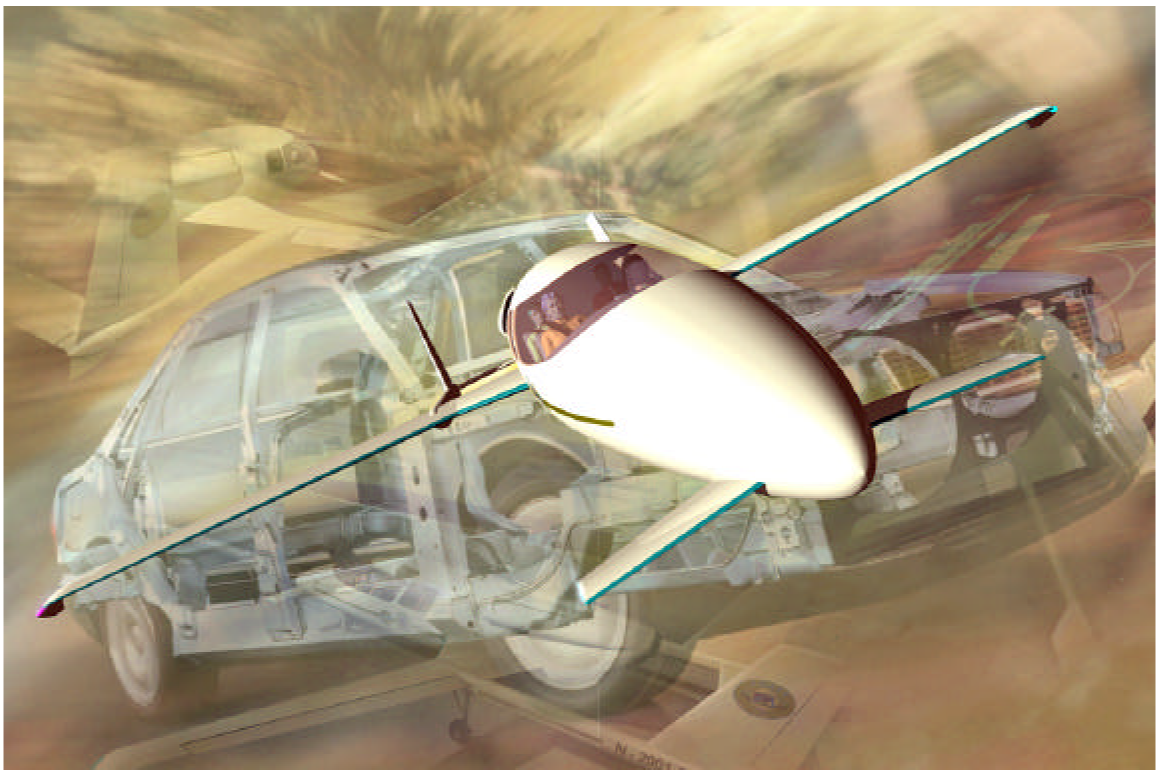
Guide for Low Cost Design and Manufacturing of Composite General Aviation Aircraft



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ABSTRACT

The Advanced General Aviation Transportation Experiment (AGATE) is a joint venture among NASA, the FAA, and numerous aircraft manufacturers and selected suppliers. The goal of AGATE is to develop the technology to create a single-engine, four-passenger aircraft with improved avionics and crashworthiness features that will sell for approximately $100,000. AGATE has many Work Packages, covering Flight Systems, Propulsion Sensors and Controls, Integrated Design and Manufacturing, Ice Protection Systems, AGATE Integrated Platforms, Flight Training Curricula, Systems Assurance, and Program Management.

The Low-Cost Manufacturing Group within the Integrated Design and Manufacturing (ID&M) Work Package is producing this Design Guide. The purpose of this Guide is to document current materials and processes being employed in manufacturing aircraft components, to summarize low-cost and emerging manufacturing technologies studied within AGATE, and to investigate automotive technology transfer. A summary at the end of this document will compare the various materials and processes to help the reader identify promising manufacturing methods for their different applications.

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1 CURRENT STATE OF THE ART

This chapter discusses the various materials and methods currently in use on certified aircraft or aircraft components.

1.1 Metallic Construction

1.1.1 Construction Techniques

Aircraft fuselage structures are classified as truss, monocoque, or semi-monocoque construction. Truss construction has tubes that are bolted, welded, or otherwise joined into a framework structure that is then covered with a fabric skin. All structural loads are carried by the truss structure. Monocoque fuselage structure consists of a load-carrying skin that is reinforced with bulkheads or longerons. Semi-monocoque fuselage structure consists of bulkheads, longerons, or other frames that carry most of the loads, and structural skins that also contribute to fuselage strength.

The line between monocoque and semi-monocoque is somewhat blurred, leading to the term "stressed-skin" construction, which provides a more useful rubric than monocoque, clearly encompassing all aircraft structures that use the covering as a load-bearing element. (Schatzberg 1998, 158)

Wings are either strut braced or cantilever. In braced structure, external struts help carry wing loading. In cantilever wing structure, all load-carrying members are contained within the wing, and the skin is generally a stressed, load-carrying member as well.

To achieve stressed-skin fuselages and wings, designers typically fasten aluminum structural skins to frames, stringers, bulkheads, spars and ribs, with some structural castings and forgings at structural hard points. The component parts are produced by conventional practices such as shearing, rolling, press braking, drop-hammer or hydro forming in dies, etc. The component parts are assembled by manual or machine riveting, bolting and a minimal amount of bonding; often in jigs and fixtures. (Cirrus 1999, 16)

1.1.2 Metallic Materials

Conventional 2024, 6061 and 7075 aluminum alloys have been used successfully in airplanes fuselages, being optimized to fulfill the desired property profiles combining strength, fracture toughness, fatigue, and minimizing corrosion resistance. But stiffness has always been a problem for designers because of the rather low Young’s modulus of aluminum. Al-Li alloys have dominated interest in the last 10-15 years because they have about 10% reduced specific weight and 15% higher stiffness. Most metallurgical uncertainties have been overcome, with the exception of crack deviation problems and some corrosion phenomena. The barrier to implementing the aluminum immediately is the two to four times higher price compared to conventional high strength aluminum alloys. Sheet, plate, and extrusion and precision forgings of 8090 and 2091 are under investigation. Depending on the outcome of the large scale qualification tests, Airbus was the first to substitute aluminum-lithium alloys for conventional high strength aluminum alloys in a number of fuselage components. (Buhr 1992, 3-5) The alloy originally developed a bad name due to incomplete knowledge of the manufacturing concerns and early use of first generation alloys. Development of new alloys significantly reduced these drawbacks. (Davies 2001)

Titanium has established its place in certain aircraft applications. Titanium alloys have greater density than aluminum, but have higher strength-to-weight and stiffness-to-weight ratios. However, they require close control to obtain consistent properties, making it generally more difficult to process. Titanium is used when heat resistant properties are required and when the local geometry is such that a denser material than aluminum is advantageous. Airframe examples main frames and very heavily loaded local structure. (Fielding 1999, 56) Titanium costs have dramatically fallen in the last five years. While not in the realm of HSLA products, they compete well in the designs that require corrosion resistance. Unfortunately, due to intermediate modulus between aluminum and steel they are not good candidates in stiffness critical designs. They are still used in these designs, if corrosion is a concern and only modest improvement in stiffness is required over aluminum. (Davies 2001)

Small quantities of steel have always been used in airframes where high strength is required in a small space. The main problem with using steel is obtaining a light structure since, although steels have strength-to-weight and stiffness-to-weight ratios comparable with aluminum, the density is nearly three times as great. This implies minimum gauge design in many places, which accentuates instability problems. Highly stressed components, such as landing gear and engine pylons, make extensive use of steel. (Fielding 1999, 56) In general aviation aircraft, steel is usually the material of choice for firewalls.

1.1.3 Friction Stir Welding

Eclipse Aviation has announced plans to build their new metallic "Eclipse 500" jet aircraft using friction stir welding. . Boeing is also using this joining process to manufacture the Delta Launch vehicle. (Sears 1999)

Friction stir welding (FSW) was developed and patented by The Welding Institute (TWI) in 1991. FSW is a highly significant advancement in aluminum welding technology that can produce stronger, lighter, and more efficient welds than any previous process. (High Tech Welding 2001)

In friction stir welding (FSW) a cylindrical, shouldered tool with a profiled probe is rotated at a high speed and slowly plunged into the joint line between two pieces of sheet or plate material, which are butted together. The parts have to be clamped onto a backing bar in a manner that prevents the abutting joint faces from being forced apart. Frictional heat is generated between the wear resistant welding tool and the material of the work pieces. This heat produces a high temperature that is below the melting temperature, which causes the work piece to soften without reaching the melting point and allows traversing of the tool along the weld line. The plasticized material is transferred from the leading edge of the tool to the trailing edge of the tool probe and is forged by the intimate contact of the tool shoulder and the pin profile. (TWI 2001) It leaves a solid phase bond between the two pieces in a fine-grained, hot worked condition with no entrapped oxides or gas porosity. (High Tech Welding 2001)

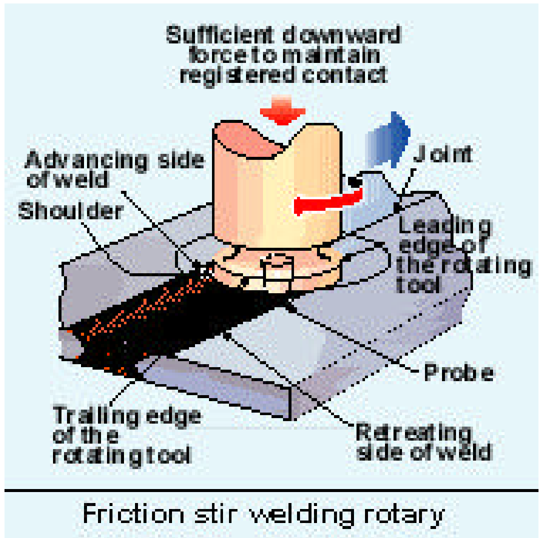


Figure 1. Friction Stir Welding Tool (High Tech Welding 2001)

The process advantages result from the fact that the FSW process (as all friction welding of metals) takes place in the solid phase below the melting point of the materials to be joined. Friction Stir Welding results in a relatively small heat-affected zone and does not use welding consumables. Since traditional heating methods are not employed, the properties of the metal in the joined area are higher than those from any other known welding process and distortion is virtually eliminated. (High Tech Welding 2001) The benefits therefore include the ability to join materials that are difficult to fusion weld, for example 2000 and 7000 aluminum alloys. The solid-state joint is of higher quality, and is 30-50% stronger than if we used traditional gas-metal welding. (Sears 1999) Other advantages are (TWI 2001):

- Low distortion, even in long welds
- Excellent mechanical properties as proven by fatigue, tensile and bend tests
- No fumes
- No porosity
- No spatter
- Low shrinkage
- Can operate in all positions
- Energy efficient
- Non-consumable tool
- One tool can typically be used for up to 1000m of weld length in 6000 series aluminum alloys
- No filler wire
- No gas shielding for welding aluminum
- No welder certification required

- Some tolerance to imperfect weld preparations - thin oxide layers can be accepted
- No grinding, brushing or pickling required in mass production

The limitations of the FSW process are (TWI 2001):

- Welding speeds are moderately slower than those of some fusion welding processes (up to 750mm/min for welding 5mm thick 6000 series aluminum alloy on commercially available machines)
- Work pieces must be rigidly clamped
- Backing bar required
- Keyhole at the end of each weld

Successful friction stir welds have been produced in (TWI 2001):

- Copper and its alloys
- Lead
- Titanium and its alloys
- Magnesium alloy, Magnesium to aluminum
- Zinc
- Aluminum-based Metal Matrix Composites
- Other aluminum alloys of the 1000 (commercially pure), 3000 (Al-Mn) and 4000 (Al-Si) series
- Plastics
- Mild steel

Single pass butt joints with aluminum alloys have been made in thicknesses ranging from 1.2mm to 50mm without the need for a weld preparation. Thicknesses of up to 100mm can be welded using two passes, one from each side, with 6082 aluminum alloy. Parameters for butt welding of most aluminum alloys have been optimized in a thickness range from 1.6mm to 10mm. Special lap joining tools have also been developed for aluminum with thicknesses of 1.2mm to 6.4mm. (TWI 2001)

The process has been used for the manufacture of butt welds, overlap welds, T-sections, fillet, and corner welds. For each of these joint geometries specific tool designs are required which are being further developed and optimized. Longitudinal butt welds and circumferential lap welds of aluminum alloy fuel tanks for space flights have been friction stir welded and successfully tested. (TWI 2001)
The FSW process can also cope with circumferential, annular, non-linear, and three-dimensional welds. Since gravity has no influence on the solid-phase welding process, it can be used in all positions, i.e. Horizontal, Vertical, Overhead, and Orbital. (TWI 2001)
The shipbuilding and marine industries are two of the first industry sectors that have adopted the process for commercial applications. (TWI 2001)

At present the aerospace industry is welding prototype parts by friction stir welding. Opportunities exist to weld skins to spars, ribs, and stringers for use in military and civilian aircraft. This offers significant advantages compared to riveting and machining from solid, such as reduced manufacturing costs and weight savings. Longitudinal butt welds and circumferential lap welds of aluminum alloy fuel tanks for space vehicles have been friction stir welded and successfully tested. The process could also be used to increase the size of commercially available sheets by welding them before forming. The friction stir welding process can therefore be considered for (TWI 2001):

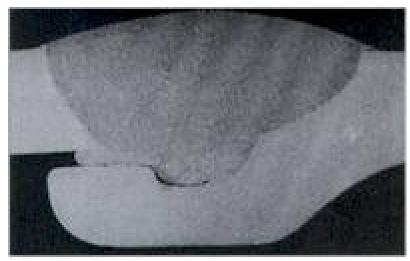
- Wings, fuselages, empennages
- Cryogenic fuel tanks for space vehicles
- Aviation fuel tanks
- External throw away tanks for military aircraft
- Military and scientific rockets
- Repair of faulty MIG welds

Friction Stir Welding works with all aluminum alloys and can be used to join dissimilar alloys and wrought products to cast products. The Friction Stir Welding process needs no post treatment grinding and straightening. Friction stir welded material can be anodized leaving the joint almost undetectable. This opens many new product opportunities not previously available. (High Tech Welding 2001)

Unlike the traditional weld shown here, Friction Stir Welding requires no backing for the weld, thus reducing weight.

Furthermore, the minimal size of the heat-affected zone offers the potential for reduced material thickness.

Less aluminum weight can mean lowered cost of production and cost of operation.



Traditional Weld

Friction Stir Weld

Figure 2. Weld Cross-Section (High Tech Welding 2001)

1.2 Composite Aircraft Construction

A composite may be broadly defined as a combination of two or more materials, each with distinctive properties. The class of composites discussed in this paper consist of fiber reinforcements in a polymeric matrix.

Polymeric composite materials are attractive alternatives to the metals used in general aviation aircraft because they are lighter weight, stronger, stiffer, and have almost no fatigue and corrosion problems. (Petersen 1998a, 1) Composites also allow the option of directional property tailoring and reduced part counts. Composite materials have the disadvantage of higher raw material costs and labor-intensive fabrication, which further leads to the parts being expensive. It is also relatively difficult to form complex shapes and hollow parts.

Composite skins have the advantage of increased load-carrying ability, thereby reducing the need for reinforcing structure, as well as the ability to reduce parts count through the integration of structural reinforcements such as longerons and stringers. For example, a fuselage skin may be laid up of glass with core, but it may also have multiple layers of carbon unidirectional fiber strategically placed in the skin to provide rollover strength or longitudinal stiffness. As this fuselage skin is laid up all at once and consolidated into a single piece, the parts count and assembly labor hours are dramatically reduced, not to mention the inherent reduction in the number of joints.

Composite sandwich skin construction allows for more interior cabin space for the same outer mold line by eliminating the need for bulkheads in the design and build process. (Miller 2001) It is also much easier to achieve true laminar wing design, because mechanical fasteners can be eliminated on the exposed wing skin surface.

Joints in components or structures incur a weight penalty, are a source of failure, and cause manufacturing problems; whenever possible, therefore, a designer avoids using them. (Schwartz 1997, 2:309) As in metallic joints, prime considerations in composite joint design are bearing, tensile, and compressive strengths in the laminate and shear and bending stresses in the fasteners. (Schwartz 1997, 2:312) Shear and bending stresses in the fasteners are typically not the most critical consideration in composite joint design. Due to the weakness of the composite material, the fastener controlled failure modes are usually far less likely than those controlled by the composite material. (Davies 2001) When constructing all-composite aircraft, structural components can be joined in one of five general ways: (1) co-curing, (2) secondary bonding with structural adhesive, (3) advanced bonding (fusion welding, etc.), (4) mechanical fasteners, and/or (5) entrapment. It is well established that relatively thin-walled structures are most efficiently joined through adhesives, whereas thick-walled structures lend themselves to mechanical fastening. (Schwartz 1997, 2:34)

Co-curing is the process of taking two or more composite components that use similar, if not identical, resin systems, and then curing them simultaneously so that the resin flows between the two pieces, and they become one piece. This is most commonly accomplished on wet layup parts that have been brought to the handle cure stage; they are then post-cured together. Parts fabricated from pre-impregnated materials that are individually laid up may also be co-cured when they are physically pressed together with vacuum pressure, autoclave pressure, etc., during the cure cycle, often with the addition of a film adhesive.

Secondary bonding with structural adhesive is the “gluing together” of two or more individual pieces. For all but wet layup parts, this is performed after the pieces are fully cured. The surface is prepared for bonding; this generally consists of abrading and solvent cleaning the two surfaces to be bonded. An adhesive is mixed, generally consisting of a resin system with inert filler materials, and then applied to the prepped surface. The second piece is placed over the adhesive and the entire assembly is cured.

Advanced bonding incorporates techniques such as ultrasonic welding, induction bonding, dual resin bonding, resistance heating, and focused infrared energy. (Schwartz 1997, 2:310) Unfortunately, these techniques are limited to thermoplastic resin systems (described in Section 1.3.1 below).

Mechanical fasteners can also be used to join together composite components, although their use generally takes away many of the advantages of composite construction. For example, the use a fastener to join together a rib to a wing skin would lessen the effectiveness of an otherwise-laminar airfoil. They also break into what would otherwise be continuous fiber reinforcement, reducing the composite effectiveness. Fasteners are more commonly used to join metal components to composite structure. Hard points, consisting of locations in the composite layup suitable for drilling holes, are used when a fastener such as a bolt or rivet will be installed. Notched strength of composites becomes very critical when holes will be drilled in the structure. Except for special circumstances where a test is required by the Federal Aviation Regulations (FAR's), general aviation manufacturers have the option of testing full-scale structure with the holes in place, or they may test coupons for notched laminate knock-down factors that can be used in subsequent structural analysis. It is also permissible to use theoretical knockdown factors in analysis, but these tend to be very conservative. One area where the FAA has yet to allow an exclusively bonded structure is the wing-to-fuselage joint. In today's composite aircraft such as the Raytheon Premier, the Lancair Columbia, and Cirrus Design Corporation's SR20 and SR22, you will find mechanically fastened wing attachment structure.

Metallic fasteners have succeeded well with fiberglass, boron, and Kevlar fiber composites. However, graphite reinforcements present problems. Some metallic fasteners in contact with graphite composites are prone to corrosion. Aluminum is the worst in this respect; stainless steel is somewhat better. Nickel and titanium alloys show excellent compatibility with graphite, but are more expensive than aluminum or steel. Other than relying on more expensive metal fasteners, two main solutions are possible: (1) Insulate the graphite from the metal by surrounding an aluminum or steel fastener with a medium the fastener will not react with. Although this approach prevents corrosion, it complicates fastener installation. (2) Use composite fasteners rather than metallic ones. Although composite fasteners are lighter than metal fasteners, their properties are degraded by environmental factors. (Schwartz 1997, 2:311-312)

The last joining method option listed, entrapment, consists of embedding mechanical fasteners into the composite structure at the time the part is cured. The technique of incorporating metal inserts and attachments into composites is expected to provide significant improvements in both part performance and fabrication efficiency (and thereby a reduction in cost simultaneous with an increase in reliability) over adhesive bonding and direct mechanical fastening (i.e. fastening through holes, molded in or drilled, in the composite). (Schwartz 1997, 2:34)

Composite and metal components can be combined together in aircraft structure. For example, wings with composite skins can have either a conventional aluminum substructure or a composite substructure. The advantage of an aluminum substructure is that there are proven design and fabrication methods for traditional spars and ribs. This method has been used successfully on military aircraft. The main disadvantage of this method is galvanic corrosion, especially in a wet fuel cell. Another disadvantage of an aluminum substructure with composite skins is internal arcing caused by lightning strikes. A composite wing with a composite substructure similar to an aluminum substructure overcomes these disadvantages. Although, a composite substructure with composite skins may still arc when fasteners and other subsystems such as fuel and control systems are present. Any fasteners have to be electrically insulated, as would an aluminum substructure. (Petersen 1998a, 1)

In current transport category aircraft, the development of composites has mostly been derived from military and NASA prototypes. Composites are used for secondary structure, control surfaces, empennage, and some new wings and fuselages. Europe is leading the way in these applications. In general aviation aircraft, the development of composites has been derived from a combination of transport category and past metal experiences. Composites are used for most structures (pressurized fuselage, wing, propellers, and extensive bonding), some dynamic components, and utilizes some advanced material forms and processes. Barriers to expanding composite applications are manufacturing cost, non-recurring development costs, maintenance technology, limited resources with sufficient training (engineers and technicians), and stable material supplier base. In addition, there are critical safety concerns regarding bonded repair practices including: the fact that damage tolerance is not well developed (safety protected by conservative design practices, but service damage threats are not well defined), high cycle fatigue, and rate effects for dynamic components, crashworthiness, and flammability. (Ilcewicz 2000, 4-7) Current "all-composite" aircraft in production include the Lancair LC40, Cirrus Design Corporation SR20 and SR22, SNA Seawind (experimental), Morrow Boomerang (experimental), Raytheon Premier I, Raytheon Horizon, Visionair Corp. VA10 (experimental), and AASI Jetcruzer 500 (experimental).[1]

1.3 Composite Constituents

Composite constituents consist of the matrix and reinforcements. Composites are further defined by their processing technique.

When designing with composites, the designer will first set property specifications for a given component. Then the selection of materials (fiber and matrix) is done in conjunction with fabrication methods so that the fibers and the matrix can be combined efficiently and inexpensively. (Cirrus 1999, 30)

According to the Composites Fabricators Association, about 65% of all composites produced currently use glass fiber and polyester or vinyl ester resins, and are manufactured using an open molding method. The remaining 35% are produced with

[1] The term "experimental" is used to identify aircraft that have not been type certificated by the FAA.

high-volume manufacturing methods or use advanced materials, such as carbon or aramid (polyamides such as Kevlar) fiber. (Borge 2000, 71)

1.3.1 Matrix (Resin)

The role of the matrix in a fiber-reinforced composite is (1) to transfer stresses between the fibers, (2) to provide a barrier against an adverse environment, and (3) to protect the surface of the fibers from mechanical abrasion. The matrix plays a minor role in the tensile load-carrying capacity of a composite structure in the fiber-reinforced direction. However, selection of a matrix has a major influence on the interlaminar shear as well as on in-plane shear properties of the composite material. (Schwartz 1997, 2:43-44) While matrix materials may be polymers, metals, or ceramics, among others, this paper focuses on polymer matrix (plastic) composites.

