integrated with the Tsai-Wu failure criterion to simulate failure in situations where either the fibers or the matrix material are dominant. The models accurately forecast the expected trend of the impact of specimen size on OHT strength. However, the model fails to accurately estimate the OHT strength of a specimen with four 0 plies. The difference arises due to the fact that the stiffness and strength of laminates increase as the proportion of 0 plies increases.

Dharmawan et al. [89] investigations have shown inadequacies in forecasting the final failure in various stacking sequences and various orthotropy values which are defined as plate elastic modulus in the y direction over elastic modulus in the X direction and the load is applied in Y direction. A progressive method available in Abaqus explicit that uses Hashin failure criteria is used in their study. These experiments specifically identified significant differences in the expected strength of stiff laminates, especially those that incorporate plies oriented at 0° .

Knight [90] employs UMAT and ABAQUS software to determine the maximum carrying capacity of OHT specimens. He employed the maximum stress, maximum strain, Tsai-Wu, and Hashin failure criteria in their progressive failure model to analyze the cross-ply laminate and quasi-isotropic laminated composite with an open hole. The maximum stress and maximum strain criterion yielded almost identical peak failure loads and nonlinear behavior. Although the Tsai-Wu failure polynomial criterion anticipated lower peak failure loads, these predictions remain highly consistent with the test results. The Hashin criteria consistently underestimated the maximum failure loads by around 25% when considering a specific set of degradation factors.

The most recent progressive failure analysis of curvilinear path fiber in variable stiffness laminated composite is conducted by Malakhov [91]. In their study material property degradation method was used for damage progression and the values suggested

by [19,82] are used in their study, Hashin failure criteria is used for determining damage initiation.

3.5. Composite Joints

Crews et al. [92] conducted stress calculations on different laminates subjected to the load of a frictionless steel pin placed in a circular hole of the same size. The contact modeling utilized short spring elements that lack transverse stiffness, enabling the transmission of solely radial loads and establishing a frictionless interface. The stress-concentration factors were determined for specimens with varying width-to-hole-diameter (w/d) and edge-distance-to-hole-diameter (e/d) ratios. These factors represent the ratio of peak stress (shear or normal) to the nominal bearing or gross stress. The study examined several combinations of width-to-depth (w/d) and edge-to-depth (e/d) ratios in six distinct layups and noticed substantial effects on stress distributions. For instance, a quasi-isotropic laminate with a width-to-depth ratio (w/d) of 10 exhibited a tensile stress concentration factor that was 20% higher than that of a specimen with an infinitely wide width.

Lessard and Shokrieh [93] aimed to assess the viability of utilizing a two-dimensional model to predict the behavior of pin-loaded structures. They conducted an investigation employing five distinct failure criteria to detect damage in pin-loaded structures. The study focused on three configurations featuring varying width-to-diameter ratios and different edge distances to diameter ratios for laminated composites of $[0, 45, 90]_s$ orientation. Their findings revealed that the maximum error percentage, in comparison to experimental results, was notably less than 11%.

Okutan [94] conducted a parametric analysis using both experimental and numerical methods. The study utilized a 2D FEM developed by Lessard [90] to simulate pin-loaded GFRP (Glass fiber-reinforced polymer) laminates. The analysis only considered cross-ply laminates. It was assumed that the pin was rigid and that there was no friction at the pin-friction interface. The implementation of Hashin failure criteria

was used to establish a precise definition of failure. Once the failure index exceeded unity, degradation rules were implemented for each mechanism. This involved reducing certain terms inside the lamina stiffness matrix to zero, depending on the form of failure. Using this modeling approach, many configurations were conducted for various e/d and w/d ratios, ranging from 1 to 5 and 2 to 5, respectively. The author states that the ultimate failure predictions were approximately accurate within a margin of 10% compared to the experimental data. The study demonstrated that ply orientations have a significant impact on the strength of the specimens. The $[90/0/90]_s$ configurations consistently outperformed the $[90/0/90]_s$ specimens by an average of around 24% across all sample geometries. Moreover, it was noted that the maximal bearing strength was only achieved when the specimens had enough edge-distances and widths, as failure due to shear-out or net-tension would occur otherwise.

Camanho et al. [95] investigated the occurrence of delamination in bolted composite joints using a three-dimensional finite element analysis (FEA) model. The researchers determined that the application of pressure by the bolt and the friction between the washer and laminate had a significant impact on the beginning and buildup of damage. The 3D model was used to examine the failure patterns of bolted composite joints, specifically in terms of bearing, net tension, and shear out.

Xue [96] developed a model of pin-loaded laminated composite plates for progressive failure analysis. The complicated process of bearing failure in composite laminates is described by an elastoplastic damage model that takes non-linear behavior into account. The model makes use of LaRC05 failure criteria and a stiffness degradation model based on CDM. When calculating the interlaminar stresses, delamination is taken into consideration by utilizing the virtual crack closure approach that is provided in the Abaqus software. The experimental result and validated result for $[45/-45/90/0]_{2s}$ pin loaded laminated composite are in good agreement.

Raju et al. [97] did a thorough inspection of the right rear lug as part of their investigation into the 2001 accident regarding the composite vertical tail of the Airbus

A300-600R. The analysis team conducted progressive failure analysis (PFA) to predict the behavior of the right rear lug, which carries loads above design limitations. They used user-defined material for damage progression and Hashin failure criteria for this purpose. The analysis of the failure of the right rear lug under loading conditions similar to a certification test conducted in 1985 included two modeling approaches: 3D solid elements with shell elements as well as layered shell elements. Both modeling approaches demonstrated good agreement with one other and with experimentally ascertained failure loads, modes, and locations. Right rear lug and substructure and damage progression in the lug are shown in Figure 9.

McCarthy et al. [98] examined the varying clearance of bolt holes in bolted joints through progressive failure analysis. Additionally, McCarthy et al. [99] devised a new technique that employed cubic spline interpolation to accurately represent the non-linear shear characteristics of notched and un-notched laminated composites with varying ply orientations. The damage was determined using a ply discount method based on Hashin's criteria. Additionally, the shear response was predicted using the spline methodology and the maximum strain failure criteria. This model exhibits high accuracy in predicting tensile strength and modulus, but, it tends to underestimate the ultimate transverse strain. However, this is only the case when shear stresses are the dominant factor.

3.6. Design Optimization of Laminated Composite

In designs where stress concentration is present, tailored fiber orientation has the ability to both avoid stress from concentrating and ensure fiber continuity.

In order to improve the tensile loading of a plate with a central hole, Hyer and Charette [100] created a laminate with fibers arranged along the major stress direction. It was demonstrated that the tensile load of laminates containing curvilinear fibers increased by up to 60% when compared to a quasi-isotropic laminate. By placing the fibers in a curvilinear direction, Hyer and Lee [98] additionally raised the buckling

load of the structure. They discovered that curvilinear fibers carry the strain from the plate's center to its edge, increasing the structure's buckling load.

Tosh and Kelly [101] optimized the fiber alignment in a plate with a hole by aligning it with the principal stress vector and load direction, resulting in enhanced structural strength. The tensile tests conducted on the manufactured specimens demonstrated a significant enhancement in specific strength, with improvements of more than 60% for open-hole and more than 80% for pin-loaded configurations detected.

In a recent study conducted by Zhu et al. [102], the orientation of fibers was adjusted to align with the main stresses in order to raise the strength of a laminate. The Trust-Region-Reflective algorithm was employed to improve the tensile strength of a composite laminated plate that included a circular cutout. The maximum strength has been assessed using the Hashin fiber failure criterion, Hashin matrix failure criteria, maximum stress, and Tsai-wu criteria. The strength of curved fiber-optimized laminates is 36% higher than that of regular laminates.

Huang et al. [17] presented an effective optimization technique, utilizing a Genetic algorithm, to optimize the curved fibers of variable stiffness composite. The design variable was defined as the path function of fibers, whereas the objective function was determined as the compliance matrix. This method was evaluated using a beam that was totally clamped and a composite plate with a hole.

Khani et al. [16] employed a multi-step optimization approach to maximize the failure load of laminates with significant cuts. These laminates were of the quasi-isotropic, constant stiffness, and variable stiffness varieties. The results indicate that variable stiffness plates exhibit greater strength compared to quasi-isotropic and constant stiffness plates.

Matsuzaki et al. [14] employ a curved fiber path to enhance the strength of a construction containing a hole. The Christensen fracture criterion is employed to de-

terminate the onset of damage, while multi-objective optimization is utilized to enhance the mean curvature and reduce the failure index. The test specimens were manufactured using the AFP method. The results indicate a 34.4% increase in the strength of the structure when a curvilinear design is adopted.

Although there is much potential for composites with variable stiffness to benefit from laminated composite materials, curved fiber placement cannot be achieved manually and requires a significant investment in automatic fiber placement machinery.

3.7. Composite Patches Used for Repairing Structures

Various endeavors have been undertaken to enhance the durability of the composite laminated plate with a cutout by adhering external patches with distinct characteristics. The load-bearing capacity of repaired and strengthened structures is significantly influenced by factors such as patch thickness, fiber orientation, stacking sequence, patch shape, and patch material. Several studies have been undertaken to examine the impact of these factors on a laminated composite plate with cutouts when subjected to tensile loading.

Gong et al. [103] utilized external circular patches to repair a composite structure that had a cutout. They utilized the Tsai-Wu and Hoffman criteria to identify the point at which damage occurs, and their optimization goal was to minimize the failure index. The study took into account the impact of patch in-plane stiffness, stacking sequence, patch thickness, adhesive thickness, and adhesive modulus.

Campilho et al. [104] employ numerical and experimental analysis to investigate the tensile characteristics of circular strap repairs made from three-dimensional carbon-epoxy adhesively bonded materials. The experimental phase assessed the elastic stiffness, strength, and failure mechanisms of various patch thicknesses and overlap lengths. An analysis was conducted to examine the factors affecting the strength of single-strap and double-strap repairs. It was determined that the drop in strength

was caused by two specific issues: early patch debonding and axial laminate splitting. These problems occurred when a small overlap length patch was utilized. The study found that the most favorable results were achieved by a double-strap repair with a 15 mm overlap length. Nevertheless, throughout the range of values investigated, the thickness of the patch had a relatively minor effect on the strength of the repairs.

Kashfuddja and Ramji [105] examined the impact of various patching configurations on the repair of carbon fiber-reinforced composite laminates that experienced damage. The tested configurations included circles, rectangles, squares, ellipses, octagons, and ovals. In their study, stress concentration variables and peel stresses were taken into account to determine the optimal patch design for improving the effectiveness of the repair. The patch size and adhesive thickness were optimized using a genetic algorithm. This optimization process was carried out using a three-dimensional finite element model and a failure criterion based on the failure index per Hashin's formulation in three dimensions. Their research has shown that using a two-sided patch repair structure resulted in a notable reduction in stress concentration, which is essential for effectively repairing carbon fiber panels that have been degraded. Furthermore, they emphasized the significance of taking into account patch designs that are determined by the order of layers. By employing a genetic algorithm in tandem with Finite Element Modeling to optimize the dimensions of patches and the thickness of adhesive, they found optimum configurations that could significantly drop the stress concentration value by 80%. Ramji et al. [106] also conducted a study similar to [102] for an inclined center cracked panel. according to the results, the extended octagonal patch shape performed better at lowering the Stress Intensity Factor (SIF).

Bhise et al. [107] use genetic algorithms and finite element analysis to optimize the thickness of the adhesive and the size of the circular patch. They also investigate the influence of these parameters on the stress concentration factor.

3.8. Design and Optimization of Structures with Patches

Mathias et al. [18] employ a Genetic algorithm to optimize a composite patch in order to decrease the stress level in a metallic structure with a cutout. The design variables consisted of ply orientations and coordinates of interpolation sites. The restricted area is a square of 20mm x 20mm located near the hole with a diameter of 15mm. Additionally, the maximum surface area of the patch is also considered a constraint.

Fischer et colleagues [108] have enhanced the strength of structures with intricate fiber architecture by implementing a method that utilizes an ant colony algorithm to automatically distribute patches.

In a recent work, Kussmaul [109] presented a two-dimensional shear-lag model for analyzing thin patched laminates. Their model was solved using the finite element method. The fiber patch placement technique is employed to precisely position discrete patches for the assembly of structural components. There are unavoidable fiber discontinuities occurring between patches and the parent plate, and these discontinuities have a substantial impact on the mechanical properties of the constructions.

Muc [110] locally reinforced the composite plate with an open hole and divided the design space into three regions. Zehnder et al. [111] optimize the patch configuration for real structure by considering the manufacturability of the structure.

Kenway et al. [112] study's primary goal is to minimize the structural weight of the wing box in order to produce a valid structural design for aerostructural optimization. This minimization is centered on changing the thickness of the structure while meeting certain stress requirements. Under predetermined maneuver conditions, the procedure makes use of fixed aerodynamic loads obtained from earlier aerostructural analyses. Efficient solutions are required for design optimization situations that involve a large number of design variables (such as the 476 variables described in their study), as

well as objective and constraint assessments that take several minutes each. Under these circumstances, the only possible option to guarantee accurate results in a feasible time limit of less than two days is a gradient-based optimization algorithm. The picture depicted in Figure 10 illustrates the internal organization and configuration of structural design variables. The thickness variables related to the structural skin are arranged in a grid consisting of 18 stations in the spanwise direction and three stations in the chordwise direction. This configuration consists of a total of 54 variables, including both the upper and bottom skins. Results of thickness distribution and failure index for initial and optimized configuration were also reported. The optimization of takeoff gross weight resulted in a 4.2% reduction in takeoff gross weight, accompanied by a 6.6% decrease in fuel consumption.

3.9. Blending Rules in Laminated Composite

By adopting the stacking sequence table (SST) concept, Irisarri et al. [113] optimized the sequence of ply-drops in laminated composites with ply-drops. The series of ply-drops that ensured the transition between interconnected laminates was described by the SST. In their investigation, numerous design guidelines that had not been looked into in previous studies were taken into account. They used a benchmark problem to test their approach, and the lightest designs they produced were more lightweight than those from earlier research and took into account a lot of design standards.

3.10. Reinforcing Around Stress Concentration

The research conducted by Schalper and Kress [114] focused on laminates with overlapping patches. They utilized the ghost layer method to reinforce the structure and prevent first-ply failure by local reinforcement. Schalper and Kress [11] enhanced the local reinforcement design method, previously proposed in their study on eigenfrequency problems, to improve the structural integrity of a plate containing a hole. The objective of the optimization was to increase the strength of the plate with a hole by a factor of two by the implementation of local reinforcement doublers. Consequently, the

load was set at twice the initial point at which the structure failed. The orientation of the doublers has been predetermined as 0, 45, and -45 degrees. Figure 11 depicts the reinforcement outlines of the improved solution.

3.11. Conclusions

Based on the identified gaps in the literature, the chosen topic holds significant importance for several reasons:

- **Addressing Progressive Failure:** Despite numerous studies exploring various failure criteria and damage progression methods, there remains a gap in understanding how these factors interact under different loading conditions and laminated composite with different stacking sequences. By investigating these effects under particular test cases, this research aims to streamline and economize the process of determining laminate strength without the need for multiple costly tests.
- **Variable stiffness laminated composite:** The utilization of Automated Fiber Placement (AFP) for tailored fibers manufacturing presents economic and designing these structures is computationally challenging. This study aims to use patch design in which the manufacturing is a lot easier the defects that might be introduced by fiber steering with AFP may be eliminated and integrate these considerations into the optimization process.
- **Reinforcement for Modified Structures:** Literature lacks comprehensive insights into reinforcing structures when undergoing revisions or introducing new cutouts without the need for extensive structural changes. This research will explore strategies to reinforce structures without compromising the overall integrity.
- **Optimizing for Final Failure:** Notably, no single study has specifically optimized structures considering their final failure. This research seeks to fill this gap by focusing on optimizing structures while considering the final failure point, contributing significantly to the understanding and improvement of structural design and reducing weight in laminated composite structures.

4. METHODOLOGY

4.1. Progressive Damage Analysis (PDA) in composites

Analytical method, which involves closed-form solutions, and numerical solutions, which involves Finite Element Analysis (FEA) methods are the two main methodologies researchers usually use when doing Progressive damage analysis. Stresses and displacements close to the crack tip can only be analytically computed for simple cases. Numerical solutions using finite element methods are necessary for the majority of complicated geometries in real-world engineering situations. Hashin [4], Puck and Schürmann [6], and other failure criteria provide four essential components for PDA in composites within the framework of a numerical methodology.

- 1 layer-by-layer analysis of strains and stresses.
- 2 Application of lamina fracture criteria.
- 3 Integration of degradation models to consider the influence of partial damage, which may not always lead to complete failure of the laminate.
- 4 The routine's implementation, which combines the aforementioned sequences to repeatedly simulate the progressive fracturing process.

The occurrence of initial lamina failure and the corresponding failure mode could be predicted by utilizing suitable failure criteria, and accurately predicting failures after damage initiation necessitates a comprehensive comprehension of failure modes and damage propagation mechanisms, This can be accomplished by modifying the lamina's material properties using a model of material degradation [33].

4.2. Failure Criteria

Damage initiation criteria are expressed through mathematical equations that forecast the onset of material damage and the corresponding stress conditions. Laminate strength can be predicted using a variety of failure criteria, including the maximum stress, maximum strain criterion, Tsai-Wu, Tsai-Hill, Hashin, Puck, Cuntze criteria, and LaRC05 criteria etc. The majority of them are macroscopic criteria, which implies that in addition to material stresses, the input data include ply or laminate strengths.

4.2.1. Hashin

The Hashin criterion defined damage initiation for fiber tension, fiber compression, matrix tension, and matrix compression modes, which can be coupled or uncoupled. These damage initiation criteria are as follows

1. Tensile fiber failure for $\sigma_{11} \geq 0$

$$\left(\frac{\sigma_{11}}{X_T}\right)^2 + \alpha \left(\frac{\sigma_{12}}{S_L}\right)^2 \geq I_{ft} \quad , \quad \text{where} \quad 0 \leq \alpha \leq 1 \quad (4.1)$$

2. Compressive fiber failure for $\sigma_{11} < 0$

$$\left(\frac{\sigma_{11}}{X_C}\right)^2 \geq I_{fc} \quad . \quad (4.2)$$

3. Tensile matrix failure for $\sigma_{22} > 0$

$$\left(\frac{\sigma_{22}}{Y_T}\right)^2 + \left(\frac{\sigma_{12}}{S_L}\right)^2 \geq I_{mt} \quad . \quad (4.3)$$

4. Compressive matrix failure for $\sigma_{22} < 0$

$$\left(\frac{\sigma_{22}}{2S_T}\right)^2 + \left[\left(\frac{Y_C}{2S_T}\right)^2 - 1 \right] \frac{\sigma_{22}}{Y_C} + \left(\frac{\sigma_{12}}{S_L}\right)^2 \geq I_{mc} \quad , \quad (4.4)$$

where

- σ_{ij} : Stress components
- α : determines the contribution of shear stress to the fiber tension mode
- X_T and X_C : Tensile and compressive strengths in fiber directions
- Y_T and Y_C : Tensile and compressive strengths in transverse directions
- S_L and S_T : Longitudinal shear strength and transverse shear strengths.

When any of the indexes exceeds the value 1, damage initiation occurs. The user must provide material properties mentioned above. Strength values that are measured by standard methods do not match precisely with the damage initiation because of application conditions. To precisely predict damage onset in laminated composite, the transverse tensile and shear strength must be replaced by the in-situ strength of unidirectional lamina. Finite element analysis software does not differentiate between nominal and in-situ strength values. The in-situ strength values can be obtained from two alternatives; one is to calculate those strength values in terms of the unidirectional ply strength ply thickness. The other one is to adjust the values using laminate composite experimental data.

4.2.2. Puck Failure Criteria

The limitations of classical failure criteria, such as maximum stress and maximum strain, and Tsai-Hill criteria, is they cannot distinguish the matrix and fiber failure under tensile and compressive loading. Hashin proposed four different failure modes for unidirectional composite material, as mentioned before [4]. Puck and Schürmann [6] used phenomenological models to describe damage and fracture behavior while also placing it on a physical basis. When a biaxial stress condition is applied to a lamina, their model not only predicts stress levels that lead to fracture initiation but also indicates the direction of the cracks. Puck's failure criteria are classified into two modes:

1. Fiber failure (FF), which is further subdivided into tensile and compression failure modes.
2. Inter Fiber failure (IFF), which is divided into three different stress state modes: A, B, and C.

Additionally, this failure criteria proposes an equation of a new fracture plane that is generated during the damage of a composite laminate.

4.2.2.1. Fiber Failure (FF). The fiber damage is usually caused by stresses in the fiber direction. For determining failure in fibers maximum stress criteria used in ANSYS APDL. The equations are as follows

$$\frac{\sigma_1}{X_T} = 1 \quad \sigma_1 > 0 \quad (4.5)$$

$$\frac{\sigma_1}{X_C} = 1 \quad \sigma_1 < 0 \quad (4.6)$$

where σ_1 is tensile stress along the fiber direction and X_T , X_C are tensile and compressive strength of the material correspondingly.

4.2.2.2. Inter-Fiber Failure (IFF). A macroscopic crack form through the matrix material is defined as inter fiber failure or matrix failure. Adhesive Fracture of the interphase between matrix and fiber and also cohesive matrix fracture are encompassed in inter-fiber failure. An initial matrix crack is formed and propagates to the fiber borders.

Inter-fiber failure (IFF) can occur in several forms, such as transverse tension and compression, transverse shear, and longitudinal shear, depending on the type of loading. These fracture planes for unidirectional (UD) carbon composites could be found in [5]. A straight crack aligned perpendicular to the force is exhibited under transverse tension or longitudinal shear, whereas an inclined crack is formed during transverse compression

and transverse shear. Although the presence of matrix damage causes a rearrangement of stresses in the laminate, the lamina can still support some load. Formation of inter-fiber failure cause subsequent damage because of stress concentration and delamination may be observed particularly around the crack tip.

4.2.2.3. Plane Stress Mode:. In the case of plane stress, inter-fiber failure can be explained in the cutting plane, where the principal stress in fiber direction is zero. The following is how the condition that distinguishes modes A, B and C and the equations are as follows

$$\sqrt{\left(\frac{\tau_{21}}{R_{\perp\parallel}}\right)^2 + \left(1 - P_{\perp\parallel}^{(+)} \frac{R_{\perp\parallel}^{(+)}}{R_{\perp\parallel}}\right)^2 \left(\frac{\sigma_2}{R_{\perp\parallel}^{(+)}}\right)^2} + P_{\perp\parallel}^{(+)} \frac{\sigma_2}{R_{\perp\parallel}} = 1 \quad \text{mode A : } (\sigma_2 \geq 0), \quad (4.7)$$

$$\frac{1}{R_{\perp\parallel}} \left(\sqrt{\tau_{12}^2 + \left(P_{\perp\parallel}^{(-)} \sigma_2\right)^2} + P_{\perp\parallel}^{(-)} \sigma_2 \right) = 1 \quad \text{mode B: } \sigma_2 < 0 \quad \text{and} \quad 0 \leq \left| \frac{\sigma_2}{\tau_{21}} \right| \leq \frac{(R_{\perp\perp}^A)}{|\tau_{21c}|}, \quad (4.8)$$

$$\left[\left(\frac{\tau_{21}}{2(1 + P_{\perp\parallel}^{(-)}) R_{(\perp\parallel)}} \right)^2 + \left(\frac{\sigma_2}{R_{\perp\parallel}^{(-)}} \right)^2 \right] \left(\frac{R_{\perp\parallel}^{(-)}}{-\sigma_2} \right) = 1 \quad (4.9)$$

mode C: $\sigma_2 < 0$ and $0 \leq \left| \frac{\tau_{21}}{\sigma_2} \right| \leq \frac{|\tau_{21c}|}{R_{\perp\perp}^A}.$

The superscript A indicates that the fracture resistance is associated with the action plane. The assumption of $\tau_{21c} = R_{(\perp\parallel)} \sqrt{1 + 2P_{\perp\parallel}^{(-)}}$ is valid and also this formulation $P_{\perp\perp}^{(-)} = P_{\perp\parallel}^{(-)} \left(\frac{R_{\perp\perp}^A}{R_{\perp\parallel}} \right)$ is correct in the case of plane stress, and it leads to

$$R_{\perp\perp}^A = \frac{R_{\perp\parallel}}{2P_{\perp\parallel}^{(-)}} \left(\sqrt{1 + 2P_{\perp\parallel}^{(-)} \left(\frac{R_{\perp\parallel}^{(-)}}{R_{\perp\parallel}} \right)} - 1 \right) \quad (4.10)$$

since the failure criteria functions and the functions for their associated stress exposure factors f_E are the same, Stress exposure factors for three modes may be expressed as

follows

$$f_{E, \text{Mode A}} = \frac{1}{R_{\perp\parallel}} \left(\sqrt{\left(\frac{R_{\perp\parallel}}{R_{\perp}^{(+)}} - P_{\perp\parallel}^{(+)} \right)^2 \sigma_2^2 + \tau_{21}^2 + P_{\perp\parallel}^{(+)} \sigma_2} \right), \quad (4.11)$$

$$f_{E, \text{Mode B}} = \frac{1}{R_{\perp\parallel}} \left(\sqrt{\tau_{21}^2 + \left(P_{\perp\parallel}^{(-)} \sigma_2 \right)^2} + P_{\perp\parallel}^{(-)} \sigma_2 \right), \quad (4.12)$$

$$f_{E, \text{Mode C}} = \frac{\tau_{21}^2}{4R_{\perp\parallel}^2 (1 + P_{\perp\parallel}^{(-)})^2} \left(\frac{R_{\perp}^{(-)}}{-\sigma_2} \right) + \left(\frac{-\sigma_2}{R_{\perp}^{(-)}} \right). \quad (4.13)$$

4.2.2.4. 3D Stress State. Puck and Schürmann established the following Inter Fiber Failure (IFF) criteria for a generalized 3D stress condition of a UD composite element rotated by an angle θ_{fp} from x_2 to the x_n direction, based on Mohr's fracture hypothesis that the stresses on the fracture plane determine a material's fracture limit. All of the following formulations are based on this hypothesis. The stresses acting on the prospective fracture plane, σ_n , τ_{nl} , and τ_{n1} , are easily determined by rotating along the first axis. The failure criteria function is then formed in the fracture plane using the associated stresses and strains.

