

**Mechanical response of carbon fiber/polyphenylene sulfide commingled fiber &
polymethacrylimide foam core sandwich composites**

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Abstract

Composite materials are desired for aerospace and other high-tech applications due to their impressive physical properties and resilience. Carbon fiber is used extensively in the aerospace industry in the form of prepreg materials for structural applications such as wing bodies, storage compartments, seats, and other areas. In order to aid in the light-weighting of aircraft, sandwich structures can be utilized. These sandwich structures often have a stiff face material surrounding a dense and light weight core material. The sandwich construction used in aerospace typically uses carbon or glass composite facing (skin) and foam, Nomex, or aluminum honeycomb core. The aim of this research is to combine two high performance constituents to create a sandwich composite capable of meeting performance criteria for aerospace conditions – high strength and high impact toughness under elevated temperatures. To create the sandwich structures, a face material was chosen to be a carbon fiber/polyphenylene sulfide (CF/PPS) commingled fiber composite panel with a closed cell polymethacrylimide (PMI) foam as the core material. The design of the sandwich structure of a ½ inch foam core with a 1.0-1.5 mm thick face of CF/PPS was chosen for this study. The CF/PPS panels were created through a filament winding process of the commingled fibers followed by a hot compression molding process to consolidate the fibers into a composite panel. The panels were then bonded to the PMI foam through the usage of an epoxy resin system from Huntsman through hot compression. Non-destructive testing was performed on a representative panel prior to testing to examine the bonding between the face and core materials. The sandwich was tested for Core Shear at room temperature (RT) and Static Indentation testing was performed at RT and an elevated temperature to examine the impact energy response. It was found that the sandwich absorbs twice the energy (11.3 vs 5.6 J) at room temperature when compared to testing at 100 °C.

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List of Abbreviations

CF – carbon fiber

GF – glass fiber

PPS – polyphenylene sulfide

RT – room temperature

CFRP – carbon fiber reinforced plastic

PMI – polymethacrylimide

ILSS – interlaminar shear strength

PP – polypropylene

HDPE – high density polyethylene

GFRP – glass fiber reinforced plastic

ASTM – American society for testing and materials

PET – polyethylene

TGA – thermogravimetric analysis

Wt.% – weight percent

PAUT – phased array ultrasound

FTIR – Fourier-transform infrared spectroscopy

SEM – scanning electron microscopy

HG – heat gun

Chapter 1 – Introduction

Composite materials are becoming more and more prevalent in industries where high physical and mechanical performance and lightweight, i.e. high strength to weight ratios are needed. They are being used to reduce the weight in vehicles to help improve fuel efficiency, they are used in aerospace applications due to their corrosion and heat resistance, they are used in the marine industry for the stiffness and light weight. The applications of composite materials are endless and new innovative systems are being developed every year. CF has become the staple of the aerospace industry due to its high performance, low density and thermal properties. Up to 50% of the structural weight of an aircraft today is comprised of Carbon Fiber Reinforced Plastics (CFRP) or composite materials [1]. CF is inherently light, stiff but has low impact toughness. In order to improve the impact properties of CF a thermoplastic polymer is often combined with the CF to create a hybrid system. CFRP can be easily processed, has improved impact toughness, and high temperature resistance of upwards of 200-400 °C depending on the matrix. In typical aerospace applications, CF would be impregnated with an epoxy matrix in the form of a prepreg. The prepreg is in the form of a unidirectional tape or fabric mat that can be laid up by hand lay-up or by a robot to the desired form/shape and then placed into an autoclave to enable the layers of prepreg, i.e. resin matrix and the fibers to compact, consolidate and cure to produce finished product. The constituents within the CFRP, namely the fiber reinforcement provides load beaming and the matrix holds the fibers together [2]. For composite materials in aerospace applications, tensile, compression, impact, and interlaminar shear are examples of the types of properties of interest to designers.

Sandwich composite constructions are another step forward regarding composites for aerospace applications. The CFRP is combined with a secondary material to improve the physical properties without drastically increasing the weight. For instance, a part that is a solid (monolithic) composite panel would be heavier than a sandwich structure, with a composite shell and a foam core assuming they are the same size. With a sandwich structure, the desired stiffness and performance can be attained while limiting the weight of the part with the use a honeycomb or foam core. When flexural load beaming, impact

and compression resistance are desired, a sandwich structure could be the ideal route. When looking at sandwich composites, the construction can vary depending on the desired strength and stiffness ratios regarding the weight of the sandwich. The stiffness of a sandwich composite can be increased significantly by simply increasing the core thickness. Figure 1 gives an example of how a designer can increase the stiffness of the composite without significant difference in the weight of the composite [3]

The main advantages of sandwich composites are the properties that can be tailored to meet environmental demands, the variety of materials that can be used as a core or face, high rigidity, low density, superior vibration resistance and damage tolerance [4]. Examining different sandwich structures in the industry has shown that aluminum cores, steel skins, wood cores, foam cores, composite skins, and honeycomb variants are prevalent. The face material is often made of steel, aluminum, composite materials with a core material of foam, honeycomb, or wood [4]. The sandwich construction of choice for this research is focusing on a composite constituent of CFRP face and a polymer foam core. The rationale behind choosing these materials was the ability to easily bond the two as well as the ease of manufacturing. A honeycomb structure can be quite complex and difficult to process, whereas a high-performance closed cell foam is readily processed.

	Solid Material	Core Thickness t	Core Thickness $3t$
			
Stiffness	1.0	7.0	37.0
Flexural Strength	1.0	3.5	9.2
Weight	1.0	1.03	1.06

Figure 1 - Sandwich construction effect on stiffness - Hexcel

It was determined that a commingled fiber-thermoplastic resin system was desirable to investigate for the usage in sandwich composite construction. There is not much data regarding commingled fibers for aerospace applications and the chosen fiber system is high performing, thermoplastic resin based, and high temperature resistant, which meets the needs of aerospace. Combining the high performing commingled fibers with a high temperature foam should yield a sandwich composite that can withstand the necessary environmental conditions for aerospace. Commingled fibers are a blend of reinforcement and matrix filaments within the overall fiber tow [5]. With this system, the thermoplastic polymer acts as the matrix for the CF. The CF is the reinforcing constituent. Whereas in a traditional prepreg material, the CF is suspended in an epoxy matrix. The commingled fibers chosen were a CF/PPS 50/50 % by volume blend. During processing, the commingled fibers will be processed and manufactured into a composite plate to act as the face material for the sandwich. The core material chosen was a high temperature polymethacrylimide (PMI) foam.

The construction of the sandwich composite will take place in stages. The commingled fibers will first be consolidated into a plate through a filament winding and a hot compression process. This composite plate will then be bonded to the foam with an epoxy adhesive. The sandwich panel will then be evaluated through non-destructive testing to examine the bonding between the two materials prior to destructive testing. Destructive testing for this research will involve both flexural and impact testing of the sandwich panel. The chosen tests are based on the sandwich composite – flexural and impact are the most relevant tests for the application. Samples will be acquired from multiple sandwich panels. To ensure the consistency of production, the settings for all processes will remain the same and the non-destructive testing will serve as validation of the sandwich composite manufacturing. The objective of the sandwich construction is to show that the sandwich can perform at the desired environmental conditions. The conditions to be tested will be room temperature as well as an elevated temperature to simulate the expected condition for an aerospace application. The impact properties between the room temperature and elevated temperature testing will be compared to determine how much of an effect temperature will have on performance. Analysis of the failure mechanism of each testing criteria will be performed using optical microscopy.

The limitations of this work involve both manufacturing time and equipment access. This work relies on coordination between several entities to get access to equipment, materials, and analysis software. The size of the panels made is limited by the manufacturing equipment available. The number of samples to be obtained from each sandwich composite panel is limited by this. It is estimated that 2 samples for flexural or 2 samples for impact testing can be obtained from 1 sandwich panel. From this, it is determined that a total of at least 3 panels will be needed for flexural testing (6 samples) and 2 panels for impact testing (4 samples).

The thesis report will be presented in the following manner: literature review of all facets of the project (commingled fibers, sandwich composites, aerospace applications, filament winding, mechanical property characterization, etc.), materials/methodology, results/discussion, and conclusions/future work.

Objectives of the thesis:

- Characterize CF/PPS as a face sheet for sandwich construction
- Evaluate the performance of the sandwich construction
- Quantify the relationship between impact properties and temperature

Chapter 2 – Literature Review

2.1 - Carbon Fiber Reinforced Plastics

Carbon fiber reinforced plastics (CFRP) are a polymer composite system where the CF acts as the reinforcing material to a matrix. The matrix in this system can be an epoxy or another thermoplastic polymer for instance. One of the main advantages of CFRP are the higher performance results when compared to metallic alloys [2]. CFRPs are becoming more popular in many industries such as automotive, aerospace, sporting goods, and medical devices. By their nature, CFRP have good chemical resistance, high mechanical properties at low densities, and strength characteristics that can be altered for a specific load situation [6]. Forintos also describes carbon fiber in multifunctional parts as being capable of de-icing and protecting an aircraft wing from a thunder strike and even the capability of storing energy. CF reinforcements can also be used as sensors for temperature/humidity as well as embedded sensors for health monitoring and data collection [6]. When evaluating the use of CF, the electrical properties play a large role in its selection for reinforcement in composite structures [6]. In order to understand the electrical properties of CF, Forintos states that the hexagonal carbon rings require high energy in order to separate, which provides macro level strength to carbon fiber while the free electrons make it a good electrical conductor. To produce CF, polyacrylonitrile (PAN) is widely used and accounts for 90% of world carbon fiber [6]. Forintos provides a good summary of the production of carbon fiber from the spinning of PAN with comonomers into a stretching process with slow heating afterward. The fibers are then carbonized by heating to 1500 °C in an inert atmosphere and then treated at a higher 1500-3000 °C temperature.

In the aerospace industry, CFRP are often used in the form of a prepreg material. Prepregs are fabrics where the carbon fibers are impregnated with epoxy resin. These fabrics are used in hand lay-up processes where they are laid out on a mold in differing orientations based on the desired performance. The fabric lay-up orientation affects the overall properties of the CFRP laminate.

Zhou examined five CFRP laminates with different layup patterns as seen in Figure 2 [7]. The layup pattern column designates the orientation of each prepreg sheet in the laminate. Each sheet is referred to as a ply, with all the plies together becoming the laminate. Zhou et al. concluded that the impact responses of the laminates significantly depended on the proportion of ply orientation as well as the stacking sequence of the laminates. They found that the more continuous plies in the same orientation lead to weaker mutual support between the adjacent plies, which resulted in more severe damage and more energy dissipation during testing. More severe interlaminar damage occurs due to the overall interlaminar performance of the laminates being weak [7]. It can be seen in Figure 3, Zhou observed fiber-matrix debonding and fiber breakage from the tensile testing. As the load is applied to the composite, the fibers share the load and with continued loading, the fibers begin to break, and the load is transferred to the matrix until the specimen fails. Another failure that CFRP can experience is delamination between the layers in addition to the above described fiber matrix cracking.

2.2 - Sandwich Composites

Sandwich composites are used when strength and stiffness to weight ratio needs to be high. The core material allows for a lighter weight structure, whereas the outer face provides stiffness and strength. The general schematic for a sandwich panel is shown in Figure 4 [8]. Figure 4 details the construction of a honeycomb sandwich. The layout consists of two outer face sheet materials, the adhesive layers and the honeycomb core. The adhesive layer is used to bond the face sheet to the core material. The core material can be made up of foam, honeycomb, or wood [9]. Depending on the application, the appropriate core material would be chosen. For structural applications, the face sheet carries the in-plane and bending loads while the core supports the face sheets by transferring the load and bearing the through-the-thickness shear load. A descriptive image for these phenomena can be seen in Figure 5 [3].

Laminate details.

Type	Layup pattern	The proportion of the plies with 0° fiber
T1	$[90_4/0_4]_{4T}$	50%
T2	$[45_2/0_2/-45_2/90_2]_{2S}$	25%
T3	$[45_2/0_3/-45_2/90]_{2S}$	37.5%
T4	$[-45_2/0_3/90/45_2]_{2S}$	37.5%
T5	$[90/0_3/45_2/-45_2]_{2S}$	37.5%

Figure 2 - Laminate details – Zhou et al.

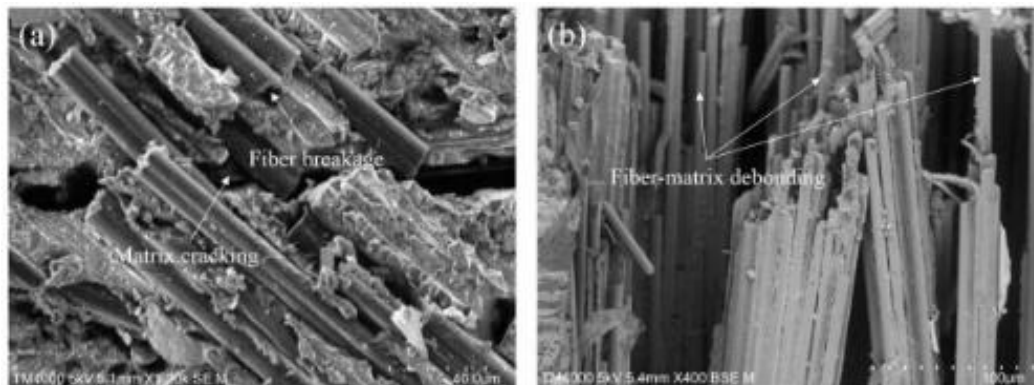


Figure 3 - SEM imaging of tensile fracture – Zhou et al.

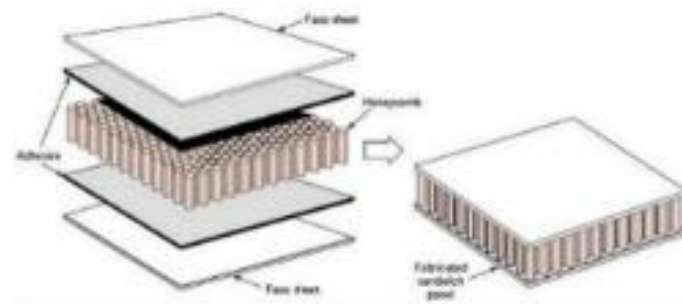


Figure 4 - Sandwich panel schematic – Batchu

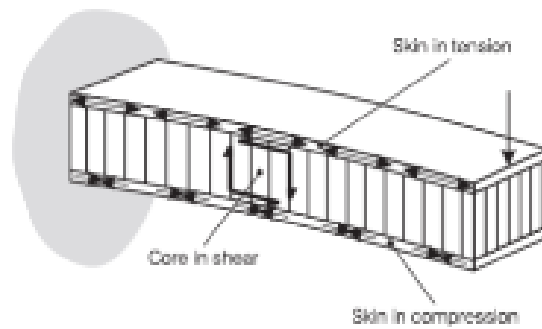


Figure 5 - Loading of sandwich panel- Hexcel

It is often of interest to predict the material properties of a face sheet material. When examining these properties, the rule of mixture can be used to predict the average modulus of the face in the fiber direction for a unidirectional composite [9]. The rule of mixture is expressed by Equation 1:

$$E_1 = E_f v_f + E_m v_m \quad (\text{Eq. 1})$$

where v_m and v_f are the volume fractions, E_f is the fiber modulus and E_m is the matrix modulus. It is also possible to determine the properties of the face material through experimental testing following ASTM D3039 for tensile testing, ASTM D790 for flexural, ASTM D5379 v-notch impact, and ASTM D2344 for short beam interlaminar shear strength. The tests should be performed in different directions in order to determine the in-plane shear modulus of the laminate [9]. PMI foams have perhaps the best properties of commercial foams, but the price is the highest [9]. PMI is brittle, but it has a high temperature resistance which makes it ideal for use with epoxy prepreps suitable for aerospace applications. PMI is comprised of a monomer mixture of methacrylonitrile, methacrylic acid, and the option of other monofunctional vinyl-unsaturated monomers, a propellant, and a cross-linking agent [10]. The PMI foam structure is a recurring structure of the elements show in Figure 6 [10]. Table 1 shows commonly used core materials for sandwich construction [9].

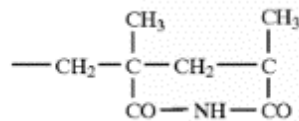


Figure 6 - Chemical structure of PMI - Werner Geyer et al.

Table 1 - Sandwich composite common core materials - Zenkert

Density kg/m^3	G_c MPa	$\hat{t}$ MPa	λ $W/m^\circ C$
<i>Balsa Wood</i>			
96	108	1.85	0.0509
130	134	2.49	0.0588
180	188	3.46	0.0710
<i>Polyurethane foam</i>			
30	3	0.2	0.025
40	4	0.25	0.025
<i>Polystyrene foam</i>			
30	8	0.25	0.035
60	20	0.6	0.035
<i>Polyvinyl chloride foam (linear)</i>			
80	18	1.1	0.034
<i>Polyvinyl chloride foam (cross-linked)</i>			
45	18	0.5	0.024
80	31	1.0	0.028
100	40	1.4	0.030
130	52	2.0	0.034
200	85	3.3	0.043
<i>Polymethacrylimide foam</i>			
52	19	0.8	-
75	29	1.3	-
110	50	2.4	-
205	150	5.0	-

Stefan et al. found that sandwich structures comprised of composite faces and core materials (foams, honeycombs, etc.) are believed to have value for aircraft structures because they fulfill mechanical and non-mechanical functions, such as thermal and vibration insulation [4]. The use of sandwich composites also results in weight reduction and the reduction of manufacturing and operating costs [4]. Stefan et al. found that the typical face and core materials are generally bonded together with an adhesive to help facilitate the load transfer between the two. An epoxy paste adhesive or a structural film adhesive may be used. Stefan et al. observed that fiber breakage was the primary failure mechanism in CFRP composites. This meant that the CFRP composites were suitable for application as skins of sandwich composites since there was no delamination failure observed [4].

He et al. covered the necessity of low velocity impact response testing of sandwich composite structures due to the common damage scenarios in industrial applications [11]. They claim that it is essential to study damage tolerance and evaluate the residual properties of the sandwich structure after impacts. In previous studies, He et al. observed that the residual compressive, tensile, and flexural properties of the sandwich after impact were adopted to determine if the structure could further carry out its function. Their study showed that the impact damage and residual flexural strength depend on the impact energy and the impact location, it was observed that the residual flexural strength decreased significantly when the impact energy was lower than 10 J and there is a larger reduction in the residual flexural strength and absorbed energy for the short span case compared to the long span [11]. The authors also ran numerical simulations to compare with experimental results, determining that the predicted results are consistent with the experimental. Davies et al. investigated the impact damage and compression-after-impact (CAI) strength of sandwich composites with carbon face sheets and aluminum honeycomb cores [12]. The results of their study showed that the sandwich panel with a thin core had a robust energy absorption while the sandwich panel with a thicker core was penetrated easily and had a lower CAI strength.

The flexural response of CFRP aluminum foam sandwich composites were investigated by Yan et al. They used a closed cell aluminum foam as the core material with a CF fabric with unidirectional fibers as the face sheet. Their choice to use aluminum and carbon fiber was due to the good energy absorption ability of aluminum and the high strength of carbon fiber to offset the weak mechanical properties of the aluminum [13]. Their investigation found that the peak load increased with the aluminum foam core density, which meant that the bending strength of the aluminum foam core sandwich increased with the foam core density. This was attributed to the mechanism that when aluminum foam is loaded under compression, the load is mostly experienced by the cell edge and cell wall of the foam. When the compressive stress exceeds the yield strength of a cell edge or wall, the edge will collapse or the wall will crack [13]. They realized that when the foam density was increased, the wall thickness and edge improved, which lead to an increase of the strength and stiffness of the foam. The failure modes of their aluminum foam sandwich composite can be seen in Figure 7. The conclusion of Yan et al. study was that carbon fiber is a good choice to reinforce aluminum foam core, but the adhesion between the core and carbon fiber sheets needs improvement. From this, it can be gathered that the bonding between the face and core materials is important in evaluating the mechanical properties.

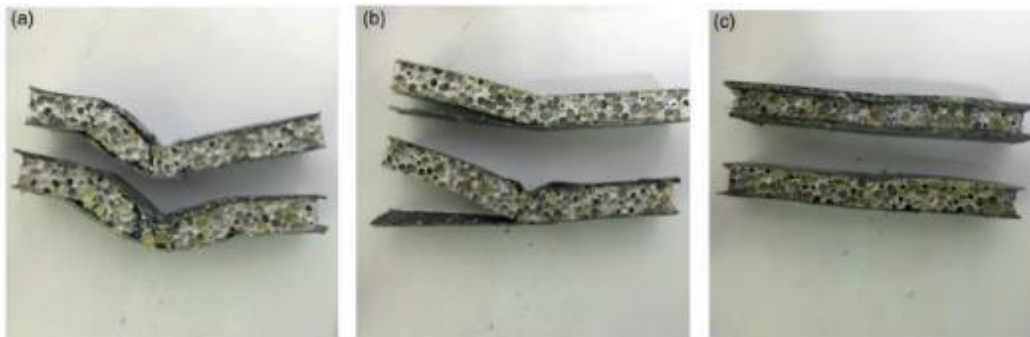


Figure 7 - Failure mode of the AFS specimens: (a) core shear; (b) interface de-bonding I; (c) interface de-bonding II – Yan et al.

A poor bond will result in a weaker sandwich composite panel. Steps to improve the bonding between the face and core materials need to be taken to ensure that the performance of the sandwich is maximized. In order to improve the bonding between CF/PPS and the PMI foam core, abrasion of the CF/PPS surface is required [14]. Light abrasion of the CF/PPS surface will provide a better key to the film adhesive as described by Plexus through a brushing process where a coarse, stiff brush is used to rub the surface.

For a sandwich composite structure to have sufficient strength, a good bond between the face and core needs to be achieved. Grunewald explored fusion bonding as a method of joining, which is based on the intermolecular diffusion of the polymers [15]. Another approach is adhesive joining where a joint is created by mechanical interlocking of the surface materials as described by Grunewald. The focus of their research was to examine the fusion bonding process of CF/Polyether ether ketone (PEEK) skins to a polyethylenimine (PEI) foam core. The bond quality would be assessed by the predicted tensile bond strength and compared to the tensile properties of the core. Grunewald describes the fusion bonding process as the application of heat and pressure to the interface between the two parts to be joined. In order to do this, the temperatures need to be above the glass transition temperatures of the amorphous polymers and above the melting temperature for the semi-crystalline polymers. By meeting these temperature guidelines, they noted that the viscosity was lowered, and the polymer molecules were allowed higher mobility. Once the temperature is reached, the parts are joined together and placed under pressure. The five steps of fusion bonding are surface arrangement, surface approach, wetting, diffusion, and randomization [16]. In the literature, these steps are combined into two mechanisms: 'intimate contact' (steps 1-3) and 'healing' (steps 4-5). Intimate contact is the amount of surface area that is physically in contact between the interfaces of both parts and healing refers to the interdiffusion of polymer molecules across the interface as described by Butler et al [17]. Butler found that by applying temperature and pressure, the materials soften, and the asperities can be deformed which increases the surface area. When examining intimate contact, the Mantell-Springer model assumes rectangular elements represent irregular surface which are then deformed by pressure [18].

The model also accounts for material properties and conditions that vary with time. Mantell describes the degree of intimate contact D_{ic} as expressed by Equation 2:

$$D_{ic} = \frac{b(t)}{w_0 + b_0} \quad (\text{Eq. 2})$$

where b_0 is the width of the initial rectangular elements, w_0 is the initial width between the rectangular elements and $b(t)$ is the width after time t of deformation. This can be seen in Figure 8 from Mantell [18]. Further manipulation of the intimate contact equation with assumption that the element volume stays constant and the law of conservation of mass, as shown in Equation 3:

$$D_{ic} = \frac{1}{1 + \frac{w_0}{b_0}} \left[1 + 5 \left(1 + \frac{w_0}{b_0} \right) \left(\frac{a_0}{b_0} \right)^2 \int_0^{t_p} \frac{P_{app}}{\mu_{mf}} dt \right]^{\frac{1}{5}} \quad (\text{Eq. 3})$$

where a_0 is the initial height, $a(t)$ is the height after time t of deformation, μ_{mf} is the fiber-matrix viscosity, P_{app} is the applied pressure and t_p is the time of pressure application [18]. With the above theory from Mantell, Grunewald developed a model for predicting the bond strength between skin and core. Their model predicts that sandwich composites manufactured with skin pre-heat temperatures above 290 °C and a core kept at room temperature will have a sufficient bond strength, which was superior to the tensile strength of the core.

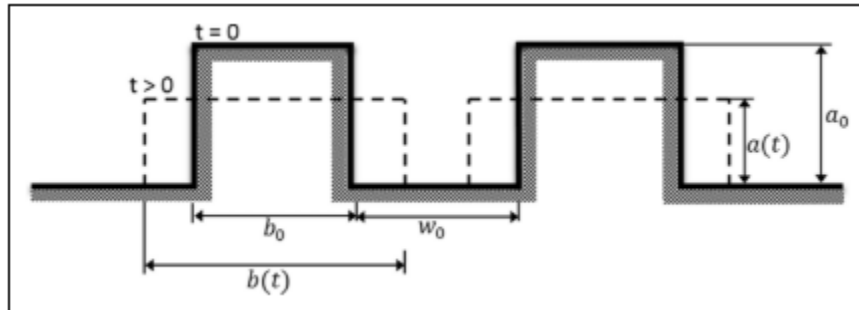


Figure 8 - Illustration of the deformation of the surface asperities - Mantell and Springer

Their model also shows that a pre-heating of the core would probably have a positive effect on bond strength.

When considering sandwich composites for a structural application, it is important to investigate the performance of the material in hygrothermal conditions. Fang et al. investigated the hygrothermal aging of polymer sandwich structures used in civil engineering applications [19]. Fang et al. determined that a hygrothermal environment, as defined to be a combined moisture and elevated temperature, causes swelling, plasticization, and degradation of the matrix, which degrades the mechanical properties of the sandwich composites. Fang et al. study focused on the degradation mechanism of the face and foam core in the combined hygrothermal condition using scanning electron microscopy (SEM) and Fourier-transform infrared spectroscopy (FTIR) analysis to examine the changes in microstructure and chemical structures. They also used an analytical model to consider the effect of hygrothermal aging on bending stiffness of the face material as well as the face/core interface bonding. The conclusion of the Fang et al. study showed that the tensile strength of GFRP face sheets experienced a significant downward trend after 4320 hours of aging in the combined hygrothermal conditions. The main reasons for the mechanical property changes in GFRP face sheets was due to resin hydrolysis and debonding at the interface. They also found that the flexural strength and modulus lost 15.62% and 7.98% respectively for GFRP face sheets aged for 4320 hours. The compressive strength and modulus of their polyurethane cores decreased by 31.77% and 44.74% respectively after aging. From this, it was observed that the degradation of the foam core was much faster than the GFRP face sheets. This means the GFRP played an important role in protecting the core material. Figure 9 summarizes the aging test results for Fang et al load displacement curves.

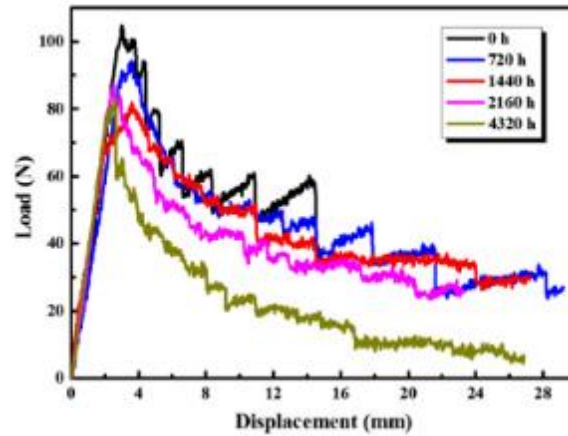


Figure 9 - Load displacement curve for hygrothermal aging- Fang et al.

Analyzing the failure of sandwich composites is of interest to determine the characteristics of the structure under expected loads for its application. Long et al. investigated the failure analysis of a foam sandwich laminate under impact loading [20]. Their study was done with drop weight testing of different impact energies, foam densities and laminate stacking sequences. They examined the failure within the structure using ultrasonic microscope and cross section observation. Their results showed that the delamination of the face material followed the debonding law of laminates and the penetration of the top plate would change the shape of the delamination region as well as stop the expansion of the delamination area as the impact energy was increased. The setup they used for their drop weight testing can be seen in Figure 10. The resulting impact studies by Long et al examined the impact energy of 7, 17 and 31 J. They took sectional images of the samples after impact and this can be seen in the Figure 11. They observed that the impact area increased with the impact energy. Based on their findings, Long et al. concluded that the sandwich structure performed differently in failure characteristics before and after the penetration of the face. They found that the delamination appears only in the top face before penetration and after penetration the delamination region becomes an annular. Another conclusion was that a sandwich structure with a hard core is more vulnerable to delamination than one with a soft core, but this was negligible when there was penetration of the structure.

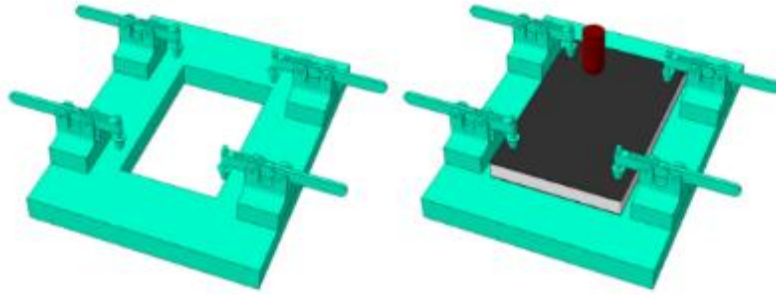


Figure 10 - Drop weight testing drawing by Long et al.

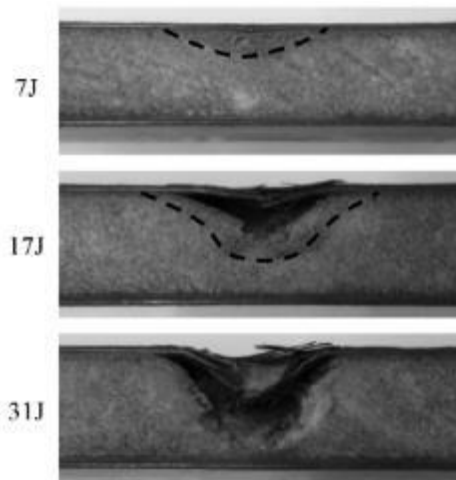


Figure 11 - Impact testing imaging - Long et al.

2.3 - Commingled Fibers

Commingled fibers are becoming more prevalent in the field of composites due to the combination effect of different material systems. A reinforcing fiber can be commingled with a thermoplastic polymer to create a two-part system of fiber reinforcement and matrix. The thermoplastic polymer would act as the matrix in this system where the fiber would be the reinforcing material, such as glass fiber or carbon fiber. The advantages of this type of system are numerous. For instance, the processing of a thermoplastic system can be more advantageous to a thermoset system for research and development. Thermoplastics by nature can be re-used and reprocessed after going through heating cycles whereas thermosets cannot be re-melted. For research, this is an added benefit in that process optimization can take multiple iterations to reach the desired properties and processing conditions. Being able to experiment with the processing conditions with the same system without having to recreate a sample or part is possible with thermosets. Thermoplastics also have an added benefit of impact resistance, which is desired for automotive and aerospace applications. The stiffness of a part can be increased through the addition of carbon or glass fiber, while the thermoplastic matrix such as PA6, PP, or PPS can provide the ductility and impact resistance.

Laberge-Lebel examined the manufacturing process of braided thermoplastic composites through the use of carbon/nylon commingled fibers [5]. They state that the main drawback of a thermoplastic composite is the high viscosity of their matrices, almost 100-1000 times more viscous than thermosets. In order to combat this obstacle in processing, commingled fibers were developed in the 1990s. During their study, Laberge-Lebel implemented a nylon wrapping strategy to prevent the commingled fibers from unmingling during tow manipulation. After this process, they performed a braiding of the yarns. They determined that the consolidation process of the commingled yarns took place inside each tow where the melted resin surrounds a set of cylindrical dry fiber bundles. The braid was then compression molded to produce laminates for flexural and tensile testing. Their results show that the maximum flexural strength was observed as the angle approaches 0 degrees for the braiding angle. Figure 12 shows the depiction of Laberge-Lebel's braid and consolidation plate [5].

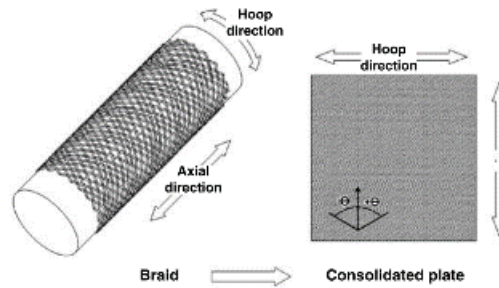


Figure 12 - Sample geometry – Laberge-Lebel

The braid angle can be seen in the consolidated plate that they then tested for flexural and tensile properties.

Hasan et al. examined the use of carbon fiber/polyamide 6,6 commingled yarns for textile reinforced thermoplastic composites [21]. As mentioned by Laberge-Lebel, Hasan also notes that the use of reinforcement fibers with the thermoplastic fibers in hybrid yarns will minimize the mass transfer distance of the thermoplastic melt in the composite. During their review, Hasan found the hybrid yarn manufacturing process has several methods to combine carbon filament yarn with thermoplastic components through twisting, powder coating, DREF friction spinning, spreading, stretch breaking and commingling by means of compressed air. The commingled process has advantages over the other processes due to higher productivity, higher flexibility in the manufacturing, such as the ability to customize the fiber matrix ratio and yarn fineness. The downside of the commingling process is that there is damage in the brittle reinforcement (carbon fiber in this case) due to the air pressure and special care is necessary when processing. Hasan found that the carbon fiber surface properties must be considering during the process due to the interaction between fiber and matrix playing a large role in adhesion properties as well as long term performance of the carbon fiber reinforced composite. In general, carbon fibers are surface oxidized and sized to improve adhesive to the matrix according to Hasan. The process chosen by Hasan involved homogenous mixing of both components, which required overfeeding of the filaments while keeping the speed of the feeding rollers faster than that of the drawing roller. The setup for their commingling process can be seen in Figure 13 [21].

