APPLICATIONS OF COMPOSITE MATERIALS

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APPLICATIONS OF COMPOSITE MATERIALS

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Foreword

The technology of high performance fiber composites has been with us for only one decade. Although fiberglass has been available for many years, the discovery of boron fiber in the early 1960's, followed quickly by graphite and other fibers, ushered in a new era of structural composites which included the rediscovery of fiberglass for critical, highly loaded structures.

At the present time there are several hundred advanced composite structures which are flying, and the technology which was developed primarily for aerospace is being quickly adapted to commercial applications, including machinery, sporting equipment, and storage tanks, among others.

The rapid developments in composite technology, which occurred primarily in the 1960's in the aerospace field, are chronicled in this book. Because this field is advancing rapidly, the material in this book is not completely up-to-date; however, it is still remarkably valid in providing a review of the fundamental technological base in this field.

M. J. Salkind

Stratford, Connecticut
October 1972

Dedication

To
Dr. Isaac H. Schwartz
my good friend and mentor
M.J. Salkind

Related ASTM Publications

Composite Materials: Testing and Design (Second Conference),
STP 497 (1972), $36.50

Composite Materials: Testing and Design,
STP 460 (1970), $31.00

Interfaces in Composites, STP 452 (1969), $16.50

Contents

R.R. June[1] *and J.R. Lager*[1]

Chapter I – Commercial Aircraft

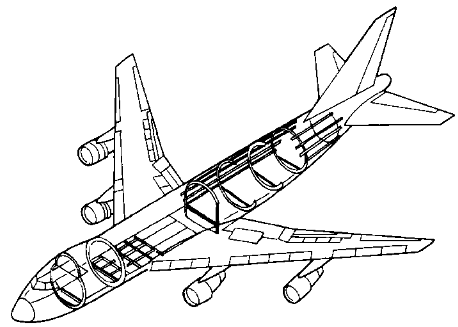
REFERENCE: June, R.R. and Lager, J.R., "Commercial Aircraft," *Applications of Composite Materials, ASTM STP 524,* American Society for Testing and Materials, 1973, pp. 1-42.

ABSTRACT: The use of composite materials offers considerable potential for reducing structural weight and, therefore, increasing productivity of commercial aircraft. The application of composites must be performed selectively, as some structures offer considerable potential for cost effective use, whereas others are more cost effective as metal structures. Heavily loaded beams, columns, and stiffness critical control surfaces are at present the major areas of application of composite materials.

KEY WORDS: composite materials, fiber composites, aircraft, composite structures, cost effectiveness, boron, graphite, fiberglass reinforced plastics

The aircraft industry has taken an intense interest in advanced fibrous reinforced composites. Proper use of these new materials offers the potential for reducing the weight of aircraft structural components by as much as 50 percent. Basic structural elements such as beams and columns offer the most potential for cost effective weight reduction. More complex components such as control surfaces, while advancing the state-of-the-art, offer less potential. An estimate of the potential weight reduction for a typical subsonic aircraft is shown in Fig. 1. Although the estimated average structural component weight reduction of 20 percent seems conservative, it can affect a rather large percentage of the net airframe weight (50 percent) and results in a total estimated weight saving of 19 500 lb. The structural efficiency of advanced composites is exemplified by the fact that this weight saving is cost effective and is accomplished through the use of only 9750 lb of composite material. For unidirectional loading, advanced composites offer a significant weight reduction with the added advantages of being relatively easy to fabricate, analyze, and design. Because of this, it is felt that initial commercial applications of advanced composites will be in beam flanges, columns, longerons, stringers, and frames with the incorporation of advanced structural concepts (for example, honeycomb sandwich) to provide the additional structural stability required.

[1] Members, Advanced Structural Design Unit, Commercial Airplane Group, The Boeing Company, Seattle, Wash.



■ AIRPLANE EMPTY WEIGHT (LB)	354,000
■ NET AIRFRAME WEIGHT (LB)	195,000
■ STRUCTURAL WEIGHT CONVERTIBLE (LB)	97,500
■ 20 PERCENT WEIGHT REDUCTION (LB)	-19,500
■ COMPOSITE NET AIRFRAME WEIGHT (LB)	175,500
■ COMPOSITE AIRPLANE EMPTY WEIGHT (LB)	334,500
■ COMPOSITE USED (LB)	9,750

FIG. 1–*Estimated potential airframe application of advanced composite materials.*

The basic concept of composite design is not new to the aircraft industry. Typically, aircraft structures use a variety of proven structural materials as shown in Table 1. For each specific application, a material is chosen which best suits the design criteria involved. Advanced fibrous composites offer to the designer a new material system with some unique structural properties.

Many fibers having the prerequisite strength and stiffness to fall in the advanced fiber category have become commercially available in recent years in various shapes, sizes, amounts, and prices. Boron and graphite continuous

TABLE 1–*Structural materials summary*

	Percent of Structural Weight	
	Boeing 707 Subsonic	Boeing SST Supersonic
Aluminum	72.4	1.2
Steel	15.5	8.9
Magnesium	2.7	. . .
Titanium	0.2	78.9
Nonmetals	0.9	4.2
Miscellaneous	8.3	6.8
Total	100%	100%

filaments have received the most attention because of their availability at a reasonable price, and very high specific strength and stiffness values when used to reinforce an epoxy matrix. Boron has come to the fore primarily because of some early deficiencies of graphite composites, namely, low interlaminar shear and compressive strength caused by the low transverse strength of the fiber and the difficulty of achieving a good bond at the fiber matrix interface. These deficiencies are rapidly being eliminated.

Many new structural materials have in the past fallen short of their expected potential because of an increase in only one of the important structural efficiency parameters, strength and stiffness. Beryllium is a material which is six times better than aluminum when only stiffness is considered, but because of its low strength and brittleness can only be used where strength is not a major consideration. Unidirectional fiberglass is four times stronger than aluminum, but because of its low stiffness has been restricted in its usage. Boron filament is six times stronger and stiffer than aluminum and, therefore, is not restricted in its expected potential. Unidirectional boron composites are strain compatible (Fig. 2) with aluminum, titanium, and steel, which means that when used in conjunction with these structural metals, the metal is working near its ultimate capability at a critical strain level for the composite.

The basic structural efficiency potential of advanced composites is indicated in Fig. 3 where they are compared with common structural materials on a strength and stiffness basis. Equal length tension bars designed to break at an applied load of 1000 lb will have a weight dependent only on their density and tensile strength in the direction of the load. Unidirectional boron and graphite composites are shown to be very light when compared to the other structural

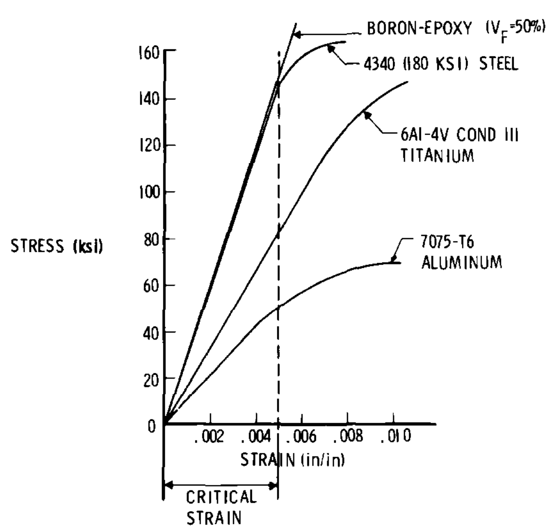


FIG. 2—*Stress-strain comparison.*

materials on this strength basis. If each tensile bar is designed to deflect an equal amount under an applied load of 1000 lb, its weight would depend on its density and Young's modulus or stiffness. Again, in this comparison, unidirectional boron and graphite composites are very light when compared to the other structural materials. This combination of high strength and stiffness with low density for unidirectional advanced composites, together with their strain compatibility with aluminum and titanium, offers the designer a material which, with proper use, can significantly reduce the weight of aircraft structural components.

	COMMON STRUCTURAL METALS (LB)		FIBROUS COMPOSITE MATERIAL (LB)	ADVANCED FIBROUS COMPOSITE MATERIALS (LB)			
	ALUMINUM	TITANIUM	UNIDIRECTIONAL FIBERGLASS	UNIDIRECTIONAL GRAPHITE-EPOXY	UNIDIRECTIONAL BORON-EPOXY	BIDIRECTIONAL BORON-EPOXY	ISOTROPIC BORON-EPOXY
	P = 1000	P = 1000	P = 1000	P = 1000	P = 1000	P = 1000	P = 1000
WEIGHT IF EACH BREAKS AT P = 1000 LB	10.00	7.70	2.05	2.25	2.28	4.56	6.84
WEIGHT FOR EQUAL DEFLECTION WHEN P = 1000 LB	10.00	9.75	8.58	1.69	1.98	3.96	5.94

FIG. 3–*Structural efficiency potential.*

Materials

Boron and graphite filaments are perfectly elastic until failure and show considerable scatter in strength values. A simplified single fiber strength model might consist of a chain with brittle links which have a variety of strengths. A tensile strength test on this model would show a scatter in strength results, and the stress associated with the peak of the distribution function would depend on the length of the test specimen. A useful composite material is obtained when these filaments are encased in a ductile, low strength, low modulus matrix material which transfers load from fiber to fiber through shear and localizes the effect of a single fiber failure by redistributing the load near the failed fiber ends to adjacent fibers. Total composite failure is then governed by the statistical distribution of single fiber failures.

The matrix material determines the efficiency with which fiber properties can be transferred to the composite. Its stiffness supports the fibers against buckling in compression, its shear strength transfers load between fibers, and its toughness helps to retard the propagation of cracks. The matrix material must also bond to the fiber and should be void free. A composite material which retains its strength and stiffness at high temperatures must have a matrix material which is structurally stable for long periods of time at the working

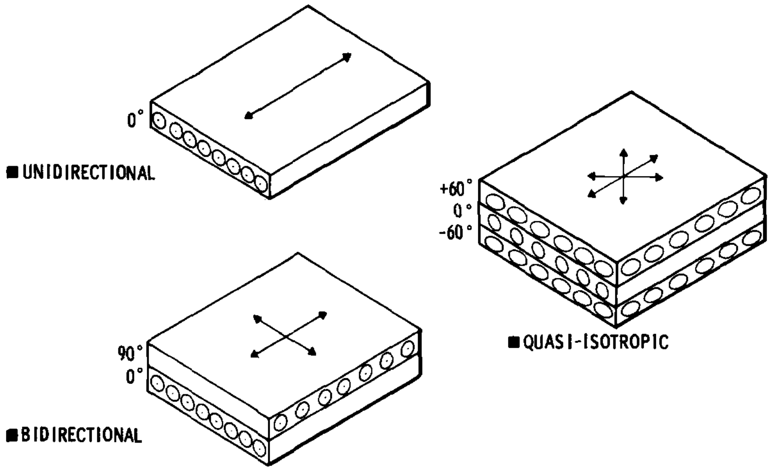


FIG. 4—*The composite.*

temperature. Polymides provide a matrix material with potential for use at high temperatures, while epoxy materials cover a range of useful matrix moduli at lower temperature. The epoxies adhere well to boron fibers and result in composite laminates which are void free and of very high quality. Polyimides, because of their volatile releasing action during cure, can result in composites with various degrees of void content and fiber-matrix bond. Significant progress has been made in solving these problems.

The basic unit utilized in the fabrication of composite structures is the unidirectional tape. These unidirectional tapes can be laminated (See Fig. 4) in the desired directions and numbers and cured using the appropriate adhesive cure cycle, resulting in composite laminates with the desired properties in the various directions.

Manufacturing

Unidirectional boron-epoxy and graphite-epoxy tape fabrication has progressed to the point where large sheets can be fabricated at a reasonably low cost with very high quality. These tapes are commercially available in continuous or rectangular sheets of various widths and sizes. Unidirectional sheets are layed up in the desired pattern on an appropriate tool. The laminate is then vacuum bagged and cured under the appropriate cure cycle in an autoclave. The cured composite material can then be used in the fabrication of a composite structure as shown in Fig. 5. There are, of course, a considerable number of desirable alternate methods of fabricating composite structures. Fabrication methods used in the initial development stages of advanced composite development will no doubt be replaced by more sophisticated, automated techniques for production as discussed in Chapter VI.

The difference in thermal coefficient of expansion between fibrous composites

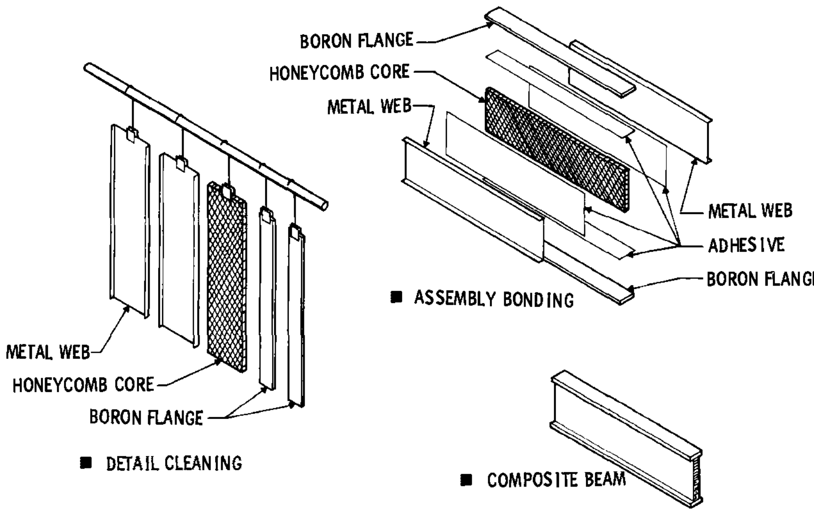


FIG. 5–*Composite beam layup.*

and metal causes some difficulty in the fabrication of hot bonded composite structures. If the coefficients are relatively close, symmetrical layups will result in unwarped final components with tolerable residual stresses. Combining materials with a fairly large difference in coefficient of expansion may require cold bonding. Fabrication of unsymmetrical components may also require the use of cold bonding of possibly curved tools to result in the desired final shape.

Some unique but solvable problems are also encountered in machining composite materials and structures involving various combinations of boron filament, epoxy, aluminum, and titanium. The main problems arise from the extreme range of hardness of the materials involved. Boron filaments have a hardness just slightly less than diamond and, therefore, do not lend themselves to machining by conventional tools. Diamond and silicon carbide abrasive tools do an acceptable job of machining boron-epoxy composites. Machining composites containing boron-epoxy and conventional metals can be accomplished by means of ultrasonic techniques.

Analysis

Optimum weight savings from the use of advanced fibrous composites will be realized only through conceptual design. A direct substitution of materials proves to be a relatively inefficient method of designing with these new materials. Conceptual design need not necessarily be associated with highly complex computerized analysis techniques, but rather can be used effectively with the assistance of basic strength of materials relationships and a common sense appreciation of composite materials and structural mechanics. Finite element structural analysis computer programs are indispensable for the analysis of structures large enough to make hand calculations impractical.

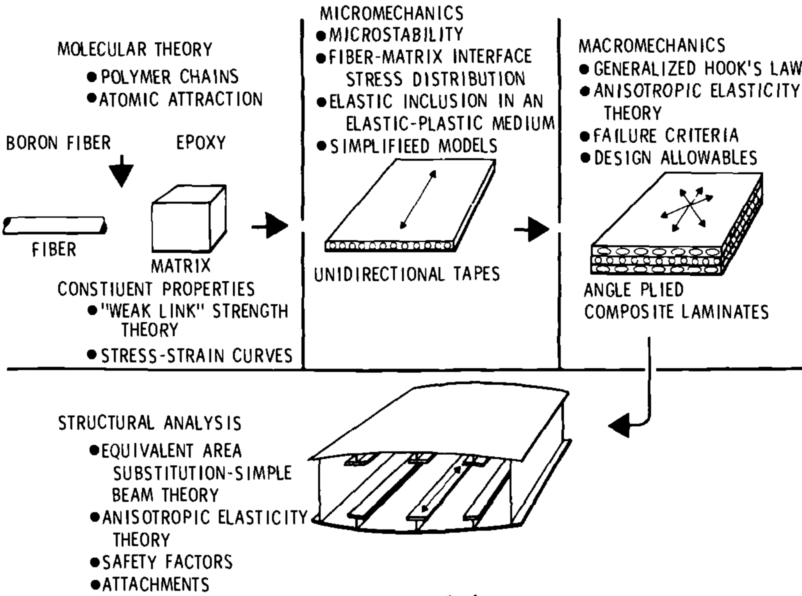


FIG. 6—*Analysis.*

Analysis can be broken down into the general categories, shown in Fig. 6, of micromechanics, macromechanics, and structural analysis. Micromechanics deals with the determination of the properties of unidirectional composites from the known properties of the basic constituents. This includes fiber-matrix interface stress, fiber microstability, and residual stress distribution. Using the unidirectional tape as a basic input, macromechanics considers the determination of angle-plied laminate properties. Structural analysis of advanced composite reinforced structures utilizes existing techniques such as equivalent material substitution, finite elements, anisotropic elasticity, and general energy methods.

When strong, stiff fibers are encased in a lower strength, lower modulus matrix material, the resultant composite material possesses some of the desirable properties of each constituent. An external load applied to a unidirectional composite with discontinuous fibers is transferred to the fibers by shear through the matrix. The fiber-matrix interface bond transfers load by shear to the fiber until a maximum value is reached at a distance $\ell_c/2$ from the fiber ends, where ℓ_c is defined as the critical fiber length. The ultimate unidirectional tensile strength is then dependent on the strength of the constituents, the volume fraction of fibers, and the ratio L/ℓ_c.

The ability of a fibrous composite to resist compressive load is dependent mainly on the ability of the matrix material to support the filaments against buckling [*1*].[2] This assumes that the fibers are quasi-isotropic, such as boron or glass, and not anisotropic like graphite. An anisotropic fiber has the added problem of being able to buckle or break down internally at a lower applied load than would cause overall fiber buckling. This is known as composite microinsta-

[2] The italic numbers in brackets refer to the list of references appended to this paper.

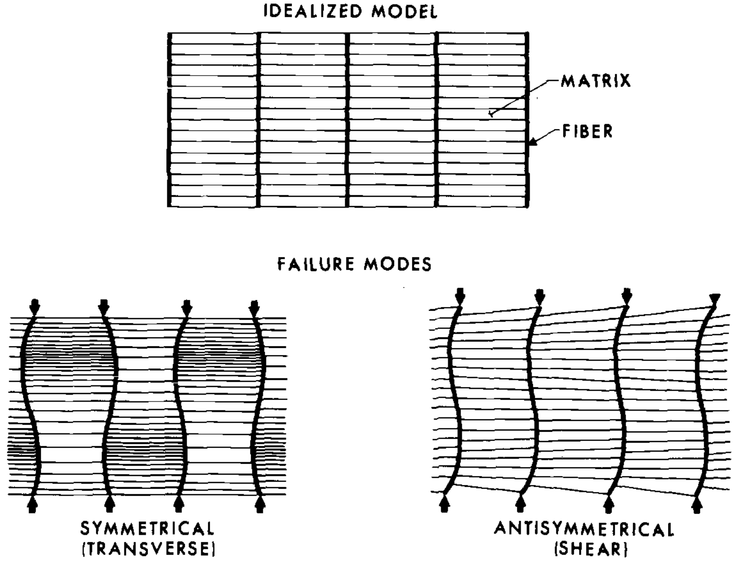


FIG. 7–*Micro-stability failure modes.*

bility. In order to theoretically predict the compressive strength of a fibrous composite material, the idealized model (see Fig. 7) was proposed by Rosen [*2*]. The energy method was used to predict buckling in each of the two modes indicated. The shear buckling mode predominates for composites with fiber volume fractions of interest for use as structural materials. The compressive strength (σ_c) of unidirectional boron-epoxy composites [*1*] has been shown to be given by the expression

$$\sigma_c = \frac{0.63\, G_m}{(1 - V_f)}$$

where G_m is the shear modulus of the matrix material and V_f is the volume fraction of fibers.

Fracture toughness of fibrous composite materials is effected by (1) critical transfer length (ℓ_c,)–if the critical transfer length is large and the discontinuous fiber length is less than L_c, failure will occur by fiber pull-out rather than a fiber tensile failure; (2) volume fraction of fibers–at high fiber contents the composite acts more like the brittle fibrous phase than the relatively ductile matrix phase; (3) weak interface–a weak bond between fiber and matrix allows a crack propagating in the matrix, perpendicular to a fiber, to be deflected parallel to the fiber leaving the fiber unbroken. An increase in composite toughness is made at the expense of other properties, notably, composite strength.

A typical curve of constant temperature plastic creep strain versus time is shown in Fig. 8 for a constant stress applied to a unidirectional composite containing discontinuous fibers with a constant L/d ratio, where L is fiber length

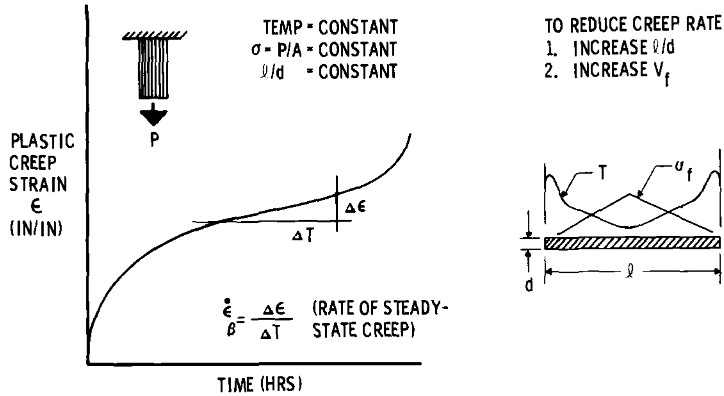


FIG. 8–*Creep.*

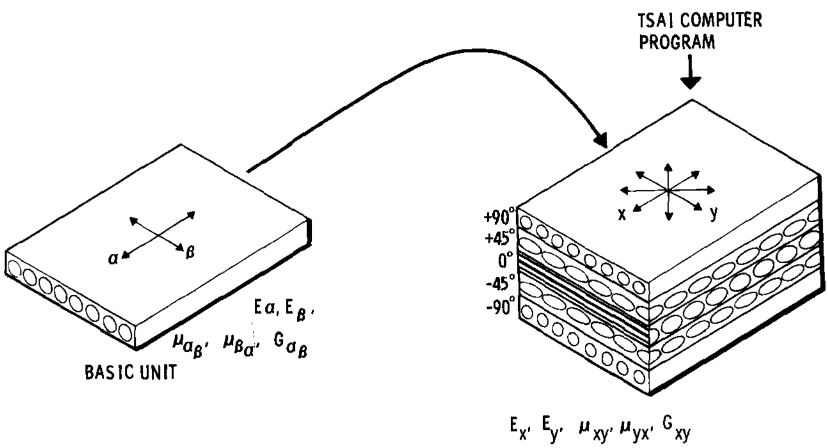


FIG. 9–*Macromechanics.*

and d is fiber diameter. An increase in L/d or in V_f will reduce the creep rate.

Table 2 summarizes the effect of fiber volume fraction and matrix modulus on the mechanical properties of boron-epoxy composites. An increase in one property is usually made at the expense of others.

From the known properties of unidirectional tape, the properties of angle-plied multilayer laminates, such as the one shown in Fig. 9, can be predicted with the aid of computer programs based on anisotropic linear elasticity theory. One such program, developed by S.W. Tsai [*3*], very accurately predicts laminate stiffness and has been extended [*4*] to predict ultimate composite strength. An example of the change in properties with change in orientation angle θ is shown in Fig. 10. Predicting the ultimate strength of laminates involves the accurate piecewise linear approximation of the continuous

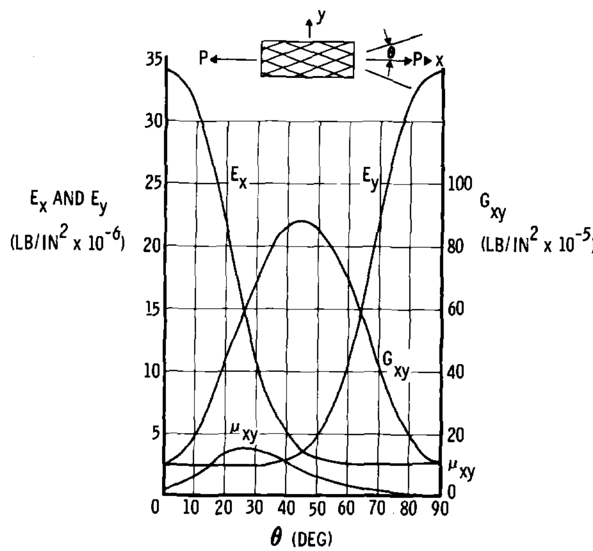


FIG. 10—*Boron fibers cross laminated at angle ± θ and loaded at angle θ = 0.*

TABLE 2—*Effect of fiber volume fraction and matrix modulus on mechanical properties of boron-epoxy composites.*

	Desirable V_f	Desirable Matrix Modulus
Tension	medium to high	medium
Compression	medium to low	highest possible
Modulus	highest possible	doesn't matter
Interlaminar shear	medium	high
Fracture toughness	medium	low
Fatigue	medium	medium
Creep	high	high

	Low	Medium	High Possible	Highest
V_f(%)	15 to 35	35 to 50	50 to 65	65 to 78
Matrix Modulus	150 000 to 350 000	350 000 to 500 000	500 000 to 650 000	650 000 to 900 000

nonlinear behavior of angle-plied laminates. Accurate prediction of angle-plied laminate properties is important for prediction of stresses, deflections, and buckling loads associated with structures which can be broken down into basic plate or shell structural elements for analysis purposes. Finite element and energy method analysis techniques and their associated computer programs are indispensable for the prediction of total structure behavior. For structures which are reinforced by the use of strategically located strips of unidirectional advanced composite material, the analysis sequence bypasses the macromechanics computer programs and goes directly from the micromechanics associated with unidirectional tapes to structural analysis using appropriate conventional analysis techniques. Boundary and attachment problems associated with the use of large flat sheets of multidirectional laminates and the increased analysis complexity suggest the desirability of incorporating composites as stiffening and strengthening unidirectional strips where possible.

An equivalent area substitution approach with basic mechanics of materials relationships is very useful for the analysis of beams and columns. Beam lateral stability and column overall stability must always consider shear effects because of the low stiffness in directions other than parallel to the fibers. Plate and shell analysis utilizes anisotropic elasticity theory and requires solution of differential equations of high order. The solution to these equations and, in particular, those associated with stability are obtainable only through very tedious and approximate computerized numerical techniques. Complex structures composed of basic beam, column, and plate elements can be handled by means of computerized finite element analysis methods.

Cost Effectiveness

Aircraft structural weight saving through the use of a material which is more expensive than the one it replaces must be made in a cost effective manner. Cost

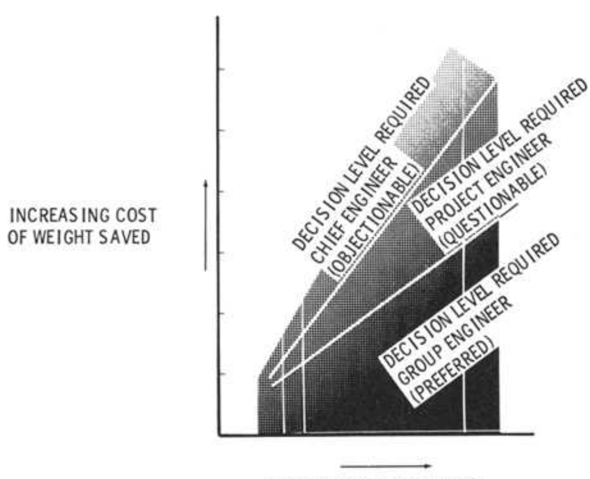


FIG. 11–*Decision level for proposed new structure.*

studies and experience gained on existing subsonic commercial aircraft have indicated that weight saved at a cost of $50 to $150 per pound or less, depending on the particular aircraft, will be economical over the life of the aircraft. The decision level required for the amount to be spent to save weight on proposed new structure is shown qualitatively in Fig. 11. The amounts are flexible depending on the stage of development of the particular aircraft system.

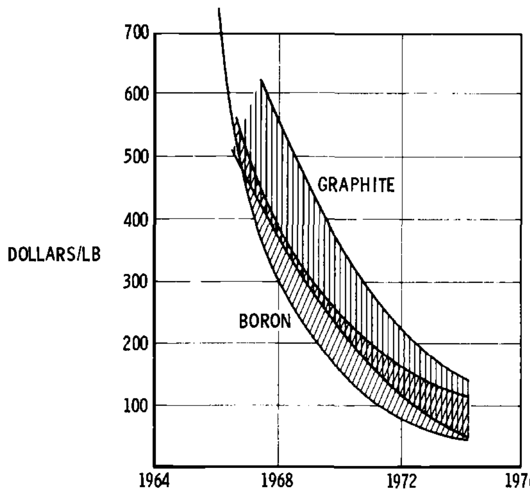


FIG. 12–*Filament costs.*

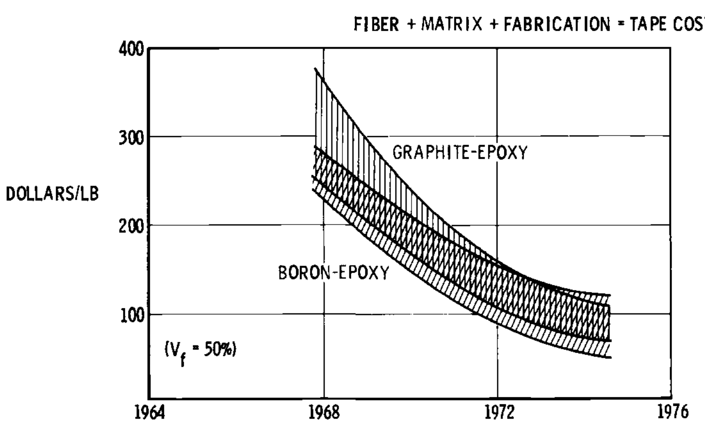


FIG. 13–*Composite tape costs.*

Boron and graphite fiber costs have been reduced significantly in recent years and have the potential of leveling off at $50 to $100 per pound or less in the near future as shown in Fig. 12. Potential fiber costs in this range allow them to be considered for use on commercial aircraft, but necessitate that they be used in a very judicious manner. Indiscriminate use of these advanced fibers could result in aircraft components which, although lighter than conventional components, become very costly and hard to justify on a cost effectiveness basis.

Present and predicted future costs of unidirectional boron and graphite tapes containing 50 percent by volume of fibers are shown in Fig. 13. Although a pound of composite tape contains considerably less than a pound of fibers, the added fabrication and matrix costs bring the cost per pound of tape back to just slightly less than the cost per pound of fibers.

The basic cost effectiveness relationship is

$$(1 - W_s)C_{\text{comp}} \leq C_{\text{conv}} + W_s(V_w)$$

where W_s is the weight savings fraction, C_{conv} is the cost per pound of the conventional structure, C_{comp} is the cost per pound of the proposed new composite structure, and V_w is the value of saving one pound of structural

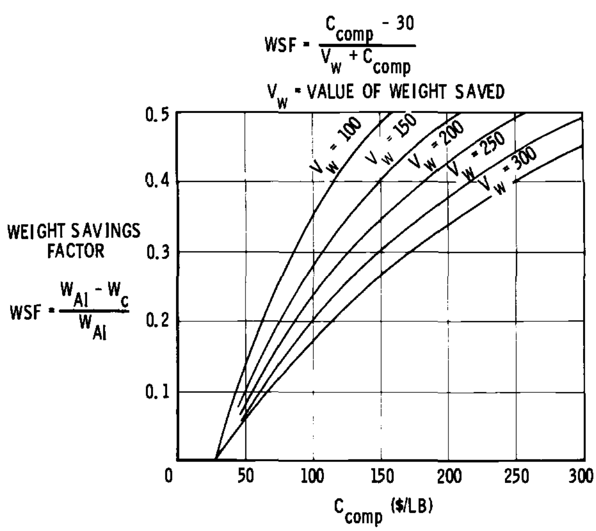


FIG. 14–*Cost effectiveness–aluminum versus composite structure.*

weight. This relationship states that the cost of a proposed composite structure must be less than or equal to the cost of the conventional structure that it replaces, plus the value of the weight saved. Writing the above equation in the form

$$W_s \geqslant \frac{C_{\text{comp}} - C_{\text{conv}}}{V_w + C_{\text{comp}}}$$

allows us to determine the amount of weight saving necessary to cost effectively replace a conventional structure by a composite structure. Assuming that a

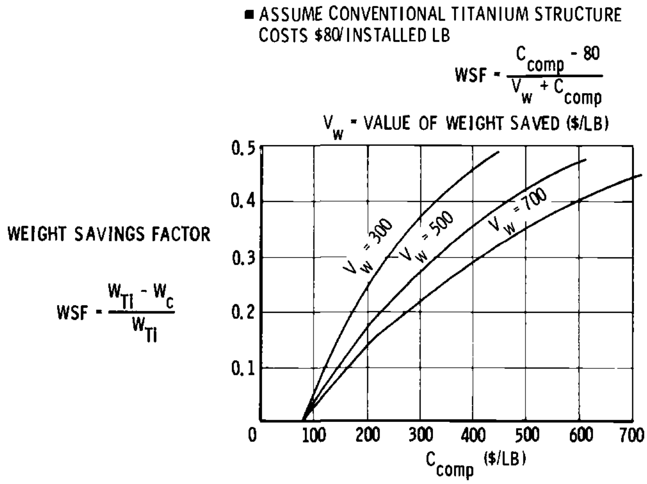
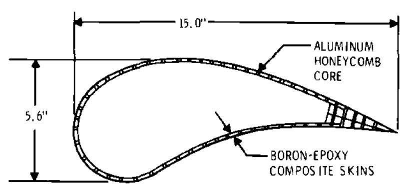


FIG. 15—*Cost effectiveness—titanium versus composite structure.*

typical conventional aluminum subsonic aircraft structure costs $30 per pound, the curves in Fig. 14 allow the determination of the percent of weight saving necessary to cost effectively replace the conventional structure by a composite structure for a known value of a structural weight saving in dollars per pound.

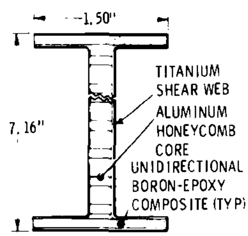


■ WEIGHT OF CONVENTIONAL ALUMINUM FOREFLAP, W_{conv}	20.0 LB
■ WEIGHT OF BORON COMPOSITE–Ti FOREFLAP, W_{comp}	15.0 LB
■ WEIGHT SAVED (LB), $W_{conv} - W_{comp}$	5.0 LB
■ WEIGHT SAVING FACTOR, $W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$	0.25
■ COST OF CONVENTIONAL ALUMINUM FOREFLAP, C_{Tconv}	$729
■ COST PER POUND OF CONVENTIONAL ALUMINUM FOREFLAP $C_{conv} = \frac{C_{Tconv}}{W_{conv}}$	$37/LB
■ BORON COMPOSITE FOREFLAP, C_{Tcomp}	$1987
■ COST PER POUND OF COMPOSITE FOREFLAP, $C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$	$132/LB
■ COST PER POUND OF WEIGHT SAVED, $C_{ws} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv} - W_{comp}}$	$252/LB

FIG. 16—*707 foreflap cost.*

Similar curves are shown in Fig. 15 for the replacement of a conventional titanium structure costing $80 per pound.

An example of a cost effectiveness analysis is shown in Fig. 16 for a proposed 707 foreflap structure. A weight reduction of 5 pounds from the conventional 20-pound aluminum foreflap results in an actual weight saving fraction W_s of 0.25. The cost of the existing 707 foreflap structures is $37 per pound.



- WEIGHT OF CONVENTIONAL ALUMINUM BEAM W_{conv} — 16.5 (LB)
- WEIGHT OF BORON COMPOSITE Ti BEAM W_{comp} — 9.0 (LB)
- WEIGHT SAVED $W_{conv} - W_{comp}$ — 7.5 (LB)
- WEIGHT SAVING FACTOR, — 0.45

$$W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$$

- COST OF CONVENTIONAL ALUMINUM BEAM C_{Tconv} — $161
- COST PER POUND OF CONVENTIONAL ALUMINUM BEAM — 10 ($/LB)

$$C_{conv} = \frac{C_{Tconv}}{W_{conv}}$$

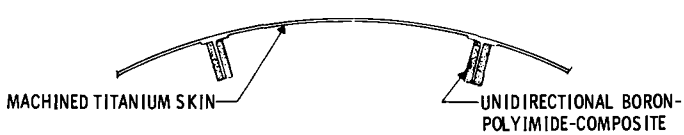
- COST OF COMPOSITE BEAM C_{Tcomp} — $964
- COST PER POUND OF COMPOSITE BEAM — 106 ($/LB)

$$C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$$

- COST PER POUND OF WEIGHT SAVED — 107 ($/LB)

$$C_{ws} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv} - W_{comp}}$$

FIG. 17—*707 floor beam cost.*



- WEIGHT OF CONVENTIONAL TITANIUM SKIN, W_{conv} — 3.6 (LB/SQ FT)
- WEIGHT OF BORON-POLYIMIDE-TITANIUM SKIN, W_{comp} — 2.2 (LB/SQ FT)
- WEIGHT SAVED, $W_{conv} - W_{comp}$ — 1.4 (LB/SQ FT)
- WEIGHT SAVING FACTOR, $W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$ — 0.42
- COST OF CONVENTIONAL TITANIUM SKIN, C_{Tconv} — 370 ($/SQ FT)
- COST PER POUND OF CONVENTIONAL TITANIUM SKIN, $C_{conv} = \frac{C_{Tconv}}{W_{conv}}$ — $103/LB
- COST OF BORON-POLYIMIDE-TITANIUM SKIN, C_{Tcomp} — 680 ($/SQ FT)
- COST PER POUND OF BORON-POLYIMIDE-TITANIUM SKIN, $C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$ — $309/LB
- COST PER POUND OF WEIGHT SAVED, $C_{WS} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv}\ W_{comp}}$ — $221/LB

FIG. 18—*SST body panel cost.*

Amortizing nonrecurring costs over 200 airplanes and using a projected fiber cost of \$150 per pound results in a composite foreflap cost of \$1937 or \$132 per pound. It is therefore costing \$1937 -729 = \$1258 to save 5 pounds of structural weight, or \$252 per pound.

▪ WEIGHT OF CONVENTIONAL TITANIUM BEAM, W_{conv}	7.3 (LB)
▪ WEIGHT OF BORON-EPOXY TITANIUM BEAM, W_{comp}	6.5 (LB)
▪ WEIGHT SAVED, $W_{conv} - W_{comp}$	0.8 (LB)
▪ WEIGHT SAVING FACTOR, $W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$	0.11 (LB)
▪ COST OF CONVENTIONAL TITANIUM BEAM, C_{Tconv}	\$480
▪ COST PER POUND OF CONVENTIONAL TITANIUM BEAM, $C_{conv} = \frac{C_{Tconv}}{W_{conv}}$	66 (\$/LB)
▪ COST OF BORON-EPOXY TITANIUM BEAM, C_{Tcomp}	\$762
▪ COST PER POUND OF BORON-EPOXY/TITANIUM BEAM, $C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$	120 (\$/LB)
▪ COST PER POUND OF WEIGHT SAVED, $C_{WS} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv} - W_{comp}}$	352 (\$/LB)

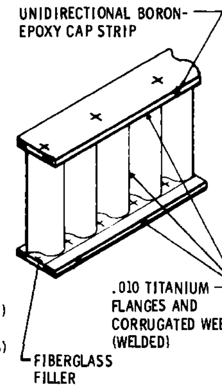


FIG. 19–*SST floor beam cost.*

Similar studies have been made for proposed new composite floor beams, compression panels, and control surfaces (Figs. 17 through 21) with the results plotted in Fig. 22. Proposed applications falling in the upper left area of the graph are the most desirable from a cost effectiveness viewpoint, and those falling farthest down and to the right are least desirable. As would be expected, simple, highly efficient structures such as floor beams are the most cost effective, while more complex structures such as control surfaces are less cost effective.

▪ WEIGHT OF CONVENTIONAL SKIN AND STRINGER PANEL, W_{conv}	4.78 (LB)
▪ WEIGHT OF BORON COMPOSITE -TI PANEL, W_{comp}	2.25 (LB)
▪ WEIGHT SAVED, $W_{conv} - W_{comp}$	253 (LB)
▪ WEIGHT SAVING FACTOR, $W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$	0.53
▪ COST OF CONVENTIONAL SKIN AND STRINGER PANEL, C_{Tconv}	\$96
▪ COST PER POUND OF CONVENTIONAL SKIN AND STRINGER PANEL $C_{conv} = \frac{C_{Tconv}}{W_{conv}}$	20 (\$/LB)
▪ COST OF COMPOSITE PANEL, C_{Tcomp}	\$228
▪ COST PER POUND OF COMPOSITE PANEL, $C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$	101 (\$/LB)
▪ COST PER POUND OF WEIGHT SAVED, $C_{WS} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv} - W_{comp}}$	52 (\$/LB)

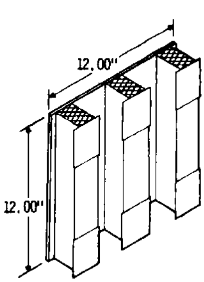
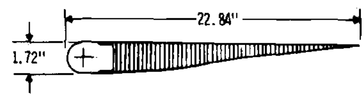


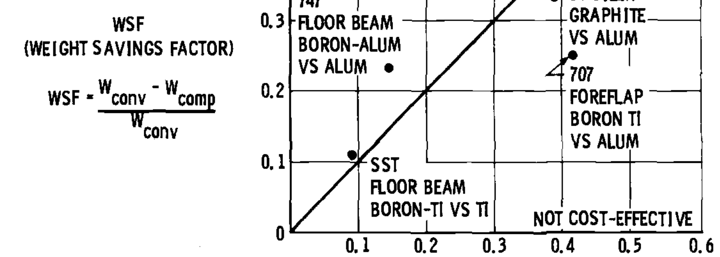
FIG. 20–*707 compression panel cost.*



▪ WEIGHT OF CONVENTIONAL ALUMINUM SPOILER, W_{conv}	14.10 (LB)
▪ WEIGHT OF GRAPHITE-EPOXY SPOILER, W_{comp}	8.92 (LB)
▪ WEIGHT SAVED, $W_{conv} - W_{comp}$	5.2 (LB)
▪ WEIGHT SAVING FACTOR, $W_s = \frac{W_{conv} - W_{comp}}{W_{conv}}$	0.37
▪ COST OF CONVENTIONAL ALUMINUM SPOILER, C_{Tconv}	\$526
▪ COST PER POUND OF CONVENTIONAL ALUMINUM SPOILER $C_{conv} = \frac{C_{Tconv}}{W_{conv}}$	37 (\$/LB)
▪ COST OF COMPOSITE SPOILER, C_{Tcomp}	\$1153
▪ COST PER POUND OF COMPOSITE SPOILER, $C_{comp} = \frac{C_{Tcomp}}{W_{comp}}$	130 (\$/LB)
▪ COST PER POUND OF WEIGHT SAVED $C_{ws} = \frac{C_{Tcomp} - C_{Tconv}}{W_{conv} - W_{comp}}$	121 (\$/LB)

FIG. 21–*737 spoiler cost.*

▪ ASSUME
- $V_w = 100$ FOR 707, 737
- $V_w = 150$ FOR 747
- $V_w = 500$ FOR SST



$$\text{COST FACTOR} = \frac{\Delta C}{V_w + C_{comp}} = \frac{\text{COST}_{COMP} - \text{COST}_{CONV}}{\text{VALUE OF LB + COST OF WEIGHT SAVED}_{COMP}}$$

FIG. 22–*Cost comparisons.*

Design

Good structural design represents the best compromise between design requirements and constraints. The criteria established for each particular design designates the amount of emphasis to be placed on each factor. The designer now has a class of materials which is very strong and stiff in a single direction and can be tailored to fit a particular design. This option of varying load carrying ability with direction necessitates that load path be well defined or oriented into desired directions. Attachments and stress at the boundaries of composite structures are of special interest because of the anisotropy of advanced composites. An effective design must consider manufacturing complexity so that it does not become excessively costly or time consuming. The relatively high cost of advanced fibers suggests that cost will be influential in guiding the design approach. Reliability is an important aspect of aircraft structural design and necessitates that a proposed new structure be at least as reliable as the one which it replaces. The potential of proposed new structures is usually judged by weight savings, all else being equal. The savings can be taken in range or increased payload, but for design comparison purposes, weight is more definable and understandable. Consideration of all these factors, combined with common sense, will result in a design which demonstrates the vast potential associated with advanced fibrous composites.

The many unique features of advanced composites require that the designer establish a set of design guidelines which are by no means rigid yet enable the development of a consistent design approach. The following is a list of proposed general guidelines which are consistent with the requirements of minimum weight, cost effectiveness, and reliability inherent in aircraft structural design.

Put Fibers in Direction of Principal Stresses

Typical properties of unidirectional and multidirectional boron-epoxy composites are shown in Fig. 23. Multidirectional angle-plied laminates may be of

(1) PUT FIBERS IN THE DIRECTION OF PRINCIPAL STRESSES

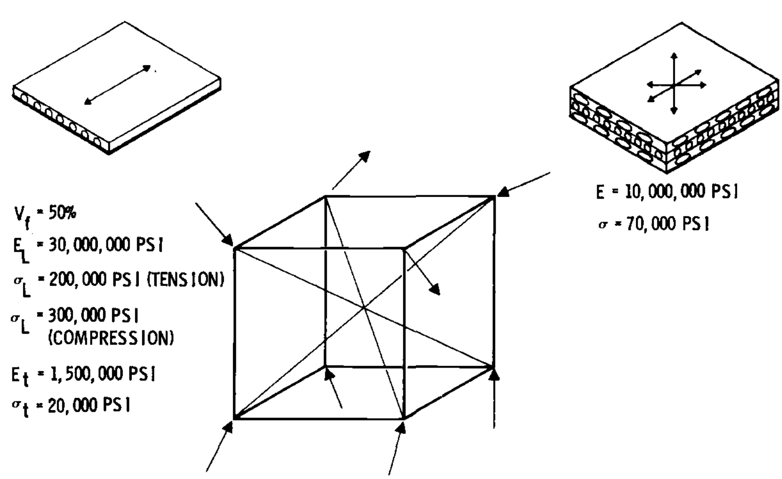


FIG. 23–*Load path.*

use for certain stiffness critical applications, but in general, the complex unpredictable stresses associated with transition regions at the boundary could limit their effective usefulness. The basic requirement of a structure is to transmit and react loads in space. Any system of concentrated forces can be brought into equilibrium by a three-dimensional space truss consisting of only tension and compression members. With a new material available which is several times stiffer and stronger in a single direction than existing structural materials, it seems logical that the greatest weight saving is going to be realized only after the basic function of a structure proposed for redesign is reevaluated considering the desirability of axial load paths.

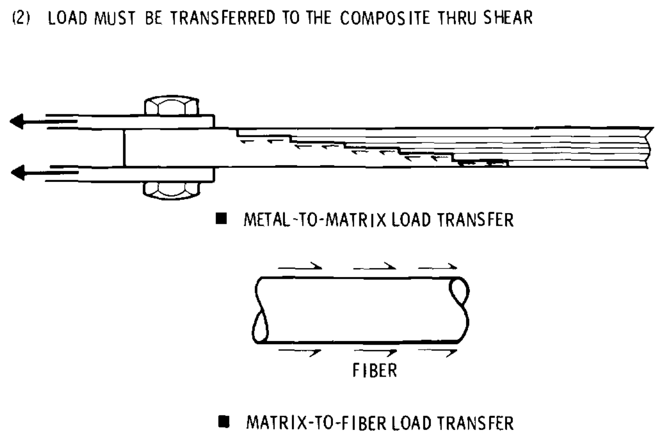


FIG. 24–*Attachments.*

Load Must be Transferred to the Composite Through Shear

Acceptance of advanced fibrous composites requires confidence in structural adhesive bonding. Optimum material properties are obtained only when external load is sheared into each layer of fibrous composites, as illustrated in Fig. 24. The matrix material serves to transfer load to each fiber through the matrix-fiber interface bond. Proper attachment geometry and good adhesive bonding insure that each fiber in the composite is carrying its share of the overall load.

Do Not Cut Holes in Highly Loaded Regions of Fibrous Composites

Theoretical investigations of stress concentrations in anisotopic plates indicate that concentration factors are generally higher than those obtained for isotropic materials. The photoelastic stress patterns seen in Fig. 25 indicate a stress concentration factor for unidirectional boron-BP 907 epoxy composite considerably higher than that for aluminum. Common structural metals flow plastically at points of high stress concentrations resulting in a redistribution and relief of troublesome stresses. The only relief for unidirectional composites at stress concentration points is that fiber strength is a funcion of length, and, therefore, a high load over a very short length will allow a higher overall failure stress. The

(3) DO NOT CUT HOLES IN HIGHLY LOADED REGIONS OF FIBROUS COMPOSITES

$$K = \frac{\sigma_A}{\sigma_\infty} = 1 + \sqrt{2\left(\frac{Ex}{Ey} - \mu_{xy}\right) + \frac{Ex}{Gxy}}$$

FIG. 25–*Stress concentrations.*

(4) USE ISOTROPIC METALS IN COMPLEX STRESS AREAS.

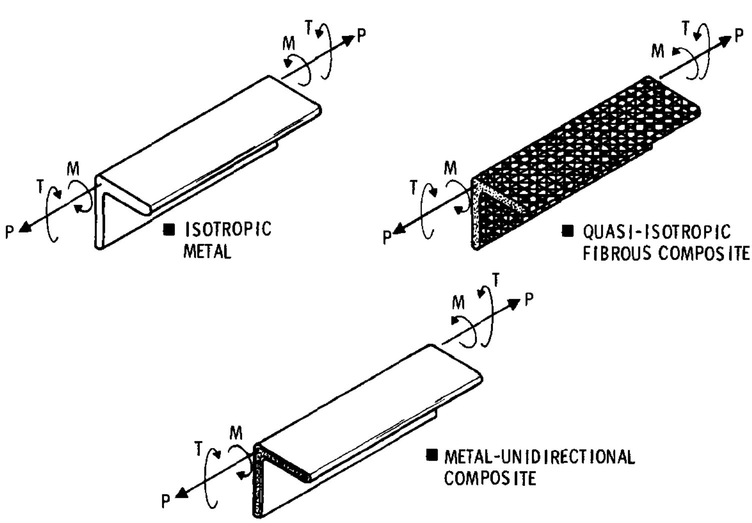


FIG. 26–*Complex stresses.*

(5) HIGHEST PAYOFFS ARE IN AREAS WHERE LOAD PATH IS UNIDIRECTIONAL AND WELL DEFINED

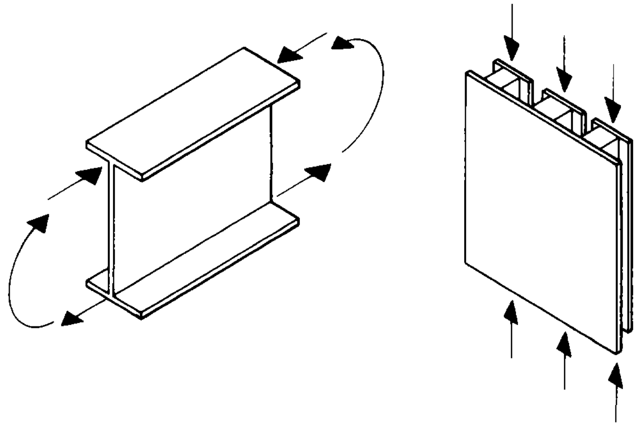


FIG. 27–*Unidirectional load path.*

use of angle-plies for reducing the stress concentration at holes appears promising.