Stress exposure factor f_E For positive normal stress applied on any sectional plane with an inclination angle of θ can be described as

$$f_E = \sqrt{\left[\left(\frac{1}{R_{\perp}^{(+)}} - \frac{P_{\perp\psi}^{(+)}}{R_{\perp\psi}^A} \right) \sigma_n \right]^2 + \left(\frac{\tau_{nt}}{R_{\perp\perp}^A} \right)^2 + \left(\frac{\tau_{n1}}{R_{\perp\parallel}} \right)^2} + \frac{P_{\perp\psi}^{(+)}}{R_{\perp\psi}^A} \sigma_n \text{ for } : \sigma_n \geq 0. \quad (4.14)$$

For negative normal stress acting on any random sectional plane with an angle θ , 3D

stress state mode can be written as follow

$$f_E = \sqrt{\left(\frac{\tau_{nt}}{R_{\perp\perp}^A}\right)^2 + \left(\frac{\tau_{n1}}{R_{\perp\parallel}}\right)^2 + \left(\frac{P_{\perp\psi}^{(-)}}{R_{\perp\psi}^A}\sigma_n\right)^2} + \frac{P_{\perp\psi}^{(-)}}{R_{\perp\psi}^A}\sigma_n \text{ for } : \sigma_n \leq 0. \quad (4.15)$$

The angle of the fracture plane θ_{fp} affects the stress exposure f_E . To determining angle of fracture plane that ranging between -90° and 90° , f_E is calculated for these angles and the plane with the greatest stress exposure f_E is the plane where fracture is most likely. In determining failure, the angle of the fracture plane is critical; for instance, $\theta_{fp} = 90^\circ$ shows a high likelihood of delamination.

The values of the inclination parameters for CFRP advised by Puck are utilized, as indicated in Table 1 Inclination Parameters values for CFRP.

Table 4.1. Inclination Parameters values for CFRP

Material	$p_{\perp\parallel}^{(-)}$	$p_{\perp\parallel}^{(+)}$	$p_{\perp\perp}^{(-)}$	$p_{\perp\perp}^{(+)}$
CFRP	0.3	0.35	0.25-0.3	0.25-0.3

Material tensile and compression strength in fiber direction and also in matrix direction as well as These four inclination parameters need to be specified in order to determine failure initiation in Composite materials. Unfortunately, if considerable Non-linear deformations are present, Puck's model underestimates ultimate failure stresses and failure envelopes. Another flaw in Puck's phenomenological method is the inclusion of numerous non-physical material properties that might be difficult to measure without extensive knowledge of a certain material system.

4.3. Damage Progression Models

A proper technique should be used to determine the damage progression until the failure occurs after the damage initiation. Because of the significant residual

strength and damaged tolerance that a damaged composite structure may exhibit after the initiation of damage and also the capability of composite structures in damage accumulation, relying solely on failure initiation criteria for design purposes leads to inaccurate predictions and conservative design solutions. As a result, it's critical to understand the nonlinear behavior of composite materials as a result of damage accumulation because the pace and direction of damage propagation affect the structure's damage tolerance and final failure [74]. For material degradation after damage has occurred, two techniques have been examined in this study.

4.3.1. Material Property Degradation Method (MPDG)

How the material degrades is determined by the damage evolution law. When it has met the chosen initiation criterion and the stress exposure reaches a value of one, Further loading will degrade the material property. Material property degradation techniques change the elasticity matrix of the damaged element. In ANSYS, to utilize progressive failure degradation, undamaged material must be linearly elastic. There are two damage evolution choices in ANSYS: instant stiffness reduction (MPDG) and continuum damage mechanics (CDM). A damaged material's constitutive relationship is defined as follows

$$\tilde{\sigma} = [D]\epsilon \quad (4.16)$$

$$\sigma = [D]_d\epsilon \quad (4.17)$$

where $\tilde{\sigma}$ is actual stress measured in the undamaged domain and σ is effective averaged stress over the entire domain in which both damaged and undamaged domains is included. In above equations ϵ stands for Total elastic strain, $[D]$ is undamaged elasticity matrix and $[D]_d$ is damaged elasticity matrix. The damaged elastic matrix of a

fiber-reinforced composite is given by

$$[D]_d = \begin{bmatrix} \frac{c_{11}}{1-d_f} & c_{12} & c_{13} & 0 & 0 & 0 \\ c_{21} & \frac{c_{22}}{1-d_m} & c_{23} & 0 & 0 & 0 \\ c_{31} & c_{32} & \frac{c_{33}}{1-d_m} & 0 & 0 & 0 \\ 0 & 0 & 0 & \frac{c_{44}}{1-d_s} & 0 & 0 \\ 0 & 0 & 0 & 0 & \frac{c_{55}}{1-d_s} & 0 \\ 0 & 0 & 0 & 0 & 0 & \frac{c_{66}}{1-d_s} \end{bmatrix} \quad (4.18)$$

where $[C]$ is compliance matrix of undamaged material, d_f is fiber damage variable, d_m is matrix damage variable and d_s is shear damage variable. Based on the damage variables, the material stiffness is instantaneously decreased when MPDG method is used.

The values of the damage variables cannot be chosen as zero because this creates FE software convergence difficulties, these variables are chosen a value near zero. Moreover, these variables are determined based on the failure mode by ANSYS solver as follows

$$d_f = \begin{cases} d_f^+, & \text{if } \lambda_f^+ > 0 \\ d_f^-, & \text{if } \lambda_f^- > 0 \end{cases} \quad (4.19)$$

$$d_m = \begin{cases} d_m^+, & \text{if } \lambda_m^+ > 0 \\ d_m^-, & \text{if } \lambda_m^- > 0 \end{cases} \quad (4.20)$$

$$d_s = 1 - (1 - d_f^+)(1 - d_f^-)(1 - d_m^+)(1 - d_m^-) \quad (4.21)$$

In which d_f^+ , d_f^- are accounted for fiber tension and compression respectively, d_m^+ and d_m^- taken into account as matrix cracking and matrix crushing. Moreover, ANSYS solver uses actual stress and damage initiation criteria in the domain to calculate the fiber tension failure λ_f^+ , fiber compression failure λ_f^- , matrix tension failure λ_m^+ and matrix compression failure λ_m^- . These parameters, such as tensile fiber stiffness reduction,

compressive fiber stiffness reduction, tensile matrix stiffness reduction, and compressive matrix stiffness reduction, must be defined by the user in the MPDG technique. Choosing these variables directly affects the results of ultimate failure of FRP composite. The review paper conducted by Garnich and Akula [115] used the FEM to conduct a thorough assessment of material degradation models for progressive damage analysis of FRP composites. One of the key problems in this sector, according to the authors, is to accurately measure the residual stiffness and strength of the damaged material at a specific region. Multiple failure mechanisms, directionality of failure, interaction between the undamaged and damaged material, and numerical implementation problems all contribute to the difficulty of simulating Progressive failure analysis in composite.

In modeling isotropic materials, sudden property degradation models give a quick but unrealistic solution. Despite the simplicity of modeling damage progression, the sudden degradation method appears to be appropriate for simulating fiber failure due to the catastrophic nature of fiber breakup. Carbon fibers have great strength and high stiffness; thus, these building materials can carry significant amounts of high loads. When a fiber is damaged under a progressively applied load, the load carried by that fiber is redistributed to undamaged fibers. This redistributed load causes an overload in undamaged fibers and a sudden fracture of the structure. This cascading event generally occurs in a relatively short amount of time. As a result, using a sudden stiffness reduction to simulate fiber failure is sufficient.

On the other hand, sudden property degradation does not offer any information regarding damage evolution, and gradual material degradation models appear to replicate reality accurately, especially when considering the progressive nature of damage buildup in the matrix phase of the composite via crack density progression.

Camanho and Matthews [95] employed a traditional method to analyze the gradual deterioration of mechanically connected joints in composite laminates. They considered the impact of damage by deliberately and abruptly decreasing the engineering properties of the orthotropic material. The reduction in stiffness is achieved by utilizing

an internal state variable d , which is multiplied by the initial modulus of the material, when a given failure condition is met. In fiber tension failure they multiplied the elastic modulus in longitudinal direction to 0.07 and 0.14 when fibers failed due to compression. When matrix failure determined the multiplies transverse elastic modulus of the materil and G_{12} and G_{23} to 0.2, when matrix failure occurs due to compression, they multiplied these values with 0.4.

The traditional failure criteria, such as maximum stress, maximum strain, Tsai–Hill, and Tsai–Wu, cannot be used to generate a matrix failure index. As a result, for an accurate assessment of composite laminate structural failure, an advanced failure criterion, such as the Puck criterion, must be used. The fiber dependent failures and matrix dependent failures could be separately predicted using the proposed model.

4.4. Pin Loaded Structures Strength

The static strengths of single pin-loaded specimens are defined in order to assess their strength.

4.4.1. Net-Tension Strength

The stress at the moment of failure in the net-tension section is determined by

$$(\sigma_t)_{\text{ult}} = \frac{F_{\text{ult}}}{(w - d)t}. \quad (4.22)$$

The variables in the equation are as follows: F_{ult} represents the ultimate failure load of the structure, W represents the joint width at the net-section, t represents the joint thickness, and D represents the hole diameter.

4.4.2. Bearing Strength

According to ASTM D5961 [[114]] The bearing force is the total force that a bearing specimen carries. Bearing strain is the measure of the relative distortion of a hole in a bearing specimen. The calculation involves dividing the displacement of the bearing hole caused by the bearing force by the hole's diameter. Bearing strength is the precise measurement of the bearing stress recorded at a significant point on the bearing stress/bearing strain curve. There are two commonly acknowledged categories of bearing strengths: offset strength and ultimate strength. Bearing stress is the result of dividing the bearing force by the bearing area. The calculation of the bearing strength of a composite material is as follows

$$(\sigma_b)_{\text{ult}} = \frac{F_{\text{ult}}}{d \cdot t}. \quad (4.23)$$

4.4.3. Shearing Strength

The shearing strength is given as

$$(\tau_s)_{\text{ult}} = \frac{F_{\text{ult}}}{2 \cdot e \cdot t} \quad (4.24)$$

where the edge distance is e , which runs parallel to the load, is the distance between the center of the hole and the free edge.

4.5. Modified Simulated Annealing

In order to search for the globally optimum laminate designs, a modified simulated annealing which is similar to the direct search simulated annealing that is proposed by Montaz et al. [116]. A key distinction in modified simulated annealing lies in its utilization of a collection of configurations, similar to the approach employed by genetic algorithms that involve populations. In contrast to the SA algorithm, which only

searches the neighborhood of a single point, the method searches the neighborhood of all the present points in the set.

Acceptance of a newly derived configuration results in the replacement of an existing configuration, excluding the optimum configuration. This ensures that the best current configuration is not lost, thereby avoiding one of the most significant drawbacks of the standard SA, where many good current solutions are replaced by worse solutions, especially at the early stages of the optimization process.

In direct search simulated annealing, the acceptance of a configuration generated during iterations leads to the replacement of the worst current configuration, presenting a drawback that becomes particularly evident with a high number of design variables. Following the initiation of the optimization process, the current configurations rapidly converge around local minimums, excluding the worst one, which is frequently updated by higher-cost configurations at high temperatures. As new configurations are generated in the proximity of the current ones, the search becomes confined to a small portion of the feasible domain, except for the infrequent trials where the new configuration is created around the worst current configuration. To address this issue, an alternative replacement scheme is implemented.

In order to ascertain the design that is optimal on a global scale, it is critical to explore a vast solution domain. In order to generate a new configuration in the closest vicinity, it is therefore advantageous to permit a significant variation in the present configurations, as opposed to implementing minor perturbations.

4.5.1. Modified Simulated Annealing Procedure

An N -number of initial configurations are generated at the beginning of the optimization procedure by arbitrarily selecting values for the design variables within the design domain. N equals $(9 \times NN)$, with NN representing the quantity of design variables. The design variables consist of the patch length (LST), patch width (HST),

layer orientations θ_1 and θ_2 . For this configuration 36 initial configuration generated. For ply orientation variables steps is chosen to 1 or 15 degree through the design variable domain that predefined in the analysis this lead to have discrete values for fiber orientation for optimization. For patch width and length no increment is chosen and these variables could have any values between the predefined domain and these variable value could be considered as continuous variables. In each iteration of Simulated Annealing (SA), a random process must generate a new configuration. First, a random configuration from the list of ones that already exist is chosen. Then, arbitrary modifications are made to the fiber angles and patch sizes to produce an altered setup. New design variables are defined as

$$LST_{new} = LST_{current} + r_{rand}\Delta L_{max}, \quad (4.25)$$

$$HST_{new} = HST_{current} + r_{rand}\Delta H_{max}, \quad (4.26)$$

$$\theta_{1_{new}} = \theta_{1_{current}} + r_{rand}\Delta\theta_{1_{max}}, \quad (4.27)$$

$$\theta_{2_{new}} = \theta_{2_{current}} + r_{rand}\Delta\theta_{2_{max}}, \quad (4.28)$$

here $LST_{current}$, $HST_{current}$, $\theta_{1_{current}}$ and $\theta_{2_{current}}$ patch length, patch width, and layer orientations of the patch that is randomly chosen in the currently configuration. r_{rand} are randomly chosen number between -1 and 1. The ΔL_{max} , ΔH_{max} , $\Delta\theta_{1_{max}}$ and $\Delta\theta_{2_{max}}$ represent the maximum variations that can be applied to the respective design variables to create new patch configurations. The objective functions are determined once the initial laminated composite configurations are selected randomly. The acceptance of a newly generated trial configuration, At , in Simulated Annealing (SA) depends on its objective function value, ft , which is calculated as follows: When a new configuration is generated, its corresponding objective function value is found, determining the setup's acceptance probability.

When the new objective function of new generated configuration is smaller or equal than worst-best configurations objective function the probability is equal to 1 as

shown below.

$$P_{\text{trail}} = 1 \quad \text{if } F_{\text{new}} \leq F_{\text{wb}} \quad (4.29)$$

where the objective function value of the worst-best configuration is indicated by F_{wb} and the objective function value of the new configuration is denoted by F_{new} . The acceptance probability is indicated as P_{trail} . When the objective function of the new generated configuration is bigger than the objective function of worst-best configuration the acceptance probability is calculated as below.

$$P_{\text{trail}} = \exp\left(-\frac{F_{\text{new}} - F_{\text{wb}}}{T}\right) \quad \text{if } F_{\text{new}} > F_{\text{wb}} \quad (4.30)$$

Then the calculated acceptance probability is compared with randomly generated number $[0,1]$.

Acceptance occurs when the computed value exceeds the configuration of randomly produced values; rejection occurs when it does not. One of the poor configurations is replaced if the trial design is accepted. When Temperature which is control parameter is maintained as constant value these iterations referred to as inner loop or i_{th} Markov chain. A new inner loop starts when the control parameter, T , is decreased after a predetermined number of iterations. Any configurations with better objective function than worse objective function are considered as enhancement. There are currently N configurations in use, and each time one is approved, it takes the place of one that was chosen at random from the $2 \times NN$ worst ones. Following every acceptance and replacement, the set is ranked in order of quality. Every Markov chain has a minimum length of $3 \times NN$. Each i_{th} Markov chain iterates, unless a configuration is generated that is superior to the best configuration in the current set. When the number of improvements in the previous Markov chain is less than 10% of the maximum Markov chain length, the length of the Markov chain decreases as follow

$$L_{\text{M_new}} = \text{nint}\left(L_{\text{M_ini}}\left(1 + 2\left(\frac{Rs_{\text{LST_c}}}{Rs_{\text{LST_INI}}}\right)\right)\right) \quad (4.31)$$

where L_{M_new} is the current Markov chain length, L_{M_ini} is initial length of Markov chain, Rs_{LST_c} the current maximum variation of the length parameter and Rs_{LST_INI} is the initial maximum variation of the length parameter. The initial value of Rs_{LST_INI} predefined for every optimization for different case studies by taking account the maximum and minimum values of design space. The maximum variation of the optimization parameters is lowered by 5% and the new maximum value of variation for each parameter is computed as follows

$$Rs_{LST_n} = 0.95 \cdot Rs_{LST_c}, \quad (4.32)$$

$$Rs_{HST_n} = 0.95 \cdot Rs_{HST_c}, \quad (4.33)$$

$$Rs_{\theta_{1n}} = 0.95 \cdot Rs_{\theta_{1c}}, \quad (4.34)$$

$$Rs_{\theta_{2n}} = 0.95 \cdot Rs_{\theta_{2c}}. \quad (4.35)$$

The inner loop and the outer loop make up the optimization algorithm. The temperature parameter is iterated at a constant value in the inner loop, also known as the Markov chain. The temperature parameter is lowered following the conclusion of each Markov chain, which also lowers the likelihood that a configuration that is less desirable than the worst-best configuration will be accepted. The only other situation in which a Markov chain's temperature parameter is not reduced is when a new configuration that outperforms the existing optimal configuration is discovered. The temperature parameter's quantity of decrement is determined by the ratios of acceptance and step size, which are described as

$$R_{acc} = \frac{X_{acc}}{X_{it}} \quad (4.36)$$

$$R_{step} = \frac{Rs_{LST_c}}{Rs_{LST_INI}} + 0.01 \quad (4.37)$$

where X_{it} indicates the number of iterations completed in the current Markov chain and X_{acc} indicates the number of acceptable configurations. By having these values a

constant that is defined as follows is used to control the temperature decrease. To find a temperature reduction factor ratio of acceptance and step size compare with each other and when the value of $R_{\text{acc}} > R_{\text{step}}$ the the temperature reduction factor is $\alpha = 0.8$ and when $R_{\text{acc}} \leq R_{\text{step}}$ the α is chosen to be 0.999. The following is the reduction scheme of temperature control parameter

$$T_i = \alpha T_{i-1} \quad (4.38)$$

The probability that a configuration with a higher cost is accepted decreases with decreasing T . Since the control parameter is initially set so high, The temperature parameter's starting value is set at 2. Acceptability decreases at low temperature parameter values, making it unlikely that configurations with poor objective functions will be accepted. Similar to how temperature affects microstructural changes during physical annealing, the control parameter governs the convergence of the optimization procedure. The temperature is high enough in the early stages of optimization such that, regardless of the value of the objective function, the algorithm can accept nearly any configuration that is generated arbitrarily. A value of 0.01 is added to the specification of R_{step} in order to gradually lower the temperature parameter at first. Consequently, α is initially employed in greater quantities. A new configuration that is more inefficient than the existing configurations should practically never be approved at this stage since it should require such low values when the temperature drops dramatically. When the difference between the worst and best configuration's objective function values is less than 0.15%, the optimization is considered to be completed.

5. PROGRESSIVE FAILURE ANALYSIS of OPEN-HOLE TENSILE COMPOSITE LAMINATES

5.1. Effects of Damage Initiation and Progression in CFRP

5.1.1. Introduction

A parametric study is conducted in this case study to find the effects of different damage initiation factors and material property degradation factors for different open hole configurations. Hashin and Puck failure criteria are used to determine damage initiation. Two approaches in continuum damage mechanics for damage progression are studied. Different damage degradation parameters for four damage modes are investigated.

5.1.2. FEM

Detailed progressive failure analyses are carried out on a structure subjected to open-hole tension. To numerically simulate the structural behavior, a finite element method (FEM) is used. This requires constructing a finite element model that precisely represents the geometry and physical characteristics of the structure being studied. The analysis aims to capture the sequential failure mechanisms that may occur in a structure with a hole. In particular, open-hole tension specimens are considered. The FEM method enables a thorough investigation of the structure's stress distribution, strain patterns, and possible failure modes. The ANSYS APDL environment requires input parameters for the analysis, including material properties, geometric dimensions, and loading conditions. The software's scripting features facilitates the creation of a finite element model, and simulations are conducted to observe the structural behavior under open-hole tension.

A well-articulated collection of assumptions forms the basis of the subsequent analysis. For a correct representation of the orthotropic laminated composite material, all the elastic constants must be input. Second, it is assumed that the material behaves in a linear-elastic manner. Lastly, it is assumed that the laminae inside the composite are perfectly bonded. The evaluation does not consider delamination using an interlaminar model, and its exclusive emphasis is on intralaminar failure. Finally, the analysis is conducted under the plane-stress condition, and it is assumed that the laminate thickness is small in comparison to the lateral dimensions of the plate. The analysis is based on these assumptions, and the required steps for model generation are described below.

5.1.3. Two-Dimensional Modeling

The use of two-dimensional modeling simplifies the complicated behavior of composite structures, making analysis of their failure mechanisms easier. When compared to three-dimensional modeling, this approach is more cost-effective because it uses fewer processing resources and less time. Despite this, two-dimensional models may correctly describe the behaviors of composite structures with cutouts or joints, such as stress distribution and load transfer. Two-dimensional modeling facilitates parametric research and optimization by allowing the modification of various design parameters in order to perform a thorough analysis of their influence on composite structural failure. In the early stages of design, engineers can benefit from two-dimensional modeling. It enables them to make sound decisions and identify critical areas that require further investigation.

5.1.4. Generating Areas for FEM

The plate area is created by specifying the width and length of the composite plate. A square area is created around the hole to refine the mesh near the cutout because a precise value of the stress gradients adjacent to the hole is needed for this model. Then, a Boolean subtraction is performed to create the circular hole. To

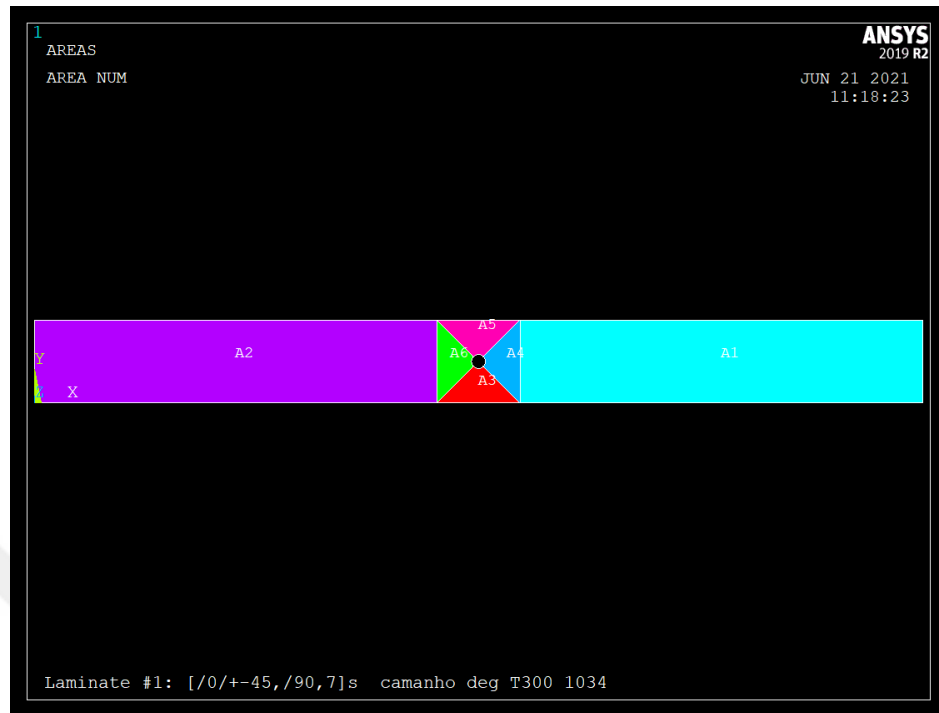


Figure 5.1. Generated areas for modeling OHT.

increase the mesh quality and use mapped mesh, four-sided regions need to be defined. To do so, four lines are created to split the square area that was created around the hole. Once these areas are created, the proper meshing method is specified by the user. In Figure 5.1, the numbered regions are displayed.

5.1.5. Element Types

It is essential to define each part of the geometry with one or more elements depending on the types of structures and analysis. The selection of the right element type by considering its application is critical because it directly affects the Formulation and solution of the finite element model. Using an element with more nodes may give somewhat accurate results. However, computation time and the cost of the analysis will often increase. In this study, SHELL181 elements are used to construct the finite element model.

The SHELL181 elements exhibit robust and accurate performance across a wide range of loading conditions, making them suitable for both thin and thick shell problems. Their formulation permits changes in thickness as a function of in-plane deformation, avoiding transverse shear locking issues. All these elements, except for the small-strain ones, incorporate finite membrane strains in their considerations.

5.1.6. Mesh Generation

The two-dimensional model of the composite plate with open-hole is meshed with SHELL181 element as depicted in Figure 3. Two distinct regions are present in the Finite element model: areas with localized stress and areas without stress concentration. The region around the hole of the composite plate is refined by reason of stress concentrations are expected at the vicinity of the hole. Different mesh refinements are used around the hole of the composite plate for testing the analysis. To reduce computation time and cost, global and local meshes are optimized by reducing the element size around the hole region and by using coarser mesh sizes in global areas. The Finite element model geometry creation process are described in previous sections; this methodology is the same for all configurations. A batch file containing user-written code is utilized to create these models.

5.1.7. Laminate

A completed composite laminate with a cutout, which is used for validation, and an enlarged view showing its 20 individual laminas can be seen below in Figure 5.2 and Figure 5.3. In all models, 2D shell elements are employed. For clarity and descriptive purposes, the elements' shape is displayed in these figures, providing a visual representation of the 2D model. Composite shell sections are composed of layers of diverse materials arranged in different orientations. During the process of analysis, the utilization of integrated shell sections allows for the computation of the cross-sectional characteristics by means of numerical integration throughout the thickness of the shell. The ability to define several material points over the thickness enables

different material responses to occur at each point, but these values could be (1,3,5,7 or 9). The initial number of integration points for each layer is three. However, when there is only one layer and plasticity is present, the number of integration points is increased to a minimum of five throughout the solution process.