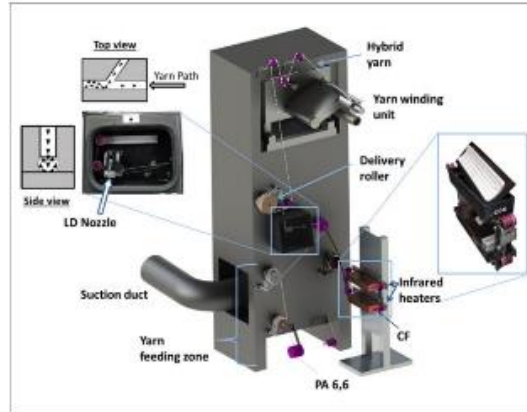


Figure 13 - Comingling air texturing machine - Hasan et al.

The IR heaters in the system in Figure 13 were used to investigate the effect of heating on the individual filaments in the carbon fiber yarn in hopes to improve the mixing with the thermoplastic filament. The chosen temperature for their examination was 430 °C, because carbon fiber yarn breaks frequently above this temperature. Hasan concluded that the damage in the carbon fiber yarn due to the air pressure in the comingling process was compensated in the overall composite due to a bridging effect as a result of the good impregnation by the surrounding thermoplastic matrix. The increase in air pressure caused the brittleness of the tested composite to increase, which resulted in lower impact strength. The heat treatment of the carbon fiber yarn showed a higher adhesion between fiber and matrix, which Hasan confirmed with the increase in tensile strength in the 90° direction. It was seen that the application of heat can improve the mixing and adhesion process for the comingled system.

Through investigation into comingled fibers, the use of comingled fibers in sandwich construction is lacking. There was a study done by Kumar et al. that examined the use of comingled glass fiber polypropylene (PP) composite for a skin and a high density polyethylene (HDPE) foam as the core of a sandwich composite [22]. They examined the use of IR heating to bond the glass fiber PP to the HDPE core through cold pressing. The bond layer between the face and core was improved using ethylene-propylene copolymer (EPC) tie-layers with high melt flow. Kumar heated the PP to a temperature of 190 °C on the resin rich side before placing the tie-layer, then the cold

press was used to apply pressure. After this, they heated both the PP face and the HDPE foam to 200 °C before placing them together and applying cold pressure to enable the fusion bonding between the melted surfaces. They determined that this process was feasible for creating an all-olefin laminate system.

When examining the use of commingled fiber systems in composites, the effect of temperature and aging is of concern. Prabhakaran et al. investigated the environmental effect on mechanical properties of CF/PA6 commingled yarn composites [23]. The aim of their study was to examine the influence of moisture absorption and temperature on the mechanical properties of CF/PA6. It is well known that PA6 is sensitive to moisture and requires drying and storage conditions during its usage for composites manufacturing. The laminate system they tested was 52% carbon fiber by volume and 10 mm thick. The aging process took place with samples exposed to 100% RH at 60 °C for different lengths of time up to 2500 hours and then they were dried at 23 °C and 50% RH. The construction of their laminate system was done using quasi-isotropic setup with an autoclave consolidation as seen in Figure 14 [23]. Prabhakaran outlines the autoclave technique as follows: prepreg material is placed and stacked in a one sided mold, then it is bagged and sealed under vacuum bagging film, vacuum is drawn to remove air, pressure is applied outside the bag to increase consolidation pressure, mold is heated to temperature where the nylon will wet the fibers, and finally the mold is then cooled to solidify the material into a composite laminate. The Figure 15 represents the heating, pressure, and cooling cycles of the autoclave setup [23]. The layup goes through cycles of heating and cooling under pressure in an autoclave. The result is a consolidated commingled fiber composite. Prabhakaran then tested flexural and ILSS after different aging conditions as well as at different temperatures. The results of their flexural tests can be seen in Figure 16 and Figure 17 [23]. The flexural strength of the laminate decreases after aging as well as decreases with increased temperature of testing. They found that aging past 500-1000 hours did not have much of an effect on the flexural performance of the commingled carbon fiber/nylon system. The results of their study are important in understanding the effect of extreme conditions on thermoplastic polymer systems that are to be used in automotive and aerospace applications.

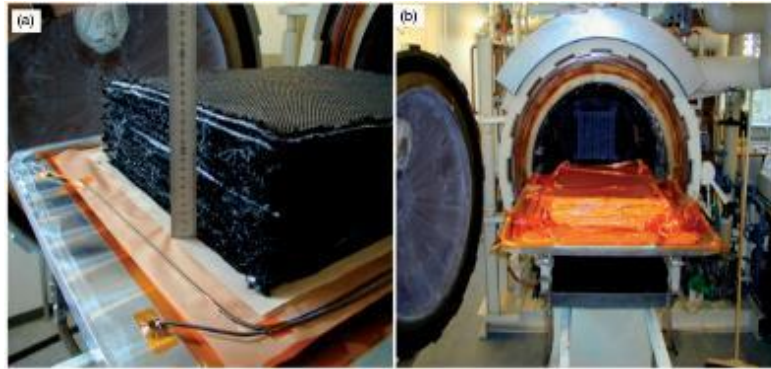


Figure 14 - Autoclave setup - Prabhakaran et al.

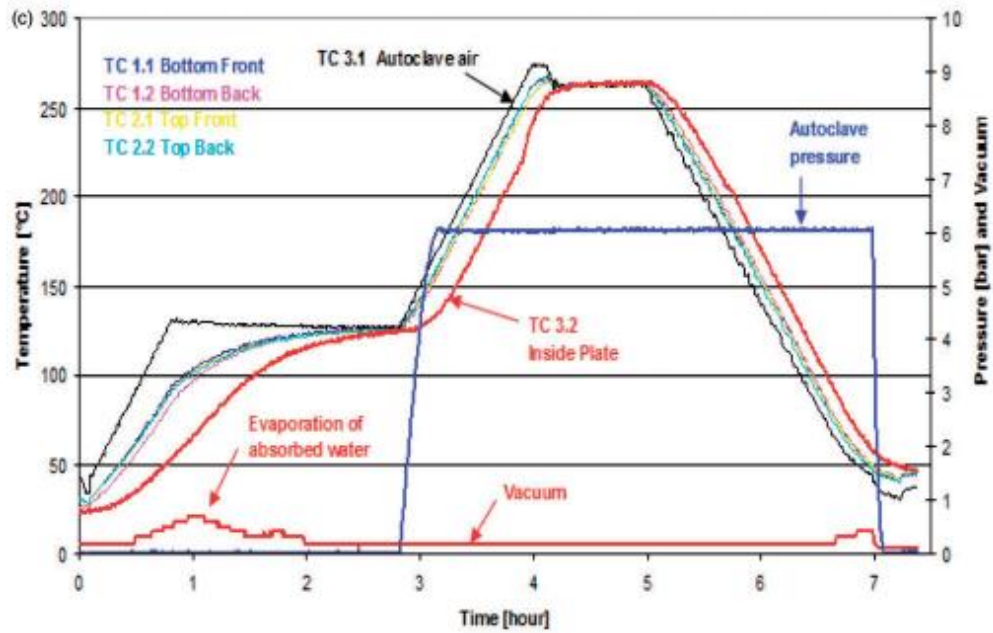


Figure 15 - Autoclave processing conditions - Prabhakaran et al.

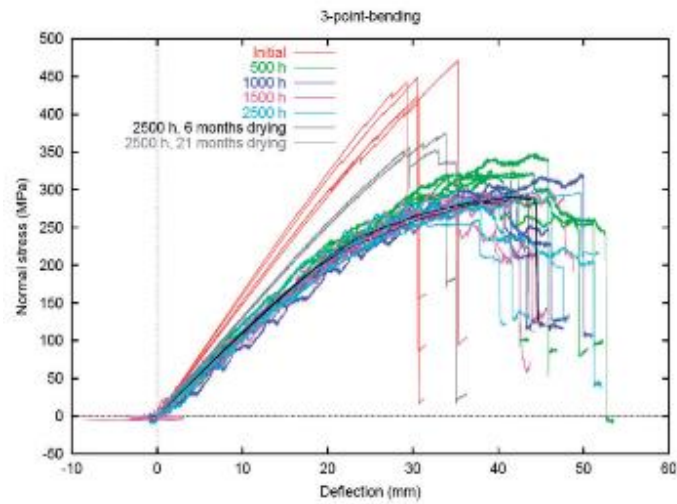


Figure 16 - Flexural test after aging - Prabhakaran et al.

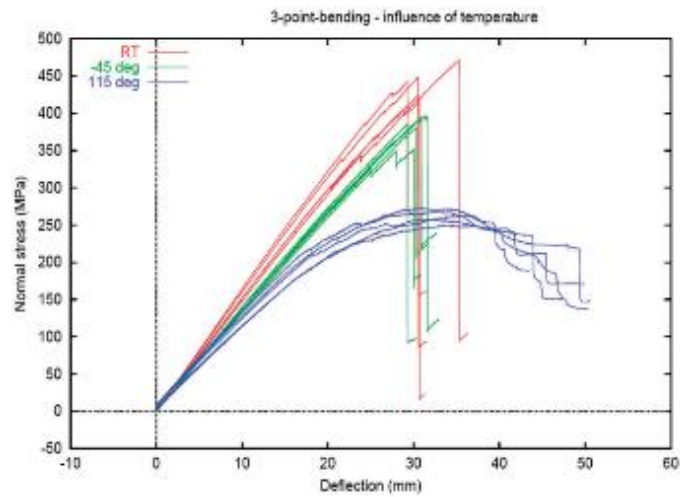


Figure 17 - Flexural testing at different temperatures - Prabhakaran et al.

2.4 - Aerospace Applications of Composites

The wide use of composites has been increasing in recent years within the aerospace industry. Boeing is moving towards building aircrafts that are primarily composed of composite materials. The Boeing 787 is an example of the shift towards composites for aerospace applications as seen in Figure 18 [1]. Based on Figure 18, Boeing is using composites in 50% of the 787 construction. The structures in the plane that require high strength are typically the CFRP composites as described earlier, whereas lower strength requirements may feature a glass fiber reinforced plastic (GFRP).

With the implementation of CFRP into aircraft, there comes with it a need for extensive testing and simulation to make sure the material can hold up under excessive usage. Fatigue testing is of importance for aerospace since these aircrafts are subjected to thousands of uses in their lifetimes. An examination on performance of CFRP after fatigue and thermal aging was conducted by Garcia-Moreno et al. In aerospace components, one of the greatest concerns is with long term aging effects since oxidation affects the properties of the polymer including failure performance [24]. It is well known that fibers are relatively stable, but the matrix and interface with the fibers can degrade with temperature. The glass transition temperature of thermoset polymers defines the region where the material has high stability in properties.

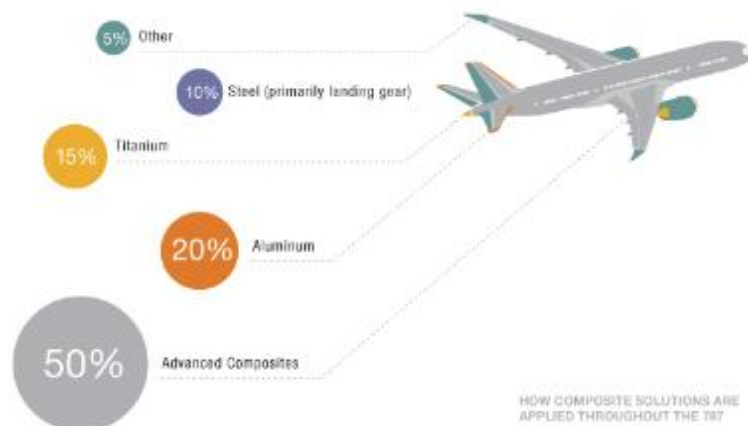


Figure 18 - Boeing 787 breakdown - Hale

There are two aging processes for polymers: physical and chemical. Physical aging is reversible in that the polymer is exposed to elevated temperatures below the glass transition and the structural integrity is not compromised. Whereas chemical aging is irreversible and results in a reduction of the molecular weight due to changes in cross-link density and depolymerization [24]. Garcia-Moreno found that there was a slight increase in the mechanical properties of CFRP at the first stage of aging because it is a consolidation phase. After that initial phase, a degradation phase begins and the mechanical properties of the composites decrease significantly due to the weakening of the fiber-matrix interface [24]. They found that composite systems for aircraft have service temperatures of 100-120 °C for a continuous basis and 135 °C for short periods. Depending on the location of the material on the aircraft, even higher temperatures can be observed. Based on this, it is important to understand and evaluate the mechanical properties of composite systems for aerospace applications under high temperatures that could lead to degradation. The polymer matrix material is the most affected by the high temperature exposure [24]. This corresponds well to the chosen material for the thesis research, CF/PPS, since it has a high glass transition temperature and consolidation of this system occurs around 420 °F.

The thermal aging study done by Garcia-Moreno can be seen in Figure 19. They examined 4 different temperatures for several different aging durations for both flexural and impact tests. The results observed by Garcia-Moreno were that the average impact strength decreased as the temperature increased. The glass transition temperature of the matrix they used was 150 °C. Samples that were aged at temperatures of 230-250 °C were observed to have extensive thermal degradation resulting in reduction of impact resistance and mechanical performance [24]. The results of Garcia-Moreno's study on both flexural and impact strength can be seen in Figure 20 and Figure 21 [24]. As seen in this study, it is important to examine the effect of temperature on the performance of composite materials to determine the suitability to aerospace applications.

Thermal Ageing Impact Samples		Thermal Ageing Flexural Samples	
Temperature (°C)	Time (days)	Temperature (°C)	Time (days)
150	28, 35, 42	150	10
190	3, 10, 20	190	10
210	3, 10, 20	230	10
230	3, 10, 20	250	10

Figure 19 - Thermal aging study parameters – Garcia-Moreno

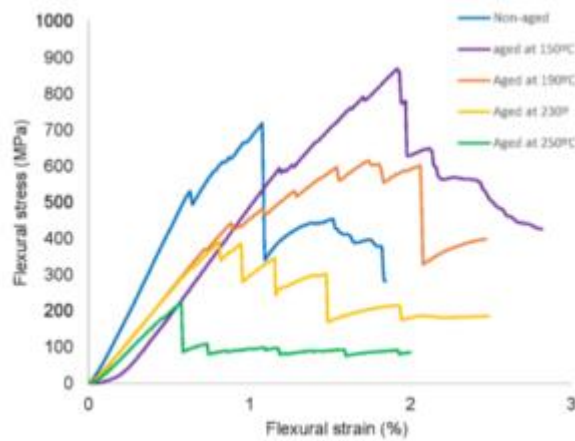


Figure 20 - Flexural stress of aged laminates at different temperatures – Garcia-Moreno

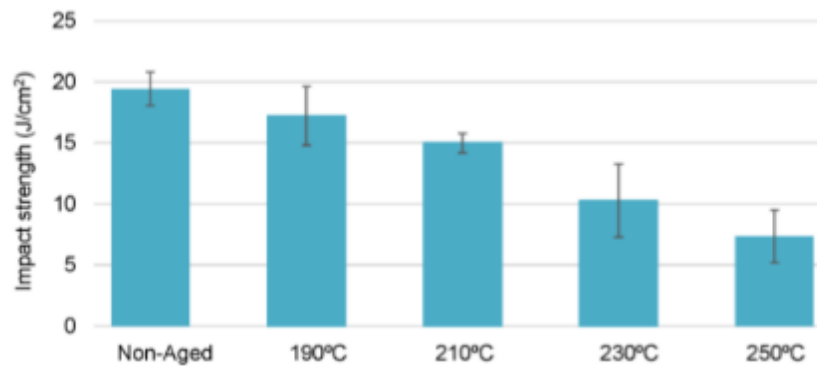


Figure 21 - Impact strength of aged laminates at different temperatures – Garcia-Moreno

2.5 - Filament Winding Process

The filament winding process is used to create composite material products for a variety of applications, but the primary use of filament winding appears to be for the construction of pressure type vessels or tube structures. Ma et al. studied the energy absorption properties of carbon/aramid fiber filament winding composite tubes [25]. In their study, they chose carbon fiber and aramid fiber as the reinforcement with a common epoxy resin as the matrix. They then manufactured different structures of composite tubes, which were then heat treated for extended periods and tested for energy absorption. The tube structures they created can be seen in Figure 22 [25]. They constructed tubes using the filament winding process and then applied compression to examine the failure with an application for automotive industry. These CFRP tubes are then crushed under a load as seen in Figure 23. They found that the energy absorption of carbon/aramid CFRPs increased after heat treatment compared to composites without any treatment.

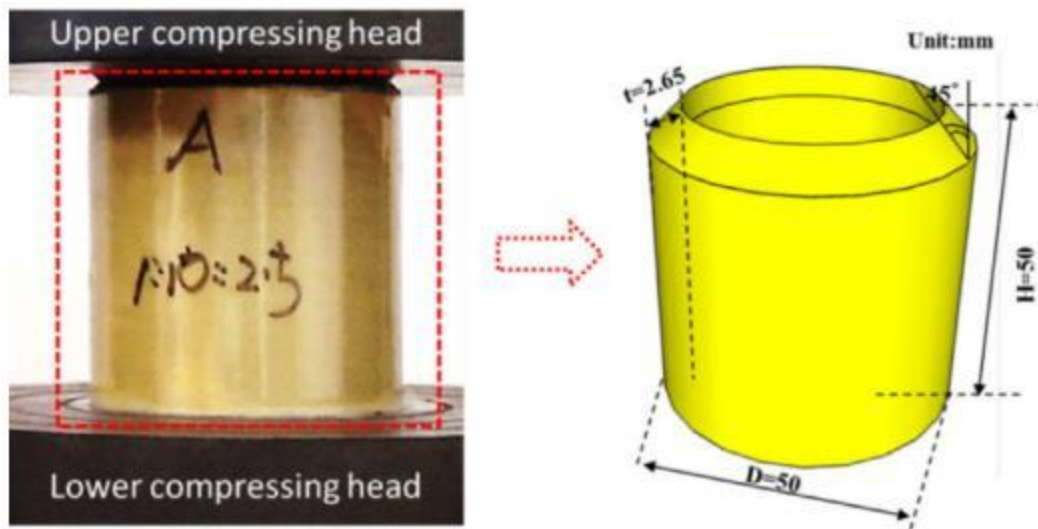


Figure 22 - CFRP tubes for energy absorption - Ma et al.

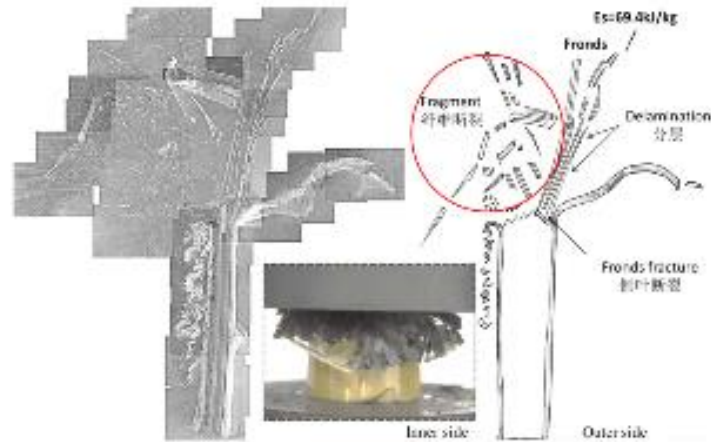


Figure 23 - Failure of tube structure - Ma et al.

Mantell describes the filament winding process for composite tubes as having the ability to apply both heat and temperature during winding [18]. During the filament winding process that they describe, the roller makes more than one pass over a given area. This results in an interface experiencing different pressures and temperatures even though the applied heat and pressure do not change. These different passes must be considered when calculating consolidation and bonding. The setup for this method is shown in Figure 24. Perillo et al., characterized CFRP and GFRP composite tubes that were created through filament winding [26]. They examined two different composite tube structures, one using glass fiber and vinyl ester and the second using carbon fiber and epoxy. They found that when high strength and good corrosion properties are necessary, then a filament wound composite could be ideal. Filament wound composites are replacing metal alloys in high-pressure vessels, aerospace vehicles, transportation tubes in oil/gas industries, and tubular structures [27]. Perillo noted that there are downsides to winding onto a flat plate when creating sandwich composite samples: there can be higher void content and lower fiber fraction compared to the actual components in filament wound material. Perillo used a PVC pipe to prevent damage to the mandrel during winding.

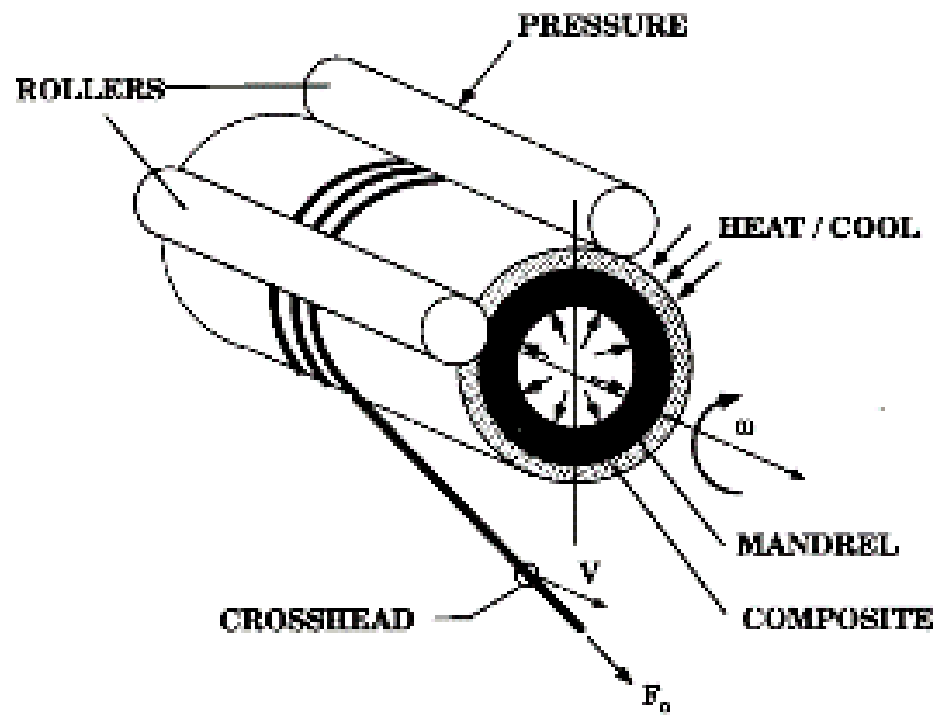


Figure 24 - Filament winding process - Mantell

After the winding, the composite was cured at room temperature for 24 hours and then post cured at 100 °C for 1.5 hours for the GFRP. For the CFRP, the tubes were cured for 2 hours at 100 °C and then 4 hours at 140 °C. They determined the fiber volume fraction for their material systems using an ignition loss test following ASTM D 2584. Perillo et al. conducted a split disk test to evaluate the material strength and modulus in the fiber direction of the tubes. The concluding failure envelopes for the two systems they tested can be seen in Figure 25 [26]. The results of their study showed that they could use Puck's failure theory to accurately predict the failure envelope of the GFRP and CFRP tubes.

When filament winding, the properties of the fibers can be affected by the process. The influence of pretension on the mechanical properties of carbon fiber during filament winding was examined by Akkus [28]. When filament winding, pretension is used to layout the carbon fiber onto the cylindrical tubes in the desired way. According to Akkus, there can be a considerable amount of damage to the fibers during the fiber movement through the pulleys in the pretensioning unit. They examined the effects of roving tension, pulley diameter, and contact angle between the pulley and the fiber. Their study determined that the angle between the pulley and the amount of force applied to the carbon fiber is what causes the damage on the carbon fiber. The results of their study showed that a change to the roving tension (pretensioning) can reduce the tensile strength of carbon fiber by 10 to 43%.

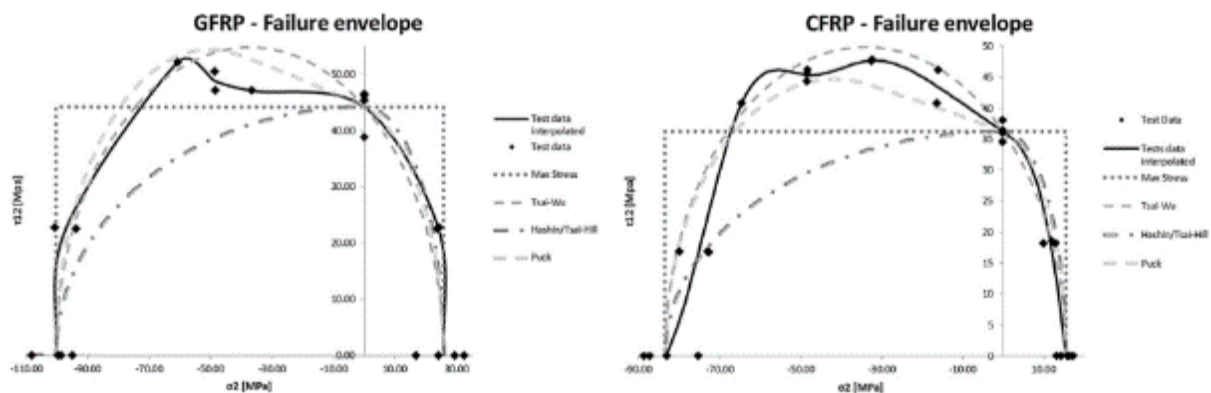


Figure 25 - Failure envelopes for GFRP/CFRP - Perillo et al.

Akkus states that the pretensioning force on the carbon fiber is kept within a specific range by the pretensioning unit as seen in Figure 26 [28]. Each pulley applies tension to the carbon fiber as it goes through the system. When changing the pulley's position and the pulley's diameter, a change in the contact angle with the fiber will occur. The total contact angle with the carbon fiber will affect the strength because of the bending stresses and friction between the carbon fiber and the pulleys. The Figure 27 shows the positioning of pulleys and their effect on the contact angle with the carbon fiber [28]. From their results, Akkus concluded that the damage to the carbon fiber increases as the total contact angle increases due to the increased friction. The damage to the carbon fiber also increases as the pretension force increases, with higher forces causing higher breakage and neps (small entanglement of fibers). They also found that a smaller diameter pulley will result in a greater damage to the carbon fiber because of the bending effects increasing.

With filament winding there are two different main methods for producing filament wound products: wet winding and dry winding [29]. Xu et al. describes the wet and dry winding methods where the wet winding has the fiber passing through a resin bath and winding onto a rotating mandrel, then the dry winding just pre-impregnated fibers placed on a mandrel [29]. They also describe a semi-dry winding method where the tows are heated by a set of drying equipment before the winding onto a mandrel. Their study investigated modular design for filament winding system due to the highly customizable process. The Figure 28 shows their schematic for a filament winding system [29]. Risteska et al. studied the flexural properties of hybrid composites made through filament winding [30]. They examined trapezoidal parts out of carbon and glass fiber using an epoxy resin system from Huntsman. They wound the fibers onto a trapezoidal mandrel with pins on both sides. The shape of the mandrel used did not allow for uniform winding tension during the process. The results of this showed different thickness and quantity of voids in the specimen walls and corners. The resulting flexural properties showed that a lower void content had a strength of 501 MPa compared to that of 359 MPa for higher void content. They concluded that void content can be reduced if a contact method or higher winding tension was used during winding. From this, proper design of the winding machine and specifications must be done to limit the creation of voids during winding.

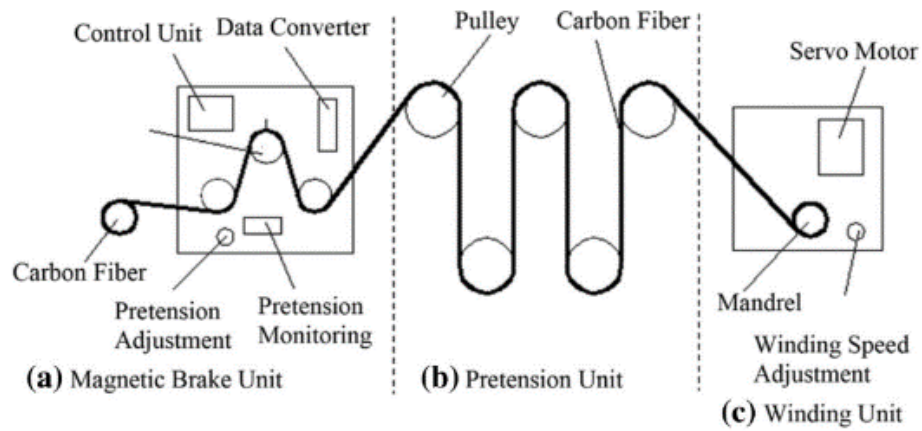


Figure 26 - Filament winding pretension system – Akkus

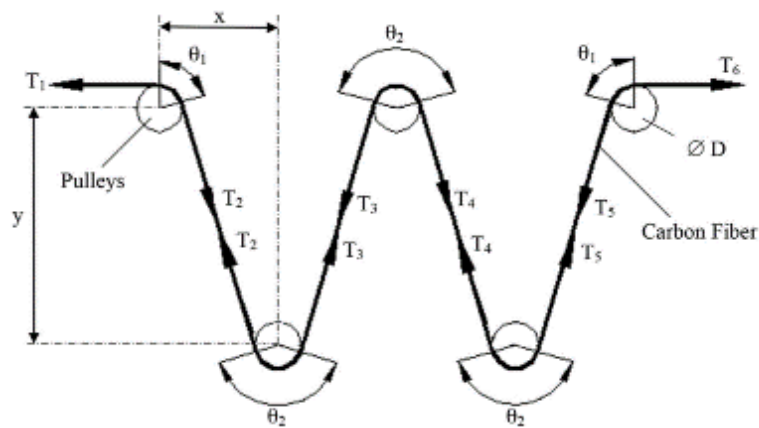


Figure 27 - Position of pulleys - Akkus

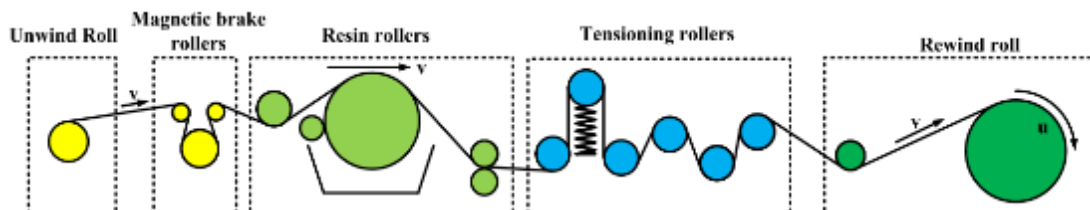


Figure 28 - Schematic of filament winder - Xu et al.

Almeida Jr. et al. examined the creep and interfacial behavior of carbon fiber reinforced epoxy laminates made through filament winding [31]. According to Faraz et al., creep is an important material property for a long term point of view due to the service durability and safety concerns [32]. This is especially important for aerospace and automotive applications where lives are at risk and the materials need to perform and last for long periods of time. Because of this, thermosets are more heavily used in aerospace because of their creep resistance being better than thermoplastics. Based on this, Almeida Jr. et al. determined that examining creep behavior in CFRP laminates is of interest. They created flat laminates using a rectangular steel mandrel in a filament winding system with the assistance of a shrink tape to help consolidate the laminate throughout the curing process. They then cured the laminates under hot compression of 6 ton for 4 hours at 130 °C. The resulting laminate was 12 layers and 4.2 mm thick. The results of Almeida Jr et al. study showed that the creep performance was dependent on fiber orientation. They also found that the short beam strength was lower for 30° fiber orientation due to the high anisotropy of the composite. The failure at 30° was more dependent on the matrix and the interface. When examining the 0° fiber orientation, they noticed that the samples failed by interlaminar shear as the mid-plane delamination was present. The results of their creep behavior study are summarized in Figure 29 [31]. Their results matched the two models for creep behavior at the 30 °C and 60 °C conditions. The Findley model was better for the higher temperature testing as observed in the data.

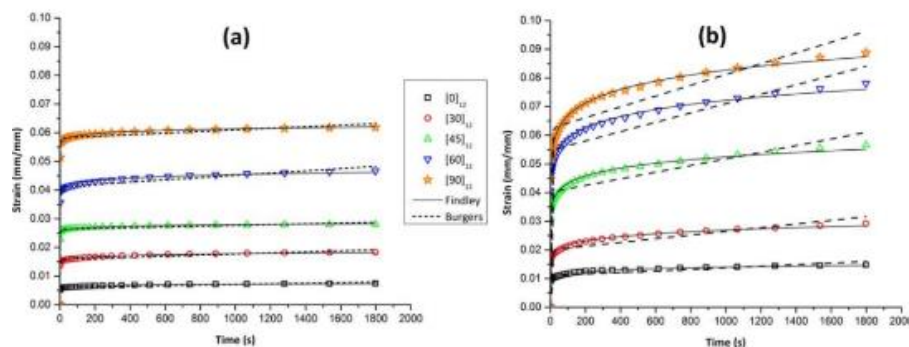


Figure 29 - Creep behavior of CFRP at 30 °C (a) and 60 °C (b) - Almeida Jr. et al.

