NEW MATERIALS AND STRUCTURES

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Abstract

The history of aircraft structural failures, particularly those resulting from fatigue, have prompted researchers to develop both new materials and structural forms to reduce the effects of fatigue. The development and refinement of fracture mechanics and crack propagation theory have enabled a fairly rigorous definition of the problem. The results are metals having slower crack growth rates, new design concepts for metals such that the historical problems are circumvented, and the new advanced composite materials exhibiting different properties and damage characteristics. The application of these new materials, particularly the composites, will result in new design techniques; the composites require that the designer add a new dimension to his design, that of designing the material.

I. INTRODUCTION

From the beginning of the industrial revolution, one of the problems that has consistently plagued designers has been that of fatigue. Probably the earliest well documented fatigue problem was that of the German railroad axles. In recent time the aircraft industry has experienced numerous instances of fatigue. Some of the more notable ones have been the experiences of the Comet, the F-111, and the C-5. Figure 1 graphically depicts what the problem is. This figure shows the crack that precipitated the catastrophic failure of the wing of one of the F-111 aircraft during flight. Note the small size of the initial flaw. The material was D-6AC steel, a very high strength steel, but unfortunately a steel that was extremely notch sensitive. The very short critical crack length of this steel resulted in a catastrophic failure from an extremely small initial flaw size. In this case, the flaw, internal to the wing, could not have been detected by nondestructive inspection techniques. Not only was the flaw internal, and it did not penetrate to the outer edge of the steel, but the outer edge of the steel was covered with an aerodynamic fairing consisting of a thin aluminum skin bonded to an aluminum honeycomb, in turn bonded to the steel structure beneath it. Because of the historical perspective of this type of problem, continuing research efforts over the years have been expended to develop new materials and design concepts that will enable aircraft designers to design structures having greater damage tolerance and durability. Research efforts have been proceeding in two directions; one in attempting to modify existing materials, i.e., metals, such that they are considerably more crack resistant (metals having a much greater toughness and consequently greater critical crack length and metals which have slower rate of crack growth), especially in agressive environments, and to develop new design concepts such that the cuase of past failures are reduced in magnitude through improved design concepts. Research efforts have also been expended in development of new materials. Notable among the new materials are the advanced composites. This paper looks briefly at first the metals and then the advanced composites.

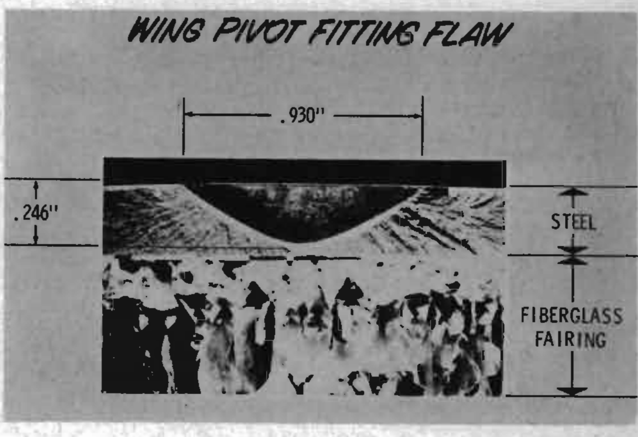


FIGURE 1

II. NEW METALS AND STRUCTURES

In recent years, the metallurgists have developed several new alloys that show great promise for aircraft structures. To overcome the fatigue problems of the high strength 7075 aluminum alloy, we now see 7050 and 7425 aluminum. 7425 aluminum has a K_{IC} value of approximately 40 to 50 KSI $\sqrt{\text{inch}}$. We see improvements in the titanium alloys; the recrystallized annealed condition titanium 6-4 shows characteristics of approximately 80 to 110 KSI $\sqrt{\text{inch}}$. And in the high strength steel, the 10 nickel steel has the value of 200 to almost 500 KSI $\sqrt{\text{inch}}$. These materials with their large K_{IC} portend long growth times to critical size. In comparison, the D-6AC steel that was used in the F-111, although it has a very high strength and can be heat treated in the order of 220 to 240 KSI, has a K_{IC} value of only approximately 90 KSI $\sqrt{\text{inch}}$. The achievement of that greater K_{IC} has been achieved at the cost of ultimate tensile strength. In addition to changing the basic characteristics of the materials, alternate design concepts are also being developed. Rather than using thick sections of metals, the use of numerous thin sheets of metal bonded together changes the overall characteristics of failure from that of plain strain to plain stress, and, in addition, a failure occurring in any one of the plates, as it approaches the bond line, i.e., the adhesive line, is effectively slowed down and prevented from transferring to the adjacent plate. Other concepts being pursued include the use of spar webs integral with covers or spar caps bonded to covers eliminating or reducing the need for the single shear bolts between covers and spars thereby reducing the number of stress concentrations.

