NEW COMPOSITE DESIGN AND MANUFACTURING METHODS FOR GENERAL AVIATION AIRCRAFT STRUCTURES

Alan Anderson, Curtis Longo and Paul Teufel
Toyota Aviation Business Development Office
Toyota Motor Sales, USA Inc.
19001 South Western Avenue, MS32
Torrance, CA 90509

Keywords: *Composite, Co-cure, RTM, Filament Winding*

Abstract

For most of the last 50 years of FAA-certified general aviation aircraft design and fabrication very little technological progress has occurred, especially in small aircraft construction. Technological improvements in aerodynamics such as laminar flow airfoils could not be exploited, since the exterior surface smoothness required for these airfoils could not be easily achieved with conventional aluminum manufacturing methods. Composites have been incorporated into recently introduced small aircraft, however the design and manufacturing methods used (generally hand layup of prepreg material into open molds) have for the most part not resulted in cost or weight savings compared to aluminum.

New methods of composite manufacturing and design were deemed necessary by the Toyota Aviation Business Development Office in order to fully take advantage of the benefits of composite structures such as laminar flow aerodynamics, yet still achieve a cost-competitive product. This paper describes the technology developments created by Toyota and vendors during the design, fabrication, and testing of the Toyota Advanced Aircraft Proof of Concept (TAA POC) vehicle. Distinctly different technologies were developed with the focus placed on demonstrating new design and manufacturing concepts for the fuselage and wing. Specifically, filament winding technology was employed to fabricate the fuselage and resin transfer molding (RTM) on the wing. Both parts used unique materials and internal tooling methods to produce composite structures primarily as co-cured assemblies.

1 Introduction

1.1 Background

Until recently, nearly all general aviation (GA) structure was constructed of thin gage aluminum, formed as required, and then fastened together. Slow flying aircraft accepted the drag penalty of protruding head fasteners while faster aircraft accepted the fabrication and installation cost penalties associated with flushing the fasteners. Further, most small aircraft employ airfoils developed by NACA some 70 years ago [1], whereas, recent aircraft designs have employed custom laminar flow airfoils that demonstrate significant drag reduction when compared to the earlier NACA sections. The resulting more efficient laminar flow wings require contour, waviness and smoothness that are difficult to obtain in conventional riveted aluminum GA construction and are costly in formed and/or machined aluminum [2].

In recent years, composite construction for GA aircraft has made progress, moving from owner-built experimental aircraft to recently certified production aircraft. The total number of parts required for these composite airframes is considerably less than similar aircraft of aluminum. However, structure is still assembled from several individually laminated and molded pieces subsequently joined with adhesive and/or mechanical fasteners. This ‘joining’ to permanently assemble structure adds weight and cost and has been a source of quality concerns. A comparison of total parts among small GA aircraft wings is shown in Fig. 1.

Wing Construction	Structural Elements	Total Parts
Thin Gage Riveted Aluminum	33	70
Current Production Composite	22	31
Integrated Composite Potential	17	9

Fig. 1. Small GA wing total parts comparison

Notwithstanding this joining, whether bonding and/or mechanical fastening, many recently introduced small aircraft have selected composite construction to take advantage of the aerodynamic efficiency improvement derived from laminar flow airfoils and the weight and cost reductions associated with fewer parts.

1.2 Goals

For Toyota to contribute to and thereby become mutually beneficially involved in the GA industry, it was believed necessary for Toyota's Aviation Business Development Office (ABDO) to develop new design concepts and manufacturing methods for composite structure that result in first, aerodynamic efficiency through accurate molding of structure, second, reduced weight through structure integration, and third, low cost through process innovations. ABDO elected to work with wing and fuselage structure for a four-place aircraft as others have developed lightweight and low cost designs and manufacturing processes for empennage and control surface structures. Wing and fuselage structure developments were part of a much larger effort encompassing a Toyota Advanced Aircraft Proof of Concept (TAA POC) airplane.

For the TAA class of aircraft, the wing produces approximately 25% of the total aircraft drag. Laminar flow airfoil and wing technology offered the potential to significantly lower this value. More recent laminar flow airfoil development has produced airfoils capable of attaining laminar flow over 70% of the wing's chord. However, these airfoils suffered from high pitching moment and associated high trim drag. In addition to these maladies, older laminar flow airfoils also suffered from severe loss of lift when the surface became contaminated and lost laminar flow [3].

For the TAA, wing structure development actually began with the design and wind tunnel testing of a proprietary laminar flow airfoil to increase aerodynamic efficiency. This airfoil was designed to attain a moderate 50% of chord laminar flow with low pitching moment and very small lift loss when laminar flow is lost. As a result, wing contour, waviness and surface smoothness to obtain laminar flow became very important and made composite construction highly desirable. However, for the joining rational explained, current composite construction would have required extensive finishing to obtain laminar flow and would have resulted in wing structure exceeding weight and cost targets.