As previously noted, filament winding is also used to create pressure vessel shells. Gusev et al. discusses the use of carbon fibers for high-pressure vessel shells [33]. According to Gusev et al., shells constructed from polymer composites are used in military and civilian applications. The primary method for creating these shells is through filament winding. As the winding process is ongoing, the fibers are soaked in a binder during the process. Filament winding allows for the reinforcing fibers to be oriented in a way such that the distribution of the principal stresses will assure high operational reliability. Polymer composite shells can then be used as tanks for various applications where high internal pressures are observed, such as solid fuel rocket engine bodies in Figure 30 [33]. Gusev et al. evaluated different fiber fillers for the shell winding process. The fiber filler of the shells was selected based on the main load acting on the part as well as the need to minimize the mass of the part. They examined carbon fiber, glass fiber and aramid fibers for the use in shell construction. Based on their process, they were able to determine the highest performing fiber for their application based on the strength to weight ratio. This strength to weight ratio is a criterion used for many applications such as automotive and aerospace when determining if a composite is suitable to replace a metal.



Figure 30 - Rocket shell body - Gusev et al.

Almeida Jr et al. examined CFRP filament wound composites exposed to hygrothermal conditioning [34]. Their process used filament winding to create flat laminates that were then exposed to hygrothermal conditions before testing. After winding, a polyester shrink tape was used to wrap and consolidate the laminate resulting in elimination of voids and improving ply compaction. The weathering of the specimens was done for 60 days at 80 °C and 90% RH. The samples were then dried before placing in the hygrothermal chamber to ensure that they had reached mass equilibrium and had no moisture from atmosphere. The shear stress of the weathered samples can be seen in Figure 31. Almeida Jr. et al. found that the mechanical properties of the filament wound laminates decreased due to aging. They also noted that the 90° samples were more strongly affected by the aging process. The reason for the 90° samples being more strongly affected could be attributed to the weakening of the fiber matrix interface. It was found that filament winding is heavily used to create carbon fiber reinforced composites, but there is not much investigation into using filament winding to create the face materials for sandwich composites. Most of the literature involves examining tube structures created through the filament winding process or examining a flat laminate. There have been a few studies described above that examine filament winding as a method for producing laminates and then examining the aging effects as well as processing condition effects on the mechanical properties of the composite.

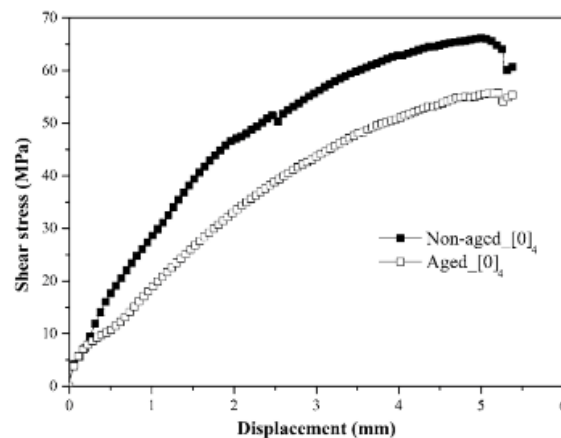


Figure 31 - Shear stress - Almeida Jr et al.

2.6 - Carbon Fiber – Polyphenylene Sulfide

When examining materials used in sandwich construction, most of the face materials consist of some form of carbon fiber reinforced plastic. This can range from polypropylene (PP) to nylon (PA6) to polyphenylene sulfide (PPS) depending on the application. For aerospace, the needs of each application require high performing materials with high temperature stability and high stiffness. PPS is an ideal candidate for use in aerospace since PPS has a high melting temperature of 280 °C [35] and high strength when compared to other thermoplastic polymers. With the addition of carbon fiber reinforcement, a CF-PPS hybrid system is suitable for aerospace applications due to the added stiffness from the carbon fiber. Vieille et al. examined the influence of temperature on mechanical properties for carbon fiber reinforced PPS laminates [35]. In order to evaluate the performance of these materials for an aerospace application, the high temperature performance is of interest. Walther investigated the tensile performance of CF-PPS laminates at 30 °C and 140 °C, showing a tensile strength decrease by 17% and a modulus decrease of 7.7% as the temperature was increased [36]. Vieille et al. tested woven prepreg laminate plates for their study with a carbon fiber volume fraction of 50%. The tested plates were tested after different stacking sequences with a [0/45/0/45/0/45/0] as quasi-isotropic. Vieille tested compressive, flexural, interlaminar shear (ILSS), and bolted joint tests to examine the performance of CF-PPS at elevated temperatures. The results of their study showed that the ductile behavior of the matrix (PPS) at elevated temperatures is detrimental to the compressive strength, flex properties and ILSS strength. The joint strength of the laminates decreased by 21% when the temperature was increased to 120 °C.

Kim et al., characterized injection molded thermoplastics with carbon fiber reinforced PPS as the material [37]. Their study showed that PPS can reduce the weld line formation of liquid crystal polymer and improve the processability of the liquid crystal polymer hybrid material. Another investigation by Xu et al. investigated the effect of air plasma treatment on interfacial shear strength of carbon fiber reinforced PPS [38]. They cite PPS as widely used in defense and military applications due to its high thermal stability, chemical resistance, and flame resistance. The issue with neat PPS is that it has

low impact toughness and tensile strength, but this can be improved with carbon fiber addition. Carbon fiber is chemically inert, which leads to poor compatibility with matrix [39]. Polymers in general are hydrophobic with relatively low surface energy, which results in inadequate adhesion bonding [40]. Xu et al. note the major approaches for enhancing the interphase of carbon fiber reinforced composites as modification of the polymer matrix and surface treatment of the carbon fibers. In an aim to improve the bonding, they implemented plasma treatment to modify the CF-PPS interface. They found that plasma treatment of the carbon fibers reduced the interfacial shear strength of the micro-composite by 13.7% whereas the plasma treatment of the PPS fibers increased the interfacial shear strength of the micro-composite by 17.1%. Liu et al. examined the interfacial micromechanics of CF-PPS composites through sample annealing treatment and microbond testing [41]. They modified the surface of the CF with air dielectric barrier discharge plasma and examined the surface with SEM and X-ray photoelectron spectroscopy (XPS). The Figure 32 shows the micro droplet setup. The result of Liu et al. study showed that annealing treatment increased the interfacial shear strength by 30%, but the plasma treatment reduced the interfacial shear strength by 36 MPa. They claim that this could be due to the removal of the size layer coating on the carbon fiber surface.

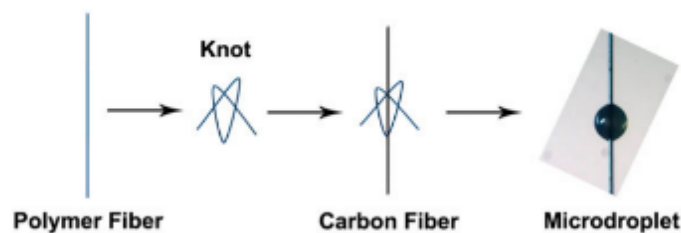


Figure 32 - Microdroplet formation - Liu et al.

When using fiber reinforcement of a polymer, the properties of the hybrid system will differ from the neat polymer and an understanding of the effect of fibers on glass transition temperature is of interest. Rebenfeld et al. examined the effects of fibers on the glass transition temperature of PPS composites [42]. Rebenfeld et al. state that the extent and nature of crystallization of a thermoplastic polymer is controlled by the thermal history. The crystallization rate of is related to the glass transition temperature, which is in the region of 85 °C for PPS. The authors assumed that the introduction of fiber fillers into the polymer matrix would increase the glass transition temperature due to the chain mobility restrictions and wall effects. Ma et al. determined that the addition of carbon fiber increased the glass transition temperature by 5 °C [43]. From the results of Rebenfeld et al. study, they found that the glass transition temperature depressed in the PPS prepregs due to the finish on the fiber surface. The reduction was only in a 3 – 5 °C range, but they assumed an average value and conclude that the PPS polymer at or near the fiber surface may be much lower in glass transition temperature than the average.

PPS is also used in other applications, such as electrical and optical-fiber cables, chip carriers, printed wiring board substrates and components for electronic capsules as described by Lu et al [44]. The reason for PPS in these applications is due to the conductive nature that can be applied through doping with strong oxidizing agents. Pure PPS is brittle, which limits its applications even though it has excellent properties. Lu et al. aimed to improve the impact strength of PPS by heat treatment and the addition of nano-particles. A higher molecular weight polymer has better mechanical properties due to high density of chain entanglements and imperfect crystal structure [45]. In order to increase the molecular weight, heat treatment can be used. Lu et al. concluded that heat treatment increased the molecular weight of PPS by 27.8%. The addition of nano-SiO_x improved the impact properties of the PPS by increasing the impact strength by 90% while lowering crystallinity by 27.3% when compared to the pure PPS. Diez-Pascual et al. examined the properties of CF-PPS laminates that were enhanced by inorganic nanoparticles [46]. Pascual et al. added inorganic fullerene-like tungsten disulfide nanoparticles to a CF-PPS laminate to examine the effect on morphology, thermal, mechanical, and tribological properties. They found that the addition of 1.0 wt.% of the nanoparticles increased the flexural strength and modulus by 17% and 14% respectively

without a loss of toughness. The wear rate and coefficient of friction was also decreased due to the lubricant role of the nanoparticles. The introduction of these types of additives to thermoplastic composites allows for new developments in high performance structures.

Fatigue loading of PPS composites was studied by Baere et al. through the use of dumbbell-shaped specimens [47]. Fatigue behavior is important for composites due to the life duration of the application. For automotive and aerospace industries especially, the materials used must be able to endure many cycles throughout the lifetime. The traditional fatigue tensile testing is done with a rectangular specimen, but the implementation of the dog-bone type specimen by Baere et al. allowed for higher failure stresses under quasi-static tests and higher maximum load levels under fatigue loading conditions. They concluded that very little permanent deformation occurs, and very little stiffness degradation is present. For stress levels lower than 675 MPa, lower frequency yields a larger number of cycles to failure for their study. Figure 33 and Figure 34 show the dog-bone specimen and the fatigue cycles respectively [47]. The fatigue loading under 675 MPa has a higher number of cycles to failure as inferred through Figure 34. There have also been studies done into the use of recycled carbon fiber as the reinforcement for PPS composites. Stoeffler et al. investigated the use of recycled carbon fiber for PPS composites [48]. With an increased demand of carbon fiber, the use of recycled materials can provide an added benefit. If recycled carbon fiber can be used instead of virgin carbon fiber without loss of performance, costs of production of composites can be greatly reduced. Carbon fiber demand for aerospace applications was 7000 trillion worldwide in 2011 and is expected to increase to 19,700 trillion by 2020 according to Das et al [49]. Figure 35 shows the carbon fiber demand projections for four major applications [49]. The demand alone justifies investigation into recycling of current products after end cycle. The price of virgin carbon fibers is between \$32/kg and \$65/kg whereas reclaimed carbon fibers from CFRP waste is \$18-26/kg [50].

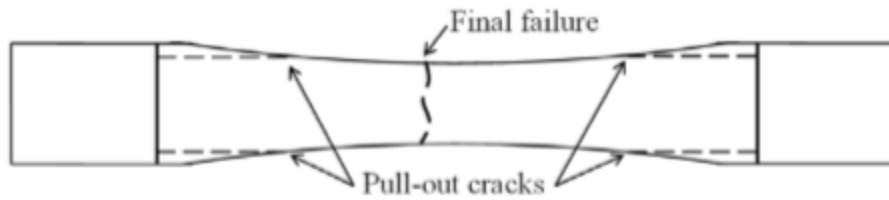


Figure 33 - Dog-bone specimen failure - Baere et al.

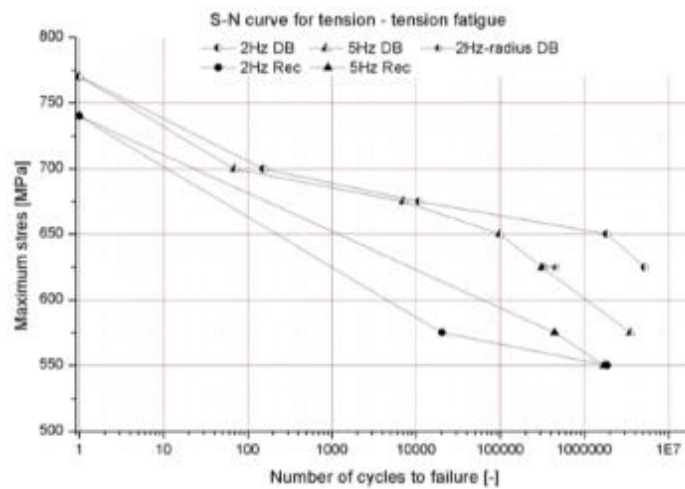


Figure 34 - Fatigue loading to failure - Baere et al.

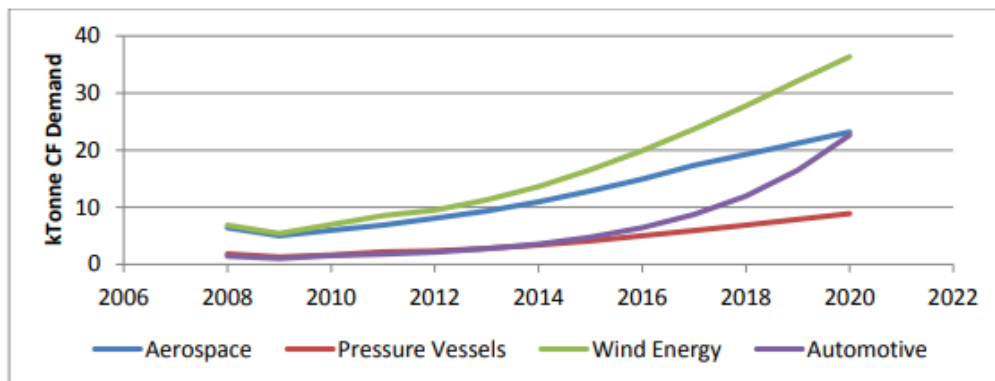


Figure 35 - Carbon fiber demand projections - Das et al.

The reclamation of carbon fibers can be done through pyrolysis which thermally degrades the matrix and retrieves the fibers and catalytic conversion which decomposes the polymer matrix into low molecular weight hydrocarbons using solvents [51]. Stoeffler et al. used aerospace thermoset composite waste for pyrolysis to reclaim carbon fibers and then reinforced PPS through twin screw extrusion. The results of their study showed that the mechanical properties of PPS composites reinforced with recycled carbon fibers were similar if not better than those using virgin carbon fibers. The results of Stoeffler study for tensile properties comparing recycled vs commercial products can be seen in Figure 36 [48]. The recycled carbon fiber composites are highlighted in red and are similar to that of the commercial grade carbon fiber composites. This evidence shows the potential for reinforcement of PPS with recycled carbon fiber.

EL-Dessouky et al. investigated spread tow technology in creating a carbon fiber thermoplastic composite [52]. According to Dessouky, there is a method of achieving ultra-lightweight composites by spread-tow technology where a conventional 12k carbon fiber tow is thinned by increasing tow width from 5 mm to 25 mm. This reduces the weight per unit area by around 500%. They used a PPS thermoplastic film to stabilize and impregnate the woven carbon fiber spread-tow fabric. The resulting composite laminate had a 55% weight carbon fiber composition. They then tested the flexural characteristics of both spread-tow composite and a conventional woven 60/40 wt.% CF/PPS prepreg. A schematic of their spread-tow setup can be seen in Figure 37. The results of their study showed that spread-tow fabric reduces the void content and resin rich areas within the composite, which enhances the mechanical properties. The mechanical properties of the two tested laminates showed that the flexural strength for the spread-tow was 829 MPa versus 748 MPa for conventional with half of the weight.

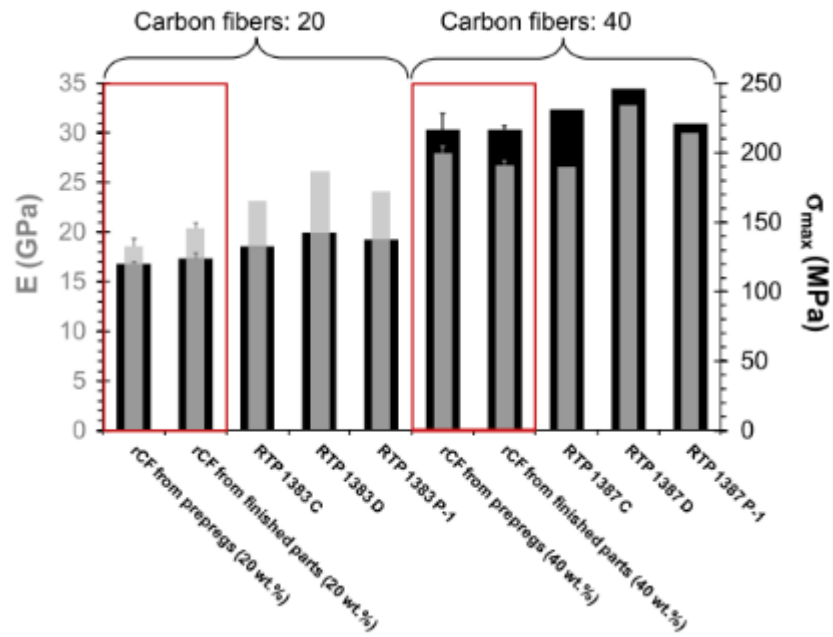


Figure 36 - Recycled CF vs commercial tensile properties - Stoeffler et al.

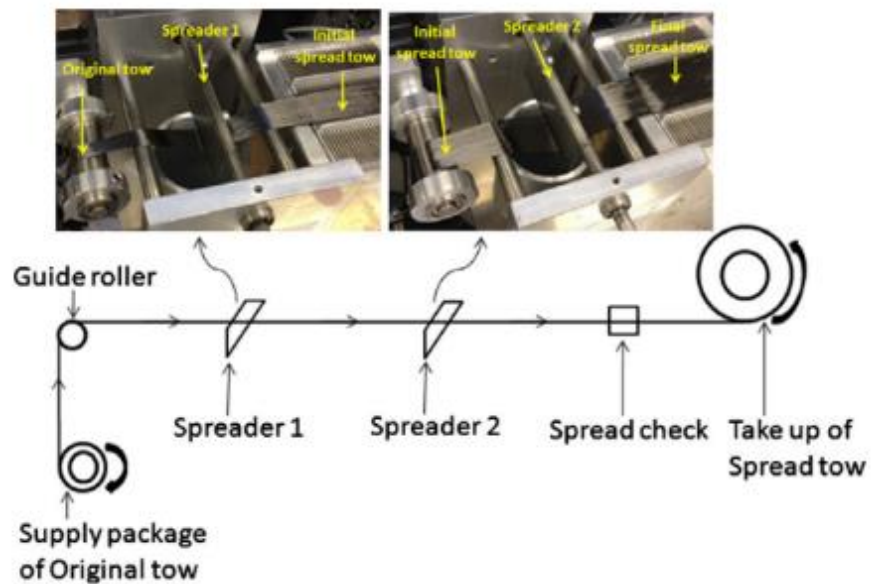


Figure 37 - Schematic for spread-tow of CF - EL-Dessouky et al.

McCallum et al. examined the influence of skin-core residual stress and cooling rate on the impact response of CF-PPS [53]. The importance of low-energy impact damage was noted by the author due to the damage leading to subsurface matrix cracking, fiber fracture and delamination. According to McCallum et al., the cooling rate in a thick laminate tends to be slow in the core of the laminate when compared to the skin. This process results in a stress distribution of parabolic nature with compressive residual stress in the surface plies and tensile stresses in the core. A method of reducing these stresses is annealing where the laminate temperature is raised above the glass transition temperature of the matrix allowing the residual stresses to relax in the semi crystalline composite [53]. In their study, McCallum et al. ran numerical simulations to estimate the influence of skin-core residual stress and cooling rate. These simulations were then compared to drop tower impact test results. The Figure 38 shows the impact energy results for the simulations. The Figure 39 show the results of their drop tower impact study compared to the simulations. The results of their study showed that the impact force increases with increasing impact energy for both fast and slow cooling, while fast cooling resulted in lower delamination extent when compared to slow cooling.

2.7 - Uniqueness of This Thesis Work

While there have been many investigations into PPS as a matrix for CFRP applications, there is a gap in the space of commingled fibers. Most of CF-PPS literature shows investigations into laminate made through hand lay-up or prepreg type systems. These systems are also often only evaluated as a laminate and not for use in a sandwich composite. The combination of commingled CF-PPS into a sandwich construction shows potential for current and future studies that are beneficial to aerospace and automotive industries. The current study focuses on the use of commercial grade products, but the availability of future recycled carbon fiber could be another area of investigation that could build on the current sandwich composite studies.

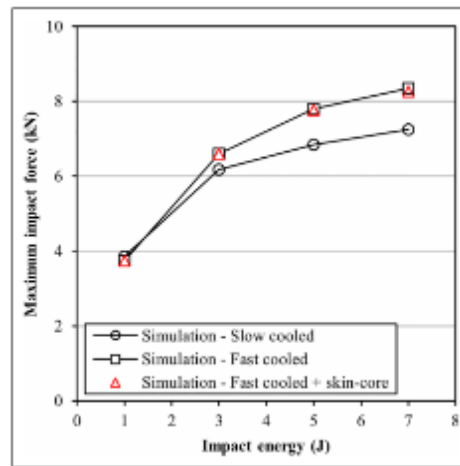


Figure 38 - Impact results of simulation - McCallum et al.

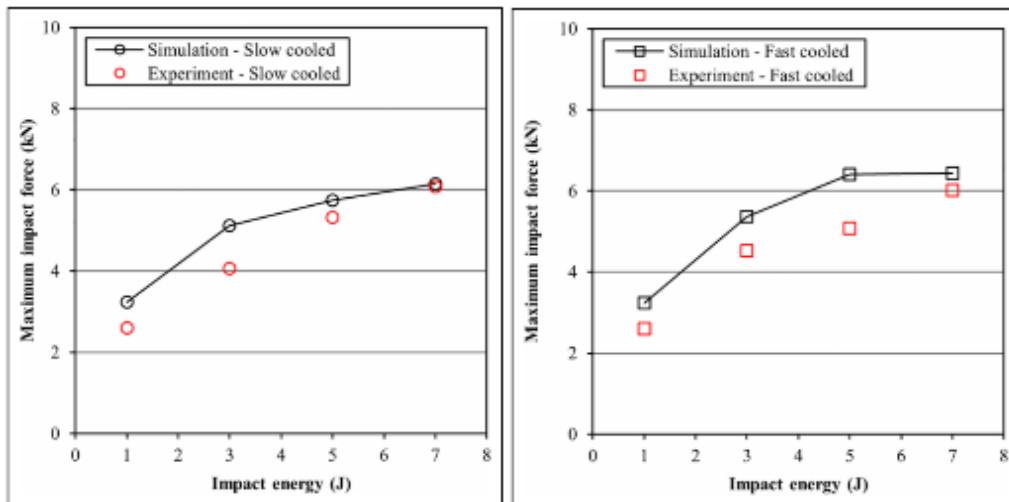


Figure 39 - Experimental impact results vs simulation - McCallum et al.

Chapter 3 – Materials and Methodology

3.1 - Sandwich Composite Materials

The sandwich composite for this research was comprised of three sections. The face material was chosen to be a CF/PPS commingled fiber of 12k tow from Concordia Fibers. The core material was a Rohacell PMI foam with a thickness of 0.5". PMI foam was chosen due to higher heat resistance and strength when compared to polyurethane and polyvinyl chloride foams according to Kornienko et al. [54]. From Kornienko et al., the glutarimide ring structure of PMI is formed through heat treatment and adds to the chemical stability of the compound. The commingled fibers were processed into a composite panel of 1-1.5 mm thickness to serve as the face material. They were then bonded to the foam core with the use of an epoxy adhesive from Huntsman, Araldite 1568/Aradur 3492.

3.2 - Processing of Sandwich Composites

In order to create a composite panel from the commingled fibers, a filament winding machine was used to ensure uniform tension throughout the layers as well as a unidirectional fiber direction. The winding process used tensioners and pulleys to keep tension in the CF/PPS spooled material while it was wound onto a steel plate of ¼ inch thickness. The dimension of the steel plate was 11.5"x11.5". The chosen width for the composite panel was 8 inches. This results in a 11.5"x8" panel.

After winding, the commingled fiber plate was consolidated using a Carver Hot Press Auto Series NE, Model #3895 4NE1000, with capability of 1000 °F and 30 tons of pressure with 12"x12" platens. Cooling is by air and water. Loctite Frekote 770-NC was used to prevent the CF/PPS from sticking to the caul plates during consolidation. Once the consolidation was complete, the consolidated CF/PPS plate was cut free from the steel plate to retrieve two panels. The two panels were then trimmed to remove the outermost material and obtain a clean plate. The trimmed plates were then bonded to the

foam core with an epoxy adhesive and the Carver Hot Press was again used to reach the 50 °C cure temperature of the adhesive for 8 hours under 0.5 ton of pressure. The result was a finished sandwich composite panel. The finished sandwich composite panel was examined with ultrasound to ensure the bonding between the face and core materials was sufficient. After this, flexural (core shear) were cut to the appropriate size of 8"x3". These samples were not able to all be taken from the same sandwich panel, so it was necessary to manufacture multiple panels in order to reach the necessary number of samples. It was determined that 3 panels for flexural were needed to obtain 6 flexural samples. For impact, it was determined that static indentation would be a suitable test method. For static indentation, a single panel was used for testing with the local test point being changed for each indentation test. The area of contact is so small, that the same panel should be enough surface area to perform multiple tests. Appendix A.1 details preliminary failures during the sandwich construction process.

3.3 - Filament Winding Process

The filament winding process used an Entec model 5K30-180-4-1 filament winder with Wimax-2 control system capable of 30-inch max diameter and 4 axes of operation. The software setup for the winding process was setup to have the following criteria in the Table 2 to obtain the desired CF/PPS plates. The band width set the extent of each loop overlapped during the winding process. The diameter was based on the size of the steel plate used during winding; this was chosen to be 12 inches to allow for the filament winder head to have enough clearance of the rotating plate during the winding process. The part length is what determines the width of the finished panel. Head shaft length is how far from the rotating system that the winding begins on the steel plate, basically where the start of the winding process would occur. This was chosen to be 3 inches for the winding to begin around 1.5 inches into the steel plate so that upon completion of an 8-inch wind, there would be 1.5 inches left on both sides of the steel plate. This ensured that the fibers were centered on the steel plate. The tension was chosen to be 15 lbs. with a voltage of 27 V to provide a consistent wind. The winding process can be seen in Figure 40.

Table 2 - Filament winding settings

Setting	Value
Band width (inch)	0.1
Diameter (inch)	12
Part Length (inch)	8
Head Shaft Length (inch)	3
Winding Classification	Hoop
Number of Plies	6
Tension (lb.)	15
Voltage (V)	27

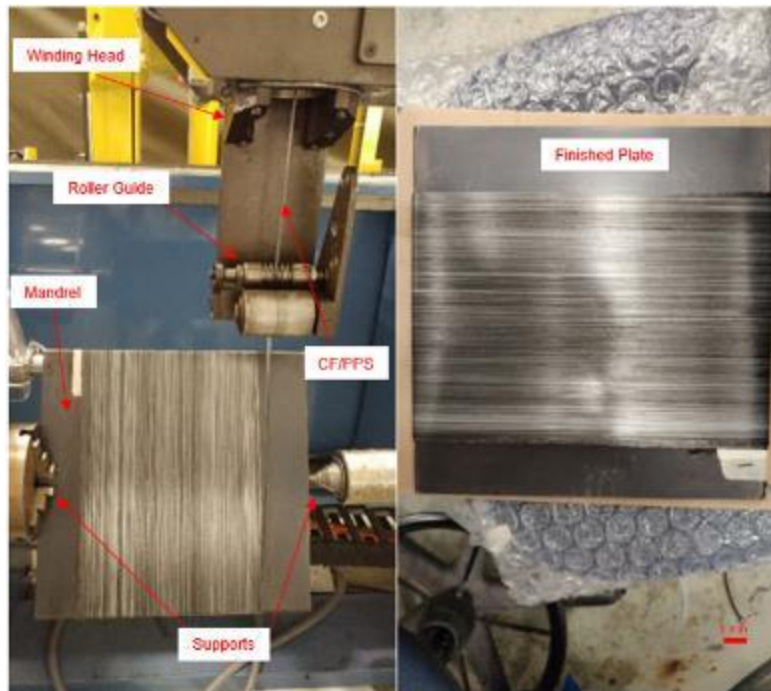


Figure 40 - Filament winding of CF/PPS panels

3.4 - Compression Molding Process

The compression molding process took place in two stages for the manufacturing of the sandwich composite structure. Firstly, the CF/PPS commingled fibers needed to be consolidated into a solid panel from their dry fiber form. This was done using the Carver Hot Press as mentioned earlier. The settings used to achieve the proper consolidation of the CF/PPS panel are outlined in Table 3. This process for consolidation applied pressure in stages and had set dwell times to ensure that the CF/PPS material was fully saturated with the temperature required for flow. The melting point of PPS is around 280 °C. This was used to determine the temperature stages for each segment in order to have the material held under pressure at the necessary state to allow the PPS in the commingled fibers to flow and consolidated around the CF. The PPS acts as the matrix in this system with the CF as the reinforcement. Since this is a thermoplastic polymer, PPS can be reprocessed if needed. For instance, if the consolidation after the first attempt of hot compression was poor, it is possible to do a 2nd round of hot compression without damaging the material. The consolidation of the CF/PPS panels can be seen in Figure 41. After consolidation in this first compression cycle, the CF/PPS is removed from the steel with an oscillating saw. The removed CF/PPS panels can be seen in Figure 42.

Table 3 - Carver hot press recipe for CF/PPS

Segment	Pressure (ton)	Time (minutes)	Temperature (°C)
1	2	10	260
2	3	10	280
3	4	15	300
4	4	1	36



Figure 41 - Consolidation of CF-PPS through carver hot press

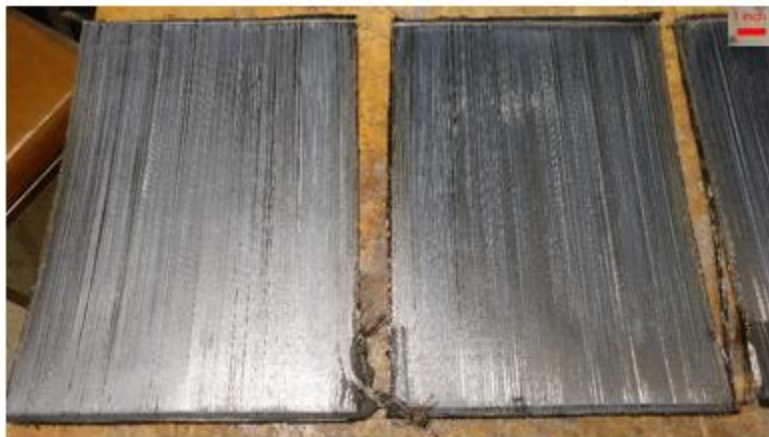


Figure 42 - After removing panels from steel mandrel

A tile saw was then used to trim the two CF/PPS plates to an appropriate size of 10"x8" for the final sandwich construction press. The CF/PPS plates were measured for thickness and found to be of average thickness of 1.18 mm. This matches well with the chosen thickness ratio of 10:1 for the core to face (12.2:1.18 actual).

The second hot compression process for the sandwich compression was to bond the CF/PPS to the foam core. This was done with an epoxy adhesive – Huntsman Aradur 1568/Araldite 3492 at a ratio of 100g resin to 28g hardener plus a glass spacing bead (9.8 mil diameter – McMaster-Carr) at 0.5% by wt. to provide a uniform bond line. In order to determine the appropriate conditions for bonding the CF/PPS to the foam core, trials were done on smaller 3"x3" samples to test the bonding. The foam is graded for use up to 130 °C (266 °F) and 44 psi. For the first trial, a temperature of 82 °C was used for an 8-hour cure time under 0.5 ton of pressure. This resulted in a compression of the foam from 12 mm to roughly 7 mm and was not ideal. It was assumed that this compression was due to the conditions being too much for the foam, the elevated temperature combined with a pressure higher than 44 psi resulted in compression of the foam. The next trial was done at the same pressure, but the temperature was lowered to room temperature (24 °C) for 12 hours. The foam did not compress, and the bond appeared to be strong between the core and face materials. Construction of the full sandwich panels was then conducted at a slightly elevated temperature in order to decrease the processing time to 8 hours. The epoxy gel time decreases as the temperature increases. It was assumed that the foam could withstand a higher temperature while keeping the pressure low at 0.5 ton since the surface area is much larger for the 10"x8" size for the actual sandwich panel. The construction of the sandwich panel was done by applying the epoxy to each face sheet and then stacking them with the foam in the center. Figure 43 shows the starting point for epoxy application. The bonding was done at 50 °C for 8 hours and 0.5 ton of pressure in the carver hot press as seen in the Table 4.

