

**ANKARA YILDIRIM BEYAZIT UNIVERSITY**  
**GRADUATE SCHOOL OF NATURAL AND APPLIED**  
**SCIENCES**



**EVALUATION OF HOLE FILLING REPAIR OF CARBON**  
**FIBER REINFORCED POLYMER COMPOSITE**  
**LAMINATES**

**M.Sc. Thesis by**

**Yeliz CAN**

**Department of Mechanical Engineering**

**May, 2022**

**ANKARA**

**EVALUATION OF HOLE FILLING REPAIR  
OF CARBON FIBER REINFORCED  
POLYMER COMPOSITE LAMINATES**

**A Thesis Submitted to**

**The Graduate School of Natural and Applied Sciences of**

**Ankara Yıldırım Beyazıt University**

**In Partial Fulfillment of the Requirements for the Degree of  
Master of Science in Mechanical Engineering, Department of  
Mechanical Engineering**

**by**

**Yeliz CAN**

**May, 2022**

**ANKARA**

**M.Sc. THESIS EXAMINATION RESULT FORM**

We have read the thesis entitled “**EVALUATION OF HOLE FILLING REPAIR OF CARBON FIBER REINFORCED POLYMER COMPOSITE LAMINATES**” completed by **YELİZ CAN** under the supervision of **PROF. DR. FAHRETTİN ÖZTÜRK** and we certify that in our opinion it is fully adequate, in scope and in quality, as a thesis for the degree of Master of Science.

Prof. Dr. Fahrettin ÖZTÜRK

---

Supervisor

Assoc. Prof. Yasin SARIKAVAK

---

Jury Member

Prof. Dr. Zafer EVİS

---

Jury Member

Prof. Dr. Sadettin ORHAN

---

Director

Graduate School of Natural and Applied Sciences

## ETHICAL DECLARATION

I hereby declare that, in this thesis which has been prepared in accordance with the Thesis Writing Manual of Graduate School of Natural and Applied Sciences,

- All data, information and documents are obtained in the framework of academic and ethical rules,
- All information, documents and assessments are presented in accordance with scientific ethics and morals,
- All the materials that have been utilized are fully cited and referenced,
- No change has been made on the utilized materials,
- All the works presented are original,

and in any contrary case of above statements, I accept to renounce all my legal rights.

**Date:** 18.05.2022

**Signature:**

**Name & Surname: Yeliz CAN**

## ACKNOWLEDGMENTS

I would like to take this opportunity to thank my supervisor Prof. Dr. Fahrettin ÖZTÜRK and for his support, guidance, encouragement, and enthusiasm. It has been an honor and a pleasure for me to work with him. I would also like to express my gratitude Manager Faruk OĞUZ and his team at Turkish Aerospace Industries, Inc. I am grateful for the support of Turkish Aerospace Industries, Inc.

In addition, I would like to thank Captain Kadir Aytuğ AYGÜN, Head of Material Department at the Ministry of National Defense, and my colleagues in Material Control and Coordination Branch for their invaluable support in this thesis.

Finally, I would like to thank my family for being with me throughout my all life.

**2022, 05 May**

**Yeliz CAN**

# EVALUATION OF HOLE FILLING REPAIR OF CARBON FIBER REINFORCED POLYMER COMPOSITE LAMINATES

## ABSTRACT

Carbon-Fiber Reinforced Polymer (CFRP) composites have been recently used in many industries specifically aerospace due to their advantages such as rigidity, high strength, low density, and corrosion resistance. The use of composite materials is increasing, in parallel with the damage to which these materials are exposed is also increasing in number. However, despite all the benefits they provide, these materials are not easy to repair. Consequently, various damage repair methods have been developed for composite materials and this issue gaining importance.

In this thesis, minor damage to aircraft structures was repaired by the plug repair technology. Firstly, prepreg carbon fiber composite HEXPLY M91/IM7/34RC/UD/194/12K plies were used to produce composite laminate plates. These plies were laid in the atmosphere-controlled environment. Throughout the lay-up process, peel plies were used as backing materials after compaction and vacuum operations were implemented. Then, holes were drilled on these composite laminate plates for plug repairing. The holes were divided into three groups of six specimens and were filled with different materials containing Loctite 9396 adhesive and various amounts of chopped glass fibers. After repairs, Manual Ultrasonic Pulse Echo (MUPE), a non-destructive test method, was used for quality control of the repaired specimens. MUPE results indicated that the repaired specimens were high quality and suitable for testing. Subsequently, tensile tests were carried out for 24 specimens by using an Instron 5985 machine with 250 kN capacity.

The results of this study demonstrated that an improvement by up to 5% was achieved with the repair using 50% chopped glass fiber and Loctite 9396 adhesive. It was observed that the mechanical properties of the composite structure were enhanced with the plug repair technology.

**Keywords:** Plug repair, composite, adhesive, carbon fiber.

# KARBON ELYAF TAKVİYELİ POLİMER KOMPOZİT LAMİNATLARIN DELİK DOLGU ONARIMININ DEĞERLENDİRMESİ

## ÖZ

Karbon-Elyaf Takviyeli Polimer (CFRP) kompozitler sertlik, yüksek mukavemet, düşük yoğunluk ve korozyon direnci gibi avantajlarından dolayı son zamanlarda birçok endüstride, özellikle havacılıkta kullanılmaktadır. Kompozit malzemelerin kullanımı artmakta, buna bağlı olarak bu malzemelerin maruz kaldığı hasarlar da sayıca artmaktadır. Sağladıkları tüm faydalara rağmen, bu malzemelerin tamiri kolay değildir. Sonuç olarak, kompozitlerin hasar onarımı için çeşitli yöntemler geliştirilmiş ve bu konu büyük önem kazanmıştır.

Bu tezde, uçak yapılarındaki küçük hasarlar dolgu onarım teknolojisi ile onarılmıştır. İlk olarak, kompozit laminat plakalar üretmek için prepreg karbon elyaf kompozit HEXPLY M91/IM7/34RC/UD/194/12K katları kullanıldı. Bu katlar atmosfer kontrollü ortamda serildi. Serim işlemi boyunca, sıkıştırma ve vakum işlemleri uygulandıktan sonra destek malzemesi olarak soyma katları kullanıldı. Ardından, bu kompozit laminat plakalar üzerinde dolgu onarımı için delikler delindi. Delikler altı numuneden oluşan üç gruba ayrılmış ve Loctite 9396 yapıştırıcı ve çeşitli miktarlarda kırılmış cam elyafı içeren farklı malzemelerle doldurulmuştur. Onarımlardan sonra, onarılan numunelerin kalite kontrolü için tahribatsız bir test yöntemi olan Manuel Ultrasonik Darbe Eko (MUPE) kullanıldı. MUPE sonuçları onarılan numunelerin iyi kalitede ve test için uygun olduğunu gösterdi. Ardından, 24 numune için 250 kN kapasiteli Instron 5985 makinesi kullanılarak çekme testleri yapıldı.

Bu çalışmanın sonucu, %50 kırılmış cam elyafı ve Loctite 9396 yapıştırıcı kullanılarak yapılan onarımla yüzde 5'e varan bir iyileşme elde edildiğini göstermektedir. Tapa onarım teknolojisi ile kompozit yapının mekanik özelliklerinin geliştiği gözlemlenmiştir.

**Anahtar Kelimeler:** Tapa onarımı, kompozit, yapıştırıcı, karbon elyaf.

## CONTENTS

M.Sc. THESIS EXAMINATION RESULT FORM .....	ii
ETHICAL DECLARATION .....	iii
ACKNOWLEDGMENTS.....	iv
ABSTRACT .....	v
ÖZ .....	vi
NOMENCLATURE.....	ix
LIST OF TABLES.....	x
LIST OF FIGURES .....	xi
<b>CHAPTER 1 .....</b>	<b>1</b>
INTRODUCTION.....	1
1.1 Review of Related Works .....	3
1.2 Aim of the Study .....	7
<b>CHAPTER 2 .....</b>	<b>9</b>
COMPOSITE REPAIR TECHNOLOGY .....	9
2.1 Composites in Aircraft .....	10
2.2 CFRP Usage in Aircraft .....	11
2.3 Damages in Composites.....	12
2.3.1 Damage Detection .....	12
2.4 Composite Repair.....	14
2.4.1 Repair Procedure .....	15
2.4.2 Plug Repair.....	17
<b>CHAPTER 3 .....</b>	<b>19</b>
EXPERIMENTAL METHOD .....	19
3.1 Lay-up Process .....	20
3.2 Curing Process .....	24
3.3 Ultrasonic Inspection .....	27
3.4 Machining Process .....	27
3.5 Plug Repair Process .....	29
3.6 Inspection of the Repair .....	30
3.7 Tensile Test.....	33
<b>CHAPTER 4 .....</b>	<b>35</b>
RESULTS AND DISCUSSION.....	35

4.1 Results .....	35
4.1.1 Results of Lay-up Process .....	35
4.1.2 Results of the Repair Process .....	36
4.1.3 Inspection Results .....	36
4.1.4 Results of Tensile Test .....	37
4.2 Discussion .....	53
<b>CHAPTER 5 .....</b>	<b>55</b>
CONCLUSION .....	55
<b>CHAPTER 6 .....</b>	<b>56</b>
FUTURE WORKS .....	56
<b>REFERENCES .....</b>	<b>57</b>
<b>CURRICULUM VITAE .....</b>	<b>61</b>

## NOMENCLATURE

### Acronyms

ASTM	American Society for Testing and Materials
BVID	Barely Visible Impact Damage
CFRP	Carbon-Fiber Reinforced Polymer
FEA	Finite Element Analysis
FRP	Fiber-Reinforced Polymer
FSH	Full Screen Height
kN	Kilonewton
lb	Pound
MPa	Megapascal
MUPE	Manual Ultrasonic Pulse Echo
NDI	Non-Destructive Inspection
NDT	Non-Destructive Techniques
SEM	Scanning Electron Microscopy
UD	Uni Directional
UI	Ultrasonic Inspection

## LIST OF TABLES

<b>Table 2.1</b> Carbon fiber properties [26] .....	11
<b>Table 2.2</b> Repair procedure for Loctite 9396 adhesive and glass chopped fiber plug repair in carbon fiber laminate .....	18
<b>Table 3.1</b> Technical properties of carbon fiber reinforced epoxy prepreg composite HEXPLY M91/IM7/34RC/UD/194/12K .....	20
<b>Table 3.2</b> Stacking sequence of carbon fiber preregs .....	22
<b>Table 4.1</b> Tensile test results .....	37
<b>Table 4.2</b> Test results for the unrepaired specimens .....	40
<b>Table 4.3</b> Test results for the repaired with Loctite adhesive specimens .....	42
<b>Table 4.4</b> Test results of the repaired with 30% chopped fiber + Loctite 9396 adhesive .....	45
<b>Table 4.5</b> Test results for the repaired with 50% chopped fiber + Loctite 9396 adhesive .....	47

## LIST OF FIGURES

<b>Figure 2.1</b> Scarf repair [34] .....	14
<b>Figure 2.2</b> Patch repair [34].....	15
<b>Figure 2.3</b> Damaged composite structure repair process [34].....	16
<b>Figure 2.4</b> Plug repair process .....	17
<b>Figure 3.1</b> Carbon fiber prepreg .....	21
<b>Figure 3.2</b> Peel ply .....	23
<b>Figure 3.3</b> Vacuum bag process .....	24
<b>Figure 3.4</b> Autoclave .....	24
<b>Figure 3.5</b> Graph of curing process .....	25
<b>Figure 3.6</b> Cure cycle graphs of specimens.....	26
<b>Figure 3.7</b> Carbon fiber prepreg plate before drilling the hole .....	27
<b>Figure 3.8</b> Technical drawing of the specimen .....	28
<b>Figure 3.9</b> Tab and plate.....	28
<b>Figure 3.10</b> Repaired specimens a) with Loctite 9396 adhesive, b) repaired with 30% chopped fiber + Loctite 9396 adhesive, c) with 50% chopped fiber + Loctite 9396 adhesive and d) unrepaired specimen .....	30
<b>Figure 3.11</b> Ultrasonic Pulse Echo experimental setup.....	32
<b>Figure 3.12</b> Screen of Manual Ultrasonic Pulse Echo .....	32
<b>Figure 3.13</b> Tensile test experimental setup.....	33
<b>Figure 3.14</b> Instron 5985 tensile testing system.....	34
<b>Figure 4.1</b> Failure mechanism of unrepaired specimens.....	38
<b>Figure 4.2</b> Failure behavior of the unrepaired specimen 2.....	39
<b>Figure 4.3</b> Graph of force-displacement of the unrepaired specimens .....	40
<b>Figure 4.4</b> Failure mechanism of repaired with Loctite 9396 adhesive .....	41
<b>Figure 4.5</b> Graph of force-displacement of the repaired with Loctite adhesive specimens .....	42
<b>Figure 4.6</b> Failure mechanism of the repaired with %30 chopped glass fiber and Loctite 9396 adhesive .....	43
<b>Figure 4.7</b> Failure behavior of the specimen 6 repaired with 30% chopped glass fiber and Loctite 9396 adhesive.....	44
<b>Figure 4.8</b> Force-displacement of the repaired with 30% chopped fiber + Loctite 9396 adhesive.....	44
<b>Figure 4.9</b> Failure mechanism of the specimens repaired with 50% chopped glass fiber and Loctite 9396 adhesive .....	46

<b>Figure 4.10</b> Graph of force-displacement of the repaired with 50% chopped fiber + Loctite 9396 adhesive .....	47
<b>Figure 4.11</b> Brittle failures of the repaired specimens .....	48
<b>Figure 4.12</b> The specimen on which no brittle failure was observed.....	49
<b>Figure 4.13</b> Failure mechanism of specimens after tensile test.....	50
<b>Figure 4.14</b> Scan photos of the second unrepaired specimen .....	51
<b>Figure 4.15</b> Scan photos of the sixth specimen repaired with the 30% chopped glass fiber and Loctite 9396 adhesive .....	52



# CHAPTER 1

## INTRODUCTION

In recent years, composite materials have been increasingly used more due to their superior properties with respect to metallic materials. They have become widespread especially in the aerospace industry [1]. Indeed, composite materials were first used in military aircraft in the 1960s, and later in civil aircraft. The aircrafts where composite materials have been excessively used as follows: Boeing 757, Boeing 767, Boeing 777, Airbus A310, A320, A330, and A340 [2]. Composite materials have used in half of all materials in the Boeing 787 [1, 3]. In the aircraft, composite materials are used in various parts. These materials were seen in secondary structures in the 1980s and used in the tail part of the A320 [4].

Fiber-Reinforced Polymer (FRP) composites have recently been used in many industries due to their advantages. They have low density and high strength, which are especially preferred in the aerospace industry. Carbon Fiber-Reinforced Polymer (CFRP), which is one of the fiber reinforced composites, includes carbon fibers and plastic resin. It has generally been preferred due to features such as rigidity, high strength, low density, and corrosion resistance [5]. In recent years, CFRP composites have drawn great attention in the primary structures of aircraft due to these advantages.

CFRP composites are preferred in the primary and secondary structural applications in aircraft [6]. They are used in aircraft such as Boeing 757 to reduce weight and increase strength. In fact, this has both reduced weight and fuel consumption by 20% in this type of aircraft [3]. A 380 is the first aircraft which used CFRP composite in wing box. The weight of aluminum wing box was 8.8 tons, whereas that of composite wing box was 5.3 tons. The weight reduction has been achieved by the use of the composite materials in the wing box [7]. The same weight reduction was achieved with the usage of carbon composites in the A310 aircraft, which had earlier used the aluminum alloy. In the A320, these materials are used in panels and secondary structures [2].

Using composite in aircraft not only helps to reduce weight but also provides convenience in production. The fin box made of CFRP contains fewer fasteners than the fin box made of metal [2]. According to the Soutis [2], when the structures contain fewer fasteners, manufacturing processes are easier.

Energy saving has always been important in aircraft. As composite materials have replaced metallic materials in aircraft, aircraft weight has significantly decreased. This weight reduction has increased energy efficiency, which has reduced fuel consumption [6]. One kg reduction in the total weight of aircraft saves 2900 liter of fuel in a year [2]. This weight reduction is observed to be around 800 kg in A320. In A310, CFRPs are also used in the primary structures. This reduces the weight of the structure by about 400 kg.

Composite materials are also used in the tail of C17, the weight of which has been decreased by 20% [2]. Composites are also used in the A350-XWB, and its fuel consumption has decreased by a quarter [1]. This weight reduction, achieved by composite materials, has been effective in reducing emissions in the aerospace industry [8].

Composite materials can be damaged while the material is being produced or during its service life. The most common type of damage during manufacturing processes is porosity. Impact damage is frequently seen during the service life of the aircraft [1].

Despite these advantages, maintenance and repairs of composites can be costly and difficult in air vehicle operation. As CFRPs are expensive materials, the repair of these composites is critical [2].

