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Composite Chronicles: A Study of the Lessons Learned in the Development, Production, and Service of Composite Structures

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COMPOSITE CHRONICLES: A STUDY OF THE LESSONS LEARNED IN THE DEVELOPMENT, PRODUCTION, AND SERVICE OF COMPOSITE STRUCTURES

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INTRODUCTION

The development of advanced fiber composites in the 1960's brought aircraft designers a new material option comparable to the introduction of aluminum some 40 years earlier. Carbon fibers, with moduli and strengths comparable to steel and a density half that of aluminum, created visions of 50% weight saving for airframe structure. Although such weight savings have been achieved on a few specific components, the added weight associated with load introduction, the need to satisfy multiple design conditions, design and producibility requirements that usually require a balance of in-plane properties, accessibility for maintenance, inspection, and damage repair, and production cost constraints have made weight savings of 15-30% a more realistic and achievable goal.

As typically happens with the introduction of new technologies, advanced fiber composites have had their share of difficulties along with many notable successes. A few programs involving utilization of composites have experienced unforeseen problems and premature failures during development testing. Many more have been very successful, have provided significant weight savings, and service experience has been excellent. Thousands of safety-of-flight components are in production and are providing excellent service on more than forty different US and foreign military and civil fixed-wing aircraft and twenty different helicopters. Most of these composite components, as well as components on technology development and demonstrator aircraft, are identified in the Appendix to this report.

The major issues today are associated with the materials, manufacturing, and repair costs and not with structural performance. The structural problems and failures that have occurred were primarily caused by deficiencies in the detail design of joints, cut-outs, and discontinuities, designs that did not make proper allowance for the lack of ductility and anisotropic mechanical properties of the composite materials. Other problems were associated with communications between engineering and manufacturing personnel, especially when the people and facilities were located hundreds of miles apart. In most instances, design changes corrected the structural problems and most companies are now using the "Concurrent Engineering" approach with collocated design-production teams to improve communications.

Given the problems and failures that have occurred, and are still occurring, one must ask whether the problems experienced are similar and inherent in the nature of advanced composites or whether new problems continually arise because of lack of technical understanding of the materials and their behavior. Furthermore, when programs are successful, is there an underlying reason that should be recognized, understood, and applied in the future?

In an effort to find answers to these questions, NASA Langley Research Center contracted with Analytical Services and Materials to conduct a study of past composite aircraft structures programs and determine the lessons learned during the course of those programs. The study was focused on finding major underlying principles and practices that experience showed could have a significant effect on the development process and should be recognized and understood by those responsible for making effective use of composites for aircraft structures. Published information on programs was reviewed and interviews were conducted with personnel associated with current and past major development programs. In all, interviews were conducted with about 56 people representing 32 organizations. Most of the people interviewed have been involved in the engineering and manufacturing development of composites for the past twenty to twenty-five years. Several of the people interviewed were retired from prominent positions in government and industry. Their insights and reminiscences of lessons learned, and sometimes forgotten, are invaluable.

ORGANIZATIONAL ISSUES

The various organizational issues, needs and barriers associated with transfer of composites technology were discussed at many of the meetings. There was general agreement about the needs, but some differences of opinion about barriers, some of which are dependent on company organization and vary from company to company.

ORGANIZATIONAL BARRIERS

When advanced composite materials first became available during the late 1960's, and when the Air Force, Navy and NASA began to fund the first advanced composites development programs, there was considerable lack of support and skepticism of the projected benefits and usefulness of composites for aircraft structures by industry.

The companies which were most successful had a few senior corporate executives who became champions for composites. These companies assigned some of their most capable people to composites development programs and set up teams to work with and respond to the Government's initiatives. Even though significant advances were made by the early 1970's, upper management in some companies were still raising concerns about risks and questioned the predicted weight saving potential of composites. Twenty years later, there are still some senior corporate executives and engineers who feel that composites have no place in airframes. A senior government representative suggested that top management education in composites is more necessary now than ever. Management should understand that problems invariably arise in composites structures programs, but these problems have generally been resolved satisfactorily.

ORGANIZATIONAL NEEDS

"Concurrent Engineering", whereby a new product or system is developed jointly and concurrently by a team composed of designers, stress analysts, materials and processes, manufacturing, quality control, and support engineers, as well as cost estimators, has generally become the accepted approach to improve the quality and performance and reduce the development and production costs of complex systems.

Most of the aerospace companies have implemented concurrent engineering approaches in one form or another. Boeing, for example, has formed "Design Build Teams" (DBTs) for development of the Boeing 777. The DBT hierarchy for a typical major structural component is shown in Figure 1. The Boeing structures teams are composed of ten to twenty people from the various engineering, manufacturing, quality control, and cost estimating departments. Where necessary, teams also include people from Boeing procurement and from subcontractors and suppliers. The teams, each of which are collocated, are responsible for producing a final design, cost estimates, production planning, etc. of the structural components and subcomponents. All drawings and interfaces with other teams and subcontractors are made using the Boeing computer network.

Lockheed, GD, Northrop, Vought and other companies have all implemented similar systems for new programs. The Grumman "Task Teaming" approach also subdivides the development effort into tasks and then collocates small multidiscipline teams to perform the tasks. McDonnell Douglas versions are known as "Integrated Product Development" (IPD) and "Integrated Product/Process Development" (IP/PD).

The concurrent engineering approach, in combination with collocation of small multidiscipline teams of people, has provided significant cost and schedule benefits to the design of composite structures. Problems are identified early and are solved by the Team. Organizational barriers are broken down, and the people assigned to a team learn about the different technologies and the specific and interrelated problems associated with design, production, operations, and costs.

TECHNOLOGY TRANSFER

Many of the people interviewed implied that almost all of the critical advanced composites technology developed in the US has been developed by industry in a research and development (R&D) environment by a small number of R&D people funded by DoD, NASA, or Independent Research & Development (IRAD) programs. Very little of the critical technology has actually been developed by universities, research centers, or government laboratories. The fact that the technology was developed in a R&D environment has caused most companies to have major problems with the transfer of the technology and experience to people working on production programs. In four Navy and four commercial aircraft programs, major composites design and producibility problems were caused by the lack of composites experience of the people working on the programs. *

One solution to problems in technology transfer among R&D and production groups has been to assign the experienced R&D people to the production program. The production program gained from their expert knowledge and the less experienced people working

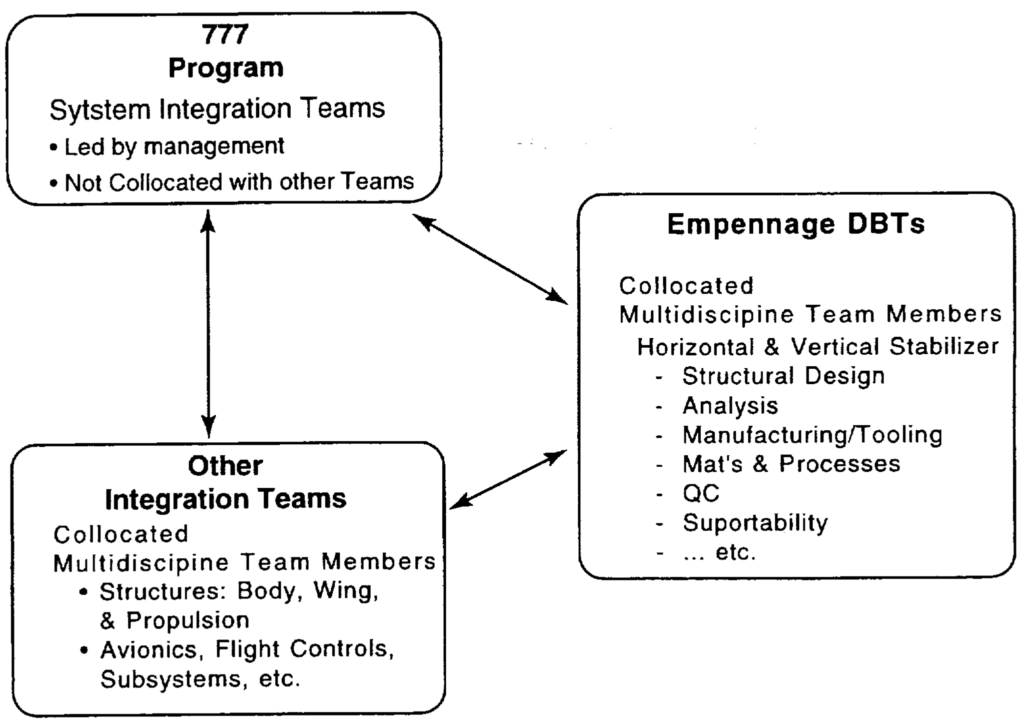


Figure 1. Organization of a structures Design Build Team .

* Programs that were cited as having suffered from a lack of experienced composites personnel included Navy programs for the V-22 wing and fuselage, A-6 wing, AV-8B wing, and A-12. Civil programs included Lear Fan, Starship, Boeing 777-300 elevator, and L-1011 vertical satabilizer.

on the program benefited from the "expert's" experiences. However, other R&D and less important production programs suffered because the number of experienced composites people was limited. In other companies, this situation was ameliorated by assigning experienced engineers and manufacturing people to work with, and learn from, composite specialists on R&D programs. Experienced metal design and manufacturing engineers can be trained in the critical elements of composites design and manufacture in a relatively short time. Some suggested that the training could take as little as a few weeks if the engineers experienced in metals design are working closely with experienced composites engineers on composites programs. Many of these "composites production program experts" were later assigned to management or engineering positions on other production programs or were transferred back into R&D where their "real world" experiences from the production program added to their capabilities.

Additional problems have occurred when the design approach, structural analysis methods, materials, processes, tooling, etc. developed under R&D programs have been applied to full-scale structures. R&D program budgets have generally constrained the size and complexity of demonstration components, and in some cases, failed to identify real world problems. Many R&D designs were very structurally efficient and demonstrated weight savings, but were not producible without major design changes.

Very little of the technology developed by one company under an R&D contract was ever transferred to other companies via technical reports, presentations, lectures and courses. Composites technology, which is highly interactive and must cross many disciplinary boundaries, was best accomplished by having experienced and inexperienced people working together. On programs that involved a team of two or more companies, technology was readily transferred between companies by having people from each company working together, i.e., collocated.

The transfer of technology and know-how into and out-of some classified military programs has been a major problem, particularly when one of the programs has special restrictions and security requirements. In one program, for example, the components for an unclassified program were produced in a restricted area. The restrictions even precluded the Government's program representatives from visiting the area to see first-hand how their components were being made.

LESSONS LEARNED

a. Composite structures technology requires far more interaction than conventional aluminum structures technology.

b. Structural design, certification, and test requirements as well as materials, processes, manufacturing, tooling, quality control, product support, and cost issues must be addressed and understood from the start.

c. The production and operational costs of composite structures must be competitive with counterpart metal structures. Weight savings are a bonus, but cost is the driver.

d. The "Concurrent Engineering" approach, using small collocated multidiscipline teams, resolves problems up front and is being used by the majority of US aircraft companies. By resolving problems early, the costs and time spent on rework, modifications, and changes are reduced.

e. Small companies and composites R&D organizations of large companies with engineering and production capability located in the same facility successfully practiced "Concurrent Engineering" before the term was invented.

f. The "Concurrent Engineering" approach enables designers to become familiar with manufacturing and QC technologies, capabilities, and problems, and vice versa. Designers should spend time on the shop floor in the production facility.

g. There is no substitute for composites experience. Experienced engineers and technicians are a very valuable commodity and are a key toward assuring program success.

h. Composites technology is transferred by people working together and not by reports, presentations, and lectures.

i. Problems have been caused by shortages of composites–experienced people. Many companies have sent their key people to outside tutorials and training courses and have implemented in-house composites training courses for engineers, technicians and production personnel. Some of the most effective courses are taught in-house by experienced company design and manufacturing engineers.

j. Companies that have engineering and production facilities located in the same area generally appear to have had fewer problems and have resolved problems more rapidly than companies with facilities that are far apart geographically.

STRUCTURAL DESIGN, ANALYSIS, AND TEST

DESIGN AND CERTIFICATION REQUIREMENTS

Composite structural design and certification requirements were identified as a major concern at twenty-eight of the thirty-five meetings with industry and government organizations.

The differences among the design and certification requirements for composite structures specified by the FAA, the US Air Force, and the US Navy, have caused major design problems and duplication of very expensive material and component certification test programs. Some structural components designed to meet the requirements of one agency had to be extensively redesigned to meet the requirements of another. As a result, design, certification, and production costs increased enormously, and, because the composite weight savings and cost projections were based on a different set of requirements, the weight and performance targets for the aircraft were not achieved, adversely affecting aircraft capability, competitiveness and price. Most of the government and industry people interviewed felt that the basic requirements should be uniform and that some of the requirements cause unnecessary weight and cost penalties to composites structures.

Twenty-two different US civil aircraft types, which were certificated under FAR requirements, are in service with the US Air Force, Navy, and Army. An additional ten foreign aircraft types, that were certificated under country-of-origin or JAA requirements, are in service with the US Air Force, Army, and Coast Guard. Five of the six JPATS contender aircraft are derivatives of foreign trainers, and one of these has a composites airframe. The Slingsby T-3A trainer, which will soon enter Air Force service, is almost entirely made from composites. A list of these aircraft, their country of origin, and the certificating agency is given in Table 1. Except for the AV-8 and the trainers, all the aircraft are used for transport or as electronic surveillance platforms. The AV-8A was essentially the same as the British Aerospace (BAe) GR. Mk 3 Harrier and was certificated by the British military. The AV-8B, also designated the GR. Mk 5, was a major redesign of the AV-8A by McDonnell Douglas and BAe and was certified to US Navy standards.

Certification to foreign standards is causing some problems for the US services, which have to decide if the original certification procedures and tests are satisfactory by US military standards. The standards also have to be reviewed by the maintenance organizations, which have to inspect and repair damage on the basis of the original civil or foreign certification requirements.

The current FAA damage tolerance requirements were criticized as being overly

US MILITARY DESIGNATION	ORIGINAL DESIGNATION	COUNTRY OF ORIGIN & CERTIFICATION
C-9A/B/C	McDonnell Douglas DC-9	US/FAA
KC-10A	McDonnell Douglas DC-10	US/FAA
C-12F	Beech Super King Air 200	US/FAA
C-18A	Boeing 707-300	US/FAA
C-20A/B	Gulfstream III, IV	US/FAA
C-21A	Learjet 35	US/FAA
C-22A	Boeing 727-200	US/FAA
C-23A	Shorts Sherpa	UK/CAA
VC-25A	Boeing 747-200B (Air Force I)	US/FAA
C-26A	Fairchild Metro III	US/FAA
C-27A	Alenia G222	Italy, MIL
C-29A	British Aerospace 125 Series 800	UK/CAA
VC-137A/B	Boeing 707-135	US/FAA
VC-137C	Boeing 707-300 (Air Force I)	US/FAA
E-3A/B	Boeing 707-320 (AWACS)	US/FAA
E-6A	Boeing 707-320 (TACAMO)	US/FAA
E-8A	Boeing 707-320 (JSTARS)	US/FAA
E-9A	de Havilland DHC-8 Dash 8M	Canada/TC
EC-18B/D	Boeing 707-320	US/FAA
F-21A	IAI Kfir (Dassault Mirage)	Israel/France
AV-8B	British Aerospace Harrier (AV-8A)	UK/US, MIL
T-1A	Beech 400T (Mitsubishi Diamond)	US/Japan/FAA
T-3A	Slingsby T67M260 Firefly	UK/CAA
CT-39A	Rockwell Sabreliner	US/FAA
T-41A/C	Cessna 172	US/FAA
T-43A	Boeing 737-200	US/FAA
T-45A	British Aerospace Hawk	UK/US, MIL
T-47A	Cessna Citation S/II	US/FAA
HU-25A	Dassault Falcon 20	France
U-27A	Cessna 208 Caravan	US/FAA
HH-65A	Aerospatiale Dauphin 2	France
TH-67A	Bell 206-B3	US/FAA

Table 1.– Civil aircraft in use by US military and the certificating country and agency.

severe by many of the people interviewed. These requirements specify that a structure with barely visible impact damage (BVID) must be capable of sustaining design ultimate flight loads in the most adverse temperature/ humidity environment throughout the life of the aircraft. (Federal Aviation Regulations (FAR); FAA Advisory Circular, 1984) Pressurized fuselages with BVID must have sufficient residual strength to withstand the combined effects of critical ultimate flight loads in combination with normal operating internal pressure and external aerodynamic pressure. Residual strength must be established by component or subcomponent tests, or by analysis supported by test evidence. The effects of temperature, humidity, and other environmental factors that may result in material property degradation must be addressed in the damage tolerance evaluation (FAA Advisory Circular, 1984; Evaluation of Composite Structure, Fed Reg, 1986).

The design maneuver limit load factors for Commercial Transport Category Airplanes weighing more than 50,000 lb are +2.5/-1.0g. (FAR Part 25.337) The current FAA requirements call for BVID residual strength capability of load factors of +3.75/-1.5g with BVID. Airbus certification requirements call for only limit load (+2.5/-1.0g) residual strength capability.

Boeing Commercial Airplane Group (Dost, 1993) discussed a revised approach that is similar to metal practices and USAF requirements for composites. The current FAA requirements are shown in Figure 2. An approach proposed by Boeing is outlined on Figure 3. Boeing suggests reducing the design load requirement for non-visible and barely-visible impact damage to a level between limit and ultimate that would account for "Real World" damage scenarios and limit load conditions based on the maximum load per fleet lifetime. The current requirements for easily visible damage would not be changed.

The US Air Force damage tolerance requirements specify that the structure must be capable of carrying the maximum load the member might encounter during a specified inspection interval or, for noninspectable structure, during a lifetime. This load (P_{xx}) is defined as a function of the specific degree of inspectability in a given inspection interval. (Damage Tolerance Criteria, AFSG-87221A) The changes to the commercial aircraft requirements recommended by Boeing would bring them into line with the USAF requirements.

The US Navy composite requirements are, in some respects, more severe than either the FAA or the USAF requirements. As an example, the structure must withstand design ultimate load, adjusted for the effects of environment and material variability, with clearly visible damage. The Navy specifies no yielding at ultimate load (versus limit load and maximum expected flight load, respectively for FAA and USAF requirements).

The Navy requirements currently prohibit use of honeycomb sandwich structure and aramid/epoxy. Because of corrosion caused by a mix of aviation fuel and salt water, the Navy has also barred the use of polyimide- and bismaleimide-matrix materials.

The three agencies also have different requirements for simulation of low energy impact damage and different test procedures to determine the fatigue strength and residual strength of components or the full-scale fatigue test article.

STRUCTURAL DESIGN

General. – The most successful composite development programs have invariably used an integrated engineering team approach to design, development, and production integration. The team is usually structured to include personnel from design, analysis, materials processing, tool design, production engineering, quality control, and, in some instances, costing. Ideally, the team is collocated to facilitate communication.

The integrated team approach of itself cannot guarantee success. The team members must possess the skills and experience needed to work with composites. Personnel with limited experience will not be able to cope with the complexities inherent in composites design and application.