The term "plastics" encompasses organic materials (carbon, hydrogen, nitrogen, etc.) of large molecular weight that can be shaped by flow. The term usually refers to the final product, with fillers, plasticizers, pigments, and stabilizers included, versus the resin - the homogeneous polymeric starting material. Plastics are polymers that are created by the chemical bonding of many identical or related structural units. (Borge 2000, 70)

A thermoplastic is a class of polymer in which the molecules are held together by weak secondary bonding forces that can be softened and melted by heat, then shaped or formed before being allowed to "freeze" again. The heating and cooling process can be repeated many times without significant chemical change. (Borge 2000, 71)

A thermoset is a polymer that solidifies irreversibly when heated due to a chemical reaction involving cross-linking between chains. Polyester, vinyl ester, and epoxy resins are most often the thermoset matrix of choice. (Borge 2000, 71) Polyester is the lowest cost resin and is commonly used in boats, vehicles, and household items. Epoxy resin shrinks less and is more expensive but provides better strength-to-weight ratio. (Cirrus 1999, 30)

Thermoplastics are easier to process than thermosets. Cycle times are shorter. Thick parts can be fabricated with no residual stresses due to the absence of an exothermic reaction. In addition, thermoplastics can be hydroformed; thermosets can not, giving thermoplastics an advantage in the potential ease and quickness of manufacturing. (Petersen 1998a, 3) Thermoplastics in general exhibit better flexural and impact performance and superior resistance to solvents; thermosets tend to have better compressive strength and abrasion resistance and significantly better dimensional stability. (Borge 2000, 71) Other thermoplastic advantages include significantly greater vibration damping, reprocessability, recycleability, very low volatile content, and an infinite shelf life with no refrigeration. Thermoplastic bonding methods include fusion techniques as well as the use of conventional adhesive. Repair is easier. Disadvantages include the fact that thermoplastic materials require a higher temperature for consolidation than thermosets require for curing. Thermoplastics require a high temperature autoclave and high temperature bagging materials, which are expensive. (Petersen 1998a, 3-4)

The selection of any resin system involves a compromise in properties. Thermoplastic systems, in particular, are particularly vulnerable to solvent degradation. Thermoplastic resins must be selected to resist solvent degradation, such as from fuel. In addition, the resin system must have proper viscosity characteristics, fairly long pot life, and high toughness. Unfortunately, the thermoplastic resins that are best meet these properties, and therefore suitable for general aviation aircraft, have a relatively low glass transition temperature (Tg), especially in relation to their high processing temperature. The Federal Aviation Administration (Federal Aviation Administration) currently requires the minimum Tg in structural components to be the maximum operating aircraft temperature plus 50°F. This follows the guidelines published in Mil-Handbook-17E (Volume 1E, Chapter 2, Section 2.2.8). To the extent of our knowledge, the lowest approved operating temperature on a composite general aviation aircraft is 150°F. The FAA will accept 180°F as a maximum operating temperature without further analysis. (Federal Aviation Administration 2000, 7) This means a prospective resin system must have a Tg of anywhere from 200 to 230°F. None of the thermoplastics that have been evaluated by general aviation aircraft manufacturers for this paper have had glass transition temperatures in that range. Aircraft manufacturers have the option of defining different temperature zones in the aircraft. It may be possible to define a location on the aircraft with a lower maximum temperature that would allow the use of a thermoplastic system. In addition, the FAA has lately opened the door to other methods of showing a resin system has sufficient strength at temperature. It has been proposed that showing a resin system retains two-thirds of its strength at the maximum operating temperature, when at the maximum operating temperature plus 50°F, would be equivalent to showing the resin system has a Tg equal to the maximum operating system plus 50°F. Or, shown as an equation, where MOT stands for maximum operating temperature:

$$\text{Strength}_{(\text{MOT}+50°\text{F})} = ?\ \ \text{Strength}_{(\text{MOT})}$$

1.3.1.1 Crashworthiness Study

As part of a crashworthiness study by an aircraft seat manufacturer, a composite subfloor was developed and evaluated for fabrication methods and projected costs. Thermosets and thermoplastics were compared to determine the most suitable material. All information in this section is from Mason-Reyes and Labun 1997.

The thermoset manufacturing process included the following steps:

1. Marking and cutting the appropriate fiber orientations of the thermoset material
2. Hand-laying the laminates so that they conformed with the tool's contours
3. Vacuum-bagging and autoclave-curing the completed layups on the composite tooling
4. Removing the parts from the autoclave and demolding the cured parts from the composite tool
5. Trimming, cutting, and assembling the parts into the final test element configuration

The thermoplastic manufacturing process included the following steps:

1. Marking and cutting the thermoplastic prepreg, in a manner similar to thermoset

2. Fabricating flat blanks by laying up uni-directional thermoplastic prepreg and consolidating them in a heated press
3. Loading the blank into a shuttle frame, inserting it into an infrared (IR) oven, and heating it to its processing temperature
4. Shuttling the heated blank out of the IR oven and positioning the blank between two matched molds
5. Clamping the molds onto the blank and placing it under high pressure
6. Opening the tool and removing the formed element, once the blank was cooled
7. Trimming, cutting, and assembling the parts into the final test element configuration

Material, labor, and tooling costs associated with the fabrication of the thermoplastic and thermoset subfloor test specimens were compared. The initial cost of the thermoplastic system is high, due to the high tooling cost associated with thermoplastic processing. However, this large cost is offset by the significant reduction in labor costs. Labor costs were the biggest difference between thermoset and thermoplastic production costs. The reason for such a significant difference is the cost associated with the time-consuming hand-layup of thermosets. The cost analysis shows that the break-even point between the two composite materials occurs after the 12th aircraft. After that point, the cost for a thermoplastic subfloor will decrease as the number of completed subfloors increases.

1.3.2 Reinforcements

Reinforcing materials may be a variety of fiber types, with the most common polymer matrix reinforcements being glass, carbon, and aramid.

Glass fiber is an inorganic, synthetic, multifilament material. Glass fibers are the most common of all reinforcing fibers for polymeric matrix composites. Glass fiber composites are strong, low in cost, nonflammable, nonconductive (electrically), and corrosion-resistant. Disadvantages are low tensile modulus, relatively high specific gravity, sensitivity to abrasion with handling, relatively low fatigue resistance, and high hardness (which causes excessive wear on molding dies and cutting tools). (Schwartz 1997, 2:2)

Carbon fibers are commercially available with a variety of tensile moduli. In general, the low-modulus fibers have lower specific gravities, lower cost, higher tensile and compressive strengths, and higher tensile strains to failure than high-modulus fibers. Among the advantages of carbon fibers are their exceptionally high tensile strength-to-weight ratios as well as their tensile modulus-to-weight ratios, very low coefficient of thermal expansion, and high fatigue strengths. The disadvantages are their low impact resistance and high electric conductivity. (Schwartz 1997, 2:5)

Aramid fibers, such as Kevlar, possess a tensile strength seven to eight times that of steel wire and have an excellent stability against temperature change. As a reinforcement, aramid fibers have been used in many applications where light weight, high tensile strength, and resistance to impact damage are important. The major disadvantages of aramid fiber-reinforced composites are their low compressive strengths and the difficulty in cutting and machining them. (Schwartz 1997, 2:3-4) There is also sensitivity to processing temperature and thermal expansion and contraction during processing. This can cause failures from moisture intrusion and delamination of composite parts. Aramid fiber manufacturers have improved the fiber's processing performance, but it is still a concern when proposing an Aramid design. (Davies 2001)

When a reinforcing material is being selected, the designer must also consider the form the fiber is used in, such as chopped fiber, woven fabric, or three-dimensional braid.

1.3.2.1 Chopped

Chopped strands are produced by cutting continuous strands into short lengths. The ability of the individual filaments to hold together during or after the chopping process depends largely on the type and amount of the size applied during fiber manufacturing operation. Strands of high integrity (i.e. those that hold together through the chopping process) are called "hard," and those that separate more readily are called "soft." Chopped strands ranging in length from 3.2 to 12.7-mm are used in injection and spray molding operations. Longer strands, up to 50.8-mm in length, are mixed with a resinous binder and spread in a two-dimensional random fashion to form chopped strand mats. (Schwartz 1997, 1:27)

1.3.2.2 Unidirectional

Unidirectional materials are available as rovings or "uni" prepregs. A roving is a group of untwisted parallel strands wound in a cylindrical forming package. Rovings are used in continuous molding operations such as filament wind and pultrusion. Rovings can be lined up and preimpregnated with resin into a tape product, which is referred to as "uni" prepreg. Uni prepreg is used in hand and machine layup techniques to provide highly directional properties.

1.3.2.3 Textile Materials

Textiles consist of both two- and three-dimensional fabrics and braids.

NASA initiated a research program in the early 1980's to investigate the potential of textile-reinforced composites as a cost-effective method of producing damage tolerant primary aircraft structures. Several textile material forms, including 2-D and 3-D weaving and braiding, multiaxial warp knitting, and through-the-thickness stitching were evaluated. Structural performance, fabricability, scale-up potential, and manufacturing cost were the primary focus of the research. As preliminary design databases were generated, NASA and US Aircraft manufacturers began to focus on the textile processes that offered the most promise. The seven textile processes listed in Table 1were evaluated.

Table 1 Application Potential of Textile Reinforced Composite Materials for Aircraft Structures

Textile Process	Advantages	Limitations
Low Crimp Uniweave	High in-plane properties Good tailorability Highly automated preform fabrication process	Low transverse and out-of-plane properties Poor fabric stability Labor intensive ply layup
2-D Woven Fabric	Good in-plane properties Good drapeability Highly automated preform fabrication process Integrally woven shapes possible Suited for large area coverage Extensive data base	Limited tailorability for off-axis properties Low out-of-plane properties
3-D Woven Fabric	Moderate in-plane and out-of-plane properties Automated preform fabrication process Limited woven shapes possible	Limited tailorability for off-axis properties Poor drapeability
2-D Braided Preforms	Good balance in off-axis properties Automated preform fabrication process Well suited for complex curved shapes Good drapeability Up to 10-foot diameter braids	Unable to get 90 degree fiber orientations Expensive machine setups on larger braiders for small runs
3-D Braided Preforms	Good balance in-plane and out-of-plane properties Well suited for complex shapes	Slow preform fabrication process Size limitation due to machine availability

Textile Process	Advantages	Limitations
Multiaxial Warp Knit	Good tailorability for balanced in-plane properties Highly automated preform fabrication process Multi-layer high throughput material suited for large area coverage	Low out-of-plane properties
Stitching	Good in-plane properties Highly automated process provides excellent damage tolerance and out-of-plane strength Excellent assembly aid	Small reduction in in-plane properties Poor accessibility to complex curved shapes

Source: (Dexter 1998, Fig 1) with comments by (McCabe 2001)

Four textile materials were selected for further evaluation: multiaxial warp knit (stitched and unstitched), 2-D triaxial braid (stitched and unstitched), 3-D braid, and knitted/stitched. (Dexter 1998, 1)

Folding or postforming of triaxial braided preforms offered the most flexibility in achieving small cross-section complex shapes. Multiaxial warp knitting proved to be the best process for large area multiaxial multiplayer broadgoods, but structural shapes have to be achieved through postforming and stitching. Through-the-thickness stitching proved to be the best textile process to achieve improved damage tolerance. NASA and Boeing demonstrated that through-the-thickness stitching of dry textile preforms could provide a 100% increase in damage tolerance of composite wing structures compared to laminated tape construction techniques. (Dexter 1998, 4)

One of the limiting factors of the three-dimensional fabrics is the development of repair concepts. Methods to reinfuse resin-starved areas must be developed and repair concepts to restore damaged structure to original strength must also be developed. (Dexter 1998, 4)

1.3.3 Preforms

Preforms are needed for all Liquid Composite Molding (LCM) methods. A preform is a stack of dry reinforcing materials (e.g. glass cloth, core, inserts) that will later be resin impregnated. Complex preforms provide LCM products with anisotropic properties. It is advantageous to have net-shape preforms prior to resin impregnation to eliminate secondary cutting and trimming processes. "Binders" are used to hold the reinforcing materials together prior to adding the resin. Binders can be as simple as a spray adhesive, or as complex as a low energy curing thermoset that is cured by ultra violet light. Mechanical stitching is also used.

1.4 Composite Processes

The processes identified below for polymeric composite materials are currently in use on certified composite aircraft. While other processing options exist, this section is designed to identify those actually in use.

1.4.1 Chopped Gun

Spray-up is applicable to glass and carbon but is of little use for aramids, owing to the difficulty of chopping the fiber cleanly. The resin, catalyst, and chopped reinforcement are simultaneously sprayed onto an open mold shape. Rollers or squeegees are then used to remove entrapped air and work the resin into the reinforcement. (Schwartz 1997, 2:28) This method is the most cost-effective fabrication for low to high production rates and is commonly used to make fiberglass swimming pools and boats. The fact that this method produces heavier parts with a lower strength to weight ratio of other methods limits its usefulness in aircraft (Cirrus 1999, 31), although it can still be used for non-structural parts such as interior panels. In addition, all directional property tailoring is lost; thereby reducing one of the initial benefits associated with aircraft composites. However, some may consider it an advantage since this method yields nearly isotropic properties, reducing the stress concentration problem associated with composite directionality.

Composite research firms are working on developing directional spray layup techniques. However, the process is currently not in use on any composite general aviation aircraft.

1.4.2 Hand Layup

Hand layup is the most labor-intensive process, but requires minimum tooling. It is used for prototypes and short production in the automotive industry and is well suited for low-volume, labor-intensive, large components such as boat hulls. Manual methods remain the most reliable for recognizing and removing problems such as wrinkles, and for quick, reliable pattern recognition. (Cirrus 1999, 30-31) Hand layup can be performed with prepreg materials or dry fabric that is reinforced with resin at the time of layup.

Hand layup of prepregs into laminates of the desired reinforcement and part geometry is a labor-intensive step and has accounted in large part for the cost of high-performance composite structures. Fabrication of hand layup parts represents more than 70% of the cost of producing finished components. (Schwartz 1997, 2:26)

1.4.2.1 Wet Layup

Wet layup is saturating dry reinforcing materials with resin at the time of part layup. Machines are available to essentially dip fabric forms into resin and provide a fairly consistent fiber-to-resin ratio. More commonly though, hand methods are used to force resin into the reinforcement. Once the plies are wetted out, they are placed in an open mold by hand, and may or may not be bagged. Room temperature or elevated cure may be used, based on the resin system. Pressure intensifiers, top caul sheets, or a top matched mold may be used to assist in consolidation. The part may also be vacuum bagged or autoclaved. Advantages are low tooling and processing costs. Disadvantages are high labor time and potential for personnel sensitization due to contact with resin.

1.4.2.2 Prepreg

Prepregs are thin sheets of fiber impregnated with predetermined amounts of uniformly distributed polymeric matrix. The resin is B-staged and kept in refrigerated storage until ready for layup. The prepreg is placed in a mold and bagged, usually with bleeder and breather materials. The entire assembly is cured at temperature with either vacuum or autoclave pressure to assist in part consolidation. Matched die molding may also be used, where a mold is placed on top of the prepreg instead of bagging materials. Advantages are much the same as for wet layup; low tooling and processing costs. Costs are elevated over wet layup due to refrigerated storage and the requirement for elevated temperature cure. Prepreg has reduced variability when compared to wet layup due to the automated resin impregnation system. Prepreg also dramatically reduces the hazards associated with personnel exposure to resins.

1.4.3 Automated Layup Methods

Automated layup methods include filament winding and fiber placement. Filament winding is most often used on cylindrical-shaped components such as pipes and ducts, storage tanks, and gas cylinders. In filament winding, a mandrel is rotated about one axis as a yarn or roving is wrapped around its surface. Matrix is applied to the filament by passing it through a bath of liquid resin prior to application on the mandrel. The filament is continuously wound on the mandrel, usually in complex patterns controlled by automated machinery. When the required wall thickness is reached, heat curing is performed either by autoclave, oven, or heating the mandrel. Once cured, the mandrel is removed through an open end in the part. Mandrels for tanks and pressure bottles are often inflatable, soluble, or collapsible so they can be removed through small end ports. Because the reinforcement is continuous and its orientation closely controlled, very efficient structures can be produced consistently by this method. (Grayson 1983, 643)

Fiber placement combines the differential tow payout capability of filament winding with compaction and cut-restart capability. Raytheon currently has several fiber placement machines from Cincinnati Machine that layup the Premier fuselage. During fiber placement process, individual prepreg fibers, called tows, are pulled off spools and fed through a fiber delivery system into a fiber placement head. Here they are collimated into a single fiber band and laminated onto a work surface such as a table, mandrel, or mold. (Evans 1997, 1)

When starting a course, the tows are restarted and compacted onto a surface. As the course is being laid down, the processing head can cut or restart any of the individual tows. This permits the width of the band to be increased or decreased in increments equal to one tow width. During the placement of a course each tow is dispensed at its own speed, allowing each tow to independently conform to the surface of the part. For example, when the head laminates a curved path, the outer tows of the fiber band pull more length than the inner tows. Compaction is achieved with heat and pressure at the time the tows are placed, removing trapped air and minimizing the need for vacuum debulking. (Evans 1997, 1-2)

Fiber placement works with common aerospace fibers such as glass, carbon, and Kevlar. The fibers must be impregnated with resin and formed into tows or slit tape. (Evans 1997, 5) Ideal fiber placement material has no tack at 70°F and high tack at 80 to 90°F. Low tack is needed when the material is pulled off the spool and guided through a fiber delivery system, but high tack is needed when being compacted on the surface. (Evans 1997, 9)

Tooling design is critical to high quality fiber placement. The tool design must pay special attention to surface stiffness, deflections due to an offset load, sag of the tool due to its own weight, and the rotational natural frequency of the tool. The electronic data for the machine's coordinate system must accurately represent the finished tool. The tool must be stiff enough so that it does not deflect as the head compacts tows on it. Heavy tools that are not symmetrical about the shaft can cause the shaft to twist. (Evans 1997, 13-14)

Two tooling approaches are used for fiber placement. One involves using the same mandrel to layup and cure the part. The other approach is to use one tool to layup the part and a separate tool to cure the part. When the second approach is used a separator film is placed between the part and the tool surface, so the part can be easily removed from the mandrel. (Evans 1997, 17)

Fiber placement is a very repeatable process and requires only a small amount of in-process inspection, but it is important to perform a rigorous first article inspection to verify that the part program fabricates a part that meets all of the design requirements. The detailed first article inspection should not need to be repeated unless a part program has been changed. After the first article inspection, subsequent parts need to have each ply visually inspected for gaps and overlaps, lost tows, twisted tows, wrinkled tows, and bridging tows. (Evans 1997, 47)

In addition to the initial cost for the fiber placement machine, the tooling for fiber placement usually costs more than tooling for a similar hand layup part, making non-recurring costs higher. Developing the process plan usually takes the same amount of effort as a typical part fabricated by hand layup. The time it takes to program a part depends on the number of plies and ply boundaries in the part. The greater this number, the longer it takes to prepare the CAD data and generate the programs. It usually takes about one half as much time to generate the programs as it does to prepare the CAD data. (Evans 1997, 49-50)

Recurring costs consist of in-cycle and non-cycle time. The in-cycle time is the time during which the machine is actually in the process of laying down material onto the part. The non-cycle time consists of machine preparation, tool preparation, tool installation, machine alignment, vacuum debulks, ply inspections, and machine time. Typically in production, 30% of the time it takes to fabricate a part would be non-cycle time. (Evans 1997, 50)

1.5 Tooling

The tremendous start-up investment required for constructing production tooling makes it difficult for small companies to "tool up" for production. The high cost also prohibits implementing design changes that would require re-tooling. Hence, it is important to produce durable, stable, temperature-tolerant molds at low cost not just to reduce initial investment, but to also allow design changes to be more readily implemented.

1.5.1 Industry Tooling Study

A study was conducted by a study by a general aviation manufacturer to develop and test a low-cost composite tooling system and design. The first stage of research involved evaluating several tooling resins and surface coats to determine the most effective system. The second stage explored the effects of variables in the design and construction of the tools. All information in the following sub-sections is taken from Kreimendahl (1999).

1.5.1.1 Phase 1

Machining metal billets to the desired contour has been the predominant method for producing composite part production tooling. These metal tools, while durable and dimensionally stable, are prohibitively expensive to manufacture for many general aviation aircraft manufacturers. When composite tools have been used in production, they have mostly been autoclaved prepreg construction, keeping costs high.

Phase 1 of the study evaluated four different tooling resin systems with their associated surface finish product. Regardless of the resin system used, the general construction of each of the molds was fairly consistent. The surface coat was applied to the surface of the pattern (also called the "plug") to an even thickness. Once ready, layers of fiberglass were added over the surface coat and wet with the tooling resin to form the surface laminate. The laminate was then vacuum-bagged to the pattern while the resin cures. The overall thickness of the surface laminate was approximately one-quarter inch. This thickness provides adequate support for the tool surface, yet is thin enough to allow good heat transfer during cure. Once the surface laminate was cured, the backing structure was added. The structure consisted of a frame that supports the perimeter of the tool with contoured ribs spaced along the tool. Once the tool had cured at room temperature, it was released from the pattern and trimmed. It was then put into an oven for post-cure, which is generally about 50° F greater than the desired service temperature. The tool is not supported by the pattern during post-cure, because the pattern is generally not able to withstand the oven cycle, due to added complications and expense that would be required. Once the tool was post-cured, it was inspected for vacuum integrity, surface quality, and dimensional accuracy.

From the several resins tested, it was determined that low viscosity and long working time were desirable characteristics of the tooling resin. Low viscosity facilitated wetting out the cloth and allowed trapped air to escape. The long working time allowed more plies of glass to be added per debulk step, in addition to providing more time for trapped air to escape.

By contrast, the ideal surface coat had medium to high viscosity and short working time. Both of these characteristics aid in keeping the surface coat in place where it was applied. Another important quality of surface coat is its repairability, as the tools will inevitably be damaged. Some surface coats allow the damaged area to be removed, sanded, filled with a new surface coat, and contoured to blend with the rest of the tool surface.

It should be noted that epoxies as well as surface coats are very sensitive to small variations to the proportions in which their components are mixed. An error of only a few percent can yield mixtures which don't cure, cure unevenly, do not bond to other materials, or at the very least, exhibit inconsistent properties. It is also crucial that the components be mixed together thoroughly.

1.5.1.2 Phase 2

In Phase 2 of the study, flap and spar mold sections were fabricated using one of the resin systems from Phase 1.

The time that the surface coat was allowed to set up before the fabric was added was varied together with the instruments to apply the surface coat. Various cloth orientations were experimented with, together with various types of cloth (weight and weave). Another variation in the surface laminate was the use of 18-inch squares to form three layers of "patchwork." This was an attempt to maintain vacuum integrity in the finished tool even if the surface coat is penetrated. The theory is that there are fewer continuous paths through the tool. Bagging with and without a separator film was attempted. Also the amount of bleeder material was varied.

All of the molds were in good condition after room temperature cure and being released from the patterns. However, upon post-curing some of the molds warped, including twisting and bowing. Up to 0.5-inch deflection was measured over the 11-foot span of the tool. Another significant problem was the appearance of bubbles in the surface coat after post-curing. The bubbles seemed to generally be between the surface coat and the laminate; however, a few were found under the first few layers of the laminate. Often the bubbles only formed near changes in geometry, such as radii. Marking can be described as small, but visible, deformations in the tool, usually occurring near the structural backing ribs of the tool. It appeared as if the surface between the ribs was either sagging or bowing upward. Cracks were also discovered in the surface coat near the radii of some of the tools. Upon post-cure of the tools, the weave pattern of the cloth laminate was sometimes visible through the surface coat. Although the texture of the surface coat was smooth to the touch, this "print-through" can be significant in operations where no secondary finishing of the part is performed.