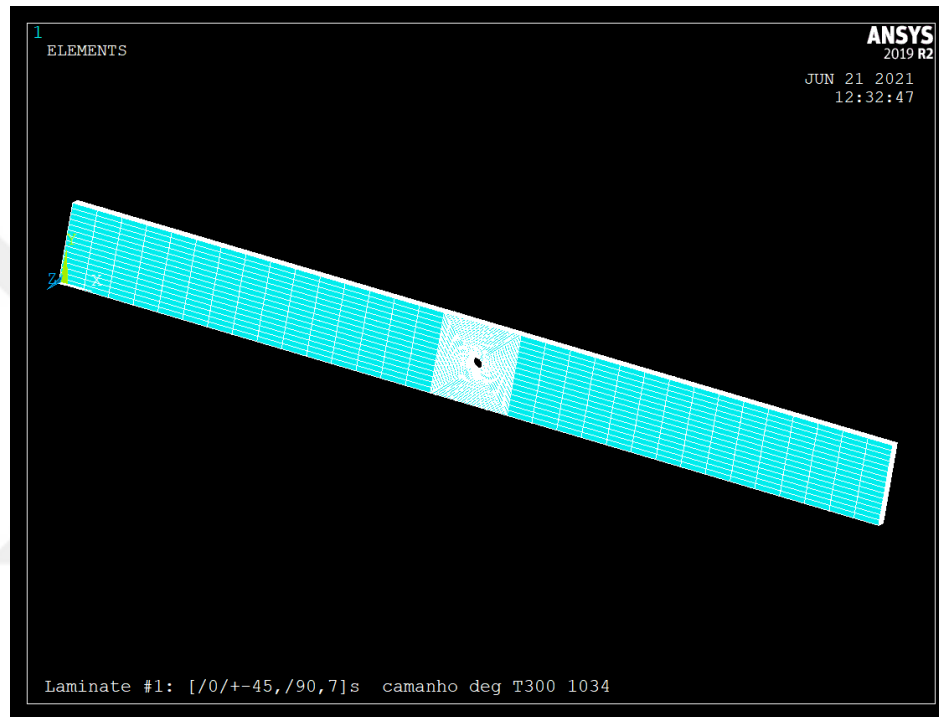


Figure 5.2. Meshed composite laminate using SHELL181 elements.

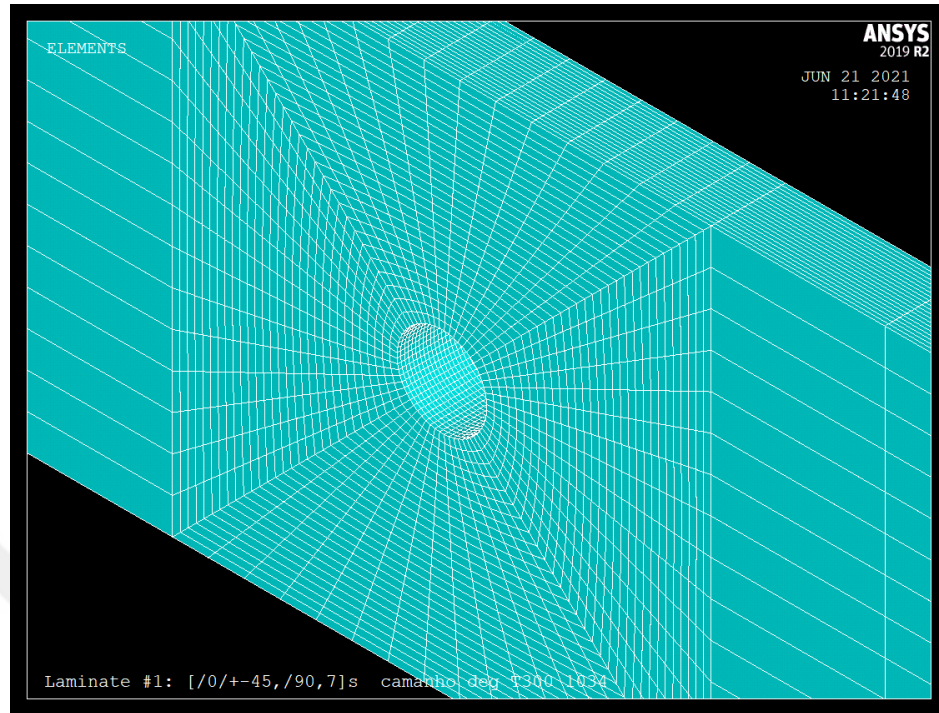


Figure 5.3. Enlarged view of the laminas in the laminate.

5.1.8. Boundary Condition

Accurate finite-element analysis requires proper displacement boundary conditions to ensure results of an analysis that are consistent with the problem intent. Following the principles of statics, adequate displacement boundary conditions are necessary for quasi-static analysis to assure structural stability [117]. It is critical to ensure that the model is not over-constrained since this would cause the stress field to be distorted, misleading results, and a singular stiffness matrix to be produced. The SHELL181 element is composed of four nodes and possesses six degrees of freedom. These degrees of freedom include three translational degrees (U_x, U_y, U_z) and three rotational degrees ($\theta_x, \theta_y, \theta_z$). The stacking sequence of carbon fiber reinforced polymer (CFRP) composite laminates exhibits both balance and symmetry with respect to the mid-plane. Consequently, the coupling effects of tension and shear, as well as bending and torsion, are eliminated in this study. At one end ($x = 0$), all degree of freedom of the composite laminate is considered to be zero along the left edge, resulting in encas-

the boundary condition ($U_x = 0, U_y = 0, U_z = 0, \theta_x = 0, \theta_y = 0, \theta_z = 0$). Displacement is applied to the right edge of laminated Composite. Figure 5.4 depicts the boundary conditions and loading.

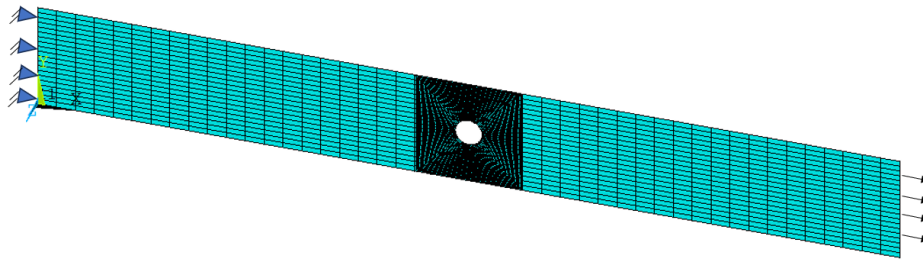


Figure 5.4. Boundary condition and loading of FRP composite with cutout.

5.1.9. Geometric nonlinearity

Unidirectional fiber-reinforced composite lamina is an example of an orthotropic material; because the characteristics of orthotropic materials vary along three mutually orthogonal axes, the response of these materials to a load depends entirely on their orientation towards the load. When a load is applied parallel to the fibers, a small change in the orientation of the UD fiber composite has a considerable impact on the outcomes.

Additionally, the geometry of the lamina and laminate undergoes changes as the load increases. Consequently, the local coordinate systems of the elements and the normal of the face of the elements also experience greater changes. This makes it crucial to understand how the software will generate the equilibrium equation for the actual condition of the fibers regarding to the deformed geometry.

There are two methods for creating the equilibrium for static analysis in the ANSYS APDL program: one is nonlinear and based on actual geometry during analysis, and the other is geometrically linear and based on the original geometry. Geometrically nonlinear analysis is highly suggested when predicting CFRP composite failure

and post-failure, particularly in cases of shear stress or substantial deformation. When analyzing CFRP with significant deformation, it is crucial to consider the direction and magnitude of applied loads, as well as laminate stiffness, in order to accurately evaluate geometrically nonlinear CFRP composites.

For nonlinear problems, ANSYS APDL employs the Newton-Raphson method. The load is divided into a sequence of load increments that can be applied throughout many load steps. Prior to each solution, the Newton-Raphson technique assesses the unbalance load vector, which represents the disparity between the restoring forces and applied loads. Subsequently, the program executes a linear solution by utilizing the unbalance loads and verifies if convergence has been achieved. In the event that the convergence requirements are not met, the load vector that is not in equilibrium is reassessed, the stiffness matrix is revised, and a fresh solution is generated. This iterative process persists until the problem reaches convergence. In this thesis, NLGEOM is activated to include geometric nonlinearity in the analysis by utilizing the Newton-Raphson approach.

5.1.10. Load Application

Forces can be exerted on the nodes of elements or on the surfaces of elements. The integration of distributed forces, such as pressure, is performed over the surface of each element in order to generate equivalent nodal forces. When dealing with complex geometry, it is generally more advantageous to utilize distributed forces rather than applying forces directly to nodes or elements. This is because identifying the nodal force for each refinement can be a hard task, whereas finite element algorithms are capable of automatically calculating the nodal forces based on applied loads and mesh [117].

In order to solve nonlinear analysis, it is necessary to apply the load in small increments. At each increment, the structural response is computed using the stiffness of the elements and the deformation from the previous increment. This technique facilitates the solver in achieving equilibrium at each load increment. The displacements

in the nodes and forces are updated once the equilibrium equation is satisfied, allowing the solution to move to the next increase in load. Tests of structural parts and components can be categorized as either load-controlled or displacement-controlled. The objectives of the study or test will dictate the method of choosing, be it load-controlled or displacement-controlled. In material testing, load-controlled tests are commonly used to investigate how materials react to loads. When doing structural analysis, it is common to employ displacement-controlled tests to find out how the structure reacts to certain displacements or deformations. In displacement-controlled tests, the analysis are often carried out at a pre-determined and constant rate of displacement. This means that the applied load adjusts dynamically based on the stiffness of the structure to ensure a consistent rate of displacement. However, challenges may arise in situations where large deformations or structural failures occur.

In such cases, the solver may encounter difficulties in achieving convergence and solving the equilibrium equations in each time step. To address this issue, a strategy involves defining very small time increments, and activating automatic time stepping. This approach allows the solution to adapt incrementally to the evolving conditions, facilitating convergence even in the presence of significant deformations or structural failures. For these cases which is the case we are going to study through this thesis very small time increment is defined and also by activating automatic time stepping, solution could be converged. Effects of these time steps also studied in following sections. This study involves a geometrically nonlinear analysis where a displacement-controlled load is given to the right edge of the specimen until a displacement of 1.6 mm is reached at the edge of the model.

5.2. Model Verification

5.2.1. Case 1: PFA of T300/1034C Laminated Composite

A progressive damage study of rectangular CFRP composite laminates loaded under in-plane tension with a central through-thickness hole is performed. In this work, twelve Open hole tension configurations are investigated that same as those investigated by Tan [19], Chang and Chang [83]. Some of these configurations were also studied by Miami [74] and Moure et al [118].

The CFRP composite laminates were manufactured from T300 carbon fiber and 1034C epoxy. The material properties are listed in Table 5.1, and these properties are employed in the finite element modeling of CFRP composite specimens. The results are compared to experimental data from previously mentioned studies. For CFRP composite laminate, the values of $p_{\perp\parallel}^{(-)} = 0.3$, $p_{\perp\parallel}^{(+)} = 0.35$, $p_{\perp\perp}^{(-)} = 0.3$, $p_{\perp\perp}^{(+)} = 0.25$ are taken for inclination parameters of Puck failure criteria.

This study employs both the continuum damage mechanism (CDM) and material property degradation (MPDG) technique for damage evolution. In the CDM method, the values of the damage variables are updated based on the failure mode and damage status, and the degradation value begins to increase by increasing the damage to the maximum value of one; when this value reaches one, it indicates that complete failure of that layer in particular failure mode. However, in the sudden stiffness reduction method (MPDG), which is a simpler form of the CDM method, four parameters need to be defined. Parameters used for Tensile fiber stiffness reduction (FT), compression fiber stiffness reduction (FC), Tensile matrix stiffness reduction (MT), and compression matrix stiffness reduction (MC) are shown in Table 5.2. The stiffness reduction ranges from zero to one, zero representing no damage in the element and one representing total damage of the layer. Dimensions of plate width and hole diameter of 12 configurations of CFRP composite plate containing a hole is shown in Table 5.3. The length of the laminated composite is 203.2 mm for all configurations. The stacking sequence of the

CFRP composite laminate for different configurations can be seen in Table 5.3. Four different plate widths and two different hole diameters with different plate widths over hole diameter ratios are investigated for the laminated composite structure. In order to determine the effect of the stacking sequence of laminated composite with a hole, three different stacking sequences are investigated.

Table 5.1. T300/1034C properties

Property	Units	Value
E_1	GPa	146.85
E_2	GPa	11.37
G_{12}	GPa	6.18
ν_{12}	-	0.3
ν_{23}	-	0.42
X_T	MPa	1730.58
X_C	MPa	-1378.95
Y_T	MPa	66.53
Y_C	MPa	-268
S	MPa	58.7
G_{IC}	N/mm	0.228
G_{IIC}	N/mm	0.455
Ply thickness	mm	0.13081

Table 5.2. Parameter values for each material degradation model

	<i>FT</i>	<i>FC</i>	<i>MT</i>	<i>MC</i>
P-0	0.93	0.86	0.8	0.6
P-1	0.9	0.86	0.65	0.6
P-2	0.93	0.9	0.65	0.65
P-3	0.95	0.86	0.65	0.65
P-4	0.93	0.86	0.7	0.7
P-5	0.9	0.86	0.8	0.6
P-6	0.9	0.86	0.75	0.6
P-7	0.9	0.86	0.78	0.6
P-8	0.88	0.86	0.8	0.6

Table 5.3. T300/1034C laminate with cutout for different configurations

Configuration	Plate width (mm)	Hole diameter (mm)	Stacking sequence
C-1	19.5	3.175	$[0/(\pm 45)_3/90_3]_s$
C-2	38.1	6.35	$[0/(\pm 45)_3/90_3]_s$
C-3	15.24	3.175	$[0/(\pm 45)_3/90_3]_s$
C-4	25.4	6.35	$[0/(\pm 45)_3/90_3]_s$
C-5	19.5	3.175	$[0/\pm 45/90_7]_s$
C-6	38.1	6.35	$[0/\pm 45/90_7]_s$
C-7	15.24	3.175	$[0/\pm 45/90_7]_s$
C-8	25.4	6.35	$[0/\pm 45/90_7]_s$
C-9	19.5	3.175	$[0/(\pm 45)_2/90_5]_s$
C-10	38.1	6.35	$[0/(\pm 45)_2/90_5]_s$
C-11	15.24	3.175	$[0/(\pm 45)_2/90_5]_s$
C-12	25.4	6.35	$[0/(\pm 45)_2/90_5]_s$

The load step is divided into three load steps. In the first load step, 0.4mm displacement is applied to the model, and in the second load step, 0.8 mm edge displacement is applied; and in the last load step, edge displacement is set to 1.6 mm. predefined time steps values for each load step are shown in Table 5.4.

Table 5.4. Time Step Configuration

	Main	Max	Min	Auto Time Step
T-0	100	1000	20	on

The OHT analysis results of CFRP composite with a stacking sequence of $[0/(\pm 45)_3/90_3]_s$ and C-4 dimensions are analyzed as a case study. The Puck failure criteria is utilized as the damage initiation criterion, and sudden stiffness reduction (MPDG) is employed as the technique for material property degradation. The study investigate the damage progression in greater depth for this case study. The stiffness reduction parameters are obtained from P-4, as specified in Table 5.2.

By considering reaction force-displacement which is shown in Figure 5.5, this curve have a sharp load drop in 1.22 mm edge displacement, and 1.58mm displacement corresponding to the displacement at ultimate failure, 1.04 mm, 1.22mm and 1.58 mm displacement is chosen for further investigation.

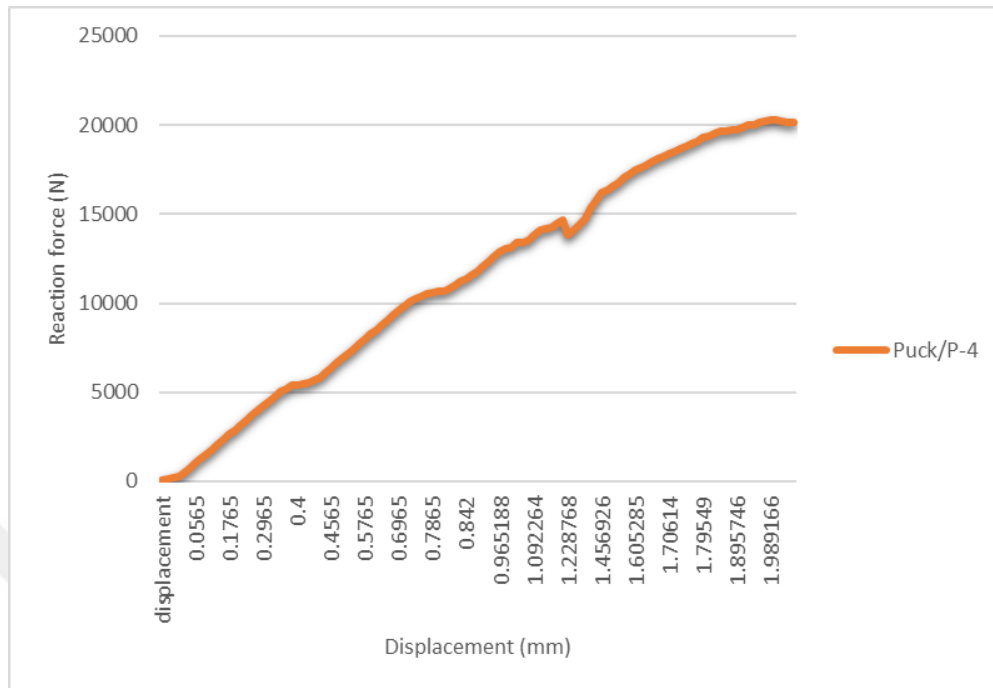


Figure 5.5. Reaction Force vs displacement of C-4 composite plate (Puck criteria / P-4) for MPDG parameters.

For undamaged material, the damage status value is zero; when the load on the element increases, and the failure index reaches to one at each layer in any element, damage is determined and the progression of damage continues as the load increases. The red domain for particular damage mode means that the degradation value is maximum.

Damage status of layer 1, layer 2 and layer 10 of the laminated composites for 1.04mm, 1.22mm and 1.58mm displacement, are shown in Figure 5.6, Figure 5.7 and Figure 5.8 respectively. The angle of the fibers in Layer 1 is zero and parallel to the loading direction. Determined damages in layer 1 are fiber failure due to the tension as shown in Figure 5.9.

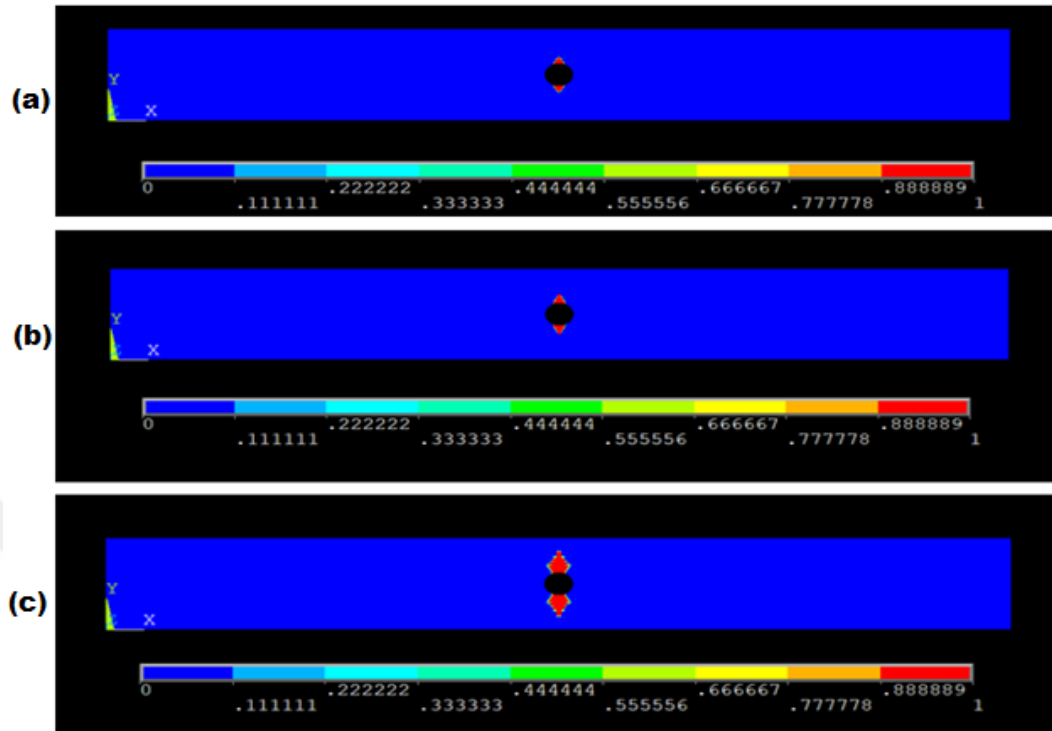


Figure 5.6. Layer 1 damage status of CFRP composite. a) 1.04mm edge displacement
b) 1.22mm edge displacement c) 1.58mm edge displacement.

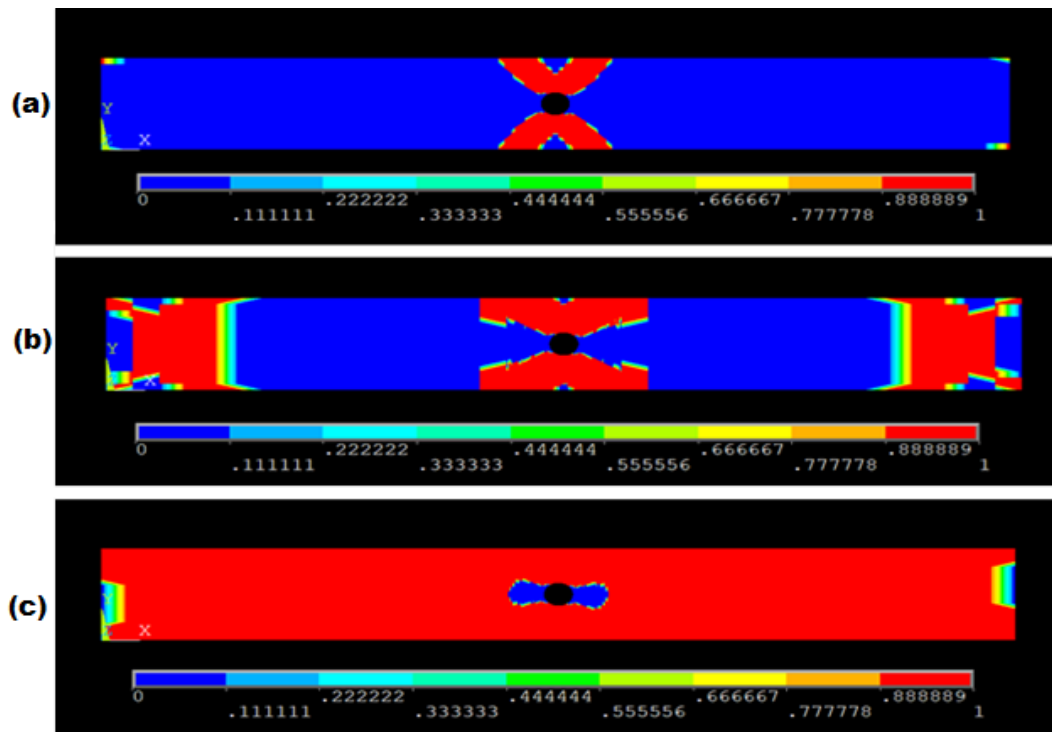


Figure 5.7. Layer 2 damage status contour a) 1.04mm edge displacement b)
1.22mm edge displacement c) 1.58mm edge displacement.

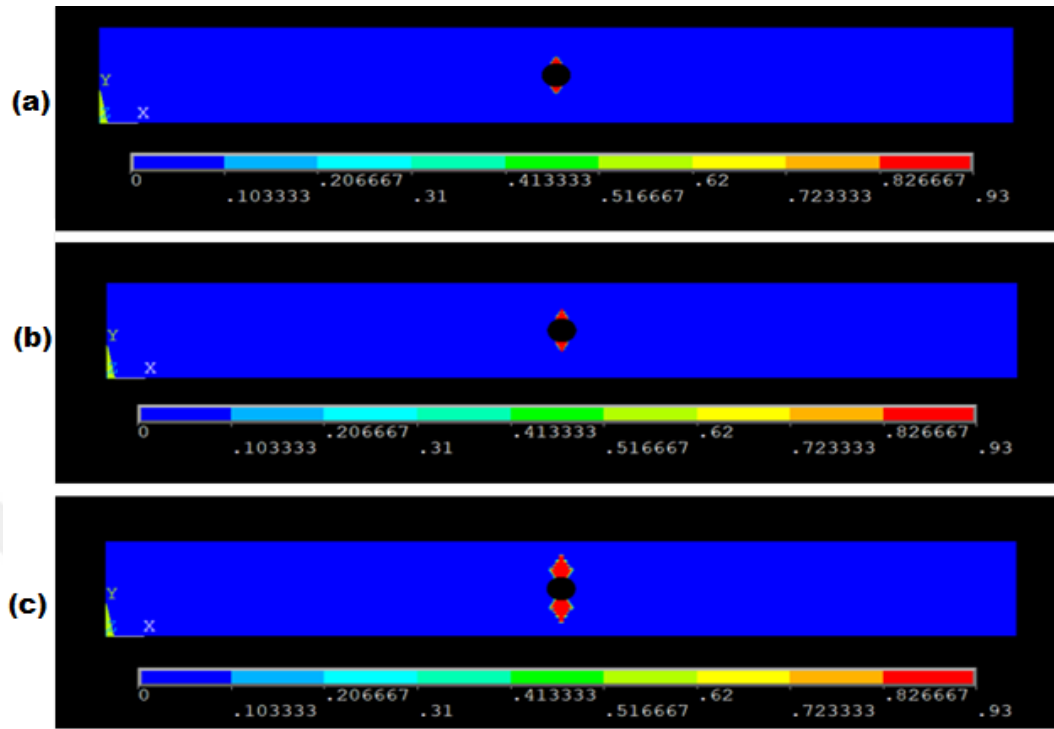


Figure 5.9. Layer 1 fiber failure contour a) 1.04mm edge displacement b) 1.22mm edge displacement c) 1.58mm edge displacement.