In the past, the personnel involved in the design of composites for a production program often came from the R&D elements of the organization. This should not be considered unusual because the R&D organizations were the first to work with and develop an understanding of composites. There is also benefit in having engineering personnel cycle between product development programs and R&D because exposure to the "real world" of product development makes them more cognizant of the constraints and specific requirements of a production environment and could help engineers formulate more focused research efforts.

Another mark of most successful programs is the implementation of a building-block approach to development. This approach implies that the design, fabrication, and test of major structural elements are taken in steps so that potential problems with the design, either

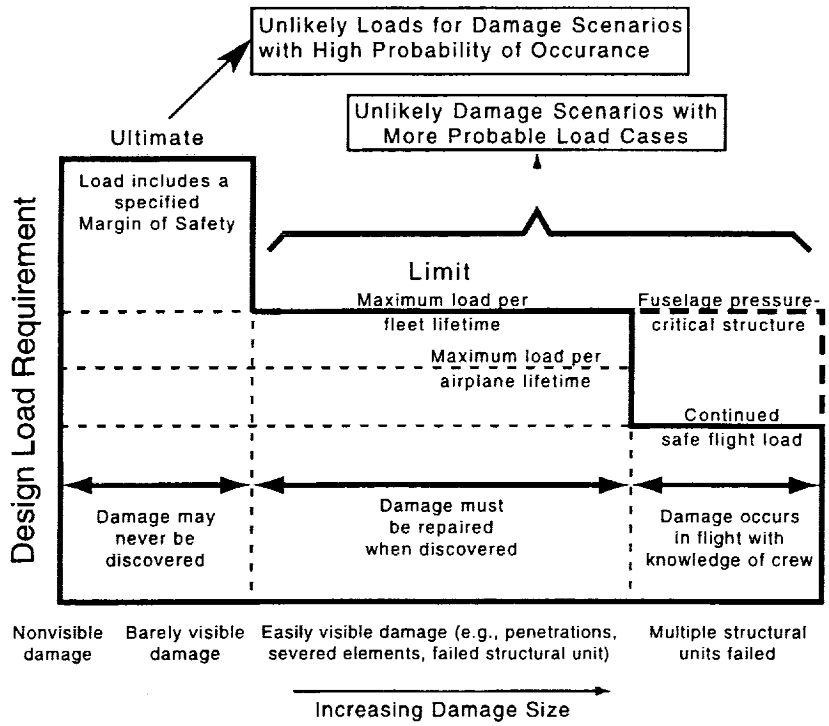


Figure 2. Current FAA damage tolerance design requirements.

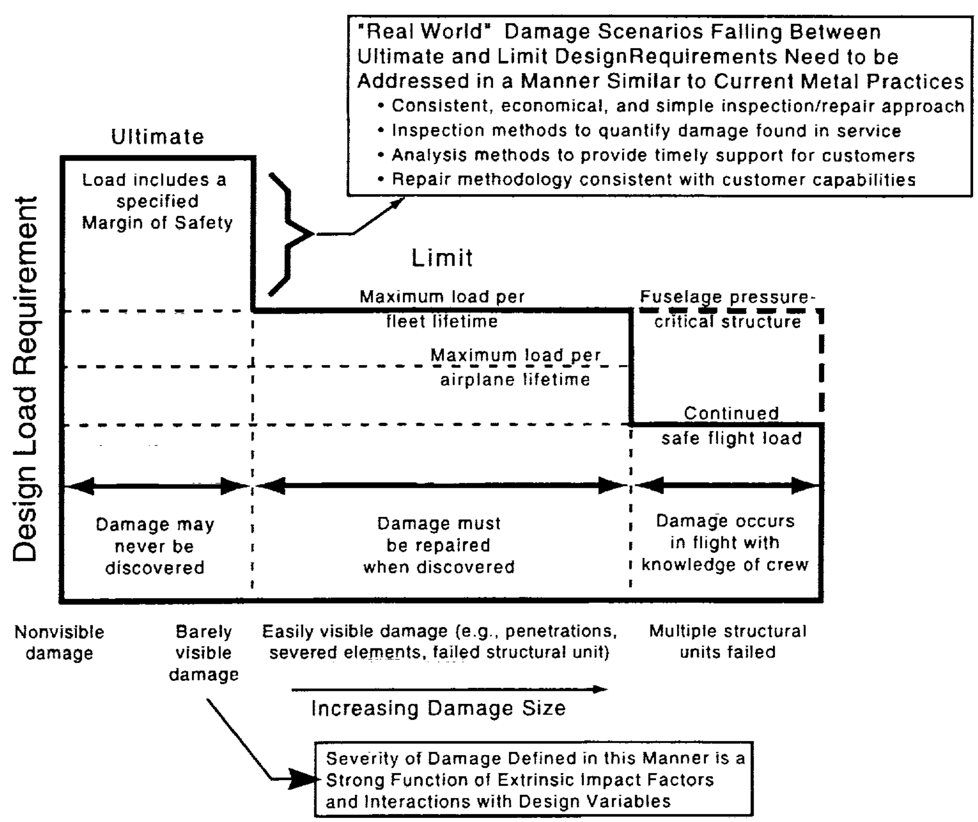


Figure 3. Revised damage tolerance design requirements.

structurally or from the fabrication perspective, are uncovered at the earliest time and with the smallest investment in tooling and test complexity. Each "block" builds on the knowledge gained in the previous step. Program management must understand that composites are a complex material system and their effective use is not a straight-forward and simple engineering effort. New applications will require more time and effort than conventional structures. The time and other resources required to implement a building-block approach must be a part of the program plan. The building-block approach has been shown to reduce risks and, in the long run, keep overall program costs to a minimum.

The advantages of using composites has been demonstrated in many programs. Success has, in some cases, led to over enthusiasm. Designers bent on making the most use of composites in a new design often try to use composites where there is no benefit to doing so. One example is to use composites for many small parts such as clips and brackets when metals would do the job effectively at much lower cost and little, if any, additional weight. Composites should be used only where careful and thorough studies show a clear benefit. The studies must include the availability and cost of facilities, the company's experience and ability to implement their ideas, and the tradeoff of benefit with risk.

Designers often over optimize their designs in an effort to obtain maximum weight savings. Weight savings should not be measured on a part by part basis but should be assessed globally. Highly optimized structures do not leave room to account for manufacturing discrepancies, inherent defects, and inevitable load increases. As program requirements change or difficulties arise, designers should have an acceptable fall-back position. To the greatest extent possible, the designer must understand all of the requirements at the start of the preliminary design. These include the customer's specific requirements and or constraints. The customer must, after all, be willing to accept advanced composites and have the infrastructure to support them in service including inspection, maintenance, and repair capabilities.

Design manuals and documented design practices are beginning to emerge for composites but have not reached a level comparable to that of conventional metal structure. The problems encountered on a program and their resolutions are often not well documented. Companies rely heavily on the experience and expertise of the individuals assigned to a program. As the number of new aircraft development programs gets fewer and more protracted, companies have a more difficult time maintaining their technical base. Experience on one type of aircraft may not necessarily transfer to another type. For example, helicopter manufacturers have experienced problems transferring their technology and experience to fixed-wing aircraft. The design and analysis tools available are adequate for today's requirements. However, the tools must be applied with a thorough understanding of composite behavior so that critical areas of the structure receive adequate analyses.

Composite designs have occasionally been criticized for appearing as "black aluminum." Such criticism is not always valid. Certain structural configurations are appropriate for certain types of loading regardless of the material used. On the other hand, a one-for-one replacement of aluminum elements with composites is probably inappropriate. Each application must be evaluated and the selection of material and structural configuration should always be based on sound trade-off studies based on a knowledge and understanding of how the material will function in each particular application.

Program management assigned to a major development effort involving composite structures should have composites experience. Management must be sensitive to the lessons learned in the past in order to avoid repeating past mistakes. Program schedules must be realistic. New applications of composites will undoubtedly cause some surprises. Most all can be overcome successfully by applying the practices that have proven successful in the past. These include an integrated design-development team and a building-block approach to validation and certification.

Laminate Design. – Although the tailorability of composites is generally regarded as one of their principal attributes, designers have had considerable success by staying with basic families of laminates for most applications. Simple ground rules include the use of a basic 0/90/±45 laminate with at least 10% of the total plies in each direction. Some also prefer that there be no more than 50% of the plies in any one direction. At points of load or geometric discontinuity, the number of plies in the principal load direction can be reduced to "soften" the structure and reduce the stress concentration factors. The optimization of the laminate should consider producibility, particularly if an automated process such as tow placement will be used.

In general, fiber dominated ply designs, that is, designs wherein the fibers are aligned with the principle load directions, have worked well. Use of the fiber-controlled 0/90/±45 family makes laminate design and analysis essentially independent of matrix strength and also facilitates layup and in-process inspection. Furthermore, keeping the reinforcement flat reduces the out-of-plane loads that can cause delamination failures. Hybrid laminates, viz., aramid-carbon mixtures, have been troublesome in some applications because of the fibers' thermal mismatch which can cause matrix cracking.

Detail Design. – The building block approach is an excellent method for developing and validating the details of the design. The final design must be validated at full scale, however. In all aspects of the design, but in particular the design details, the designer must pay attention to producibility and dimensional tolerance requirements. Specifying unnecessarily restrictive tolerances can drive up costs. On the other hand, maintaining close dimensional tolerances at mating surfaces can keep assembly costs down and avoid the potential for assembly-induced damage. A good working knowledge of manufacturing processes or, if possible, experience in a manufacturing facility would not only be helpful to designers but may be essential if the costs of composites are to be competitive with metals. The design and the method of construction must be worked together, particularly if some automated production methods are being considered. Again, the importance of the input of manufacturing, tooling, and quality control personnel in the design process cannot be overemphasized. All the disciplines involved must work concurrently to evolve a design that will meet the goals for schedule, weight, and cost. Complex designs should always be challenged. Simplification of the design can often lead to significant cost savings with little effect of weight.

Companies have found that the problems invariably occur during the early phase of production. These problems can be more successfully resolved if design engineers are available at the manufacturing facility during the first two to three months of production.

Although it is well known that out-of-plane loads should be avoided in composite design, failures still occur because of the inadvertent introduction of out-of-plane loads. Joints, structural discontinuities, and other areas of stress concentration must receive careful attention. Bolted joints are the preferred method for the introduction of out-of-plane loads.

Generally, joints will be the critical element affecting component strength. The designer must rely on a combination of detailed structural analysis and full-size element tests to verify the design. Because joint strength is dependent on actual laminate thicknesses and bolt size and placement, subscale tests cannot be used to verify a joint design.

Overall Design. – Composites have always held the potential for reducing the part count through the cocuring of large assemblies, and a number of companies have had success with designs that minimize the number of parts that have to be joined in separate assembly operations. The larger the part, however, the more likely that the part will become complex and consequently the tooling may also become complex and costly. The benefits of reduced assembly costs must be weighed against the possible increase in tooling costs and the risks involved in curing larger parts.

Early in the development of composites, designers tended to avoid post-buckled, stiffened skin designs. Experience has since shown that such designs can be used successfully and provide a viable alternative to honeycomb sandwich construction..

During the course of a development program, the design loads often increase and the load spectrum may become more severe. The structural designer and the tool designer should consider, early on, how increased loads will be accommodated.

Most companies and government program offices favor designs that avoid the use of secondary bonding. Local disbonds have occurred in service in secondarily bonded adhesive joints. Cocuring and cobonding, however, appear to work satisfactorily.

Honeycomb sandwich structures have had the most disbond problems in service. Because of these problems, the Navy, beginning in 1984, imposed a ban on the use of honeycomb structures on Navy aircraft. The Air Force has also had problems with honeycomb structures on their aircraft. Often, foreign object damage (FOD) or damage induced during maintenance will produce a site for water ingress. During subsequent high speed flight at high altitudes, the build up of vapor pressure will cause skin-core disbonds. Freeze-thaw cycles can induce damage as well. The recurring problems with honeycomb structure has led one transport aircraft manufacturer to avoid its use on new aircraft and to express serious reservations about its suitability for use on the High Speed Civil Transport (HSCT).

The inspectability of structures, both during production and in service, must be considered in the design. Designing for inspectability includes consideration of the type of equipment that will be available to the field inspection units. As noted above, large monolithic structures with reduced part count are desirable. This approach must be tempered with the need to inspect critical interfaces that could get buried within the structure. Removable fasteners should be used in areas where access to the interior is needed for inspection.

STRUCTURAL ANALYSIS

Finite Element Analyses (FEA) have been applied successfully to most composite designs. In regions of high stress gradients, such as around cut-outs and at ply and stiffener drop-offs, a fine mesh must be used. Except at cut-outs and discontinuities, general laminate theory and the modeling of the laminate as an orthotropic plate are satisfactory. At discontinuities, out-of-plane loads are introduced and a 3-dimensional FEA must be used. The problem in many cases has been the identification of a discontinuity. A number of major structural failures have occurred during development programs because the discontinuity was not adequately addressed in the analysis. In most all cases, a post-test analysis of the critical region showed clearly the inadequacy of the design and guided a satisfactory redesign. Generally, detailed analyses should be performed whenever there is an abrupt change in load path or the load in the composite is being taken out or introduced through a fitting. Several designers noted the importance of designing and analyzing the details first and then filling in the design of the spaces between.

Although teaming has become very commonplace in today's environment, problems can and often do arise when companies working on the same project are using different analysis tools. An obvious solution is for all team partners to use the same analysis tools. But, difficulties can still arise when team members are not familiar with the analysis tools imposed by the team leader or when team members do not want to share their "proprietary" codes. When planning joint programs, the planners must be aware of the need for coordination of analytical activities as well as providing for the usual physical interface controls.

STRUCTURAL TEST

Several techniques and criteria are currently in use for dealing with the effects of low energy impact damage (LEID) and the definition and generation of barely visible impact damage (BVID). Several organizations felt that techniques for generating damage, defining BVID, and tests to evaluate the effects of impact damage should be standardized. Standardization would eliminate confusion, permit direct comparison of test data, and reduce costs.

As noted above, subscale components are not useful for evaluating joint designs. The evaluation of design details and the demonstration of damage tolerance should also be done at full size.

Design allowables are usually considered a material property determined by coupon tests. The apparent design allowables can be affected by structural configuration and application. In some cases, generating design allowables from a few selected component tests can be more realistic and cost effective than running hundreds of coupon tests.

As discussed earlier, a well planned test program is an integral part of the building-block approach to development. Certification tests should be a part of the initial plan. Experience has shown that composites do not present a high technical risk when the program includes a 2-3 year building-block validation program. All of the structure should be static tested. When the structure satisfies all the static test conditions, experience has shown that fatigue will not be a problem. Techniques for taking into account the effects of moisture, temperature, impact damage, etc. during the static testing remains a problem. An approach that at least one company has found successful is to evaluate the effects of temperature and moisture at the coupon and subcomponent level and make comparisons between predicted and measured performance. These ratios are then applied in the design of the full-scale component to provide the margins needed.

COST CONSIDERATIONS IN DESIGN

There was nearly universal agreement that costs and not weight savings have become the major driver in the application of advanced composites. For some military applications, improved performance, including stealth, can still dictate the need for composites. But, affordability remains a major design constraint, even for military aircraft.

The need to achieve both weight savings and retain affordability presents a challenge to the designer. The real cost of a weight-optimized structure is often much higher than estimated. Production costs, and particularly assembly costs, should drive the design. In order to achieve a cost-effective design, an integrated design and manufacturing team is essential. Often, slight changes in design can lead to part simplification and reduced manufacturing costs with little or no weight penalty. Designers must understand the ramifications of design details on producibility and be prepared to explore non-traditional approaches in order to reduce costs. As noted in the section on Assembly, monolithic, integral structures can be used to reduce part count and assembly costs. Part commonality is another effective method for reducing costs. Often slight changes in design can lead to significantly more part commonality. Designers must understand that not everything on a so-called composite airplane needs to made of composites. Small parts, for example, can be very expensive when made of composites by conventional methods and metal may be the most cost-effective choice.

The high material costs associated with composites, about 25 times that of aluminum, means that manufacturers need to minimize the amount of scrap. Again, close coordination of design and manufacturing and an understanding of the manufacturing methods available can help to minimize scrap. In a production environment, for example, one manufacturer was able to maintain a buy-to-fly ratio of about 1.1. High material costs are one reason thermoplastics have not been used in any great quantity. Also, the compressive strength of thermoplastic composites has not been as good as that of thermosets. Except in some special applications, the manufacturing cost savings projected for thermoplastics have not been realized.

New US and foreign tactical aircraft are making extensive use of composites for both primary and secondary structures. For these newer aircraft, composites make up about 20 to 35% of the airframe weight. Regardless of the type of material used for the airframe, the costs of military aircraft are considerably higher than those of commercial transports.

Based on current program cost estimates, the fly-away prices of some newer military aircraft structures range from $590/kg ($1300/lb) for the McDonnell Douglas C-17 and $680/kg ($1500/lb) for the Lockheed F-22 to about $1800/kg ($4000/lb) for the Northrop B-2B. These prices reflect the limited production rates and quantities. In contrast, the prices of commercial turbojet and turboprop transports range from $90-140/kg ($200-300/lb). (Aerospace Facts & Figures, 1992-93; Hadcock, 1985, 1989; McCarty, 1990; Harris, W., 1993) Although it may appear that the higher costs associated with composites might be easier to accept within current overall structure costs, composites will still have to "buy their way" onto the aircraft by demonstrating improved performance at an acceptable cost.

The costs associated with assembly of mechanically attached composite joints are very high and require close tolerances at faying surfaces and rigid control of the thicknesses of the parts to be joined. Unless matched tooling or machining is used to match faying surfaces, liquid and/or structural shimming must be used to prevent the introduction of out-of-plane loads in the composite parts at the joint during assembly. The high costs of special fasteners, their installation, the control, inspection and measurement of the thicknesses of the composite parts (to assure proper fastener selection), and shimming are the primary reason for minimizing mechanical joints in the design. These cost issues are described in more detail in the section on Assembly.

Another element of overall cost that is not always apparent to the designer is the cost of in-service support. Inspection and repair personnel must also be a part of the overall design team to help assure that neither the weight nor the cost of maintenance and repair negate the assumed benefits of the initial weight savings.

At present, the experience of the integrated design team is the best tool available for keeping costs down. Tools that can help the designer optimize on costs as well as weight would be a useful addition to design technology.

DESIGN R&D NEEDS

Opinions on current R&D needs were quite varied. Some felt that most R&D programs contributed little to the real problems encountered in the application of composites. They felt that most research was unfocused and failed to address the problems and issues associated with production development programs. Some felt that research, rather than focusing on long range needs, should always be tied to a development program and focus on near-term (3 to 5 year) needs and solutions. Others stated that a R&D program should concentrate resources on full-scale components with real design features rather than on coupon/design allowables programs.

Because fewer resources are going into new (particularly, military) aircraft development, the present time was viewed by some as a good time to concentrate R&D on the development and improvement of overall composites technology. This development would include integration of design concepts, material forms, and manufacturing methods.