Therefore, the goal of wing development was to create a one-piece, laminar flow wing surface from composite materials with highly integrated structure (see Fig. 1) that would be superior to competitive aircraft in the areas of aerodynamic efficiency, weight and cost. The initial wing concept is depicted in Fig. 2.

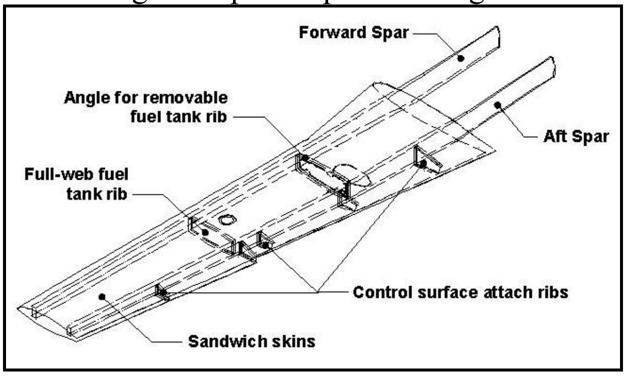

Fig. 2. Initial wing concept

Fuselage efforts began with the study of construction methods used for metal and composite fuselages for both production and experimental aircraft. Composite fabrication methods studied included automated fiber placement (tow placement, tape lamination, filament winding, chopped fiber spray-up, short oriented fiber preforms), resin transfer molding (RTM), vacuum assisted resin transfer molding (VARTM) and resin film infusion. Based upon this construction methods study and the results from the design trade studies, it was concluded that a fuselage made from carbon-fiber/epoxy and aramid honeycomb core would be best. For cost reasons, a non-traditional form of carbon-fiber/epoxy would be used.

Therefore, the goal of TAA fuselage development was to create a one-piece, closed-shape composite fuselage, of minimum wetted area, with highly integrated structure that would be superior to competitive aircraft in the areas of aerodynamic efficiency, weight and cost. The initial fuselage concept is depicted in Fig. 3.

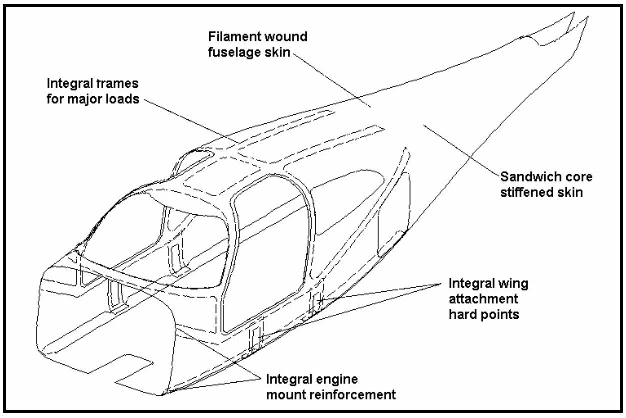

Fig. 3. Initial fuselage concept

2 Design Trade Studies

Prior to fabricating structure, various studies were conducted to assure obtaining desired aerodynamic efficiency and meeting the weight and cost targets. Collectively called design trade studies, efforts in aerodynamics, structure design, cost, weight analysis and materials were conducted and the results used to guide design decisions and manufacturing process selections.

2.1 Aerodynamic Studies

A study of previous research into contour, waviness and surface smoothness requirements necessary to achieve laminar flow was conducted [4,5,6]. Based on this research it was determined that molding the airfoil of the wing as a continuous surface was desirable. Further, obtaining laminar flow without rework and/or excessive finishing was important to meet cost and weight targets.

Information regarding potential tool fabrication methods indicated that achieving laminar flow could be accomplished with available tooling technology. As a result, the following as-fabricated wing shape targets were created:

1. Contour: ±0.76mm/0.030-inch (maximum form error from nominal)
2. Chord and span waviness: ±0.25mm/0.010-inch individual wave maximum with ≤0.0025 section ratio (h/λ) (longer wavelength deviations from nominal)
3. Roughness: ≤120μm/50μinch (R_a) (shortest wavelength irregularities of a surface)
4. No steps or gaps in airfoil contour until reaching control surface interfaces.

Tolerance selections for the fuselage shape were also driven by tooling technology. However, since developmental tooling was proposed, some additional risk was introduced in meeting the following as-fabricated fuselage shape targets:

1. Contour: ±1.27mm/0.050-inch
2. Waviness: ±0.42mm/0.017-inch individual wave maximum with ≤0.0042 section ratio
3. Roughness: ≤120μm/50μinch
4. No step up into the air stream, no external fasteners, and no gaps greater than 2.5mm/0.10-inch.