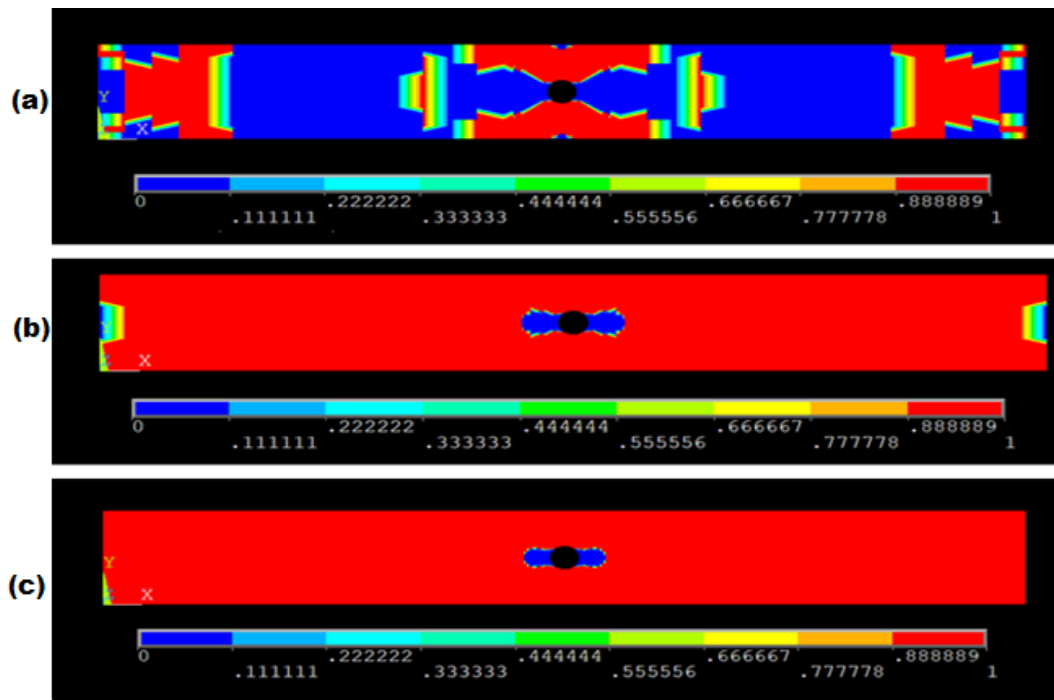


Figure 5.8. Layer 10 damage status contour a) 1.04mm edge displacement b) 1.22mm edge displacement c) 1.58mm edge displacement.

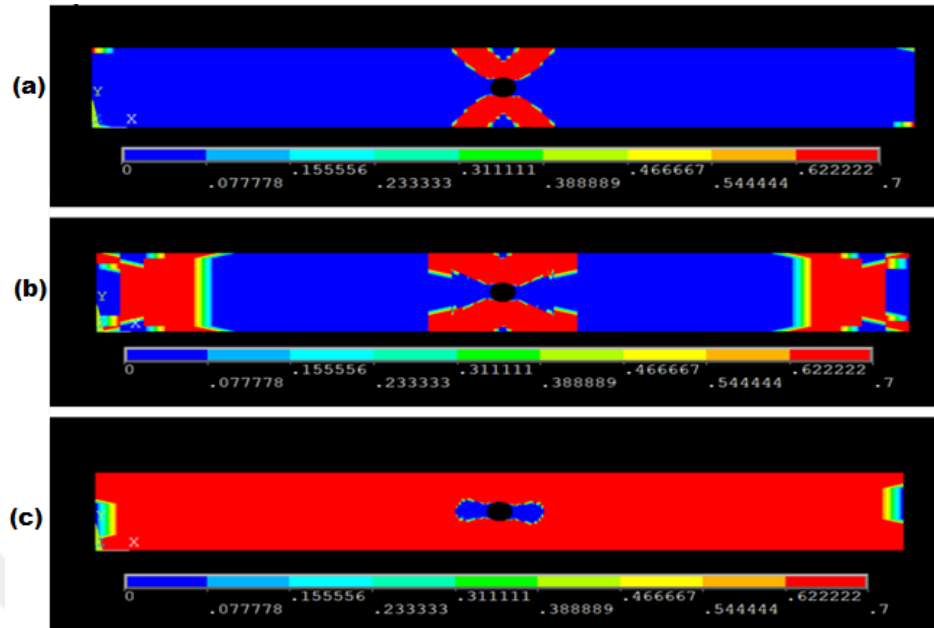


Figure 5.10. Layer 2 matrix failure contour a) 1.04mm edge displacement b) 1.22mm edge displacement c) 1.58mm edge displacement.

Matrix failure is dominant in layers 2 and 10 because the angle of fibers in layer 2 is $+45^\circ$ and the angle of fibers in layer ten is 90° . Additionally, in our case, since tension is applied to the specimen, matrix tension failure indexes are inspected. The matrix failure index of layer two and layer ten are shown in Figure 5.10 and Figure 5.11, respectively. The failure index at any element reached 0.7 means that matrix failure occurs and the stiffness of those elements reduced. The 0.7 value indicates the value of parameters that are chosen by a user as stiffness degradation parameters. The ultimate failure load in these studies corresponds to when laminated composite stiffness reaches 80% of initial stiffness. Comparison between the response of all configurations for Hashin failure criteria as damage initiation and both sudden stiffness reduction and continuum damage method with experimental results can be seen in Figure 5.12. By comparing these results for damage progression methods considered when Hashin failure criteria is used for determining damage we could come to a conclusion that CDM method gives more conservative ultimate failure for these configurations than MDPG in when P-0 parameters for damage degradation values were used as the parameters for stiffness reductions.

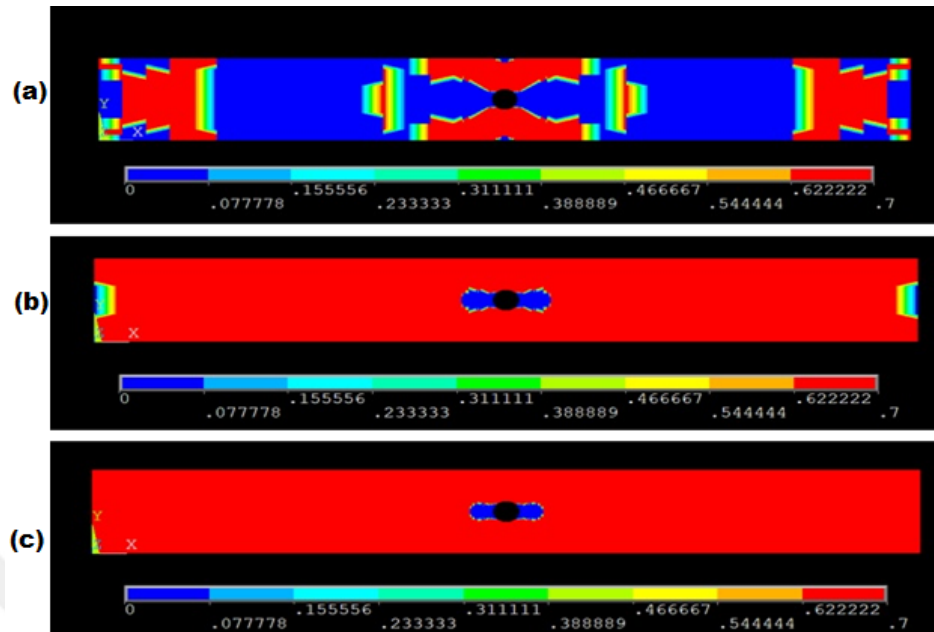


Figure 5.11. Layer 10 matrix failure contour a) 1.04mm edge displacement b) 1.22mm edge displacement c) 1.58mm edge displacement.



Figure 5.12. Experimental vs. MPDG vs. CDM damage evolution methods for all configurations of T300/1034C.

Predicted failure strength for all configurations is shown in Table 5.5. The formula $\sigma_u = \frac{F}{w \times t}$ is used to calculate the final failure stress in the model. F is reaction force when stiffness reaches 80% of the initial stiffness and w is the width of the laminate, and t is the thickness of the laminate. The error percentage of all configurations and the effect of different failure initiation and damage evolution parameters from experimental results are shown in Table 5.6.

Table 5.5. Percentage of error in predicted ultimate failures load of different configurations with experimental data

Config.	Exp.	Hashin P-0	Puck P-0	Puck P-1	Puck P-2	Puck P-3	Puck P-4	Puck P-5	Puck P-6	Puck P-7	Puck P-8	Hashin CDM
C-1	277	259	215	346	327	326	317	219	223	205	218	238
C-2	256	231	215	327	323	318	301	222	231	216	218	218
C-3	226	247	208	323	320	314	288	218	216	208	213	237
C-4	236	225	212	299	299	293	250	212	227	212	212	207
C-5	191	183	140	154	156	154	151	142	150	146	147	167
C-6	159	174	185	157	157	160	154	146	151	147	133	148
C-7	135	180	185	153	153	154	152	131	149	149	130	168
C-8	160	161	178	155	156	160	152	142	147	134	137	141
C-9	236	230	186	300	299	288	248	181	188	190	189	192
C-10	204	209	187	289	276	248	221	191	198	193	187	190
C-11	178	215	182	289	284	277	228	184	185	189	186	204
C-12	186	198	188	266	236	222	202	184	187	184	190	179

Table 5.6. comparison of the results of different failure criteria and damage evolution methods with experimental results

Config.	Hashin/P-0	Puck/P-0	Puck/P-1	Puck/P-2	Puck/P-3	Puck/P-4	Puck/P-5	Puck/P-6	Puck/P-7	Puck/P-8	Hashin/CDM
C-1	-7%	-22%	25%	18%	18%	14%	-21%	-20%	-26%	-21%	-14%
C-2	-10%	-16%	28%	26%	24%	17%	-13%	-10%	-16%	-15%	-15%
C-3	9%	-8%	43%	41%	39%	27%	-4%	-4%	-8%	-6%	5%
C-4	-4%	-10%	27%	27%	24%	6%	-10%	-4%	-10%	-10%	-12%
C-5	-4%	-27%	-20%	-18%	-20%	-21%	-25%	-22%	-24%	-23%	-13%
C-6	10%	17%	-1%	-1%	1%	-3%	-8%	-5%	-7%	-16%	-7%
C-7	34%	38%	14%	14%	14%	13%	-3%	11%	11%	-3%	25%
C-8	1%	11%	-3%	-3%	0%	-5%	-11%	-8%	-16%	-15%	-12%
C-9	-3%	-21%	27%	27%	22%	5%	-23%	-20%	-20%	-20%	-19%
C-10	2%	-8%	42%	35%	22%	8%	-6%	-3%	-5%	-8%	-7%
C-11	21%	2%	62%	59%	56%	28%	3%	4%	6%	5%	15%
C-12	7%	1%	43%	27%	20%	9%	-1%	0%	-1%	2%	-4%

5.2.2. Case 2: PFA of AS4/8552 and IM7/8852 laminated Composite

The three OHT configurations studied in the current case are the same as those qualified by Clarkson [119] for AS4/8552 carbon/epoxy and by Marlett for IM7/8552 carbon/epoxy specimens [80]. The aerospace industry commonly employs these materials because of their exceptional mechanical properties, which perform well under diverse environmental conditions experienced by aircraft. In the qualification test of these prepregs, a number of different batches were tested, and there are very reliable sources for using the provided test results because all of the parameters used in these qualification tests are kept constant. In this case study, we also want to validate our model, and besides damage initiation criteria and damage evolution parameters, we would like to validate our model for different materials, in which AS4 fiber is standard modulus carbon fiber and IM7 is intermediate modulus carbon fiber.

The PFM is performed by using Puck and Schürmann's failure theory and material property degradation implemented in the Ansys Mechanical APDL. For generating a finite element model, all of the steps used in Case 1 are used for this Case except some of the differences that are described in the following.

The composite laminate, in this case, is made up of sixteen thin laminas for quasi-isotropic and 20 thin laminas for the other two configurations, each having a thickness of 0.18796 mm and 0.18288 mm For AS4/8552 and IM7/8552, correspondingly. These test specimens were tested according to the ASTM D5766, and the dimensions of the plate are $L = 203.2\text{mm}$ and $W = 38.1\text{ mm}$ with an open hole $R = 3.175\text{mm}$. Different engineering elastic constants and strengths for each laminate in the material principal coordinates are shown in Table 5.7 and taken from [80, 119]. We take $p_{\perp\parallel}^{(-)} = 0.3$, $p_{\perp\parallel}^{(+)} = 0.35$, $p_{\perp\perp}^{(-)} = 0.3$, $p_{\perp\perp}^{(+)} = 0.25$ as suggested by Puck and Schürmann for CFRP [6].

Table 5.7. Mechanical properties of IM7/8552 and AS4/8552 provided by NCAMP

	IM7/8552	AS4/8552
F1t (MPa)	2500.66	1995.82
E1 (GPa)	158.51	127.27
F1t (MPa)	2238.17	1871.72
F2t (MPa)	64.05	63.91
E2 (GPa)	8.96	9.2
F1c (MPa)	1716.38	1397.56
E1c (GPa)	138.17	111.48
F2c (MPa)	285.71	267.86
E2c (GPa)	9.72	9.85
G12 (GPa)	4.68	4.82
F12 s5% strain (MPa)	91.14	91.56
F12 0.2s (MPa)	53.50	55.15
v21	0.018	0.021
v23	0.45	0.42
G23 (GPa)	3.08	3.24

First coupon had quasi-isotropic lamination that consists of 25 percent 0-degree 50 percent ± 45 degrees and 25 percent 90-degree fiber direction and had $[45, 0, -45, 90]_{2s}$ lamination scheme that is denoted as OHT-1. The second coupon that considered as soft laminate is made up of 10 percent 0 degree and 80 percent ± 45 degrees and 10 percent 90-degree fiber direction. Stacking sequence of this coupon is $[45, -45, 0, 45, -45, 90, 45, -45, 45, -45]_s$ that is denoted as OHT-2. The third coupon has $[0, 45, 0, 90, 0, -45, 0, 45, 0, -45]_s$ lamination scheme that contains 50 percent 0 degree and 40 percent ± 45 and only 10 percent 90-degree fiber direction that is denoted as OHT-3. For these three configurations, the lamination scheme is shown in Table 5.8. The laminate is loaded by applying axial displacements at the right edge while the left edge is fully constrained in all directions. The meshed model is shown in Figure 5.13.

Table 5.8. Lay-up configuration of OHT

Configuration	Stacking sequence
OHT-1	$[45, 0, -45, 90]_{2s}$
OHT-2	$[45, -45, 0, 45, -45, 90, 45, -45, 45, -45]_s$
OHT-3	$[0, 45, 0, 90, 0, -45, 0, 45, 0, -45]_s$

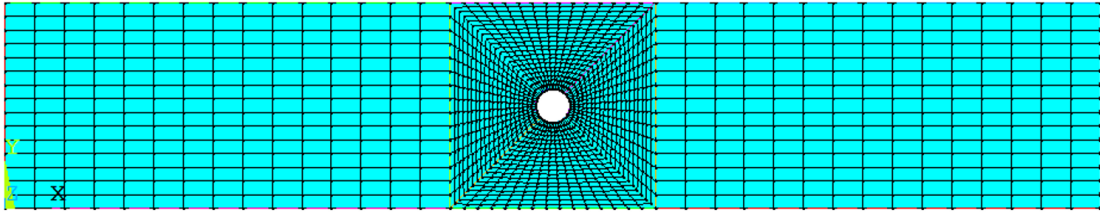


Figure 5.13. 2D Meshed model for OHT Case 2.

The new degradation parameters are chosen for this progressive analysis and are shown in Table 5.9. Also, the new sub steps values ($T4$) are chosen for this and upcoming PFA and the effect of these values are discussed in the Case 1.

Table 5.9. Material property degradation parameters used for OHT Case 2

Parameter	Value
FT	0.999
FC	0.9
MT	0.8
MC	0.8

5.2.2.1. Results for OHT Made with IM7/8852. The progressive failure model is created for IM7/8852 carbon epoxy specimens, and the predicted results are compared with experimental results reported by Marlett [80]. The presently predicted load-displacement response of the OHT1, OHT2, and OHT3 of laminate composite with

IM7/8552 material are shown in Figure 5.14. According to Figure 5.14, for quasi-isotropic laminated composite with an open hole, Maximum load is predicted at 44117 N when the specimen is subjected to 1.39 mm displacement. For soft specimens, the predicted ultimate failure load is 42737 when the edge displacement is 1.59mm. For hard OHT3 configuration, the ultimate failure is predicted at 75514 N, and the displacement is 1.20mm. When we compare the predicted displacement at failure loads, these results are in good agreement with the logic that hard laminates should fail earlier than soft and quasi-isotropic laminates. The strain and displacement values are not available in the qualification test data in order to compare these results.

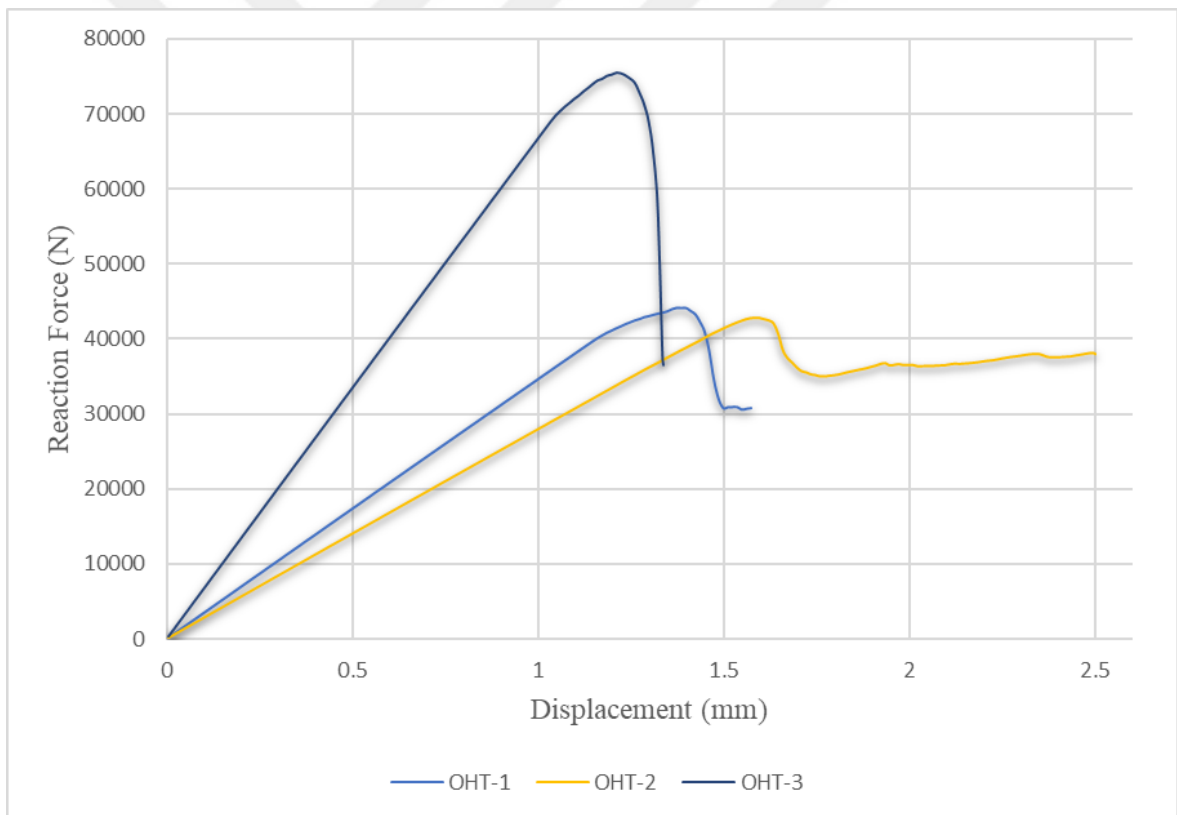


Figure 5.14. Experimental vs. PFA predicted ultimate failure strength for the three different tested laminates manufactured with AS4/8552.

Although standard test and manufacturing methods in controlled environment were used for qualification of composite material but Variations in laminated composite results occur due to material and manufacturing defects, variations in conditions, and differences in testing parameters. As can be seen in Table 5.10 Reported Maximum and Minimum Values for different OHT with IM7/8552 there is about 20% variations in maximum and minimum values of reported test results for some configurations. Therefore, designers used A-basis or B-basis values for different reliability requirements during the design of laminated composites.

Table 5.10. Reported maximum and minimum values for different OHT with IM7/8552

Configuration	Min. Strength (MPa)	Max. Strength (MPa)
OHT1	373.14	445.47
OHT2	275.17	316.88
OHT3	543.99	656.17

A comparison between normalized experimental results for IM7/8852 carbon-epoxy with an open hole and the predicted results from our model, considering the specified degradation factors, is presented in Table 5.11.

Table 5.11. Comparing the results of IM7/8552 OHT specimens with PFA results

Configuration	Strength (MPa)	PFM (MPa)
OHT1	406.79	395.84
OHT2	300.96	306.77
OHT3	597.02	541.88

All of the predicted results are in good agreement with the experimental results. Predicted results for OHT1 and OHT2 are also in between the maximum and minimum reported test results and for hard specimens predicted result is 0.3% under estimate the strength of the OHT3 in comparison with minimum test results. When comparing the results with normalized averaged reported values, the percentage of the error is lower than 10% for all of the configurations.

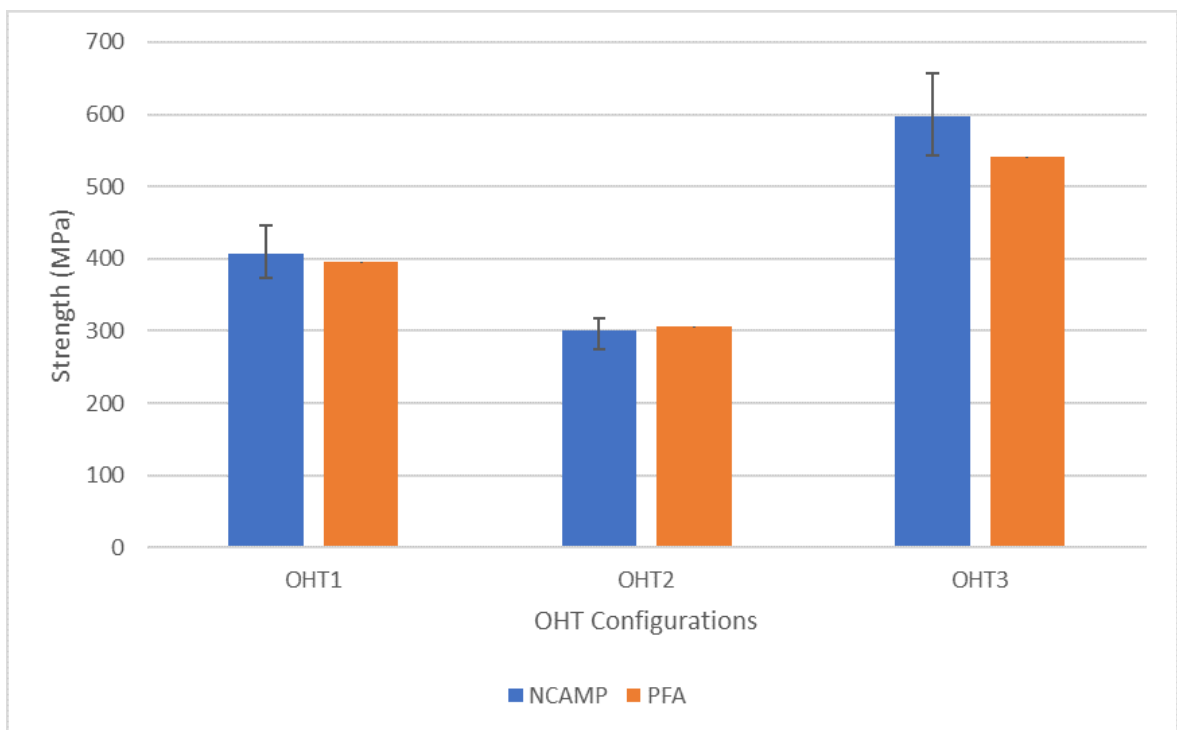


Figure 5.15. Experimental vs. PFA predicted ultimate failure strength for the three different tested laminates manufactured with IM7/8552.

In terms of relative differences, for the three configurations, OHT1 and OHT3, the PDM under-predicted the failure strength by 3% and 9%, respectively, and for OHT2 with soft lamination, it Over-predict the failure strength by 2% in relation to the NCAMP results for IM7/8552 laminated composite with an open hole.

5.2.2.2. Results for OHT Specimens Made with AS4/8852. The progressive failure model is created for AS4-8552 carbon epoxy specimens, and the predicted results are compared with experimental results reported by Clarkson [119]. The minimum and maximum values of experimental results are also shown in Table 5.12. The normalized NCAMP reported test values for all specimens at room temperature compared with the predicted failure strength in Table 5.13.

Table 5.12. Reported maximum and minimum values for different OHT with AS4/8552

Configuration	Min. Strength (MPa)	Max. Strength (MPa)
OHT1	309.64	342.25
OHT2	259.51	280.54
OHT3	437.26	504.21

Table 5.13. Comparing the results of AS4/8552 OHT specimens with PFA results

Configuration	Strength (MPa)	PFM (MPa)
OHT1	328.39	324.92
OHT2	270.06	263.61
OHT3	472.63	433.44

Figure 5.16's bar chart demonstrates how well the numerical model predicts the reported experimental ultimate strength in general. The expected failure strengths are found to be smaller than the experimental averages for all layups, with the largest disparity occurring for the OHT3 example. However, there was some dispersion in the tested specimens' failure strength.