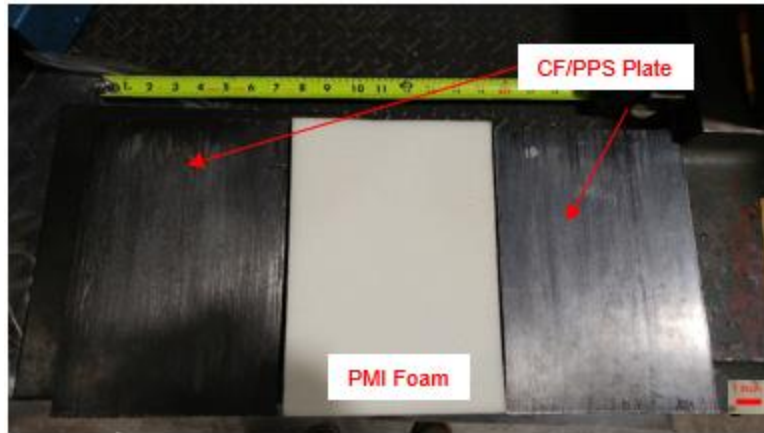


Figure 43 - Sandwich construction

Table 4 - Carver hot press recipe for sandwich composite

Segment	Pressure (ton)	Time (minutes)	Temperature (°C)
1	0.5	480	50
2	0.5	15	26

The results of the first sandwich panel construction appeared to show a strong bond between the face and core. Figure 44 shows the finished sandwich construction. The remaining 5 panels were processed under the same conditions. The 3"x3" sample trial from before was cut using a tile saw to observe possible delamination from the cutting process, none were observed. To provide enough time for the epoxy to fully cure, all panels were kept for several days at room temperature before cutting the samples. The finished composite panels were then dimensioned and cut using the tile saw as shown in Figure 45 and Figure 46. The tile saw used was a Kobalt 10-inch wet tabletop sliding table tile saw Model #KWS S10-06 with a 1/8-inch-thick blade.

3.5 - Ultrasonic Inspection/Nondestructive Evaluation (NDE)

To examine the bond between the face and core materials, ultrasound NDE was used. A sandwich panel before any testing and a core shear failure sample were examined to compare the interface before and after bonding. The equipment used was an Olympus Omni-scan SX Ultrasound unit that does traditional and phased array. The transducer used was a phased array 5 MHz transducer, linear array with 64 elements with contact wedge to give normal incidence. To produce the C-scans, an encoder was used with 12 steps per millimeter. Each sample was scanned in several passes to create an overall image of the interface. Higher amplitude readings were shown in red with lower readings in blue.



Figure 44 - Finished sandwich after bonding



Figure 45 - Tile saw for sample cutting



Figure 46 - Samples to be cut from sandwich panels

3.6 – Thermogravimetric Analysis

Thermogravimetric analysis (TGA) was performed on the CF/PPS to determine the volume fraction of CF in the face sheet. Nine (9) samples were tested from different locations within the CF/PPS panel to obtain an accurate volume fraction for the face sheet. TGA was also performed on the dry commingled fiber to compare to the finished panel. TGA is a test method in which the mass of the sample is measured over time as the temperature is increased to a chosen temperature. The sample is placed within a sample pan inside the furnace of the TGA machine and then the temperature is increased at a constant rate in inert atmosphere (Nitrogen). The data is compiled into a plot of % of initial mass on the y-axis and temperature on the x-axis. The TGA used in this work is a Mettler Toledo model # TGA/SDTA 851 and a TA Instruments TGA Q50.

3.7 - Core Shear Testing

Samples were cut from the sandwich composite panel in accordance with ASTM C393 [55]. The size of the samples followed the standard geometry, which calls for 8"x3" samples for flexural tests. Two flexural samples were cut from each sandwich panel. The flexural testing was performed on an MTS 810 Material Test System with a 100 kN load cell. A rubber protection material of 70A hardness was used between the fixture heads and the sandwich panel to prevent surface failure during testing. This was in order to ensure that the failure would be within the material and not the fixtures causing the face material to collapse. The testing rate was 6 mm/minute with a span length set for 6 inches. Figure 47 and Figure 48 show the samples before testing and the flexural core shear testing setup. The core shear stress and facing stress for the sandwich construction were calculated based on the test results. These values were then compared to literature and the expected values based on rule of mixtures. Microscopy was done on the tested samples to determine the failure modes.

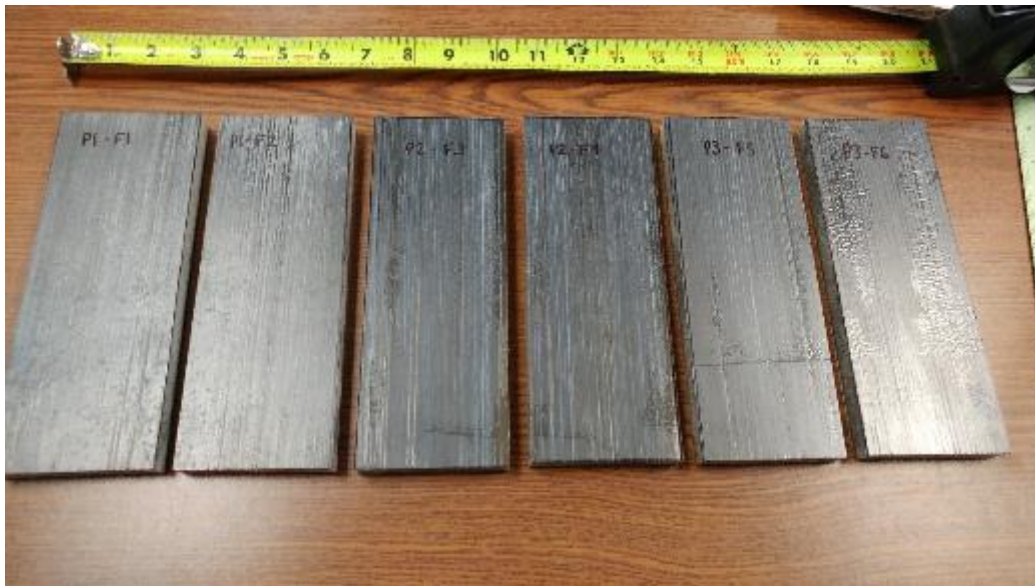


Figure 47 - Core shear samples before testing



Figure 48 - Core shear testing setup

Core shear testing was chosen due to shear in the structure being the weakest link in sandwich design. For instance, where geometries change in sandwich application. It is necessary to understand when materials fail in core shear as well as to understand the mechanisms of failure. Most sandwich structures are designed for flexural loading conditions, which correlates to a core shear test under 3-point bending. Limits to manufacturing capability also played a role in choosing core shear, due to the max composite face sheet size of 11"x8" as the limit for sandwich dimensions.

3.8 - Static Indentation Testing

Static indentation was used in lieu of the standard drop weight impact testing following ASTM D6264 [56]. Shengqing et al. found that when examining the failure mode maps of low-velocity impact (drop weight) tests and quasi-static indentation tests, the modes are similar [57]. Allen et al. compared quasi-static indentation and low-velocity impact on composite pressure vessels [58]. They found that the results show a force displacement response that follow a similar pattern within 10% for all tested cases. Both test modes apply a force to the surface of the sandwich panel and damage is initiated through the face sheet with cracking and extends to the core through crushing. The quasi-static indentation test was done by pressing a $\frac{3}{4}$ inch steel ball into the surface of the sandwich supported by a rigid steel plate at a set rate of 1.25 mm/min according to ASTM D7766 [59]. This rate was chosen for the foam core due to low compression strength. The testing will be terminated before penetration of the back face of the sandwich. The sandwich is will be rigidly backed by a $\frac{3}{4}$ " steel plate. A full sandwich panel was tested with local points on the sandwich being tested under static indentation at both room temperature and elevated temperature. The elevated temperature testing was done by heating the sandwich panel in an oven up to 100 °C and allowing the panel to equilibrate at this temperature for 4 hours. Subsequent tests for elevated temperature were equilibrated for 15-30 minutes at 100 °C due to time constraints. The chosen temperature was determined by taking 80% of the suggested operating temperature given by the manufacturer (130 °C). During testing, a portable heat gun was used to maintain the elevated temperature on the sample until testing was completed. The heat gun kept a

constant source of heat on the testing area. After the panel cooled down to room temperature, it was then tested to examine the effect of the thermal cycle on energy absorption. This test of the panel after the room temperature cool down was compared to the previous room temperature testing. The max energy as well as the absorbed energy were calculated using Simpson's 3-point rule for integration following ASTM D6264. Performance of the sandwich was evaluated between the two conditions. Microscopy was done after testing to examine the damaged areas.

The setup for static indentation can be seen in Figure 49. The panel was supported by the 3/4" thick N18 cold rolled steel plate (12"x12") during testing. The 3/4" diameter steel ball was used to indent the surface under the loading rate of 1.25 mm/minute. The panel was shifted after each indentation test to a new location for subsequent testing until 5 sample locations had been tested. This was done due to a limited number of sandwich panels that were manufactured during this thesis. Immediately following testing, the indent depth was measured by creating a peripheral area measurement around the damaged area and then measuring the max indent depth in the center. Elevated temperature setup is shown in Figure 50. Thermocouples were used in 4 locations to monitor temperature during heating/cooling and the first test of static indentation with the heat gun. Additional images of static indentation methodology can be seen in Figure 96 and Figure 97 in Appendix A.4. A summary of the steps for the high temperature testing can be seen below:

1. Heat the sandwich in the oven to temperature of 100 °C, let soak at temperature for 15-30 minutes
2. Transfer sandwich to testing frame for static indentation
3. Apply heat to the testing location using the handheld heat gun to maintain temperature in a range of 85-105 °C
4. Continue heat application until the static indentation testing is completed

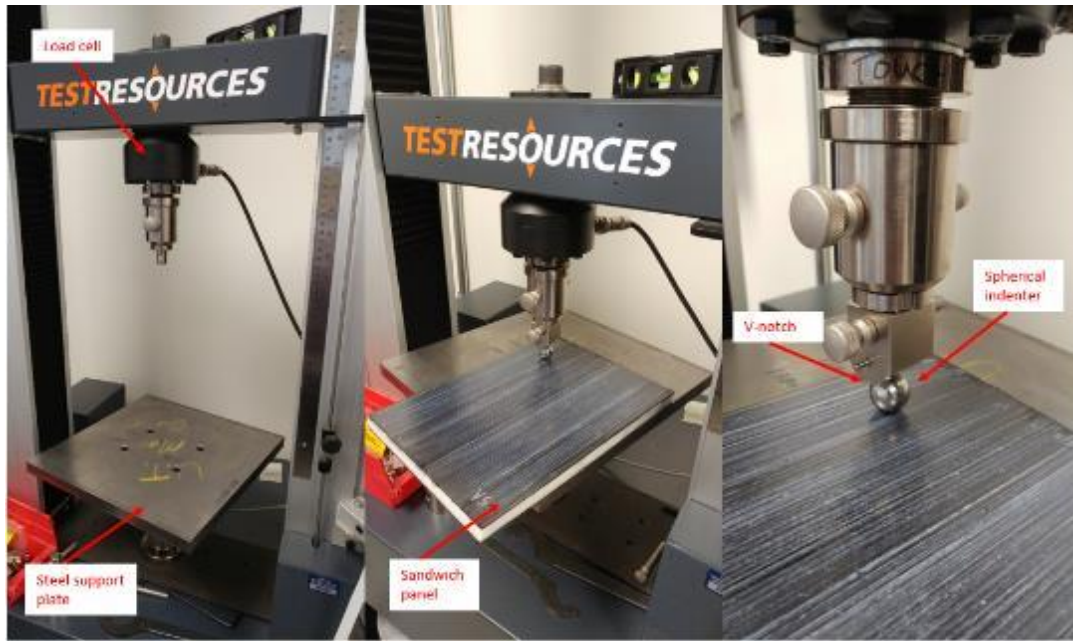


Figure 49 - Static indentation test setup

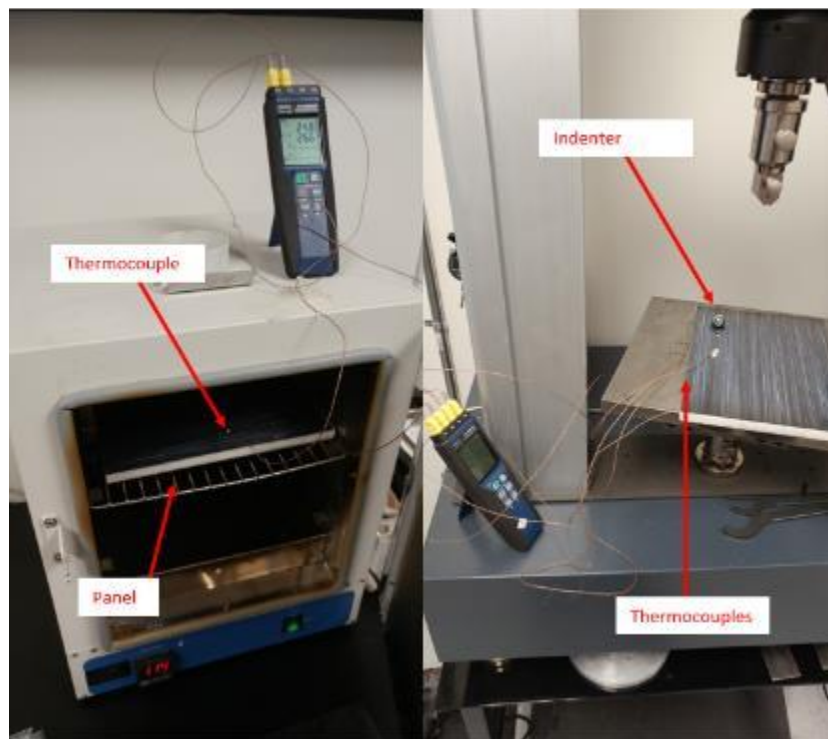


Figure 50 - Elevated temperature setup for static indentation

3.9 - Optical Microscopy

In order to evaluate the failure mechanisms of the sandwich panel during flexural testing, optical microscopy was done of the failed samples. The CF/PPS and core material were then examined to determine which mode of failure was present for each sample. The microscopy was done using a Dino-lite edge digital microscope, AM4815ZT, with magnification in the range of 20x to 150x. Microscopy was done on the core shear samples, the CF/PPS flexural and ILSS samples, and the static indentation samples after testing to examine the failures. The setup for optical microscopy can be seen in Figure 51. The microscope is supported by a stand and positioned above or to the side of the sample for imaging. The microscope software was used to improve image quality and alter settings in order to produce sufficient images for inspecting failures.



Figure 51 - Optical microscope setup

3.10 - Finite Element Analysis (FEA)

Finite element analysis using AutoCAD Inventor Nastran was performed to simulate the stress experienced by the sandwich during the 3-point bending test. Simulations were done for the full sandwich as well as just the foam core. These were then compared to experimental data.

3.11 - Characterization of CF/PPS

The face sheet for the sandwich construction was characterized for flexural, interlaminar shear and impact properties. After core shear testing, one of the face sheets from sample #1 was removed from the foam. The CF/PPS sheet was then sanded down using a belt sander to remove the epoxy and foam from the surface. Then it was cut into flexural samples for testing. Eight samples were tested for flexural strength and the flexural modulus. Microscopy was done on the samples to examine the failure mode.

In order to characterize the remaining properties of CF/PPS, a second panel was wound using the filament winding process as described earlier. Instead of using 6 plies, 14 plies were wound in a goal to create a 3.5 mm thick commingled fiber composite. The plate was consolidated following the same recipe as with the face sheet processing. A second consolidation was done on one of the resulting plates to further consolidate the fibers. After the 2nd consolidation, the plate incurred some residual stress build up. These plates were then cut using the tile saw into flexural, ILSS and Izod (impact) samples following ASTM guidelines. ASTM D790 was used for flexural testing [60]. ASTM D2344 was used for ILSS short beam strength composites [61]. ASTM D256 was used for IZOD testing [62]. Testing for flexural strength and ILSS were performed with a 3-point bending setup on a Test Resource Model 313 series tensile frame. The IZOD testing was done on a Tinius Olsen Model Impact 104, Model #IT504 plastic impact machine as shown in Figure 95 of Appendix A.3. The results of these tests will give an understanding of the ultimate stresses the face material can withstand before failure. The IZOD impact results can be used to compare to the static indentation of the constructed sandwich composite.

Chapter 4 – Results and Discussion

4.1 - Flexural Core Shear Results

The constructed sandwich composite structures were tested with ASTM C393 [55]. Samples were cut to the size of 8"x3" per the guidelines. Before testing, samples were conditioned in a desiccator for 14 hours overnight at room temperature to remove moisture from the foam. Conditions of the testing apparatus were at room temperature. Testing was done at the suggested 6 mm/minute crosshead rate. The fixture for the 3-point bend testing was a 3"x1" steel bar with 70A rubber padding to prevent failure of the face prematurely. Once failure was observed, testing was stopped. The setup for testing can be seen in Figure 52. The system used for testing was an MTS 810 Material Test System with a 100 kN load cell and a 3-point bend fixture for sandwich testing. The samples for this testing were cut from 3 total sandwich composite panels, with 2 samples coming from each panel. The dimensions of each sample were measured and recorded before testing with 3 measurements taken for each and a corresponding average calculated for length, width, and thickness. The sandwich thickness was measured with a Mituyo micrometer C/N 293-340-30 before testing. The sandwich width was measured with a Mituyo caliper CD-6" ASX. The corresponding dimensions can be seen in Table 5.

Results of the flexural testing of the sandwich beams all showed failure within the core. Some of the tested samples did experience delamination of the face to core, but this is most likely due to the shear force incurred by the core material. Therefore, the mode of failure for the sandwich beams was core shear. The load applied to the samples at the constant 6 mm/minute rate as per the ASTM C393 standard for core shear testing. The average and standard deviation for the max load, core shear ultimate stress and facing stress were calculated for all samples and are summarized in Table 6.

Table 5 - Core shear sample dimensions

Panel	Sample	Length (mm)	Width (mm)	Thickness (mm)
1	1	201.7	75.70	15.085
1	2	201.8	75.50	15.063
2	3	203.1	75.66	15.087
2	4	203.0	76.44	15.091
3	5	204.4	75.88	14.893
3	6	204.2	76.20	14.858

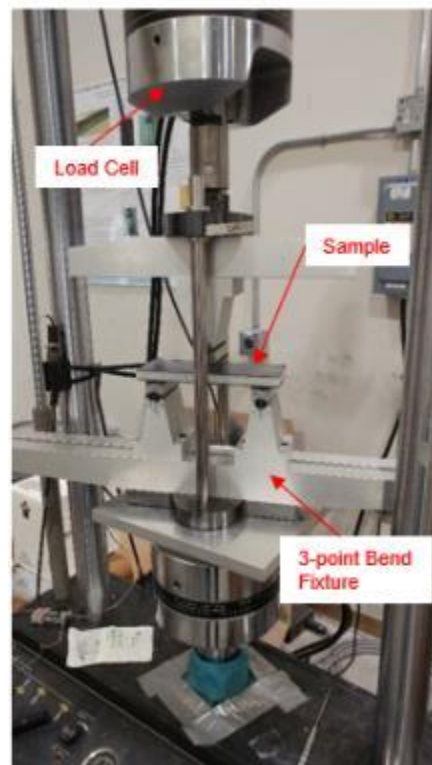


Figure 52 - Core shear testing setup

Table 6 - Core shear stress CF-PPS/PMI sandwich

Sample	Max Load (N)	Core Shear Ultimate Stress (MPa)	Facing Stress (MPa)
1	3745.86	1.778	113.32
2	3551.75	1.694	107.96
3	3410.85	1.619	103.18
4	2583.46	1.215	77.44
5	3193.97	1.533	97.70
6	2613.66	1.250	79.63
Average	3183.26	1.515	96.54
Stdev	487.53	0.234	14.89
CV	15%	15%	15%

The core shear stress and facing stress were calculated following the equations below, taken from ASTM C393 [55]. The equation used to calculate core shear ultimate stress can be seen in Equations 4 and 5:

$$F_s^{ult} = \frac{P_{max}}{(d+c)b} \quad (\text{Eq. 4})$$

Where P_{max} is the maximum force prior to failure, d is the sandwich thickness, c is the core thickness, and b is the sandwich width. The equation used to calculate facing stress is below:

$$\sigma = \frac{P_{max}S}{2t(d+c)b} \quad (\text{Eq. 5})$$

where t is the facing thickness and S is the span length. The facing thickness used in calculations was an average thickness taken from the measurement of all the face sheets, this value was 1.177 mm. The span length was set at 6 inches according to the ASTM C393 standard. The results can be seen in Figure 53 for the load vs displacement curves of each sample. The load displacement curves were plotted up until the max load failure occurred where testing was stopped. The average displacement until failure occurred between 4 to 5 mm where the total thickness of the sandwich was around 15 mm.

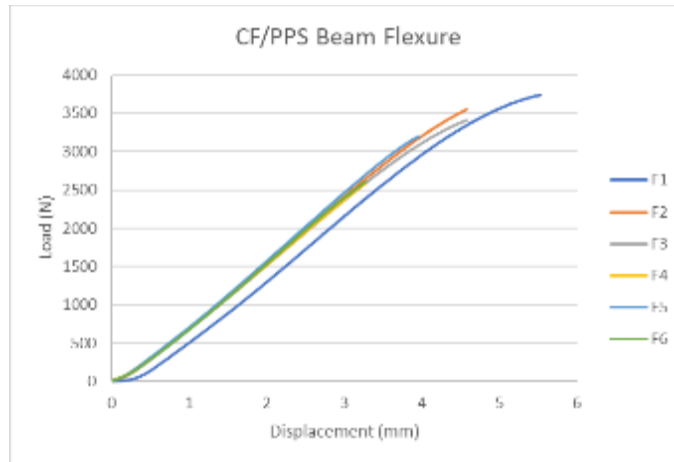


Figure 53 - Load vs displacement for CF-PPS beams

Core shear stress and facing stress values were determined for other loading conditions to determine a representative model for this sandwich construction. The application of this sandwich would be in a scenario where the max load would not be reached in order to prevent failure of the material. In a likely design situation, a factor of safety of 2 would most likely be used, resulting in an expected load of 1500 N based on the experimental results of the core shear testing. From this load, the core stress for the sandwich structure was found to be 0.734 MPa with a facing stress of 47.53 MPa. These are roughly 50% of the experimental results for an average 3183 N load at failure. This is as expected since there is a linear relationship between the load and the stress of the core/face materials as seen in the load vs displacement curves as well as Equations 4 and 5. All other values in the equations are constant for the chosen sandwich structure, with only the load changing. Figure 54 shows this relationship with the chosen loads and the load giving a safety factor of 2 as the red square and the experimental load results shown by the green diamond. According to the manufacturer of the Rohacell 71 IG-F foam, the shear strength of the foam is 1.3 MPa [63]. The average core shear stress from experimental testing was 16% higher than this value. This difference is not significant when compared to the experimental results because the CV of the experimental testing was 15%. There were 2 observed samples that experienced a failure at 1.2 MPa, which is below the manufacturers stated value. This shows that the results of the core shear testing are as expected and give data that matches previously documented core shear stress values for the chosen foam core.

The average density of the tested core shear samples was found to be 331 kg/m³. This was determined by weighing each core shear sample and dividing the weight by the volume of each sample. The strength to weight ratio for the core stress can be calculated by dividing the core stress by the weight of the sandwich structure. By doing this, a strength to weight ratio of 19.9 MPa/kg was obtained for the sandwich structure. This is largely affected by the tested sandwich design. The chosen design was for a thin sandwich to ease manufacturing as well as cost for this study. To improve this strength to weight ratio, a different face to core thickness ratio could be chosen to result in less stiffness of the sandwich construction.

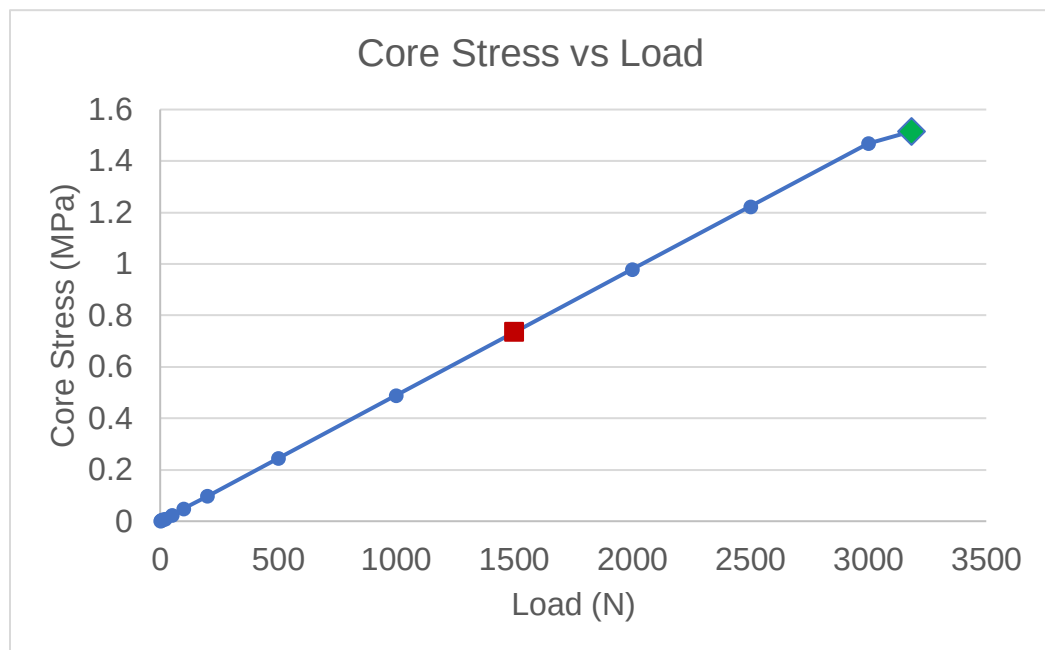


Figure 54 - Core stress vs load. Red data point is factor of safety of 2. Green data point is experimental results.

4.2 - Mechanisms of Failure – Core Shear

The failure mechanisms observed in this study for the 3-point bending testing were all core shear failures. Figure 55 shows the failure for each of the tested samples. Core shear resulted in the stoppage of the test. There was also some observed separation of the face sheet from the core which appeared to be a delamination between the face sheet and the core material, but this is due to the shear forces in the core during failure that caused the delamination. The damage to the core cells was the primary failure. For example, this can be seen in delamination in sample number 3 (see Figure 55), the bottom face has separated from the core. The core shear failures are occurring between the loading and the bottom supports, on either side of the loading nose with 3 failing on the left side of the nose and 3 failing on the right side of the nose. The resulting failures show that the construction of the sandwich composites was properly done to ensure an acceptable failure mode. The core failed before the face of the material which was expected. This is due to the face having better mechanical properties when compared to the core. The design of the sandwich using a core to face thickness ratio of 10:1 ensured that the core would fail during testing.

4.3 - Microscopy Imaging Results

Microscopy was performed on two of the core shear samples to be a representation of the tested population of samples. Imaging was done at 20x and 50x magnification for each sample to examine the crack formation and failure modes. It was observed during testing that each sample failed in the core due to shear, which resulted in separation of the face sheet from the core. This can be seen in Figure 56. A crack propagation begins at the face and continues through the thickness of the core until the core fails in shear. It was determined that the most likely origination of the crack was at the top of the panel where compression was applied by the 3-point bend test. The crack then propagated downwards at an angle towards the bottom face. Delamination can be seen on the bottom face with pieces of the foam still sticking to the CF/PPS face sheet. This shows that the failure was within the core and further confirms core shear failure as

the mode of failure. The adhesive was strong enough to withstand the shear forces and the core failed before the adhesive. The foam cells do not all fail at the same time. This may be attributed to local stresses in the core and the small amount of variability within the core cells and the adhesive variation within the contact zone. This phenomenon can be seen in Figure 57 as shown by the arrows. Between Figure 56 and Figure 57, the foam fails in different areas in the proximity of the CF/PPS face sheet. The microscopy on sample #1 is shown in Figure 58. The mode of failure is very similar to sample #2 as shown in Figure 56. The angle of failure is about the same between the two samples. Sample #1 has a more noticeable delamination from the top face as well as a straighter crack formation. Sample #2 had a more jagged crack. The reason for the crack propagation occurring in a 45° angle from the face is due to that being the path of least resistance through the core material.

4.4 - FEA Results

FEA was performed on the sandwich structure to model the performance of the material under the 3-point bend conditions experienced during core shear testing. Fixtures were set to alloy steel with the beam set as a PMI foam using the material properties provided by the manufacturer [63]. The analysis was performed with a loading scheme setup to test to a deflection of 4.6 mm, which was based on the average displacement during testing for core shear. The results showed that the foam experienced a max stress of 1.582 MPa for von mises with a max displacement of 4.953 mm. This is close the experimental results of a core shear stress of 1.515 MPa and a displacement of 4.6 mm. The model was done in AutoCAD Inventor Nastran, an add-on FEA tool to Inventor. There are limitations within the software that made it difficult to generate a proper sandwich construction for modeling. The results for the foam core can be seen in Figure 59 and Figure 60.