Some time has passed since a forum has been held involving Government, Industry, and Academia to specifically discuss present and future R&D needs. One of the first forums was Project Forecast sponsored primarily by the Air Force and held in the early 1960's. This was followed in 1972 by Project Recast which served as a guide for composites research through the 1970's. The most recent forum, conducted under the auspices of the National Research Council, was chaired by Professor James W. Mar and was completed in about 1986. A major recommendation of the committee was for NASA to institute a bold new initiative in composites aimed at the development of technology that would reduce the cost of composites. The present Advanced Composites Technology (ACT) Program was a direct result of the Committee's recommendations. It is now appropriate to once again assess the status of composites, the near-term outlook, and future R&D needs. The present interest in high speed transports and results of studies that indicate the need for composites to achieve the weight targets essential for economic viability are more reasons to address the most appropriate course for composites R&D.

LESSONS LEARNED

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

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

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

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

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

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

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

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

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

j. The number of mechanical joints should be minimized by utilizing large cocured or cobonded subassemblies. Mechanical joints should be restricted to attachment of metal fittings and situations where assembly or access is impractical using alternative approaches (see also, the section on Assembly).

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

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

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

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

MATERIALS & PROCESSES

MATERIALS

Nearly all composites engineers hold strong views on material selection and properties determination and how these have influenced composite structures development. Most agreed that a unified approach was needed for determining material properties. Earlier efforts by NASA and continuing coordination of test methods by SACMA are beneficial and permit direct comparison of test data. Many would like to see uniform material specifications and standards for a few selected material systems and suggested the Government could take the lead in defining these. There were others who believed just as adamantly that standard specifications were not needed. Companies add to the problems of myriad specifications by writing their own and asking materials suppliers to show that their product will meet those specifications. The cost to qualify a new material and generate design allowables can be as much as $3-5 million.

The current Air Force Manufacturing 2005 program is looking at the potential cost savings in using commercial specifications for many of the systems they procure including composites. Other efforts at standardization of materials and processing specifications are under way through the auspices of the Aircraft Industries Association (AIA) and the Great Lakes Composites Consortium. Because of the large number of material systems in use, standardization of repair materials would benefit the user by reducing the number of expensive material systems that have to be stored and periodically monitored to assure their viability. Out of date materials must often be scrapped, thus adding to the cost of composite maintenance. As more and more composites go into service, the problem of maintaining replacement materials for repair and rebuilding of older structures is of increasing concern. Generic composite materials that can be used with a broad range of composite materials are urgently needed for repair.

Development risks can be reduced by not locking into a single fiber/matrix system, particularly if the fiber/matrix system is relatively new. Preferably two material systems that will meet all requirements should be available at the start of a product development program. Unduly high risks are incurred if a materials development program is undertaken in conjunction with the product development. If some materials development is underway during the preliminary phases of a program, there should be a specific cut-off point early in the program so that design can proceed with known (and not projected) performance values.

The Navy has had bad experience with polyimide- and bismaleimide-matrix systems in service. The combination of aviation fuel and salt environment degrades the matrix material. The Navy now considers these materials unacceptable on Navy aircraft. Bismaleimides are proposed for use on the F-22 in areas where there is no contact with aluminum and moisture is not a problem. The need to isolate carbon fiber composites from aluminum or steel because of galvanic interaction has been long recognized and has been dealt with effectively by using an adhesive layer and/or a thin glass-fiber ply at faying surfaces.

Continuing concern over the effects of low energy impact damage has led to the development of toughened epoxy systems, and there is a strong tendency for designers to use the latest and "best" material systems. In many instances, an untoughened system can do the job reliably at a much lower cost than a toughened system. For example, the Navy has had good in-service experience with a well-characterized, untoughened epoxy system that has been on the market for many years.

Material costs can be a significant portion of overall component costs. In addition to the cost of the material alone, the part design along with the method of construction can affect the amount of scrap and the buy-to-fly ratio. At a buy-to-fly ratio of 2:1, for example, carbon/epoxy material purchased at $16/kg ($35/lb) is actually costing $32/kg ($70/lb) on the part, not including processing and lay-up costs. Buy-to-fly ratios ranging from 1.1:1 to 2.2:1 were quoted by interviewees for ongoing production programs. With the cost of aluminum at less than $2.3/kg ($5/lb) and emerging advanced alloys such as aluminum-lithium projected to cost \$5 -\$7/kg (\$10-15/lb), the cost of composite material is still a significant factor in their effective application. Costs associated with incoming material inspection, handling, and storage are higher for composites than for metals. Inspections done by the material supplier, provided they can be certified to the satisfaction of the buyer, should not have to be repeated at the manufacturing facility. Quality control costs usually can be reduced further when automated process controls are used.

None of the people interviewed who have had experience with thermoplastics were optimistic about their viability for extensive use on aircraft. Currently, with costs ranging from \$55-80/kg (\$120-175/lb), thermoplastic-matrix composites are very expensive compared to most thermosets. Furthermore, properties, especially in compression, make them of questionable use for primary structure. Although considerable R&D funding has been applied to the development of thermoplastics, many problems remain unresolved and some question the efficacy of continued Government funding for their development.

Material selection is a critical element of a development program and selection must be based on a thorough analysis that includes consideration of performance, cost, schedule, and risk.

PROCESSES

Processing depends on material, configuration, tooling, and the fabrication method being used. There are, however, some general comments that can be made and would apply to most materials and applications.

Process controls must serve two purposes: they must assure consistent properties and also provide dimensional control on thickness and overall geometry. The process controls must be compatible with the tooling. Tool design can affect heat-up rates, for example, and dictate the extent to which temperature can be controlled. Materials that permit a broad processing window can alleviate some of these problems, and also problems associated with batch processing various parts in an autoclave.

Several techniques have been used successfully to control part thickness and overall part quality. These include:

1. Material specifications that affect resin flow during cure.
2. Improved control of resin content in no-bleed resin systems.
3. Use of higher than normal pressure during cure.
4. Frequent compaction cycles, particularly of thick parts.
5. Intermediate partial cures for thick and/or large parts when material out-time could be a problem.
6. Post cure of all parts.

Continuing process control and process monitoring are required during production to assure that neither the process nor the material is changing. Tag-end specimens can be used to check the processing of every part. Verification of the process should include tests that check the critical structural properties of a laminate, particularly for primary structure. Any proposed changes in processes during production must receive careful evaluation and validation before being approved. Experienced process engineers must be available to work with designers in the early phases of the program and with manufacturing personnel once the part starts into production. Processing problems that occur during development and in production can often be categorized and related thus leading to some generic procedures and solutions that can be applied on future programs. Careful records must be kept and thorough assessments made in order to gain the full benefit from past experience.

Engineering drawings and Materials & Processes specifications tend to be highly detailed and complex. Although the detail is necessary for thorough documentation, it is not easy to follow during the fabrication process on the factory floor. Several companies have developed handbooks that describe the fabrication of a part in a step-by-step process. Engineering drawings are replaced by several successive perspective views or cross-sections of a part showing how each ply or element is to be installed, and detailed material specifications are replaced by short narratives describing the procedures to be followed at each step of the fabrication. An example of a fabrication process sheet is shown in Figure 4. The descriptive manual aids in the inspection process as well as the fabrication.

A number of techniques were suggested for reducing fabrication costs. Some of these included:

1. Use of no-bleed prepreg and adhesive prepreg systems.

2. Automated tow placement using no-bleed tows.

3. Pultrusion of constant section stiffeners.

4. Resin injection molding (RIM) or resin transfer molding (RTM), using stitched or woven preforms, for complex shapes and small parts.

5. Use stack gas rather than nitrogen in autoclaves.

JOINTS AND ATTACHMENTS

As noted above, most companies favor cocuring or cobonding over secondary bonding. For example, cocuring was used very effectively on the F-15 speed brake where carbon/epoxy skins ranging in thickness from 4 to 72 plies were cured and adhesively bonded to aluminum honeycomb core in a single autoclave cycle. Cocuring has also been used effectively for the F-14 horizontal stabilizer and the F-18 wing skins where the composite has been cured and bonded to stepped titanium joint plates in a single operation. Cobonding has also been an effective method of bonding precured composite stiffeners or frame sections during autoclave cure of an uncured skin.

In some cases, however, secondary bonding might have to be used. For example, if cocuring a large complex part with internal tooling, some portion of the part might purposely be left unbonded during the initial cure to permit access for removal of the internal tooling. If secondary bonding is used, great care must be taken to assure near perfect fit-up of the faying surfaces. Close-tolerance machining of faying surfaces may be required prior to bonding.

In all bonding and cocuring operations, problems such as core slippage and crushing, skin movement, and ply wrinkling can occur. Sometimes a two-step curing process can be cost effective because of the significant improvements in quality and process repeatability.

Just as secondary bonding requires a near perfect interface, mechanical joints also require very close fit-up in order to prevent any out-of-plane loads being induced by forcing adjoining surfaces into place during assembly. Whenever possible, mating surfaces should be tool surfaces to help maintain dimensional control. If this is not possible, either liquid shims or, if the gap is large, a combination of precured and liquid shims, should be used in all mechanically fastened joints. Another approach is to cocure the component parts with a very thin steel sheet between the joint interfaces. The steel sheet is removed after the parts have been cured, and assembly completed using mechanical fasteners.

LESSONS LEARNED

a. Trying to conduct materials development in conjunction with a product development program creates undue risks.

b. Because of the high cost of composite materials, designs and manufacturing methods must attempt to minimize scrap.

c. Experienced designers and process engineers must be readily available during the early phases of production

M. COMPACT IN PLACE. EACH FULL LENGTH HAT PLY MUST BE COMPACTED.

NOTE: GLASS BARRIER PLIES MAY BE ASSEMBLED, COMPACTED ON A BLADDER TABLE, INSTALLED ON HATS AND COMPACTED AS A UNIT

N. VERIFY HAT LOCATION WITH TOOL WITH CHECK TEMPLATES. THIS WILL INSURE THAT MANDRELS HAVE NOT MOVED.

HAT COMPACTION

A. APPLY ONE PLY PERFORATED SEPARATOR FILM OVER HAT AND EXTEND ONTO SKIN.

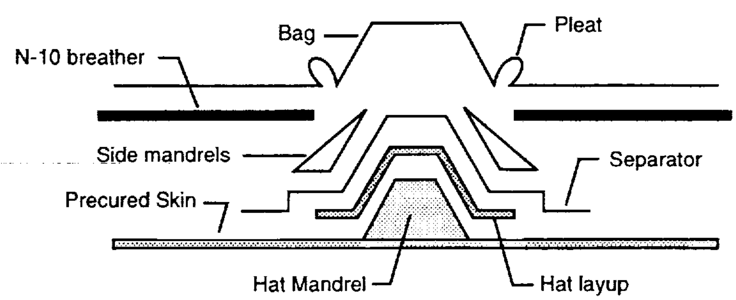
B. LOCATE SIDE MANDRELS.

NOTE: ENSURE SIDE MANDRELS HAVE BEEN COVERED WITH TEFLON TAPE AND ARE SEATED SQUARELY IN THE FLANGE/HAT RADIUS.

C. APPLY BREATHER AROUND PERIPHERY TO CREATE A MANIFOLD. APPLY BREATHER STRIPS IN BAYS BETWEEN HAT LAYUPS. EXTEND FULL WIDTH OF BAY BETWEEN HAT FLANGES.

D. MUD, BAG, AND SEAL. APPLY TO TOOL. APPLY VACUUM BAG OVER PART AND SEAL TO TOOL.

E. VACUUM COMPACT EACH PLY FOR 5 MINUTES AT 22 INCHES HG MINIMUM. REFERENCE FIGURE 6. CHECK THAT SIDE RUBBERS ARE LOCATED PROPERLY AND SEATING INTO HAT RADIUS.



NOTE: ASSEMBLIES -005, -007, -009, -011, -013 MAY BE WORKED SIMULTANEOUSLY

LAYUP -500 FRAME

PLY NO.	DASH NO.	ORIENT	MATERIAL
1	-029	0	DMS2224 W

Figure 4. - Example of process description from shop manual.

to help correct production problems and assess and validate any proposed changes in production processes.

d. Handbooks that pictorially describe the manufacturing process are easier to interpret than engineering drawings and result in fewer layup and processing errors.

e. Cocuring and cobonding are preferred over secondary bonding. Secondary bonding requires near perfect interface fit-up.

f. Mechanically fastened joints require close tolerance fit-up. Liquid or structural shimming is usually required to assure a good fit and to avoid damage to the composite parts during assembly.

MANUFACTURING AND TOOLING

MANUFACTURING

General. – A recurrent theme throughout all the discussions was the usefulness of a well integrated design-build team. For production programs, the team should include mostly "generalists" with multidisciplinary experience and training who understand the interactions among the various disciplines and appreciate the ramifications of decisions made at each step of the development process. Specialists in each field must be available to support the overall effort, but there is no substitute for experience.

Experience gained in an R&D program is not easily transitioned into the production environment unless the people who have the experience are assigned to the production program. Differences in personnel skills, component scale, "real-world" interfaces, schedules, and cost targets must be understood and taken into consideration when production program decisions are made based on prior R&D experience. Many R&D programs have not addressed manufacturing technology, which is the key issue in affordability of composites.

The building-block approach to development provides the best approach to solving design and manufacturing problems and reducing program risk. The probability of success is greatly increased when a program includes the time and resources for an integrated engineering, process validation, testing, and manufacturing development program. Manufacturing development needs to take place at a scale that will test and validate the concepts proposed. The use of sub-scale components, often as a cost-saving attempt, can prove risky and lead to erroneous conclusions.

Cost has become the principal concern in the development, application and utilization of composites, and rightfully so. All aspects of the manufacturing process must be considered when determining costs. Nonrecurring costs include tool design and fabrication; software for automated tape-laying machines (ATMs), tow placement machines, and nesting programs for NC cutters; and manufacturing methods and shop instructions. Recurring costs include composite and other material procurement and waste disposal; layup, autoclave cure and post cure; part trimming; installation of detail parts in assembly fixtures; special fastener procurement; subassembly; and final assembly. Generally, labor costs are slightly higher than material costs, but layup costs can be reduced considerably through the use of automation. Composite material costs are very much higher than the costs of aluminum alloy sheet and plate and care must be taken to minimize composite material waste and scrap.

Simply building low-cost elements does not guarantee low cost structures if the elements do not have the needed quality. Assembly costs, cited by many as the major manufacturing cost element, can soar if parts fit poorly in assembly and an excessive effort is spent in shimming, hole preparation, and fastener selection. A seemingly simple, minimum part count design will not be cost effective if the tooling becomes too complex and costly and part layup and removal becomes difficult, time consuming, and risky. Some "build-to-print" programs intended to foster competition and reduce manufacturing costs did not achieve their goals because the original design did not pay adequate attention to producibility and resulted in major manufacturing problems. Manufacturing personnel have often prided themselves in their "can-do" attitude when instead they should challenge designers to justify a design that makes manufacturing difficult. Producibility is a key element of cost reduction and usually cannot be addressed with the fabrication of one or two prototypes. A full-scale production program is often needed to identify and cure producibility problems.

Automation – As noted above, automation of the layup process can produce cost savings. In general, however, automation is not cost effective if production rates are low (1 aircraft per month, for example). Also, automation is most cost effective on larger parts and may not be cost effective at all on small parts. When automation is planned, the design should be optimized for the process to be used. The process must meet the tolerance requirements for ply placement, or else the design must be changed to accept the tolerances achievable. Again, the need for integration of all aspects of the design and the manufacturing process is apparent.

Incorporation of automation into the fabrication process does not necessarily mean that the part is automatically built to net shape on a contoured tool. Automated methods have been used effectively on a number of programs to lay down flat laminates that are then trimmed to shape and placed onto the mold form. Good quality and repeatability has been achieved using this process. For small parts, automated cutting of plies from broadgoods and manual placement on the tool may be the most cost effective approach. Furthermore, automated processes, because of their repeatability, can often reduce quality control costs.

Assembly. – Some of the discussion presented in the section on Joints and Attachments also applies to assembly. As stated above, assembly costs can be high. Installation of mechanical fasteners can cost as much as $100 per fastener because of the time required for hole preparation, measurement and inspection of each hole, and the cost of special fasteners. Also, fastener grip lengths must be based on actual thicknesses (including shims) at the fastener location and not on nominal thicknesses. Fasteners made of materials such as titanium or A286 stainless steel must be used to avoid galvanic interaction with the carbon fibers. In spite of the potential for high costs, mechanical fasteners generally are used in the assembly process and are preferred over secondary bonding. If secondary bonding is used, enough fasteners should be used to carry limit load without relying on the bond.

Large cocured parts reduce assembly costs. The interfaces to other parts must be carefully controlled, however, to avoid the out-of-plane loads that can be induced if mismatched parts are forced together. Also, good dimensional control of thickness can help reduce assembly time and costs. Because tolerances are more critical for composites, the structural designers and tool designers must know, early on, the capability of the proposed manufacturing process to maintain dimensional tolerances. If necessary, design and tooling changes must be made to meet tolerance requirements. Some extra effort spent on design and tooling can usually be recouped through reduced assembly costs.

Shimming is commonly used to bring mating surfaces into alignment and some manufacturers plan for 100% liquid shimming at final assembly. If possible, the mating surfaces should be tool surfaces in order to maintain the best possible dimensional control. Most manufacturers believed that the cost of quality tooling and dimensional control on mating surfaces was more than offset by reduced labor and inspection costs at assembly. The close involvement of manufacturing/assembly personnel in the design process is essential to strike the proper balance between design requirements and the need for producibility.

TOOLING

Tooling is a critical element of the manufacturing process. Nearly all manufacturers interviewed commented on tooling and had experienced a tooling problem at one time or another. Tool design, including tool material selection, must be an integral part of the overall design process, especially with cocured structures. Tools designed and built by another organization without close coordination with the design, materials and processes, and manufacturing personnel that will ultimately be responsible for the product can lead to serious problems. The capability now exists to couple the three-dimensional design of a part to the three-dimensional definition of tooling and include the capability to perform a finite element analysis of the tool with thermal effects. At the present time, even with this coupled design capability, most manufacturers would expect some tooling changes to occur before or during the early phases of production.

Manufacturers have had success with a number of different tooling materials. Aluminum tools have been used successfully on small parts, but are generally avoided for large parts and female molds because of the thermal expansion mismatch with CFRP. Invar is often used for production tooling because of its durability and low coefficient of thermal expansion. Electroformed nickel also produces a durable, high quality tool but is more expensive than some of the other materials. Steel or Invar tools are needed for curing high temperature resin composites such as polyimides and bismaleimides. Steel, although not as dimensionally stable during heating and cooling as Invar, has been used successfully in a number of applications. A summary of some tooling used on past and current programs is given in Table 2.

The use of CFRP tools has been both successful and disastrous. A key to the successful use of CFRP tooling is to build a tool of very high quality. Most problems have stemmed from tools with poor surface quality and internal porosity. Tools made from CFRP have the obvious advantage of a coefficient of thermal expansion that matches that of the CFRP part. Also, the heat-up of CFRP tools is controllable and uniform and the tools are easily repaired. In general, CFRP tools are lighter than metal tools and therefore easier to handle and transport. Although the life of a CFRP tool is probably less than that of a metal tool, some manufacturers have made up to 300 parts on carbon/epoxy tools and as many as 1000 on carbon/bismaleimide tools. Postcuring

APPLICATION	COMPONENT	TOOL TYPE	TOOL MATERIAL
C-17	Main Landing Gear Doors	OML	Steel
	Control Surfaces	IML	Aluminum
	Inlet Ducts	OML	Invar
B-2	Wing Skins	OML	CFRP
	Wing Substructure	OML/IML	Steel and CFRP
Boeing 777	Stabilizers	OML/IML	Invar
V-22	Wing	IML	Invar
	Fuselage	OML	CFRP
A-6	Wing Skins	OML	Steel
Starship	Wing Skins	OML	CFRP
	Fuselage Skins	OML	CFRP
	Detail Parts	IML	Steel
Lear Fan	All Parts	OML/IML	CFRP
F-14	Horizontal Stabilizer Skins	IML	Steel

Table 2. Examples of moldform tooling used on some past and current programs.

carbon/epoxy tools at 200° C has been found to improve their stability and life.