Final results show that it is wise to use a balanced, symmetrical, quasi-isotropic design for the surface laminate. By using a symmetrical laminate with alternating plies of 0/90 and +/-45, tool warpage was eliminated. Also, it is better to use more plies of a thinner, lighter weave rather than a few thicker, heavier ones to more closely approximate in-plane isotropy. Slight variations of the symmetry of the laminate are acceptable. For instance, tight-weave plies and chopped-strand mat can be used as the first few layers to reduce the amount of print-though. Also, a few thick plies may be added at the plane of symmetry to reduce labor.

As for application of the surface coat, an even thickness is the main goal. Proper tackiness of the surface coat before application of fiberglass is also a factor. The tests indicate that it is best to allow the surface coat to set up until it is sticky to the touch, but does not come off on the finger.

All of the application tools tested (brushes, squeegees, and rollers) worked well. Depending upon the viscosity of the surface coat, brushes and squeegees may create ruts. Rollers, while they do provide even application, are not re-usable and therefore incur an additional expense.

1.5.2 Additional Tooling Technologies

A new method, patented by General Magnaplate Corp. of Linden, NY, manufactures tools with a reported 60% cost reduction compared to machined steel or aluminum ingot tooling, reduced delivery time from four months to one week, and weight reductions from 35,000 pounds to 400 pounds. The method is the trademarked Custom Moldmaking Process Technology (CMPT). CMPT tools are built to part net shape off of a fiberglass or foam “splash” of the component supplied by the customer. The tools are developed by a method similar to fiberglass composite layup in terms of warp and fill, only they use metal instead of fiber-reinforced prepreg. Molten metal is laid down in a crisscross woven pattern that allows formation of a tool face with low voids and low oxide contamination but with the same density as a heavier, more expensive, machined ingot. CMPT tools can be sealed to provide vacuum integrity for composite parts manufacture up to 750°F, or they can be made porous to allow the wicking of volatiles. Different surface coatings can be applied to the mold depending upon the customer’s requirements. Thickness of tool faces is on the order of 0.5-inch. A simple but effective tool face supporting jig or table is integrally molded into the tool face or fabricated separately and fastened to the tool.

1.6 Composite Material Qualification

The governing rules for aircraft and aircraft components are defined in Title 14 of the Code of Federal Regulations (CFR). Within Title 14, the rules for the Federal Aviation Administration are collectively called Federal Aviation Regulations (FAR’s). Part 23 of the FAR’s defines the rules for general aviation aircraft. There are other Parts to cover other aircraft categories; Part 25 is for transport category aircraft and Parts 27 and 29 are for rotorcraft. Other FAR Parts cover engines, propellers, and many other aviation-related subjects.

Within FAR Part 23, composite material requirements are defined in paragraphs 23.601, 23.603, 23.605, and 23.613. (Federal Aviation Administration 2000, 3) The latest revision regulations are made available on the FAA web site, www.faa.gov. Copied below are the latest revisions in Part 23 as of October 2000.

23.601 General
The suitability of each questionable design detail and part having an important bearing on safety in operations, must be established by tests.

23.603 Materials and workmanship
(a) The suitability and durability of materials used for parts, the failure of which could adversely affect safety, must--
(1) Be established by experience or tests;
(2) Meet approved specifications that ensure their having the strength and other properties assumed in the design data; and
(3) Take into account the effects of environmental conditions, such as temperature and humidity, expected in service.
(b) Workmanship must be of a high standard.

23.605 Fabrication methods
(a) The methods of fabrication used must produce consistently sound structures. If a fabrication process (such as gluing, spot welding, or heat-treating) requires close control to reach this objective, the process must be performed under an approved process specification.
(b) Each new aircraft fabrication method must be substantiated by a test program.

23.613 Material strength properties and design values
(a) Material strength properties must be based on enough tests of material meeting specifications to establish design values on a statistical basis.
(b) Design values must be chosen to minimize the probability of structural failure due to material variability. Expect as provided in paragraph (e) of this section, compliance with this paragraph must be shown by selecting design values that ensure material strength with the following probability:
(1) Where applied loads are eventually distributed through a single member within an assembly, the failure of which would result in loss of structural integrity of the component; 99 percent probability with 95 percent confidence.
(2) For redundant structure, in which the failure of individual elements would result in applied loads being safely distributed to other load carrying members; 90 percent probability with 95 percent confidence.
(c) The effects of temperature on allowable stresses used for design in an essential component or structure must be considered where thermal stresses are significant under normal operating conditions.
(d) The design of the structure must minimize the probability of catastrophic fatigue failure, particularly at points of stress concentration.
(e) Design values greater than the guaranteed minimums required by this section may be used where only guaranteed minimum values are normally allowed if a "premium selection" of the material is made in which a specimen of each individual item is tested before use to determine that the actual strength properties of that particular item will equal or exceed those used in design.

Part 21 of the FAR's controls, among other things, continued quality assurance requirements for aircraft manufacturers. An applicable rule from there is 21.33(b)(2), copied below from the FAA website as of October 2000.

> 21.33 Inspection and tests
> (b) Each applicant must make all inspections and tests necessary to determine--
> (1) Compliance with the applicable airworthiness, aircraft noise, fuel venting, and exhaust emission requirements;
> (2) That materials and products conform to the specifications in the type design;
> (3) That parts of the products conform to the drawings in type design; and
> (4) That the manufacturing processes, construction and assembly conform to those specified in the type design.

These rules mean that, in practice, aircraft manufacturers must control composite materials through four processes – initial material qualification, supplier quality assurance, incoming receiving, and in-process verification. Each manufacturer is allowed to propose to the FAA the ways they will show compliance with these rules. The FAA publishes Advisory Circulars (AC's) that describe acceptable means, but not the only means, of showing compliance to the rules. The FAA also publishes Policy Letters that are less formal than AC's, but still offer acceptable means of showing compliance to FAR's. AC 20-107A (Federal Aviation Administration 1984) has been the standard for composite material qualification, describing initial material qualification, as well as full-scale test guidelines required under other rules. AC 21-26 (Federal Aviation Administration 1989) provides guidance on acceptable means of demonstrating compliance of quality control systems for the manufacture of composite structures.

These guidelines have resulted in the creation of composite material databases by aircraft and aircraft component manufacturers that were then kept company proprietary. Anyone else who wanted to certify the same material and process had to redo initial material qualification testing, to the tune of hundreds of thousands of dollars per material system. The databases were created using the methods in Mil-Handbook-17E, whose guidelines require five batches of material and six replicates per batch for each type of test/property being evaluated.

Those FAA guidelines are in the process of being superseded for initial material qualification of polymer matrix composite materials under draft Policy Statement published in the Federal Register on June 13, 2000. (Federal Aviation Administration 2000)

This method is already proven for prepreg materials. Under these guidelines, an interested party, such as a composite material manufacturer, funds the development of A- and B-Basis design values using three batches of materials, with six replicates per batch, tested at four environmental conditions.[2] Composite end users must then show "material equivalence" to the FAA approved material database. The Policy Letter defines material equivalence testing as the testing of a single batch of material with eight replicates per test condition at a single environmental condition (room temperature, dry). This procedure significantly reduces the economic burden on aircraft fabricators to certify the materials and processes. Equivalence testing is used to prove that the manufacturing techniques and processes employed by the end-user will produce equivalent strengths and stiffness as the initial database generated by the material manufacturer. The policy letter also gives guidance on continuous quality control testing.

This policy has led to a draft technical report in September of 2000 titled "Material Qualification and Equivalency for Polymer Matrix Composite Material Systems," DOT/FAA/AR-00/47. This document is intended to become the standard FAA guidance material for polymer matrix composite initial material qualification for general aviation manufacturers, as well as transport category, rotorcraft, and all other manufacturers covered under the Federal Aviation Regulations.

The new published policy does not affect the glass transition temperature (Tg) requirement discussed in Section 1.3.1. For now, all composite resin systems, whether used in reinforced laminates or in structural adhesives, require a Tg equal to the maximum operating temperature plus 50°F.

These guidelines are directly applicable to prepreg composites, either hand laid up or fabricated with mechanical methods such as tape laying or fiber placement. Wet layup, or other methods that involve resin mixing and infusion of dry reinforcements at the aircraft manufacturer, require individual qualification plans. Many of the same philosophies described in the Policy Letter can likely be followed, such as number of batches and replicates per test condition. However, additional testing is required for other material/process combinations to take into account variables such as resin/hardener mix ratio, cure cycle variations, etc. Structural adhesives also require individualized certification plans, as do non-homogeneous laminates, such as those that include honeycomb core or foam centers.

Louis F. Vosteen of Analytical Services & Materials, Inc. and Richard N. Hadcock of RNH Associates wrote a paper in 1995 under contract to NASA Langley. They conducted a study of past composite aircraft structures programs to determine the lessons learned during the course of those programs. The study was focused on finding major underlying principles and practices that experience showed could have a significant effect on the development process and should be recognized and understood by those responsible for making effective use of composites for aircraft structures. Published

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[2] The reader is encouraged to read Mil-Handbook-17E or the FAA guidance materials for further information on the definition of such terms as "A-Basis," "B-Basis," "batch," "replicate," and "environmental condition."

information on programs was reviewed, and interviews were conducted with personnel associated with current and past major development programs. In all, interviews were conducted with about 56 people representing 32 organizations. Most of the people interviewed have been involved in the engineering and manufacturing development of composites for the past twenty to twenty-five years. In the paper they summarize "Lessons Learned" from a review of aircraft that have been FAA certificated with composite structure. (22-23)

1. Design and certification requirements for composite structures are generally more conservative than for metal structures.

2. There are no reported aircraft accidents involving failure of primary or safety-of-flight composite structure.

3. Design and certification of composite structures are expensive and costs are much higher than for metal structures. The effort and costs associated with design and certification have often been underestimated.

4. All design and certification requirements must be thoroughly evaluated and understood at the start of the program. Certification requirements of the FAA and the various foreign certification offices can differ and require different approaches.

5. Design and certification test data generated under a military aircraft program has only rarely been transferable to a commercial aircraft program or a program sponsored by another military department and vice versa. This practice has resulted in considerable duplication of effort.

6. Successful programs have made effective use of integrated development teams that include personnel experienced in design, analysis, materials and processes, tooling, quality control, production, and cost analysis.

7. Experience gained in R&D programs does not readily transfer to production unless the people with the R&D experience participate actively in the production development.

8. Successful programs have used a building-block approach to development. Program managers with prior composites experience usually understand the necessity of realistic schedules that allow a systematic development effort.

9. The use of a basic laminate family containing 0/90/±45 plies with a minimum of 10% of the plies in each direction is well suited to most applications, generally assures fiber dominated laminate properties, and simplifies layup and inspection.

10. The number of mechanical joints should be minimized by utilizing large cocured or cobonded subassemblies. Mechanical joints should be restricted to attachment of metal fittings and situations where assembly or access is impractical using alternative approaches.

11. Large, cocured assemblies reduce part count and assembly costs. If the cocured assembly requires overly complex tooling, however, the potential cost savings from low part count can be easily negated. Producibility must be a key consideration in the design.

12. Structural designs and the associated tooling should be able to accommodate design changes associated with the inevitable increases in design loads.

13. Standardization of techniques for inducing impact damage and assessing its effects would eliminate confusion and permit direct comparison of test data and transfer of results to other programs.

14. Designing for producibility is generally more cost effective than optimization for weight savings.

1.7 Lightning Protection

Lightning protection was evaluated by Lightning Technologies, Inc., under contract to NASA AGATE. A "Direct Effect Handbook" was the result of the contract. All information in this section is taken from that handbook (Lightning Technologies, Inc. 1998) and summarizes the important points of composite aircraft design to incorporate lightning protection.

1.7.1 Lightning Effects

The effects of lightning on skins, both metallic and composite include:

1. Melting or burning at lightning attachment points
2. Resistive temperature rise
3. Magnetic force effects
4. Acoustic shock effects
5. Arcing and sparking at bonds, hinges and joints
6. Ignition of vapors within fuel tanks

Not all materials suffer from these effects equally. Aluminum skins will suffer most from melting at lightning attachment points. While they will be subject, like composites, to acoustic shock damage, their greater ductility and malleability will likely enable htem to survive. Composites will suffer the most from acoustic shock waves. It must be emphasized, that carbon composites are conductors, albeit resistive conductors. They are therefore subject to the same influences as metal structures, although in different degree. They are, for example, subject to magnetic forces, as well as arcing and sparking at bonds and resistive heating. Non-conductive composites, such as fiberglass and aramid fiber reinforced plastics will be subject to dielectric breakdown and puncture.

Aircraft structures include the outer skins of the aircraft, together with internal framework, such as spars, ribs, frames and bulkheads. Lightning currents must flow between lightning entry and exit points on the aircraft and tend to spread out as they flow between attachment points, using the entire airframe as a conductor. Any conductive material, metal, or conductive composite with which most of these structures are fabricated becomes part of the conductive path for lightning currents.

1.7.2 Protection Requirements and Plan

The primary focus of the design guidelines for lightning protection is to minimize lightning effects caused by severe lightning attachment to the aircraft and enable compliance with the lightning direct effects protection requirements contained in the Federal Aviation Regulations (FAR's). The objectives of lightning protection design direct effects are:

1. Prevent catastrophic structural damage
2. Prevent hazardous electrical shocks to occupants
3. Prevent loss of aircraft flight control capability
4. Prevent ignition of fuel vapors

The FAR's state that compliance can be shown by either bonding components to the airframe or by designing components so that a strike will not endanger the airframe. In this context, the term "bonding" refers to electrical connections among components sufficient to withstand lightning currents. Unfortunately, the emphasis on bonding has led some designers to conclude that bonding, by itself, will provide adequate lightning protection for an aircraft and that little else needs to be done. To them, a lightning protected aircraft has meant a "bonded" aircraft. Verification of this "bonded" status has, in turn, been signified by attainment of a specified electrical resistance among "bonded" components. This is achieved by allowing metal-to-metal contact among parts and verified by a dc resistance measurement.

Criteria like the 2.5 milliohm bonding specification in MIL-B-5087B have taken on an importance all of their own, to the neglect of the real purpose of the design. Whereas electrical continuity among metal parts of an aircraft is important, there are many other features of a successful protection design that are of equal or greater importance.

Experience has shown that the most successful lightning protection design and certification programs have occurred when the work is conducted in a logical series of steps. In this case, success means achievement of a satisfactory protection design and compliance with the regulations; all with a minimum impact on weight and cost. The specific steps and order of occurrence may vary somewhat from one program to another, but most programs include the following:

1. Establish the lightning zone locations. Lightning zoning is a functional step in demonstrating that the aircraft is adequately protected both from direct and indirect effects of lightning. The purpose of lightning zoning is to determine the surfaces of the aircraft which are likely to experience lightning channel attachment and the structures which may experience lightning current conduction between pairs of entry and exit points. There are three major divisions.
   - Zone 1 is defined as regions likely to experience initial lightning attachment and first return strokes.

- Zone 2 is defined as regions which are unlikely to experience first return strokes, but which are likely to experience subsequent return strokes. This will happen where the aircraft is in motion relative to a lightning channel causing sweeping of the channel backwards from a forward initial attachment point.
- Zone 3 is defined as regions which are unlikely to experience any arc attachment, but which will have to conduct lightning current between attachment points.

Regions 1 and 2 are further subdivided into specific lightning attachment zones.

- Zones 1A and 2A are areas where long hang-on of a lightning channel is unlikely because the motion of the aircraft with respect to channel causes the arc root to move across the surface of the aircraft in the opposite direction from the direction fo motion.
- Zones 1B and 2B are regions where the lightning channel is unlikely to move during the remainder of the flash because the location is a trailing edge or a large promontory from which the relative motion of the aircraft and channel cannot sweep the attachment point further.
- An additional zone, 1C, is defined as one which, by virtue of the change in current parameters along a lightning channel and the time taken for sweeping of the attachment point across the surface of the aircraft, the threat to the aircraft is reduced.

2. Identify systems and components that are performing flight critical or essential functions. Determine of any of these structures and systems could be damaged or upset by direct or indirect lightning effects.

3. Establish protection criteria. In this step, the specific criteria for each of the structures and systems in need of protection should be decided upon. For direct effects, this will include definition f the degree of physical damage that can be tolerated by flight critical/essential structures, and establishment of ignition free criteria for the fuel tanks.

4. Design protection. In this step, specific design additions, changes or modifications are made to the flight critical/essential structures, systems, and components to enable them to meet the protection criteria established in Step 3.

5. Verify protection adequacy. Verification of protection can be accomplished by analysis, similarity to previously proven designs, and/or by test.

1.7.3 Protection Methodology

Conductive composites such as carbon fiber reinforced composites, have adequate conductivity to prevent electric field penetration and prevent internal streamers. Non-conductive composites with conductive coating will also prevent electric field penetration when coatings approach 100% coverage.

Non-Conductive composites are prone to punctures by lightning initiated streamers since electric fields are able to penetrate these skin materials and induce streamers from conductive objects within. Therefore, when employed as an exterior skin, they may be punctured by a lightning flash that contacts a conductive object beneath the skin. The high amplitude return stroke currents can then result in significant damage to these materials. Unprotected airframe skins fabricated of fiberglass or other non-conductive composites are subject to such puncture and damage.

There are two basic ways of providing protection for non-conductive composites. One employs diverter strips or bars on the exterior surface to serve as preferred streamer initiation points and intercept lightning flashes, while allowing the skin to be transparent to electromagnetic waves. This is the approach used for protection of radomes and some antenna fairings. The other method is to apply an electrically conductive materials over the exterior of the structure. This latter method provides the most effective lightning protection and should be employed whenever possible. It also provides improved protection of enclosed systems against lighting indirect effects.

Protective materials include arc or flame sprayed metals, woven wire fabrics, expanded metal foils, aluminized fiberglass, nickel plated aramid fiber and metal loaded paints. All of these approaches must be applied to the exterior surface of a composite skin, and there should only be one layer of protective materials. Keeping the lightning currents on the outside of the aircraft will significantly minimize effects to systems and personnel. Dielectric coatings cannot protect the entire aircraft from lightning attachment. On the contrary, the dielectric may actually increase damage at locations where the attachment occurs. Dielectrics should only be used to provide protection for small regions or components of the aircraft.

Summary of test results of lightning protection on fiber-reinforced composite skins:

1. Conductive paints do little to protect against a severe strike with a conductive composite beneath.
2. Unperforated (solid) foils work well in aluminum, copper or the more resistive nickel. Application and maintenance problems persist however. Most designers will prefer the perforated and expanded variety.
3. Typically, protection from les conductive materials (such s nickel or stainless steel) will not perform as well as more conductive materials. There is likely to be little difference in maintenance, apart from galvanic concerns. There is also likely to be little difference n application.
4. A layer of significant dielectric strength placed over a conductive layer (such as CFC) will typically increase damage to the conductive layer when attachment occurs.
5. Fiberglass non-conductive panels can have puncture prevented when protected by expanded foils of either copper or aluminum.

Within a fuel tank, the principal challenge of designing fuel cells lies in protecting the fuel system from ignition by lightning. Protection of fuel systems will involve one of more of the approaches:

1. Containment: Designing the structure to be capable of containing the resulting over-pressure without rupture.

2. Inerting: Controlling the atmosphere in the fuel system to ensure that it cannot support combustion.
3. Foaming: Filing the fuel systems with a material that prevents a flame from propagating.

The following design approaches should be utilized to the extent practical to minimize potential ignition sources within aircraft fuel tanks.

1. Design the fuel tank structure to minimize the number of joints, fasteners and other potential arc and spark sources in fuel vapor areas.
2. Provide adequate electrical conductivity between adjacent parts of structures
3. Provide a barrier to separate remaining arc or spark products from the fuel vapor.
4. Design fuel system components to interrupt potential current.

2 LOW-COST MANUFACTURING METHODS

This chapter discusses low-cost manufacturing methods that have been studied by composite aircraft manufacturers. Most low-cost studies have focused on Liquid Composite Molding (LCM) that encompasses Resin-Transfer Molding (RTM) and Resin-Film Infusion (RFI) techniques.

Liquid composite molding allows the user to lay up all of the dry fabric into a mold, arrange the core shapes where they need to be, close the mold or apply bagging materials, and then inject the resin into the part. With the exception of prepreg materials, traditional composite processing methods have been to wet out the dry fabric layers by hand individually as they are laid into the mold. After this, the vacuum bagging materials are hurriedly attached to the mold with hopes that the debulking and bleeding will take place before the resin "kicks," or begins to gel. This process results in relatively low fiber volume fractions, high resin wastage and more than desirable human contact with hazardous chemicals. Prepreg materials reduce such disadvantages, but are more expensive to store and process, as they must be stored in a freezer and then almost all are cured with heat in an oven or autoclave.

Liquid Composite Molding methods have numerous benefits. There is no need to "rush" the layup due to a short working time for the resin. There is no limit on the amount of time one may spend to arrange the fabric and core shapes, as they are dry. It is possible to pull vacuum on the part to check ply orientation, core location, etc., before the resin is added. Better fiber volume can be achieved without an autoclave since only the fabric is compacted during the debulk process. Worker contact with resins is minimized. Freezer storage is not required.

One of the disadvantages of LCM is the fact that the mold system is often defined by trial-and-error. Molding methods are difficult to simulate unless used on simple parts with resin injection at relatively low pressure and constant porosity of the part throughout injection. FiberSIM, CAD-integrated composite design software from Composite Design Technologies, and LCMFLOT, mold-filling analysis software from Liquid Process Performance Prediction, have been linked to enable companies using LCM to more accurately simulate the flow of resin through the mold without the need to build and test a prototype mold. The new simulation method provides an accurate description of resin flow during injection by detecting spatial variations of porosity due to composite material draping. This method can incorporate the significant effect of local porosity variations in the curved regions of complex or large parts, ribbed parts made of an assembly of flat panels, parts of variable thickness, or parts containing sandwich elements. (Automotive Engineering 2000, 123)

2.1 Resin-Transfer Molding (RTM)

RTM is the process by which dry fabric reinforcements are placed into a mold that consists of two matched halves. When this mold is closed, it forms a hollow pressure tight cavity. A liquid resin and reactive catalyst are then transferred from a holding tank into the closed mold cavity. This action infuses or wets the fabric surrounding all the fibers with the resin system. The mold is heated for a set period of time allowing the liquid resin and catalyst to cure and harden into a solid. When the cure is complete the part is removed from the mold.

Resin transfer molding has the potential to overcome the disadvantages of traditional hand layup by being less labor intensive and having the ability to form complex shapes and hollow parts. RTM has some advantages over prepreg construction. Prepreg is stiff and difficult to form into shapes. RTM does not require secondary bonding. RTM has better surface finish because it has hard tooling on all sides. The main disadvantages of RTM are associated with tooling. Resin flow is difficult to control, making mold design and temperature critical. Tool tolerances are important. This combines to make tooling costs high, and there are additional costs for pumps and a press.

An emerging, and potentially lower cost version of RTM is Vacuum-Assisted RTM or VARTM. VARTM utilizes male or female tooling combined with a vacuum bag much like an autoclave or oven cure process. This process offers the advantages of RTM without the expense of matched metal die tooling. VARTM has opened up the potential applications for RTM to larger, more complex structures; however, it has not been studied under AGATE contract and is not presented further in this section.