Composite materials are expensive materials, so the repair of composite materials is more cost-effective than replacing them. Damage type and size, as well as where it occurs must be known to decide whether to repair or replace them. In metallic structures, dents have been observed to occur after damage, yet in composite materials, no visible effect has been seen. In composite materials, damages occur during service or due to external reasons. These damages may not be visible to the naked eye, so certain

techniques have been applied for damage assessment. This is important for optimum repair.

Composites are more complex than metallic materials; for this reason, specific repair techniques have been developed. There are many steps to be followed for the repair of the composites. Firstly, the location of the damaged is determined. Then, the depth of the damage from the surface is examined. Next, a suitable repair method is chosen. According to whether these repairs are permanent or temporary. This is followed by preparation of the damaged area for repair. Finally, non-destructive methods are used for quality control of the repair [9].

Non-Destructive Techniques (NDT) can be used to detect damages. These techniques are diverse, and no single technique can detect all types of damage. Since composites have inherent non-homogenous and anisotropic material behavior, identifying the location of the damage can be difficult. If the damage is repairable, repair technology should be used instead of replacement of the damaged composite component. Because composite materials are expensive, it might not be economical to replace the damaged region of the composites. It is also important that the method to be used is cost-effective [10].

Many studies have been done in the literature on the maintenance and repair of composite structures with modern methods, which are reviewed in the following section.

## **1.1 Review of Related Works**

With the increasing use of composites in aircraft structures, studies about composite repairs have gained importance. This study investigates several scarf repair, external patch repair, and plug repair methods. Based on the findings, the effectiveness of these three repair methods for minor damages in aircraft is discussed.

One of the repair methods for aircraft structure is composite patch repair. In this repair method, the composite patch is adhered, and applied to restore the strength of the structures [10]. Adhesively bonded patch repair is an effective repair method to increase strength. According to the data obtained from the test result of Caminero et al. [11], strength of the unnotched specimen is 996 MPa. On the other hand, the repair with a

single patch is 685 MPa, and that with patches on both sides is 704 MPa. The recovering strength was found to be about 70% of the unnotched strength. The results showed that this type of repair is effective. This work, proved that, in a double patch repair, stress concentration at the edge of the hole is lower than that in a single side patch repair. Baker [12] studied adhesively bonded patch repair. In this study, the wing skin of aircraft A15-5 was repaired with boron/epoxy as a patch material, and FM 73 as an adhesive material. After patch repair, the patch was controlled periodically, and the repaired area of the wing skin was monitored for several years. No crack was observed.

Three different specimens, i.e. undamaged, damaged, and repaired were investigated by Soutis et al. [13]. The steps followed in this study were similar to steps followed by Caminero et al. [11]. A hole with a 10 mm diameter was drilled at the center; then, hole patches with 35 mm in diameter were bonded on each side of the specimen. For damage location and type identification, scanning electron microscopy was applied. Compression test of CFRP laminates was performed by Instron servo-hydraulic machine. Firstly, failure mechanism was examined by using X-ray radiography and Scanning Electron Microscopy (SEM), and then a 3D Finite Element Analysis (FEA) was used to find a stress field in the repaired specimens. Damages was estimated by means of both. Afterward, predictions were compared with the experimental results. One finding of this study was that damages tend to occur near the holes, in which was also confirmed by the study of Cheng et al. [14]. Additionally, the laminate with a hole was found half as strong as the original one.

Many studies exist in the literature that changed the repair parameters. The effects of several repair parameters on two types of external bonding patch repair were pointed out by Çetinkaya et al. [15]. The repair parameters were applied to wet lay-up patches and prepreg patches. These parameters included patch bonding area, bonding length, adhesive, and patch thickness. In this study, unidirectional tensile test was applied to the specimens. Ultimate failure loads were presented; a hole was drilled and used as a damage, which was to be repaired with composite patches in laboratory. The change of parameters influenced the results. As the patch thickness increased, so did the ultimate failure loads of the structures in prepreg bonding. In the case of wet layup patches, the failure loads were increased by up to three plies, and the loads were decreased after three

plies. Parallel to both prepreg patch length and wet layup length increase, the failure loads increased by up to 40 mm in length. When the length exceeded 40 mm, failure load decreased in wet layup patches. An increase in the bonding area led to an increase in failure load of repairs. The results showed that maximum ultimate failure load can be obtained with adhesive thickness of three plies. According to the results of this study, the increase in patch thickness caused an increase in weight in the aircraft structure. Also, if bonding area increases, weight also increased. For this reason, a rise in patch thickness and bonding area should be optimum. Soutis et al. [13] also reported difficulties caused by the increase in the total weight of the structure and in the size of the patches.

Researchers have tried to achieve more effective results by changing different parameters. Stresses must be transferred to the patch and minimize adhesive failure. To this end, Bouiadjra et al. [16] proposed improving the adhesive properties. Patch thickness is another important parameter for reducing stress. Stress decreases with increasing patch thickness. The effectiveness of patch repairs depends on the appropriate patch geometry. For this reason, the effects of the geometry of composite patches have been investigated in the literature. Benyahia et al. [17] used four different types of composite patch geometry to investigate this effect. They observed that the elliptical patch is the most suitable, whereas the trapezoid patch is the most ineffective one among the patches used. Patch weight can be reduced by changing the geometry of the patch. Bouiadjra et al. [18] investigated two types of patches, which are rectangular and trapezoidal, and the trapezoidal patch proved more advantageous due to reduced weight. The effect of patch geometry on repair was also studied by Kashfuddoja and Ramji [19]. Different types of patch geometry, which are manufactured CFRP laminates have were used. The patch shapes were circle, rectangle, square, ellipse, octagon, and oval. The damaged CFRP laminates were repaired by an adhesively bonded patch. Caminero et al. [11] compared scarf repair and external bonded patch repair. Types of specimens have been used in this study. While the first specimen consisted quasi-isotropic woven M21/HTA carbon-fiber epoxy, the second one consisted a quasi-isotropic unidirectional M21/T700 carbon fiber-epoxy. Scarf patch repair was applied to the first specimen, which had impact damage, while external bonded patches were used for the second specimen. Certain steps were followed for the second specimen with

composite patch repair. A hole was drilled at the center, and the damaged area was removed. Then external patches were bonded to the specimen. The epoxy adhesive was used for bonding. In an electromechanical testing machine, tensile tests were performed both one-side and two-sides patch. Digital Image Correlation was used to examine the damage. The results pertaining to the FEA and Digital Image Correlation Technique were compared on external bonded patch repairs, and they were found compatible. In this study, it was observed that drilling had caused damage around the hole. Depending on the type of the damage around the hole, both strain distributions and strain concentrations were changed. In addition, one side repair cannot attain a complete repair, only protecting the damaged area. For bonded composite patch repair, Digital Image Correlation is an efficient and practical technique. Like many other researchers, Cheng et al. [14] derived conclusions about scarf repair and external bonded patch repair. In this study, it was concluded that external bonded patch repair was less efficient than the scarf repair, which. The scarf repair, on the other hand, requires special equipment and causes a waste of time. The external bonded patch repair causes stress concentration. Indeed, high stress concentrations were observed around hole edges in this study. Different types of parameters have to be used to decrease stress concentrations. Tensile behavior of composites which are repaired by external patch was handled by Cheng et al. [14]. Tensile test was performed to determine the load transfer to the repaired part. Numerical and experimental results were used to obtain the optimum patch design. The FEA was used to determine stress distribution in the repair region. Infrared thermography has been used to investigate failure modes. In this study, it was found out strength of structures can restore up to 90 % of their original state with patch repairs.

Based on a comprehensive literature review, comparative analysis of patch, scarf, and plug repairs were made. It was observed that the part repaired on both sides with external patch repair provides up to 70% of the original strength. In addition, while this repair is performed, the weight of the structures is increased. On the other hand, the scarf repair is more effective than the adhesively bonded patch repair, although it requires special equipment. The plug repair can be preferred for it is time saving and facilitates the repair process.

The plug repair is more cost effective and time efficient than the patch and the scarf repairs. Michelis [20], obtained stress reduction in the plug repair. The damaged area of the laminate was cut in a circular shape and removed. Then the hole was plugged with a filling material. In this study, which is based on the basic concept of the scarf, patch repairs, and a comparative approach, firstly, the scarf repair was studied, and it was selected to be used for major structural damage. It was reported that performance of the repair was advantage but the complexity of the repair method was a disadvantage. Regarding the patch repair, more than 70% of the structures cannot be repaired and this technique is used only for minor damages. Plug repair is also used for minor damage but the authors still recommend it for this minor damage since plug repair is easy to apply.

In the study of Vlattas [21], composite laminates were produced by using prepregs with hand lay-up process. Specimens were produced with 24 plies in the following dimensions: 215 mm in length and 50 mm is width. A hole was drilled at the center of each as a damage, and these holes were filled with filling material. This study examines, the patch and the plug repair techniques, analyzing the strength values of undamaged, damaged, and repaired laminates comparatively by an Instron servo-hydraulic machine. Stress concentrations of damaged and repaired specimens were calculated using a 2D Finite Element Analysis. Stress concentration at the edge of the hole was reduced by means of plug repair. It was concluded that the stress concentration factor of the repaired specimens was lower than the stress concentration factor of the unrepaired specimens.

## **1.2 Aim of the Study**

Composite structures have long been preferred to reduce weight in aircraft. However, the biggest problem encountered in these materials is the damage. Therefore, the increasing usage of composites in aircraft has led to the need for new technologies for repair.

This study is focuses on repair of minor damage of aircraft structures and development of effective repair methods. Composite materials are costly, so the repairing is often preferred to replacing the damaged composite component. One way to do this is the plug repair, which is a practical method used for minor damage [22].

Ideal for cosmetic repair, this type of repair does not provide a permanent repair. This thesis, investigates whether the plug repair can provide a permanent repair, exceeding its presently common cosmetic use.

Three different specimens were tested in this study: undamaged, with holes drilled as damage, and with holes repaired with adhesive plug repair. By applying the tensile tests to these three specimens, the study intended to evaluate the effectiveness of the repair. ASTM D3039, often used to specify tensile properties of composites, was performed for polymer matrix reinforced composites [23]. D695 is a standard test method for open hole tensile strength of polymer matrix composite laminates [24]. In this test standard, the most common specimen is a rectangular shape. An open hole tension was employed using the adhesive plug. The mechanical properties of the composite structure are expected to increase with the plug repair technique.

# CHAPTER 2

## COMPOSITE REPAIR TECHNOLOGY

Advanced material applications are needed to reduce weight and improve performance [25]. Thus, in recent years, the usage of composite materials has increased, due to their superior properties. Having advantages over conventional metals in terms of their structural applications, their use has become widespread in various industries in recent years [1].

Composite materials have very critical in aerospace vehicles and components, as it is used in primary structures such as wing, fin, and control surfaces. These non-metallic materials have features that can compete with advanced metallic materials such as Al-Li. However, unlike metallic materials, they are not simple to produce because of these properties and their complicated designs.

Thanks to composite materials, weight reduction and high reliability are ensured in all aerospace vehicles. Aerodynamic performance has increased in aircraft with the use of composite materials. They are effective in all-weather operations, as well as ensuring lightning protection and erosion resistance [26].

The material selection of aerospace components is very critical. Weight reduction, which also reduces fuel consumption, is one of the primary criteria in material selection. However, while aiming to reduce the weight of the aircraft, there should be no loss in performance. In metal structures, quality assurance testing is minimum, which reduces the cost. Nevertheless, it is conducted by non-destructive tests for composite materials, which increases the cost. Considering the fact that it is absolutely necessary to apply quality control applications to the primary structures of the aircraft and that the usage of composite materials has increased in large aircraft, one can see that it is a costly alternative. Indeed, approximately 16% of the total weight of the Airbus A380 is composite materials. In large commercial aircraft, composite materials have been used in both primary and secondary structures. Therefore, as the amount of composite used in the aircraft increases, so does the cost of the tests applied for quality control [27].

## 2.1 Composites in Aircraft

Composite materials are frequently encountered in the aviation industry, including military and commercial aircraft as well as helicopters. The amounts of composites used in airframe structures applications have increased over years.

They are primarily used in secondary structures and as the technology, and usage of composite materials has advanced, they are now used in primary structures as well. Composites are suitable for complex shapes. As they allow less use of fasteners and joints, stress concentration is reduced [28].

In 1968, while 1.5% composite material is used in the F-15 Eagle which is a military aircraft, this rate increased to 26% in the AV-8B Harrier II Aircraft in 1978. While the composite rate of the A380 among commercial aircraft was 24%, this rate increased to 52% in the A350. Composite materials are used in various parts of an aircraft. For instance, it is used in wing, empennage, and fuselage skins of Boeing MDD Harrier aircraft. These materials are also used in the flight control structure panels, doors, and engine nacelle of the A380 [25].

In military aviation, composite materials are increasingly used due to the advantage of weight reduction. The first aircraft to use composite is the Lear fan 2100. Composite materials are used in the body and propeller blades, doors, control surfaces, and wing boxes. These materials have been used for doors and control surfaces of Boeing 757 and Boeing 767. The fin box of the Airbus A300/A310 aircraft was made of carbon fiber composites. Composite materials are applied in different proportions in different aircraft. For example, they were used in 26% of AV-88 Aircraft, while 50% composite was used in the Advanced Tactical Fighter Aircraft. They were used in the entire Beech Starship Aircraft [6].

Empennage of the Boeing 777 was produced by the CFRP, and 1500 lb weight saving was achieved by doing so. Half of the structural parts of the Boeing 787 were made of composite materials, and 20% reduction in total weight has been achieved by using these materials. [28, 29].

## 2.2 CFRP Usage in Aircraft

In the face of difficulties that may arise in the complex design of aircraft, materials such as carbon fibers are used to produce high performance structures. Advanced composites such as carbon fiber have been widely used in military and civil aircraft. Carbon fiber reinforced composites reduce weight, resist corrosion and fatigue, and ensure a high degree of optimization. Despite all these advantages, these materials are exposed to impact, thus internal and production defects.

Carbon fiber reinforced composites are fibrous and anisotropic [28]. Different production techniques are used due to the anisotropic and inhomogeneous structure of non-conventional carbon fiber composites. Therefore, various techniques are applied in quality control for these composites. The usage of different production and quality control techniques requires the development of computer technology and analysis. Despite its weaknesses, it is widely in the aerospace industry, thanks to the overwhelming benefits. Although carbon fiber composites comprise an extensive portion of aircraft, their total structural weight remains low. For example, in F15 aircraft, the composite weight is 2% of the total aircraft weight [28].

Carbon fibers are used in various aerospace vehicles such as satellite, military aircraft, and helicopter. These materials are applied on solar panels of satellite; wings, fin, control surface, fuselage panels of military aircraft, and all structural parts of advanced helicopters [26]. Table 2.1 presents the carbon fiber properties commonly used in aerospace industry. While the F18 aircraft uses composites by 19%, this ratio is 22% in the F22, and 40% in the Eurofighter [28].

**Table 2.1** Carbon fiber properties [26]

Fiber	Density (g/cm <sup>3</sup> )	Modulus (GPa)	Strength (GPa)	Application Areas
Carbon	1.77-1.80	220-240	3.0-3.5	Widely used for almost all types of parts

The use of the CFRP in transport aircraft is advantageous because it reduces weight and accordingly provides fuel consumption. In 1983, the CFRP was used in the rudder of Airbus A300 and in the vertical fin of Airbus A310. With fin material, which is manufactured from metallic materials, 2000 parts were used. However, when it was produced from composite material, the number of parts decreased to 100. Such a remarkable decrease in the number of parts, reduces both the total weight and production cost significantly [28, 29].

While impact damages on metals are deformed locally, these damages cannot be seen visually in the CFRP. That is why damage detection is a challenging issue. Density of the CFRP is half that of the aluminum, and its strength is much higher. Nevertheless, it is not as simple to manufacture them as it is to manufacture aluminum alloys. Furthermore, the repair knowledge of carbon fiber materials is not as advanced as that of aluminum alloys [30].

## **2.3 Damages in Composites**

Damages occur in composite materials during the production stage or its service life. They can occur for various reasons such as hail stones, runway debris, and dropping of tools. The damages inside which are invisible, reduce the compression strength. Reduction in compression strength is measured to be around 60% by Barely Visible Impact Damage (BVID).

Various damages occur due to mechanical and environmental conditions. A common type is impact damage caused by mechanical factors [29]. Composite airframe components are easily damaged in service. These service damages are; voids, delamination, surface damage, wrong drilled holes, mechanical damage, abrasion, penetrations, disbands, hole elongation, dent, edge damage, and impact damage [27].

### **2.3.1 Damage Detection**

Detecting the type and size of composite damage has been a challenging issue ever since the first usage of composites. In fact, damage mechanisms are complex due to the uniqueness of the industry, so, different types of non-destructive inspection (NDI) techniques are used. These techniques are costly but essential to obtain high-quality

composite structures. Structural damages need to be precisely located and measured, and internal properties of structures need to be analyzed accurately, which can be accomplished by non-destructive testing [6,27].