2.2 Structure Design Studies

Studies were conducted throughout the design process to minimize structural elements and total parts. For example, the use of mechanical fasteners to permanently join structure was avoided due to high fastener cost, the cost of installation labor and the relatively poor capability of composite structures to react the resulting concentrated loads.

Generally, wing and fuselage structural elements were co-cured or bonded and designed to be fail-safe. As a design philosophy, the only reasons not to co-cure an element are first, to allow removal of the internal tooling necessary for molding the integrated elements or second, to simplify systems installation. Even the passenger and baggage doors were co-cured with the fuselage to ensure fit upon assembly. Wing and fuselage structural elements that were impractical to co-cure were secondarily bonded.

A design trade study was performed to assess the merits of sandwich skins vs. solid laminate skins (with and without stiffeners, as required). Results indicate that sandwich construction would be lighter weight as carbon-fiber/epoxy skins of small GA aircraft are generally sized more for stability (buckling) and damage tolerance than for strength. Only minimum thickness skins are needed for strength. Two 0.19mm/0.0075-inch thick plies of carbon-fiber/epoxy with a core of honeycomb or foam were determined to be the superior solution for structural stability and damage tolerance. Additionally, sandwich structures can be designed to deform very little when subjected to flight loads enabling the wing to maintain its shape and thus sustain laminar flow.

2.3 Cost and Weight Studies

Numerous cost and weight studies were conducted to compare various fuselage and wing design concepts, manufacturing methods and materials (see Fig. 4 and 5).

For the wing, it was determined that an RTM process, with integrated structural elements, would best achieve cost and weight targets. See Fig. 4, Case 3. Although both full and half span concepts were studied, full span concepts were believed to be high risk because of the lack of a transfer resin with suitably long transfer time. The half span concept selected included co-cured, continuous sandwich skins with two integral sandwich spars, two inter-spar ribs and all trailing edge ribs. Studies assumed that the needed materials and technologies would be developed.

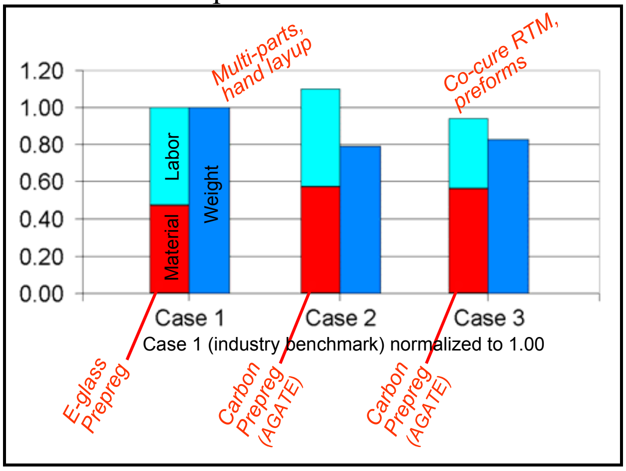

Fig. 4. Wing cost and weight trade studies

For the fuselage, it was determined that filament winding was the optimum method (Fig. 5, Case 27). The one-piece, integrally stiffened co-cured fuselage concept included many integral structures including wing attachment hard points, engine mount reinforcements and frames for major loads. Studies assumed use of low-cost carbon fibers, bulk resin and development of necessary technologies.

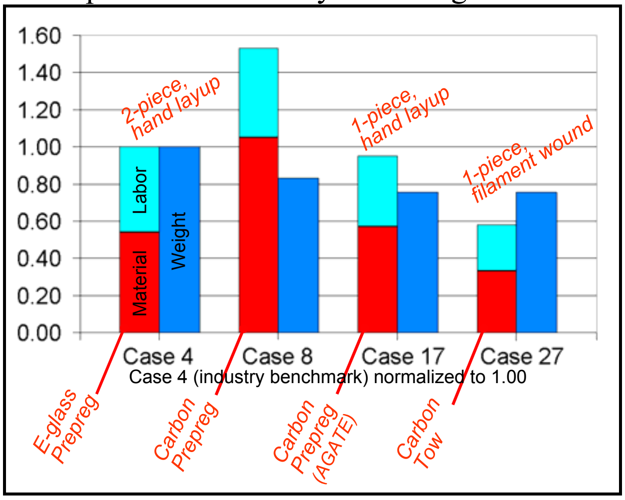

Fig. 5. Fuselage cost and weight trade studies

2.4 Materials Studies

Various material studies were conducted in conjunction with the structure design studies to determine the best materials for fuselage and wing structure. Candidate fiber included carbon and E-glass. Aramid honeycomb core and foam cores of polymethacrylimide (PMI), polyvinyl-base, and polyamide materials were the principle sandwich core candidates. Candidate resins for the fiber matrix and for adhesive to bond face sheets to the core included both 132°C/270°F and 177°C/350°F cure epoxies.

