Composite materials, composite structures, composite systems

M. van Tooren
m.j.l.vantooren@tudelft.nl
Fokker Aerostructures, Papendrecht
The Netherlands

C. Kasapoglou and H. Bersee
c.kassapoglou@tudelft.nl
Faculty of Aerospace Engineering TU-Delft, Delft
The Netherlands

ABSTRACT

The first part of the history of composites in aerospace emphasised materials with high specific strength and stiffness. This was followed by a quest for reliable manufacturing techniques that guaranteed sufficiently high fibre volume fractions in complex structural parts with reasonable cost. Further improvements are still possible leading, ultimately to an extension of the functionality of composite structures to non-mechanical functions. Reduction of material scatter and a more probability-based design approach, improved material properties, higher post buckling factors, improved crashworthiness concepts and improved NDI techniques are some of the evolutionary measures that could improve the performance of current composite structures. Modular design, increased co-curing, hybrid material structures, hybrid fabrication methods, innovative structural concepts and reduced development times are more revolutionary steps that could bring today's solutions further. Manufacturing engineering is also important for achieving revolutionary change. Function integration such as embedded deicing, morphing,, and boundary-layer suction are among the next steps in weight and cost reduction, but now on the system level.

1.0 INTRODUCTION

The development of aircraft structures has been highly influenced by the advancements in structural materials. Where the first structures were compiled from, almost, natural materials like wood and steel, latter developments towards alloys, polymers and inorganic synthetic fibres changed the outlook considerably. First driving criteria were strength and stiffness, constrained by availability of both the material itself and of proper jointing techniques. Wood and steel were easily available as were nails, screws, bolts and nuts. High density could be handled by proper choice of half product like thin hollow tubes that allowed thin walls with relatively high compression strength (buckling). Using welding gave light weight cost-effective structures. The further developments in harvesting and modifying lightweight metals gave the possibility of load carrying skins for modestly big aircraft. Discrete stiffening and strengthening using stiffeners, frames and ribs allowed much more integral components (integral with respect to functions like mechanical strength, close-out of air, comfort). Adjustability based on a limited set of preforms was quite extensive. Further possibilities to tune but now also very locally the strength and stiffness of a structure came with the developments of artificial fibrous materials and polymer resins. These composite materials have changed the lower end of the general aviation and the high end military aircraft industry tremendously. Both are performance driven but in different fields and for both industries composite materials make a real difference even though the production techniques for these structures come with severe challenges. Only recently the civil aircraft industry is developing towards large scale integration of composite materials in their bare structure. This has everything to do with the improvement and cost decline of manufacturing equipment. The improvement in fibre placement techniques from hand-lay-up, through tape-laying towards fibre steering techniques beyond filament winding finally supports

Paper No. 3725. Manuscript received 12 May 2011, revised version received 30 June 2011, accepted 4 July 2011.
A version of this paper appeared in at the RAeS 2010 Structures and Materials Group conference, London, UK.

https://doi.org/10.1017/S0001924000006527 Published online by Cambridge University Press

the intrinsic design freedom and mechanical tunability of long fibre polymer composites. Parallel to the fibre reinforced polymer developments and their manufacturing techniques, important improvements in metal alloys took place and a first application of mixed fibre composites and sheet metal found its way into the airframe of the A380. All the ingredients are now available and partially proven (scale-up issues, reliability, durability, certification being some of the remaining tasks) to take up the challenges formulated by the aircraft designers: morphing structures, embedded flow control, and the more electric aircraft. These challenges require the step from composite structure to composite system. In this keynote address we would like to touch some of the aspects of these developments.

2.0 SHORT HISTORICAL OUTLINE OF COMPOSITES IN THE CIVIL AIRCRAFT INDUSTRY

In the aircraft industry the term composites is the term used for a range of structural materials that consist of at least two different constitutive materials with two different macroscopic physical appearances. In general one can say that a fibrous material is combined with a resin-like material. The fibrous material is often applied in flat pre-forms called plies. The second constituent, the resin-like matrix, is mostly a polymer, a 'plastic', although metal is also applied as matrix material. The terminology already suggests a strong link with the wooden era. The fibrous material being the plywood and the resin the adhesive with which the plies are glued together like with a resin. In fact, there is only a blurry border between composites and wood. In the 1930s The Fairchild and Haskelite companies jointly developed a bag-moulding technique for producing aircraft parts of phenol plywood, termed 'Duramold'. This material was a combination of natural wood and a synthetic resin and was applied in the Fairchild F-46, which completed its first flight on 5 December 1937. It showed excellent performance characteristics, but it failed to sell, probably due to its wood-like image. The F-46 fuselage consisted of two half-shells moulded on the same die and then assembled with casein glue. Due to the high buckling strength of Duramold, the fuselage required very few internal stiffeners, thus simplifying assembly. The Duramold fuselage clearly required just a fraction of the labour needed for a comparable all-metal fuselage, which could require up to six person-months to complete.

Glass fibres were conceived in the 1930s. Fibreglass reinforced plastics were first applied in aircraft during World War 2. A key application was for lightweight, electro magnetically transparent coverings for airborne radars. But fibreglass was also promising as a structural material, easily surpassing aluminium alloy in specific tensile strength, though far inferior to metals in stiffness. Researchers also discovered that glass fibres did not combine well with phenol resins, unlike cellulose fibres. By 1943, a new class of resins had become available, the polyesters, which bonded well with glass fibres and required very low moulding pressures. These new resins made it possible to develop aircraft structures using fibreglass plastics. In 1943 the Aircraft Laboratory at Wright Field designed and built a glass-fibre monocoque fuselage for the Vultee BT-15 training aircraft. This fuselage used sandwich construction, with thin sheets of glass-fibre-reinforced plastic bonded to a balsa-wood core. There were large objections against the use of balsa wood. Therefore research into synthetic substitutes like various honeycomb structures and plastic foams was initiated. Even with the low elastic modulus of the fibreglass plastic, the sandwich structure proved stiffer than the metal fuselage. Even in the 1930s, it had been clear that low stiffness posed the main barrier to widespread use of fibre-reinforced plastics in aircraft structures. Researchers worked with a variety of exotic materials, some highly toxic, like beryllium, others tremendously expensive, like boron. One of the most promising materials was carbon. By 1964 a commercially viable technique for producing high-stiffness carbon fibres with almost nine times the specific elastic modulus of steel had developed. By the mid-1980s, enthusiasts were proclaiming a new materials revolution, the shift from metal to composites.