In metal structures, quality assurance testing is done minimally, which reduces the cost. By contrast, these control tests performed for composite materials are non-destructive tests, which increases the cost. However, it is utterly necessary to apply quality control applications to the primary structures of the aircraft [28].

As the use of composites in aircraft increased, the importance of non-destructive inspection also increased. Inspection of thick primary structures of aircraft is difficult and non-destructive inspection of these structures is more critical [29]. That is non-destructive inspection is vital for critical structures of aircraft. Fortunately, non-destructive tests and inspection effectively assesses, manufacturing quality, and detects damage [27].

Different methods are used depending on the type and development of damage. It is not possible to control all types of damages with a single NDT method. For example, visual inspection is performed for surface damage such as dents, visible crack, and punctures. This inspection method is a simple and low-cost inspection method, requiring minimum equipment. When visual inspection is possible, there is no need to apply other costly non-destructive techniques. Tap test is another method used commonly, so its development is also important. However, it is not suitable for inspection of minor damage. Therefore, in this study uses, NDT methods to repair minor damage in composite. More precise results are obtained by using more advanced methods such as ultrasonic inspection (UI).

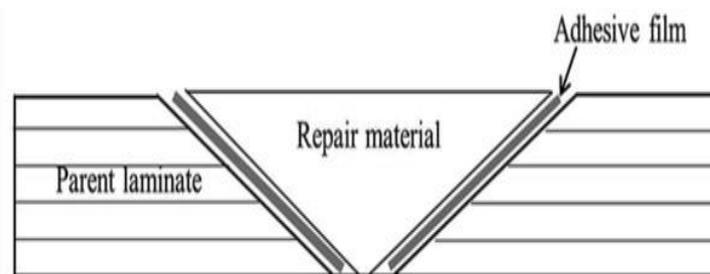
In the last decade, ultrasonic techniques proved promising for non-destructive inspection, and they are becoming an effective alternative [31]. With this method, which is suitable for detecting subsurface damages, the depth of the defect in the laminate is also determined. It can also inspect defects such as delamination, void, and disbanding [6, 32]. However, ultrasonic inspection is difficult to use in complex shapes [33]. Still ultrasonic testing and radiography are the most frequently used destruction inspection methods in the aerospace industry. Finally, it should be noted that, as the use of

composite materials in large aircraft expands, the amount of composite used in the aircraft increases, so does the cost spent for the tests applied for quality control [27].

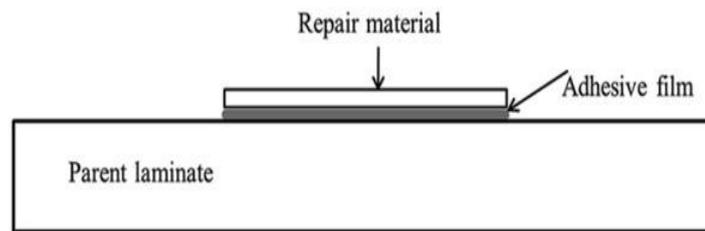
## 2.4 Composite Repair

Whether the damages in composite materials can be repaired or not is decided according to certain criteria. If the damage is negligible, then the repair process is not implemented. If the repair process is not economical, the structure is replaced rather than repaired. If the structure is considered to be critical, the decision is often made in favor of repair. Structures are divided into three groups according to their criticality: primary, secondary, and tertiary. Primary structures are critical for the operation of the aircraft, while tertiary structures do not affect it [27].

External patch repair and scarf repair, which are mostly used in aerospace industry composite materials, are shown in Figure 2.1 and Figure 2.2. In terms of application, the two repairs are different from each other. In a scarf repair, special equipment is required to remove the damaged material from the parent material, so this type of repair is suitable for thick laminate composites. External patch repair, however, is a simpler and faster type of repair. Both are types of bonded joint repair and they are complex repair [34].



**Figure 2.1** Scarf repair [34]



**Figure 2.2** Patch repair [34]

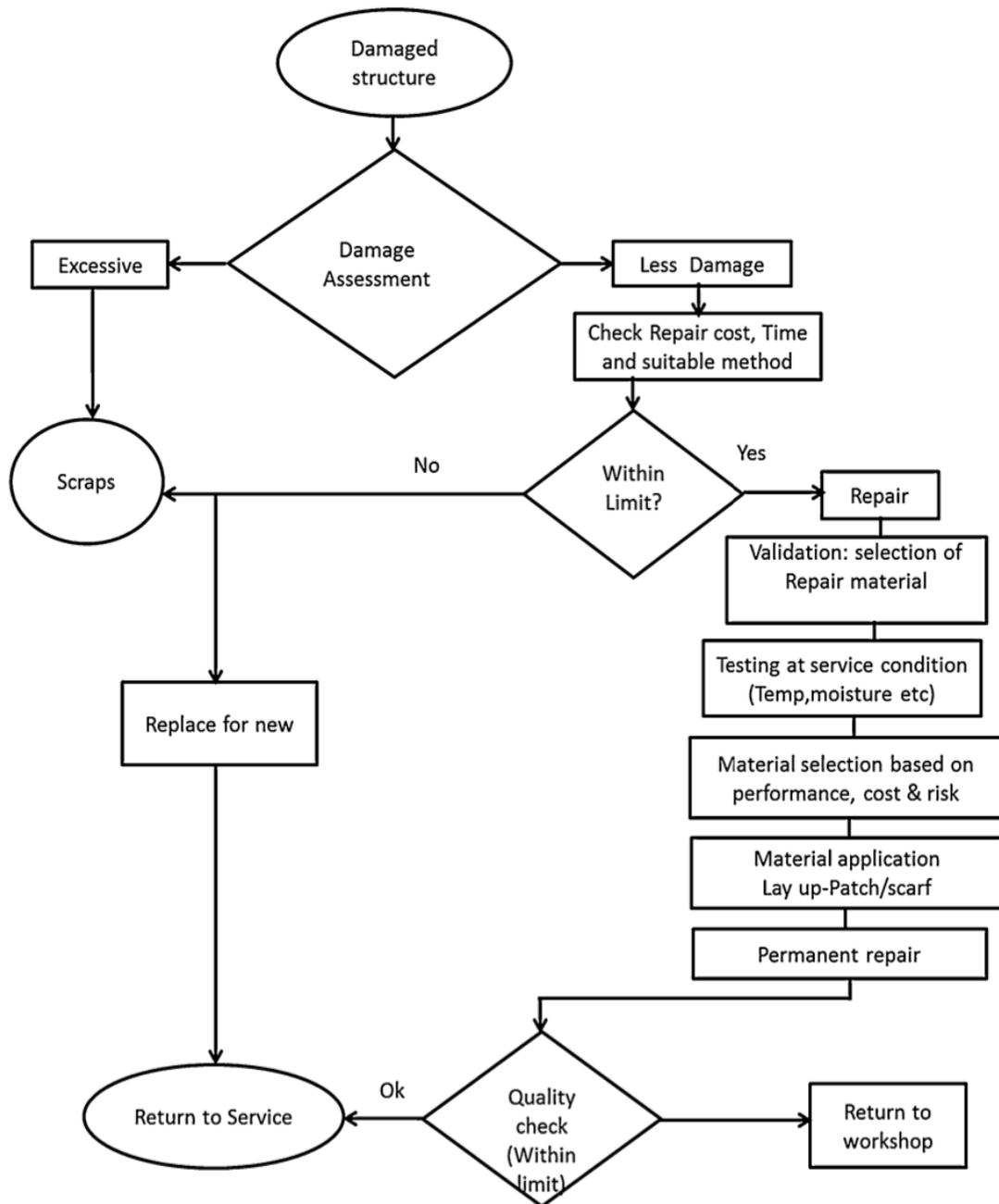
### 2.4.1 Repair Procedure

For the repair of composite materials, firstly an appropriate NDT is used to determine the damaged area and its size. The scheme of the repair to be performed is planned. Then, the damaged area is removed. Repair scheme is applied as planned [25]. Figure 2.3 illustrates the steps of the repair process laid down by Budhe et al. [34].

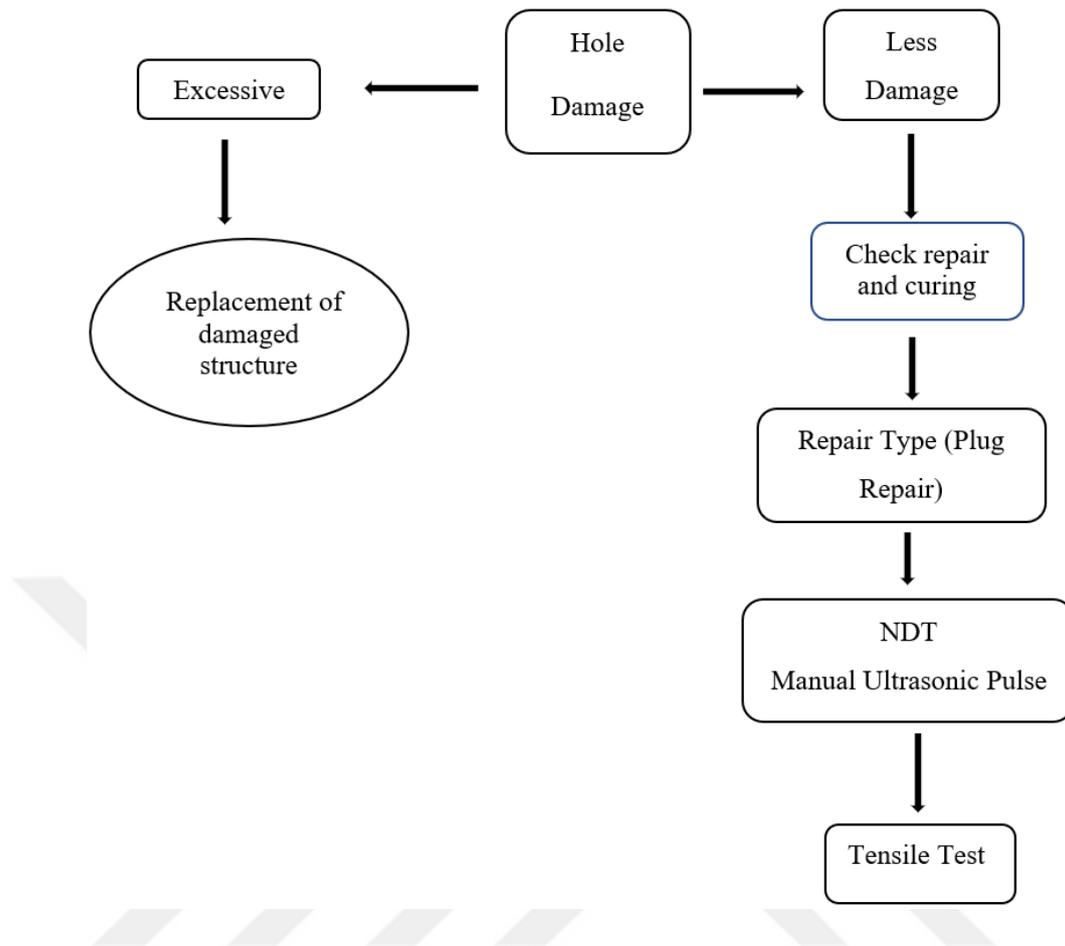
In this study, a diameter of 25,4 mm hole was used as a damage and whether repairing or replacing it was investigated. The method to be applied for composite repair was determined. Adhesive and chopped fiber to be used for plug repair was selected. Autoclave and oven time for curing was taken into consideration. After the plug repair, type of non-destructive method to be used for quality control was determined. The steps for this plug repair are seen in the Figure 2.4.

Autoclave, drying and curing, curing temperature are important parameters for repair. The drying procedure is applied to remove moisture before repair. Drying and curing are the longest operations in repair. If the drying time is reduced, the total repair time decreases.

The repair time applied to the composite should be as short as possible and the simplest repair that can be effective should be preferred. The purpose of the repair is to restore structural stiffness and strain. The repair should result in minimum weight gain, but maximum success in returning the structure to its original state. The type of repair is applied according to the type of structure [27].



**Figure 2.3** Damaged composite structure repair process [34]



**Figure 2.4** Plug repair process

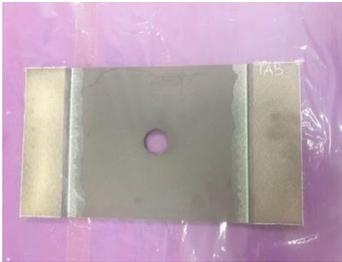
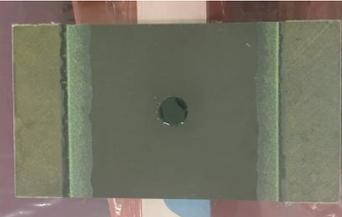
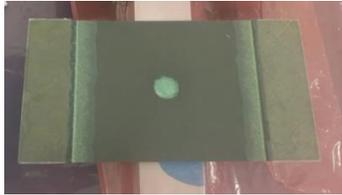
### 2.4.2 Plug Repair

External patch repair and scarf repair are the most used methods in aerospace industry for composite materials, but both have drawbacks. Material removal is a challenging issue in the scarf repair. In the patch repair, the surface preparation must be excellent, or else the repair efficiency decreases. Performance of bonded patch repair depends on material and geometrical parameters.

In the aerospace industry, it is critical that the repair be of superior quality and efficient, yet at the same time low cost. The plug repair is low cost and time saving compared to the patch and the scarf repairs.

Procedure for the application of Loctite 9396 adhesive and glass chopped fiber plug repair on a damaged carbon fiber laminate is summarized in Table 2.2.

**Table 2.2** Repair procedure for Loctite 9396 adhesive and glass chopped fiber plug repair in carbon fiber laminate

Step 1	Hole 25.4 mm in diameter is drilled as a damage of all 24 specimens.	
Step 2	<p>Glass fiber piece is added to the lower surfaces of all the 18 specimens. (a)</p> <p>Then, release film and metal plate which are backing elements, are added on glass fiber piece. (b)</p>	 <p>a) Glass fiber piece</p>  <p>b) Release film and metal plate</p>
Step 3	<p>The hole is filled with</p> <p>a) Loctite 9396 adhesive and</p> <p>b) Glass chopped fiber and Loctite adhesive.</p>	 <p>a) Loctite 9396 adhesive</p>  <p>b) Glass chopped fiber and Loctite adhesive</p>

# CHAPTER 3

## EXPERIMENTAL METHOD

In this thesis study, minor damages to aircraft structures were repaired by the plug repair technology. The specimens were 364 mm in length, 330 mm in width, and 3.68 mm in thickness (22 plies). The drilled holes have a diameter of 25,4 mm. The dimensions of the specimens and hole can be seen in Figure 3.8. These processes were carried out for the plug repair in the following order:

- Prepreg carbon fiber composite HEXPLY M91/IM7/34RC/UD/194/12K plies were used to produce composite laminate plates.
- After the lay-up process, curing is performed in the autoclave which took approximately three hours.
- An ultrasonic inspection was applied to examine porosity in the specimens.
- Composite laminate plates were accepted as damaged with the criterion of 25.4 mm through hole drilling. This process was applied with OKUMA CNC machine.
- Repairing process was executed in atmosphere-controlled environment. The holes were divided into three groups of six specimens and were filled with different materials which are Loctite 9396 adhesive and various amounts of chopped fiber.
- After the repairs, Manual Ultrasonic Pulse Echo (MUPE) was used as NDT method for quality control of the repaired specimens.
- All test specimens were prepared; ASTM D3039, which is a standard tensile test, was performed using tensile test machine Instron 5985.

Lay-up process, repairing process, and all tensile tests were implemented in atmosphere-controlled environment. The temperature of the atmosphere-controlled area was  $23\pm 2^{\circ}\text{C}$ .

### 3.1 Lay-up Process

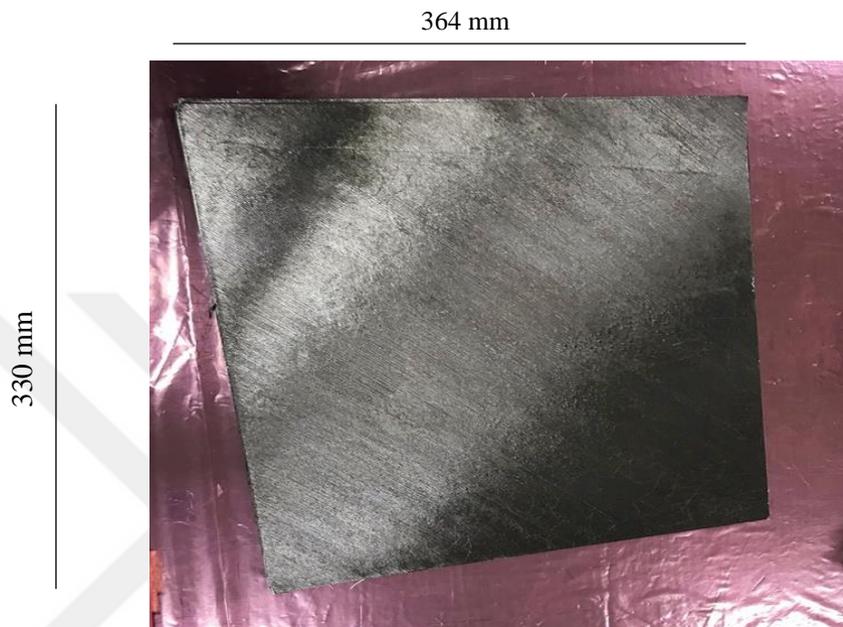
Composites have been produced using prepregs since 1970. Composite production is carried out by hand or automatedly. While hand lay-up is generally preferred for large parts, automated tape lay-up is applied in empennage parts such as rib. Hand lay-up is the process of cutting prepregs in ideal sizes and laying them. Since this process is done manually, it is slow and has a margin of error [33].