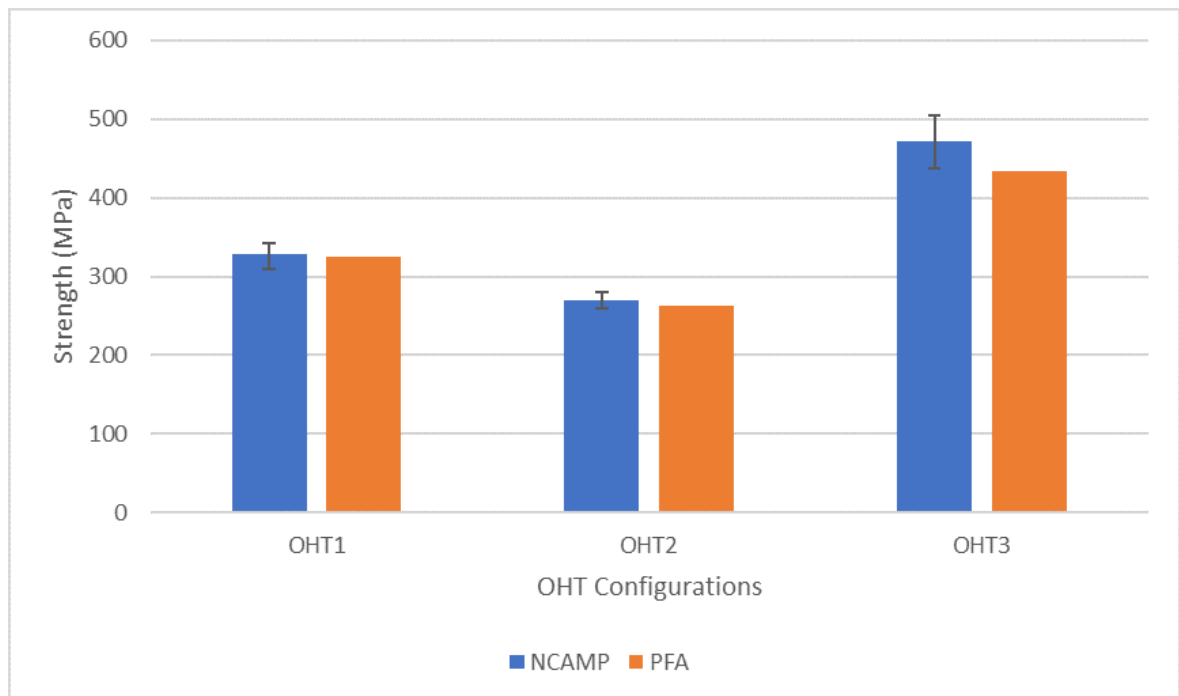


Figure 5.16. Experimental vs. PFA predicted ultimate failure strength for the three different tested laminates manufactured with AS4-8552.

For the three configurations, OHT1, OHT2, and OHT3, the PDM under-predicted the failure strength by 1%, 2%, and 8%, respectively, in relation to the NCAMP results for AS4-8552 laminated composite with an open hole.

5.2.3. Case 3: Schläpfer's Model Verification for Optimization

The progressive failure analysis of the OHT $[0/45/-45]_s$ laminate that has also been studied by Schläpfer and Kress [11] is investigated in this case. The first reason to validate this model is that we are going to optimize the final failure of the model by reinforcing the plate. Secondly, to fully validate our progressive model, when the ratio of the hole diameter over the width of the plate changed, the failure modes and stress redistribution in the model changed, and some studies investigated the effect of this ratio [120]. In Case 2, this ratio is 6, and in the present study, this ratio is 4.

A model is generated for a rectangular composite plate with a circular cutout in the center of the plate, a length of 200mm and a width of 50mm, and the hole radius was 10mm. The layer thickness is 0.25mm. The model geometry and the material properties are the same as those used in [11], and the meshed model is shown in Figure 5.17. The material property that is used in PFM is shown in Table 5.14. Firstly, the effect of timesteps is investigated in this case study. Also, a mesh convergence study is conducted, and according to that, the best element size is used in the analysis.

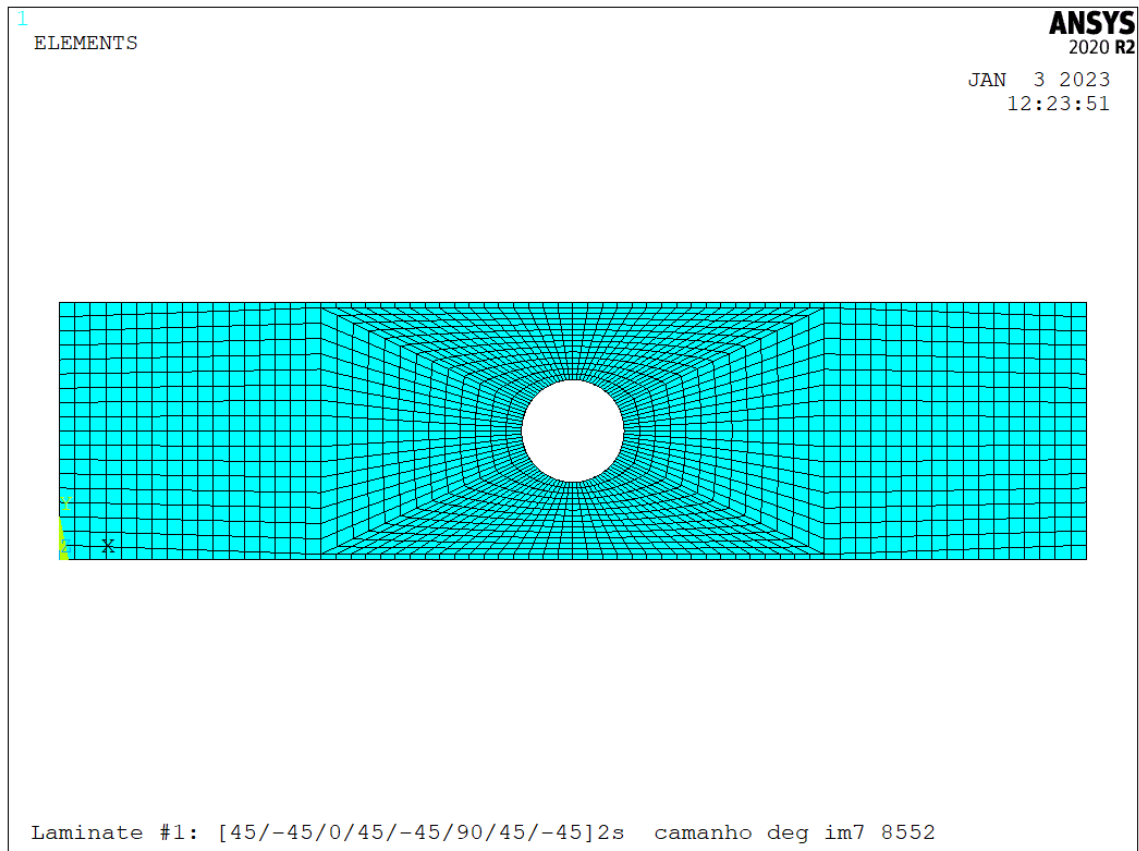


Figure 5.17. 2D meshed Model for Case 3

Table 5.14. Carbon-epoxy mechanical properties

Property	Units	Regular CFRP
E_1	GPa	130
E_2	GPa	9
G_{12}	GPa	4.25
ν_{12}	-	0.34
X_T	MPa	2330
X_C	MPa	-1500
Y_T	MPa	47
Y_C	MPa	-235
S	MPa	79

5.2.3.1. Effects of Time Increment on Progressive Failure Analysis. In Finite Element Analysis (FEA), the time step plays a crucial role in determining the accuracy, stability, and computational efficiency of the simulation. The time step represents the intervals at which the analysis calculates and updates the solution over time. The choice of time step size significantly impacts the accuracy of predicted failure load with progressive failure analysis because material degradation happens during the analysis, and the stiffness matrix of each element changes in each iteration if failure happens. If the time step is too large, it leads to inaccuracies, numerical instability, or even an uncoversgent solution. Conversely, a very small time step increases computational cost without a significant improvement in accuracy. Optimal time step selection is involved in balancing computational efficiency and accuracy through iterative adjustments. Sensitivity analyses and convergence studies were done to find the most suitable time step for our cases, and the results for different time steps are shown in Table 5.15. ANSYS also changes those time increments automatically by considering different convergence criteria when the auto time step is turned on. Most efficient results are obtained when the auto time step was on, the maximum value for the time step was set to 400, and the Minimum value was set to 200.

Table 5.15. Effect of different time steps in PFA results

Trial	Main	Max	Min	Auto Time Step	Ultimate Failure Load (N)
T1	50	1000	20	on	23248
T2	50	1000	20	off	20187
T3	300	1000	200	on	21382
T4	400	1000	200	on	21509
T5	400	1000	20	on	24214

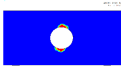
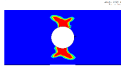
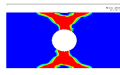
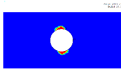
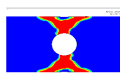
The best results are obtained when T4 time steps were used in the analysis. Therefore, the number of substeps for this loading condition was taken as 400. also, automatic time stepping is activated, and the maximum time stepping could be 1000,

and the minimum of that could be 200 in this analysis.

First ply failure and Last ply failure results of PFA for OHT $[0/45/-45]_s$ laminate. The predicted first ply failure load for this case from PFM is 11.33KN, which is only 1.7% higher than the experimental results and 13.2% higher than the first ply failure detected by DIC. According to experimental results, the first failure occurred when the load reached 11.14KN. When first ply failure accounted for designing of laminated composite it severely underestimates the load carrying capacity of the laminate.

The ultimate failure load for this case was predicted to be 21.51kN, which is slightly less than the experimental value of 22.79. Comparing predicted results with Schläpfer's [11] testing data, the FEA result underestimates the final failure load by 5.6%. The predicted final failure load is 21.51 KN, and damage progression for loads lower than this value and loads higher than these values are shown in Table 5.16. The color of the elements is used to indicate the damage status. Blue elements indicate no damage, while red elements indicate fully damaged.

Table 5.16. Damage progression in Case 3 Layers

	11326 N	17278 N	21509 N	20137 N	beyond failure
0	first layer	No-Damage	No-Damage		
+45	second layer				
-45	third layer				

The first signs of failure in the laminates are observed at 11.33 KN, specifically in the +45 and -45 degree plies of the second and third layers. Similar damage patterns are also seen in the symmetric plies, but no damage is found in the zero-degree plies. As the load and displacement increase, the damage in the ± 45 plies also progresses. At 17.28 KN, no failures are detected in the zero-degree plies. As the load continues to increase and reaches the final failure point, damage begins to appear in the zero-degree layers, and the damage in the ± 45 plies spread to the edges of the laminate. As the displacement was slightly increased after the final failure, the load started to drop, and the damage in the ± 45 plies spreads to the edges of the laminate, and a substantial portion of fibers in the zero direction were also damaged.

6. PROGRESSIVE FAILURE ANALYSIS IN CFRP UNDER COMPRESSION

6.1. Introduction

The design of composite structures often involves cutouts and holes for joints or accessibility, which might become crucial regions under tensile or compressive strain. Compression failure of fiber composite laminates containing an open hole is commonly triggered by the propagation of a micro-buckle originating from the edge of the hole [120,121]. This process differs from tension, as it is commonly acknowledged that fiber breaking is a primary factor leading to ultimate failure. In the aerospace sector, it is usual to determine the allowed strengths of notched designs by considering the stress concentrations in features such as fastener holes and accounting for them in the gross section stress [123]. These notched strengths act as restrictive design allowances for numerous structures. The ASTM D6484 is the most commonly utilized standard test procedure for measuring the Open Hole Compression (OHC) strength. This test is utilized to assess the effects of flaws and defects on aeronautical structures and examine the structure's ability to withstand damage, known as damage tolerance. The damage events in OHC specimens may lead to catastrophic failure. As a result, it is critical to evaluate and predict these structures' slow degradation behaviors in order to ensure their safety and dependability. To understand the response of composite materials under compression loads and to develop accurate predictive models ensuring the structural integrity of aerospace components, the implementation of progressive failure analysis becomes crucial. By performing progressive failure analysis of these structures, one could reduce the necessity for several tests if good correlation with experimental results could be achieved.

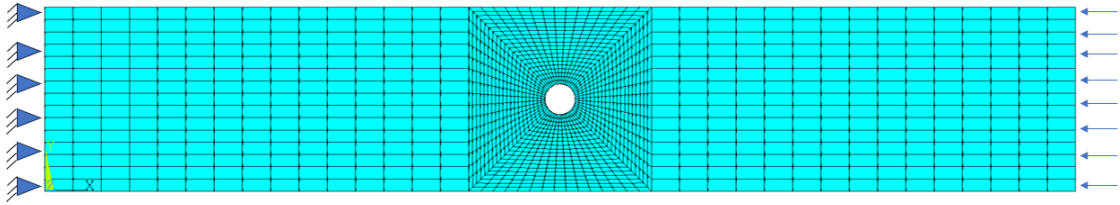


Figure 6.1. Boundary condition and meshed part of OHC

6.2. Problem Description

To find the OHC strength of laminated composite, PFA is performed. All material properties are taken from [80], shown in Table. Same as OHT models puck failure criteria is used and FPPP damage degradation parameters are used for damage progression in the specimens. The loading condition is shown in Figure 6.1. The laminates stacking sequences also same as OHT test case except OHC1. Staking sequence for OHC1 specimens is $[45, 0, -45, 90]_{3s}$ and the number of plies is 24. ASTM D6484 test method that is used for testing these specimens by considering the fixture of the OHC test all of the transitional degrees of freedom in Z direction are constrained and both transitional and rotational degree of the freedom of the left edge of laminate is fully constrained. To apply compression to the models -2.5mm displacement is applied to the right edge of the models.

6.3. Results of OHC

The predicted specimen load versus edge displacement responses of OHT1, OHT2, and OHT3 configurations are shown in Figure 6.2. The percent errors are calculated using the experimentally reported values and are shown in Table 6.1. Good overall agreement is noted for OHT1 and OHT2 cases in terms of final failure strength. The Hard laminate strength is overpredicted by about 18%, whereas the strength for the other laminates agrees within less than 5%. One reason for this difference could be delamination, which is not considered in this analysis. The quasi-isotropic and Soft

laminates tests show a slightly nonlinear response after the first load drop in comparison with the hard laminate. This small nonlinearity in these laminations could be due to the shear nonlinearity characteristic in $\pm 45^\circ$ laminates in which soft and quasi-isotropic lamination have more percentage of these laminates.

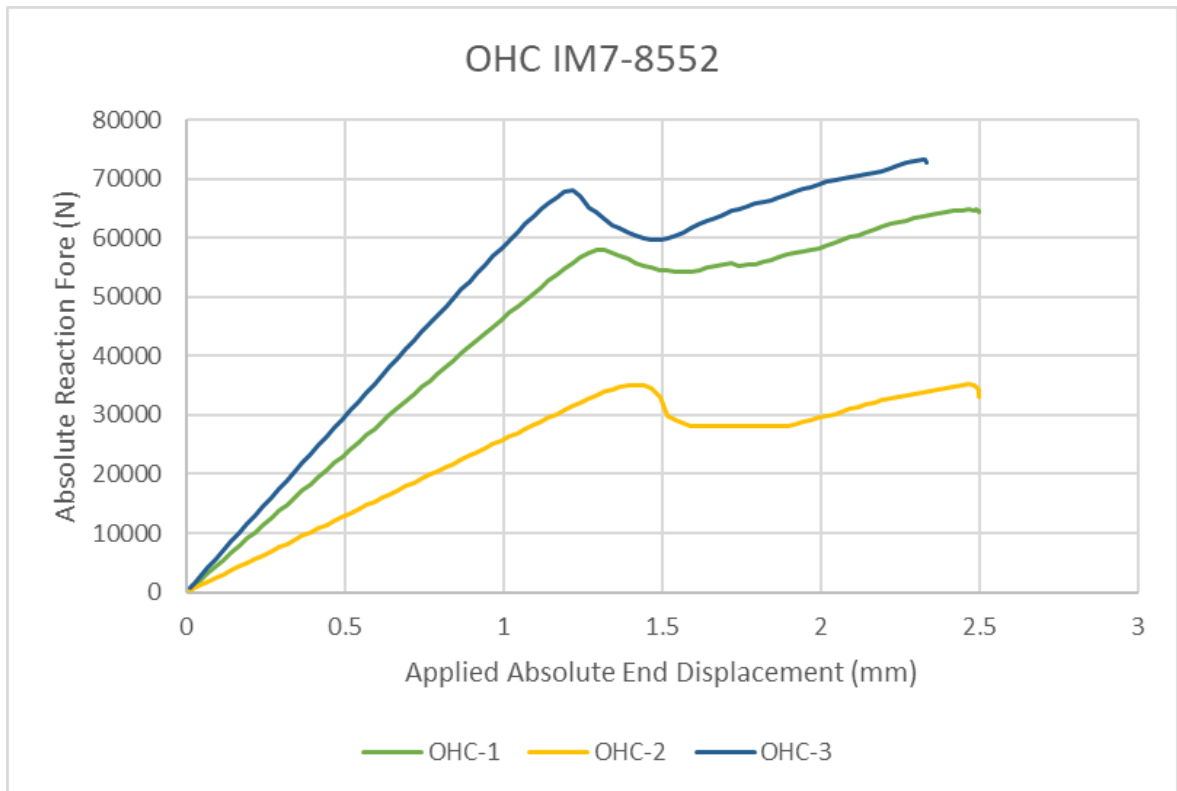


Figure 6.2. Predicted absolute reaction vs. displacement for OHC specimens

Table 6.1. Comparing the OHC PFM results with experimental results provided by NCAMP

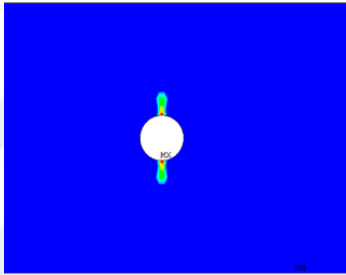
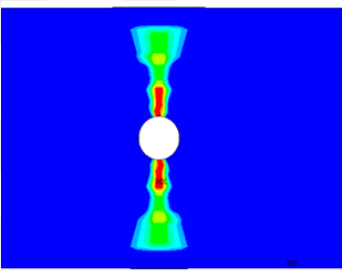
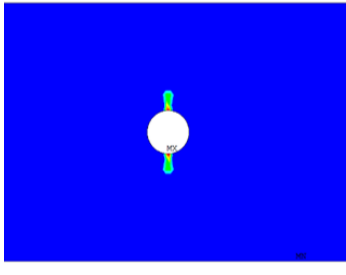
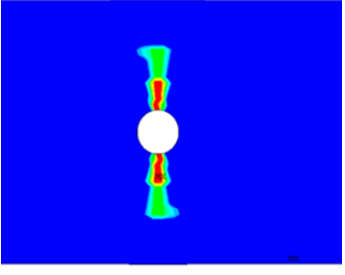
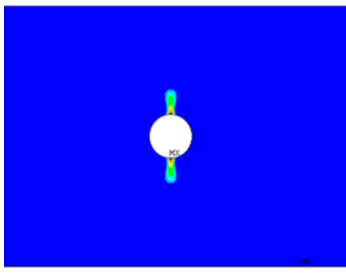
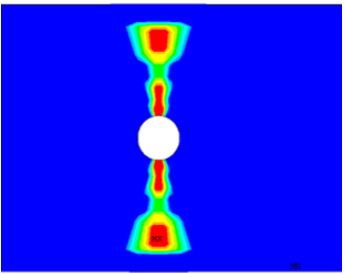
Configuration	NCAMP (MPa)	PFM (MPa)	Error %
OHC1	338.16	353	4%
OHC2	267.33	268	0%
OHC3	435.72	516	18%

6.3.1. Damage Progression in OHC1

The progression of damage at layer 1, layer 2, and layer 4 in OHC1 is shown in Table 6.2, Table 6.3 and Table 6.4, respectively when the peak ultimate failure load happened in the structure happened in which the load is 51015N and the edge displacement is $U_x = -1.29$ mm is chosen as load step 1 and by increasing the displacement to $U_x = -1.76$ mm the reaction load is predicted as 55412N is chosen as $U_x = -1.76$ mm to investigate these layer behaviors. The first ply has 45-degree angle, and the second ply has 0-degree angle, and also fourth ply has 90-degree fiber orientation. In all layers, damage initiates from the hole edge, and the cap $U_x = -1.29$ mm displacement is applied because of these damages, the stiffness of the structure is reduced, and by further applying displacement, these damages in all layers progress.

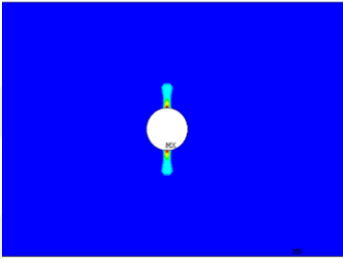
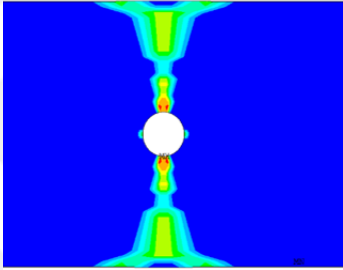
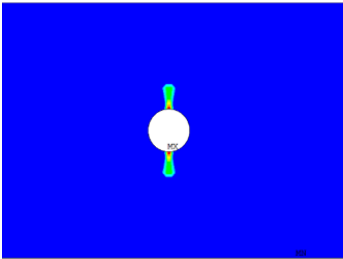
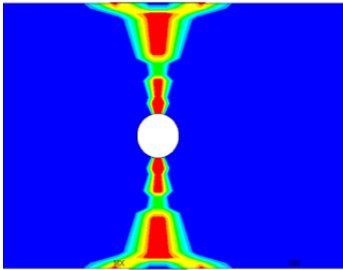
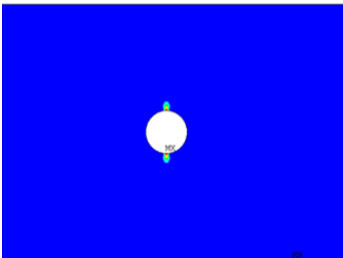
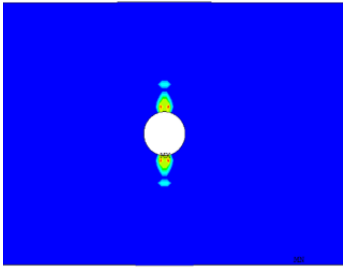
In layer 1 with 45-degree fibers at $U_x = -1.29$ mm, the amount of fiber and matrix failure are the same, but when the displacement increases, matrix failure is more than fiber failure as shown in Table 19. These layers carried the compression load by transferring the compression load into shear.

Table 6.2. Layer 1 damage progression in OHC-1

		$U_x = -1.29$ mm	$U_x = -1.76$ mm
Layer 1	Stat		
	FC		
	MC		

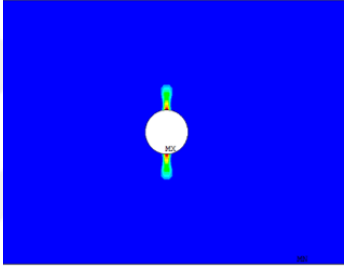
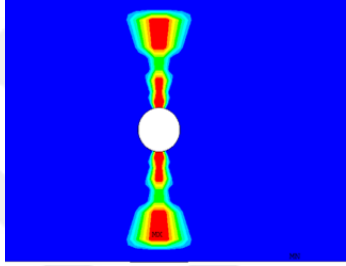
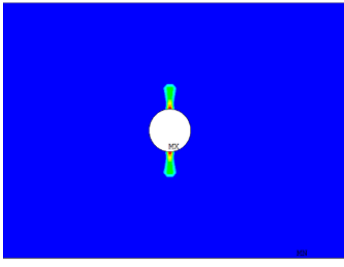
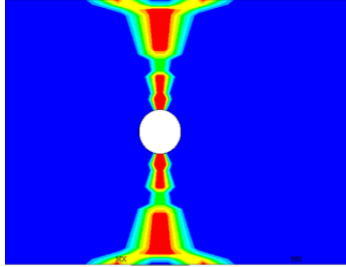
In layer 2 with 0-degree fibers, matrix and fiber compression are observed, but the amount of fiber failure due to compression is significantly higher than matrix compression failure as shown in Table 6.3. When -1.76 mm displacement is applied these fiber compression failure reaches to the edge of the plate.

Table 6.3. Layer 2 damage progression in OHC-1

		$U_x = -1.29\text{ mm}$	$U_x = -1.76\text{ mm}$
Layer 2	Stat		
	FC		
	MC		

In layer 4, where the fiber orientation is at 90 degrees (perpendicular to the loading direction), there is no observed fiber damage, shown in Table 6.4. However, matrix failure is evident due to compression. Although the matrix damage in $U_x = -1.76$ mm does not extend to the edge of the plate, a substantial portion of the fibers at 90 degrees near the hole experiences failure across the width of the plate.

Table 6.4. Layer 4 damage progression in OHC-1

		$U_x = -1.29$ mm	$U_x = -1.76$ mm
Layer 4	Stat		
	FC	No Damage	No Damage
	MC		

To provide a clear illustration, the element shape is activated. For $U_x = -1.76$ mm, the damage status of fiber compression and matrix compression is depicted in Figure 6.3 and Figure 6.4, respectively. A fiber compression damage status of 0.9 indicates complete failure of these fibers due to compression; thus, these elements are incapable

of carrying further load. Fiber damages have also extended to the edge of the plate in certain fiber orientations, with a significant portion of the fibers along the width and near the hole edge experiencing failure.

