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Manufacturing Process Considerations When Transitioning From Thermoset to Thermoplastic Composite Material for Urban Air Mobility Propellers

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MANUFACTURING PROCESS CONSIDERATIONS WHEN TRANSITIONING FROM THERMOSET TO THERMOPLASTIC COMPOSITE MATERIAL FOR URBAN AIR MOBILITY PROPELLERS

by

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ABSTRACT

Aerospace manufacturing has seen continually increased use of carbon fiber reinforced polymers in aircraft structures. The favorable strength to weight ratios, as well as the increased resistance to fatigue and corrosion, compared to metallic structures, provide a common argument for further research and development of composite airframes. Two main classes of carbon fiber reinforced polymers can be distinguished, thermosetting polymers and thermoplastic polymers. Thermoset polymers have been relied on for decades for the ability to reliably produce strong laminates for aircraft structures. By comparison thermoplastic polymers experience less comparable utilization on primary structures despite boasting similar or greater performance capabilities due to their processing requirements. The path to bridging the utilization gap between the polymers begins with advancing thermoplastic composite manufacturing strategies to allow engineers to adapt thermoset designs for thermoplastic manufacturing.

The shift from thermoset to thermoplastic composites is needed to meet the high manufacturing rate required for urban air mobility (UAM) vehicles. The manufacturing rate of thermoset composites are increased through parallel manufacturing lines due to the long cure cycle of thermoset material. The consolidation cycle for thermoplastic composites is significantly shorter than a thermoset cure cycle, therefore, the manufacturing rate for thermoplastics can be increased without the need for multiple manufacturing lines. Another advantage in favor of thermoplastics is the capability to

remelt and reconsolidate the polymer without negatively impacting its mechanical properties. The latter opens the opportunity to use fusion joining methodologies to assemble structures without the need for fasteners or adhesive, and thus lower the structural weight of the part. Induction welding is the fusion joining methodology used in this investigation. Induction welding allows for localized heat generation in the thermoplastic composite without contact between the structure and induction coil, therefore, making it an ideal candidate to explore the fastener free assembly of thermoplastic propeller blade demonstrator.

The research presented in this thesis focuses on the transition from using a thermoset to thermoplastic composite for the manufacturing of a UAM propeller blade demonstrator. The transition process involves a reevaluation of the internal structure of the propeller blade, and creation of a manufacturing plan to laminate, consolidate, and fusion join the individual components of the propeller blade. The objective is to showcase different aspects one must consider when transitioning from a thermoset material system to a thermoplastic one. In addition, the application of induction welding highlighted to achieve a fastener-free assembly which can aid in increasing the manufacturing rate to meet high production rate required for UAM vehicles.

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LIST OF ABBREVIATIONS

AAM .... Advanced Air Mobility

AFP .... Automated Fiber Placement

ATL .... Automated Tape Laying

ATP .... Automated Tow/Tape Placement

CFRP .... Carbon Fiber Reinforced Polymer

EASA .... European Union Aviation Safety Agency

FAA .... Federal Aviation Administration

FUBACOMP .... Full Barrel Composite Fuselage

HTC .... High Temperature Cure

LAA .... Los Angeles Airways

LTC .... Low Temperature Cure

OoA .... Out-of-autoclave

PAEK .... Polyaryletherketone

PEEK .... Polyetheretherketone

PEKK .... Polyetherketoneketone

PPS .... Polyphenylene Sulfide

RKAFP .... Reverse-Kinematic AFP

RTM .... Resin Transfer Molding

TP .... Thermoplastic

TPC .... Thermoplastic Composites

TS .... Thermoset

TSC .... Thermoset Composites

UAM .... Urban Air Mobility

VBO .... Vacuum Bag-Only

CHAPTER 1 INTRODUCTION

The effort to pursue sustained performance improvements in the field of aerospace engineering drives the development of manufacturing methods for advanced materials. Carbon fiber reinforced polymers (CFRPs) represent a composite material which continues to be the dominant material of choice within multiple industries, including the aerospace industry. CFRPs components outperform their metallic counterparts with greater strength-to-weight ratios, better corrosion properties, internal structure (e.g., the layup), and their malleability (the possibility to have composites tailored designed for specific load cases and applications). In Section 1.1, the types of polymer resin systems are reviewed, and which advances can be made concerning urban air mobility. Next, the urban air mobility (UAM) market is briefly introduced in Section 1.2, during which the need for high-rate manufacturing and the research focus of this thesis becomes apparent. Finally, in Section 1.3, the need to evaluate the redesign for thermoplastic (TP) materials and processes is highlighted.

1.1 POLYMER RESIN SYSTEMS

CFRP composites consist of two types of distinctly different constituent elements: a reinforcing carbon fiber structure, and a matrix fill. The type of polymer used for the matrix can be divided into two characteristic groups: thermoset (TS) polymers and TP polymers. The choice to select one of either polymer systems will allow the resulting

composite certain advantages and disadvantages, affecting the thermal, mechanical, and chemical performance of the composite, and the manufacturing strategies to produce the composite.

Thermoset composites (TSCs) have a much longer history of utilization in the aerospace industry and remain primary the polymer choice for CFRPs. Thermoplastic composites (TPCs) are far from a novelty material system, yet still do not have the same level of utilization as TSCs. The justification for using TPCs over TSCs is formed by highlighting associated advantages which can offset the processing favorability of TPCs.

1.2 URBAN AIR MOBILITY

The UAM (also referred to as advanced air mobility, (AAM) transportation market has been evolving from concept to near reality over the last decade. The UAM market envisions vertical-takeoff, short-range airborne travel of two to six passengers to replace and/or supplement conventional means of transportation, like trains, busses, taxis, etc. UAM aircraft would function like existing services such as trains, busses, and taxis, traversing a specified route for single to small groups of passengers. Most companies developing their own concept aircraft have opted for two additional key design features: (i) the aircraft be powered by electric batteries; and (ii) the aircraft must be capable of autonomous flight, increasing the number of paying passengers by reducing the need for a pilot. UAM aircraft are commonly referred to as eVTOLs, which is an abbreviation for electric vertical take-off and landing. Beyond these key similarities, aircraft configurations vary by company, producing concept vehicles with different ranges, passenger capacities, and method of lift and thrust generation. Some UAM craft based on different

configurations are pictured in Figure 1.1 which includes Lilium Jet (left), Joby Aviation's eVTOL (middle), Volocopter's Volocity (right).

Figure 1.1 Various UAM craft concept renders [1]

The emerging UAM market has profited largely from partnerships with composite material suppliers. Some examples include Joby Aviation (Santa Cruz, CA), and Lilium Jet (Munich, Germany) which have chosen to partner with Toray Industries (Tokyo, Japan), along with Vertical Aerospace (VA, Bristol, U.K.) forming an agreement with Solvay Composite Materials (Alpharetta, GA) [2] [3] [4]. [1] These companies are stressing the importance of composite utilization in the airframe, citing range and speed maximization [1]. This runs parallel with the continuing increase of composite material deployment in aerospace applications, with the Boeing 787 and Airbus A350XWB exceeding 50% composite materials by operational empty weight, the latter of which is planned to further increase composite weight fraction to 53% [5] [6] [7].

The companies developing eVTOLs base their business models on helicopter airlines of the last century: provide a service to bypass congested roadways over short distances. Helicopter airlines first saw service in 1954, provided by Los Angeles Airways (LAA), a company originally focused on providing airborne mail delivery between key mail hubs of the greater Los Angeles area, shown in Figure 1.2 [8], but later pivoted to

passenger service. In essence, LAA can be seen as the precursor to modern commercially available short-range rotorcraft travel, an industry UAM would eventually seek to challenge [8].

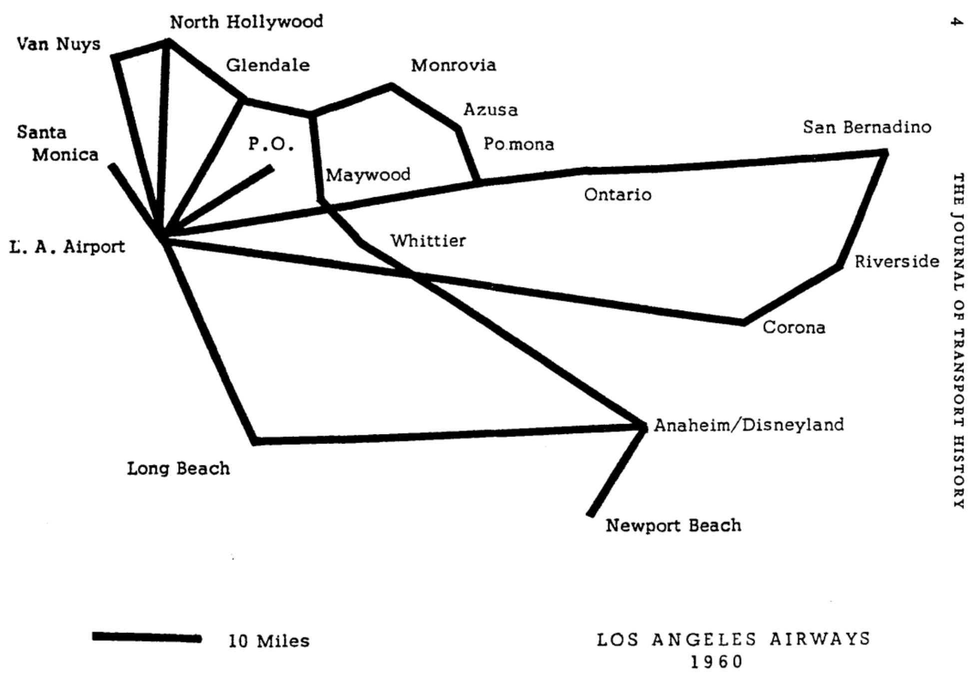

Figure 1.2 Los Angeles Airways service map [8]

1.3 REDESIGN FOR THERMOPLASTICS

For UAM companies to succeed it is crucial that they produce their aircraft in large quantities at a manufacturing rate comparable to the manufacturing rates seen in automotive industry. Most of the UAM aircraft designs use a multi-rotor configuration, the propellers must be manufactured at a high rate, repeatedly and reliably. To meet the required high production rates, a shift from TSCs to TPCs is explored. High-rate manufacturing techniques such as stamp forming can be used on TPCs, and this in combination with the fact that TPCs can be locally melted and reconsolidated without

negatively impacting strength, can enable the high-rate manufacturing UAM industry is looking for. The latter, the possibility of locally melted and reconsolidated TPCs, allows for fastener-free composite assemblies (reducing structural weight), recyclable structures, and improve the repairability of the composite structures. The transition from TSCs to TPCs also requires a transition in design and manufacturing methodologies.

The work presented focuses on the design and manufacturing of a TP propeller blade, a component that must be manufactured in large quantities. With the propeller blade as a demonstrator, the following research questions are addressed in this thesis:

- *What adaptations must be made to the design of propeller blade to transition from a TSC to a TPC material?*
- *Which manufacturing and assembly technologies must be used to increase manufacturing rate and reduce structural weight?*

To answer the research questions, this thesis is structured as follows: a review of the state of the art in CFRP manufacturing, with emphasis on comparisons of TP and TS manufacturing methods, followed by a detailed record of the process followed to adapt a TS propeller blade design to be manufactured using TP materials. A discussion of the manufacturing outcomes is given following the manufacturing record and final conclusions and future work remarks are made to resolve the research questions related to this thesis.

CHAPTER 2 STATE OF THE ART

The background of the different topics discussed in this thesis is provided in this chapter. This includes, but is not limited to, a comparison between TS- and TP-resin based composite material, and different composite manufacturing, curing/consolidation, and joining methodologies. The comparison includes an analysis of the individual polymer matrix characteristics and their implications on the composite material performance and manufacturability. The background on which later chapters are built and the foundation to discuss merits of TPCs as an alternative to TSCs.

2.1 CARBON FIBER REINFORCED POLYMER COMPOSITES

The production of CRFP composites represents a class of manufacturing material gaining receiving increased attention. CFRP composites are found in applications across various industries by replacing existing conventional building materials, such as wood and metal alloys in both aerospace and automotive structures [9]. For decades, substantial research has been conducted to further understand, develop, and improve the performance of composite materials. A significant interest has been the separate development of polymer resin systems used as the matrix constituent in composites. The choice of resin has an enormous impact on the specific performance of a composite, providing desirable mechanical (e.g., strength, stiffness) and chemical (e.g., corrosion resistivity, thermal stability) properties and aiding in the manufacturability of the composite.

There are two main types of polymer resin systems chosen for CFRP matrices: TP polymers and TS polymers. The differences in the chemical structure of these polymers have implications on the methods of design, manufacturing, cost-, and structural utilization. Specific implications can include the ease of infusing the resin system within the chosen fiber reinforcing structure, the ease of manufacturability, the processing parameters and capital investment required to stack and cure the laminate, the cycle cost, and the ability of the final composite to perform under the necessary structural, chemical, and environmental requirements. Typically, the choice of polymer is not only affected by the direct performance of the resulting laminate, but also includes the cost related to the effort, equipment and material associated with each method, especially when comparing the cost of integrating a new polymer resin system against maintaining an existing system.

2.2 THERMOSET COMPOSITES

The most common distinction made between TS and TP polymer structures comes from the crosslinks formed in TS polymer chains during the curing process. Curing is the chemical process by which a polymer is rendered into a final, solidified form. This difference defines the TS polymer at the resin level, the laminate structure, and manufacturing approach.

2.2.1 THERMOSET RESIN SYSTEMS

The physical properties of TS polymer, both before and after curing define how it is utilized in the composite manufacturing process. The class of TS polymers includes common advanced and engineering plastics such as epoxies, polyesters, polyurethanes, cyanate esters, polyimides, vinyl esters, phenolics, and silicone. Applications for TS resins

vary based on mechanical and thermal requirements, with some polymers strong enough to withstand aerospace structural requirements while other polymers are better suited for lower-intensity applications with other performance advantages. One example is phenolic composites which are well-suited for applications such as aircraft interiors due to favorable flame, smoke and toxicity properties [10]. When referring to TSCs in this study, epoxy will be the matrix material used as baseline for the comparisons. This is due to the widespread use of epoxy in primary aerospace structural composites thanks in part to wide range of operating temperatures, compatibility with toughening agents and the ability to manipulate the curing process to control cross-link formation [11].