In this study, carbon fiber reinforced epoxy prepreg composite HEXPLY M91/IM7/34RC/UD/194/12K plies were used to produce composite laminate plates. This epoxy prepreg ply is a uni directional (UD) tape, which is frequently used in the aviation industry due to its low weight and high strength. Thickness of the cured ply was 0,184 mm. Other technical properties of the epoxy prepreg are presented in Table 3.1.

**Table 3.1** Technical properties of carbon fiber reinforced epoxy prepreg composite HEXPLY M91/IM7/34RC/UD/194/12K

Longitudinal Modulus (MPa)	17000
Transverse Modulus (MPa)	8800
In-plane Shear Modulus (MPa)	5500
Longitudinal Tensile Strength (MPa)	2700
Transverse Tensile Strength (MPa)	105

The shape of laminate plate which was determined as rectangular, had the dimensions of 330 mm and 364 mm. The thickness of the laminate was 3.68 mm. Figure 3.1 displays carbon fiber epoxy prepreg plies which were laid in the atmosphere-controlled environment.



**Figure 3.1** Carbon fiber prepreg

Prepreg plies are increasingly used in high performance applications in the composite industry and other sectors. According to Rubino, the prepreg system is fast and versatile; for this reason, it is used for urgent repair [35]. In this study, each specimen consists of 22 plies, 20 of which are carbon fiber prepreps and two are peel plies. The sequences and directions of the carbon fiber prepreg plies are shown in Table 3.2. Carbon fiber epoxy prepreg plies are implemented according to the lay-up sequence configuration. In this composite plate, the compaction process was implemented for four times. During the lay-up process, the compaction process was performed for the first ply and then for every fourth ply. The processes were realized through a vacuum bag or by hand and gained low void percentage on the composite laminate plate.

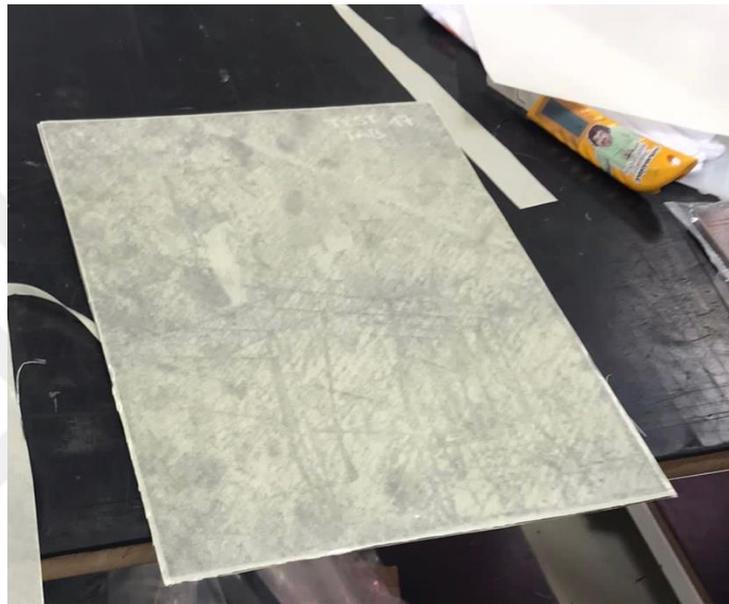
In the stacking sequence, the lay-up process is usually performed in four different angles: 0 degree, +45 degree, -45 degree and 90 degree [33]. In this sequence, carbon

fiber epoxy prepregs are laid using an angle of 45 degree. Laying the prepregs at an angle of 45 degree increases the tensile test results of the specimens.

**Table 3.2** Stacking sequence of carbon fiber prepregs

<b>GROSSPRINT PLANNING</b>			
<b>LINE</b>	<b>PLY</b>	<b>MATERIAL DESIGNATION</b>	<b>DIRECTION</b>
1	P000	M21/48%/BI9842 WET PEEL PLY	OPT
2	P001	TMS-N192-17-01	[-45]
3		COMPACTION	
4	P002	TMS-N192-17-01	[+45]
5	P003	TMS-N192-17-01	[-45]
6	P004	TMS-N192-17-01	[+45]
7	P005	TMS-N192-17-01	[0]
8		COMPACTION	
9	P006	TMS-N192-17-01	[-45]
10	P007	TMS-N192-17-01	[+45]
11	P008	TMS-N192-17-01	[-45]
12	P009	TMS-N192-17-01	[+45]
13		COMPACTION	
14	P010	TMS-N192-17-01	[90]
15	P011	TMS-N192-17-01	[90]
16	P012	TMS-N192-17-01	[+45]
17	P013	TMS-N192-17-01	[-45]
18		COMPACTION	
19	P014	TMS-N192-17-01	[+45]
20	P015	TMS-N192-17-01	[-45]
21	P016	TMS-N192-17-01	[0]
22	P017	TMS-N192-17-01	[+45]
23		COMPACTION	
24	P018	TMS-N192-17-01	[-45]
25	P019	TMS-N192-17-01	[+45]
26	P020	TMS-N192-17-01	[-45]
27	P021	M21/48%/BI9842 WET PEEL PLY	OPT

In the lay-up process, peel plies were used as backing materials after compaction and vacuum operations were implemented. Vacuum was applied to check leaks in the specimens. Upon completion of the lay-up process, a peel ply was applied to the top and the bottom plies. These peel plies have no structural importance and are used as a backing element. The peel ply which prevents the composites from deteriorating during autoclave, is shown in Figure 3.2.



**Figure 3.2** Peel ply

After the peel ply was applied to both the top and bottom of the specimens, final vacuum bag was implemented so that no air would remain between the plies. The vacuum operation, which took up to 15 minutes, is shown in Figure 3.3. The vacuum bag was put into the autoclave, where thermocouples are located.



**Figure 3.3** Vacuum bag process

### **3.2 Curing Process**

After the lay-up process, curing was applied to the specimens. Pressure and temperature are important in the curing process, which takes place in an autoclave, as shown in Figure 3.4. Firstly, pressure, and then heat, was implemented.



**Figure 3.4** Autoclave

Curing and drying take the longest time among all repair processes. Time required for curing is approximately three hours, and autoclave process is 400 minutes. Figure 3.5 illustrates the duration of autoclave process.

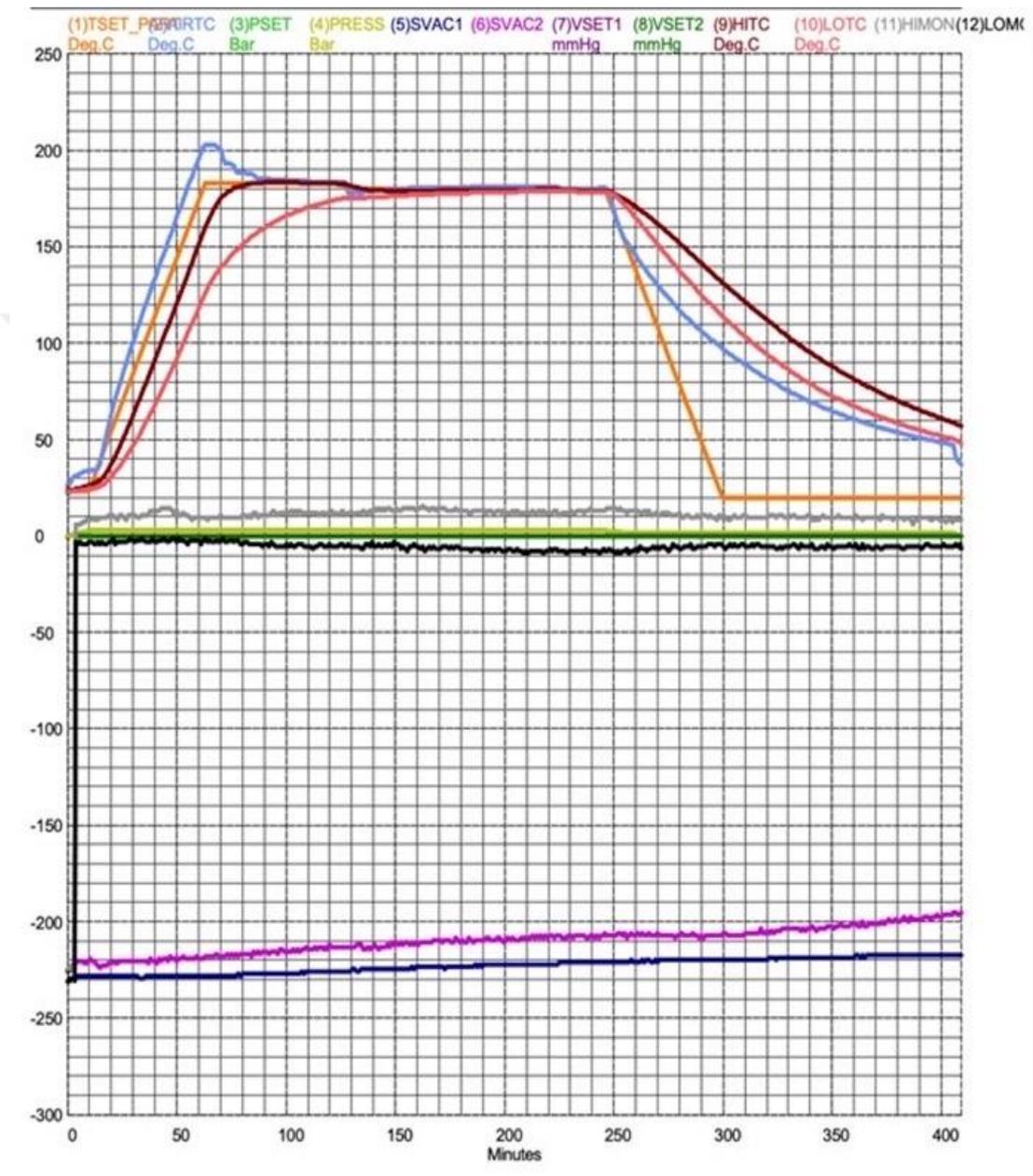
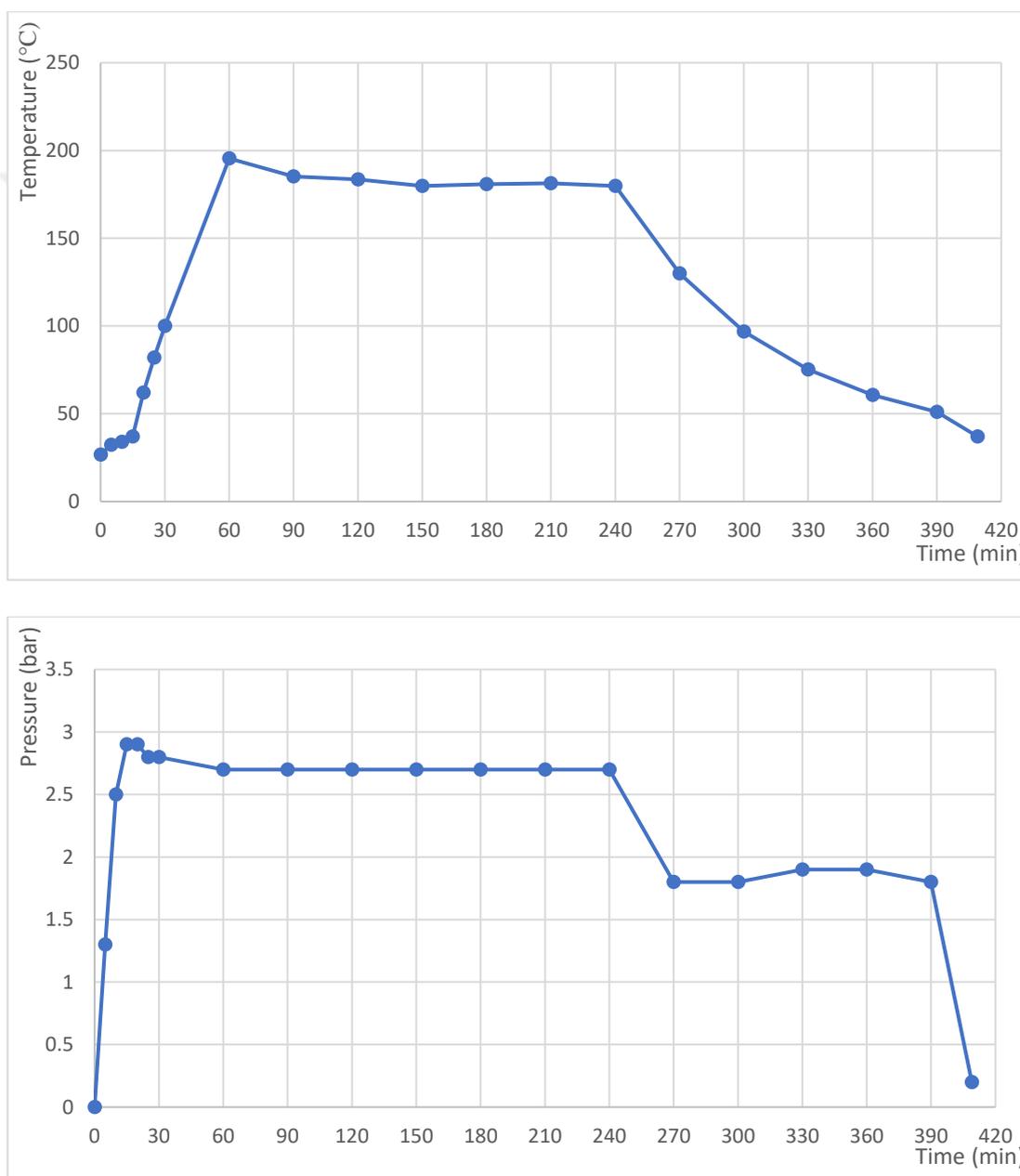


Figure 3.5 Graph of curing process

Figure 3.6 shows the cure cycle graph of the specimens. (2)AIRT in the graph shows the temperature inside the autoclave. (4)PRESS is the pressure exerted on the specimens from the autoclave. (9)HITC indicates the maximum temperature of the specimens while (10) LOTC shows the minimum temperature of the specimens. The highest temperature of the specimens is measured as 202.9 °C at 64 minutes. The highest pressure is measured as 2.9 bar between the 15 and 21 minutes.



**Figure 3.6** Cure cycle graphs of specimens

After the curing process, demolding was performed. Burrs were cleaned from the specimens. After the demolding, ultrasonic inspection was applied.

### 3.3 Ultrasonic Inspection

After the curing process, an ultrasonic inspection was applied to examine porosity in the specimens, which is performed after demolding. Ultrasonic inspection is a NDT necessary for the analysis of internal properties of structures. In the last decade, ultrasonic techniques have turned out to be highly promising for non-destructive inspection and an effective alternative [31]. Figure 3.7 illustrates the carbon fiber epoxy prepreg plate, which has the dimensions of 330 mm and 364 mm. It was subjected to curing process and ultrasonic inspection.