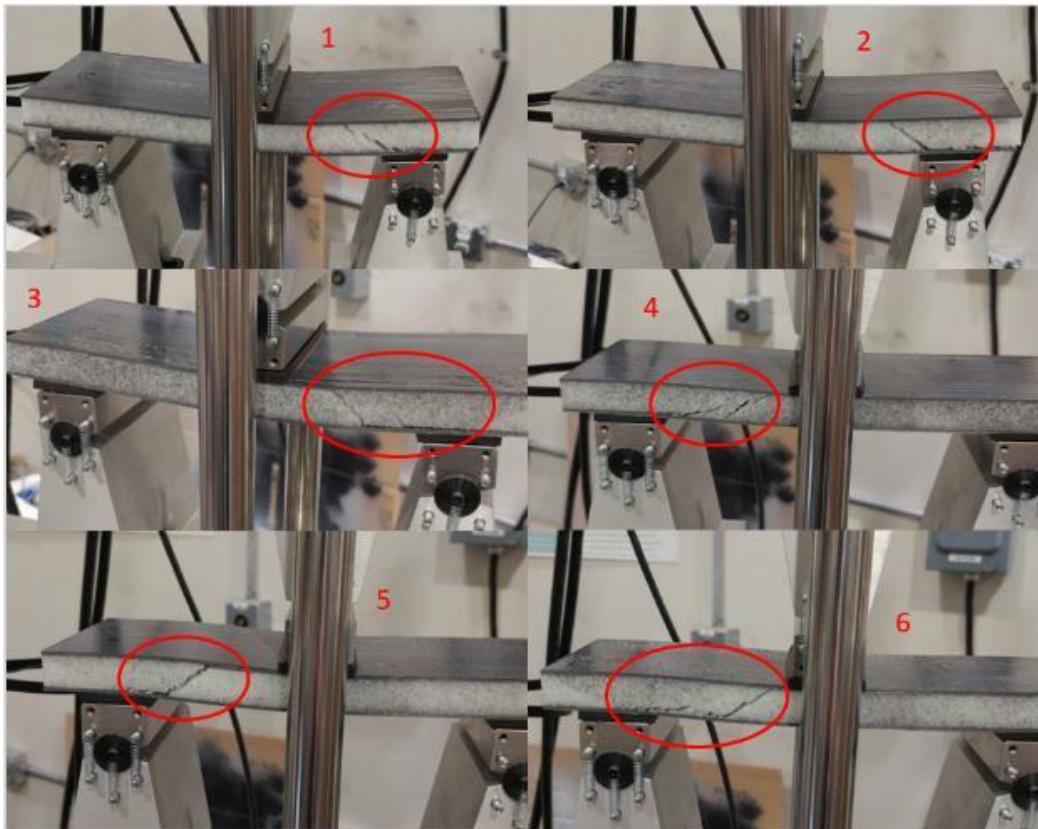


Figure 55 - Core shear failure for samples 1 through 6

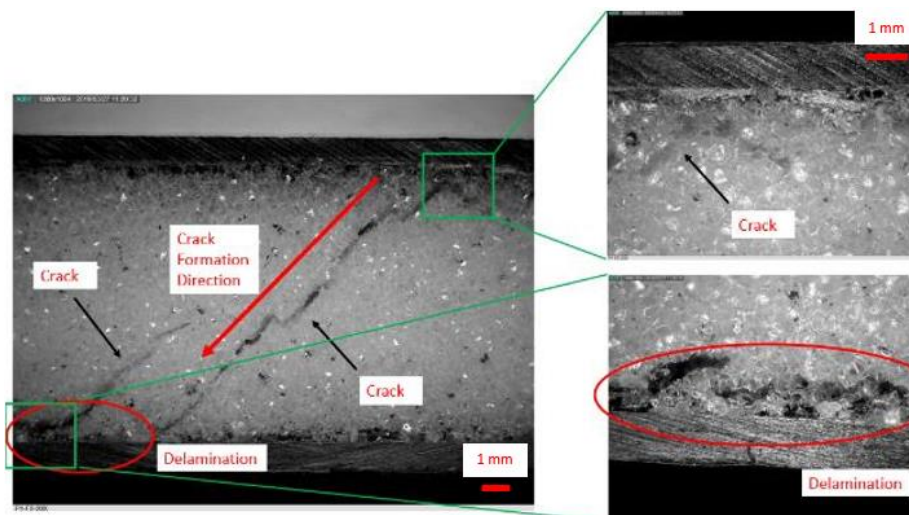


Figure 56 - Microscopy of core shear sample #2

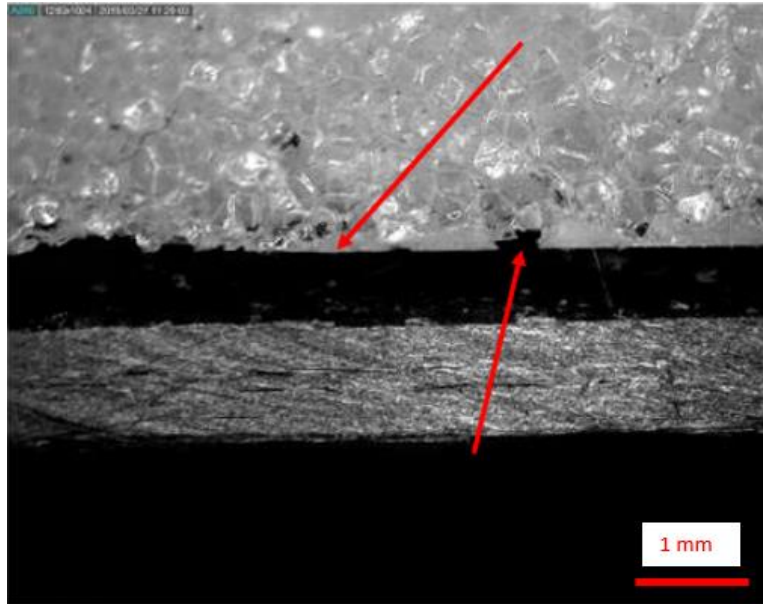


Figure 57 - Uneven stick on PMI core face

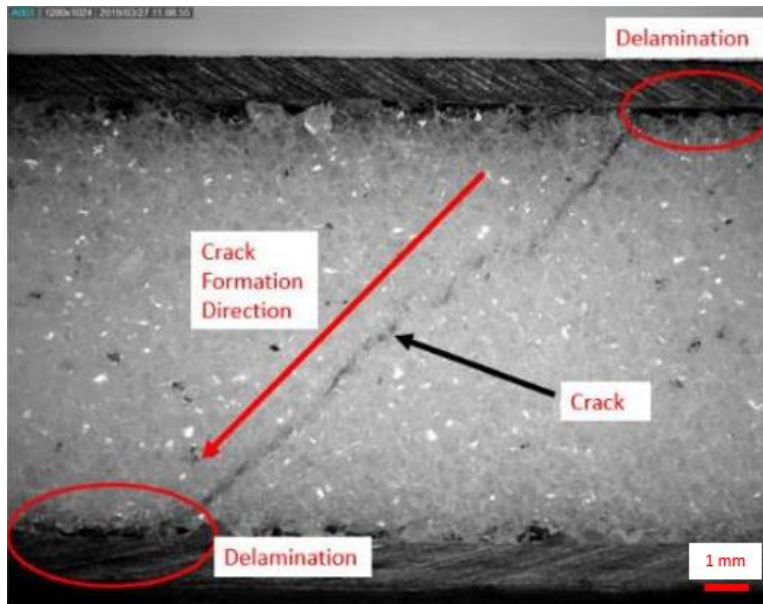


Figure 58 - Microscopy of core shear sample #1

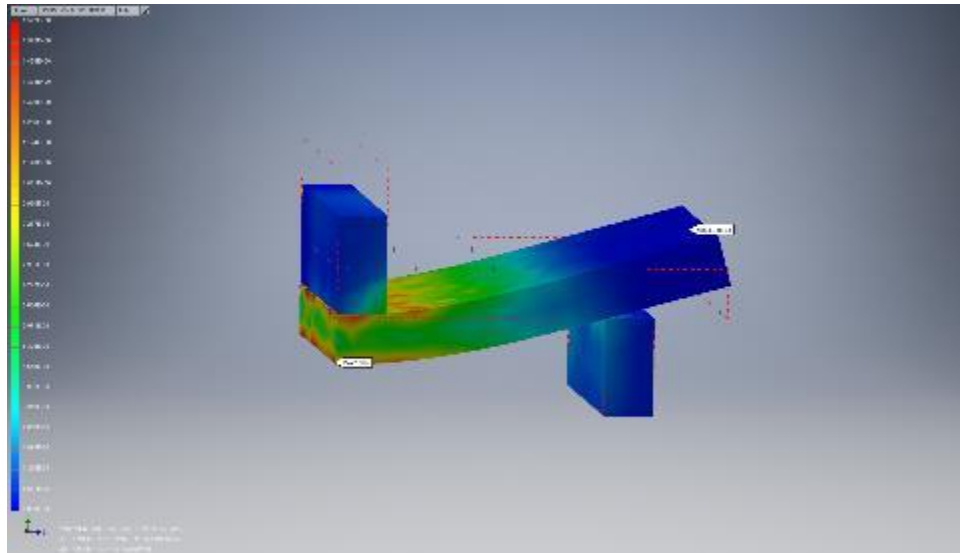


Figure 59 - Von mises stress for foam core model of 3-point bending

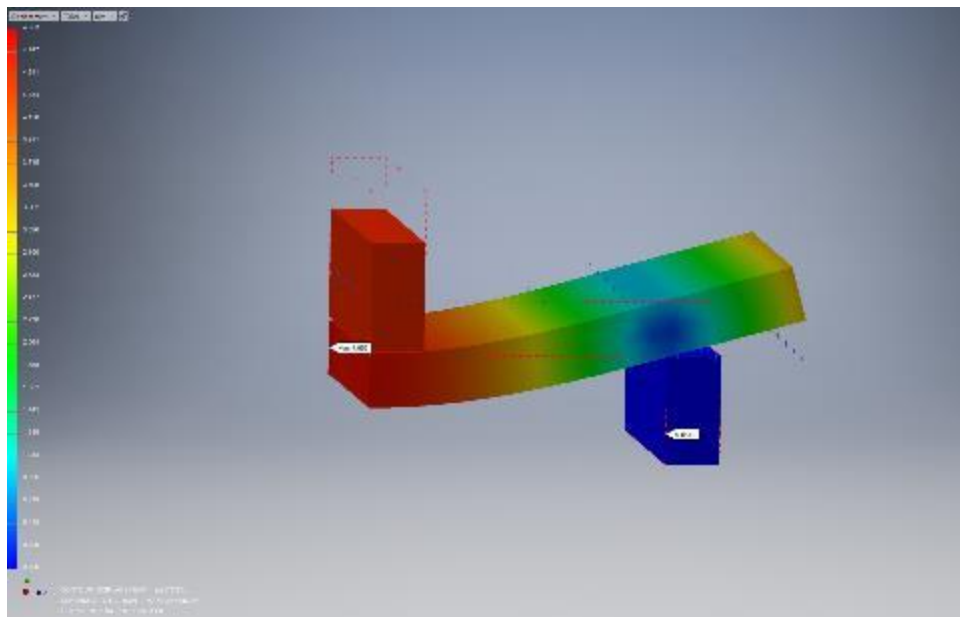


Figure 60 - Displacement for foam core model of 3-point bending

In replacement of the FEA for the sandwich structure, a numerical model was analyzed for the core shear stress for different core thicknesses of sandwich construction using MATLAB. The model was based on the proposed sandwich design weight minimization by Triantafillou et al. [64]. They examined the optimization of a sandwich design based on the desired failure criteria. The main failure modes examined were yielding and wrinkling of the face and yielding of the core in shear. For this thesis research, core shear was targeted failure mechanism for the sandwich design. From this a core to face thickness ratio of 10:1 was chosen to ensure the core shear failure. A model can be developed for the core shear failure using the following Equation 6 from Triantafillou et al. [64]:

$$q = \frac{C_4 \left(\frac{\rho_c}{\rho_s} \right)^B * \sigma_{ys}}{0.33} * \frac{c}{l} \quad (\text{Eq. 6})$$

where q is the load, C is a constant, ρ_c is the density of the core, ρ_s is the density of the solid unfoamed core material, B is the exponent for the foam shear strength density relationship, σ_{ys} is the yield strength, c is the thickness of the foam core, and l is the length of the beam. B was given as $3/2$, densities were known, and the foam core thickness and length were also known. Using these values along with the experimental load and core shear stress values, the constant C was determined. The equation was then plotted with core stress on the x-axis and load on the y-axis. The plot was generated for three different thickness of the core (0.25", 0.5", and 1"). The 0.5" core was the size used in this thesis. The plots show that as the thickness of the core increases causes the load to increase for a corresponding core shear stress and conversely as the thickness decreases, the load will decrease for a corresponding core shear stress. The constant was assumed to be constant between the different core thicknesses but could be experimentally determined by performed the core shear testing for thicker sandwich structures to get more accurate models. This can be seen in Figure 61.

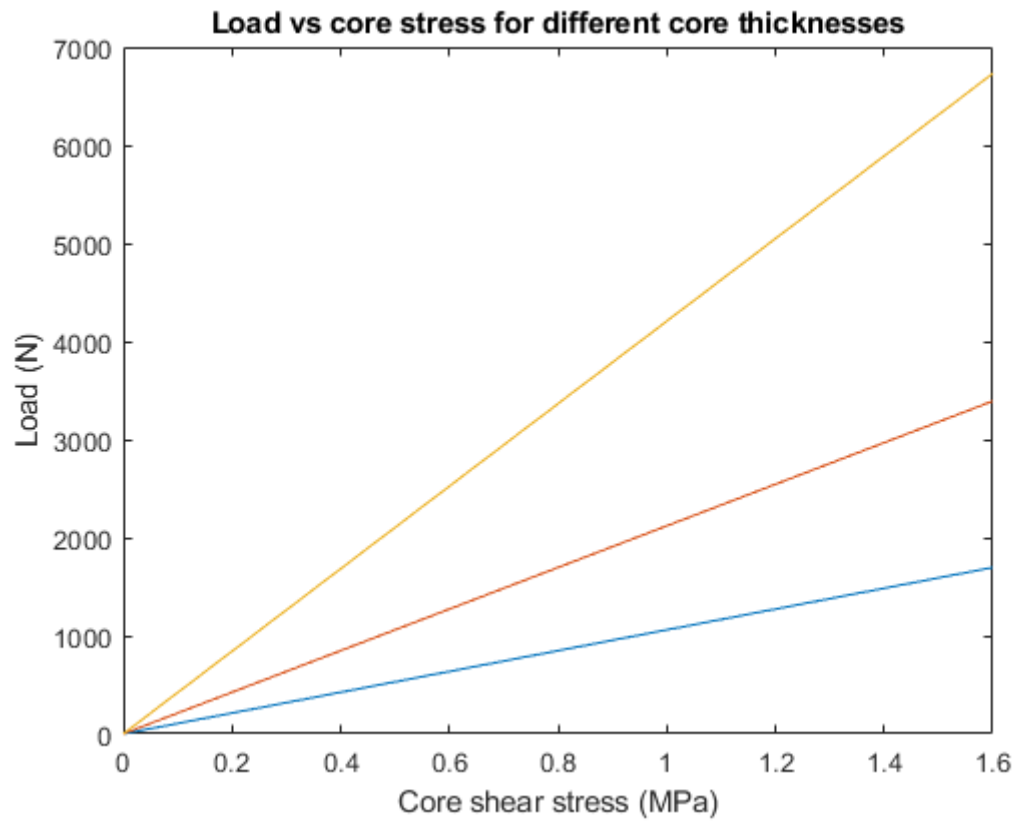


Figure 61 - Load vs core stress for 0.25\" (blue), 0.5\" (red), 1\" (yellow) core thicknesses

4.5 - Ultrasonic Inspection/Nondestructive Testing

Ultrasonic inspection was performed by phased array ultrasound (PAUT) on the sandwich panels to examine the bonding between the face and core materials as well as examined the bond after core shear testing. The settings for the ultrasound can be seen in Figure 91 in Appendix A.2. Ultrasound uses sound waves to reflect off materials with different density or stiffness, or difference between past properties and new properties resulting in a reflection back. This is explained by Snell's law. With the equipment used, there is a time measurement of the wave as it travels from the transducer to the back wall of the desired area of examination until it reaches back to the transducer. This time delay measurement is the basis on which the interface plots are constructed. Each sample tested was done in several passes over the sample surface from one edge to the other until the whole surface was examined. Each image was then stitched together to get an overall image for the sandwich panels. Panel 4 and core shear sample #1 were examined in this study.

Overall, the results of the nondestructive testing were difficult to understand. There was either good adhesion at the interface between the CF/PPS and foam core or a lack of adhesion and possible delamination. When inspecting the imaging results, one thought is that the high amplitude response in the images as seen by the Red color could represent good adhesion between the interface. This is due to the face around the edges of the panel where a low amplitude is observed. It is typical to see poorer bonding around the edges of this kind of sandwich structure as opposed to in the center of the panel. This is due to the contact force during bonding as well as the heating differential during the curing cycle. Adhesive was also applied in a manner such that there would not be overspill after the cure cycle when pressure was applied, this meant that when applying adhesive there was a gap of around 1 cm around the edge of the panel to prevent overflow. Another assumption could be that the Red amplitude could represent a discontinuity or air pocket, a concentration of adhesive, or surface variance of the foam itself. It was noted during microscopy of the core shear samples that the foam did not fail in the same locations during testing, possibly due to uneven surface finish of the foam. The results of the scans can be seen in Figure 62 and Figure 63.

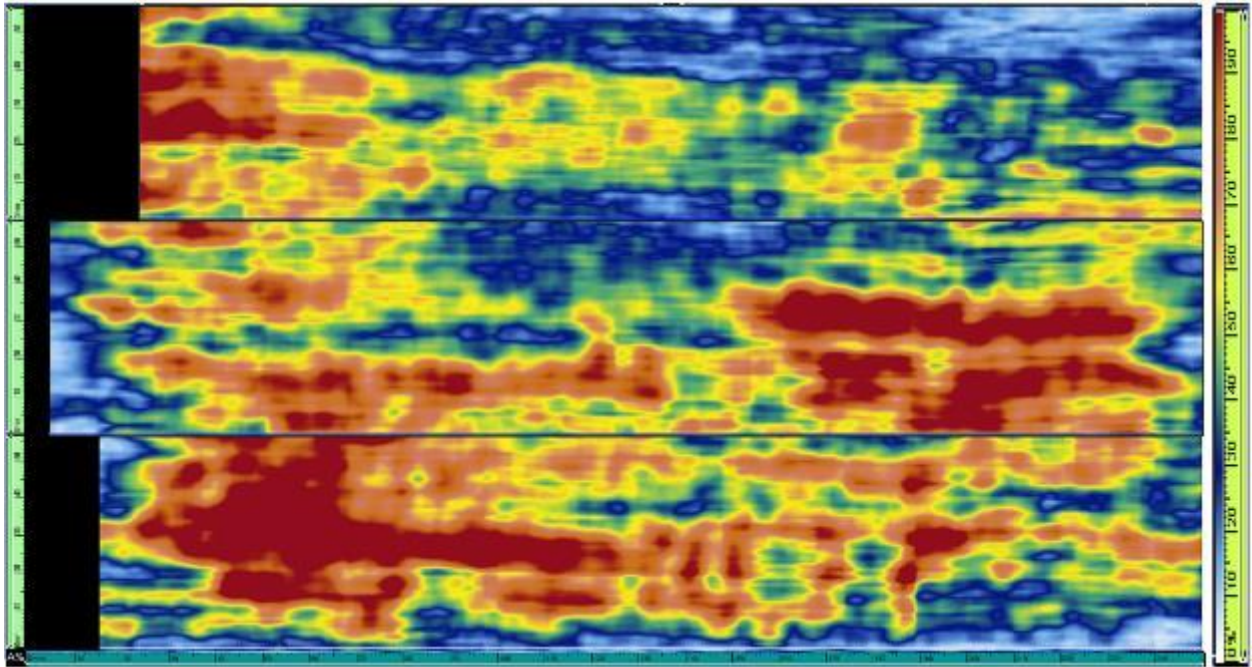


Figure 62 - Ultrasound inspection of panel 4 - untested sandwich panel

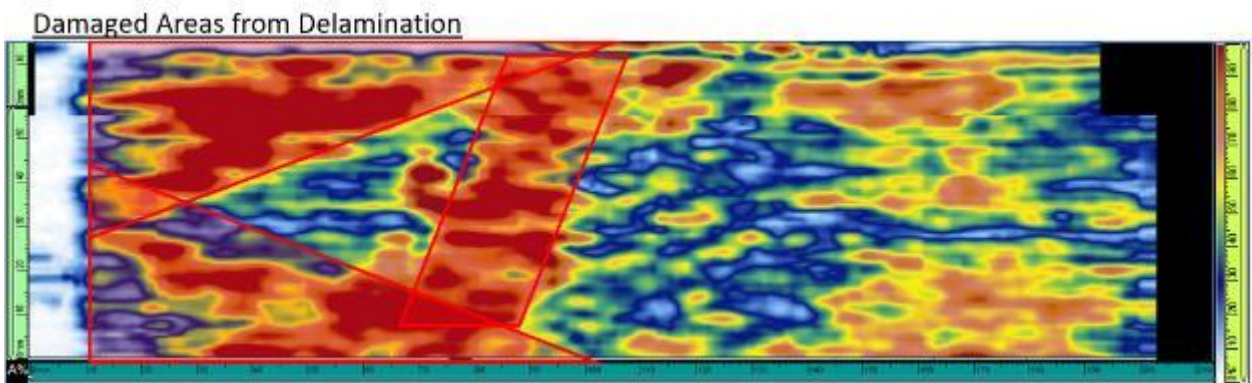
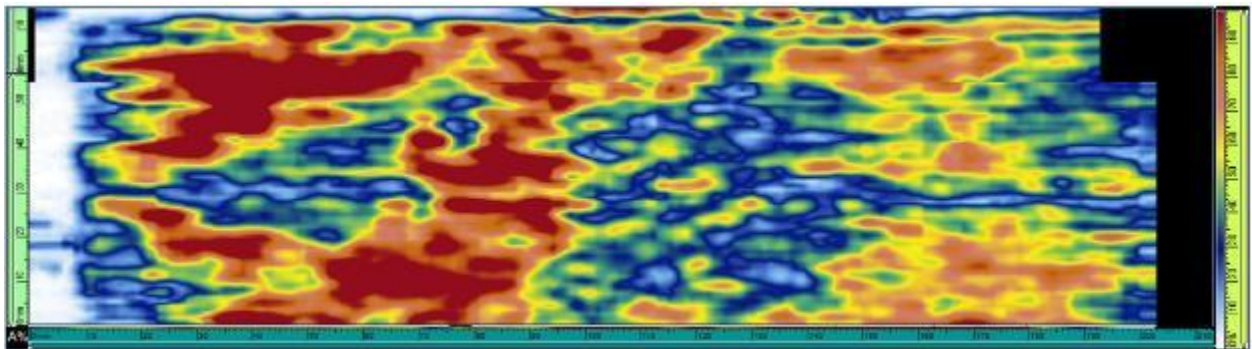


Figure 63 - Ultrasound of core shear sample #1 - delamination

The ultrasound results show potential delamination between the face and core materials for the core shear sample #1. The red highlighted areas have very high amplitude responses and would be the potential delamination. Based on visual inspection, there were delamination in these areas of the tested sample due to the core shear. It is easily noticeable in Figure 64. This could confirm the assumption that higher amplitude (red) would mean delamination/air pocket within the interface. A reason for the blue amplitude around the edge of the panel on the left of Figure 63 is due to the contact solution used during inspection. The gel/coupling could have gotten onto the side of the sample and worked its way inside the interface on the edges. This would propagate the sound farther in, causing blue instead of red for the amplitude response. In order to get a better understanding of this interface, a more precise transducer would be needed along with further testing. The current transducer only has a wavelength of 0.45 mm, which means that any anomaly under 0.45 mm is hard to detect. The bond-line thickness of this sandwich construction is less than 0.45 mm. This means that the inspection of the bond-line is qualitative and cannot be quantified at the present time of this study.



Figure 64 - Delamination due to core shear, visible inspection to compare to non-destructive evaluation

4.6 - Static Indentation Results

Static indentation was performed at room temperature and at an elevated temperature of 100 °C, which is 20% less than the stated max operating temperature of the Rohacell foam core, per manufacturer. Testing was conducted on a single sandwich panel that was 10"x8" in size for each environment. Each test was done on a different local area of the panel by simply repositioning the panel beneath the indenter after each test to an untested area. Load was applied to the sandwich at a rate of 1.25 mm/minute following the D7766 standard. The load selection was based on the core consisting of a low compression strength foam. Testing was carried out until a prescribed displacement that was determined from a dummy sample to ensure that results were comparable between each test as opposed to testing until a specific load was reached. The chosen displacement where loading would stop and unloading would begin was 7.3 mm, which is 50% of the thickness of the sandwich panel. This allowed for testing to stop before the back-side of the panel was damaged. Once this displacement was reached, unloading was done at the same rate as loading at 1.25 mm/minute. Immediately after testing, the dent periphery and max dent depth were measured using a Pittsburgh Dial Indicator, resolution 0.01 mm on an eclipse magnetic stand, with 4 mm diameter surface contact as seen in Figure 99 of Appendix A.4. This was done by measuring the thickness of the sample around the edge of the dent in 15 locations as well as measuring the depth in the center. The area of the dent was then calculated. For this study, the dents were observed to all be elliptical in shape due to the unidirectional fibers. Failure on the surface was in the direction of the fibers, since the laminate is weakest in that direction. After testing, the panel was cut using the tile saw to observe the damage in the cross-section through optical microscopy.

The results of the room temperature testing showed an average max load of 2306.9 N with an average dent depth of 4.58 mm. The dent depth had a CV of 7%, so the surface behaved consistently once the indenter was removed. The max displacement of the indenter was 7.3 mm, due to the testing setup. From this, there was an observed relaxation of the surface since there is a difference of 2.72 mm between the max displacement and the dent depth. This is due to the stiffness of the composite face. The

curves for load vs displacement were generated with the loading and unloading segments in order to calculate the Max Energy as the energy required for the indenter to reach the maximum displacement and the Absorbed Energy as the energy absorbed inelastically during the complete cycle. These calculations were based on the following Equations 7 and 8:

$$E_{max} = E(\delta_{max}) = \int_{\delta_0}^{\delta_{max}} F(\delta) d\delta \quad (\text{Eq. 7})$$

$$E_a = E(\delta_f) = \int_{\delta_0}^{\delta_f} F(\delta) d\delta \quad (\text{Eq. 8})$$

where δ_{max} is the maximum indenter displacement, δ_0 is the indenter displacement at initial specimen contact, F is the contact force at indenter displacement, δ is the indenter displacement during the test, and δ_f is the indenter displacement at the end of the unloading cycle. In order to calculate these integrals, a numerical method called Simpson's rule (3-point rule) was used. This method uses a parabolic fit to the curve to evaluate the area under the curve. The resulting integral calculation was done using Equation 9:

$$\int_0^{2h} F(\delta) d\delta = \frac{1}{3} h (F_0 + 4F_h + F_{2h}) \quad (\text{Eq. 9})$$

where h is the equally spaced interval being evaluated. For this test, the interval h was determined as the max displacement divided by 2 for the Energy Max calculation and the displacement at the end of the unloading cycle divided by 2 for the Energy Absorbed calculation. The results for the room temperature static indentation testing can be seen in Figure 65 and Table 7. The max energy was found to be 11.3 J and the absorbed energy was found to be 6.45 J for the sandwich panel.

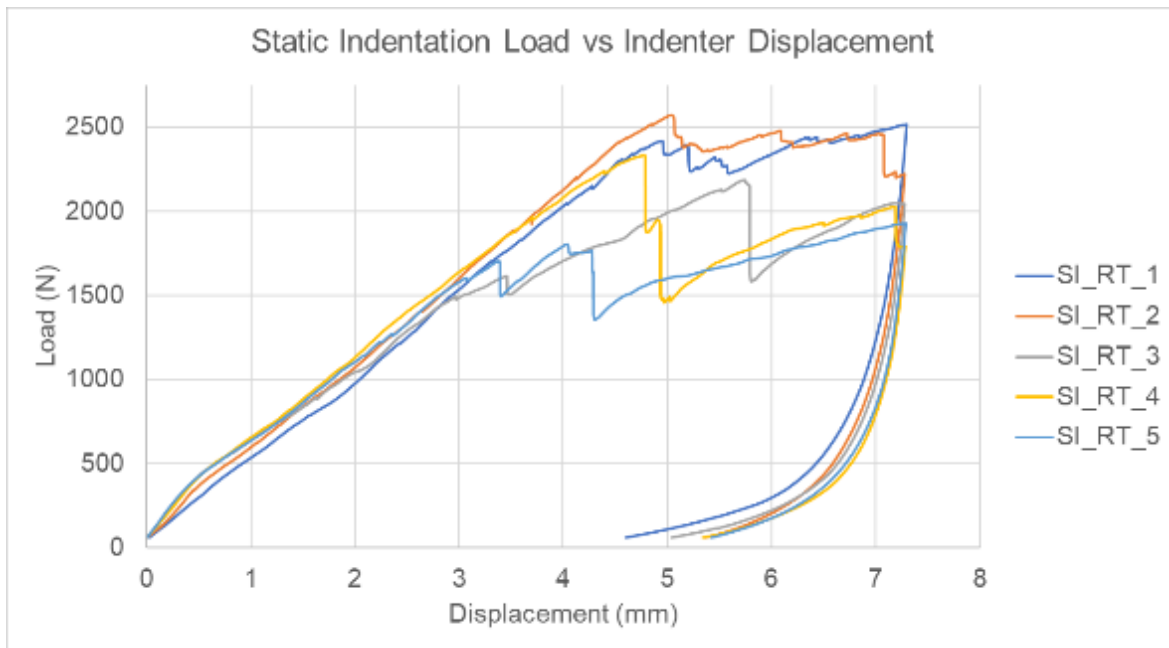


Figure 65 - Static indentation at room temperature, loading/unloading cycle

Table 7 - Static indentation summary for room temperature testing

Sample	Max Load (N)	Indent Depth (mm)	Energy max (J)	Energy Absorbed (J)
SI_RT_1	2517.1	4.77	12.16	5.34
SI_RT_2	2572.0	4.58	12.18	7.24
SI_RT_3	2186.0	4.05	10.20	6.10
SI_RT_4	2332.2	4.72	11.63	6.77
SI_RT_5	1927.1	4.79	10.32	6.79
Average	2306.9	4.58	11.30	6.45
Stdev	261.7	0.31	0.97	0.74
CV	11%	7%	9%	11%

Static indentation was also performed at an elevated temperature of 100 °C. This was done by heating the sandwich panel in an oven to 100 °C and keeping it at temperature for a 4-hour cycle. After 4 hours, the sandwich panel was removed for testing. In order to maintain temperature during testing, a Portable Cable Heat Gun 1500W was used. The heat gun was aimed at the area of testing to keep temperature elevated throughout the testing. Once the loading/unloading cycle was complete, the heat was removed, and the sandwich panel could cool down to room temperature. The panel was then tested again at room temperature to see if there was a difference between the heat gun testing and a room temperature test after the oven heat cycle. The result of the room temperature test showed a load of 2x that of the heat gun test. Due to this result, a 2nd room temperature test was done to confirm. The result of the 2nd room temperature test showed similar load of 2x the heat gun test. These values corresponded to the previous room temperature results and it was determined that a heat cycle alone where the sandwich is heated to a high temperature and then cooled back down to room temperature did not influence energy absorption. Based on this result, the remainder of the elevated temperature tests were carried out using the heat gun to maintain temperature. To achieve this, the sandwich panel was reheated in the oven up to 100 °C between each test and held at temperature in the oven for 15 minutes. A static indentation test after a 30-minute hold time and 15-minute hold time were done to determine if there was a difference between 4 hours, 30 minutes, and 15 minutes. The results observed showed that there was no difference, hence a dwell time of 15 minutes was chosen. Four total tests were conducted with the heat gun method for elevated temperature. The results showed an average load of 1379 N, average max energy of 5.63 J and average absorbed energy of 3.36 J. The results can be seen in Table 8 and Figure 66. The load displacement curves show the drastic difference between an elevated temperature testing response and a room temperature response. There is a difference in dent depth for two of the samples measured at elevated temperature, but this is due to the measurements not being done until the very end of testing after all the heat cycles. From this, there is an observed change in dent depth due to the environmental changes.

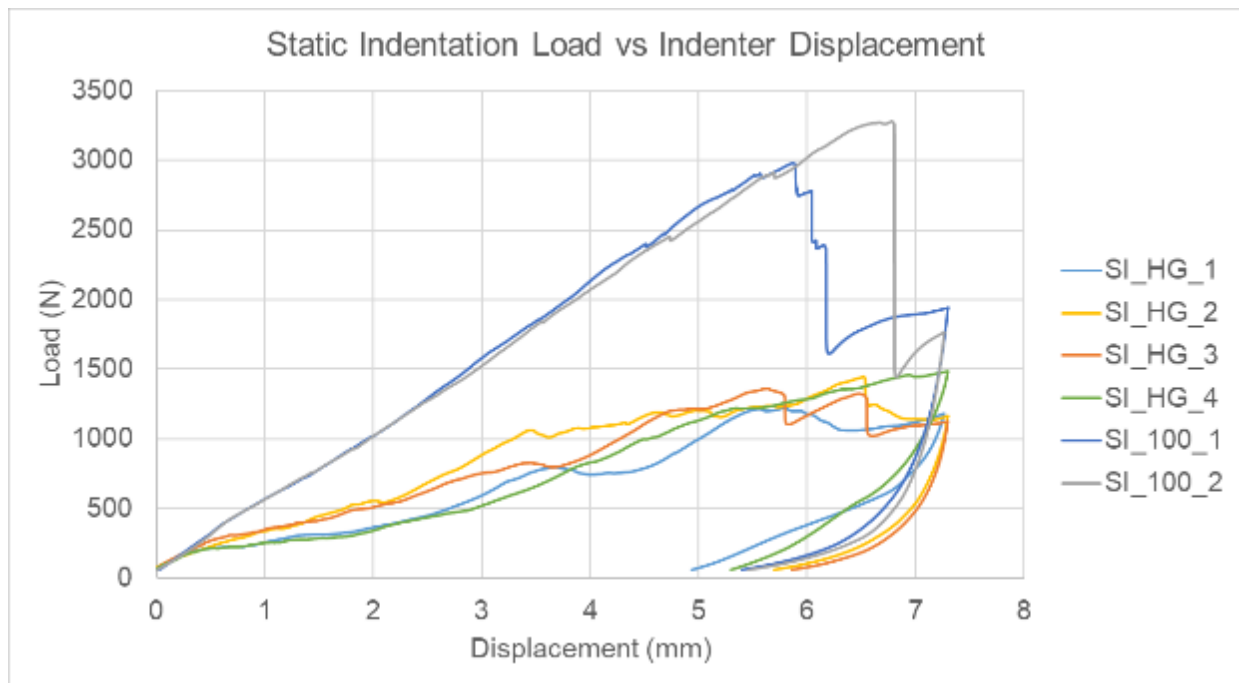


Figure 66 - Static indentation at elevated temperature of 100 °C

Table 8 - Static indentation summary, elevated temperature

Sample	Max Load (N)	Indent Depth (mm)	Energy max (J)	Energy Absorbed (J)
SI_100_1	2980.3	4.54	11.78	7.62
SI_100_2	3280.4	4.48	11.31	7.65
SI_HG_1	1222.8	4.66	5.35	2.31
SI_HG_2	1446.4	3.76	6.46	4.33
SI_HG_3	1360.8	3.43	5.33	4.05
SI_HG_4	1485.5	4.92	5.36	2.75
Average	1378.9	4.19	5.63	3.36
Stdev	116.34	0.71	0.56	0.98
CV	8%	17%	10%	29%

The static indentation testing showed a difference between room temperature and 100 °C testing. There is an observed decrease in the max load the sample can withstand of around 40% when heat is applied to the sample. There is also a reduction in energy absorbed and max energy of 50% between room temperature and elevated temperature. These results show that heating of the sandwich causes degradation to the sandwich structure and confirms previous studies shown in literature [20][23]. The temperature chosen was at 70-80% of the max operating temperature of the foam. It can be seen from these tests that the foam does begin to degrade, and the mechanical performance of the sandwich is reduced when operating at higher temperatures. The method of testing for this study could be improved by developing an oven to encase the entire testing apparatus so that heating of the sandwich is more consistent and uniform throughout. This would give a better understanding of mechanical performance under a higher temperature. The results of this study with the localized heating of the heat gun to maintain temperature do show this reduction in mechanical performance, but the extent of the reduction could be higher in this study due varying temperature differentials across the sandwich.

The static indentation testing was done in lieu of low velocity impact testing since there are comparable results between the two tests. Shengqing et al. investigated failure modes of quasi-static indentation and low-velocity impact tests on composite sandwich panels [57]. They found that the failure modes between the two tests were similar. From this, quasi-static indentation should give a suitable replacement for low velocity impact tests on sandwich panels. The results of the thesis study showed that static indentation resulted in an absorbed energy of 6.45 J for room temperature. Flores-Johnson et al. investigated static indentation of sandwich panels with CF face sheets and foam core similar to this thesis study [65]. They found that a hemi-spherical indenter resulted an absorbed energy of around 3.75 J for a core of Rohacell 71 WF. This is less than the absorbed energy observed during this thesis study, but it is close in magnitude. From this, the results of this thesis study are comparable to previous studies in literature. Dikshit et al. investigated quasi-static indentation response of inkjet printed sandwich composite structures [66]. They observed an indentation energy of 6.87 to 8.82 J. The composite configurations in their sandwich structures differ from this thesis research, but the values for indenter energy are relatable to the indenter energy observed in this study of 11.3 J.

The sandwich panels were inspected after testing with optical microscopy to examine the effect of indentation on the core. It was found that the core experienced localized crushing under the face where the indenter was applied. The face sheet experienced cracking along the fiber direction and after unloading, there was a spring back of the face leaving the core crushed and a gap between the face and core material. These results suggest that permanent damage is done to the core from the static indentation while the CF/PPS face sheet can recover several millimeters after the load is removed. The load displacement response from the testing shows that after the onset of failure, there is a drop off in load followed by load recovery. This is most likely due to the cells in the foam collapsing onto each other and compacting together thereby increasing the stiffness of the foam as well as localized density as air pockets within the foam are removed as seen in Figure 106 of Appendix A.4. The load experienced after this begins to increase as the displacement of the indenter increases. The microscopy of room temperature and elevated temperature cross sections can be seen in Figure 67 as well as Figure 100 through Figure 105 in Appendix A.4.