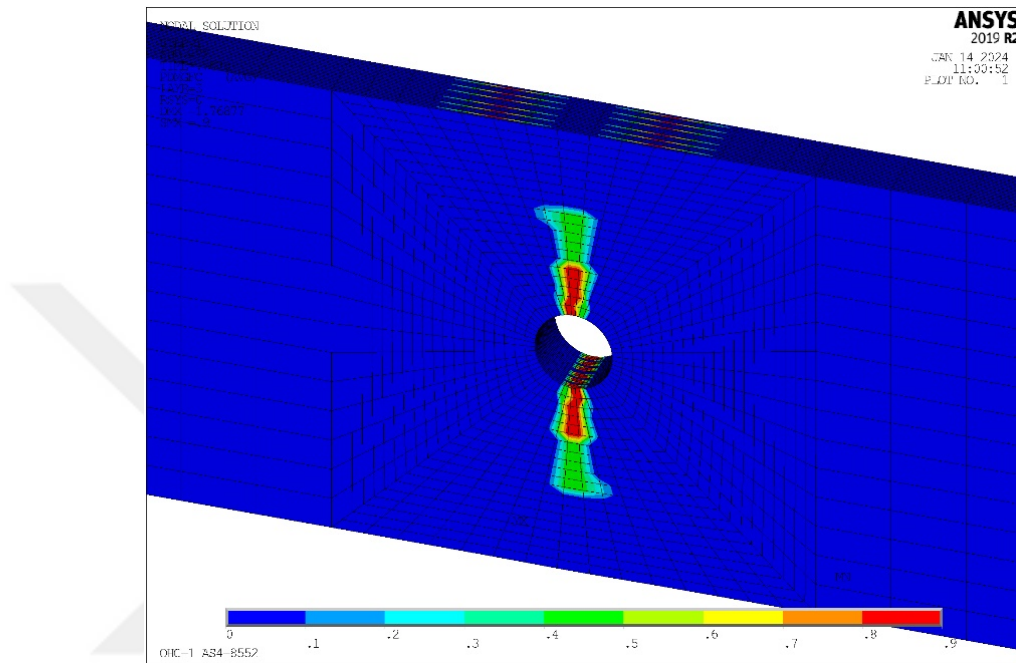


Figure 6.3. Fiber compression damage status for OHC-1 in $U_x = -1.76$ mm

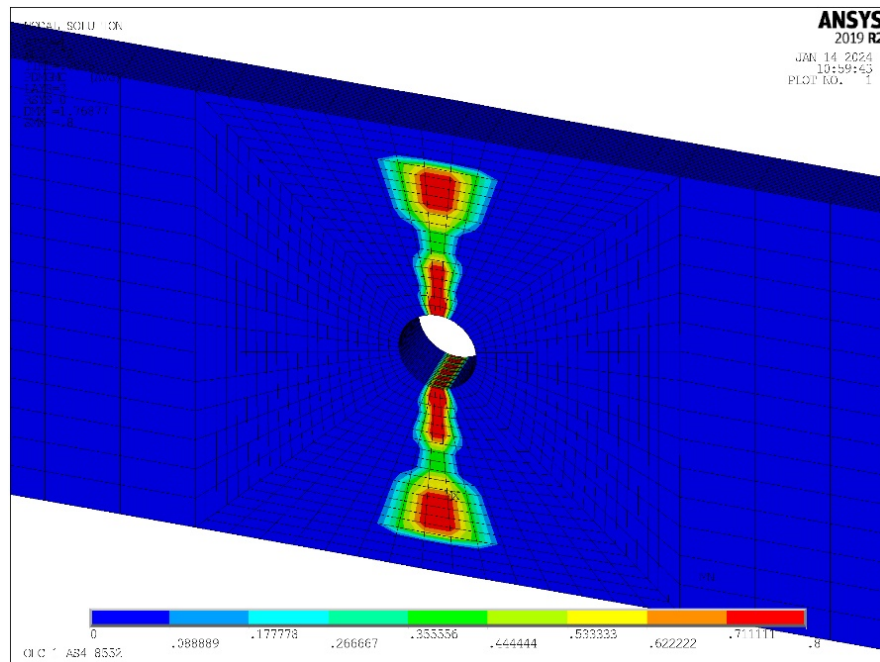


Figure 6.4. Matrix compression damage status for OHC-1 in $U_x = -1.76$ mm

7. PROGRESSIVE FAILURE ANALYSIS OF PIN LOADED CFRP

7.1. Introduction and Overview PFA of Pin Loaded Laminated Composite

This chapter presents the development of a progressive failure model using the ANSYS APDL finite element code. The model is specifically designed for analyzing pin-loaded composite assemblies. The utilization of ANSYS APDL scripting involves the creation and execution of finite element models, taking into account a range of user-defined parameters such as plate dimensions, hole diameter, stacking sequence, ply-thickness, lamina elastic moduli, and strengths, as well as additional parameters related to finite element analysis (FEA). By applying the Puck failure criteria and the discussed damage progression methods throughout the entire area, and by assessing the concentrated force applied to the cutout region, which showed failure indices surpassing unity, estimations were made regarding the maximum load-bearing capacity. The reaction force obtained from these simulations is used to estimate the bearing strength of the specimen.

7.2. Finite Element Modelling

The ANSYS APDL implicit solver is used in numerical simulations due to its efficacy in dealing with static and low-speed dynamic events. Laminated composite plate and pin assemblies, which follow to the specifications stated in the ASTM D5961 standard [114], are created in ANSYS using APDL scripts, as illustrated in Figure 7.1. The dimensions of the model are the same as those of the experimental specimens, as the plate length is $L = 139.7$ mm and the width of the plate is $W = 38.1$ mm (1.5 inches). There is a hole in the centerline of the plate with a diameter of 6.35 mm (0.25 inch), located at a distance of 19.05 mm (0.75 inch) from one end of the plate. The lamination is same as OHT test case and shown in Table 7.1. All parameters utilized for damage initiation and progression in the OHT case study are also applied in this

particular case study. This study examines three frequently used lay-up configurations in the aerospace industry, to explore their behaviors and investigate the progression of damage within these lamination combinations.

Table 7.1. Lay-up configuration of Single-Shear Bearing FEM

Configuration	Stacking sequence
SSB1	$[45, 0, -45, 90]_{2s}$
SSB2	$[45, -45, 0, 45, -45, 90, 45, -45, 45, -45]_s$
SSB3	$[0, 45, 0, 90, 0, -45, 0, 45, 0, -45]_s$

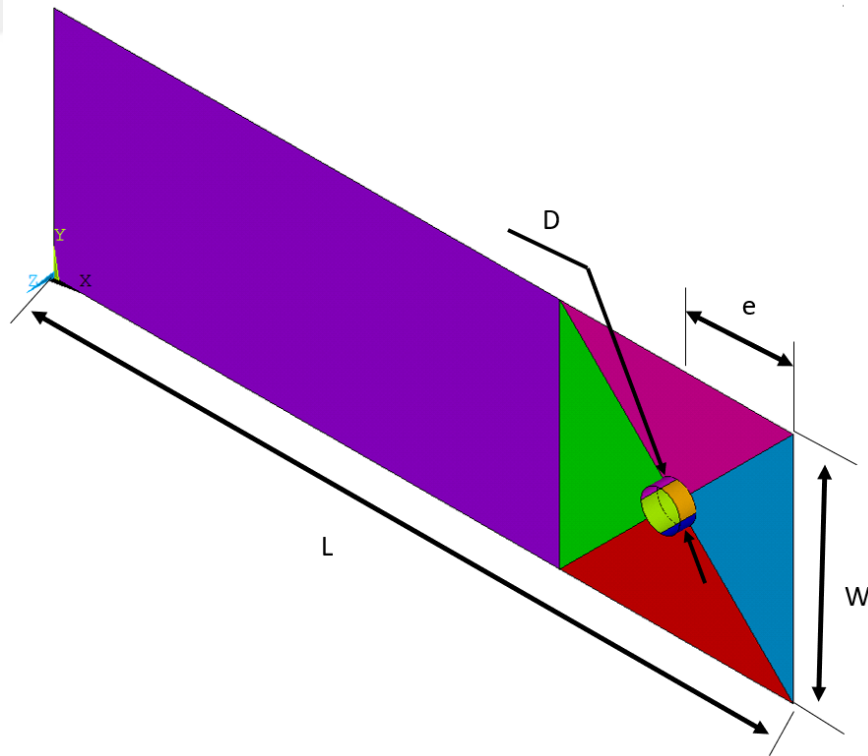


Figure 7.1. Single Shear Bearing FEM model geometry

7.2.1. Meshing

For the composite panel, 2D linear SHELL181 elements with full integration and 6 degrees of freedom (DOF) per node were used. This type of element formulation is suited to thin and moderate thickness composites. The precision in simulating composite shells is determined by the Mindlin-Reissner shell theory, often known as the first-order shear-deformation theory [122]. The global mesh size is set to around 2.0. To achieve precise simulation of pin-hole contact and accurate failure prediction, a higher mesh density is necessary in the region where failure is expected to happen. Area near the hole edge is expected to fail due to the pin compression force. Therefore, local element seeds were applied in that particular area to refine the mesh size in this region. Figure 7.2 displays an annotated image of the plate and pin assembly with mesh.

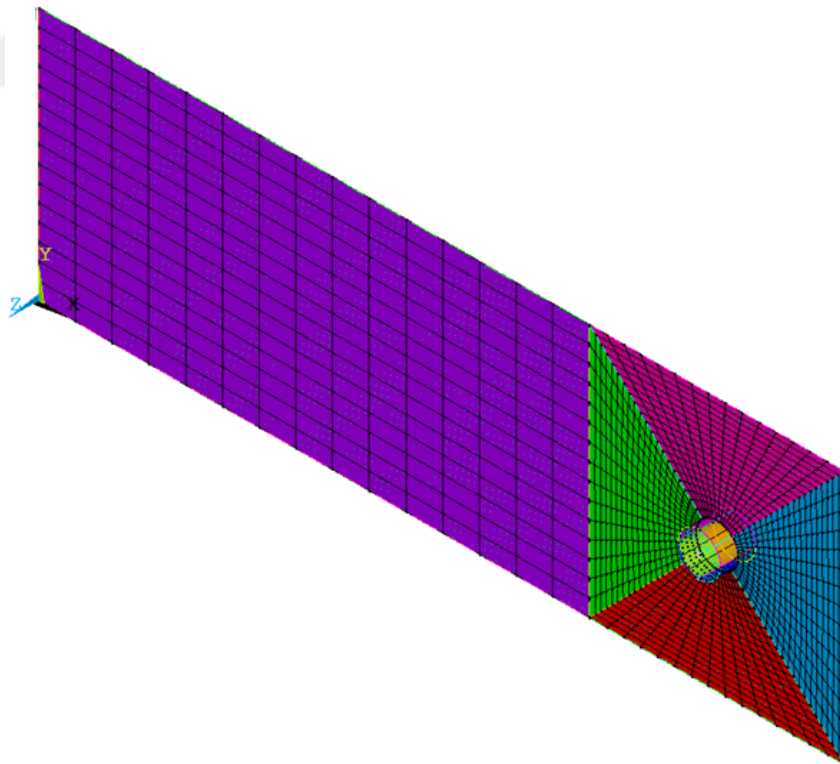


Figure 7.2. Pin loaded structure meshed parts

The pin was modelled as a discrete rigid shell. This simplification is adopted because pin stiffness was shown to have minimal influence on strength predictions for pin-loaded composites in earlier publications [93]. The TARGE170 element is selected for the curved surface of the pin, and the eight-node Quadrilateral form is chosen to accurately represent the curved shape of the elements.

7.2.2. Mechanical Properties

Uni-direction Composite ply properties are defined in ANSYS APDL, where the number of plies and ply angles relative to the mesh orientation—are inputted, and all ply thicknesses are set to a nominal value of 0.18796 mm, which is normalized values reported by [119]. For all plies, moduli and strength properties for the previously utilized AS4/8852 prepreg system are defined and implement—values are provided in Table 8. These properties are taken from a qualification report of AS4/8552 Uni-direction prepreg system that is reported by [119].

7.2.3. Contact

Using the pair-based contact with Augment Lagrange approach, which involves defining a contact pair between the curved pin and plate edge, load transfer between parts is made easier, and by using Augment Lagrange contact algorithm the degree of penetration in relation to the permissible tolerance is assessed. In ANSYS APDL, one can describe the contact between the shell edges using the CONTA177 element, which is suitable for 3D deformable lines [122]. Elements utilized for the hole's edge are contact177 and this element type is compatible to use with TARGE170 target element and are useful for both two- and three-dimensional geometries. This method enforces contact constraints using tracking techniques. Because incorporating friction between the pin and plate increases calculation time and expense by injecting more nonlinearity into the analysis, a frictionless contact is chosen.

7.2.4. Boundary Conditions

In order to horizontally move the pin, a reference point is established at an arbitrary position within the pin and used as the control point for applying constraints. By defining tie-type constraints between the reference point and the pin's nodes, it is possible to indirectly apply a displacement to the pin. Displacement is applied ($U_x=2$, $U_y=0$ and $U_z=0$) to the reference point and left edge of the specimen is fully constraint. To simulate the fixture used in this test case, the degree of freedom in z direction is constrained.

7.3. Single-Shear Bearing PFM Results

Predicted reaction force–bearing strain of all three configurations is depicted in Figure 7.3. For the SSB1 configuration, the corresponding 2% offset bearing strain in this case is 0.06 strain when the reaction force is predicted as 10896.72 N. For ultimate bearing strength of pin loaded quasi-isotropic laminated composite, this value was reported as 934.58 MPa which corresponds to 17847.52 N. The predicted result for ultimate failure load is 15389.96 N. In the SSB2 arrangement, the corresponding 2% offset bearing strain is 0.072 strain when the estimated reaction force is 12634.94 N. The ultimate bearing strength of the pin-loaded soft laminated composite is given as 950.58 MPa, which corresponds to 17847.52 N. The predicted result from PFM for the ultimate failure load is 20014.61N. It can be seen in the figure 5 the soft laminate deform more than the quasi-isotropic and hard laminate in elastic region of the loading and show lower stiffness in comparison with other two configurations. However, as a result of stress redistribution in the plate, the bearing strength can be regarded as a laminate with a high bearing strength. Nevertheless, when designing laminated composites, designers must take into account these deformations, which can potentially lead to issues such as fuel leakage due to significant deformation at the joints of these laminates. In SSB-3 configuration, the maximum load when first load drop is happened is 1485.87 and this is corresponding to 0.2275 mm pin displacement in the structure. Predicted load for 2% bearing strain offset is 17423.64 N and this is occurred when

0.3675mm displacement applied to the pin. The ultimate failure of this specimen predicted to be 21048.94N that is when 0.7125mm displacement is applied to the pin and corresponding bearing strain is 0.112205.

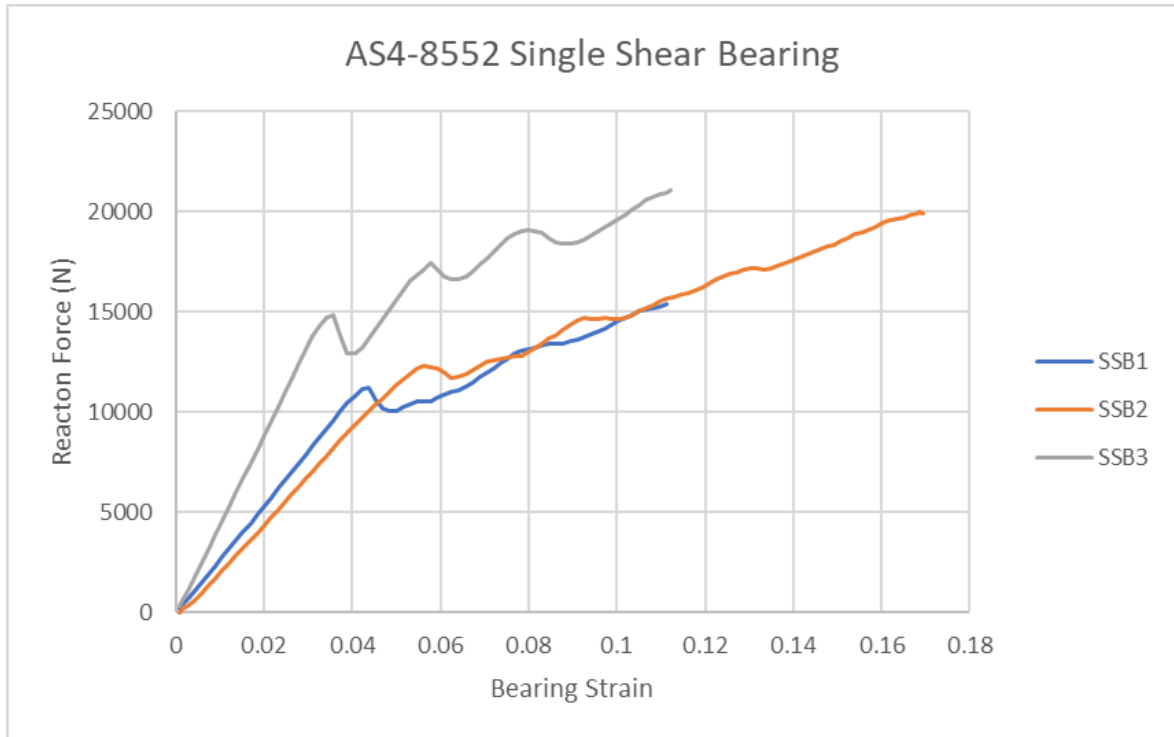


Figure 7.3. Reaction force versus bearing strain for all SSB configurations

7.3.1. Damage Initiation and Progression for SSB2

For the SSB2 configuration, the first matrix and fiber failures are investigated in detail to find the first ply failure loads in this case of laminate composite. Failure initiates at six and fifteen ply with a 90-degree angle and matrix failure is determined with puck failure criteria. No damage is determined in other plies when the pin displacement is 0.2475mm and the corresponding load is 9355.674 N. Figure 7.4 depicts the failure location. Matrix tension failure is observed in the edge of the hole perpendicular to the loading direction.

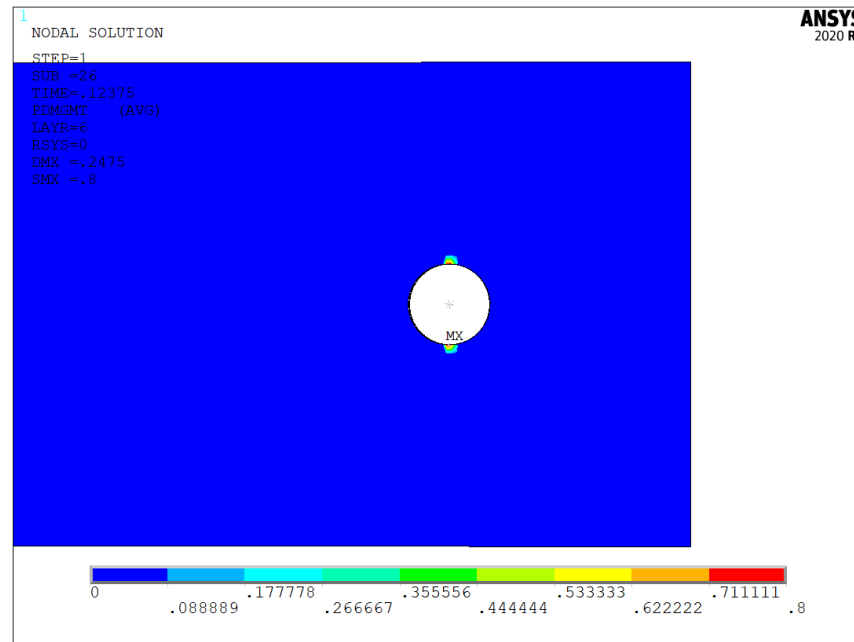


Figure 7.4. First matrix tension failure in 90-degree plies for SSB2 specimens

By increasing the load by applying more displacement when the reaction force reaches 10042.5 N, in all ± 45 degree plies matrix compression failure happens, and for first layer the location of matrix compression failure is shown in Figure 7.5. Additionally, the first fiber failure in compression is determined in this time step and shown in Figure 7.6. Fiber failure is observed in the third layer, oriented at zero degrees, near the hole's edge, where the pin exerted compressive force onto the laminated plate. The first fiber compression failure occurs when the reaction force is 12349.74 N and for the first layer with 45-degree orientation the failure location is shown in Figure 7.7.

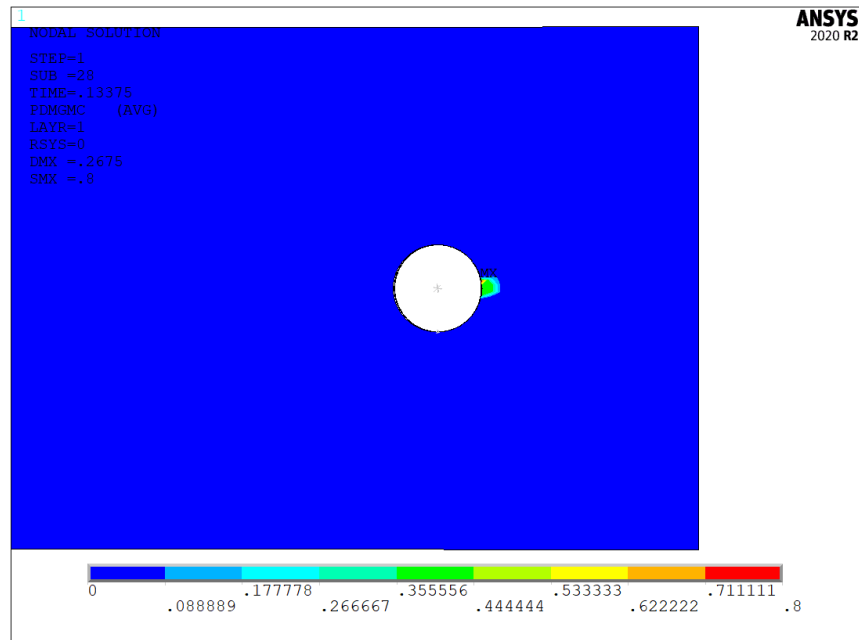


Figure 7.5. First matrix compression in SSB2

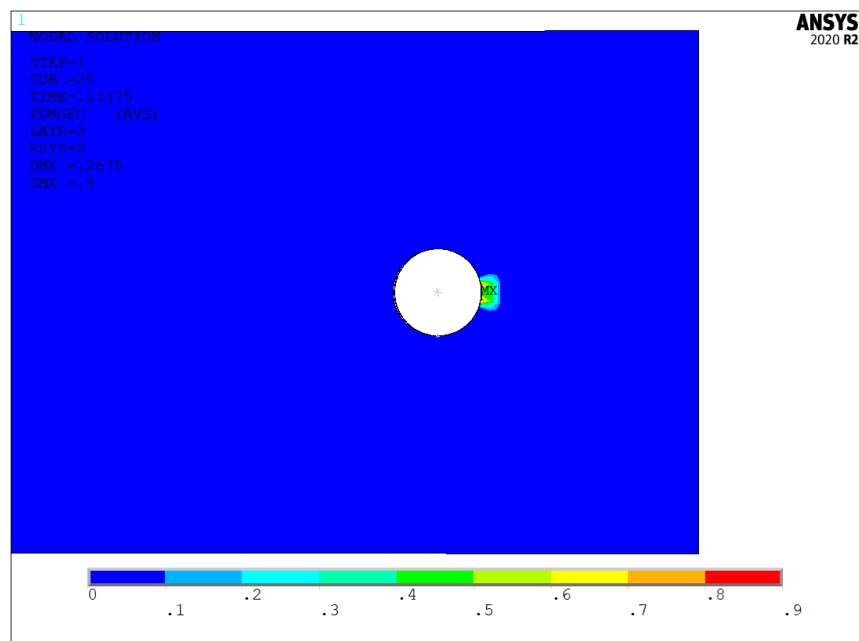


Figure 7.6. First fiber compression in SSB2

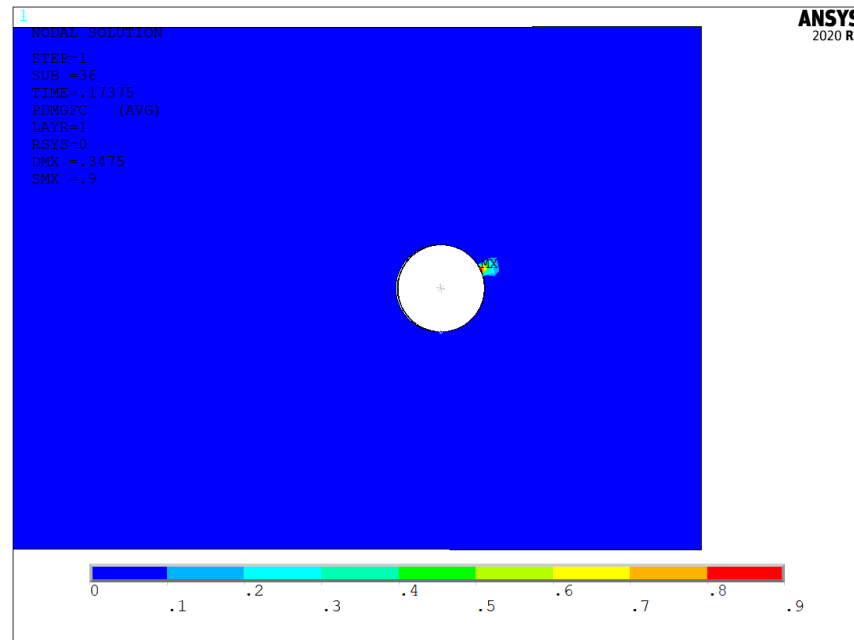
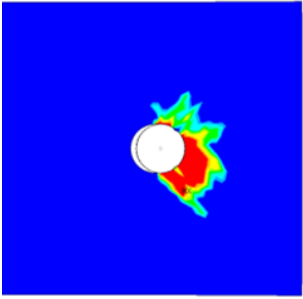
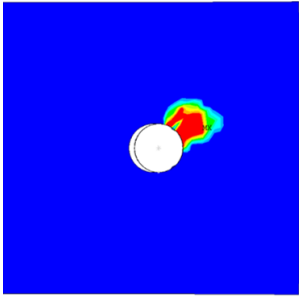
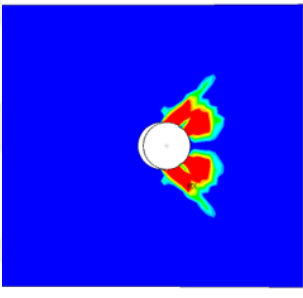
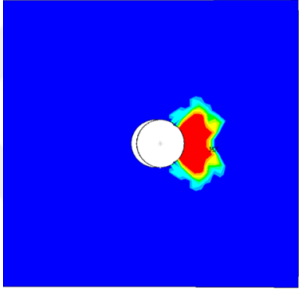
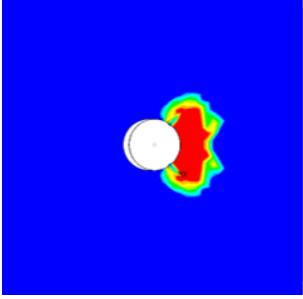
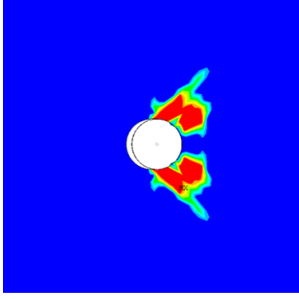


Figure 7.7. First Fiber failure in 45-degree plies for SSB2 model

7.3.2. Damage in Different Plies for SSB2

The study into matrix and fiber failures under compression for the ultimate failure load revealed the status of four layers, as depicted in Table 7.2. The pin displacement when ultimate failure load detected is 1.07mm and corresponding force is 20014.62 N. When the first layer with a 45-degree orientation was considered, a considerable amount of failure was observed in these layers. Due to the high deformation in the hole edge region, fibers also fail at 90 degrees along the displacement direction. In 0-degree fibers, matrix failure due to the high compression forces and fiber failure due to the compression is observed in this time step. The displacement applied to the pin was 2 mm, but the analysis could not converge due to high distortion elements near the hole edge.