The world's first all-composite powered aircraft (with non-natural fibres) to fly was the Learfan. Conceived by Bill Lear in 1954, work commenced upon it in 1977 and the first flight took place on 1 January 1981. The glass fibre fuselage was built of four sections, top, bottom and two sides. The wing was laid up as a one-piece box structure with removable tips, leading edges and trailing edges. The four carbon fibre tip-to-tip spars were formed as 'I' sections built of back-to-back 'C' sections. The whole wing was assembled in a vertical fixture. It was also highly innovative with respect to its side-by-side engines which drove the propeller through a gearbox. The drive shafts were of aluminium tube reinforced with carbon fibre to stiffen them and to adjust their critical resonance. Underfunding combined with technical problems in the certification phase terminated the project.

The Beech Starship twin P&W PT6A-67 turbine canard design was designed to fill the same niche in the market as the Lear Fan. It is a true 'twin' with the engines driving two pusher propellers. The variable geometry canard can be swept forward to a span of 7·3m for take-off and landing while in flight they may be swept back 30o to a span of 6·4m to reduce drag in the cruise condition. Originally the fuselage of the Starship was intended as a filament wound structure but this was abandoned. The 16·5m wing is a bonded box structure. The fuselage is a sandwich structure with carbon facings and a Nomex honeycomb core. The wing and fuselage are cured in a very large autoclave. The Starship went into production but was not a commercial success. Part of its technology has been used in the Raytheon Premier I, a business aircraft with a composite fuselage and a metal wing.

The introduction of composites has been very successful in the lower end of general aviation segment. Airframes for gliders are almost universally made of glass-fibre polymer composites. In the newer designs for the two and four seats segment, composite designs are also dominating the market. The very light jets market is not (yet) mature enough to draw any conclusions on dominant airframe technology in this segment. Airframes of regional jets are still mainly made from aluminium.

At the other end of the civil transport aircraft spectrum, Boeing and Airbus dominate the large transport aircraft segment, an evolutionary increase of the utilisation of composite materials has taken place. Taking Airbus as an example, in 1972 the A300 had the fin leading edge and various fairing panels made of glass fibre. From 1985 to 1987 usage was extended considerably with a major component, the fin box, being built largely from carbon fibre reinforced plastic (CFRP). Two years after the introduction of the composite fin box, 60,000 hours had been flown without damage. Five years after the introduction of the CFRP rudder in 1982 on the A310, 800,000 flying hours without damage has been recorded. Today all Airbus aircraft, including the A380, use this fin. In 1988 the horizontal tail plane followed in composite materials. In the A340, also the pressure bulkhead and the keel beams were developed in composites. For the A380, the evolution was continued by introducing a composite centre wing box. Interesting development is the use of thermoplastic based composites in some secondary parts. This allows the use of physical processes in the assembly, namely welding. This reduces the need for secondary bonding steps and therefore creates opportunities for cost savings. A revolutionary step is made by Boeing by developing the all-composite 787, so including composite wings and fuselage. It goes without saying that such a revolution is creating a large range of technical challenges.

In the 1980s two interesting new 'materials' emerged. Both were laminates made of alternating aluminium alloy sheets and fibre reinforced polymer prepregs. Both try to combine the durability,

https://doi.org/10.1017/S0001924000006527 Published online by Cambridge University Press

reparability and ease of fabrication of aluminium alloys with the high specific strength and fatigue resistance of composite materials. The first material, Arall, combines aluminium with aramid fibre reinforced epoxy. Aramid has poor compression characteristics, limiting the use of Arall to loading schemes with a positive R-value. To overcome this problem a second material, Glare, was developed using glass fibre and a toughened epoxy instead of the aramid/epoxy. The first large scale application of Glare is found in the top fuselage shells of the A380. The incompatibility between forming principles of composites and metal make it difficult to create parts with high and or/complex curvatures.

In the business jet aircraft segment a mix of materials and structural types is implemented. No clear winner can be indicated in this segment although one can say that at the higher end of this market the percentage of composite in the airframe is substantially higher than at the lower end.

Composites, including fibre metal laminates, still have a long way to go before dethroning light alloys as the principal material for aircraft structures. The main difference with the earlier substitution of wood by metal is that a large range of optimised aircraft for different speed, range and payload segments has evolved. It is very unlikely that a one size fits all replacement of airframe technology will improve these different optima. In the recent years a renewed interest in flow control has developed related to climate change concerns. Also the drive for more electric aircraft comes from these concerns. Both developments require a redefinition of the functionalities of aircraft structure.

Composites can offer fatigue free, corrosion free, aerodynamically smooth solutions. They can also be part of more complex systems that answer the required extended functionalities. Whether that is useful and possible for a specific segment should be the result of a very thorough and critical analysis. In the next sections we will investigate this issue further.

3.0 SHIFTING THE BALANCE

There is no and most likely never will be a single combination of manufacturing process, material, and design concept that is a clear winner with respect to significant improvements in aircraft performance. First of all one would have to define what significant improvement means. A more traditional performance based definition could be 'a more than 15% improvement over a certain baseline in range or payload or fuel carried'. That is, if 787 and/or A350 are used as the baselines, the clear winner would give 15% range increase (for the same number of passengers) or payload increase (for same range). This kind of 15% improvement from a single concept could be termed revolutionary change. A more life-cycle based definition could be: a 15% reduction in DOC coming from reduction in fuel consumption, maintenance cost, crew cost or any other cost item felt by the operator.

There is a series of evolutionary changes that when put together, could give such an improvement and probably more. Some of them are here already but not implemented to the fullest and some are around the corner and we have not defined them very well yet. Here are some of them starting from the more 'exotic' and moving to the more realistic. They can be viewed as challenges to the industry and material suppliers.