Temperature monitoring for the static indentation part of this study was done using thermocouples for the heating/cooling cycles with 3 thermocouples inside the core at various locations and 1 thermocouple on the surface. This was done to understand how the sandwich responds to heat at each location and how long it would take to reach the desired temperature as well as cool down to room temperature. Thermocouples were used on the surface during the heat gun testing to monitor the temperature as the heat gun was applied. Data was recorded for the oven heating and cooling cycle as well as the first heat gun trial. It was found that the sandwich panel takes around 30-40 minutes to reach temperature of 100 °C during heating and 15 minutes to cool down to room temperature after being removed from the oven. There was an observed temperature gradient of 3-5 °C between the face and core of the sandwich structure. The temperature study and thermocouple diagrams can be seen in Figure 107, Figure 108, and Figure 109 in Appendix A.4. It was observed that the heat gun for maintaining temperature has variability in a range of 85-105 °C. This explains the larger variability in the elevated temperature results. Even with the increased variability, the elevated temperature static indentation results were all significantly lower than the room temperature testing.

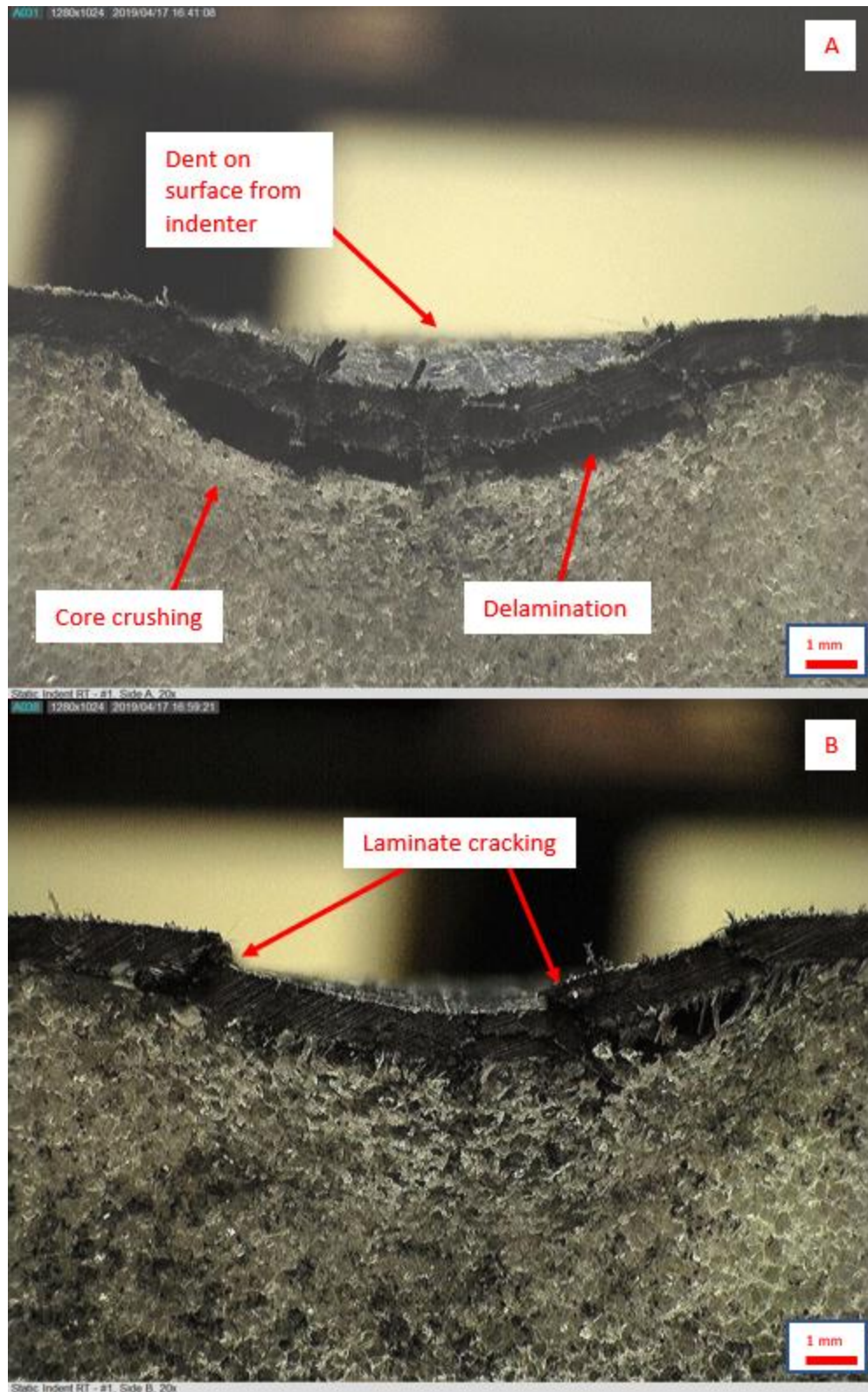


Figure 67 - Static indentation microscopy, RT sample #1, 20x, side A (top) and side B (bottom)

4.7 - Characterization of CF/PPS

The face sheet of the sandwich composite was characterized for flexural, interlaminar shear, and impact. The results of the first phase of characterization, the face sheet removed from core shear sample #1, can be seen in the Table 9. The resulting flexural strength of the CF/PPS was found to be 1415.45 MPa and the flexural modulus was found to be 121.94 GPa. The flexural strength observed for the face sheet is relatively high compared to similar tests from literature. For instance, Vieille et al. obtained a flexural strength of 578 MPa and flexural modulus of 44 GPa for their CF/PPS laminate with quasi-isotropic layup [35]. The reason for the experimental results of this study being much larger than Vieille et al. can be attributed to the performance of unidirectional fibers compared to a cross-ply quasi-isotropic hand layup. Since the flexural testing of this study was performed with the direction of the fibers along the flexural sample length, the strength of the material is much stronger in this direction. Thickness of the laminate does influence the behavior of the composite and the way that the load is carried through the system.

Table 9 - CF/PPS flexural sample results – face sheet

Specimen	Width (mm)	Thickness (mm)	Peak Load (N)	Flex Strength (MPa)	Flex Modulus (GPa)
CF-PPS-F1	12.767	1.117	511.63	1223.77	112.54
CF-PPS-F2	12.203	1.182	604.78	1351.51	116.92
CF-PPS-F3	12.710	1.059	623.48	1666.51	132.04
CF-PPS-F4	12.307	1.160	485.00	1115.82	102.23
CF-PPS-F5	12.690	1.142	604.55	1391.75	120.06
CF-PPS-F6	12.687	1.167	726.50	1602.00	127.96
CF-PPS-F7	12.653	1.100	583.70	1453.63	137.28
CF-PPS-F8	12.540	1.105	610.29	1518.59	126.45
Average				1415.45	121.94
Stdev				185.59	11.33
CV				13%	9%

Kumar M. et al. studied the effect of thickness and fiber orientation on flexural properties of composites [67]. They found that the flexural stress and young's modulus increased with a decrease in laminate thickness. Racz et al. studied the relationship between flexural properties and specimen aspect ratio in unidirectional composites [68]. They determined that unidirectional composites have a transition in the failure mode from shear delamination to fiber yield when the span to thickness ratio is increased. The span to thickness ratio of the face sheet flexural test for this thesis work was higher than the typical 16:1 ratio, with a ratio of 22:1. Based on literature, it is reasonable to see higher flexural strength due to a thinner specimen during testing. The load vs displacement graphs for each sample from the flexural testing of the face sheet can be seen in Figure 68. The results have a tight fit with sample #6 as a potential outlier (green curve in Figure 68), but the corresponding CV of 13% is acceptable for deviation between the samples. The loading phenomena observed is typical of fiber-resin matrix systems. The material exhibits a linear response till peak load, followed by a sudden load drop, representative of crystalline thermoplastics. At maximum value, matrix microcracking is observed, then the fibers pick up the load and there is a slight load recovery after the first drop from max load. Under flexure, the composite can continue carrying load after the matrix failure, but this load is significantly less (30%+).

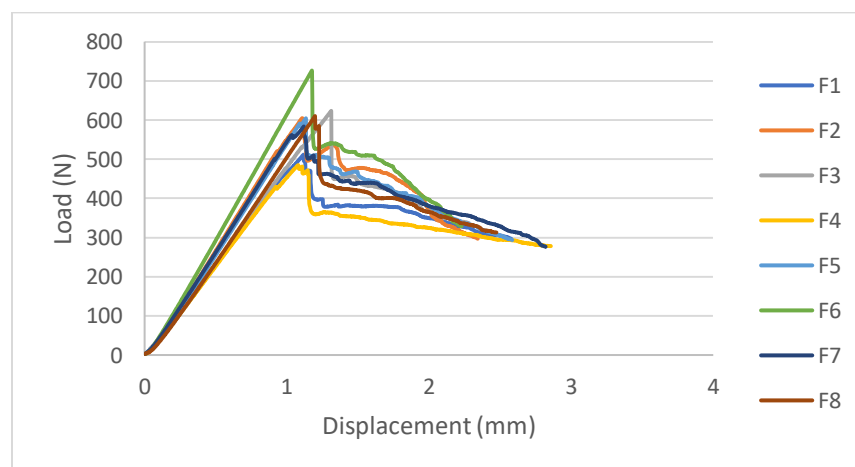


Figure 68 - Flexural load vs displacement for CF/PPS face sheet (1.177 mm thickness)

The CF/PPS was also characterized for a thicker panel with a desired thickness of 3-3.5 mm. To achieve this, 14 layers were wound during the filament winding process and then consolidated into panels as described earlier. The resulting panels for testing was measured to be 2.63 mm average thickness using a Mituyo micrometer. From this panel, flexural and ILSS samples were cut for testing. Another panel was made with 18 layers to achieve a higher thickness for IZOD testing, but the resulting panel came to be 2.7 mm thickness. IZOD samples were cut from this panel for testing. Flexural samples were cut following ASTM D790 to a width of 12.7 mm and a length of 60 mm to provide a larger overhang during testing with a span length of 42 mm (16 times thickness). The flexural, ILSS and IZOD testing was done on both 0- and 90-degree fiber orientation directions. The results of the flexural testing can be seen in Figure 69 and Figure 70 for 0- and 90- degree respectively. As seen earlier with the face sheet flexural testing, the load is linear with respect to displacement up until failure where the matrix cracking begins, and the fiber picks up load carrying. The post curve for each sample shows a drop after max load, followed by slight load recovery before the sample cannot support the load as it continues to decrease over time.

The flexural results from the 14-layer CF/PPS panel differ from the face sheet panel due to the thickness of the samples being 2.6 mm for 14-layer vs 1.117 mm for the face sheet. As described earlier, the thickness can affect the flexural strength of the composite due to the failure of the beam as well as the span to thickness ratio changing. The span to thickness for the 2.6 mm samples was 16:1 whereas the 1.117 mm samples had a 22:1 span to thickness ratio. The peak load experienced by the thinner face sheet was less than 50% that experienced by the 14-layer laminate. The load vs displacement curve was also much steeper for the 14-layer compared to the face sheet (1287 N/mm vs 537 N/mm). This can most likely be attributed to the increase in layers sharing the load before ultimate failure during flexure. The face sheet only had 6 layers of CF/PPS.

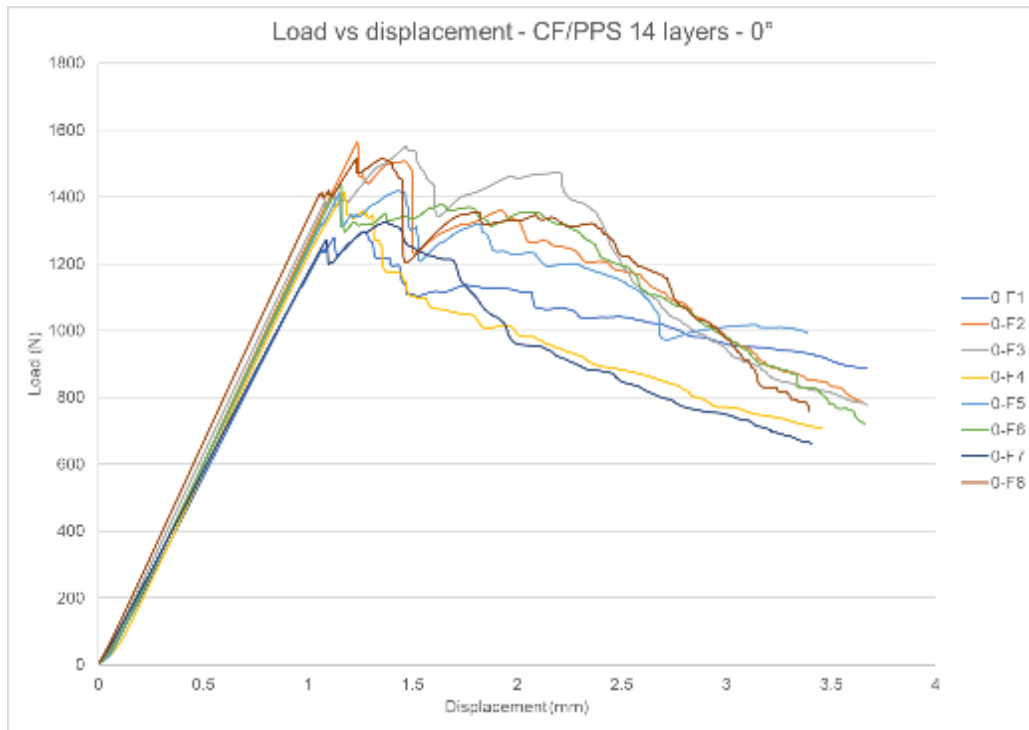


Figure 69 - Flexural results for 14-layer CF/PPS - 0 degree

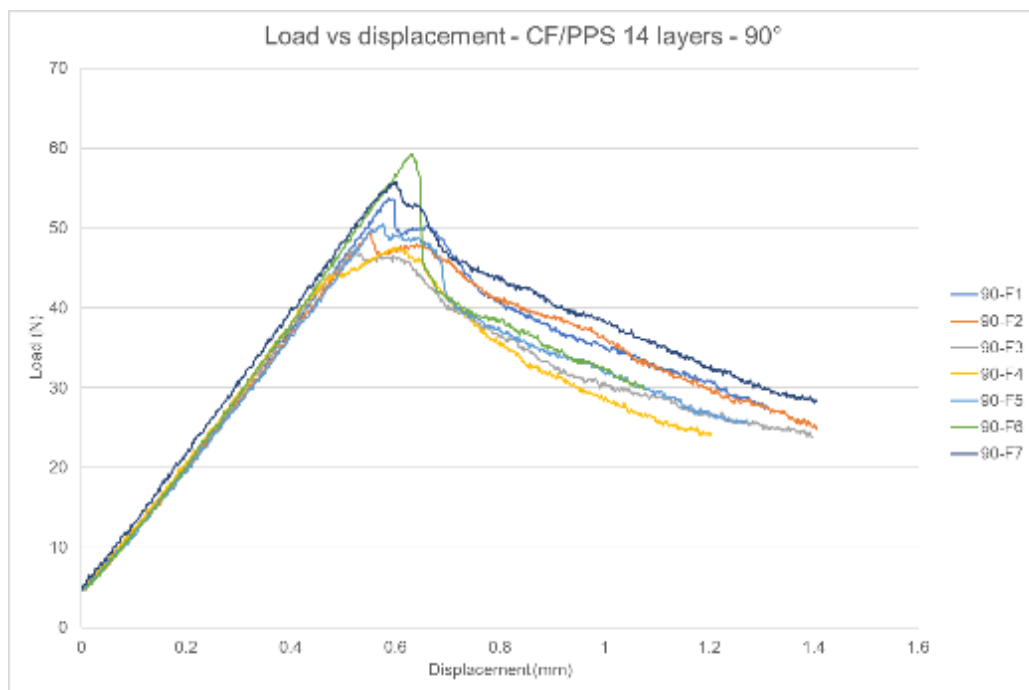


Figure 70 - Flexural results for 14-layer CF/PPS - 90 degree

The resulting flexural strength and flexural modulus were calculated following ASTM D790 using Equations 10 and 11:

$$\sigma_f = \frac{3PL}{2bd^2} \quad (\text{Eq. 10})$$

$$E_B = \frac{L^3 m}{4bd^3} \quad (\text{Eq. 11})$$

where P is the load, L is the support span, b is the width of the sample, d is the depth (thickness) of the sample, and m is the slope of the tangent to the initial straight-line portion of the load-deflection curve. The resulting calculations are summarized in Table 10. The flexural strength for the 0-degree orientation was found to be 1057 MPa for the 14-layer vs 1415 MPa for the face sheet of the sandwich. This difference is due to the lower span to thickness ratio for the 14-layer samples. Between the flexural strength and flexural modulus, there was an observed 33% reduction in strength and 15% reduction in modulus when increasing the thickness of the CF/PPS laminate to 2.6 mm from 1.13 mm. The 90-degree flexural results were found to be 3.5% of the 0-degree flexural results for strength and 6.5% for modulus. This large reduction in performance is attributed to the uni-directional fiber panel, the strength in the 90-degree direction is not influenced by the fibers and is based on the bonding between the layers (matrix-fiber interfacing). When compared to literature, the observed flexural strength is much larger than previous studies on CF/PPS. Values for flexural strength ranged from 235 to 578 MPa at room temperature testing for previous studies [35][48][46]. There was one study by El-Dessouky et al. that showed flexural strength of 829 and 748 MPa for two different laminate systems that were done through a satin 5:1 weave pattern [52]. These values are comparable to the flexural strength results of 1057 MPa from this thesis work.

Table 10 - CF/PPS flexural results summary

Sample	Thickness (mm)	Orientation (°)	Flexural Strength Average (MPa)	Flexural Strength Stdev (MPa)	CV (%)	Flexural Modulus Average (GPa)	Flexural Modulus Stdev (GPa)	CV (%)
CF-PPS Face Sheet - 0°	1.13	0	1415.45	185.59	13.11	121.94	11.33	9.30
CF-PPS 14 layer - 0°	2.62	0	1057.40	67.08	6.34	105.78	3.76	3.56
CF-PPS 14 layer - 90°	2.70	90	36.66	3.21	8.75	6.89	0.22	3.26

There were 3 types of failure modes observed for the flexural testing of 0-degree orientation: face buckling with tension side cracking, compression failure, and interlaminar shear. Most of the failures were face buckling with tension side failure, which is as expected for composite materials of high stiffness. Sample 0-F4 had a failure on the compression side with a crack forming off center as well as some face buckling and tension side failure. Sample 0-F5 had an interlaminar shear failure during testing on the side to the right of the loading nose. Sample 0-F7 had an interlaminar crack form as well as compression failure. The rest of the samples all failed on the tension side with a small amount of face buckling due to the loading nose contact. For the 90-degree orientation, all the sample failure modes were observed to be tension side failure. This is as expected, and it occurred rather quickly due to primary strength in that direction being the bond between each layer within the laminate. Microscopy was performed on the flexural samples using a Dino-lite edge digital microscope with magnification from 20-80x. The failure of sample 0-F1 can be seen in Figure 71 and Figure 72. There was face buckling on the compression side due to the loading nose which caused expansion of the fibers out of the side of sample resulting in crack formation. There was also an interlaminar crack formation starting at the center of the sample. The tension side of the sample had a crack form down the middle of the sample during tension failure.

The samples were placed between a C-clamp in order to apply a compressive force to open the self-healing cracks within the composite for microscopy. Imaging of the compression face buckling and interlaminar shear failures can be seen for 0-F5 in Figure 73 and Figure 74. The face buckling occurs on the compression side of the sample where the loading nose makes contact. There are ridges that form on the face as the sample undergoes compression. The interlaminar shear failure was observed to propagate from the center of the sample and travel towards the right of the loading nose. The reason for the interlaminar failure could be due to a weakness in the bonding between two of the layers within the laminate which failed and caused the sample to split apart under the load. This failure is a sign of a failure of the interface (matrix-fiber). Compression side failure was observed in 0-F4 as seen by crack formation on the compression surface where the loading nose made contact. This can be seen in Figure 75. There is a large crack formed on the compression side of 0-F4 with some microcracking between fiber

layers on the bottom side of Figure 75. This was a rare failure during testing and is most likely due to the compaction process magnifying a local area weakness where the 1st or 2nd layer would be weak compared to the rest.

ILSS testing was performed on the 14-layer CF/PPS panels following ASTM 2344 [61]. Once again, the 0- and 90- degree orientations were tested. A 3-point bend setup was used for ILSS testing with a span length of around 10.5 mm. The peak load experienced by the 0-degree orientation were on an order of 10x higher than that of the 90-degree orientation. This is as expected, as discussed for the differences between flexural testing of 0- and 90-degree orientations. The resulting average ILSS strength of the 14-layer CF/PPS panel was found to be 75.65 MPa in the 0-degree direction and 6.44 MPa in the 90-degree direction. The CV% for the testing was 6% and 10% for 0- and 90-degree respectively. When comparing to literature, there were a few studies that examined the ILSS of CF/PPS laminates with varying ply lay-ups. Franco et al. tested a 3.9 mm laminate of 0/90/45 layup and obtained 60.5 MPa for ILSS [69]. Jaeschke et al. tested ILSS on a 3.1 mm thick laminate with 0/90 layup and obtained 72 MPa [70]. Kadlec et al. tested ILSS on a 2.45 mm thick laminate with 0,90/+45 layup and obtained a range of ILSS from 63.3 to 76.7 MPa [71]. These values are all relatively close (within 20%) of the ILSS results from this thesis work for the 0-degree direction. The results of the ILSS testing can be seen in Figure 76 and Figure 77 for 0- and 90- degree respectively. The failure of the ILSS samples is like that of the flexural testing in that the load is linear with displacement up until failure begins to occur where the load tapers off to a maximum point before failure within the sample is enough to result in a drop of load. It was observed during testing that the load begins to rise again after this drop off due to the small size of the sample and the support nodes eventually picking up the load as the sample is compressed. It appears that sample 0-l8 is an outlier from the rest.

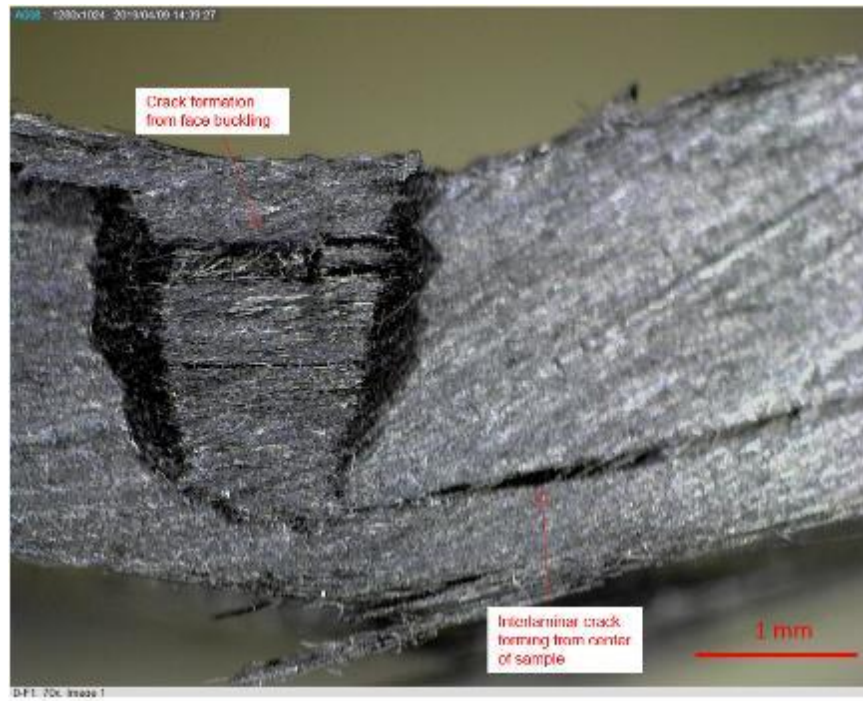


Figure 71 - Microscopy of 0-F1 flexural failure side view



Figure 72 - Microscopy of 0-F1 flexural failure bottom view (tension side)

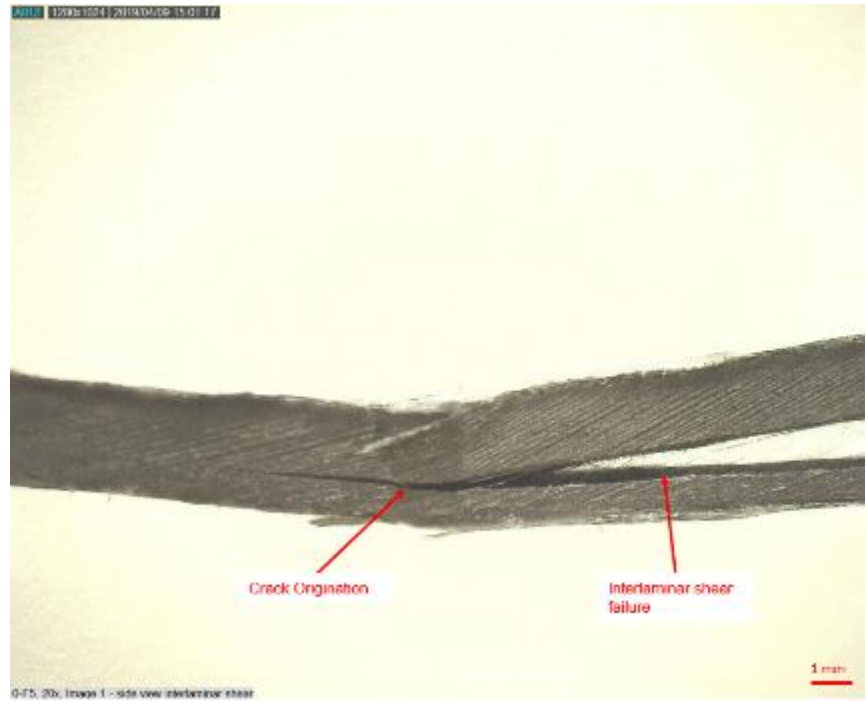


Figure 73 - Microscopy of 0-F5 flexural failure side view

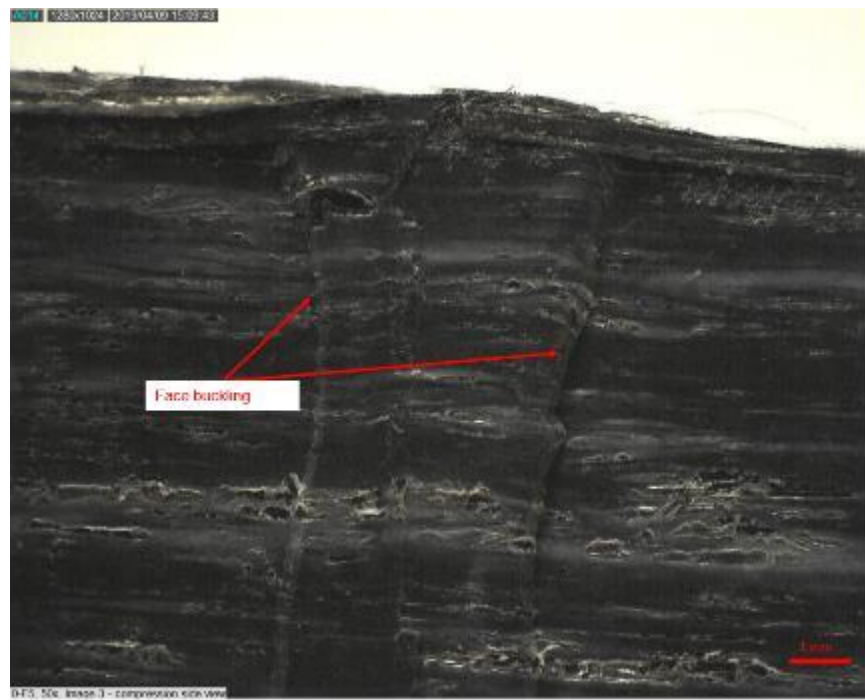


Figure 74 - Microscopy of 0-F5 flexural failure top view (compression side)

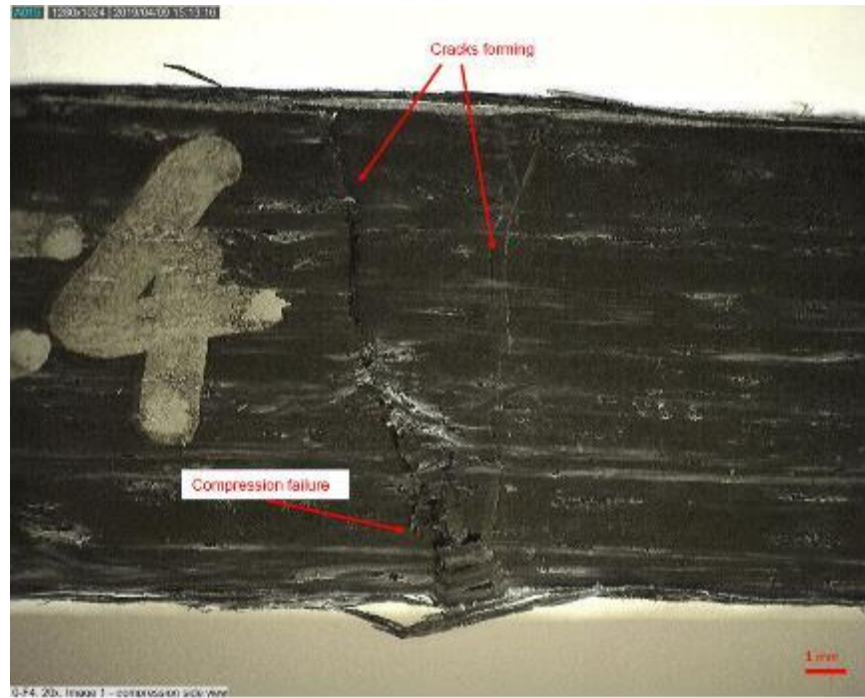


Figure 75 - Microscopy of 0-F4 flexural failure top view (compression side)

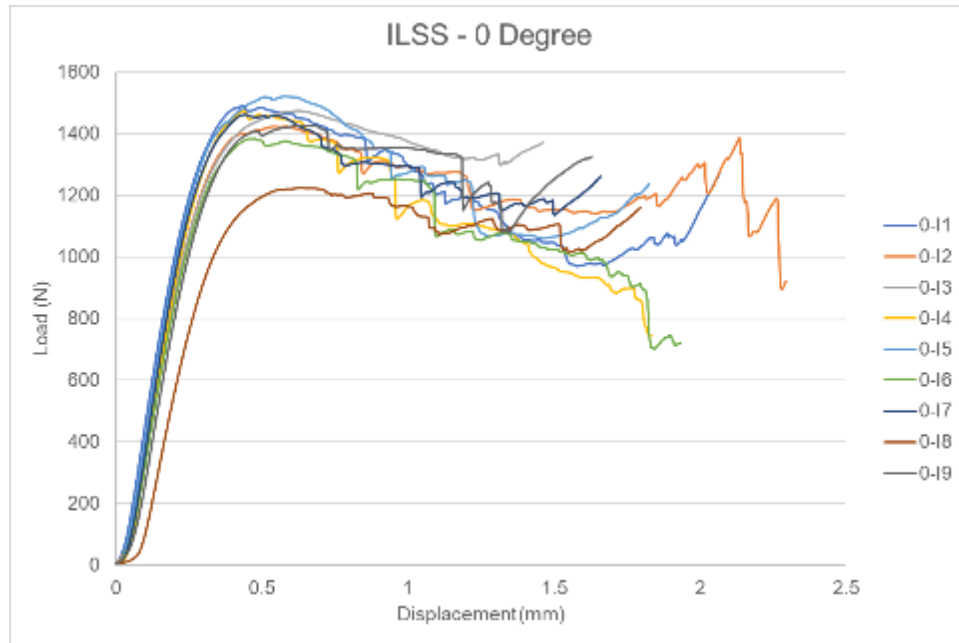


Figure 76 - ILSS for 14-layer CF/PPS - 0-degree orientation

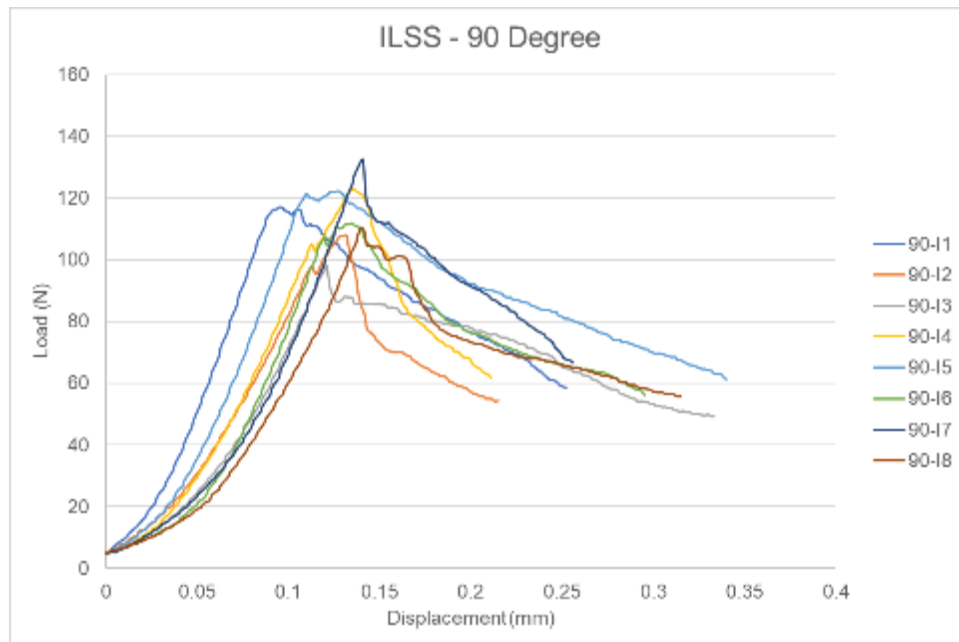


Figure 77 - ILSS for 14-layer CF/PPS - 90-degree orientation

The failure of the 0-degree ILSS samples was all by interlaminar shear with some compression buckling from the loading nose. A few of the 0-degree samples experienced a minor crack formation on the tension side of the sample. These failure modes are as expected for the ILSS testing of composite materials. The 90-degree samples all failed on the tension side with a crack forming. This is the same failure as the flexural testing and is a result of the interface being weak compared to the fibers. Microscopy was done on the ILSS samples to observe the failure modes. The failure of the 90-degree sample 90-I4 can be seen in Figure 78. The tension side cracking propagates over 50% of the way through the thickness of the sample. A tension side view of the 90-degree sample failure for 90-I6 can be seen in Figure 79. There was difficulty in obtaining an image of the tension side crack due to the size of the sample (15 mm by 5.2 mm). The failure of the 0-degree samples were primarily interlaminar shear where the layers separated in several locations within the thickness of the sample. This originated in the center of the specimens under the load and propagated towards one side of the specimen. Imaging of the tension side failure mechanism was also done to show the small crack formation. Figure 80 shows the interlaminar shear of sample 0-I1. This is present in all 0-degree samples. The tension side crack formation can be seen in Figure 81. The crack originates in the center of the sample and there is one crack that forms along the fiber direction towards the right. The primary crack is very jagged and forms in a direction off center.