Use Isotropic Metals in Complex Stress Areas

At corners or where stress magnitude and direction change with time, or where loads are not known accurately, metals are more forgiving and should be considered. For a typical structural component, as shown in Fig. 26, where the load in one direction predominates but smaller loads must be carried in other directions, unidirectional fibrous composite can be efficiently utilized to transmit a large portion of the high axial load, and thin metal used to carry all secondary stresses. Added benefits from the use of metal in conjunction with

(6) SHEAR EFFECT MUST BE CONSIDERED IN COMPRESSION STABILITY ANALYSIS

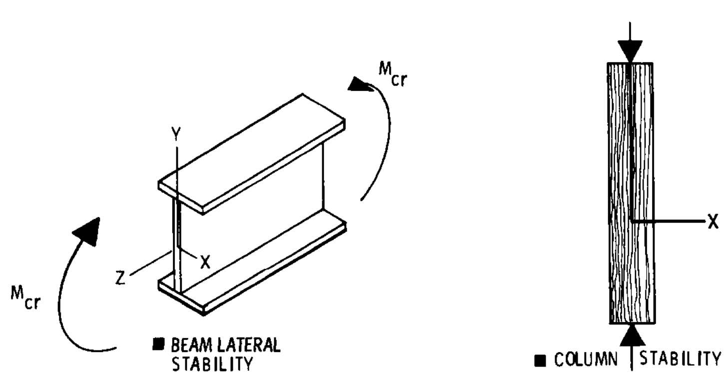


FIG. 28–*Shear effect.*

unidirectional composites are that metal provides a protective shield, facilitates attachments, and allows detection of fatigue cracks well before ultimate fatigue failure.

Highest Payoffs are in Areas Where Load Path is Unidirectional and Well Defined

Metal which carries an axial load, as in beam or column flanges in Fig. 27 can be replaced by unidirectional fibrous composite with a resultant weight reduction proportional to the ratio of the specific strengths or moduli. These ratios, being of the order of three to five, yield weight savings of 66 to 80 percent. Total structural weight saving is then highly dependent upon the percentage of the existing structure which is convertible by means of unidirectional composite application.

Shear Effect Must be Considered in Compression Stability Analysis

The interlaminar shear modulus, G_{xy}, of unidirectional composites is relatively low when compared to its high axial stiffness. The utilization of the axial stiffness in compression structures requires that the low shear modulus does not initiate premature buckling. Plate, column, and beam buckling theory must be reexamined so that out-of-plane stiffnesses are carried along and appear in the final solutions (Fig. 28).

Choose Matrix Material, Fiber, and Volume Percent Fibers to Optimize Desired Properties

The ability to tailor fibrous composite materials to optimize desired properties is a desirable feature which must be exploited to the maximum extent possible for the attainment of optimum designs (Fig. 29). The range of material properties available for existing matrix materials and fibers, along with the options of changing fiber location and amount, results in a wide range of composite material capabilities. Trade studies must be conducted for each

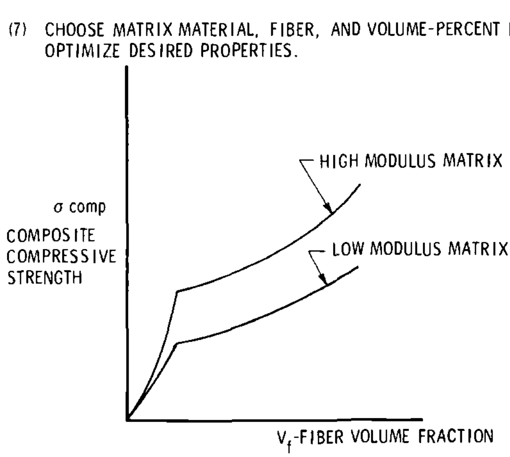


FIG. 29–*Optimum properties.*

proposed design usage because of the interaction of composite material properties.

Allow for Reasonable Analysis Capability

The five material elastic constants for unidirectional tape increase to 21 as composites tend toward three-dimensional anisotropy. The use of finite strips of

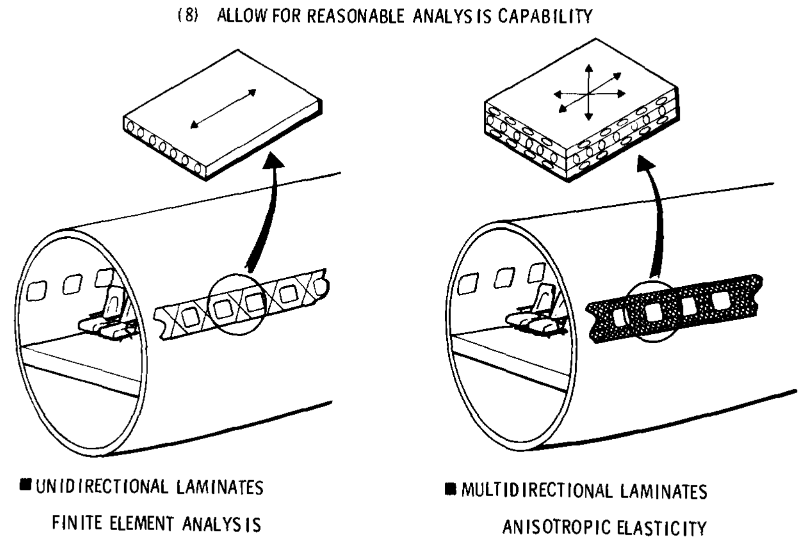


FIG. 30–*Analysis capability.*

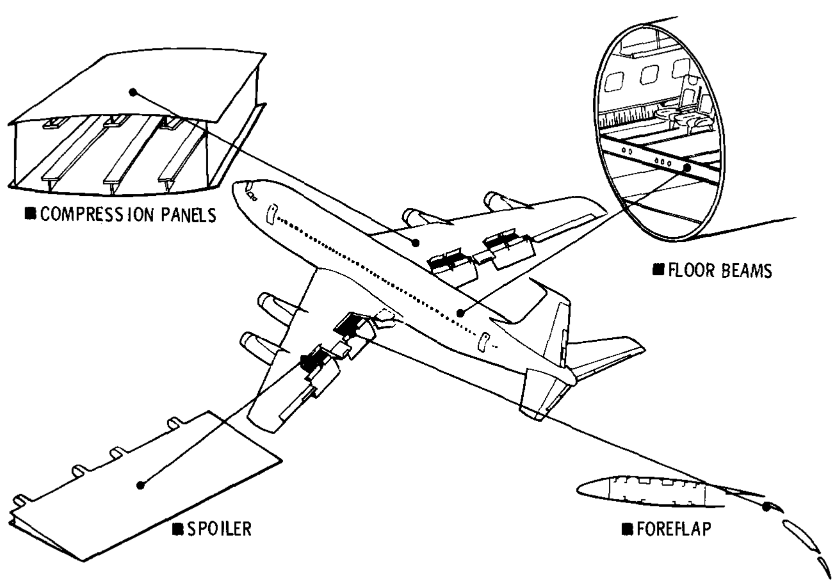


FIG. 31–*Airframe application of composite assemblies.*

unidirectional composite along with isotropic metals allow structural analysis to be accomplished utilizing existing finite element structural analysis programs (Fig. 30). Anisotropic elasicity solutions for basic anisotropic plate and shell structures, although of academic interest, are of limited practical value because of the complexity of even the most basic problems.

The key to the extent and timing of the impact of advanced composites on the commercial aircraft industry is the design confidence developed through proper use and understanding of advanced composites under a consistent design approach.

Applications

Using the design approach presented, several aircraft structural components have been identified as having a high potential for weight reduction when redesigned using boron-epoxy composites. Structures incorporating boron-epoxy composites can be grouped into three general categories: (1) structures redesigned using advanced unidirectional composites to result in a lower weight cost effective structure, (2) more complex, multidirectionally loaded structures redesigned using advanced composites to aid in isolating and solving attachment and fabrication problems and to verify in-service performance, and (3) existing structures using advanced composite material as a least-added weight "fix" to meet minimum or increased performance requirements.

Two components falling into each of these three general categories are discussed in the following section. Floor beams and compression panels, shown in Fig. 31, are examples of structures falling into the first category. These components are 45 and 53 percent lighter, respectively, than the conventional structures which they replace and prove to be cost effective for subsonic commercial aircraft use. Control surfaces such as the spoiler and foreflap (in Fig.

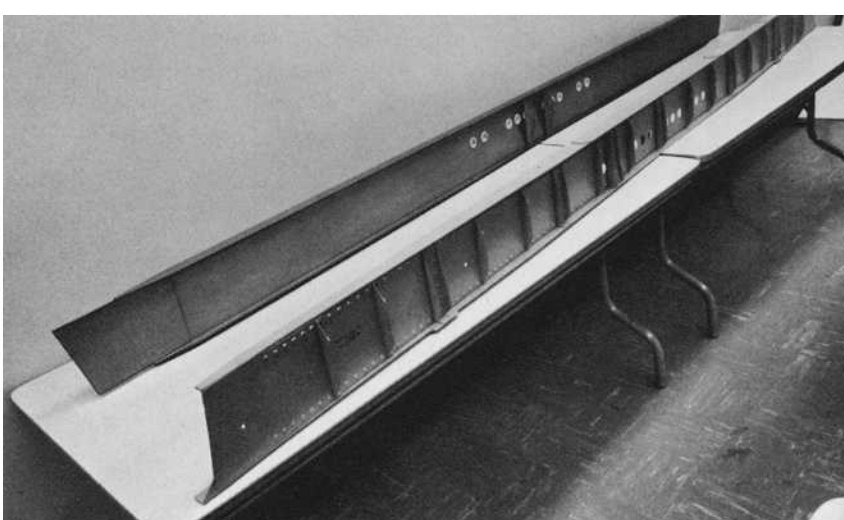


FIG. 32–*Floor beams.*

31) are examples of structures falling into the second category. These components are 33 and 25 percent lighter, respectively, than the current production parts. Examples of structures falling into the third category are ceiling panels and seats.

Floor Beam

An initial step toward the use of advanced composites in commercial aircraft structures has been taken through the design, analysis, fabrication, and test of an aircraft floor beam with boron filament-epoxy flanges and a titanium-aluminum honeycomb web [5]. The beam is designed to replace an existing Boeing 707 web-stiffened aluminum floor beam. The composite beam and the aluminum beams are shown in Fig. 32. In order to achieve a cost-effective design, it was necessary to utilize, to the maximum extent possible, the strength and stiffness of the composite.

The composite beam was designed to the same criteria used in the aluminum beam presently in the Boeing 707. The most critical considerations were the following:

1. a fixed beam depth of 7.16 in.,
2. equivalent beam stiffness (EI),
3. beam and fixity equal to 33 percent,
4. transverse beam stability not considered due to the stabilizing influence of the seat tracks on the compression flange, and
5. capability of withstanding loads imposed by a 9*g* forward ultimate condition. The beam shears and moments resulting from this condition are shown in Fig. 33.

Results from the analysis for the conventional aluminum beam and the composite beam are shown in Table 3. The composite beam crosssection was first converted to an equivalent all titanium beam as shown in Fig. 34. This

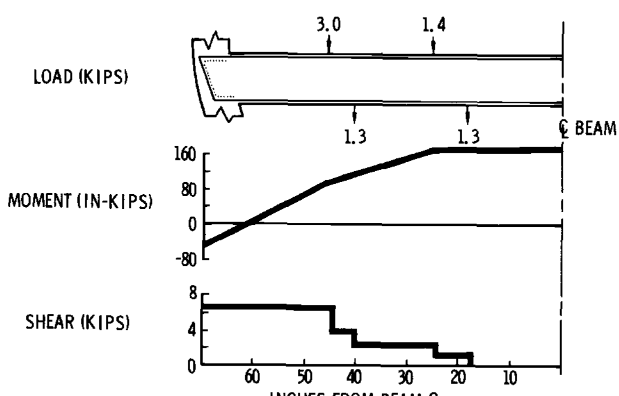


FIG. 33—*Shear and moment diagrams.*

TABLE 3–*Beam analysis.*

	Boron Composite Beam	Aluminum Beam
Moment of Inertia $I = \int y^2 dA$ (in.^{4})	8.22	11.77
Stiffness $EI \times 10^6$, psi	134.80	117.73
Flange stress at failure, psi	306 000 in boron fibers	50 200
Web shear stress, psi	45 800	16 200
Midpoint defection, in.	3.05	3.30

converted beam was then analyzed in the conventional manner used with beams made of isotropic materials. The geometry changes were made by changing the areas of the constituent materials using the ratio of their Young's moduli to the modulus of titanium. These area changes were made by changing the widths of the layers; the thicknesses and location from the neutral axis remained the same.

The floor beam considered is attached to a body frame through the bolt pattern shown in Fig. 35. Solid fiberglass filler 1/4-in. thick is used in the area around the bolts to stabilize the thin titanium web skins. A 0.01-in. titanium doubler is used at the ends of the beams to resist bolt bearing loads. Provisions for seat track and floor panel attachments, while not made on the test beam,

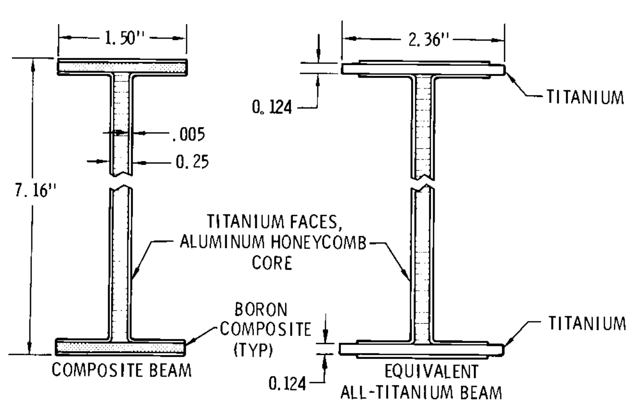


FIG. 34–*Beam cross sections.*

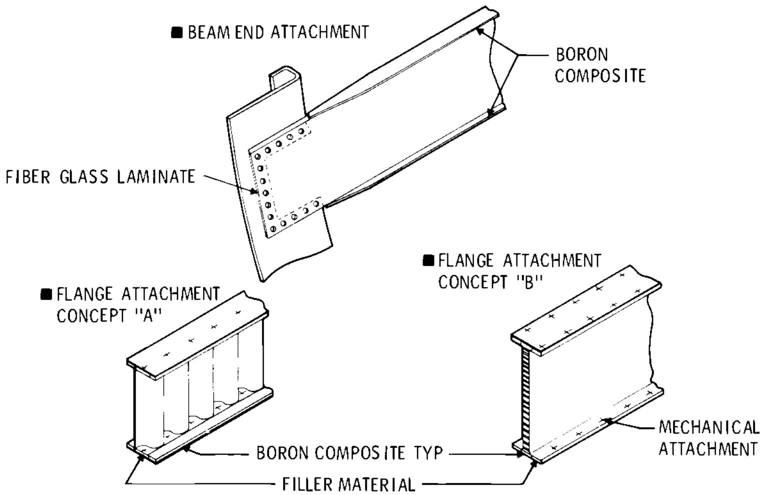


FIG. 35–*Typical attachments.*

were considered. A subsequent beam includes secondary attachment provisions as shown in concept "B" on Fig. 35 with an increase in beam weight of approximately 1/2 lb.

The beam flange "caps" were precured in the mold prior to assembly of the beam. The mold was designed to produce the caps having finished dimensions except for excess on the length. The top of the mold consisted of a thin steel caul plate back up with a silicon rubber seal and a steel pressure bar. This seal, as well as seals on the end of the mold, was necessary to prevent extrusion of the thin boron filaments and the liquified adhesive from the mold during cure. At curing temperature, the adhesive has a low viscosity, slightly greater than water.

Boron tapes were loaded into a mold, a diaphragm constructed, and vacuum

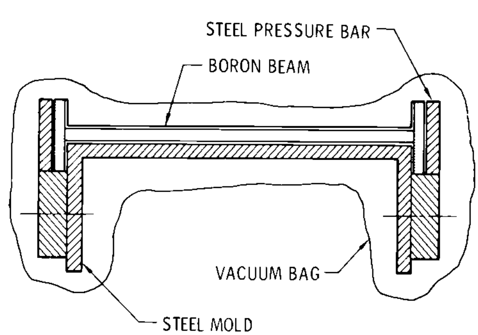


FIG. 36–*Boron beam bonding tool.*

applied. The part was cured under vacuum in an autoclave. The as-cured finish and flatness were acceptable.

The ends of the caps were trimmed by climb grinding. One-thousandths thick cuts were made with a 3/32-in. silicon-carbide, 80 grit, cutoff wheel. To prevent delamination, it was necessary to make the final passes from the opposite side. The grinder was run at 3500 ft per min, with soluble oil fluid used for cooling.

The beam assembly shown in Fig. 36 was bonded in a single step using an epoxy adhesive and a steel tool. The precured boron caps presented no new problems and bonded as easily as more conventional materials. The boron caps were cleaned prior to bonding by lightly abrading the surface and washing them with a solvent.

The problem introduced by bonding materials having different coefficients of expansion was not serious in the case of this beam. In the longitudinal direction, the difference in coefficient of expansion between the titanium and the composite is small (5.7 times 10^{-6} in./in.-F versus 3.1 times 10^{-6} in./in.-F, respectively). In the transverse direction, the difference is large (5.7 times 10^{-6} in./in.-F versus 28 times 10^{-6} in./in.-F, respectively), however, the cap is narrow and the resulting small distortion in the transverse direction was acceptable.

Stiffness of the beams was determined by measuring the first resonant frequency. For comparison purposes, the conventional aluminum beam and the boron composite beam were each mounted in the fixture shown in Fig. 37 and

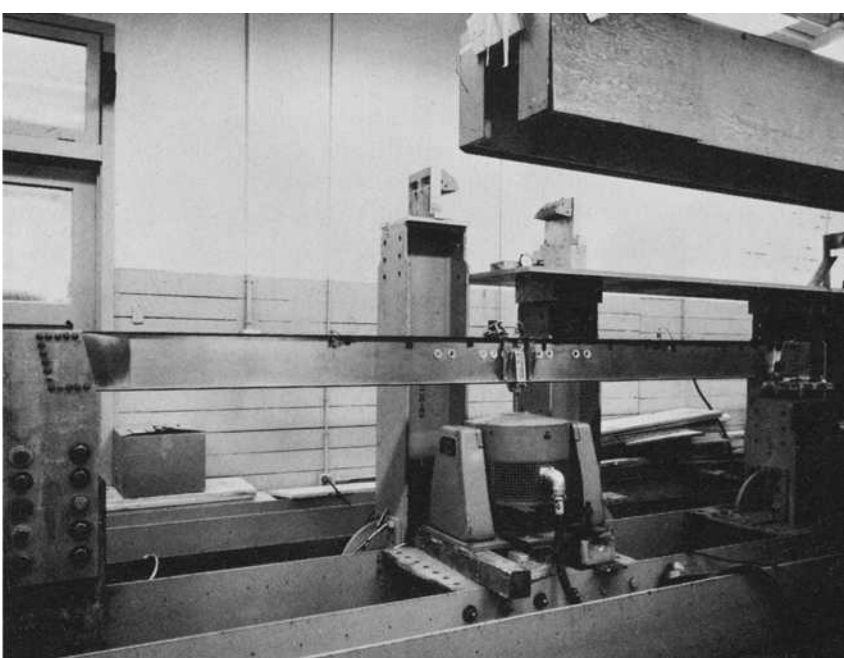


FIG. 37—*Vibration test setup.*

subjected to mechanical vibration by means of a shaker attached at the mid-span. The first resonant frequency in the principal direction occurred at 68 Hz for the aluminum beam and 73 Hz for the boron composite beam. The *EI* stiffness of the beam was designed to be identical, and therefore, the increase in natural frequency of the boron composite beam was due mainly to its smaller mass.

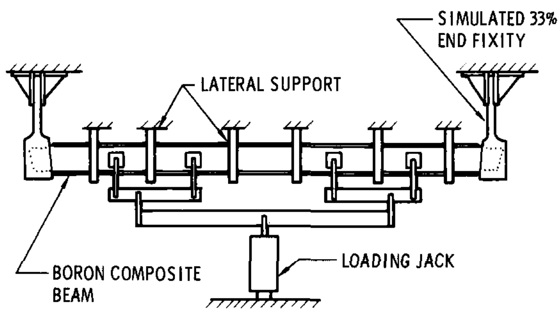


FIG. 38–*Static test setup.*

The beam was next subjected to an ultimate static load test. Figure 38 is a schematic drawing of the test setup with simulated seat tracks in place to provide lateral stability of the compression flange. The beam failed at a boron fiber stress of 316 000 psi. The maximum moment at failure was 15 percent higher than the ultimate design moment. The 16 percent increase in stiffness predicted by theory was verified by measured midpoint deflection.

A cost effectiveness study, summarized in Fig. 17 indicated that weight was saved at a cost of $107 per pound for this application of boron composites, assuming a boron filament cost of $150 per pound. Acceptance of a proposed new subsonic commercial aircraft structure at this cost per pound of weight saved would require a decision from a fairly high level of management.

Table 4 summarizes the results of this study. While the amount of boron filament contained in the beam was only about 20 percent of the total beam weight of 9 pounds, it was located and utilized so as to take maximum advantage of the properties of the advanced composite.

Compression Panels

Wing upper surfaces, as shown in Fig. 39, offer potential for weight savings with the use of advanced composites because of their high intensity of compressive loading. Consistent with the design approach, unidirectional boron composite is utilized to resist the major portion of the compressive load with conventional metals utilized for secondary load carrying and arranged in a manner such that the three major stiffnesses $(EI)_x$, $(EI)_y$, $(GJ)_{xy}$ are preserved. The use of honeycomb sandwich construction is usually required to preserve the

TABLE 4—*Floor beam test results.*

	Boron Composite	Aluminum
Weight (lb)	9.17	16.5
Number of fasteners	22	458
Number of detail parts	23	41
Maximum moment, in·lb	192 000 test result	165 000 design ultimate
EI (lb-in.^{2} x 10^{-6})	134.8	117.7
Fiber stress at failure, psi		
calculated from simple beam theory	306 000	...
measured	318 500	...
Midpoint deflection, in.		
calculated from simple beam theory	3.05	...
measured	3.20	...

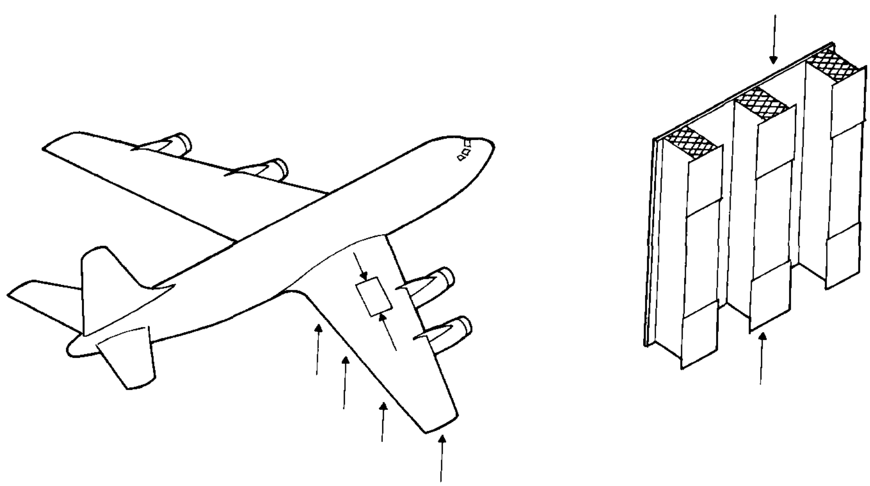


FIG. 39—*Wing upper surface.*

$(GJ)_{xy}$ stiffness because of the low shear modulus, G_{xy}, of unidirectional composites. The basic differential equation governing the buckling of orthotropic panels loaded in axial compression is

$$D_1 \frac{\partial^4 w}{\partial x^4} + 2D_3 \frac{\partial^4 w}{\partial x^2 \, \partial y^2} + D_2 \frac{\partial^4 w}{\partial y^4} = N_x \frac{\partial w^2}{\partial x^2}$$

where N_x is the axial load per unit width of panel, w is the displacement perpendicular to the panel as a function of x and y, and D_1, D_2, and D_3 are the average rigidities of the panel given by

$$D_1 = \frac{(EI)_x}{(1 - \mu_x \mu_y)} \qquad D_2 = \frac{(EI)_y}{(1 - \mu_x \mu_y)}$$

$$D_3 = \tfrac{1}{2}(\mu_x D_2 + \mu_y D_1) + 2(GJ)_{xy}$$

where μ_x and μ_y are Poisson's ratios and $(EI)_x$, $(EI)_y$ are the stiffnesses corresponding to the x and y directions, respectively, and $(GJ)_{xy}$ is the torsional rigidity.

Optimum minimum weight design requires that the maximum possible amount of unidirectional advanced composite material be utilized while holding $(EI)_x$, $(EI)_y$, and $(GJ)_{xy}$ rigidities at or near their original values. The equivalent area substitution method can be used to compute composite structural rigidities with the appropriate equivalent area substituted for the advanced fibrous composite for each of the three separate stiffnesses. An example of equivalent sections for computation of the three rigidities is given in Fig. 40.

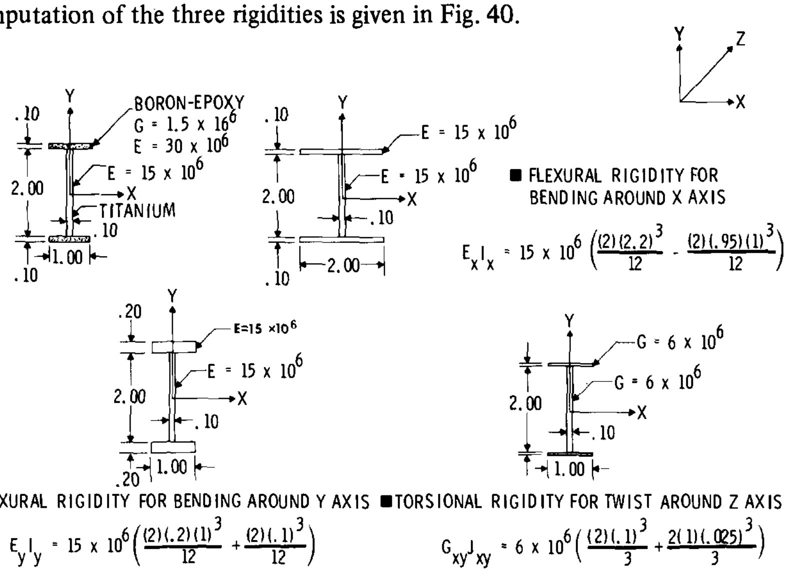


FIG. 40—*Equivalent rigidities.*

Conventional skin and stringer panels, as shown in Fig. 41, can be replaced by structures incorporating unidirectional composites in a very efficient manner when these directional rigidities are considered and compensated for, Fig. 42.

The feasibility of stiffening compression panels with boron-epoxy composite was demonstrated by the design, analysis, fabrication, and test of the panel shown in Fig. 43. This panel proved to be 53 percent lighter than an equivalent conventional all titanium skin and stringer design. The load carrying capacity

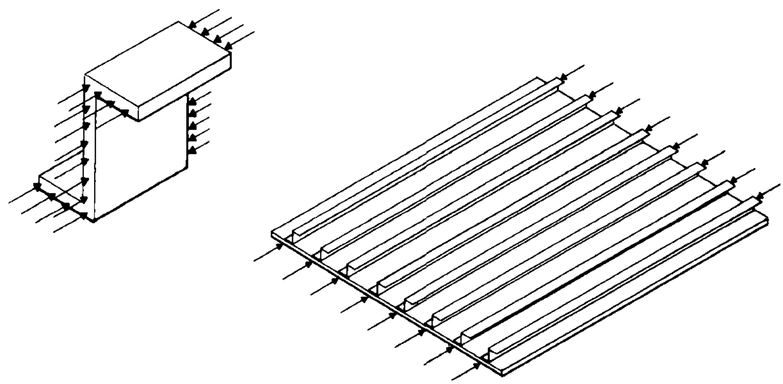


FIG. 41—*Conventional skin and stringer compression panel.*

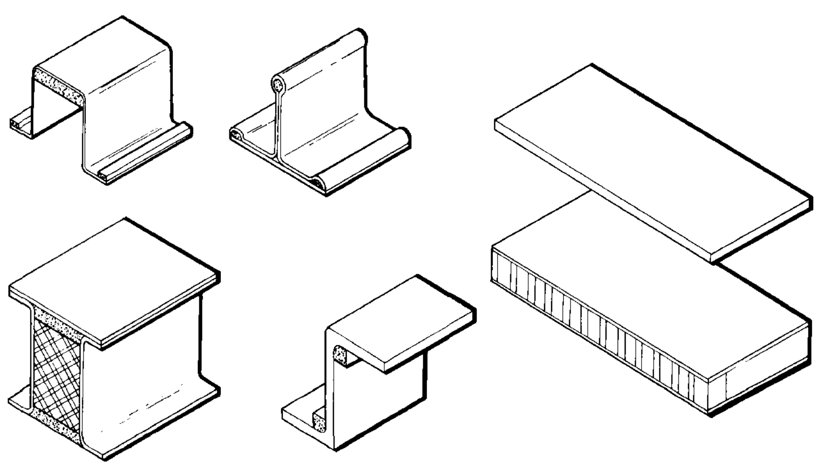


FIG. 42—*Composite concepts.*

was chosen for design purposes as that which is typically sustained by the upper surface of a Boeing 707 wing panel at mid-span. An end view of the composite panel is shown in Fig. 44 with dimensions, material sizes, and locations indicated. The panel was tested in compression.

Strain gages on the skin and stringers were monitored to verify that a uniform load was being transmitted to the panel section. Panel failure occured at an overall strain associated with a stress of 100 000 psi in the titanium and 340 000 psi in the boron fibers. Failure was caused by local instability of the titanium. A side view of the panel after failure is shown in Fig. 45. A comparison of the efficiency of the composite panel tested against two other conventional compression panel configurations is shown in Fig. 46. The allowable effective stress shown includes the weight of honeycomb and adhesive and, therefore,

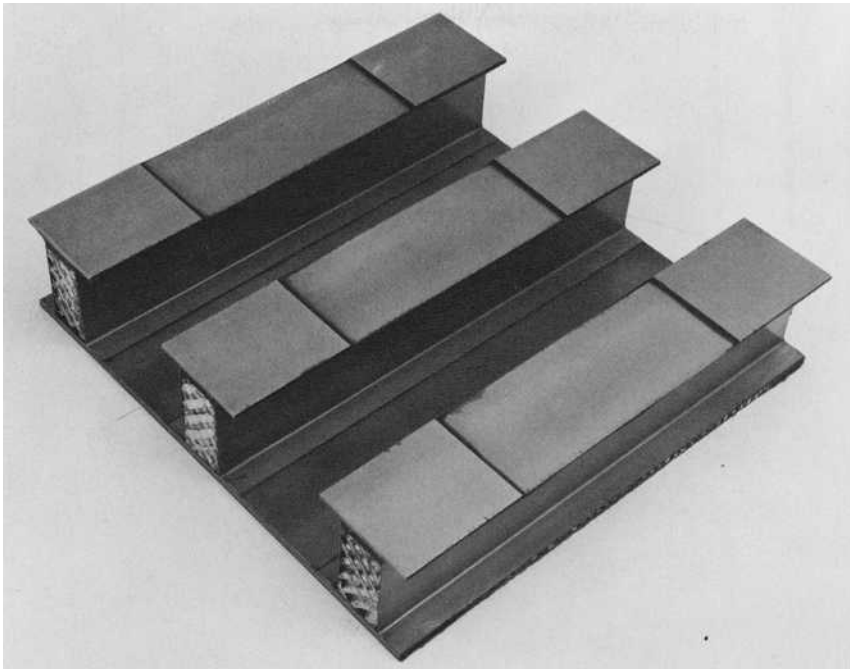


FIG. 43–*Composite compression panel.*

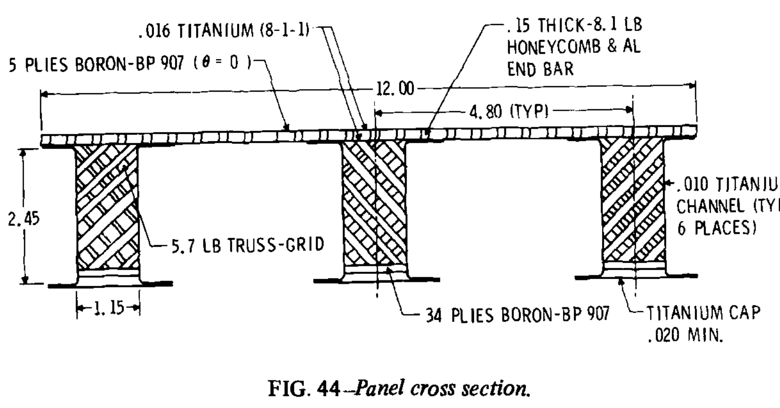


FIG. 44–*Panel cross section.*

gives a direct indication of weight efficiency. A cost study indicated that this particular application saved weight at a cost of \$52 per pound saved (Fig. 20). This puts it well within the range of being cost effective for potential use on subsonic commercial aircraft.

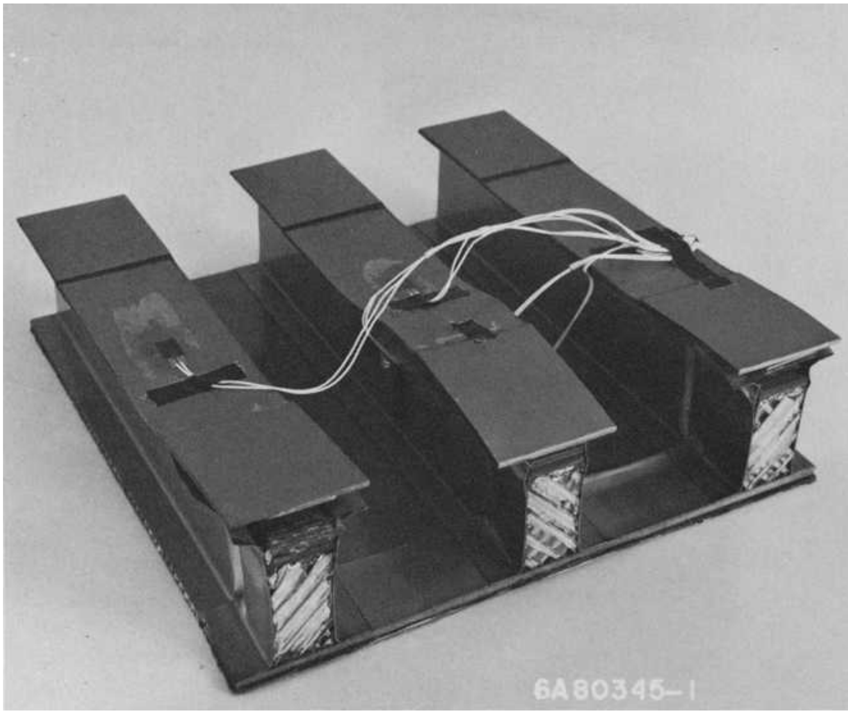


FIG. 45–*Panel failure.*

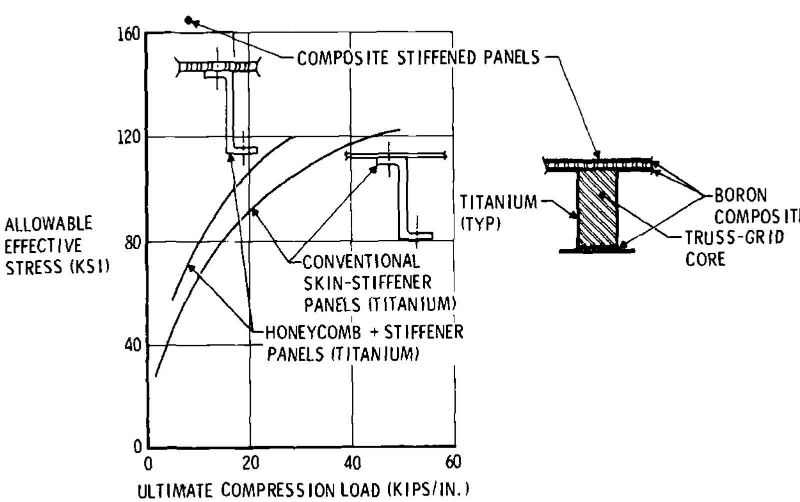


FIG. 46–*Compression panel efficiency.*

Control Surfaces

Extensive future commercial aircraft use of advanced composites requires the determination of performance under in-service environments. In addition to specimen testing to determine the effect of extremes in environment, it is desirable to perform in-service evaluations of structural components. Those components should preferably be readily installed to facilitate flight testing and not protected from in-service damage. Control surfaces such as foreflaps and spoilers are examples of this kind of structure. Because these components are relatively small and lightly loaded, and have problems with attachements, multidirectional laminates, and minimum gage, they offer only marginal

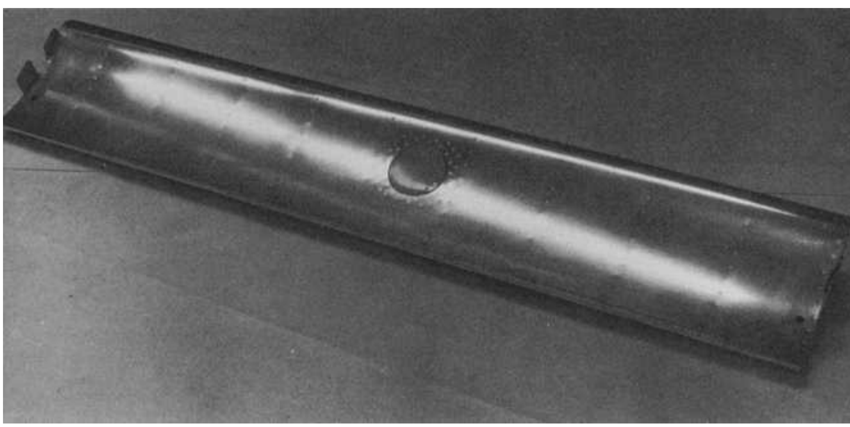


FIG. 47–*707 foreflap.*

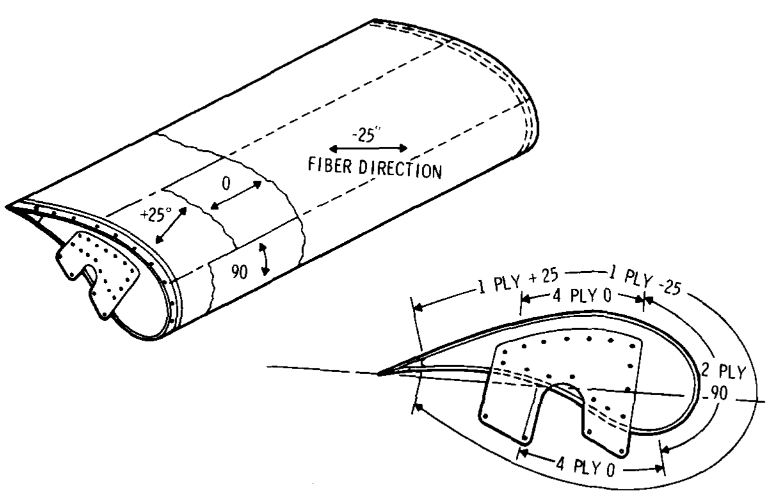


FIG. 48–*Composite foreflap layup.*

potential for cost effective weight savings when redesigned using advanced composites. However, because of the extremes of environment seen by these components, they will play an important role in the rapid development and acceptance of advanced composites for use on commercial aircraft by allowing considerable service experience to be gained in the near future.

Foreflap—The conventional Boeing 707 rib stiffened aluminum foreflap shown in Fig. 47, has been redesigned as a monocoque structure with boron-epoxy, aluminum honeycomb skins and titanium end attachments. The composite

FIG. 49—*Composite foreflap.*

foreflap is shown in Figs. 48 and 49. For analysis and design purposes, the shell structure is assumed to behave as a beam in bending under the distributed load shown in Fig. 50. The maximum moment at mid-span due to this loading determines the amount of unidirectional composite required on the top and bottom of the foreflap. Bending shear, when added to torsional shear due to the location of the shear center, determines the amount of angle-ply material needed. Shear stresses can be carried by the minimum possible number of

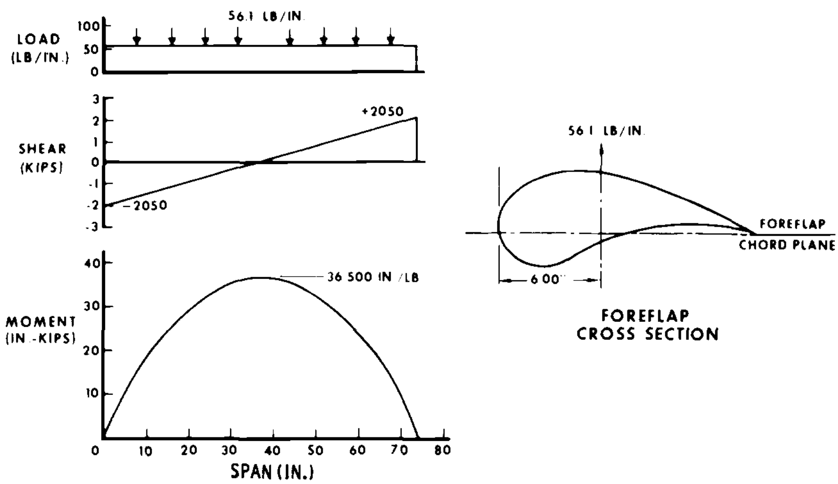


FIG. 50–*Spanwise loading.*

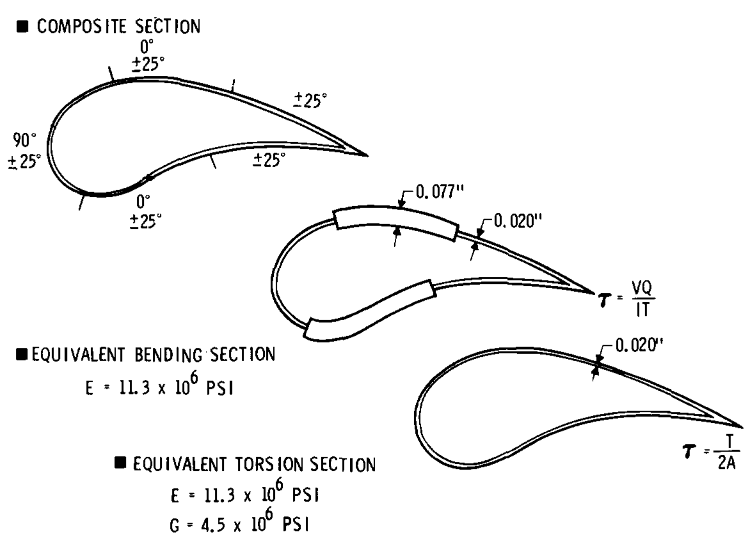


FIG. 51–*Foreflap equivalent sections.*

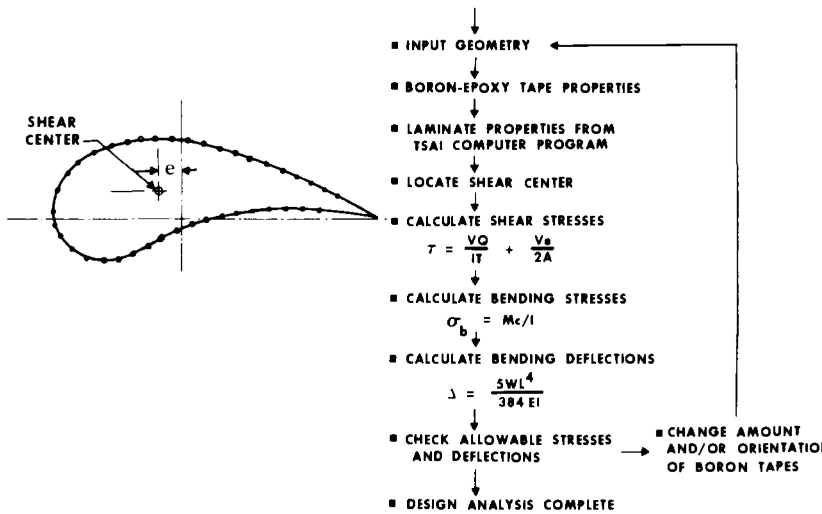


FIG. 52–*Analysis program.*

composite plies, which is one sheet at $+\theta$ and one at $-\theta$ on each side of the ¼-in. aluminum honeycomb. The orientation angle, θ, is chosen such that the shear strength is just adequate to handle the shearing stress. As the angle required gets smaller, the contribution of the angle-plies to the bending stiffness and strength increases. This allows for a reduction in the amount of unidirectional material. A computerized shell analysis using finite elements would be preferable and is possibly essential; however, initial analysis has used elementary strength of material relationships together with an equivalent area substitution technique. The equivalent areas for bending and torsion are shown in Fig. 51. To facilitate analysis, the structure was divided into sections with the area of each section assumed for analysis purposes concentrated at the center of the section. The appropriate geometry for a particular design is then input to a computer program which computes all stresses and deflections based on the simplified theory. A general flow chart for the analysis is shown in Fig. 52. Location and orientation of the boron-epoxy plies can be varied until all stresses and deflections satisfy the design criteria.

Fabrication, static, fatigue, and flight tests were successfully completed. Figure 53 shows the composite foreflap installed for flight testing.

A cost effectiveness study, summarized in Fig. 16, indicates that the Boeing 707 foreflap, redesigned using advanced composites, may not be cost effective. Future commercial aircraft, in which the value of weight saved is higher, could possibly incorporate advance; composite foreflaps or similar control surfaces on a cost effective basis.

Spoiler–The Boeing 737 flight spoiler (Fig. 54) is another example of a control surface which aids in determining in-service performance of an easily

FIG. 53–*Composite foreflap installation.*

replaceable component. Cross-plied graphite-epoxy skins are used to replace the existing aluminum skins and resist torsional loads. Loading and shear, moment and torsion diagrams used to design the composite component are shown in Figs. 55 and 56. The cost study summarized in Fig. 21 indicates that the weight saved on the spoiler is nearly cost effective for use on the 737.

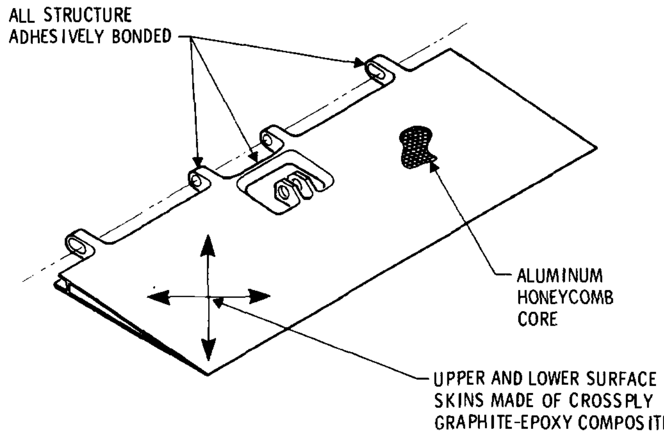


FIG. 54–*737 flight spoiler.*

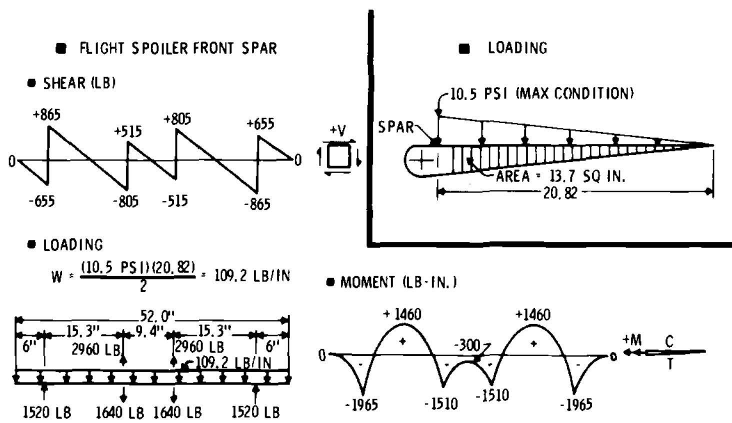


FIG. 55–*Spoiler loading.*

Ceiling Panel

Many structural components which fall short of expected stiffness or strength requirements can look to advanced composites for a least added weight fix. One such example is a ceiling panel (Fig. 57) which during verification testing was found to have a lower natural frequency than expected. One third of a pound of boron composite was used to stiffen the panel to meet minimum design requirements where 1½ pounds of fiberglass would have been required to do the same job.

Seat

A fiberglass seat (Fig. 58) proposed for aircraft use was another example of a structure falling short of expected performance. During verification testing it was determined that the static deflection of the fiberglass seat under service loading was in excess of the maximum amount allowed. The addition of one pound of boron composite was sufficient to stiffen the seat, whereas 4½ pounds of fiberglass would have been required.

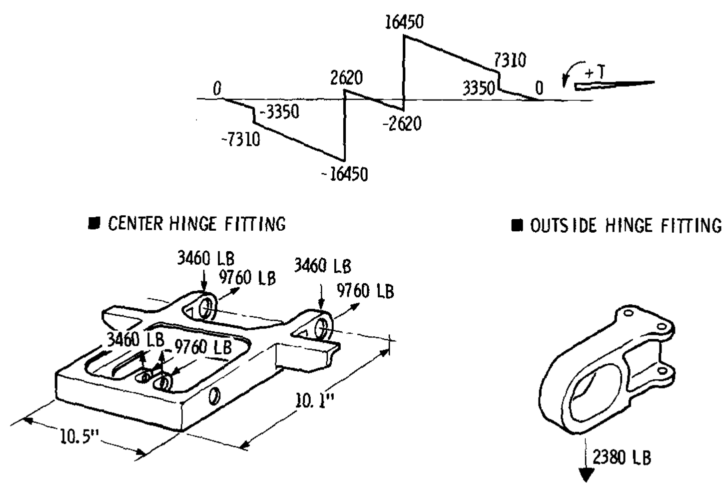


FIG. 56–*Attachment loading.*

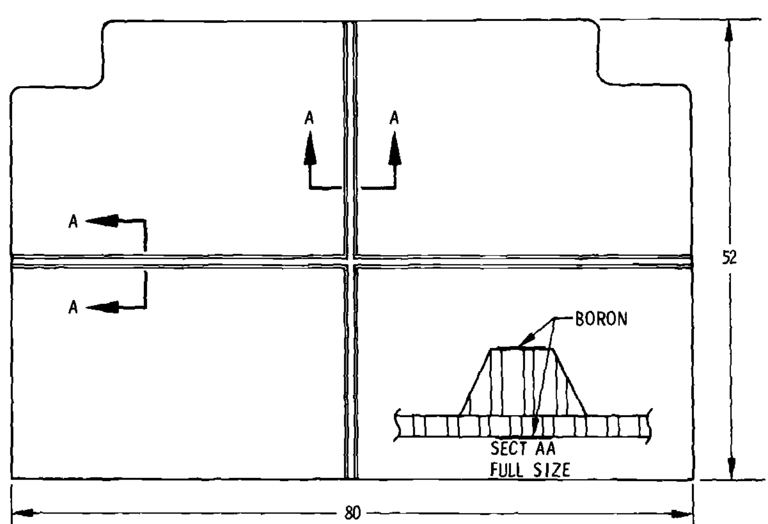


FIG. 57–*Interior ceiling panel.*

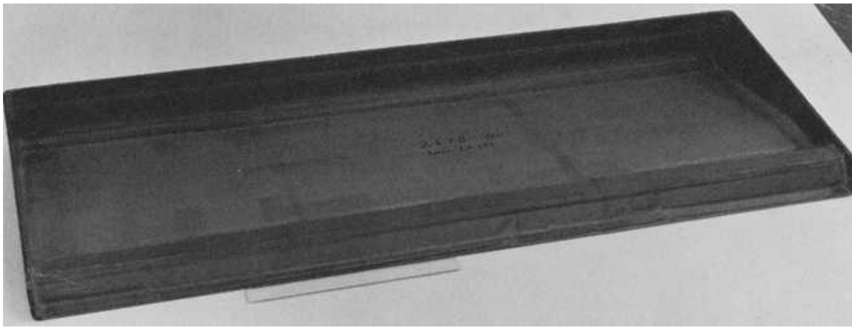


FIG. 58–*Composite seat.*

Conclusions

Commercial aircraft design of the future must incorporate the best combination of materials and design concepts in order to meet increased demands for high performance. Advanced fibrous composites make available to the aircraft designer a new class of materials which has the potential of reducing total structural airframe weight by a significant amount. The relatively high price and anisotropic nature of these materials necessitate that a design approach consistent with commercial aircraft design requirements be developed. This approach should enable weight to be saved at a cost within commercial aircraft guidelines and in a manner which does not sacrifice reliability or structural integrity. The design approach presented has been illustrated by the design and test of several aircraft components which incorporate conventional metal and boron-epoxy composite in a manner which allows the component to be easily analyzed and fabricated, and which saves a considerable amount of weight at a reasonable cost.