3.1 Evolutionary change makers

3.1.1 Reduction of material scatter and more use of reliability-based design

Today's composites have some properties (e.g. transverse compression strength or shear strength) that have coefficient of variations (CVs) higher than 10%. This means that the B and A Basis values that must be used in design are at least 20% lower (for B-Basis and more so for A-Basis) than the mean strength. There is a huge potential for weight reductions if the scatter around the mean was decreased. Hand-in-hand with this is a reliability-based design. Today the ratio of ultimate to limit load is 1·5. A factor, that historically, has served us well. But this factor corresponds to different reliability values for different properties or loading cases. While, historically, the purpose of this factor has been (a) to protect against unknowns (for example a failure mode we had not anticipated) (b) to cover inaccuracies in determination of internal loads (c) to cover against inadequate analysis methods, today, with the advent of computational methods and our understanding of material performance under all kinds of conditions, we have to be able to say that we can do a better job against the three items mentioned, that the 1·5 factor was meant to protect us from. Instead we should decide on a reliability value (0·99 or 0·999?) and use that. If we do that, then, depending on the property and failure mode, there will be a different factor we should use which would be in most cases less than 1·5 and in some cases, maybe higher than 1·5. The net result is a weight reduction.

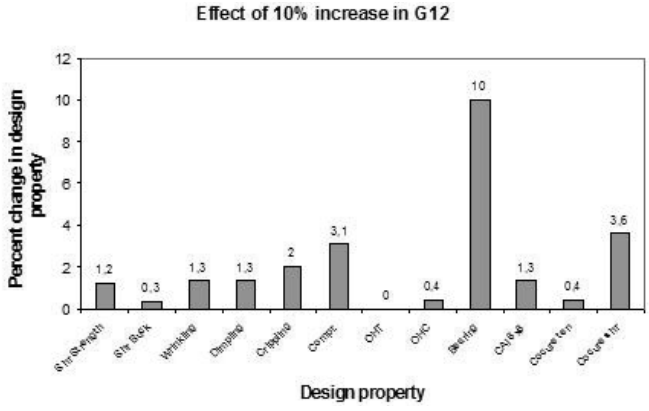


Figure 1. Effect of material properties on structural performance.

3.1.2 Improved material properties

Not much work has been done in seriously relating some basic material properties to actual design values being used. We tend to gravitate towards high strain to failure and forget about other properties that may be even more important. For example, Fig. 1 shows how a 10% increase in the basic ply shear stiffness G12 affects the majority of design properties used in structural design: Shear strength, shear buckling load, sandwich wrinkling, sandwich dimpling, flange or web crippling, compression strength, open hole tension and compression strengths, bearing strength, compression after impact, co-cured (out-of-plane) tension strength, co-cured (transverse shear) strength.

Similar trends are observed for other properties, but not as drastic, if there is a 10% increase in compression strength of the basic ply, or the energy release rates in modes I and II.

3.1.3 Use of higher post-buckling factors

Getting post-buckling factors higher than two without running into all kinds of problems under static and (even more so) fatigue loading is very difficult and rarely done. Yet, going to a post-buckling factor of five reduces the weight of the panel more than 20%. Interestingly, going to a post-buckling factor of ten 'buys' us only another 5-10% (diminishing returns). The skin-stiffener separation mode is still viewed as a show stopper. Good analysis methods at high post-buckling factors are still not there yet. And, worst of all, understanding how a post-buckled structure performs under fatigue loading, how damage is created and grows is still very much an open issue.

https://doi.org/10.1017/S0001924000006527 Published online by Cambridge University Press

3.1.4 Improved crashworthiness concepts

A significant percentage of a fuselage is designed by crash conditions. Unlike metals which absorb energy by yielding and successive formation of plastic hinges, composites absorb energy through judicious sequencing of local failure mechanisms. However, the methods used today for composite fuselages are rather crude and very conservative. The main reason is the complexity of the crash absorbing mechanisms that can be built into a composite structure. These range from controlled and sequential material failure starting with matrix cracks, continuing on with and local fibre failures and creation of delaminations — the latter growing and/or branching out to other ply interfaces — to local structural failure and load redistribution to adjacent structure. Each of the failure mechanisms absorbs a certain amount of energy, with delamination growth absorbing the largest. In addition, allowing the structure to undergo large deflections before failure, in particular significant hole elongation in bolted joints, provides another mechanism for energy absorption. Each of these failure mechanisms is currently a topic of research and must be understood well, in particular under dynamic loading (very rapid load introduction). We still don't understand dynamic effects well. Analysis of full-scale crash tests in helicopters have shown that the methods currently used overestimate the local forces created and underestimate the deflections before collapse of the cabin. If we understood these phenomena better we could design a controlled failure sequence (with proper use of triggering mechanisms to force the failure mechanism desired during each stage of the crash) that would eliminate some of the conservatism in today's approaches and reduce weight. In a simplified scenario that on-going research has shown to be quite promising, delaminations are triggered in laminates under in-plane compression and they start to grow with the delaminating layers bending away from the laminate plane. As the delaminations grow, the delaminating layers undergo large deflections, which, eventually, cause failure of the delaminating members. The latter 'fly-off' absorbing a lot of energy in the form of kinetic energy. By properly designing the stacking sequence and triggering mechanism, new delaminations will continuously form before each piece of delaminating member breaks off, thereby creating a self-repeating mechanism that can absorb significant amounts of energy (as much as 100kJ/kg).