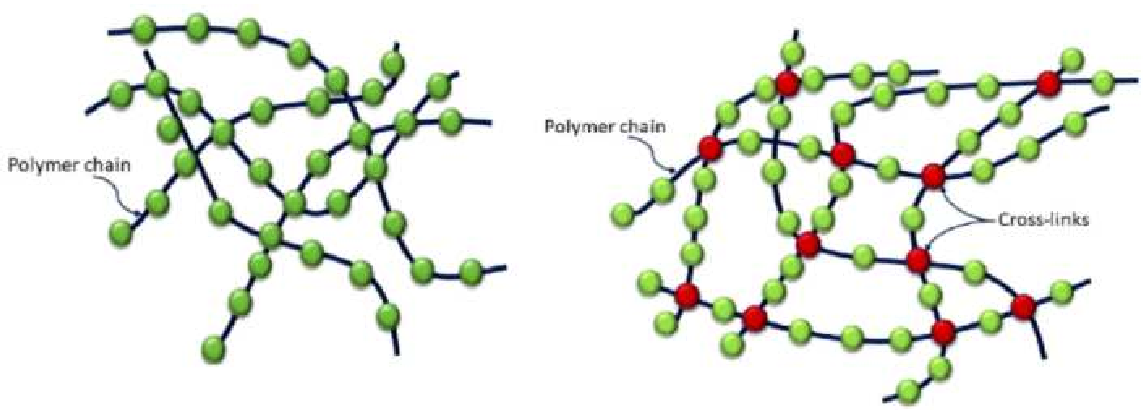

Figure 2.1 Thermoplastic (left) and Thermoset (right) molecular structures [12]

TSCs have a much longer history of utilization in the aerospace industry. At room temperature, most TS polymers are either liquid or easily melted solids with a viscosity that makes the polymer desirable for simple infusion methods, such as resin transfer molding (RTM). The ease of epoxy-based infusion provides and extreme advantage for composite designers to incorporate post-layup resign impregnation, something that is less

than desirable when using TP resins [13]. TS polymers are characterized by their cross-linked molecular structure which is formed by chemical reactions during the curing process which renders the polymer mass into a strong solid form. The formation of crosslinks is irreversible, similar to that of constituent colors of a painting which cannot separated once mixed and dried. The presence of crosslinks in a polymer structure changes the nature of the polymer. Crosslinks prevent atomic and molecular movement within the polymer. The glass transition temperature (T_g) of a polymer increases when crosslinks are present. The rise of a polymer's T_g is used to predict the degree of crosslink formation that has occurred in the polymer [10].

Another effect of the crosslink formation is the change in molecular weight. Crosslinks connect polymer molecules together, increasing the molecular weight with each link. Changes in the molecular weight of a polymer affect the temperature at which the polymer melts. Prior to crosslinking, as a polymer's temperature increases, the melting point is reached, and the polymer becomes a liquid. Further increasing the temperature at the liquid state eventually renders the polymer in a gaseous state. This transition is referred to as the decomposition point as passing this temperature range causes the polymer to degrade, shown in Figure 2.2. The presence of crosslinks not only greatly increases the melting point of the polymer, but also increases the melting point beyond the decomposition point. Practically, this means that the cross-linked polymer will decompose before melting occurs which is a departure from typical phase change progression and earns cross-linked polymers the "non-meltable" moniker [10].

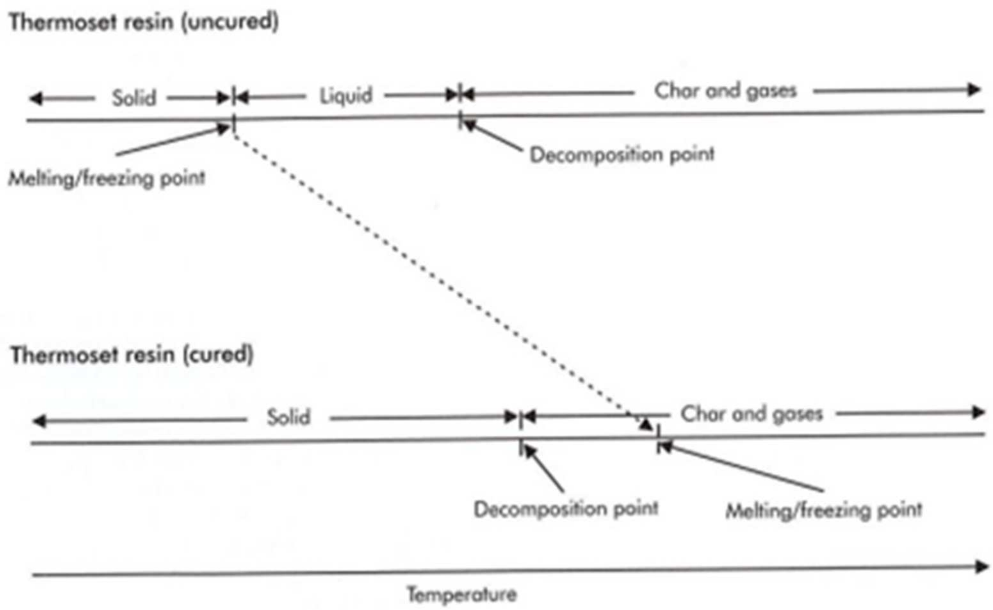

Figure 2.2 Comparison of phase changes in uncured and cured thermoset resin [10]

A major drawback of TS polymers is the inability to be reshaped after curing. While this presents some challenges in the layup and joining stages, it also makes the material difficult to dispose of. In one example, wind turbine blades, many of which are currently made from TS composite materials, represent exceptionally large structures currently in use. Upon retirement, smaller composite parts could be sent to landfills. However, in the European Union, the Directive on Landfill Waste has banned the disposal of oversized composite trash as large as these blades. The next logical disposal option is to recycle the material for a secondary use, however recycled TSCs cannot usually be melted, remolded, or reprocessed [14].

There are several TS polymers that populate the composites market. Some TS polymer families such as polyesters experience wide utilization across the composites market, however, due to lesser performance capabilities, cannot be used for high-

performance applications. For this need, manufacturers turn to epoxy as a low-cost material which is extensively used in high-performance applications alongside carbon fibers as the reinforcing material. Epoxy resin is cured by chemical reaction with amines, anhydrides, phenols, carboxylic acids, and alcohols. Epoxy variants can be mixed based on the desired application. Epoxy recipe changes/variations can modify the cure rate, required processing time, drape and tack of the material. Additionally, toughness and temperature resistance can be improved using recipe optimization and using TP additives.

As resins, epoxy polymers are made from a polymer branch chain, a three-member ring epoxy group (oxirane group) and an organic component. Epoxy rings are found at the ends of the epoxy polymer. Epoxy rings are the locations where cross-linking occurs. It is important to note that cross-linking is a product of curing the resin, as opposed to synthesizing the resin, or polymerization. Epoxies are synthesized by condensation reactions which also produce byproducts which must be removed from the uncured resin. Cross links in the polymer are formed by chemical reactions. The maximum cross-linkage of an epoxy polymer is dependent upon how many ends the branch chain possess which an epoxy ring can be attached to, increasing the cross-linkage. Higher cross-linkage can increase the thermal tolerance of the resulting resin [10].

2.2.2 THERMOSET COMPOSITE PERFORMANCE

The chemistry of cross-linkages precipitates predictable thermo-mechanical properties in TS laminates. Because of the alternative phase change behavior, TSCs are favorable for high-temperature applications. The cross-links within the composite matrix also lend to a stiffened surface, which presents challenges for TS laminates distributing impact loads [15]. Epoxy composites are recognized for inherent drawbacks such as low

toughness and crack resistance. This results in typical failure presentation such as interlaminar fracture due to impact damage, void growth, and delamination which create shear stresses in the material [16] [17]. A subsequently damaged composite panel will quickly succumb to fast crack propagation under dynamic loading [18]. TSCs also must contend with weakening due to fluid absorption. A small number of studies have made note of weakening toughness in epoxy-based composites when exposed to environmental fluids, such as water and hydraulic fluids [19] [20] [21].

While 80% of the plastics industry relies heavily on TP polymers, TS-based fiber reinforced composites represent the polymer composite market majority, surpassing 90% of the CFRP market [22]. The global market for TSCs is expected to reach $54 billion by 2026 [12]. The current state of TSC manufacturing represents almost a century of research, development, and industry reliance.

2.3 THERMOPLASTIC COMPOSITES

TPCs represent the other side of the polymer matrix composite coin. TP resins find use as both the defined subset of polymer matrix composites as well as additive agents for TS resin systems, working to toughen the resin mixture and increase delamination resistance [23] [24]. TPCs are defined in contrast of TSCs by having valuable damage tolerance, solvent resistance, unlimited shelf life, reprocessability, and high impact resistance [25]. TPCs also present the well sought-after potential of a process known as “in-situ” manufacturing, in which a TPC laminate can be cured during the layup process as opposed to the typical separate curing stage that follows layup. The possibility for the significant cost reduction found in the opportunity of in-situ manufacturing and fusion bonding drives much of the interest in TPC manufacturing development [26].

2.3.1 THERMOPLASTIC RESIN SYSTEMS

TP resin systems are characterized by “loose” or straight-link polymer chain structures. TP polymers are completely formulated or polymerized prior to contact with fiber reinforcement, meaning that the chemistry of the polymer does not change once infused. TP polymers are characterized by strong carbon bonds which give the resin system a considerable amount of strength. This is significant as the TP polymer is strong in the base form while TS polymers gain strength, not from the existing chain structure, but rather the curing process and the formation of cross-links. The structural differences illustrate why TPCs require greater heating and pressure requirements to consolidate within the laminate, as the energy needed to break and reform the carbon-carbon and carbon-hydrogen bonds is greater than the energy needed to initiate the TS curing process [10].

The curing of TPCs is different than the curing of TSCs due to the respective post-cure linkage structure. The term “curing” can be applied to either TS or TP polymers, however, a more proper term for the TP process would be “consolidation” since the polymer mass is not changing, only being redistributed throughout the fiber arrangement.

The list of TP polymers considered for high-performance composite matrix applications is far shorter than that of the list of TS polymers used for the same purpose. All TPs considered for high-performance applications share a high degree of aromaticity. When selecting TP polymers to utilize, Polyphenylene Sulfide (PPS) Polyetheretherketone (PEEK), Polyaryletherketone (PAEK) and Polyetherketoneketone (PEKK) are common choices for high-performance applications [10].

TPCs do not experience considerable property change during consolidation and thus characteristically have lower T_g values than aerospace-quality epoxy resins. While, as shown in Table 2.1, toughened epoxy resins may not have a significantly higher T_g value when compared to high-performance TP resins such as PEEK, the difference in behavior at and above the T_g value may influence the selection of one resin system over another. The second considerable difference between the two resin systems, also shown in the Table 2.1, is the difference in ductility. Epoxy resins are known to be strong, but stiff and brittle, while TP resins see much more elongation prior to fracture. This is important for composite designers which may need to minimize elastic deformation. Advanced TPCs exist as a favorable option for working environments full of heat and moisture as most TP resins boast a much lower water absorption than TS resins [10]. In general, TPCs are known for high productivity, high damage tolerance, high impact resistance, and recyclability.

Table 2.1 Comparison of example TSC and TPC physical properties [10]

Property	PEEK	PPS	Epoxy (toughened)
Density (g/cm^3)	1.32	1.34	1.3
Melting range ($°C$)	340-380	280-290	N/A
Tensile Strength (MPa)	90-100	75-85	60-80
Young's Modulus (GPa)	4	4	3-4
Elongation at break (%)	40-50	5-10	2-8
Maximum Water Absorption at Room Temperature	~0.5	0.05	1-5

2.3.2 THERMOPLASTIC COMPOSITE PERFORMANCE

TPCs have some advantages over TSCs, largely attributed to the reformability of the resin. Where TSCs are strong yet brittle due to crosslinking, TPCs are more ductile leading to greater material toughness and impact resistance. TPCs also have greater

fire/smoke/toxicity resistance when compared to epoxy TSCs [27]. Although not directly to this study, TPCs also exhibit fewer anisotropic mechanical properties when in woven form, moving away from the traditional anisotropic nature of composites when compared to traditional aerospace materials such as metal alloys [28]. The greatest challenges for TPC performance come from heating requirements and certification bias. Currently, epoxy-based composites have the advantage to operate in temperatures which exceed TP resin operating parameters. This is again due to the difference in thermophysical changes each cured resin system progresses through as increasing heat is introduced. The second challenge is certification, while research progresses to improve TPC production methods, the last hurdle to overcome is validation. As with aircraft structure designs, aerospace materials must be certified and validated by the relevant regulatory body. This can be especially challenging for composite designers as each new resin formulation presents a newly uncertified material which must be rigorously tested prior to approval by either the Federal Aviation Administration (FAA) for the United States or the European Union Aviation Safety Agency (EASA) for the European Union.

2.4 COMPOSITE LAYUP METHODS

Composite layup is one of the key fabrication methods used for producing composites components. As with other manufacturing industries, the final product can be produced using multiple different fabrication approaches. Some composite fabrication methods such as pultrusion and extrusion are excellent choices when a key need of the manufacturing process is a long component with a constant cross section. One other method, fiber winding, provides an effective processes strategy for final products such as

pressure vessels. However, when complex surfaces are needed composite layup is the key choice. The process of laying down layer by layer of composite laminates allows for complex surface fits, a wide range of part sizes, and close control of the fiber structure throughout the laminate.

When manufacturing composites, materials suppliers provide two different classes of material based the needs of the manufacturer. These classes are differentiated by the presence of uncured resin in the fiber structure. The shorthand name for materials which have been pre-impregnated with resin is "prepreg" while materials that are delivered devoid of resin are known as "dry fiber." Dry fiber materials require an additional step in part manufacturing as the fibers structure of the laminate is impregnated with resin after layup using methods such as RTM [29]. Both classes of material are used extensively in the aerospace composite field, however the comparisons drawn in this review will strictly relate to prepreg materials as no dry fiber materials were utilized during the manufacturing process of the propeller blade demonstrator. The impregnation of TP resins into fiber structures can increase the difficulty of manufacturing as opposed to TS resins which are equally suited for pre-layup and post-layup impregnation. For this reason, TPC materials are more likely to be utilized in prepreg form.

2.4.1 COMPOSITE HAND LAYUP

Composite hand layup is possibly the most labor-intensive composite fabrication method. Using hand layup, the stacking of composite laminates is handled strictly by a team of technicians carefully arranging each layer. In an increasingly automated world, hand layup still maintains relevance. The advantages and disadvantages between hand

layup and automated layup closely mirror the elementary comparisons drawn between 3D printing and injection molding, respectively. Hand-layup provides a low startup cost method for producing new composite parts during the development stage of a design. The absence of robots and other automated machines allows for easily applied variability of the design with much less preparation. Hand layup, shown in Figure 2.3, is also optimal for low-output manufacturing lines. The cost of automation and machine maintenance may pale in comparison to the profit of producing a large output of components, but for a small manufacturing environment, the cost-benefit analysis of implementing expensive manufacturing solutions justifies relying on manual processes. Hand layup also allows for the use of composite fabrics, or weaves, which are multi-directional layers of material produced by a material supplier and impossible to exactly reproduce with an automated layup process.

Figure 2.3 Blue Force Technologies hand layup [1]

2.4.2 AFP/ATL COMPOSITE PRODUCTION

The composite layup process can benefit significantly from automation for complex surfaces and high volume design cases. Automated layup is divided into two different types of machines: Automated tape laying (ATL) machines, and automated fiber placement (AFP) machine. Both machine types can fall under the general automated process known as automated tow or tape placement (ATP). The difference between machines is the compatible material width for the machine. Material produced for ATL machines is commonly referred to as “tape” and is made to have a width of up to 12in. Material produced for AFP machines is referred to as “slit-tape” or “tows” as material spools are often slit from the original width to the compatible width, typically between 0.25in and 1in [30]. Though the material width is reduced for AFP layup, AFP deposition heads are designed to layup multiple tows in parallel [5].