Acknowledgments

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Efforts of members of the unit and supporting groups made this chapter possible. Their contributions are gratefully acknowledged.

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L.E. Hackman[1]

Chapter II – Military Aircraft

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ABSTRACT: Military aircraft have utilized fiber reinforced composites for more than twenty years. The bulk of this use has been glass fiber reinforced plastics, which has provided both improved structural efficiency and lower cost. Recently, the advent of high modulus composites has led to its use in highly loaded, stiffness critical wings and control surfaces, as well as other structures.

This chapter details the major structural design approaches used for fiber composites in aircraft and summarizes several applications.

KEY WORDS: composite materials, fibers, aircraft, composite structures, structural analysis, design, wings, fuselages

The spade work to prepare reinforced plastics for big jobs in tomorrow's highspeed aircraft is well under way. Plans and projects in the works will pave the way for a wide use of these plastics in planes, missiles, and auxiliary components.

The studies aim to take advantage of such properties of this type of material as their favorable strength/weight ratio, low coefficient of expansion, good flexibility characteristics, and aerodynamic smoothness.

One of the earliest applications of high-strength plastics–a role that practically launched the material into the field of primary structures–is a glass-plastic outer wing panel which has been under flight evaluation for several months at Wright Air Development Center, Dayton, Ohio.

These introductory statements could have been taken from nearly any current aviation publication, but were taken from *Aviation Week* of 1 June 1953. The referenced articles referred to the development of the AT-6 glass fiber reinforced plastic (GFRP) wing outer panel initiated in 1945 by the Air Force.

Development History

These developments of GFRP in the aircraft structural field began over 20 years ago (Fig. 1). Initial steps were taken by Owens-Corning in the 1930s with the development and commercial production of glass fibers.

GFRP materials were first developed and designed for airframe structures at the U.S. Air Force Wright Air Development Center in 1943. The first such structure fabricated was the aft fuselage of the Vultee BT-15 [*1*][2]. The

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[2]The italic numbers in brackets refer to the list of references appended to this paper.

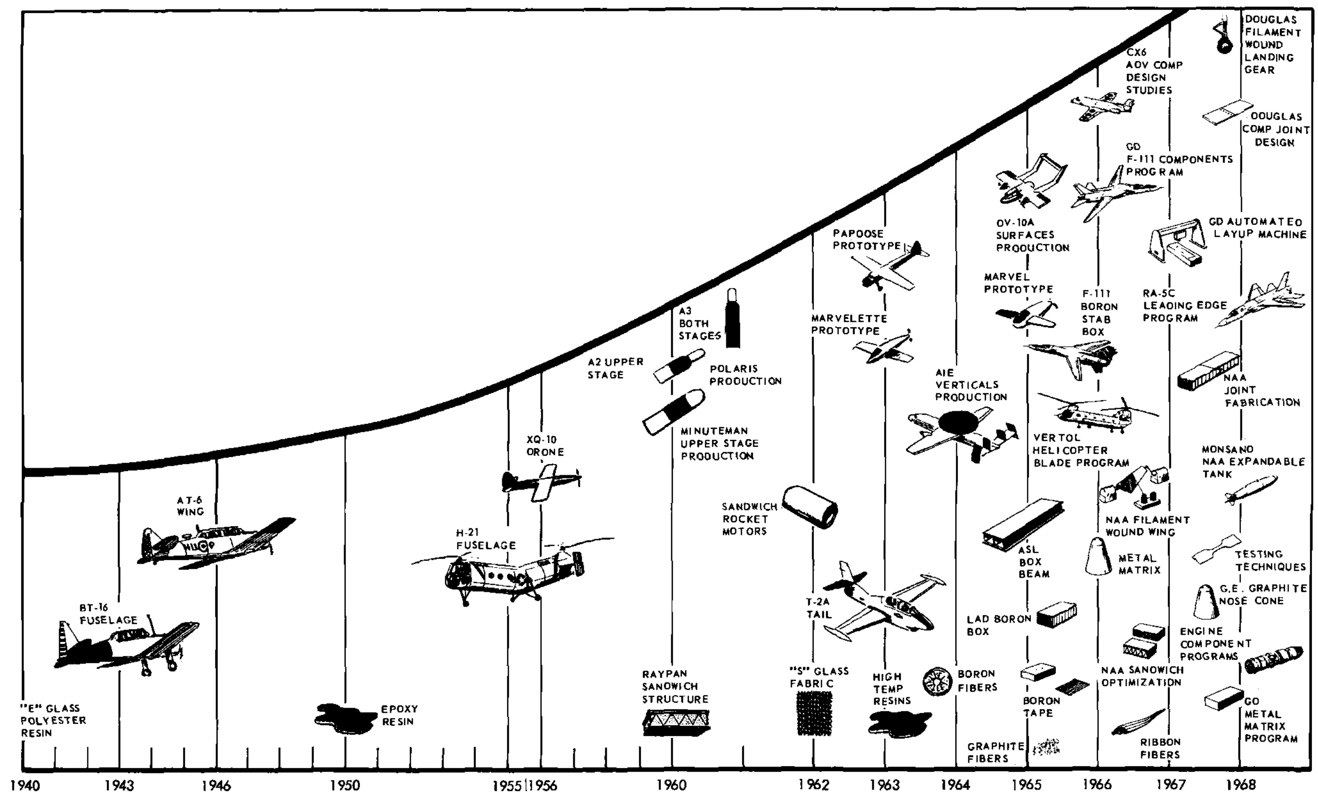


FIG. 1—*Chronology of composites usage.*

construction was a balsa wood core with glass fiber skins impregnated with a polyester resin system. Development work in the area of core materials for use in sandwich skins to overcome the inherent low modulus of early GFRP resulted in the fabrication of radomes for the Black Widow fighter, the B-17, and the B-29.

The next significant step in the development of reinforced plastic for airframe structures was taken at Wright Field with the fabrication of the outer wing panels for the AT-6 airplane [*2*]. Although these wing panels were substantially heavier than the comparable aluminum wing panels, the ultimate load was also substantially higher. Test data proved an increased strength-to-weight ratio of approximately 13 percent for the plastic structure. However, many problems still remained with the use of polyester resins, such as weathering and temperature properties.

With the synthesis of epoxy resins in the 1950s, aircraft engineers were provided with a reinforced plastic system of greatly improved weathering ability. An Air Force contract in the mid 1950s initiated a study with Vertol for the use of GFRP in the fuselage of the H-21 helicopter [*3*]. From this program it was concluded that lower cost structures are potentially possible with the use of GFRP. Piper Aircraft in 1958 investigated GFRP as a primary aircraft structural material and produced a reinforced plastic aircraft which flew in April 1962. This aircraft was fabricated using GFRP skins with a paper honeycomb core.

In 1959 the development of a Fairchild surveillance drone system was begun using GFRP [*4*]. The significant contribution of this program was the use of the GFRP structure as an integral fuel tank.

During the 1950s many applications were found for GFRP particularly in the nonstructural areas of operational aircraft. Today reinforced plastics are being used to an even greater extent, but the role of GFRP has remained one of a secondary or nonstructural nature in the general case, even though its structural potential has been realized by many engineers.

With the development of the COIN (counter insurgency) aircraft concept in the early 1960s, GFRP was suggested as a structural material which would provide the desirable qualities at low cost, with reduced maintenance, and improved corrosion resistance. In early 1963, North American Aviation, under contract to the Navy's Aeronautical Structures Laboratory, Philadelphia, Pa., undertook a GFRP structures research and development program [5]. This program was established to evaluate the static strength, fatigue, and weight characteristics of a typical military aircraft component using available glass fiber reinforced plastic technology. To supplement this structural evaluation, additional experimental and theoretical programs were conducted under a North American Aircraft Independent Research and Development program to provide insight into other design considerations such as repair ability, vulnerability, cost, noise attenuation, performance, and corrosion.

The first major structural application of GFRP to military aerospace systems came in the 1960s with the production of Polaris and Minute Man missile systems. The rocket cases were the ideal application of the high tensile strength of glass fibers. Substantial performance gains were realized on these programs over the conventional metal cases. In the mid 1960s the army sponsored a program with Mississippi State University for the investigation of a high lift low drag aircraft configurations [6]. This program resulted in the fabrication of two all GFRP aircraft (Marvellet and Marvel prototypes). During the same time period the Grumman Aircraft Corp. initiated a production program for the vertical stabilizers on the A1E aircraft. Even though the requirement was for electronic purposes, it did require the application of GFRP to the primary structure of the vertical stabilizer and became the first production application.

It was also in the mid 1960s that boron fibers became available to the industry in limited quantities, and the first advanced filamentary composites hardware programs were initiated. The two major boron application programs were the General Dynamics F-111 horizontal stabilizer program and the Boeing Vertol helicopter blade program which is described in detail in Chapter IV. Both of these programs have involved the extensive development of design and fabrication techniques with boron filamentary composites. Substantial advantages have been shown in both programs with the application of boron composites to the military air vehicle system. The development and manufacturing techniques were also considered very important at this time for the continuing progress of filamentary composites. Programs were initiated to develop two major fabrication concepts. General Dynamics was awarded a contract to develop a production tape lay up machine. North American Aviation was awarded a contract to demonstrate the application of filament winding to an aircraft lifting surface structure. Both programs have been successful in meeting their objectives and establishing a new basis for thy fabrication of filamentary composite structures.

The Air Force in the last several years has entered into many development contracts directed toward re-entry vehicles, aircraft power plants, aircraft wing and fuselage structures, and all types of structural applications in military aircraft. This great interest by the Air Force and participating contractors is culminating in the application of filamentary composites to near term systems such as F14, F15, and B1.

General Advantages and Applications

Many advantages are to be gained through the use of fiber reinforced plastics (FRP) in primary airframe structures. Multicontinuous fiber load paths and bonded joints provide inherent "fail-safe" characteristics for high load-carrying capability after damage. These fail safe characteristics have been demonstrated many times in the static test of filamentary composite structures. The T2A GFRP horizontal tail program was the first of these demonstrations in which 85 percent of the failing load was sustained by the structure after failure. This was compared to the 20 percent failing load sustained by the corresponding metal structure. A GFRP flap structure after failing at 140 percent of limit load was actually able to be loaded to 150 percent without a total structural collapse. This is only one of the characteristics of filamentary composites which can be attributed to the multiple load path nature of the material. The capability of tailoring material properties to meet load intensity and direction adds to the strength and stiffness in addition to weight advantages of filamentary composites.

Good notch insensitivity with attendant low crack propagation greatly improves the fatigue properties of filamentary composite materials. Early programs on full scale structures have shown substantial increases in the fatigue life of filamentary composite structures over their metal counterparts. On the basis of fatigue life to weight ratio, filamentary composites show substantial improvements over titanium which is the best of the metal materials in fatigue. The improvements can be expected to be as much as 100 percent over titanium.

Another advantage is the fact that the structure can be made of a minimum number of large, one-piece subassemblies. The capability of molding filamentary composite materials and their adapation to sandwich type structures greatly reduces the number of parts in the structure. The reduction in the number of parts manifests itself in many ways: lower tooling costs, fabrication costs, handling costs, and finally, greater reliability through a reduction of the number of joints and discontinuities in the structure. With the use of sandwich structures, optimum aerodynamic contours are easily fabricated and remain in shape at high load levels.

Absolute corrosion resistance is a characteristic uncommon in all metal structures. No such mechanism as electrochemical corrosion exists in the reinforced plastic materials. However, a degradation from weathering can occur in reinforced plastic materials, primarily from ultraviolet or moisture penetration. This can be more readily controlled by protection of the material than is the case with metal corrosion.

Composites exhibit high structural damping properties (especially in sandwich construction) which greatly reduce vibration and noise transmission. In certain military aircraft it is necessary to protect the crew or provide an environment which will allow long periods of sustained flight. Such an environment is enhanced by the reduction of vibration and noise transmission inherent in a filamentary composite structure.

The high resistance to impact of filamentary composite materials helps localize damage and maintain contours around the areas of damage. The inherent elastic behavior of the materials up to failure provides the characteristic that either the material fails totally or does not fail at all. Projectile impacts are localized to an area only slightly greater than that of the projectile itself, whereas metal structures cracks are initiated which may tear the skin materials over a large area. The stress concentrations around cracks in metal are extremely high and tend to assist propagation, whereas in filamentary composite materials the stress concentrations remain low and no crack propagation occurs in the area of damage. Composites also offer increased flight performance due to reduced parasite drag coefficient of the smooth skin finish [6]. Again, the molding characteristics of filamentary composite materials fabricated on external mold line tools provide contours as smooth as the tool itself, and in the application of large parts, reduces the number of skin joints and fasteners on the external mold line. The work at Mississippi State University under the Army Marvellet program has demonstrated reductions of as much as 40 percent in the parasite drag or approximately 15 to 20 percent of the total drag.

Minor field repairs are facilitated by the lack of crack propagation, and such repairs can be limited to nonstructural patches sufficient to maintain a smooth aerodynamic surface. More thorough structural repairs can be achieved at times of major overhauls.

An improved cost effectiveness has been shown in recent studies of the application of filamentary composite materials to military aircraft structures using combinations of glass and advanced filamentary materials, even at the present day costs of boron and graphite [7]. For the greatest effectiveness, the advanced filamentary materials must be limited to the specific application of solving major stiffness problems in the airframe structure. With substantial reductions in fiber costs, more extensive use of the advanced materials can be made and should show improvements in their effectiveness.

Reinforced plastic materials provide a capability for both active and passive radar nonreflective systems. The use of such systems in the basic design of a military aircraft can reduce the detection range for aircraft substantially. The capability of a GFRP lifting surface to pass radar energy through the surface reduced the detection range by 50 percent. In many cases the designer thinks only of the weight savings as a potential improvement with filamentary composites. However, there are other major advantages one can exploit in aircraft systems through the use of filamentary composites. For example, an aircraft required to loiter on target for a long period of time could use as high an aspect ratio as possible for aerodynamic efficiency. The aspect ratio used in conventional aircraft is a trade off between wing structure, flutter requirements, and a reasonable weight for the aircraft. However, because of the high stiffness to weight ratio of the composite materials, one can extend the aspect ratio significantly with filamentary composites over that of the similar aircraft designed with conventional metal materials. This allows the designer to trade off between a major weight saving in the aircraft and to improve the performance by changing the aspect ratio.

Another example is the case of the high speed air superiority fighter in which drag is of prime interest to the performance. One of the parameters determining aircraft drag is the wing thickness. The high stiffness to weight and strength to weight ratios of advanced composites allows the designer to consider reducing the wing thickness without paying severe weight penalties. Maintaining an equal weight to the conventional metal aircraft the thickness can be reduced significantly to produce a lower drag aircraft. Through these types of design trade offs and greater structure flexibility, significant changes in aircraft configurations can be expected in the future.

Design Considerations

Aircraft structures have been designed in the past to meet specific requirements based on Government specifications, such as FAR 25 and MIL-A-8860. There are no basic reasons why such criteria cannot continue to apply in the area of filamentary composites. Some of these criteria, which inherently relate to isotropic or homogeneous materials, need redefinition before being applicable to composite design, but the fundamental requirements are still valid. The following design requirements are restated as fundamental criteria independent of material characteristics. Where redefinition of terms is necessary in order to recognize the anisotropy of composites, several approaches have been expressed by various sources and are presented for guidance.

Strength and Load Criteria

The aircraft structure shall have sufficient strength to sustain ultimate load without failure. Limit load establishes one of the basic strength levels for the design of an airplane. Limit load for a given design condition is defined as, "the maximum load expected to be encountered in service." Of all the limit loads on a part, the most critical for design is referred to as "design limit load" for the part within the specified design limits. "Limit stress," in turn, is defined simply as the stress magnitude resulting from limit load.

Ultimate loads consist of the corresponding limit loads multiplied by an ultimate factor of safety. However, in a few individual cases, additional factors are specified in the derivation of ultimate loads for reasons of added safety, rigidity, quality assurance, and wear. Also, in certain particular cases, such as landing conditions, ultimate loads are directly specified. Design ultimate stress is the stress magnitude resulting from design ultimate load.

Deformation Criteria

The aircraft structure shall be capable of withstanding limit load without excessive deformation and with negligible permanent set. The cumulative effects of elastic, permanent, and thermal deformations resulting from the application of limit design conditions shall not interfere with the functioning of any aircraft subsystem, adversely affect its aerodynamic characteristics, nor require repair or replacement of any part.

Fatigue Life Criteria

The aircraft structure shall be capable of sustaining the operational spectrum of loading and enviroment for a period equal to the specified service life times a multiplying factor. (The Air Force requires a full-scale test demonstration of a factor ranging from 2.0 to 4.0, while the Navy requires that a factor of 2.0 be demonstrated by analysis and tests.)

Material Design Allowables Criteria

The foregoing requirements and definitions do not involve the characteristics of the materials which are utilized in the design of the aircraft. These material characteristics are the basis upon which material design allowables are established—the statistically defined stress levels beyond which the material may not be loaded without high probability of violating a basic design requirement. The characteristics of a filamentary composite laminate can be established by empirical, semiempirical, and analytical approaches, as described in the following paragraphs.

In the empirical approach, the mechanical properties of each laminate orientation to be used are determined directly by tests. This approach has the disadvantage of time and cost, because tests of many separate orientations may be required for a single design project. This approach also seriously handicaps the optimization process since the variations with orientation configuration are unknown.

In the semiempirical approach, the mechanical properties of an uniaxial lamina are established by tests, and analytical procedures are then used to calculate the characteristics of any laminate orientation. The validity of the analytical extrapolation of properties may be verified by spot-check tests if desired, but the expensive comprehensive test programs of the empirical approach are avoided.

The analytical approach uses micromechanics theory to predict the properties of the uniaxial lamina based upon the fiber and matrix properties. The laminate properties can then be determined analytically as in the semiempirical approach. This totally analytical approach is highly advantageous in the optimization of aircraft structures in the preliminary design phase.

However, differences of opinion exist within the industry with regard to the application of laminate characteristics to the design process. These differences of opinion involve the definition of such basic concepts as limit and ultimate allowable stresses. The more prevalent approaches to material allowables criteria which are currently employed by various members of industry are delineated in subsequent paragraphs. Because of the lack of sufficient data on the response and failure of actual aircraft structural components, it cannot as yet be firmly established which of these approaches is the most appropriate for general aircraft design purposes.

Criterion A [8]—The proportional limit of a cross-plied composite laminate is the stress level which represents the onset of permanent damage, since above this level the material exhibits a permanent reduction in elastic modulus. The proportional limit, therefore, should be selected as the limit allowable stress (the maximum stress used for limit design), since permanent damage is not acceptable. The stress level at two-thirds ultimate allowable stress is too far into the permanent damage region, in characteristic boron-epoxy stress-strain curves, to be acceptable as the limit allowable stress. Ultimate allowable stress is simply defined as the maximum attainable stress in the material; the point at which complete failure occurs.

It should be noted that this criterion represents an extremely conservative approach and is presented to illustrate one limit of the range of philosophies which have been considered. However, this philosophy is not currently used by any major organization active in advanced composite programs.

Criterion B [9]—The material allowables criteria can be stated as follows:

1. Design ultimate loads shall result in a stress that does not exceed the ultimate allowable stress for the laminate used, where ultimate allowable stress is the maximum laminate stress attainable without rupture of any lamina.

2. Design limit loads, as defined by the vehicle specifications, shall result in a stress that does not exceed the limit allowable stress for the laminate used, where limit allowable stress is that stress beyond which no lamina suffers intolerable degradation of stiffness and permanent deformation.

As an illustration of the manner in which limit and ultimate values are

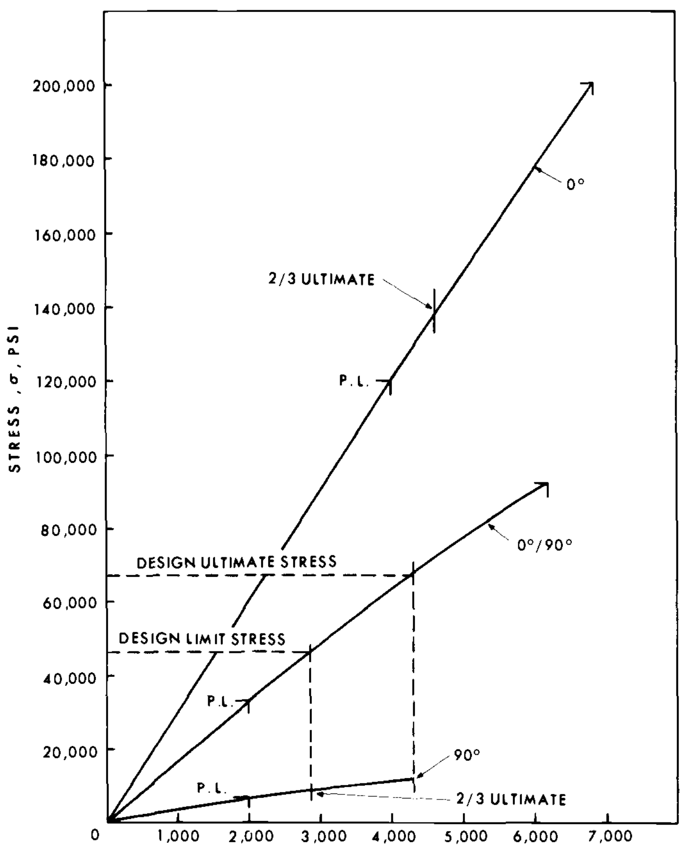


FIG. 2—*Tension design allowables typical boron-epoxy 0/90 laminate [2].*

established, and the significance of the design criteria, consider the uniaxial tension behavior of a $(0/90)_C$ laminate in a Narmco 5505 material system (50 percent volume content). Typical 0, 90, and 0/90-deg tension stress-strain curves for the Narmco 5505 system are shown in Fig. 2. For this example, assume that the ultimate condition is 1.50 times the limit condition.

The second statement of the criteria requires knowledge of the limit allowable stress for the laminate. For this example, the tension limit and ultimate value for the 0 and 90-deg directions of a basic lamina are required. The ultimates for both of these directions are obtained from test data and, because of the relationships between limit and ultimate assumed earlier, the limit design value for each of the directions cannot be greater than two thirds of the corresponding ultimate. For the 0-deg direction, two thirds of ultimate will exceed the proportional limit, and the design limit has been established as the proportional limit. It has been hypothesized that for this direction, the proportional limit may also be the yield point, where yield is uniquely defined as the onset of inelastic action. Exceeding this point in the major contributor of laminate strength and stiffness (the 0-deg laminae) may permit significant inelastic strains and resulting degradation in modulus which cannot be tolerated at limit loads. For the 90-deg direction, two thirds of ultimate exceeds the proportional limit, and the design limit has been established at two thirds of the ultimate. It has been hypothesized that any resulting degradation of the modulus in the 90-deg direction of the basic lamina will be insignificant in a major load carrying direction of an oriented laminate and that no damage to the laminate will occur. This postulation is being evaluated with fatigue tests.

The limit and ultimate allowable stresses which result for the 0/90-deg example problems are shown in Fig. 2. Note that for this particular problem the behavior of the 90-deg direction completely controls the points which are available for design, and the design ultimate allowable stress is approximately 27 percent less than the true material ultimate. If higher design allowable stresses had been selected while continuing to hold the 1.50 factor between limit and ultimate, then before the $(0\text{-}90)_C$ laminate could reach ultimate allowable stress, the 90-deg direction would have ruptured. For this problem, rupture of the 90-deg direction does not actually cause immediate catastrophic failure of the laminate, but additional capability of the $(0/90)_C$ laminate is not available in design because of the violation of the first statement in the design criteria.

The shear stress-strain curve has been shown to be continuously nonlinear, and the limit design point has been selected such that the secant modulus to the limit design point is not less than 70 percent of the initial modulus. The value of this permitted maximum degradation was arbitrarily set in the absence of fatigue or damage information. The ratio of ultimate to limit in this case is many times greater than 1.50. The stress-strain curve for compression loading in the 0-deg direction has been shown to be linear to failure, and the value of the design point has been established at two thirds of the ultimate. The stress-strain curve for compression loading in the 90-deg direction has been shown to be linear to a proportional limit, but becomes nonlinear thereafter. The design value for compression at 90-deg has been selected so that it exceeds the proportional limit by the same ratio as the limit design value in tension exceeds the tension proportional limit at 90-deg. Selected in this manner, the ration of ultimate to limit is greater than 1.50.

Because of the nonlinearities that exist in these curves, linear mathematical models will incur inaccuracies when operating near ultimate of the material. For this reason, a "Limit Design Philosophy" appears to be most reasonable and will be utilized for design with advanced composite materials. The Limit Design Philosophy requires that limit design loads be utilized and design values be selected in the area of the proportional limits.

Criterion C [10]—To restrict the applications of advanced composites by the requirement that their response exhibit no nonlinearities may be unduly conservative. Many useful ply designs are markedly nonlinear at stress levels far below their ultimate design strengths. For example, the use of 45-deg fiber orientations in spar webs, which has become traditional in glass composite design, would have to be ruled out because of its distinctly nonlinear tension and compression characteristics even at modest bending stress levels. Material nonlinearities in themselves need not be avoided indiscriminately. They can often be adequately accounted for in design so that there are no substantial added risks inherent in their use. For example, in elastic buckling of plates or shells, the use of a reduced effective modulus to account for material nonlinearities is common practice.

Small nonrecoverable strains are usually acceptable up to a point. The main problem associated with plastic or nonrecoverable strain in cyclic load situations comes from the fact that such response can sometimes be related to short fatigue life. However, this is not a general rule. For example, a number of tensile fatigue tests ($R = 0.5$) have been performed on 45-deg boron-epoxy. This material is probably the most nonlinear laminate that can be made. These specimens have all indicated that, despite the material nonlinearities, the fatigue life was adequate for the application intended (although it is admitted that this application required a rather mild spectrum).

In general, the plastic flow of any structure (micromechanics models included) which is subject to cyclic loading either shakes down after a finite number of cycles into a state wherein subsequent loading cycles lead to essentially linear response, or else continues to experience a progressive increase in the nonrecoverable strain with each subsequent load cycle. They may also fall into a third category wherein steady-state plastic flow reversal occurs in some part of the structure during each load cycle. The cases of progressive or reversed plastic flow usually are equivalent to a short fatigue life. The shakedown case usually represents a safe cyclic load design. There is reason to believe that all three forms of cyclic response are possible with certain laminate designs and load histories. Presently, the only available means for establishing the validity of any composite design under repeated loadings is to perform the necessary number of spectrum fatigue tests to provide the required confidence. However, in the future, nonlinear micro- and macromechanical analyses, which are already beginning to appear, do hold some promise of providing both insight and the analytical means for verifying the safety of a ply design short of an extensive fatigue test program.

In summary, the C criterion postulates that, except for those designs which are fatigue-critical, the ultimate allowable stress governs the design of boron-epoxy composites.

Criterion D [11]—The basic structural requirements of this criterion for components designed with advanced composites are essentially identical to the requirements defined previously, except that 1.15 times limit load is established as the level for which there should be only negligible permanent deformation. There is also one additional requirement. Fatigue tests indicate that the degradation of the composite, in the form of surface delamination or crazing, can start at about one third of the life of specimens cycled to a maximum stress, well above the first loading run proportional limit stress. This delamination causes a reduction in static strength and suggests an additional requirement; no significant degradation of the composite or reduction of static strength should be allowed to occur within the life of the aircraft. This effect will be studied in more detail as more test results become available.

In material allowables philosophy, Criterion D, is in agreement with Criterion C, which postulates the design control exercised by ultimate allowable stress. This criterion expresses this point in the following manner: multidirectional boron-epoxy laminates fail at about 0.7 percent strain, with negligible permanent set prior to failure. Therefore, if the structure is designed so that the ultimate strength of the material is not exceeded at design ultimate load, the criterion that there should be negligible permanent deformation at 1.15 times limit load will be automatically satisfied.

Re-entry Vehicle Design Requirements

Re-entry vehicle (R/V) system design generally involves a wide variety of technical activities which include trajectory ballastics analysis, aerodynamic and thermal analyses, structural design, material component and system testing, and other activities. The transition from standard R/V structural materials to those involving advanced composites will not affect all aspects of the design process. For example, determination of trajectory characteristics, aerodynamic loading and heating, etc., will be independent of the structural materials being used. The present discussion will emphasize those phases of design most affected by the introduction of advanced composites.

The most critical types of environment affecting the structural design of a re-entry vehicle are the re-entry and hardening loads. The first of these (re-entry loads) consists of external pressure, bending moments, axial loads, and shear loads, combined with axisymmetric temperature gradients. Although effectively static, these loadings and temperature gradients change with time so that more than one critical loading condition must be evaluated.

The hardening loads consist of combined dynamic response loads resulting from hardening requirements. These are the result of overall vehicle response due to blast or impulsive loading. Since the loads change along the length of the vehicle and with time, many combinations of internal shell loads at a given station must be evaluated to assess the capability of the structure. For blast loads, failure will tend to occur through quasi-static buckling, whereas, for impulsive loading, the failure takes the form of dynamic stress failure. The major requirements of advanced composite structures are governed by the re-entry, blast, and impulsive loading environments.

Design Cycle

Major changes in the design cycle have been prompted by the composite materials concept itself. Previously, we have had composite structures in the form of sandwich constructions. We are now, however, talking of a true composite material, a material in which fibers of one material are embedded in a matrix of a second material. The interactions of these two materials in meeting specific loading requirements are the subject of a major design consideration. The requirement for the structures engineer to design the material with which he will work is the most drastic departure from the normal design routine. Some may look upon the task as simply one of selecting constituents as before. This is only a minor part of the total problem. To achieve a proper design, one must consider the arrangement of these constituents to make up the final material. This arrangement is affected significantly by the design loading condition and must be considered for each structural element. One of the major advantages of filamentary composites is the ability to orient the fibers in the directions in which they can most efficiently react the loads without accepting the penalty of unnecessary material. Very seldom in engineering design will the engineer find what might be called an isotropic loading condition. An isotropic orientation of these new fibrous materials may show some advantages over other isotropic materials, but an even more significant payoff is effected by an oriented design.

Because of the many material configurations, the industry has found itself confronted with an extremely large materials testing task. An approach providing thorough test data for each laminate configuration which may be used in a structural application presents an endless task of coupon testing. Therefore, new approaches were necessary to reduce this task to a reasonable size and cost so as to make this new and valuable materials concept practical for use in aircraft structures. To achieve this purpose emphasis was placed on the development of micro- and macromechanics analysis methods to produce basic laminate allowable properties and combine these individual laminates into a reliable laminate structure. The micromechanics analysis techniques and their validity are discussed in further detail in Ref *12*. A comparison of analytical predictions with test results are presented in Table 1.

A major changeover in analysis techniques has also been required. A greater emphasis on orthotropic and anisotropic methods development has been seen in

TABLE 1–*Comparison of experimental and theoretical strength and stiffness results from boron 104 cloth 828-1031 composite [12].*

Fiber Orient (0 deg is load axis)	Composite Tensile Strength, ksi	Composite Compression Strength, ksi	Composite Tensile Modulus, ksi	Principal Poisson's Ratio	Composite Shear Modulus
0	102.8	162.8	29 000	0.15	1 300
	(100.9)	(151.3)	(30 250)	(0.24)	(892)
±15 deg	84.7	117.0	24 000	0.61	...
	(89.7)	(120.6)	(24 120)	(0.76)	(2 420)
±30 deg	34.3	31.3	12 000	0.96	...
	(33.2)	(30.7)	(10 500)	(1.28)	(6 030)
±45 deg	17.7	23.4	3 400	0.83	7 500
	(13.8)	(16.2)	(3 230)	(0.86)	(7 840)
±60 deg	11.6	30.6	3 500	0.30	...
	(11.8)	(21.5)	(2 640	(0.34)	(6 030)
±75 deg	...	27.6	...	...	...
	(12.5)	(27.5)	(3 050)	(0.09)	(2 420)
90 deg	8.6	27.3	3 700	0.005	1 300
	(13.2)	(26.3)	(3 240)	(0.02)	(892)
0, ±10, ±20,	59.9	97.6	...	...	...
±25, ± 35 deg	(62.0)	(82.4)	(18 320)	(1.26)	(3 830)

NOTE–Theoretical values are in parentheses.

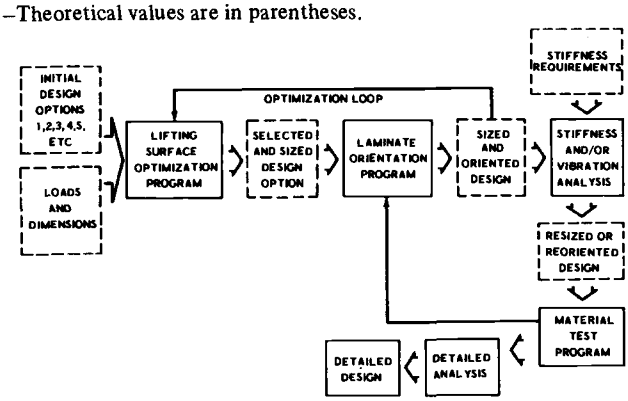


FIG. 3–*Current design cycle [12].*

the past several years. It is the development of these methods which are required to finally provide full understanding of filamentary composite materials. With the added complexities of these analyses, the greater availability of computer systems has greatly aided the development of design techniques for filamentary composites. The development of these techniques began where the development of metal techniques left off and has matched or improved upon the homogeneous design and analysis methods.

In the past several years, drastic changes have taken place in the design cycle for filamentary composites. Figure 3 presents a design cycle used in the past.

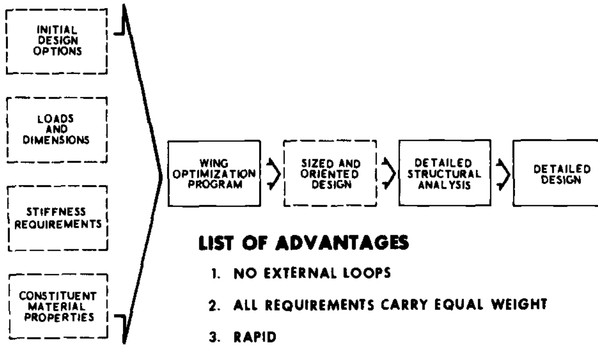


FIG. 4–*Optimum design cycle [12].*

Figure 4 shows a newer, single pass design cycle developed on Air Force Contract AF33(615)-5150. It is this design cycle which makes the complex design problem of filamentary composites well within the realm of competing with the metallic structural design approaches. With this design cycle an engineer is capable of producing a filamentary composite design concept for a wing structure with greater completeness in equal or less time than that required for metal structure.

This design cycle is initiated with the design of the material itself (Ref *13* was an initial design concept). The function of this computerized design method is to derive the laminate orientation which will best support the load requirements at any particular point in the structure. The method is based upon first determining the polar diagram of the applied normal loads. As many as twenty different flight or loading conditions can be reviewed by the computer program. The loading conditions may also include such life factors as creep, fatigue, and temperature effects. From these many different load conditions and life requirements of the specific loading conditions, a polar diagram envelope of the critical load is plotted. Having generated such a composite critical load diagram, the different laminates can be built up, one by one, in such a manner as to best resist the total applied load at all angles of the polar diagram. The computer program will use as many as fifteen different laminating materials to generate the optimum material and laminate orientation. This computer program has been used in several design applications, the most recent (for which verifying test data has been obtained) being the Aeronautical Structures Laboratory box-beam program. The unidirectional fiber laminate orientation of the skin elements was first derived for the initial honeycomb sandwich box-beam. A series of tests were performed on this ply material to establish tension, compression, and shear allowables as well as tension, compression and shear moduli. Figure 5 presents a predicted polar strength diagram for the specific orientation with test points located along the curve. The agreement of these test data with the theoretical curve provided the necessary confidence in this computer program method of laminate design.

Modifications of this technique have been made in the industry to provide capabilities for filamentary composite material design. The newer techniques

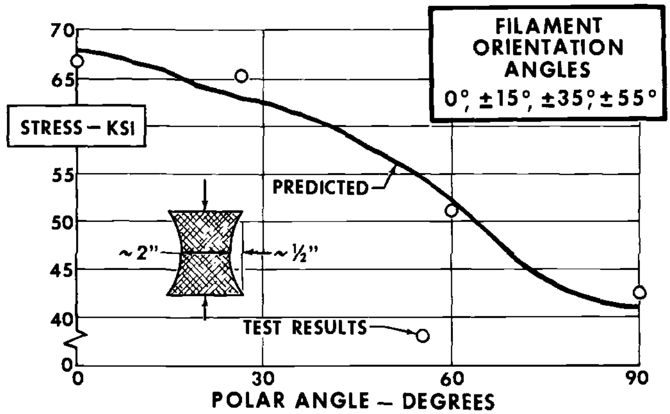


FIG. 5–*Unidirectional scotchply S-glass compression tests.*

provide the most basic material design capability accounting for compatibility, basic element failure criteria, and micromechanics of fiber spacing and packing as they affect the laminate properties in conjunction with a total structural design technique.

As a next step in the overall structural design the engineer must consider the elements of the structure itself. A series of computerized element design programs, which can be used in studying design concepts and the structural efficiency of the various fabrication concepts, have been developed in the industry. The capabilities provided by these element design techniques have also been included into the total structural design techniques recently developed. One such program designs with honeycomb sandwich type structural elements. The computer program is capable of selecting facing thicknesses, core depths, core density, and cell size as determined to be the optimum configuration for the loads involved. Other element programs are available which include fluted or truss core sandwich, honeycomb sandwich with concentrated cap areas, and skin stringer type constructions.

In addition to the material and element design considerations, the engineer must also consider the design of the overall configuration. This overall configuration primarily involves the optimum location of spar and rib elements in conjunction with the skin elements. The configuration design process involves investigating element panel widths and lengths and includes the design of the spar and rib structures for weight purposes. In the past this configuration design has been accomplished for a single lifting surface cross section. With the development of the new structural design technique the redundancies of spar and rib structures are being accounted for in the total configuration design to provide a more accurate internal structural loads distribution.

All of the design problems as discussed in this section are accomplished in the new computerized design technique in a single pass operation (Fig. 4). This is made possible only through the use of large computer systems and new concepts of structural optimization. The optimization concept must still be one minimizing a weight equation. The technique of accomplishing this was first investigated by Luscan Schmidt. Through such a technique the optimization of the material, structural elements, and the total structural configuration for strength, stiffness, manufacturing tolerances, and any other requirements necessary may be performed through a single pass operation. With such a design technique, the aircraft design engineer can now handle the extremely complex problems of filamentary composite design with the same ease and accuracy as the common metal design techniques used in the aircraft industry.

This technique establishes the general design configuration such as material orientations, material thicknesses, element sizes, as well as the rib and spar spacings. The use of such a design cycle must be followed with the more conventional final design steps of structural analysis and detailed design drawings. Even though the computer program has located the structural elements and sized them, it establishes no basis for their joining or integration. This phase of the final design is entirely up to the structures design engineer and his imagination. The designer is required to develop the overall design

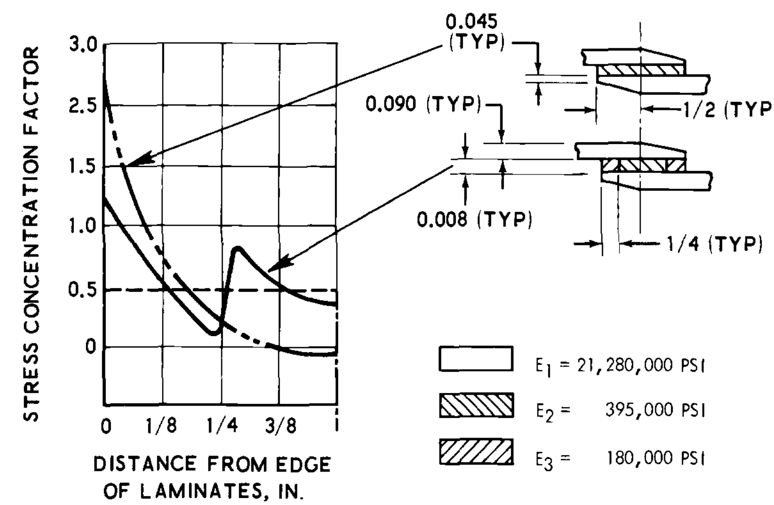


FIG. 6–*Bonded joint shear distribution* [17].

and fabrication concept. For such a task the designer must be familiar with the tooling and fabrication techniques used for filamentary composites and solicit the aid of the tooling and manufacturing engineers. Concepts, such as integration of spars with skins, or ribs with skins, are of great value in the long run in providing a low cost tooling and fabrication concept.

The structures analyst must also analyze the details of the structure, even though the sizing is already established on the basis of stress equations and is satisfactory to meet the requirements of the loading conditions present. Local problems such as joints and cutouts must be analyzed in greater detail for local reinforcement and stress distributions. This task can be accomplished through the use of several specific analysis programs. One is a mechanical joint analysis program, capable of analyzing inelastic mechanical joints to failure. A second joint analysis method used is a discreet element method developed for elastic micromechanics techniques. This program has been modified to include inelastic effects in the substructural elements, so that inelastic analyses of adhesive bonded joints can be made (Fig. 6). A third computer program is capable of providing a detailed stress analysis of a laminated plate structure with stress concentrations, reinforcements, cutouts, etc. The technique of laminated plate analysis provides a stress distribution in each laminate throughout the plate (Fig. 7). This family of computerized techniques provide an automated design capability for filamentary composite structures.

To achieve the greatest potential from this new design concept, the engineering organization itself may require changes. The successful use of such a concept requires major changes in the relationship between existing organizational groups. Of greatest importance is the relationship between the structures and materials technologies. Where once the structures engineer could accept

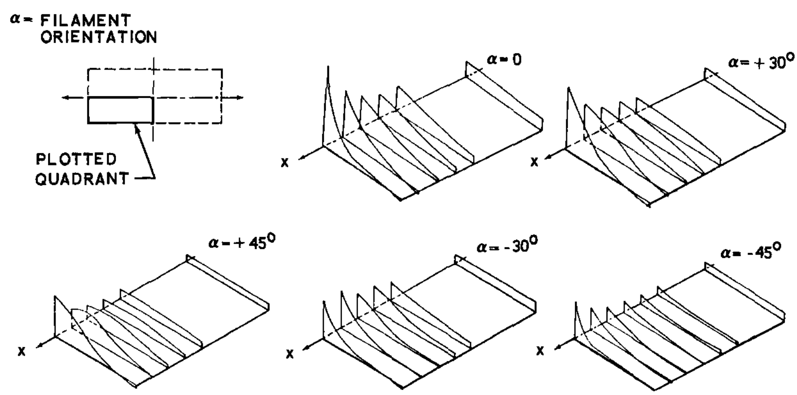


FIG. 7—*Laminated plate analysis.*

standardized materials and material properties from the materials engineer, he is now involved in designing the material. The structures engineer must maintain a closer liaison with the materials engineer to provide all the information needed for designing the materials to meet specific structural applications. The material design must take into account not only the basic elements of the fiber and matrix, but must also consider their interfacial bonds and fabrication processing to achieve a cost effective performance. The achievement of such things as permissible void content and correct resin content depends largely upon the fabrication processing techniques and the knowledge of the materials engineer. Many companies have organized groups which include both structures and materials engineers so as to ensure the proper integration of these technologies.

The introduction of filamentary composite materials earlier into the design phase must also be a major consideration. Their use can make designs that were impractical for metal construction practical. The increased efficiencies of strength to weight ratio and stiffness to weight ratio of the advanced filament systems may provide the capability of using different wing plan form areas. If the use of such plan forms substantially improves the performance of the aircraft, and the weight requirements for the vehicle can be met then an improved cost effectiveness for the vehicle may be attained. A recent design study on Air Force Contract AF33(615)-5150 has demonstrated this point [*12*]. Unique unidirectional properties of boron were found to be specifically applicable to the CX6 wing design concept being studied. The highly efficient use of unidirectional boron in spar caps was found to add significantly to the torsional stiffness of the wing when using two individual structural boxes (Fig. 8).

Even if filamentary composites are not considered early in the design phase, the structures engineer must enter into this early phase with an estimate of material configurations for specific design applications. Where the advanced design engineer previously could use standard metallic systems and allowables,

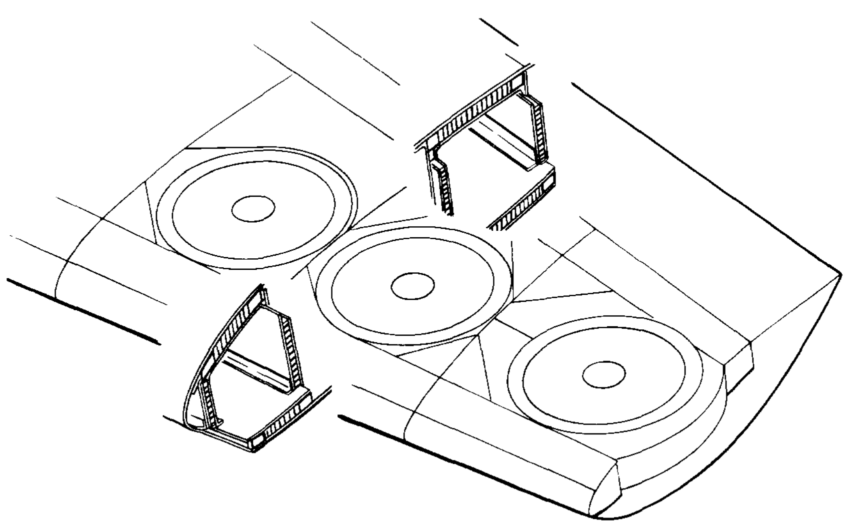


FIG. 8–*CX-6 boron composite wing [12].*

he must now consult a structures engineer who is capable of estimating material configurations and allowables for specific structural applications such as wing skins, fuselage skins, spar and rib constructions. Figure 9 presents a skin thickness for a typical wing design with an optimum and nonoptimum fiber orientation. The 17 percent weight savings shown could significantly effect trade off studies.

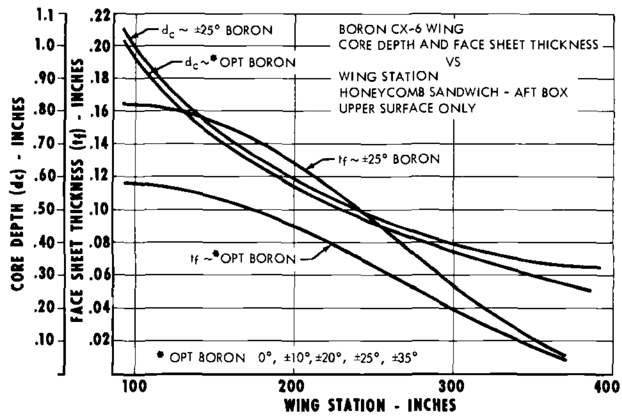


FIG. 9–*Skin and core thickness plot [12].*

Applications

A brief application study performed to measure the impact of boron composites in aircraft structural systems indicated that the most promising use lies in the lifting surface skin covering applications. Therefore, the majority of development programs have dealt with lifting surface structures.

T-39 Wing Box

This program was performed to demonstrate the capabilities of reinforced composite materials in airframe structures. The technical approach was to design, fabricate, and test a simulated T-39 wing center section box. Two boxes were built and tested, one statically at North American Rockwell, Los Angeles Division, and the other in fatigue at Wright-Patterson Air Force Base.

A study of the relative weights of the basic box cross section (excluding splice plate area) indicated an estimated boron box weight of 9.33 lb compared with 14.77 lb for the present T-39 type, integrally stiffened aluminum skin design. This results in a potential weight saving of 37 percent.

The static bending condition was carried to failure which occurred while holding 60 percent of ultimate load. The failure originated in the corner of the outer face sheet of the upper (compressive) cover at the intersection of the splice plate and panel edge. The failure extended diagonally across the panel. Examination of the failure indicated a high stress concentration at the panel corner adjacent to the splice plate. The plate provided a lateral restraint to the adjacent composite material because of the large difference in Poisson's ratio.

The fatigue specimen was identical to the static test wing box specimen. To determine more accurately the strains in the panel corner, strain gages were placed as close to the corner as possible. Some gages were also added to the splice plates to afford a comparison with the adjacent composite material stress distribution pattern. The fatigue spectrum was developed from the 15 000-h service requirement and the 3.0 life multiplying factor specified for the T-39. The fatigue tests were completed with no apparent damage to the specimen.

F-111 Horizontal Stabilizer Box

A development program for a boron-epoxy horizontal stabilizer for the F-111 demonstrated the potential of advanced composite materials. The stabilizer is flutter-critical and, therefore, designed both for stiffness and strength. A 27 percent weight saving is realized from the substitution of boron-epoxy for the aluminum skin and portions of the understructure. Because the composite surface is lighter and slightly stiffer, an additional weight saving can be realized in the support structure.

Three general types of construction were considered for preliminary design: full depth sandwich, sandwich panel, and stiffened plate. In order to form a basis for comparisons, production break locations, pivot structure, leading edge, trailing edge, and tip structure were identical for each concept. The weight comparisons were based on a single station outboard of the pivot fitting area on each of the designs.

The full depth sandwich was chosen for design development. Major factors which led to the selection were: weight, complexity comparisons, and consideration of the previous development experience with the scaled version of the component.

A test component was built simulating the structural box portion of the horizontal tail. Its purpose was to determine the feasibility and problems of application of the boron-epoxy composite against a typical aircraft component. The component consisted of boron-epoxy skins, fiber glass spars and honeycomb core, and titanium root rib, pivot fitting, and tip rib.

Reduced loads wery applied in both the up and down directions for initial flight load conditions. Up to 40 percent of design ultimate was applied in several instances. The part was then loaded to failure. Failure occurred in the attachment area in the region of the fitting at a load of 89 percent of the design ultimate or 133 percent of the actual limit design stresses.

The rate of transfer of load from the laminate to the scarfed titanium was considerably higher than anticipated. Strain compatibility between the titanium and the boron-epoxy laminate was known to be a critical parameter in the design. However, practical considerations of machining costs, bonding reliability, and the complexity of the strain relationships dictated an empirical approach to the design and, in the final analysis, an arbitrary design of the scarf. Failure did not initiate from this area, although it is believed that it would have done so within the next 10 to 15 percent increase in load.

In the fatigue test, four lifetimes of the simulated F-111 horizontal tail loading spectrum were applied without any evidence of degradation or imminent failure. Rather than continue with further cycling, a residual static strength test was considered to be the most productive in providing structural information. The loading condition for test to failure was the supersonic condition, the same as for the static demonstration article. Test failure occurred at 75 percent of ultimate in a manner similar to the static test failure. The failure was confined to the tension cover.

The results of this fatigue test program are considered to be significant in verifying the fatigue resistance of boron composite materials in component applications.

F-111 Horizontal Stabilizer

This component development program involved the design, fabrication, and testing of real structures [*14*] see Figs. 10*a* and 10*b*. A boron-epoxy airflow deflector door was designed and fabricated and placed in service on the F-111, in January 1967. Two other advanced composite flight test articles were installed on the same F-111, in March 1967. An initial observation of these articles showed excellent serviceability for the three-flight test articles after more than one year in operation. The major objective in this program involved the design and fabrication of the complete full scale horizontal stabilizer. The stabilizer has been fabricated and static tested in 98 percent of design ultimate load with a failure occurring in the metal substructure. Revisions in this design have been made prior to proceeding with the flight test article. A fatigue test requirement of four life cycles has been met by a fatigue test article. The results of this program have shown a weight saving of approximately 30 percent.