3.1.5 Improved NDI techniques

Some of the structure is designed by the minimum flaw size (e.g. 6mm delamination) that the inspection techniques available in the field during service are able to find reliably and consistently. Even more so, impact damage is based on BVID (barely visible impact damage) which condemns most of the structure to carry a lot of extra weight. This is a choice of convenience to keep the cost of inspections down by doing visual inspections. However, if the costs of NDI techniques such as x-ray scattering or laser interferometry can be reduced sufficiently, the BVID requirement could be changed to something like 0·8BVID. This means that reliable, easy to use inspection methods can be used to detect damage that is 80% of that corresponding to BVID. This in turn would lead to, at least 10% weight reductions for parts designed by BVID today. 'Better' methods (better here may only mean cheap enough to be able to use in the field) will allow us to open up some of the design criteria. Another issue is the lack of a reliable method that can find 'kissing' bonds. These are bondlines where there are no voids or apparent contamination, yet, the surface preparation and adhesive formulation lead to a large portion of the bond being inoperative. Currently, to work around that, the designer must show that even if 50% of the bond is lost the structure can meet limit load. A better NDI technique could reduce the weight and make use of adhesives more prevalent leading to further weight reductions by elimination of fasteners and fabrication cost reductions by reducing assembly time.

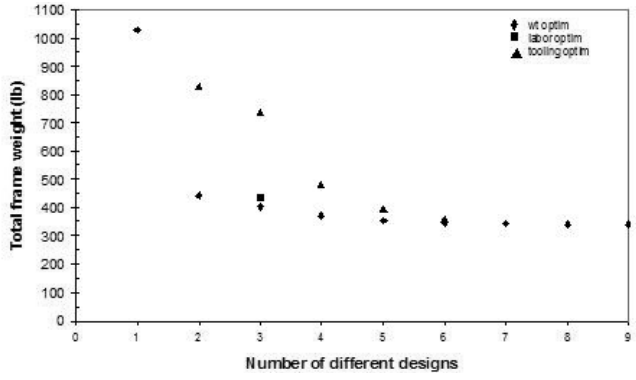


Figure 2. Effect of structural compromises on weight.

3.2 Contributors to revolutionary change

In addition to the evolutionary changes discussed in the previous section, there are some ideas that may contribute towards revolutionary changes. The ideas themselves are not new but viewed from a fresh perspective they can be quite promising:

3.2.1 Modular design

Optimising for weight means that each part is different than almost any other part that performs the same function at a different location in the aircraft. The associated recurring fabrication cost and non-recurring tooling cost are astronomical. If we are willing to accept a small weight increase, we can find 'optimum compromises' where we minimise the number of different parts thus minimising the number of tools and benefiting more from learning curve effects (within the same aircraft making 15 identical parts means that the 15th part has much lower cost than the first; to get the same kind of improvement in cost, in today's one of a kind design mentality we would have to wait until aircraft 15.). An example applied on fuselage frames can be seen in Fig. 2. A 9% increase in weight reduces the number of different designs from nine to four and the associated tooling cost by 40-50%. Similar improvements can be obtained with stringers, skins, or, even, entire fuselage barrel sections. The reduced number of modules in each case means that some of them are over-designed because they are used in other locations in the aircraft which are more highly loaded. But optimisation can help us pick the optimum compromise in terms of minimum weight increase and maximum decrease of tooling cost and recurring labour for fabrication. It is recognised here that minimising the weight is paramount in achieving performance objectives for a vehicle and such a weight increase might be unacceptable in some programs. Still, the specific priorities of a manufacturer at any given time, might favour decreased non-recurring cost if development funds are hard to come by. In a not so far-fetched scenario, one could have 'just in time tailored' aircraft where based on anticipated number of passengers, modules corresponding to the different barrel sections are assembled on the fly within two to three days to create the aircraft tailored for the route selected. Of course, this requires very efficient reliable assembly methods for the different barrels but it does not hurt to dream. The modular design will also reduce repair and maintenance cost since 'remove and replace' will be routine and easy because most parts will be replaceable. A part that needs repair is removed and replaced so the aircraft can fly again. Then, off-line, the part that needs repair can be fixed and used in another aircraft at a future time.

3.2.2 Increased co-curing

One of the main reasons composites have not achieved their potential is that we do less co-curing than we could/should thus

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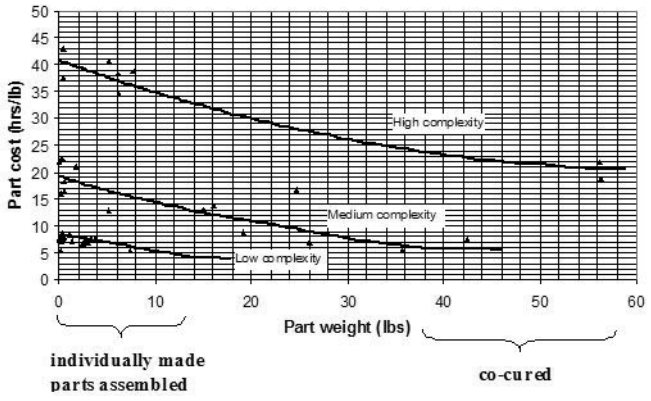


Figure 3. The effect of co-curing on part cost.

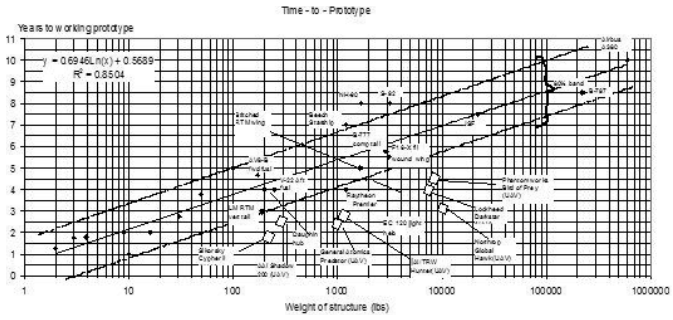


Figure 4. Relation between structural weight and development time.

increasing the cost. There is a risk with large co-cures. If something goes wrong, the repair or scrap cost can be prohibitive. But there is a proportionally large reward. Figure 3 shows the effect of co-curing. As the part size increases the cost per weight of the total structure decreases drastically.

3.2.3 Hybrid fabrication methods

The advent of automated fabrication methods (e.g. fibre placement) has helped reduce the cost of composites but, in general, they are still more expensive than current metal parts (this is a bit unfair to composites because metals are way down on a learning curve that started in the 40s while some composite processes are just starting so we are at unit 1; however, the airline does not care about that; if they can buy a metal aircraft at 70-80% of the composite cost, they probably won't opt for the weight savings of the composite). In addition, there are specialised fabrication methods (pultrusion, press-moulding) that can make specialised parts (stringers, fittings) at very low cost, in some cases even lower than metals (e.g. press-moulded composite fittings versus machined metal fittings). Combining more than one fabrication methods in one large (co-cured?) part would combine the advantages of the individual processes and generate a much lower cost part.