The use of molded rubber and trapped rubber tools has had limited success and those that have used it usually would not do so again. Rubber can be used successfully in local areas as a pressure intensifier, such as inside radii on stiffeners of cocured structure.

All tools require periodic inspection to assure dimensional control. Although analyses can be done to predict the distortion or "spring back" of a part after it is removed from a tool, the problem is usually solved through trial and error methods to define tool modifications. The "spring back" problem is generally more pronounced on metal tools than on CFRP tools.

The decision on whether to use inner (IML) or outer mold-line (OML) tooling depends on many factors. As stated earlier, tooling the interfaces that will later be joined helps to maintain close dimensional tolerances and usually simplifies assembly. Outer mold-line tools, on the other hand, generally provide more flexibility for design changes such as skin thickness changes. Often the decision to use OML tooling, particularly for parts with an air-passage surface, is based on the feeling that the mold surface will produce a better aerodynamic surface. The differences between IML and OML surfaces may be inconsequential for the application in question and should be challenged if overall part quality and cost could be significantly affected.

Quality tools are essential to the production of quality parts. If possible, production quality tooling should be used during the development program to validate the tooling concepts and materials. The additional costs that might be associated with quality tooling are more than offset by the benefits of producing parts of consistently high quality. As one manufacturer stated, "You can't make good parts on bad tools."

LESSONS LEARNED

a. Generalists with multidisciplinary experience are valuable assets to a concurrent engineering team.

b. R&D experience can be best transferred to a production program if the people with the experience are assigned to the program.

c. The building block approach facilitates process validation and manufacturing development.

d. Most of the costs of a composite part are associated with manufacturing. Effective use of automated processes can reduce both fabrication and quality assurance costs.

e. Designing for producibility is essential. Assembly costs as well as part fabrication costs must be considered when selecting a design and manufacturing process.

f. Automation may not be cost effective if production rates are low or part sizes are small.

g. Dimensional tolerances are more critical in composites than in metals. Dimensional control of mating surfaces can reduce assembly costs and avoid damage to parts during assembly.

h. Selection of tool material is dependent on part size and configuration, production rate and quantity, and company experience.

i. Tools often require modifications. Tool designers should anticipate the need to modify tools to adjust for part springback, ease part removal, or maintain dimensional control of critical interfaces.

j. Quality tools are essential to the production of quality parts and a cost effective element of low-cost production.

QUALITY CONTROL

GENERAL

Currently, costs associated with quality control (QC) for composites are much higher than for metal structures. The costs can range from 15-40% of the manufacturing costs with the higher rates associated with R&D or development programs and the lower rates with established production programs. The effort expended on review board actions related to discrepancies is also higher for composites. Much of the QC costs can be attributed to nondestructive inspection (NDI) of completed parts. In many programs, the policy is to do 100% C-scan ultrasonic inspection of all parts. In addition to in-process inspections, post-assembly inspection is essential to verify assembly process and assure the part has not been damaged in the assembly process.

Techniques intended to reduce QC costs have been implemented by some manufacturers. These include:

1. Automated process controls to assure repeatability.

2. Integrate QC with manufacture in real time.

3. Evaluate specifications to be certain they are not unnecessarily restrictive. Relax on controls that have no direct bearing on part quality.

4. Review past Materials Review Board (MRB) actions to see if disposition could have made with out board review. Alter criteria as appropriate.

5. Assure that QC issues are considered as a part of design and producibility.

6. Zone the structure based on structural criticality and key the sensitivity or degree of inspection to the structural requirements.

NONDESTRUCTIVE INSPECTION

Ultrasonic C-scan is the most commonly used NDI technique. Other techniques, including X-ray, shearography, and thermography, are used to a lesser extent or in special cases where C-scan is not sufficient. The detection of foreign materials that can find their way into a layup may require more than one NDI technique. The techniques currently in use require considerable manual interpretation of the results, usually by an experienced NDI engineer. The use of automation and expert systems in the evaluation of NDI records could have some cost saving potential.

EFFECTS OF DEFECTS

Programs to establish the effects of typical manufacturing defects should be instituted early in the development program. The program should define QC and NDI accept/reject criteria. Often the components made during the early phases of the program, before all procedures, processes, and tooling have been fine tuned, will contain defects. These articles should be tested to establish boundaries on acceptable defects.

LESSONS LEARNED

a. Automated processes can help to reduce QC costs.

b. Focus inspection and controls on aspects of the process and part that have a direct bearing on part quality and performance.

c. Determine and understand the effects of defects on part performance.

SUPPORTABILITY

GENERAL

During the course of this study, the individuals visited and the discussions held focused mainly on the design and manufacturing aspects of composite structures. At the time of the study, NASA personnel were visiting airline operators in an effort to obtain their views and concerns about the application of composites on commercial aircraft. Results of those discussions are included in a paper by Harris (Harris, C., 1993).

As shown in the list of facilities and individuals visited, two military logistics centers were included in our visits. Just as the comments of manufacturers and program office personnel were not specifically limited to design and manufacture, comments from logistics personnel pertaining to design and manufacturing have been included in the appropriate sections. The following discussion combines information from all sources, not just logistics centers.

Civil and military aircraft must be repainted at frequent intervals. Paint stripping is a special problem when composites are present since many commonly used solvents can damage epoxy matrices. For example, repair of a Boeing 767 CFRP rudder that was damaged during paint stripping was reported to cost $99,000 (Harris, C., 1993). Aircraft manufacturers and repair depots are moving towards painting with water-based paints and paint stripping by blasting with polyethylene beads.

IN-SERVICE DAMAGE & REPAIR

Service experience with composite primary and safety-of-flight structures has been very positive: no aircraft have been lost due to failure of composite structure. However, secondary composite components are being damaged repeatedly in airline and military service. Much of the observed damage occurs during aircraft servicing and maintenance that may be unrelated to the composite part (Harris, C., 1993; Donnellan, 1991). Many parts can be repaired, but the cost and time required for repair is much higher and longer than needed to repair similar damaged metal components. The extent and type of damage must be defined using NDT techniques. Damaged structure must then be removed and prepared for repair. After the repair has been completed, it must again be inspected using NDI. Repair prepreg and adhesive materials are expensive. Many different materials must be stocked that have to stored in refrigerators or freezers and have a short shelf life. Often, these materials cannot be procured in small quantities and much of the repair material must be scrapped when shelf life is exceeded. Special approved fasteners and spare parts are also very expensive and add to the cost of repair.

In addition to composite control surfaces (rudders, elevators, spoilers, and flaps) being used on Boeing's 757, 767, and 737-300, Boeing's new model 777 will have composite horizontal and vertical stabilizers and floor beams. The 777 is expected to be ready for flight in 1994. On the other hand, Boeing plans to use fewer composite structures on its new derivative 737 (737-X) than it has on the 737-300. The new 737 is aimed at the small airline market and typically the smaller operators will not have the composite maintenance and repair capabilities that the larger carriers have developed.

Aluminum honeycomb (H/C) sandwich construction has caused major problems on Air Force and Navy aircraft because of moisture ingress, core corrosion, and in the case of supersonic aircraft, debonding of the skin-to-core adhesive due to pressure build-up in the core after moisture ingestion. Stabilizers and control surfaces have had to be disassembled and rebuilt after the core has been replaced with corrosion resistant aluminum core. The Navy has prohibited use of any H/C sandwich components on the V-22, F/A-18E/F and all other new aircraft types. (Donnellan, 1991)

Composite components on Navy aircraft have performed well in service, and most of the damage has been on secondary structural components as a result of handling or manufacturing discrepancies. There have been some critical shortfalls for repair technology R&D and development of 120°C (250°F) repair materials and special repair equipment, including NDI. There have also been some problems in identifying the extent and effects of heat damage and developing repair concepts for heat-damaged composite structure, such as the inboard flap of the AV-8B. (Donnellan, 1991)

Operators would like to have more engineering and manufacturing information about specific components as well as factory rework, repairs and inspection records. Manufacturer's Standard Repair Manuals (SRM) have been criticized as being inadequate and most repairs fall outside SRM guidelines.

LESSONS LEARNED

a. Supportability has not been adequately addressed during design. Composite structures must be designed to be inspectable, maintainable and repairable.

b. Most damage to composite structure occurs during assembly or routine maintenance of the aircraft.

c. Repair costs are much higher than for metal structures.

d. Improved SRMs, engineering information, and MRB records are needed for in-service maintenance and repair.

e. Special long-life/ low-temperature cure repair materials are required.

f. Moisture ingestion and aluminum core corrosion are recurring supportability problems for honeycomb structures.

CONCLUSIONS AND RECOMMENDATIONS

Although composites technology has made great advances over the past 30 years, the effective application of composites to aircraft is still a complex problem that requires experienced personnel with special knowledge. All disciplines involved in the development process must work together in real time to minimize risk and assure total product quality and performance at acceptable costs. The most successful development programs have made effective use of integrated, collocated, concurrent engineering teams. The composition of the team should represent design, analysis, materials and processes, tooling, manufacturing, quality assurance, cost analysis, and product support. Ideally, at least one representative of each discipline would have prior applicable experience in composite structures. Program managers should understand and appreciate the need for a well-planned, systematic, development effort wherein the design and manufacturing processes are validated in a step-by-step or "building block" approach. Such an approach will reduce program risk and is cost effective.

Producibility and supportability are the key elements in the design of composite structures. The design, tooling, and manufacturing processes must function in concert to assure consistent high quality parts at an acceptable cost. Quality tooling is essential to the production of quality parts.

Not all parts are suited to composite construction. Their use should clearly warrant the added care that must be taken at every step of their development.

Some of the confusion and problems related to certification stem from differences in requirements set by the various certificating organizations. Much of the confusion could be resolved if the FAA and the military services would jointly generate a uniform set of basic certification requirements for composite structures. Additional special requirements would be needed for military aircraft composite structures to meet uniquely military design conditions such as live fire, survivability, catapult, arrested landing, etc., but these should be based on the generic requirements.

Current concerns in the industry about the value of generic R&D programs need to be addressed. An industry-government forum, similar to Project Recast or the National Research Council ("Mar Committee") activities, should be instituted to assess the efficacy of today's research activities and help set the directions for future efforts.

An interdisciplinary team of experienced, knowledgeable people working together can make the technical risk of applying composites comparable to that of any other advanced structure. Weight savings alone are no longer considered sufficient justification for using composites. Composite structures must be cost effective. The weight saved and other in-service benefits, such as durability or corrosion resistance, must have enough value to offset any added costs that may arise from the use of composites. Optimization for producibility and supportability can significantly reduce the cost of composite structures with little, if any, weight penalty or loss of structural performance.

APPENDIX

A CHRONOLOGY OF ADVANCED COMPOSITE APPLICATIONS*

Richard N. Hadcock
RNH Associates

Hundreds of different composite aircraft components have been designed, developed, and produced over the past thirty years. A few of the components have been associated with technology development, were generic, and did not fly or go into production. Other components were designed to assess in-service performance and were produced in relatively small quantities. The majority have been production components, designed from the start in composites, that have provided up to twenty years' of reasonably trouble-free service.

The first structural composite aircraft components, made from glass fiber reinforced plastics (GFRP), were introduced in the 1950-60 time frame. These components included the fins and rudders of the Grumman E-2A, helicopter canopy frames, radomes, rotor blades, etc. By 1967, the entire airframe of the small Windecker Eagle was made from GFRP. Since then, GFRP has become one of the standard materials for light aircraft and lightly loaded structural components.

Many of the lessons learned in design and fabrication of the GFRP parts were used in the initial development of advanced composites technology during the 1965-70 time period.

Boron filaments and carbon (or graphite) fibers first became available about 1965. Their high compression strength and stiffness, in combination with low density, enabled boron fiber reinforced plastics (BFRP) and carbon fiber reinforced plastics (CFRP) to be used instead of aluminum for high performance airplane structures. At about the same time, duPont introduced Kevlar®, a low density aramid fiber. Aramid fiber reinforced plastics (AFRP), which have high tension strength but very low compression strength, have been used for some lightly loaded fairings and helicopter components and as hybrid composites composed of two or more different types of fiber reinforcement for access doors and fairings.

Epoxy thermoset-matrix composites have been used for most of the composite parts. A few higher temperature parts have been made from polyimide or bismaleimide thermoset-matrix composites. Very few have been made with thermoplastic matrices.

Details of some of these components and their associated successes, problems, and failures are described in this Appendix and supplements the information obtained from the discussions with industry and Government personnel.

MILITARY AIRCRAFT

Many composite structural components have been designed and produced during the past thirty years for US and foreign military aircraft. Most of these primary and secondary structures are listed by aircraft and component in Tables A-1(a) and A-1(b). They include both development and production components. Tertiary structural components, such as landing gear doors, access doors, panels, and fairings are not included.

U.S. Military Aircraft. – Much of advanced composite structures research and development in the U.S. during the past 25 years has been associated with military aircraft applications. These R&D programs provided much of the technology base for production of composite aircraft structures. During the 1965 to 1973 time period, boron filaments were available at lower prices than carbon (or graphite) fibers. Boron/epoxy also had higher specific strength and stiffness than the then-available carbon/epoxy materials. For these reasons, boron/epoxy (BFRP) was the advanced composite of choice in the late 1960s.

*This Appendix was prepared by Richard N. Hadcock, RNH Associates, from material that he had collected outside of the present contract effort. The material collected by Mr. Hadcock provides an excellent background for the current study and shows the growth of composites use on aircraft worldwide. The information is included here with Mr. Hadcock's permission.

The first aircraft to fly with advanced composite control services were some 50 USAF McDonnell F-4 aircraft that were fitted with boron/epoxy rudders in the late 1960s. About the same time, a USN Douglas A-4 was flown with a boron/epoxy flap.

Boron/epoxy was also selected by General Dynamics for the F-111 composite horizontal stabilizer USAF development program. The composite stabilizer was flown in 1971. Boron/epoxy reinforcement was also bonded to the F-111 D6-ac steel wing pivot fitting to reduce stress and increase fatigue life of production F-111s. This modification incorporated a removable CFRP fairing for inspection of the pivot fitting.

In 1967, Grumman selected boron/epoxy for an IRAD program to design and build a wing box extension for the FB-111. The technology and experience gained on this and the 1968 USAF Advanced Composites Wing Structure (ACWS) program, led to the Grumman F-14A boron/epoxy horizontal stabilizer. The stabilizer was fully qualified for production in 1969 and flew for the first time in December 1970. The F-14 horizontal stabilizer was the first advanced composite safety-of-flight component to fly and go into production. (Lubin, 1971)

Rockwell International also selected BFRP for the F-100 composite wing development program for the Air Force. A wing demonstration component was tested, but the wing was never flown.

A large CFRP/GFRP overwing fairing was introduced into production for the Grumman F-14D in 1990. Five ship sets of composite overwing fairings had previously been produced for the F-14A for the Navy in-service evaluation program (Manno, 1977).

The first significant CFRP component to fly, the Douglas A-4 flap, was flight tested in 1970. Douglas Aircraft continued with the development of a CFRP horizontal stabilizer for the A-4 under a USN contract. The stabilizer failed prematurely under static test due to stress concentrations at an attachment. Though never flown, the program provided Douglas and the Navy with useful information for later programs.

Experience from the F-4 BFRP rudders and an IRAD horizontal stabilizer program resulted in the McDonnell F-15A horizontal and vertical stabilizers and rudders. The F-15A speed brake became the first carbon/epoxy production safety-of-flight component. The first F-15 flew in 1972. Approximately 1.2% of the F-15 airframe is composite. McDonnell later designed and built a BFRP F-15 wing under an Air Force development program, but it was never flown.

Both the General Dynamics YF-16A prototype, which flew in 1974, and it's competitor for the USAF Light Weight Fighter program, the Northrop YF-17, had CFRP stabilizers. General Dynamics, which had completed a USAF CFRP fuselage development program based on the Northrop F-5A center fuselage, went on to design and build a CFRP YF-16A forward fuselage in 1975. These components were structurally tested but did not fly or go into production.

The F-16A won the Light Weight Fighter competition. The YF-16A and early F-16A production horizontal stabilizers were honeycomb sandwich bonded/bolted design, which had to be proof tested to satisfy the USAF MIL-STD-1530 certification requirements. The revised design, which had CFRP covers mechanically attached to aluminum substructure eliminated the need for proof testing.

Lockheed designed and built two boron/epoxy reinforced center-wing boxes for the C-130. These have been in service since 1974.

Grumman designed and built very large composite horizontal stabilizers for the B-1A bomber, which were structurally tested in 1976. These, and the Rockwell-designed vertical stabilizer, were both fully qualified for the B-1A bomber prior to cancellation of the program in 1977. When the B-1B program was revived in 1981, Rockwell decided to revert to the original metal stabilizers.

Eleven left-hand outer wings of the Vought A-7D were designed and built under a USAF program in 1973-76. Three wings were used for flight and ground tests; the remaining eight were put into service for five years on Air National Guard airplanes. These were the first composite wing components to enter military service. About the same time, Vought designed and made 28 CFRP spoilers that were installed on Lockheed S-3 aircraft for Navy in-service evaluation. (Manno, 1977)

AIRCRAFT	YEAR†	WING BOX		FUSELAGE				STAB.		CONTROL SURFACES						
		COVERS	SUBSTRUCT	FORWARD	MID	REAR	INTERNAL	HORIZONT'L	VERTICAL	ELEVATOR	RUDDER	AILERON	FLAP	SLAT	SPOILERS	AIR BRAKE
Grummaan E-2	1960								P		P					
MD F-4E	1969										L					
Douglas A-4	1969/70							D					D			
Northrop F-5	1970				D			D								
GD F-111/FB-111	1970/73	D	D			D		D								
Lockheed C-5A	1970													D		
NA F-100	1970	D	D													
Grumman F-14A/D	1970/88							P[1]								
HS Vulcan	1971															D
MD F-15	1972	D[2]	D[2]					P	P							P
Vought A-7D	1973	L	L													P
Lockheed S-3	1973														L	
HS Harrier	1973	D[3]														
Northrop YF-17	1973							D	D		D	D	D			D
Lockheed C-130	1974	L														
GD YF/F-16	1974/6							P	P		P					
Dassault Mirage III	1975										D					
Dassault Mirage F1	1976							D				P				
Rockwell B-1A	1976						P	D	D							
MiG-29	1977							P	P		P	P	P			P
MD F/A-18	1978	P						P	P		P					P
SAAB Viggen	1978							D	D	D						
Panavia Tornado	1978							D								
Dassault Mirage 2000	1978								P		P	P				
SEPECAT Jaguar	1979	D	D													
Mitsubishi T-2	1979										D					
Dassault Mirage 4000	1979							P	P		P	P				P
TOTALS 1960-1979																

†Year of first flight or completion of R&D program.