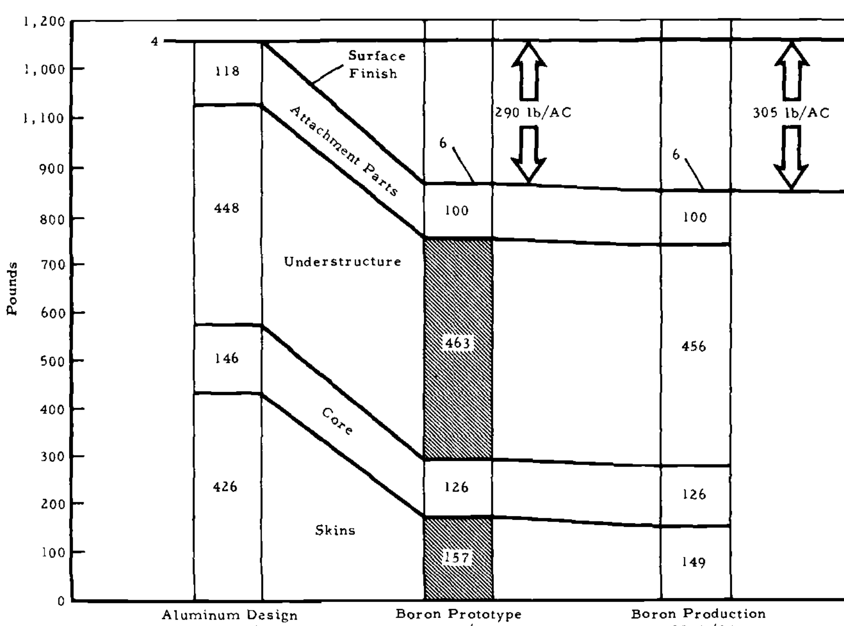


FIG. 10*a*–*Horizontal tail weight comparison [14]*.

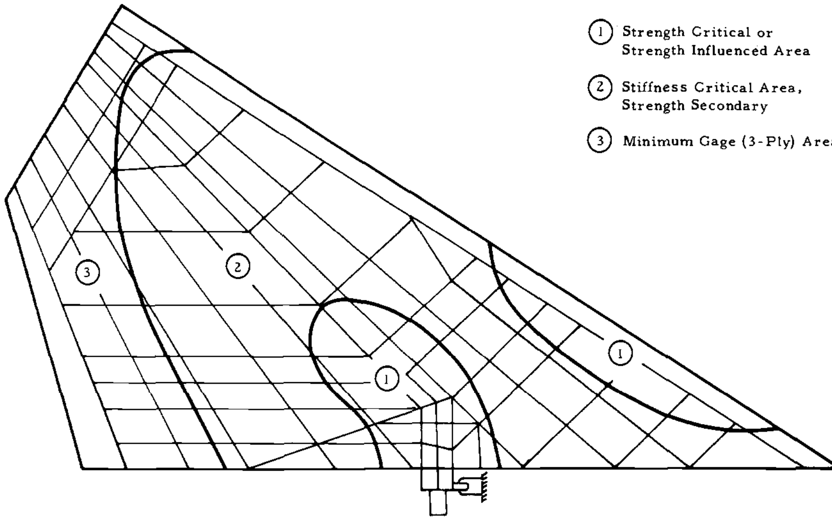


FIG. 10*b*–*Skin design influence factors [14].*

F-100 Wing Skin Program

This program involves replacement of the F-100 wing outer panel metallic skins (see Figs. 11*a* and 11*b*) with new skins developed from advanced composite materials. The design, development, fabrication, and ground testing of the composite material wing skins will be accomplished by the Los Angeles Division of North American Rockwell Corporation. The objective of this program is to evaluate the application of advanced composite materials to high performance aircraft by development and test [*15*]. The twenty-five month program is composed of three principal task areas.

The purpose of Task I is to develop a design of the F-100 outer panel wing skins of advanced composite materials. The evaluation of the design, relative to the criteria specified, is to be accomplished by ground and flight testing as specified in subsequent tasks.

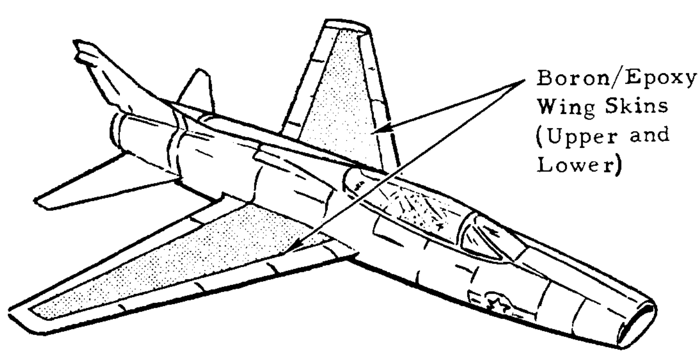


FIG. 11*a*–*Boron-epoxy wing location on F-100 aircraft.*

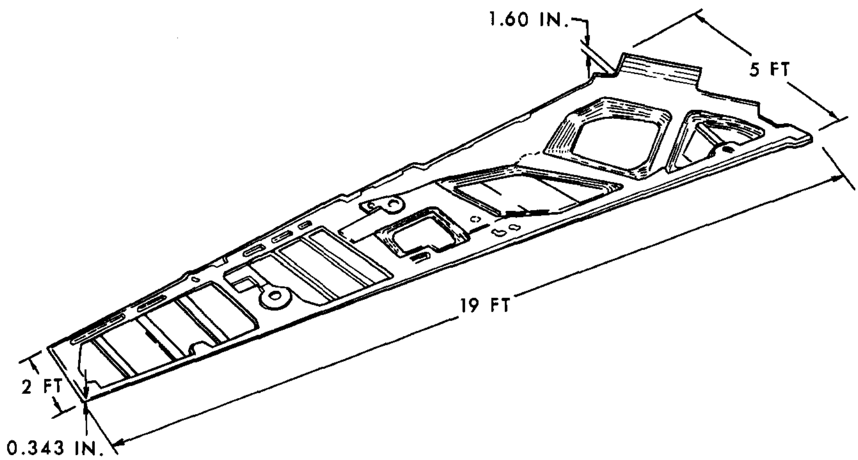


FIG. 11*b–Detail of inner surface.*

During the conduct of the design and evaluation program, the following supporting analyses will be performed to provide data for evaluation of the capabilities of the composite wing skin: strength, weight, flutter, aeroelasticity, stability and control, aerothermodynamics, and electrical systems.

The purpose of Task II is to fabricate and ground test upper and lower F-100 outer panel wing skins assembled on major static and fatigue test wings, and the flight test airplane. Ground testing will provide verification of structural intergrity in the wing skin design. Task II also includes the modification, instrumentation, ground vibration testing, and preflight preparation of the flight test airplane.

The Task III effort consists of flight testing an F-100D airplane furnished by the Government and modified as described in Task II. Instrumentation will be installed on the flight test airplane to record flight data on structural loads, and thermal and flutter characteristics.

FX Type Wing Structure

The program objective is to expand the scope of structural design technology utilizing advanced composite materials, including the capability of completely detailing the design of a highly loaded wing structural box which is representative of an advanced flight vehicle design [*16*]. The design approach will utilize boron reinforced epoxy materials to completely redesign an existing wing structure within the constraints and design criteria of the fixed contour, baseline metal wing. The approach is to obtain the maximum, practical use of composites in the basic wing structure and develop the design techniques required for the general application of composites to large, complex structural components. The effort will consist of the design, fabrication, and test verification of the composite wing box, and an evaluation of the effectiveness of indepth design with advanced composite materials.

OVIOA Wing Center Section

This component was a major test element of a contract with the Air Force to develop fabrication techniques for advanced composite joints in aircraft structures [*17*]. The box center section represents a full scale seven foot section of the wing across the fuselage including: fuselage to wing attachments, a fuel cell door in the upper skin, a lower skin splice, and access holes through the front and rear spars (Fig. 12). The lower skin splice is a scarf joint at the center line of the box. The wing to fuselage fittings are integral in the lower skin and front and rear spars. The major fuel cell door in the upper skin is fabricated of composite materials and represents the removal of one third of the upper skin area. The spar access holes are reinforced with secondarily bonded, titanium ring reinforcements. The skins are attached to the spars through an adhesive bonded tongue and groove joint. A weight saving of 40 percent has been demonstrated over the present metal construction.

T2B Filament Wound Wing

This programs' objective was to develop advanced manufacturing methods, equipment, and processes for the fabrication of efficient, reliable, low cost, filament wound fiber reinforced plastic aircraft structures [*18*]. The program involved the development of design allowables for the unidirectional S-glass and fabric composites. These allowables were developed and substantiated through simple test specimens and major structural elements prior to the fabrication of three analogue articles and the full scale demonstration article shown in Fig. 13. The demonstration article showed a 40 percent weight saving over the metal structure and met all strength and stiffness requirements of the metal structure. Static test failure occurred at 165 percent of the limit load. An analysis of the fabrication costs showed the filament winding technique to be competitive with present production costs for metal structures. During the development program certain changes in the design concept were proposed that could reduce the cost of fabrication. The filament winding or adaptation of the filament winding technique to the production of filamentary composite structures does offer a potential for low cost automated fabrication techniques.

CH-47A Rotor Blade

A development program was initiated to produce and flight test a full-scale helicopter main rotor blade. Subsequent to a feasibility program, a contract was awarded to the Boeing Company, Vertol Division, to develop and test advanced filament reinforced composite main rotor blades for the U.S. Army CH-47A helicopter. Applications analysis has shown that rotor and drive shafting

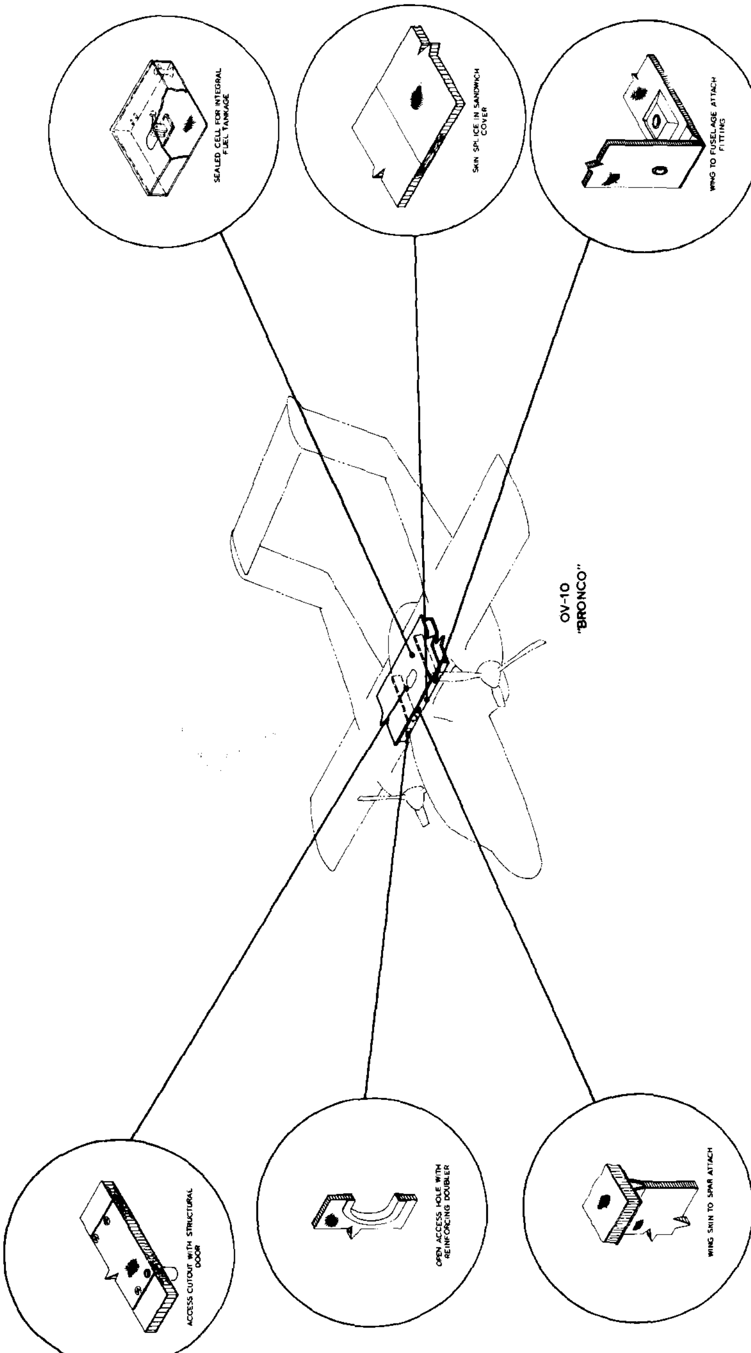


FIG. 12—*Boron-epoxy wing box demonstration article.*

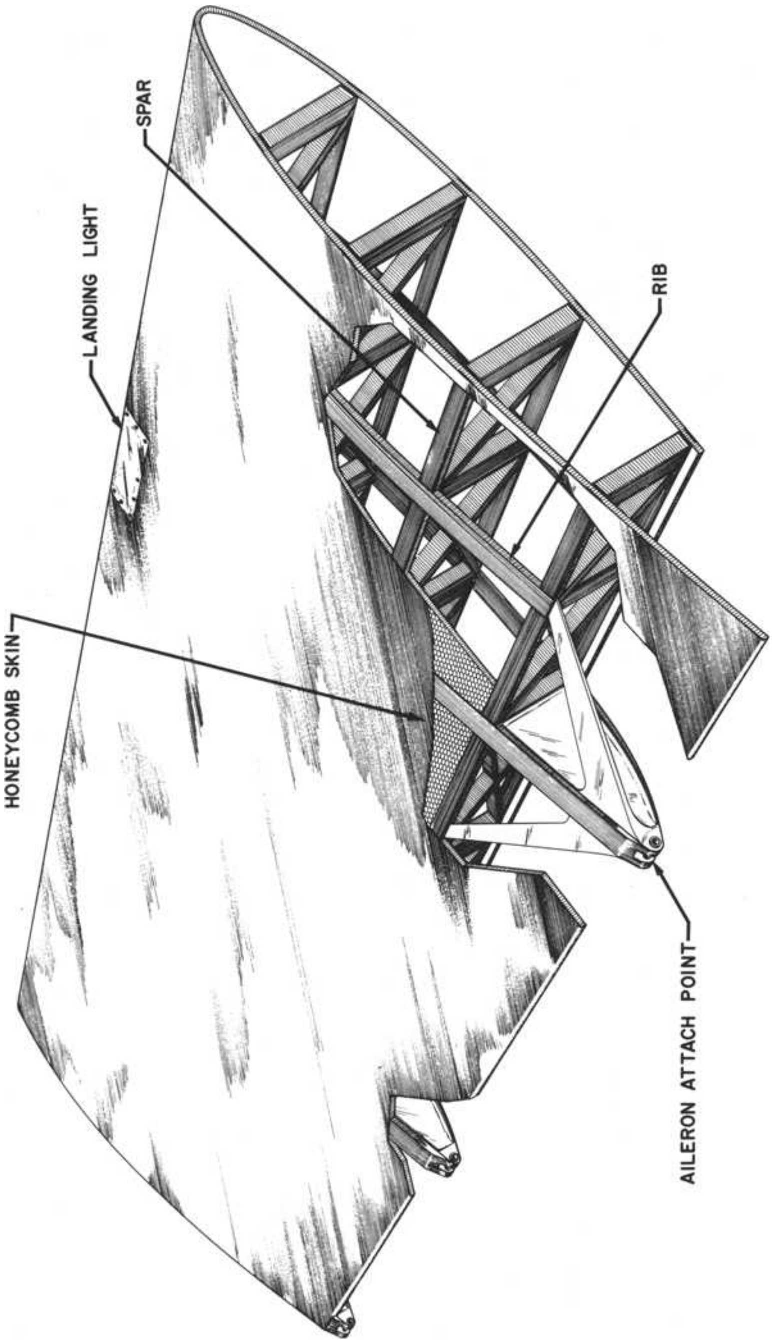


FIG. 13—*Filament-wound aircraft wing structure [18].*

components are the most promising boron applications in helicopters. They show a component weight saving of 40 percent to 50 percent by using boron. This is discussed in more detail in Chapter IV.

Missile Interstage

The Convair Division of General Dynamics (in conjunction with Harvey Aluminum) under contract to the Air Force, greatly advanced the technology of metal matrix composites in the development of a missile interstage [*19*]. The interstage was fabricated as a skin stringer structure. The stringers included boron in an aluminum matrix. The interstage section was successfully tested and demonstrated a significant weight saving for missle interstage structures.

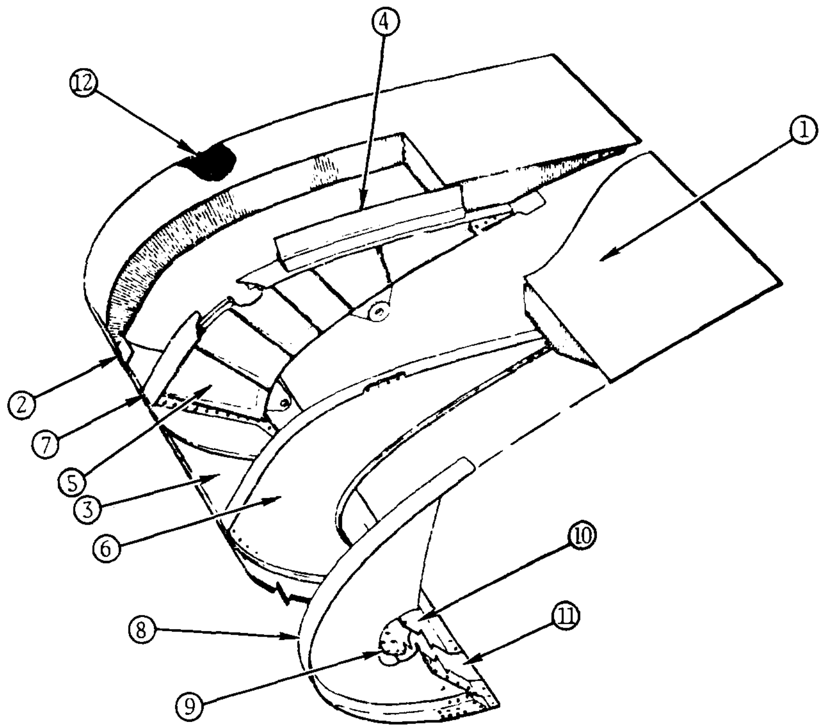
Re-entry Vehicle

A program has been used to investigate advanced composite materials applied to re-entry vehicles (R/V) and relate them to a conventional material design where possible. The advanced fibers and matrix materials which are presently available, suggest the possibility of a completely integrated heat shield and structure in which the bond layer, generally considered to be a weak link in the system, can be eliminated. The resulting advantages are:

1. The internal components and payload are volume-restricted to location in the vehicle. With a thick-walled shield-structure, the internal components must be moved aft, resulting in an aerodynamically unstable re-entry vehicle. To correct this situation, the design engineer must add forward ballast, with a resulting R/V weight increase. With the high strength and stiffness of a high modulus composite, the shell wall thickness and the number of stiffener rings can be reduced, resulting in increased internal volume and forward shifting of payload, decrease of ballast, increased stability, and lighter design.
2. The ablative heat shield thickness required for the re-entry vehicle depends on the thickness required for ablation plus the thickness required for insulation. The insulation thickness required depends upon the allowable temperature that the structural material and bond or both can withstand and still survive the re-entry aerodynamic loadings. If the structure can maintain a high strength and modulus at elevated temperature, the shield insulation thickness can be reduced, thereby, increasing the internal package volume and reducing R/V weight. A major incentive for using advanced composites is the potential for an increase of R/V structural temperature capability.
3. The weight of the basic structure is also important. Advanced composites are competitive with metallic structures because of their high specific modulus strength characteristics at room and elevated temperatures.
4. Shield structure thermal compatibility is important in the design of an ablative shield structure. If the gross thermal expansion coefficients of the shield and structure are compatible, the system can undergo a large temperature change with minimized thermal stress problems. Advanced composite structures provide greater thermal compatibility with heat shield materials of interest.

Propulsion System Applications

Advanced fiber-reinforced composites can be used to good advantage in current propulsion system components. Even greater potential is present in the possibility of optimizing the design of future power plants, taking cognizance of the peculiar capabilities of advanced fiber composites.



1. Eyebrow Honeycomb Panel
2. Eyebrow Panel Edge Members and Lighting Diverter
3. Lower Leading Edge Honeycomb Panel
4. Insert Retainer Fitting
5. Track Rib and Nose Former
6. Intermediate Rib and Nose Former
7. Track Rib Cap
8. Nbd End Rib
9. Stringer Channel
10. Lower Trailing Edge Stiffening Web
11. Lower Trailing Edge Skin
12. Plastic Casting Resin Core Fill

FIG. 14–*Subscale model C5A boron slat [20].*

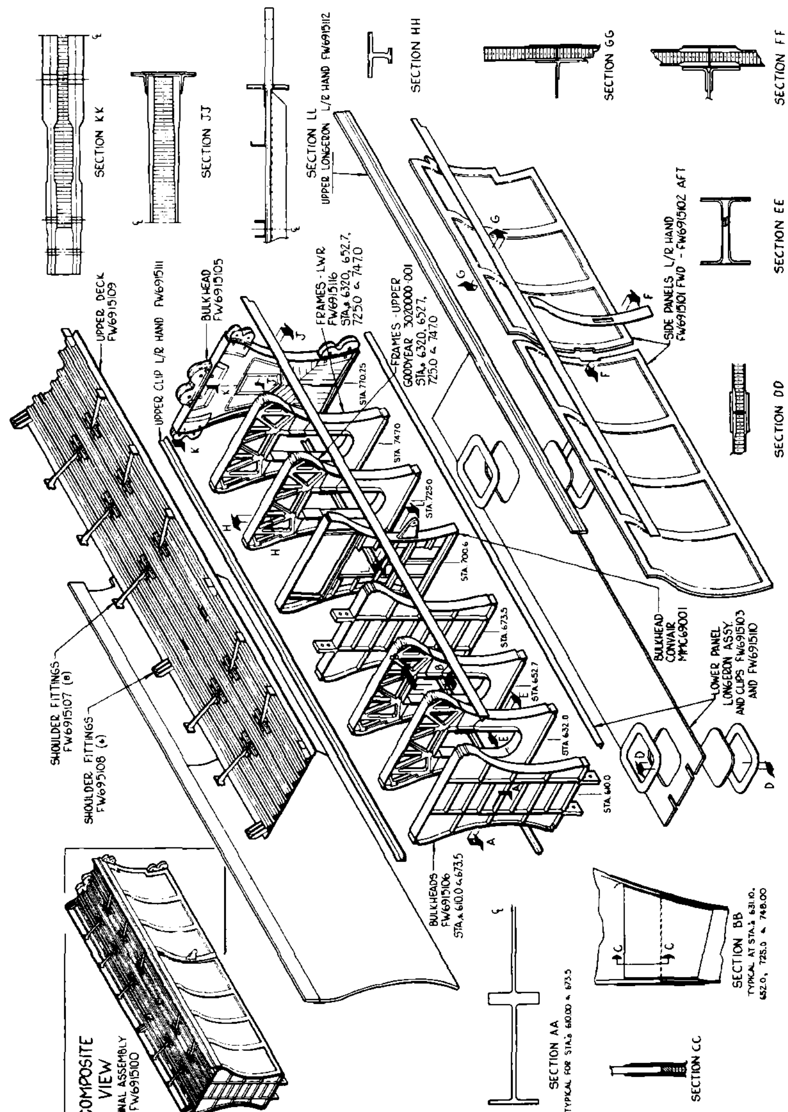


FIG. 15—*F-111 advanced composite fuselage structure [21].*

C5A Leading Edge Slat

The Lockheed Corporation of Marietta, Ga., is developing a boron leading edge slat for the C5A [*20*]. The slat is one of the largest structures presently being developed under contract with the Air Force and uses boron-epoxy skins to provide a substantial weight saving in the leading edge structures (Fig. 14).

F-111 Fuselage Shell Program

The objective of this program is to advance the technology associated with the application of advanced composite materials into the areas of plate and shell applications and into the related hard-point requirements [*21*].

The materials to be considered will include laminates of boron-epoxy, carbon-epoxy, glass-epoxy, boron-aluminum, boron-high-temperature plastic, and molded plastic matrix composites.

The component development effort will include development of solutions peculiar to advanced high performance, fighter type aircraft fuselage. This effort will culminate in the fabrication and test of a 14-ft long full-scale section of the aft fuselage of the F-111.

Figure 15 shows the center fuselage shell from Station 770 forward to Station 610. This section is fuel tankage with internally machined frames and bulkheads. Bulkhead 770 center section is loaded at the corners by horizontal tail and nacelle loads and, as noted, by vertical tail loads. The Station 700 bulkheads mount the upper nacelle frames.

Internally machined aluminum longerons are located at the four corners as shown. The skin panels are sandwich contruction except for the lower aft panel which is ring-stiffened sheet.

The composites' fuselage component is patterned after the F-111 section depicted in Fig. 15. The section from Station 770 forward to Station 673 is defined as the test section. The area forward of Station 673 is used for full airplane loads. Composite construction will be used throughout. The section forward of Station 673 will be designed to component reaction loads. The test section will be pressurized to F-111 fuel pressure requirements.

Bulkhead 770 will be designed to full F-111 loads, and the load introduction points will be realistically located. The frame design, used at Station 747, 725, 652, and 631, will be a joint design-fabrication effort with Goodyear Aerospace Corporation, Akron, Ohio. The bulkhead at Station 700 will be a boron-aluminum part designed and fabricated by the Convair Division of General Dynamics. The component test reaction bulkhead, Station 673 and 610, will utilize the Bulkhead 770 design and tooling but will be tailored to test loads.

Composite sandwich skin panels with mechanical joints to the substructure are planned for the side and lower panels. An all composite hat-stiffened sheet design is planned for the upper panel. Mechanical joining will also be utilized. Longeron designs of all composite or composite/metal combinations are contemplated for use with the all composite panels.

Conclusion

Major improvements in the performance of military aircraft may be expected from the many attributes of the material discussed. Two of the primary problems remaining are the development of effective manufacturing and quality assurance techniques. Another problem area, which can significantly change the type of construction used, is that of mechanically fastened joints. Even though mechanically fastened joints are practical in composite materials, there are penalties associated with using this type of joining method. Greater efficiencies can be obtained with adhesive bonding of structural elements.

The new graphite materials are very attractive from the viewpoint of both economics and structural efficiency. However, the compatibility problem of graphite with a matrix system offers a great deal of room for improvement. Even though the initial fiber finish problem seems to have been relatively well solved, the negative coefficient expansion of the graphite fibers sets up high thermal stresses within the matrix materials which affect the overall material performance. From an economic standpoint graphite offers the greatest potential. Boron fiber costs have been slowly decreasing and appear to be leveling off, whereas the cost of graphite fiber continues to be reduced. Graphite costs are projected to be as low as $50 per pound in the next several years. One of the economic aspects one must consider when dealing with high priced materials is the problem of scrapage. If we consider the scrap rate of the conventional metal used in aircraft we find it is as much as 75 percent to 85 percent of the purchased material. From statistics on glass reinforced plastics, which should be similar to the advanced filamentary composites, a maximum of 5 percent scrapage occurs. This factor offers an improvement in the cost of a final aircraft and makes advanced composites more competitive with materials such as titanium.

Filamentary composites are destined to see extensive use in the design of military aircraft systems during the 1970s and culminate in a family of advanced aircraft of the 1980s. We are only now making a break into the potential of filamentary composites and understanding their use in advanced applications.

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Chapter III – VTOL Aircraft

REFERENCE: Salkind, M. J., "VTOL Aircraft," *Applications of Composite Materials, ASTM STP 524*, American Society for Testing and Materials, 1973, pp. 76-107.

ABSTRACT: Fibrous composites offer significant potential for structural improvement in VTOL (vertical take off and landing) aircraft. In addition to the potential for light weight fuselage structures, composites offer the unique capability of providing dynamic tuning of the fuselage. Composites provide substantial potential for helicopter rotor blades because of improved fatigue capability, good damage tolerance, and ability to be molded in complex aerodynamic configurations.

This chapter summarizes the major design considerations in VTOL aircraft and reviews composite hardware which has been developed to date.

KEY WORDS: composite materials, fibers, aircraft, helicopters, rotary wings, fatigue (materials)

Vertical takeoff and landing (VTOL) aircraft consist of helicopters (Fig. 1), tilt-wing (Fig. 2), tilt-rotor (Fig. 3), and direct lift-thrust aircraft (Fig. 4). These aircraft offer a variety of advantages such as intercity travel between midcity terminals and important roles for military applications. As a group, they offer substantial improvements in performance through the wise use of composite materials. Because of the vertical lift requirement of VTOL aircraft, their forward flight capability is considerably restricted relative to conventional fixed wing aircraft. As a result, improvements in structural efficiency, made possible through the use of composite materials, will allow greater improvements in systems performance than for fixed wing aircraft. In addition, VTOL aircraft contain a large number of dynamic components which make them favorable candidates for the use of composite materials, since the ability to vary elastic properties allows for dynamic tuning.

Direct lift-thrust aircraft are similar in construction to conventional fixed wing aircraft, except that deflected thrust or lift engines are required for vertical movement. Several such aircraft, including the Dornier DO.31, Hawker-Siddeley Harrier, and the Dassault Mirage III-V, have reached flight status. Since the wings do not provide lift during vertical movement, the thrust required for lift is considerably greater than that required for cruise. The thrust-to-weight ratio of

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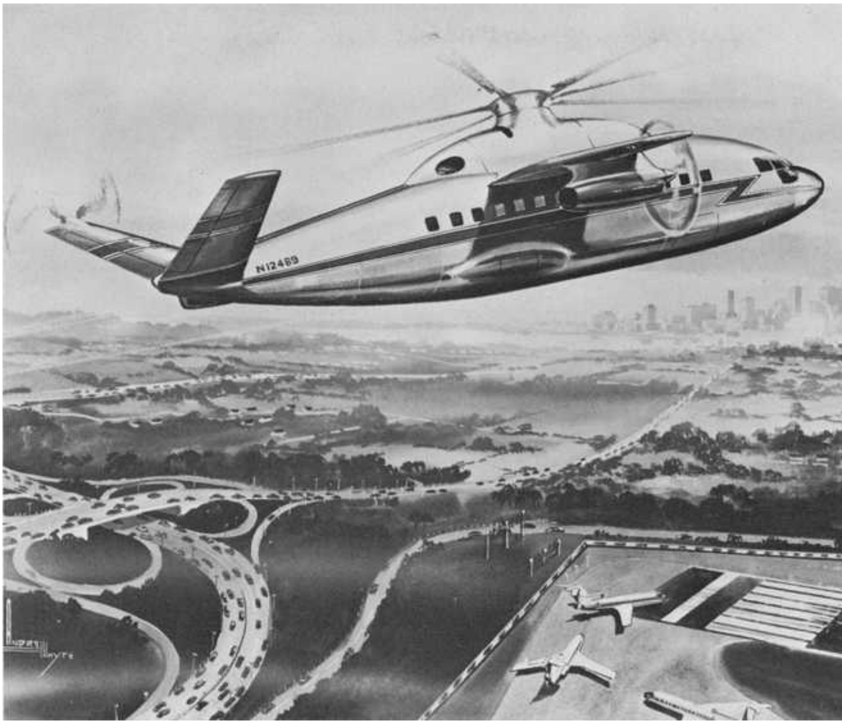


FIG. 1–*Compound helicopter, Sikorsky S65-200 (Sikorsky Aircraft).*

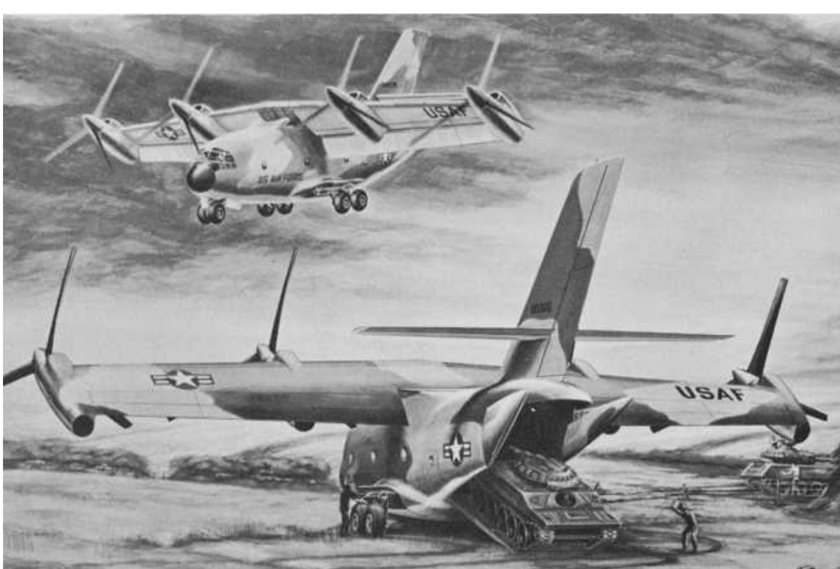


FIG. 2–*Tilt-wing aircraft (The Boeing Co., Vertol Div.).*

FIG. 3—*Tilt-stowed rotor aircraft, Sikorsky Trivertiplane (Sikorsky Aircraft).*

FIG. 4—*Direct lift aircraft, HFB 600 (Messerschmidt-Bölkow-Blohm).*

FIG. 5–*Compressor rotor of RB 162 engine with fiberglass composite blades (Rolls-Royce, Ltd.).*

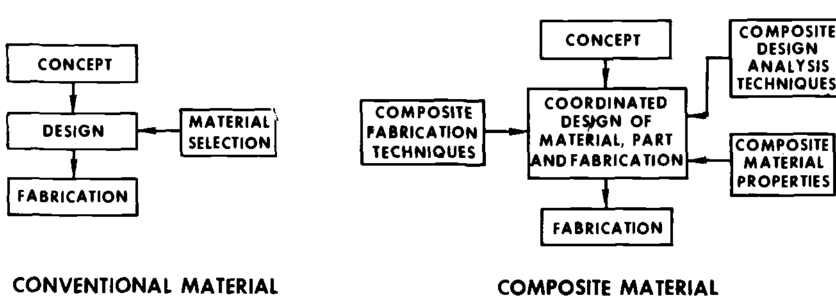


FIG. 6–*Design philosophy for conventional and composite materials.*

the lift engines must, therefore, be considerably higher than that for cruise engines, and the weight savings made possible by the use of composite materials becomes even more important for such engines. The Rolls-Royce RB162 lift engine [*1*],[2] used in the Mirage III-V, derives much of its structural efficiency through extensive use of glass fiber reinforced plastic (GFRP) including the compressor case, vanes, and blades (Fig. 5).

The helicopter, tilt-wing, and tilt-rotor aircraft contain a substantial number of dynamic components such as rotor or propeller blades, shafting, hubs, and control links, which are prime candidates for the application of composite materials. Because these aircraft have complex dynamic loadings, the ability to dynamically tune the airframe through the use of composites is also important. An overall system weight saving of 25 percent is projected for VTOL aircraft using composites. Because of the additional design flexibility of composites, new structural concepts creating new systems attributes are possible.

Design

Design Philosophy

One of the most significant advantages of composite materials is the exceptional flexibility of properties which accrues to the designer. The fact that there are not only one or two composite materials, but an infinite variation of fibers, matrix, fiber orientations, and fiber contents, allows a new degree of freedom in structural design and has a profound effect on the whole materials-design-fabrication process. As seen in Fig. 6, the design procedure for conventional materials involves a selection from a finite catalog of materials, each having well defined properties. Although the strength of metals can be varied somewhat by processing, the elastic properties are fixed and are approximately isotropic. The structural metals used in aircraft are primarily steel, titanium, and aluminum with respective Young's modulus values of 30, 16, and 10 million psi. But, what if the optimum design of a rotor blade, for example, having a specified aerodynamic geometry and aeroelastic characteristics, requires a material having a modulus of 22 million psi? The design must be compromised to incorporate one of the off-optimum metals such as steel, titanium, or aluminum. In addition, once the airfoil shape and bending stiffness are fixed, the torsional stiffness is also fixed because of the isotropy of conventional metals. The only opportunity for elastically tailoring a rotor blade of fixed airfoil shape is through changing the sectional properties by varying the internal geometry; however, this is extremely limited by the requirements for mass balance. As a result of this relative inflexibility of design with conventional metals, there have been cases of helicopter rotor blades which have developed disastrous dynamic instabilities in service. Serious fuselage dynamic problems have also occurred in helicopters through the improper selection and design of metal structures.

What will the design function with composites be? As seen in Fig. 6, the design

[2] The italic numbers in brackets refer to the list of references appended to this paper.

activity will be considerably expanded to include a concurrent material design while a closer team effort among structural designers, analysts, and materials engineers will be necessary. Such a change appears to be occurring in several aerospace companies seriously involved in the use of composite materials. Instead of selecting from a finite catalog of materials, the designer will have a wide choice of composite properties, computerized material, structural design analysis, and optimization procedures. Early examples of such routines are presently operational throughout the aerospace industry. The designer now has a new degree of freedom in that he can select the fiber orientation and spacing to optimize the elastic properties and the elastic anisotropy. This is in addition to his conventional variables of material selection and shape.

It should be noted that the expanded design activity associated with composite materials requires better definition of the loads on a component, because the fiber directions are chosen to react the principal stresses. The use of more sophisticated analytical techniques is required, and the increased complexity and repetitive nature of the design process calls for the extensive use of digital computers.

The changed design procedure associated with composite materials also has a profound effect on the manufacturing procedure, and in turn, consideration of the fabrication technique must be incorporated into the design process itself. Since the fabrication of a composite component is one of synthesis, involving building up the component layer by layer, the fabrication sequence and the design of the laminate orientations are intimately related and must be designed together. This requires the participation of manufacturing personnel in the design-materials-fabrication team and will require organizational adjustments for implementation.

An additional consequence of the advent of composites is on the manufacturing procedure itself. As mentioned previously, composite fabrication is a synthesis process, whereas metal fabrication is an analysis process in which a large piece of metal is cut down to form the finished shape. In the case of complex forgings, it is not unusual for 75 percent of the starting material to wind up as chips on the shop floor. The machines used for cutting metals must, of necessity, be massive to apply the large cutting forces necessary and still maintain close dimensional tolerance. Many different forgings must be procured and inventoried to account for the varied part shapes. Also, long lead times are common for large forgings, exceeding one year in some instances. Because composite components are built up from tapes of preimpregnated fiber, the waste is small because they are formed nearly to finished dimensions. In the aerospace industry at the present time, the waste for most composite parts averages less than 10 percent. Machines used for tape laying need not apply large forces and can, therefore, be of light construction and operate at greater speeds. The procurement and inventory costs and lead time problems are also reduced, because many shapes can be made from the same tapes.

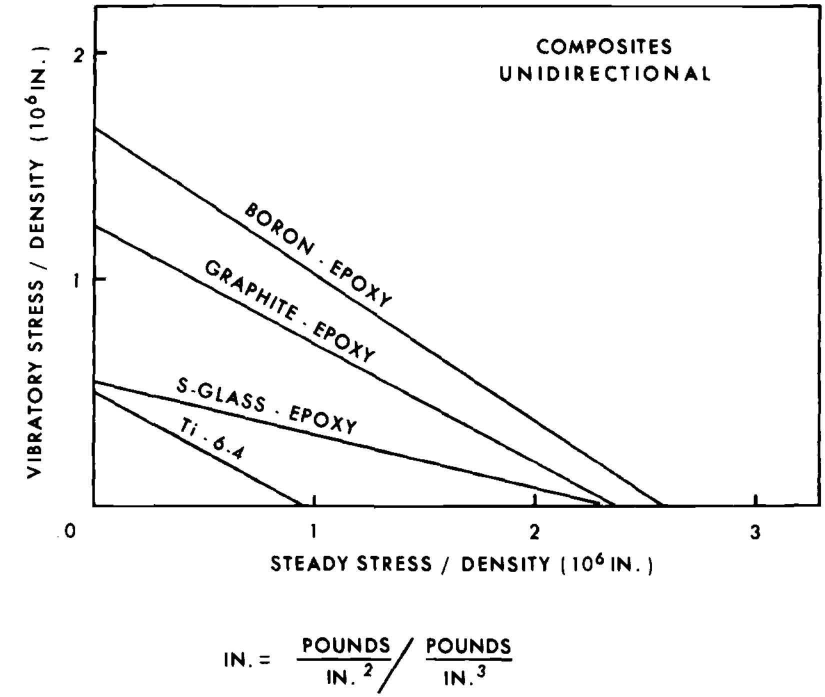


FIG. 7–*Fatigue properties of fiber composites and metals at 10^7 cycles.*

Design Properties

The major material property considerations for VTOL aircraft are fatigue strength, elastic properties, and static strength and toughness [*2, 3*]. Dynamic components such as rotor heads and blades are designed on the basis of high cycle fatigue, and (as seen in Fig. 7) fiber composites exhibit considerably better fatigue properties than metals. The data obtained from small specimen tests must be reduced for reliability (typically three times the value of the standard deviation), surface finish, and size factor, and verified by full scale testing before a design allowable is established. At the present time there is a very limited amount of full scale fatigue test data for composites, and reliable design allowables are, therefore, not readily available. Such testing has been done for fiberglass reinforced epoxy rotor blades and has been described in detail by Jarosch and Stepan [*4*].

There are two major concerns with respect to the use of composites in fatigue: mode of failure and low cycle fatigue behavior. Metal parts exhibit cracks when they begin to fail in fatigue, and the cracks generally propagate in a predictable manner to failure. Thus, the metal part may be inspected at specific intervals and removed from service prior to failure. Composites, on the other hand, do not fail in the same manner. Fatigue damage can consist of resin crazing (matrix failure), fiber

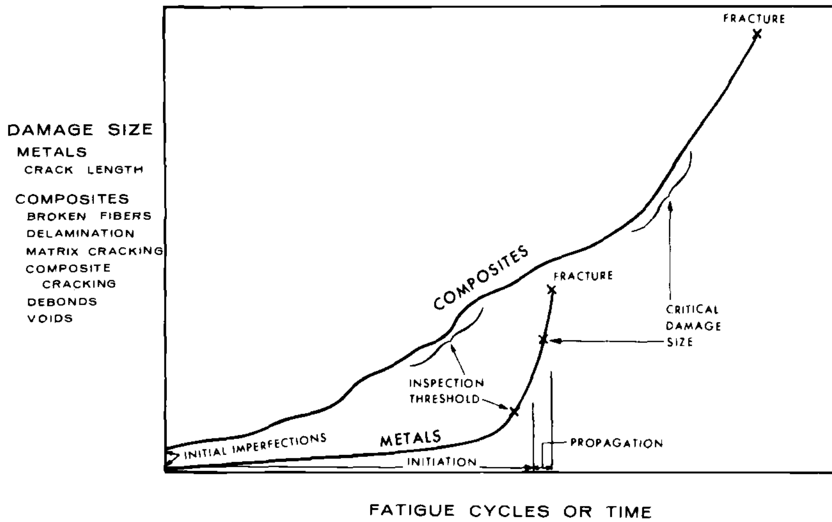


FIG. 8–*Fatigue damage behavior of composites and metals.*

debonding, delamination, fiber cracking, or composite cracking either separately or in combination. In addition, crack propagation often does not occur in a predictable manner, such as is the case for metals, but often changes direction.

The difference between fatigue behavior of a composite and that of a metal structure is depicted schematically in Fig. 8. The primary mode of damage in a metal structure is cracking. Cracks propagate in a relatively well defined manner with respect to the applied stress, and the critical crack size and rate of crack propagation can be related to specimen data through analytical fracture mechanics. In general, the crack initiation time, defined as the time to detectable cracking (inspection threshhold), occupies a large part of the fatigue life of a metal part. It should be noted that all structures have some initial damage in the form of microcracks, surface imperfections, inclusions and other stress risers, and that much of the so-called crack initiation time involves propagation of this damage to detectable size.

The major difference with composite structures is that there is no single damage mode which dominates. Matrix cracking, delamination, debonding, voids, fiber fracture, and composite cracking can all occur separately and in combination, and the predominance of one or more is highly dependent on the laminate orientations and loading conditions. In addition, the unique joints and attachments used for composite structures often introduce modes of failure different from those typified by the laminate itself.

Referring to Fig. 8, the composite damage propagates in a less regular manner and damage modes can change. Present experience with composites, although limited, indicates that the rate of damage propagation in composites changes in a more uniform manner than that of metals and does not exhibit the two distinct regions of initiation and propagation. Although, as mentioned above, the crack initiation range in metals is actually propagation, there is a significant quantitative difference in rate. This quantitative difference appears to be less apparent with composites. This observation is quite subjective and apparently dependent upon the observer's definition of initiation. Some investigators have observed matrix crazing and other indications early in their tests, but have reported short time rapid propagation because they define the latter based upon their experience with metals as crack propagation. Indeed, composite cracking may occupy only a small part of the fatigue life at the very end, but we can certainly make use of all the earlier indications which are prevalent.

It is expected that composite materials will be more damage tolerant than metals. Again, this expectation is based upon limited experience and will depend upon the laminate orientation (unidirectional composites are subject to splitting) and loading conditions, but in general, it can be argued that each fiber is a

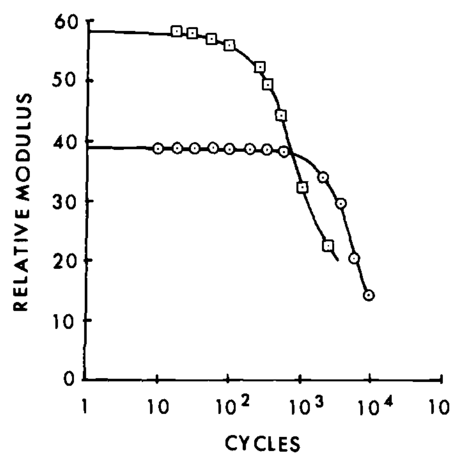


FIG. 9—*Variation of the stiffness of fiberglass-epoxy during torsional fatigue [9].*

separate load path and that a composite is, therefore, highly redundant. Our present analytical fracture mechanics tools must be supplemented for use with composites before we have a better understanding of this behavior. Several investigators have indicated that, in general, composites exhibit good fracture toughness [*5-8*] and, unlike metals, increasing fracture toughness with increasing strength [*8*]. It is reasonable to predict the critical damage size in composites to be greater than that for metals (Fig. 8), although the multiple failure modes make this value a band for composites. Similarly, the inspection threshhold is depicted as a band in Fig. 8, because there are multiple failure modes and multiple inspection methods.

The problem then is to determine the critical mode or modes of failure and develop detection schemes in order to ensure fail-safety in critical components. One such procedure involves the determination of changes in the static or

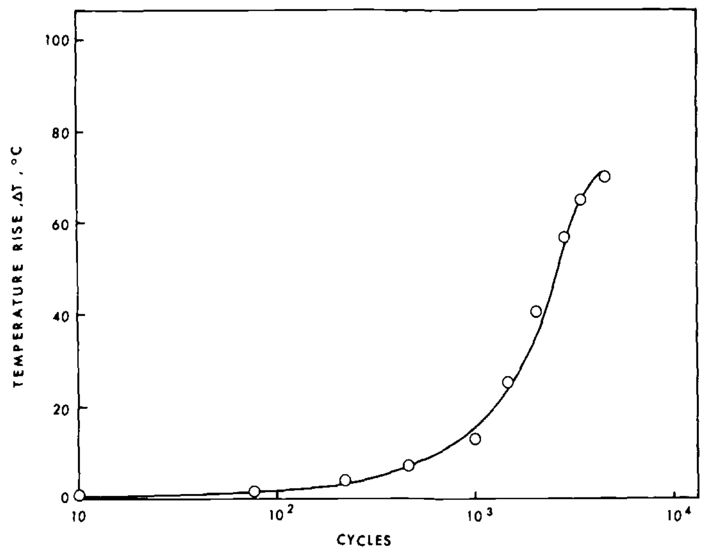


FIG. 10—*Temperature rise during flexural fatigue of glass fiber reinforced polypropylene [10].*

dynamic stiffness properties of the component. A change in the resonant frequency or damping behavior of a part is an indication of damage. As seen in Fig. 9 [*9*], the elastic properties of composites can show substantial changes early enough in the fatigue life to allow safe detection and removal from service. This characteristic may provide excellent fail safety for rotor blades in that the aeroelastic behavior may degrade noticeably long before the part has sustained damage of critical size. Rolls-Royce developed resonant frequency tests for acceptance and in-service inspection of their Hyfil (graphite-epoxy) fan blades. This method may not, however, be sufficiently sensitive to detect very localized damage in large components such as helicopter rotor blades. Other detection schemes such as temperature rise measurements (Fig. 10 [*10*]), embedded conducting wires, sonics, ultrasonics, infrared, dye penetrant, and visual inspection will probably be used separately or in conjunction with dynamic measurements.

The large advantage of composites over metals in high cycle fatigue does not hold true for low fatigue as shown in Fig. 11. Dynamic components undergo a spectrum of loading which includes a small number of high stress fatigue cycles due to stop-start, maneuvers, and the ground-air-ground cycle. For composite materials, the poorer low cycle fatigue behavior may be the controlling design factor such that the excellent high cycle fatigue properties cannot be completely utilized.