III. ADVANCED COMPOSITES AND STRUCTURES

The desire to circumvent the cracking problem of metals was certainly one of the driving forces in the development of the composite materials. Early research in whiskers demonstrated that indeed fibrous materials could be manufactured which exhibited superior strength and stiffness; this together with the experience gained using fiberglass, and the desire to improve the modulus of the fiberglass, lent impetus to the development of the advanced composites. The advanced composites make use of some of the lightest elements known. That light weight, together with the high strength achievable through the use of these materials in the whisker or fiber form, adds to their attractiveness as a structural material. The specific strength and the specific modulus of the advanced composites considerably exceed the similar properties of the metals; the aluminums, titaniums and steels. Figure 2 graphically depicts how the efficiency of the advanced composite materials compares with other materials used in aircraft structures. The first advanced composite filament was boron, an element characterized by light weight, and high strength and stiffness. In the early 1960's, the development of boron fiber, i.e., boron deposited on a tungsten substrate, presented the aircraft designer with a new material exhibiting very high strength, high modulus, and very light weight; and indeed the goal of increased modulus was achieved with a sixfold increase over the fiberglass. The early boron fibers were, however, very expensive. Continuing fiber research resulted in the development of graphite fibers. The graphite fibers initially were lower in cost and, although they did not exhibit quite the efficiency of the boron fiber, they nevertheless made possible the development of aerospace materials with very attractive properties. The cost of both fibers has dropped considerably in recent years. Projections of graphite fiber costs predict a continued decline while the cost of boron has probably reached a minimum unless there is a marked increase in demand for the boron fiber. The steps used to create an aircraft structure from the advanced composite materials are as shown in Figure 3. In the upper left-hand corner we see the fiber manufacture; in the case of boron, a very fine tungsten wire is run through a reactor in which boron is deposited on that tungsten substrate. Likewise, the graphite is formed by running a precursor fiber through a reactor in which heat first carbonizes the precursor and then continued exposure at higher temperatures graphitizes that carbon fiber. The resulting fibers from either process are wound on spools. The boron fiber is 4 mil in diameter; the graphite fibers, on the other hand, are exceedingly small in diameter, and come from the reactor in bundles or tows, i.e., groups of very fine fibers. The resulting fibers, wound on spools, are then used in the formation of a tape, i.e., the impregnation of the fiber, either the boron or graphite, with an epoxy material as shown in the upper right-hand part of Figure 3. We now see a whole spectrum of matrix materials whose strengths, elasticity and temperature capabilities vary widely; and of course that variance is reflected in their costs. In addition there are varying degrees of compatibility between the various matrix materials requiring judicious selection if it is necessary to mix materials; this is particularly true when designing a hybrid material, i.e., a material, or structure, having both boron and graphite plies or some other composite combination. As shown, groups of fibers are run through an impregnating system. In the case of boron, the fibers (one fiber diameter in thickness laid across a very fine fiberglass substrate) are impregnated with an epoxy material. In the case of graphite, the bundles or tows of fibers are combed out and impregnated with the matrix material, generally the graphite/epoxy tape is 5 to 6 mils thick; and in either case, the resulting tape with a suitable backing material is wound on a spool. That tape then is used in the layup of the desired aircraft parts. The use of tape now permits the designer to not only design the part, but to design the material, for now the designer can choose to use material in those areas where there are high stresses and to reduce the amount of material where the stresses are lower (or to change the orientation or quantity of fibers as the primary load paths change). The designer must carefully consider how he uses that tape, for the tape in the direction of the fiber is quite strong, while across the fiber (or, in other words, tending to pull the fiber away from the one adjacent to it) we find that the tape is characterized by the strength of the matrix material itself. Consequently, we find that there are different strengths depending on the orientation of load upon the tape. It is from this base that the design of the material evolves. The difference in strength is reflected in different SN curves, as shown in Figures 4 and 5. The fiber-dominated laminate, i.e., a laminate in which there are sufficient plies of zero degree fibers, fibers running parallel to the load, exhibits an SN curve that is virtually flat as shown in Figure 4; i.e., very slight decreases in loading magnitude give greatly extended fatigue life. In the matrix-dominated, or orthotropic, laminate a $\pm 45^{\circ}$ boron epoxy laminate, as Figure 5 presents, where the load is acting in large part against the matrix material, we find a somewhat steeper SN curve and a considerably lower strength. This in part highlights one of the problems for the designer when using the advanced composite materials. He must choose the orientation which best suits the need of the structure. Maximum strength of course would be achieved if all the fibers ran in the one direction; however, we know that there are seldom, if ever, cases in which the actual stresses in a structure are limited to one direction. Consequently, the designer must choose laminate configurations that give him the strength and fatigue life he needs. This means that there are cases in which he is going to vary the laminate orientation considerably. Changes in orientation and in thickness will occur throughout a structure, throughout a wing skin for example, as the bending moment and torsional loads change over the span.