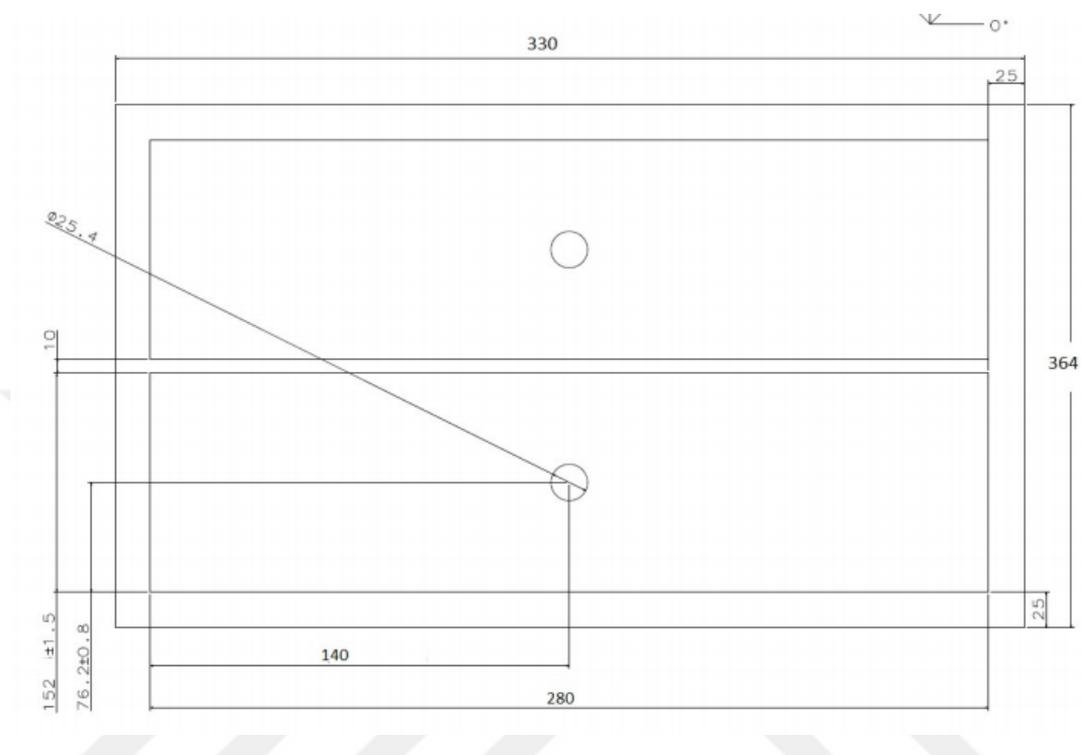


**Figure 3.7** Carbon fiber prepreg plate before drilling the hole

### 3.4 Machining Process

It was mentioned that damage scenarios of aircraft structures were simulated in this study. Composite laminate plates with 25.4 mm hole drilling were accepted as damaged. This process was applied with OKUMA CNC machine. The dimensions of the specimens and hole are presented in Figure 3.8.

After the drilling operation, tabs and plates were joined with film adhesive in the atmosphere-controlled environment. Figure 3.9 illustrates the joining of the tab and plate. As a final process, autoclave was applied to the specimens.



**Figure 3.8** Technical drawing of the specimen



**Figure 3.9** Tab and plate

### 3.5 Plug Repair Process

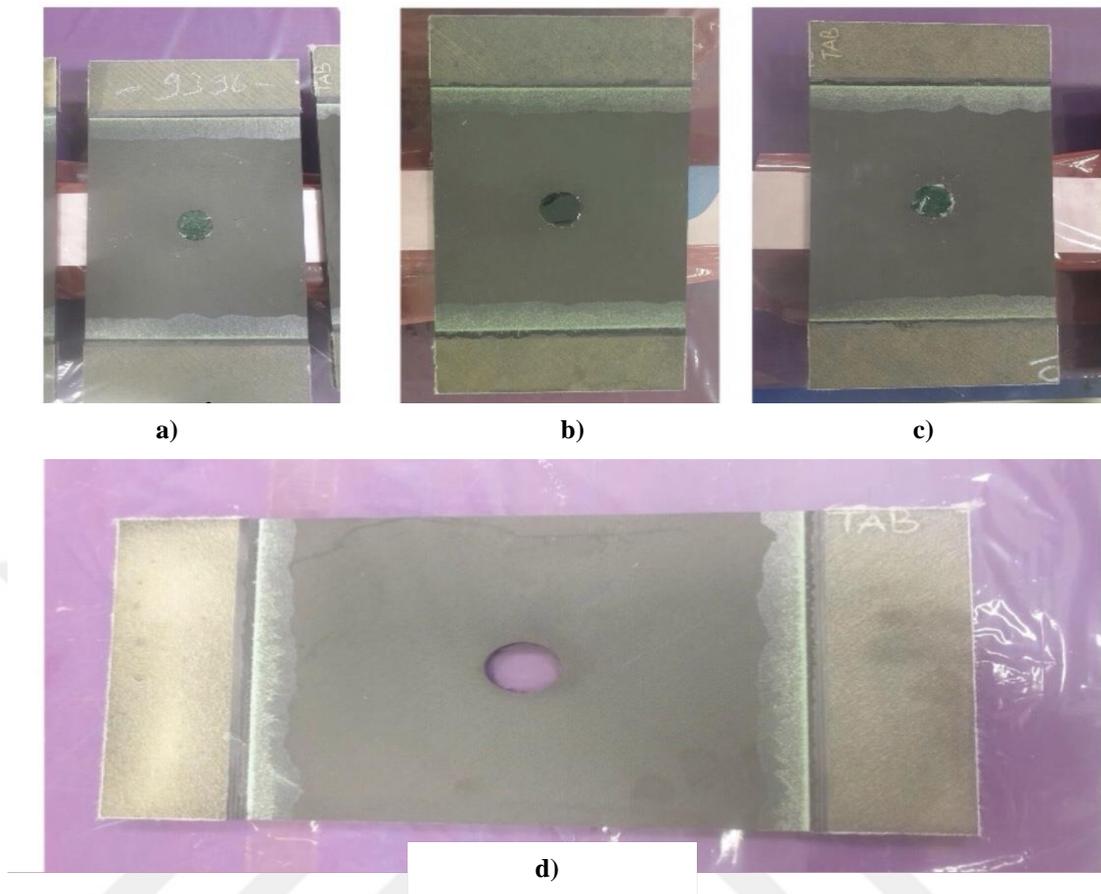
Composite laminate plates were accepted as damaged according to the criteria of 25.4 mm hole drilling. The holes on the laminate plates were repaired with three different plug repair technology methods. The repairing process was executed in atmosphere-controlled environment. The holes were divided into three groups of six specimens and were filled with different materials, which are Loctite 9396 adhesive and various amounts of chopped fibers.

LOCTITE EA 9396 Aero, which is an epoxy paste adhesive, is also known as Hysol EA 9396. This adhesive cures in 25°C within three to five days and has a low viscosity. In this research, Loctite adhesive was applied in atmosphere-controlled area at 25°C. It cured in the oven for 90 minutes at 80 °C to cure [31].

Before the repair process, some treatments were applied to the specimens. Glass fiber piece was added to the lower surfaces of all the 18 specimens. Then, release film and metal plate, which are backup elements, were added onto the glass fiber piece. These materials prevent specimens from dispersion during the curing process.

Three different repair processes were applied to the specimens, which were in three groups of six. In the first group, only epoxy 9396 adhesive was used to fill the hole. In the second and third groups, epoxy adhesive and chopped fiber were used collectively. In the second group, the holes on the specimens were filled in the following ratio: 30% chopped fiber and 70% adhesive 9396. In the third group, this ratio was as follows: 50% chopped fiber and %50 adhesive 9396. Figure 3.10 displays the types of repairs and the unrepaired specimens.

After the hole filling process, the upper surfaces of the specimens were covered with glass fiber pieces; then, release film and metal plate were added, as is done on the lower surfaces. Vacuum process was implemented to three groups. After the vacuum procedures, 18 specimens, which had been repaired, were kept in the oven for curing 90 minutes at 80 °C.



**Figure 3.10** Repaired specimens a) with Loctite 9396 adhesive, b) repaired with 30% chopped fiber + Loctite 9396 adhesive, c) with 50% chopped fiber + Loctite 9396 adhesive and d) unrepaired specimen

### 3.6 Inspection of the Repair

After the repairs with Loctite adhesive and glass chopped fibers, Manual Ultrasonic Pulse Echo (MUPE) was used as NDT method for quality control of the repaired specimens. The MUPE is a standard inspection method for laminate composites. With this method, porosity, layer porosity, delamination, layer and multilayer delamination, foreign object damage, debond, and unbonding layer can be observed.

It is well-known that this inspection was used to monitor, porosity and layer delamination. Porosity was observed in the repaired region in all the 18 specimens used, and no measurement was made in this area. When the porosity was high, noise occurred. Back wall echo decreased with increasing the porosity. Therefore, with this technique,

measurement was made around the repaired hole, not on the repaired area. Layer delamination was observed around the repaired hole region.

In this set up, probe and screen are used. Probe, at the tip of the MUPE device, is applied to various points from around the hole towards the edge. In this inspection, the probe diameter is  $\frac{1}{4}$  in and its angle is 0 degree. The defect size to be measured should be larger than the probe diameter; in this context, defects are greater than  $\frac{1}{4}$  in.

In this inspection method, back wall echo is used as 80% full screen height (FSH). The FSH refers to the height of the display on a screen. The amplitude of an echo is described by its relative height to the screen. An echo that reaches the top of the screen is indicated to be FSH or 100% FSH. A value 80% FSH is common in this technique [38]. Device frequency is 5 MHz.

The MUPE device is calibrated using an inspection reference panel suitable for similar test parts and according to the thickness, material type, speed, and sound. This technique is performed for the inspection of 24 specimens, six of which are drilled only and, 18 of which are repaired.

Figure 3.11 illustrates the MUPE experimental set up. Before the process, the parts are cleaned with water or a high viscosity gel. In this inspection, only water is used as the surface of the specimens is not rough. Gel or water ensures that the roughness on the surface does not disturb the signals.



**Figure 3.11** Ultrasonic Pulse Echo experimental setup

In six specimens with holes, the probe is applied to various points from around the hole towards to edge. The same process is applied to the repaired specimens. The probe was not applied on glass chopped fiber and Loctite 9396 adhesive filling in the repaired 18 specimens.



**Figure 3.12** Screen of Manual Ultrasonic Pulse Echo

Figure 3.12 shows the result of the inspection. In this technique, ultrasonic signals are sent on the inspected parts. The distance between the second and third pick shows the thickness of the inspected parts. Between the second and the third picks on all inspected parts exists, a pick shorter than the second and third one. This pick is influenced by laying type and laying direction. The red line in Figure 3.12 is produced by the standard procedure, and back wall echo is used as 80% FSH.

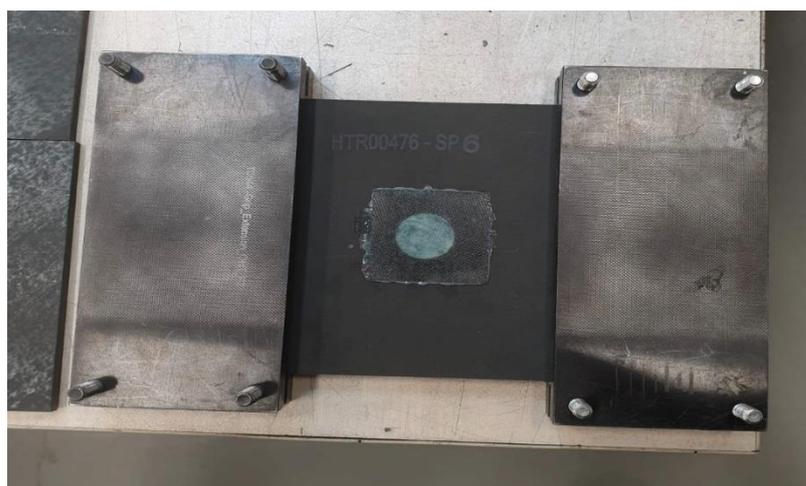
### 3.7 Tensile Test

Before the tensile test, the specimens were marked and divided into four groups consisting of six specimens.

- The specimens in the first group were unrepaired, with holes only.
- The second group was repaired with only Loctite 9396 adhesive.

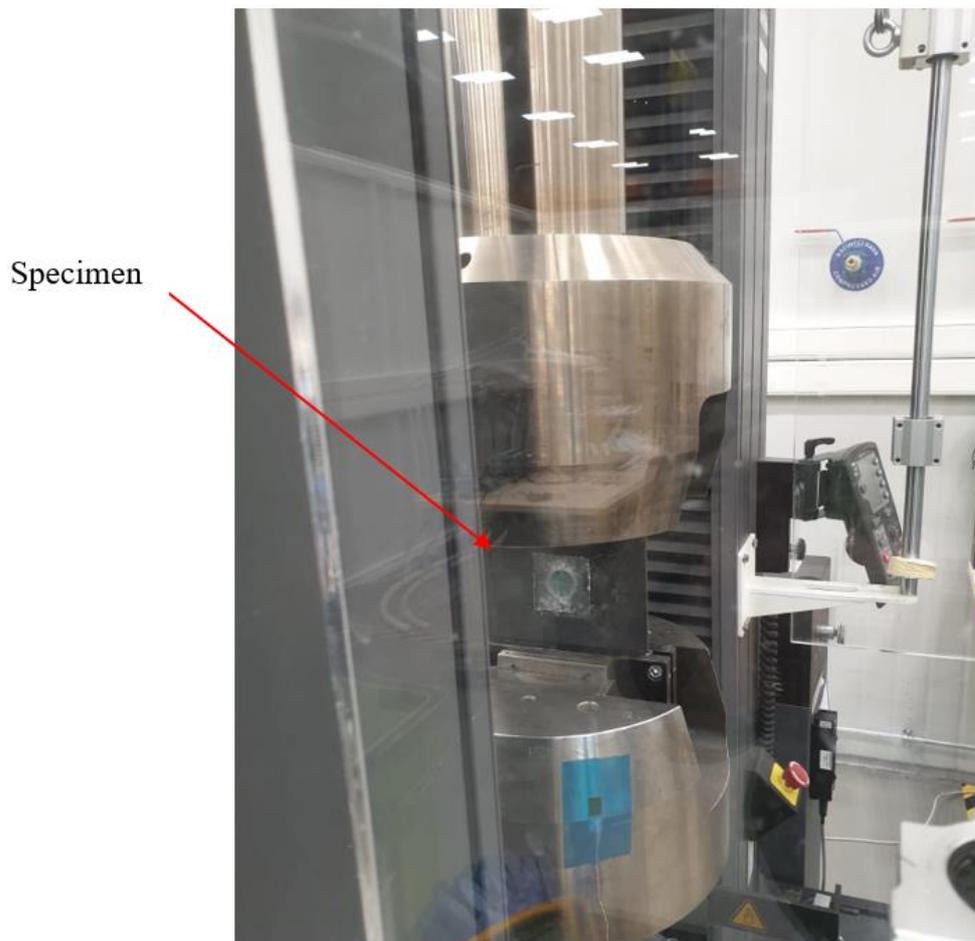
The third and last groups included repaired specimens:

- In the third group, specimens were repaired with 30% chopped fiber and Loctite 9396 adhesive.
- The final group was repaired with 50% chopped fiber and Loctite 9396 adhesive.



**Figure 3.13** Tensile test experimental setup

All test specimens were prepared as shown in Figure 3.13; the standard tensile test was applied using machine Instron 5985. Tabs are implemented at the gripping area. Figure 3.14 illustrates how the specimen was prepared for the tensile test. ASTM D3039, which is a standard test method for tensile properties of polymer matrix composite materials. Tensile tests for 24 specimens were carried out by using an Instron 5985 machine with 250 kN capacity, and at a typical test speed of 2 mm/min. Tensile tests were implemented in a  $23\pm 2^{\circ}\text{C}$  atmosphere-controlled area.



**Figure 3.14** Instron 5985 tensile testing system

# CHAPTER 4

## RESULTS AND DISCUSSION

### 4.1 Results

In this research, the effects of the plug repairs on minor damages in aircraft structures were investigated. The plates were produced by using HEXPLY M91/IM7/34RC/UD/194/12K carbon fiber preregs. A total of 24 specimens were used, and plate dimensions were 364 mm in length, 330 mm in width, and 3.68 mm in thickness. The plates consisted of 22 plies, 20 of them being carbon fiber preregs. After applying the lay-up process, autoclave process, ultrasonic inspection, machining, repairing, curing, final inspection, and tensile test were carried out. Most of these processes were implemented in the atmosphere-controlled environment. The results drawn are presented in the following section.

#### 4.1.1 Results of Lay-up Process

During the lay-up process, precautions were taken to increase the tensile strength, prevent defects in the specimens and deterioration during autoclave process. In the lay-up process, more carbon fiber preregs could be laid using an angle of 45 degrees. Laying the preregs at an angle of 45 degrees increased the tensile test results of the specimens. Compaction process was performed after four or five plies of batches. It was realized through a vacuum bag or by hand and was effective in preventing potential defects on composite specimens. Upon completion of the lay-up process, a peel ply was applied to the top and the bottom ply. These peel plies had no structural importance and were used as a backing element. They prevented the composites from deteriorating during autoclave. After the peel ply was applied to both top and bottom of the specimens, final vacuum bag was implemented so that no air would exist between the plies.

#### **4.1.2 Results of the Repair Process**

The holes on the laminate plates were repaired with three different plug repair technology methods. The repairing process was conducted in atmosphere-controlled environment. The holes were divided into three groups of six specimens and were filled with different materials which are Loctite 9396 adhesive and various amounts of chopped fiber. Before the repair process, some treatments had been applied to the specimens. A glass fiber piece was added to the lower surfaces of all the 18 specimens. Then, release film and metal plate which are backup elements, were added onto the glass fiber piece. After the hole filling process, the upper surfaces of the specimens were covered with glass fiber pieces; then, release film and metal plate were added, as was done on the lower surfaces. These materials prevent specimens from dispersion during curing process. During the repair, it is difficult to fill 30% and 50% of the chopped glass fiber in the hole volume. Their weight having been determined by precision weighting, the chopped fiber and Loctite 9396 adhesive is filled into the specimens. Since this process is done by hand, there is a margin of error. The application of adhesive with or without glass chopped fibers both result in effective repairs from the visual inspection perspective. No separation/gap was seen between the parent laminate plates and the plug repair. Similarly, no shrinkage was observed on the adhesive in any of three repair scenarios.