Izod impact tests were conducted on CF/PPS samples in both the 0 and 90-degree orientations for notched samples as well as the 0-degree orientation for unnotched samples. Testing was performed on a Tinius Olsen model impact 104 machine, model IT504 plastic impact. The setup was a swinging pendulum was set weights to transfer the potential energy into kinetic energy as the samples were impacted in order to measure the break energy and impact strength of CF/PPS. The weights used were 37 N with a capacity of 22.6 J. The results can be seen in Figure 82.



Figure 78 - Microscopy of 90-I4 ILSS failure side view

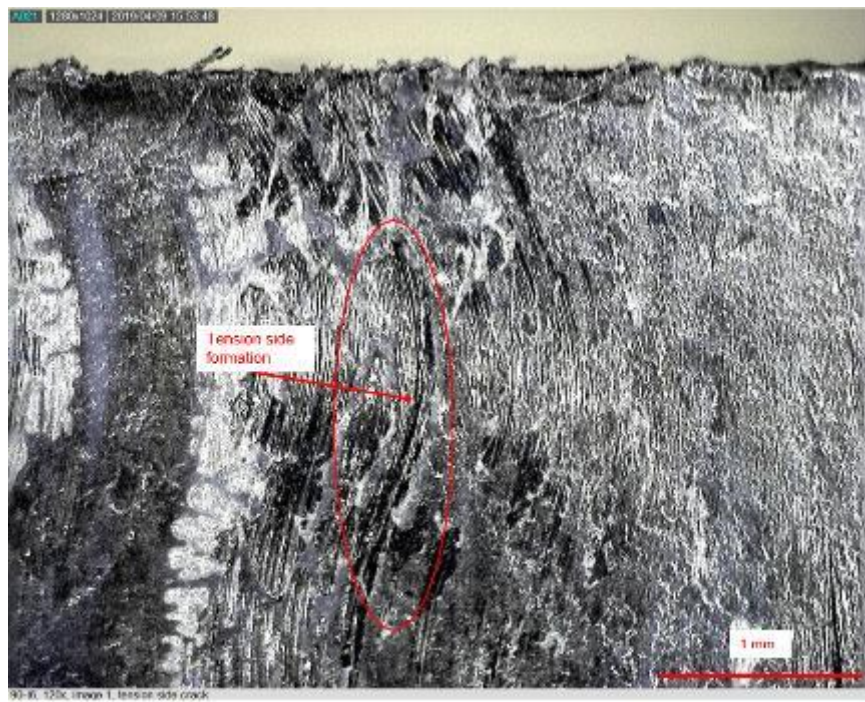


Figure 79 - Microscopy of 90-I6 ILSS failure bottom view (tension side)

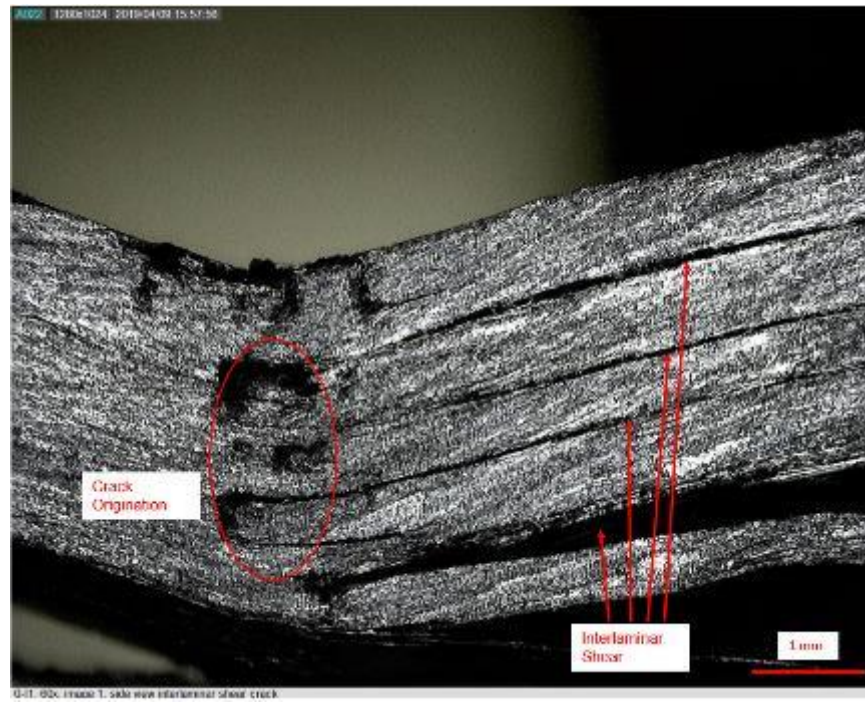


Figure 80 - Microscopy of 0-I1 ILSS failure side view

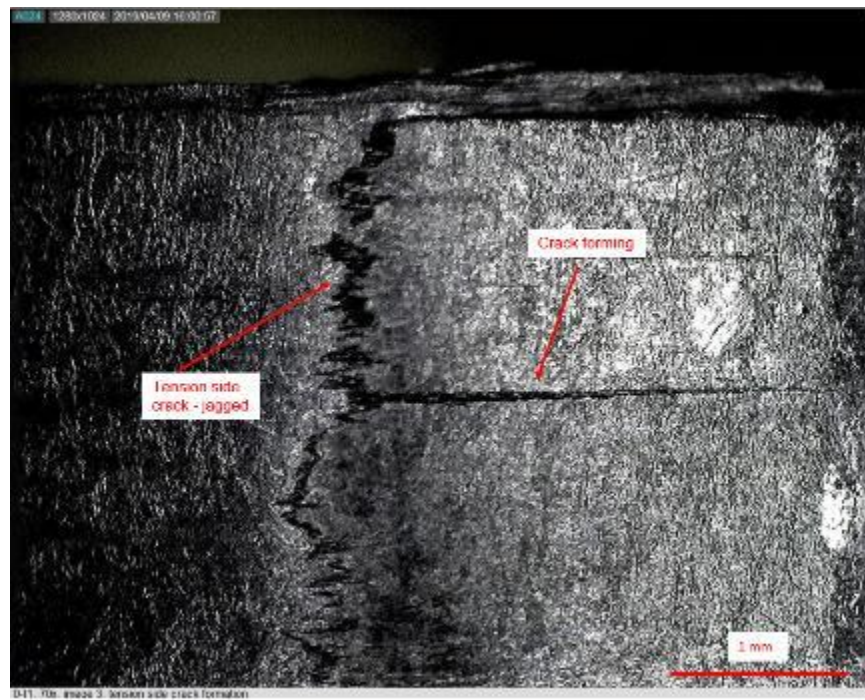


Figure 81 - Microscopy of 0-I1 ILSS failure bottom view (tension side)

The 0-degree samples were able to absorb a much larger amount of energy compared to the 90-degree samples, which is as expected. The unidirectional fiber laminate is weak in a 90-degree direction the fiber direction. There were two observed failure modes for the 0-degree notched samples: hinge and partial. The failure mode was complete break for all the 90-degree notched samples. The unnotched 0-degree samples had two failure modes: no break and partial. The notch allows for the assistance of the composite to fail, but when it is unnotched, the fibers are that much stronger, and the failures were mostly observed to be partial breaks. The average impact strength for the 90-degree samples was 40 J/m. The average impact strength of the 0-degree notched samples was 1867 J/m for hinge breaks and 3975 J/m for partial breaks. The partial breaks for the notched samples correlate well with the failures of the unnotched samples. The unnotched samples had an average impact strength of 3396 J/m and 3018 J/m for no break and partial break respectively. There is a large variance within the unnotched samples of 20%. This is most likely due to the different failure locations during testing and the unique failures. Since there is no notch to assist in the breaking of the sample, the sample will fail in a location of weakness which can vary between each sample due to a local area weakness such as a void or an interface weakness. A summary of the data collected from Izod testing can be seen in Table 11. The failures of the Izod samples can be seen in Figure 93 and Figure 94 in Appendix A.3.

TGA was performed on 9 CF/PPS samples that were heated up to 1000 °C at a rate of 20 °C/minute. CF/PPS samples were extracted to be between 6 - 9 mg in weight. The samples were taken from the face sheet of the Core Shear Sample #1, the same face sheet used for the flexural testing earlier. TGA was done to determine the weight fraction of CF within the face sheet. The dry fiber was known to be 50% volume fraction CF, but this can change during processing due to filament winding as well as hot compression. The results of the TGA can be seen in the representative plot of the CF/PPS behavior in Figure 83. TGA results show that the resulting samples kept between 74.04% and 79.30% of their mass when heated to 1000 °C. At this temperature, the PPS and any other compounds within the laminate will be vaporized and only the CF will be left. This gives an average fiber wt.% of 77.17% with a CV of 2.15%. TGA was performed on the dry commingled fiber material to evaluate the CF wt.% before the face sheet consolidation

process. The results of the dry fiber TGA showed a wt.% of 72.75% with a CV of 1.4%. There is a difference in the CF wt.% of the finished panels and the dry fiber where the CF increases from 72.75% to 77.17%. This increase is less than 10% and is most likely due to the filament winding and hot compression process where resin can be lost due to the tensioning during winding and the pressure application during panel consolidation. The CF is stronger than the PPS matrix, and having a larger wt. % of carbon fiber will increase the flexural properties. Additional TGA results can be seen in Figure 110 and Figure 111 in Appendix A.5.

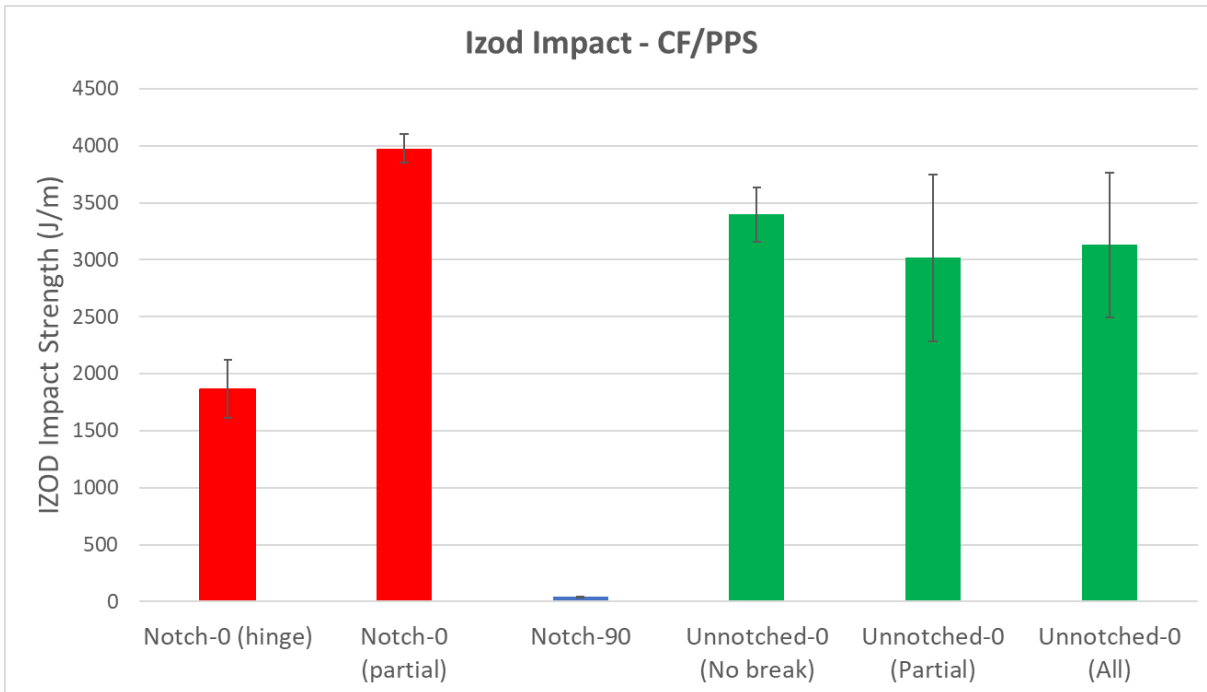


Figure 82 - Izod impact strength for 0,90 notched and 0-unnotched

Table 11 - Izod impact data summary

Name	Orientation	Break Type	Break Energy (J)	Strength-1 (kJ/m ²)	Strength-2 Avg (J/m)	Strength-2 Stdev (J/m)
Notch-0 (hinge)	0-deg	hinge	5.11	146.919	1867.2	257.3
Notch-0 (partial)	0-deg	partial	10.94	308.393	3975.7	126.0
Notch-90	90-deg	complete	0.11	3.168	40.2	7.4
Unnotched-0 (No break)	0-deg	no break	8.96	282.023	3396.0	235.6
Unnotched-0 (Partial)	0-deg	partial	7.86	255.701	3018.3	731.5
Unnotched-0 (All)	0-deg	ALL	8.19	263.598	3131.6	634.3

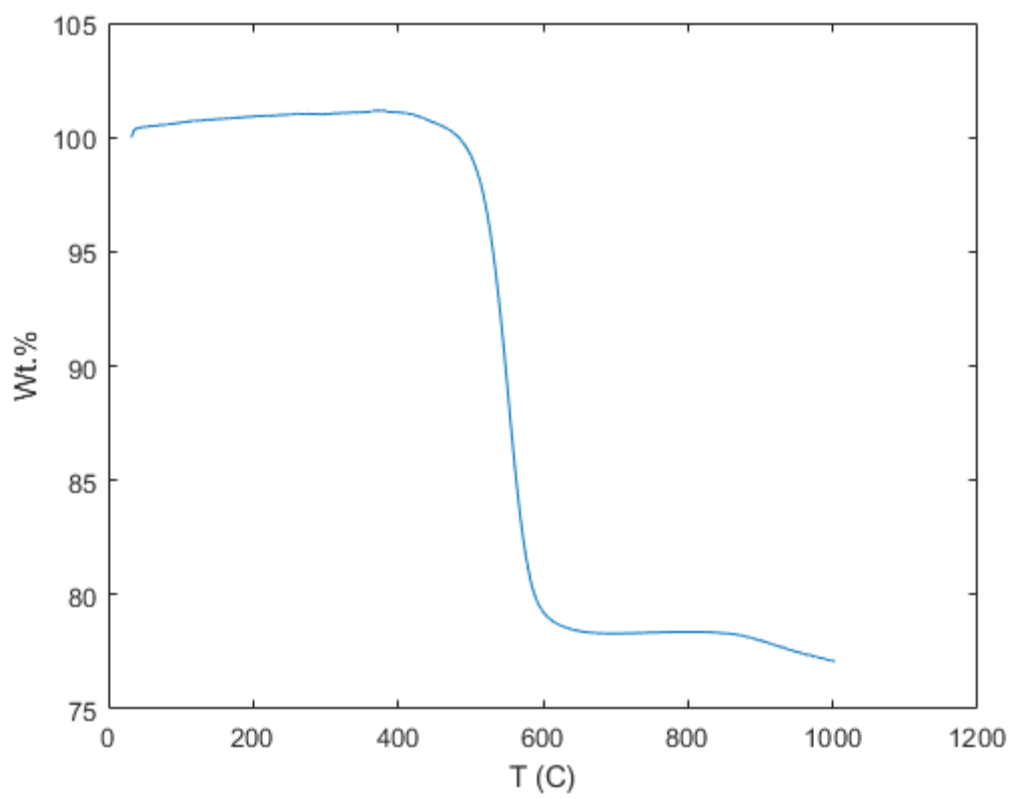


Figure 83 - TGA results of CF/PPS

Chapter 5 – Conclusions and Future Work

Concluding Remarks on CF/PPS & PMI Foam Core Sandwich

The design of the CF/PPS-PMI foam sandwich was done to ensure that the core would fail during core shear testing by choosing a core to face thickness ratio of 10:1. The resulting core shear tests showed that the sandwich design was successful in that the core failed during testing. The face material was characterized during this study to gain an understanding of its mechanical properties. The characterization showed that the face is much stronger than the core, as expected. Simulations of different core thicknesses confirmed that an increase in the core thickness would result in higher load resistance during core shear testing, which is as expected based on previous literature. The assumption of this work was that higher temperature would influence the performance of the sandwich structure and the evaluation of this effect for static indentation confirmed this. The absorbed energy of the sandwich at room temperature was twice as much as that at an elevated temperature of 100 °C. Previous literature has shown that temperature and thermal aging of composite materials does reduce the mechanical property performance. The results of this thesis research confirm that mechanical performance decreases under high temperature conditions. The load to failure for high temperature static indentation was 40% lower than that for room temperature static indentation. The max energy and absorbed energy saw a reduction of 50% and 48% when the sandwich panel was tested at elevated temperature when compared to room temperature. There was no observed statistical difference between damage resistance during the room temperature testing before and after the heat cycle (heating from RT to 100 °C, holding for 4 hours, and then cooling back to RT). Further investigation is needed to determine if repeated heat cycles or longer heat cycles would compromise the sandwich structure. For aerospace applications, materials are at elevated temperature during flights before returning to room temperature upon landing. The designed sandwich structure appears to withstand a considerable amount of load for each testing condition performed. The break energy of the CF/PPS was found to be between 5 and 8 J during Izod characterization, which is comparable to the 11.3 J energy for static indentation at room

temperature. This increase in energy can be explained by the foam core providing additional reinforcement to support the load. By comparing the impact response of the CF/PPS from Izod testing and the static indentation testing results, it can be concluded that the PMI foam core accounts for 41% of the energy absorption, while the CF/PPS face sheet accounts for 59%.

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Appendix

A.1 – Experimental Failures During This Thesis Research

The use of commingled fibers as a face material was chosen along with a PMI closed cell foam as the core. A foam of 1-inch thickness was used with the assumption of a 10:1 ratio to the thickness of the face, which resulted in a desired face thickness of 2.5 mm. The foam was tested first in the Carver Hot Press at the necessary conditions to consolidate PPS. These were 2 ton of pressure and a temperature of 280 °C for 15 minutes. During this trial, the foam began to deform and expand under the elevated temperature before reaching the necessary temperature to process PPS. It was observed, that the foam began degrading before reaching 220 °C. This phenomenon can be seen in Figure 84. It was assumed that the CF/PPS fibers would protect the foam during processing. Thus, the CF/PPS was wound directly onto a foam plate of 10"x10". The filament winding process for this required low tension to prevent the foam from crushing under the fibers during winding. The local crushing on the edge can be seen in Figure 85. The tension was set to only 5 lb. force during the winding of 10 layers of CF/PPS onto the foam. After the winding, the foam was trimmed in order to fit into a bar mold setup for the hot press. Each of the 4 corners of the foam plate had foam exposed to the air where no fibers were present as seen in Figure 86. During the hot pressing of this first sandwich trial, the foam began to degrade before reaching 180 °C. The set pressure of 1 ton steadily began to rise on its own to 2 ton, pointing towards a compromised foam. The foam continued to deform and expand, resulting in burning of exposed areas and collapsing of the edges. After the cycle finished, the edges of the foam were cut off and the inside of the sandwich was observed to be a dog-bone shape. The foam had melted and flowed out of the center of the plate towards the exposed corners resulting in less material left in the center. The fibers did not protect the foam from the high temperatures, another process needs to be used or a change to the sandwich structure needs to be done. This can be seen in Figure 87.



Figure 84 - Foam expansion during hot compression trial



Figure 85 - Local crushing of foam during filament winding



Figure 86 - Hot compression trial of filament wound foam



Figure 87 - Foam degradation during compression trial

The foam was changed from a 1-inch thick foam to a $\frac{1}{2}$ inch thick foam to see if manufacturing could be easier. The CF/PPS was again wound directly onto the $\frac{1}{2}$ inch foam plate, but more issues arose during this process that were not observed during the 1-inch foam trial. The $\frac{1}{2}$ inch foam started to buckle and crack down the center of the foam perpendicular to the fiber winding direction. Two separate winding trials failed during winding because the foam collapsed as seen in Figure 88. The foam collapsed in the bottom right corner of the image on the left side of Figure 88 and it cracked down the center in image on the right side. Winding directly onto the foam was determined to not be a means of creating the sandwich structure. The winding process was altered to an already established commingled fiber plate manufacturing process where the commingled fibers are wound onto a steel mandrel of $\frac{1}{4}$ inch thickness. The steel mandrel has beveled edges to prevent fiber breakage during winding and is of 11.5"x11.5" size to allow for creating plates up to 10.5"x10.5" in size. Using this method, plates of 10"x8" were able to be created from commingled CF/PPS fibers. The $\frac{1}{2}$ inch foam was kept as the core material, which meant that the CF/PPS face had to be around 1.2 mm in thickness to meet the 10:1 chosen ratio for sandwich construction. It was determined that 6 layers of CF/PPS during winding would create a sandwich panel of the desired thickness after consolidation. CF/PPS was wound directly onto the steel, then consolidated in the carver hot press, and finally being cut off the steel mandrel using an oscillating saw. In order to bond the face and core materials, an adhesive needed to be used. A PET film was chosen as the bonding agent between the two materials. A trial was performed to bond a 3"x3" sandwich panel together using the PET film. The film required processing at 175 °C for 10 minutes in order to melt and then a cooldown to room temperature to finish the curing. This trial resulted in the foam compressing from 12 mm to 2 mm in thickness. This process is shown in Figure 89.

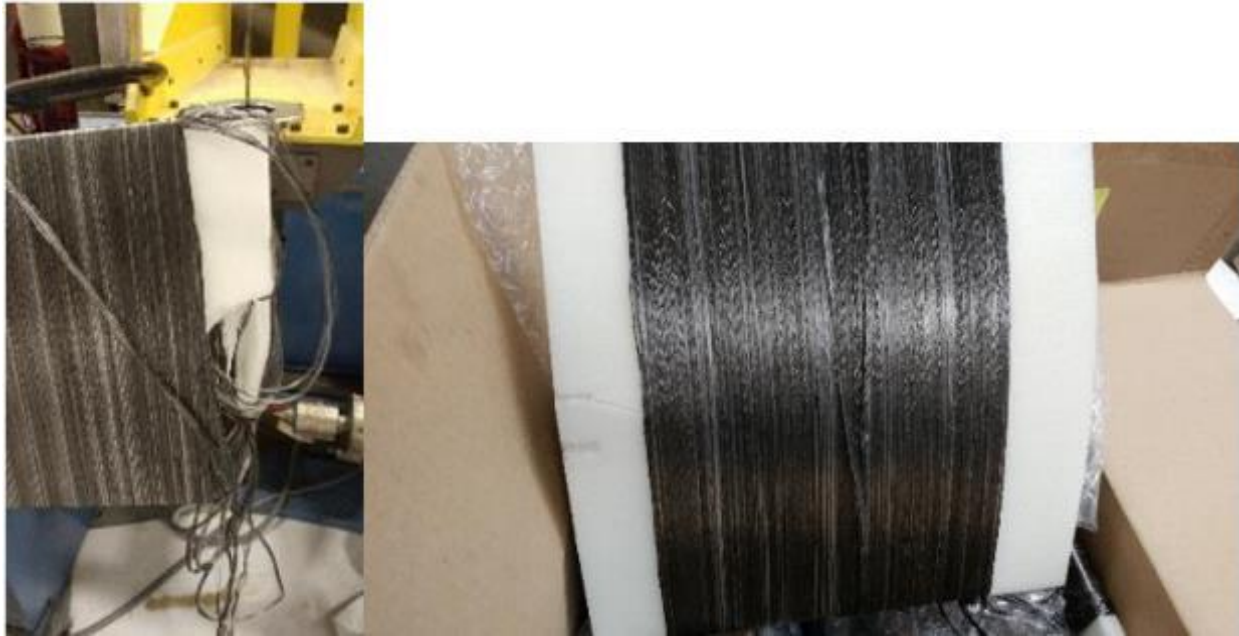


Figure 88 - Filament winding onto 1/2-inch foam - failures during winding process

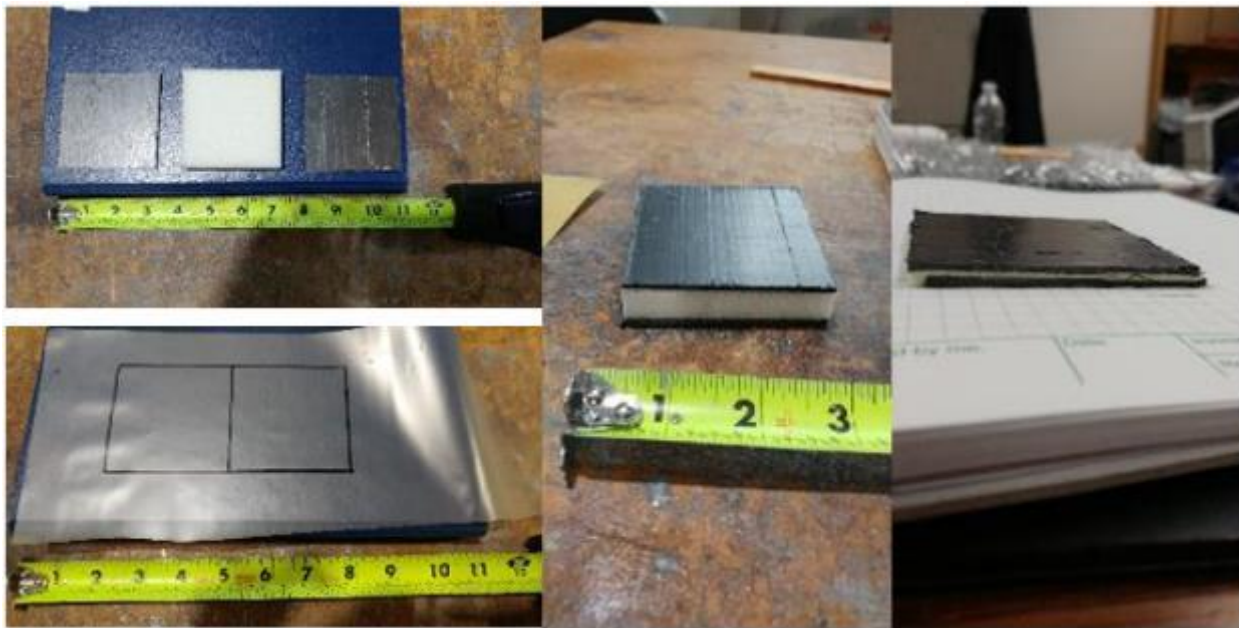


Figure 89 - PET film trial for bonding

The bond was indeed strong between the foam and the CF/PPS, but the compression resulted in an unusable sandwich. The PET film cannot be used to bond the face to the core. It was determined that an epoxy adhesive could be used to bond the two materials. On hand was a Huntsman Aradur 1568/Araldite 3492 two-part epoxy. The required ratio for the epoxy was 100g of resin to 28g of hardener. Trials began with this epoxy resin system using the same 3"x3" sample size. First the epoxy was tested at 82 °C for 8 hours and 0.5 ton, this resulted in the foam compressing from 12 mm to 7 mm thickness. Figure 90 shows the comparison between the trials for bonding with the epoxy trial on the right compared to the PET film trial on the left. There was an observed improvement in that the foam did not compress as much when the epoxy was used for bonding. The bond was strong, but the compression was concerning. Again, an epoxy trial was done using a 3"x3" sample, but instead it was carried out under room temperature with 0.5 ton of pressure. The pressure was applied for 12 hours and the bond appeared to be strong. There was no compression of the foam core. Sandwich construction then proceeded using the Huntsman epoxy.