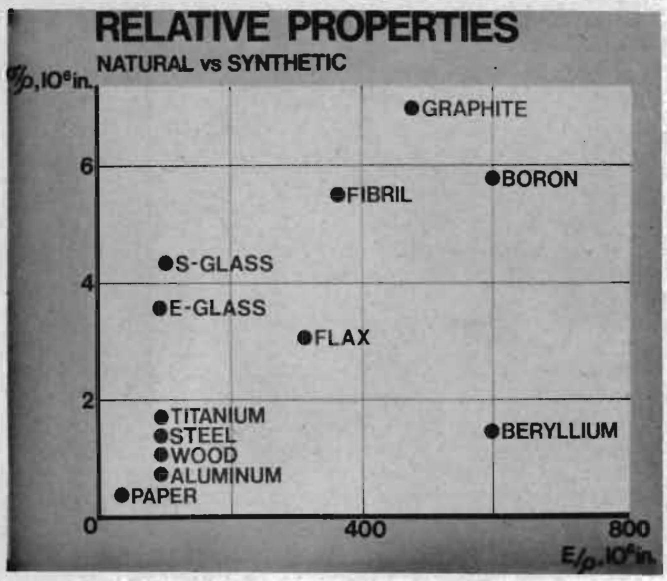


FIGURE 2

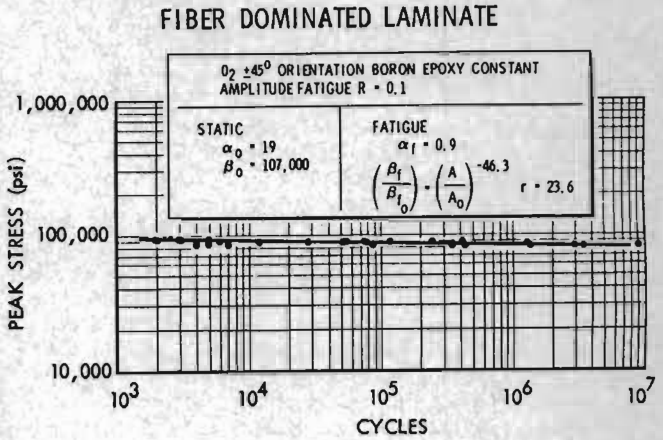


FIGURE 4

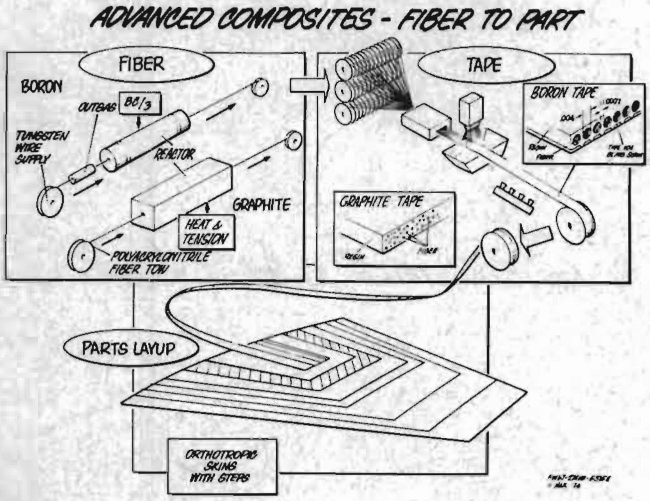


FIGURE 3

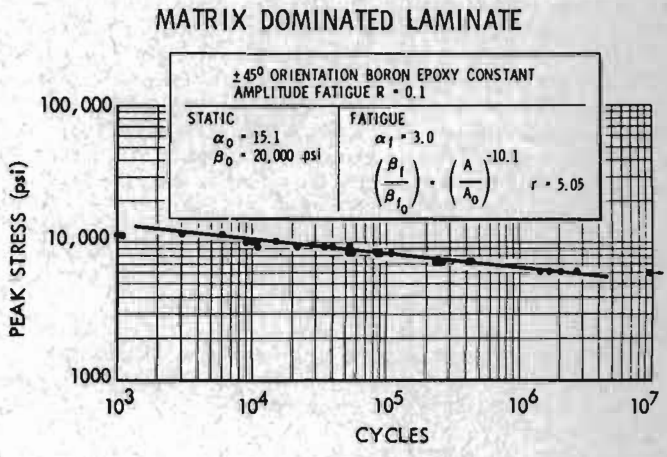


FIGURE 5

One of the more intriguing characteristics of the composite materials is their damage characteristics. The composite materials do not, as metals, exhibit a discrete crack formation and propagation under fatigue loading. Rather, the composite materials develop areas of damage at stress concentrations in highly stressed areas under repeated loading. The action of fatigue loading, when that loading is at a stress level approaching that of the ultimate strength of the material, tends to break down the matrix in the area of stress concentrations, but that very breakdown of matrix has the effect of redistributing the load in the general area of damage, reducing the load levels, and in a sense, erasing the effects of the stress concentration. The results of this can be seen in Figure 6. As noted, the average static strength of an unnotched specimen is as shown. Other specimens with various size stress concentrations have a lower static strength as a result of the stress concentrations. The composite materials are sensitive to the size of a stress concentration as depicted in Figure 7, although as can be seen the effect of size is non-linear and does diminish as the size of the hole increases. However, after being subjected to fatigue cycling, it is rather obvious that all the specimens have essentially the same net area strengths, even though some have a stress concentration. This is a direct result of the action of the high load cycles on the stress concentration, (the breakdown of the matrix material, the redistribution of those loads, and the actual increase of the static strength of the materials). This then indicates that the material will have a very long fatigue life. And indeed we have seen this to be true; constant amplitude fatigue tests conducted on both notched and unnotched specimens at stress levels approaching that of static failure has resulted in very good fatigue life if the specimen did not fail on the first load cycle. This then points up another characteristic of the advanced composites, that of a rather broad variability compared to the metals. In the aluminums we see a Weibul shape parameter of about 4.0 for fatigue scatter while the graphite epoxy exhibits an α of about 1.2 to 1.5; however, the number of cycles which can be sustained by the graphite/epoxy is about three orders of magnitude greater than the aluminum (this is shown in Figure 8). Consequently, even though the variability of the composite is great, its greater fatigue life makes it a more reliable aerospace structural material.

One of the potential benefits of the advanced composites appears to be the ability to design damage tolerance or crack arrestment into the structure. Because of the simultaneous changes in strength, fracture toughness, and modulus of the laminate due to the angle at which the plies are laid up, we can design areas of different properties in an otherwise constant thickness laminate. Out of research in the design of joints in the advanced composites was developed the concept of softening strips, i.e., the design of a laminate with areas of material having a lower modulus than the laminate incorporating zero degree plies in the surrounding area. Unidirectional fiberglass epoxy has been used for this purpose and in some instances we find the use of $\pm 45^{o}$ graphite epoxy for the softening strips.

$0/90_{2S}$ GRAPHITE-EPOXY FATIGUE RT, R=0.1

Specimen	Average Static Strength (psi)	Average Residual Strength After 5×10^6 Cycles (psi)
No stress concentration	83,500	77,200
0.063-in.-dia. hole	68,900	77,700
0.063-in.-dia. hole + 0.018 x 0.004 notch	72,200	82,800
0.063-in.-dia. hole + 0.078 x 0.004 notch	58,300	78,300