Table 7.2. Damage in different layers at ultimate failure load for SSB-2

Layer Orientation	Matrix Compression	Fiber Compression
45		
0		
90		

7.3.3. Damage Progression in Different Plies for SSB3

In this case, we focused on examining the maximum load at the first load drop, 2% offset bearing load, and ultimate load to thoroughly investigate the propagation of

damage in the selected plies. Damage status could be used to determine damage in any ply without taking into account the different failure modes in the analysis. The damage status for ± 45 , 0, and 90-degree plies is detailed in Table 7.3. The maximum load at first load drop is 14858.87 N and the corresponding 2% offset bearing load is 17423.64 and final failure load predicted as 21048.94079 for SSB3 configuration which is considered as hard laminate.

7.3.4. Evaluating the Predicted Results with Experimental Results

PFM of the above-mentioned configurations is performed and results of the predicted value for ultimate bearing strengths of the pin loaded joints is compared with the experimental values. Ultimate bearing strength and also 2% offset bearing strength were reported in the qualification of AS4/8552 laminated composite[119]. Table 7.4 and Table 7.5 present a comparative analysis between predicted values and experimental results for 2% offset bearing strength and ultimate bearing strength across all configurations. The results exhibit good agreement with experimental data, particularly for hard laminates, where the model's deformation is comparatively lower than the other two configurations.

Table 7.3. Damgae progression in different layers for SSB-3

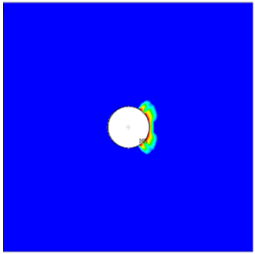
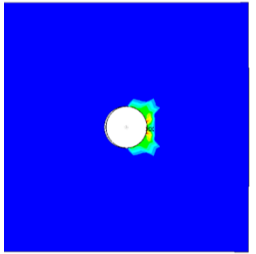
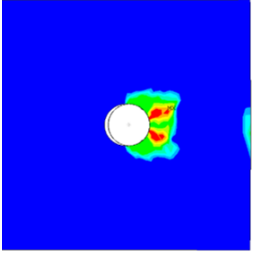
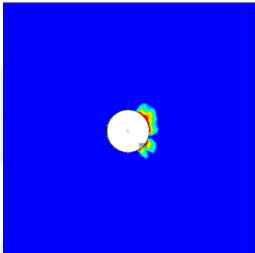
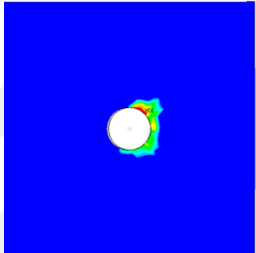
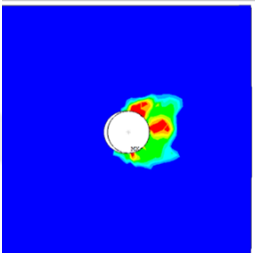
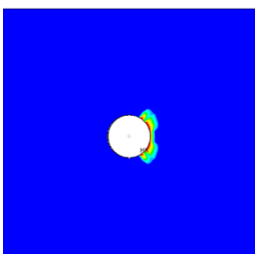
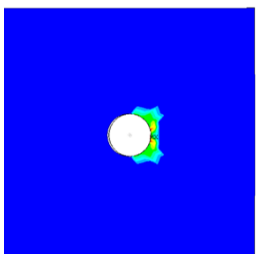
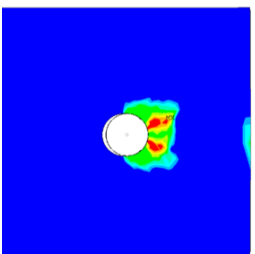
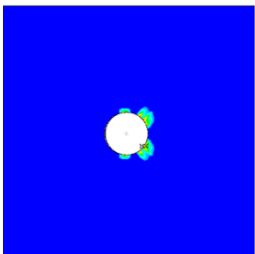
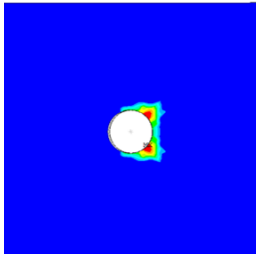
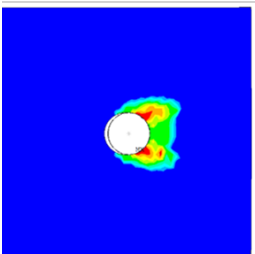
Orientation	First load drop	2% strain offset	Ultimate
45			
0			
-45			
90			

Table 7.4. Comparing the predicted PFM 2% offset Bearing Strength results with reported experimental results for all SSB Configurations

Configuration	2% offset Bearing Strength (MPa)	Corresponding Bearing Load (N)	Predicted Load from PFM (N)	Error Percent- age
SSB1	711.81	13593.34	10896.72	-20%
SSB2	671.93	16039.56	12722	-21%
SSB3	720.92	17208.93	17423.64	1%

Table 7.5. Comparing the predicted PFM Ultimate Bearing Strength results with reported experimental results for all SSB Configurations.

Configuration	Ultimate Bearing Strength (MPa)	Corresponding Bearing Load (N)	Predicted Load from PFM (N)	Error Percent- age
SSB1	934.58	17847.52	15389.96	-14%
SSB2	950.58	22691.23	20014.62	-12%
SSB3	919.35	21945.67	21048.94	-4%

To consider the variability in experimental results, the maximum and minimum values from the reported test data were also compared with the predicted results. Figure 7.8 and Figure 7.9 illustrate the comparison for determining the bearing load across all configurations. Our progressive failure model consistently underpredicts the ulti-

mate bearing strength of pin-loaded laminated composites across all configurations, even when comparing the estimated results with the minimum values of experimental results for quasi-isotropic and soft laminates. However, for the hard laminate configuration, the predicted results fall within the range of the maximum and minimum reported experimental values for both 2% offset bearing strength and ultimate bearing strength.

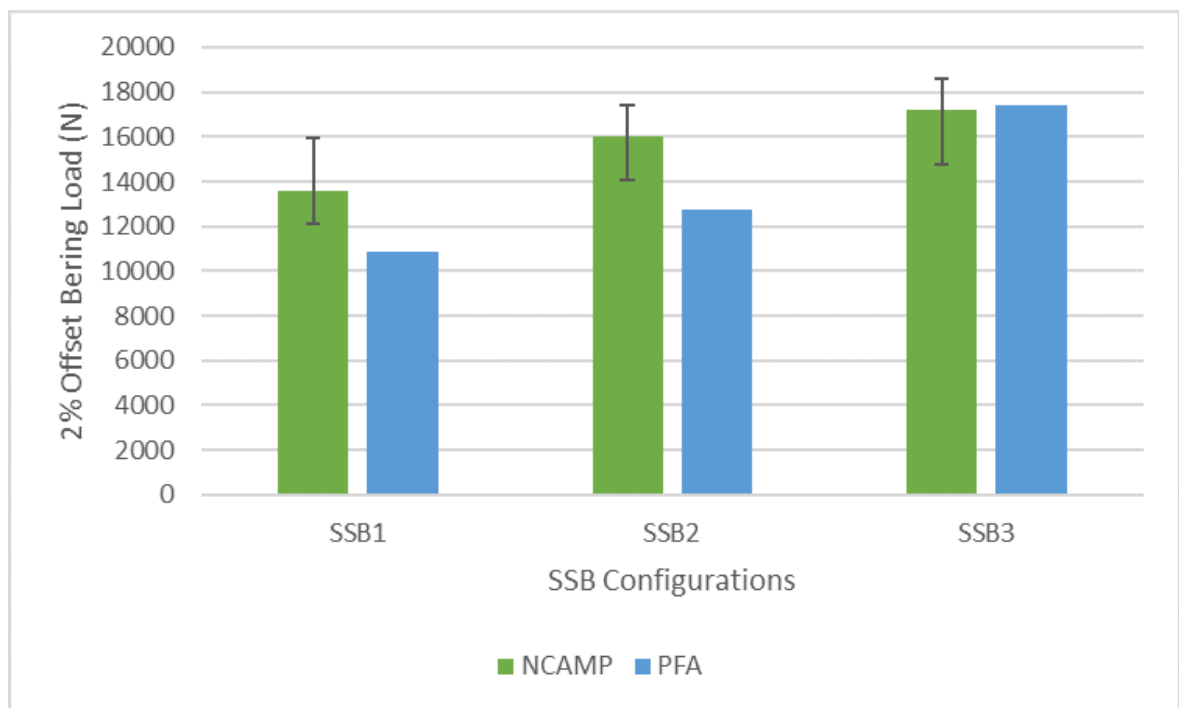


Figure 7.8. Comparing the predicted results with considering experimental result variation for 2% offset bearing of all specimens

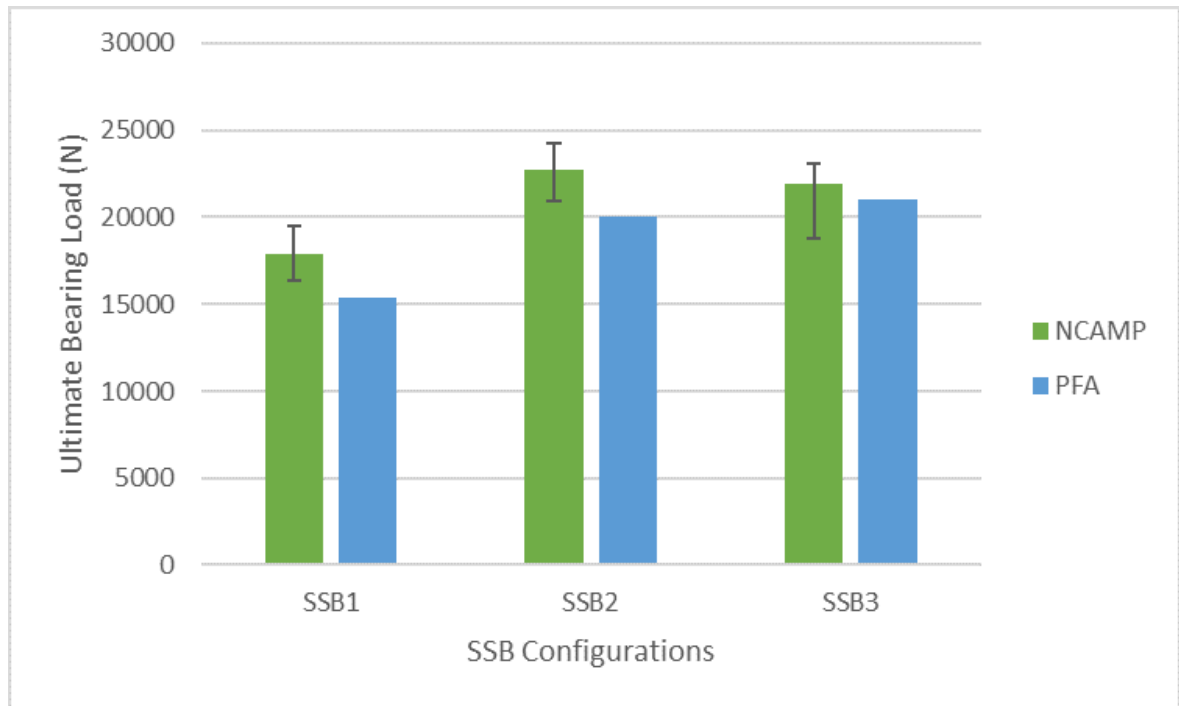


Figure 7.9. Comparing the predicted results with considering experimental result variation for the ultimate bearing of all specimens

While the analyses mentioned earlier do a decent job of predicting ultimate failure, many researchers agree that 3D models can do a better job and allow for the analysis of more complicated joint configurations and loading situations [29,123]. The presence of inter-laminar pressures at the free edges and the reliance on 3D effects for the bearing failure mode are contributing factors. Bolt tilting and bending, bolt pre-tension, and secondary bending are all factors that exhibit three-dimensional variation [98].

7.4. Conclusion:

- When comparing the first ply failure load, for example, in SSB-2, the first matrix failure was observed at 9355 N, and the first fiber failure load is detected to be 10042 N, with predicted ultimate failure load, which is 20014N, over 53% and 50% difference observed correspondingly with the final failure load.
- Ultimate bearing strength could be predicted very well for hard-laminated com-

posite structures

- For soft and quasi-isotropic laminates which contain a lower percentage of 0-degree fiber along the loading direction, the model underpredicts the final failure load in these structures with a results 14% lower than the experimental values

When designing a pin-loaded composite structure, one should take into consideration the significant deformation in the loading direction, especially for soft and quasi-isotropic laminates. This becomes crucial if other design considerations demand minimal deformation in the joint.



8. OPTIMIZATION

8.1. Introduction

The presence of cutouts in laminated structures reduces mechanical strength by concentrating stress around these openings. Identifying the best local reinforcement is critical for increasing the strength of such structures. Local reinforcements can alter load paths while also strengthening the entire structure. It is critical to evaluate the impacts of various parameters in order to identify the proper design space for the optimization problem. The influence of reinforcing dimensions and orientations on the open-hole tension specimen is investigated using parametric studies. After establishing the design space, multi-objective optimization is used to determine the most effective configurations for these reinforcements to strike a balance between multiple objectives.

8.2. Effect of Patch Parameters

To find the optimum local reinforcement of composite plate with cutout effect of parameters that affects the ultimate failure load of the component should be investigated. Well-known parameters that affect the ultimate failure of patched components are Geometric properties, shape, thickness, and stacking sequence of the patch. It's essential to consider the effects of input parameters, damage mechanisms, and their relationships in order to create an acceptable and optimal reinforcement design. A parametric study is conducted to determine the model's sensitivity to the patch's length, the patch's width, and the orientation of the plies of the patch.

A model is created for a rectangular composite plate with 200mm length and 50mm width that contained a circular cutout with a 20mm diameter in the middle of the plate. The stacking sequence of the plate is $[0, 45, -45]_s$ and. The nominal thickness of each ply is 0.25mm, which is the same as the specimen configurations used in [11]. The model is created following the steps outlined in Case Study 3 of Chapter

5. However, a key distinction in the model generation lies in the splitting of the plate and the subsequent meshing of the reinforced areas with the new laminated stacking sequence. While the base plate's stacking sequence remains constant, additional plies are introduced to both the upper and bottom surfaces of the plate. The layer thickness for these patches is set at 0.125 mm.

The meshed part with patches is shown in Figure 8.1 and Figure 8.2. As previously stated, 2D SHELL181 is used in the model to facilitate the visualization of the model, and element shape is activated to depict the layer's thickness. The applied boundary condition is shown in Figure 8.3. The developed model predicts the patch effect on the behavior of the composite plate with cutout under tensile loading and finds the ultimate failure load of the component.

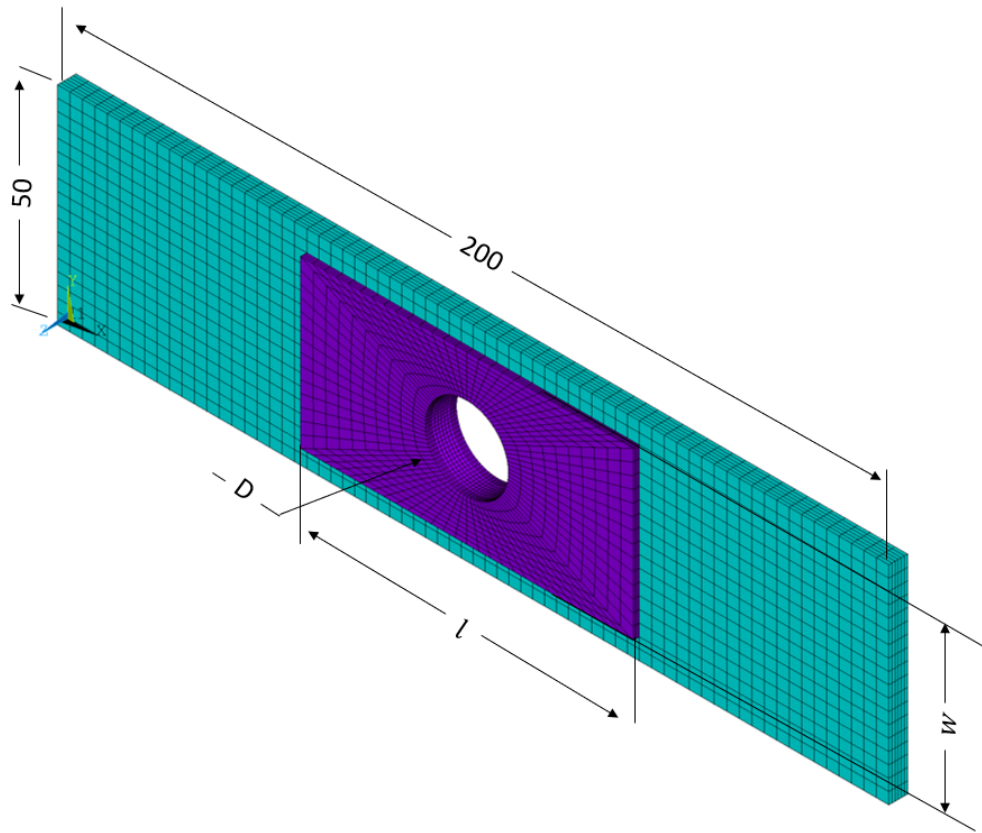


Figure 8.1. A stiffened plate

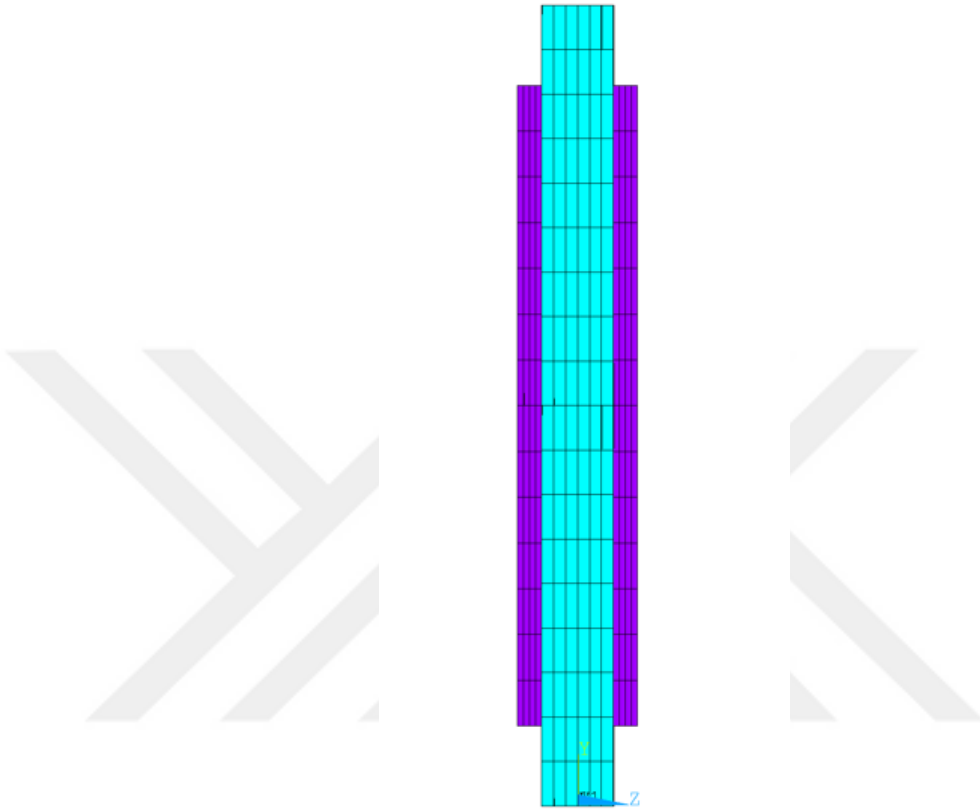


Figure 8.2. A plate with two local reinforcements

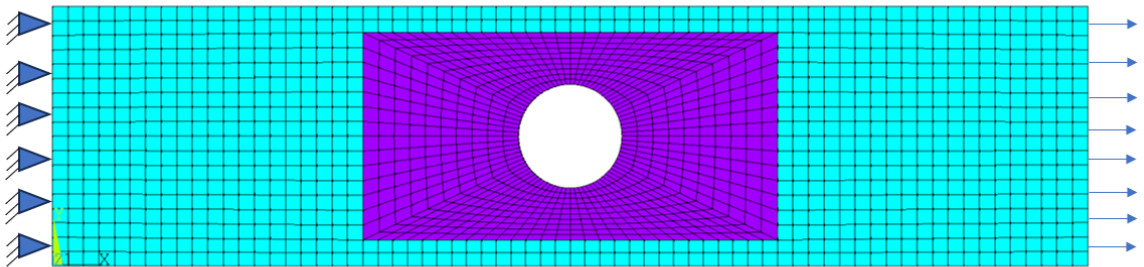


Figure 8.3. A plate with two local reinforcements

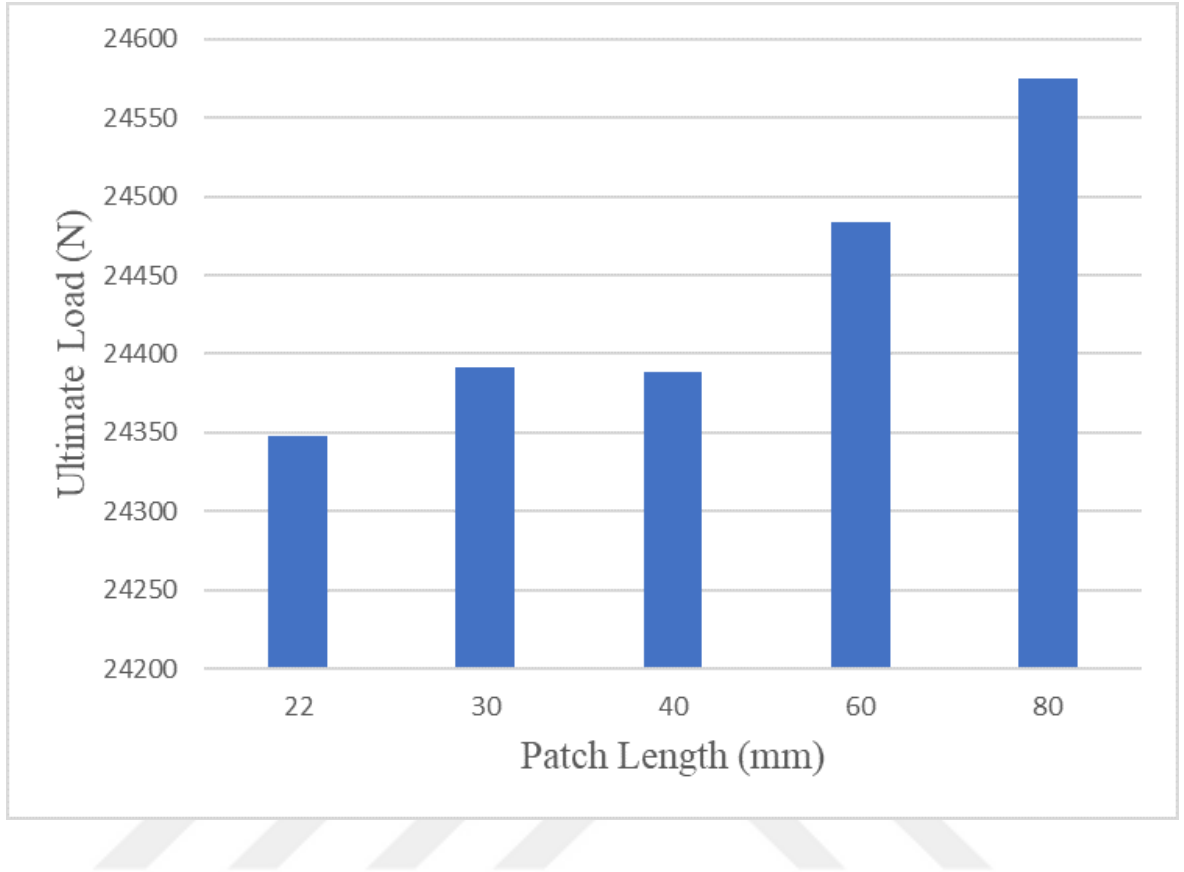


Figure 8.4. Effect of various patch lengths on final failure load of reinforced composite plate

8.3. Effect of Patch Length on Final Failure Strength

Five configurations are modeled to investigate the influence of the patch length on the ultimate failure load of the stiffened plate, and the anticipated ultimate load findings are provided in Figure 8.4. In these models, the patch width is taken as 40mm for all configurations, and the stacking sequence is $[45, -45, 45, -45]$.

8.4. Effect of Patch Width on Final Failure Strength

Ultimate failure loads are obtained for various patch widths for the stiffened composite plates. For applied patches, a constant length of 80mm and a stacking sequence of $[45, -45, 45, -45]$ are used. A comparison of the results for different patch

widths is shown in Figure 8.5.