[1] 1970 — P = Production

[2] 1975 — L = Limited Production

[3] Ferry wing tip — D = Development

Table A-1. Composite components on military aircraft.
(a) 1960-1979

In 1978, the McDonnell Douglas F/A-18A was the first military aircraft to designed with an advanced composite wing. The F/A-18 was a derivative of the Northrop YF-17 with modifications to meet Navy requirements. The weight increases due to a new landing gear, arrester hook, wing folding, etc., required a larger wing and increased fuel capacity.

A composite wing was selected for the F/A-18 to save weight. The wing covers are CFRP mechanically attached to aluminum substructure. In addition to the wings, CFRP covers are used for the horizontal and vertical stabilizers (produced by

Northrop), the center fuselage upper skin panels, the speed brake, flaps, and various fuselage access doors and panels. CFRP accounts for 9.5% of the structure weight (Weinberger, 1977; Kandebo, 1993a). The heavier McDonnell Douglas F/A-18E/F, scheduled to fly in December 1995, has CFRP wings that are 25% larger and stabilizers that are 36% larger than the F/A-18A. Much of the center and rear fuselage is also composite. These changes will increase the use of CFRP materials to 18% of the structure weight (Kandebo, 1993a).

The McDonnell Douglas AV-8B had its origins in the British Aerospace AV-8A, Harrier, two of which were modified into development YAV-8Bs by McDonnell and were first flown in

		WING BOX		FUSELAGE				STABILIZER		CONTROL SURFACES						
AIRCRAFT	YEAR†	COVERS	SUBSTRUCT	FORWARD	MID	REAR	INTERNAL	HORIZ*	VERTICAL	ELEVATOR	RUDDER	AILERON	FLAP	SLAT	SPOILERS	AIR BRAKE
Alphajet	1980	D[1]	D[1]					D[2]								D
Vought VSTOL A	1980					D										
Fuji MT-X	1981								D				D			
Grumman VSTOL A	1981				D											
MD AV-8B	1981	P	P	P				P			P					
Lockheed C-141	1981					D[3]										
GD F-16XL	1982	D	D						D			D	D			
Antonov An-124	1982						P									
Northrop F-20	1983							D								
Grumman X-29	1984	D														
Dassault Rafale A	1986	D	D	D	D			D	D		D	D	D			D
BAe EAP	1986	D	D					D								
IAI Lavi	1987	D	D					D	D		D	D	D			D
Grumman A6-E	1988	P[4]	P[4]													
SAAB Gripen	1988	P	P					P	P			P		P		P
Bell/Boeing V-22	1989	P	P	P	P	P	P	P	P	P	P	P	P			
Northrop B-2A	1989	P	P	P	P	P	P						P		P	
Lockheed F-117	1990								P							
Northrop YF-23	1990	D	D	D	D	D	D	D			D	D	D	D		D
Lockheed YF-22	1990	D	D	D	D	D	D	D	D		D	D	D	D		D
MD C-17A	1991									P	P	P			P	
Rockwell/MBB X-31	1991	D		D								D	D			
Dassault Rafale C/M	1991	P	P	P	P				P		P	P	P			P
Eurofighter EFA	1992	P	P	P					P		P	P	P			P
GD/MD A-12	1992	D	D	D	D	D	D									
MD F-18E/F	1995	P			P	P		P	P		P					P
Mitsubishi FSX	1995	P	P					P	P		P	P	P			
Lockheed F-22	1995	P	P	P	P	P	D	P	P		P	P	P	P		
TOTALS 1980-1995																
TOTALS 1960-1995																

* Including canards
† Year of first flight or completion of R&D program
[1] 1987, Dornier [2] 1980, Dassault [3] Cargo doors [4] Boeing

P = Production
D = Development

Table A-1. Concluded. (b) 1980-1995

1978. The modifications including replacing the aluminum wing with a CFRP supercritical wing that had 14% more wing area and 50% more internal fuel volume. The production AV-8Bs, which first flew in 1981, have a CFRP wing (including substructure), horizontal stabilizer, forward fuselage, rudder, wing flaps, fuselage fairings, and strakes (Weinberger, 1977, Watson, 1982).

The AV-8B inboard flaps, inboard fairings, and the strakes are high temperature carbon/bismaleimide. All the other composite components are carbon/epoxy. With the larger wing, the range-payload capability of the AV-8B is approximately twice that of the AV-8A. Advanced composites account for 26.3% of the structure weight of the AV-8B.

About the same time, the US Navy became interested in a multipurpose V/STOL aircraft to replace the Grumman E-2 and the Lockheed S-3. Composite fuselage development contracts were awarded to Vought (LTV) for development of rear fuselage structure and to Grumman for development of the center fuselage. Both these designs were capable of significant elastic post-buckling capability and were also designed to be exceptionally damage tolerant.

Vought built a 6 ft full-scale section of the rear fuselage using CFRP stiffeners, longerons, and bulkhead webs with AFRP skins. Grumman built a 25 ft long, 10 ft deep, and 7 ft wide section of the center fuselage using CFRP skins that incorporated GFRP crack-arrestment strips, reinforced by integrally molded CFRP hat-section stiffeners and "J"-section frames. Tests at Naval Air Development Center (NADC), Warminster, demonstrated both post-buckling capability and capability to hold limit load with significant low energy impact damage as well as ballistic damage.

The first wing made from high temperature carbon/polyimide, the General Dynamics F-16XL, flew in 1982. The F-16XL has a gross weight of 48,000 lb compared with the F-16C gross weight of 37,500 lb. The carbon/polyimide wing covers, made using inner mold line (IML) moldforms, are bolted to aluminum substructure. The F-16XL wing is twice the area of the standard F-16 wing and carries 80% more internal fuel. The F-16XL did not go into production; the two prototype airplanes were acquired by NASA in 1989 and are currently being used for flight test programs. The Japanese FS-X, which is under development, is a derivative of the F-16 and will have a Mitsubishi-designed composite wing.

The Grumman X-29A forward-swept-wing technology demonstrator aircraft first flew in 1984. The X-29A wing had CFRP covers mechanically attached to a substructure composed of titanium and aluminum. The X-29A wing is divergence critical so the wing covers were designed using aeroelastic tailoring to preclude divergence by coupling wing bending and twist. This coupling was accomplished by orienting the outer wing cover laminate axis at a different angle than the wing geometric axis. Two X-29A aircraft were built and are now in storage at NASA Dryden Flight Center after completing about eight years of flight testing. (Hadcock, 1985)

The CFRP wing of the Navy Grumman A-6E was designed and built by Boeing to replace the aluminum wing, which had a relatively short fatigue life. The A-6E wing is much more complex, larger, and more highly loaded than the wing of the AV-8B. Its geometry is identical to the metal A-6E wing, with the same wing-to fuselage attachments, fold joints, store stations, and control surfaces.

Design studies of the USAF Advanced Technology Bomber (ATB) were initiated in 1979. The ATB became the B-2A when the Northrop/Boeing/Vought team won the development contract in 1981, and the first flight took place from Palmdale, California, to Edwards AFB on July 17, 1989. Almost all the skin and much of the substructure are CFRP and other composite materials. The B-2A wing has a span of 172 feet and a wing area of 5,140 square feet. The span and wing area are only slightly less than the wing of a Boeing 747 (Jane's All the World's Aircraft, 1993-94 and prior ed.).

Northrop, the B-2A prime contractor, was responsible for overall design, and designed and made the leading edges, the crew station assembly, wing tips, elevons, and fixed trailing edge assemblies. Boeing designed and manufactured the outer wings and the wing-fuselage center section. Vought designed and manufactured the intermediate wing section that included the engine bays, inlets, and the main landing gear bays and doors. Final assembly was performed at the Air Force plant in Palmdale, CA. At the start of the program, Northrop set up an advanced CAD/CAM system that was shared by Boeing and Vought.

Following the success of the Bell XV-15 tilt rotor research program, which was sponsored jointly by NASA and the US Army, the DoD outlined a requirement for a Joint Services Vertical Lift aircraft, which became the JVX program. In 1982, the Bell/Boeing team proposed a tilt rotor configuration in response to the JVX program solicitation and was awarded a $200-million contract for preliminary design (Jane's; Air International, May 1989).

The $1.8-billion full-scale development (FSD) contract was awarded to the Bell/Boeing team for the aircraft in June 1985. Allison won the engine competition with a derivative of the T-56 turboprop in December 1985. The development program consisted of six flight test prototypes plus a static and a fatigue test prototype aircraft. The first prototype V-22A flew on March 9, 1989. The fifth prototype was lost on June 1, 1991. Boeing Helicopters designed and built the prototype fuselages. Boeing Defense and Space designed and built the prototype wings under a subcontract from Bell. Grumman designed and built the empennage under a subcontract to Boeing Helicopters.

The airframe of the Bell/Boeing V-22A Osprey multi-mission tilt-rotor aircraft is almost entirely integrally stiffened CFRP, which accounts for approximately 70% of the airframe weight. The rotor blades have hollow GFRP/CFRP spars with honeycomb sandwich trailing edges. The original V-22 program schedule called for limited production of 12 aircraft to start in 1990 and reaching a peak of 132 aircraft by 1996. A major redesign was still being funded in FY 1993. A critical design review to freeze the design of preproduction aircraft was scheduled for late 1994. Four engineering and manufacturing development (EMD) aircraft are scheduled to be delivered beginning in 1996. The design of the subsequent low-rate production aircraft will be fixed in 1997 (Kandebo, 1993c).

The stabilizers of the Lockheed F-117A, which provide yaw control, were originally made from aluminum (the prototype airplane flew in 1981). These have been made from carbon/thermoplastic since 1990 and, except for some small components on the F-22A, appear to be the only production application of carbon/thermoplastic structures.

In 1986, the Lockheed YF-22A and the Northrop YF-23A were selected as the USAF Advanced Tactical Fighter Demonstration/Validation (Dem/Val) program winners. Both aircraft had made extensive use of CFRP composites. The YF-22A was designed and built by a consortium composed of Lockheed, Boeing, and General Dynamics. Northrop teamed with McDonnell for the YF-23A.

The Northrop YF-23A first flew in June 1990 followed two months later by the Lockheed F-22A. Lockheed was selected to proceed with the F-22A EMD program in April 1991. Lockheed has overall program responsibility as well as design and fabrication of the forward fuselage, inlets, wing leading edge flaps, trailing edge ailerons and flaperons, and the vertical and horizontal stabilizers. General Dynamics (now part of Lockheed) is responsible for the fuselage center section and Boeing is responsible for the wing (Jane's).

Both the YF-22A and the F-22A have CFRP wing, fuselage, and empennage surfaces. Advanced composites, titanium, and aluminum accounted for 23%, 23% and 35%, respectively, of the structure weight of the YF-22A. The composites portion was 13% thermoplastic (TP) matrix and 10% thermoset (TS) materials. Following live fire tests and a reevaluation of the TP materials, the distribution on the F-22A will be 26% composites (22% TS, 4% TP), 30% titanium, and 14% aluminum. The F-22A is scheduled to make its first flight in 1995 with full production starting in 1998 (Morrocco, 1993a, 1993b).

Much of the airframe of the General Dynamics/McDonnell Douglas A-12 Navy attack bomber was going to be made from carbon/bismaleimide composites. This program was terminated by the Navy in 1991 because of schedule delays, cost overruns, and Navy concerns about corrosion of the BMI matrix in a jet fuel/sea water environment.

The USAF C-X program for a heavy lift military transport was won by McDonnell Douglas in July 1982 and became the C-17A. Full scale development was approved in February 1985 and the first development aircraft first flew in September 1991. Four airplanes were on order in FY 1992 and eight in FY 1993. Composite components consist of control surfaces, fairings, and engine nacelles, which are similar to the components currently in service on commercial airliners. About 15,000 lb of composite materials are used in the C-17A, accounting for some 8% of the airframe weight. Most of the composite components are supplied by subcontractors. The components and manufacturers include (Jane's; Parker, 1989a):

AFRP wing trailing edge panels and flap hinge fairings:	Aerostructures Hamble
CFRP winglets and main and nose landing gear doors:	Beech
CFRP ailerons, elevators, and rudders:	Grumman
CFRP/AFRP wing-fuselage fillets:	Heath Techna
CFRP tail cone:	Martin Marietta
CFRP/AFRP main landing gear pods:	Northwest Composites
CFRP spoilers:	Textron
CFRP engine nacelles and AFRP stabilizer leading edges:	Vought

Foreign Military Aircraft. - Some of the major European aircraft companies initiated advanced composite structures development programs in the early 1970s. Foreign military aircraft and their associated composite components are included in Tables A-1(a) and A-1(b).

The British government and aircraft industry have been involved in development of polyacrylonitrile (PAN) precursor CFRP structures since 1964, when the Royal Aircraft Establishment filed the patent for high-strength, high-modulus PAN-based carbon fibers. British Aircraft Company (BAC) and Hawker Siddeley Aviation (both are now part of British Aerospace PLC) were involved in structural development of CFRP components. In 1968, Hawker Siddeley began a CFRP structures development program that resulted in design, fabrication, and flight test of a Harrier ferry wing tip (which incorporated an additional fuel tank), an airbrake flap for the Vulcan bomber, and six CFRP rudder trim tabs that were installed on BAC Jet Provost trainers to obtain service experience (Sanders, 1971; Fray, 1991; Molyneaux, 1978).

About 1974, BAC initiated a cooperative program with Grumman to develop CFRP engine bay doors for the Jaguar ground attack aircraft. This was followed by the BAC-MBB Tornado CFRP Taileron development program and the BAC CFRP wing development program, which used the Jaguar wing as the baseline.

Around the same time, Hawker Siddeley was working with McDonnell Aircraft in the development of the AV-8B Harrier. Hawker Siddeley (now BAe) manufactures all the CFRP horizontal stabilizers for the U.S. AV-8B as well as the British Harrier GR Mk 5 and Mk 7 V/STOL close support aircraft.

The BAe Experimental Aircraft Programme (EAP) technology demonstrator program was initiated in 1982 as a result of joint British-German-Italian European Combat Aircraft studies. The German team members, MBB and Dornier, dropped out of the program in 1983, but BAe and Aeritalia went ahead with the program. Flight testing began in 1986. The wing covers and substructure were all CFRP and were made using the BAe co-bonding process. One wing was built by BAe in England, the other by Aeritalia in Italy. The canards had CFRP covers bonded to honeycomb/metal substructure. The remainder of the airframe and control surfaces were metal (Braybrook, 1986).

In 1985, SAAB contracted BAe to design and manufacture the first 3 ship-sets of CFRP wings for the JAS 39 Gripen multi-role light fighter. SAAB is manufacturing the production aircraft wings and designed and built the remainder of the structure, which includes a GFRP vertical stabilizer, canards, control surfaces, and landing gear doors. CFRP materials account for 30% of the structural weight of the Gripen (Braybrook, 1986). The first SAAB Gripen flew in December 1988 but crashed in February 1989 just before landing due to loss of control. The test pilot survived the crash with very little injury, possibly because the composite wings broke off cleanly as the aircraft hit the runway (Jane's).

SAAB started an advanced composite development program with a British company in the 1970s to design and build a CFRP elevator for the canard of the Viggen. The elevator development was followed by a cooperative program with Grumman to develop a CFRP vertical stabilizer for the SAAB JA 37 Viggen. Grumman made the prototype stabilizers, which incorporated parts designed and fabricated by SAAB. These were flight and ground tested by SAAB who then produced some 50 stabilizers for the Swedish Air Force to obtain CFRP structures in-service experience.

Avion Marcel Dassault-Bréguet Aviation (Dassault), with financial assistance from the French government, started development of CFRP structures for their military aircraft in the early 1970s. Their first design was the rudder of the Dassault Mirage III fighter. One flight test and one ground test article were built and tested in 1975. This was followed by the Mirage F1 CFRP horizontal stabilizer program in 1976. Four prototype stabilizers were built and flight and ground tested. Dassault then decided to design CFRP ailerons for the F1. The prototypes were flight and ground tested in 1977. These were 26% lighter than the aluminum ailerons, so they were put into production at a rate of 7 ship sets per month. More than 730 Mirage F1 had been produced by the beginning of 1990 (Chaumette, 1982).

Dassault introduced the Mirage 2000 multirole fighter in 1978 and the Super Mirage 4000 in 1979. Both aircraft had CFRP vertical stabilizers and rudders (the Mirage 4000 stabilizer was also a fuel tank), inner and outer elevons, and landing gear doors. The Mirage 2000 avionics access door and the Mirage 4000 canards were also CFRP. Only prototypes of the Mirage 4000 were built, but approximately 500 Mirage 2000 fighters have been produced since 1984 (Chaumette, 1982).

Dassault followed with a program to develop a CFRP horizontal for the Bréguet-Dornier Alpha Jet. The stabilizer utilized CFRP covers bonded to full-depth honeycomb core outboard, transitioning to a bolted multi-spar substructure inboard (similar to the SAAB Viggen Vertical Stabilizer design). The stabilizer was designed using an automated optimization computer program and was projected to be lighter and less expensive than the metal baseline (Chaumette, 1982).

About 1980, Dassault and Aerospatiale were sponsored by the French government to design, build, test, certify and fly a composite wing for the Dassault Falcon 10. The development was jointly funded by Dassault and Aerospatiale. The wing was fully certified and was flown in the mid-1980s. The Falcon 10 wing program provided Aerospatiale with the know-how and confidence needed for the ATR 72 CFRP wing, described later in this section. It also gave Dassault the know-how and confidence needed to baseline a CFRP wing for the design of the Rafale (Chaumette, 1982).

The Dassault Rafale 'A' prototype first flew in July 1986, and the first production Rafale 'C' flew in April 1991. The Rafale is a land-based or carrier-based multi-role fighter and is about the same size and weight as the F/A-18. CFRP components account for about 35% of structural weight and include the wing, canards, vertical stabilizer, control surfaces, landing gear doors, and some fuselage panels. Significant use is also made of aluminum-lithium alloys and superplastic-formed diffusion-bonded (SPF/DB) titanium parts for the leading edge flaps and hot fuselage structure (Interavia, 1985).

Dornier used the Alpha Jet wing as the baseline for their CFRP wing development program during the mid 1980s. The Alpha Jet ground support/trainer aircraft was used by both Dornier and Bréguet as a test bed for CFRP structures, shown shaded in Figure 6-1B. The Alpha Jet wing had CFRP spars and integrally stiffened covers and with aluminum ribs and fuselage attachment plates (Rose, 1986).

The latest Western European fighter to utilize significant quantities of CFRP structure is the British-German-Italian-Spanish Eurofighter EFA (European Fighter Aircraft), which first flew in 1992. CFRP structures include the wings, forward fuselage, vertical stabilizer, and control surfaces. The canards are metal (Jane's).

The Israeli Aircraft Industries (IAI) Lavi fighter full scale development program was started in October 1982. Grumman was contracted to design and build the CFRP wings and vertical stabilizer and IAI designed and built the CFRP canards. The first prototype aircraft flew on December 31, 1986.

Following the Lavi flight tests, the wing geometry was changed and the areas of the control surfaces were increased. This required a major wing redesign. The original wing had CFRP skins bolted to CFRP substructure. The redesigned wing retained CFRP for the covers, but the substructure was changed to aluminum. There was a significant cost saving and only a small weight penalty changing to aluminum substructure. Since the Lavi production program was canceled, IAI has used the aircraft for technical development and flight testing (Jane's).