FIGURE 6

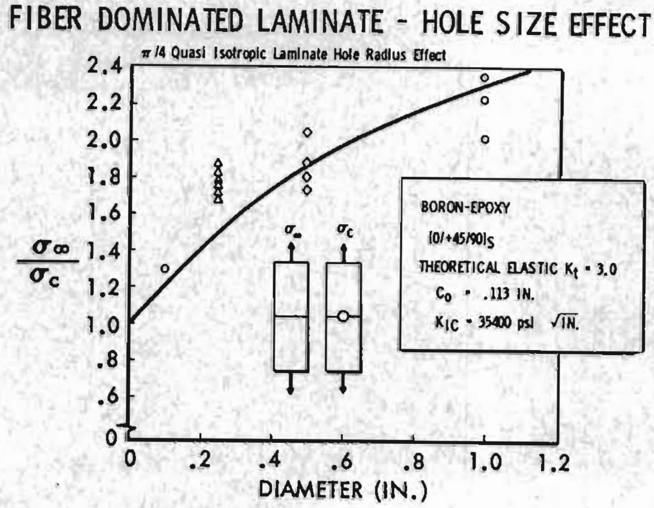


FIGURE 7

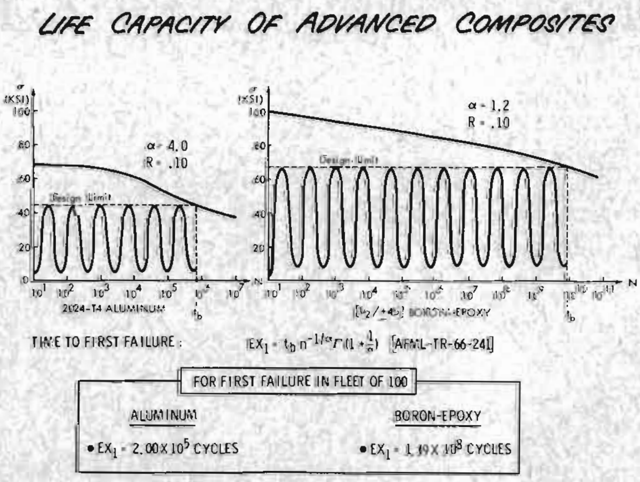


FIGURE 8

The intent of the softening strips in the areas of both the joints or the buffer strips for crack arrestment is to reduce the modulus in this localized area such that stress intensities that would cause crack propagation in the higher strength, high modulus areas will be absorbed as it enters into the lower modulus softening strip area, thus slowing down and effectively stopping the growth of the crack. Figure 9 portrays how the layup provides for the softening strip when using $\pm 45^\circ$ material for that strip. In the required areas the plies of the 0° and 90° material are eliminated and replaced with plies of the $\pm 45^\circ$ material. The figure also shows the material properties of the base material and the softening strip material; the difference in strength and modulus is very apparent. Earlier it was pointed out that the composites do not crack as the metals do under fatigue loading, but rather develop areas of damage to the matrix material. However, under static loading we do find that from a stress concentration a crack may develop and propagate, depending on the makeup of the laminate, and the effect of a very high load cycle, particularly on a damaged structure, may react as a static load would causing rapid damage extension in the form of a crack. The effect of the buffer strips in the case of crack arrestment has the effect of slowing down and indeed stopping that crack. Tests run on actual specimens containing a buffer strip have indeed demonstrated the ability to stop the crack from continued growth. In Figure 10, we see the results of tests actually conducted to determine the ability of the concept to function as designed. Similar specimens were built both with and without crack arrestment strips. A center notch in the specimen provided the necessary stress concentration, and, upon the application of static load induced damage, in the form of a crack, which as it grew, reduced the strength of the material. However, in the specimen with the buffer strips built into it, as the crack entered the buffer strip, the area of much lower modulus, and high toughness was able to absorb the damage and stop the progression of the crack. This results in a lower bound strength, rather than a reduction in strength to zero. Figure 9 depicts this, showing how at the point at which the crack reaches the buffer strip, the strength from then on out is constant. Of course, when the load is increased to such a magnitude that it exceeds the strength of the buffer strip, then a catastrophic failure would occur. However, with the buffer strips installed, the specimen was able to maintain the load rather than having the crack continue to propagate resulting in a total failure of the specimen in the unbuffered section. The same concept of course applies to the use of softening strips in the area of bolted attachments. The use of a high toughness material in the area around the bolts permits a reduced modulus permitting the loads of the bolts to be distributed into the general area, rather than introduce that concentrated load on higher modulus material, which would tend to induce rapid failure. Figure 11 presents some empirical data developed during evaluation of the concept. As can be seen, the application of load to a hole in a softening strip had but little effect upon the load in the basic material due to the absorption of that load in the otherwise lightly loaded strip, i.e., lightly loaded due to the lower modulus carrying but little of the net tension load.

A side benefit is achieved in fabrication in that the softening strip is inherently easier to drill through, thereby reducing the time required for assembly and eliminating the need for special drills to use for the very high modulus materials, especially boron.