#### **4.1.3 Inspection Results**

After curing process, demolding was performed, and burrs were cleaned from the specimens. This was followed by an ultrasonic inspection checking for porosity in the specimens. Consequently, the machining process, applied with OKUMA CNC, accepted 25.4 mm hole drilling as damaged. No nonconformity was observed around the holes after drilling. The laminate materials were of superior quality and passed the NDT.

After the repair processes, the MUPE was used as the NDT method for quality control of the repaired specimens. In six specimens with only drilled holes, the probe at the tip of the device was applied to various points from around the hole towards the edge. The process was repaired for the repaired specimens. Probe was not applied to glass chopped

fiber and Loctite 9396 adhesive fillings. Minor picks were observed between the second and third picks, which were shorter than the second and third picks. This shorter pick is shown in Figure 3.12. If it crosses the red line in the figure, it indicates that there is a defect. As a result of the inspections made on a total of 24 specimens, no such situation was encountered. The MUPE, which is a standard inspection method for laminate composites, revealed that the specimens were in high quality and suitable for tensile testing.

#### 4.1.4 Results of Tensile Test

When the test specimens were prepared, the standard tensile test was applied using the Instron 5985 machine. An increase was expected in the mechanical properties of the materials with the plug repair. It was concluded that the strength of specimens repaired with both chopped glass fiber and Loctite 9396 adhesive was higher than the strength of the specimens filled with only the Loctite 9396 adhesive.

Tensile test results for four groups (unrepaired, repaired specimens with Loctite 9396 adhesive, repaired with 30% chopped fiber + Loctite 9396 adhesive and repaired with 50% chopped fiber + Loctite 9396 adhesive) of 6 specimens are presented in Table 4.1. Mean maximum load and mean ultimate strength are shown in this table, revealing the expected increase in the values.

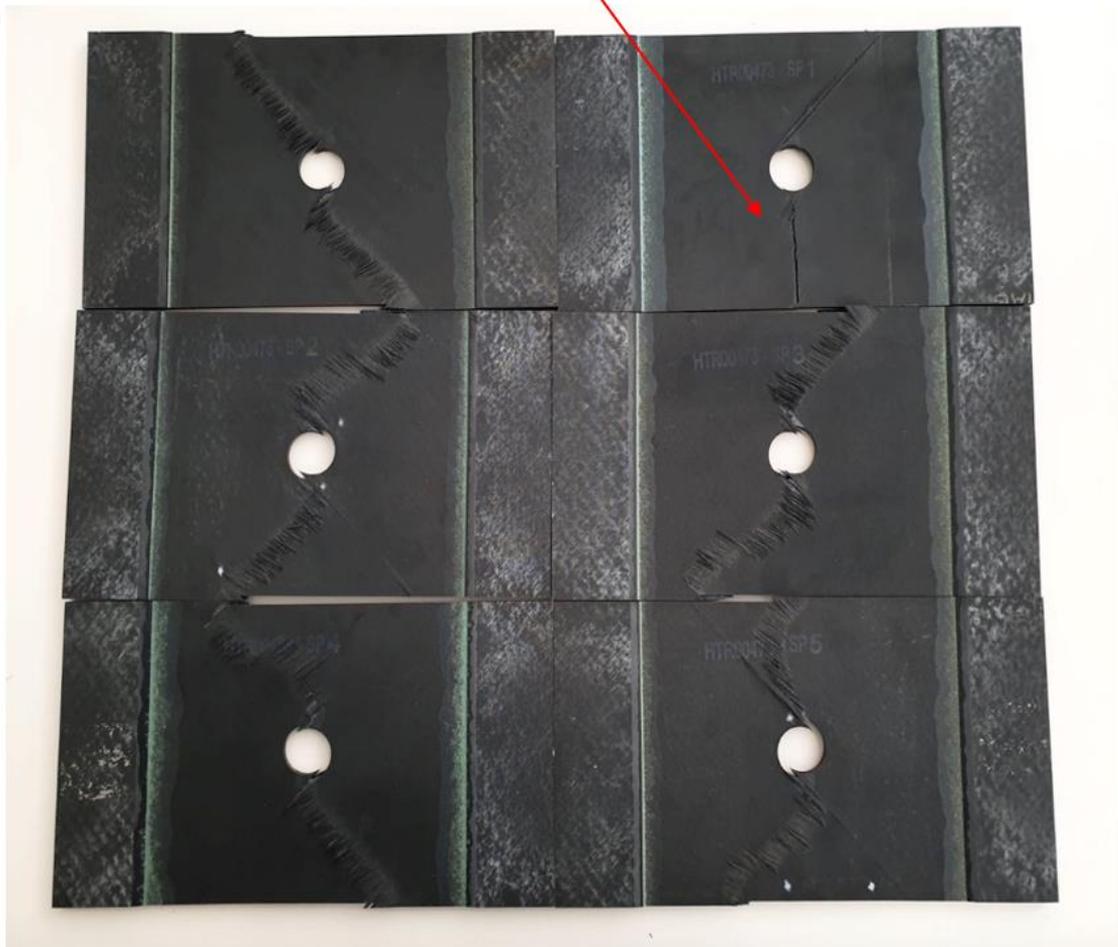
**Table 4.1** Tensile test results

	Mean Maximum Load (kN)	Mean Ultimate Strength (MPa)
Unrepaired Specimens	136,1	235,2
Repaired Specimens with Loctite 9396 Adhesive	141,7	246,2
Repaired with 30% chopped fiber + Loctite 9396 adhesive	142,1	245,7
Repaired with 50% chopped fiber + Loctite 9396 adhesive	143,9	248,3

The results of the tensile test demonstrate an improvement up to 5%, a result of the repair using 50% chopped glass fiber and Loctite 9396 adhesive. Maximum mean load is increased from 136.075 kN to 143.948 kN with the plug repair.

Figure 4.1 illustrates the follow-up tensile test of unrepaired specimens. Firstly, tensile test was applied to the six unrepaired specimens. Figure 4.1 shows the failure behavior of these six specimens. As can be seen here, the failure behavior of the second specimen which is shown in Figure 4.2, is different from the others.

Different failure behavior

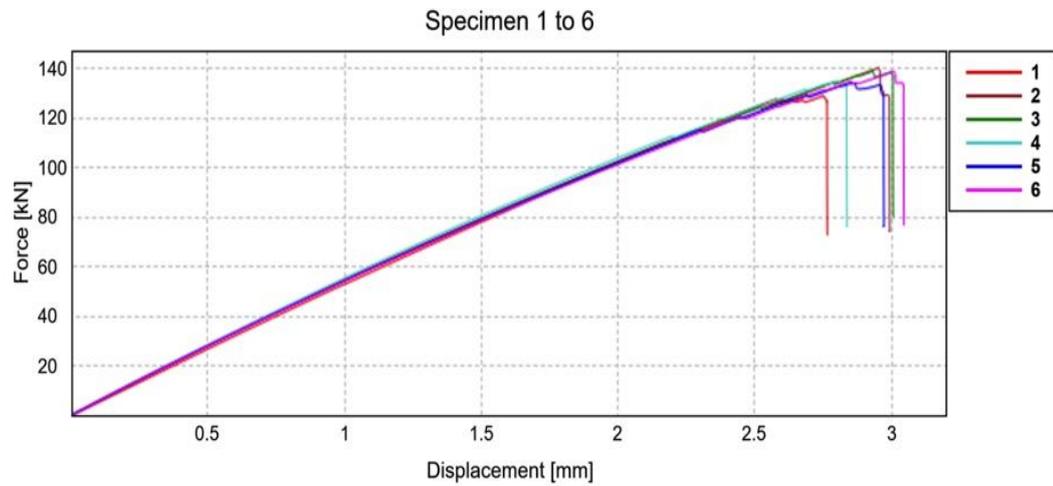


**Figure 4.1** Failure mechanism of unrepaired specimens

Figure 4.3 illustrates the force-displacement of unrepaired specimens. The maximum load and ultimate strength values measured according to the tensile test result are also shown in the Table 4.2. According to the tensile test results, the maximum load and ultimate strength are on the specimen 2.



**Figure 4.2** Failure behavior of the unrepaired specimen 2



**Figure 4.3** Graph of force-displacement of the unrepaired specimens

**Table 4.2** Test results for the unrepaired specimens

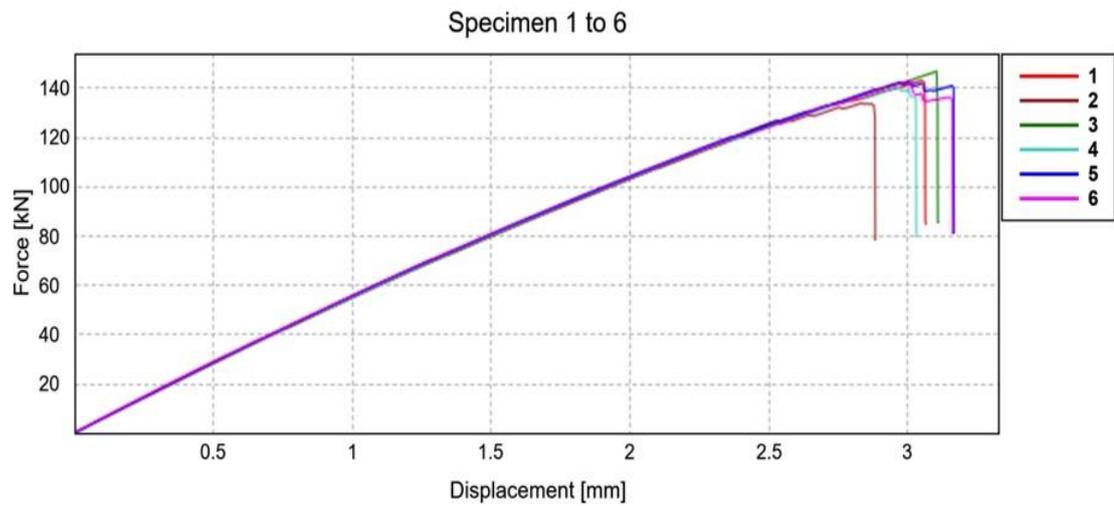
Specimen Number	Width	Thickness	Hole Diameter	Maximum Load	Ultimate Strength
	mm	mm	mm	kN	MPa
1	151.98	3.812	25.32	129.14	222.9
2	152.09	3.785	25.34	140.28	243.59
3	152.01	3.790	23.36	138.96	241.21
4	152.08	3.825	25.36	135.15	232.33
5	152.04	3.810	25.35	134.07	231.45
6	152.04	3.810	25.34	138.88	239.76
Maximum	152.09	3.825	25.36	140.28	243.59
Minimum	151.98	3.785	25.32	129.14	222.9
Mean	152.04	3.805	25.34	136.07	235.21
Standard deviation	0.04	0.01	0.02	4.16	7.76
Coefficient of variation	0.03	0.39	0.06	3.06	3.30
Standard error	0.01	0.004	0.008	1.7	3.18

Figure 4.4 illustrates the tensile test of the repaired with Loctite 9396 adhesive specimens. The test was applied to the six repaired specimens. Figure 4.4 shows the failure behavior of these six specimens. No different failure behavior is observed in this type of repair.



**Figure 4.4** Failure mechanism of repaired with Loctite 9396 adhesive

Figure 4.5 illustrates the force-displacement of the repaired Loctite adhesive specimens. The maximum load and ultimate strength values measured according to the tensile test result are also shown in Table 4.3.

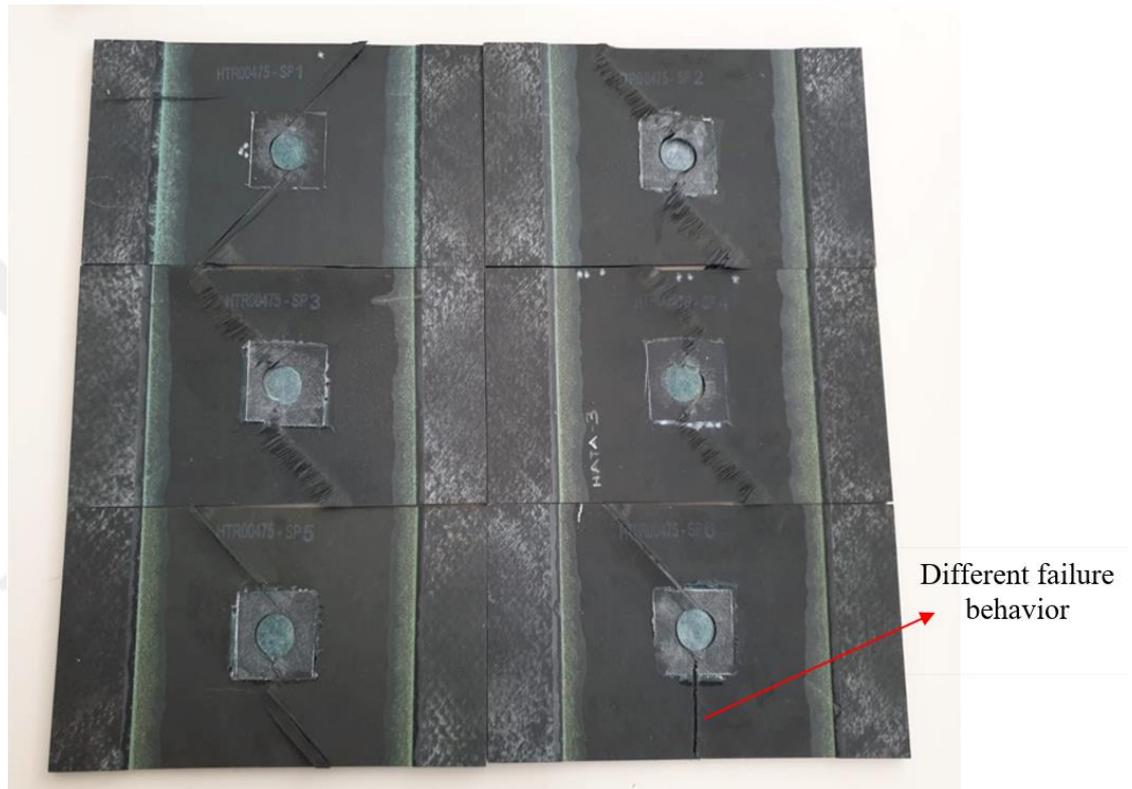


**Figure 4.5** Graph of force-displacement of the repaired with Loctite adhesive specimens

**Table 4.3** Test results for the repaired with Loctite adhesive specimens

Specimen Number	Width	Thickness	Maximum Load	Ultimate Strength
	mm	mm	kN	MPa
1	152.06	3.77	143.113	249.645
2	152.03	3.81	134.119	231.545
3	152.01	3.805	146.801	253.806
4	152.02	3.79	140.508	243.872
5	151.99	3.77	142.358	248.443
6	151.98	3.77	143.298	250.098
Maximum	152.06	3.81	146.801	253.806
Minimum	151.98	3.77	134.119	231.545
Mean	152.02	3.786	141.699	246.235
Standard deviation	0.03	0.02	4.24	7.88
Coefficient of variation	0.02	0.49	2.99	3.2
Standard error	0.012	0.008	1.73	3.22

Figure 4.6 illustrates the situation after the tensile test of the repaired with 30% chopped glass fiber and Loctite 9396 adhesive specimens. The test was applied to the six specimens as usual and Figure 4.6 shows the failure behavior of these six specimens. As can be seen in Figure 4.7, the failure behavior of the last specimen is different from the others.

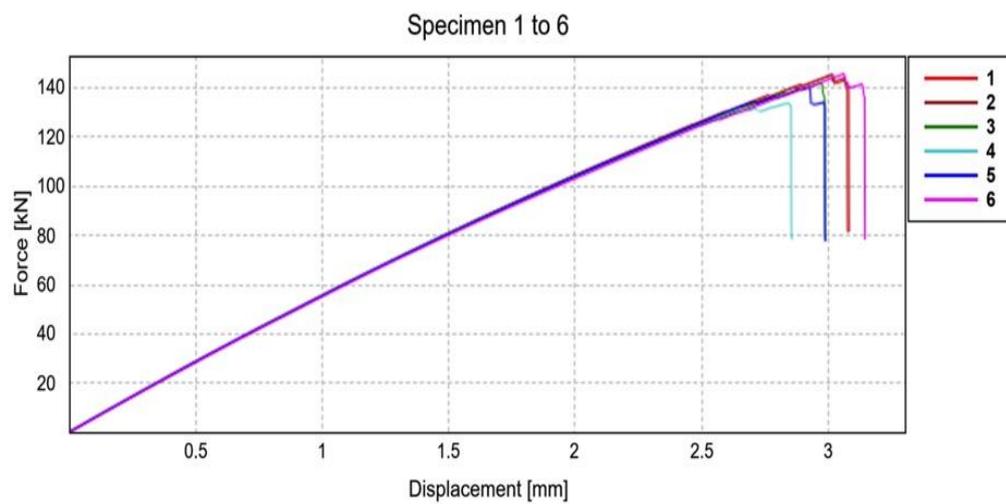


**Figure 4.6** Failure mechanism of the repaired with %30 chopped glass fiber and Loctite 9396 adhesive

Figure 4.8 illustrates the force-displacement of repaired with 30% chopped fiber and Loctite adhesive specimens. The maximum load and ultimate strength values measured according to the tensile test result are also shown in Table 4.4. A noteworthy result is that the maximum load and ultimate strength are on the specimen 6.