2.1.1 Industry RTM Study

A general aviation manufacturer performed a study in 1998 to fabricate both a horizontal stabilizer and inboard flaps utilizing different RTM techniques, which is documented in Petersen (1998b). The horizontal stabilizer was later structurally tested and documented in Petersen (1998c). All information in this section and the following sub-sections is taken from those references.

The RTM press was pneumatic and used fire hoses to provide clamping force. The top platen was heated with rod heating elements and fixed to an I-beam support structure. The I-beams were individually raised and lowered to provide different shut heights to the press. The I-beams had to be level to avoid damaging the platen. The tool rested on fire hoses that were on a heated cart. Inflation pressures were up to 150 psi. A data acquisition system was required to measure thermocouples, pressure transducers, and flow meters.

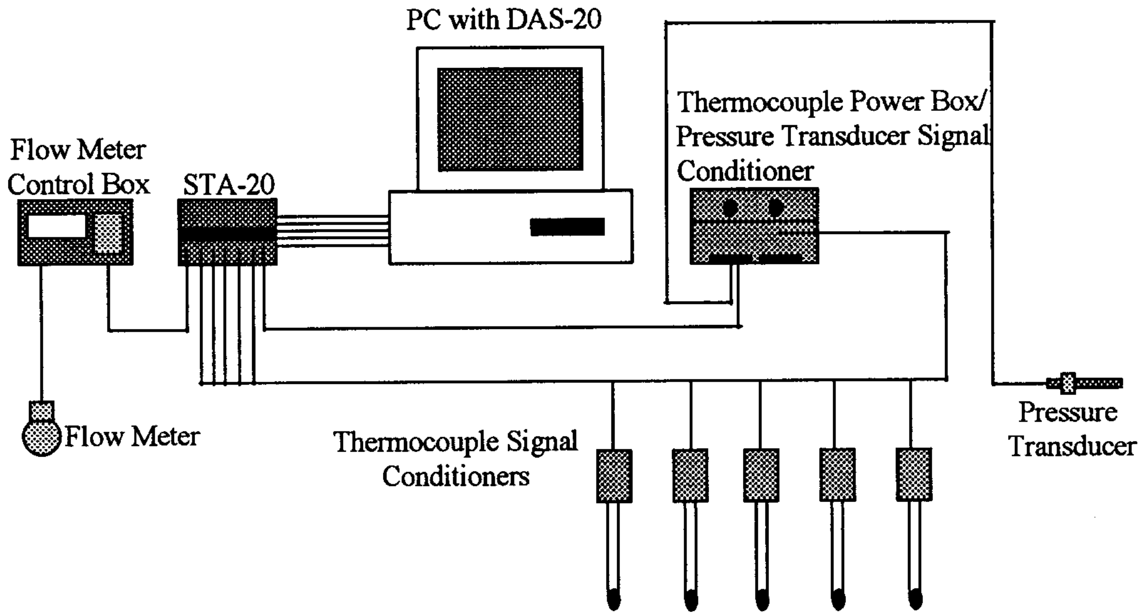

Figure 3. Schematic of Data Acquisition System

Comparing the RTM horizontal stabilizer to a conventional aluminum one, RTM took 15-20 labor hours to build; an aluminum one is estimated at 100 hours. For the flap, RTM was 20 labor hours to build, with a normalized part weight of 0.98; honeycomb composite structure has an estimated build time of 118 labor hours and a normalized weight of 1.0.

2.1.1.1 Horizontal Stabilizer

The layup consisted of 1-5 plies of plain weave fabric over the upper and lower tools plus braid over the internal mandrels. Ciba RTM6 epoxy resin was selected for the resin system because it has good viscosity characteristics and fairly long pot life. However, degassing was required, which added to the total injection time. It was difficult to clean the resin out of the pump after fabrication. The resin is toughened, but is still not very tough compared to the toughest prepregs. The horizontal stabilizer required 1.7 gallons of resin and took about 30 minutes to infuse.

The horizontal stabilizer tool had female outer mold tool halves made of epoxy resin with small aluminum spheres (BB's) embedded in the resin. The material was selected because the coefficient of thermal expansion (CTE) matched that of the RTM composite. The mandrels were tapered so that they could be pulled out of the stabilizer after it was cured. Some were pulled from the root and some were pulled from the tip.

The outer molds were clamped together in a press and heated. A vacuum pump removed the air and resin was pumped into the mold. The seal between the molds was a double O-ring that had a low power vacuum pump producing a vacuum between the O-rings to help hold the tool halves together and keep resin from leaving the mold cavity.

Several cracks developed in the molds during their first use due to thermal stresses caused by uneven heating and cooling of the molds. Epoxy filler was placed in the crack, but did not seal. The RTM resin from the next part eventually sealed the mold. A mold material that can withstand thermal cycling without cracking is needed for production.

The leading edge mandrel was reinforced with steel rods along the entire length. The trailing edge mandrel was not reinforced and broke several times. For production, all mandrels should have steel reinforcing rods.

Steps used to fill the horizontal stabilizer mold:
1. Apply mold release to all tooling components.
2. Lay a cloth preform into the bottom tool.
3. Hang mandrels vertically and apply trim sock.
4. Slide each mandrel into place one at a time, being careful to engage the tang in the slots equally at each end to avoid racking.
5. Insert Teflon blocks in the spaces between the mandrel tangs. Wrap the skin preform cloth around the mandrel system.
6. Place o-rings in the o-ring grooves.
7. Lower the top tool half in place.

The first problem encountered during horizontal stabilizer fabrication was fiber pinching at the leading edge when the two halves of the mold were closed. The first solution was to insert a metal cuff, which prevented pinched fibers but was covered with a thin coat of resin that flaked off. The final solution was a one ply precured composite cuff.

Full-scale production was performed by attaching the vacuum pump to the main vacuum port in the tool. Thermocouples were connected. The tool was heated to the injection temp of 250°F. Vacuum was applied and checked for leaks. Resin was released into the tool and the vacuum port clamped off. Resin was pumped at 80 psi until the tool was full. The press pressure was 100 psi. Several parameters are critical to part quality: good vacuum; correct temperature; press pressure must be at least 20 psi greater than the resin pressure; the preform must be properly compressed so that the tool is closed and the correct tolerances applied. After the two-hour, 350°F cure, the part was removed from the tool. This must be accomplished while the tools are still hot due to the difference in tool and part CTE, leading to possible warpage during cooling.

Five horizontal stabilizers were made. The first one had many dry spots because the mold warped and cracked during the first use and could not hold vacuum. Repairs and resin infusion into the tool from the second part sealed the tool and resulted in several good parts.

The horizontal stabilizer was subsequently structurally tested. It showed that strong, high-quality parts can be produced by RTM. The strength of the horizontal stabilizer met or exceeded the design requirements. The stabilizer was designed to carry limit load without buckling. The test results showed it met this requirement. The final failure load exceeded the ultimate design load. This structural test shows the horizontal stabilizer had good quality and no structural flaws causing unexpected failures.

2.1.1.2 Flap

The flap layup was similar to the horizontal stabilizer, in that it consisted of several layers of plain weave fabric on each mold half in addition to fabric braids over the internal mandrels. 3M PR520 epoxy resin was selected for the flap. The main advantage of this resin is toughness, which is important for a flap that may be impacted with runway debris. The main disadvantage is that it is expensive.

The flap tool was CNC machined from solid aluminum. The tooling system had nine primary tooling pieces: the two outer mold line halves, the five inner mold line mandrels, and the two mandrel end fixtures. The lower half tool had one injection port, one primary vacuum port, and two auxiliary vacuum ports. The upper half tool was equipped with a vacuum port that provided vacuum between the o-rings that seal the mold halves. A pneumatic demolding system was required to force the mold halves apart. Each mandrel is equipped with a tang at each end for fixturing into the slots of the mandrel fixtures. Flaps were constructed similarly to the horizontal with the following exceptions: The flap did not have a “c” channel at the trailing edge. The flap had only one braid per mandrel. The flap has no additional plies between the braids, which form the spars. Four plies of plain weave were placed in the lower half of the mold. Braids were pulled over the mandrels and placed in the mold. The skin plies were wrapped around and butt joined on the upper surface with film adhesive used to keep them in place.

Steps used to fill the flap mold:

1. Apply mold release to all tooling components.
2. Lay a cloth preform into the tool bottom.
3. Hang mandrels vertically and apply trim sock.
4. Position mandrel fixtures on flat surface.
5. Insert tangs of the smallest mandrel into sockets of the mandrel fixtures.
6. Pin in place.
7. Slide each mandrel into place one at a time, being sure to engage the tangs in the slots equally at teach end to avoid racking.
8. When all mandrels are engaged in the slots, gently pin the mandrels in place.
9. Insert Teflon plugs in the lower demold sockets of the fixtures.
10. Lower the mandrel/preform system into the tool bottom.
11. Insert Teflon plugs into the upper demold sockets of the fixtures.
12. Wrap the skin preform cloth around the mandrel system.
13. Place o-rings in the o-ring grooves.
14. Lower the top tool half in place.

The same infusion process was followed for the flap as was followed for the horizontal stabilizer, except the press pressure was 150 psi and the pump pressure was 100 psi. The tool did not fill the whole press area. It is recommended that a fixture the same height as the tool be put in the press to make the stresses uniform on the press and the tool. Special procedures may be required to remove the part from the tool, such as using air pressure within the mandrel to release the part.

2.1.1.3 Lessons Learned

1. Tooling must be selected that can withstand thermal cycling without cracking.
2. Mandrels should be steel reinforced.
3. The press should be sized to the tool, or additional fixtures used in the press to ensure uniform stresses on the press and the tool.

2.1.2 Second Industry RTM Study

A different general aviation manufacturer performed a study in 1999 to demonstrate the capability to produce a RTM structure that has blind cavities requiring the use of innovative lost core technology. The project was a manufacturing technology demonstration program, and was not intended to evaluate a specific design of a given structure. The purpose was to prove the feasibility of the process and to learn more about using the lost core RTM process to produce large complex structures.

All information in the following sub-sections is taken from Bowden (1999).

2.1.2.1 General

RTM creates solid parts of resin and fiber reinforcement. To make stiffer and lighter parts they must be hollow or filled in the center with a low-density core material. This feature makes RTM more complex. The dry fabrics must be wrapped around the core before they are placed in the mold. The fact that the resin cannot flow through the core and must go around it further complicates the flow path and makes wetting the fabric difficult.

The most common method of producing a hollow RTM part is making a cored section with a removable mandrel or tool insert, as used in the study discussed above. The largest disadvantage is that the part design must allow the mandrel to be removed, which limits the design and functionality of the part.

In some cases, mandrel removal cannot be designed into the part. The mandrel must then be an inflatable material that can expand or collapse, similar to the system discussed below in Section 3.3. Inflatable mandrels generally do not last very long in a production setting and must be periodically replaced due to the rigors of exposure to heat and resin catalysts.

The other option is a lost core mandrel. The lost core material is used just like the permanent mandrel with the fabric wrapped over it and then placed in the mold. It is then melted and or washed out of the part through drain holes after the part has been cured. Lost core material has the disadvantage of having to be reproduced each time a part is produced, hurting economic justification of such a process.

The general aviation manufacturer partnered with Fiber Dynamics Inc. (FDI) in Wichita Kansas to assist in this investigation using their proprietary lost core formulation. FDI has demonstrated production rates on recreational bicycle parts of 12,000 per year using this new core chemistry. This evaluation was undertaken to evaluate the technical feasibility of the lost core RTM process for large aerospace structures and how the technology competes with traditional sheet metal materials.

2.1.2.2 Test Article

The part candidate for trade study had to meet the following basic criteria:

1. It had to be both large and complex enough to ensure the technology was pushed to its limits
1. There had to be an existing sheet metal structure similar enough to be able to make reasonable cost and weight comparisons.
2. It had to be a structure that could be realistically commercialized or incorporated into a new or existing design.

This led to the selection of a center one-third span of a typical sheet metal horizontal stabilizer found on a business jet aircraft. Part length was limited to less than 86 inches by press platen size. The leading edge was omitted due to tooling cost constraints. The composite part cocured all skins, spars, ribs, and stringers in one layup and injection cycle. The layup could easily be scaled up to a wing or down for a horizontal stabilizer sized to general aviation aircraft.

The part was modified to use good design practices in a composite part. The bulbed T-stringers were converted to hollow hat sections. The C-shaped ribs were converted to I-shaped cross-sections. The rib lightening holes were reinforced with a ring doubler. Flanged holes would have been preferred but would have complicated the tooling further. Cross section thicknesses were analyzed and modified to provide a comparable strength and durability with respect to the sheet metal design. Most areas were reduced in thickness, but some were increased, especially in the bearing areas for fitting attachment. Most layups were three- or four-ply quasi-isotropic layups. Spar caps incorporated several large stack-ups of 0° graphite uni tape interspersed with +/- 45° double bias knitted carbon tape.

The material used for the lost cores was developed by Fiber Dynamics. The chemistry was not revealed. It is referred to by “FD180-CF.” FD180-CF lost core material is a low density, moldable compound, which melts and washes out of the cured part at elevated temperatures. The material is completely recyclable and generates no hazardous waste through the core removal process.

Commercially available 3M Spray Adhesive 77 was used to position the dry fabrics. Three types of fabrics were used: a plain weave, a uni tape with the fibers stabilized with a non-crimping stitch, and a two-ply double bias +/- 45° uni tape. The two plies were also stabilized with a non-crimping stitch. Shell EPON Resin 9504 and Shell EPI-CURE 9554 Curing Agent were selected for the resin system, based on viscosity, pot life, and cost. Viscosity was particularly important due to the complex nature of the stabilizer features.

2.1.2.3 Equipment

A 50-ton hydraulic press was used for mold closure and clamping. Mold temperature was controlled with oil-heated platens. The oil-heated platens were secondary platens that were isolated from the hydraulic press platens with a phenolic, mineral fiber sheet. Resin injection was performed using an FDI fabricated meter/mix/injection RTM pump consisting of dual metering pumps with 10-130 cc/min flow capacity at 30" Hg and 85 psi. Pressurization was supplied by shop air pressure to the tanks. Resin temperature was maintained by heating the resin tanks and the resin hose to the static mix head.

2.1.2.4 Tooling

An above average CATIA user generated a solid model of the part in 40-50 hours. Engineering resources and a structural analysis were not expended to create a production-quality composite design. This study was only a proof of concept effort to verify that a composite structure with blind cavities could be fabricated.

Prior to the start of tooling fabricating, a core sizing experiment was performed. A rectangular block of lost core material the approximate size of the largest internal bay was fabricated and covered with carbon fiber/epoxy skin, cured, and processed through the lost core removal stages. This experiment confirmed the lost core material's CTE and molding shrinkage factors were suitable for application on the internal bays and the removal process did not induce any processing defects or damage.

The core molds were sized based on the proposed cure cycle in the press. The coefficient of thermal expansion (CTE) of the core allows it to correctly expand and match the inner mold surfaces of the stabilizer at temperature during cure, thus generating the desired laminate thickness and fiber volume. With traditional lost core materials, if the cure cycle were to change it could require new tooling to ensure accurate fiber volume in the finished part. The CTE of the lost core formulation used in this experiment can be modified, allowing the injection and cure cycle temperatures to change without tooling modifications.

The lost core tooling for the internal bays was comprised of matched die tools machined from aluminum mold plate. The RTM tooling for the outer mold surfaces consisted of a two piece matched die mold with four removable mandrels all machined from vacuum cast aluminum mold plate.

2.1.2.5 Part Fabrication

Plies were kitted into preform layups of the spars, ribs, and skins. Spar plies were hand cut oversized and net trimmed after layup on a plywood preform assembly fixture. To aid in ply positioning, 3M Spray Adhesive 77 was used as a tackifier. All rib plies were hand cut oversized and net trimmed after layup on lost cores. All skin plies were pre-plied before cutting in order to retain shape and minimize fiber fraying.

After preform layup, all further layup and assembly was performed in the RTM stabilizer mold in the following sequence:

1. The lower skin was positioned in the mold and aligned to the forward edge of the cavity.
2. The forward spar preform was removed from the layup fixture, overall length was verified in the mold, and the preform was placed at the forward edge of the tool cavity.
3. The lower forward hat stiffener lost cores and plies were placed in the tool cavity.
4. The eight forward lost cores, carrying the forward rib plies, were placed in the tool cavity, starting with the inboard cores and working outboard. The hat stiffener lost cores and plies were aligned and interlocked with the internal bay lost cores as the internal bay lost cores were moved into place. The lost cores for all internal bays were placed with an allowance for thermal expansion by leaving a 0.060-inch gap between the rib plies of mating cores.
5. The center spar preform was removed from the layup fixture, overall length was verified in the mold, and the preform was placed against the aft surface of the forward lost cores.
6. The lower aft hat stiffener lost cores and plies were placed in the tool cavity.
7. The eight aft lost cores, carrying the aft rib plies, were placed in the tool cavity, starting with the inboard cores and working outboard as described above for the forward lost cores.
8. The aft spar preform was removed from the layup fixture, overall length was verified in the mold, and the preform was placed against the aft surface of the aft lost cores.
9. All upper hat stiffener lost cores and plies were placed in their respective positions interlocked in the upper surfaces of the internal bay lost cores.
10. The two split trailing edge mandrels were placed against the aft surface of the aft spar preform
11. The upper skin plies were positioned in the RTM tool and aligned to the forward edge of the forward spar.
12. Any stray fibers were trimmed and removed from the mold parting line surfaces and silicone seals.
13. The RTM tool was closed and clamping pressure maintained with the hydraulic press.

The RTM tool assembly was preheated to 130°F and was vacuum desiccated for 60 minutes prior to injection. The resin was preheated to 120°F and the injection rate was set at 77 mm per minute. Resin injection pressure was monitored throughout the injection cycle. The injection rate was adjusted during the cycle to maintain an injection pressure of 40-60 psi.

Resin was injected at butt line (BL) 0. Two vacuum ports were located at the outboard ends of the RTM tool and were connected to a vacuum trap with nylon tubing. After about 30 minutes of rein injection, resin with air bubbles would begin exiting at the vent ports. As the injection cycle progressed, the tool backpressure increased with resin viscosity, which required a reduction in the injection rate to keep the injection pressure below 60 psi. Resin injection was continued until no air bubbles were visible exiting the

RTM tool at the vent port locations. The typical cycle from start to end was 70-90 minutes long.

The stabilizer was cured 4 hours at 130°F in the RTM press, followed by a freestanding oven post cure for 4 hours at 275°F. The RTM tool was opened with the hydraulic press and the outboard mandrels were removed. The trailing edge mandrels and cured part assembly were lifted out of the RTM tool via lifting holes in the trailing edge mandrel with forklift assist.

Lightening holes common only to the inboard cavities in the upper skin were drilled and functioned as drain holes for those cavities. Drain holes in the forward spar, aft spar, and outboard ribs were drilled specifically for lost core material removal. Core removal was also accomplished through the lightening holes in each internal rib after they had been trimmed. The horizontal stabilizer was processed through FDI's proprietary three-step lost core removal process. All lost core material was removed from the internal cavities and recovered for molding subsequent cores. Following lost core removal, the part was trimmed and the lightening holes de-flashed by grinding.

2.1.2.6 Results

Three parts were fabricated in the study. During pre-heating of the first article, it was discovered that without a gap between cores, friction between the fabric and the mold surfaces prevented a full linear spanwise expansion of the cores, and the rib fabric was not pushed tight against the outboard mandrels. Also during cure of the first article, the tool temperature was increased to 140°F for the cure cycle. The resulting thermal expansion of the lost cores coupled with the resin injection pressures generated enough force to open the hydraulic press 0.020 – 0.030-inch. On subsequent fabrication cycles, the cure cycle was maintained at 130°F.

The first article took 181 hours to fabricate; the second, 135; and the third, 110. It should be noted that the parts were fabricated by hand. There were no efforts to create the tooling or use the techniques normally associated with a production basis. Tooling required approximately 1000 hours to fabricate, including part solid model time, tooling concept design, tooling modeling and programming, tooling machining time, and tooling finish and assembly.

The parts weighed 20.1 – 20.6 pounds. Scaling the composite structure to full span and including the leading edge, a comparison against the equivalent steel metal structure yields a weight savings of approximately 27%. This weight savings has the potential to be increased 5 - 10% with structural efficiency – reducing the number of ribs, and tailoring the skin/spar cap thickness and stiffness.

Labor hours are estimated at 50 - 70 hours under a full production set up with 80% labor efficiency. When compared to a similar sheet metal structure, the labor hours average a total of 75 hours each.

At the time of the study, resin and hardener were $2.25 per pound. Dry carbon fabric was $25 - 30 per pound. Cured part raw material at 55% fiber volume fraction was $17.50 per pound. Aluminum alloy 2024-T3 coil, in thicknesses of 0.20 – 0.40 inch is $3.39 – 3.79 per pound.

2.1.2.7 Recommendations for Future Study

1. Layup time could be significantly reduced if pattern templates and/or powered cutter were used for cutting ply shapes.
2. Investigate braided preforms for use on the internal bay lost cores to reduce layup labor and improve damage tolerance.
3. Modify the design on the hat stiffeners to allow the use of braided sleeving instead of plain weave fabric to reduce layup labor.
4. Use braided preforms for the skin plies. Make the skin plies common to the forward spar preforms allowing them to function as part of the spar web. This would allow continuous skin plies from the upper aft spar cap around the OML surface to the lower spar cap.
5. Use a press capable of processing all lost core molds at one time.
6. Increasing hydraulic press tonnage to react against at least two times the desired injection pressure, thereby ensuring maximum resin injection flow rates, could reduce resin injection time.
7. Flow rate development should be pushed as fast as possible to maximize press utilization.
8. Consider molding the aft spar flanges net in order to eliminate the routing operation.
9. For a production design, perform the trade off to increase skin stiffness to reduce the number of ribs. Consider the incorporation of a thin low-density foam core or syntactic foam core or additional plies on the skin surfaces. This would also help to minimize rib, spar, and hat stiffener mark-off.
10. Maintain an inert gas over the resin/hardener inside the pressure pot to reduce moisture absorption and increase pot life.
11. Inspection trade studies. This complex geometry poses many challenges to verify part integrity due to the blind cavities and multiple surfaces. Further NDI studies are required to determine the threshold of detectability of defects commonly found in composite part.
12. Static and fatigue testing of the articles.

2.2 Resin-Film Infusion (RFI)

Resin-Film Infusion (RFI), as discussed in this report, is similar to Resin-Transfer Molding (RTM) in that it uses dry preforms that are then injected with resin. However, RFI has only one hard tool surface; the other side is vacuum bagged. In addition, the resin is not pumped into the part under pressure – instead, vacuum applied elsewhere on the part is used to pull the resin.

The preform is placed in the mold, and then vacuum bagged. Several ports in the vacuum bag are to inject resin and several are to pull vacuum. The resin and/or mold are heated prior to and/or during the infusion process. The resin flow is started while vacuum is pulled. Vacuum and resin flow continues until the entire preform is wetted out. The resin sources are removed and the part is cured. Vacuum may or may not be applied during the cure cycle, depending on the characteristics of the resin system – gel time, cure temperature, etc. Part quality is very dependent on the design of the resin infusion / vacuum system.

There are many variations to this technique that seem to be developed by each part manufacturer to fill their individual needs. One commercially available system is Seeman Composite's SCRIMP system, which is discussed below in Section 2.2.1.4.