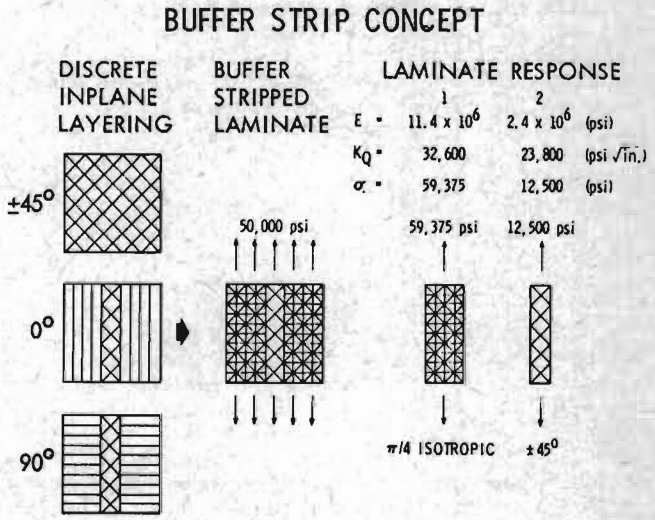


FIGURE 9

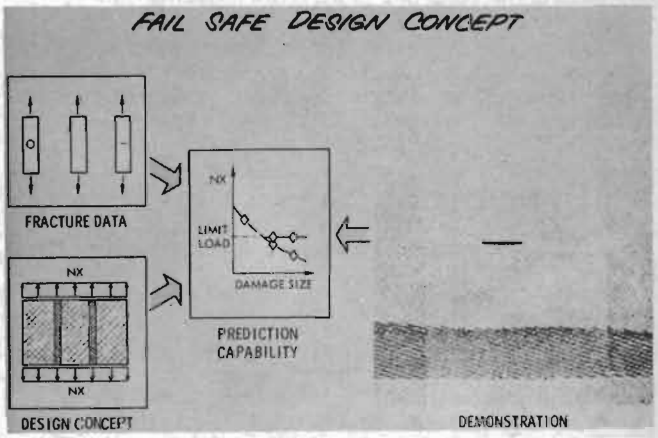


FIGURE 10

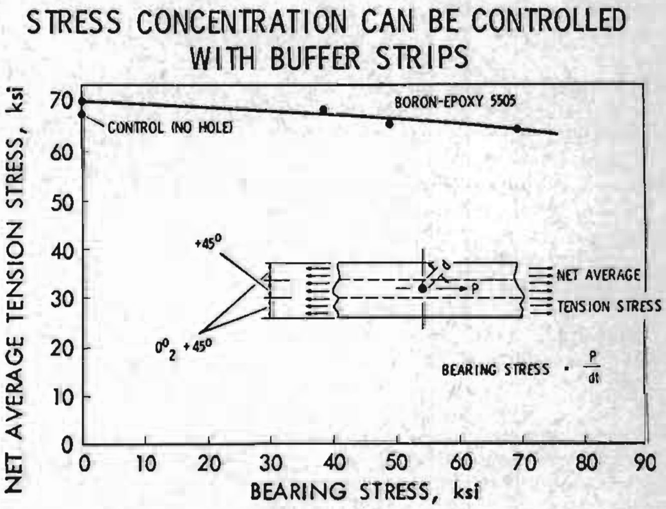


FIGURE 11

Previously, the point was made that the designer is now faced not only with designing the basic structure but also with designing the material that goes into that structure. This highlights another major requirement in the use of the advanced composites, i.e., the need for a very strong design and manufacturing interface during the design operation. Because of the nature of the material, the fact that the structure is built up using discrete layers of material rather than carved away out of a billet of material means that the designer must work very closely with the manufacturing people in order to design a part that can be easily and economically fabricated. In order to achieve low cost in the manufacture of advanced composite components, the design should be such that there is a minimum of cutting, trimming and hand fitting. Additionally, the tooling should be such that the layup of the material is simplified as is the bagging prior to curing.

The whole concept of tooling, particularly for the graphite epoxies required extensive design. Because of the coefficient of thermal expansion of the graphite epoxy, considerable thought must be given to the design of the tools used in the curing process. The extremely small coefficient of thermal expansion of the graphite epoxy means that tools must either match that coefficient of thermal expansion or must be designed such that the finished cured part will have the correct shape and dimensions, i.e., parts of rather simple shape, even though the size might be rather large, could be made on tools whose design considered the thermal mismatch resulting in a correct contour of the finished part. It is possible to fabricate metallic tooling and still achieve the proper finished contour. However, for large parts that have complex contours it appears best to fabricate tools using graphite epoxy material for the tool itself. This, of course, is just one more reason why there is such a great need for a strong design/manufacturing interface during the design operation. The conversion of the tape to finished part involves the placement of the tape on the tool; this may be achieved by various means, hand layup of the tape, placement of the tape by the use of a simple aid to move the roll of tape such that placement by the technician is facilitated, or by very sophisticated machines which are programmed to lay the tape according to computer direction. Various approaches to layup are followed. In some cases the tape is laid up in single plies on a mylar film template, each ply being trimmed to contour according to the template instructions. When the plies are individually laid on mylar templates they must all be brought together in their proper sequence, laid one on top of the other as the templates are removed, to achieve the required part geometry. In other cases the layers of tape are laid directly on plies already laid up, being trimmed as per tooling instructions. For parts of rather simple contour some success has been experienced wherein large sheets of graphite/epoxy have been previously laid up and the required shape cut out of the sheet as needed and draped onto the tool. Prior to cure of the laid up part it is necessary to prepare the layup for cure; this is accomplished by the addition of a separator material, bleeder plies to absorb the excess resin, other plies to permit the flow of air to facilitate the evacuation of any possible air entrapped during the layup process. Finally a barrier to seal the layup and permit the application of a vacuum to both remove the entrapped air and to compact the fiber/matrix during that time when the matrix material softens and flows under the application of heat. That barrier, or bag, must be pliable enough to conform to the shape of the part to permit maximum compaction; the bag may be made from polyvinyl chloride film and in some cases, as for a production run, the bag may be made from a reinforced silicon rubber such that it is reusable.

Let us now look at the results that have been achieved in using the advanced composites materials. Over the years a series of aircraft components have been designed, fabricated, tested and, in some cases, even test flown. As would be expected, an orderly progression from simple, lightly loaded secondary structure parts through complicated, primary structure, has been pursued. The materials used have been both the graphite epoxies, and the boron epoxies. In many cases, only one or a few of a kind have been fabricated. Among those pieces which were fabricated and ground tested, i.e., static and/or fatigue tested, were the F-5 main landing gear door and the F-5 horizontal tail. There have been some trailing edge panels for the F-111 aircraft put into service, as well as parts such as a main landing gear door for the F-111, as shown in Figure 12. These have been tested to prove their ability to sustain their design load and their service experience has been exemplary; the F-111 MLG door suffered considerably less damage than the aluminum baseline design. There has been limited production of some parts in order to achieve some degree of service experience; parts such as the C-5 slat of which 13 were put into service after proof of concept through static and fatigue tests. The experience with these slats indicates approximately two-thirds as much maintenance has been required by the baseline aluminum slats; the F-4 rudder, 50 of which were fabricated and put into service and have provided one of the primary bases for predicting maintenance requirements. One of the more successful programs has been the F-111 horizontal stabilizer program. In that program, a static test article was fabricated using boron/epoxy skins over full-depth aluminum honeycomb and static tested successfully. A completed F-111 horizontal stabilizer is shown in Figure 13. This was followed by a fatigue article which was fabricated and also tested successfully; then two ship sets were fabricated and test flown, again with excellent results establishing the feasibility of using the advance composites for primary load carrying structure. These stabilizers provided a 27% weight savings over the baseline aluminum parts. Because of the experience from the F-111 horizontal stabilizer program, there was enough confidence generated to permit the use of advanced composite empennage structure on both the F-14 and the F-15 aircraft. The F-111 horizontal stabilizer utilized the boron epoxy material, as do the F-14 and F-15.