With some differences, the process of layup using either AFP or ATL is remarkably similar. While automated layup machines may vary in configuration, however, each manufacturing system will consist of the same key mechanisms. There are two main locomotive configurations for AFP/ATL machines: gantry or robot. Gantry systems utilize linear rails for movement while robotic systems rely on an robotic manipulators to perform the same movement [31]. Every machine must have a mechanism for mounting and unspooling preform material. The design of the material storage unit reflects the pot-life requirements of the material to be stored, which will be discussed in the next section. The crucial element of the AFP/ATL process is the deposition head. The deposition head houses the feeding and cutting mechanisms which applies tension to the unspooled material and cuts the tape/tow(s) once the material has been deposited. The head also contains a form of

compaction mechanism which applies pressure to the tool during layup. Additionally, any AFP/ATL head will also contain a heating element. The heating element works to improve the bond between layers of the laminate as it is laid up. Heating elements vary in design, with popular choices including laser heating, infrared heating, hot gas torch heating, and more recently flash lamp heating. Similar to the choice of material storage design, the heating needs are driven by the material of choice. The proper heating of a part substrate can greatly improve the final composite material performance by reducing layup defects such as high interply void presence [32]. As will be discussed in the next section, TSC layup can be accomplished with minor applied heat, while TPC processors look for methods which can produce large amounts of heat.

Because of the working material width, ATL is best suited to produce large components with less surface complexity than components better suited for AFP layup. For this reason, AFP layup receives significantly more development attention from the aerospace industry. The separation of a single, wide material sheet into multiple smaller individual tows promotes improved surface conformity and reduction of material waste.

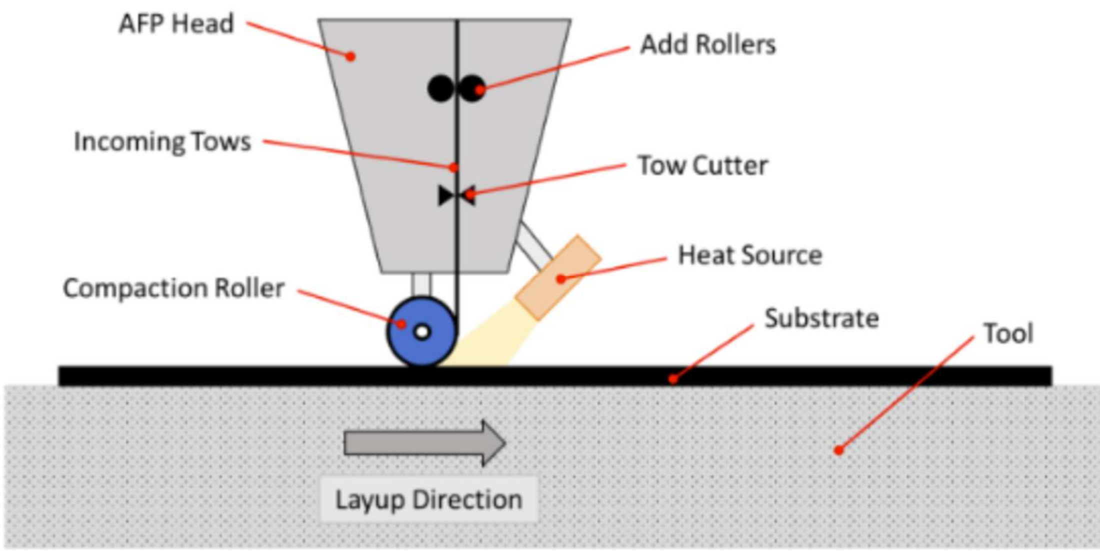

Figure 2.4 Typical AFP process [31]

AFP is not a perfect process. AFP layup suffers faults similar to other automated processes adapted from manual processes. The unspooling of tows from the material spools and feeding and cutting during layup can provide challenges for reliable layup, regardless of the material choice. Conventions must be taken in the design of a machine such that the mechanisms within the feed and guide assembly are best mated with the material. This may include employing guide surfaces that TSC tows are less likely to stick to or a feeding mechanism that provides constant tension on TPC tows.

2.4.3 THERMOSET AFP

Being the material, more effort and experience has been invested in improving automated TSC layup, with TS layup relying on the characteristics of the uncured resin in TSC tows to facilitate interlaminar bonding during layup. Accommodation for the usage of TSC material in automated layup begins with the material storage. As previously noted,

TSC material has the disadvantage of a limited usable life. In the case of Hexcel's HexPly M21, an epoxy matrix that is used in the wing box construction of the Airbus A350 [33], the material life for the uncured epoxy in three different states is defined in Table 2.2 and Table 2.3.

Table 2.2 Conditional lifespan definitions for TS resin [34]

Type of Life	Definition
Tack Life	The time, at room temperature, during which prepreg retains enough tack for easy component lay-up
Out Life	The maximum accumulated time allowed at room temperature between removal from the freezer and cure.
Shelf Life	The maximum storage life for HexPly® M21 prepreg, from date of manufacture, when stored continuously, in a sealed moisture-proof bag, at -18 °C (0 °F). To accurately establish the exact expiry date, consult the box label.

Table 2.3 Conditional lifespans for HexPly M21 resin [34]

Type of Life	Life Span	Required Temperature (°C)
Tack Life	10 to 15 days	23 (room temperature)
Out Life	30 days	23 (room temperature)
Shelf Life	12 months (from date of manufacture)	-18

The flexible nature of TSC prepreg material lends towards a number of potential defects during automated tow layup. Layup defects present on the surface of a laminate in variations of longitudinal folds, deviation from the tow path, loss of interlaminar bonding, and partial or complete dropping of a tow. A review by Chevalier et al. considers a number of defect studies to examine the relationship between AFP layup defects and reduced mechanical performance. The review found that the evidence supported a conclusion that

the observed defects had a greater negative impact on fatigue strength than quasi-static strength [35].

2.4.4 THERMOPLASTIC AFP

TP-AFP layup, much like the utilization of TPCs, is a developing topic of composite research. A key challenge in the adaptation of AFP technologies from TSC layup to TP-AFP layup is the applied heating necessary for interlaminar bonding. While TS-AFP is not completely dependent on the AFP machine's applied heat to achieve some level of interlaminar bonding, TP layup is entirely dependent on the ability of the heat source to melt the resin, promoting interlaminar bonding. This may involve an increase of applied heat from TS layup by over 100°C. Process parameter development is needed for this topic as a favorable relationship between laying speed and applied heat is needed to remain profitable. The laying speed must not be increased to such a degree that the applied heat is increased to such a temperature that causes material degradation.

An additional hurdle for TP-AFP layup is the the feeding mechanism must accommodate for the increased rigidity of the prepreg tows. Rigidity in the wound spool leads to the tendency of TP tows to unspool if not kept in tension, leading to uncontrolled unspooling of the material which demands a cycle stop and reset of the tow feeding system.

Although not a component of this study, TP-AFP cannot be discussed without the inclusion of in-situ consolidation. In-situ consolidation is the process by which a TPC laminate is cured during the layup, as opposed directly after using a separate curing process. The already substantial increase in heat from TS-AFP layup is further raised to

the temperature region nearing 600°C. As the processing temperature is increased, other parameters are modified, such as deposition rate and compaction force/pressure to mitigate the potential degradation of the polymer under such substantial heat [36]. With varying results, additional heating/compacting passes are made with the goal of improving interlaminar bonding. In-situ consolidation has been a recent topic of study in the composites research community; however, it is yet to find validation at the production scale. The promise of the pursuit and development of in-situ consolidation is a redefinition of the TPC manufacturing process in which massive potential cost reductions could be made through the elimination of costly autoclave curing cycles.

2.5 COMPOSITE CURING TECHNOLOGY

The greatest challenge in advocating for the use of TPCs over TSCs comes from the layup and consolidation stage. As previously discussed, TSCs need only be heated to the temperature necessary to promote the cross-linkage of the molecular chains in the TS resin. TPCs on the other hand must be heated to the melting point to achieve movement of the TP resin among the fibers. The most reliable methods of curing/consolidating CFRPs are large autoclaves, ovens, and compression molding machines. Each of these methods have advantages and disadvantages as reviewed in the following subsections.

2.5.1 AUTOCLAVE CONSOLIDATION

Large autoclaves that are designed to process CFRPs consist of a large, cylindrical chamber which can be precisely heated and pressurized. When curing/consolidating CFRPs, parts are set up in the autoclave chamber with necessary tooling and pressurized

under a vacuum bag. Once pressurized, the chamber is sealed and, a recipe of heating and pressurization is followed relative to the material system and laminate size before cooling, depressurization, and part removal.

Autoclave processing cycles have several advantages in curing/consolidating CFRP laminates. The use of a vacuum bag allows for complex curvature parts, such as wing skin and fuselage sections, to receive uniform pressure. Additionally, the large size of autoclaves allows for sections of full-scale passenger aircraft components to be processed, as opposed to processing smaller sections and assembling the structure, creating stress concentrations. A proper example of the size advantage is the case of the autoclave used to cure the center wing box of the large Boeing 787 Dreamliner.

Designed and partially built by the Taricco Corporation in California and assembled and operated by Fuji Heavy Industries in Japan, the autoclave designed to cure the center wing box of the Boeing 787 one of the largest autoclaves ever constructed at the time. The autoclave measures 23ft long with a 23ft diameter [37].

As this is the processing step for TSCs where molecular structure of the resin is changed, producing crosslinks, it is crucial that the curing recipe be carried out as stated by the material supplier. Here is there is a key tradeoff that cannot be unequivocally labeled as an advantage or disadvantage related to TSC processing. Using Hexcel's HexPly® M21 [34] as an example of an aerospace grade TSC, there are specified temperatures and heat-up rates which must be followed to achieve a quality composite. For a monolith part that is less than 15mm thick, the recommended recipe involves first pressurizing the autoclave to 7 bar and reducing the vacuum pressure of the part to -0.2 bar once the autoclave pressure

reaches 1 bar. Once pressurized, the autoclave is set to increase heat at a rate of 1-2°C/minute until a temperature of 180°C is reached, with a tolerance of 5°C. This temperature is held for 120 minutes before the part is cooled at a rate of 2-5°C/minute and the autoclave is vented once the part temperature reaches 60°C. This cure cycle is designed to both promote resin flow and the chemical reaction necessary to produce crosslinks in the resin. Hexcel's recipe allows for small 5-minute tolerance for the hold time; however, the overall recipe is still quite fixed. This is not the same for TPC consolidation cycles. As the word choice indicates, a consolidation cycle is only concerned with applying enough heat to the TP matrix such that the semi-cured resin can consolidate within the laminate, achieving even distribution throughout the plies. Therefore, either resin system may have an advantage as related to autoclave processing, as TSC's require strict curing cycles but use less heat while TPC's require more heat to reach consolidation temperature but have looser constraints regarding time held under heat.

2.5.3 COMPRESSION MOLDING

Two heated platens pressed together with the desired pressuring using a hydraulic press (a hot press) can be used instead of a vacuum bag and pressurized cell to cure/consolidate the composite laminate. Compared to autoclave processing, compression molding has the advantage of lower associated cost, high-efficiency, low stress, small bucking deformation, good mechanical stability, and excellent product repeatability [22].

Within the industry of CFRPs, compression molding processes fall into two subprocesses based on the form of preform material, either bulk charge or sheet materials. These processes are illustrated in Figure 2.5.

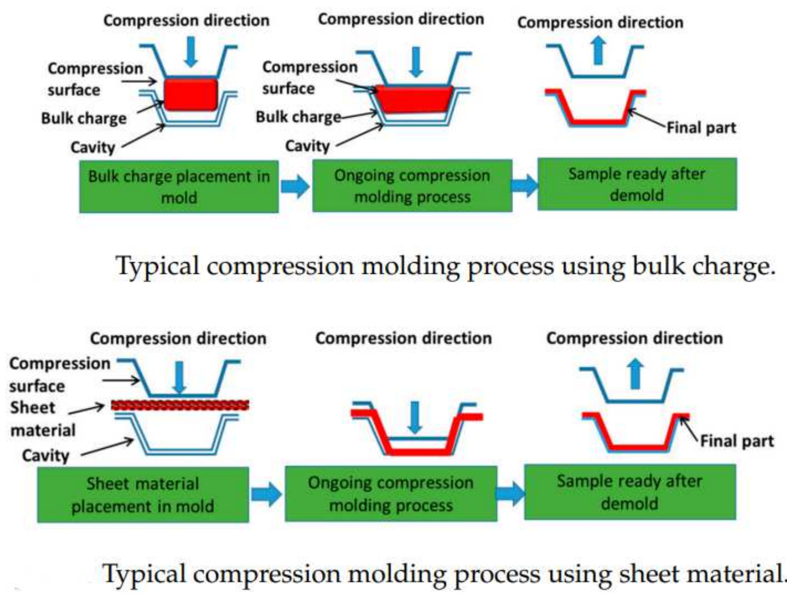

Figure 2.5 Typical compression molding processes [38]

Of these two subprocesses, molding sheet material is more applicable to parts produced by AFP. Geometrically, compression molding is limited to parts which can be cured with unidirectional pressure. While this constricts the diversity of parts that can be made by compression molding to a greater degree than parts that can be made by autoclave process, the compression mechanism has fewer points of failure compared to vacuum bag tooling, which is commonly a manual process and suffers from defects in the tooling setup.

2.5.2 OUT OF AUTOCLAVE TECHNIQUES

There is an additional method of curing that is not available to conventional TPCs and provides a significant advantage for TSCs. Out-of-autoclave (OoA) resins are TS resins which do not require an autoclave to achieve quality parts. The motivation of OoA resins

stems from the inherent high costs of setup and operation of an autoclave, as well as the associated tooling cost and the eventual limitation of the vessel dimensions as part sizes are increased. An OoA process relies on vacuum bag-only (VBO) curing and may receive the heat necessary to achieve a curing reaction from a number of non-autoclave sources, including heating blankets, conventional ovens, and heated tooling [39]. OoA may be used for prepreg composites as well as infusible resins paired with dry fiber laminates for first time cures as well as repair cycles. Additionally, a compromise between high temperature cure (HTC) resins and OoA resins are low temperature cure (LTC) resins, which do not completely eliminate the need for closed cell heating solutions (i.e., autoclaves) but do considerably reduce processing energy needed for achieving desirable cure characteristics [40].

A key concern of OoA processes as related to conventional autoclave and compression molding processes is the ability of OoA to produce parts without increased void content [41]. This can be mitigated by conforming to out-of-storage life spans for TSC prepregs properly managing off-gassing related to the curing reaction while resin, with some oven cured parts having negligible mechanical performance differences from autoclave cured parts. As with TPC processing methods, further research and development continues to improve OoA part quality within the composite manufacturing industry. One such example of this is the Quickstep method, an OoA process that works as a TS/OoA analogue to TPC compression molding in terms of expedited production. The challenge for the Quickstep method is precise control of the chemical reactions during cure while maintaining higher ramp rates in heating [42].