The high specific stiffness (stiffness to density ratio) in the fiber direction coupled with the ability to substantially vary the anisotropic elastic properties of composites makes them extremely attractive for the design of VTOL aircraft. As

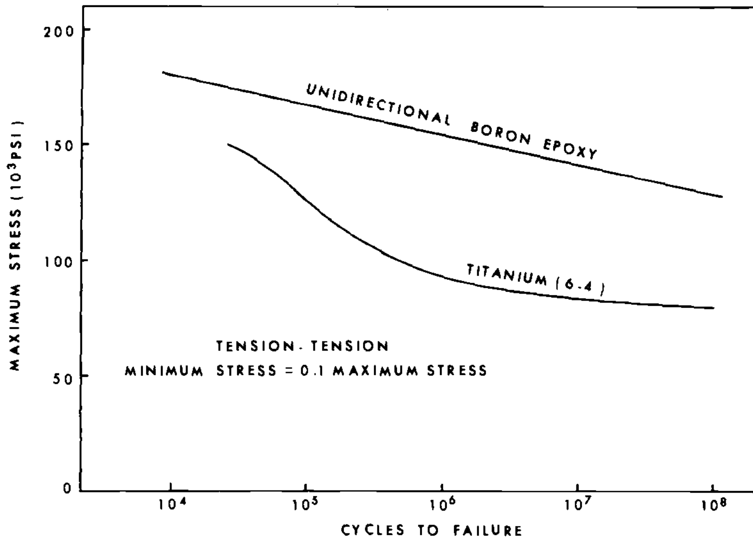


FIG. 11—*Characteristic fatigue behavior of composites and metals.*

seen in Fig. 12, the specific stiffness of unidirectional boron and graphite composites are considerably higher than all structural metals with the exception of beryllium. Beryllium has not found wide use because of its poor fracture toughness and relatively high cost. The high specific stiffness of composites is

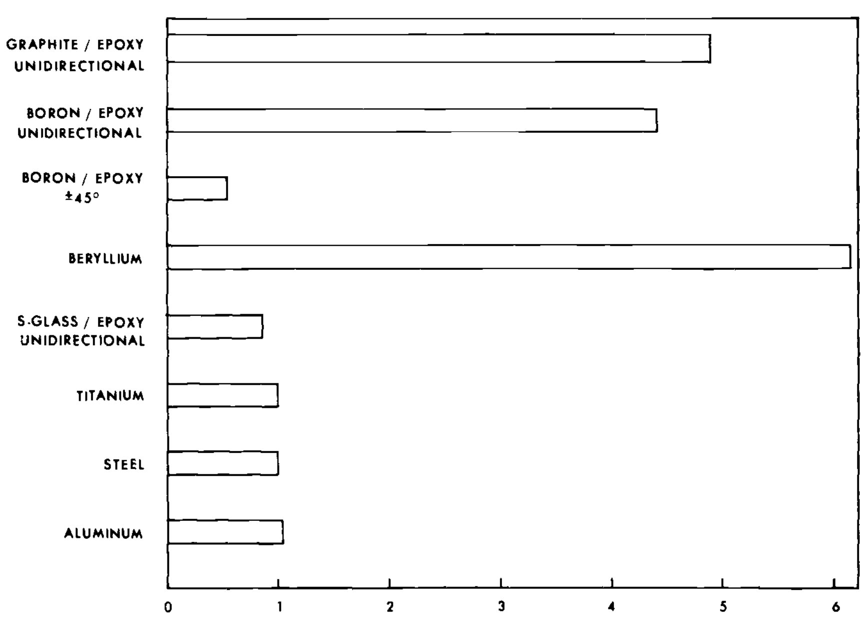


FIG. 12—*Specific tensile modulus of structural materials.*

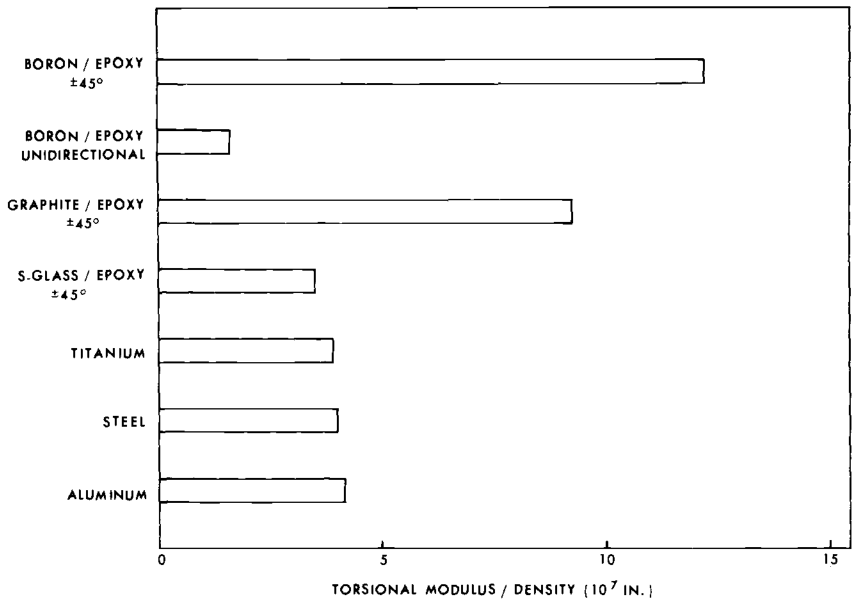


FIG. 13–*Specific torsional modulus of structural materials.*

structurally very efficient for such applications as rotor blade spars, control rods, and airframe stiffeners. High specific torsional stiffness for applications such as rotor blade skins and drive shafts is accomplished by using ±45-deg fiber orientations (Fig. 13). In addition to the high specific stiffness which allows greater structural efficiency than metals, the anisotropy of these properties allows use of new structural design concepts. The effect of varying the lamina orientation and the fiber orientation within a lamina on the modulus is seen in

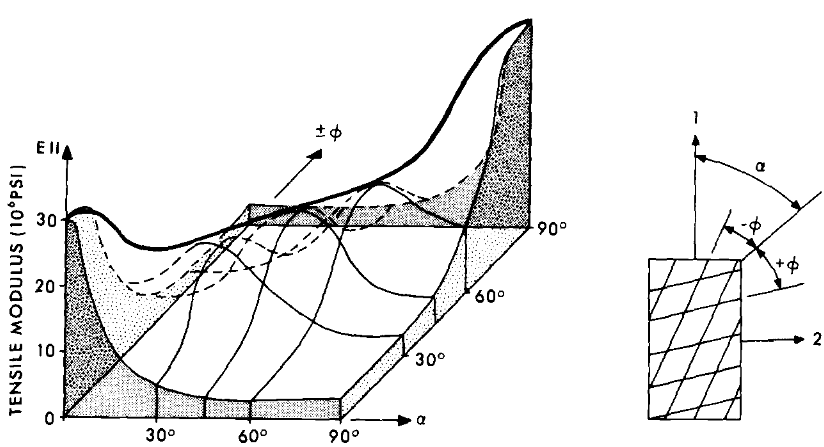


FIG. 14–*Variation of boron-epoxy tensile modulus with fiber orientation.*

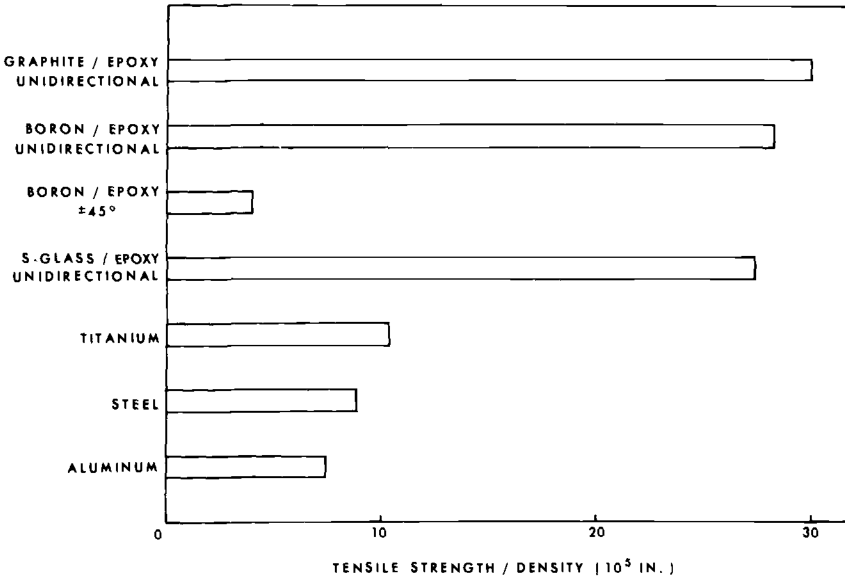


FIG. 15–*Specific tensile strength of structural materials.*

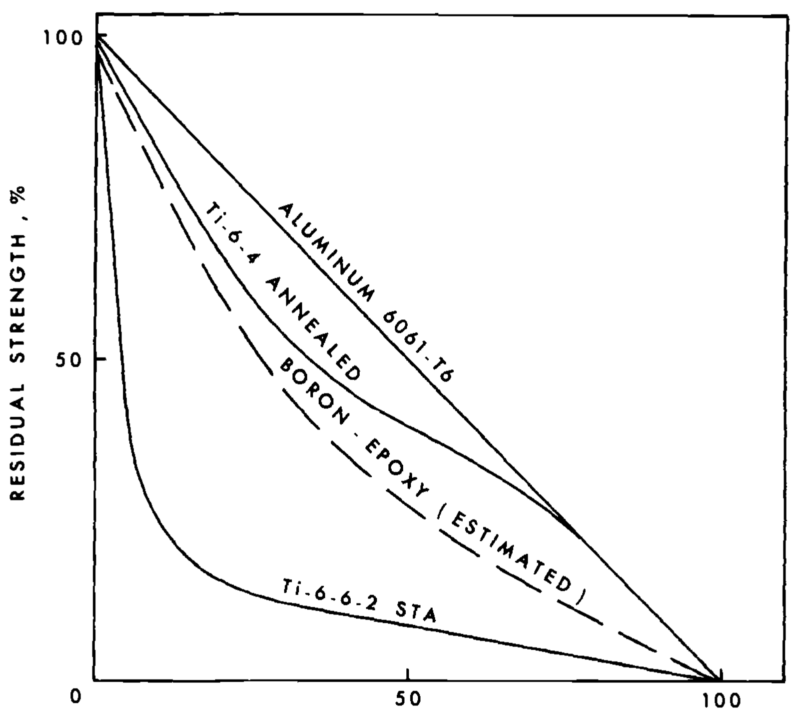


FIG. 16–*Residual strength of sheet materials [3].*

Fig. 14. Such a wide variation in properties allows the designer a considerable latitude and the capability to change the fiber, matrix, and fiber content to further tailor the elastic properties of his structure.

The specific strength properties of composites also offer advantages over metals as seen in Fig. 15. This property allows structural improvements in the airframe and other statically-loaded structures. Coupled with strength is the consideration of fracture toughness for these applications. Although toughness has been shown to be highly directional [*5,6*] composites generally exhibit relatively good fracture toughness. Some preliminary data for boron-epoxy is compared with structural metals in Fig. 16. It should be noted that unidirectional resin matrix composites readily split parallel to the fibers and some cross plies are almost always needed to allow handling of the structure.

Applications

Rotor Blade

Perhaps the most important use of composite materials for VTOL aircraft is for the main rotor blade or rotary wing. The ability to mold almost any aerodynamic shape and tailor the aeroelastic properties makes the payoff in performance significant through the use of composites. The application of composites for rotor blades dates back to the early days of helicopters. Molded wood and fiberglass blades were used for many early VTOL's including the Sikorsky VS-300, Piasecki H-21, Bell H-13, Hiller H-23, and are presently in production for the Kaman HH-43 Huskie. A more recent and somewhat more sophisticated use of fiberglass-epoxy is embodied in the main blade of the MBB BO-105 [*4*] and its derivative used for the SNIAS SA-341. By far, the most significant use of advanced composite materials for a rotor blade is the main blade of the Boeing CH-47 helicopter which is described in detail in Chapter IV.

The tail rotor, or rotary rudder, of a VTOL aircraft experiences a somewhat different loading environment from that of the main rotor. The blades do not see the significant cyclic stresses experienced by the main blade, yet have stringent aeroelastic requirements. An experimental composite tail rotor blade for the Sikorsky S-61 is seen in Fig. 17 [*3,11,12*]. This blade was designed to give higher chordwise stiffness than the production metal blade as seen in Fig. 18. The construction, shown in Fig. 19, consists of a D-shaped boron-epoxy spar with the fibers oriented in the spanwise direction, cross-plied fiberglass-epoxy skin and ribs, and a unidirectional boron-epoxy trailing edge insert for stiffness. The root end attachment was accomplished by splaying the boron-epoxy from the spar into a wider area in the root, and interleaving it with fiberglass-epoxy and metal shims to carry the bearing load of the attachment bolts. The composite blade weighs 5.85 lb compared with 6.15 lb for the aluminum production counterpart.

The composite tail rotor blade was subject to laboratory fatigue, whirl, and flight tests. The fatigue test consisted of imposing a steady load of 8500 lb

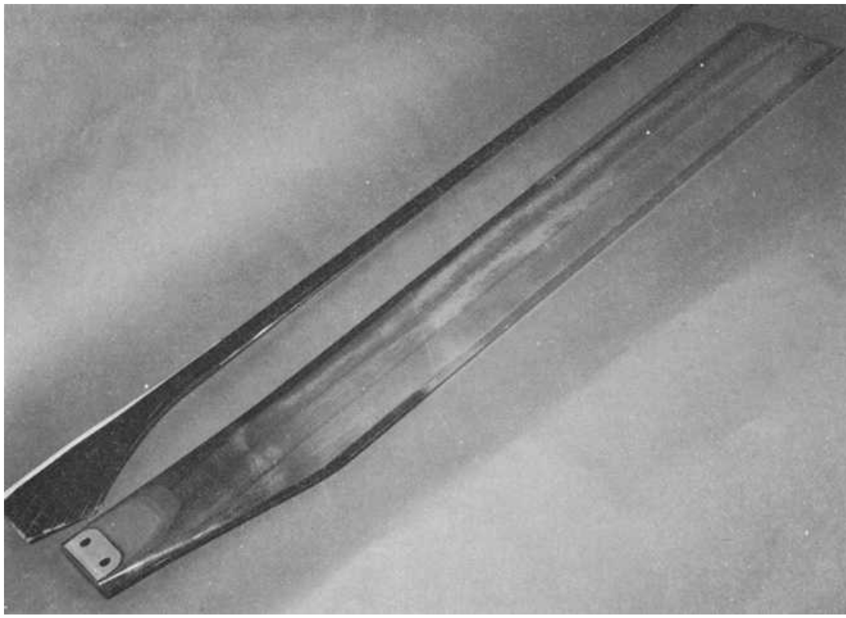


FIG. 17–*Boron and glass fiber reinforced epoxy S-61 tail rotor blade and unfinished boron-epoxy spar [12].*

(normal operating) and an edgewise bending moment of ±6000 in·lb, which is four times the normal cruise load. There was no indication of damage after fifteen million cycles, and the vibratory load was increased to ±7000-in·lb. After eleven million cycles, delamination in the root end attachment and cracking of the metal shims occurred.

A full set of five blades was then whirl tested, found to be flight-worthy, and flight tested in December 1968 (Fig. 20). This was the first flight of a primary aircraft structure fabricated from advanced high modulus composite material. The flight test included hovering at 104 percent of normal rated rotor speed, turns, sideward and rearward flight, takeoff and climb at maximum power, level flight at 120 knots, and an autorotation descent with power off.

A unique application of the anisotropic properties of composite materials was made by Kaman [*13*] for an experimental tail rotor for the Bell UH-1 (Fig. 21). The two-bladed rotor was constructed using a single continuous unidirectional fiberglass-epoxy spar which carried completely through the hub from the tip of one blade to the tip of the other. The high strength and stiffness of the unidirectional fiberglass-epoxy in the spanwise direction reacted all the centrifu-

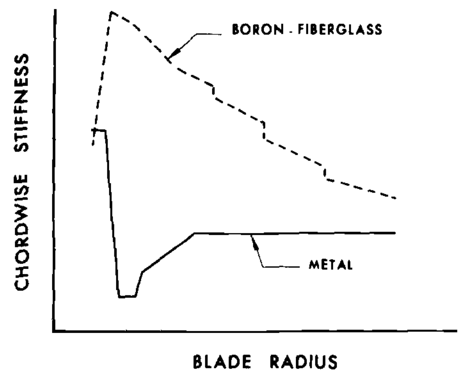
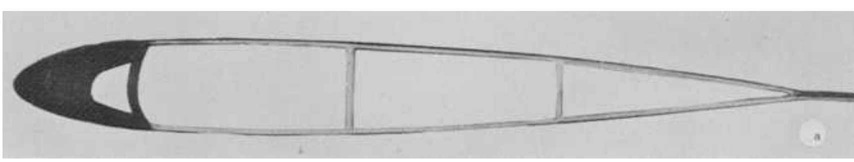


FIG. 18–*Relative chordwise stiffness distribution of S-61 production metal blade and experimental composite blade.*



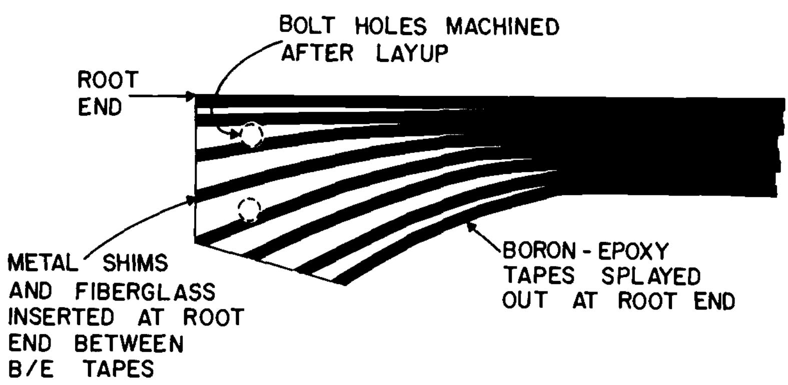


FIG. 19–*Construction of S-61 composite tail rotor blade: (a) blade cross section, (b) root end [3].*

gal and bending loads. The pitch control links were attached to an aerodynamic cover which was bonded to the spar, and the low torsional stiffness of the unidirectional spar allowed sufficient pitch motion without the necessity for pitch bearings. A similar concept utilizing high modulus composites for a hingeless main rotor has been investigated by Arcidiacono and Cheney.

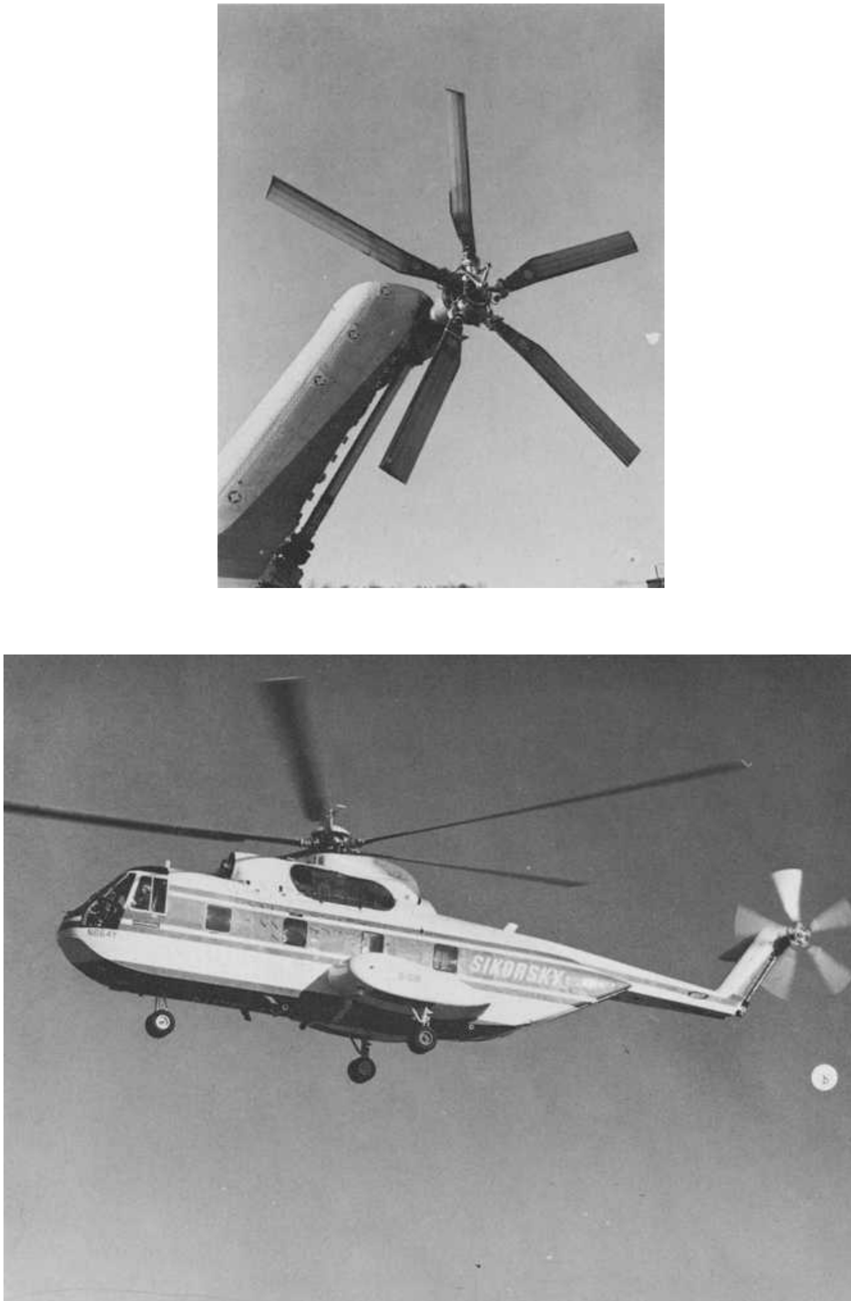


FIG. 20—*Flight test of S-61 composite tail rotor blades: (a) tail rotor, (b) aircraft in flight.*

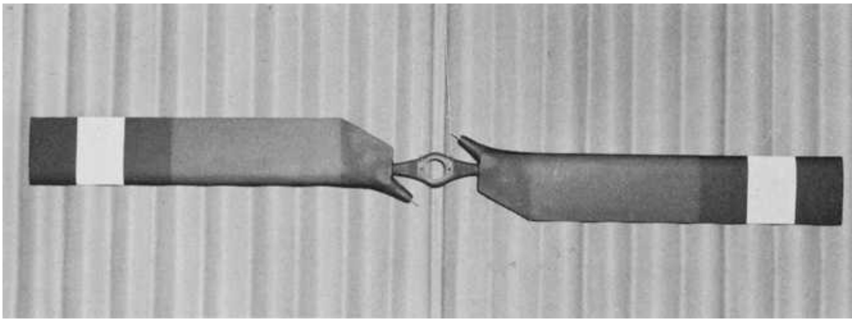


FIG. 21–*Single fiberglass spar UH-1 tail rotor (Kaman Corp.).*

Drive System

A feature common to all VTOL aircraft, except the direct lift engine machine, is the extensive use of connecting drive shafts. These are used to transfer power to the tail rotor in most helicopters, between the two rotors in tandem helicopters, and for interconnection of drive systems in tilt-rotor and tilt-wing aircraft such as the VFW-400 seen in Fig. 22. These drive systems typically consist of sections of metallic tubes, one to four feet in length, separated by bearings and couplings to compensate for misalignment (Fig. 23). They operate at several thousand rpm and are designed to have their fundamental frequency well above operating speed. Since their primary function is to convey torque, they are designed for torsional strength and buckling stability.

A section of the S-61 drive system made from boron-epoxy is seen in Fig. 24. It consists of a 57-in. long, 3-in. diameter, ±45-deg boron-epoxy sandwich tube bonded to steel end fittings. Initial experiments with single wall boron-epoxy tubes at ± 45 deg were unsuccessful because although stronger than aluminum, the extreme anisotropy of the composite resulted in torsional elastic instability at low loads [*14*]. The use of a sandwich structure consisting of two concentric tubes with a low density core of foam or honeycomb has overcome the stability problem and allowed composite tubes of high structural efficiency to be produced as seen in Fig. 25. A double wall boron-epoxy drive shaft was mounted on an S-61 helicopter in 1969 and successfully passed a 2-h ground run-up endurance test.

Surprisingly, the composite drive shaft was found to be easier to balance dynamically than its production metal counterpart. Because the metal shaft is precision machined and the composite shaft is molded and adhesively bonded, it was expected that the latter would have a less uniform mass distribution. The better dynamic balance of the composite shaft was attributed to two factors: the mass distribution being concentrated nearer the end fittings (nodes); and the high damping capacity of the ±45-deg composite material.

Although composite shafting provides weight savings and better damping than

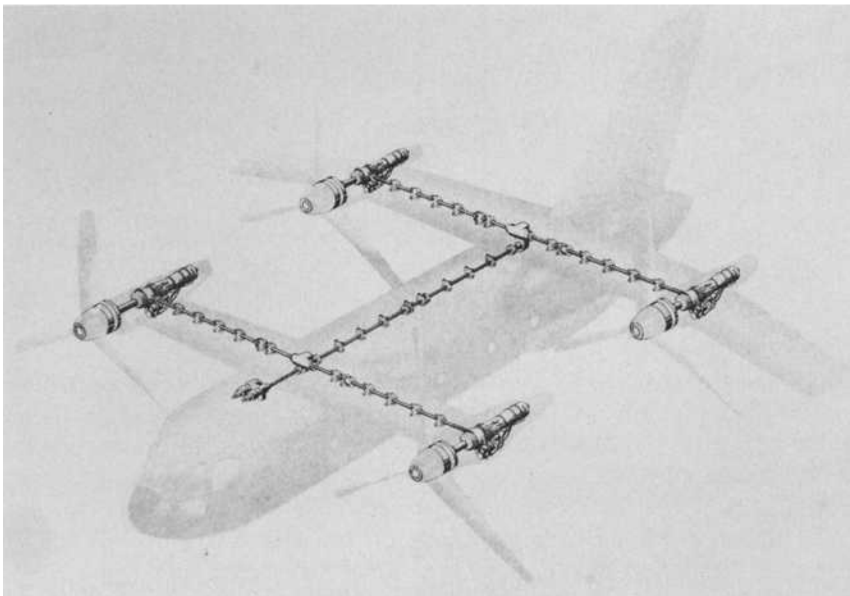


FIG. 22–*Interconnecting drive system of the VC-400 tilt-wing aircraft (Vereinigte Flugtechnische Werke).*

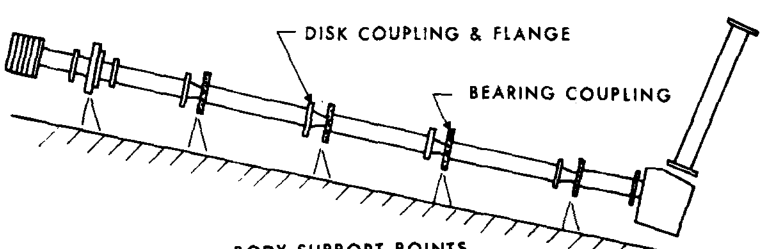


FIG. 23–*S-61 helicopter tail rotor drive shaft system.*

FIG. 24–*Boron-epoxy tail rotor drive shaft.*

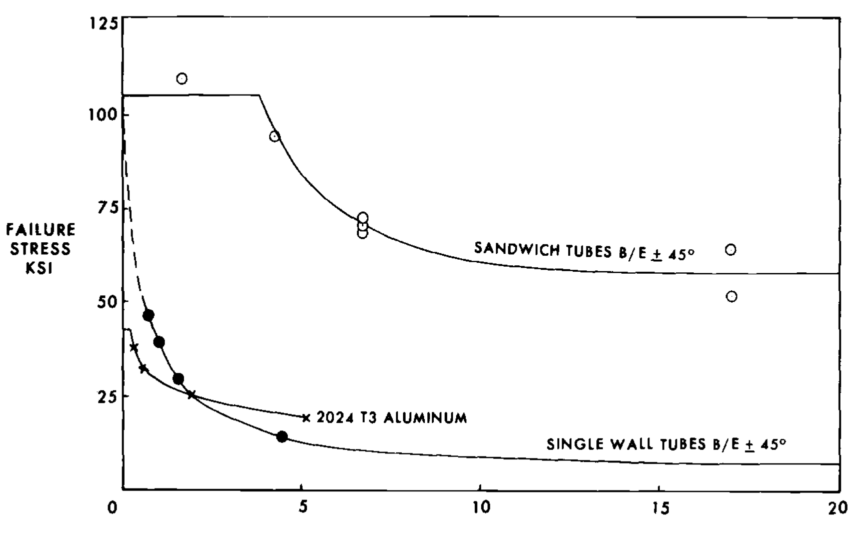


FIG. 25–*Torsional strength of composite and metal drive shafts.*

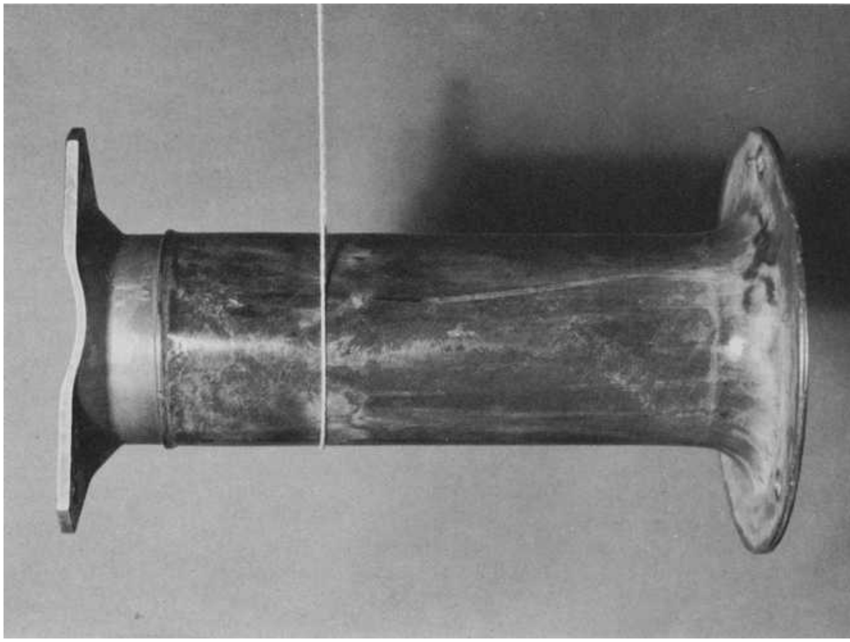


FIG. 26—*Shaft having integral end fitting and heavier conventional bonded metal end fitting suspended by string from its center of balance.*

metal shafting, perhaps a more important factor is the ability of the composite properties to be tailored to provide longer lengths and reduce the number of bearings and couplings in the drive system. In addition to the ±45-deg laminae to react torque, axial layers can be added to react bending and provide dynamic tuning. A recent example of this is a 92-in. long graphite-epoxy drive shaft designed and built by Bell Helicopter and Whittaker Corporation [*15*].

As with most composite structures, a vital consideration is the end attachment. The adhesively bonded metal end fittings seen in Fig. 24 are structurally adequate, but would not be considered safe for flight because the strength of the adhesive bond could not be substantiated by nondestructive testing (NDT). Present NDT methods are capable of detecting de-bonds and voids, but cannot detect weak or no-take bonds. This is presently a serious limitation of composite materials because adhesive bonds permit ease of manufacture.

The integral end fitting seen in Fig. 26 provides a continuous load path from end to end by forming the fibers from the tube into a bell shape and interleaving them with metal shims to take the bearing loads required of the bolted attachment. In addition to eliminating the bonded end attachment, the integral end fitting weighs less than half of the bonded metal end fitting as illustrated in Fig. 26.

The excellent structural efficiency of composites for drive shafts makes them promising for other applications such as engine input shafts, main rotor shafts, and for a variety of airborne and land-based rotating machinery. In addition to shafting, composites have been considered for gear box housings [*16*] and for large ring gears. In both cases, the higher stiffness would reduce gear mesh tolerance, considerably reducing gear wear.

Control System

Control systems of helicopters consist of a series of pushrods and linkages, bellcranks, and rotating and stationary swash plates. The stiffness of this system must be matched to that of the rotor blades, because the control load response of the rotor is a function of the total system stiffness. As helicopters fly faster and carry greater loads, the necessity for greater system stiffness is increased. This can be obtained in the rotor blades by increasing torsional stiffness using high modulus composites, but the control system must also be stiffened. This

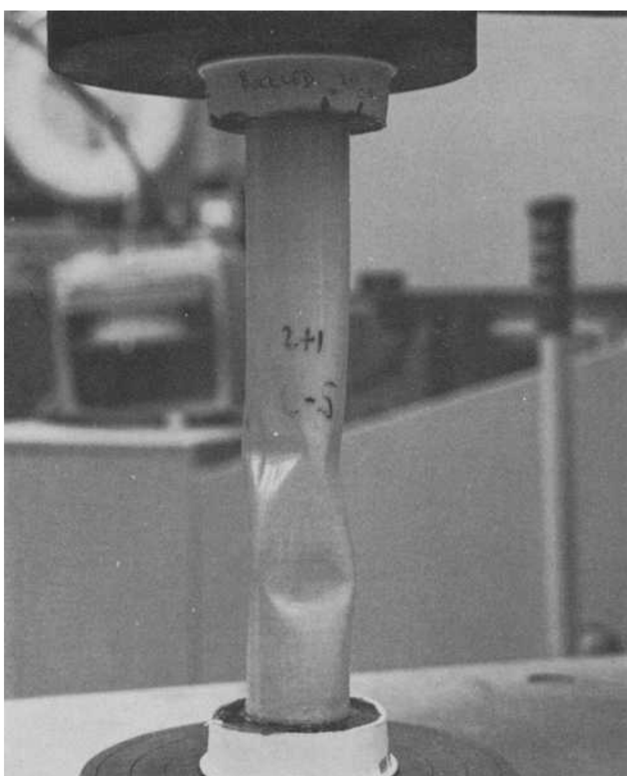


FIG. 27–*Compression test of fiberglass composite control rod section exhibiting buckling.*

can be effected by reinforcing metal swash plates with high modulus composites and using composite bellcranks and pushrods, such as that seen in Fig. 27. An additional feature of composites for control systems is their high damage tolerance which is critical for such safety-of-flight items.

Airframe

The airframe of a rotary wing aircraft is unique in that it requires dynamic tuning to preclude rotor-fuselage responses which can cause undue vibration. A classic example of such a problem was encountered with the Army/Sikorsky CH-54 Skycrane aircraft as seen in Fig. 28. The vertical rigidity of the tail cone was originally designed for static loading. During flight testing, an interaction between the one-per-revolution vibration of the rotor and certain size loads suspended at certain cable lengths caused a vertical bounce in the airframe. This was solved by increasing the vertical rigidity of the tail cone as given by the upper curve in Fig. 28.

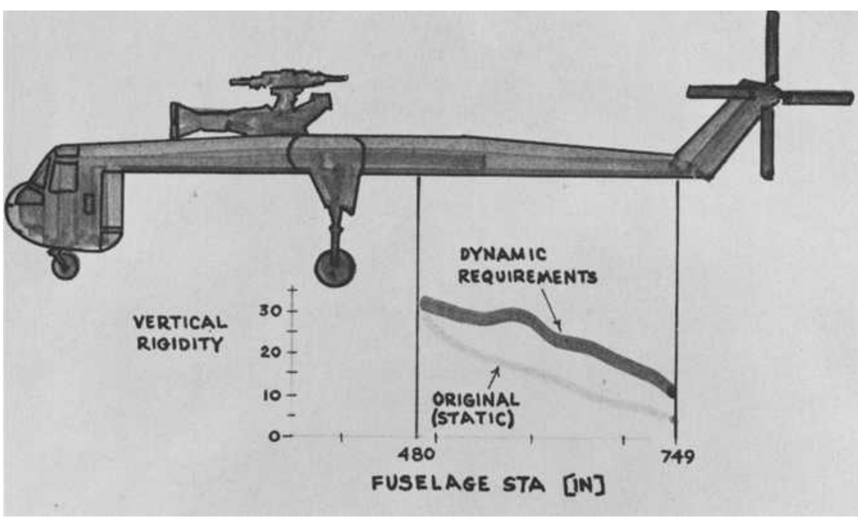


FIG. 28—*Vertical rigidity design for CH-54 tail cone.*

The increased stiffness was accomplished by increasing the thickness of the upper and lower aluminum skins from 0.040 to 0.125 in. for a total weight increase of 160 lb (Fig. 29). Because the requirement was for stiffness only, the same result could be accomplished by using boron-epoxy bonded to the upper and lower stringers for a weight saving of 130 lb, as seen in Fig. 29. By adhesively bonding precured segments of unidirectional boron-epoxy to the vertical leg of the stringers (Fig. 30), the aircraft could be assembled by riveting through the aluminum without changing tooling or making attachments to the boron-epoxy. A production aircraft with this configuration was introduced into service with the U.S. Army in March 1972.

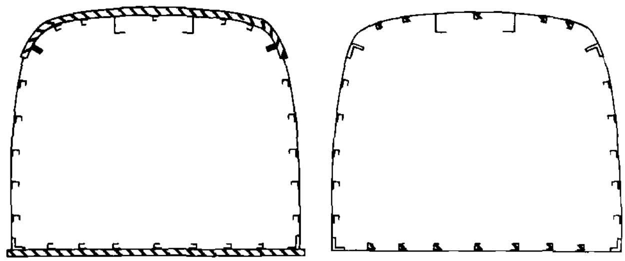


FIG. 29–*Alternate designs for increased vertical rigidity of CH-54 tail cone.*

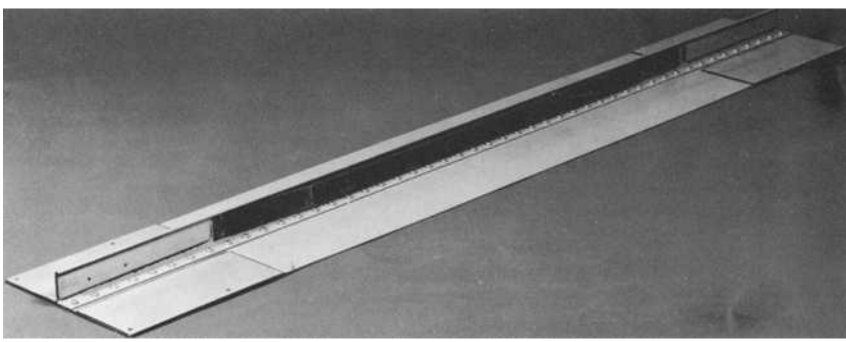


FIG. 30–*Aluminum stringer with adhesively bonded boron-epoxy.*

Another concept is the all composite stringer, seen in Fig. 31, which would be bonded to the skins and would result in reduced drag by decreasing the fuselage diameter as well as resulting in reduction in weight empty.

For low speed helicopters such as cranes, the use of truss construction for the fuselage appears particularly attractive for the use of composites. Because trusses utilize directionally loaded tension-compression members, the use of anisotropic high modulus composites can be extremely efficient. Collings and Steinlein [*17*] have projected weight savings of more than 50 percent for composite compression tubes compared with metal tubes (Fig. 32). An estimate of the weight savings possible through the use of a composite truss fuselage has been made for the CH-54 helicopters in Fig. 33 and is summarized in Table 1. Note that in addition to the structural weight savings, the open truss allows the downwash of the rotor to pass through the fuselage with less vertical drag than for the conventional semimonocoque fuselage, thus allowing an additional increment in payload. Because the truss structure is less streamlined than the closed conventional structure, there is an increase in forward drag which would reduce the cruise speed of the CH-54 with a truss tail cone by an estimated 3 knots.

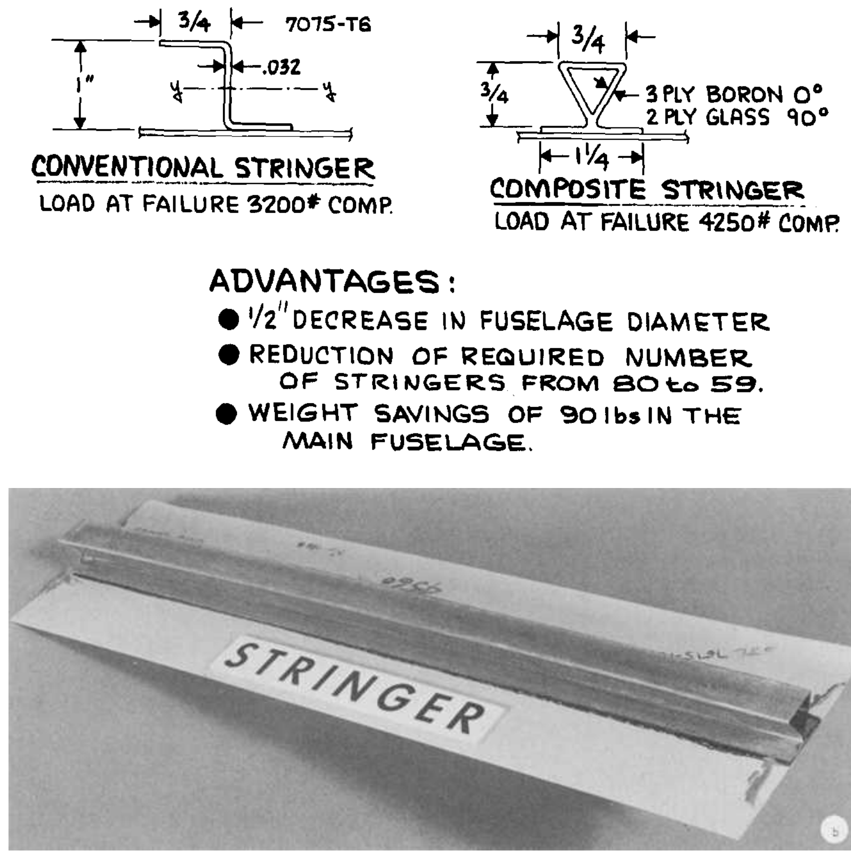


FIG. 31—*Composite stringer (a) comparison with metal stringer for compound aircraft (b) boron-epoxy stringer adhesively bonded to skin.*

TABLE 1—*Weight savings for CH-54 helicopter with truss.*

	Fuselage Center	Landing Gear	Tail Cone	Vertical Drag
Present design Aluminum skin and stringers	1445 lb	600 lb	387 lb	3060 lb
Truss design High modulus composite	850 lb	300 lb	160 lb	2450 lb
Weight saving	41%	50%	59%	610-lb increased payload

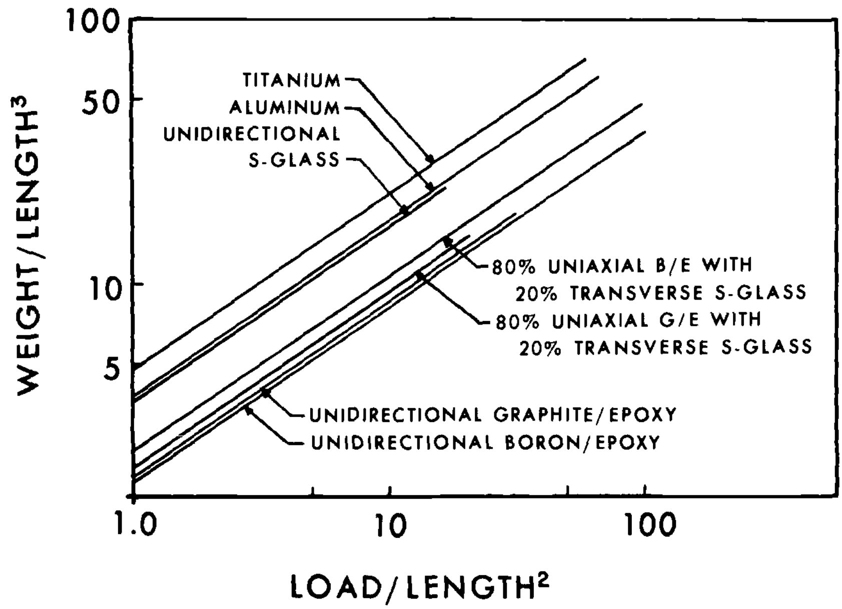


FIG. 32–*Structural efficiency of optimum compression struts [17].*

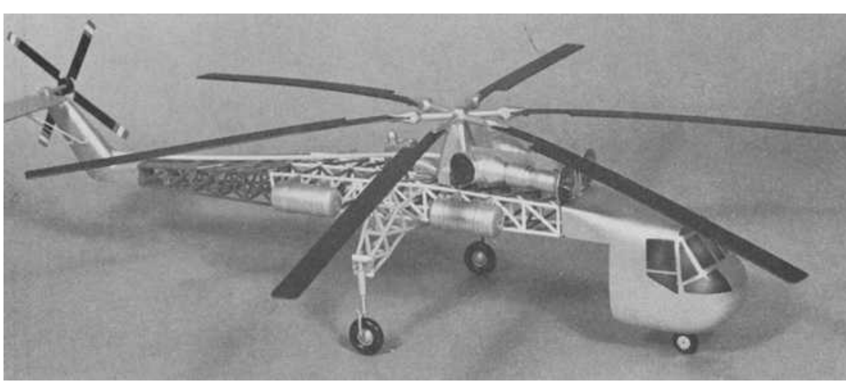


FIG. 33–*CH-54 crane helicopter with truss fuselage.*

Another example of the weight savings made possible by a composite truss is the compression boom of the twin lift system seen in Fig. 34. This concept uses two crane helicopters to lift a load which is beyond the lifting capability of one and consists of a compression boom and cable tension members. The boom designed for use with two CH-54 helicopters consists of an aluminum tubular truss weighing 1400 lb. A comparable truss designed for boron or graphite-epoxy would weigh 600 lb.

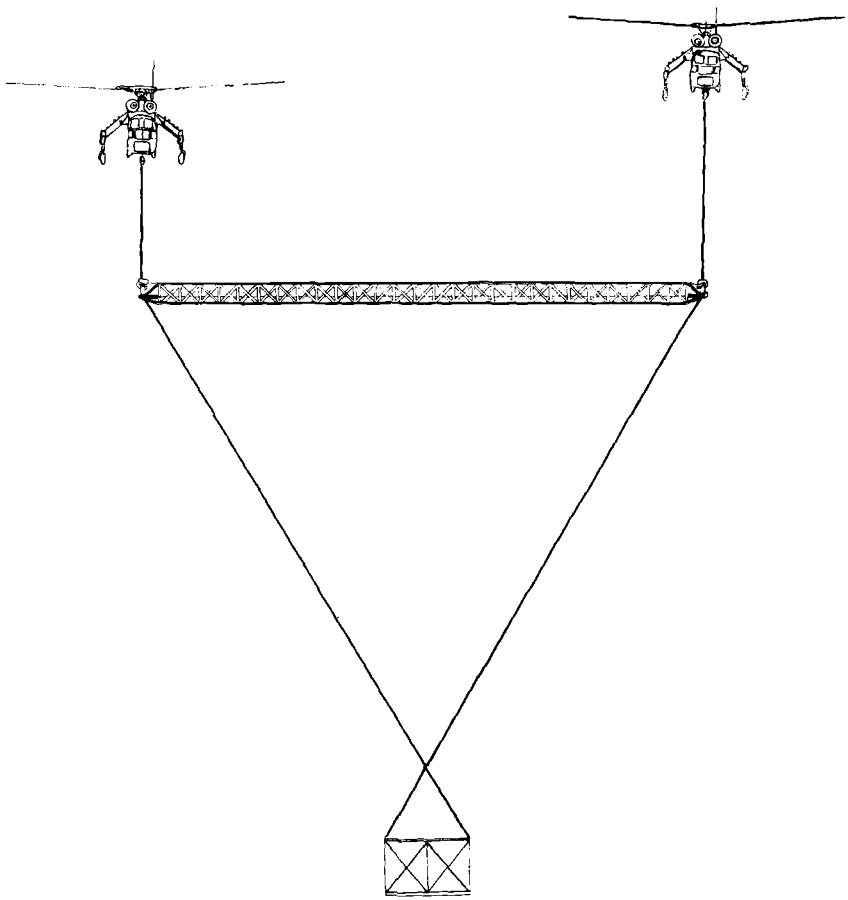


FIG. 34–*Twin lift system with truss compression boom.*

In order to demonstrate the concept of fabricating a composite truss, the tail skid of the CH-54, seen in Fig. 35 was redesigned for use with composite tubes [*17*]. Geometry constraints associated with end clearance and the shape restrictions placed on the end fittings by the existing hardware resulted in the design of an off-optimum tube. The final design was a 2.0-in. diameter tube having six layers of unidirectional boron-epoxy, two layers of S-glass-epoxy at 45 F and a circumferential S-glass-epoxy wrap on the outside for damage protection. The end fittings consisted of aluminum cylinders with tapered ends as seen in Fig. 36. The end fittings were adhesively bonded to the tubes and bolted to provide a mechanical backup for the adhesive bond. Additional plies of 45-deg glass-epoxy were inserted between boron layers at the ends to provide additional bearing strength. The specimens shown in Fig. 36 were tested in

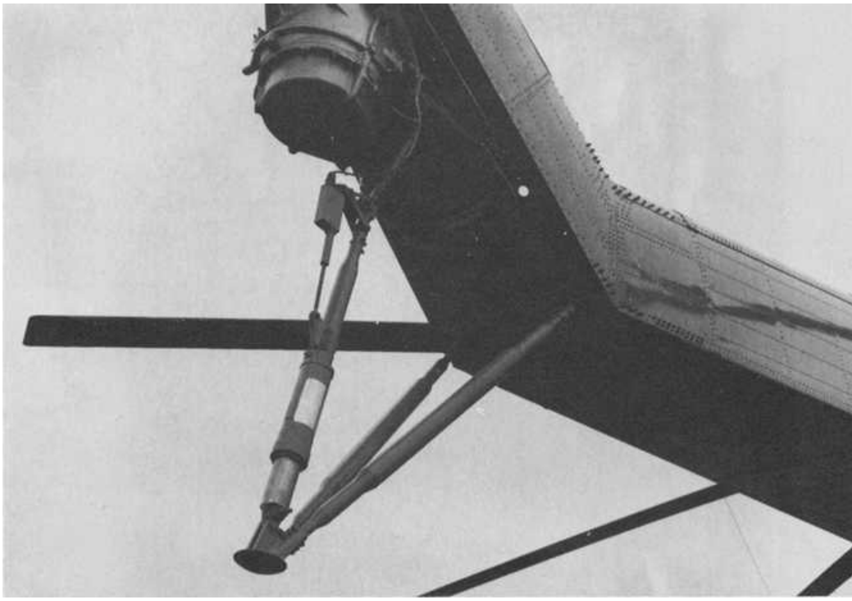


FIG. 35–*CH-54 tail skid.*

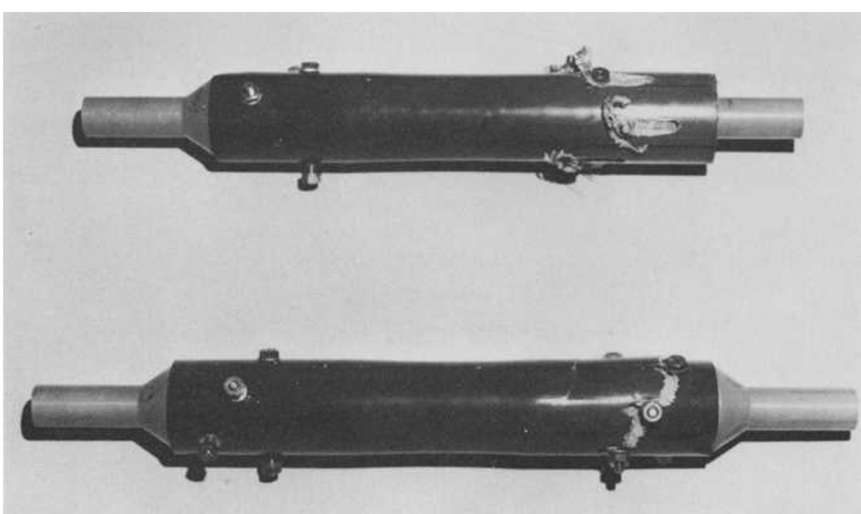


FIG. 36–*Compression* (top) *and tension test of bolted and bonded end fittings.*

compression and tension to demonstrate the structural integrity of the end fittings. In both cases the adhesive bonds failed at loads varying from 15 000 to 17 000 lb, and the bolts allowed an additional 2000 lb of load before failure. The design limit load for the configuration was 9300 lb.

FIG. 37–*Composite tail skid.*

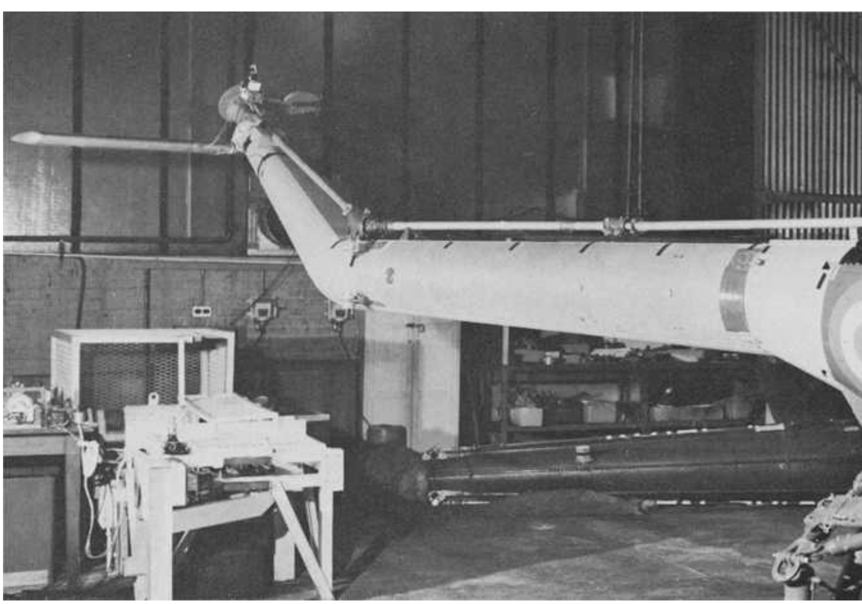
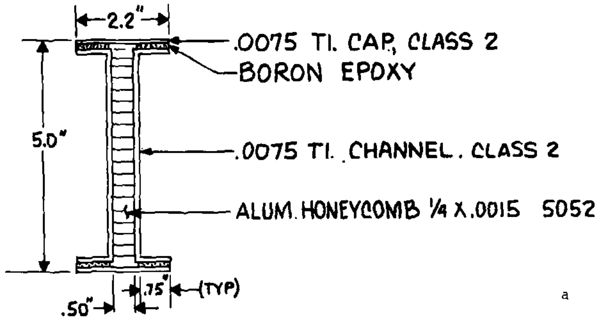


FIG. 38–*Fiberglass reinforced plastic tail cone for the Westland Wasp Helicopter (Westland Helicopters Ltd.).*



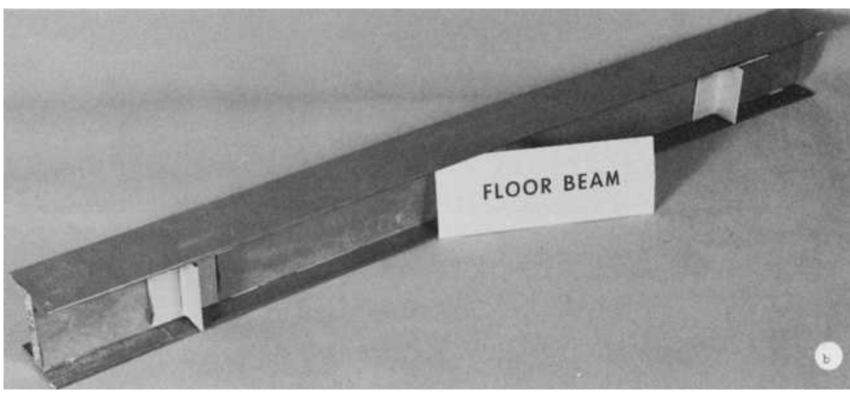


FIG. 39–*Hybrid composite-metal floor beam: (a) construction, (b) test section.*

The composite tubes were fabricated using an automated rolling table with a teflon coated aluminum mandrel. The finished composite skid seen in Fig. 37 weighed 21.25 lb compared with 23.24 lb for its metal counterpart. Each composite tube weighed 1.1 lb compared with 3.2 lb for the aluminum tube.