2.2.1 RFI Feasibility Study

A general aviation manufacturer performed a comparative evaluation between various RFI techniques – SCRIMP, the Martin method, Film Technologies' Quick Draw VARTM, and the company's own classic bleeder/breather method. All information in this section is taken from Lafeen (1996).

The following criteria were evaluated:

1. Feasibility
2. Relative cost
3. Availability of "special" materials
4. Producibility, when compared to wet layup baseline
5. Amount of waste material, compared to the baseline
6. End part quality
7. Fiber volume
8. Worker exposure to resins
9. Labor intensity
10. Learning curve

2.2.1.1 Classic Breather/Bleeder Method

The manufacturer did not have great success with their classic breather/bleeder method. They attempted to manufacture an 18-inch square panel from five layers of 7500 glass cloth. They used a single resin port in the center of the part with perimeter vacuum, but then varied the location of the bleeder and breather media and locations and resin system. Finally using a manifold to distribute the resin over the panel surface produced good resin dispersion with relatively few voids.

They proceeded to manufacture a flap skin using the same techniques, with the exception of heating the resin prior to infusion. Even with the use of an injection manifold and heated, very low viscosity resin, the injection times were too long to allow complete wet out of the fibers and also void removal before gel using the classic bleeder/breather technique. They were unable to apply the traditional RTM methodology of injecting resin through the length of the laminate to the flap with good results.

2.2.1.2 Martin Method

This method is very similar to the classic bleeder/breather method with the exception that once the resin was flowing into the laminate, workers would use small, flexible, plastic squeegees or spreaders to assist the resin through the part. It was difficult to obtain a quality part with this method; voids were still present. In addition, there was not a substantial labor and material savings over traditional wet layup.

The benefits of the Martin Method are that one can still obtain the good compaction with dry fibers before wetting out the fabric and that worker exposure to resins and vapors is quite low.

2.2.1.3 Film Technologies' Quick Draw

Film Technology Inc. in Texas has a product they call Quick Draw VARTM film. This is a bagging film that is permanently, specially, wrinkled to produce a path for volatiles and resin to migrate through.

The advertised benefits of this product and method are that the special shape of the film allows resin and volatiles to flow on top of the laminate and then through the laminate to speed up the whole infusion process. Another advertised benefit of the process is that since there are potentially no other items between the film and the laminate, ultraviolet light curing resin systems may be used with it.

Despite repeated attempts to contact the company, the general aviation manufacturer performing the study was unable to obtain a sample of the product for evaluation.

2.2.1.4 SCRIMP

SCRIMP is a trade name for the patented Seeman Composites Resin Infusion Molding Process. The process was evaluated during a tour of the Seeman facility in Gulfport, MS.

The SCRIMP process uses a similar schematic principle as described for the other RFI methods, but uses a proprietary "medium" that allows the resin to infuse the part very rapidly and also allows entrapped air and volatiles to escape during the injection process and even until the gel of the resin. Due to the proprietary nature of the process, it is not possible to discuss exactly how the process works or what this "medium" is.

During the factory visit, the general aviation manufacturer witnessed many different products being fabricated by the SCRIMP process. The size of the parts made by the SCRIMP process was not a factor. According to Mr. Seeman, one of the interesting features of the SCRIMP system is that one can inject a part that has a different number of plies in different areas, such as hard points, and also foam core sandwich components without difficulty. They also have the technology to start and stop the flow of resin during the injection at certain locations on the part if need be.

2.2.1.5 Lessons Learned

The high fiber volumes that can be obtained from RFI is most likely due to the fact that during the debulking process, only dry fabric is compacted, not fiber and resin. This alone is a substantial benefit over wet layup and prepregs. Another substantial benefit is that unlike prepregs that must be debulked many times during a large layup, bag side RFI has only one debulk.

With respect to the classic bleeder/breather method, after trying different methods of faster infusion and void removal, it was decided that this method would not be suitable for use. After visiting Seeman Composites to witness the SCRIMP process, the study's author decided the SCRIMP process effectively solved all of the problems with the system.

The Martin Method, which is basically a manually assisted variation of the classic bleeder/breather method, was quickly abandoned in favor of SCRIMP, because there was no substantial labor savings with the process. The study's author believed the Martin method made an impractical process more labor intensive.

After trying different methods of bag-sided RFI, it became evident that if a general aviation manufacturer were to get involved with the process, SCRIMP appears to be the best available. While there is no real earth-shaking technology going on with SCRIMP, it appears Mr. Seeman has spent a lot of time (and presumably money) figuring out how to do it and then patented it. According to Mr. Seeman, since the license fee includes hands-on training and help in solving the customer's individual need, it can become almost a "turn-key" operation. In the opinion of the study's author, the SCRIMP process has enough benefits over wet layup and even prepregs to be considered very seriously in any material and process evaluation for composite general aviation aircraft components.

3 ADVANCED MANUFACTURING METHODS

While the goal of this guide is to identify manufacturing methods that reduce cost, the "advanced technologies" identified here are discussed knowing they have equal or higher costs associated with them, but have other technical advantages.

This chapter discusses advanced manufacturing methods that have been studied by general aviation manufacturers. These methods are Hydroforming, Internal Pressure Molding, and Electron Beam Curing.

3.1 Hydroforming

Hydroforming in the automotive industry generally means forming tube with liquid pressure on the inside and hard molds on the outside. The automotive industry also recognizes hydroforming to mean forming metal sheet between a hard metal tool on one side and a flexible bladder with liquid pressure on the other. Within aviation, however, hydroforming also refers to thermoplastic sheets that are formed a hard metal tool on one side and a pressurized silicone rubber (or other solid elastomer) on the other side.

Hydroformed parts have the potential to require fewer labor hours than conventional machining of aluminum or hand layup of thermoset composites. The reduction of labor hours, and resultant reduction in costs, is the main advantage of hydroforming over other fabrication methods. In addition, a pressurized elastomer and metal mold produce more uniform pressure on the part than two metal molds, which results in a higher quality thermoplastic part.

3.1.1 Industry Hydroforming Study

All information in this section is taken from Petersen (1998a).

The dimensions of a composite rib are critical because a composite rib is usually bonded into the substructure. The bondline thickness needs to be controlled for stress considerations. The height of the rib is critical because the height of the rib and spar must be the same so the skin will fit flush. The dimensions of an aluminum rib in an aluminum substructure are not as critical because an aluminum rib is flexible and fastened with rivets.

Hydroformed thermoplastic ribs have the potential to require fewer labor hours than conventional machining of aluminum ribs. The labor hours to produce a rib for various methods are shown in Table 2

Table 2 Estimates of the Labor Hours Required to Fabricate a Rib

Method	Labor Hours
Conventional machining of aluminum rib	3
Hand layup of graphite/epoxy composite	3
Aluminum rib manufactured using high-speed machining after numerical control programming was done	1/3
Hydroformed thermoplastic composite	2/3
Hydroformed thermoplastic composite with automated tape laying machine and processes optimized for production	1/4

The laminates for hydroforming were all made at Automated Dynamics Corp. The ribs were made at both the aircraft manufacturer and Automated Dynamics Corp. Three methods were used to consolidate the thermoplastic before hydroforming to try and reduce the cost of thermoplastic consolidation. The methods are: (1) fiber placement with a tape laying machine, (2) tacking loose plies with a hot iron, and (3) autoclave.

A thermoplastic laminate was heated until it was soft and began to sag. The mold was heated to a temperature below the softening point of the thermoplastic. The laminate was placed between the two mold halves, which were pressed together with a large force to form the material into a rib. The material was also cooled in the mold.

Ribs manufactured at the aircraft manufacturer were made of APC(PPS)/AS4 a thermoplastic graphite composite (polyphenylene sulfide with a graphite fiber) from ICI Fiberite in Orange, CA. APC(PPS) is a semi-crystalline thermoplastic resin that has excellent corrosion and solvent resistance; excellent flame, smoke, and toxicity performance; very low moisture absorption; and a use temperature close to the glass transition temperature. The physical and mechanical property data for APC(PPS)/AS4 composite is shown in Table 3

Table 3 Properties of Fiberite APC(PPS)/AS4 Composite

Physical Property Data

Density (73°F) (kg/m^3)	1400	Service Temperature (°F)	150
Glass Transition Temperature (°F)	185	Processing Temperature (°F)	600

Mechanical Property Data

Laminate Property (RTD)	Sample Orientation	APC/AS4
Tensile Strength (ksi)	0°	250
Tensile Modulus (Msi)	0°	17
Compression Strength (ksi)	0°	160
Flexural Strength (ksi)	0°	210
	90°	13
Flexural Modulus (Msi)	0°	17

The rib preforms, for ribs fabricated at the aircraft manufacturer, were manufactured as approximately 15-inch squares 5 plies thick. They were autoclaved with a maximum pressure of 100 psi and a maximum temperature of 635°F.

The preformed sheet was hung vertically on a fixture that was suspended from a track above the heater and press assembly. The laminate was heated until it started to droop when the fixture was shaken. The fixture was quickly slid down the track into position within the press molds. The female, metal side of the mold had steel cutting rings to make the lightening holes. The male, silicone rubber side of the mold had locating pins on the outside of it to assure correct line-up with the other mold half. The ribs were hydroformed with 70 pounds of force. The rib was allowed to cool in the mold for approximately one minute before removal.

Primary difficulties were the bunching of fibers at the corners of the ribs and at the lightening holes caused by the aluminum lightening hole inserts being unable to punch holes through the graphite fibers and by the rib cooling too rapidly in the mold. Steel cutting rings were placed around the circumference of the lightening hole inserts in the aluminum mold. Also, the aluminum mold was heated to 250°F and held at that temperature for one hour prior to molding to reduce the defects.

For ribs manufactured at Automated Dynamics Corporation, the rib preforms were manufactured as eight plies thick using APC(PPS)/AS4 from Quadrax, which is no longer available. The ribs were made without the lightening holes in the web of the rib. Tooling costs for the rib were approximately $7000 in 1996 and consisted of a steel female mold half and a silicone rubber male mold half attached to an aluminum sub-structure. Three consolidation methods were tried: (1) fiber placement with a tape laying machine; (2) tacking loose plies with a hot iron; and (3) autoclave. For the fiber placement method, an automated tape laying machine was used. The thermoplastic tape was heated with a torch before it was laid on the laminate. The laminate was then cut to shape by hand. For the loose plies method, loose plies of unidirectional composite were tacked together with a soldering iron to form laminates. They were heated and hydroformed for the purpose of comparison to the other methods in both properties and cost. For the autoclave method, a small number of preforms were fabricated using hand layup. The preforms were consolidated in an autoclave and cut into shape using hand techniques. This is a proven method to produce the best possible consolidated part. The disadvantage of this method is that it is slow and expensive. It is used as a baseline for comparison with the other methods.

The laminates were all hydroformed in the same manner. The laminates were placed on top of the steel mold, which had been preheated within a predetermined range. The temperature was between 300 and 350°F for the most successful ribs. An infrared heating panel with 4.75 watts/in^2 was then positioned over the steel mold. A thermocouple was taped to the laminate. When the laminate was at the selected temperature, the elastomeric mold was pressed into the steel mold half, forming the composite rib between the two molds. A clamping force of 10 tons was used.

Common difficulties were wrinkles, or sticking to the mold, which was caused by too high of pre-set temperature. From external appearance, the autoclave ribs do not appear to have better quality than the other two methods. Dimensional tolerances were fairly tight. Height was closely controlled, but the angle of the flange bends varied from 90 degrees, which can probably be overcome by bending the flexible flange during bonding to the wing skin.

The time to fabricate and the weight of the material for fiber placed laminates is shown in Table 4

Table 4 Time and weight for fiber placed laminates

Fiber placement of 3' x 6' panel from which 20 ribs were cut	8 hours (24 minutes each piece)
Material used per rib, including scrap	0.41 lbs.
Cut laminates using a template and band saw	Approximately 5 minutes each
Heat and hydroform	6 minutes each

The time and weights in the table above could be improved by changing the methods. A one-inch wide prepreg tape was used in the fiber placement. Changing to a three-inch wide tape would cut the fabrication time by about half. Three laminates were cut with a band saw. Cutting the laminates with an automated system, such as a water jet, would decrease the time required and material loss. The time to fabricate and the material weight that would be required for improved methods is shown in Table 5Note that the improvements described would result in a large decrease in time and weight. These are values that could be achieved in a production environment that has been optimized.

Table 5 Predicted time and weight for optimized fiber placed laminates

Fiber place laminates with 3-inch wide tape	Approximately 4 hours (8 minutes each piece)
Material used per rib, including scrap	0.27 lbs.
Water jet cut 30 pieces from panel	Approximately 3 minutes each
Heat and hydroform with a higher wattage IR panel	4 minutes each

Another method to fabricate the rib used loose ply laminates. The loose, individual plies were cut from unidirectional tape. They were stacked into eight-ply laminates which were tacked together at several spots using a hot soldering iron and then placed into the press, heated, and hydroformed. The results from these hydroforming experiments were more variable than those obtained from the fiber placed laminates both in the positioning of the laminate on the mold and splitting apart of the plies during hydroforming. Table 6 shows the time to fabricate and the weight of the material used for loose ply laminates. The time and weight for the loose ply method was approximately the same as for the fiber placed laminate method.

Table 6 Time and weight for loose ply laminates

Cut shaped laminate from unidirectional tape	Approximately 25 minutes each
Form laminate stacks and spot weld together	Approximately 4 minutes each
Material used per rib	0.39 pounds.
Heat and hydroform	Approximately 7 minutes each

3.1.2 Future Work

There are still some problems with the processes used here. The fibers bunched around the edges of the ribs, which can be reduced by heated mold halves. The fibers were pulled around the lightening holes. Cleaner cutting around the lightening holes may be obtained with the addition of metal cutting surface underneath the silicone rubber mold to push against the cutting ring. More experimentation is needed to perfect the method.

A device to hold the laminate in tension during hydroforming would help solve the problems of fiber bunching, pulling and wrinkling. This method should also be tried with fiberglass thermoplastic. The time required to produce a rib was approximately 35 minutes for both the fiber placement method and the loose ply method. The time to produce a rib using fiber placement method could be reduced further with a method optimized for production and should be tried.

Further work is also necessary to find a thermoplastic material with a higher Tg. Even though the Tg is relatively high, it is still below the FAA guideline for structural composites for most general aviation aircraft.

3.2 Internal Pressure Molding

All information in the following sub-section is taken from Petersen (1998d).

3.2.1 General

In internal pressure molding (IPM), prepreg is placed in a mold along with an internal bladder that forms the hollow portion. The mold is closed and the part cured. IPM produces net shaped parts, which means no machining is required. IPM has some advantages over conventional hand layup of prepreg. The primary advantage is that complex, net shape hollow structures can be fabricated without any secondary bonding. Additionally, inserts and special reinforcements can be co-cured in the part. Since IPM can be used with any prepreg system, a material that is already certified can be used, thus eliminating additional material certification. The disadvantage of IPM is that the part must be hollow and be formed with an internal bladder.

IPM has some advantages over resin transfer molding (RTM). For the RTM method, fiber with no resin is placed in the mold. The mold is closed and resin is pumped into the mold. Both methods can produce net shape parts. IPM with bladder molding can have more complex hollow inner structure because is uses a flexible bladder. At the present time, RTM has only been done with solid mandrels. Unless multipiece mandrels are used, the solid mandrels used in RTM must have constant tapers which allow them to be removed from the part after cure.

IPM can produce complex hollow composite parts. A hard tool is used to produce the outer mold line and a removable bladder is used to produce the inner mold line. Prepreg is laid up in the bottom mold half according to the laminate description. The bladder is inserted in the hollow portion of the prepreg and the bladder fitting is fed through the outlet in the tool. The tool is closed and placed in the press. The tool is heated while the bladder is inflated to the specified pressure. When the part is cured, it is removed from the tool. The bladder is deflated and removed through a hole in the part, which can be as small as 0.25-inch diameter.

The IPM method will tolerate variations in the amount of resin and the weight per area of the fabric up to about 5%. The results of variations in resin and fabric are similar to conventional bag molding. The internal bladder applies a constant pressure on each part. For parts with low resin content, the wall will be thinner than normal. For parts with high resin content, the wall will be thicker than normal.

3.2.2 Industry IPM Study

The goal of this project was to demonstrate that internal pressure molding could be used to manufacture high quality, low-cost composite parts for general aviation aircraft. The project involved making two components – a quarter window frame and a spoiler subelement. The purpose of the window frame is to close out the honeycomb sandwich fuselage shell and provide support for the window. Various designs and manufacturing concepts using compression molding of sheet-molded compound were considered for the project. IPM was selected for several reasons. IPM parts can be fabricated using continuous fiber prepreg, which has several advantages over sheet molding. Material allowables were available, continuous fibers were stronger than the sheet molding compound with short fibers, and the extra strength was needed in the window frame. The tooling costs for compression molding were also higher than for IPM, because IPM only requires outer mold line tooling. Compression molding has tooling for both inner and outer mold lines. Plain weave along with some unidirectional prepreg was placed in the aluminum tooling. The part was cured in a pneumatic press with heated platens. The pressure was over 100 psi and the cure temperature was 250°F. A single cure was used.

A spoiler subelement was also selected to demonstrate IPM on a part with an integral, composite attachment lug. The subelement was about 11 inches long and contains one lug. The actual spoiler was approximately 60 inches long, has three attachment lugs, and a control actuator attachment point. The spoiler had one spar and no internal ribs. The tool produced a hard outer mold line. The rear hollow section was made with a solid aluminum mandrel, while a bladder was used in the front hollow section. Both plain weave and unidirectional tape prepreg were used in the part. The ply layup in the attachment lug consisted of uni wrap and titanium straps that were added to form a load path between the attachment lug and the spoiler. This part demonstrated that IPM could be used for composite parts with metal inserts. After the part was cured, a hole was drilled and a steel bushing was inserted.

Six quarter-window frames were fabricated showing IPM could be used for low cost manufacturing. The time to cure and remove the part from the tool is less than one hour. The IPM method required much fewer hours than hand layup, as shown in Table 7The weight of the window frame is also less than the window frame made by hand layup.

Table 7 Comparison of Hand Layup to IPM

	Hand Layup	IPM
Labor to layup prepreg (hours)	50	Est. 10
Weight	1	0.92

The spoiler had excellent aerodynamic surface finish. The void content was low because both internal pressure and external force on the mold were applied during curing. The cost of a spoiler made by IPM is much less than the cost of a spoiler made by aluminum casting. The weight of the spoiler made by IPM is less than the weight of a spoiler made by aluminum casting. The data in Table 8is for a complete, approximately 60-inch long spoiler, with three integral attachment lugs and a control actuator attachment point. The cost to produce an 11-inch spoiler subelement was used to estimate the cost of a 60-inch spoiler. The data is normalized for cast aluminum.

Table 8 Comparison of Aluminum Casting to IPM

	Aluminum Casting	IPM
Relative Cost	1	0.40
Weight	1	0.68

3.2.3 Future Work

It would be useful to fabricate a complete, composite spoiler by another method for comparison with IPM.

3.3 Electron Beam Curing

Electron beam (E-beam) curing is in the very developmental stage when compared to the other two methods discussed in this chapter.

Electron beam curing is being investigated as a way to reduce the cost of composite fabrication and increase the ease of building complex shapes. Conventional thermoset composite materials are cured with heat and pressure. E-beam composites are cured by high-energy electrons and/or x-rays in a very fast, non-thermal, non-autoclave cure process. (Oak Ridge National Laboratories 2000) E-beam curing composites have the potential to be manufactured with lower cost than conventional composites and with similar material properties.

E-beam curing offers many advantages for producing high quality, low-cost parts for general aviation aircraft. The manufacturing process is faster because e-beam curing requires minutes rather than hours required for conventional curing. The parts are stronger and have more consistent dimensions because the curing is at low temperatures, 150°F, which produces no significant residual thermal stresses. Cheaper tooling materials such as foam or wood may be used because curing is at low temperatures. Also the tooling may last longer because there is no thermal cycling. Large complex parts can be cured as a unit eliminating the need for secondary bonding. (Petersen 1998e, 1)

E-beam curing uses high-energy electrons from a high-energy accelerator to initiate polymerization and cross linking in certain polymers. E-beam curable epoxies are conventional epoxies with small amounts of cationic initiators or free radicals added to initiate curing when electron beams are applied. The accelerator is similar to a high power x-ray machine. In a typical electron beam facility, the material is placed on a conveyor belt and moved at a constant rate of speed past an accelerator, which produces the electron beam. The beam is scanned back and forth across the part to provide uniform exposure. The accelerator is located in a vault with much shielding to protect people from radiation. The high cost of the vault is a barrier to the use of e-beam curing composites. (Petersen 1998e, 2-3)

There are still many potential advantages to electron beam curing. There is a 25-60% cost reduction compared to thermal processing for aerospace structures. There are reduced tooling costs and shorter cure times. In addition there are reduced costs related to compliance with environmental, safety, and health regulations. (Oak Ridge National Laboratory 2000)

The highest benefits are likely to come from increased design options. Electron beam curing provides uniform and efficient cross-linking without hot spots. Thick parts can be cured in one cycle, up to 12 inches thick. Low temperature materials can be integrated, like foam core, which could not survive an elevated temperature cure cycle. There are no part size limitations, and theoretically in-situ fabrication and curing is possible. (Oak Ridge National Laboratories, 2000) Significantly higher fiber volumes in filament wound composites can be achieved with e-beam curing than with oven curing. Deficiencies still exist in the interfacial properties, particularly interlaminar shear strength. (Janke 1999)

3.3.1 Manufacturer Study

One general aviation manufacturer performed a study to demonstrate that e-beam curing can be used to manufacture high quality, low cost parts for general aviation. Only the first phase of the study was completed, with limited results. Information in this section is taken from Petersen (1998e).

Flat panels and double lap shear coupons were made using electron beam curing composite material and very low-cost tooling. The quality of the specimens was marginal because of the tooling used. The wood caul sheets gave off water, which interfered with curing and caused the glass transition temperatures to be lower than expected. The double lap shear coupons needed tooling on the sides to prevent resin flow and sliding of the laminas. The void volumes were satisfactory. The electron beam curing material appears to be suitable for low-cost manufacturing and it is recommended that more test coupons be made and tested.

3.3.2 Conclusions

While electron beam curing can reduce the cost of tooling, care must still be taken to select proper materials, as demonstrated in the industry study. Electron beam curing of composites does seem to offer a viable alternative for oven cure thermoset materials in aviation applications. However, it remains to be seen if a general aviation manufacturer would be willing to make the up-front cost commitment for an electron beam vault to support such a project. While an electron beam curing facility such as a university or government facility could be used to develop material databases, eventually a true manufacturing facility would be required to show equivalence, as described in Section 1.6, and make full-scale parts for testing to show their compliance with all applicable federal aviation regulations.

If a material supplier can provide an FAA-approved material database, then an end-user can better weigh the pros and cons of electron beam curing vs. thermal curing. In addition, resin manufacturers will need to improve their interlaminar shear properties.

4 LOW-COST NON-DESTRUCTIVE INSPECTION TECHNIQUES

4.1 Introduction

Currently most general aviation manufacturers use visual, coin-tap, or ultrasonic inspection methods, or some combination thereof, for inspecting composite parts and assemblies. While these techniques can be quite effective in locating disbonds, they have some potential drawbacks. All require extensive training and significant expertise for accurate interpretation of the results. Additional ultrasonic inspection equipment can be expensive to purchase and is typically slow for large area measurements.