Regarding fibers, it was determined that polyacrylonitrile (PAN) based standard modulus carbon fiber in high-count tow represented the immediate future - the next five years - if the high-count tow could be spread to the thin gage required for sandwich construction.

Aramid honeycomb and PMI foam were the preferred core materials for both weight and temperature capability with honeycomb core being the easiest to form to GA airfoil and fuselage shape requirements.

Resin studies revealed that multiple developments were necessary. The first, a tough, two-part RTM resin capable of transfer at room temperature (with cure occurring at 132°C/270°F) for molding highly integrated wing sized parts. Second, a resin on a scrim was needed to bond a barrier onto core materials. The barrier prevents the transfer resin from filling the core materials and the scrim support the barrier during resin transfer. And third, a toughened resin for use with filament winding of the fuselage with skin-to-core adhesive properties. Additionally, a film adhesive for bonding skin plies to core was selected from aerospace industry offerings to reduce the risk associated with flying a developmental filament wound honeycomb sandwich fuselage.

3 Technology Development

Process developments were performed in order to successfully fabricate structures using RTM technologies (half-span wing) and filament winding (one-piece fuselage).

3.1 RTM Technologies

Successful RTM fabrication of wing structure required technology development in two main areas: sealing of core against transfer resin ingress and co-curing of large integrated structures.

3.1.1 Sealing of Core

Without sealing, honeycomb core cells would quickly fill with resin due to the pressure used to transfer the resin. Resin is transferred using three atmospheres of pressure in addition to full vacuum in the mold part cavity. Closed-cell foams were investigated and found to absorb resin to an unacceptable level. To seal the core, a suitable barrier was created. The requirements for the core sealing barrier were:

1. Withstand pressure during transfer of resin
2. Bond core to face skins with resulting bond strength greater than core strength
3. Compatible with the components in the other resins used
4. Meet typical GA aircraft cure (132°C/270°F) and service (82°C/180°F) temperatures
5. Laterally stabilize the shape of honeycomb cores.

For the resin barrier, trials were conducted with a variety of plastic films. Polyetherimide (PEI) film was determined to have the best overall performance. In addition to satisfactory bonding and sealing, PEI film was relatively low cost and lightweight. A lightweight glass fiber fabric (scrim) was used in conjunction with an epoxy resin to support the film barrier and bond the film to the core.

3.1.2 Co-curing of large integrated structures

Resin transfer molding of a part the size and structural complexity of the TAA half-span wing posed multiple challenges.

Major tooling challenges included size and complexity of the internal tooling necessary to facilitate structure integration. For development, all of the rigid components were machined from aluminum alloy and then hard anodized to provide durability. See Fig. 6. The leading edge D-section tool was fabricated as a single piece (removable from wing root end) and the trailing edge tooling was interlocking, multi-piece elements fitted between trailing edge ribs (removable directly aft). Proprietary collapsible, semi-rigid tools, ‘plastic bladders’, were used in the area between the spars to provide forgiveness of preform layup bulk. The ‘fuel tank bladder’ is visible in Fig. 6.

Fig. 6. Wing mold base and some internal tooling

Materials efforts consisted of core material selections, fiber preform evaluations and transfer resin development. Core materials selected include aramid honeycomb (wing skin and ribs) and PMI foam (spar webs). Fiber preforms investigated included both dry and 'tackified' (impregnated with low resin content) versions of woven and stitched fabrics and filament wound broadgoods. Ultimately, tackified woven fabric was selected to decrease development time and risk.

Development of a resin transferable at room temperature with sufficient transfer time was the major material development. No aerospace RTM resin was found to have sufficient transfer time for a part the size and complexity of the half span wing. Requirements for the resin included the following:

1. Viscosity suitable for resin transfer at room temperature using three atmospheres pressure
2. Minimum transfer time of 120 minutes
3. Chemically compatible with preform resin
4. Wet $T_g \geq 110°C/230°F$
5. Resin cure at 132°C/270°F using three atmospheres of pressure
6. Damage tolerance greater than benchmark carbon-fiber/epoxy prepreg material
7. Low cost relative to other proprietary aerospace RTM resins.

These requirements were provided to a resin system formulator who developed and evaluated several potential room temperature transferable systems with the resulting resins used to make subcomponent parts. Finally, the selected resin was used to fabricate multiple left and right half-span wings.

3.2 Filament Winding Technologies

Technologies developed to support fuselage fabrication include winding of spread fiber that is in-process impregnated with toughened resin and then wound on a mandrel that can be made to 'push-out' to an outer mold line (OML) mold when internally pressurized.

3.2.1 Fiber Winding

Winding was selected as the method of fiber and resin placement because of the ability to use relatively low-cost carbon fibers (12,000-end) and bulk resin. Further, the winding equipment automatically places the fiber and resin for the fuselage skin on a mandrel capable of pushing the wound material out to the surface of an outside mold to achieve consolidation and a smooth exterior surface. The fuselage was a winding challenge because of its complex shape: squarish firewall area, relatively flat bottom in the cabin area, concave area forward of the windshield and the aft 'wasp body'.