There is an effort to make exceptions to the exemption of TP resins from OoA processes. Several studies have focused on a new class of TP resin: Elium® liquid TP resin, developed by Arkema [43] [44] [45] [46]. Elium® resin provides a TP resin that is designed to circumvent the conventional challenges of infusing TP resins by delaying the complete formulation of the resin until after infusion has taken place. This achieved by a mixture mechanism similar to two-part epoxy resins that require a resin and hardener agent to cure and produce crosslinks. The Elium® resin is monomeric prior to infusion and when mixed with an initiator, solidifies the polymer [47]. Studies of this kind which highlight the possibly of an easily infusible TP resin could provide the means for further development if the cured resin can perform to the necessary mechanical standards.

2.6 COMPOSITE JOINING METHODS

Like metallic airframe structures, composite structures are assembled by joining individual components together. The methods by which composite components are joined belong to one of three methods: mechanical, chemical, or fusion bonding. Because fusion bonding is a method not readily available to TSCs, mechanical and chemical joining see far more industry utilization [48]. Each of these methods have advantages and disadvantages which has led to manufacturers favoring one method over others based on the application. The implementation of effective composite joining, especially in cases of retrofits and repairs is crucial as the composite joint represents one of the most common locations of failure in an aerospace composite structure [49].

2.6.1 MECHANICAL JOINING

Mechanical fasteners are joining tools that can be utilized for both composite and non-composite applications. The most common mechanical fastener used in aerospace applications is the rivet. Rivets can be installed relatively quickly using a two-step process which involves first drilling the clearance hole and then installing the rivet. The generalized geometry of a mechanically fastened joint includes at least two components with an overlapping area and a hole created through the components to accommodate the fastener. When considering the design of a mechanical joint in a composite structure, the main considerations are given to the joint configuration, and the fastener type and material. Mechanical fasteners have a reputation of reliability due to past use in metallic airframe structures, however in composite structures these same fasteners will encounter challenges [10].

Mechanically-fastened joint structures are typically designed to best support the localized stress concentration at the fastener location. These joint structures often include a variation of a lap joint. For a composite designer to utilize mechanical fasteners, the joint design necessary to support the fastener must also comply with the net shape of the assembly. Illustrated in Figure 2.6, a double lap joint offers a great reduction in the potential shear forces experienced at the joint. However, this is achieved by adding additional material to the joint, which may not be an option for designs set to minimize weight.

The size of the fastener is also a design measure that must be balanced. As logic would suggest, the size of a fastener increases the strength of the fastener. Simultaneously, increasing the drilled clearance hole necessary for the fastener decreases the strength of the laminate. This is because CFRP composites derive strength from the fiber structure and

severing fibers weaken the greater structure. For this reason, the shaft diameter of a fastener may not be so large that the laminates are weakened beyond an allowable measure. Also of importance is the head/end diameter of the fastener. The minimal head/end diameter must be large enough that the compressive force applied by fastening the joint together does not crush the internal plies of the laminate. A summary of advantages and disadvantages is shown in Table 2.4.

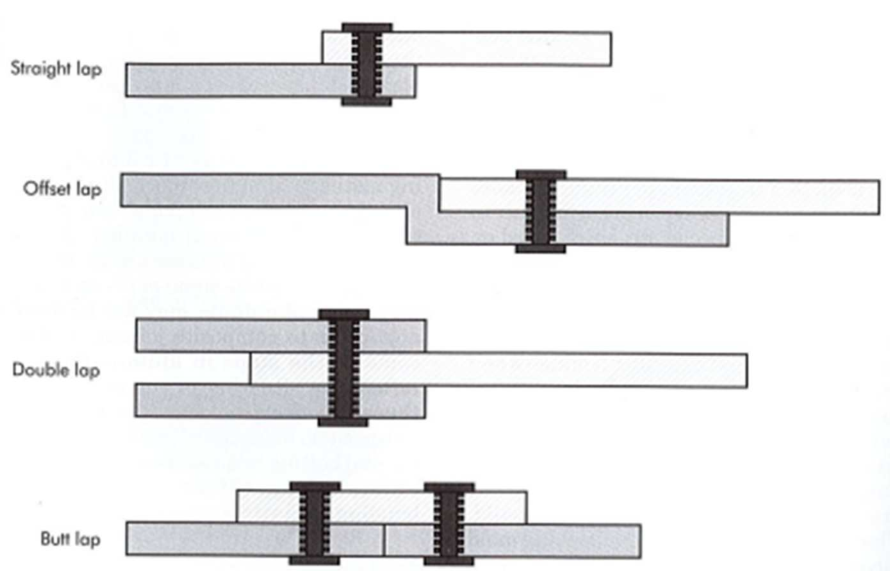

Figure 2.6 Examples of common mechanical joint designs [10]

Table 2.4 Advantages and disadvantages of mechanical joining

MECHANICAL JOINING	
ADVANTAGES	DISADVANTAGES
Low Creep	Drilling holes through composite laminates reduces the strength of the laminate
Stronger in peel loading than adhesive bonds	Applies compressive force through the joint
Easy to inspect/Possible to remove	Added weight to the structure
Little to no surface preparation	Fasteners may promote internal corrosion
Less sensitive to thermal, water and solvent based degradation	Time intensive process for manual or automated joining
Can join composite and non-composite structures together with negligible design variance	High stress concentrations at the fastener location

2.6.2 CHEMICAL JOINING

Chemical joining, also referred to as adhesive joining, is the process of mating two composite joints by way of an adhesive layer. Unlike mechanical joining, adhesive joining reduces the stress concentration by increasing the effectively bonded area without decreasing the strength of the laminate. Adhesives provide greater variability to the composite designer, both in joining materials and joint geometries. The geometric difference is most notably illustrated in the scarf and stepped lap joints shown in Figure 2.7, which are variations of butt joints with complex mating surfaces. These joints are able to avoid the shear stresses that lap joints experience by placing the joined laminates within the same plane rather than being offset. Chemical joining presents more challenges with TPCs than TSCs. Other than favorable mechanical properties such as impact resistance, TPCs are typically favored for having a much lower water absorption than TSCs. This advantage acts in the inverse when considering chemical joining. For an adhesive to

properly adhere to a TPC surface, surface treatments such as atmospheric plasma and ultraviolet light treatments are needed to properly prepare a surface when considering adhesive bonding [50].

When contrasting joining technologies, the greatest weaknesses of adhesive bonds can be derived from material surface issues. Adhesive bonding can become exceedingly challenging when heavy surface decontamination is needed. In a production setting, this is possibly the most important area that demand close quality control. It is also difficult to assess the performance of an adhesive bond through non-destructive testing methods. This has prompted the use other methods for surface analysis to extrapolate bond strength, such as optically stimulated electron emission and laser-induced breakdown spectroscopy [51]

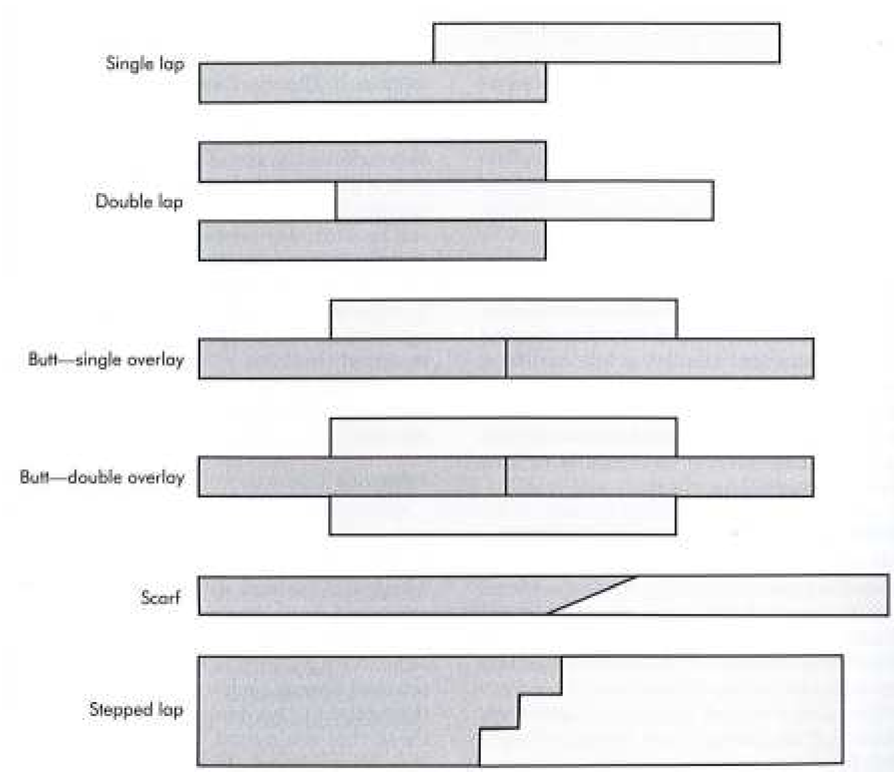

Figure 2.7 Examples of common chemically bonded joints [10]

The chemistry of each adhesive is chosen based on the performance requirements of the joint and the composite matrices of the laminates to be bonded. One example of this is in [14], which aimed to compare the performance of a two similar wind turbine blades, one made from an epoxy composite and one from a TP composite. A great effort was made during the production of both blades to keep them as structurally identical as possible. One key consideration given to the TP blade was that the epoxy adhesive, a common structural adhesive choice to use for epoxy composites, was found in single lap shear characterization to be not compatible with the TPCs and a polymethyl methacrylate (PMMA) adhesive was chosen to better represent how TP wind turbine blade would perform.

The application of adhesive bonds requires much more attention to detail than that of mechanical fasteners. Proper environmental control must be maintained throughout the bonding process to ensure that the adhesive does not become contaminated during the preparation and bonding process as well as that the chemical/physical state of the adhesive does not change prematurely. If the laminates to be bonded have already been shaped to fit the joint geometry, then the surfaces must be prepared for the adhesive. Typically, this means decontaminating the surfaces of any substances that may have accumulated either through storage or the manufacturing processes used to cure the laminates. This may include process by-products such as residual mold release agents utilized in the remove of the cured laminate from a mold tool. This decontamination may be achieved by cleaning with a solvent, after which a secondary cleaning agent would be used to remove any solvent residue. The surfaces would next be physically altered by a form of surface sanding to increase the roughness of the surfaces, promoting a greater bond adherence. After surface preparation is complete, the laminates may be aligned for bond application. There may be an intermediary step between surface prep and bond application in the event that the chosen adhesive is a two-part formulation, requiring an additional hardening agent which cures the adhesive when mixed. Typical applied adhesive layers are kept between .004in and .008in (0.1-0.2mm) [10]. To complete the joint, the laminates are joined together in a compressive tool to remove air from the bond and the adhesive is cured using machines which can also be used for laminate cures, such as compression molding machines and autoclaves. Similar to original assembly, adhesive joining may also be used for repair operations. While this is commonplace, adhesive joining has many complications that TPC fusion bonding methods do not, including complex instrumentation, potential damage of metal components due to

bulk heating and the inability to repair thick structures [52]. A summary of advantages and disadvantages is shown in Table 2.5.

Table 2.5 Advantages and disadvantages of chemical joining [52] [10] [53]

CHEMICAL JOINING	
ADVANTAGES	DISADVANTAGES
Low stress concentrations	Heavy reliance on careful and proper surface preparation
Low added weight to joint	Requires additional materials to bond with non-composite surfaces
Less sensitive to cyclic loading	Long cycle times due to multi-step processes and cure time
Bond strength does not reduce for thin laminates	Different composites matrices require different adhesives
Allow for end-to-end joints	May require high-temp tooling

2.6.3 FUSION JOINING METHODS

Fusion joining may be considered a type of chemical joining but can be differentiated by the lack of a foreign bonding agent. Fusion bonds are created by joining two laminates together using their common resin system. Depending on the resin system, this may be performed during or after a laminate has been cured/consolidated. Fusion joining can be attractive as a joining method because the process essentially creates a larger laminate joined by an unreinforced area, rather than multiple laminates joined with a foreign agent. Fusion joining also dismisses the need for post-cure shaping of joint areas, such as cut-outs, and minimizes conventional joint concerns such as questionable leak resistance in adhesive joints [54]. It is worth noting that TSCs are not completely excluded from the fusion bonding process, however when fusion bonding TSC laminates, TP films and hybrid interlayers are still required to create a strong bond. Efforts have been made on

this topic to maintain the relevance of TSCs in advanced composite joining technologies [55].

2.6.3.1 CO-CURING

The only fusion bonding process available to both TSCs and TPCs is co-curing. Co-curing is the process by which preform laminates with a shared resin system are set into fixtures such that during the curing process the resin flows between both laminates creating a strong bond at the interfacing surfaces. This method is an efficient assembly process for bonding stringer components to larger fuselage sections. As shown in Figure 2.8, this is achieved by applying opposing pressures against the bonding surfaces and also utilizing any internal fixtures needed to achieve complex geometries such as the hat stiffener in Figure 2.9. The advantage of co-curing here is that the multiple hat stiffeners can be co-cured to the larger fuselage section as shown in the second figure below, contrasting the two different tooling approaches to bonding multiple stiffeners [56]. A summary of advantages and disadvantages is shown in Table 2.6.

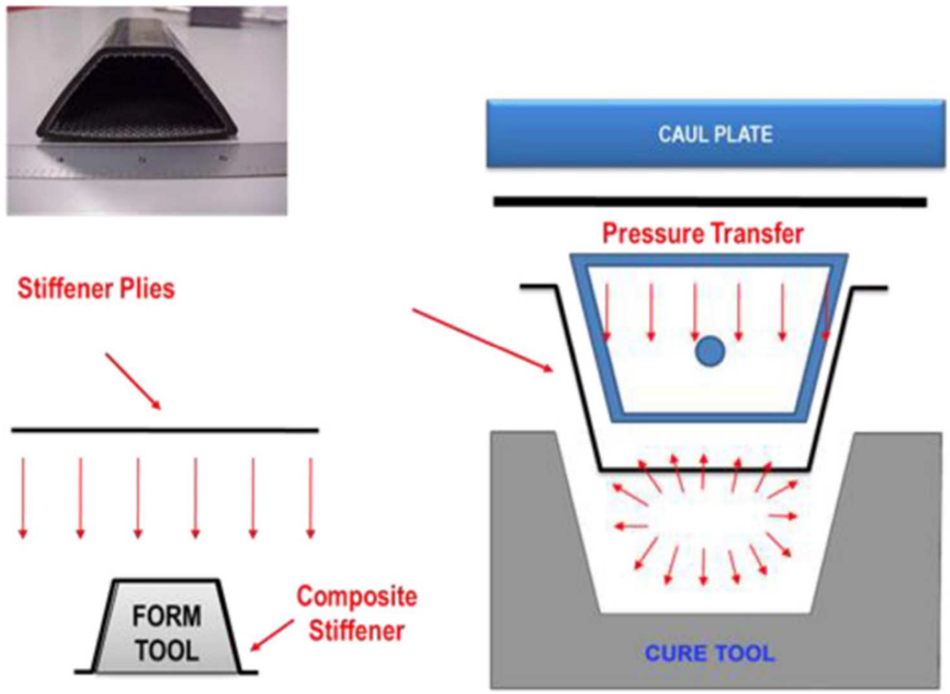

Figure 2.8 Co-curing process of a fuselage and hat stiffener

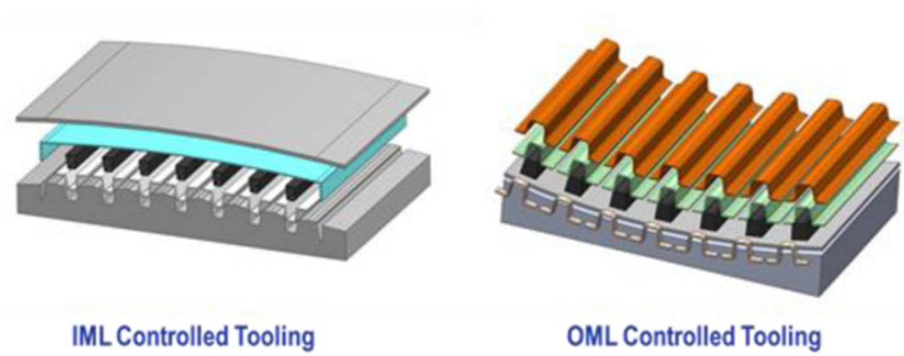

Figure 2.9 IML controlled tooling for co-curing manufacturing process

Table 2.6 Advantages and disadvantages of co-curing

CO-CURING	
ADVANTAGES	DISADVANTAGES
No cost of additional bonding materials	Complex Tooling Required
Strong bond strength	Challenges related to dissimilar matrices
No large stress concentrations	No method of inspection
No added weight	Longer processing time to promote resin flow
Little to no surface preparation	Challenge of uniform pressure application

2.6.3.2 THERMOPLASTIC BONDING METHODS

The ability to remelt TPCs allows for consolidated laminates to be joined by melting the shared resin at the interface of the laminates. The fusion joining methods yet to be discussed differ from standard co-curing in that the process does not necessarily require full-scale curing equipment such as an autoclave because laminates can be joined after consolidation has taken place. In a review [57], TPC fusion joining techniques were classified into three classes based on the type of joining used: (i) thermal welding; (ii) frictional welding; and (iii) electromagnetic welding. Subsets of welding methods associated with the three main welding types are listed in Figure 2.10. Advantages and disadvantages of fusion joining techniques are tabulated in Table 2.7.