NASA, together with an experimental aircraft manufacturer, investigated two low-cost techniques that can meet the challenges of the inspection of these materials for differing types of defects. (Cramer, Gravinsky and Semanskee 1998) The first technique is a low-cost thermal method that utilized thermochromic liquid crystal sheet to sense temperature changes induced by actively heating the material being inspected. The second technique is called shadow moiré. This technique uses optical methods to detect small displacements in the material surface due to subsurface delaminations. Both methods allow large areas to be inspected quickly. Additionally, the thermal method also provides size and shape information of the defects.

The investigation focused on inspecting and identifying two types of defects – debonds in secondary bonds and debonds between core and skins. All information in the following sections is taken from Cramer, Gravinsky, and Semanskee (1998).

4.2 Thermochromic Liquid Crystal (TLC) Sheets

Infrared (IR) thermography has been used extensively for the detection of defects in bonded structures. Typical systems employ an IR radiometer (or an IR camera) that measures the thermal radiation and converts this to a measure of the surface temperature. Application of this technology for nondestructive evaluation (NDE) usually consists of actively heating the material's surface a few degrees centigrade. The responses of the material to the external application of heat can be recorded and used to calculate variations in material properties, which can be indicative of defects in the material. A disadvantage of this technology for application to general aviation materials is the cost, which can be $100,000 or more for a complete system.

One cost-effective thermal NDE technique is thermochromic liquid crystal sheets. TLC sheets are optically active mixtures of organic chemicals that react to changes in temperature by changing color. TLC sheets can be obtained commercially in a number of different forms. This study used coated sheets, which are available commercially and consist of a thin film of liquid crystals sandwiched between a transparent polymer substrate and a black absorbing background. The TLC sheets cost approximately $25 for a 12-inch square reusable sheet. After application of the sheet to the surface being inspected, a small amount of heat is injected through the sheet, such as with a 500-Watt quartz lamp, causing a temperature rise in both the sheet and the structure. The heat source is removed, and the sheet inspected during the cooling process. If a delamination is present, a temperature gradient will develop and be evident by non-uniform color changes in the TLC sheet.

For TLC to work as an inspection method, good contact must be achieved between the substrate and the sheet. Self-adhesive sheets are available, but lose their tackiness after four or five applications. Sheets without self-adhesive can be applied to the surface with standard ultrasonic gel. Even distribution of the gel is important and clean up necessary afterward. A vacuum bag system can also be used to provide good thermal contact between the sheet and the substrate. It may even be possible to implement a system where the TLC sheet is built into the vacuum bag used in the initial curing of the parts and thus allow an initial quality control inspection to be performed during the manufacturing process. While this potential exists, it was not investigated during the study.

It is important to select a TLC sheet with the correct ambient temperature range for the environment it will be used in. The sheet's full range of temperature is on the order of 5°C from ambient. If ambient temperature is too high or too low, the sheets will not be useful.

The TLC sheets were successfully tested to find debonds in secondary bonds between two four-ply glass skins. Debonds were clearly identified as color changes in the sheet, which could be recorded with conventional photos, video, or digital cameras and archived for later reference.

4.3 Shadow Moiré Instrument

Optical interference techniques such as shearography and holography have been used extensively for nondestructive evaluation. Shearography is sensitive to derivatives of the out-of-plane displacement of a body under load, while other full-field methods such as holography typically contour the surface displacement directly. Both of these techniques require some external load be applied to the part under inspection. The external load can be applied in any number of ways such as heating, vibration, pressurization, or mechanical loading. These techniques are typically quite expensive with commercial systems easily costing $100,000 or more, which has been a limiting factor for application by general aviation manufacturers.

Another optical technique, which has proved to be both low-cost and effective for the detection of disbonds between glass skins and foam core material, is the shadow moiré method. This method uses optical techniques to detect small displacements in a surface that is not experiencing loading. In the case of glass skins bonded to foam core, these displacements are due to disbonds that occur between the skin and the core. In the shadow moiré method, low-frequency patterns can be observed when light leaving a source passes through an optical grating. The light is then reflected from the surface of the test article, back through the grating, and viewed at from a different direction. Since two rays of light leaving the source can have different path lengths on reflecting due to the presence of the grating, a perceived interference pattern occurs.

If the surface is flat and parallel to the grating, the pattern will be a regular spacing of fringes. On the other hand, if surface deformation is present, there will be an additional localized path length change resulting in changes in the localized fringe density. In severe deformation cases, the fringes tend to form rings around the deformation.

Because the shadow moiré method requires that the surface being inspected be a diffuse reflector, depending on the surface coating, it may be necessary to lightly coat the surface with something such as talcum powder. During the study, it was not necessary to perform any additional surface coating other than the gel coat already present on the parts.

This method was successfully used in the study to identify core/skin debonds. The method was inexpensive; using laser printed gratings onto transparency film and a home video camera light with an appropriate slit attached for the light source. Visual inspection of the resulting fringe pattern is possible and recording, if necessary, can be done by any optical means desired.

5 AUTOMOTIVE TECHNOLOGY TRANSFER

5.1 Introduction

The premise of this section is that some current and near-future automotive design, materials, and manufacturing technologies can be identified and beneficially adapted to future general aviation airplane production.

Information transfer from the automotive industry to the aviation industry can be divided into two broad categories – manufacturing technology and manufacturing philosophy. Both will be discussed in this chapter. There is also a section that summarizes what AGATE members noted during a tour of a low production run automotive factory and a section on "Final Thoughts."

Manufacturing Technology covers many products including materials, processes, electronics, and crashworthiness, among others. The AGATE Low-Cost Manufacturing Group is principally interested in manufacturing materials and processes, which is what that section will primarily focus on.

Manufacturing philosophy will be divided into two sections to discuss the Toyota Production System and to summarize the advice given by an automotive consultant during AGATE Low-Cost Manufacturing Meetings in Cincinnati, Ohio in October 1999 and in Detroit, Michigan in March 2000.

5.2 Manufacturing Technology

There are two ways to integrate automotive technology into airplane design and production. One method is to find parts that are made for cars and use them on airplanes. This takes advantage of large volume production and low parts costs, but may create problems with availability later on and may create difficulties with certification. This method is widely done in Europe. The second method would be to use the automotive suppliers and their manufacturing processes and equipment to design aircraft parts, but it may be difficult to convince them to work with the low volume numbers.

It is well known that automobile production far exceeds general aviation production, but perhaps not as well know is just how disparate the numbers are. For example, in 1998 for every general aviation private airplane produced, there were 30,000 private motor vehicles produced and sold. This disparity suggests that, with production quantity being a significant factor in production cost, there are likely to be meaningful barriers to extracting and adapting automotive design and production technologies to broad and relevant general aviation usage. (Cirrus 1999, 11-12)

Automakers have demonstrated their willingness to pay a premium for lightweight solutions that enable them to meet automotive weight requirements. Automakers reveal that they routinely pay up to twice as much for alternative lightweight closures. (ULSAC 1999) The reason for this is that a 10% weight reduction in an automotive body equals a

6% increase in fuel economy and a 5-7% weight reduction in the power train. Glass fiber composites have the potential to reduce body weight by 20-30%; carbon fiber composites by 40-60%. (Warren 1999)

Despite this, conventional automotive construction is still a welded steel frame with stamped steel body panels, and non-structural plastic interior panels. While several all-aluminum cars exist, they are still considered innovative. Ultra-light steel structures are also considered somewhat novel, as are some of the new processing methods such as sandwich construction and hydroforming. Structural composites are almost unheard-of for mass-produced vehicles, although the race car industry has long used composite frames and body panels due to their superior strength-to-weight ratio.

Automotive materials can be grouped into four categories: steel; aluminum; non-structural plastics and composites; and structural plastics and composites. Their raw materials and process options will each be explored. Other products and technologies will also be briefly discussed.

5.2.1 Steels

Steel has been, and continues to be, the basic material used in automobiles. Over the years processes have evolved to reduce the cost of fabricating steel. These processes have been based on large-scale production that involves very expensive tooling for forming steel panels. Complex and large dies capable of handling large steel body panels are inherently expensive and require high volume production conditions to be competitive. (Cirrus 1999, 25)

Thirty-five sheet steel producers have joined together for designing lightweight steel autobody structure and auto closures. They formed two consortiums, The UltraLight Steel Auto Body (ULSAB) and UltraLight Steel Auto Closure (ULSAC). The consortiums have commissioned studies to assist their automotive customers with viable weight reduction solutions that maintain performance and affordability. The ULSAB study focused on reducing weight of the automotive body structure. The ULSAC study investigated doors, hoods, decklids and hatchbacks.

Both studies used the same materials and processes to make their final design recommendations.

1. High Strength Steels (yield strength of 210-550 MPa)
2. Ultra-High Strength Steels (yield strength greater than 550 MPa)
3. Tailor Blanks
4. Hydroforming
5. Laser Welding
6. Steel Sandwich Materials

5.2.1.1 High Strength and Ultra-High Strength Steels

Ultra-light high-strength steel can reduce car weight by up to 36%, but high-strength steel is harder to form and exhibits varied moduli. (Technology Century 1999, 26) When using very thin high strength and ultra high-strength steels, several common automotive design techniques are required to minimize dent resistance and oil canning. Feature lines were used on outer panels to stiffen unsupported areas. Inner panel structures were designed to provide good support to outer panels. Sheet metal hydroforming was used to increase effective outer panel dent resistance through work hardening. (ULSAC 1999)

In addition, door inner panels have non-linear weld lines in the tailored blank to save mass and improve formability. Laser welding is used to attach parts to hydroformed frame. Steel sandwich structure was often proposed to reach mass targets. However, since steel sandwich material is not yet widely used in large production quantities and is more costly, 0.6-mm sheet steel can be used for inner hood panels. This alternative demonstrates performance results that are similar to the sandwich material at reduced cost, but with a slight sacrifice to mass savings. Adhesive bonding is used to hem flanges for structural performance on grilles. Lift gate design was a hydroformed tube hatch, laser welded to a conventional inner panel. A sheet hydroformed outer panel is then hemmed to the inner panel. Tube hydroforming and assembly process lends inherent integrity, while sheet hydroforming provides excellent dent resistance at a thinner gauge. Sheet hydroformed outer panels hemmed to stamped tailored blank inner panel is lower-cost alternative. Another alternative is a complete hydroformed frame, eliminating the necessity of a full inner panel. A ring provides intrinsic structural integrity while saving mass. The last alternative is continuous laser welded joined inner and outer panels providing a fluid pressure seal for the hydroforming process. In this process, internal forming pressure expands the parts into the molds. The sheet hydroforming process contributes further to local panel stiffness through work hardening of the inner and outer panels. To contribute to torsional stiffness, the glass is bonded to the frame using urethane. (ULSAC 1999)

5.2.1.2 Tailor Blanks

"Tailor blanking" is welding together steel sheets of various thicknesses to make one non-uniform sheet. The new sheet is then formed like conventional, uniform thickness sheet stock through stamping, additional welding, etc. Tailor blanks reduce the total number of parts, which results in a reduction in the number of dies for part stamping and therefore reduces total cost. It reduces spot welds and increases dimensional accuracy due to a reduction in assembly steps.

5.2.1.3 Hydroforming

Hydroforming can be performed on tubular or sheet steel.

Tubular hydroforming provides the opportunity to make a complex structural component from a single tube, instead of several stamped sheet components. Hydroformed tubular structures are manufactured by first welding a tube which is then bent and pre-formed. The pre-formed tube is then hydroformed into the final shape. (ULSAB 1998)

Sheet hydroforming allows components to be manufactured from thinner gauge steel by achieving a work-hardened effect, especially in the center of the sheet where the degree of stretch is normally minimal. Using the thinner sheet stock reduces final component weight. Typically, additional thickness is required to meet dent resistance requirements.

With sheet hydroforming, the work hardening effect is achieved by using fluid pressure to stretch the blanks towards the punch. In the second step, the punch forms the panel back towards controlled fluid pressure. These parts have excellent quality because there is no metal-to-metal contact on the outer part surface. (ULSAB 1998)

The hydroforming process employed by the automotive industry should be explored by the aerospace industry for a number of applications. The process is employed by a number of companies to manufacture radiator frames, rear axle casings, engine cradles, seat frames, and space frames. (Xia 2000) This process could be used to fabricate the crashworthy engine mount for general aviation aircraft.

5.2.1.4 Laser Welding

Laser welding can be used to improve static and dynamic strength of joints, in areas where access is available on only one side, and for good aesthetic appearance at joint areas. Laser welding also has the benefit of a small heat-affected zone, which reduces dimensional distortion and property changes. (ULSAB 1998)

5.2.1.5 Steel Sandwich Structure

Steel sandwich structure is composed of a thermoplastic (propylene) core between steel skins. It is used where bending stiffness is the chief design criterion. It can be deep drawn, shear cut, laser cut, drilled, adhesively bonded, and riveted, but can not be welded. (ULSAB 1998)

5.2.2 Aluminum

The automotive world enjoys a tremendous cost advantage by employing stamped steel space frames with attached steel metal panels or by using monocoque steel construction. The small airplane must stay with semi-monocoque aluminum construction to preserve a higher stiffness per unit weight. (Cirrus 1999, 16)

The Audi A series car is constructed of an aluminum space frame that results in a 40% reduction in weight with equal or increased rigidity. Overall, aluminum substitution can be done at four times the current cost for steel. Any body panel or structural component made from aluminum can exceed the stiffness per unit weight of steel components, while for the same component size, the stiffness is about one-third that of steel. (Cirrus 1999, 18)

The trend to substitute steel sheets with aluminum sheets has been somewhat limited by problems associated with recycling. Aluminum is emerging as the material of choice and the search is on for manufacturing technologies that can produce good surface finish on large surfaces by using new forming technology. Advances have been made in material properties with high-strength aluminum, hardened aluminum, hot-formed aluminum, and sandwich aluminum. Along with this, advances have been made in deep drawing, hydroforming, welding (resistant, arc, and laser), bonding (adhesive, weld, and mechanical) and thin-wall vacuum casting. The cost of aluminum stock has continued to become more and more competitive. (Cirrus 1999, 25)

Aluminum is lighter and, if recycled, potentially cheaper. Automakers have been stepping up their use of aluminum, notably in engine blocks and cylinder heads. But aluminum sheet in body panels has been an on-again, off-again proposition. International Body Engineering Conference Chairman Bernard Robertson, calls aluminum "invariably more expensive" as a raw material, with higher fabrication and painting costs. (Technology Century 1999, 26) Several aluminum car frames were on display at the 2000 SAE World Exposition in Detroit; they consisted mainly of extruded profiles, stamped sheets, and cast parts. Where aluminum was used, it was mainly MIG-welded and riveted.

The most recent trends in aluminum, as witnessed at the SAE world exposition in Detroit in March of 2000, were vacuum-formed castings. These castings are made by heating a thin plastic film on a pattern (plug). A vacuum tightly draws the film over the pattern, which is then surrounded by a flask. Next, the flask is filled with dry, unbonded, extremely fine sand and vibrated so that the sand is tightly packed. After a second of sheet is placed on the flask, a vacuum draws out the air, and the completed mold is then stripped from the pattern. Each half of the mold is made in a similar fashion and then aluminum is poured directly from the furnace into the closed halves. The mold is held under vacuum to retain its shape. After the mold cools, the vacuum is released and the sand and completed castings fall free. (Harmony Castings 2000)

Vacuum casting can allow zero degree draft, thin walls and tighter tolerances than conventional sand castings. Table 9, taken from Harmony Castings (2000), compares different casting processes.

Table 9 Comparison of Aluminum Casting Processes

Process	1. V-Process Castings	2. Sand Castings
Description	Extremely fine sand is “vacuum packed” around pattern halves. The pattern is removed and metal poured into the cavity. The vacuum is released and the casting removed.	Treated sand is molded around a wood or metal pattern. The mold halves are opened and the pattern removed. Metal is poured into the cavity. The mold is broken and the casting removed.
Typical Size Range	Up to 150 lbs.	Ounces to tons
Tolerances	± .010" for the first 1", then add ± .002 inches/inch. Add ±.020" across parting line	± 1/32" to 6", then add ± .003 inches/inch. Add ±.020" to .090" across parting line
Surface Finish	125 – 150 RMS	200 – 550 RMS
Min. Draft Required	None	1 to 5 degrees
Min. Section Thickness	0.125”	0.25”
Typical Order Quantities	All	All
Typical Tooling costs	$3,000 to $14,000	$800 to $4,000
Nominal Lead Times	Samples: 2 to 6 weeks Production: 2 to 6 weeks after approval	Samples: 2 to 6 weeks Production: 2 to 6 weeks after approval

Process	3. Investment (Lost Wax)	4. Permanent Mold
Description	A metal mold makes wax replicas. These are joined and surrounded by an investment material. Wax is melted out and metal is poured into the cavity. The molds are broken and the casting removed.	Molten metal is poured into a steel mold. The mold is opened and the casting is ejected.
Typical Size Range	Ounces to 20 lbs.	Ounces to 100 lbs.
Tolerances	± .003"to 1/4" ± .004" to 1/2" ± .005" to 3", then add ± .003 inches/inch.	± .015" to 1", then add ± .002" inches/inch. Add ±.010" to .030 across parting line
Surface Finish	63 - 125 RMS	150 – 300 RMS
Min. Draft Required	None	2 to 5 degrees
Min. Section Thickness	0.060”	0.1875”
Typical Order Quantities	Under 1000	500+
Typical Tooling costs	$3,000 to $20,000	$5,000 to $25,000
Nominal Lead Times	Samples: 8 to 10 weeks Production: 5 to 12 weeks after approval	Samples: 8 to 20 weeks Production: 10 to 12 weeks after approval

Process	5. Plaster Mold	6. Die Casting
Description	Plaster slurry is poured into the pattern halves. After setting, the mold is removed from pattern, baked, assembled and metal poured into the cavity. The mold is broken and the casting removed.	Steel dies, sometimes water cooled, are filled with molten aluminum. The metal solidifies, the die is opened and the casting ejected.
Typical Size Range	Ounces to 50 lbs.	Ounces to 15 lbs.
Tolerances	± .005" to 2", then add ± .002 inches/inch. Add ±.010" across parting line	± .002 inches/inch. Add ± 015" across parting line
Surface Finish	63 – 125 RMS	32 – 63 RMS
Min. Draft Required	0.5 to 2 degrees	1 to 3 degrees
Min. Section Thickness	0.070”	0.030” to 0.060”
Typical Order Quantities	Prototypes up to 250 pieces	2500+
Typical Tooling costs	\$3,000 to \$15,000	\$10,000 to \$100,000
Nominal Lead Times	Samples: 2 to 10 weeks Production: 4 to 8 weeks after approval	Samples: 12 to 22 weeks Production: 8 to 14 weeks after approval

Source: Harmony Castings (2000)

Quick transfer of this technology to aircraft will probably by stymied by FAR 23.621 – “castings factors” – that puts an almost prohibitive inspection and safety factor on all castings. Without changing this rule, the largest advances in cost and weight savings cannot be realized by the general aviation industry. The latest revision regulations are made available on the FAA web site, www.faa.gov.

5.2.3 Plastics and Composites

The use of plastics in the automotive industry can be traced back to the industry’s infancy, mainly in such items as electrical components and interior fittings. The concept of actually designing vehicles around plastics came much later. Henry Ford began experimenting with composites around 1940, initially using compressed soybeans to produce plastic-like components. These days the increase in the use of plastics and composites can be linked to legislative actions, such as the Clean Air Act Amendment, which have resulted in increased emphasis on electric, hybrid, and alternative-fuel vehicles, as well as improved fuel economy of conventional vehicles. According to the Oak Ridge National Labs, 75% of a vehicle’s energy consumption is directly related to factors associated with vehicle weight, and it identifies as critical the need to produce safe and cost-effective lightweight vehicles. (Borge 2000, 72)

Substituting a material with another is often a “copy and paste” of the previous solution. In the past, some thermoplastic body panels failed or were not as successful as expected because the characteristics of the thermoplastics were not taken into consideration initially. Recent successes owe to the fact that the unique characteristics of the new materials were considered during the early stages of development. (Borge 2000, 74)

The choice of plastics is often a compromise between design goals and technical feasibility from the manufacturing point of view. Two key factors specific to the automotive industry have driven the innovation in plastics industry. One is sheer volume of material needed even to replace a single part in over 10 million automobiles that quickly adds up to large volume. Secondly, the constant model changes make it possible to replace old materials by new materials through improved product and process. (Cirrus 1999, 32)

5.2.3.1 Non-Structural

The considerations behind the use of plastics are many – such as end-use temperature, environment, flammability, life cycle, part strength and consolidation, volume usage, and well as government regulation. Two fabrication methods are potentially suitable for general aviation aircraft use – vacuum forming and compression molding for large panel parts. In addition, two chemical families of plastic polymers are potentially suitable for general aviation aircraft use – amorphous polymers and crystalline polymers. Amorphous polymers are materials of random molecular structure, gradual melt point, and low chemical resistance. Examples include polystyrene, ABS, and polycarbonate. Crystalline polymers are materials with an organized molecular structure, sharp melt point, and high chemical resistance. Examples include HDPE, PP, and Nylon. (Cirrus 2000, 32-33)

For parts with complex shapes, molding techniques are most suitable. Molding reduces secondary operations, but requires higher initial investment in tooling. For large volume production, the production cost is comparable to metal forming processes. Vacuum forming is a simple, low-cost fabrication process for very lightweight materials with complex shapes. The process consists of taking a sheet of material, heating it to where the material is soft, and then placing it over a mold. A vacuum is applied to the mold and atmospheric pressure forces the plastic into the mold. The part is cooled and removed from the mold and retains the shape of the mold. (Cirrus 1999, 33)

A two-stage molding process is commonly used to manufacture radiator shrouds and seals. In this process, the harder part is molded first, which is then transferred to a second set of dies that forms the added softer component. The transfer is automated in high-volume applications. This process may be appropriate for manufacturing the baffles in the cooling system of air-cooled aircraft engines.