Because of the shape, a numerically controlled winding machine with a sophisticated head was necessary. The winding machine automatically oriented and positioned the spread and resin impregnated fiber band onto the revolving surface of the fuselage shaped winding mandrel. Fiber spreading technology was developed to allow the use of four, 12,000-end carbon fiber tows at a fiber band areal weight of $95 g/m^2$ for the fuselage skin and for helical wound (±45°) broadgoods material. For unidirectional wound (0°) broadgoods material, a fiber band areal weight of $150 g/m^2$ was the target. A heated, in-line impregnation system was developed to enable the use of a toughened epoxy resin matrix – a resin not normally used with filament winding.

3.2.2 Winding and Push-out Mandrel

Filament winding required the use of a mandrel upon which the fibers and resin for fuselage skin plies are directly placed. A winding mandrel for the fuselage shape was required to possess the following characteristics:

1. Stable surface for winding, lay up of frame plies and core placement
2. Sized to allow the OML mold to close around the wound part
3. Necessary features like channels for lay up of frame plies
4. Necessary winding aids including end turnarounds
5. Index to OML mold in the area just forward of the firewall flange
6. Apply pressure for consolidation, that is, push-out and hold the wound fuselage out to the OML mold surface during cure
7. Collapsible for extraction after cure.

A suitable fuselage winding mandrel (see Fig. 7) was designed and fabricated. Obvious features include channels for frames, recesses for baggage bulkhead frame and pockets for wing attachment lugs.

Fig. 7. Fuselage winding mandrel

The developmental OML mold used (see Fig. 8) consists of three carbon-fiber/epoxy pieces that are bolted together along flanged edges. An aluminum insert was used to eliminate negative draft at the lower firewall flange. The mold completely encloses the wound fuselage when end plates are in place.

Fig. 8. Right side, bottom and insert of OML Mold

4 Fabrication Processes

This section lists the steps used for fabrication of the TAA POC wing and fuselage articles, including the use of the new RTM and winding technologies. Although production scenarios were created for costing purposes, no attempt was made to simulate production during fabrication of these articles.

4.1 Wing Fabrication

Wing fabrication consisted of the following:

1. Prepare materials including core sealing
2. Prepare all preforms (forward and aft spars, lower and upper skins, fuel tank ribs, and trailing edge ribs)
3. Position upper skin preform in mold
4. Assemble trailing edge rib preforms to trailing edge tooling and locate on upper skin in mold
5. Position aft spar preform
6. Locate semi-rigid inter-spar tooling and inter-spar rib preforms on upper skin
7. Position forward spar preform
8. Locate cabin area spar extension mandrels
9. Lap lower skin preform leading edge with leading edge of upper skin
10. Position D-shaped leading edge mandrel and then complete locating lower skin preform
11. Locate remaining spar extension mandrels, mold top, mold end and then vacuum seal the resulting mold cavity
12. Install in press (or autoclave) and then perform vacuum integrity check
13. Pressurize semi-rigid tooling (bladders)
14. Transfer the RTM resin at room temperature using pressure and vacuum
15. Heat to cure temperature and hold (2 hours)
16. Cool and then open mold
17. Remove wing from tooling
18. Trim and inspect.

For production, the manual cutting, kitting, handling and positioning of tools and preforms used for this development effort would be replaced by mechanized or automated processes typical to current composite part production.

4.2 Fuselage Fabrication

Fuselage fabrication consisted of the following:

1. Filament wind broadgoods for frame plies and skin reinforcements
2. Prepare materials and details (including core details and inserts)
3. Setup fuselage winding mandrel, including winding aids
4. Hand lay up frame plies, wing attachment lug plies (including inserts) and baggage bulkhead flange plies
5. Position frame mandrels
6. Wind inner skin plies, hand locating skin reinforcement plies and inserts, and then cut skin plies to allow them to be pushed out
7. Place film adhesive and core details

8. Wind outer skin plies, hand locating skin reinforcement plies where required
9. Remove winding aids and then cut and trim skin plies at window openings and at ends to allow push-out to OML mold
10. Place mandrel and wound fuselage in OML mold and close mold
11. Seal part cavity with end plates, perform vacuum integrity check and then place in oven
12. Apply pressure inside winding mandrel to consolidate fuselage out against the mold
13. Heat to cure temperature and hold (2 hours)
14. Cool, remove end plates and winding mandrel and then remove fuselage from OML mold
15. Remove mandrels from hollow frames
16. Trim and inspect.

For production, many of the part fabrication tasks performed manually (cutting and kitting of plies for frames, skin reinforcement and other structural elements) would be automated and tool handling would be mechanized.