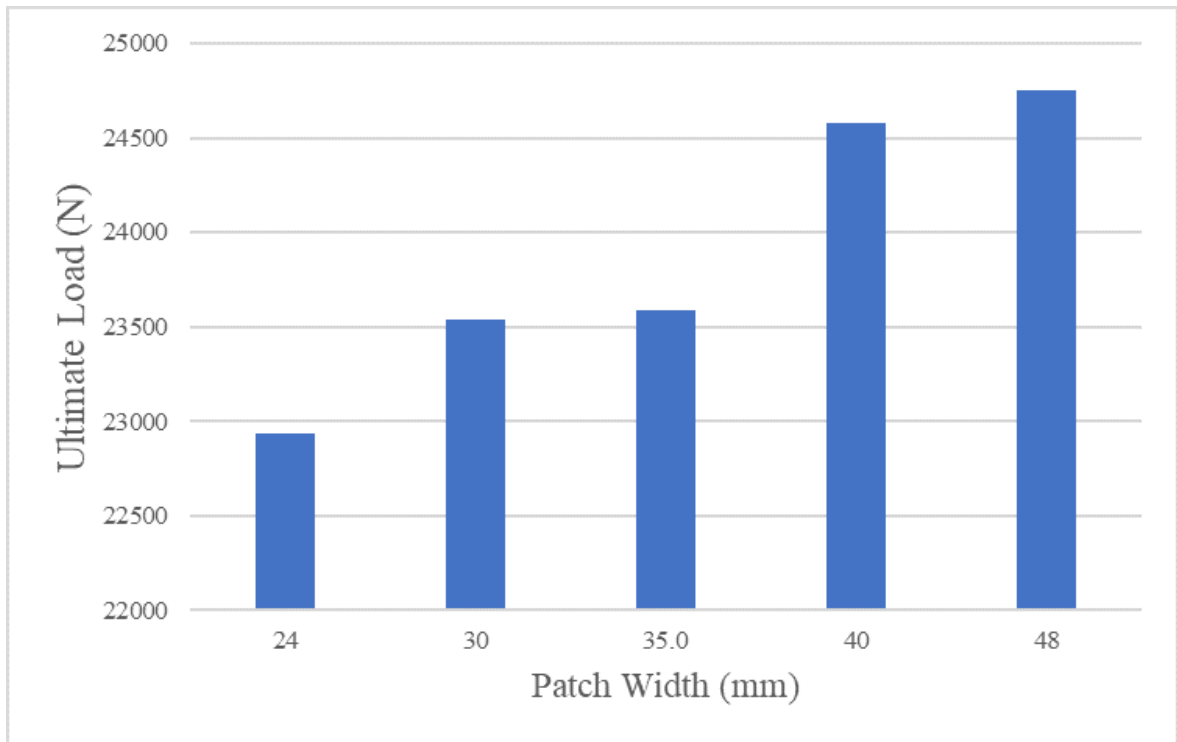


Figure 8.5. Variation of final failure strength of composite plate according to the patch width

Noteworthy insights for design purposes can be gained from Figure 8.5. The Predicted results show that increasing the width of the patch significantly affects the final failure load of the patched structure, and the ultimate load for the chosen configuration increased 23% above the unpatched plate for structures with 48mm patch width.

8.5. Effect of Patch Orientation on Final Failure Strength

One parameter that has a large effect on the ultimate failure of reinforced structures is patch fiber orientation. To determine the effect of the patch layers' orientation on the ultimate failure load of the stiffened plate, seven configurations are modeled, and the predicted ultimate load results are shown in Figure 8.6. The reinforcement

of the host structure includes two patches, one for the upper surface and one for the bottom surface. Each patch had 80mm length and 40mm width and had four layers with 0.125mm thickness and $[\theta_1, -\theta_1, \theta_1, -\theta_1]$ stacking sequence.

Regrading Figure 8.6 One may expect that to increase the ultimate failure load, the optimum fiber angle should tend to 0 degrees as this would align the fibers in the axial direction parallel to the loading direction. Similarly, Figure 8.6, Shows that when the fiber orientation of the patch is 0 degrees, the best results are obtained for the specified configuration.

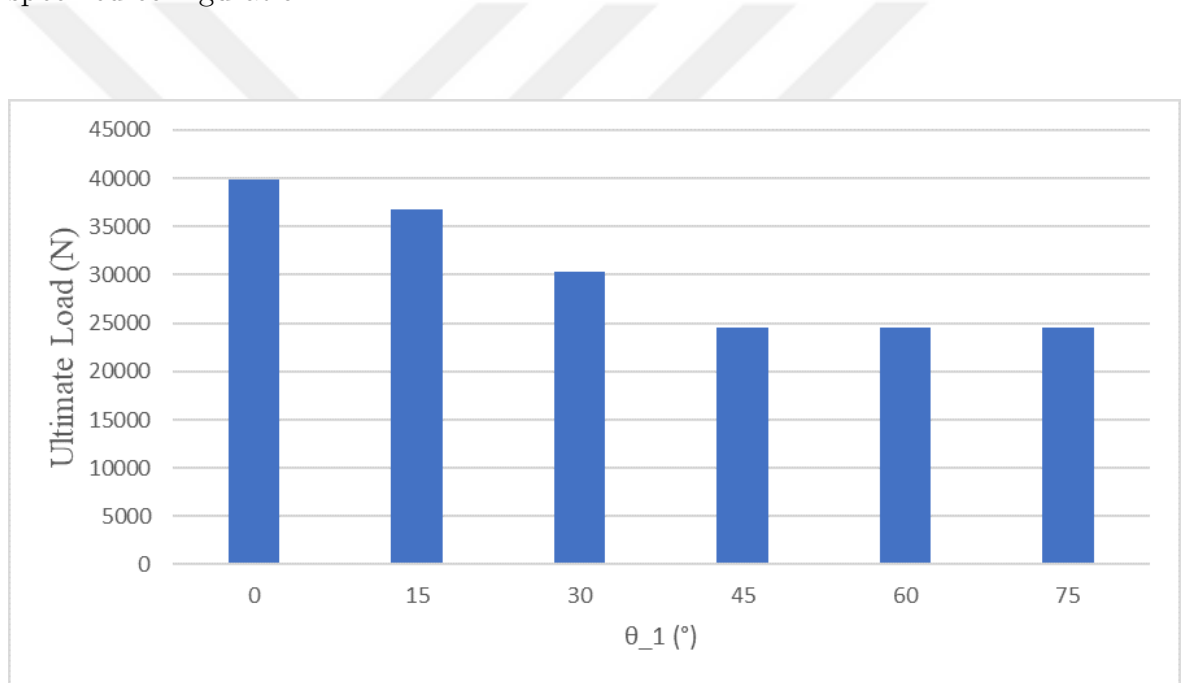


Figure 8.6. Effect of various orientations on the ultimate failure load value of patched composite plate

8.6. Case 1.1: One Rectangular Patch Optimization (15-Degree Increment)

The methods for improving ultimate failure load outlined in the earlier section are used to find the ideal values for the patch configuration parameters $(w, l, \theta_1, \theta_2)$, as illustrated in Figure 8.7, to increase the strength of the reinforced specimen while

adding the least amount of weight to the structure. The design space variable for ultimate failure load optimization of the reinforced composite plate with a hole in the patch shape parameters are defined by defining the values of the l for the length of the patch and the w for the width of the patch as well as fiber orientation of these patches with the stacking sequence of $[\pm\theta_1, \pm\theta_2]$. The thickness of each ply in the patch was set at 0.125mm, while the thickness of the base specimen's layers was chosen to be 0.25mm. Based on previous parametric studies, the upper and lower limits for the four parameters in this case study are presented in Table 8.1.

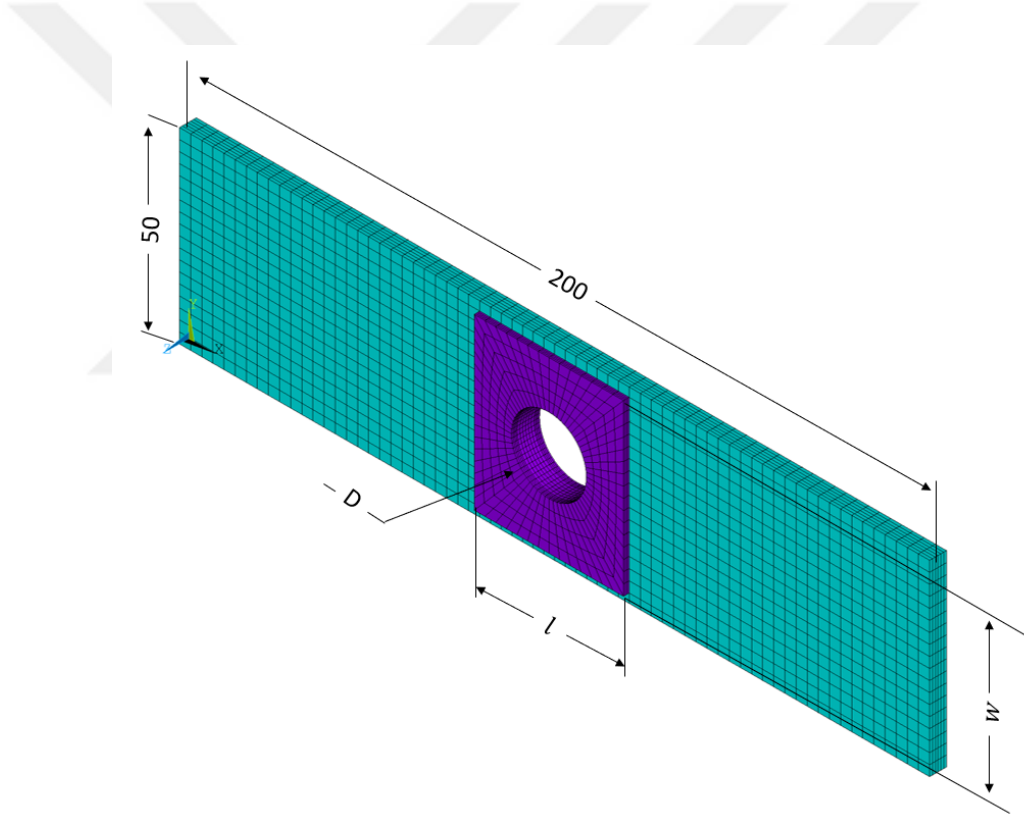


Figure 8.7. Configuration for reinforced CFRP plate with one rectangular patch on each side

The objective function was defined as

$$F_{\text{Obj}} = w_{\text{mass}} \frac{w_r}{w_u} + w_{\text{Load}} \frac{F_{\text{ult-u}}}{F_{\text{ult-re}}} \quad (8.1)$$

Table 8.1. Design space variables range

Parameter	Value
Min w	28
Max w	48
Min l	28
Max l	60
Min θ_1 & θ_2	-75
Max θ_1 & θ_2	75

where w_r is the reinforced structure weight, and w_u is the weight of the structure without reinforcement. $F_{\text{ult-u}}$ and $F_{\text{ult-re}}$ are the ultimate load of the unreinforced specimen and the final failure load of the reinforced structure, respectively. The weighting coefficients for w_{mass} and w_{Load} were set as 0.3 and 0.7, respectively; also, the ultimate failure load was defined as the maximum load of a structure during the loading. The mass of the unreinforced plate in this study is 22.61 grams. The ultimate failure load of unreinforced is 21342 N.

In this Case, the results of the optimization of one rectangular patch on each side of the plate are shown in Table 8.2. Fiber orientation variables, in this Case, range from -75 degrees to 75 degrees, with increments of 15 degrees. That means that the orientation of the fibers can be adjusted to -75, -60, -45, -30, -15, 0, 15, 30, 45, 60, and 75 degrees In order to find the optimal values for the cost function.

Table 8.2. Result of optimal final failure load and weight for One rectangular patch with a 15-degree fiber angle increment

Trial	F_{Obj}	w (mm)	l (mm)	θ_1 (°)	θ_2 (°)	Rmax (N)	Mass (g)
1	0.6639	47.3	47.94	0	0	48605	26.87
2	0.6660	46.92	41.32	0	0	47243	26.36
3	0.6676	46.3	43.62	0	0	47246	26.48
4	0.6678	46.32	46	0	0	47571	26.66
5	0.6680	46.24	40.44	0	0	46737	26.25
6	0.6683	47.78	50.2	0	0	48340	27.07
7	0.6683	46.16	41.6	0	0	46842	26.33
8	0.6687	47.8	48.88	0	0	48088	26.98

The best Objective function value for this case is 0.6639319, and the predicted ultimate failure load for this specimen is 48.60 KN. When these results were compared with the unreinforced laminated plate, the failure load increased by 126% by just adding approximately 19% of the weight of the unreinforced laminated composite plate. The best configuration's ultimate load-carrying capability per weight of the structure is 1.8 (KN/g).

8.7. Case 1.2: One Rectangular Patch Optimization (1-Degree Increment)

For the same case study, the fiber orientation range was kept the same as the previous study, but the increment value changed from 15 degrees to one degree in order to increase the design space for the fiber orientation angles and may find better solutions for that case. The results are shown in Table 8.3.

Table 8.3. Results for one rectangular patch with a 1-degree increment

Trial	F_{Obj}	$w(\text{mm})$	$l(\text{mm})$	θ_1 (°)	θ_2 (°)	Rmax (N)	Mass (g)
1	0.6621	47.52	42.28	6	-2	45668	25.24
2	0.6622	46.36	46.16	-3	4	46034	25.45
3	0.6624	47.78	42.3	7	3	45658	25.26
4	0.6625	46.64	45.96	3	-3	46011	25.45
5	0.6626	46.28	39.82	4	0	45135	24.99
6	0.6640	46.92	42.68	7	2	45396	25.23
7	0.6643	46.8	36.12	1	2	44476	24.75
8	0.6644	47.86	44.56	9	-6	45707	25.44

The predicted ultimate failure load for the best configuration is 45.67KN, which is 114% more than the unreinforced laminated composite plate. The weight of the patches is 2.63 gr, which is about 11% of the unreinforced specimen. The ultimate load-carrying capability of the structure increased significantly by attaching these two rectangular patches. Additionally, the best patch configuration is achieved by stacking the patches in the sequence of $[\pm 6, \pm 2]$, and the values for w and l are less than the w value from the previous case, resulting in a lighter overall weight for the structure.

The cost function for the best configuration in this case is also lower than the best configuration in Case 1.1. This demonstrates that by expanding the range of options for fiber orientation, the structure's capacity to bear load is improved, but it needs different manufacturing capabilities like AFP or ATL machines to place these fiber orientations.

8.8. Case 2: Two Rectangular Patches Optimization

We study another configuration to find the optimal design of the reinforced, laminated composite plate with a hole under tension. Two rectangular patches are attached to each side of the plate, as shown in Figure 8.8. Two-dimensional shell 181 elements are used in the analysis to clearly depict the patch elements' shape turned on. Table 8.4 shows the design variables used in this study. We choose a step size of 15 degrees to adjust the fiber orientation variable.

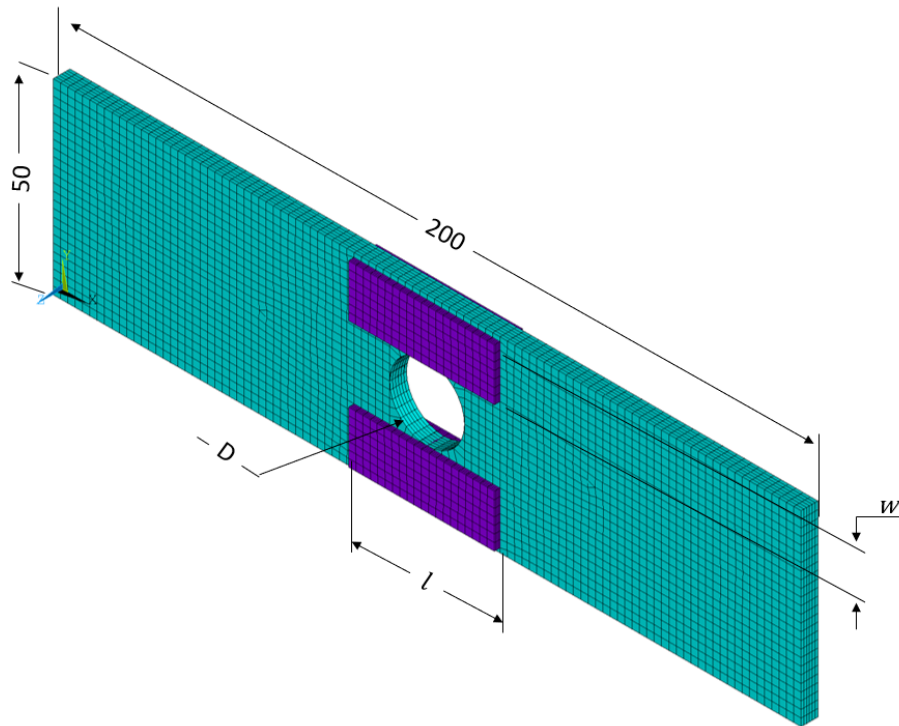


Figure 8.8. Configuration for reinforced CFRP plate with two rectangular patches

Table 8.4. Design space variable range

Parameter	Value
Min w	4
Max w	14
Min l	28
Max l	60
Min θ_1 & θ_2	-75
Max θ_1 & θ_2	75

The top eight configurations resulting from the optimization process are presented in Table 8.5 for this case study. The gain in ultimate failure load in case of the best configuration for the reinforced structure is 130% by reinforcing the near hole region with four rectangular patches with 17.4 mm width and 58.94 mm length for each patch. The added weight value is 1.59 grams, which is only for 7% of the unreinforced specimen weight.

Table 8.5. Results for Two rectangular patches with a 15-degree fiber angle increment

Trial	F_{Obj}	$w(\text{mm})$	$l(\text{mm})$	θ_1 (°)	θ_2 (°)	Rmax (N)	Mass (g)
1	0.6258	8.70	58.94	15	15	49039	24.20
2	0.6269	9.77	58.14	-15	-15	49241	24.37
3	0.6269	7.93	58.26	-15	15	48533	24.04
4	0.6269	9.69	58.36	-15	-15	49211	24.37
5	0.6271	7.94	57.56	-15	-15	48471	24.03
6	0.6277	7.27	59.26	15	0	48200	23.95
7	0.6278	7.94	54.84	-15	-15	48221	23.96
8	0.6289	8.37	59.26	0	-15	48440	24.15

When the best configuration results are compared with one rectangular patch on each side the value of w is lower than in the previous study but the values of the l is larger than in the previous Case1.1 study. Additionally, fiber orientation changed from the value of near zero degrees to 15 degrees. This demonstrates that changing the patch configuration from one patch to two patches on each side of the specimen to achieve the best ultimate failure, more narrow patches with [15,15] stacking sequence could generate a better load path and could reduce stress concentration in these types of structures under tension load more efficiently. The objective function is 0.6258, and when we compare it with Case1.1's best configuration, the objective function is 0.6621. We found out that this configuration is more efficient than Case1.1. The best configuration has a ratio of ultimate load-bearing capacity to structure weight of 2.02 (KN/g), which is nearly double that of the base structure, which has high stress concentrations near the hole.

8.9. Case 3: Two Different Rectangular Patches Optimization with Ply Drop

Another configuration is studied to find the optimal design of the reinforced, laminated composite plate with a hole under tension. To decrease the stress concentrations near the ply drop, blending rules were studied by some researchers and also applied in the industry. As the thickness of the plate increased in the reinforced regions to decrease this drop angle and reduce stress concentration in these regions, the design space split into three different regions. By doing this, resin pockets in the ply drop region and stiffness variation difference between the base plate and reinforced region could be reduced. Therefore, more efficient designs could be obtained when micromechanics of laminated composite are considered.

The stacking sequence in the base plate is $[0, 45, -45]_s$ to add reinforcement to the base, plate the stacking sequence of purple region is $[\theta_1, -\theta_1, 0, 45, -45]_s$ which two rectangular patches reinforce the both side of the plate and these patches have $[\theta_1, -\theta_1]$ stacking sequence. Two other patch is introduced in red region on both side of the base plate and the red region has $[\theta_1, -\theta_1, \theta_2, -\theta_2, 0, 45, -45]_s$ lamination scheme that shows two smaller rectangular patches with $[\theta_1, -\theta_1, \theta_2, -\theta_2]$ stacking sequence reinforce the region around the hole. The dimension of the base plate is the same as the previous case studies, and two other parameters introduced for smaller patch dimensions and generated area for this model are shown in Figure 8.9.

One of the rules in blended laminates is the distance between the ply drops should be ten times the thickness of the dropped thickness. Additional constraints must be introduced for newly generated configurations, and design space must be changed to apply this rule in the optimization. Additionally, by dividing the design space and adding additional variables to the design space, better optimization results could be obtained because this configuration could add a smaller weight to reinforce the structure.

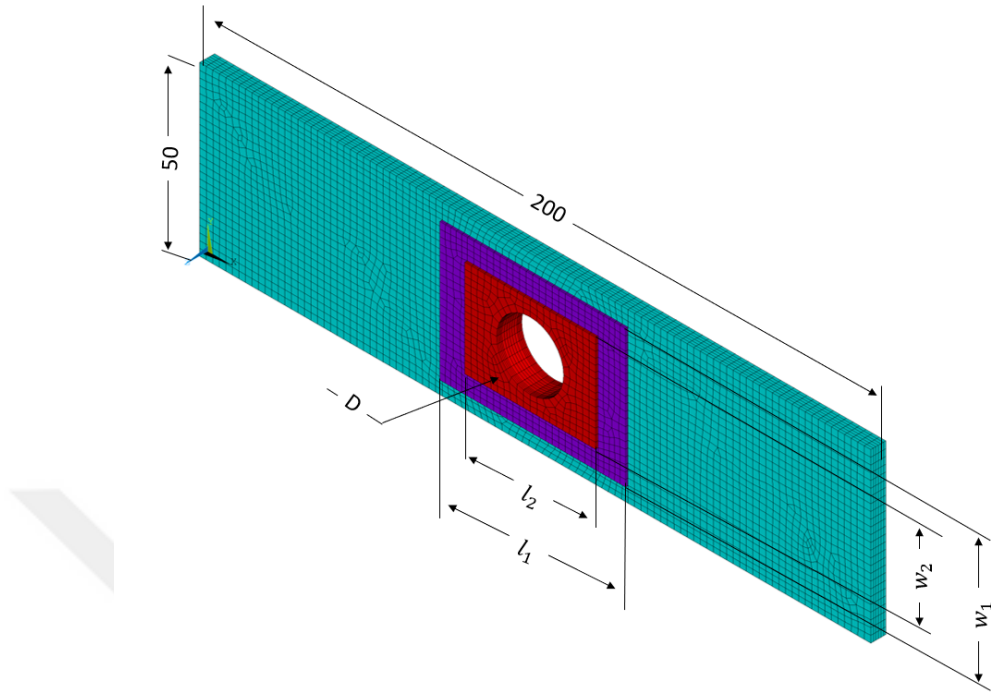


Figure 8.9. Ply drop optimization configuration model

The maximum and minimum values of the length and width of each patch that could be used to generate the model are shown in Table 8.6. An additional constraint is defined in the model generation for the generated values for patch length and width in smaller and bigger patches. The difference between the newly generated value for the length and width of the patches must be bigger than 2.4 mm, and smaller patches cannot be bigger than the big patches. The step size for adjusting the fiber orientation variable was chosen to be 1 degree.

Table 8.6. Design space variable range

Parameter	Value
Min w_1	33
Max w_1	48
Min w_2	28
Max w_2	43
Min l_1	33
Max l_1	60
Min l_2	28
Max l_2	55
Min θ_1 & θ_2	-75
Max θ_1 & θ_2	75

The top ten optimization results for this case study are shown in Table 8.7. In a purple region with the bigger patch, the average width and length for the ten best configurations is approximately 47.5 mm for width and 52 mm for length. Ply orientation varied between -4 and 3 degrees. The design space for width is limited to the value between 33 mm and 48 mm; the average value of the width in best configurations is 47.5, which shows that aligning the fiber at near zero degrees and covering almost the plate width gives the best ultimate failure load. For small reinforcement, these generated the best results in the layer orientation change between -2 and 2 degrees. The average width and length of the best generated configuration is 32.7 mm and 30.9, respectively. The best configuration's ultimate load-carrying capability per structure weight is 1.95 (KN/g).

Comparing the best optimized configurations are shown in Table 8.8. The best objective function is found when Case 2 configuration is used. After that, the Case 3 reinforcement configuration has a smaller objective function than the two configurations in Case 1.

One of the rules in blended laminates is the distance between two ply drops should be ten times the thickness of the dropped thickness. Also, other blended rules are investigated. One may use multi-step design optimization and apply all of these rules that generally affect the manufacturing and mechanical performance of the structure. In the first step, the group of optimum designs could be determined. Then, another constraint in the design space could be considered in the second optimization, and another optimization study with this optimum configuration obtained from the first step could be used as design space.

Table 8.7. Ten best optimization results for test Case 3

Trial	F_{Obj}	w_1(mm)	w_2(mm)	l_1(mm)	l_2(mm)	θ_1 (°)	θ_2 (°)	Rmax (N)	Mass (g)
1	0.6382	47.7	33.06	53.26	30.1	3	-1	48483	24.87
2	0.6384	47.54	30.72	50.12	30.34	-2	-2	48092	24.7
3	0.6397	47.52	36.76	54.66	28.12	0	2	48398	24.94
4	0.6397	47.62	29.68	49.94	38.06	-3	-2	48199	24.85
5	0.6398	47.52	32.6	54.68	28.66	0	2	48216	24.87
6	0.6399	47.66	36.04	49.98	30.42	-3	-2	48124	24.82
7	0.6399	47.48	31.28	54.78	32.4	0	2	48336	24.93
8	0.6403	47.62	34.12	49.98	32.08	-3	-2	48046	24.82
9	0.6408	47.28	29.3	49.1	29.66	-4	-2	47529	24.6
10	0.6409	47.6	33.72	54.66	29.9	0	2	48171	24.93

Table 8.8. Comparing the best configuration of optimized case studies

Case	F_{Obj}	Rmax (N)	Mass (g)
Case 1.1	0.6639	48605	26.87
Case 1.2	0.6621	45668	25.24
Case 2	0.6258	49039	24.2
Case 3	0.6382	48483	24.87

9. CONCLUSIONS

In this study, the ultimate strength of composite plates with a cutout is maximized by optimizing local reinforcements in the form of additional rectangular layers. The ultimate strength is computed by implementing a progressive damage model based on the Puck failure criterion using a finite element method. The optimum reinforcement designs are obtained using a modified simulated annealing algorithm. The accuracy of the model predictions is verified by comparisons with open-hole tension, open-hole compression, and single-shear bearing test results.

The methodology proposed in this study is applied to the optimal reinforcement design for an open-hole tension test specimen. Three different reinforcement configurations are tried and all of them are found to be equally effective in improving the ultimate failure strength. Although improving the first-ply failure strength is not a direct objective of this study, the first-ply failure strengths of the three optimized reinforced configurations are compared. The configuration with single rectangular layers is found to be the most effective in that respect. The improvement achieved by using this optimal local reinforcement is 184.2% increase in the first-ply failure strength and 121.5% increase in the ultimate failure strength with 13.6% increase in the material use. The full potential of the design methodology proposed in this study should be further investigated considering different types of loads and notch geometries.

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