The 2001 Explorer Sport Trac is Ford’s first truck to include an SMC composite box. It is a vinyl ester with random glass fibers that produces a part 20% lighter than the traditional steel pickup box. A SMC tri-door closure system on the Ford Excursion is 15% lighter than a comparable system made from sheet metal. Tooling costs for the composite system were approximately 75% less than the costs associated with steel doors. (Borge 2000, 74)

A look inside the passenger compartment of virtually any vehicle shows the dominance of plastics. According to experts, the passenger compartment accounts for 56% of the total usage of automotive plastics. PVC was once the almost universal surface in car interiors. It has largely been displaced, including GM's announcement of its intention to replace PVC on all new vehicle interior panels by 2004. TPO is a common replacement. TPO is a material made by combining rubber with PP. It is the lightest and lowest-cost form of plastic. Rubberized PP is classified as TPO when the material contains at least 20% rubber. Delphi offers deep draw capability with TPO, meaning that the material can be vacuum-formed to meet complex shape and contour demands. (Borge 2000, 78)

The new active air-intake manifold for Rover is made of a glass-reinforced nylon from DuPont. Switching from aluminum to nylon reduced weight by 40% and costs by approximately 30%. The manifold offers reduced complexity while delivering significantly more torque than the passive aluminum manifold it replaces, incorporating patented overmolding process using dissimilar materials to create all-plastic flap valves. The flap valves, the valve's frame, shaft, and flap are all formed successively in the mold and joined during the injection-molding process, made possible by using two different types of nylon for the valve's frame and flap. Once molded, the valve assemblies are then ultrasonically welded to the manifold, eliminating any assembly steps. The intake manifold is made using the lost-core process, while the plenum cover and the manifold are joined using vibration or shell welding. (Borge 2000, 81)

New roof module technology is under development that is produced through a process where a polyurethane composite is layered between the vehicle's outer roof skin and its interior headliner. The roof exterior – which can be constructed of steel, aluminum, or plastic – is pre-formed and painted. Polyurethane foam is injected between the exterior skin and the interior fabric. During the process, aluminum or steel coils are pre-painted to match the vehicle color, then formed using a deep draw process that allows the material to flow so that the material does not stretch, and in turn the paint is not broken or marred. During the application of the polyurethane foam, resin and fiberglass combinations can be varied throughout the roof, enabling softer areas above passengers and stiffer locations for mounting handles and light housings. (Borge 2000, 87)

5.2.3.2 Structural

Carbon fiber is not often used in the automotive industry because of its high cost, though one recent use is in Ford's 2000 SVT Mustang Cobra R. Air inlets designed into the Cobra R's fog light bezels are used to provide extra cooling for the front brakes. Air ducts run from these inlets to special carbon-fiber heat shields around the inside of the brakes to intercool the rotors. The heat shields were developed by Multimatic Motorsports and were used by 1999 Cobras in the Motorola Cup racing series. The main automotive application for carbon fiber continues to be for moving parts in the engine and transmission. Aramid fibers are used in moving parts where lubricity and dimension consistency are more important than strength or rigidity, such as clutch belts and greasefree ignition switches. The Ford Equator concept truck displayed at the North American International Auto Show in January 2000 featured Kevlar bumpers, fenders, wheel wells, and lower trim panels, making the parts resistant to stone damage. (Borge 2000, 71-72)

The Department of Energy (DOE) is funding a $1.8 million project at Virginia Tech and Clemson University aimed at the development of low-cost carbon fiber for use in making lightweight automotive parts. The approach is to develop a new polymer to serve as a precursor for the carbon fiber. Currently, carbon fiber suitable for automotive use costs around $8 per pound. The research team hopes to develop a carbon fiber that can be produced for less than $5 per pound. (Borge 2000, 72)

Bayer has developed hybrid technology that links metal and plastic by combining them into one component. An example is the 2000 Ford Focus which has a plastic / metal composite grill opening reinforcement (a structural body component) that consists of two metal stampings of 220 bake-hardened mild steel with a nominal thickness of 0.5 mm and heat-stabilized polyamide 6 resin with a nominal thickness of 2.5 mm. The metal stampings are placed in a mold and resin flows into and around them, mechanically locking to the metal and forming a single integrated unit. The component part maintains dimensional stability after being e-coated and painted. (Borge 2000, 74)

Structural injection molded components require glass-fiber reinforcement for stiffness purposes, but during the injection-molding process the fibers orient with the flow of the polymer, which can lead to anisotropic shrinkage during cooling and warpage of the part. Although jigs can be used after molding to achieve part flatness, this can lead to residual stress in the part, causing warping or early mechanical failure in service. Nylon 66 resin is a widely used base polymer for under-the-hood automotive components because of its balance of properties, such as good heat performance, resistance to oils and other chemicals, and toughness. However, it has relatively high shrinkage, leading to larger differences in the flow/cross-flow direction for simple glass-fiber reinforced materials. M.A. Hanna has overcome this problem by using both glass-fiber and mineral fillers. (Borge 2000, 80)

Hexcel Corp.'s prepreg (a familiar product in the composite aircraft industry) is used for composite leaf springs, weighing up to 60% less than their steel counterparts, while absorbing energy more readily than steel and providing a more comfortable ride. Composite leaf springs are non-corrosive and resistant to salt damage as well as oil, gas, and battery acid. They are molded into shape in a single operation where prepreg is taken from a roll, cut into shape, and placed into a mold tool. (Borge 2000, 80-81)

For an aircraft, "structure" is defined as the components that carry airloads – skins, spars, stringers, ribs, bulkheads, etc. All of these components are currently manufactured out of composite materials in various aircraft. Comparable "structure" in automobiles would be the chassis and body frame that alone carries impact and rollover loads. Skins on automobiles provide crash absorbing features, but are not load-carrying members. True automobile structure is not manufactured out of composite materials, except in "exotic" cars, such as racecars, where weight and stiffness are more important than cost. Other "structural" automotive applications include fuel tanks, which have burst pressure requirements, and leaf springs.

Another type of "exotic" car include affordable "world cars," engineered for economical manufacture and assembly in nations with emerging economies. Composites are used extensively in these vehicles, even for structural members such as fames and roof supports, eliminating the need for expensive metal stamping equipment – a plus for developing countries that cannot afford to invest in traditional manufacturing methods. Just starting to take hold in niche markets, these limited-production vehicles represent a lower-cost alternative to conventional cars and trucks. Alternative vehicles, engineered specifically for fiber reinforced plastic materials, contain fewer parts; they weigh less, offer better fuel economy, and are impervious to the ravages of corrosion. The world's first car to make extensive use of composites and plastics is the Paradigm, a four-door five-passenger sedan scheduled for production late 2000 in China. The car will have an FRP composite chassis with frame members made of pultruded fiberglass/vinyl ester epoxy and body formed from vacuum thermoformed non-reinforced thermoplastic body panels. The total cost of tooling for the Paradigm's chassis and body is about $300,000. The chassis is constructed of a center beam and two outside beams joined by five cross-car frame members. The pultruded composite rails are twice a strong as steel at a quarter of the weight. In the future, a thermoplastic resin matrix will be used instead of thermoset epoxy due to recyclability and the ability to use vibration welding to join members. In vibration welding, the interfacing parts are subjected to high-amplitude, low frequency energy, which produces heat. The polymer melts, welding the parts together and forms a strong joint in a matter of seconds. (Stewart 2000, 17)

In damp environments, composites are an ideal material to withstand rapid corrosion. Automotive Design & Composites has developed the Baja, a sport utility vehicle with a molded thermoset composite chassis and precolored thermoformed thermoplastic body panels. A steel subframe, which might later be replaced by a pultruded frame similar to the one on the Paradigm, supports the tub. The Baja's hood, bumper covers, dashboard, and removable half doors are vacuum formed of thermoplastic material. Thermoset components are used to mold other components, including a lower chassis tub and an upper roof structure. The composite parts are molded via a vacuum-assisted resin transfer molding process, using vinyl ester epoxy and glass. (Stewart 2000, 17)

Composite Automobile Research Ltd. has developed the WorldStar, a lightweight, inexpensive utility vehicle, targeted at developing countries. It is made with a glass-reinforced polyester structure on a tubular steel frame. The body is produced with open molds, using either sprayup and/or hand layup processing. The hand laid up sandwich structure core has a total of 25 open molds, which are used to manufacture the WorldStar in its various body styles. Between 100 and 200 hours of labor are required to build each WorldStar. Raw materials and parts to produce the car cost manufacturers about $4,000, and the vehicle is expected to sell for les than $7,000, with actual prices varying according to labor cost. (Stewart 2000, 18)

General Motors introduced its EVC1 in 1996. The two-passenger sports coupe weighs 2970 lbs., including its 26 lead-acid batteries. Doors, roof, hood, and trunk lid are compression-molded from sheet molding compound (SMC). The polyurethane fenders, quarter panels, rocker panels, wheel skirts and belly pan are reaction injection molded. The battery tray, a structural component molded from glass/polypropylene, was named the most innovative use of plastics in 1997 model cars by the Society of Automotive Engineers. GM continues its efforts to incorporate lightweight materials in vehicles, as evidenced by the Chevrolet Triax concept vehicle introduced in October at the 1999 Tokyo Motor Show. The Aerodynamic SUV is built on a steel chassis with separate body shell molded of composites and plastic materials. Horizontal body surfaces are molded from SMC, while resin injection molding (RIM) is used on vertical panels. (Stewart 2000, 18)

An all-composite compressed natural gas tank is currently being manufactured by Advanced Technical Products that is 50 to 70% lighter than Type 1 all-metal tanks or Type 2 and 3 tanks with metal liners and composite overwrap, and carries 20% ore fuel. The TuffShell tank features a high-density polyethylene liner overwrapped with carbon and glass fiber/epoxy, and cushioned by polyurethane foam and two impact-resistant exterior layers of glass/epoxy with aluminum end fittings. (Transportation Composites Newsletter 1999, 1)

According to International Body Engineering Conference (IBEC) chairman Bernard Robertson, "[Automakers] must radically rethink vehicle design and fabrication techniques to capitalize on the unique characteristics of high-performance plastics." (Technology Century 1999, 27) Several years ago, Chrysler designed a Composite Concept Vehicle (CCV) consisting of four huge PET plastic moldings glued together and bolted on a steel frame. But at $6000 per car frame it was still too pricey. In addition, automakers have toyed for years with super-strong composites with carbon fiber. In fact, they routinely use them during vehicle development and on show cars, but when they go into production they "switch to a material we can afford." (Technology Century 1999, 27)

5.2.4 Other Technologies

In addition to the materials and processes discussed above, there are a number of other products and processes that can be transferred from the automotive industry to aviation. AGATE members explored these technologies when they attended the SAE World Congress Exposition in Detroit in March 2000. AGATE members considered the following items interesting:

- New vendor sources for low-cost sensors, transducers, instruments were identified that could significantly lower the cost of GA instrument panels and monitoring systems. These items have received extensive testing and environmental evaluation, plus they operate is a much harsher environment than do airplane systems.

- Information on engine exhaust systems, acoustic and thermal insulation of exhaust systems, new fabrication techniques for exhaust systems, and identification of expert consultants on exhaust system design and tuning.

- Information on new molding materials and processes for lightweight engine baffling and ducting. Some of these materials may provide superior acoustic attenuation while also providing low-cost fabrication processes that are durable and lightweight.

- Sources of low-cost transducers, telemetry systems, and ribbon-wire technology were identified that are directly applicable to efforts to develop a low-cost drag reduction system for airplanes. This is a Project that may be able to reduce aerodynamic drag by electromagnetic stimulation of the boundary layer airflow. Initial tests have looked promising; however, development of the transducer has been stymied by the lack of a good process for building the ribbon-wire transducer. The new sources of technology identified above should aid in developing a more robust transducer.

- In general, it appeared that the subject automotive technology was 10% to 20% of the cost of similar aviation technology.

- Electronics currently being used (or nearing use) in the automotive industry look like the AGATE goals for avionics technology (GPS, moving maps, Internet, collision avoidance systems) and for some of the safety technology

- The automobile data bus works with automotive electronics that look a lot like airplane avionics; is the automotive data bus transferable to the airplane data bus?

- There are innovative manufacturing processes that might be very applicable to the aviation industry. It will be necessary to stop thinking of the FAA and the certification process as an obstacle and to start thinking about how to prove the new processes meet the requirements.

- Vacuum-process castings (if the “casting factor” requirement could be modified).

- Lead-free soldering

- Memory metals

- Practical magnetic refrigeration (to replace air conditioning)

- Helicoils

- Pierced nuts (in place of welded nuts), “Multi-Fastener” stamp in self-piercing nuts, and AKH Fastening System self-piercing rivets (require both side access, but were flush and filled)

- New windshield adhesive

- Hydroformed tubes (this was compared to hydroformed sheet parts that the aircraft industry regularly uses)
- Vacuum pressure formed plastics
- Impact extrusions
- Ground studs (high amperage connectors)
- Dual-shot injection molding process
- New parts marking technologies, including a new ATA specification
- Thinner, lighter sound foam
- Pressure imaging systems
- New crash simulation software

5.3 Manufacturing Philosophy

5.3.1 Toyota Production System (TPS)

AGATE Low-Cost Manufacturing Group toured a Toyota automobile plant in Kentucky during their October 1999 meeting in Cincinnati. The focus of the tour was an introduction to the Toyota Production System (TPS). This section summarizes the notes handed out during the tour.[3] (Toyota Motor Corporation 1995)

5.3.1.1 General

The Toyota Production System is a framework of concepts and methods for enhancing corporate vitality. It enables companies to achieve continual gains in productivity while satisfying customers' expectations for quality and prompt delivery.

Basically, the TPS is based on the knowledge that eliminating waste will increase quality and reduce cost. TPS has its roots in Henry Ford's system in that it still includes a conveyor belt, a division of labor, and an integrated supply chain. But it is different in that it also stresses mutual trust between labor and management. Employees in the Toyota system are in charge of their own jobs. Through their teams, they run their own worksites, they identify and take the initiative in implementing improvements, and they are rewarded for their initiative. In addition, the conveyor belt concept has been extended to include every step of production from customer order to automobile delivery and beyond, with maintenance and repair.

[3] Whether précised or directly quoted, all information in this section comes from the referenced Toyota notes.

The basic concepts of the TPS are unchanging, but they can be implemented differently. To benefit from the TPS, companies must satisfy three basic conditions:

1. Top management must make a strong and visible commitment to the system, must participate directly in implementing the system, and must instruct middle-level managers to do likewise.

2. All employees must participate in the system.

3. The companies must put in place a solid framework for cultivating capable leaders and for providing employees with necessary practical skills.

5.3.1.2 Just-In-Time Production

A core element of the TPS is "just-in-time" production; i.e. *making only what is needed, only when it is needed, and only in the amount that is needed*. Just-in-time (JIT) production, though simple in principle, requires dedication and careful, hard work to implement properly. Managers and employees must learn to: create leveled production; use the pull system; have continuous-flow processing; and establish takt time.

Leveled production distributes work evenly among the steps in the production sequence at all times, where as conventional batch production concentrates work on different processes at different times.

In the pull system, all work is in response to demand. This is in contrast to conventional push systems, where manufacturers produce goods and then try to find buyers for those products.

Continuous flow processing is an ideal system where items proceed one at a time though the production sequence. That one-at-a-time production is impractical, however, in work where dies, molds, or other tools to produce different items, as in forging, casting, stamping, and molding. In those kinds of work, processes must attempt to emulate continuous-flow production.

Takt is the pace of sales in the marketplace. In the plant, takt is quantified as the quotient of daily working hours divided by the number of orders that must be filled that day. Each step in the assembly process must take the same takt time to complete. Multi-skilled operators are required to make takt time work. As takt time goes up and down, fewer or more operators are added to the line, respectively. Assigning more people to a line means that each operator handles a narrow range of work. Assigning fewer people means each operator handles a broader range of work. Flexibility in allocating work is possible because people master a broad range of skills in the TPS, and "multiprocess handling" is followed, instead of "multimachine handling."

5.3.1.3 Jidoka

The principle of stopping work immediately when problems occur and preventing the production of defective items is basic to the TPS. That principle is called jidoka. Equipment is designed to detect abnormalities and to stop automatically whenever they occur. Operators are equipped with means of stopping the production flow whenever they note anything suspicious. The mechanical and human jidoka prevents defective items from progressing into subsequent stages off production, and it prevents the waste that would result from producing a series of defective items.

5.3.1.4 Standardized Work and Kaizen

Standardized work is a tool for maintaining productivity, quality, and safety at high levels. Three elements are used in structuring standardized work: takt time; working sequence; and standard in-process stock. Takt time, as described earlier, is the pace of sales in the marketplace. Standard in-process stock is the minimum number of work pieces that are needed to have on hand in a process to maintain a smooth flow of work. Standardized work provides detailed, step-by-step guidelines for every job in the TPS. Team leaders determine the most efficient working sequence. They make continuing improvements – Kaizen – in that sequence. That produces a new set of standardized work instructions that is then continuously improved and refined through Kaizen.

5.3.2 Automotive Manufacturing Consultant

As mentioned in the Abstract, the goal of AGATE is to develop the technology to create a single-engine, four-place aircraft with improved avionics and crashworthiness features that will sell for approximately $100,000. Lancair and Cirrus are two aircraft manufacturers within AGATE producing new certified products that fit the requirements, however, each are selling for significantly more than the target price. Each company involved in AGATE has worked to decrease the cost of certain components, however the entire picture of manufacturing management has never been specifically addressed. During Low-Cost Manufacturing Group meetings held in Cincinnati, OH in October 1999, and in Detroit, MI in March 2000, a consultant[4] was brought in to discuss manufacturing philosophies – primarily focusing on automotive industry lessons and "lean manufacturing." The following are some of the important points from the presentations.

5.3.2.1 Management

- The two jobs of management are to set priorities and to remove roadblocks.

- It's often been said that, "Business is War." Remember that no war was ever won without money and allies. Select the best suppliers (see Ten Commandments for OEM/Supplier Partnership in "Design" Section below). Develop partnerships with competitors to reach common goals, as the automotive industry has done.

- Changing a company is 90% psychology and 10% technology. Everyone within the company needs to be receptive to new ideas and be willing to accept that they might not currently know the best ways, but they can learn them. Leaders must be powered with profound knowledge, committed to a vision that they will lead with honor.

[4] Sandy Munro of Munro and Associates, Troy, Michigan

Managers must also be committed to the vision and serve the people that work for them by removing obstacles.

- Time is your worst enemy and your best friend. You cannot buy back a second of time so use it wisely.

- Stop saying, “It can’t be done.” That is fear and arrogance speaking. Arrogance will equal opportunity for your competitor.

- Do not set obtainable targets and goals for your company. Dream big and go for all of the profit.

- If you know yourself and know your enemy, you will win every battle. If you know yourself, but don’t know your enemy, you will win one, lose one. If you don’t know yourself and don’t know your enemy, you will lose every time.

- Allow plenty of up-front time, which is not just spent in engineering. Make sure to understand your target market as well. Design problems can not later be fixed on the factory floor; they can merely be worked around.

- Break down the wall between engineering and manufacturing.

- Benchmark against other industries. Profound knowledge will always be found on the outside.

5.3.2.2 Generating Profit

- Faster, Better, Cheaper + Best Practices = Profitable
  Speed is the most important element to generating a profitable manufacturing environment. Getting your product to market first will almost guarantee profitability. Therefor, nothing and no one should be allowed to get in the way of speed. Remember it is not the big that eat the small, but rather the fast that eat the slow. This is where small companies can succeed by being quicker and more agile than big companies. However, Best Practices must also be followed. Designs should not be made and changed on a whim (don’t run fast with your eyes closed). There should not be any one person in a company with the authority to implement a change. Best Practices include coordination, part standardization, “design for quality,” “design for manufacturability,” and “concurrent manufacture and design.”

- There are two modes companies work under – Save Money and Make Money. In the Save Money Mode, the company is running scared with short-term objectives. They cut labor and material, do not use analytical methods, have a time chart mentality, are hip-shooters, and are operating in a reactionary mode. In the Make Money Mode, the company makes investments with time and money, has long term goals, leads the market, has an analytical approach, and suffers from minimal internal politics.

- Put price on parts on floor-people will handle them better if they know how much they cost.

- Increase up front time by 20% and you decrease time to market by 50%.

Percent Up-Front Cost	**Cost Area**	**Percent Influence on Final Cost**
5%	Design	70%
50%	Material	20%
15%	Labor	5%
30%	Overhead	5%

- Determine the cost of the product by first selecting the selling price and required profit so that: *Price - Profit = Cost*. Do not determine the selling price of the product by calculating costs and adding profit so that: *Cost + Profit = Price*.

- It is possible to make money on small production runs if the products are designed for low-cost manufacturing.

- Automakers work to "Six Sigma" quality. Six Sigma is defined as 3 failures per million products, and refers to product reliability. Seven Sigma is defined as 3 failures per billion operations, and refers to process reliability. When trying to obtain these quality rates, a reduction in parts automatically reduces the number of inspections. The more parts there are, the more chances there are for failure when trying to achieve six sigma.

- As a rule, an airplane is built 3 times. According to the consultant, 80% of labor in aircraft should disappear.

5.3.2.3 Design

- See what everyone has seen but think like no one has thought.

- The best design is the simplest (Einstein), but the first design is never the simplest.

- Cannot have serviceability and reliability—choose one or the other for a priority. Remember for aircraft you have the choice of "Safe Life" or "Fail Safe" design. "Safe Life" is fatigue certification with definite replacement after X number of hours. "Fail Safe" is certification for the life of the aircraft.

- Beware of the "Ugly Baby Syndrome." An "Ugly Baby" is developed at the concept phase and you can not put it back. So what happens? It gets dressed up to make it look OK. It hangs around for about 18 years because the factory will "adopt" the ugly baby and resist re-design. The moral of the story is that the concept phase is extremely important to prevent creation of an unworkable design that ends up staying around because no one wants to throw away something they've started working with.

- Design must look good, function well, be built with quality, and at a profit. How to identify a poor design: it will have a functional failure if it is assembled wrong; it has small features that are critical to proper assembly; it has subtle differences in orientation; or it is a design so confusing that it can't be assembled by inexperienced personnel.

- Teamwork is the difference between good and bad designs. Here are Ten Commandments for an OEM/Supplier Partnership:
  1. Select the best quality partner. If you want the best products, you need the best quality suppliers. Your quality is only as good as the poorest quality supplier you have.
  2. Benchmark the world. Don't settle for the first candidate to come along. Take time to survey all the possible candidates to ensure you get the best possible match.
  3. Search out companies that innovate. Technologies and materials change, and if you want to be successful, your partners must be at the forefront of these changes, or you will all be left behind.
  4. Choose a partner you can work with. Once the honeymoon is over, you'll want a partner that is with you through good times and bad.
  5. Find a partner who believes in your company and your product. If they use your product and reinvest in your company, they share your customer's insights as well as a stake in your company.
  6. Select a partner that stays aware of the competition—both your competition and its own.
  7. Pick a partner that is committed beyond its shipping dock. A true partner not only makes trips to your plant to ensure its products are being installed and used correctly, but also talks to the ultimate customer, to see how its product can be improved to add more value.
  8. Help your partner grow. Share knowledge on new materials, technologies and techniques, and help them become better, so they can help you become better.
  9. Partners should be open, honest, and willing to share the risk. If a supplier is not willing to share its portion of warranty and investment costs, it is unlikely they will stand by you in bad times. Conversely, if an OEM isn't willing to accept its share of a problem, true improvement will never come.
  10. Partners should share the rewards. This way, what is good for one partner is good for the other.

- Design for a blind one-armed builder (BOB) to manufacture the part. Design the parts to fixture themselves one to another so the product is "Poka Yoke" (Error Proof). Be cognizant of the subtle differences that could result in misaligned or misplaced parts.

- The engineer must understand the available processes and design the product to meet the target cost. The engineer should know the cost of each part and process they use and think decisions through entire part manufacture to account for total cost. Minimizing the number of parts will minimize final cost.

- Avoid expensive secondary fastening operations. Threaded fasteners are the biggest enemy of quality – At Ford 70-90% of all mechanical failures are due to fasteners.

- Design out handling problems – Think Bulk Storage.

- Design parts that are easy to insert and align. Use gravity in assembly processes; don't fight it.

- Question servicing and simplify or eliminate packaging. Do not expect to profit from a design that will require customers to purchase special tools that the company would manufacture and sell.

- When evaluating a design, involve the people who will be responsible for manufacturing and assembling it.

- Look for ideas outside your own industry.

- Don't modify – redesign!

- The designer should think, "How would this be put together, or how can I design it to be built, by a robot?"

- The two most important things to remember when doing lean manufacturing is to (1) design for BOB, and (2) when in doubt, throw it out (parts count).

- The part value challenge: Does it have to move? Does it have to be a different material? When in doubt, *throw it out!*

- Piece cost is the great American mirage. Piece Functional Design vs. Design for Bob. "Any fool can make a piece-functional design."

- It is the Engineer's job to reduce waste.