5 Results

Results to be addressed from TAA POC half-span wing and one-piece fuselage fabrication include (1) RTM and winding technology development results, (2) actual part weights and estimated production costs, (3) overall quality as judged by visual, geometric and dimensional inspections, and (4) aerodynamic efficiency as confirmed through flight testing.

5.1 Technology Development Results

As previously discussed, RTM technologies used for wing manufacture included the sealing of core against transfer resin ingress and co-curing of large integrated structure. Winding technologies used for fuselage manufacture included fiber spreading with in-process resin impregnation and direct winding on a fuselage shaped mandrel with push-out to a mold for consolidation.

5.1.1 RTM Technology Results

From process development subcomponent parts and completed half-span RTM wing assemblies it was found that the scrim supported film barrier was effective in preventing transfer resin from filling the core. Both honeycomb (see Fig. 9) and foam core examinations confirmed this result.

Fig. 9. Sectioned panel showing effective core sealing

Flatwise tensile testing of core-to-barrier and barrier-to-skin bonding confirmed failure in the core. Barrier failure did not occur even when core having strength twice that of the baseline core was used. Further, lap shear testing with barrier material between plies confirmed that skin-to-skin bond failure occurs before barrier-to-skin bond [7].

As previously explained, using a transfer resin for co-curing of large integrated structure requires the resin to have certain properties. Viscosity of the admixed two-component transfer resin was shown to have an 800cP viscosity at room temperature with at least two hours of transfer time making it ideal for fabrication of large assemblies having many integrated structural elements. Through testing, the transfer resin was shown to be chemically compatible with the preform resin as well as having the desired cure profile. Transfer resin toughness was tested and demonstrated to be equal to or greater than the resin used in the tackified tape and fabric preform materials.

5.1.2 Filament Winding Technology Results

Regarding fiber spreading for the broadgoods used for frame plies and skin reinforcements, a variation of ±7% was seen in 190g/m^2 fiber areal weight (FAW) for helical wound (±45° wind angle) broadgoods material with variation of ±3% calculated for 150g/m^2 FAW hoop wound broadgoods. As expected, FAW for the directly wound fuselage skins (nominally a ±45° wind angle) showed much greater variation (186 to 291g/m^2) as a result of fuselage circumference changes with results reflected in finished fuselage weights [8].

In-process resin impregnation of the fiber band was maintained at 40±5% resin content for helical wound broadgoods (a variation greater than expected) and 30±3% for hoop wound material (within target) with fuselage skin winding reflecting slightly greater variation than helical winding because of the fuselage shape.

Regarding the push-out winding mandrel, results revealed that the mandrel for this fuselage must be made undersized (at least 4mm/0.15inch per surface) to allow the OML mold to close around the wound part but not more than 8mm/0.30-inch to preclude fiber bridging. Further, the maximum allowed length of fiber than can be made to push out to the OML mold in the fuselage skin is a function of the fuselage shape, specifically corner radii. Further, fiber length may not span two fuselage radii or exceed 1/3 the circumference of the fuselage when wound at a 45° angle. An exception to this finding is evidenced in the fuselage tail cone area where fiber successfully traversed from BL 0.00 on the top of the fuselage to BL 0.00 on the bottom of the fuselage. To facilitate push-out to the OML mold, fuselage skin plies were cut and spliced.

5.2 Weight and Cost

The average weight of all finished half-span wings (four lefts and two rights) was within 3% of the calculated design weight. However, estimated production cost of the half-span wings exceeds the production cost target by a significant amount as a result of using the tackified carbon fiber fabric and the PMI foam. By studying the POC wing structural testing results, design improvements that include changes to the spars have been identified that reduce the use of the expensive PMI foam. Further, use of wound broadgoods in place of the expensive tackified carbon fiber fabric was identified as a potential significant cost reduction that would make the production cost target achievable.

Actual fuselage weights exceeded target design weight by approximately 20% with film adhesive for skin-to-core bonding and fuselage skin winding penalty being the major excess weight contributors. (Note: target design weight did not include film adhesive.) For production, the self-adhesive properties of the winding resin would be relied upon for skin-to-core bonding. Further, testing performed on fuselages indicate that lowering resin content would still yield acceptable quality and result in less weight. The estimated production cost of fuselages using the developed winding technologies was calculated to be significantly less than the production cost target almost entirely due to the use of low-cost fiber and resin and direct winding of the fuselage skins.

5.3 Overall Quality

Visual, geometric, and dimensional inspections were conducted on all wing and fuselage structures. In typical aerospace manner, in-process and final part visual inspections were used to document and correct deviations from known acceptable quality. (Note: proof load testing of the TAA POC wing and fuselage confirmed that the resulting structures were flight worthy.)