An interesting application of composite materials is the fiberglass reinforced plastic (GFRP) tail cone for the Westland Wasp seen in Fig. 38. The composite structure was considerably less complex and required fewer parts than its metal counterpart.

The beams and frames in the fuselage cabin represent a substantial portion of the airframe weight and a potential area for weight saving. A floor beam for a large passenger helicopter was designed using composite material (Fig. 39). It consists of titanium skin with a low density honeycomb core in the web and high modulus boron-epoxy in the beam cap. If such a structure were utilized for a floor with no intermediate support posts (clear hold for cargo), a 60 percent weight saving is projected when compared with an aluminum beam. If the subfloor hold could be broken up by intermediate floor posts, then the high stiffness would not be utilized as efficiently, and a 25 percent weight saving would accrue.

Cargo floors are designed to resist deflection, impact, puncture, and wear. The hybrid composite cargo floor in Fig. 40 is 20 to 40 percent lighter than metal cargo floors. It consists of a sandwich structure of high modulus with a low density core to resist deflection, a balsa wood layer to absorb energy due to impact, and a titanium skin to provide a puncture and wear resistant surface. The boron-epoxy bottom skin can be seen in the inverted section in the upper part of Fig. 40.

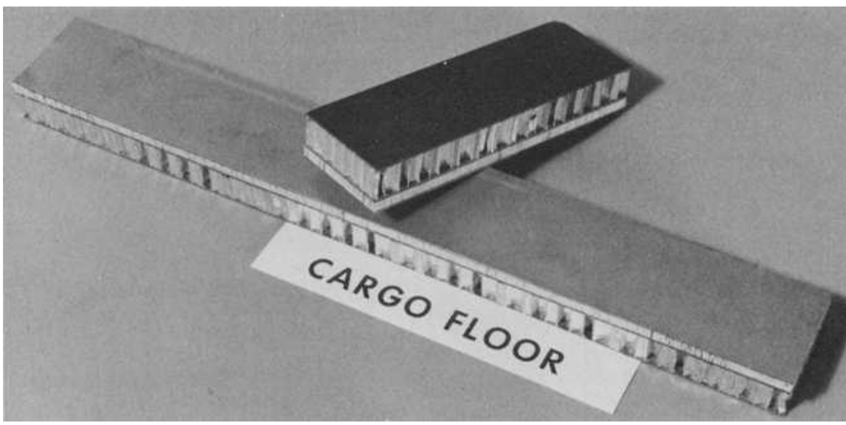


FIG. 40—*Composite cargo floor.*

Conclusions

VTOL aircraft generally cost more in terms of initial cost and operating cost per pound of payload than fixed wing (CTOL) aircraft. The additional cost is warranted when a vertical operating mode is necessary. As a result of this differential, weight savings made possible by composite materials are usually more cost effective for VTOL aircraft. In addition, the more stringent dynamic environment of VTOL aircraft favor the use of composite materials for dynamic tuning. Also, the use of composite materials promises substantial improvements in structural reliability and fail-safety.

It is expected that within the next decade, there will be profound advances in VTOL aircraft resulting from the advent of composite materials. Although the application of composites will also result in improved productivity in CTOL aircraft, the greater potential that is possible with VTOL aircraft will likely make them more competitive in the future.

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Chapter IV – Helicopter Rotor Blades

REFERENCE: Pinckney, R.L., "Helicopter Rotor Blades," *Applications of Composite Materials, ASTM STP 524*, American Society for Testing and Materials, 1973, pp. 108-133.

ABSTRACT: The development of fiber reinforced resin-bonded structural composite materials has created a new degree of design flexibility for the helicopter engineer. The development and commercial availability of glass fibers, carbon or graphite fibers, and boron fibers, with their attendant widely varying stiffness properties and material densities, enable the specialist both to design his structure to utilize the excellent fatigue properties of the materials available and to design the inherent properties of the material which he intends to use in his advanced VTOL systems.

Filament reinforced epoxy structures are compared with metal structures commonly used for helicopter rotor blades. Test data on several fiber reinforced epoxy materials and development test results of full-sized reinforced epoxy rotor blade segments are given. The manufacturing techniques, quality assurance provisions, and economic considerations of the cost of materials versus component performance improvements are also discussed.

KEY WORDS: composite materials, fibers, rotary wings, rotors, helicopters, fatigue (materials)

This chapter presents general information on the application of advanced composite materials to new product lines. Specifically, it deals with work carried out by The Boeing Company, Vertol Division, in the application of advanced structural composite materials to helicopter rotor blades. Figure 1 illustrates the relative size and shape of various blades which have been flown on the CH-47 helicopter. The two lower blades are a constant airfoil steel spar construction representative of current production. The upper blade is an advanced airfoil design fabricated from fiber reinforced composite materials.

Design Advantages

In blade applications there are three major advantages to be realized from the application of fiber reinforced composites.

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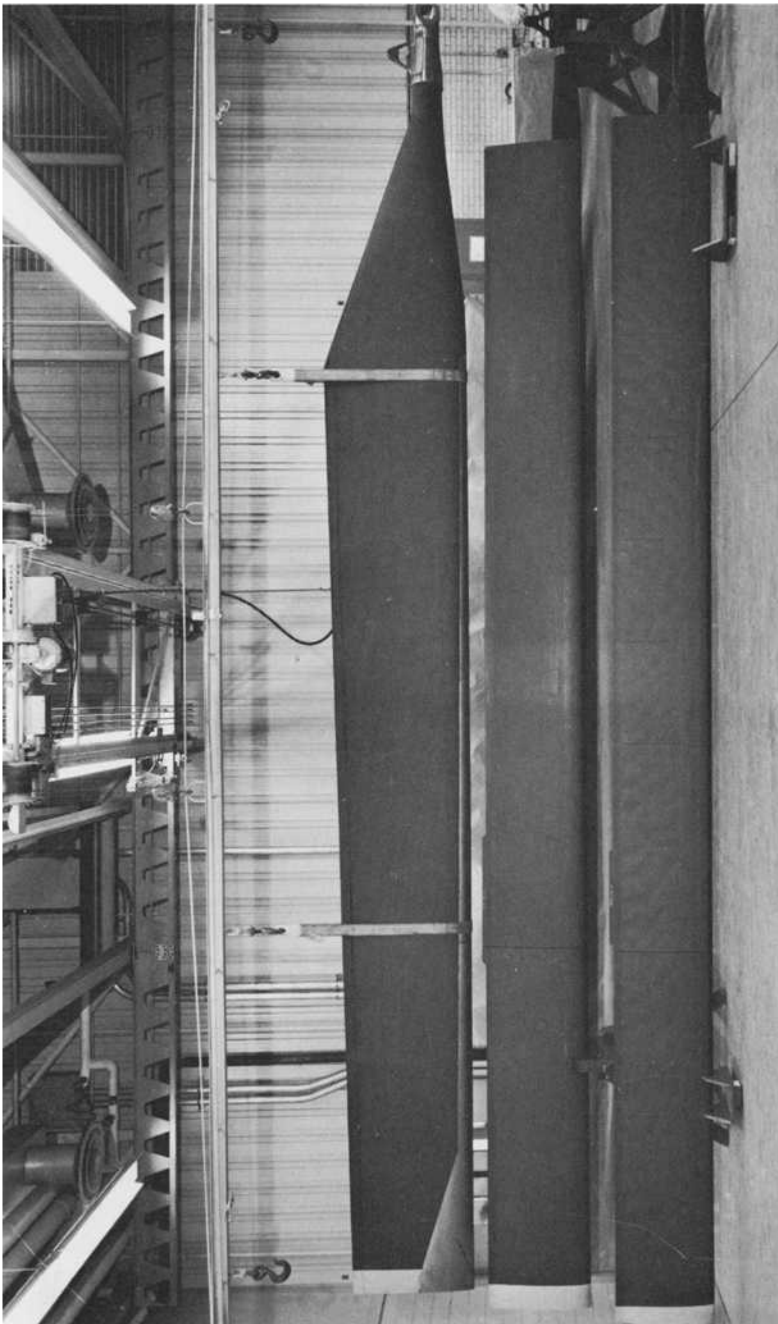
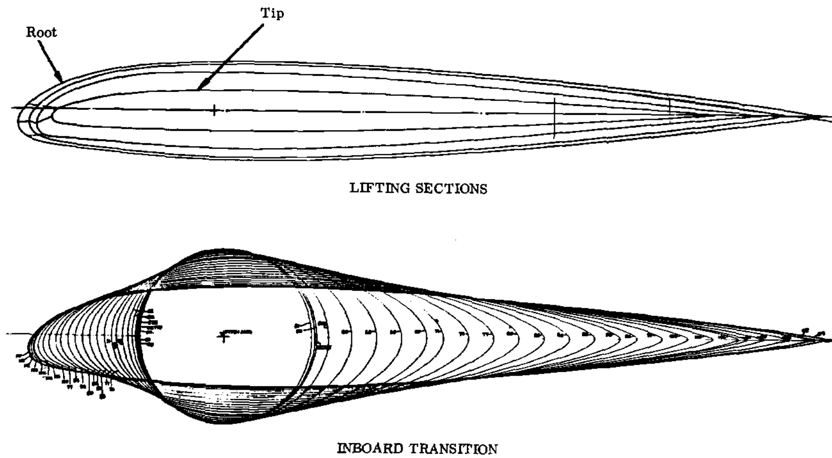


FIG.1—*Rotor blades, CH-47, current production and advanced airfoil composite structures.*



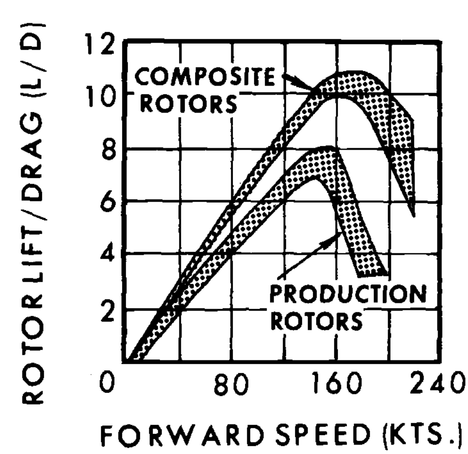


FIG. 3—*Improvement in rotor efficiency, lift/drag, attainable with advanced geometry.*

Airfoil

The first advantage is in airfoil configurations. The open die concept of blade molding and the honeycomb stabilized construction used in the development of the composite blade are not limited to shapes which can be extruded, machined, or rolled, as is the case with more conventional tubular metal spar designs. Complex airfoil shapes are possible with small increases and, in some cases, a significant decrease in component manufacturing costs. Thus, improved aerodynamic efficiency is possible through the use of composite materials. The composite blade in Fig. 1 has an airfoil chord ranging from 39 to 29 in., and an

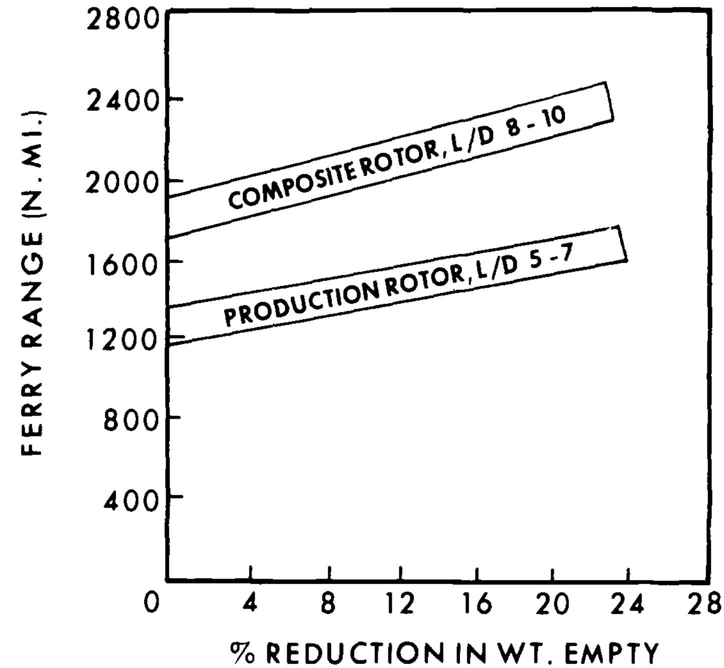


FIG. 4—Range improvement with advanced geometry blades.

airfoil thickness of 12 percent at the maximum chord and 6 percent at the blade tip. Typical blade airfoil sections are shown in Fig. 2. The materials and tooling concepts now available allow the aerodynamicist to select each individual airfoil section along the blade to satisfy a particular performance objective. Figures 3 and 4 show the increased helicopter performances attainable with advanced airfoil composite structures.

Weight

The second recognizable advantage of composite materials to helicopter systems comes from weight savings that translate directly into increased performance or payload capability. The improvement in operational payload with weight empty reductions indicates that a 10 percent reduction in weight empty for the CH-47 helicopter, for example, will yield a 30 percent increase in payload. Reduction in weight empty, through the use of advanced composite materials throughout the vehicle structure, will also translate into additional fuel capacity which improves helicopter range (Fig. 4). A composite material application resulting in a 10 percent reduction in weight empty of the helicopter will yield a 40 percent increase in operational range.

The rotor designer frequently finds that due to total system requirements, there is a minimum blade weight required. Such factors as centrifugal force and coning angles on rotor system loads, frequencies, and stored rotor inertia requirements, sufficient to allow nominal pilot reaction time in case of power

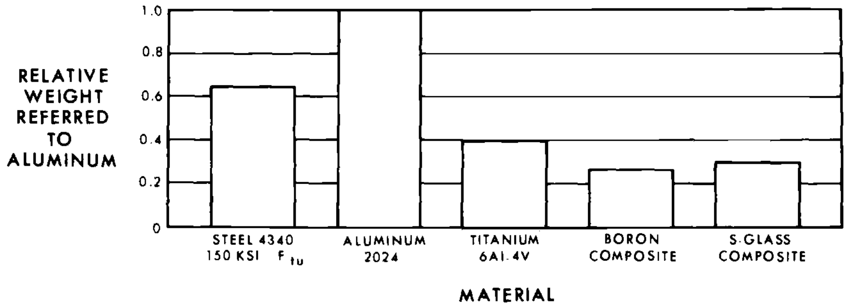


FIG. 5—*Comparison of material relative weights required to sustain high cycle fatigue loads.*

failure, may well dictate a blade weight and mass distribution in excess of the structural material weight required when composite materials are used. The relative structural efficiencies of metallics and composite materials under fatigue loading conditions are shown in Fig. 5.

There are certain applications, however, where the utilization of composite materials can effect significant reductions in blade weight. A significant application is in attaining suitable chordwise balance. Normally, rotor blades are balanced about their quarter chord, and as a result, a 1-lb reduction in trailing edge weight will result in a 3-lb reduction in the over-balance weight required in the blade leading edge.

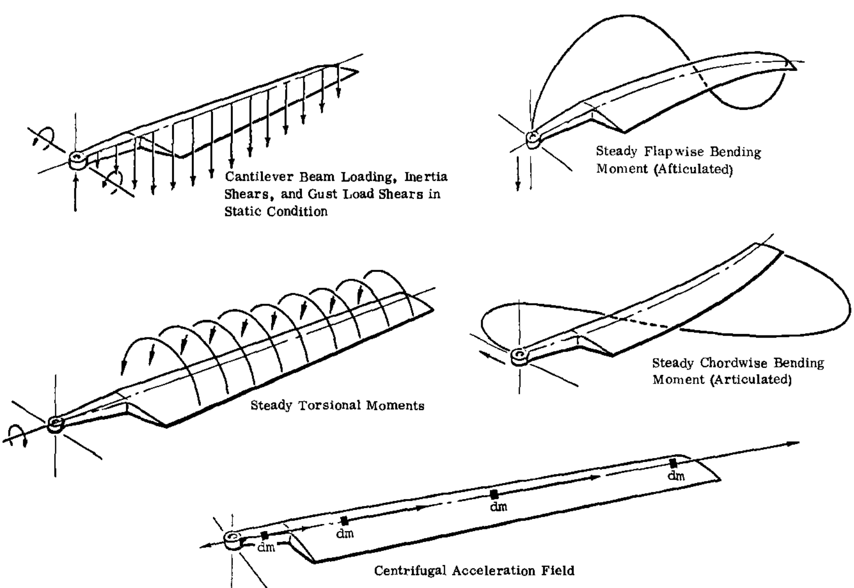


FIG. 6—*Principle steady blade loading.*

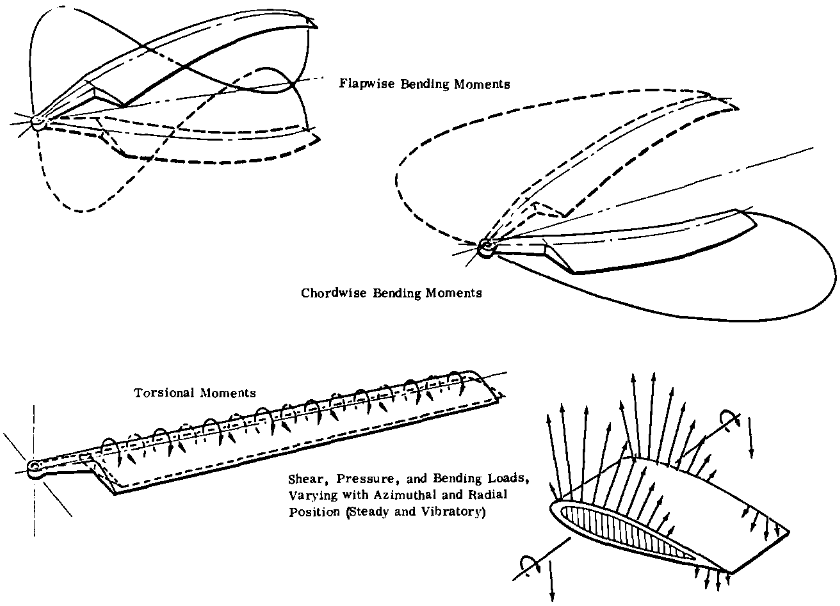


FIG. 7—*Principle cyclic or vibratory blade loading.*

Dynamics

The third, and perhaps most significant advantage of the use of composite materials, is the ability to tailor the dynamic frequencies and structural responses of the blade element to its operating parameters. Figures 6 and 7 show the static and dynamic loading environment in which typical blades operate. Blade first mode flapping frequencies, for example, can be changed not only by the classic method of mass distribution, but also, by the selection of specific fibers having a high or low modulus, concentration and distribution of the fibers in the resin matrix, and the angle of fiber orientation. All of these tools are available to the designer to control the frequencies within the blade. For

TABLE 1—*Effect of materials on blade dynamics.*

Construction		Flap Stiffness (EI) Constant		Torsional Stiffness (GJ) Constant		Weight Constant	
Spar	Skin and Torsion Wrap	Relative Weight	Relative Torsional Stiffness	Relative Weight	Relative Torsional Stiffness	Relative Flap Stiffness	Relative Torsional Stiffness
S-Glass epoxy	S-Glass epoxy	1	1	1	1	1	1
S-Glass epoxy	boron-epoxy	0.93	2.7	0.85	1	1+	3.7
S-Glass epoxy	carbon-epoxy	0.88	3.3	0.83	1	1+	5.1

example, significant increases in blade torsional rigidity can be obtained with minor effects upon first mode longitudinal frequencies by increasing the quantity of fibers oriented at ±45 deg to the blade span. The dynamic changes resulting from material variations are shown in Table 1 for a typical main rotor system.

Chordwise Tuning—High modulus composite materials are better suited to "tailoring" blade natural frequencies than are metals or glass fiber materials because of their higher stiffness to mass ratios. As an example, chordwise (in-plane) tuning is achieved through varying the axial stiffness (product of area and tensile modulus, AE) of a continuous trailing edge strip. Therefore, a significant change in frequency can be achieved with boron or graphite composites over glass fibers, since a major stiffness increase (AE) can be obtained for essentially the same trailing edge weight.

In one instance, the integration of high modulus materials into the trailing edge of a composite blade, maintained at a relatively constant weight for other design reasons, was found to change the first mode chordwise natural frequency from approximately 4.5 to 6.0. Again, the stiffness properties of the materials can be changed by varying the fiber angle and volume ratio to achieve the desired AE characteristics.

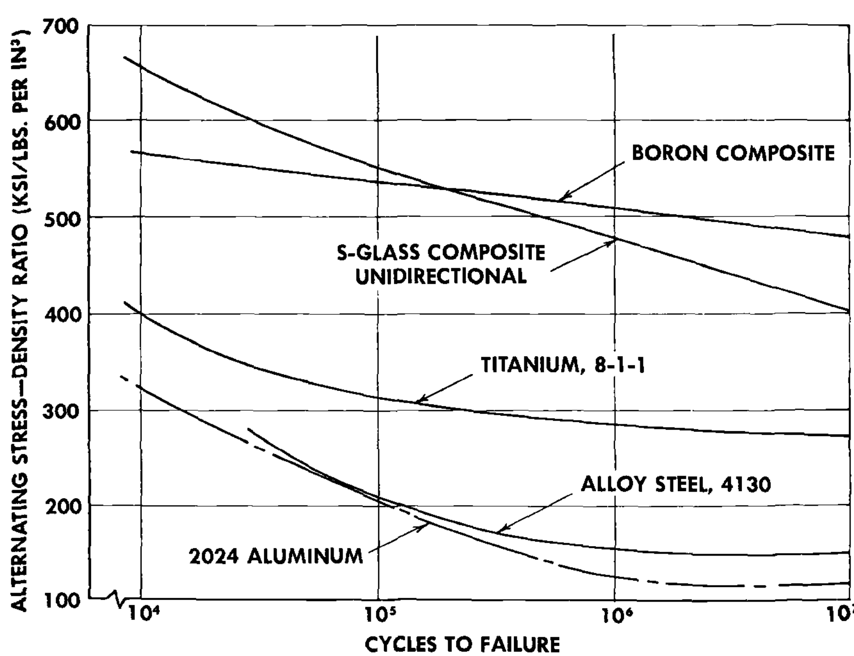


FIG. 8—*Typical tension-tension fatigue data (R = 0.1).*

Torsional Tuning—For a given rotor-helicopter system in which blade length is fixed by system design, rotor efficiency can be improved by increasing blade chord length and decreasing the airfoil thickness. This change in design has two adverse effects on the blade torsional responses: (*a*) decreasing the torsional stiffness, due to a reduction in blade cross sectional area; (*b*) increasing the polar moment of inertia exponentially, as a result of increasing chord length. The net result for thin tipped advanced rotors is a significant reduction in torsional stiffness for any given material. Due to the relatively low shear modulus of fiberglass, torsional frequencies on a fiberglass blade are low by comparison to metal blades. Equivalent frequencies may not be achieved even with the expense of additional material and resultant weight, because the polar moment of inertia will also increase due to the added mass. The use of boron or graphite, however, permits the achievement of comparable torsional frequencies without weight penalty, and may actually result in blade weight reductions due to the higher GJ obtainable at a reduced polar moment of intertia.

Materials and Design Considerations

Alumino silicate or S-glass filament reinforced polymer materials are readily available in collimated fiber tape and sheet forms and were among the first materials evaluated for use in primary load carrying airframe structures. Initial composite blade work used E-glass materials for the construction of a constant airfoil blade and is reported [*1*][2]. Later, advanced filaments such as boron [*2*] and graphite became available and are currently being evaluated in laboratory specimens and full scale hardware. Boron helicopter rotor blades were developed under Air Force contract [*3*], and extensive development of graphite engine components is reported [*4*]. All of the composite materials mentioned above have shown one or more desirable properties which make them suitable for advanced aerospace applications. Figure 8 shows the cyclic fatigue performance obtained with aluminum, steel, titanium, glass, and boron materials. Data available [*4*] on graphite materials indicates that a similar performance regime may be expected from these.

For design purposes, the testing of tubes, laminates, and honeycomb sandwich beams has established relationships between filament orientation and elastic and mechanical properties. Typical relationships are shown in Figs. 9 and 10. Limited data on the results of creep testing is available. However, this property does not appear to be a problem at present. Flight testing of glass fiber helicopter blades has shown no permanent deformation, and tests reported in the literature [*4*] and shown in Fig. 11, do not indicate that significant creep of the material occurs.

Limited quantitative data are available at this time on many of the other engineering properties of composite materials of interest to the helicopter designer. For instance, some fatigue evaluation of S-glass materials has been

[2]The italic numbers in brackets refer to the list of references appended to this paper.

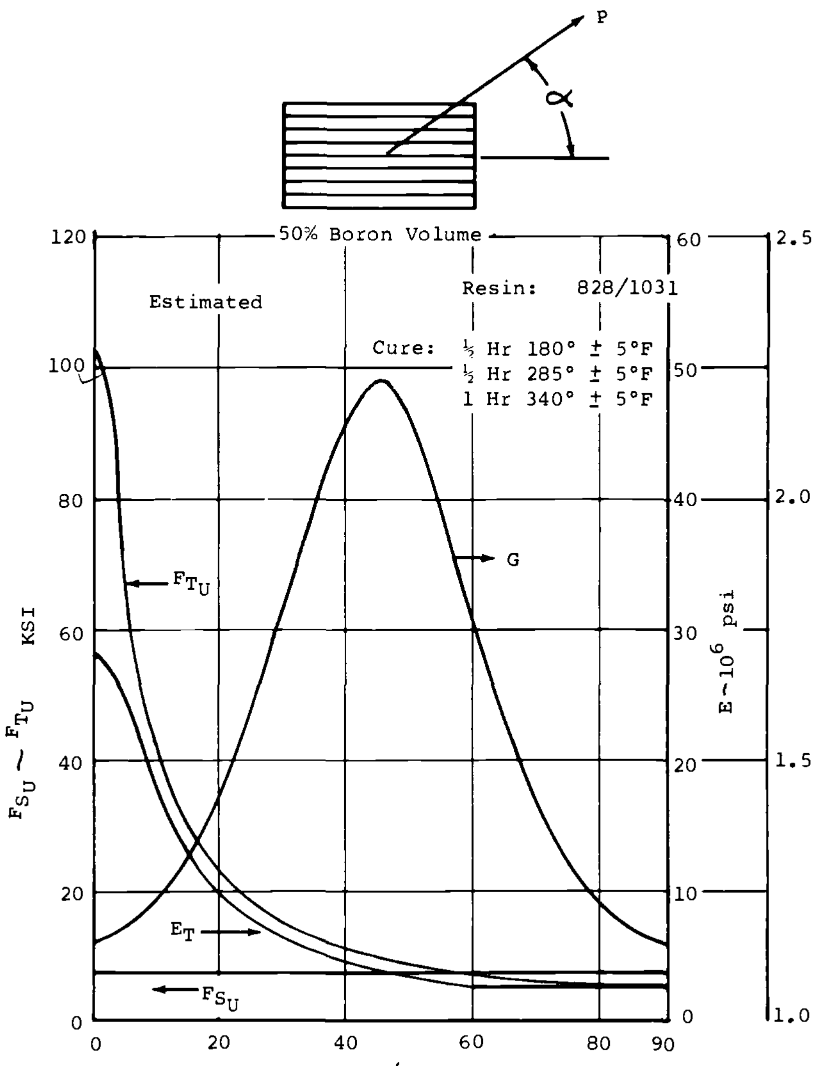


FIG. 9—*Effects of load direction on unidirectional properties.*

made on sandwich beams and laminates after exposure to actual weathering, artificial weathering, and condensing humidity at no load [*5*]. However, work is incomplete on the effects of the above environments and others, including temperature cycling while the materials are under combinations of steady and cyclic loading.

The effects of fabrication defects, notch sensitivity, and impact loading have not been thoroughly explored for composite materials performance. Many examples of the use of polyester and epoxy matrix-glass composites can be

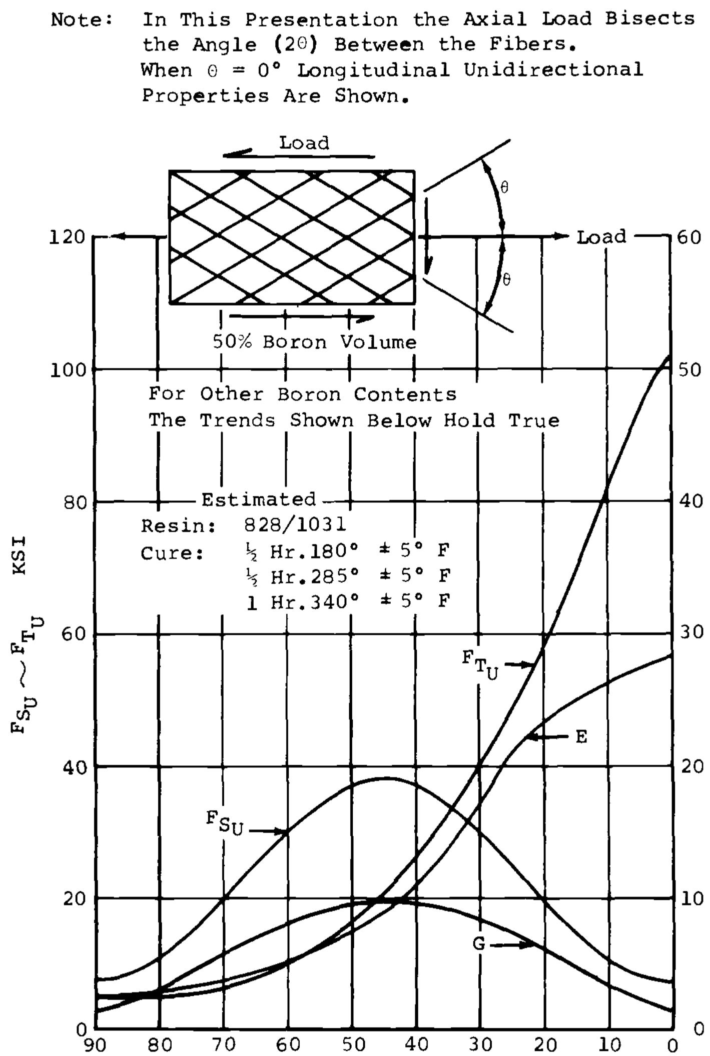
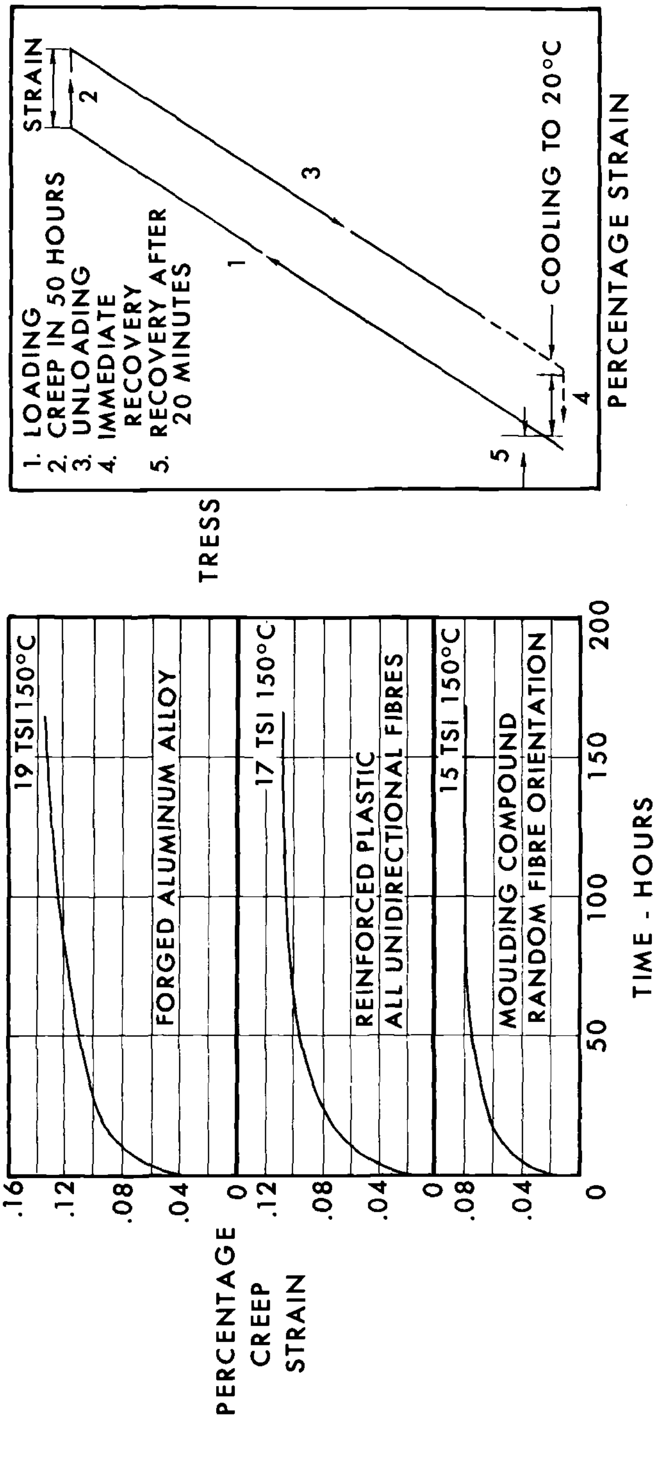


FIG. 10—*Fiber orientation effects.*

found in the general literature which pertains to industrial usage in applications such as chemical piping, fume ducts, building components, automotive applications, and wind tunnel and water tower cooling fans. In such applications the performance of composites under severe environmental climates and long term cyclic loading has been most successful. Although these data are most encouraging and indicate that composites are fully capable of maintaining their superior performance, their long term performance in actual rotary wing applications remains to be demonstrated.



4.11a
CREEP PERFORMANCE

4.11b
TYPICAL CREEP RECOVERY DIAGRAM UNIDIRECTIONAL MATERIAL

FIG. 11—*Typical creep behavior of structural materials [4].*

Master Dimensioning System

One of the major advantages for the use of composite materials in blade construction is the capability of producing complex geometric forms. A major consideration must be the lofting and subsequent tool fabrication required to provide close tolerances for the sophisticated airfoil geometry (Fig. 2). A technique has been utilized to minimize lofting and tool template production which also establishes the basis for automated machining of full scale tooling and certain blade details. All structural component surfaces have been mathematically defined using a procedure called "Master Dimension Identifier" (MDI). Key contours are established using second degree or cubic equations to establish key airfoil stations. All surfaces are then faired mathematically between the airfoil stations. The output of the computerized program is a complete *x-y-z* coordinate definition of every point in the blade structural component surface. The data are stored in a model 360 computer core, from which tapes can be prepared for any two-dimensional section, which is automatically lofted by the *X-Y* Gerber plotter.

Three-dimensional machining also can be performed by preparing input tapes for use on large numerically controlled (NC) milling machines. This provides the potential for multiple series tooling with identical shapes by automatically machining the required tapered contours and twisting into the assembly fixtures. The system also fabricates the matched tooling for internal bags and mandrels, and machines the honeycomb blade components.

Tool and Fabrication Concepts

A number of processes are available for the manufacture of composite blade structures. These include press laminating, matched die molding, and positive

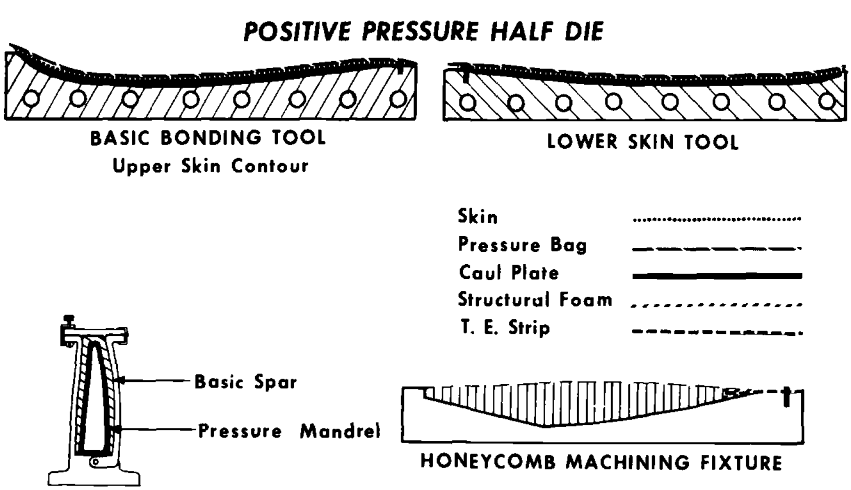


FIG. 12–*Principle tooling components.*

fluid pressure molding. The process used for the advanced blades utilizes an autoclave, wherein molding pressures of 100 psig are employed to form the spar against pressure stabilized external molds. Autoclave pressure is applied to the internal bags and reacted by autoclave pressure on the external tools.

The aluminum honeycomb core is installed and bonded at 25 to 30 psig. Any one of several resins or adhesives are suitable for this operation. Primary resin prepreg systems are bonded directly to the core, and precured skins are bonded using structural adhesive films curing at approximately 350 F. Good results have also been obtained with epoxy adhesive film systems which cure at 250 F. A sketch of the tooling principles utilized is shown in Fig. 12. Figure 13 identifies the locations of the various elements such as external skins and the unidirectional and cross-ply buildups in the basic spar.

The prototype nature of the current advanced blade programs led to a decision to use less costly, conventional plaster master and transfer techniques and fiberglass tooling, rather than NC machined metal tools. Lofted contours and master templates were fabricated using the MDI system, and all blade component check templates were produced in the same manner.

Initial blade fabrication development centered around the root end portion of the blade spar. This section transitions from a closed round tube 8 in. in diameter to an open C-section at the maximum blade chord area (Fig. 13). This section includes the root end fittings, inner and outer bias tape spar torsion

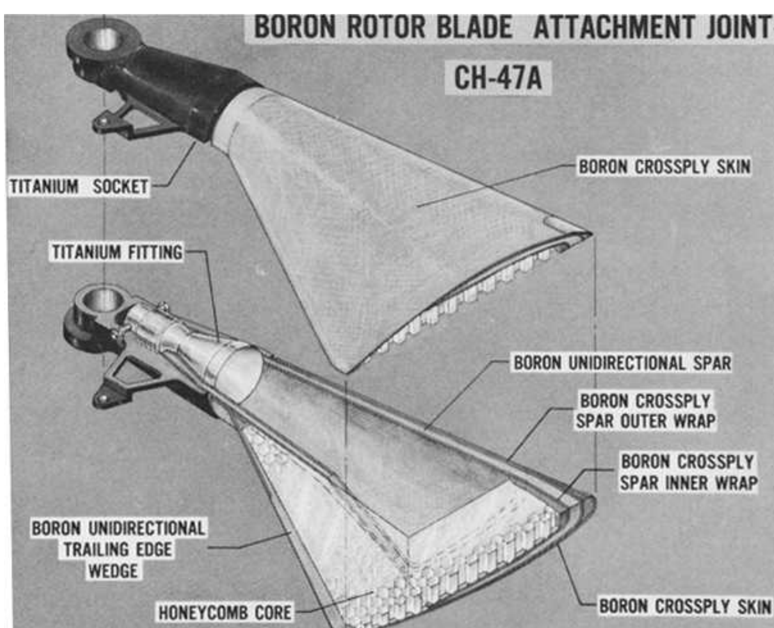


FIG. 13–*Rotor blade details, CH-47 composite blade.*

wraps, and the unidirectional spar material. Tooling principles, material layup, and cure techniques were most critical in this segment of the component. The large number of plies of tape material in the layup necessitated a debulking or compaction process to remove entrapped air and to assure a rough dimensional fit of the outside contour molds and the proper inner contours necessary to assure uniform pressure during the cure cycle.

The problems were resolved in the fabrication of an S-glass specimen after several development cycles. The same tools and processes were then used to reproduce a geometrically similar boron reinforced component.

Tool design requirements for the fabrication of large heavy laminates, such as main rotor spar, have been found to be much more stringent than for conventional metal bonding operations, or for the production of more conventional glass fiber structures common within the aircraft industry. The construction of large blade components having a nonuniform material mass leads to problems in uniform heat up and cooling rates, which in turn, can lead to rather high thermal stresses and attendant warpage problems during component cure and subsequent assembly. The use of relatively thin glassfiber tooling and the balanced pressure concepts, previously described, were found to provide a suitable answer to this problem. In addition, proper tool placement and, in some cases, autoclave circulating air baffling were found to improve component temperature uniformity.

There is, of course, an exothermic chemical reaction involved in most common resin matrix curing systems, and this can lead to disastrous results in the curing of heavy laminates if proper cure cycles are not used. The laboratory examination of the potential exotherm for the proposed resin fiber system by means of full thickness laminate cure tests is recommended. It has been found that multiple step thermal cure cycles with intermediate temperature holding periods will control the exothermic reactions of the commonly used resin systems available at present. The system must be carefully developed, and its effect on the composite structural properties must be determined. The usual tests to determine laminate qualities such as density, porosity, interlaminar shear, and transverse tensile strength provide an excellent guide for the determination of the optimum cure cycle. The results must be checked to insure that the expected structural design properties have been attained.

Full Scale Component Testing

The full scale rotor test and evaluation program was designed to substantiate the flight test structural adequacy of the new materials and the design and fabrication concepts incorporated in the composite rotor blades.

Static and Fatigue Tests

During the fabrication development program, the first components were inspected by nondestructive techniques, and then torn down for destructive examination and analysis. In this manner, the effect of cure cycle changes on laminate physical properties, dimensional accuracy and component fit up, and the actual condition of NDT indicated anomalies could be evaluated. Based on these results, a spar root end section and two fully completed blades were fabricated for sectioning and full scale static and dynamic fatigue evaluation.

The static test program included the following:

(*a*) spar section root end retention strength,
(*b*) full length blade natural frequency determination, and
(*c*) chord bending and flap bending proof loads.

The dynamic combined load fatigue program included:

(*a*) root end blade attachment,
(*b*) mid-span blade segments,
(*c*) outboard blade segments,
(*d*) tip balance hardware area, and
(*e*) chordwise cyclic air loads.

The glass reinforced spar component was statically tested to nearly 2.5 times the design limit load (330 000 lb) in pure tension to evaluate the attachment design. The redundancy of the joint was demonstrated when the bond between the glass and inner titanium fitting, Fig. 13, failed at approximately twice design limit load. While the applied load dropped off slightly as the load redistributed, the structure continued to carry load to 2.5 times the limit. At that load a test fixture bolt failed, causing secondary failure of the outboard end of the component.

A similar boron reinforced component (Fig. 14) was subjected to combined axial load (tension) and cyclic bending for approximately 20 x 10^6 cycles at successively higher load levels between 100 and 200 percent of design steady-state flight loads, Fig. 15. Testing was discontinued when apparent failure

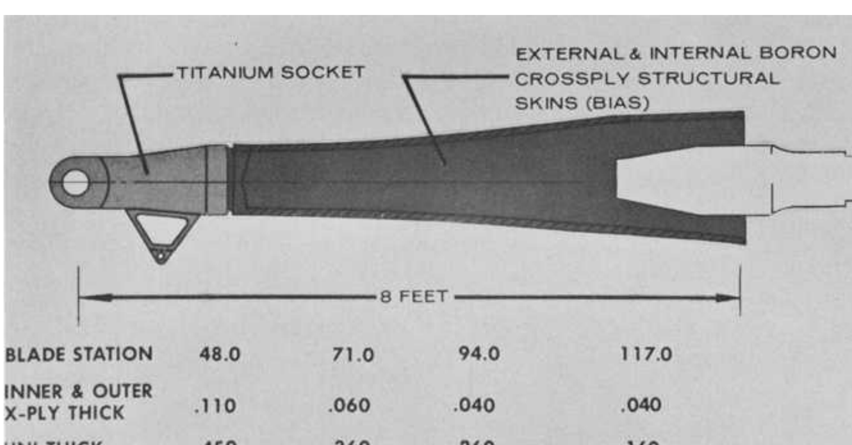


FIG. 14–*Spar root end configuration, six month demonstration article.*

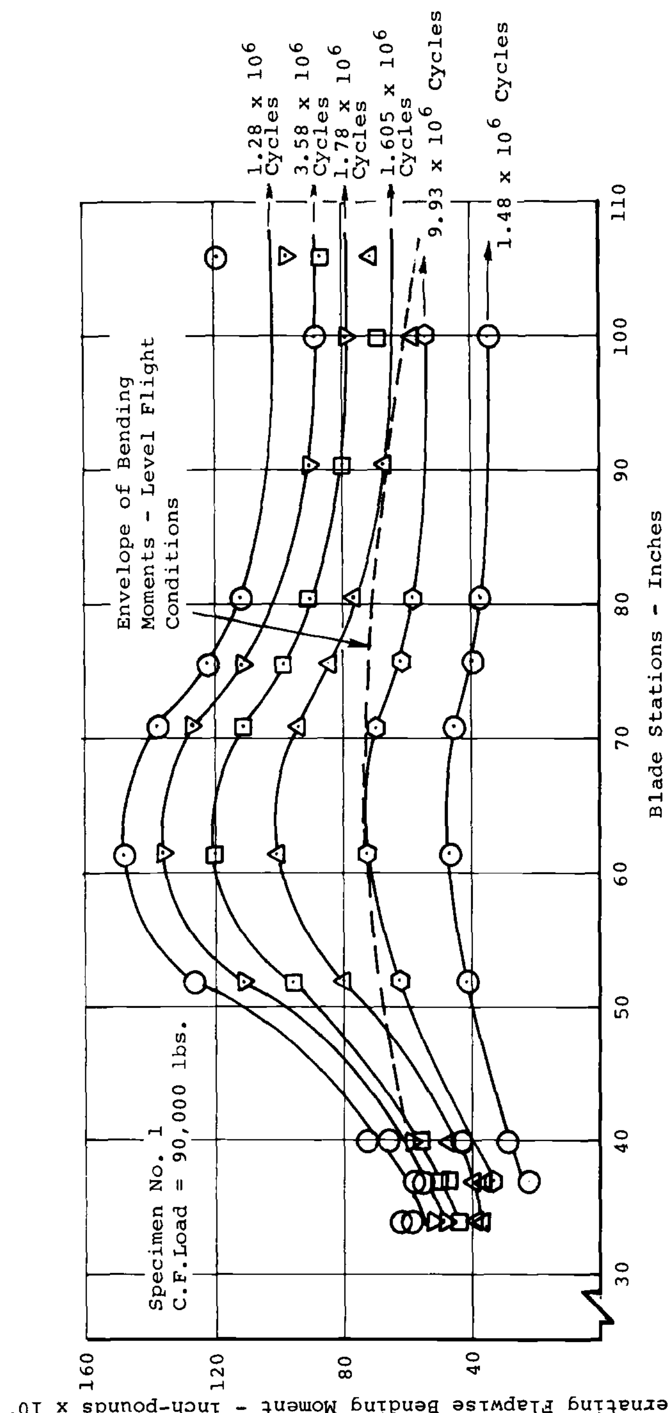


FIG. 15–*Root end specimen test load history.*

TABLE 2–*Advanced geometry blade fatigue test summary.*

Blade Test Area	Run No.	Flight Fatigue Load %	Applied Load: C.F. lb	Applied Load: Flap Moment, in · lb	Applied Load: Chord Moment, in · lb	Applied Load: Torsion Moment, in · lb	No. of Test Cycles, $\times 10^6$	Test Frequency, Hz
Root end	1	100	90 000	± 71 000	±24 000	...	10	8.2-10.4
	2	150	106 000	±110 000	±32 000	...	5	8.4
	3	200	106 000	±171 000	±46 000	...	1.33	7.5
	1	150	106 000	± 75 000	...	...	1.87	14.5
	1	150	106 000	± 75 000	...	...	3.51	12.5
Midspan	1	100	70 000	± 36 000	±23 500	...	10	16
	2	150	70 000	± 54 000	±30 350	...	5.62	15
	3	200	70 000	± 72 000	±40 470	...	2.75	12.5
	1	100	70 000	± 36 700	±20 700	...	10	15.4
	2	150	70 000	± 54 000	±30 350	...	3.23	...
Outboard span	1	100	43 500	± 27 700	...	11 000 ±10 500	1.69	...
	1	100	43 500	± 23 000	±10 300	...	10	13.8
	2	150	43 500	± 34 000	±19 200	...	5	13.3
	3	200	43 500	± 46 000	±25 600	...	3.4	13.3
Tip area	1	100	Loading simulates pressure distribution, both				10	...
	2	150	steady and alternating, over the airfoil section				5	...
	3	200	from 25 percent chord to trailing edge.				1.3	...
	1	100	Specimen fixed at leading edge.				10	...
	2	150					5	...
	3	200					.27	...

occurred in the specimen during post test calibration with a static loading equivalent to 283 percent design level flight flap bending moment. The specimen was strain gauged to monitor applied moment loads. Measured strains remained linear with increases in alternating flapwise bending moments to a measured maximum strain of 3300 μin./in. No permanent deformation or hysteresis was detected. Surface temperature remained approximately 10 deg over ambient. Measured forcing loads and phase relationships between forcing function and maximum deflection of the specimen indicated no direct association with structural damping. The results of full blade section fatigue testing of all glass fiber components is summarized in Table 2.

The fatigue testing of the major blade sections listed in Table 2 was conducted in special heavy structural steel frames equipped with vertical or chordwise flexure plates at each end of the test specimens. The spanwise steady force required to simulate the flight centrifugal force was obtained by means of calibrated spring banks acting through one of the flexure plates. Blade bending moments were induced by hydraulic actuators acting on a flexure plate. The actuator forces were controlled by pressure regulation, and the force frequency was controlled by electro magnetic valves actuated by means of an amplified frequency generator source. The load reversal hydraulic actuators and the specimen were thus operated at the resonant frequency of the blade section. Frequently, it was found necessary to add large bob weights to the blade sections in order to achieve the higher bending moments. None of the static or fatigue tests conducted indicated any danger of structural failure in any of the blade components in flight. Further, it was impossible to drive the loads high enough to induce valid fatigue failures in all of the blade sections. The full scale tests provided little data to verify material design allowables or the design safety factors utilized, although the program did generate stresses in some areas in excess of the basic material fatigue allowables. The program has indicated that composite structural materials can be utilized for primary load carrying structures, and that test design and technology must be further improved in order to produce fatigue failure in the new performance regime of composite materials. By this means, full scale component failure modes and stresses can be compared with like results obtained on laboratory bench test specimens. Such valid comparison data will allow for more realistic design structural factors to be utilized in future work.