Advanced composite structure development by the Soviet (now Russian and Ukrainian) aircraft companies appears to have lagged behind that of the rest of Europe. Fighter aircraft applications include the CFRP vertical stabilizers, the rear portion of the horizontal stabilizers, rudders, ailerons, and flaps of the Mikoyan MiG-29, which first flew in 1977 and entered operational service in 1983 (Fricker, 1988).

Ukrainian military transport applications include some 12,000 pounds of CFRP, AFRP, and GFRP in the airframe of the large Antonov An-124 Ruslan (Condor), which first flew in 1982. The even larger An-225 Mriya (Cossack), which flew in 1988, was reported to utilize composites for 25-30% of the structure weight (Jane's, DeMeis, 1988).

In Japan, Fuji designed and built some CFRP rudders that were flown on the Mitsubishi T-2 supersonic trainer in 1979. Fuji went on to design, build and test a vertical stabilizer for their MT-X advanced trainer contender in 1981. Fuji lost the prime trainer contract to Kawasaki. The elevators and rudder of the Kawasaki T-4 are made from CFRP. (Private communication, Fuji Industries, Utsonomiya, Japan, 1981)

Mitsubishi proceeded with CFRP primary structures development and is currently designing and building an all CFRP wing for the FS-X. Mitsubishi was appointed prime contractor of the FS-X program, which is a modified General Dynamics F-16C. The first of four prototypes is due to fly in 1996. General Dynamics, Kawasaki, and Fuji are the major subcontractors.

COMMERCIAL TRANSPORT AIRCRAFT

Most of the composite structural components designed during the past twenty-three years for US and European commercial transports and business and private aircraft are listed by component type in Table A-2. These include both development and production primary and secondary structures. Tertiary structural components, such as wing and stabilizer fixed leading edges and trailing edge panels, landing gear doors, access doors, fairings, cabin floors, and engine nacelles and inlets are not included in the table.

U.S. Turbojet Transports. – Most of the US airliner components, designed and built during the 1972-1986 time period, were developed under the NASA Langley Research Center Aircraft Energy Efficiency (ACEE) program and a predecessor flight service evaluation program to establish a long-term durability data base for composite materials and structures. The NASA programs included limited production and airline service evaluation of various components. It also included a program to determine the long-term effects of exposure to moisture, ultraviolet radiation, fuels, and hydraulic fluids on the mechanical properties (NASA CP-2321, 1984).

The first airliner advanced composite component to fly was a Boeing 707 boron/epoxy fore-flap, which was flown in 1970.

Under the NASA flight service evaluation program, Boeing designed and built 108 carbon/epoxy spoilers that entered service on Boeing 737 aircraft with six different airlines in 1973. Some spoilers were later made from different carbon/thermoplastic materials, but had to be taken out of service because the matrix was degraded by hydraulic fluid (NASA CP-2321, 1984).

Under the NASA ACEE program, Boeing continued with ten CFRP elevators that entered service in 1980. These were followed by four Boeing 737 CFRP horizontal stabilizers that were installed on two aircraft in March 1984. These horizontal stabilizers were the first commercial transport CFRP primary structures certified for airline service. (NASA CP-2321, 1984)

Douglas Aircraft designed and built thirteen CFRP upper rudders and three boron/aluminum aft pylon skins for the DC-10 under a NASA program. Additional rudders were built under the ACEE program. The rudders first entered service in 1975. Some of the upper rudders, which are a multi-rib post-buckled design, are still in service. This design approach was later used for the CFRP MD-11 ailerons. The aft pylon skins were the first metal matrix composite components to enter airline service (NASA CP-2321, 1984).

Douglas followed by designing and building a multi-spar vertical stabilizer for the McDonnell

AIRCRAFT	YEAR†	WING BOX		FUSELAGE				STAB.		CONTROL SURFACES					
		COVERS	SUBSTRUCT	FORWARD	MID	REAR	INTERNAL	HORIZONT'L	VERTICAL	ELEVATOR	RUDDER	AILERON	FLAP	SPOILERS	AIR BRAKE
AIRLINERS															
Boeing 707	1970												D		
Boeing 737-200	1973/84							L[1]						L[2]	
MD DC-10	1975-78								L		L				
Boeing 727	1980									L					
Boeing 767	1981									P	P	P		P	
Lockheed L-1011	1982								D			L			
Boeing 757	1882									P	P	P	P	P	
Airbus A300-600	1983													P	
Boeing 737-300	1984									P	P				
Airbus A310	1985								P	P	P			P	
Airbus A320/A321	1987							P	P	P	P	P	P	P	
Ilyushin Il-96	1988						P						P		
Tupolev Tu-204	1989									P	P			P	P
MD MD-11	1990						P			P	P	P	P	P	
Airbus A330/A340	1993/91							P	P	P	P	P	P	P	
Boeing 777	1995						P	P	P	P	P	P	P	P	
TOTALS		0	0	0	0	0	3	4	6	10	10	7	7	10	1
REGIONAL TRANSPORTS															
Embraer Brasilia	1983					P[3]							P		
SAAB 340							P						P		
ATR 42	1984									P	P	P	P		
Fokker 100	1986											P	P		P
ATR 72	1988	P	P							P	P	P	P		
Dornier 328	1991					P	P	P	P	P	P	P	P		
TOTALS		1	1	0	0	2	2	1	1	3	3	4	6	0	1

†Year of first flight or completion of R&D program.

[1] 1984 P = Production

[2] 1973 L = Limited Production

[3] Tail cone D = Development

Table A-2.- Composite components on commercial transport aircraft.

Douglas DC-10 in 1977. The stabilizer was still in service with Finair in June 1993.

Lockheed participated in the NASA flight service program with eighteen AFRP fairing panels for the L-1011. These panels entered airline service in 1973. In the ACEE program, eight CFRP ailerons were designed by Lockheed and built by Avco and entered service in 1982 (NASA CP-2321, 1984). Lockheed also designed and built a CFRP L-1011 vertical stabilizer. The vertical stabilizer failed during static test partly because of

Composite Component	Boeing 737	Boeing 757	Boeing 767	Boeing 777
Ailerons	Boeing	Boeing	Alenia (Italy)	CASA (Spain)
Elevators	Boeing	Boeing	Alenia (Italy)	HDH (Australia)
Rudder	Shorts (UK)	Boeing	Alenia (Italy)	ATA (Australia)
Spoilers		Grumman	Alenia (Italy)	Grumman
Inboard Flaps		Shorts (UK)		Grumman
Outboard Flaps		CASA (Spain)		Alenia (Italy)
Floor Beams				Rockwell
Ldg Gear Doors:				
Main			Fuji (Japan)	JADC (Japan)
Nose			Boeing	Shorts (UK)
Fairings		Heath Techna	Fuji (Japan)	JADC (Japan)
Nacelles	Various	Various	Various	Various

Table A-3. Boeing composite component suppliers.

the method of load introduction. Extensive long term environmental and cyclic load tests were performed on spar and skin panel components, but no flight articles were built.

Boeing started design of both the model 757 and the 767 in the late 1970s. The 767 made its first flight in 1981 followed by the 757 in 1982. Boeing decided to baseline CFRP composites for the elevators, rudders, spoilers, landing gear doors, and engine cowlings for both these airplanes. The flaps of the 757 are also CFRP. Composite components account for about 3,400 lb of structure on both the 757 and 767.

When Boeing introduced the 737-300 in 1985, CFRP composites were selected for ailerons, elevators, the rudder, fairings, and engine cowl doors. Composites account for 1,500 lb of the structure.

The Boeing 777, scheduled to fly in 1994, utilizes about 18,000 lb of composites, some 9000 lb of which are CFRP. CFRP components include the entire tail, control surfaces, floor beams, main landing gear doors, and engine nacelles. Other composite components include wing-fuselage fairings, and wing fixed trailing edge panels. The CFRP horizontal and vertical stabilizers are made by Boeing. Many of the other composite components are being supplied to Boeing by U.S. and foreign subcontractors (Table A-3).

The structure of the McDonnell Douglas MD-11, which first flew in January 1990, includes almost 9,500 lb of composites. Components include CFRP elevators, winglets, ailerons, outboard flaps, spoilers, wing fixed trailing edge panels, tail cone, engine cowls, center engine inlet duct, cabin floor beams, and AFRP/GFRP wing-body and aft body fairings. Almost all these composite components are produced by subcontractors. Some of the suppliers include: Fuji (Japan): outer ailerons; Embraer (Brazil): outboard flap; Mitsubishi (Japan): tail cone; Westland (UK): flap vanes; Heath Techna (US): center engine inlet duct (Therson, 1989, Colucci, 1991).

Foreign Turbojet Transports. – The European consortium, Airbus Industrie, uses about 4,000 lb of composites on the A300, which first flew in 1972. A300-600 components include CFRP/GFRP elevators and rudders, CFRP spoilers, nose landing gear doors, and main landing gear leg fairings, GFRP wing upper surface trailing edge panels, and AFRP wing-body fairings and flap track fairings.

Use of composites was extended in 1985 by changing the Airbus A310 vertical stabilizer material from aluminum alloy to CFRP. The CFRP vertical stabilizer, built by MBB (DA) in Germany, also contain a balance fuel tank on the long-range A310-600. About 7,400 lb of composite structures are used on the A310.

Composites use was further extended in 1987 on the Airbus A320. Both the horizontal and vertical stabilizers are CFRP as well as the elevators, rudder, ailerons, spoilers, flaps, wing leading and trailing edge access and fixed panels, landing gear doors, and engine cowls and doors. Fairings are GFRP and AFRP. Composites account for about 9,000 lb or 15 % of the structure of the A320. The larger Airbus A330/A340 uses compos-

Composite Component	Airbus A300-600	Airbus A310-300	Airbus 320/319/321	Airbus 330/340
Horiz Stabilizer			CASA***	CASA
Vert Stabilizer		DA	DA	DA
Ailerons			BAe	Aerospatiale
Elevators	CASA	CASA	CASA	CASA
Rudder	DA*	DA	DA	DA
Inboard Spoilers	Aerospatiale	DA	BAe	BAe
Outboard Spoilers	DA	DA	BAe	BAe
Inboard Flaps			DA	DA
Outboard Flaps			DA	Textron (US)
Wing LE Panels	BAe**	BAe	BAe	BAe
Wing TE Panels	BAe	BAe	BAe	BAe
Ldg Gear Doors:				
Main	Fokker	CASA	CASA	ATA (Australia)
Nose	CASA	CASA	Aerospatiale	Fokker
Fairings:				
FWD Wing-Fuse	Aerospatiale	Aerospatiale	Aerospatiale	Aerospatiale
Rear Wing-Fuse		Belairbus	Aerospatiale	Aerospatiale
Nacelles	Rohr (US)	Rohr (US)	Rohr (US)	Rohr/Grumman (US)

*Deutsche Airbus **British Aerospace ***Construcciones Aeronáuticas, S.A.

Table A-4.- Airbus composite component suppliers.

ites for similar components, but, although the total weight of composite structure is much higher, the percentage weight dropped to 12% (Parker, 1989b).

The Airbus A320 and A330/340 CFRP horizontal stabilizers are designed and are built in Madrid by CASA, Airbus Industrie's Spanish partner (Barrio Cardaba, 1990, Marsh, 1991). The A310, A320, and A330/340 CFRP vertical stabilizers are designed and built by the Deutsche Airbus division of MBB (DA) at Stade in Germany. Suppliers of composite components to Airbus are listed in Table A-4 (Jane's).

The Soviet Ilushin Il-86 wide-body airliner, which entered limited service with Aeroflot in 1980, had CFRP cabin floors. The derivative Il-96, which first flew in 1988, has CFRP flaps and cabin floors. The horizontal and vertical stabilizer leading edges are also composite but are probably GFRP or a CFRP/GFRP mix (Jane's).

The Tupolev Tu-204 medium-range airliner structure is about 18% by weight composites (approximately 20,000 pounds). The Tu-204, which first flew in 1989, is the Russian counterpart of the Boeing 757. CFRP components include spoilers, airbrakes, flaps, elevators, and the rudder. Other composite components include part of the wing skins, stabilizer leading edges, and wing-fuselage fairings (Jane's).

Turboprop Transports. – Advanced composites are used extensively for control surfaces (ailerons, elevators, rudders, and flaps) of turboprop regional transport aircraft. Many of these are made from AFRP or a mix of AFRP with local CFRP reinforcing.

The de Havilland Canada Dash 8, which has been in airline service worldwide since 1984, uses AFRP for the wing and stabilizer and flap leading edges, wing tips, flap trailing edges and shrouds, wing-to-fuselage fairings, and engine nacelles. AFRP components account for about 2,000 lb, or 10% of the structure weight. Approximately 350 Dash 8s are currently in service.

The first commercial transport airplane with a CFRP wing to enter airline service (in 1989) was the Avions de Transport Régionale ATR 72. Avions de Transport Régionale is a French/Italian consortium composed of Aérospatiale and Alenia. The ATR 72 is a derivative of the 42-passenger ATR 42, which was certified and entered airline service in 1985 (Pilling, 1988).

The complete outer wing of the ATR 72 is made from CFRP instead of aluminum because the wing weight was critical to operating performance and costs. The wing, designed and manufactured by Aérospatiale, incorporates fuel tanks and the design is based on much of the experience which Aérospatiale gained from their joint program with Dassault from the Falcon 10 CFRP wing program.

Aérospatiale manufactures the wing components that are assembled at Toulouse, France. Other ATR 42 and ATR 72 composite components produced by Aérospatiale include the wing fixed leading edge and trailing edge panels, ailerons, wing-to-fuselage fairings, and engine nacelles. *

Alenia manufactures the fuselage and tail of both the ATR 42 and ATR 72 in Italy. Composite components include the elevators, rudder, landing gear doors and fairings, and tailcone. Final assembly of the ATR 42 and ATR 72 takes place at Toulouse (Pilling, 1988).

* Private communication on composites use on the ATR 72 supplied to R. Hadcock by Avions de Transport Régional, Blagnac Cedex, France, November 1993.

The first commercial transport to enter airline service that has significant portions of the fuselage made from CFRP was the German DA Dornier 328. The rear fuselage, pressure bulkhead, and nose cone are made from CFRP. The wing-fuselage and the main landing gear fairings and doors are made from mixed CFRP/AFRP. The wing flaps, ailerons, and wing tips, as well as the complete tail of the Dornier 328, are all CFRP. CFRP/AFRP mixed composites are used for the wing fixed trailing edges and the dorsal fin (Jane's).

GENERAL AVIATION

Business Aircraft. – Business aircraft are listed in Table A-5 together with their associated composite structures components. The German Claudius Dornier Seastar light amphibious flying

		WING BOX		FUSELAGE		STABILIZERS		CONTROL SURFACES			
AIRCRAFT	YEAR†	COVERS	SUBSTR	SHELL	INTERNAL	HORIZ*	VERTICAL	ELEVATORS	RUDDERS	AILERONS	FLAP
LearFan 2100	1981	L	L	L	L	L	L	L	L	L	L
Dassault Falcon 10	1983	D	D								
Dornier Seastar VT01	1984			D	D	D	D	D	D		
Dassault Falcon 50	1984									P	
Avtek 400	1984	D	D	D	D	D	D	D	D	D	D
Gulfstream IV	1985				P			P	P	P	
Beech Starship 2000	1986	P	P	P	P	P	P	P	P	P	P
Piaggio Avanti	1986					P	P	P	P		
Cessna Citation	1987							P	P	P	P
Dornier Seastar CD2	1987	P	P	P	P	P	P	P	P	P	P
Avtek 400A	1991	P	P	P	P	P	P	P	P	P	P
TOTALS		6	6	6	7	7	7	9	9	8	6

* Including canards
† Year of first flight

P = Production
L = Limited Production
D = Development

Table A-5. Composite components on business aircraft.

boat is included in the list.

The only three business aircraft that have been almost entirely made of advanced composites are all American-made. All three programs have had major structural problems that caused weight growth, extension of design and development schedules, and cost increases.

The first of these composite aircraft was the LearFan 2100, which was conceived by Bill Lear shortly before his death in 1978. Almost the entire airframe was made from CFRP and AFRP. All primary structure was sheet-stiffened construction.

The prototype first flew in 1981. The airframe was modified following wing and fuselage failures during structural test. The modified airframe did satisfy structural certification requirements and was certified by the FAA. The structural modifications required to meet certification in combination with premature failure of the gearbox delayed the program, increased development costs, and caused most of the 200 orders to be withdrawn. The first and only production aircraft finally flew in 1983. LearFan Corporation declared bankruptcy in 1985 because of delays and difficulties in FAA and British CAA certification and other financial problems (Jane's; AWST, Jan 12, 1981; Whitaker, 1981; Wigotsky, 1983).

The second all-composite aircraft was the Beech 2000 Starship. The Starship configuration was originally conceived in 1982 by Burt Rutan and went into production in 1988. The Starship has an airframe made almost entirely from CFRP-Nomex® honeycomb sandwich construction. An 85% scale proof-of-concept vehicle was flown in 1983. The first of three Starship prototypes flew in February 1986. Major structural modifications had to be made to the wing to satisfy FAA damage tolerance requirements, and to the fuselage following premature failure during structural test. The first production Starship, which was flown in April 1989, had a take-off gross weight of 14,400 lb compared with an original target weight of 12,500 lb. The weight increase in combination with aerodynamic efficiency that was lower than expected based on the performance of the scale vehicle, reduced range and performance. Because of limited aircraft orders, Beech decided in 1993 to terminate production at 50 aircraft (Jane's; Abbott, 1989; Aerosp Eng, Apr 1990).

The third composite airplane was the Avtek 400 light corporate transport.* Avtek Corporation has produced one proof-of-concept Model 400 that first flew in 1984. The airframe, designed by Dr. Leo Windecker, is 72% Kevlar®/epoxy and 18% carbon/epoxy by weight. Following flight and wind tunnel tests, the aircraft was redesigned and incorporated so many major changes that the current Model 400A is essentially a new design. CFRP is used for the wing spar caps and webs, and the rudder. Two ground test aircraft are being built for FAA certification tests to FAR Part 23 requirements. Current investors in Avtek include duPont, the State of Alabama, the government of Malaysia and various foreign companies. Avtek is currently looking for about $70-million for the flight tests and structural tests needed to complete the FAR Part 23 certification program. They have orders for 89 airplanes (about two years of production) (Jane's).

Other US business aircraft CFRP applications include the Gulfstream IV engine support structure and the pressure bulkhead as well as the ailerons, rudder, and spoilers, which were designed and are made by Lockheed. The elevators, rudder, ailerons, and flaps of the Cessna Citation V are also made from composites.

The Claudius Dornier Seastar was designed in Germany by a team led by the late Dr. Claudius Dornier (who had no connection with Dornier GmbH). The Seastar is a light twelve-passenger STOL amphibian, designed to operate from grass, water, snow or ice. The airframe is made almost entirely from GFRP with some CFRP reinforcing. Design was started in January 1982 and the prototype Seastar was flown from July 1984 until it was damaged landing on Lake Constance in July 1985 (Air Int, Oct 1988).