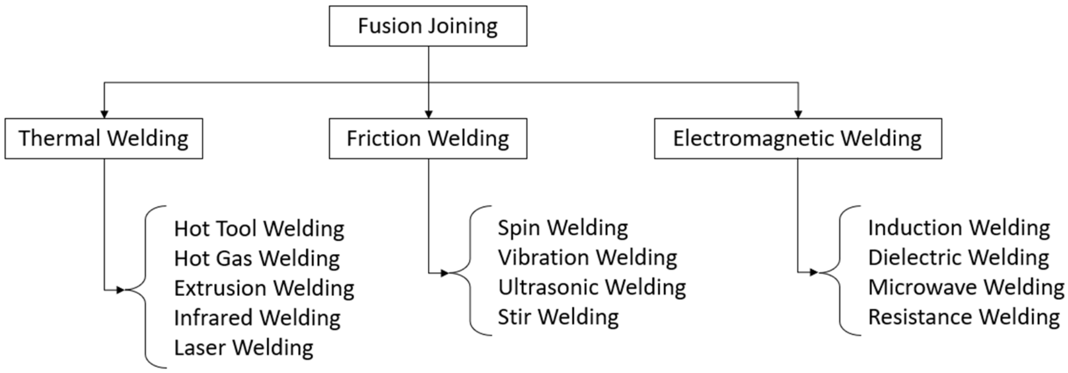

Figure 2.10 Fusion joining technologies [57]

Bulk heating relates to single-laminate consolidation methods that can be applied to TPC joining. Bulk heating includes any heating method which joins TPCs by applying equal heat to the entirety of both laminates. This method of fusion bonding has the advantage of relying on existing curing machines with only small tooling modifications. As with co-curing, complex tooling may be needed based on the geometries of the laminates to be bonded. Additionally, as the entire consolidated polymer body is being remelted, care must be taken so that no loss of polymer occurs through polymer squeeze out as the laminates are remelted and pressurized together at the joint surface. In some instances, a then TP polymer veil may be included between laminates at the bond surface to add additional bonding polymer. This polymer veil may not necessarily be made of the same resin system as the bonded laminates.

Frictional heating as a bonding method can be seen in the composites industry as well as the metals industry such as with friction stir welding. Frictional heating is the process by which vibrations are induced at the bond surface to promote semi-local heating thereby creating a bond. Friction heating is a common joining method for unreinforced

plastics; however, the induced vibrations can have a damaging effect on the fiber structure if the laminates are not designed for the process. Ultrasonic welding (high frequency, low amplitude vibration) (see Figure 2.11) is a preferred method of frictional heating for TPCs.

Electromagnetic heating for TPCs (see Figure 2.12) is most commonly seen in resistance welding and induction welding. Resistance welding relies on the inclusion of an electrically resistive element which is set in the contact area between constituent laminates. Current is run through the element, promoting heating of the bond area through direct conduction [58]. Induction welding (see Figure 2.13) works by creating eddy currents which are used to generate the required heat to melt the TP polymer. These currents are induced in the electrically conductive CFRP by a time-varying magnetic field [59]. Because polymers are not conductive, the carbon fibers within the laminate as well as optional susceptors are used at the weld surface to promote heating. Induction welding has several advantages as a bonding method, one of which is safety. As both an advantage and a drawback to the method, the effective range of the induction coil is typically short (less than 1in). While this poses challenges for complex geometries as the coil must be able to travel with a small variance as close to the weld line as possible, it also translates to a minimal safety risk to operators. Methods such as bulk heating or friction welding produce large amounts of widespread heat or involve high-speed components, warranting a wide safety radius. Automated induction welding may be carried out using robotic or gantry travel systems allowing for optimal repeatability. The challenge of induction welding is that variability in the melt behavior at the weld line is common. Intricate process parameter development is required to ensure constant bond success.

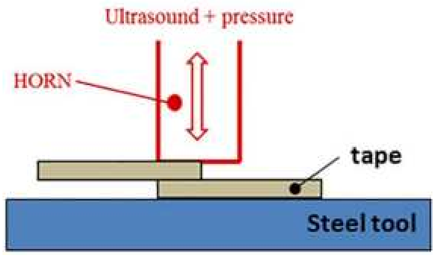

Figure 2.11 Ultrasonic welding sketch [60]

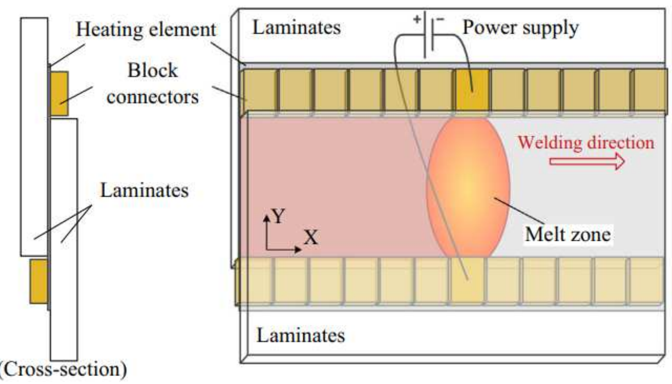

Figure 2.12 Continuous resistance welding joints [58]

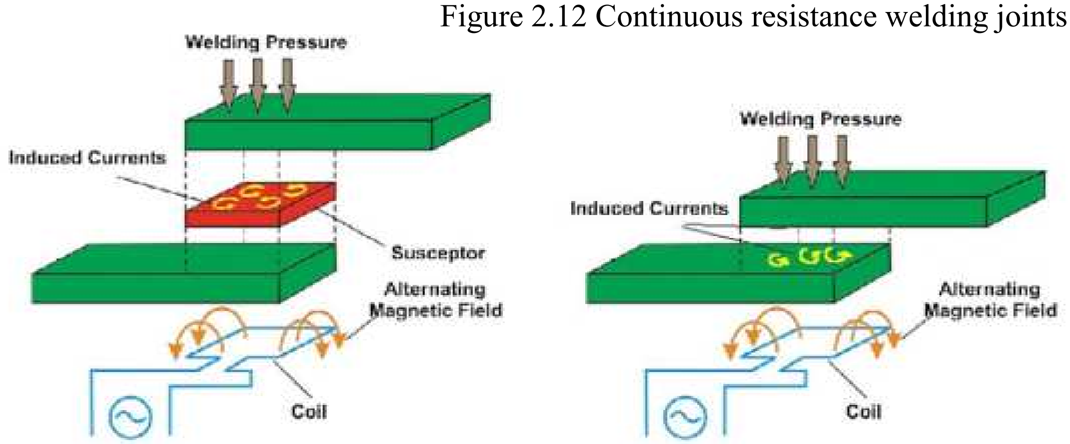

Figure 2.13 Induction welding processes: susceptor and susceptorless heating [61]

Table 2.7 Advantages and disadvantages of fusion joining

FUSION JOINING	
ADVANTAGES	DISADVANTAGES
Potentially no added weight	Parts must be designed for certain joining methods
Process is highly automatable	Unreliability of some methods
Processes can be applied to primary joining as well as repairs	Complex tooling required

CHAPTER 3 MANUFACTURING APPROACH

This chapter discusses the process steps to design, fabricate the laminates, and tooling produced to manufacturing the TP propeller blade demonstrator. First, the different stages of design process are discussed. Second, the methodology used to modify the component models, generate process plans, and design complementary tooling are presented. Third, and last, the manufacturing of the different laminates used for fusion joining of the TP propeller blade demonstrator is elaborated upon.

3.1 DESIGN DISCUSSION OF PROPELLER BLADE VARIENT

Several revisions were made to a generic TS blade model used for this research. While the propeller blade skin did not receive significant changes, the full spar could not be produced with the automated layup configuration used in this research setting. The full spar model followed the curvature of the blade skin into the blade's winglet structure. The first revisions made to the spar model were based on the decision that the spar laminate would be designed such that the necessary tooling would fit on a conventional rotary axis mandrel, mounted on either end of the tool. To comply with this decision, the spar model was trimmed to an arbitrary point closely resembling the point along the spanwise length that the spar cross-section began to no longer be centered on the relative spanwise axis. This ensured that the spanwise axis could be synonymous with the tool axis during layup process programming.

The equipment and software used during the manufacturing process is shown in Table 3.1. Each of the machines and software packages are recorded alongside the relevant application. Multiple methods of composite curing were utilized in the manufacturing process, necessitating the need for several different types of curing machines. Multiple design software packages were used to complement the available machine compatibilities.

Table 3.1 List of machines and software and respective applications

Machine/Software	**Application**
Dassault Systèms Catia V5	Composite surface modeling
Autodesk Inventor/Inventor HSM	Composite tool modeling
Haas VF-5/50 VMC mill	Precision composite tool fabrication
Ingersoll Machine Tools Lynx AFP	Automated layup
Heraeus Noblelight Humm3®	Automated layup heating
Gerber Scientific DCS 1500	Precision composite cutting
Wabash Genisis hot press	Composite compression molding
Wisconsin oven	Composite oven curing
Bondtech autoclave	Composite autoclave curing

3.1.1 SELECTION OF INTERNAL CONFIGURATION

Three internal configurations were considered for the TP propeller blade. Differences between configurations do not affect the design of the blade skin or the main spar. Each configuration is meant to represent general structures which would reinforce the TP propeller blade differently and require different assembly methods. The intention was to select an internal configuration with structure and manufacturing in mind.

The first configuration to be considered (Concept 1) was derived from the generic TS blade model. The internal structure consisted of a second and third spar connected in series to the main spar as shown in Figure 3.1. For Concept 1, all internal minor components (spars, ribs, and webs) are joined to the main spar at a single location, note all components are also joined to the skin panels. The minor spar structure would also take the shape of a diagonal and spanwise offset. This concept reinforces most of the skin surface while providing a clean load path to the root of the propeller blade.

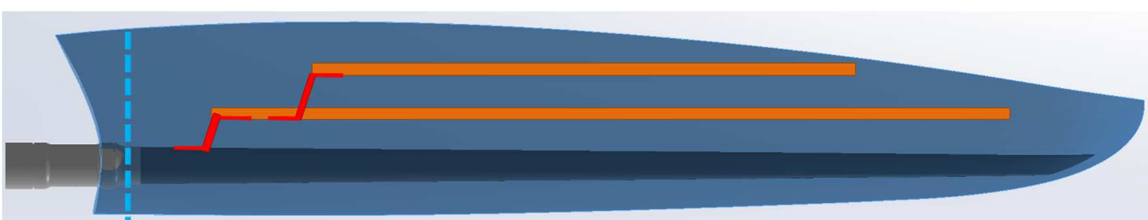

Figure 3.1 Concept 1 configuration

For the second configuration (Concept 2) the spar-based reinforcement was replaced by rib-based reinforcement (see Figure 3.2) . This configuration is meant to resemble the structure of natural inspirations, such as the reinforcing veins of a maple leaf seed pod. This concept maintains the same strength of the Concept 1 with the load instead distributed to the main spar at several locations along the spanwise length. From a

manufacturing point of view, this configuration has the potential for more complex tooling and assembly design as the number of joints has increased while the average surface area of each joint decreases as the ribs extend to the wing tip.

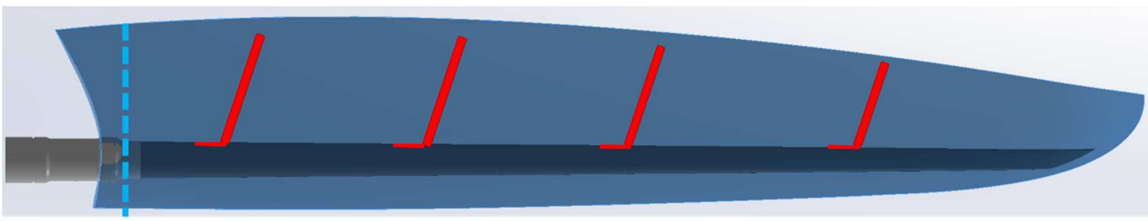

Figure 3.2 Concept 2 configuration

The final concept (Concept 3) considered bears resemblance to the Concept 1, except Concept 3 only incorporates a single minor spar (see Figure 3.3). With only a single minor spar, the minor spar displacement is set further away from the main spar. The interface section for second spar is increased to better transfer the loads to the root section.

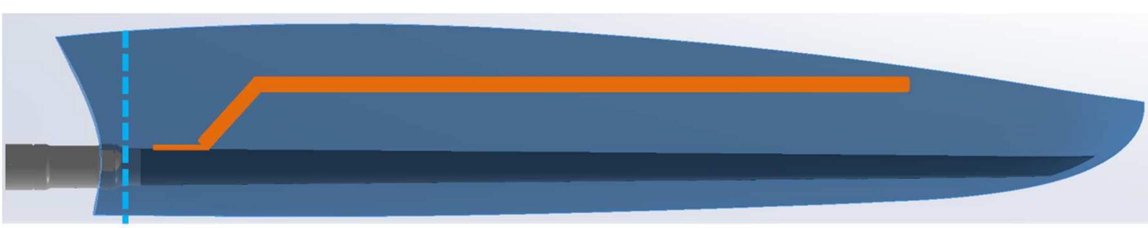

Figure 3.3 Concept 3 configuration

To determine which concept was the best a trade-off was made between the different concepts based on the criteria listed in **Error! Reference source not found.**. Five

ndependent individuals with direct interest to the research project voted on these metrics, after which the average of the votes was recorded as listed in the Table 3.3. Each concept was meant to be rated independently of other configurations, resulting in unique value instead of comparative value. Rating criteria were also assigned a weight factor depending on the importance of the corresponding criterion. Based on the result of the trade-off (Table 3.3), Concept 3 was chosen as the most-suitable configuration.