**Figure 4.7** Failure behavior of the specimen 6 repaired with 30% chopped glass fiber and Loctite 9396 adhesive



**Figure 4.8** Force-displacement of the repaired with 30% chopped fiber + Loctite 9396 adhesive

**Table 4.4** Test results of the repaired with 30% chopped fiber + Loctite 9396 adhesive

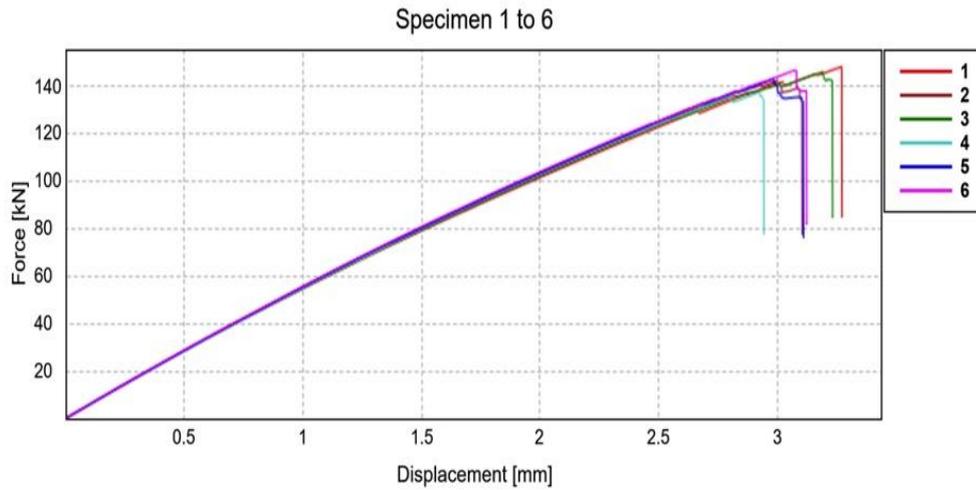
Specimen Number	Width	Thickness	Maximum Load	Ultimate Strength
	mm	mm	kN	MPa
1	152.16	3.8	145.24	251.191
2	152.07	3.835	145.398	249.316
3	152.02	3.8	141.970	245.761
4	151.99	3.825	133.798	230.146
5	152.02	3.765	140.372	245.254
6	152.2	3.790	145.644	252.487
Maximum	152.2	3.835	145.644	252.487
Minimum	151.99	3.765	133.798	230.146
Mean	152.08	3.802	142.071	245.693
Standard deviation	0.09	0.03	4.59	8.14
Coefficient of variation	0.06	0.66	3.23	3.31
Standard error	0.03	0.012	1.88	3.33

Figure 4.9 illustrates the condition after the tensile test of the repaired with 50% chopped glass fiber and Loctite 9396 adhesive specimens. The test was conducted for the six specimens again and Figure 4.9 shows the failure behavior of these six specimens. No different failure behavior was observed for the specimens.



**Figure 4.9** Failure mechanism of the specimens repaired with 50% chopped glass fiber and Loctite 9396 adhesive

Figure 4.10 illustrates the force-displacement of the repaired with 50% chopped glass fiber and Loctite adhesive specimens. The maximum load and ultimate strength values measured according to the tensile test result are also shown in Table 4.5.



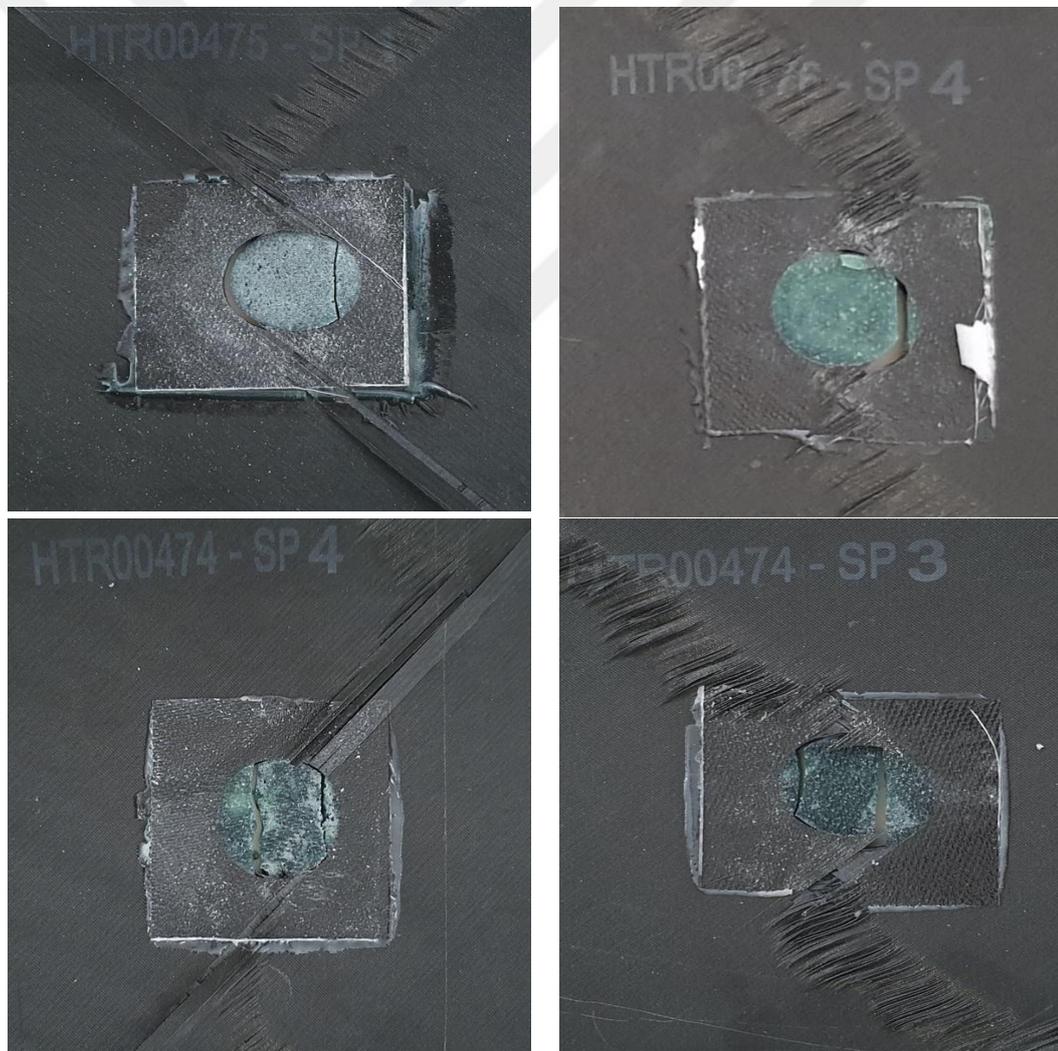
**Figure 4.10** Graph of force-displacement of the repaired with 50% chopped fiber + Loctite 9396 adhesive

**Table 4.5** Test results for the repaired with 50% chopped fiber + Loctite 9396 adhesive

Specimen Number	Width	Thickness	Maximum Load	Ultimate Strength
	mm	mm	kN	MPa
1	152.02	3.8	147.988	256.178
2	152.01	3.84	142.518	244.156
3	152.03	3.81	146.013	252.079
4	152.07	3.82	137.381	236.493
5	152.05	3.81	142.954	246.765
6	152.01	3.805	146.836	253.866
Maximum	152.07	3.84	147.988	256.178
Minimum	152.01	3.8	137.381	236.493
Mean	152.03	3.814	143.948	248.256
Standard deviation	0.02	0.01	3.88	7.3
Coefficient of variation	0.02	0.37	2.69	2.94
Standard error	0.008	0.004	1.59	2.99

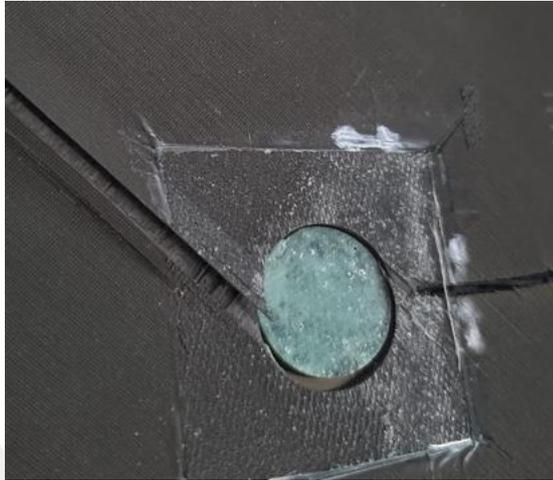
According to the Roylance [37], coefficient of variation (C.V.) is in the range of 1-10% for tensile strength. As seen in Table 4.2, Table 4.3, Table 4.4 and Table 4.5 C.V. values are 1-10%. In this thesis, three different repaired scenarios were applied, and tensile tests were repeated for four different groups of six specimens. It was confirmed that number of test repeat was sufficient in the study [37].

In this thesis, holes made on 18 specimens were repaired with the plug repair method. Based on the experimental observation, some specimens had brittle defects. (Figure 4.11)



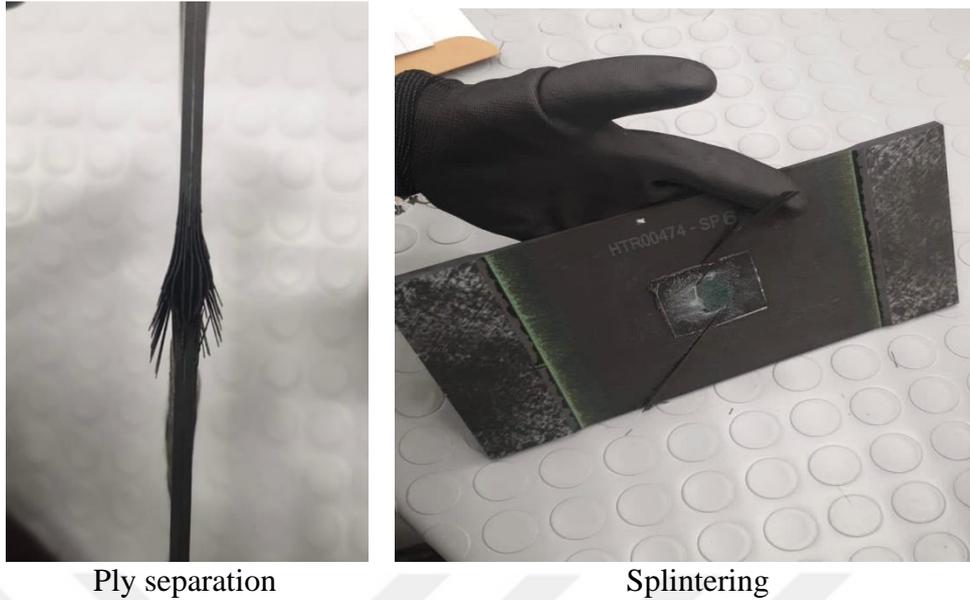
**Figure 4.11** Brittle failures of the repaired specimens

It was observed that some of the filled hole regions remained almost the same as the original state. Figure 4.12, for example illustrates the situation of a specimen repaired with the 30% chopped glass fiber and Loctite 9396 adhesive.



**Figure 4.12** The specimen on which no brittle failure was observed

In composite laminates, different types of failures can be observed. Strength and stiffness of the composite may sometimes decrease due to this failure behaviors [39]. Similarly, different failure types were observed after the tensile test. Ply separation and splintering, which are illustrated in Figure 4.13, were observed in the specimens.