3.2.4 Innovative concepts

Many innovative concepts are killed during preliminary design because the comfort factor is low. Companies are unwilling to take large risks because of funding problems and time constraints. Yet, some innovative concepts (e.g. giant braided fuselage barrels, three-dimensional performs with RTM for increased out-of-plane strength, composite fittings, etc.) could help bring the next revolution in transport aircraft.

The same is true for fibre steering. The clear advantage in design freedom and optimisation potential is challenged by the unconventional approach needed for material qualification and proof of structural strength. The technique is demonstrated successfully on the Advanced Composite Cargo Aircraft, or ACCA.

Besides varying fibre direction from point to point, one could also think how to have the best resin system at each location. Currently no mixing of different resin systems in a single composite structure is done. This could however have significant advantages. The application of thermoset based composites and thermoplastic based composites in one part could help to reduce cost in manufacturing and assembly. Thermoplastics enable more physics based assembly methods like resistance, induction and laser welding. However, the base material is more expensive than baseline epoxy. Combining skins made from thermosets with locally placed thermoplastic sheets that allow for welded connections to items like frames, stiffeners and ribs could combine advantages of both worlds. Adding to this selective reinforcement using metal sheets would give the designer a large range in design options.

3.2.5 Reduced development times

From design inception to first prototype, it takes a very long time. This ties in investments in factory floors (tooling, fabrication equipment, etc.) and stifles innovation. Figure 4 shows that for every ten-fold increase in structural weight, the time to prototype increases by approximately two years. For large aircraft like the 787 or the A380 this translates to seven to nine years total time, which is too long. Development times must be cut down. This means reduced certification costs (for example by more certification by analysis), reduced material qualification times etc. Granted, some of the delays are unrelated to the technologies themselves as strategic decisions to launch now or later and staffing profiles can have a significant effect but there is a lot of room for improvement even with these external factors in place. It is also interesting that very different products (a small cover versus an entire wing for example) very different missions (military or civilian), very different funding sources (R&D versus military customer), very different material forms (RTM, RTM stitched, hand laid up, fibre placed, etc.) all fall, on the same curve. For the design phase considerable improvement can be obtained with technology like knowledge based engineering and multi-disciplinary design optimisation. Encouraging results are obtained by several research and industrial parties.

4.0 MANUFACTURING THE NEXT GENERATION OF COMPOSITE STRUCTURES AND SYSTEMS

The major draw back of composites is their higher manufacturing and material costs compared with metallic structures. Consequently, much of the research on composites is focused on the development of affordable composite materials and manufacturing techniques which don't require an expensive autoclave. Another focus is on the development of manufacturing techniques which can be automated as much as possible, e.g. automatic tape laying and robotic fibre placement or can produce integrated structures thus reducing the assembly costs like resin transfer moulding and vacuum infusion.

4.1 Automation

The standard composite manufacturing technique in the aerospace industry is still prepreg hand lay-up followed by autoclave curing. Although the prepreg hand lay-up process results in high quality parts, it is very labour intensive. It was found that irregular placement of the tape could cause inconsistency in the quality of the cured laminates. To improve the quality and it's consistency but

https://doi.org/10.1017/S0001924000006527 Published online by Cambridge University Press

most of all to reduce the manufacturing costs, aerospace companies began to develop tape-laying machines. The current tape laying machines consist of a floor mounted gantry system with two side rails. A cross feed bar containing the end effectors or delivery head moves over these side rails. Rolls of tapes with a width from 3 to 12″ can be laid down. The width of the tapes depends on the curvature of the product. For flat tape laying 6 or 12″ rolls are being used to maximise the lay-up rate, while for contoured tape laying 3 or 6″ rolls are being used.

The labour savings achieved with tape-laying machines ranges up to 86% over hand lay-up. Tape laying also offers a reduction in waste which is in case of carbon fibre prepregs a significant cost factor. With hand lay-up, waste occasionally occurs at the end of each strip from the same side of the lay-up, which causes waste at the end of every tape cut, with the exception of 90° cuts. Most tape-laying machines, however, turn around 180° at the end of a strip and lay the next strip adjacent to the previous one (parallel to it) but in the opposite direction. Because the two strip ends on one edge of the ply have the same angle, there is no waste of material with this procedure.

However, the tape laying process is limited in the complexity of the parts to be laid up. Due to the width and the stiffness of the tapes, only slightly (single) curved parts can be made. In case of small lay-up radii, the tape will buckle and can not be laid onto the tool. Typical parts being produced by tape laying are the skins of the horizontal and vertical tail plane as well as the skins of wings.

In order to be able to produce more complex parts in an automated way, in the 1980s the fibre placement machines were being developed. Fibre placement combines the advantages of high precision fibre lay-up and differential tow payout capability of filament winding with the cut-restart and layer compaction capabilities of tape laying. The individual tows are fed through the delivery head or end effector and pressed onto the tool surface. Multiple tows, even up to 32, are placed during a course. Every single tow can be dispensed at its own speed and cut independently, thus allowing each tow to conform independently to the surface. Consequently, the fibres are not restricted to geodesic paths like in filament winding. They can be steered to meet specified design goals.

The materials used in fibre placement are fibres which are fully impregnated. The fibres are formed into tows of a width ranging from 3 to 4·6mm. Slit tapes are also used and are fabricated by running a 3″ wide tape through a slitter, creating smaller width tapes. Advanced fibre placement delivery heads contain hot air heating and cooling capabilities. Heating is used to increase the tack for the tow-to-tow adhesion and improve the compaction during lay-up. The cooling is to decrease the tack during cutting, clamping and restarting.