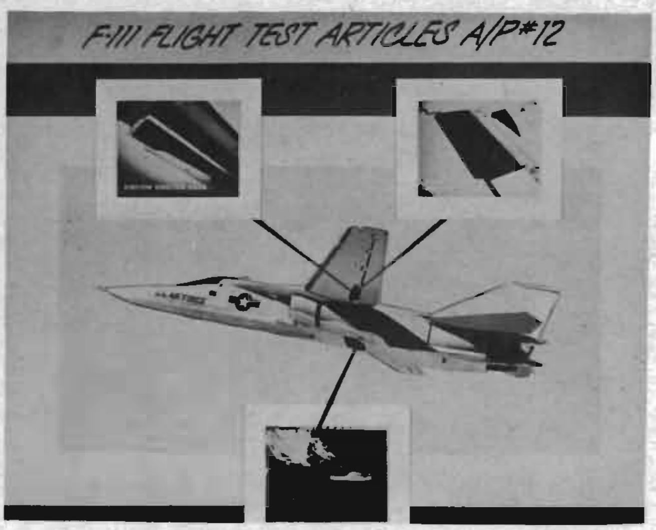


FIGURE 12

FIGURE 13

The initial approach to the design of primary wing structure was the Grumman mid-wing box; this program involved the design, fabrication and test of a two-bay specimen from the mid-wing area of a highly loaded wing. The specimen was a four-spar, three-rib design using boron/epoxy for the skins and spar and rib webs; the skins and spar webs were honeycomb sandwich. The spar caps were titanium due to the then inability to transfer the loads from the skins to the webs. The design also included provisions to induce both flap and slat loads. The first specimen fabricated was tested statically to a failure at 350°F at 123% of design ultimate load, a second specimen was fatigue tested to four lifetimes of fatigue which was then followed by a residual strength test at elevated temperature to a failure at 90% of design ultimate load. The design represented a 16% weight savings over the baseline titanium design. The second wing program approached replacement of the F-100 wing skins; the goal of the program was the design, fabrication and static test of boron/epoxy skins for an F-100 wing. This was the first attempt at the fabrication of reasonably thick skins which were highly loaded; the loading requirements dictated the need for titanium shims to be inserted during layup in the highly loaded local bearing areas of the skin attachment. After fabrication the skins were installed on a substructure (spars and ribs) which had seen prior service. This test program was not a success, the upper, or compression, skin failing in tension during the negative bending test, this due either to insufficient strength of the fatigued substructure or to inadequate design concepts. However, that program did have its technological merits; the ability to design and fabricate thick composite skins which were highly sculptured on the inner surface to match the existing metallic substructure was demonstrated, the curing of thick laminates and the drilling of a large number of holes in boron/epoxy to accommodate attachment to the pre-drilled substructure as well as the large area of that wing skin also added to the technological base. A more recent program is attempting the design, fabrication and test of a wing for the F-15 aircraft. The wing, a direct replacement for the baseline metal wing, is fabricated using boron epoxy for the skins, graphite epoxy for some of the substructure, while maintaining the use of the baseline primary inboard load carrying spars made of titanium and/or aluminum. The design concept of the wing is shown in Figure 14. The successful completion of the static test on the F-15 wing will signal a go-ahead for the fabrication of a flight test wing. The advantage of the composites is strongly pointed out in Figure 15. As shown, an advanced composite wing will be lighter due to the characteristics of the composite materials in relation to the currently used metals; the comparison presented is for a small wing such as might be used on a fighter aircraft, but certainly appreciable weight savings would also be achieved on larger transport type wings. Another successful advanced composite program has been the wing pivot fitting reinforcement on the F-111. As a result of the catastrophic failure mentioned in the early part of this paper, the decision was made that stresses in the highly loaded area of the pivot could be reduced by the use of a boron epoxy patch over that highly loaded area. A retrofit program currently underway is bonding such a boron epoxy patch to the lower portion of the wing pivot fitting. The installation of that patch requires the removal of the aerodynamic fairing in that area. The location of that patch and its fairing is shown in Figure 16. Following the installation of the patch, which was at first done during production of new aircraft but is now being done at the depot level demonstrating the applicability of the advanced composites to service requirements, it is necessary to provide a new fairing over the patch, such that the fairing can be removed for inspection purposes. The design of a suitable fairing was approached from both a metals and a composites utilization. It was found that in the final analysis, although the advanced composite materials, and in the case of the fairing the material chosen was graphite epoxy, was considerably more expensive than the aluminums that would be used; the cost of the finished article out of graphite epoxy was considerably less than the cost of a similar aluminum part. As Figure 17 shows, the difference is a result of the amount of labor required to fabricate the aluminum part, compared to the reduced number of hours and number of parts required for the advanced composite part, even in the light of the much greater cost of the advanced composite material itself. Efforts have also been expended in the development of composite fuselage structure.