Table 3.2 Configuration trade off table criteria

Criteria	**Importance**	**Definition**
Weight Reduction	High	Potential for weight reduction of configuration compared to legacy and current propeller configurations.
Scaling for Production and Desired Production Rate	High	Potential for manufacturing/assembly strategies of configuration to be scaled up for feasible production rate of UAM vehicles
Individual Part Manufacturing	Low	Ease of individual part manufacturing and respective tooling
Inspectability	High	Ease of inspection for configuration external and internal components during service
Maintenance	Medium	Potential for maintenance of external and internal components for configuration, if required or allowed during service
Dimensional Accuracy	High	Repetitive dimensional accuracy of manufacturing/assembly strategies used for configuration to comply with aerodynamic and aeroacoustics requirements for OML shape
Cost Reduction	Medium	Potential for configuration to reduce manufacturing and assembly cost for feasible production rate (direct and/or indirect)
Part Count Reduction	Low	Potential for reduction of parts required to be manufactured individually, prior to assembly of final propeller configuration
Assembly of Individual Parts	High	Ease of assembly, implementation of potential assembly strategies, and required tooling for configuration leading to final propeller assembly.

Table 3.3 Configuration trade off table criteria ranking

Criteria	Criteria Weight	Concept 1	Concept 2	Concept 3
Weight Reduction	5	3.2	3.6	3.8
Scaling for Production and Desired Production Rate	4	3.4	3.2	4
Individual Part Manufacturing	2	3.6.	3.6	3.8
Inspectability	4	3.4	3	3.4
Maintenance	3	3.4	3.2	3.4
Dimensional Accuracy	5	3.6	3.2	3.5
Cost Reduction	3	3	4	3.7
Part Count Reduction	2	3.2	2.6	4.2
Assembly of Individual Parts	4	3.6	3.4	3.8
	Final Score	67.8%	66.5%	74.1%

3.2 CAD MODELING OF PROPELLER BLADE

The reference model (see Figure 3.4) for the generic TS propeller blade included an array of construction surfaces and curves (not pictured in figure). These surfaces and curves were used to define the net shape of the propeller blade, also seen in Figure 3.4. This model shows a component at the root of the propeller blade which was not included in the final demonstrator design. This component is the metallic root coupler, which was

not in the scope of design changes, and only meant as an additional component to be bonded to the completed propeller blade.

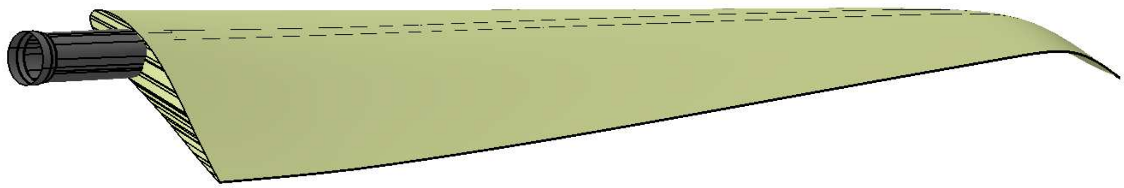

Figure 3.4 Generic thermoset blade model

Early in the design process, it was decided to produce the skin panels separately and join them at the leading edge and trailing edge. This decision reduced the amount of curvature present in the tool which improved the AFP layup process. The separated skin panels are shown in Figure 3.5, with the lower skin in green, the upper skin in blue, and the construction surface in transparent grey. This was the only modification made to the skin surface.

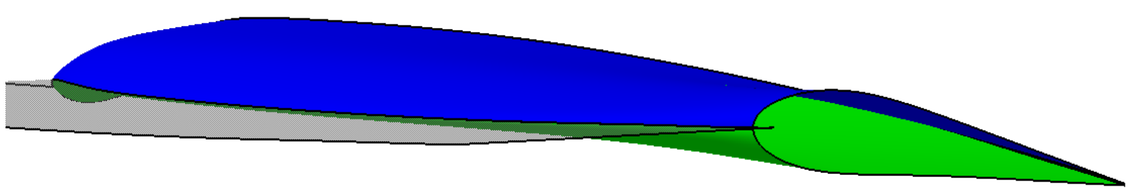

Figure 3.5 Split of upper and lower skin surface

As previously discussed, the spanwise length of the spar required a spanwise length trim to simplify the layup tooling design. The spar was trimmed to an arbitrary spanwise length of 1040mm. This length was chosen to remove most of the spar curvature as the cross section approaches the winglet. The reduction is illustrated in Figure 3.6 which shows the full-length spar in gray and with the final spar model overlaid in yellow. This was the only modification made to the spar model for tooling purposes.

Figure 3.6 Spar reduction

The purpose of the TP propeller blade is to be a manufacturing demonstrator. For this reason, the sizing of the “secondary components” does not reflect an analysis of the dimensions necessary to sustain the propeller blade under required strength and fatigue conditions. The term “secondary components” refers to all internal components which did not exist in the generic propeller blade. This includes the second and third spar, and a leading-edge doubler laminate designed to aid in bonding the skin panels together at the leading-edge. This doubler laminate was not included in the final demonstrator design. Like the trimming of the spar model, many aspects of the secondary components were defined purely to benefit the manufacturability, with the understanding that these laminates would be laid up by hand and consolidated in a compression molding machine with a

maximum working area of 2ft by 2ft. All secondary components were designed to be 8-ply laminates, also produced using Cetex® TC1225 low-melt PAEK prepreg material [62].

In addition to the sizing, the displacements of both minor spars (see Figure 3.7 and Figure 3.8)from the main spar are also not representative of a strategic reinforcing location, rather a focus on increasing the ease of tooling and laminate manufacturing. The web surfaces of both minor spars were generated using isoparametric curves on the top and bottom skin surfaces. This ensured that the webs of the minor spars followed the curve of the skin rather than running parallel with the main spar. While the minor spars do have "C" cross sections, the web sections are not perpendicular to the flange sections, as shown in Figure 3.8 below. Instead, both the cross sections and flanges twist with the upper and lower skin surfaces.

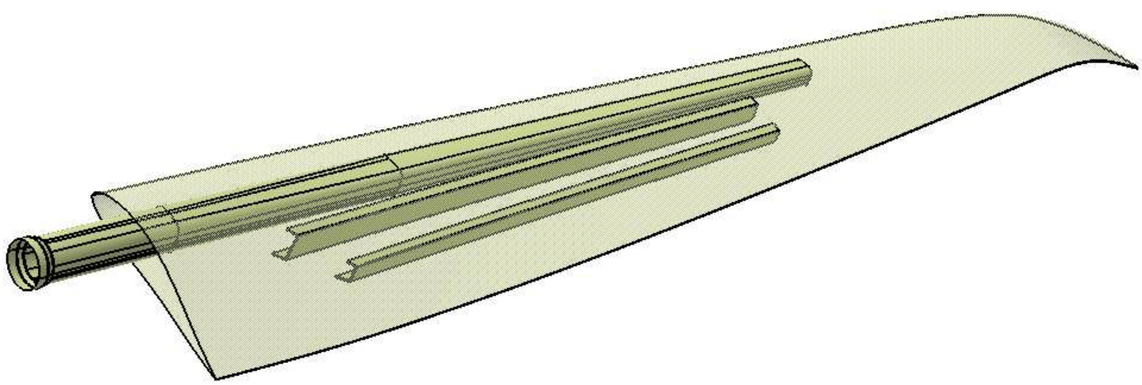

Figure 3.7 Minor Spar locations

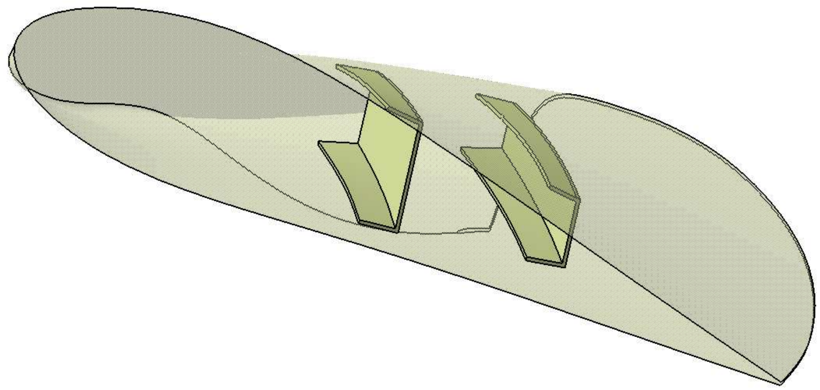

Figure 3.8 Minor Spar cross-sectional view

In addition to the minor spars, a third secondary component was designed to improve the bond formation between the skin panels at the leading-edge boundary. This component is referred to as the leading-edge doubler laminate. As with the minor spars, spanwise length of this component was trimmed to ensure the tooling could be set in the same compression molding machine. In Figure 3.9, the skin cross section at the root is shown, with the upper skin colored green, the lower skin in blue, the lightning strike material in orange, and the leading-edge doubler in red.

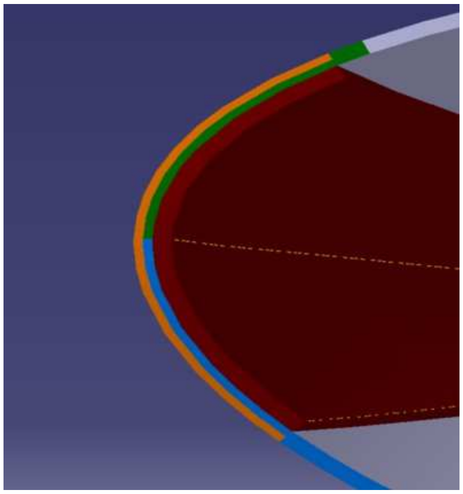

Figure 3.9 Leading-Edge cross-section at root

3.3 PROCESS DESIGN OF TPC BLADE DEMONSTRATOR

Several methods of laminate layup were considered to produce each component of the demonstrator. In addition to identifying complementary processes for the Cetex® TC1225 material, there was an interest in exploring methods which could maximize production rates.

One such method explored was a Reverse-Kinematic AFP (RKAFP) process, defined by a stationary deposition head and a mold tool mounted to an industrial robot arm performing most of the travel. This method was considered for both skin and spar laminates. The potential production rate bolster attributed to RKAFP is that, due to the relatively small size of the propeller blade net shape, several large industrial robots could work in tandem with the same stationary AFP deposition head to create a continuous automated layup cycle. Essentially, one robot could carry out a layup as the second robot, having just completed a layup, provides a preform ready for consolidation and can be prepared to carry out another layup once the first robot has completed a cycle. This

significantly decreases the potential downtime of the traditional layup process where an AFP machine has down time during part removal and tool preparation steps. Ultimately, this method could not be demonstrated due to the payload capacity and instrumentation limitations of the manufacturing center.

3.3.1 SKIN LAMINATE

With the inoperability of RKAFP, traditional AFP layup was considered next as the ideal manufacturing strategy for the skin laminate. At this stage in the process design, the available layup options were either to produce skin preforms shaped to the final contour or planar preform charges. This first option would require additional tooling while the planar layup could utilize the existing flat tool already available to the Lynx. Analysis of the AFP head reachability showed that the head would not reach the greater run out boundary needed for layup on a tool. This is due to the difference in relative axis angles needed for the AFP head to make contact. With this option also inoperable, the remaining path led to layup planar charges.

In terms of process programming, planar charges presented far fewer challenges. To produce the same preform shape using both AFP and hand layup, an oversized rectangular shape was selected as the ideal preform geometry which could then be trimmed following consolidation. After flattening the complex upper and lower skin surfaces to determine comparable planar dimensions, a rectangular layup boundary of 54in by 12.5in was determined to provide a preform with ample excess surface area.

3.3.2 SPAR LAMINATE

As the spar design also could have benefited from the use of an RKAFP process, additional methods were considered to produce the laminate. Initially, a mandrel tool assembly was developed that could be mounted to the *Lynx* AFP machine, including stepdown mandrel extensions which can mount a tool with a smaller than conventional mounting surface. Due to the large difference in size between the tools intended to be mounted to the *Lynx* AFP and the spar layup tool, an additional "free-spin" mount was designed to relieve any resistive torque from the tailstock mount designed to spin freely for multi-ton tools as opposed to this tool assembly which would weigh significantly less and have a minimum cross section of 631mm^2. The most intricate aspect of this concept was that for the entire spar to be laid up in a single cycle, the tool would have to be collapsible in some way. Two aspects of this layup presented challenges which restricted available design options: the size of the laminate and the resin system. First, in large structures, collapsing mechanisms are challenging but feasible, as seen with the development of the collapsible mandrel developed by FUBACOMP (Full Barrel Composite Fuselage) [63]. Mechanically collapsing tools require additional sealing features to be compatible with vacuum bagging processes. This additional hurdle can by bypassed by a progressively emerging technology: reformable tooling. Reformable tooling offers the utility of conventional tooling with the benefit of being able to remove internal tooling without intricate mechanisms. Unfortunately, this technology has not progressed to be able to withstand the processing pressure and temperatures required for TPC consolidation. With the challenges of size and processing parameters, advanced tooling technologies were considered yet ultimately could not be sought as tooling solutions.

Thus, the final conceptual design of the spar tool consisted of a "puzzle piece" cross-section designed to be disassembled after layup and removed from the root (see Figure 3.10). At this stage there was still concern regarding how much force the assembled tool could withstand from the AFP deposition head during compaction. Previously, TPC layup quality was controlled by manipulating laying speed, deposition compaction force, and applied heat. To maximize layup quality, past layups used maximum settings for heat and compaction, and moderately slow laying speeds. This presented an impassible problem because the compaction force previously used for TPC layup was 888 Newtons ($\approx$ 200lb$_f$), a compaction force deemed too high for the smaller sections of the layup tool. The problem remained impassible because the heating parameters had been set to maximum could not be further increased to compensate for a reduction in compaction force. The use of AFP layup for the spar tool was temporarily tabled until an upcoming upgrade to the heating element could be installed. This upgrade was delayed and did not complete installation until the alternative method had already begun tooling fabrication. The complete puzzle piece concept tool was never fabricated.