Geometric inspections of as-molded wing and fuselage shapes were conducted using portable laser measuring equipment. Wing airfoil results revealed that wing surfaces (upper, lower, leading and trailing edges) were smooth and within wing requirements for waviness with no steps in the contour. The only contour area exceeding requirements was the inboard trailing edge aft corner (see Fig. 10) which was subsequently brought within tolerance when the inboard rib was installed.

As expected, dimensional inspection of wing front and aft spars, inter-spar ribs and trailing edge ribs revealed that the internal tooling (D-section leading edge tool, inter-spar bladders and segmented trailing edge tools) did an excellent job of correctly locating these structural elements.

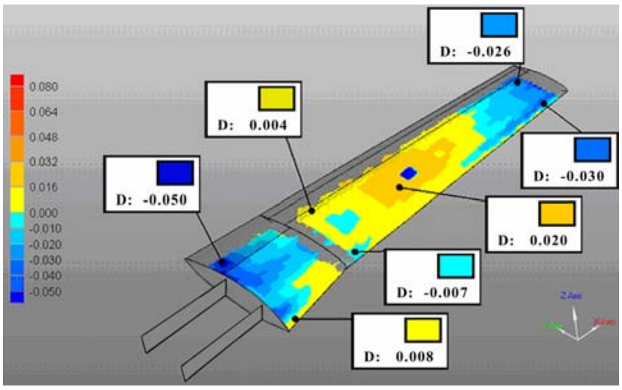

Fig. 10. Upper surface inspection data (inches)

Laser measurement of the fuselage shape confirmed that fuselages failed to meet contour requirements as a result of using the developmental OML mold which was subjected to both internal pressure and elevated temperature during fuselage cure. Resulting fuselages are rounder than designed, especially in cabin door areas and across their bottom. Fuselage bottoms are distorted a maximum of 3.3mm/0.13-inch with cabin door areas distorted a maximum of 6.9mm/0.27-inch - doors would not have fit the fuselage if they had not been co-cured. However, because the doors were co-cured, they met both step and gap targets. Regarding waviness and roughness, once initial winding issues were resolved, waviness targets were met but roughness still exceeded the target in some areas due to bridging between the mold surface and fuselage skin plies as a result of the winding mandrel being too small.

Regarding fuselage integrated structural elements, dimensional inspection of fuselage frames (see Fig. 11) revealed that the drawing shape requirements for the hollow frames were generally met, but that location control (e.g., around door openings) needs improvement as the frames were mislocated up to 8.6mm/0.34-inch. The wing attachment lugs (bottom of Fig. 11 just left of center) were also mislocated up to 5.0mm/0.20-inch forward and 3.8mm/0.15-inch low. Location control issues are the result of the winding mandrel being too small, both in length and circumference. This was confirmed using laser measurement of the push-out mandrel just prior to the start of winding.

Fig.11. Fuselage interior viewed through firewall opening

5.4 Aerodynamic Efficiency

To verify the extent of laminar flow achieved, flow visualization studies were conducted during test flights using chemical sublimation techniques successfully used by NASA in the past. The combination chosen for TAA flight testing used naphthalene with acetone as a carrier and was sprayed on the wing surface before takeoff. After in-flight sublimation the naphthalene becomes flat white.

For good visualization previous efforts required the wing be painted a dark color. However, elevated ambient temperatures dictated that dark paint not be applied to the TAA POC wing. Instead, blue food coloring was added to the solution before spraying. Sublimation of the chemicals during flight left a flat, light blue residue in the region of low energy (laminar flow) and the wing was scrubbed clean in the areas of higher energy (turbulent flow).

Fig. 12 shows that the wing attained 50 to 60% of chord laminar flow over the majority of the span (see superimposed red line). Also seen in the picture are disruptions in the flow (turbulent wedges) resulting from impurities in the spray process or in the surface finish which, because of their location on the forward area of the airfoil, were of sufficient height to trip laminar flow. The inboard third of the wing had diminishing amounts of laminar flow due to wing twist, that is, outboard washout to enhance low speed handling. The pictured test result was obtained at approximately 10 KTS slower than design cruise.

Fig. 12. A sublimation test result

In configurations such as the TAA aircraft, downstream propeller slipstream effects greatly restrict the ability to attain laminar flow on the fuselage. However, attached flow can be obtained if surface contours are smooth and undisturbed by gaps, overlaps, and steps. In-flight tuft testing was conducted over an extensive portion of the fuselage utilizing lengths of yarn taped approximately every 100mm/4-inches.

Fig. 13 shows the well attached flow over the majority of the fuselage indicated by the yarns flowing directly aft. This testing indicated that the majority of the fuselage was able to attain attached flow and showed no significant areas of separation except in the area of the wing-to-body fairing at high angles of attack as the aircraft neared stall.