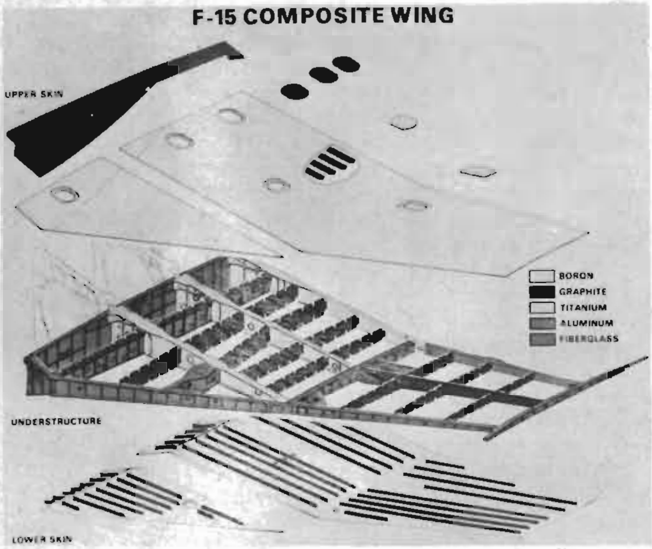


FIGURE 14

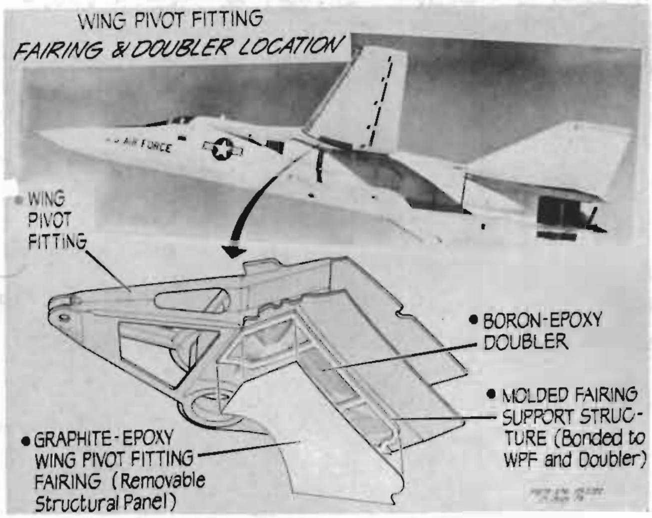


FIGURE 16

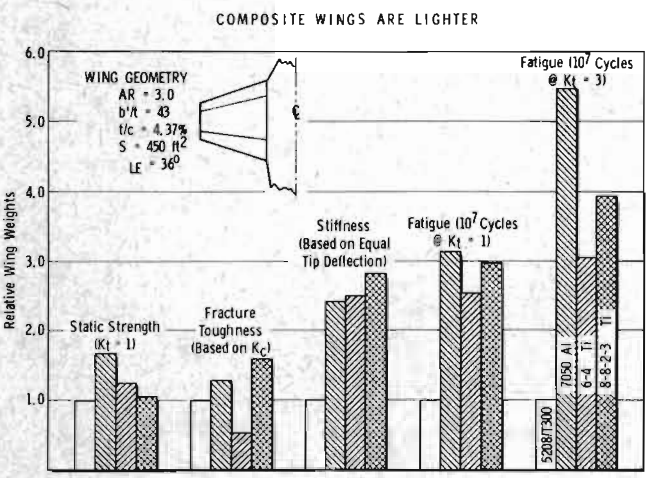
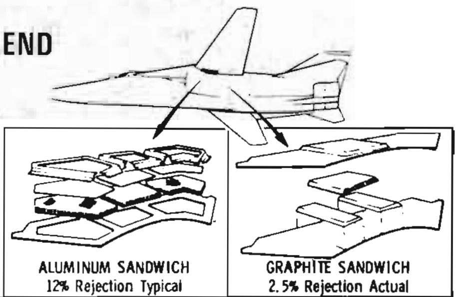


FIGURE 15

COST TREND



	ALUMINUM SANDWICH 12% Rejection Typical	GRAPHITE SANDWICH 2.5% Rejection Actual
Number of Operations	103	19
Number of Hours	31.60	7.6
Labor Costs @$13.00/hr	410.80	98.80
Material Cost	12.50	258.00
TOTAL	$423.30	$356.80
TOTAL NUMBER MANUFACTURED		200
TOTAL NUMBER ORDERED		1000

FIGURE 17

The first attempt at the design and fabrication of a composite fuselage was a section of the F-111 aft fuselage, an area between the engines used as a fuel tank. That section having the proper cross-sectional shape but idealized throughout its length, i.e., doing away with actual aircraft fuel tank taper in order to facilitate tooling, was fabricated from graphite epoxy honeycomb sandwich, i.e., the sidewalls and tank floor were sandwich construction while the tank cover was a hat-section-reinforced laminate skin. Various frame configurations were used including boron aluminum, graphite epoxy, and molded chopped graphite epoxy. The finished specimen was successfully tested to a static failure. The test setup is shown in Figure 18; the specimen failed at 115% of design ultimate load. As a result of that project, the fuselage effort was continued directed toward the design, fabrication and static test of a section of the F-5 mid-fuselage. This effort built an exact full-scale section of the F-5 fuselage including the fuel tank area, wing attachment area, engine nacelles, ducts, and splices; indeed an exact replacement for the baseline metal structure. Figure 19 shows a view of the fuselage looking inside at the frames, ducts, and bulkheads. The structure was fabricated primarily from graphite epoxy. Some boron epoxy was used in major bulkheads. Longerons were laid up in the shell as the shell was laid up. Frames and bulkheads were fabricated and then fit to the cured shell. That fitting included the installation of wet fiberglass ties which were then cured, locking the frames in position. The design of the F-5 fuselage section included the use of a failsafe approach to the wing body intersection. A failure of any one structural member in the area of the wing carrythrough attachment would not result in a catastrophic failure of the fuselage test section. To accomplish this, there was some slight change in the frames and frame spacing in the area of the wing attachment. That failsafe feature, as might be expected, required the use of some additional material, reducing the weight savings from a predicted 25% to 15%. The completed specimen was tested statically through all the conditions of the F-5 airplane. One of the concerns in designing with the advanced composites was the apparent inability of the laminate to sustain post-buckling loads. Figure 20 shows the degree of buckling experienced during the test of the F-5 fuselage. Buckling, in flat panels, has little impact upon the strength of the composites; however, buckles which may grow into curved panels can cause catastrophic failure, but this occurrence may be precluded through judicious design wherein additional stiffening plies are added to the skins adjacent to the change in contour. Following the completion of the ultimate load test without failure, the specimen was subjected to three failsafe tests in which select members of the structure were severed while the structure was at limit load. None of those induced structural failures resulted in further failure to the test section.