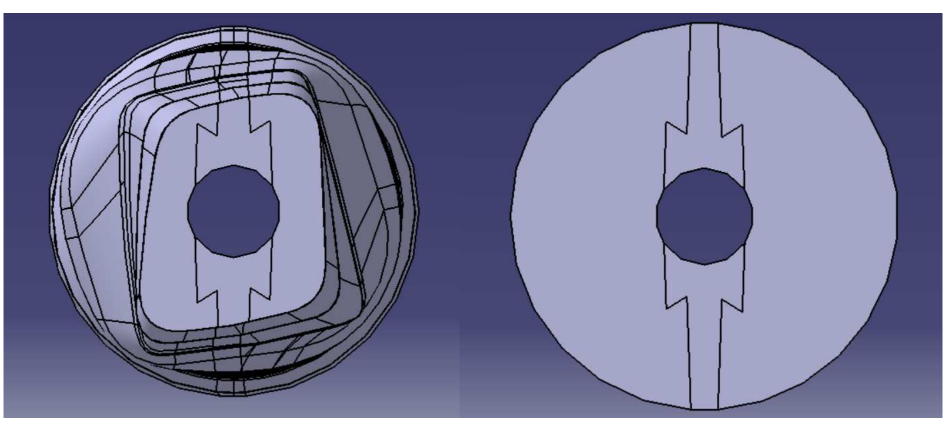

Figure 3.10 Main Spar Puzzle Piece concept cross-section

With the complete loss of rotary AFP layup, the spar laminates to be used in the demonstrator assembly were set to be laid up as planar preform charges, like the blade skins and be shaped during the consolidation stage using a compression molding process. This limited the spar to being made in halves.

3.3.3 SECONDARY COMPONENT MANUFACTURING

Due to the small size of the secondary component design, each secondary component laminate process was designed to be manufactured using hand layup and compression molding. This was the most efficient process option as the small preforms could quickly be cut and stacked to the proper pre-consolidation shape by hand and the size of the laminates had been predetermined to account for tooling which the in-house Wabash Genisis compression molding machine could accommodate.

After eliminating the non-planar layup process plans in favor of shaping preforms post layup, the only process step that required new tooling was the consolidation process. As previously stated, the process plan for the secondary components always relied on the in-house Wabash press. With that in mind, secondary component tooling for each component consisted of male and female press fit tools.

For the minor spars, 1018 mild steel was chosen as the tooling material of choice. This material is a readily available, general use tooling material. After fabricating the minor spar tools, the mold surface was found to have a surface roughness above what was desired. This machining drawback led to a new material choice for the remaining tools to be machined. For the main spar tools and the leading-edge doubler tools, 4140 HT alloy steel was chosen to decrease the mold surface roughness found in the minor spar tooling. The

worth of this design change was proven as the surface finish of the leading-edge doubler laminate showed clear improvement when compared to the minor spar tooling.

While the secondary components could be sized so that respective tooling could be machined in house, the main spar halves and skin panels required either a larger press or an autoclave to properly process the preforms. To increase the number of consolidation processes available, all four of the main spar tools (one male and one female tool for each half spar) and all four of the skin tools were designed to comply with constraints so that the tool could be mounted to an off-site compression molding machine. This allowed the tools to be used in the main facility as oven and autoclave tooling before being shipped to off-site facility to be used as compression molding tools.

Due to size requirements, the four large skin tools could not be milled using the Haas VF-5. For these tools, large material blanks were procured from Finkl Steel (Chicago, Illinois) and machine by ToolTech LLC (Springfield, Ohio). For the skin tools, P20 tool steel was chosen for high-temp capability and a high degree of machinability compared to more common carbon steals such as mild steel, a metal frequently used in general purpose tooling.

3.4 LAMINATE MANUFACTURING

Both hand layup and AFP methods were utilized when fabricating the components of the demonstrator. The choice to use either method was based on the ease of manufacturing for the individual components. The similarity between all components is that are parts were laid up as flat charges before being shaped during consolidation. The decision to produce flat charges had different motivations for each part.

3.4.1 SECONDARY COMPONENT LAMINATE MANUFACTURING

For the secondary components, hand layup represented the most efficient method. Small panels can quickly be assembled by streamlining the cutting and stacking phases. Although a Gerber vacuum table was initially used to precisely cut individual plies, consistent feedback from the fabrication team revealed that experienced fabricators could cut and size plies at a faster rate by simply using a paper slicer with a composite-cutting blade. By leaning the process of more robust equipment like the Gerber table, the required process instrumentation could be reduced to a slicer capable of reliably trimming TPC prepreg material and a soldering iron.

The method used to laminate preform plies requires all plies to be trimmed to the required preform dimensions prior to tacking. The tacking method begins with the proper orientation of the first two plies, one on top of the other. These plies are then melted together at discrete points along the boundary of the preform using a soldering iron. The quantity and pattern of the melt points is determined by the composite designer, but a small variance from the perceived ideal number of melt points does not affect the consolidation of the laminate. As TPC prepreg materials do not have tacky surfaces, the melt points act as weak bonding points to hold the preform together prior to reception into a mold tool. The weak strength of the melt points allows inspection of the preform prior to consolidation in cases where the stacking sequence must be confirmed. Once placed in complex tooling, melt points can be heard breaking as the plies shift in the mold. This method of hand layup was used to produce all the secondary component preforms and some of the main spar and skin preforms.

The minor spars were designed to be similar in shape and dimension, with less than a 50mm difference in overall length. For the small-batch manufacturing approach taken to producing these laminates, the same preform dimensions were used for both laminates. While this meant more excess material required trimming when producing the smaller third spar, the simplification streamlined the preform stacking process. This same approach would not be recommended for large scale manufacturing due to the considerable material waste. For the leading-edge doubler laminate, the flattened dimensions differed from the minor spars enough to warrant smaller preform dimensions. However, the same 8-ply stacking sequence is used for manufacturing both preform laminates.

During the secondary component manufacturing, there was an iterative process parameter development with the goal of increasing the consolidation and surface quality of the laminates. This will be discussed in the next chapter.

3.4.2 SKIN LAMINATE MANUFACTURING

AFP layup was selected to produce most of the skin and spar preforms. The skin preforms are much larger than the secondary component preforms, challenging the relative efficiency of hand layup for smaller preforms.

The skin layup was very straight forward in process design. The skin preforms were laid up using material in 0.25in wide slit-tape form, in 6-tow courses. Layup was conducted with 888N on compressive force and the Humm3® flashlamp settings were as follows: voltage set to 250V, pulse frequency set to 20Hz, and pulse duration set to 3ms. The vacuum bagging protocol consisted of a Kapton vacuum bag, GS# A-800-3G sealant tape [64], a rough-cut aluminum backing plate, and an industrial heating blanket. The purpose

of the heating blanket and aluminum backing plate was to apply as much heat to the preform during layup as possible, with the backing plate placed directly under the vacuum bag surface and the heating blanket set between the backing plate and the AFP mandrel tool. These aspects of the tooling setup were particularly driven by the need to lay the first ply, as subsequent plies would more easily cohere to previously laid plies. Using this setup, the blanket and backing plate were held in position only with vacuum pressure. For this application however, the additional heat tooling was found to cause more issues than advantages. Previous research had been performed with smaller layup boundaries approximately 12in by 12in, thus smaller backing plates. No consideration was taken that this larger plate would require mechanical fastening to the AFP tool. The need for securement was confirmed with the plate began to slide downward after prolonged time under the vacuum bag. Temporary securement was attempted using adhesive tapes, however this proved futile. To prevent the need to modify the AFP tool and/or re-program the process to align the layup on an area of the tool where fastening could be achieved, an alternative method was devised. With the presence of an oversized net-boundary, a double-sided Kapton tape boundary could be utilized closely within the laying boundary to achieve reliable first-ply adhesion, without any applied tool-side heat. This method was quickly adapted as it simplified the bagging process and once a preform was complete the tape could be easily removed either by removing the tape or trimming the adhered material prior to cure.

Three related strategies were used to consolidate the skin laminates. The consolidation strategies included an oven consolidation, autoclave consolidation and

compression molding consolidation. The oven and autoclave processes were carried out using in-house equipment.

The oven process was the first to be used and received the majority of process parameter and tooling iteration development. Increased attention was paid to the oven process based on prior challenges experienced while consolidating TPC specimens using the in-house autoclave. Conclusions from past autoclave failures indicated that high-temp vacuum bag tooling experienced increased disturbance from the gaseous fluid flow needed to maintain the high pressure within the cell during the peak heating steps of the consolidation process. To bypass this challenge, the focus was put on the process parameter development within the oven process, as the processing environment contained minimal sources of disturbance for the vacuum bag.

One preform was consolidated using a “Single-Shot” consolidation cycles. This process involved a tooling setup comprised on exclusively high-temp tooling materials, including high-temp vacuum bagging and tack tape materials. A preform was aligned on a tool, then secured under vacuum. The Single-Shot process name comes from the simplified process approach. The unprocessed preform is brought up to a temperature of 685°F, then held for 30 minutes.

Three preforms were used in a second cycle, referred to as the “2-step.” The 2-step is a derivation of the final iteration of the secondary component processes. This process involves separate thermoforming and consolidation cycles. The thermoforming cycle functioned like the full consolidation cycle with a lower temperature, holding the preform at a temperature of 350°F while the process parameters for the second step were identical

to the Single-Shot method. The purpose of the separate cycles is to form the laminate partially at a lower temperature, improving the conformity of the consolidated laminate with the target shape. For this method, separate vacuum bagging setups were used for each step. Lower-cost materials were used for the thermoform vacuum bagging due to the lower cycle temperatures. Once the thermoforming cycle had completed and cooled, the semi-formed laminate and vacuum bag were removed from the tool and the tool was cleaned. After this, the remaining step of the 2-step process is identical to the Single-shot, aside from the partially formed laminate. High-temp vacuum bagging was once again used and greater process parameters were set to achieve full consolidation. The final method used was the autoclave Single-Shot method. This method was designed to be the autoclave-equivalent to the original Single-Shot method. There is no thermoforming involved in this step. Instead, a high temperature vacuum bagging setup is once again employed. Once sealed, the preform is brought to 350°F at 45psi for 120 minutes before cooling and removal. This method was used to produce one skin panel.

The many different consolidation methods were part of an iterative process, like the process used for the minor spars, which was carried out to improve the surface finish of the skin panels. This process will be discussed further in the next chapter.

3.4.3 MAIN SPAR MANUFACTURING

Of all preforms produced, the spar preforms are the only preforms not to be made exclusively of full coverage plies. This was due to the spar ply schedule including ply boundary drops within the laminate. As the need for boundary precision is greater for a laminate with ply drops, rather than a laminate with full coverage plies, automated processes were used for both automated and hand layup. For the AFP process this meant

programming a unique boundary for each ply. At this stage, the double-sided Kapton tape method had already found success with the skin layup and was also applied to the spar layup. For the hand layup process, the precision of the Gerber cutting table was needed to ensure individual plies were cut to exact dimensions. While the width of the spar did not require exact dimensioning as excess would be trimmed post-consolidation, the spanwise length required precision cutting as well as alignment in the compression tool. While the spar tools have been fabricated and AFP and hand layup preforms produced, at the time of reporting no main spar laminates have been produced.

3.5 INDUCTION WELDING TOOLING DESIGN

To assemble the demonstrator, induction welding was the fusion bonding technology of choice. To simplify the demonstrator design to expedite the proof of concept, the demonstrator assembly was reduced from a fully representative blade model to a half-blade, consisting of a top skin panel, top spar half panel, second spar and third spar. This reduction simplified the joining process as three different welds (each spar welded to the skin panel) could be performed using the same tooling. The tooling design consisted of a six-piece pneumatic tooling assembly designed to align the four laminates and apply pressure at the weld zones to promote proper fusion bonding. The material of choice is a high-temp, non-conductive tooling board material. These pieces include a 2-piece lower tool, designed to align the skin laminate, and support the spars, three separate inserted mounting tools for the main and minor spars, and a top tool to apply resistance against the direction of applied force on the skin panel. This would be achieved by inflating pressurized bladders beneath the separate spar insert tools, driving said tools to press the

spar panels into the skin. At the time of reporting, the three spar insert tools are awaiting manufacturing.

CHAPTER 4 RESULTS AND DISCUSSION

At the time of reporting, the construction of the manufacturing demonstrator remains in progress. In addition to tooling components, a spar half laminate is needed to complete the set of demonstrator constituent laminates. For these reasons, this chapter will review the results of the controlled variance of consolidation outcomes and the development of improved process parameters.

4.1 EXAMINATION OF CONSOLIDATION OUTCOMES

After initial consolidation cycles had been completed for both minor spars and skin panels, the preliminary results demanded further refinement of the process. To inspect this, a sample section was cut from a skin laminate (see Figure 4.1). As expected, thickness measurements taken around the boundary of both first attempts at consolidation proved that the 8-ply minor spars and 7-ply skin panel had formed with thicknesses close to the theoretical thickness based on the cured ply thickness of Cetex® LM-PAEK (.14mm) [62]. Caliper measurements of a cured 7-ply sample are shown in Table 4.1. These values are compared against the theoretical cured thickness of 0.98 mm, with consistent variances attributed to difficulty in measuring ply thickness in areas of complex surface curvature. There was concern that the irregular surface defects could be emblematic of internal defects as well, however microscopy showed no irregular distribution of fibers or resin within the laminate. This indicated that most of the surface of the laminates were receiving vacuum pressure from the vacuum bag. However, the appearance of both laminates indicated that,

while consolidation pressure was applied in sections of the laminate, the total pressure was not properly distributed throughout the laminate surface area.

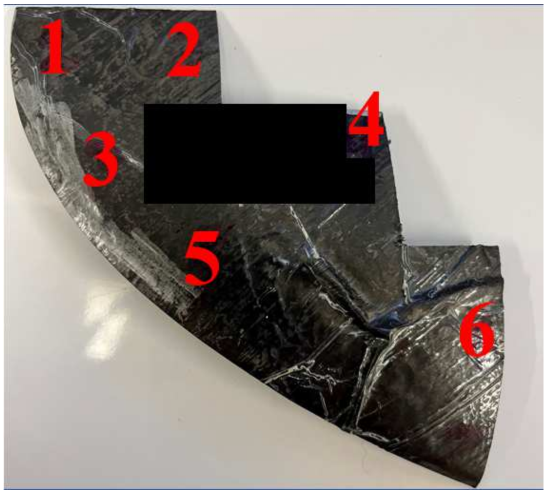

Figure 4.1 Microscopy specimen

Table 4.1 Microscopy thickness measurements

Specimen Location	Average Thickness (mm)
1	0.98
2	1.02
3	1.00
4	1.02
5	1.00
6	0.96

4.1.1 DISCUSSION OF SECONDARY COMPONENT CONSOLIDATION

The iterations done to improve the surface quality of the minor spars were largely based on the appearance of resin pooling in the laminate as well as discrete areas of fiber win. This was attributed to the concerns raised in the previous chapter regarding the surface roughness of mild steel. These defects were commonly seen in the web section of the spars and in the internal corners. To compensate for these defects several process changes were sequentially added until the final laminate met acceptable visual inspection standards. An additional process modification included inserting a thermoforming step at the start of the curing process prior to full temperature heating. This additional preparatory step included a low pressure (30% of full pressure) step with full heat, allowing the resin to flow under reduced pressure for 15 minutes before ramping pressure to full and heating for an additional 10 minutes before cooling. Once properly formed the laminate could then be re-inserted into the press, where the full recipe would take place. Another method seen as having a positive effect was the improved measures taken to properly prepare the tool prior to introducing the preform. After cleaning the tool with an abrasive pad and acetone, three coats of Frekote 770-NC [65] were applied to the tool per the material application instructions, with 5-minute intervals between coat applications and 10-minutes of rest time after the third coat prior to any further contact. To aid in surface finish and laminate removal, high-temperature Upilex® 25S release film [66] was placed between both tools and the preform, with concave sides of the film positioned away from the preform as illustrated in Figure 4.2 below.