Fig. 13. Air flow over the tufted fuselage surface

6 Discussion – Future Applications

Other applications for the wing (RTM) and fuselage (direct winding) technologies developed in support of TAA POC aircraft fabrication have been identified and include use of core sealing (for sealing and other reasons), fabrication of full-span GA wings or other large structures using room temperature transferable resin, filament wound broadgoods materials in place of commercial prepreg materials, and direct winding of other fuselages or convex objects of revolution.

6.1 RTM Technologies

Sealed honeycomb and foam cores in conjunction with resin transfer technology created for this integrated GA wing has application to other aerodynamic surfaces such as empennage (vertical and horizontal stabilizers) and control surface structures for GA as well as large commercial aircraft. Also, the technique used to seal core materials can be used to stabilize the shape of core details for other purposes and provide the added benefit of sealing them from moisture or other fluid ingress thereby preventing damage to core due to freezing.

The room temperature transferable resin developed enables even larger part fabrication than the TAA POC half-span wing. In conjunction with the ‘plastic bladders’ and the other internal tooling demonstrated, a full-span GA wing is no longer believed to be high-risk and business jet half-span wings are possible.

Use of developed winding technologies, specifically low resin content (tackified) wound broadgoods, in support of GA wing fabrication would singularly address the most negative aspect of the RTM technologies as used for the wing – the use of high cost tackified woven fabric material for the preforms.

6.2 Winding Technologies

Fabrication of a small GA aircraft fuselage as one-piece using the winding and push-out tooling technologies developed has been shown to be feasible and cost effective though not yet lightweight. The technologies of fiber spreading, in-process resin impregnation and push-out tooling generally have application to other fuselages as well as in broadgoods wound for specific or general applications.

With improvement to the push-out mandrel and OML mold tooling, application of direct winding would have immediate application for all composite GA aircraft fuselages. With the development of additional resins, application could easily be made to other fuselages (including large commercial fuselages) that requiring higher performance and to other lightweight aerospace products including commercial launch vehicles.

6.3 Production Cost Estimates

Production cost estimation studies performed after technology development confirm that the aggressive wing and fuselage cost targets established before development began can be met. By using the technologies synergistically for fabrication of GA wing and fuselage and other aircraft components the production cost of aircraft structure can be reduced significantly.

7 Conclusion

Co-curing of a one-piece half-span GA wing as highly integrated structure has been shown to be feasible and cost effective. Sealing of honeycomb and foam core used in conjunction with resin transfer was successfully demonstrated. The RTM process developed, especially the transfer resin and tooling, provides many advantages that enable other large integrated or complex parts to be co-cured. Identified potential improvements in the demonstrated wing design and manufacturing processes promise further cost reductions, especially for airfoil applications. Synergies between technologies developed for the wing and fuselage promise meeting the aggressive production cost targets for the wing.

The winding technologies demonstrated to fabricate this small GA aircraft fuselage, fiber spreading and in-process resin impregnation, are key to achieving low cost. Fabrication of the wound broadgoods and direct winding of the skins for the TAA POC fuselage demonstrated that winding can be a highly effective way of producing carbon-fiber/epoxy structure. The application of fiber spreading and in-process resin impregnation to current GA structure, especially aircraft fuselages, would have direct and immediate application and with concurrent cost savings. With development of additional resin systems, these key technologies would have application to other aerospace products including commercial aircraft and launch vehicles.

References

[1] Abbott Ira H, Von Doenhoff Albert E. *Theory of wing sections*. Dover edition, Dover Publications Inc, 1959.

[2] Barnwell R W, Hussaini, M Y. *Natural laminar flow and laminar flow control*. Springer-Verlag (New York), 1992.

[3] Holmes B, Obara C and Yip L. *Natural laminar flow experiments on modern airplane surfaces*. NASA Technical Paper 2256, June 1984.

[4] Holmes B, Obara C, Martin G and Domack C. *Manufacturing tolerances for natural laminar flow airframe surfaces*. Society of Automotive Engineers, 850863, April 1985.

[5] Payne Henry E, *Laminar flow rethink using composite structure*. Society of Automotive Engineers, 760473, April 1976.

[6] Alther G A. *A Significant Role for Composites in Energy Efficient Aircraft*. Technical Session of the 36th Annual Conference on Reinforced Plastics, Composite Institute and Society of Plastic Industries, Session 12-D, pp 1-4, February 1981.

[7] Anderson A, Simpson C and Ta'ala B. *Low cost manufacturing method for general aviation aircraft fuselage*. AIAA/ICAS International Air and Space Symposium and Exposition: The Next 100 Years, Dayton, Ohio, 2003-2766, 14-17 July 2003.

[8] Gardiner R, Maxwell M and Teufel P. *Low cost composite manufacturing method for a general aviation aircraft wing*. AIAA/ICAS International Air and Space Symposium and Exposition: The Next 100 Years, Dayton, Ohio, 2003-2768, 14-17 July 2003.