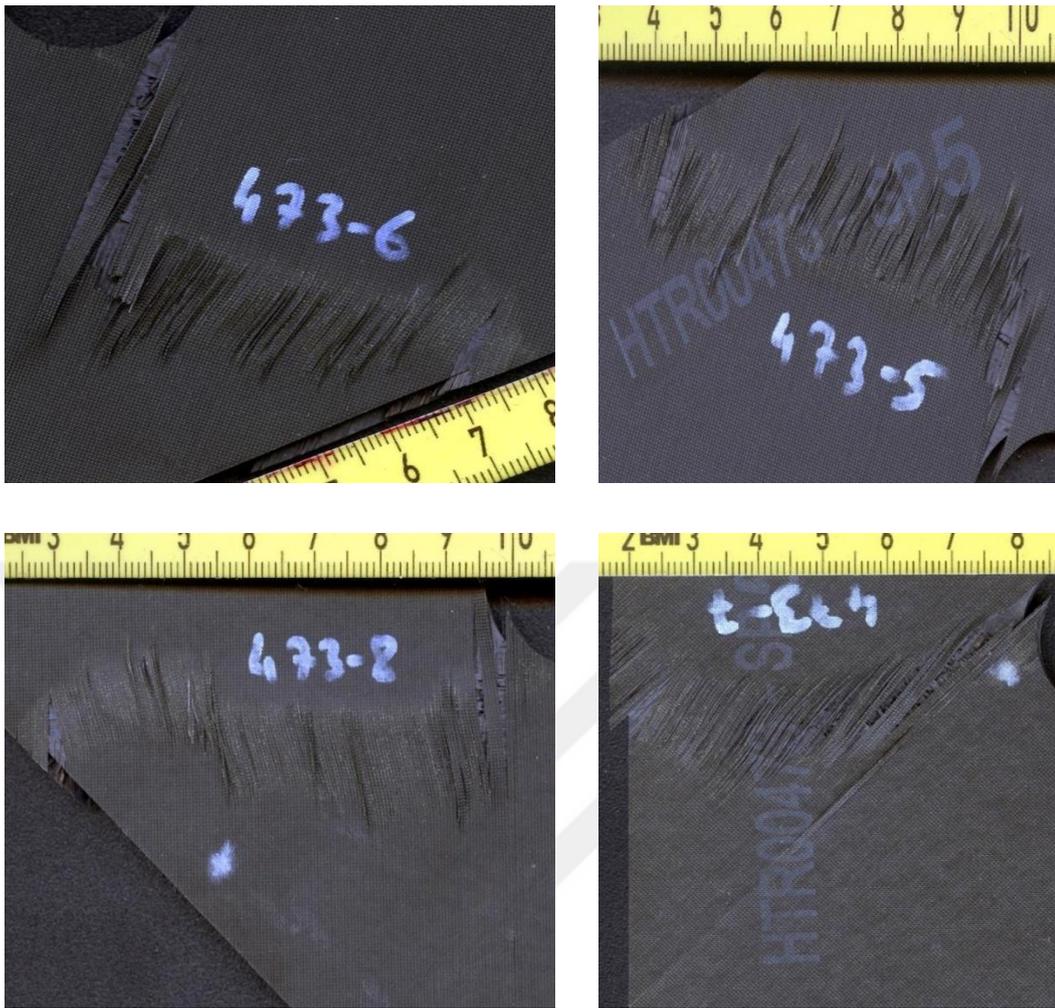


Ply separation

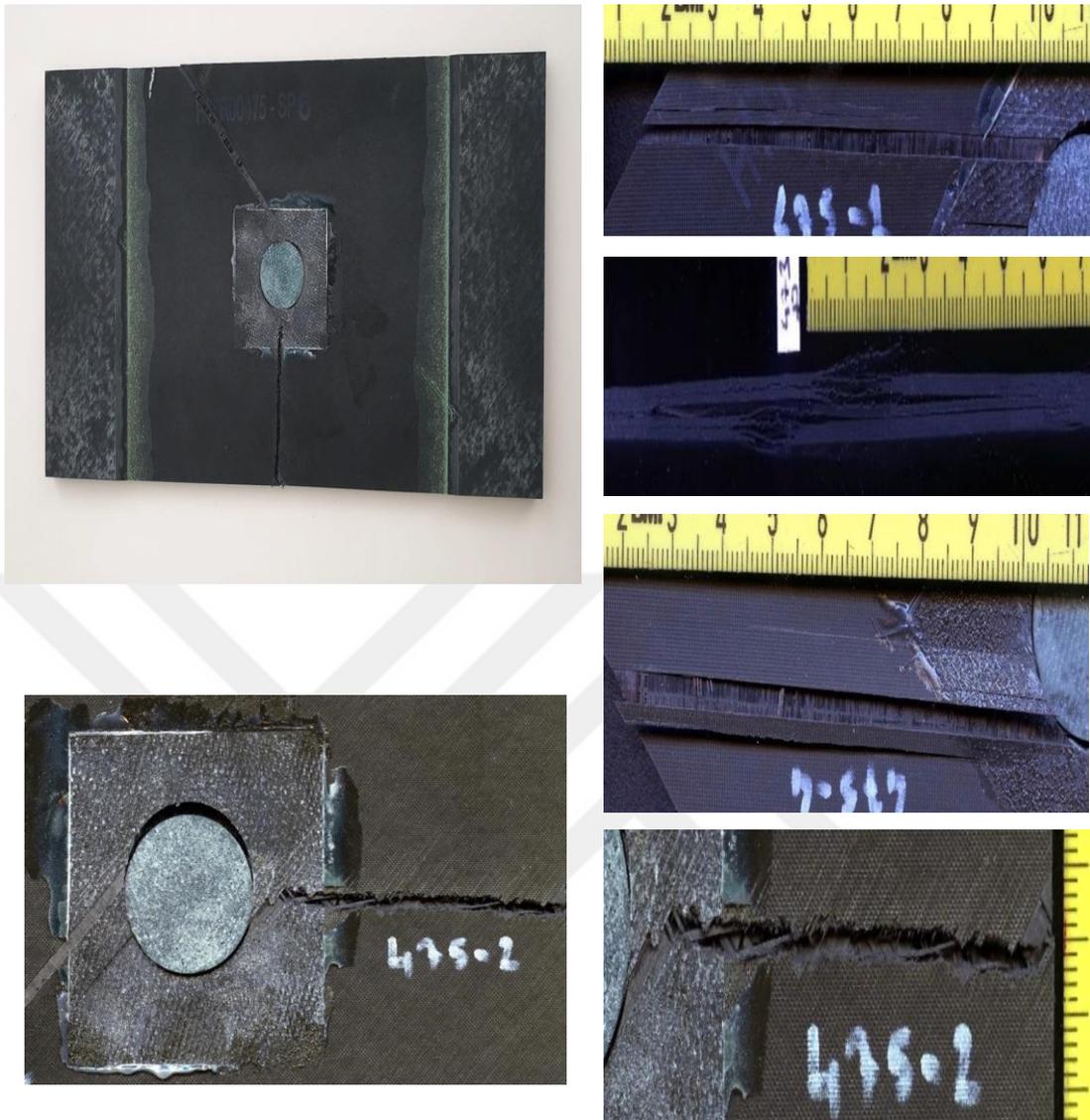
Splintering

**Figure 4.13** Failure mechanism of specimens after tensile test

The failure behavior of the second unrepaired specimen and failure behavior of the sixth specimen repaired with 30% chopped glass fiber and Loctite 9396 adhesive were different from the others. These specimens were scanned with Epson Perfection V700 Photo scan device, and different failure behaviors were examined. The scan results are shown in the Figure 4.14 and 4.15.



**Figure 4.14** Scan photos of the second unrepaired specimen



**Figure 4.15** Scan photos of the sixth specimen repaired with the 30% chopped glass fiber and Loctite 9396 adhesive

This device made it possible to detect, potential errors in the stacking sequence of the specimens in the lay-up process, and to determine foreign substances in specimens. According to results there was no errors in the sequencing of the specimens. It can be attributed to the fact that the specimens were produced from a whole single panel. Similarly, no foreign objects were observed. Such failures in specimens may be due to the local variation of the fiber to resin ratio, which was not experienced in the present study.

## 4.2 Discussion

Caminero et al. [11] examined scarf and external patch repairs and measured strength of undamaged and repaired specimens for each type of patch repair. They found the ultimate strength of undamaged specimen 996 MPa while this value was 704 MPa for double patch repaired specimens. Strength recovery was achieved by 70% with the double patch repair.

In this research, the ultimate strength of the unrepaired specimen was 235,212 MPa while this value was 248,256 MPa for the 50% chopped glass fiber and Loctite 9396 adhesive repaired specimens. Strength recovery was achieved by 94% with 50% chopped glass fiber and Loctite 9396 adhesive plug repair.

In the study of Soutis et al. [13], a 10 mm diameter hole was drilled, and a 35 mm diameter composite patch was used for repairing this hole. The non-destructive inspection manifested that the damages had occurred near the hole. The damages were also observed near the hole in the study of Caminero et al. [11]. Whereas they were seen at the edges of the holes in the study of Cheng et al. [16].

In this thesis, the MUPE results showed that the damages had not occurred near the hole in the plug repair technology. After the lay-up and autoclave processes, the specimens were subjected to the ultrasonic inspection, and it was seen that there was no porosity. Since plug repair is not a structural repair, an inspection is not needed after the repair is made, but in this study MUPE was applied to the repaired specimens, and the results were as expected.

In another study, the relationship between patch thickness and ultimate failure load was investigated. It was observed that the ultimate failure load increased as the patch thickness and bonding area increased [15]. It was concluded that the increase in the number of patches causes an increase in the weight of the structures [13].

Repairs made with the plug repair do not increase as in patch repair, especially as in double patch repair. Surface preparation must be good in the patch repair to increase repair efficiency. The performance of bonded patch repair depends on both the surface preparation and the material and geometry parameters. In this study, the upper surface

of the specimens is cleaned with water for the MUPE to be applied after the plug repair. In this inspection, only water is used as the surface of the specimens is not very rough.

Bouiadjra et al. [16] investigated scarf and external bonded patch repair interventions. It was reported that scarf repair is a more efficient repair method than external patch repair. Scarf repair is preferable for this reason, but it is a special repair method requiring special equipment Michelis [20] compared the scarf, patch, and plug repairs, concluding that, although the scarf repair is permanent, it is a type of repair that is nontrivial. The researchers stated that patch and the plug repair are used for minor damages, and the plug repair is easier than patch repair

The plug repair is a practical method to use for minor damage [22]. Repairs made with the plug repair do not increase as in patch repair. For these reasons, the plug repair is utilized in this thesis. To use this procedure effectively, two key points should be considered. First, the plug repair procedure is highly dependent on workmanship experience. It must be applied properly to get optimum results. Before being applied on real aircraft parts, it should be applied on a test part to determine the amount of adhesive/chopped fiber. In this study, the thickness of the laminate was selected as 3.68 mm. Secondly, during the application, it became clear that the thickness of the laminate was a key factor for the success of the repair.

The tensile test results pointed to differences between the failure behaviors of some specimens. This may be due to an error in the stacking sequence or a foreign object during the laying. However, in the case, the specimens were produced from a single panel to prevent sequencing errors. Epson Perfection V700 Photo scan device used to check the existence of foreign objects. The different failure behaviors can also be explained by the local differences or fiber to resin ratio.

# CHAPTER 5

## CONCLUSION

Fiber-reinforced polymer (FRP) composites have been increasingly used in various industries. They have low density and high strength, which are especially preferred in the aerospace industry. In recent years, CFRP composites have drawn great attention in the primary structures of aircraft due to their advantages. With the expansion in the usage of composite materials, a major problem has emerged damage. In composite materials, damages occur during service or due to external reasons. Since composites inherently have non-homogenous and anisotropic material behavior, finding the location of the damage can be difficult. That is, these damages may not be visible to the naked eye. Therefore, relevant techniques have been applied for damage assessment. If the damage is repairable, repair technology should be used instead of replacing of the damaged composite component. Because composite materials are costly. In other words, repair of composite materials has proved more cost-effective than the replacement.

The increasing usage of composites in aircraft has led to growing need for new repair technologies. This repair technology can be provided by the plug repair as it is a practical method. In this study, in house manufactured composite laminate plates were drilled to create holes to simulate damage. Then they were repaired by the plug repairing process. The holes were divided into three groups of six specimens and were filled with different materials, namely Loctite 9396 adhesive and various amounts of glass chopped fibers. After the repair work, the MUPE was used as a NDT method for quality control of the repaired specimens. MUPE results indicate that the repaired specimens were in high quality and suitable for testing. Then, tensile tests were carried out for 24 specimens by using Instron 5985 tensile test machine which has the capacity of 250 kN. The results of this study demonstrate that an improvement by up to 5% was achieved with the repair using the 50% chopped glass fiber and Loctite 9396 adhesive.

# CHAPTER 6

## FUTURE WORKS

The use of the composite materials in various industries, especially in the aerospace industry, is increasing day by day. Indeed, it seems that composite materials will be preferred in the future due to the advantages they offer and replace metallic materials in aircraft.

Due to the unique, non-homogeneous and anisotropic behavior of composites, a damage is not easy to detect. Thus, development of repair technology plays a key role in the applications of the primary structures of aircraft. With the increasing importance of composite materials, repair methods of these materials should also be improved.

It is recommended that further research do the following to increase the generalizability of this study's results pertaining to composite repair.

- In this study, composite laminate plates were accepted as damaged based on the criterion of 25.4 mm hole drilling. The diameters of the holes can be changed.
- A different adhesive can be applied and the effects of different adhesives on the repair should be investigated.
- In this study, glass chopped fiber was used and the contribution of chopped fiber in different ratios was investigated. To better understand the effect of chopped fiber on repair, different kinds of chopped fiber such as carbon chopped fiber may be tested.
- In the lay-up process, the carbon prepregs were mostly laid at 45 degrees. The effect of different patterns of laying of carbon prepregs and different angles should be investigated.
- Real structural repair can be made in order to verify the assumptions.
- In this study, the thickness of the laminate was selected as 3.68 mm. During the application, it was observed that the thickness of the laminate was also a key factor for the success of the repair. Future studies can address the maximum and minimum thickness limits for a successful repair on one of the three scenarios.

## REFERENCES

- [1] Ghobadi, A. Common type of damages in composites and their inspections. *World journal of mechanics*, (02), 24-33. 2017.
- [2] Soutis, C. Fibre reinforced composites in aircraft construction. *Progress in aerospace sciences*, (41), 143-151. 2005.
- [3] Hoshi, H., Nakano, K., & Iwahori, Y. Study on repair of CFRP laminates for aircraft structures. 16th International Conference on Composite Materials, 3-8, 2007.
- [4] Worries About New Composite Made Airplane. (n.d.). Retrieved from <https://www.1001crash.com/index-page-composite-lg-2.html>.
- [5] Ahmed, K. Noor., Samuel L. Venneri., Donald B. Paul., & Mark. A. Hopkins. Structures technology for future aerospace systems. *Computers & Structures*, (74), 507-519. 2000.
- [6] Katnam, K. B., Da Silva, L. F. M., & Young, T. M. Bonded repair of composite aircraft structures: A review of scientific challenges and opportunities. *Progress in Aerospace Sciences*, (61), 26-42. 2013.
- [7] Pora, J. Composite Materials in the Airbus A380 - From History to Future. In Beijing: Proceedings 13th International Conference on Composite Materials, 2001.
- [8] Lee, J. J. Can we accelerate the improvement of energy efficiency in aircraft systems? *Energy conversion and management*, (51), 189-196. 2010.
- [9] Yücel, S. B. Kompozit Onarım Bölgesinin Kırılma Davranışları Üzerindeki Etkileri, master, Dokuz Eylül University. 2015.
- [10] Gholizadeh, S. A review of non-destructive testing methods of composite materials. *Procedia Structural Integrity*, (01) 50-57. 2016.

- [11] Caminero, M. A., Pavlopoulou, S., Lopez-Pedrosa, M., Nicolaisson, B. G., Pinna, C., & Soutis, C. Analysis of adhesively bonded repairs in composites: Damage detection and prognosis. *Composite Structures*, (95), 500-517. 2013.
- [12] Baker, A. Structural health monitoring of a bonded composite patch repair on a fatigue-cracked F-111C wing. Defense science and technology organization Victoria (Australia) air vehicles div, 2008.
- [13] Soutis, C., Duan, D. M., & Goutas, P. Compressive behaviour of CFRP laminates repaired with adhesively bonded external patches. *Composite structures*, (45), 289-301. 1999.
- [14] Cheng, P., Gong, X. J., Hearn, D., & Aivazzadeh, S. Tensile behaviour of patch-repaired CFRP laminates. *Composite structures*, (93), 582-589. 2011.
- [15] Çetinkaya, Ş., Pihtılı, H., Yıldırım, H. Investigation of external bonded patch repair of laminated composite materials experimentally. *European Journal of Technique*, (8), 134-140. 2018.
- [16] Bouiadjra, B. B., Belhouari, M., Serier, B. Computation of the stress intensity factors for repaired cracks with bonded composite patch in mode I and mixed mode. *Composite Structures*, (56), 401-406. 2002.
- [17] Benyahia, F., Albedah, A., Bouiadjra, B. B. Analysis of the adhesive damage for different patch shapes in bonded composite repair of aircraft structures. *Materials & Design*, (54), 18-24. 2014.
- [18] Bouiadjra, B. B., Bouanani, M. F., Albedah, A., Benyahia, F., & Es-Saheb, M. Comparison between rectangular and trapezoidal bonded composite repairs in aircraft structures: a numerical analysis. *Materials & Design*, (32), 3161-3166. 2011.
- [19] Kashfuddoja, M., Ramji, M. Design of optimum patch shape and size for bonded repair on damaged Carbon fibre reinforced polymer panels. *Materials & Design*, (54), 174-183. 2014.

- [20] Michelis, A. Plug Repairs of Marine Glass Fiber / Vinyl Ester Laminates Subjected to In-plane Shear Stress or In-plane Bending Moment, masters Massachusetts Institute of Technology, 2009.
- [21] Vlattas, C., Soutis, C. Composite repair: Compressive behaviour of CFRP plates with reinforced holes. European Conference on Composites, (02), 87-92. 1996.
- [22] Heslehurst, R. Challenges in the repair of composite structures. Sampe journal repair paper, 2017.
- [23] ASTM D3039 / D3039M-17. (n.d). Retrieved from <https://www.astm.org/Standards/D3039>.
- [24] ASTM D695-15. (n.d). Retrieved from <https://www.astm.org/Standards/D695.htm>.
- [25] Rikard Benton Heslehurst, “*Defects and damage in composite materials and structures*”, CRC press, USA 2014.
- [26] Mangalgi, P. D. Composite materials for aerospace applications. Bulletin of Materials Science, (22), 657-664. 1999.
- [27] Alan A. Baker, “*Composite materials for aircraft structure*” 2004.
- [28] Quilter, A., Composites in aerospace applications. IHS White Paper, (444), 264. 2001.
- [29] David, K., Nondestructive inspection of composite structures: methods and practice. In 17th world conference on nondestructive testing, 2008.
- [30] Piancastelli, L., Frizziero, L., Zanuccoli, G., Daidzic, N. E., & Rocchi, I., A comparison between CFRP and 2195-FSW for aircraft structural designs. International Journal of Heat and Technology, (31), 17-24. 2013.
- [31] D’orazio, T., Leo, M., Distanto, A., & Guaragnella, C., Automatic ultrasonic inspection for internal defect detection in composite materials. Composite Structures, (41), 145-154. 2008.

- [32] Scott IG, Scala CM., A review of non-destructive testing of composite materials. NDT International, (15), 75-86. 1982.
- [33] Jawaid M., Thariq M., “*Sustainable composites for aerospace applications*”, Woodhead publishing, India, ISBN: 978-0-08-102131-6, 2018.
- [34] Budhe, S., Banea, M. D., De Barros, Bonded repair of composite structures in aerospace application: a review on environmental issues. Applied Adhesion Science, (06), 1-27. 2018.
- [35] Rubin, F., & Niscito, A., & Tucci, F., & Carlone P. , “*Marine application of fiber reinforced composites:a review*, Marine Science and Engineering, 2020.
- [36] Adhesive Technologies (n.d.). Retrieved from [https://www.henkel-adhesives.com/vn/en/product/industrial-adhesives/loctite\\_ea\\_9396\\_aero.html](https://www.henkel-adhesives.com/vn/en/product/industrial-adhesives/loctite_ea_9396_aero.html).
- [37] David, R., Statistics of Fracture. Department of Materials Science and Engineering Massachusetts Institute of Technology, 2001.
- [38] FSH Full Screen Height (n.d.). Retrieved from <https://www.bindt.org/What-is-NDT/Index-of-acronyms/F/FSH/>
- [39] Liu X., & Yao X., & Cai J., & Zeng J., & Chiu W., Failure mode analysis of carbon fiber composite laminates by acoustic emission signals. Advances in Materials Science and Engineering Volume 2021, 2021.

# CURRICULUM VITAE

## PERSONAL INFORMATION

**Name Surname** : Yeliz CAN  
**Date of Birth** :  
**E-mail** :

## EDUCATION

**High School** : Hacı Ömer Tarman Anatolian High School Ankara/Turkey  
**Bachelor** : Atılım University Mechanical Engineering Ankara/Turkey  
**Master Degree** : Ankara Yıldırım Beyazıt University Ankara/Turkey

## WORK EXPERIENCE

Internship at ASELSAN A.Ş. 07/2016 – 08/2016  
MCM Savunma - Purchasing Engineer 09/2018 – 12/2019  
Mechanical Engineer: Ministry of National Defense 12/2019 -

## PUBLICATION

**Yeliz Can**, Faruk Oguz, Fahrettin Ozturk, Analysis of Plug Repair Technology in Composite Structures, 4. International Scientific and Innovation Congress, 21-22 February 2022, Ankara, Turkey.