The damaged aircraft, which originally had been flown with an aluminum wing from a Dornier Do 28 and a composite fuselage, was repaired and rebuilt with an all-composite wing. Designated CD2, the rebuilt aircraft flew in April 1987. CD2 Seastar production started in October 1989 but the company went into bankruptcy one month later.

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*Private communication on composites use on the Avtek Model 400A supplied to R. Hadcock by Robert Adikes, Avtek Corp., September 1993.

Dornier Composite Aircraft, owned by Conrado Dornier, purchased the company in February 1990. The CD2 was certificated by LBA in October 1990 and meets the FAR Part 23 commuter aircraft certification requirements. The first production Seastar was delivered at the end of 1991 (Jane's; Air Int, Oct 1988).

Dassault and Aérospatiale were sponsored by the French government in 1980 to design and build a CFRP wing for the Dassault Falcon 10 business jet. The integrally stiffened wing covers and most of the beams and ribs in the substructure were CFRP. The main landing gear ribs, inner rear spars, and the root wing-to-fuselage lap joints were aluminum.

The Falcon 10 program included fabrication and tests of critical subcomponents. The CFRP wing was flown on a Falcon 10 in 1984 and provided Aérospatiale and Dassault with the technology and experience they used for the ATR 72 and Dassault Rafale CFRP wing programs. Both the twin-engine Falcon 50 and the three-engine Falcon 900 have CFRP ailerons (Chaumette, 1982).*

The Italian Piaggio P.180 Avanti light corporate turboprop was produced at a rate of 24 airplanes per month in 1991. The complete rear fuselage and empennage assembly of the Avanti is CFRP and was designed and is being produced by Sikorsky. Piaggio makes the CFRP canard. The Avanti prototype first flew in September 1986 and the first production airplane flew in January 1990. Full Italian certification was obtained in October 1990.

Private, Trainer, and Competition Aircraft. – A list of composite components on private, trainer, and competion aircraft is given in Table A-6. The Windecker AC-7, Eagle I, was the first all-composite private aircraft to receive FAA certification. The Eagle was designed by Leo Windecker who used GFRP for the entire airframe. The Eagle flew in 1967 and received FAA certification in 1969 (Rosato, 1969; Taylor, 1989). The Eagle was a high-performance, single engine, four-seat monoplane but did not go into production. Leo Windecker later assisted in the design and development of the Avtek 400.

*Private communication, Avions de Transport Régional, Blagnac Cedex, France, November 1993.

The Bellanca Model 19-25 Skyrocket II, a six-seat light monoplane, was also made almost entirely from GFRP. The Skyrocket was designed and built by Bellanca Engineering Inc., a company formed by August Bellanca. Design and construction of the prototype started in 1971 and the airplane first flew in March 1975. Powered by a 435 hp Continental engine, the Skyrocket II had a cruise speed of 331 mph and a range of 1,465 miles and held five FAI speed records. Bellanca was working on FAA certification in 1984, but the program was never completed. The airplane was used by NASA Langley for flight and wind tunnel tests in 1982 (Jane's; Taylor, 1989).

For the past 25 years, Scaled Composites Inc. led by Burt Rutan, has been involved in design and fabrication of many all-composite proof-of-concept and competition aircraft. These aircraft, which are made from CFRP/foam sandwich construction are not included in this report. They include the Voyager, which was the first airplane to fly around the World without refueling, the Pond Racer, the NASA AD-1 oblique wing research aircraft, the scale demonstration T-46, and the Starship.

The British Slingsby T67 Firefly aerobatic, training and sporting aircraft is an all-composite version of the French wooden Fornier RF6B. Slingsby has made GFRP high performance sailplanes for many years and used their sailplane experience to design the Firefly. The T67B gained CAA certification in September 1984 and about 200 had been sold to customers world wide by 1993.

The USAF ordered 113 Slingsby T67s, designated the T-3A, to fulfill the Enhanced Flight Screener (EFS) program. T-3As are being produced at a rate of 5 aircraft per month and are assembled by Northrop. FAA certification is being obtained to avoid the need for USA airworthiness testing. The T-3A airframe is GFRP with local reinforcement of CFRP. The structure has been modified to satisfy the USAF +6/-3g limit requirements carrying two 260 lb pilots (Jane's; Penney, 1993).

Sukhoi first flew the Su-26M single-seat aerobatic competition aircraft in June 1984 and gained both the men's and women's team prizes in the 1986 World Aerobatic Championships. The wing and empennage have CFRP and CFRP/AFRP skins. Composite materials comprise more than to 50% of the structure weight. The fuselage is made

		WING BOX		FUSELAGE		STABILIZERS		CONTROL SURFACES			
AIRCRAFT	YEAR†	COVERS	SUBSTR	SHELL	INTERNAL	HORIZ*	VERTICAL	ELEVATORS	RUDDERS	AILERONS	FLAP
PRIVATE, TRAINER & COMPETION AIRCRAFT											
Windecker Eagle	1967	L	L	L	L	L	L	L	L	L	L
Bellanca Skyrocket II	1975	L	L	L	L	L	L	L	L	L	L
FFT Speed Canard	1980	P	P	P	P	P	P		P	P	P
Slingsby T–3A	1983	P	P	P	P	P	P	P	P	P	P
Sukhoi Su–26M	1984	P	P			P	P	P	P	P	P
Rutan Voyager	1984	D	D	D	D	D	D	D	D	D	D
Avtek 400	1984	L	L	L	L	L	L	L	L	L	L
Egrett D–500	1987	L	L	L	L	L	L	L	L	L	L
Rutan AT3	1987	D	D	D	D	D	D	D	D	D	D
Rutan ARES	1990	D	D	D	D	D	D	D	D	D	D
Rutan Pond Racer	1991	D	D	D	D	D	D	D	D	D	D
Sukhoi Su–31	1991	P	P	P		P	P	P	P	P	P
FFT Eurotrainer	1991	P	P	P	P	P	P	P	P	P	P
Grob GF 200	1992	P	P	P	P	P	P	P	P	P	P
RI/DASA Fan Ranger	1993	L	L	L	L	L	L	L	L	L	L
TOTALS		15	15	14	13	15	15	14	15	15	15
RPVs & DRONES											
Ryan BQM–34E (NADC)	1971	D	D								
Boeing YQM–94A	1976	D	D	D	D	D	D	D	D	D	D
Rockwell HiMAT	1980	D	D	D	D	D	D	D	D	D	D
Ryan BQM–34F DAST	1983	D									
Boeing Condor	1991	D	D	D	D	D	D	D	D	D	D
TOTALS		5	4	3	3	3	3	3	3	3	3

* Including canards
† Year of first flight

P = Production
L = Limited Production
D = Development

Table A-6. - Composite components on private, trainer, competition, RPV, and drone aircraft.

from welded stainless steel tubing with removable composite skin panels, and the landing gear is titanium. Operating g limits are +12/–10 and the aircraft was designed to an ultimate load factor of +23. The Sukhoi Su-29 is a larger two-seat trainer version of the Su–26M, designed to limit load factors of +11/–9 solo and +9/–7 dual. The Su–31 is an all-composites higher performance follow-on to the Su–26M and first flew in 1991 (Jane's; Smith, 1993).

The German Grob company is producing various all-composite airplanes made primarily of GFRP. Grob was also a partner in the E-Systems/ Grob/Garrett Egrett-1 high altitude surveillance aircraft project, which was terminated in 1993. The company is headed by Dr. Burkhart Grob, and, like Slingsby, has had many years of experience designing and manufacturing GFRP sailplanes.

The Grob G 115 two seat light aircraft has "conventional" GFRP structure and was certificated to FAR Part 23 standards by the German LBA in 1987 and by the British CAA in 1988. It now has FAA certification. Grob is currently developing the GF 200 all-composite four seat light aircraft.

FFT, another German company, has been producing all-composite light aircraft for many years. Their FFT Speed Canard two-seat sporting aircraft first flew in 1980 and is certificated in many countries including the US (Jane's).

The FFT Eurotrainer is a two seat trainer. The airframe is primarily GFRP reinforced with CFRP. The first Eurotrainer flew in 1991 and obtained certification in 1992 (Jane's).

The last trainer aircraft listed in Table A-5 is the Rockwell International/Deutsche Aerospace Ranger 2000. Rockwell teamed with DA to develop the Ranger 2000 for JPATS. The airplane is a derivative of the German RFB Fantrainer that was first produced in 1984. The airframe is almost entirely made from GFRP with CFRP reinforcement (Jane's; AW&ST, Sep 13, 1993).

REMOTELY PILOTED RESEARCH VEHICLES AND DRONES

Various remotely piloted research vehicles (RPRVs) have been made from advanced composite materials to demonstrate performance. Some of the US RPRVs and drones are listed in Table A-6.

The Boeing YQM-94A Compass Cope was an Air Force long range, high-altitude, unmanned reconnaissance vehicle made almost entirely of glass/epoxy with some Kevlar®/epoxy. The vehicle, which had a wing span of 94 ft, had an endurance of 30 hours at 50,000 to 70,000 ft. Two aircraft were made. The first one flew on July 28, 1973, but it was destroyed in a crash nine days later. The second aircraft completed a successful flight test program in 1974 and is now in the Air Force Museum. The YQM-94A never went into production (Bowers, 1989).

An NADC program included design, fabrication, and flight test of CFRP wings using a Ryan BQM-34E supersonic drone as the baseline vehicle. Five ship sets of wings were fabricated and the first was proof-tested to 120% design limit load (DLL) for one critical 5g maneuver condition before flight. The other wings were proof tested to 100% DLL before flight (Manno, 1977; McQuillen, 1971).

The wings were deployed at Pacific and Atlantic ranges and flown on operational BQM-34Es starting in 1976. After 10 flights or three years of service, the wings were returned to NADC for dissection and small specimen testing (Manno, 1977;

The NASA HiMAT (Highly Maneuverable Aircraft Technology) RPRV was designed and built by Rockwell International to demonstrate improved transonic maneuver performance using aeroelastic tailoring. The HiMAT was a 0.44 scale model of a 17,000 lb fighter and was designed to a limit load factor of 12g. Almost the whole airframe was made out of CFRP and the anisotropic properties of the CFRP wing covers were used to provide aeroelastic tailoring (Monaghan, 1981; DeAngelis, 1982).

The NASA DAST (Drones for Aerodynamic and Structural Testing) program utilized a Ryan Firebee II BQM-34T target drone aircraft. The CFRP wing skins were mechanically attached to metal substructure and were purposely designed to have fiber controlled bending strength and stiffness but matrix-controlled torsional stiffness and shear strength. The wing was designed to encounter flutter within the flight envelope so that an active flutter control system could be investigated (Eckstrom, 1983).

During the third test flight of the DAST, divergent oscillations occurred with the flutter suppression system on. The wing failed and the aircraft crashed. The primary wing components were recovered and flight testing was resumed in 1982. Although active flutter suppression was effective, the study indicated that structural tailoring using matrix-dominated properties should be avoided (DeAngelis, 1982).

Boeing Advanced Systems designed a large twin-engine robotic aircraft in 1988 under a DARPA program. Nicknamed the "Condor", the HALE (High Altitude Long Endurance) aircraft set a high altitude record for piston engine aircraft at 66,980 ft in 1989. The "Condor" has a wing span of more than 200 ft and has an all-bonded airframe made almost entirely out of carbon/aramid/epoxy hybrid materials (Bowers, 1989).

HELICOPTER APPLICATIONS

Composite materials have been used for many different helicopter components including rotor blades, stabilizers, and fuselage structure. Many of the helicopters and their associated composite structural components are listed in Table A-7.

Rotor Blades. – In 1959, the Vertol Aircraft Corporation (previously the Piasecki Helicopter Corporation and later Boeing Vertol and Boeing Helicopters) started development of an "Optimum Pitch Blade" for the XCH-47 twin-rotor helicopter. These blades were made from E–glass/epoxy and survived a 150 hour whirl test. This success led to fabrication of ten CH-47 GFRP blades for static, fatigue, and flight tests in 1964.

The CH-47 blade test program was followed by the successful completion of a Navy-funded GFRP production blade development program. By the mid-1970s, GFRP blades had essentially replaced all 4130 steel spar blades on Boeing helicopters. The GFRP blades have a service life of at least 10,000 hours compared with a life of about 1,000 hours for the blades with steel spars. Boeing had made more than 10,000 GFRP blades for the CH-46 and CH-47 by the end of 1992.

About the same time that Vertol was developing GFRP blades, Messerschmitt Bölkow-Blohm (MBB) developed GFRP blades for the hingeless, semi-rigid rotor system for the Bo-105 helicopter. Initial flight tests of the rotor system were made using a Sud-Aviation Alouette helicopter and the first flight of a Bo-105 was made in 1967. The Bo-105 was still being produced in 1993 (Jane's).

Boeing Vertol and MBB reached a cooperative agreement for Boeing to utilize the MBB system and GFRP blade design for their Model 179, YUH-61A helicopter, which flew in November 1974. The tail rotor also had GFRP blades. The YUH-61A was the Boeing Vertol contender for the DoD UTTAS (Utility Tactical Transport Aircraft System) competition, which was won by Sikorsky with the UH-60A in 1976 (Air Int, Aug 1975).

Boeing Helicopters is responsible for the five-blade main rotor system of the Boeing/ Sikorsky RAH-66A Comanche that is scheduled to fly in 1994. Boeing is using a version of the MBB all-composite bearingless system (Jane's).

Bell Helicopters and Kaman also began to develop all-GFRP blades in the late 1960s to replace metal-spar blades. Bell blades had D–shaped aluminum spars with bonded aluminum skins; Kaman blades had aluminum spars with GFRP skins. Kaman introduced all-composite blades on the SH-2G in 1987. These blades have a service life of 10,000 hours. (Jane's; Rosato, 1969)

By the mid-1970s, Bell started producing GFRP blades for the AH-1 Huey Cobra, Model 214, and Model 222, but retained aluminum for the tail rotor blades. Many Bell AH-1 models were produced or retrofitted with Kaman-designed K-747 GFRP rotor blades between 1977 and 1988. These blades also have a service life of the order of 10,000 hours. (Jane's; Peacock, 1988)

Bell won the OH-58D, US Army Helicopter Improvement Program (AHIP), in 1981. These aircraft have the Bell four-blade rotor systems with CFRP yokes, GFRP blades, and elastomeric bearings. The first OH-58D flew in 1983 and deliveries started in December 1985. The Bell Model 406, a lighter version of the OH-58D, first flew in 1990. The Bell Model 680 4-blade rotor system ("Rotor 90") is almost entirely composites. It is 15% lighter and has 50% fewer parts that the current system (Jane's).

During the early 1970s, Sikorsky developed main rotor blades composed of hollow titanium spars with GFRP/honeycomb sandwich trailing

AIRCRAFT	YEAR†	ROTOR BLADES MAIN	ROTOR BLADES TAIL	FUSELAGE MIDSECTION	FUSELAGE TAILBOOM	FUSELAGE CANOPY FRAME	FUSELAGE INTERNAL	STABILIZERS HORIZ	STABILIZERS VERTICAL	OTHER SPONSONS	OTHER DR SHAFT
Sikorsky S-61	1959		D[1]			P					
Sikorsky CH-53	1965			D[2]		P				P	
MBB Bo 105	1967	P									
Sikorsky CH-54	1971				D[3]						
Westland Wasp	1971				D						D
Boeing CH-46/CH-47	1974	P									
Bell AH-1F	1974	P									
Sikorsky YUH/UH-60	1974/75			P		P	P	P			
Boeing YUH-61A	1974	D	D			D	D				
MD YAH-64/AH-64	1975/84	P									
Sikorsky S-76	1977			P		P	P	P			
Aerospatiale AS-365	1979	P							P		
Westland Sea King	1979	P	P			P					
BK-117	1979	P	P					P	P		
Aerospatiale AS 332L	1980	P			P			P		P	
Bell 206L	1981								P		
Bell 412	1981	P									
Kaman SH-2G	1981	P									
Kamov Ka-32	1981	P		P		P		P	P[4]		
Mil Mi-28	1982	P	P								
Agusta 129	1983	P	P		P	P	P	P	P	P	
Sikorsky S-75 (ACAP)	1985			D	D	D	D	D	D	D	
Westland Lynx	1986	P	P						P[5]		
Bell 292 (ACAP)	1986	D		D	D	D	D	D	D	D	
Boeing 360	1987	D		D		D	D			D	D
Aerospatiale AS 565	1987	P	P	P	P	P	P	P	P	P	
EH Industries EH 101	1987	P	P	P		P	P	P	P		
Mil Mi-34	1987	P		P		P	P				
MBB Bo 108	1988	P	P	P	P	P	P	P	P	P	
MD MD 520N/530N	1990	P	none		P						
Eurocopter Tiger	1992	P	P	P	P	P	P	P	P	P	
MD MD 900	1992	P	none	P	P	P	P	P	P		
Kamov Ka-62	1994	P	P	P	P	P	P	P	P		
Boeing/Sikorshy AH-66	1995	P		P	P	P	P	P	P	P	P
TOTALS		25	11	15	13	19	15	15	14	10	3

† Year of first flight or completion of R&D program

1. In 1961 2. Cargo Ramp 3. B/ep reinforced 4. Rudders 5. Partial

P = Production D = Development

Table A-7.– Composite components on helicopters.

edges. The spars were pressurized to check the integrity of the blades. These blades are used for the military UH-60 Black Hawk and the civil S-76. Sikorsky began development of a new all-composite blade for the Black Hawk in 1991. Flight tests began in October 1993. The new blades are expected to have a service life of about 10,000 hours compared to 1,900 hours for the blades with titanium spars. Full scale production should begin in 1996 (Kandebo, 1993b).

McDonnell Douglas (previously Hughes Aircraft Company) uses aluminum skins bonded to extruded aluminum spars for their small helicopter main rotor blades. The blades of the AH-64 Apache attack helicopter, which entered service in 1984, are made from GFRP tubes with stainless steel leading edges and GFRP trailing edges (Jane's).

During the past few years McDonnell Douglas has been developing a low-noise five-blade rotor system with carbon/epoxy blades. The company-funded HARP (Helicopter Advanced Rotor Program) flexbeam CFRP rotor was first flown on an MD 500E helicopter. This rotor is used in combination with the NOTAR (No Tail Rotor) system for the MD 520N and MD 900 transport/utility helicopters that first flew in 1990 and 1992, respectively. The McDonnell Douglas NOTAR system was first flown on the MD 530N in 1989 and the first production MD 520N was delivered in October 1991. With the NOTAR system, the MD 520N and MD 900 are 50% quieter than comparable helicopters (Jane's; Proctor, 1993).

Westland started production of the Lynx in the UK in 1972 under a cooperative agreement with Sud Aviation. The Lynx had rotor blades with titanium spars, based on the Sikorsky design (Westland was licensed to produce various Sikorsky helicopter models). The Sea King Mk 2, British derivative of the Sikorsky S-61D0 and the Lynx AH Mk 9, an upgraded version of the Lynx, were the first Westland helicopters to use GFRP main and tail rotor blades. The Lynx composite blades, developed under the British Experimental Rotor Programme (BERP), were first flown in 1986, and the Lynx established a world helicopter speed record of 249 mph (Jane's; Gething, 1990).

The blades of the EH Industries EH-101 are scaled-up versions of the BERP Lynx blades, designed and built by Westland. EH Industries is a Westland/Agusta partnership. The EH-101, a large military and commercial general purpose helicopter, was first flown in 1987. Production aircraft are currently being delivered to British and Italian forces (Jane's).