Similar to tape laying, in case of thermoplastic composites a heater, gas torch, is added to the delivery head. This heater melts the thermoplastic resin which enables it to adhere to some extent to the previous prepreg layer. However, since the temperature involved in thermoplastic composite processing are significantly higher than with thermoset composites, issues like the life time of the heating system as well as the compaction of the plies are not solved yet and subsequently are a topic of further research.

These unique capabilities of fibre placement resulted in a rapid implementation within many aerospace companies to fabricate a variety of composite parts. In a relatively short period of time, fibre placement has been developed into a widely accepted manufacturing process for affordable composite aerospace components. As new aircraft and aerospace programs are introduced, fibre placement will be used to fabricate many more composite parts to gain the benefits of reduced cost, improved quality, and improved performance.

4.2 Parts integration

Although tape laying and fibre placement both result in significant labour cost reductions they still require an autoclave for the curing of the prepregs. Autoclaves are a big cost driver due to the investment and operation costs, long cycle times due to inefficient heating and also because it is a batch process. Manufacturing techniques are being developed to manufacture composite parts without using an autoclave; one of these techniques is a family of processes called resin infusion or liquid composite moulding.

All resin infusion processes are based on the same principle. A thermoset resin is infused into a dry fibre lay-up or preform which has been placed inside a mould. Subsequently, the package is cured in an oven. Three main types of liquid moulding processes can be distinguished, namely Resin Transfer Moulding (RTM), Vacuum Assisted Resin Transfer Moulding (VARTM sometimes called Vacuum Infusion) and Resin Film Infusion (RFI). In RTM, two metal mould halves are being used and the resin is infused into the preform by using high pressures up to several bars. VARTM on the other hand only uses vacuum to drive the resin through the preform. The mould is closed by using a vacuum bag, a one sided mould is thus used. In RFI films of b-staged cured resin are placed on top and/or bottom of the preform. In an oven heat is applied to push the resin through the preform.

Another advantage of resin infusion is the potential to produce integrated parts. Preforms can be a simple flat lay-up or a complex 3D structure in which skin and stiffening elements are stitched or bonded together into 1 complex preform. After infusion and subsequent curing of the preform, a complete structure is manufactured. In this way the process step requiring assembly of stiffening elements onto a skin has been skipped and thus significant manufacturing cost reductions can be obtained.

4.3 Out-of-the-autoclave materials

Besides applying heat, the autoclave is necessary for applying pressure which is needed to remove the voids from between the layers and from within the layers. Extreme low void contents (<2%) are requested by the regulations, therefore, high pressures (up to six bars) are needed for the existing prepregs. Currently, material suppliers for example the Advanced Composite Group are developing thermoset resin prepregs which don't need a high pressure for removing voids. Secondly, the trend is to lower the curing temperature while keeping the *Tg* high thus reducing the cycle times and thus the costs.

A family of non-autoclave materials receiving a lot of attention in the past 15 years are the thermoplastic composites. The interest in thermoplastic composites is growing due to the development of better engineering thermoplastics and the availability of continuous fibre thermoplastic composites. Composites made with engineering thermoplastic matrices, like e.g. PEI (Polyethylenimine), PPS (Polyphenylene Sulfide), PEEK (Polyetheretherketone), have similar basic mechanical properties, such as strength and stiffness, as thermoset matrix composites. However, they have some advantages compared to thermosets that make them very interesting for the use in aircraft structures. These advantages are:

1. Potential of low cost processing
   - Indefinite shelf life
   - Fast processing cycle
   - Re-processable to correct flaws

2. Improved secondary properties
   - Excellent fire-smoke-toxicity characteristics
   - Non-flammability
   - Decreased moisture absorption
   - Improved resistance to aviation fluids
   - High temperature resistance
   - High toughness

https://doi.org/10.1017/S0001924000006527 Published online by Cambridge University Press

A major design requirement for aircraft structures is the ability of the structure to fulfil all its requirements after impact damage, particularly at non-visible or barely-visible levels. Composites sustain considerable impact damage with little or no visible external evidence. However, there may be internal delamination, which reduces strength and possibly leads to damage growth with repeated service loads. In the past the impact damage requirement has severely limited the design strain levels and/or required highly tailored designs. Tougher materials with significantly improved resistance to impact damage and delamination enable higher design strains and avoid the use of penalising factors to account for reduced structural properties resulting from damage or unforeseen loads.

Another severe requirement, which is of increasing importance for aircraft manufacturers, is the flammability and toxic emission of composite structures. Most of the engineering thermoplastic matrices are non-flammable and are virtually free from toxic emission when subjected to an external source of flame. Tests at the Ohio State University (OSU) on the BASF's Ultrason prepregs (PES, PSU) have demonstrated heat release rates which are well below the limits for aircraft interiors which have been in effective since August 1990.

Cost effectiveness is increasingly important in the aerospace industry. The potential of low cost manufacturing of thermoplastic composites is the main reason for the growing interest in these materials in aerospace industry. The main difference between thermoplastics and thermosets is their chemical structure. Thermosetting polymers are chemically reactive and solidification (curing) takes place by cross-linking. They can, therefore, be characterised as 'crosslinkable' polymers. Thermoplastics, however, require only temperature and pressure to process the different layers into a laminate and they can be characterised as 'linear' polymers. The consolidated thermoplastic composites, therefore, can be reshaped by melting and cooling, in contrast with the cured thermoset composites which can not be reshaped.

Curing of thermoset composites requires a significantly longer and more difficult cycle compared to consolidating thermoplastic composites. The complicated curing cycles of thermosets can cause errors, which require higher quality maintenance costs. With thermoplastic composites, the 'resin chemistry' has already been performed within very close tolerances by the material supplier. The thermoplastic composite is, at the end of the processing cycle, chemically the same as at the beginning. This leads to a better reproducibility of the process compared with the processing of thermosets and thus to significant advantages in quality assurance. Errors made during the curing cycle of thermosets also result in a large amount of scrap through rejected parts, due to the non-reformability of the cured composite. The cross-linkable resin of the thermoset prepreg requires refrigerated storage, while the thermoplastic prepregs can be stored in any ambient environment. Moreover, history records of storage times of thermoset prepregs have to be maintained and monitored in relation to specifications allowables. Material which is 'life expired' may be given a life extension after being subjected to repeated physical and mechanical testing in accordance with the release test requirements of the material specification. Thermoset prepregs expiring the storage date are another contribution to the scrap rate of the cured part rejection.