- Consider obtaining a copy of AIAG Design Guide, which is published by the Auto Industry Action Group. This guide gives general guidelines from the Big 3 Automakers that all suppliers can follow to be certified to produce parts for them.

- Do not let design changes happen on the floor. The difference between a Refined Design out of Engineering and a Perfect Design out of the shop should be no greater than 5%.

- Material choice must be made in conjunction with manufacturing processes and cost during the initial design process.

- Allowing too much adjustment (too open of tolerance) is undesirable when trying to reduce failure rates. Fix one location, and float the other. Don't float both.

- Most gains from Design for Manufacturing comes from combining parts and assemblies, although for current composite aircraft, the biggest gains would probably come from self-fixturing.

- The four steps of redesigning are: (1) Discovery – how much does the current design cost?; (2) Brainstorm – collect new ideas; (3) Redesign; and (4) build a Business Case.

- The evaluation criteria for a product are: (1) Investment; (2) Cost; (3) Functionality; (4) Assembly Time; (5) Weight; (6) Warranty; and for aircraft (7) Certifiability.

5.4 Viper and Prowler Plant Tour

In March 2000, during the SAE Exposition trip, AGATE members toured the plant where the Viper and Prowler are made. The plant produces about 3000 cars annually, which is similar to the numbers that airplane manufacturers are aiming at in the near future. The following was noted.

- The cycle time is up to one hour, which is considerably longer than the high volume production lines that have takt times on the order of one minute. The production was set up with input from production workers to ensure that everything would be practical.

- The Prowler has an aluminum frame and the Viper has a steel frame, which are made by subcontractors. The smaller numbers of cars produced means that less expensive and dimensionally stable tooling must be used. This results in more hand fitting and rework.

- On these cars several body parts are composite material. The hood is SMC; some other fairings are RTM. These parts are very expensive compared to aluminum or steel because of the extensive finish work that is done by hand. The hood costs more than the frame and about as much as the engine.

- The cars had a simple, welded, aluminum structure. The low cost of the aluminum body of the Prowler and the high cost of the Viper hood were interesting contrasts in the selection of materials and how material selection can greatly influence production costs.

- It was also interesting to contrast the difference in production methods for the Prowler (moderate tolerance control) and the Viper with high tolerance control. It was also of interest that the Prowler is such a successful product with tolerance control that is only slightly better than that in current kit airplanes.

- The following cost data was discussed: SMC hood - $7,000; assembled ten-cylinder engine - $8,000; dash board (with working instruments) - < $600

- Painting continues to be a significant issue and a great deal of attention is placed on protecting it. Everyone wears pads around their watches and belt buckles (including the plant manager).

- The car is run up to speeds in excess of 100 mph the very first time the engine is started.

5.5 Final Thoughts

According to Mod Works, an aircraft repair and modification shop in Punta Gorda, Florida, a Mooney aircraft has an average of three hours of maintenance for every flight hour over its lifetime. By comparison, a car has 50 hours of driving for each minute of maintenance, according to automotive consultant Sandy Munro of Munro and Associates in Troy, Michigan.

Far fewer fasteners are used on cars than on aircraft. According to Mr. Munro, 70-90% of all mechanical failures at Ford are due to fasteners. The only rivets used are either self-piercing (two-side access required) for high structural strength and low installation time or pop-rivet style (one-side access). Self-piercing rivets are used only with ductile material (metal). Welding and spot-welding are widely used.

It is apparent that for high volume cars, currently the most favored materials are a stamped steel frame covered with injection molded plastic body panels. This is quite simply because they are cheap, easy to form into complex shapes, and automated manufacturing techniques can be used. In aviation these materials are of limited value because of weight (steel) and low strength (injection molded plastic).

6 SUMMARY AND CONCLUSIONS

6.1 Comparison of Composite Materials and Processes

Table 10 compares the different polymeric composite materials and processes discussed in this paper. The following points are evaluated:

State of Technology -	Categorized as: "State of the Art" for technologies that are in research, but not production; "Developmental" for technologies that may in production for some applications, but not widely practiced; and "Proven" for technologies that are widely used in production
Potential Applications -	Description of the types of aircraft parts that may be manufactured from the material and process
Costs -	Description of the total costs, including the up-front costs of tooling and the recurring costs of materials. Description of the up-front costs of tooling and the recurring costs of materials. Will give hard numbers where available. Note this section, and the entire report, only addresses a fraction of the total cost picture one must consider when initiating the design and production of a new aircraft. Labor savings is only one factor in the "ramp cost" of the total aircraft. It doesn't include a number of other costs such as: overhead, energy, inspection, tooling, queue times, total flow times, indirect factory support, finishing, warranty and field service support. All of which have a direct correlation to and impact on NOP (net operating profit), ROIC (return on invested capital), IRR (internal rate of return) and inventory turns. Furthermore it doesn't address the DOC (direct operating costs) or insurance costs of the end product. Will give hard numbers where available.
Labor Effort -	Description of the recurring amount of labor to perform the process. Will give hard numbers where available.
Certification –	Notes on the ease or difficulty of FAA certification of the material/process.
Quality Assurance -	Notes on the ease or difficulty of quality control of the material/process.
Advantages -	List of advantages to the material/process.
Disadvantages -	List of disadvantages to the material/process.

Table 10 Composite Material and Process Comparison

Material/Process	Fiber and Resin / Chopped Gun
State of Technology	Proven
Potential Applications	Non-structural parts such as interior panels
Costs	Tooling costs are low when the resin system is room temperature cure (as often selected for non-structural parts). The gun itself is also fairly inexpensive, especially when considered to the trade-off in labor time for hand layup.
Labor Effort	Very low operator time. One person can spray a large part, such as a bolt hull, within the working time of a resin. If proper surface gel coating is incorporated, very little post-layup surface finish is required as well.
Certification	Certification is fairly simple when the use is restricted to non-structural parts. Flammability requirements will have to be met, either through testing the laminate itself, presumably with paint or other surface finish, or by covering with appropriately rated upholstery. No mechanical testing is required on non-structural parts.
Quality Assurance	Quality Assurance is straightforward when the use is restricted to non-structural parts, particularly when the flammability of the laminate is not an issue, i.e. if it is covered with an upholstery material. Minimal supplier control and incoming receiving is required.
Advantages	Nearly isotropic properties
	Low labor times associated with spraying equipment
	Little to no additional surface preparation if paint incorporated in the molding process prior to spraying the composite
Disadvantages	Low fiber volume fraction
	Low strength to weight ratio effectively restricts use to non-structural parts
	Loss of directional property tailoring

Material/Process	Fiber and Resin / Wet Layup
State of Technology	Proven
Potential Applications	May be used in any application
Costs	Tooling costs are as low as possible, particularly when a room temperature curing system is used. An elevated temperature system requires more robust tooling, an oven, and likely a vacuum system.
Labor Effort	Labor effort is extremely high, however as time goes on, labor can be reduced by the addition of an automated cutting machine and/or mechanical wet-out system.
Certification	European regulation agencies, in particular, have very well defined methods for certifying the use of wet layup materials. The FAA does not follow the same standards as the European agencies, and requires more individualized certification testing. No universal databases are currently available from which to perform "equivalence testing" as described in Section 1.6.
Quality Assurance	To make continued quality assurance as easy as possible, the aircraft manufacturer should attempt to certify a wide range of process variables (mix ratio, cure time and temperature, processing environment) from the beginning so that they don't have to be as closely controlled. As with all hand layup techniques, fiber alignment and ply orientation will be difficult to control. Again, advantages can be gained by testing "worst-case" structural test articles from the beginning.
Advantages	Lowest cost structural composite
Disadvantages	Relatively low fiber volume fraction A percentage of operators will become sensitized to the resin, and will not be able to work with it Can not use honeycomb core, must select foam cores instead

Material/Process	Prepreg / Hand Layup, Vacuum Bag or Autoclave
State of Technology	Proven
Potential Applications	May be used in any application
Costs	Tooling costs are nearly as low as possible. Prepreg requires an elevated temperature system, which requires fairly robust tooling, particularly when curing in an autoclave. A vacuum and oven system is much less inexpensive than an autoclave, particularly when curing large parts such as a one-piece spar or wing skin. Prepreg requires refrigerated shipping and storage, adding to its cost.
Labor Effort	As with all hand layup techniques, labor effort is high; but it is less than required for wet layup. Labor can be reduced by the addition of an automated cutting machine.
Certification	Prepreg has the most well defined certification policy available for composite materials, the cost of which has been significantly reduced with the latest policies from the FAA (see Section 1.6).
Quality Assurance	Raw materials can be well controlled. Hand processing can open the door to errors in layup, such as orientation or placement of plies, and often requires significant operator and inspector training to perform correctly. However, compared to other hand layup techniques that involve mixing resin, the process is much more controlled with less room for processing errors.
Advantages	Easily defined material properties with several material databases available
Disadvantages	Labor intensive Requires refrigerated shipping and storage

Material/Process	Prepreg / Fiber Placement
State of Technology	Developmental
Potential Applications	Most advantageously used for tubular structure, such as fuselages. Can also be used to place relatively flat structure, such as bulkheads.
Costs	Up-front costs are very high. The fiber placement machine itself is on the order of $4 million. There are additional costs for CNC programming and intricate tool design.
Labor Effort	Labor costs are as minimal as possible. Two people can lay up an entire fuselage within an 8-hour shift. People are required to monitor the program, load prepreg in the machine, and lay non-tape reinforcements, such as honeycomb core, on the layup.
Certification	Prepreg has the most well defined certification policy available for composite materials, the cost of which has been significantly reduced with the latest policies from the FAA (see Section 1.6).
Quality Assurance	Raw materials can be well controlled. A rigorous first article inspection is required, after which there are relatively few requirements for continued inspection.
Advantages	Easily defined material properties with several material databases available
	Automated system is accurate and requires very few labor hours
Disadvantages	Significant up-front costs for tape laying machine and tooling
	Requires refrigerated shipping and storage

Material/Process	Fiber and Resin / RTM
State of Technology	Proven
Potential Applications	Complex shapes with access on both sides, such as ribs and floors; also control surfaces with braided performs
Costs	Tooling costs are relatively high due to the requirement for matched metal dies with tight tolerance. Up-front costs also include the resin injection and heating system.
Labor Effort	Labor effort varies, based on the method used to create preforms. Fiber placed or braided preforms have the lowest operator time; hand cutting and layup of the preforms requires the most, and automated cutting with hand layup of the preform would be somewhere in the middle. The type of preform tackifier would also affect labor effort. After the preform is fabricated, little operator effort is required to place it in the mold and perform the injection process.
Certification	A unique material qualification plan would be required to take into account processing variables such as variations in cure temperature or effects of fiber distortion in the RTM process.
Quality Assurance	As the RTM process itself is automated, it requires minimal QA efforts, especially if fiber orientation/distortion is addressed up-front during material qualification. Preform fabrication could require significant QA effort if it is a hand layup technique, or be relatively low if it is automated.
Advantages	Minimal operator contact with resin No need to “rush” layup due to short working time of resin, but then followed by a fast cure cycle Increased fiber volume Better surface finish due to hard tooling on all sides Reduced labor effort after removal from the mold due to net shape
Disadvantages	Mold and injection system difficult to simulate and often designed by trial-and-error Tool tolerances must be tight, increasing cost

Material/Process	Fiber and Resin / VARTM (Vacuum Assisted RTM)[5]
State of Technology	Developmental
Potential Applications	Complex shapes with access on both sides, such as ribs and floors, large unitized shell and control surface type structures.
Costs	Tooling costs are lower than traditional RTM since matched metal dies not required. Up-front costs include the resin injection and heating system.
Labor Effort	Labor effort varies, based on the method used to create preforms. Fiber placed and braided preforms offer lowest operator time; hand cutting and layup of the preforms requires the most, and automated cutting with hand layup of the preform would be somewhere in the middle. The type of preform tackifier would also affect labor effort. After the preform is fabricated, little operator effort is required to place it in the mold and perform the injection process, which only lasts on the order of a few minutes.
Certification	The same as for traditional RTM.
Quality Assurance	As the VARTM process itself is automated, it requires minimal QA efforts, especially if fiber orientation/distortion is addressed up-front during material qualification. Preform fabrication could require significant QA effort if it is a hand layup technique, or be relatively low if it is automated.
Advantages	Minimal operator contact with resin
	No need to "rush" layup due to short working time of resin
	Increased fiber volume
	Applicable to larger structures; can use autoclave tools
Disadvantages	Mold and injection system difficult to simulate and often designed by trial-and-error

[5] Information from (McCabe 2001).

Material/Process	Fiber and Resin / RTM with Lost Core
State of Technology	State of the Art
Potential Applications	Hollow structure such as a control surface or horizontal stabilizer
Costs	As for traditional RTM, significant up-front costs are required for hard outer tooling and the resin injection system. The lost core material is reusable, making it advantageous compared to inflatable materials that do not last very many cure cycles. However, the lost core material must be reformed after each cure cycle.
Labor Effort	The part requires hand layup. However, a significant amount of structure, such as skins, ribs, spars, and stringers, are all laid up and cured at once thereby reducing post-assembly requirements.
Certification	The same as for traditional RTM.
Quality Assurance	The same as for traditional RTM with some added complication due to the hollow structure. If visual inspection is not possible of the structural components, additional techniques may be required.
Advantages	Minimal operator contact with resin No need to “rush” layup due to short working time of resin Increased fiber volume Better surface finish due to hard tooling on all sides Ability to integrate significant amounts of sub-structure in a single layup, thereby reducing post-assembly time Overcomes restrictions in mandrel design
Disadvantages	Mold and injection system difficult to simulate and often designed by trial-and-error Tool tolerances must be tight, increasing cost Core must be reformed after each cure

Material/Process	Fiber and Resin / Resin Film Infusion
State of Technology	Proven
Potential Applications	As for wet layup or prepreg hand layup, this can be used on any part. There are limitations on core material selection.
Costs	SCRIMP seems to be the most effective RFI method available, but requires licensing fees. Afterward, tooling and processing costs are similar to hand laid up prepreg.
Labor Effort	As with all hand layup techniques, labor effort is high. Automated cutting machines can be used to reduce labor hours.
Certification	European regulation agencies, in particular, have very well defined methods for certifying the use of wet layup materials, which this is. The FAA does not follow the same standards as the European agencies, and requires more individualized certification testing. No universal databases are currently available from which to perform “equivalence testing” as described in Section 1.6.
Quality Assurance	To make continued quality assurance as easy as possible, the aircraft manufacturer should attempt to certify a wide range of process variables (mix ratio, cure time and temperature, processing environment) from the beginning so that they don’t have to be as closely controlled. As with all hand layup techniques, fiber alignment and ply orientation will be difficult to control. Again, advantages can be gained by testing “worst-case” structural test articles from the beginning.
Advantages	Low cost structural composite Fiber volume fractions can be increased over traditional wet layup due to ability to consolidate preforms prior to addition of the resin Worker contact with resin is minimized, compared to traditional wet layup, because resin is infused while bagged and under vacuum
Disadvantages	Can not use honeycomb core, must select foam cores instead Infusion system design is intricate and often performed by trial-and-error

Material/Process	Thermoplastic / Hydroforming
State of Technology	State of the Art
Potential Applications	Complex shapes with access on both sides, such as ribs skin halves and floors
Costs	Tooling costs are not as low as for thermosets, because of the high temperature requirements, but only requires metal tooling on one side. Having an elastomer on the other side reduces cost somewhat. High temperature facilities are required to process thermoplastics, which increase capital investment
Labor Effort	Very low
Certification	Thermoplastics can be certified with relative ease, with respect to generating B-Basis or A-Basis design allowables. Their limiting factor will likely be their Tg.
Quality Assurance	Moderate. As with most composites, some form of NDI over and above visual inspection is likely to be required to assure part quality. With the automated system, however, it should be relatively easy to implement statistical process control and batch sampling.
Advantages	Fast production cycles and low labor time
Disadvantages	Thermoplastic limitations – high processing temperature and relatively low Tg

Material/Process	Prepreg / Internal Pressure Molding
State of Technology	State of the Art
Potential Applications	Hollow parts, with or without co-cured inserts, such as control surfaces
Costs	Tooling costs are relatively low. Hard outer tool can be fabricated from low-cost materials, depending on the cure temperature of the prepreg used in the parts. Flexible bladder inside can be used in a number of different parts.
Labor Effort	This process still involves hand layup of prepreg. However, IPM consolidates multiple parts, reducing over-all labor hours for layup and secondary bonding/assembly.
Certification	Simple. Can use an already-approved prepreg system. This processing method should not require additional certification.
Quality Assurance	As with most composites, some form of NDI over and above visual inspection is likely to be required to assure part quality. Having a hollow structure may require additional negotiation with the FAA to assure integrity, if visual inspection is not possible.
Advantages	Complex, net-shape hollow structures can be fabricated without secondary bonding Can use an already-approved prepreg material system Can co-cure inserts
Disadvantages	Requires hollow parts that must have a 0.25-inch hole for bladder removal Variations in resin content of the prepreg will affect final wall thickness

Material/Process	Modified Prepreg / Electron Beam Cure
State of Technology	State of the Art
Potential Applications	Any part, including very thick sections (up to 12 inches)
Costs	Extremely high up-front costs associated with an electron beam vault. Afterward, processing costs are 25-60% lower than thermal processing. Resin systems are more expensive than heat-activated thermosets. Tooling costs can be extremely low as the tooling doesn't have to undergo any heat or pressure cycles.
Labor Effort	This all depends on the layup method used. Hand layup methods will obviously require more labor effort than an automated layup system. Both can be used with e-beam curing composites.
Certification	E-beam curing will be new to the FAA, and will require a unique certification plan. It is likely the FAA will require a large number of tests, when compared to heat-activated prepreg or even wet layup materials, due to the newness of the technology if it will be used on primary structure.
Quality Assurance	QA effort will vary depending on the layup method used. Layups can be inspected prior to cure for variables such as ply orientation and location. E-beam curing has the advantage over RTM in that the plies should not be warped during cure due to resin flow.
Advantages	E-beam curing is a "green" process that doesn't produce volatiles Uniform and efficient cross-linking without hot spots Thick parts can be cured in one cycle, up to 12 inches thick Low temperature materials can be integrated, such as foam core, which would otherwise not withstand a heated cure cycle High fiber volumes
Disadvantages	Resin systems still weak in interlaminar shear strength Up-front costs for e-beam vault

6.2 Conclusions

Designers bent on making the most use of composites in a new design often try to use composites where there is no benefit to doing so. One example is to use composites for many small parts such as clips and brackets when metals would do the job effectively at much lower cost and little, if any, additional weight. Composites should be used only where careful and thorough studies show a clear benefit. The studies must include the availability and cost of facilities, the company's experience and ability to implement their ideas, and the tradeoff of benefit with risk. (Vosteen and Hadcock 1995, 13) When used, the designer must take advantage of composite properties and integrate them into a "composite design" instead of recreating metallic structure with composite parts. (Davies 2001) Composite aircraft design and construction should be different from metallic aircraft construction; the industry is just starting to design structure that takes advantage of composite materials. Past practice has been to attempt to design composite parts that look and form like metal, this leads to parts being less cost effective and, in many cases, too heavy and require secondary bonds. (Miller 2001)

Most new general aviation companies have very limited start-up funds. This will tend to limit them to methods that require the least-expensive tooling, with the trade-off in labor time. In addition, they will tend to choose methods such as wet layup or prepreg hand layup that can be used for every application. Cirrus Design Corporation and The Lancair Company are two examples of general aviation manufacturers who have elected to use vacuum-cure prepreg systems because they offer low up-front costs, with the penalty of higher labor hours. As a company becomes more established, they may be able to transition to more optimal materials and processes as additional funds are available. For example, Raytheon was able to use fiber placement on their Premier aircraft. They now have three fiber placement machines, each of which is approximately equal to the selling price of two aircraft. However, switching materials and/or processes mid-stream means manufactures are required to re-prove their design to the FAA. Ideally, a start-up company would have the funds to identify and start with the most ideal material/process combinations from the beginning, so that material qualification and structural tests need only be performed once.

While composites offer dramatically reduced parts counts, there are likely to be limits to how far it can be taken. While a process such as lost-core molding was demonstrated effectively on a horizontal stabilizer, attempts to manufacture an entire wing as a single component may start to have reduced benefits due to the cost of scrapping a completed wing due to a non-conformance. It's one thing to scrap a rib that was miss-drilled, it's another to scrap an entire wing when a small portion of it (such as a rib) was miss-drilled. Careful thought must be given, not only to the structural design and material processing, but to post-assembly and repair requirements.

6.3 Future Work

Much future work remains to be performed, especially in the areas of material databases and standardized testing requirements, non-destructive inspection, and composite repair.

6.3.1 Material Database and Testing Standards

The FAA has recently set standards for prepreg material database generation and material qualification. Several FAA-approved material databases have already been generated, with at least one aircraft manufacturer in the process of equivalence testing. Additional standards need to be set for other material/process systems such as wet layup, or even thermoplastics. Additional guidance on quality assurance test requirements (incoming receiving and in-process testing) would also be beneficial.

6.3.2 Non-Destructive Inspection

Non-destructive evaluation was briefly discussed in Section 4. As mentioned in that section, the most common non-destructive techniques involve visual, acoustic (coin tap or hammer tap) or ultrasonic (a-scan or c-scan) inspection. Glass composites are well suited for visual inspection, at least at the time of manufacture. Painting and assembly will reduce effectiveness in the field. Acoustic methods are very subjective and require highly skilled operators. In addition, when an indication is given of a possible defect through acoustic methods, further investigation is required. Ultrasonic inspections are the most accurate and objective. A-scan works fairly well for bond testing. It can also be used to check for delaminations, but requires coordinated equipment selection, calibration, and procedures to ensure the scan is taken the same way every time to ensure defects are found. A-scan tends to be conservative in showing defect indications which, when further investigated, often turn out to be false. C-scan is much more accurate. It can clearly show delaminations and provides a three-dimensional view of defect location. However, it is a much more expensive piece of equipment and requires the entire part be placed in a chamber for scanning, where as a-scans are portable.

The two methods discussed in this guide should be further investigated. A full investigation of all possible non-destructive inspection techniques should also be performed with an eye toward their application in the general aviation industry. The industry should work to standardize inspection methods for composite structures.

In addition, non-destructive techniques should be better employed on the fiber materials. Current standards/specifications for fibers are based on tensile strength and modulus. This is perhaps acceptable for glass, but carbon can have significantly different compressive properties even when tensile strengths and moduli are consistent. Currently, the way a composite end-user determines equivalence between different carbon fibers is through lamina testing, which then adds in resin variables. An inspection technique such as Raman spectroscopy could be used to better define carbon fiber characteristics to help end-users select comparable materials. It could also be used for in-situ inspection during fiber manufacture.

6.3.3 Repair Procedures

Composite repair is another very important factor in selection of composite materials and processes, but was not discussed in this report. Few universal standards exist, particularly in general aviation, although there are more in place for transport-category aircraft. There is also a shortage of trained repair personnel. Europe has more experience with composite repair, but it tends to be limited to wet layup structure.

Currently, composite repair tends to be unique for each aircraft, with the manufacturer's engineering department writing unique dispositions and repair instructions for each type of damage. It will be difficult to establish universal standards, given the variety of composite structure in different aircraft, but the FAA should work to publish guidance materials that standardize composite repairs to the extent possible. There is an FAA advisory circular on standard aircraft repair procedures (AC 43-16A) but it only covers metallic construction. Similar standards are necessary for composite construction.

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