Figure 90 - PET film trial (left) vs huntsman epoxy (right)

A.2 – Ultrasound Inspection/Nondestructive Testing

Report Date	Report Version	File Name	Inspection Date	Inspection Version	Save Mode
2019 / 03 / 27	OmniPC - 4.4R3	19.03.26_J Thesis_4A_S1_t3.opd	2019 / 03 / 26	NDU - 4.4R3	Inspection Data
OmniScan Type	OmniScan Serial #	Module Type	Module Serial #	Data File Name	
OmniScan SX	QC-007981	OMNISCX-PA1664PR	QC-007981	19.03.26_J Thesis_4A_S1_t3	

Setup					
A:0.00 Sk:090 L:001					
Beam Delay	Start (Half Path)	Range (Half Path)	Max. Acq Rate	Type	Averaging Factor
17.2 μ s	0.00 mm	9.60 mm	60	PA	1
Scale Type	Scale Factor	Video Filter	Pretrig.	Rectification	Filter
Compression	1	Off	0.00 μ s	HW-	None (1.0 - 17.8 MHz)
Voltage	Gain	Mode	Wave Type	Sound Velocity	Pulse Width
80	7.96 dB	PE (Pulse-Echo)	User-Defined	3000.0 m/s	100.00 ns
Scan Offset	Index Offset	Probe Skew	C-Scan Time Resolution	Digitizing Frequency	A-Scan Time Resolution
0.00 mm	0.00 mm	90.0°	10.0 ns	100 MHz	10.0 ns

Gate	Start	Width	Threshold	Synchro.	Peak Selection
I	0.00 mm	1.01 mm	45.00 %	Pulse	Max Peak
A	1.50 mm	0.66 mm	5.00 %	Pulse	Max Peak
B	4.01 mm	2.00 mm	30.00 %	Pulse	Max Peak

Calculator					
Element Qty. Used	First Element	Last Element	Resolution	Wave Type	Material Velocity
8	1	64	1.0	User-Defined	3000.0 m/s
Start Angle	Stop Angle	Angle Resolution	Focus Depth	Law Configuration	Beam Skew
0.00°	N/A	N/A	1000.00 mm	Linear at 0°	0.00°

Figure 91 - NDT settings for ultrasound inspection

A.3 – Izod Testing

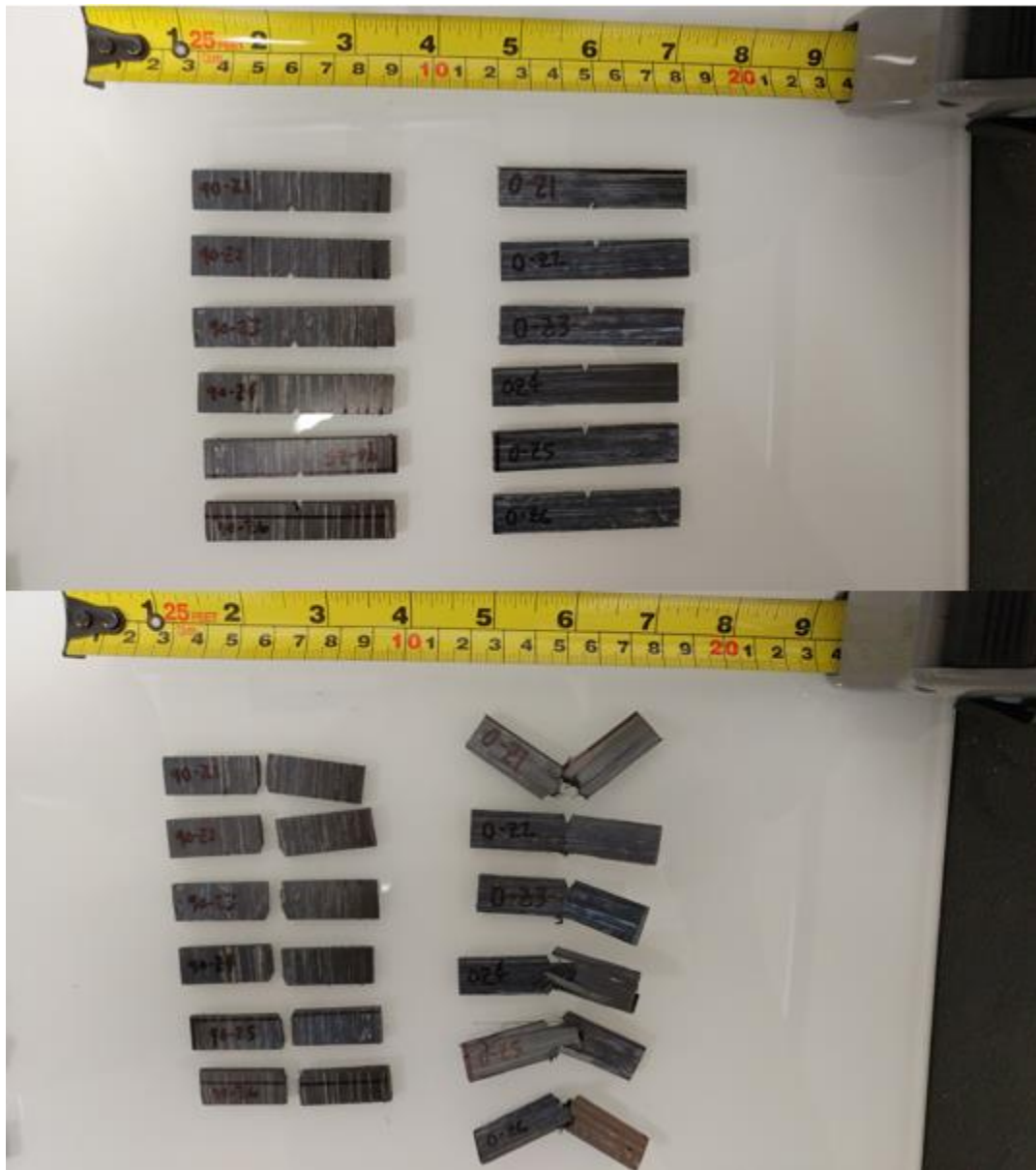


Figure 92 - Izod testing of CF/PPS notched. 90-degree before (top left), 0-degree before (top right). 90-degree after (bottom left), 0-degree after (bottom right)



Figure 93 - Unnotched samples before testing



Figure 94 - Failures of unnotched 0-degree samples

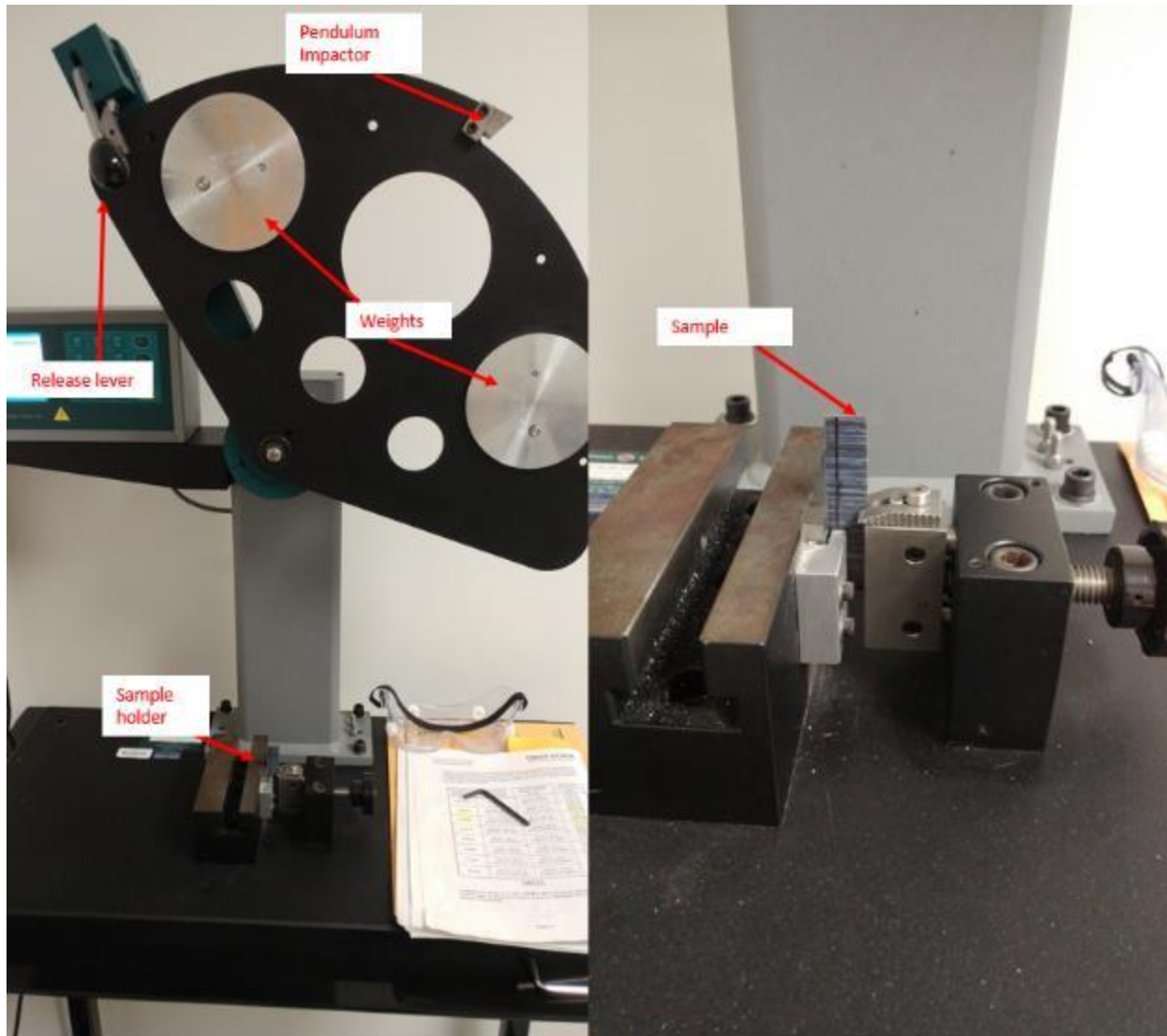


Figure 95 - Izod testing setup

A.4 – Static Indentation

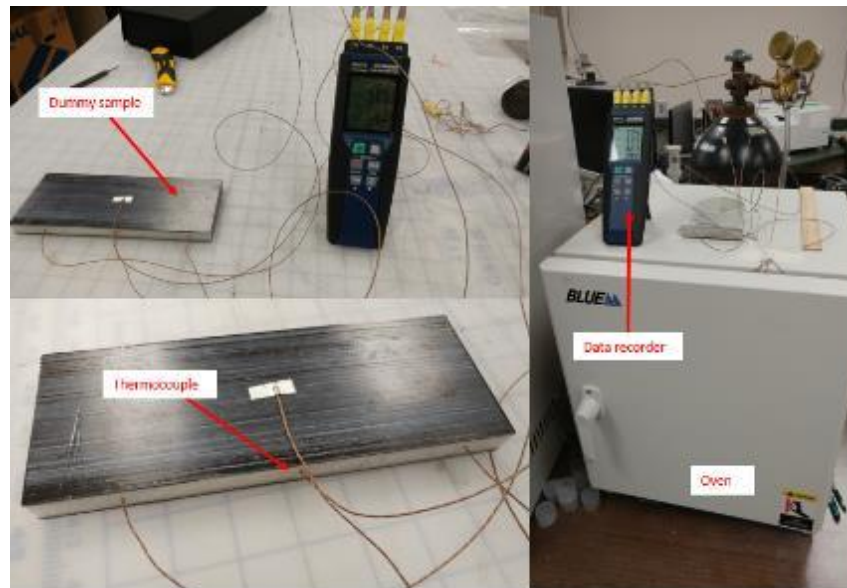


Figure 96 - Dummy sample testing for thermal cycle of sandwich panel with thermocouples. Heating to 100 C and cooling to room temperature



Figure 97 - Static indentation test setup for room temperature

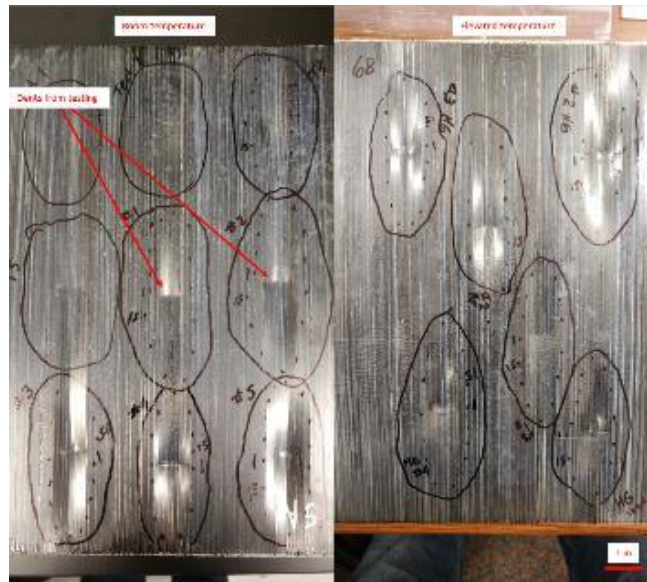


Figure 98 - Sandwich panels after static indentation testing. Room temperature on left, elevated temperature on the right



Figure 99 - Dent depth measurement process using dial indicator

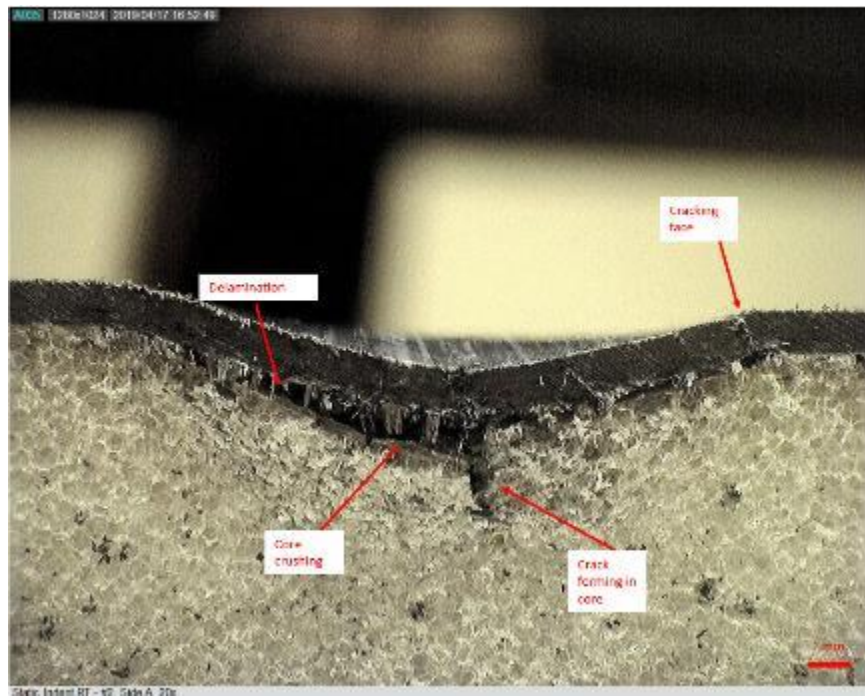


Figure 100 - Microscopy of RT static indentation sample #2, 20x

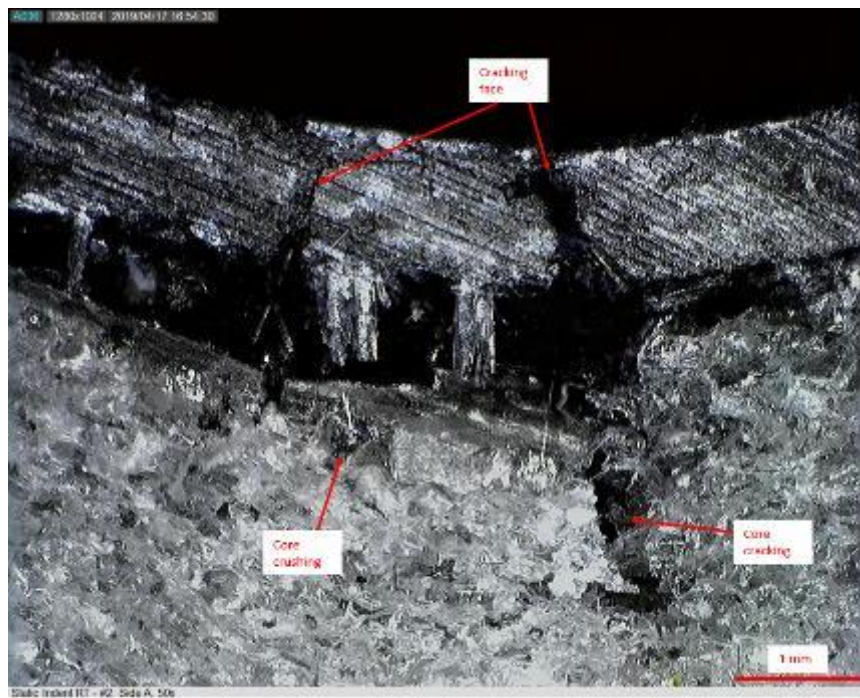


Figure 101 - Microscopy of RT static indentation sample #2, 50x



Figure 102 - Microscopy of RT static indent sample #5, top view of indent, 20x

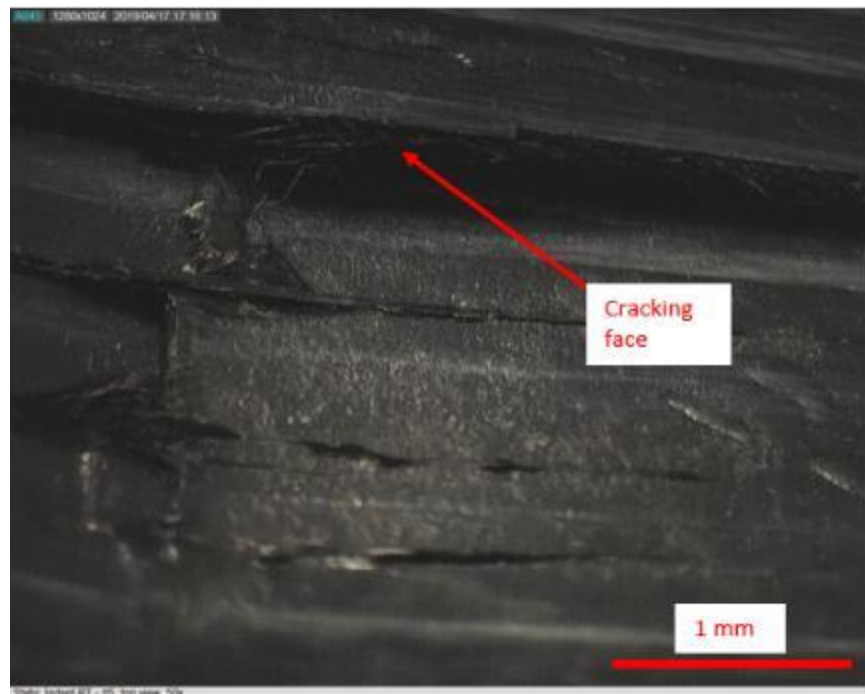
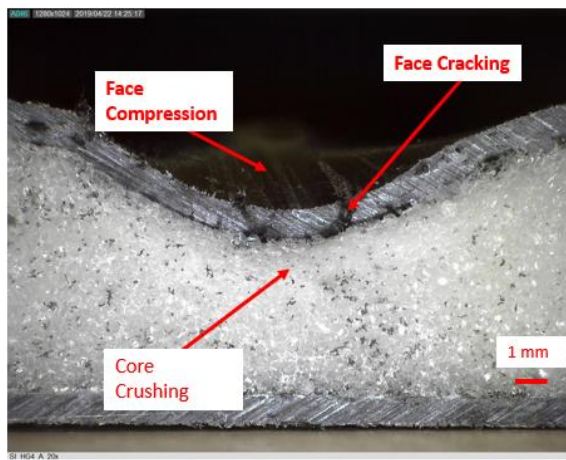
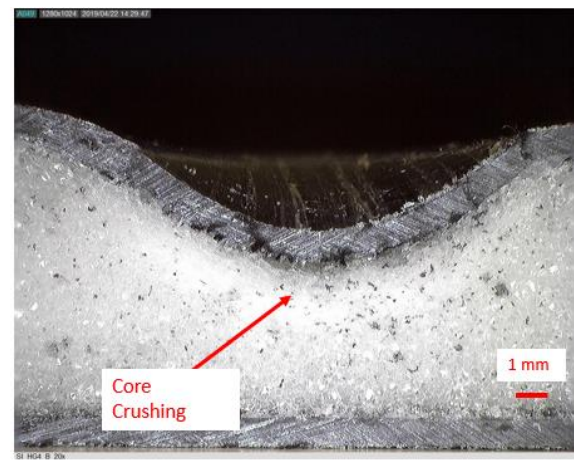


Figure 103 - Microscopy of RT static indent sample #5, top view of indent, 50x

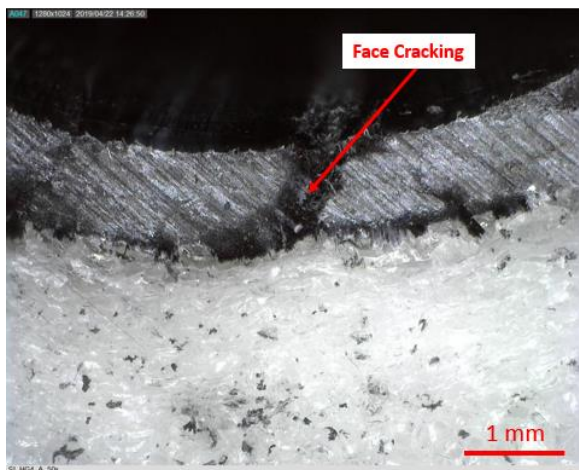


Side A

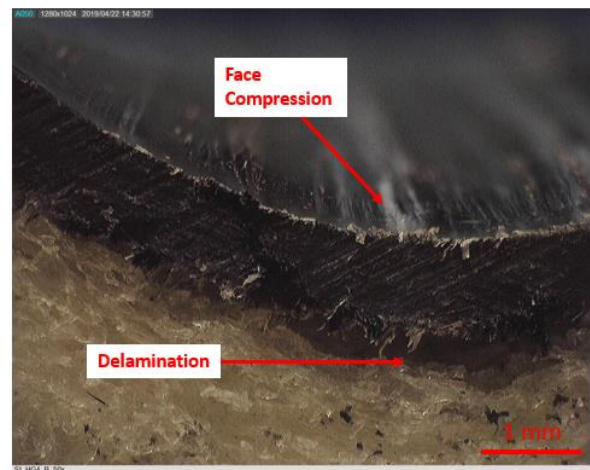


Side B

Figure 104 - Microscopy of 100 C static indentation HG #4 sample, side A and B, 20x



Side A



Side B

Figure 105 - Microscopy of 100 C static indentation HG #4 sample, side A and B, 50x

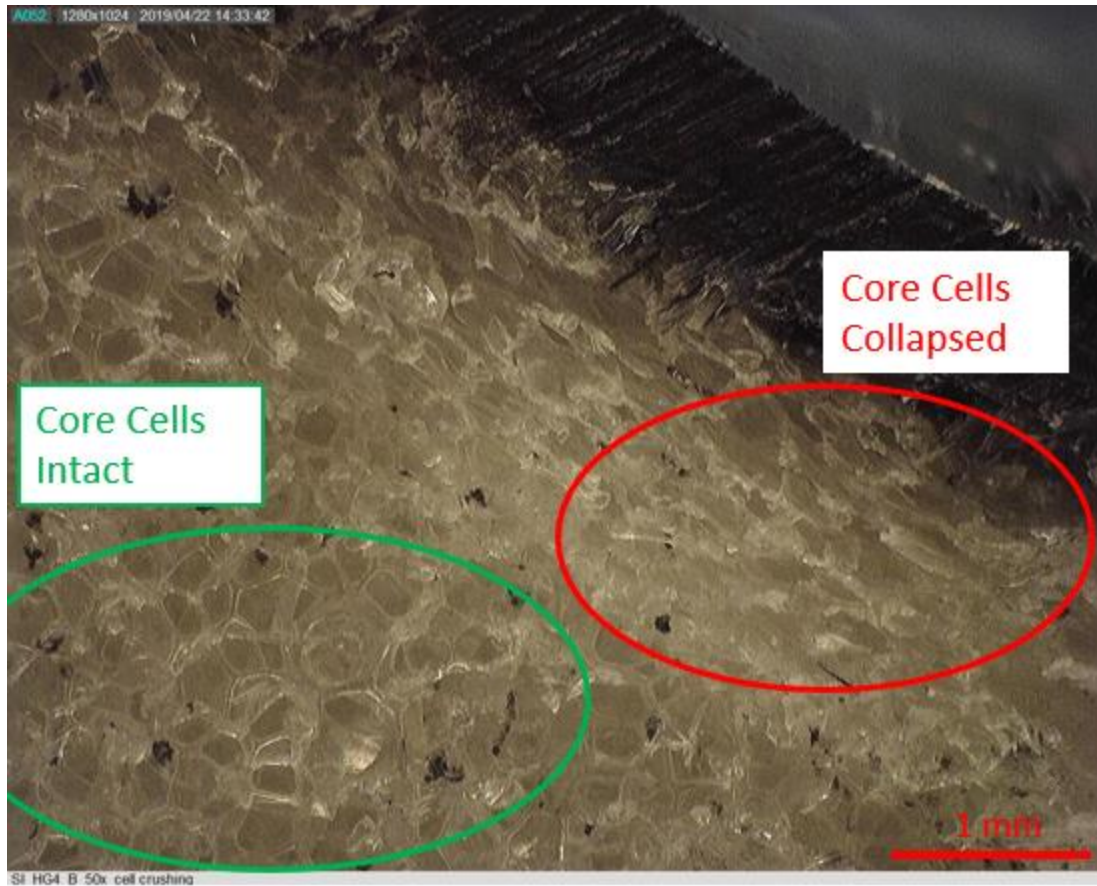


Figure 106 - Core crushing observed during static indentation testing: intact vs collapsed cells

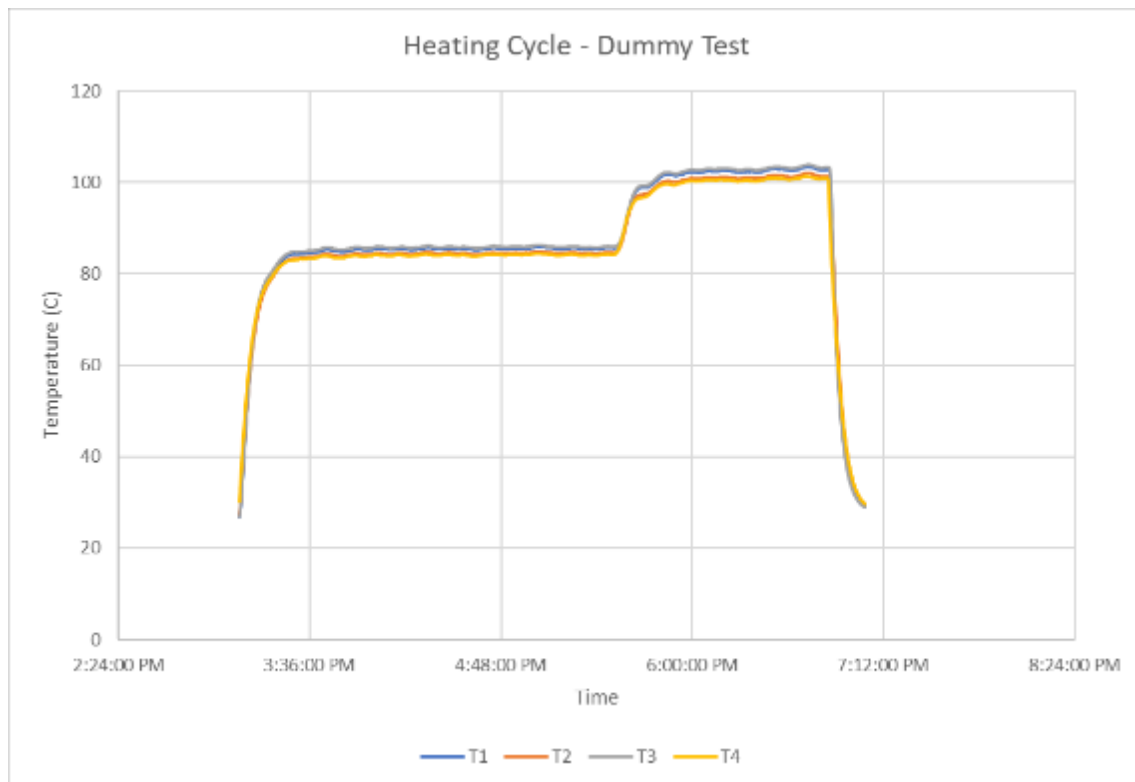


Figure 107 - Heating cycle for test dummy of sandwich panel for 100 °C. The temperature setting had to be altered from 100 °C to 120 °C in order to reach the desired temperature, as seen by the jump between 5 and 6 pm.

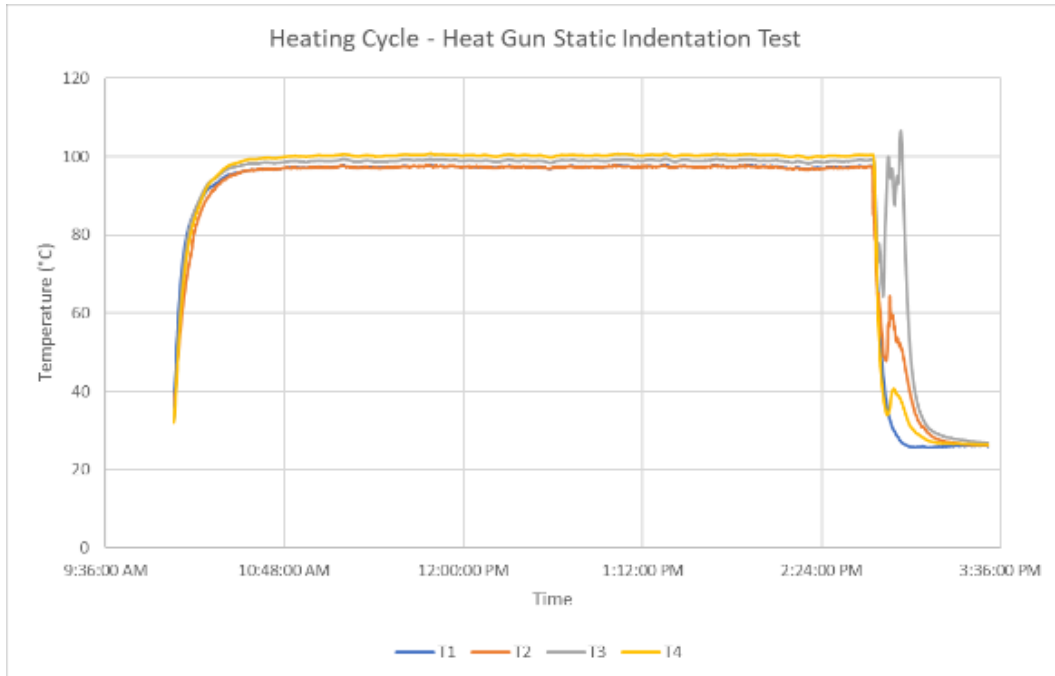


Figure 108 - Heating cycle for static indentation heat gun test. T3 is the thermocouple where testing was done, and the heat gun was aimed to keep temperature as close to 100 C as possible.

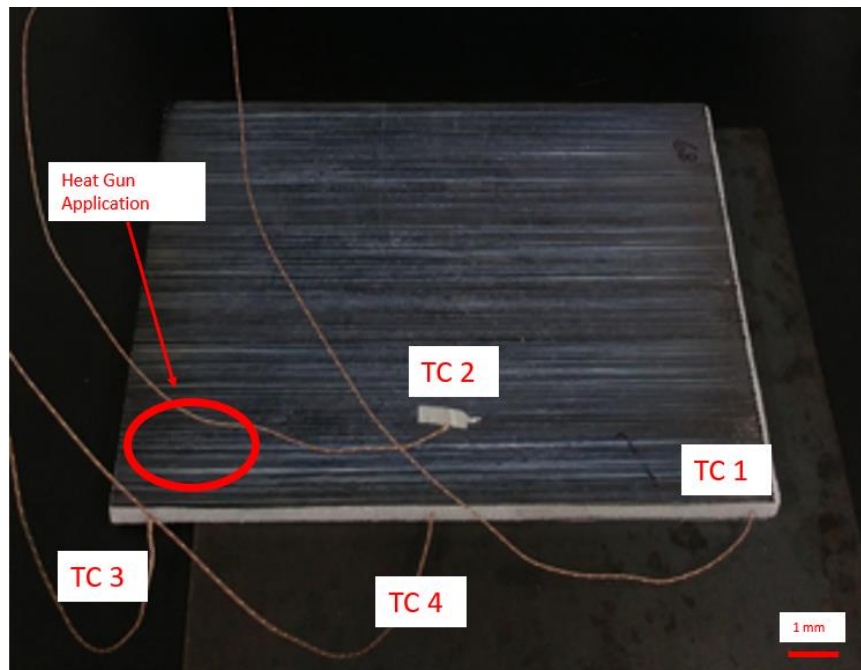


Figure 109 - Diagram of thermocouples (TC1-TC4) and heat gun application for elevated temperature static indentation

A.5 – TGA Results

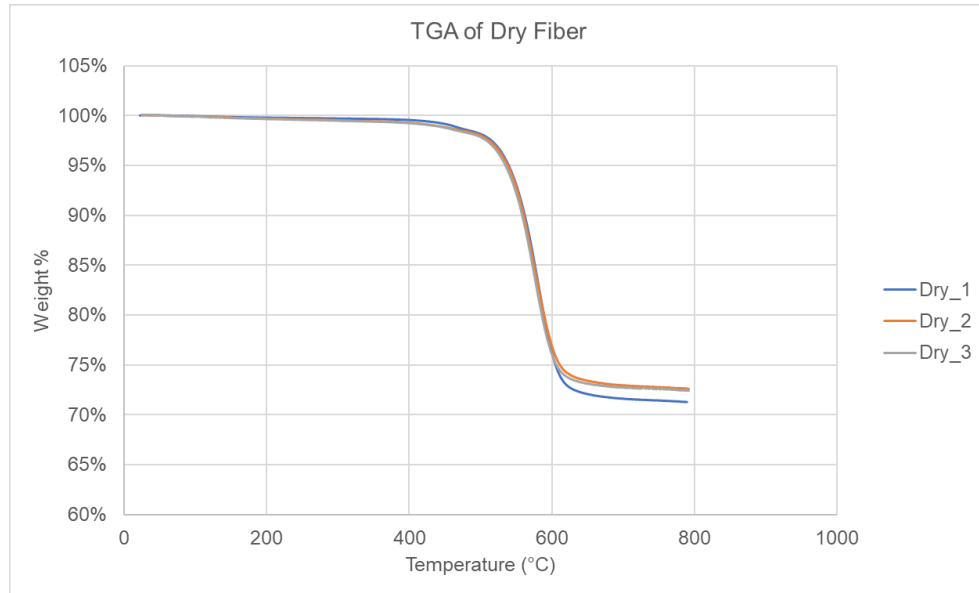


Figure 110 - TGA of dry fiber

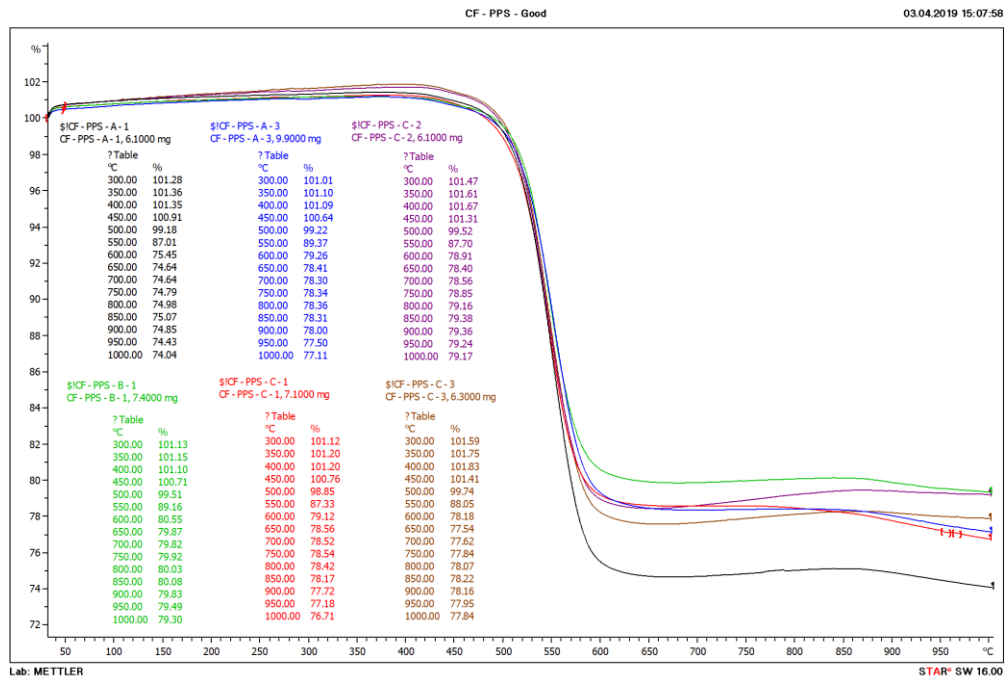


Figure 111 - TGA of CF/PPS face sheet

Vita

Jared Hughes was born in Oak Ridge, TN to the parents of Patrick and Paula Hughes. He is the first of two sons: Zach. He attended Oak Ridge High School in Oak Ridge, TN. After graduation, he went to Georgia Institute of Technology to pursue a B.S. in Polymer and Fiber Engineering. This is where he was first introduced to composite materials and developed an interest in their applications. He obtained his B.S. in Polymer and Fiber Engineering from Georgia Tech in May 2013. Upon graduation, he began work at the Centers for Disease Control and Prevention with his research focusing on the harmful constituents in cigarette smoke. After 3 years at the CDC, he sought to get back into the field of engineering. Upon learning of the Institute for Advanced Composites Manufacturing and Innovation (IACMI) and their work on advancing composite materials, he left the Centers for Disease Control in 2016 to pursue a M.S. in Mechanical Engineering at The University of Tennessee, Knoxville under the supervision of Dr. Uday Vaidya. After graduating with his M.S., he joined Boeing as a Composites Engineer.