Whirl Tower Demonstration

All eight flight vehicle blades, three forward, three aft, and forward and aft spare blades, were whirled, balanced and tracked or adjusted to fly, within 1/8 in. of each other on a 10 000 hp whirl tower. The final adjustments required were obtained by tip weight mass balance fittings and by adjustment of a full length thermo-plastic-glass fiber trailing edge tab.

The forward set of rotor blades designated for flight, including one blade instrumented to measure bending moments, chordwise tip air loads, and principle stresses near the three-quarter blade radius point, as well as blade motions, were whirl tested for approximately 60 h.

The whirl test program followed the Mil-T-8679 designations [6], except for the test time required. The blade performance was excellent. They were stable throughout the range of rpm, cyclic, and collective loading tests. No hinge or control system problems were encountered. Overspeed conditions were also evaluated up to 288 rpm. Normal flight rotor speed design was 230 rpm.

Flight Testing

A full scale flight test program was completed [7]. No structural problems were encountered, and the predicted increase in maximum gross weight, speed and altitude performance was met or exceeded. The CH-47 equipped with composite rotor blades is shown in Fig. 16, and the speed altitude performance envelopes for the CH-47C and the advanced geometry composite blade equipped CH-47C are shown in Fig. 17.

Quality Assurance

No composite material application will survive satisfactorily in a production environment unless adequate quality control techniques are established which can rapidly locate and define structural defects in a manner requiring little or no interpretation by a skilled operator. Advances in the technology presently offer radiation, ultrasonic, and infrared detection techniques which can define defects as small as 1/16 in. diameter in a 1-in. thick laminate. Distinguishing between clusters of small defects at adjacent strata and a single, larger defect at a discrete

FIG. 16–*Advanced geometry composite rotor equipped CH-47 Chinook in flight.*

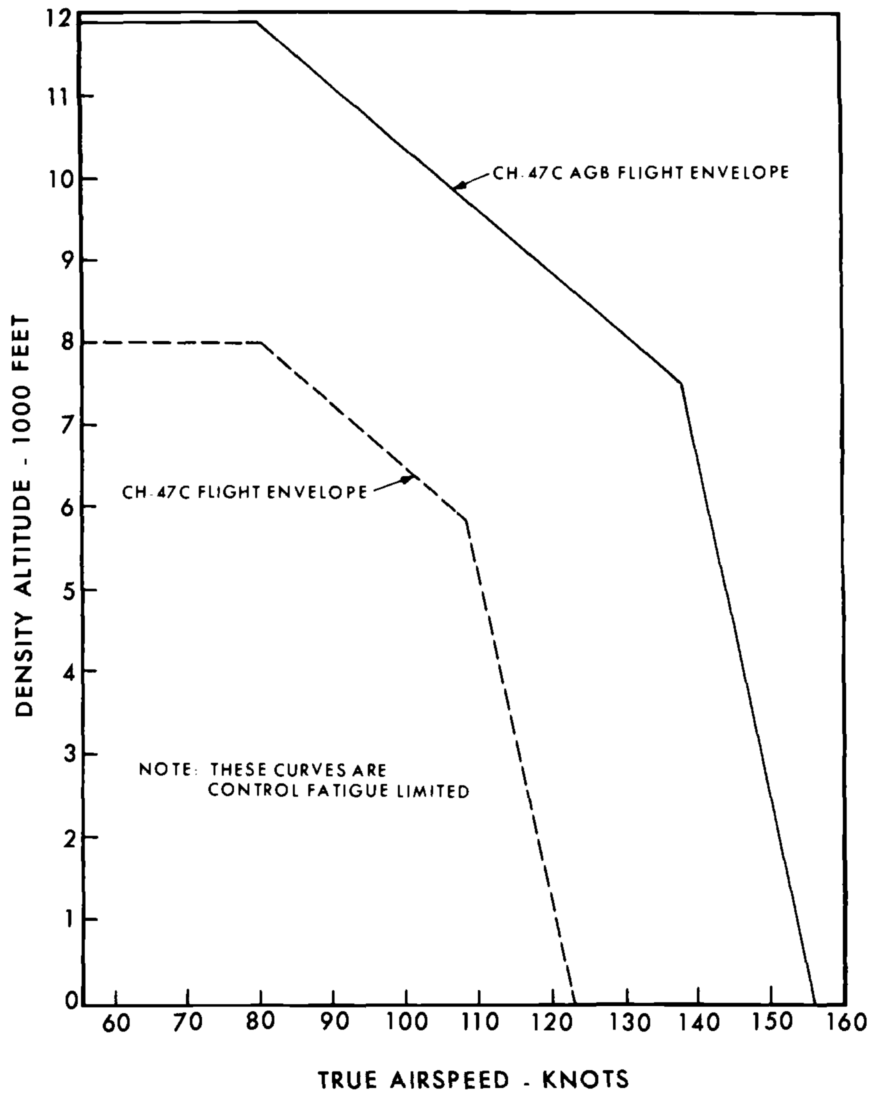


FIG. 17–*Velocity-altitude envelopes CH-47C and CH-47C with AGB (AGB 46 000 lb flight envelope evaluation 245 rpm rotor speed).*

stratum remains a problem. Automated production quality control equipment utilizing infrared, ultrasonic, and radiographic techniques are available and will isolate and define major defects. Equipment of this type was utilized throughout the blade development program. Further work remains in defining defects in sharp or re-entrant curves. In the definition process control systems and related test specimens remain a necessary and valuable tool.

Extensive use of X-ray and ultrasonic test techniques has been made on composite and bonded structures in the past, and these techniques have been further developed and applied to advanced helicopter blades. X-ray and ultrasonic records were made of the subcomponents and the completed blades. The defect limit, ¼ in. diameter, 6 in. on centers, was considered more stringent than necessary but was utilized to insure extensive defect area mapping.

The blade sections were "mapped" at the completion of fabrication and each significant test completion, that is, completion of 10×10^6 cycles at 100 percent flight load, 150 percent, etc. In addition, the flight blades were inspected before, during, and after whirl, and also before, during, and after actual flight. No increase or propagation of the noted defects could be found that were attributable to test or flight load conditions.

Application of Composites to Other Helicopter Components

The excellent dynamic and structural performance of composite materials exhibited in rotor applications has provided a sound technical base for the application of these materials to other components. Application studies have extended beyond the rotor blade and include the entire dynamic system and static structure of the aircraft, Fig. 18. An analysis of two helicopter systems, the CH-47 and CH-46 aircraft, will be found in Ref *3*.

A helicopter applications study including the rotor group, the drive system, the rotor blades (discussed previously in this chapter), and the air frame, Fig. 19, has been completed under the Ref *3* program.

The rotor group consisted of the rotor hub and the upper flight controls; the drive system was subdivided into the transmission and the drive shafting. The rotor blade was placed in a separate category since it entailed a study that was significantly larger in scope than that of the other components.

The rotor group study was essentially the study of the dynamically loaded components. In general, the components within this group are designed to a fatigue criterion, and the resulting metal hardware items are relatively heavy compared to their normal load. Typical weight savings are identified in Table 3.

The drive shaft design study featured a tube which has honeycomb stabilized walls and molded end fittings with boron fiber materials for bending strength. The end adapters were steel and were wound into the basic tube section. The weight saving per aircraft with boron applications in the drive shafting is estimated at 52 percent for the CH-46 and 47 percent for the CH-47.

The airframe study was limited to the basic load carrying elements, Fig. 19. The scope of the program did not allow for an extensive study of composite replacement structures for the shear elements, but the literature indicates that flat sheet advanced composites with stability characteristics equivalent to aluminum are also equivalent in weight per square inch. Therefore, the study centered on the payoff associated with the replacement of the beam, cap, and web stiffening elements. A method of indexing these elements was developed to account for the necessary column and crippling stability characteristics in the weight estimating procedure. A sinusoidal or bead shaped element was selected

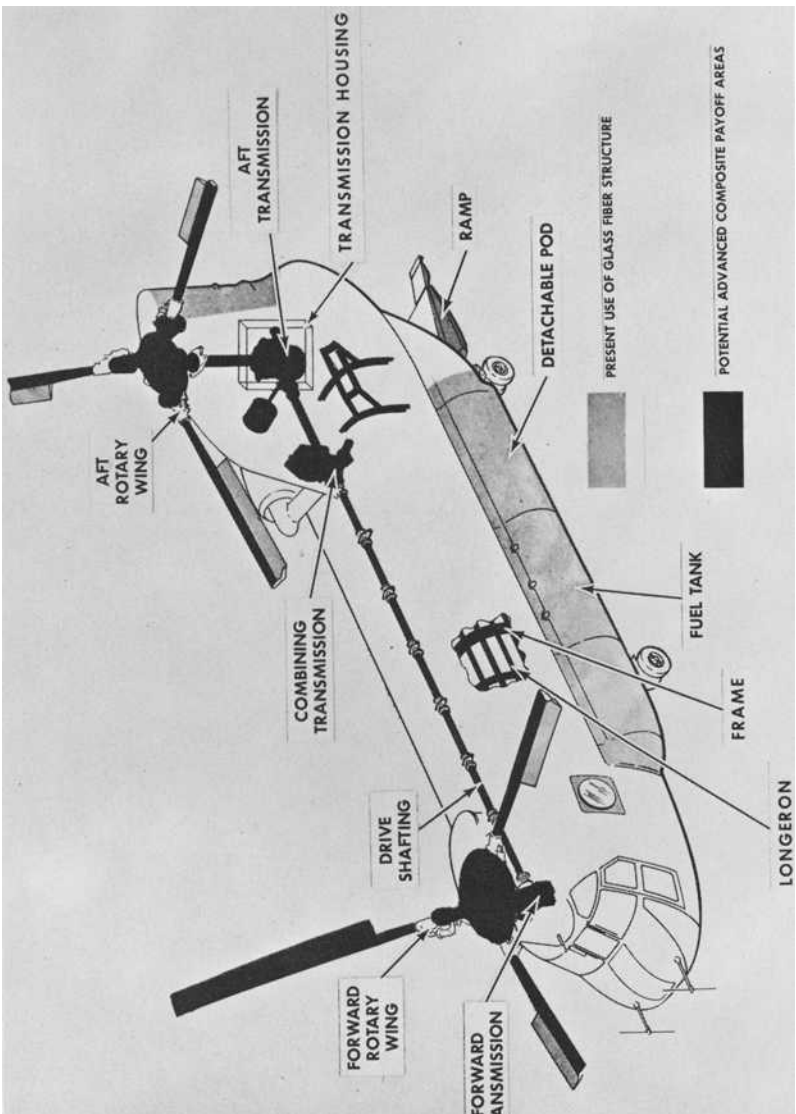


FIG. 18—*Composite materials structural applications.*

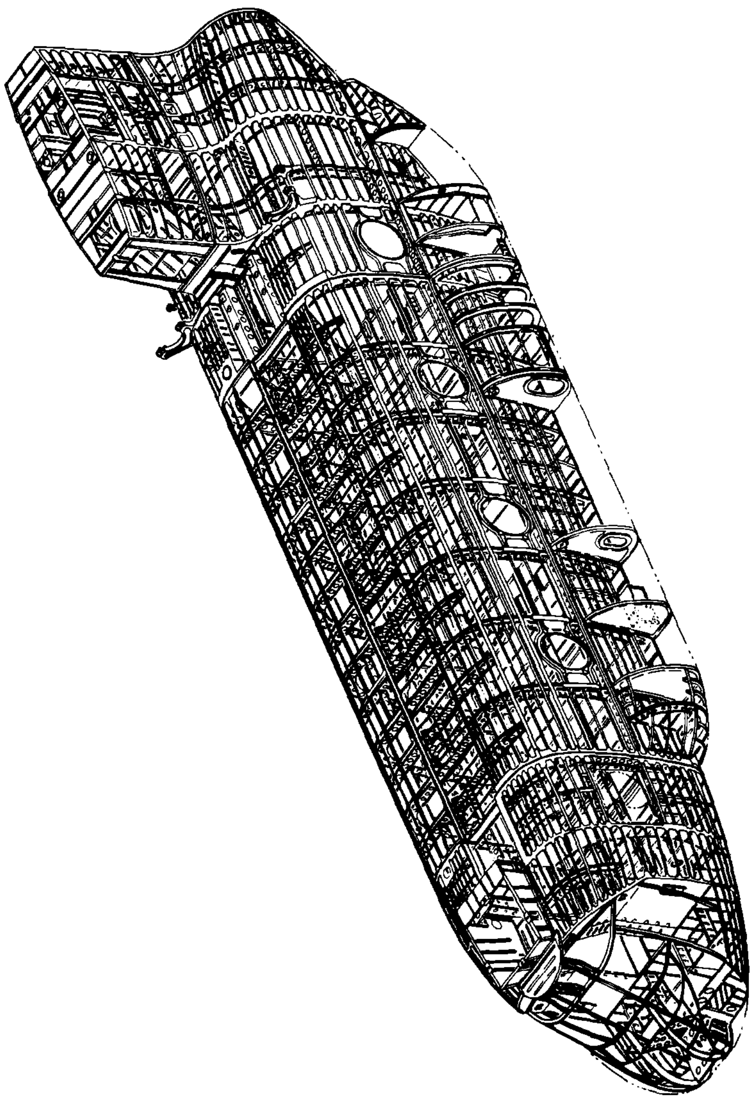


FIG. 19—*Airframe basic load carrying elements.*

TABLE 3–*Weight reduction achievable with composite materials, CH-47 helicopter.*

Nomenclature	Basic Material	Basic Weight, lb	Boron Composite New Weight, lb (F_{TU}-ksi)* 102[a]	200[a]
Rotor group:	steel	231.2	209.0	177.0
Rotor hub	aluminum	9.4	6.4	6.4
Drive arm	aluminum	2.4	1.6	1.6
Lower drive arm	aluminum	91.8	96.0	96.0
Rotating ring assembly	aluminum	75.1	79.0	79.0
Swashplate ring assembly	aluminum	409.9	392.0	360.0
	subtotal	409.9	392.0	360.0
	Δweight	...	-17.9	-49.9
	% change	...	-4.4	-12.2
Drive system transmission:				
Support assembly	magnesium	8.9	9.4	8.25
Forward transmission support	aluminum	143.9	113.9	99.0
Forward housing	magnesium	53.5	56.0	44.0
Housing assembly	magnesium	44.8	46.9	41.0
Housing assembly	magnesium	54.3	56.8	49.6
Transmission bevel gear support assembly	magnesium	8.0	8.4	7.0
Engine drive shaft assembly	aluminum	8.9	6.9	5.9
Lift bearing housing	aluminum	33.9	26.9	22.7
Aft rotor shaft	aluminum	75.6	77.0	46.1
	subtotal	431.8	402.2	324.6
	Δweight	...	-29.6	-107.2
	% change	...	-6.8	-24.8
Drive system shafting:				
Synchronizing shaft assembly	aluminum-steel	46.2	43.4	36.9
Shaft subassembly	aluminum-steel	26.2		
Synchronizing shaft assembly	aluminum-steel	23.1	17.9	15.2
Shaft subassembly	aluminum-steel	11.0	...	...
Synchronizing shaft assembly	aluminum-steel	20.0	...	...
Shaft subassembly	aluminum-steel	8.9	13.8	11.8
Fan drive shaft assembly	aluminum	1.9	1.5	1.3
	subtotal	137.7	76.6	65.2
	Δweight	...	-61.1	-72.5
	% change	...	-44.4	-52.6
Airframe:				
Structure assembly	aluminum	1766.7	1251.0	1081.0
	Δweight	...	-515.7	-685.7
	% change	...	-29.1	-38.8

[a]Potential Class A design allowables for static strength of boron-epoxy materials.

as the replacement element for the study. This approach resulted in an indicated weight saving of 38 percent in the basic structural weight of the CH-46 and CH-47.

Cost Factors

Considerable speculation exists over the future cost of high modulus fiber in tape form and, to some extent, the fabrication cost of composite components. A method was developed to treat both of the costs as variables rather than known numbers. Boron component fabrication costs were estimated at three levels: expected, optimistic, and pessimistic. With these assumed fabrication costs, raw material costs were varied to determine the break-down cost of the fiber.

The detailed analysis and results are reported in Ref *3*. Generally, results substantiated a break-even cost of boron tape materials in the range between $50 per pound if fabrication costs are high, and $400 per pound if fabrication costs are low and all mission factors favorable.

In summary, it was concluded that composite applications to the CH-47 should produce over-all system cost effectiveness if the cost of tape materials levels off at some price below $200 per pound, and if fabrication costs of the components remain at appromixately the same level as fabrication costs of more familiar metal materials.

The impact of materials costs and fabrication costs on the acceptance and utilization of composite structures is inescapable. The most thorough and careful system analysis and trade study is valueless unless it is applied to actual production hardware. Even though the results of the cost effectiveness system analysis are extremely favorable, the normal reaction of engineering management, sales personnel, and the potential customer is to resist increased acquisition costs of new equipment.

The materials selected must be capable of producing clearly superior performance. Frequently, it is not sufficient to justify an increased price on the basis of reduced maintenance costs due to increased corrosion resistance and ease of repair or improved fatigue resistance and low notch sensitivity. Other factors must be present if composite materials are to be used extensively for primary structural applications.

Composite materials engineering properties must be utilized in such a way that more conventional, and less costly, materials cannot compete. Their weight advantage is clear, but the fact that they allow the helicopter designer to work in performance regimes not possible with other materials is less easily recognized. Certain VTOL propeller designs, for instance, simply cannot be fabricated in metal. The dual speed, namely, hover and high forward speed, rotor resonant frequencies which must be avoided, make such blade designs the sole property of the composite materials designer. Here, the ability to vary EI and GJ properties with a wide range of mass distribution available is essential. Such applications will provide the impetus required for continued development and application until such time as increased production and reduced materials prices are available.

The cost of producing composite materials used in most processes available today may be broken down into raw materials, fiber, resin, scrim, weaving or collimating, and subsequent impregnation. In some cases special material forms must be added. The costs associated with component design, fabrication process selection, and the tooling and manufacturing concept utilized to produce a given blade or other component can easily outweigh the prices of today's high modulus materials. The fabrication of large sections (eliminating detail parts) and their attendant joints, multiple tools, and cure cycles are the principle factors involved. In the past, the application of composite materials in areas where other materials could do the required engineering job did not occur unless production costs were substantially lower for the composite system. This fact will not change. The cost of composite fabrication can be reduced; however, it takes the combined skills of all concerned, not just the manufacturing engineer, if these materials are to reach their full potential.

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Chapter V – Space Structures

REFERENCE: Forest, J.D. and Christian, J.L., "**Space Structures,**" *Applications of Composite Materials, ASTM STP 524*, American Society for Testing and Materials, 1973, pp. 134-162.

ABSTRACT: The use of fibrous composites for space structures offers considerable potential because of the high premium placed on weight saving. Space applications which are discussed in this chapter include a missile interstage adapter, pressure vessels, a truss supporting structure, a dish antenna, and a re-entry vehicle. Both epoxy matrix and aluminum matrix materials have been used for these applications and are included in the discussion.

KEY WORDS: composite materials, fibers, boron, aluminum, space structural forms, missiles, antennas, re-entry vehicles, trusses

The major emphasis to date in composite research and development has been directed towards aircraft applications. There are a number of good reasons for this approach, the most important being that the aircraft industry has a high production rate and utilizes very large quantities of material. Acceptance of a new material by aircraft producers is, therefore, a strong spur to high volume, low cost material production, whereas the space industry generally produces small quantities of a single product.

A number of studies made on composites for space structures generally reflects strong payoffs in terms of weight and other performance factors against the more conventional materials currently employed, and this chapter will explore some of these potential applications. The information presented on material development and the missile adapter program is largely drawn from completed research conducted by Convair under Air Force contract [*1*].[2]

Space structures include manned and unmanned spacecraft, the boosters and associated fairings and adapters which launch the spacecraft, and the re-entry vehicles which might resupply or refurbish such spacecraft. The structural configurations, environments, load intensities, and other design requirements differ radically among space applications and are generally quite different from

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[2]The italic numbers in brackets refer to the list of references appended to this paper.

aircraft requirements. To place the assessment in perspective, the following examples give the critical design conditions and their effect on composite performance as an aid to further studies.

Missile Adapter

The first large space structure fabricated from advanced composite materials was a missile adapter constructed by Convair Division of General Dynamics for the Air Force Materials Laboratory [*1,2*]. This program was aimed primarily at the development of aluminum-boron (Al-B) as a structural material and had as its major goals the following: (*a*) to develop the basic material to a usable level, (*b*) to assess the potential payoff of the material, (*c*) to develop manufacturing technology, and (*d*) to demonstrate the structural feasibility of using the composite in a realistic application.

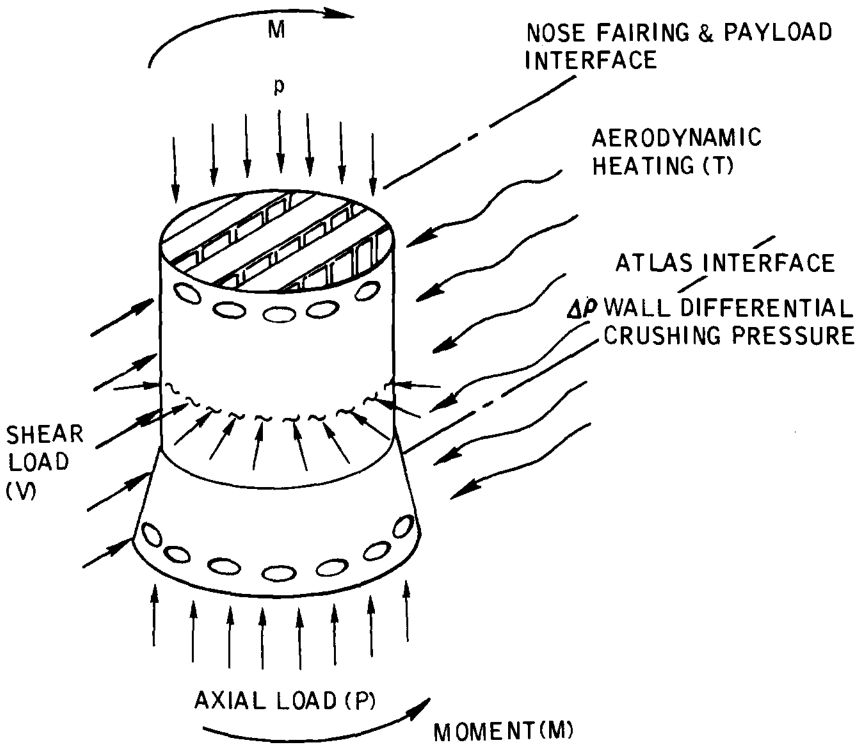
It was decided that the design, construction, and test of a major aerospace component was the most direct way to demonstrate all of the program objectives. After an initial survey of possible components, an interstage adapter, Fig. 1, was selected. Convair had built several aluminum adapters for a similar program and had conducted extensive structural analyses and tests in support of the design. The cylindrical shell portion of the adapter was complex enough in configuration and applied loading to provide a realistic demonstration article. The design of the composite adapter was required to satisfy all expected flight loads with a minimum safety factor of 1.5, maintain the original interface geometry, and incorporate joints and cutouts similar to the flight aluminum design. The program objectives were achieved with considerable success as outlined in subsequent sections.

Adapter Material

All material used in this program was prepared by diffusion bonding continuous boron filaments between foils of 6061 aluminum. The finished raw stock was in the form of flat sheet or strip with the filaments arranged in a square array in a voidless matrix (Fig. 2) without the detectable interlaminar planes characteristic of resin composites. The development and fabrication of the mill stock sheet for the program was performed by the Harvey Aluminum Company for Convair.

At the initiation of the program, material sizes were limited to a few square inches; volume fractions of reinforcement were less than 25 percent; material costs were about $6000 per pound; and property values were very erratic. At the end of the program, sheet sizes up to 4 by 4 ft were being produced with 50 volume percent (v/o) filaments and consistent properties. At the conclusion of the program material costs for sizable sheet orders were in the $500 to $700 per pound range.

Several thousand mechanical and physical property tests were conducted on aluminum-boron specimens covering 0, 0/90, and ±30-deg filament orientations.



		ADAPTER LOADS (ULTIMATE)		
		MACH 1.0	MAX σq	MAX G
CYLINDER	M	637,000 IN - LB (72,000 NEWTON METERS)	825,000 IN - LB (93,000 NEWTON METERS)	630,000 IN - LB (71,000 NEWTON METERS)
	P	26,700 LB (11,900N)	25,000 LB (11,100 N)	16,900 LB (7,500 N)
	V	7,950 LB (3,540N)	9,000 LB (4,000 N)	7,800 LB (3,500 N)
	ΔP	5.9 PSI (0.0407 MN/m^2)	2.3 PSI (0.0158 MN/m^2)	0
	T	90° F (305°K)	90° F (305°K)	+420 °F (489°K)
CONE	M	930,000 IN - LB	1,100,000 IN - LB	870,000 IN - LB
	P	16,000 LB (7,100 N)	17,000 LB (7,600 N)	12,000 LB (5,300 N)
	V	8,300 LB (3,700 N)	7,400 LB (3,300 N)	5,700 LB (2,500 N)
	ΔP	0 AVG	0 AVG	0 AVG
	T	90°F (305°K)	90°F (305°K)	+620° F (600°K)

FIG. 1—*Missile interstage adapter.*

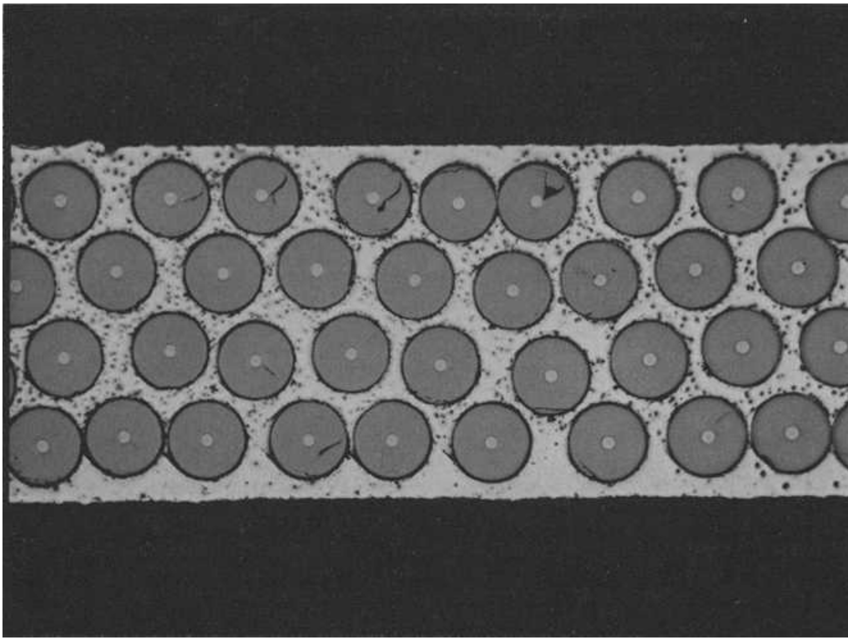


FIG. 2–*Microsection of Al-B.*

The adapter design, however, requires only 0 and 0/90-deg orientations. Design allowables used for the final component analysis are given in Table 1, while some useful physical properties are listed in Table 2. It is interesting to note that the strength property values in Table 1 are already out of date. Recent material advances have doubled the matrix shear strength and increased transverse tensile strength by a factor of three. Newer forms of coated filaments and mono-layer tapes have also eliminated many of the fabrication constraints inherent with the

TABLE 1–*Preliminary structural design allowables for boron-aluminum.*

Material	Property	Direction	Temperature Room	Temperature 700 F
Al-B, 50 v/o, unidirectional	F_{tu} (ksi)	Longitude	160	120
		Transverse	12	2
	E (msi)	Longitude	32	26
		Transverse	19	4
	endurance limit (ksi)	Longitude	90	
Al-B, 45 v/o, 0/90 deg cross-plied	F_{tu} (ksi)	Longitude	70	50
		Transverse	60	30
	E (msi)	Longitude	19	16
		Transverse	19	16
	Endurance limit	Longitude	50	

TABLE 2–*Physical properties of boron-aluminum.*

Density	0.096 lb/in.^{3}
Coefficient of thermal expansion	3.5×10^{-6} in./in./F
Resistivity	1 micro-ohm-cm
Thermal conductivity	0.25 cal/cm^2/cm/s/C

wrought sheet form. References *1* and *3* through *9* give a more detailed description of material and fabrication development of the aluminum-boron (Al-B) composite material system.

Fabrication Development

A program was performed to develop and evaluate acceptable means for fabricating structural hardware from Al-B composite material. The fabrication development investigation included various techniques for machining, cutting, forming, and joining of the Al-B material. Table 3 summarizes the methods that have been demonstrated to be acceptable. These methods have already been used to fabricate a number of structural elements and subcomponents, and are presently being used to manufacture aerospace hardware.

TABLE 3–*Demonstrated successful fabrication methods for Al-B composite materials.*

Machining and Cutting	Forming	Joining
Shearing and punching (thin sheet material only)	roll forming (room or elevated temperature)	resistance spot and seam welding
Abrasive cut-off	brake bend forming (at elevated temperature or in compression fixture)	riveting
Grinding		bolting
Electrical discharge machining (EDM)		brazing
Drilling, routing, etc. (with diamond tipped tools)		adhesive bonding

A large number of methods and techniques for machining and cutting of Al-B composite material was evaluated. Although some were completely unacceptable (for example, high-speed steel drilling and band sawing), several methods were found to be acceptable under certain conditions. These methods and their limitations are summarized in Table 4.

Techniques have been developed for successfully roll forming and bending Al-B composite sheet material. Both unidirectional and cross-ply materials were roll formed at room temperature to as small as 8-in. radii without any damage to the composite material. By use of elevated temperatures and compression

TABLE 4 – *Visual comparison of machining processes on 0.02-in. thick 50 v/o Al-B (ranked in order of least damage first).*

Rating	Process	Observation	Remarks
1	Electrodischarge machining	Almost complete absence of filament damage. Caused least damage to filaments of all the processes tried	Slow feed rate used on machine
2	Electrolytic grinding	Small amount of filament damage consisting of chipped filament ends	Slow feed rate
3	Shearing	Practically no crushing of filament ends observed. Uneven profile of cut filaments when compared with above processes	Damage to filaments increases rapidly with increase in sheet thickness
4	Abrasive cutoff	Slight chipping of filament ends	Minimum damage, fast and acceptable process
5	Grinding	Some filament edge crumbling and broken filaments	Acceptable process
6	Diamond routing	Broken and chipped filaments	Acceptable process, but needs further development
7	Punching	Crushed and chipped filaments	Acceptable process for sheet material, but wears punch rapidly
8	Diamond drilling	Broken and chipped filaments, uneven filament profile	Acceptable process, but needs further development

fixtures or both, the Al-B sheet material was successfully bent to a 4.5 t-bend radius. Examples of parts formed from Al-B composite material are shown in Fig. 3.

The joining of Al-B composite material, both to itself and to other structural materials, has been most successful. One of the most promising methods is resistance welding (spot and roll seam) which results in very high joint strengths, since full benefit of the filaments is realized. Typical strengths of individual resistance spot welds are shown as a function of nugget diameter in Fig. 4. Other successful joining methods include mechanical joints, brazing, and adhesive bonding; typical joint strengths are given in Fig. 5. A comprehensive presentation of joining of Al-B composite material can be found in Ref *10*.

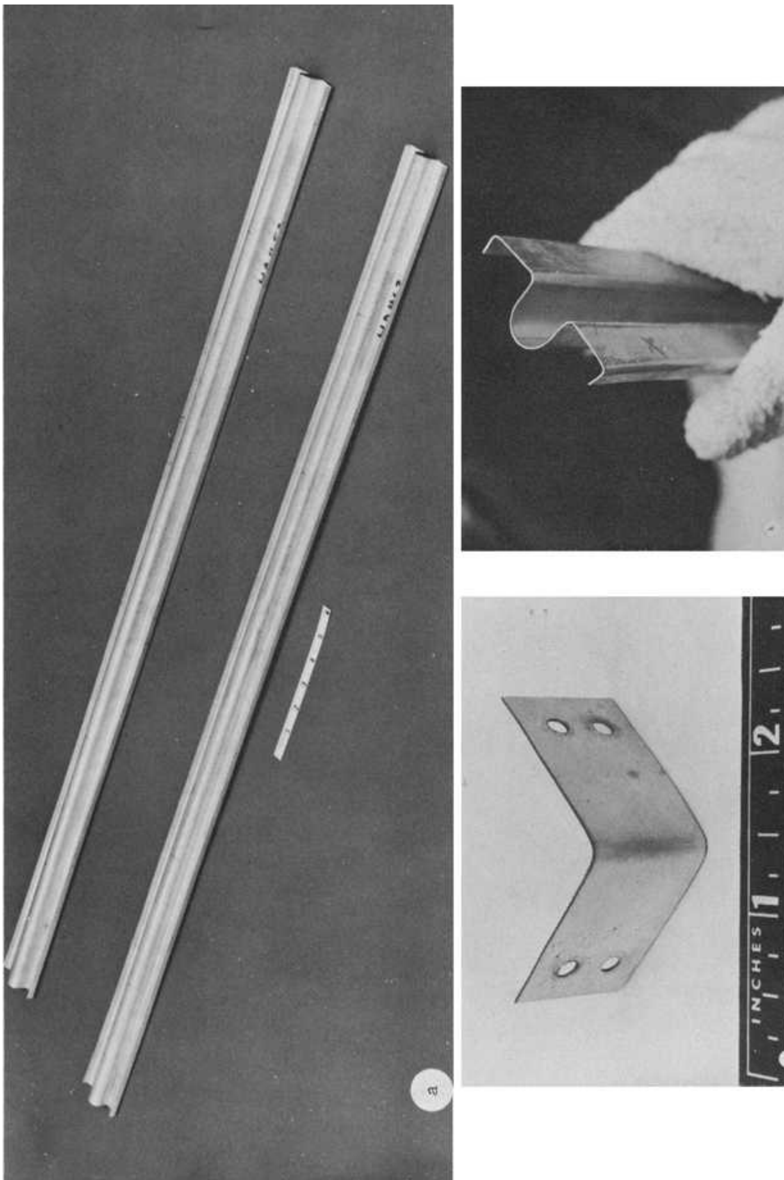


FIG. 3—*Examples of formed parts from Al-B composites: (a) 42-in. long stringers with selectively placed boron, (b) stringer detail, and (c) 45 v/o 0/90-deg cross-ply formed to 4½ t-bend radius.*

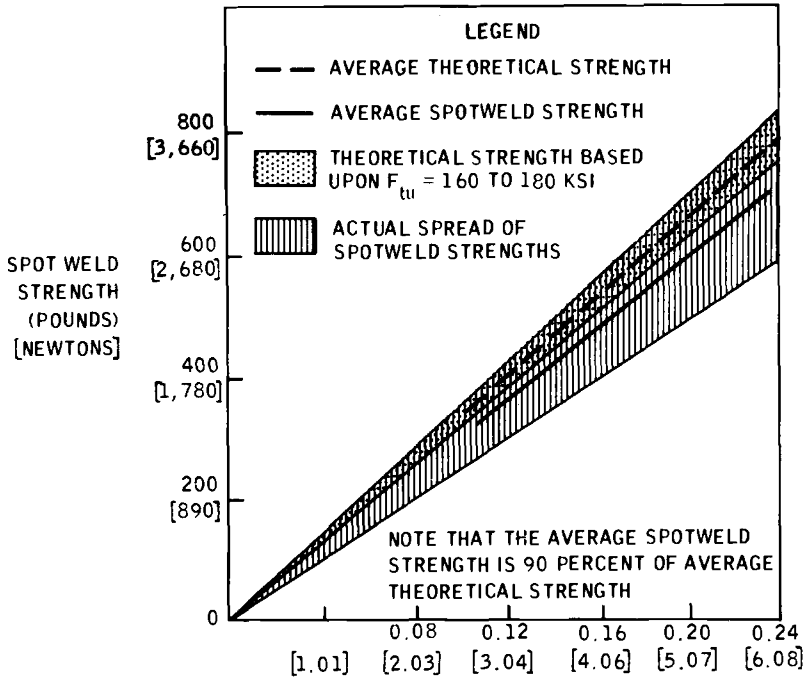


FIG. 4—*Resistance spotweld strength (0.020-in., 50 v/o Al-B composite).*

Adapter Design

All the advanced composites are very expensive to procure and fabricate, so it is imperative that an efficient structural system be used to provide the highest potential cost effectiveness. In the case of the adapter, several types of structures were studied to determine the lightest construction. The most efficient was a biaxially stiffened thin sheet structure designed to allow skin buckling under the applied loads. In this construction, shear and pressure loads are carried by the skin as a buckled membrane. Axial loads are reacted by the stable biaxial stiffeners (stringers and frames). Comparative weights of various concepts are shown in Table 5. The sandwich was heavier than the monocoque version for the short adapter cylinder. As the length of the cylinder was increased, however, the sandwich construction became more efficient, but still did not approach the buckled skin design in Fig. 6.

The design selected for further development was the stiffened sheet concept using component configurations similar to those employed in conventional sheet aluminum structures. The 40 longitudinal stiffeners were a 4-ply unidirectional layup using selectively reinforced, diffusion bonded mill stock containing 50 v/o boron reinforcement. The unique selective reinforcement concept was intended to allow easy forming of the mill stock to tight bend radii. The generous radius

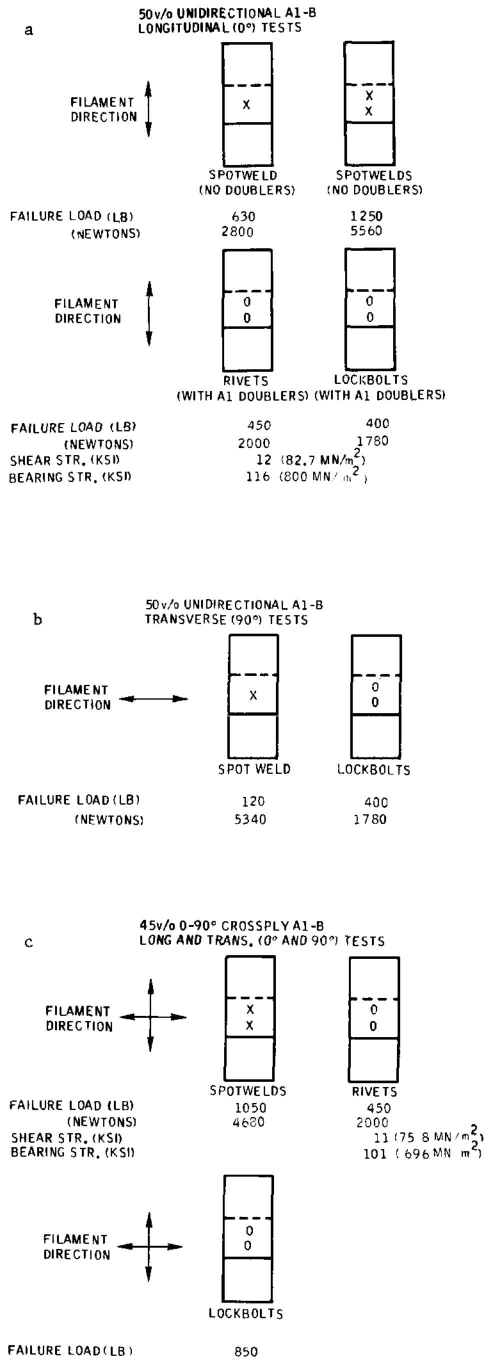


FIG. 5—*Typical strength properties of Al-B composite joints: (*a*) longitudinal, (*b*) transverse, and (*c*) cross-ply.*

TABLE 5–*Concept comparison.*

Structural System	Shell Weight, lb	Saving, %
Existing 2024 Al monocoque	121.0	0
Al-B sheet-stringer	67.3	45
Al-B monocoque	85.6	30
7075-T6 Al sheet stringer	100.6	17
Boron-epoxy sandwich	104.4	14
Boron-epoxy monocoque	93.2	23
Fiberglass sandwich	113.8	6

at the cap of the stiffener section was fully reinforced, however, and would require an 800 F forming operation.

The circumferential frames were also designed for selectively reinforced Al-B.

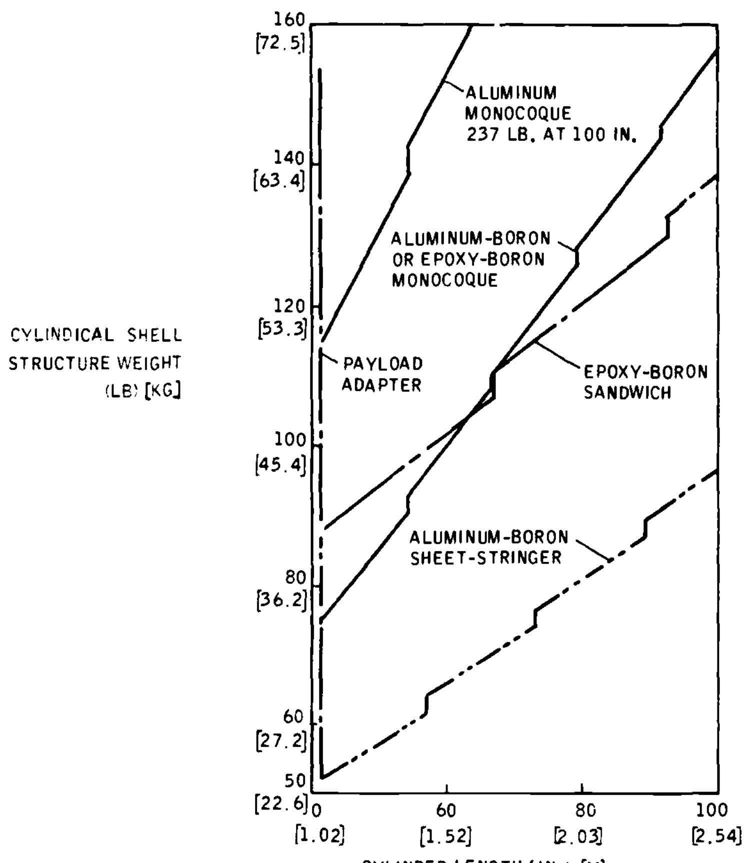


FIG. 6–*Cylinder weight distribution.*

The skin material was 50 v/o Al-B (4-plies) in a unidirectional layup. A variety of joining techniques were proposed including dip brazing, resistance spot welding, and mechanical fasteners.

The selected design was based on certain assumptions of structural behavior which are difficult to predict from theoretical considerations. The philosophy of stiffened shell construction, for instance, requires a material with significant post buckling shear strength. Also, since the shell is loaded by a substantial external pressure, the skin material would be required to snap through to reverse curvature and carry pressure as a tension membrane. The resin composites generally showed limited post buckling strength and exhibited a tendency for catastrophic brittle fracture under buckling loads. It was felt that the stronger, more ductile aluminum matrix would allow the required degree of elastic buckling and a subcomponent test program was initiated to substantiate the basic design assumptions.

Adapter Test Program

Only a brief summary of the test program is included here. The major tests and their impact on the final design are described in a rough chronological order.

Four panels, about 10 by 12 in., were tested as supported plates in an acoustical test chamber to an overall level of 152 dB, the design environment of the adapter. Two panels were a unidirectional layup and two were a 0/90-deg cross-ply. The unidirectional panels split along the support edges after about five minutes exposure. The cross-ply panels survived two exposures of 10 min each without damage.

Both unidirectional and cross-ply panels, 5 by 12 in, were tested under 6 psi crushing pressure as curved membranes to simulate the snap through skin condition. The unidirectional panel showed permanent set and severe yielding along the supported edges. The cross-ply panel behaved elastically with no evidence of filament breakage or delamination. The acoustical and pressure test behavior led to the selection of a 0/90-deg cross-ply skin for the final design. The skin filaments were oriented along the natural axes of the cylinder, rather than at ± 45 deg, for two reasons. It was felt that the 0/90-deg orientation would have more elongation during shear buckling than the stiffer ±45-deg layup and reduce the chances of fracturing the brittle boron filaments. In addition, the 0/90-deg layup provides greater stiffness along the stiffener axes, increasing the effective skin width for axial loads after buckling.

A tension field beam (Fig. 7) was constructed with a 0/90-deg Al-B web and aluminum caps and stiffeners. Buckling of the square panel occurred at an applied shear stress of 4000 psi. The beam was cycled several times to limit load, and behavior was elastic. As expected, ultimate failure occurred in web tension at an apparent shear stress of 14 000 psi. Easily visible buckles occurred, but X-ray inspection indicated no filament breakage except at the corner tension failures. The failure occurred at the predicted stress level in the manner prescribed by conventional tension field analysis. Significant post buckling shear

FIG. 7–*Tension field beam.*

strength is shown by the $\tau/\tau_{\text{critical}}$ ratio of 3.6 noted in this test.

Stiffener crippling tests were conducted next, beginning with flat elements of varying width-to-thickness ratio and progressing to formed hat sections of the desired adapter design. Theoretical methods for predicting initial buckling of orthotropic plates and an empirical method of predicting ultimate crippling strength were verified. Ultimate section crippling stresses of 86 ksi were obtained, about 2½ times the value of a comparable aluminum section.

The longitudinal stiffeners in the adapter are critical as beam columns. The beam loading from pressure and diagonal tension effects is the most severe type of load. Two beam specimens, shown in Fig. 8, were tested. The longer specimen failed on the compression side at 105 percent of ultimate load (169 ksi stress). The short specimen failed in horizontal shear at about 9000 psi. The low shear strength was attributed to delaminations present in the mill stock from which this section was formed as evidenced by ultrasonic inspection prior to testing.

Two column tests were also made using specimens similar to the stiffener

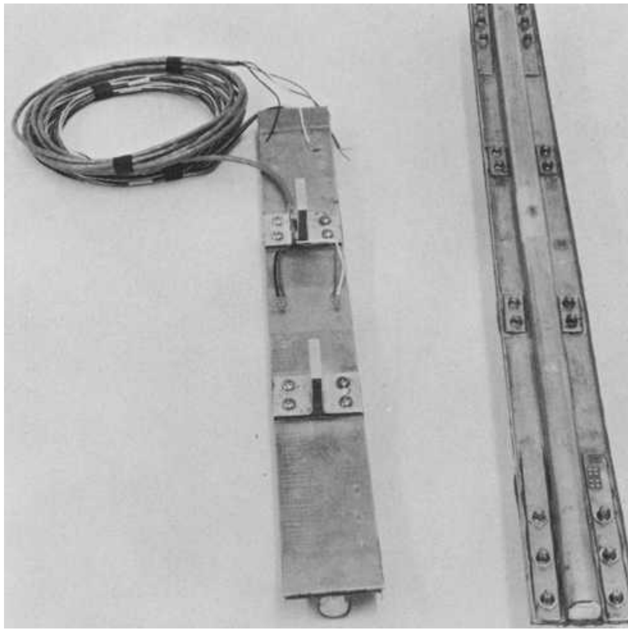


FIG. 8–*Stringer test beams.*

beams. Failing loads were well beyond ultimate flight levels but were lower than predicted column strength. Both specimens failed by pulling local spot welds to the skin in cross tension, which initiated local skin instability failures. Examination of the failures indicated improper welding (stick welds), a recurring problem in final adapter fabrication.

An extensive joint testing program to determine design allowables and failure modes was conducted. Some of the specimens are shown in Fig. 9. Some of the more interesting results were the high spot weld strengths (4 to 6 times equivalent aluminum) and the improvement in rivet strength obtained by using a thin aluminum shear doubler. On 20-mil Al-B, a 16-mil aluminum doubler will develop ultimate shear in a 1/8-aluminum rivet at room temperature and an edge distance of two diameters.

Several changes between the final design and the initial design were made as a result of the test program and a concurrent fabrication development program. Some minor changes were also made as a result of economic trade offs on the use of composites versus conventional materials. The substitution of 0/90-deg cross-ply skin for unidirectional skin has already been described. The other major change was the substitution of aluminum intermediate frames for the composite frames originally planned. The fabrication study indicated that simple forming of the composite frames was not feasible and would require an

FIG. 9–*Joint test specimens.*

expensive buildup by other techniques. This expense was not justified to achieve the small total weight saving possible in the adapter design. The small load introduction doublers at each end of the stringers were also revised from Al-B to aluminum for economic reasons. This required a change from dip brazing to adhesive bonding for joining the doublers to the skin. The mismatch in coefficient of thermal expansion of aluminum and Al-B produces severe warpage at the 1000 F brazing temperature, but not at the lower adhesive bonding temperature.

The final design, illustrated in Fig. 10, results in a 40 percent weight saving over the original adapter design.

The composite adapter (Fig. 11) was tested under maximum combined loads of axial compression, bending, and external pressure at room temperature. All loads were applied and increased simultaneously in increments of 10 percent of limit load. Shear, from the applied bending moment, caused noticeable skin wrinkling to commence at 40 percent of limit load. Audible snap through to reverse curvature from external pressure occurred at limit load (1.5 psi) and continued as load increased. Ultimate load (150 percent limit load) was reached

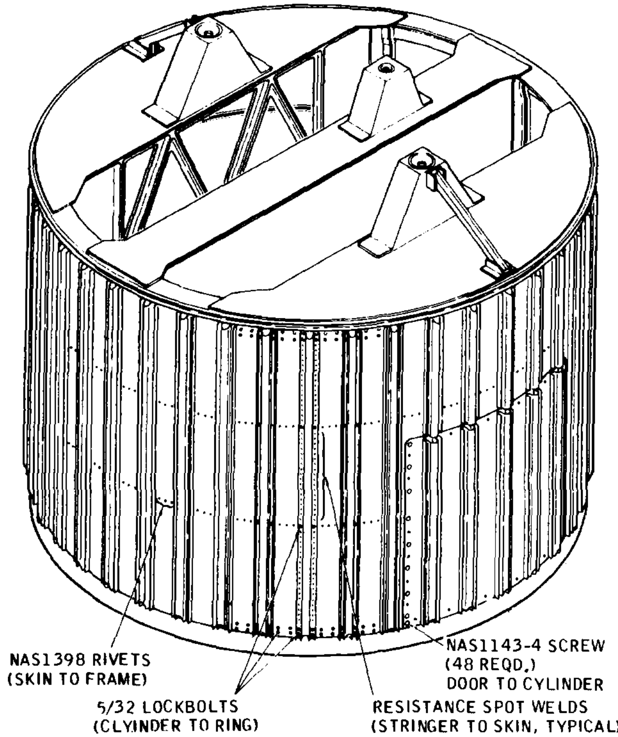


FIG. 10—*Final design of missile adapter.*

without failure and loads removed. No permanent wrinkling of the skins was observed, although some flattening of the initially curved skins did seem to occur in local areas. Loads were again applied simultaneously to final failure, which occurred at 133 percent of ultimate. The failure was a local instability collapse of three Al-B stringers on the compression side of the adapter near the base ring. The instability seemed to occur following loss of a weld attachment just above the ring in the same fashion as in the stringer column tests. No other failures were noted elsewhere in the structure.

The successful completion and test of the adapter represents the first major application of metal matrix composites to aerospace structures. The shear buckled plate stringer concept was proven feasible, and the high payoff possible in weight reduction was demonstrated. The test verified the concepts of selective reinforcement, a variety of joining techniques, and the design and analysis techniques that were employed. Much practical experience in fabrication techniques was obtained, and potential problems were identified for future study. The structural merit and high potential for aluminum-boron in aerospace structure will make this material a mandatory candidate for future designs.