In France, Sud Aviation (later Aerospatiale and now part of Eurocopter) first introduced GFRP main rotor blades on the SA 341 Gazelle, which was first flown in 1967. Tail rotor blades were aluminum. The blades had a GFRP spar and skins supported by a foam core. The current version of the Gazelle, the SA 342, was still being produced in 1992 (Jane's).

By 1990, Aerospatiale was using GFRP blades with a CFRP hub for the main rotor system and CFRP blades for the ducted fan of the AS 365 Dauphin 2 helicopter (Jane's).

MBB teamed with Kawasaki to design and produce the BK 117 multipurpose helicopter, which made its maiden flight in June 1979. The BK 117 main rotor system is similar to the MBB Bo 105 system with hingeless GFRP blades. The BK 117A was certified to FAR Part 29 requirements in Germany, Japan, and the US in December 1982, and the BK 117B model was certified by the LBA, JCAB, and the FAA in 1987/1988. The BK 117 is being produced in Germany, Japan, Canada, and Indonesia (Jane's; Air Int, Apr 1989).

Some of the other helicopters that utilize GFRP or mixed CFRP/GFRP rotor blades are the Franco-German Eurocopter (DA/MBB with Aerospatiale) Bo 108 and the PAH-2 Tiger.

The Italian Agusta A 109 and A 129 helicopter blades have AFRP spars with GFRP skins, Nomex® honeycomb core, and stainless steel leading edge abrasion strips (Jane's).

All-composite blades were first produced in Russia for Mil Mi-28 combat helicopter, which first flew in 1982. The blades are made from a mix of CFRP, GFRP, and AFRP composites with a Nomex®-type core. All the latest Russian helicopters, which include the Kamov Ka-32 and Ka-62, and the Mi-34 and Mi-38, have all-composite blades (Jane's; Fricker, 1990).

Helicopter Airframes. – During the past 35 years, use of composite materials in helicopter structures has grown from a few small access panels and canopy frames to almost all of the airframe. Composites have provided weight savings, which is particularly important to helicopter performance, as well as improved durability and corrosion resistance, and reduced drag.

Sikorsky started using GFRP materials for fairings and secondary structure of the S-61 in 1959. Use was extended to the canopy frame of the CH-53 in 1965, when GFRP materials accounted for about 5% of the structure weight.

In 1971, Sikorsky completed a NASA program that used boron/epoxy to reinforce aluminum components on the rear fuselage of the CH-54B Skycrane. Static and fatigue tests satisfied strength and life requirements and the reinforced structure was 130 pounds lighter than the aluminum baseline. A reinforced rear fuselage was put into flight service on an Army helicopter in April 1972. The helicopter was taken out of service in after it was severely damaged in a wind storm. The tail boom was not damaged and the helicopter was subsequently repaired and put back into service with the National Guard. The boron-reinforced structure approach has not been used for any subsequent Sikorsky helicopters (Rich, 1972).

Second generation composite structures included horizontal stabilizers, fuselage panels, floors, doors, and the stabilizer of the S-76 civil helicopter in 1977 and the UH-60A Black Hawk in 1978. The combination of 100 lb airframe weight savings and reduction in drag from the flush external smoothness of the fuselage increased range by about 20% (Ray, 1982).

A joint NASA/Army for the development and flight service of helicopter components complemented the NASA ACEE program for transport aircraft structures and provided additional confidence in composites. The helicopter program, which was started in 1979, included flight service of 14 Sikorsky S-76 horizontal stabilizers and rotors; 160 Bell 206L fairings, doors and vertical fins, and a cargo ramp skin for the Sikorsky CH-53 (NASA CP-2321, 1984).

Both the MBB/Kawasaki BK 117 and the Aerospatiale Dauphin (US designation HH-65A) utilize composites for horizontal and vertical stabilizers, doors, floors, etc. (Jane's).

Third generation composite structures, which use a mix of CFRP, GFRP, and AFRP for almost all of the airframe, were first demonstrated by the US Army Sikorsky S-75 Advanced Composite Aircraft Program (ACAP) and the Bell Model 292 ACAP in 1985/1986. Boeing separately developed the Model 360 all-composite twin rotor helicopter, which flew in 1987. These helicopters were technology demonstrators and did not go into production, but their development led to extensive proposed use of composites on contenders for the Army Light Attack Helicopter (LHX). The LHX contract was won by the Boeing/Sikorsky team's RAH-66 Comanche in 1991. The first RAH-66 is scheduled to fly in August 1995. The RAH-66 fuselage is being made primarily of carbon/epoxy and aramid/epoxy and has about 350 parts compared to about 6,000 parts for the UH-60 fuselage (Jane's; Parker, 1993).

Other helicopters that have largely composites airframes include McDonnell Douglas MD 900 and the German MBB Bo 108, both of which are in production, as well as the Eurocopter Tiger and the Russian Kamov Ka-62. The composite fuselage of the Bo 108 has almost 30% less drag than the Bo 105 aluminum fuselage (Jane's).

The Tiger is an attack helicopter that is being developed by Eurocopter (the DA/Aerospatiale consortium) for the German and French armies. Deliveries are scheduled to begin in 1997 (Jane's; Mordoff, 1988).

The Kamov Ka-62 is a multi-purpose helicopter. Composites account for about 50% of the airframe weight and include the main cabin shell, floors, tailboom, stabilizers, fan duct, and fan blades (Jane's).

The Italian Agusta A 129 Mangusta attack helicopter airframe is a mix of composites and aluminum. The A 129 first flew in 1983 and entered service with the Italian Army Aviation in 1990. Some 900 lb of composite materials are used for the nosecone, canopy frame, tailboom, tail rotor pylon, engine nacelles, and the stabilizers, accounting for about 45% of the airframe weight (Jane's).

EH Industries was formed in 1980 by Westland Helicopters and Augusta. The EH 101 multi-role helicopter first flew in Italy 1987. British and Italian civil certification was expected in 1993. Military variants, scheduled to enter service in 1995, include naval and land-based helicopters for the British, Italian, and Canadian forces. Composites are used for the canopy frame, forward fuselage, vertical and horizontal stabilizers, upper cowlings, and engine inlets.

Composite materials, primarily GFRP or AFRP with CFRP reinforcement, have become the standard materials for helicopter stabilizers, engine doors, cowlings, fairings, doors, landing gear doors, floor panels, stub wings, sponsons, and fan ducts. McDonnell Douglas made some AH-64 stabilizers from carbon/thermoplastics, but they were not flown or put into production (Colucci, 1991).

OBSERVATIONS AND CONCLUSIONS

The following observations and conclusions are drawn from a review of the international aircraft programs described in this Appendix as well as from the interviews with industry and government personnel.

Advanced composites are being used extensively for primary and secondary structures of many new US and foreign military and commercial aircraft. The technical risks involved with the use of composites appear no greater than those associated with metals.

Overall weight savings have been achieved by using composites instead of metals. Component weight savings can be as high as 35%. Typically, composites make up between 22% and 35% of the airframe by weight for new US and foreign military tactical aircraft. The composite horizontal stabilizer on the Airbus A320 is 15% lighter than its aluminum counterpart and the ATR 72 outer wing saves 20% (Barrio Cardaba, 1990; Pilling, 1988).

Weight savings were the major consideration when Boeing decided to introduce advanced composites on the B757 and B767 in the late 1970s. The price of jet fuel had increased from $0.12/gal in 1973 to $1.04/gal in 1981 (Aerospace Facts & Figures; Bowers 1989). By 1991, the price of jet fuel had dropped to $0.69/gal, but the prices of commercial transport aircraft have increased by almost 500 percent since 1973. The cost of fuel dropped from 30% of cash operating expenses in 1981 to 14.8% in 1991. (Aerospace Facts & Figures) In today's business environment, weight savings are not marketable unless they can be accomplished at no additional cost.

Until the mid 1980s, Boeing was the only company utilizing advanced composite materials for spoilers, elevators, rudders, and flaps of commercial transport aircraft (B757, B767, B737-300). In 1985, Airbus moved ahead by adding the A310 composite vertical stabilizer and, in 1987, they added both the horizontal and vertical stabilizers of the A320 to their list of applications. The A330 and A340 also have composite stabilizers. As of December 1991, 247 A320s were in airline service.

The European aircraft community moved further ahead when ATR introduced advanced composite outer wings on the ATR 72 in 1988 and Deutsche Airbus selected CFRP for the rear fuselage and pressure bulkhead of the Dornier Do 328 in 1991. As of December 1991, 48 ATR 72s were in airline service.

Boeing is adding horizontal and vertical stabilizers and cabin floor beams to the list of composite components on the B777, which is due to fly next summer. By that time, Airbus will have an advantage of at least eight years of production and service experience of advanced composite stabilizers. On the other hand, Boeing may use of less advanced composites on their derivative 737 (737-X) than the 737-300 because smaller airlines, who are the potential customers for the 737–X, do not have the composite maintenance and repair capabilities of larger carriers.

McDonnell Douglas is using advanced composites for most of the control surfaces of the MD-11. The aileron is a post buckling design derived from the DC-10 rudders developed under the NASA ACEE program.

The NASA Aircraft Energy Efficiency Program (ACEE) Primary Aircraft Structures Program, which ran for about 15 years from 1972 until 1987, was very successful in demonstrating the technology readiness and cost effectiveness of composite structures for commercial transports (NAA CP-2321, 1984). In retrospect, the NASA flight service experience programs and the ACEE program had enormous influence on the acceptance of advanced composite structures by industry and the aircraft operators in the US and abroad. Many of the components developed under these programs (e.g., the DC-10 vertical stabilizer and rudders and the Boeing 737 horizontal stabilizers and spoilers) are still in scheduled airline service after almost 20 years. Successful service performance of the Boeing 737 composite spoilers and horizontal stabilizers and the 727 elevators provided Boeing with the confidence and technology needed to commit to the composite elevators and rudders for the B757 and B767, and lately, to the horizontal and vertical stabilizers for the B777.

The airlines, however, are very concerned about thin-skin honeycomb sandwich composite secondary structures. These parts often get damaged in service (generally during aircraft maintenance) and the costs of repair, replacement, or leasing spare parts are very high. As noted above, Boeing may change a number of secondary composite components back to metal on the new B737-X.

Costs are universally recognized as the biggest problem associated with composites use in place of conventional metal structures. Many of the military aircraft programs have had considerable cost overruns. Some cost increases are directly associated with composite structures and some with inexperience in the design and manufacture of composites. The costs of composites maintenance, repair, and replacement parts add to the overall cost problem. Regardless of type of material, the prices of military aircraft structures are much higher than those of comparable commercial and business aircraft structures. Based upon current program cost estimates, the fly-away prices of new military aircraft structures range from $1,300/lb for the McDonnell Douglas C-17 and $1,500/lb for the Lockheed F-22 to $4,000 for the Northrop B-2B. Prices will increase further if production rates and quantities are reduced. In contrast, the current prices of commercial turbojet and turboprop transport aircraft range from $200/lb to $300/lb (Aerospace Facts & Figures; Hadcock, 1985, 1989; McCarty, 1991).

As was the case with military aircraft, costs are also the biggest problem associated with composites use in place of conventional metal structures on commercial transports. Costs include production costs, as reflected in airplane prices, as well as in-service costs associated with component inspection, maintenance, repair, and replacement.

Nearly all the people interviewed thought that the prices of composite components are higher than their metal counterparts, and that, to be marketable, their prices should be comparable. The prices of current commercial transport airframes, based on aircraft prices, have remained relatively uniform at $200/lb to $300/lb (Whitehead, 1993; McCarty, 1991).

Allowing for a 20% weight saving, prices of installed composite structures should be competitive with metal structures in the $250/lb to $350/lb range. Component prices should be in the $200/lb to $300/lb range to allow for final assembly costs.

Many composite parts are supplied by coproducers or subcontractors under fixed price contracts. Since the price is rarely broken down into individual elements, the individual cost of a unit or ship set of composite components is impossible to obtain.

As an example, the Japan Aircraft Development Corp. (JADC) is contributing 21% to the Boeing 777 project for design and production of the fuselage, center wing, and wing-body fairings for the life of the 777 program. Grumman has a 10-year, $400 million contract to produce the 777 composite inboard flaps and spoilers, and Rockwell, CASA, HDH, and Alenia have contracts to produce other composite components (O'Lone, 1991).

Since the end of the Cold War and the cutback in defense spending, there will be fewer new military aircraft opportunities and the gestation time period will probably be longer than ten years. US companies (McDonnell Douglas, Boeing, Northrop, Lockheed, Grumman, and Vought) are still the world leaders in composites technology and production experience for high performance military aircraft. However, retention and transfer of technical information and experience will be a major problem in the future. Much more reliance will probably have to be placed on use of composite technology and materials developed for commercial aircraft.

Other than the Boeing 777 and 737-X, development of any new US commercial transport aircraft is unlikely during the rest of the century. The development of the Boeing 777 is almost complete, and it is improbable that any material changes will be made at this stage of the program. Although composites could be considered on derivative aircraft, Boeing has chosen to use fewer composites on the 737-X than they presently have on the 737-300 because of the problems the smaller airlines have with repair and maintenance of composites.

Realistically, the next major US opportunity for extensive composites use will be the High Speed Civil Transport (HSCT) aircraft, which desperately needs the weight savings provided by high temperature composites. The HSCT will probably not fly before 2005 (Whitehead, 1993; Blankenship, 1991).

With a few notable exceptions, use of composites on general aviation (GA) airplanes has been very conservative and has been limited to control surfaces, flaps, fairings, landing gear doors, and engine nacelles. Certification for these components has been relatively straightforward and has not required structural testing of the complete airframe. Generally, in-service performance has been trouble free.

Five different all-composite GA airplanes have been designed and built in the US during the past twenty years. None has been an unqualified success. Private aircraft include the Windecker Eagle and the Bellanca Model 25 Skyrocket II. Business aircraft include the LearFan 2100, the Beech Model 2000 Starship, and the Avtek 400A.

FAA airframe certification was granted to the Windecker Eagle, LearFan, and Starship after lengthy and expensive test programs. The Bellanca Skyrocket was never certificated and the certification program for the Avtek 400, which first flew in 1984, has yet to be completed.

LearFan went out of business in 1985 following a number of program delays and certification problems. Two flight-test and one structural-test aircraft were built (Jane's). Windecker Research ceased operations in 1976 after completion of 8 aircraft (Jane's; Simpson, 1991).

The all-composite (primarily GFRP) British Slingsby T.67 Firefly (USAF T-3A) and the German Grob G-115 civil/military trainers appear to have avoided the financial problems of the Eagle and Skyrocket. Both aircraft obtained civil certification in the mid-1980's and combined civil and military sales have been about 320 for the Firefly and 100 for the G-115. Prior to the introduction of these trainer aircraft, Slingsby and Grob had extensive design and manufacturing experience producing high-performance GFRP sailplanes.

Avtek Corporation is building two ground test aircraft for FAA certification tests to FAR Part 23 requirements, but will require about $70-million to complete the flight tests and structural tests needed for the FAR Part 23 certification program.*

Beech Aircraft Corporation first flew the prototype Starship in 1986 following the flight-test program on an 85% scale proof-of-concept (POC) aircraft. The full-scale aircraft did not have the performance projected from the POC program. Between 1984 and 1990, structure, empty, and takeoff weights increased by 4%, 29%, and 19%, respectively. These weight increases reduced cruise speed, fuel efficiency, and range.

The airframe of the Starship, which is 67 percent composites by weight, has been blamed for the poor performance of the aircraft. It is approximately the same price as the Piaggio Avanti and Beechjet ($4-million), but is slower and has a higher approach speed than the other aircraft. The poor performance appears to have been caused by the aerodynamics of the unconventional configuration of the aircraft and by systems weight growth, and not by the weight of the composite airframe, which is only 27 per cent of takeoff gross weight (Abbott, 1986; Aerospace Eng., 1990).

In retrospect, the late 1970s and early 1980s were inopportune times for Windecker, Bellanca, LearFan, and Avtek to enter the GA marketplace. Their composite aircraft were introduced during a time when sales of GA aircraft were on the decline. In the case of US single-engine piston aircraft, sales dropped from 14,400 aircraft in 1978 to 1811 aircraft in 1983 and only 564 aircraft in 1991. In the case of twin-engine turboprop aircraft, sales dropped from 918 airplanes in 1981 to 222 aircraft in 1991 (Aerospace Facts & Figures). The newly formed companies were trying to enter this declining market with new aircraft designs made from unconventional materials. Their competitors (Beech, Cessna, Piper) had extensive aluminum airplane design and production experience, world-class reputations, and world-wide sales and support organizations.

The huge costs of full-scale engineering development and associated wind tunnel, structural, and flight testing, as well as the delays and modifications associated with certification of new

*Private communication from Robert Adikes, Avtek Corp., September 1993.

aircraft types, were not fully anticipated. These delays caused lost orders and compounded financial problems. None of the new companies had an established product line that could carry them through the lengthy development programs. Their future depended on timely delivery of their promised product. As customer confidence faded, so did their financial support.

The Beech Starship might have had a niche in the market if its performance had come up to expectations and if the program had not been delayed by structural certification problems. As it happened, the Starship seems to have ended up as a competitor to the Beechjet and the Piaggio Avanti, both of which are in the same $4 million price range and have much better performance.

Lindon Blue, president of Beech during the early 1980's when the decision was made to go ahead with the Starship program, seemed to have anticipated the Starship problems when he wrote:

'As to the execution, weights must be forced to fulfill the composite promise of 20-30 per cent, surfaces must be mirror- smooth and yield laminar flow, attention to producibility and economy must start when the CAD-CAM CRT is first switched on. Absence of any of these critical points of concentration .... will result in a product that will probably be a market bummer even if it is fortunate to get past the prototype stage.' (Blue, 1985).

The use of composite (primarily GFRP) rotor blades on helicopters has raised blade operational life from between 1,000 and 3,000 hours to 10,000 or more flight hours. Composite blades can be designed to be "fail-soft" and do not require as frequent inspection for cracks as do metal blades. In addition, blade and rotor system efficiencies have been improved because of the tailorability of composites. Composite blades are generally no lighter and are more expensive than their metal-spar counterparts. However, their longer life and reduced in-service inspection requirements make them very attractive and cost-effective for both military and civil helicopters.

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13. ABSTRACT (Maximum 200 words)

A study of past composite aircraft structures programs was conducted to determine the lessons learned during the programs. The study focused on finding major underlying principles and practices that experience showed have significant effects on the development process and should be recognized and understood by those responsible for using of composites. Published information on programs was reviewed and interviews were conducted with personnel associated with current and past major development programs. In all, interviews were conducted with about 56 people representing 32 organizations. Most of the people interviewed have been involved in the engineering and manufacturing development of composites for the past 20 to 25 years. Although composites technology has made great advances over the past 30 years, the effective application of composites to aircraft is still a complex problem that requires experienced personnel with special knowledge. All disciplines involved in the development process must work together in real time to minimize risk and assure total product quality and performance at acceptable costs. The most successful programs have made effective use of integrated, collocated, concurrent engineering teams, and most often used well-planned, systematic, development efforts wherein the design and manufacturing processes are validated in a step-by-step or "building block" approach. Such approachs reduces program risk and are cost effective.

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