As a result of the experience gained in the various programs pursued in the advanced composites, particularly those involved with the fabrication of large complex structures, it is now apparent that the designer must conceptualize new structural designs considering the manufacturing properties associated with these materials. For example, in the past we have manufactured fuselage structure from the inside out, i.e. the frames, bulkheads and longerons are fitted to assembly fixtures in sequence and then covered with the skin. The advanced composites will provide an extremely smooth aerodynamic surface if that surface is the mold side during cure. To take advantage of that characteristic it becomes necessary to fabricate the skin first and then install the internal substructure, as was done and has been shown in the photograph of the F-5 mid-fuselage. But that approach was very labor intensive and time consuming; therefore it will benefit the designer to conceptualize new structural arrangements. Initial attempts at conceptual approaches to design have resulted in designs in which both frames and longerons may be laid up and cocured with the outer skin in larger panels; assembly then has been reduced to the joining of a few larger units, i.e. panel units and substructure, to form the finished part. As this new material matures and moves into possible production applications there will be the need for intensive efforts to create those conceptual designs which will provide hardware whose acquisition cost is competitive with the metals. The characteristics of the advanced composites, their high structural efficiency, and their apparent long fatigue life forecast high payoffs for their use. But again, it is necessary to emphasize the need for a strong design-manufacturing interface in order to achieve those potential payoffs, and that need is for conceptual approaches to designing with composites and for innovative manufacturing techniques which will put those designs into production at a minimum cost, a cost below that of the metals.

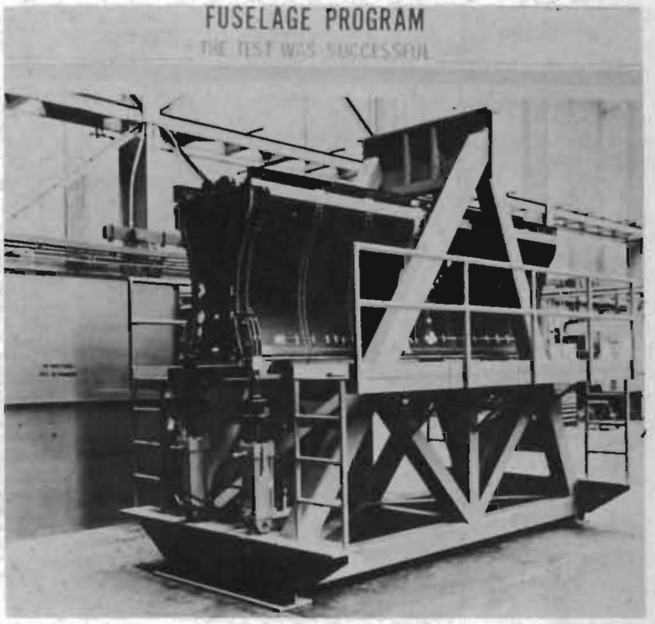


FIGURE 18

UPPER SHELL ASSEMBLY

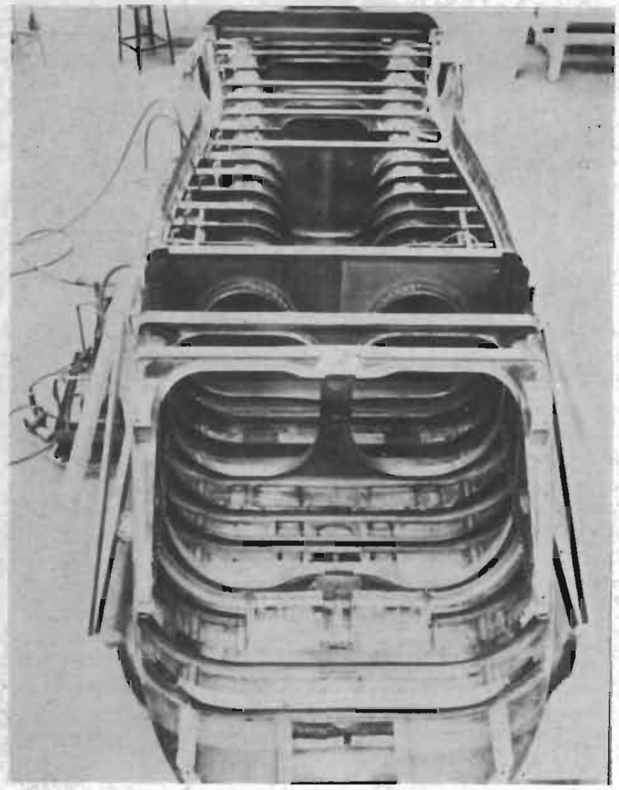


FIGURE 19

SIDE PANEL SHEAR BUCKLES

FIGURE 20

CONCLUSIONS

We see today two new thrusts in aircraft structures; efforts being expended to improve the metals through the modification of alloys and the development of new design concepts aimed at circumventing those fatigue critical problem areas which have plagued aircraft for so long. There is also the rapid expansion of the new technology of advanced composite materials; this development provides new materials which appear to be more resistant to fatigue, but which also require new approaches to design. The use of advanced composites places emphasis on both the development of new design concepts which will take advantage of the characteristics of these materials while designing around their drawbacks, and that very design process requires a strong interface with manufacturing, and tooling, to assure that the designs can be fabricated, and at the least possible cost.

D I S C U S S I O N

J. Solvey (Aeronautical Research Laboratories, Melbourne, Australia): Why did the author not mention that the special fittings needed to introduce the loads will considerably ~~reduce~~ the savings in weight?

W.R. Johnston: It is not as critical as it was thought. Early in the development and application of advanced composites to aerospace structures the thrust was to use them to replace metals, that is to directly replace metals. And at that time there were instances wherein there was a weight penalty incurred through use of special load introduction fittings. However, as the technology has matured and we have become more learned in the design of composite structures it has become apparent that direct substitution is not the optimum design condition. Rather the design should be based upon not only the load carrying requirements but also upon the characteristics of the material. This means that designs will change, particularly at the detail level, such that the attributes of the advanced composites will predominate in the designing the configuration of the component. Admittedly, even today, very high concentrated loads present a problem in designing with the advanced composites; however, it is possible in many cases to create a design such that other types of load introduction will reduce the load levels such that the composite materials are able to sustain the required loads.