The current most used manufacturing process for thermoset composites parts in the aerospace industry is hand lay-up of pre-impregnated material. Hand lay-up is one of the oldest techniques for making composite parts. It is a labour-intensive process, which makes it human error-prone, but relatively simple to perform. During the process, the individual lamina (ply) is cut by hand to the required shape from a roll of material and subsequently built ply-by-ply into a laminate. The plies are laminated on the tool, after which a vacuum bagging procedure is followed in order to cure the part. The processing cycle is in the order of hours, which lead to high labour costs.

The thermoplastic composite parts, on the other hand, can be made by using rapid transforming processes with processing cycles in the order of minutes, which results in a significant reduction in labour costs. The thermoplastic forming processes are called thermoforming and are derived from metal forming technology. The laminate is heated to the necessary processing temperature by infrared heaters, autoclave or hot platen press. At this temperature it is pressed into a mould by mechanical means or by applying a hydrostatic pressure. After forming, the laminate is cooled under pressure to a temperature below the glass transition temperature (*Tg*) after which it is removed from the mould. The most used thermoforming process is rubber forming, a stamping process in which one of the mould halves is replaced by a silicon one. Thermoplastic prepregs also lack the tack of the thermoset prepregs and therefore, do not require the stringent, clean room conditions that are necessary for the manufacture of high quality thermoset laminates.

The melt processability of thermoplastic composites also enables novel welding techniques which offer improved manufacturing cost efficiencies through very fast and robust joining. Welding is a well established joining method within the plastics industry that takes advantage of the specific property of the thermoplastic resins of flowing when heated above their glass transition temperature, Tg (amorphous polymers), or their melting temperature, *Tm* (semi-crystalline polymers). Also known as fusion bonding, it can be generally described as joining two parts by fusing their contact interfaces, followed by cooling under pressure, which enables the bond to be made and the material to recover its initial properties.

Fusion bonding has a number of advantages relative to the traditional joining techniques. It has the potential of overcoming or eliminating the typical drawbacks of mechanical fastening, such as (1) the introduction of stress concentrations in the material, (2) the possibility of causing delaminations during drilling, (3) thermal stresses due to a different thermal expansion coefficient of the fasteners relative to the composite, (4) water intrusion into the joints, (5) possible galvanic corrosion, (6) weight increase and (7) extensive labour and time requirements. It also avoids some of the difficulties related to adhesive bonding, namely the extensive surface preparation (difficult to achieve and control in an industrial environment), the use of adhesives with long curing cycles and the complications that may arise due to the chemical inertness of some thermoplastic resins.

Fusion bonding techniques are usually classified by the way of applying heat to the welding interface. From the variety of those traditionally used in the plastics industry, two are considered as the most promising for joining continuous fibre reinforced thermoplastic composites:

1. Resistance welding
2. Induction welding

It has been already proved that these three welding techniques provide high quality welds in thermoplastic composites. Nevertheless, each one has its own characteristics and limitations, so that none of them could be selected as the best solution to all kinds of materials and structures. It is believed, though, that they combine well to cover a vast range of applications.

Resistance welding is a relatively simple technique, generally limited to producing straight weld lines. It takes advantage of the heat generated by an electrical current flowing through a resistive insert placed at the welding interface. The most commonly used resistive implants are metallic meshes, which exhibit a highly reliable and repeatable behaviour. Special care has to be taken when conductive materials, i.e. carbon reinforced thermoplastics, are to be welded in order to avoid current leakage from the resistive insert to the substrates. One of the advantages of this welding technique is that, as the metallic mesh remains trapped in the joint, defective welds can be reprocessed. Likewise, the simplicity of the required equipment makes it possible to perform low-cost welding processes. Resistance welding has been used to assemble the ribs to the skin of the inboard and outboard wing leading edges of the A340-500/600 and the A380, respectively.

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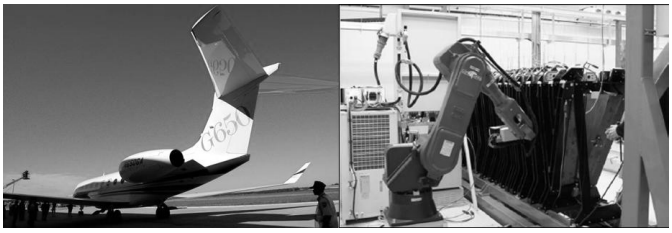


Figure 5. Welded rudder and elevator of the G650.

Induction welding is a quite flexible and easy-to-automate process that allows joining parts of complex geometries. In this welding technique heating at the welding interface is caused by an alternating magnetic field (Fig. 19). Since different induction heating mechanisms, i.e. hysteresis loses and eddy currents, can be activated depending on the nature of the materials involved in the process, this welding technique offers a set of different solutions. Non-conductive materials (e.g. glass fibre reinforced thermoplastics) can be induction welded either using ferromagnetic particles embedded in a resin layer or a metallic mesh, both located at the welding interface. On the other hand, conductive materials (e.g. carbon reinforced thermoplastics) can be induction welded either by placing a metallic mesh at the interface or by using the reinforcing fibres as the heaters. This latter solution, in which no foreign material is placed at the welding interface, means that not only does heating occur at the interface but also through the whole composite thickness. A robotic induction welder for thermoplastic composites has already been developed by Kok and van Engelen (The Netherlands).

Both processes are commercially being applied by Fokker Aerostructures The Netherlands for the outboard leading edge of the A380 (resistance welding) and the elevators and rudder of the Gulfstream G650 (Induction welding), Fig. 5.

Although existing for many years (APC-2), the implementation of thermoplastic composites into aircraft structures has been limited. However, in the past eight years there has been an increasing interest and acceptance of these materials. This resulted in a switch from usage in small parts like brackets and covers, to large secondary structures like the inboard fixed leading edge of the A340-500/600 and outboard fixed leading edge of the A380. Besides the size of the components also the weight of thermoplastic composites used in aircraft structures increased significantly, from 1 ton in the A340-500/600 to 6 tons in the A380.