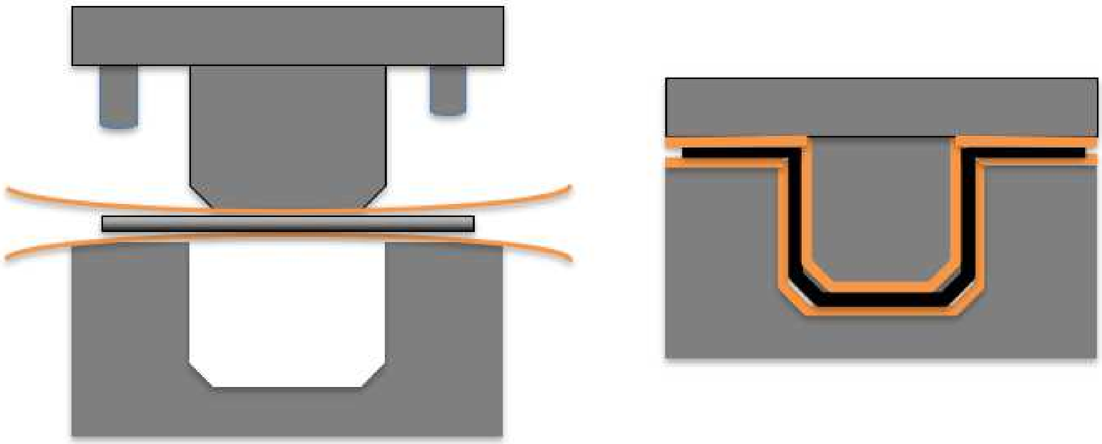

Figure 4.2 Minor Spar compression molding setup

A conclusion to be made from the challenges presented by the minor spar compression molding results is that, in addition to maintaining the proper process steps found to promote quality consolidation and surface finish, the minor spar geometries could be manipulated to produce obtuse angles between the flanges and web sections rather than the changing angles of the current design. A consistent obtuse angle between the flanges and web section eliminates undercut surfaces which affect the machinability of the tool and the ability to properly distribute pressure during consolidation. Examples of minor spar specimens at different stages of the iterative quality improvement are shown in Figure 4.3 and Figure 4.4.

Figure 4.3 Early Minor Spar iteration showing surface defects

Figure 4.4 Later Minor Spar iteration showing improved surface quality

4.1.2 DISCUSSION OF UPPER SKIN PANEL CONSOLIDATION

Due to the scope of the demonstrator changing to only include a single skin panel, the consolidation effort was focused on the upper skin panel. The defects observed in the early attempts at consolidating the skin were much more severe than the comparably minor resin pooling and fiber bunching of the previous samples. This was attributed to the use of a male tool and vacuum bag setup as opposed to male and female tools compressed together. The work of improving the surface quality of the laminate saw some increase in quality but could not completely rid the laminate of repeated defects, such as reduced but still present resin pooling and raised sections of skin surface. Inspired by the success of the

thermoforming stage used in the minor spar development, a similar process was developed to produce the 2-step cure, consisting of a preliminary heating cycle which heated the preform at 350°F for 20 minutes before removal, followed by a second high-temperature cycle where the tool was held at a temperature of 685°F for 30 minutes before cooling. This procedure, developed over a series of iterations, resulted in a final panel which showed improved surface conformation and a reduction in surface defects. The final oven consolidation laminate is shown in Figure 4.5, displaying clear raised defect lines.

Figure 4.5 Oven consolidated skin laminate

Observing these results, it was clear that the skins would benefit from a compression molding process rather than an oven process. A final iteration of a vacuum supported consolidation cycle was run using the Bondtech autoclave. The goal was to

produce a laminate with fewer defects in a more pressure-intense cycle. After the autoclave sample was completed, the same prominent defects occurred while smaller raised defects of similar shape also presented across the laminate. The main difference in defect formation between the oven iterations and the autoclave cycle is that the general height of the raised defects was reduced, likely under the increased pressure of the vacuum bag and the autoclave cell. In Figure 4.6 three skin specimens are shown, two of which are oven consolidated while the center specimen is the result of an autoclave cycle. As shown in Figure 4.6, small defects are increasingly frequent in the autoclave specimen compared to the oven specimens, which have larger, less frequent, but more prominent defects.

Figure 4.6 Comparison of consolidation specimens

CHAPTER 5 CONCLUSIONS AND FUTURE WORK

Adapting a composite design to transition between material systems presents challenges and a simple transplant of a new material into the original process plan presents a far from ideal manufacturing outcome. When transitioning from TSCs to TPCs, several considerations must be made. TSCs require much lower energy to layup and produce than TPCs, as shown by the lesser heat and pressure needed to sustain a viable layup and cure cycle. However, if a co-curing process is to be conducted, TSC preforms must be joined within a shorter timetable after the material has been removed from storage. TPC preforms have no shelf life, giving the manufacturer added flexibility to the manufacturing process and the option to assemble a completed structure in a separate location without removing fusion joining as a viable assembly method due. To draw conclusions on the adaptability of TPCs to a historically TSC industry, a wide scope must be held which encompassed the different manufacturing approaches as well as the manufacturing outcomes.

5.1 CONCLUSIONS FROM RESEARCH

The integration potential of TPCs is dependent upon the considerable effort needed to differentiate not just the material, but also the structure and processes from TSCs. It is not enough to modify process parameters in relation to differing melting temperatures and preform characteristics. The development process of the TP propeller blade illustrates the need for intentional part and process design variations which integrates the strengths of TPCs. An effective adaptation of TPC materials for an existing TSC design include the

consideration of (i) applicable layup technologies for greatest ease of manufacturing, (ii) cost and difficulty of tooling production, (iii) machinery, processing environments, and cycle times required for curing processes and, (iv) the joining technologies best suited to assemble the structure. Only TPC designs which reflect these considerations can be used to create an accurate comparison of TSCs and TPCs laminates and structures.

5.2 COMPLETION OF WELDING DEMONSTRATOR

At the time of reporting, the welding demonstrator assembly is in development. Currently, development is in progress to decrease the presence of surface wrinkling of the skin panels which would aid in proper bond formation during induction welding. The information gained from refining the skin consolidation process will then be applied to manufacture the upper half spar laminate, for which tooling and preforms have already been designed and manufactured. The completed demonstrator will serve as a physical validation of the method used to adapt the generic TS propeller blade to a TP structure.

In addition to completing the manufacturing of the remaining laminates, the three spar insert tools are also queued for manufacturing. These inserts are crucial components for supporting the main and minor spars during bladder pressurization, leading to forced contact at each weld zone. After these insert tools are completed, along with the laminates also in development, all components will be assembled into a full manufacturing demonstrator. The proper welding the components together will signal the final stage of development for the manufacturing demonstrator. Continuing research efforts will include validation of the weld strength of the demonstrator based on anticipated performance requirements.

5.3 SHORTCOMMINGS OF LAMINATE MANUFACTURING PLAN

During the process planning and manufacturing stages, shortcomings of the original project goals became apparent. A general reason for this is that the project scope was originally defined as reproducing a propeller blade originally designed for TSCs and producing a TPC version of the same propeller blade. The challenge is that to truly produce a high-quality TP propeller blade, the propeller blade must be designed, not revised, with TPC manufacturing in mind. While free design license was taken with the secondary component designs, to reduce the scope of this research, only small revisions were made to the skin and spar model. The skin panels and especially the spar are designed to benefit from TSC manufacturing methods that have no comparable TPC counterpart. To draw genuine, even-ground comparisons between TSC and TPC versions of this propeller blade, some model geometries and ply stacking sequences would need to be altered to represent how a TP propeller blade would be manufactured.

5.4 PROPOSED ALTERNATIVE MANUFACTURING PLANS

The stacking sequence and shape of the spar provided major manufacturing challenges. Early in the process planning stage, alternative preform manufacturing methods were suggested, such as composite braiding and filament winding. At the time, these were not feasible methods because the original stacking sequence was designed a combination of unidirectional and woven fabric hand layup and would not translate to winding or braiding well without stacking sequence revision. For this reason, these manufacturing strategies were not further investigated as part of the research effort. Alternative manufacturing research should focus on alternative methods of manufacturing the spar

laminate if alternative stacking sequences could be defined which better complimented those alternative manufacturing methods.

The shape of the spar is another point of concern for TPC layup: TPC consolidation requires high temperature and high pressure tooling which is challenging and expensive to manufacture when considering the original spar model that included the trimmed-off section at the winglet. Parallel to process development for the demonstrator, an alternative concept for the spar shape was drafted that could provide the bases for a spar with increased ease of manufacturing. This spar, shown in Figure 5.1 trades the original open cross section for an I-beam cross section. This spar could be made in halves split down the middle of the vertical web section, enabling the laminate to be compression molded in separate sections and then fusion bonded together. Note that the new concept retains the necessary root section geometry.

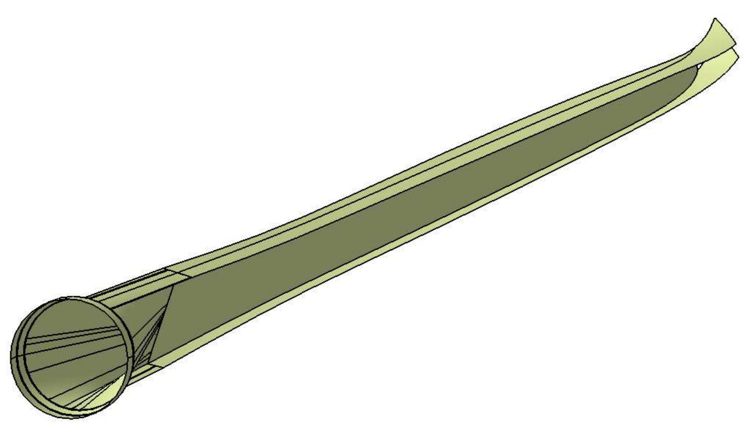

Figure 5.1 I-beam spar

For a higher rate of production, co-curing should be considered as a more viable method of fusion bonding. Induction welding as a bonding process has many positive attributes, however welding every joint of the propeller blade would likely require multiple cycle scenes involving separate tooling setups and different induction coils. While no less complex, a co-curing setup could provide a method of fusion bonding that does not require an additional cycle after the consolidation cycle to bond every composite component together.

5.5 MACHINE DESIGN RECOMMENDATIONS

Several machine system features used during the manufacturing processes were noted as considerably more beneficial to the manufacture of TSCs than TPCs. This does not reflect machine deficiencies, but rather a focus of the Lynx machine's service of the more readily used polymer composite. Two key challenges of TP-AFP layup, the prepreg material stiffness and greater layup heat application, can be overcome with intentional design features. Tension holds a greater importance within a TP-AFP system to combat the stiffened prepreg tows' propensity to re-wind if material is not held tight during travel. The Lynx utilizes a set of static, spring-driven rollers located shortly before the material feeding system to maintain tension between the material spools and said spring-driven rollers. This works well for managing tow tension in the travel, however experience during this research taught that even momentary lapses in constant tension could cause the spool to quickly unwind leading to material entanglement in the material storage unit. The proposed solution this is illustrated in Figure 5.2, where a rubber roller of the same length as the material spool would press down on the exit point of the material from the spool, keeping any brief lapses in tension from being transferred to the spool.

The other challenge of TP-AFP layup is the heating. It is completely possible to utilize the same heating element for TPC and TSC layup, if the heating element can provide both low and high heat application. The Heraeus Humm3® flashlamp system was able to provide the necessary level of heat without the wide dispersion of heat that other heating elements such as infrared and hot gas torch methods also produce. With a greater heat application, the components of the deposition head must also be protected from possible thermal damage. As the use of Humm3® involved a retrofit to the Lynx, the deposition head did not, by default, have the necessary heat shielding. Temporary solutions were made by non-permanently fastening thin metallic sheets to the most exposed surfaces of the deposition head. Thin aluminum sheet metal was found to be more than acceptable for protecting the deposition head during laying movement. A deposition head designed for the greater heat of TP-AFP operations would benefit from a removable heat shielding solution so that mechanisms are protecting during layup but also still able to be serviced without additional intricacy.

An additional challenge to managing the heat necessary for TP-AFP layup is protection of the compaction roller. Currently, for a flexible polymer roller, it is a balancing act to find an optimal trade-off between a polymer with the needed thermal operating range, and a hardness value that allows for deformation at the necessary compaction as increasing a polymer's operating temperature range typically increases the hardness value. The alternative to a thermally resistant roller is to incorporate a robust active cooling system. By default, the Lynx has an internal cooling system for the deposition head which extends to the compaction roller, however this alone cannot protect an inferior polymer roller without observable surface degradation. The topic of alternative roller temperature control

was also explored in a 2014 study by Henne et. al [67]. The study produced four compaction roller concepts, including air flood cooling, conduction cooling, and two concept rollers with internal cooling with one roller being made of rigid metal and the other silicone. The design and fabrication of cooling systems that allow for flexible polymer rollers to operate in environments out of their temperature range is yet another facet of the TPC manufacturing process that warrants further study.

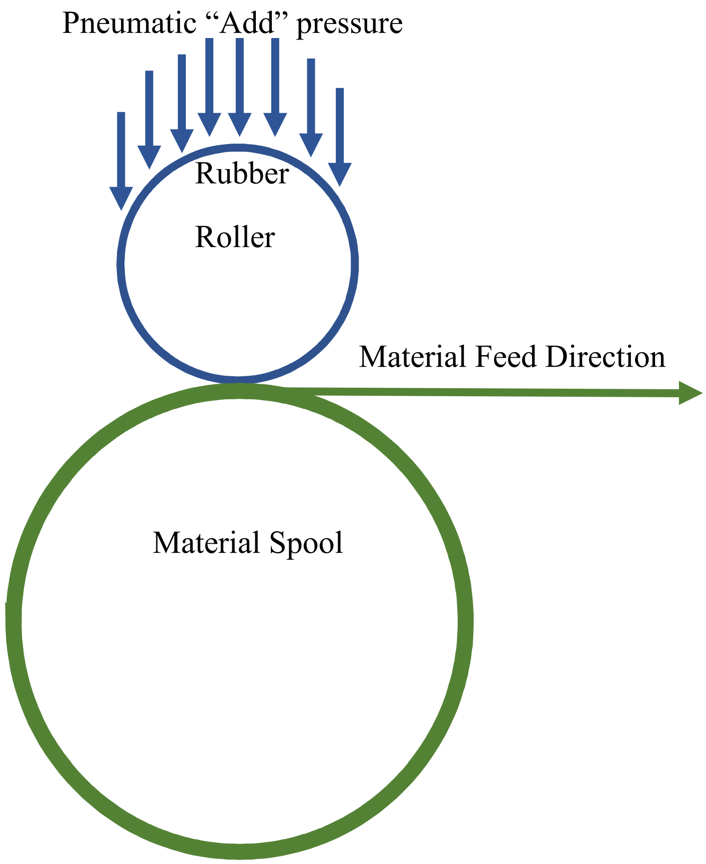

Figure 5.2 Proposed AFP spool securement

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