However, the great breakthrough of thermoplastic composites and thus manufacturing cost efficiency should come from the implementation of these materials and their associated processes in primary aircraft structures like fuselage, wings and empennage. To achieve this two steps have to be made namely development of *in situ* consolidation and affordable thermoplastic composites. For manufacturing of the skins of these primary structures tape laying or fibre placement processes need to be used. Thermoplastic composites lack the tackiness of the thermoset composites. The tackiness of the thermoset prepregs facilitates the lay-up process especially also with respect to lay-up of the first ply. Thermoplastic composites have to be heated to soften the resin before they can be placed. The state of the art is that gas torches or lasers are used to heat the material and after lay-up the whole package is consolidated in an autoclave.

To utilise the full cost efficiency potential of thermoplastic composites this should be a one step process. So lay-up and consolidation during fibre placement or tape laying. The challenge is to heat up and cool down the material as fast as possible without introducing too high residual stresses. For this novel heating systems and a profound knowledge of the formation of residual stresses need to be developed.

The thermoplastic composites can be categorised as illustrated in the figure below At the bottom are the composites based on commodity plastics like PP. These materials have a relatively low processing temperature (200-220°C) and a very favourable cost

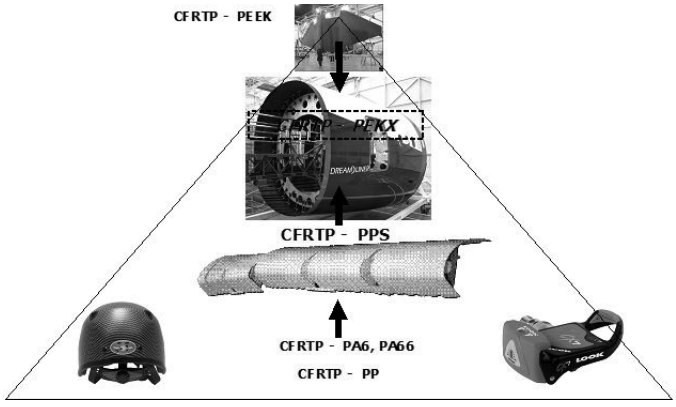


Figure 6. Categorisation of thermoplastics according to their field of application.

level. However, the properties are relatively poor (for aerospace standards!). At the top end are the composites based on the high performance plastics like PEEK. Excellent performance but unfortunately also very expensive and having a high processing temperature of 400°C resulting in even higher costs.

In the middle is the material which is currently used in the A380 and most other aircraft thermoplastic composite application. For the current applications the material functions very well, however, for the larger primary aircraft structures the properties are not good enough. A resin is needed with a higher toughness and temperature resistance but a low cost level and processing temperature are also required in other words PEEK properties for PPS processing and costs. The main thermoplastic composite suppliers Cytec and Ten Cate Advanced Composites are trying to meet this challenge by developing a PEKK (Polyetherketoneketone) composite. This material could be situated in between PPS and PEEK.

If these two challenges are met there is a potential for large cost reduction which would enable the usage of thermoplastic composites on the next single aisle aircraft of either Boeing and/or Airbus. Especially the high production series of 40 aircraft per months of single aisles will favour the automated processes of thermoplastic composites.

5.0 COMPOSITE SYSTEMS

There are many studies ongoing into developing a next generation of aerostructures with a next level of functionality. Examples are smart high lift devices and integrally de-iced leading edges and moveable surfaces. An example of the former is the eccentric beam actuator (EBA). An EBA in its most generic form consists of a rotating beam, whereby the axis of rotation does not coincide in part or in full with the central axis of the beam. By rotating such a beam, a shape change of the structure surrounding it can be achieved. EBA's have been successfully applied in the DARPA Smart Wing Project and considered as a candidate technology for future high-lift applications in the context of the EU FP7 project SADE (SmArt High-Lift DEvices for Next Generation Wings). Specifically, the EBA that is considered in SADE is to be employed as a combined actuator and support structure of a flexible skin material in either the wing's leading- or trailing-edge sections. This flexible skin is initially considered to have a relatively low out-of-plane bending stiffness, while the extensional stiffness is very high. This results in a deformable skin that essentially maintains its cross-sectional arc-length. Figure 7 shows an example of an EBA applied to a flexible skin leading-edge section. In both figures the shape of the beam and its discs is identical, the only difference being that it is rotated over 90° about its left endpoint in the latter figure.

An example of a Fibre-Metal-Laminate de-iced leading edge is shown in Fig. 8.

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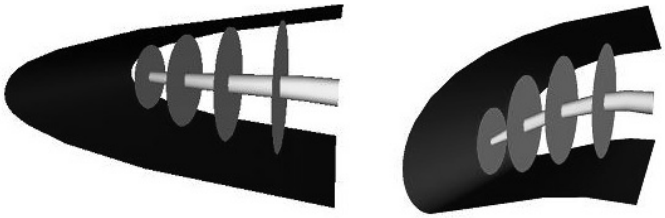


Figure 7. Eccentric beam actuator in a leading-edge high-lift device.



Figure 8. Fibre metal laminate leading edge with embedded de-icing.

6.0 CONCLUSIONS

The alternatives presented here cover the complete range of options for improving the aircraft of the future from the raw materials to design approaches and methods, to new ways of incorporating functionality and performance in the final product. At this point in time, there is no single idea or technology that is a clear winner. In fact, it will most likely be a combination of these that will break the paradigm and will form the basis of future designs. Engineers and management together must decide how to combine resources and invest in the ideal 'technology mix' that will bring about the desired performance increases of the future.

Composites have a relatively short history in aerospace. Therefore many improvements can still be expected in materials and material qualification, in the design of composite structures and in the manufacture of composite structures. Combining the structural functions with other aircraft system functionalities can open up a whole new range of possibilities. The future design team will see its design options increase even further. It will be a challenge to select and mould appropriate options for each individual component taking into account the bigger system. An open and curious mind will be a key requirement for future designers.

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