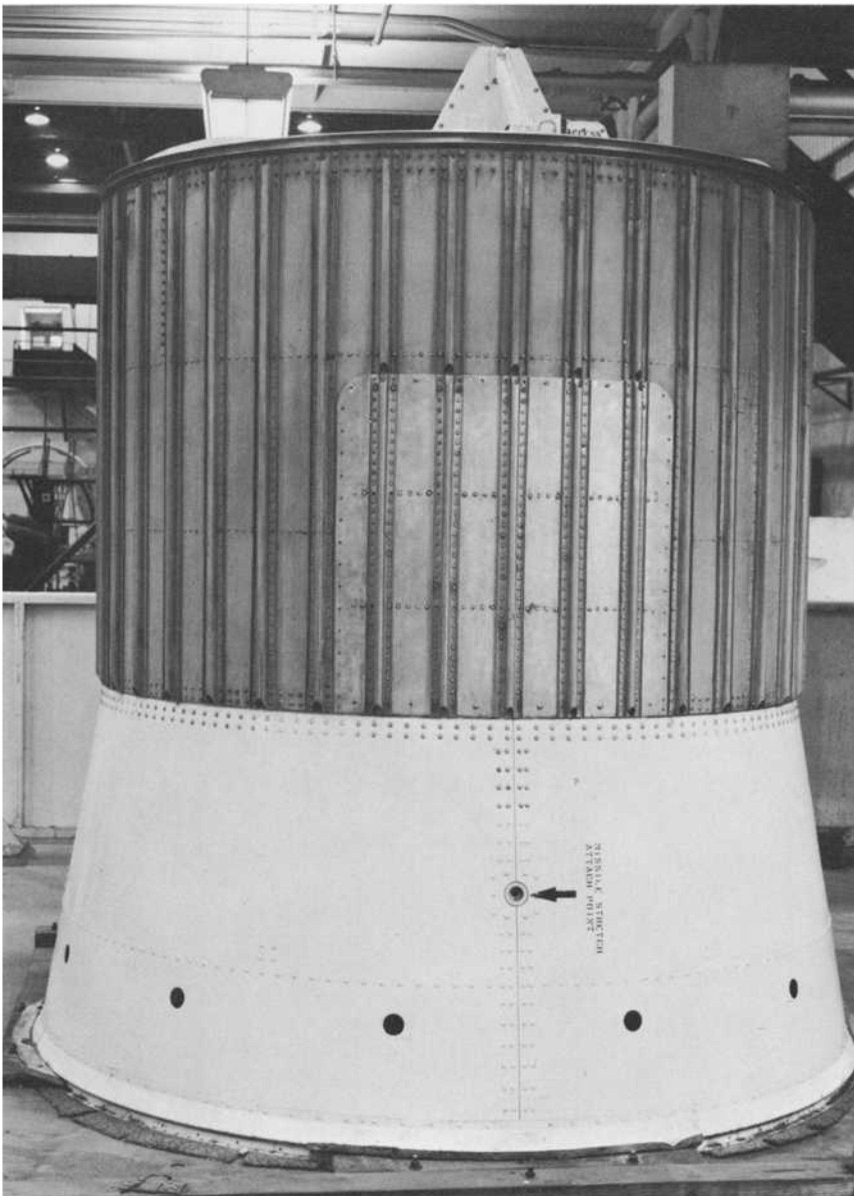


FIG. 11–*Completed adapter.*

Pressure Vessels

Pressure vessels are a common structural form for space systems. One of the more unusual applications of this type is the Atlas booster propellant tankage which also forms the body structure of the missile. The Atlas is a balloon type pressure stabilized structure composed of very thin, welded 301 type stainless steel. No stiffeners to react axial loads are used. Instead, the bulk of flight axial and bending loads are reacted by internal pressurization.

One advanced version of this missile, however, is unusually long and heavily loaded. Consequently, axial compression stresses develop in the tank skins which require gage increases in the skins to prevent buckling. A comparative study of 220 ksi ultimate strength 301 steel and aluminum-boron composite material was performed [*11*] with the results shown in Fig. 12.

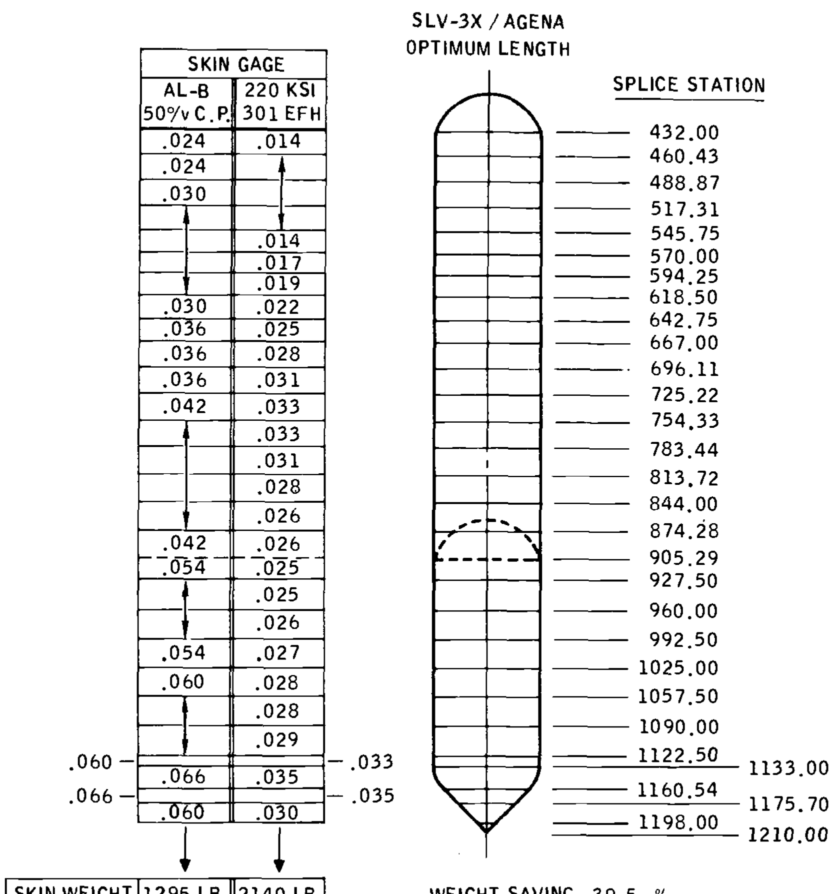


FIG. 12—*Atlas SLV-3X.*

Aluminum-boron is an obvious choice for this application rather than resin matrix materials because of the requirement for pressure tight fuel sealing of both liquid oxygen and RP-1 at pressures up to 90 psi. The resistance welding method of spot and seam joining, currently used with Atlas steel skins, can also be used with Al-B to achieve pressure tight joints. In addition, the tank skins are required to function over the temperature range of about -320 F to +800 F during flight. Current resin systems do not perform well at these extremes.

The results indicate several advantages of Al-B composite for this type of application including:

1. A potential weight saving of 39.5 percent.

2. The heavier gages provide a self-standing tank capability in the unpressurized, unfueled condition. This capability prevents the collapse of the booster if pressure is lost prior to fueling.

3. Potentially less damage during handling and transportation because of the thicker skin gages.

4. Additional weight savings are possible since many pod and auxiliary equipment supports can be aluminum rather than steel. Existing bracketry must be steel for compatability with the booster skins.

Another type of space pressure vessel is shown in Fig. 13 which is a sketch of a space station module concept studied for the NASA Manned Spacecraft Center. The pressure contained here is the oxygen-nitrogen mixture required to support the astronauts manning the module. Although the pressure is low, perhaps 10 psia ultimate, absolutely leak free joints must be maintained continuously for a very long time, perhaps 5 or 10 years.

Composites were not seriously considered for the pressure skins of this module. The reliability of a new material is too questionable for an application of this type. Considerable weight payoff is possible, however, by using

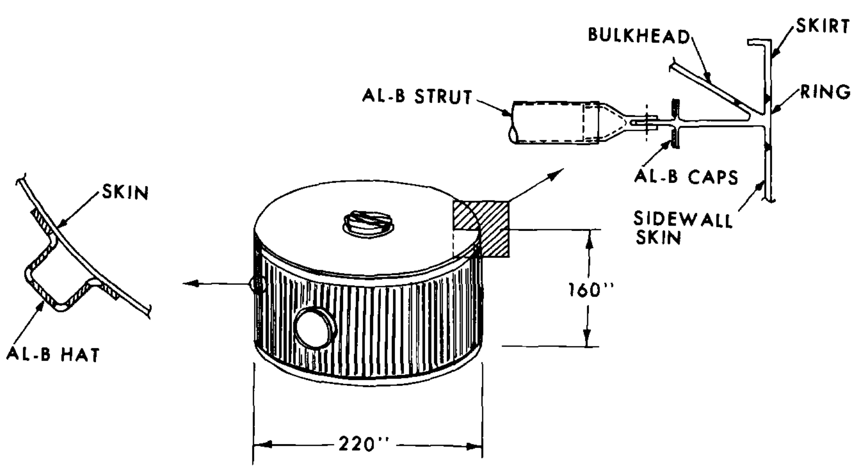


FIG. 13–*Basic subsystem module (BSM).*

composite stiffeners and rings on the basic welded aluminum pressure shell [12]. It has been shown than an overall structural weight saving of 25 percent is possible by using aluminum-boron for the longitudinal stiffeners and the two supporting compression rings at the base of each end pressure dome. The metal matrix composite is desirable here, because it can be used to produce very efficient, unidirectional stiffening members and can be spot welded to the aluminum pressure skins.

Another type of application involving pressurized structures are small gas storage tanks used for attitude control and auxiliary propulsion systems. The pressures in this kind of tankage typically run from 100 psi to several thousand psi.

For tankage requiring relatively high pressure containment (pressure x tank radius = pR = 3000 psi or higher), filament wound epoxy-boron over a thin metallic shell liner can show about a 20 percent weight saving over conventional metals. Tankage for pR values below about 2200 psi is in the minimum gage region (0.030-in. aluminum, 0.025-in. titanium). In these low pressure applications, Al-B appears quite attractive, showing 25 to 35 percent weight savings over 2219 aluminum or 6Al-4V titanium. A typical low pressure tank design (500 psi burst) in Al-B is shown in Fig. 14. Note that no liners are required if welded or diffusion bonded joining is used.

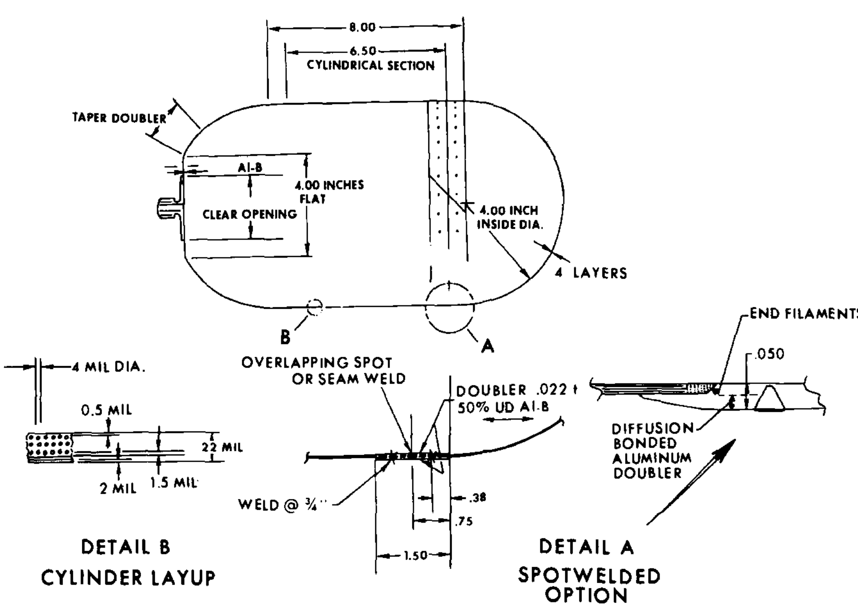


FIG. 14–*Small pressure tankage.*

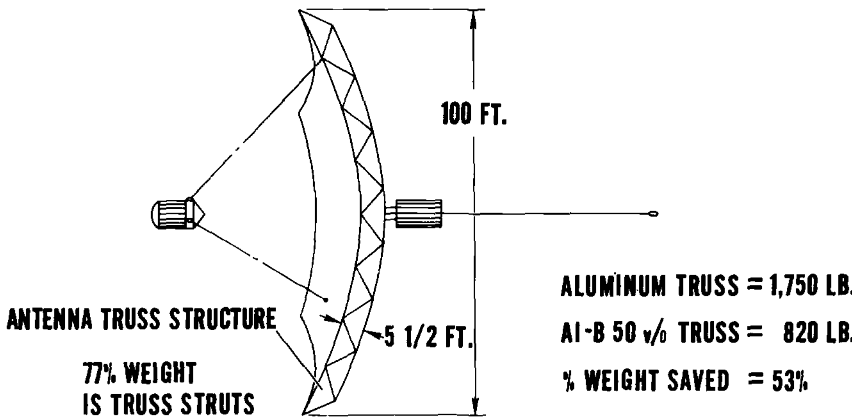


FIG. 15–*Large truss structure.*

Space Antenna Truss Structure

Serious consideration is being given by NASA to the use of parabolic antennas of up to 100-ft in diameter for space communications missions. One such structure now under development utilizes a three-dimensional erectable tubular truss structure [*13, 14*] shown in Fig. 15. Materials currently considered for the tubular members are aluminum and titanium. The aluminum is a little lighter than the titanium, but produces greater thermal distortions (which can seriously affect the *RF* gain of the antenna) because of its larger coefficient of thermal expansion. As shown in Table 6, the three advanced composites have even lower thermal expansion, and therefore distortion, than titanium.

The tubular truss members for this 100-ft structure are very long, pin-ended columns with an l/r (length/radius) requirement of about 100 in aluminum. The stiffer composites can use much larger l/r values to support the same loads. The composite tubes however, require very thin wall gages (0.010 to 0.015 in.) for

TABLE 6–*Coefficients of thermal expansion for candidate structural materials.*

Material	Coefficient of Linear Thermal Expansion Near 68 F (20 C)	
	μin./in./deg C	μin./in./deg F
Aluminum	23.6	13.1
Titanium	8.4	4.7
Stainless steel	15.3	8.5
Al-B composite	5.8	3.2
Boron-epoxy composite	4.7	2.6
Graphite-epoxy composite	-0.73	-0.40

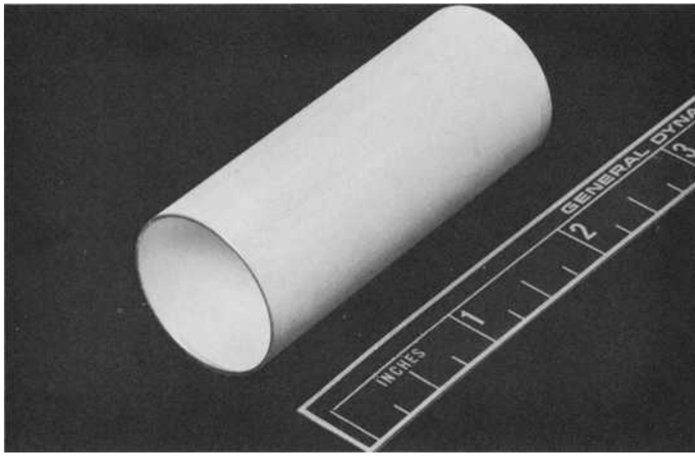


FIG. 16–*Al-B composite tubing.*

efficiency. This allows the use of only two or three plies of either boron or the conventional graphite materials. Resin based systems are not suitable for such thin-gage tubing because of their very low transverse and shear strength. Unidirectional A1-B, however, can be used in these gages.

In the 100-ft truss design, a substitution of Al-B for aluminum tubing produces a truss of equal stiffness and strength with over a 50 percent weight saving. The manufacture of such tubing has been demonstrated on a laboratory scale [*15*]. A typical tube of 1-in. diameter in shown in Fig. 16.

Mariner IV Antenna

Minimum gage considerations play an important part in materials consideration for a different type of antenna, Fig. 17. This small, aluminum sandwich antenna was developed by the Jet Propulsion Laboratory [*16*] for the Mariner IV spacecraft. The flight reflector was bonded from 0.004-in. aluminum skins and 1.7 lb/ft^3 density aluminum honeycomb core. The basic design requirements were a structural stiffness of at least 10 Hz for launch and a good contour tolerance. The antenna surface is a compound contoured segment of a paraboloid.

Surprisingly, two composites show considerable merit for this type application. One is a graphite laminate using a heat cured epoxy matrix, the other a boron-film laminate (not fibrous) deposited on polyimide sheets. Graphite tape is available from several domestic suppliers. Four plies of this material yield a laminate as stiff as a 4-mil aluminum skin at 46 percent less weight. The resulting reflector is 20 percent lighter and has about one tenth the thermal distortion of the aluminum version used on Mariner.

The basic boron-film material is available in ply thicknesses of 1/2 mil with

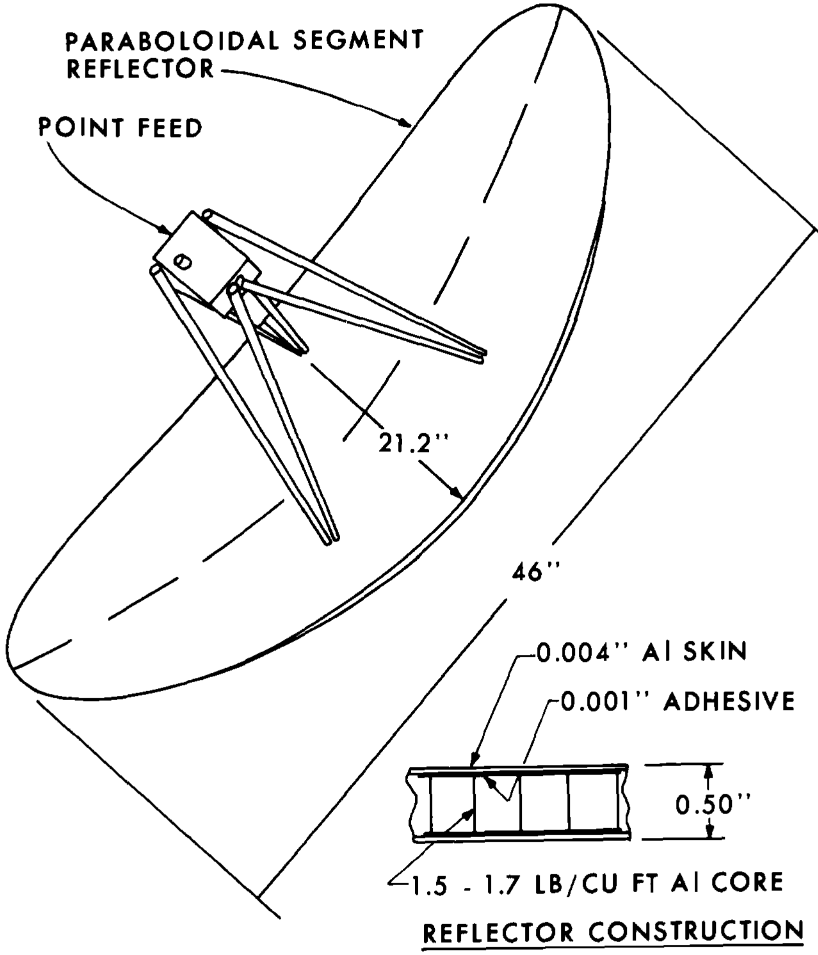


FIG. 17–*Mariner IV high-grain antenna.*

TABLE 7–*Properties of boron/PI film laminates.*[a]

Tensile strength	55 ksi (ult)
Tensile modulus	23×10^6 psi
Failure strain	3100 in./in.
Proportional limit	24 ksi
Density	0.064 lb/in.^{3}
Boron volume	44%
Approximate coefficient of thermal expansion	3×10^{-6} in./in. F
Film thickness	0.0005 in.
Anisotrophy	isotropic in-plane

[a]Data supplied by the Norton Research Corp., Cambridge, Mass.

typical properties shown in Table 7. A skin thickness of only 2 mils is required with this material to duplicate the stiffness of the aluminum skin because of its high elastic modulus. The skin weight is about 65 percent less than the aluminum skin, giving a 30 percent reduction in reflector weight. The thermal distortion is also low, about one third of the aluminum design.

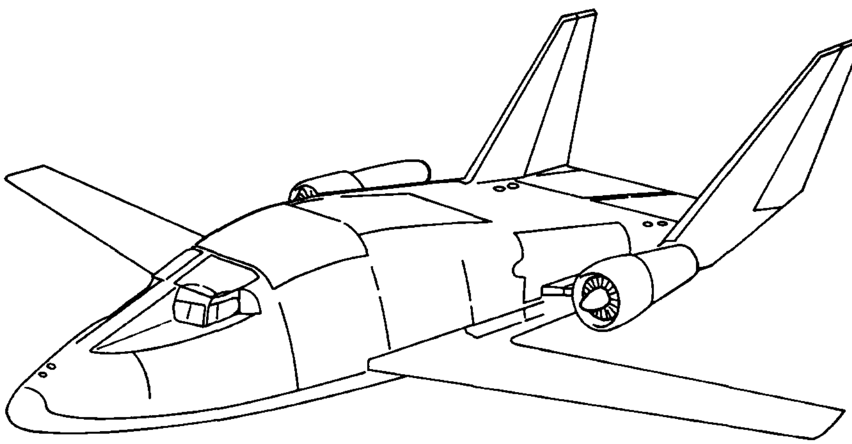


FIG. 18–*Manned re-entry vehicle.*

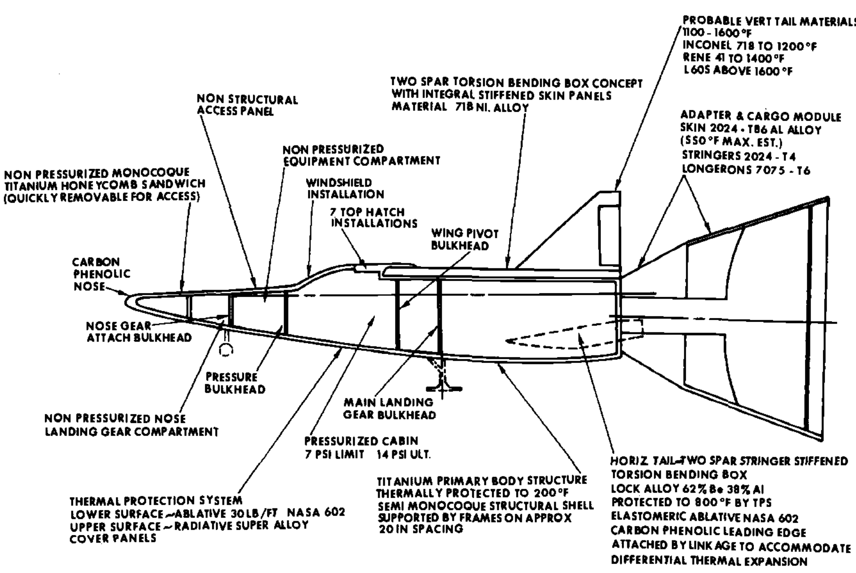


FIG. 19–*Variable geometry spacecraft materials, gliding entry vehicle.*

Re-entry Vehicle Applications

Studies have been performed [*17*] for Aerospace and NASA on variable geometry entry vehicles of the Cruise Spacecraft type shown in Fig. 18. A recent study for NASA-MSFC covers this type vehicle as well as a "triamese" type recoverable booster. The Cruise Spacecraft is boosted to orbit by a Titan class missile and then re-enters the atmosphere with the capability of powered atmospheric flight and landing. A more typical entry vehicle which does not have powered flight capability is shown in Fig. 19.

A general summary of some of the structural characteristics of the three vehicles are given in Table 8. Note that the powered entry configuration is very similar to the glide configuration.

A view of the body structure of the Cruise Spacecraft is shown in Fig. 20. Note the similarity to conventional airframe structure with main bulkheads, frames, longerons, skin panels, and access openings. The general studies of the F-106 and F-111 [*3*] apply equally well to this structure. However, an additional analysis of the frames aft of the canopy were made to check the payoff of A1-B. The results show a 42 percent weight saving as detailed in the following paragraphs. The general payoffs are 25 to 35 percent overall, with 30 to 60 percent in many components.

TABLE 8—*Structural characteristics of three entry vehicles.*

Powered Cruise Spacecraft	Gliding Re-entry (FDL)	Triamese Recovery Booster
General configuration similar to an airplane	Similar to an airplane	Similar to a booster, large tankage areas
Body L/D low	Body L/D medium	Body L/D large
Swing wings	May be wingless	Wingless
Swing engine(s)	No engines	No engines
Retractable gear	Retractable gear	Retractable gear
Extensive external insulation	Extensive external insulation	Little or no external insulation
Probable materials: Titanium body Nickel 718 wing Rene 41 vertical stabilizer Lockalloy horizontal stabilizer Al cargo adapter	Probable materials: Same as powered version	Probable materials: Main tankage probably Ni 718 Some structure titanium in front end
Structure about 30% of total weight	Structure about 40% of total weight	Structure about 15 to 20% of wet weight
Body temperature 200 F	Body temperature 200 to 400 F	Body temperature 600 to 800 F
Control surface temperature 800 to 1600 F	Control surface temperature 800 to 1600 F	Control surface temperature 800 to 1600 F

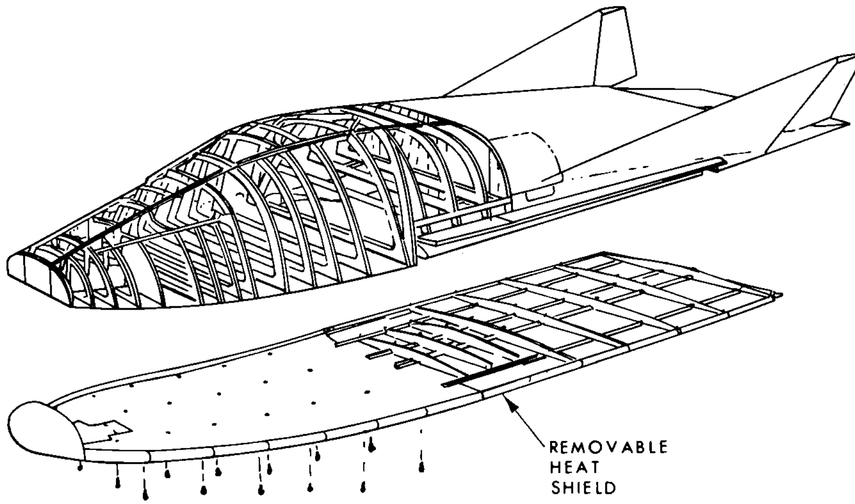


FIG. 20–*Structural arrangement, cruise spacecraft.*

One section chosen for analysis was the two bays immediately behind the windshield. This section spans the two pilot escape hatches which connect to and are divided by a centerline or hatch longeron (Fig. 21). The hatches are also latched down to the main longerons. The forward and aft frames are continuous, while the center frame is cut in four places, which for analysis purposes are considered pin jointed. The hatch and main longerons contribute to the center frame's stiffness, Station 230, by virtue of their own stiffeners and their ability to transfer loading to the adjacent frames. All frames are restricted to a maximum height of 2 in. at the top and 4 in. at the bottom, with a uniform transition between these two dimensions at the sides. Deflection limitations were established to ensure a reliable escape hatch operation and to reduce interactions between the main shell and the ablation and heat shield structure.

Advantages of the Al-B composite material are its high specific strength and density. The modulus and strength are comparable to steel, while the density is closer to that of aluminum when the fibers are unidirectional and are 50 percent of the total composite by volume. The disadvantages of the composite material lie in the limited design and fabrication experience, high cost, and highly anisotropic characteristics. It was decided to take a conservative approach to its employment in the frame designs to provide realistic weight savings for the minimum complexity and cost. A heterogeneous frame structure was decided upon where the primary frame structure was aluminum, and the composite material was bonded to the aluminum frame caps in the form of strips. The resulting frame sections are shown in Fig. 22.

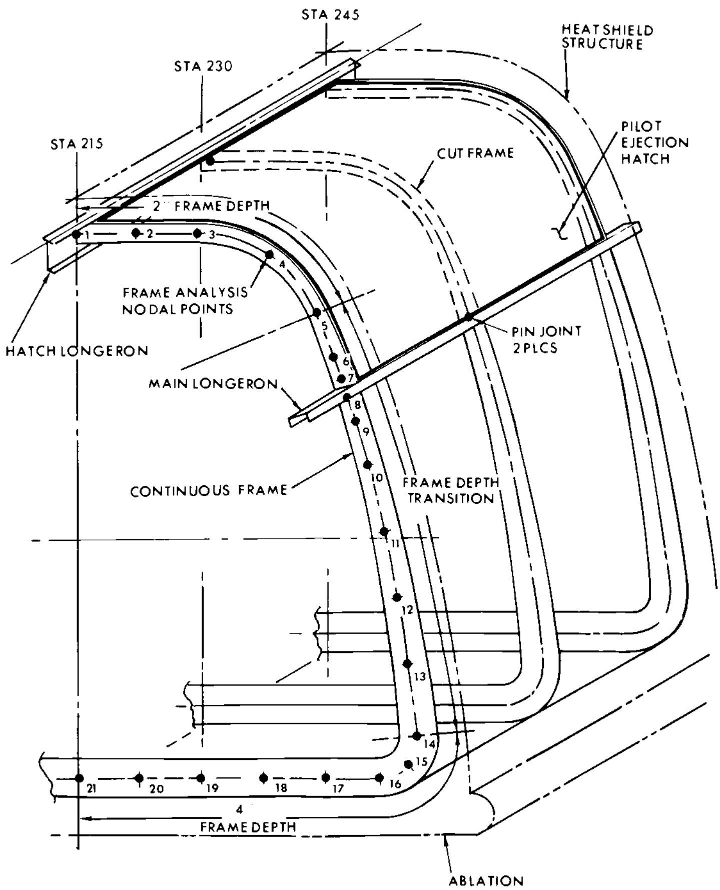


FIG. 21–*Fuselage analysis section* (scale: 1/20 full size).

The internal loading and deflections on both the titanium and composite material frames were analyzed for a pressurized loading condition. A burst pressure of 14 psi was associated with ultimate stresses and 7 psi with limiting deflections. The analysis was run for both the cut and uncut frames, with and without shell constraint loading. Half frames were analyzed, being symmetrical about the vehicle centerline, having 20 segments and 21 nodal points.

The all titanium frames (8Al-1Mo-1V) were first analyzed, both cut and uncut, by using best guess frame area and inertia distribution. Iterations were then made by varying the frame area and inertia on the uncut frame until the required deflection and stress conditions were obtained. Shell constraint loading due to the presence of the longerons was then established and input into the

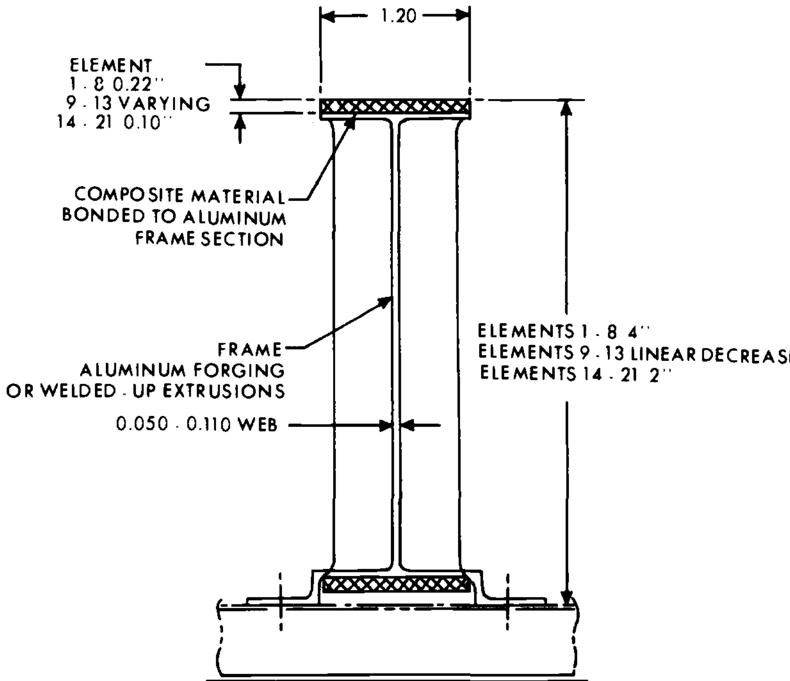


FIG. 22–*Composite frame design.*

frame analysis. Resulting stresses and deflections were found to be acceptable. Area and inertia distribution were identical for all frames, and a slight deflection variance between the cut and uncut frames was not thought to justify any further iterations.

Frame analysis of the composite frames was performed by substituting equivalent *EI* values to those finally determined for the all-titanium frames. Since both the basic aluminum frame and the composite material contribute to bending stiffness, a heterogeneous analysis was performed. This was accomplished by multiplying the aluminum cap area by the ratio of its modulus to that of the composite material, 10^7 / 32 x 10^8 = 0.31. The analysis showed that the heterogeneous frame had matching deflections with the all-titanium frame. The distribution of material and resulting weight of the differing frame constructions are given in Table 9. The potential weight savings that the aluminum-boron composite material can offer in lifting entry vehicle applications for stiffness and deflection critical structures is significant. The conservative approach taken in this study resulted in a weight saving of 42.6 percent over the all-titanium frames. This represents a potential weight saving of 0.63 psf of the vehicle surface with a minimum of manufacturing complexity and cost.

TABLE 9–*Cruise spacecraft frames, weight comparison.*

Element	Length, in.	Area Al-B, in.^{2}	Area Al, in.^{2}	Weight lb	Area Ti, in.^{2}	Weight, lb
1	6.2	0.52	0.24	0.47	1.00	0.99
2	6.2	0.52	0.24	0.47	1.00	0.99
3	8.0	0.52	0.24	0.61	1.00	1.28
4	8.0	0.52	0.24	0.61	1.00	1.28
5	5.0	0.52	0.24	0.38	1.00	0.80
6	2.5	0.52	0.25	0.19	1.00	0.40
7	2.0	0.52	0.26	0.16	1.00	0.32
8	2.5	0.52	0.26	0.20	1.00	0.40
9	5.0	0.52	0.27	0.40	1.00	0.80
10	7.0	0.47	0.29	0.52	0.90	1.01
11	7.0	0.41	0.30	0.50	0.80	0.90
12	7.0	0.35	0.32	0.47	0.70	0.78
13	7.6	0.30	0.34	0.48	0.60	0.73
14	3.3	0.24	0.34	0.19	0.50	0.26
15	3.3	0.24	0.34	0.19	0.50	0.26
16	5.6	0.24	0.34	0.32	0.50	0.45
17	6.5	0.24	0.34	0.38	0.50	0.52
18	6.5	0.24	0.34	0.38	0.50	0.52
19	6.5	0.24	0.34	0.38	0.50	0.52
20	6.5	0.24	0.34	0.38	0.50	0.52
Length = 112.2				7.68		13.74
			web stiffening	0.56		0.63
			TOTAL	8.24		14.37

NOTE–Unit Weights:

$$\text{Al-B:}\quad WT = \frac{144 \times 8.24}{15 \times 112.2} = 0.71 \text{ psf} \times 1.20 = 0.85$$

$$\text{Ti-s-1-1:}\quad WT = \frac{144 \times 14.37}{15 \times 112.2} = 1.23 \text{ psf} \times 1.20 = 1.48 \quad 100\%$$

$$\text{Weight Saving:} = \frac{1.48 - 0.85}{1.48} \times 100 = 42.6\%$$

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S. Yurenka Elliott[1]

Chapter VI – Fabrication Processing

REFERENCE: Elliott, S. Yurenka, "**Fabrication Processing**," *Applications of Composite Materials, ASTM STP 524,* American Society for Testing and Materials, 1973, pp. 163-185.

ABSTRACT: A considerable factor in the cost of a composite structure is the cost of fabrication. Because composite fabrication techniques differ substantially from metal fabrication techniques, new technology is required. This chapter summarizes progress in the areas of automated tape layup, filament winding, and molding plus secondary bonding.

KEY WORDS: composite materials, fabrication, moldings, filament winding, boron

It has been thoroughly demonstrated and almost universally recognized that a major problem associated with the use of advanced filamentary composite materials in structural applications is the development of tooling and fabricating techniques. A particular drawback has been the major reliance upon hand processing techniques due to the lack of automatic processing equipment. This suggests that if the promise of composites is to be realized, great emphasis in the future should be placed on the development of improved tooling and fabrication techniques.

The purpose of this chapter will be to present some of the latest ideas and expectations related to improvements in such techniques. The types of processes which will be described include:

1. Automatic tape layup process.
2. Continuous filament winding process.
3. Molding plus secondary bonding.

Automatic Tape Layup Process

Glass fiber reinforced plastics have been used successfully on military and commercial aircraft for many years. The structures, for the most part, have been designed using glass cloth laminates and low production hand methods. In recent years a number of aircraft companies have seriously investigated the potential of some of the newer composites, such as boron or graphite filament reinforced

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epoxy laminates for primary structural aircraft applications. Because of the much higher moduli of these newer reinforcements, they are much less "forgiving" in their tolerance of the inaccuracies of hand workmanship than the glass fibers have been. Furthermore, because of their high cost, rejects and excessive waste are intolerable. These and other considerations have therefore led to the replacement of hand fabrication techniques by automatic machine methods.

One version of this type of machine has been developed by the Douglas Aircraft Division of the McDonnell Douglas Corporation [*1*] [2] and is shown in Fig. 1. Although it was designed specifically to fabricate boron and graphite filament reinforced epoxy composite laminates, it has sufficient versatility and capacity to be useful for producing many other similar layups. The machine in Fig. 1 may be described as consisting of two parts. The first is the basic machine structure, which is the supporting mechanism and provides the power and motions for the tape laying head. The second part is the tape laying head itself, which carries, unrolls, cuts, and applies the tape.

The basic machine structure consists of four weldments. The two longitudinal beams are supported at one end by a leg assembly and at the other end by a box assembly. A cross-rail assembly provides the 96-in. longitudinal travel and a cross-slide provides the 36-in. transverse motion. The longitudinal motion is variable in speed from 0 to 15 in. per second.

The forward speed control is located on the operator's remote pendant, whereas the return speed control is located on the manual control panel.

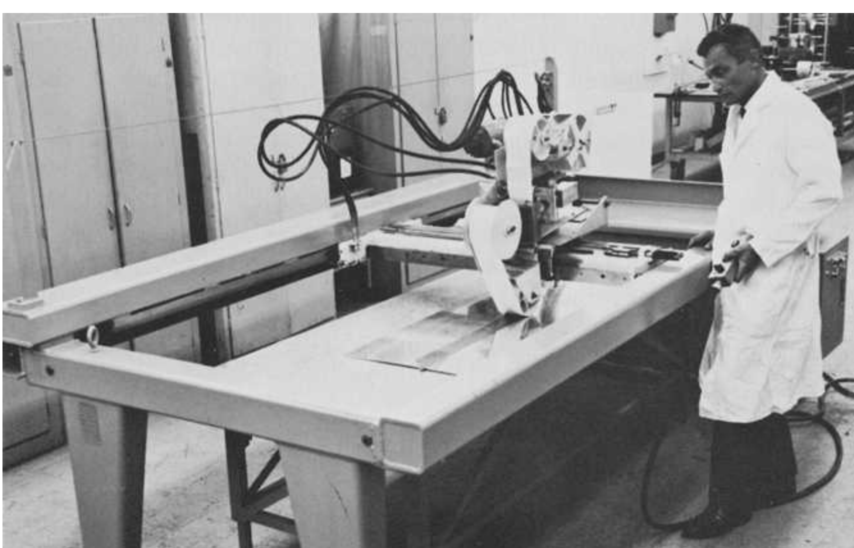

FIG. 1—*Automatic tape layup machine [1].*

[2]The italic numbers in brackets refer to the list of references appended to this paper.

Automatic reversing is accomplished by means of limit switches located on the head assembly. Trip bars located properly on the work holding table provide the outline of the part shape.

With each longitudinal reciprocation, the head is automatically stepped, transversely, a preselected distance and direction to correspond to the tape width, allowing the head to lay one complete tape layer automatically before repositioning the work for the next layer.

Orientation of the tape on the layup sheet is accomplished by manually locating the work holding table beneath the machine's working area.

The operator's controls consist of a hand-held pendant containing forward, reverse, stop, and forward speed control for the longitudinal motion. On the bottom of the pendant is a pushbutton for momentary release of the cross-slide clamping to permit manual positioning. Other manual controls are located on a panel attached to the machine structure and consist of: machine start-stop pushbuttons, selector switch for direction of transverse motion, tape laying head up/down automatic selector switch, and tape cut pushbutton.

Power for the longitudinal (reciprocating) motion is provided by a 1/2-hp dc electric motor. Power for the transverse stepping is supplied hydraulically with a cylinder-valve complex and a 1/2-hp pump. The entire machine operates from a single 115-volt ac source.

Manufacturing Flat Uniformly Thick Laminates Automatically

The manufacture of flat, uniformly thick laminates by means of an automatic tape layup process is obviously much simpler than the manufacture of panels possessing variable thickness and double curvature. Other than spools of 3-in. wide, paper-backed, boron or graphite filament prepreg tape, Fig. 2, and the automatic tape machine, the only item needed is a stainless steel caul plate which is slightly larger than the required dimensions of the panel which is to be laminated.

The caul plate is first cleaned, then coated with a waxy parting agent, and

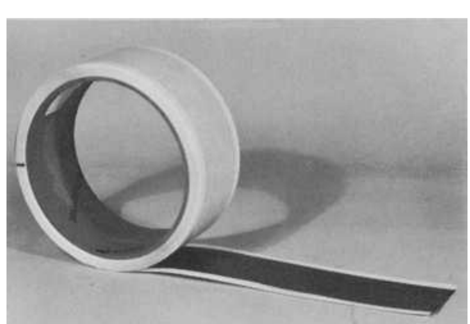

FIG. 2–*Boron filament preimpregnated.*

placed on the layup table. The layup table and caul plate are oriented so as to make the filaments lay down at the desired angle with respect to the axis of the machine. The traversing head containing the prepreg tape is positioned at one corner of the caul plate, and its automatic operation is activated.

With each forward stroke, a length of the 3-in. wide tape is rolled onto the caul plate. At the end of the stroke, the cutter (Fig. 3) is activated by the trip bar and severs the tape at the proper length and angle. The machine then automatically indexes itself one 3-in. tape width and butt lays another strip parallel to the previous one. This action is continued until the entire plate is covered with one continuous layer of tape.

Succeeding layers of tape are processed in a similar manner as the first, except that the layup table and caul plate are oriented at the proper angles with respect to the previous directions. In this manner the laminate is built up to the desired thickness with the respective filament directions of each layer oriented at the desired angles. Because the uncured resin in the prepreg tape is somewhat tacky, the entire prepreg panel will maintain its shape readily and can be handled easily with no fear of its possible disintegration or delamination. After the layup is completed, the panels can be vacuum bagged and cured in an autoclave according to the prescribed temperature and pressure cycle.

Manufacturing Curved and Variable Thickness Laminates Automatically

Structural aircraft airfoil composite shapes, such as the vertical or horizontal stabilizer and wing skins, are both doubly curved and variable in thickness, and

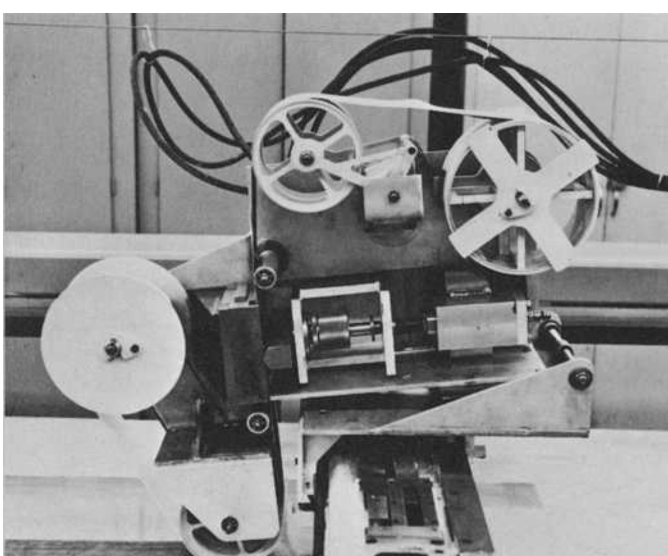

FIG. 3—*Cutting mechanism.*

hence, the automatic fabrication technique becomes complicated.

As with flat skins, a steel caul plate is cleaned, coated with a waxy parting agent, and then placed on the layup table. This caul plate is a curved shell as required by the design of the structural part. Obviously, for a cylindrical shell with a single radius of curvature the prepreg tape can still be layed up flat and curved afterwards by draping the prepreg panel over the curved caul plate. However, if the shape is doubly curved, this secondary draping and stretching of a pliable layup is not possible unless the curvature is very slight, because there is a possibility of producing either wrinkles or excessive gaps in the laminate. Consequently, the automatic tape layup technique becomes more complicated as additional degrees of freedom are added to its motions. To cover every possible translation and turn, a universal layup head with six degrees of freedom would be required (three translational and three rotational motions along or around the x, y, and z axes, respectively). In some cases it might be more advantageous to reduce the number of degrees of freedom of the layup head on such a machine by substituting a movable table or mandrel under the head which would simultaneously rotate or translate the tooling plate as the head also rotated and translated. An automatic tape layup machine with either type of capability does not yet exist, but, as described in another section, initial thinking and planning have started. In the meantime such doubly curved layups must be performed by hand labor.

Depending on the degree of curvature, the width of the prepreg tape must be changed; the tape strips becoming narrower as the radii of curvature become smaller. This will prevent wrinkling at the edges or the center portions of the tape as their relative contact lengths with the curved mold vary. Furthermore, since curved surfaces cannot be covered completely with parallel tape strips butted against each other, as in the case of flat surfaces, without bending the filaments out of their straight line paths, a certain amount of deliberate gapping and overlapping will occur. If the filaments have a small diameter (such as carbon, graphite, or glass), this will not be serious, because the small spaces and ridges in the vicinity of the joggles produced by overlapping layers will become faired out by the resin matrix. In the case of the much larger diameter filaments such as boron, however, the residual stresses produced by such kinking could become quite serious.

In certain special curved shape cases such as a hemispherical or ogive dome, the crossing of successive "diametral" strips of tape at the apex of the dome not only leads to an "infinite" laminate thickness at the apex, but also results in an unsightly "circular staircase" pattern. In such cases the filament orientations and strip lay down sequences would have to be varied so that most of the strips would lie tangent to various imaginary circles having a common center at the apex.

In any practical structural composite part there will have to be provisions for holes or cutouts of various sizes and shapes and for attachments and joints to other similar composite structures or metal members. The problems of designing, analyzing, and fabricating such cutouts and joints in conjunction with the rest of the composite structure represent an entire subject in itself and is one which has not yet been completely solved. It should be recognized, however, that the ultimate success of the application of composites to practically all structures will be determined largely by the success of the design and the fabrication of the required joints and cutouts contained within the part. In general, these areas will involve the internal, or external bonding or laminating of thin tapered metal or composite shims or doublers.

Manufacturing Problems

The most serious problem presently associated with the automatic layup fabrication technique is the lack of tackiness, or adhesion, between the 3-in. wide prepreg tape and its 4-in. wide paper backing. As currently supplied, this backing paper is coated with a waxy parting agent to make separation easier; hence, its adherence to the tape is minimal. As a result, during the passage in the machine, delamination occurs making it difficult to utilize the paper's edge as an accurate reference line for indexing the tape. Considering that when it was developed, the paper backing was intended only as a protective covering for the boron tape and not as an integral part of an automatic process, it is not surprising that such complications occur. This is not an insurmountable difficulty, however, and investigations are currently being made to select a better separator material which has the proper adhesion to the tape. At the same time the rollers of the machine are being equipped with motor drives and sprocket teeth. With the separator material correspondingly punched by the tape vendor to accept the sprockets, much more positive and accurate motion of the tape through the layup head will be assured. In traversing around the rollers of the head, the tape motion will be reminiscent of a film passing through a movie projector.

The second problem is in the mechanism which severs the tape at the end of each stroke. Boron tape, for example, is bonded to a fiberglass scrim cloth to prevent the individual filament from shifting around during handling. Because the boron filaments are very brittle, they fracture easily when dealt a strong chopping blow with a hard blade. However, occasionally a few fiberglass strands in the scrim cloth remain uncut, necessitating stopping the machine and completing the cut with scissors. When higher impact loads are triggered, the carbide-tipped cutoff knife has a tendency to fracture. Substitution of a Teflon striker plate for the hardened steel is not a solution because the Teflon soon develops abrasions. A rubber faced striker plate also has a drawback in that it does not always sever the scrim cloth. The obvious conclusion is that a guillotine or shearing type cutter should be employed. Because this does not lend itself to the flying knife technique, wherein cutting is done without stopping the machine, the machine cycle time is slower. However, the total time to complete a layup is less, because the positive cutoff eliminates the occasional hand scissor cutting.

A third problem is in the compression roller which presses the tape against the caul plate and simultaneously peels away the paper backing. In the layup head design, the boron tape makes a 90-deg turn as it curves around a quarter of the roller. Because the roller is eight in. in diameter, the bending stress imposed on the boron filament is not considered serious. However, because of its stiffness and because of the lack of adhesion between the paper separator and the boron tape, the tape does not follow the paper around its 90-deg turn each time a freshly cut edge appears, but simply drops down onto the caul plate instead of lying tangent to it. This problem can be temporarily overcome by manual redirection of the tape by the machine operator prior to the beginning of each new stroke. The permanent solution is a rearrangement of the tape laying mechanism which will minimize the change in direction of the tape from a continuous straight line. Graphite filament tapes, which have smaller diameter and lower stiffness filaments, do not have this problem.

Future Potential of the Automatic Tape Layup Process

Anticipating the possibility that various problems would arise during the development of the automatic tape layup machine, the design was kept small, rudimentary, and inexpensive so that modifications and improvements could be added as they evolved. When the materials and processing problems are thoroughly solved, the next step will be to incorporate the results of the investigations into a production machine. The present concept is that such a machine (Fig. 4) would be much larger (for panels up to six ft wide and 30 ft

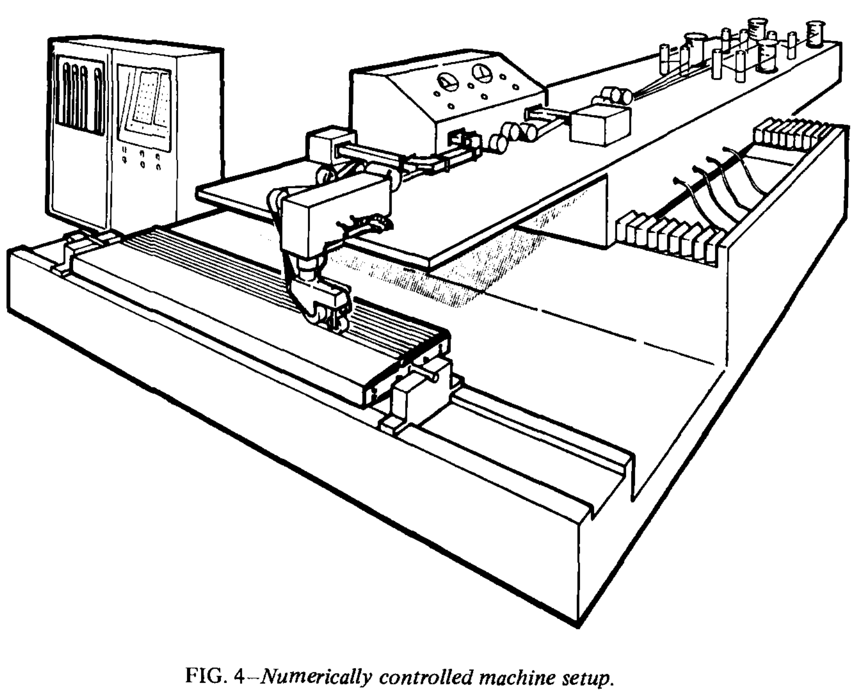

FIG. 4–*Numerically controlled machine setup.*

long), more versatile (for doubly contoured parts as well as flat), more automated (numerically controlled and with a feedback servo-mechanism drive), more universal (will accept tape from 1/8 to 3 in. wide of boron, glass, graphite, or other filaments), and more precise (with perforated paper backing for sprocket wheel positioning). With such a machine it would be possible to wrap an entire stabilizer, wing skin, or portions of a large fuselage, Fig. 5. Information obtained from the use of the rudimentary machine described would prove extremely valuable in the future planning of a fully automated machine.

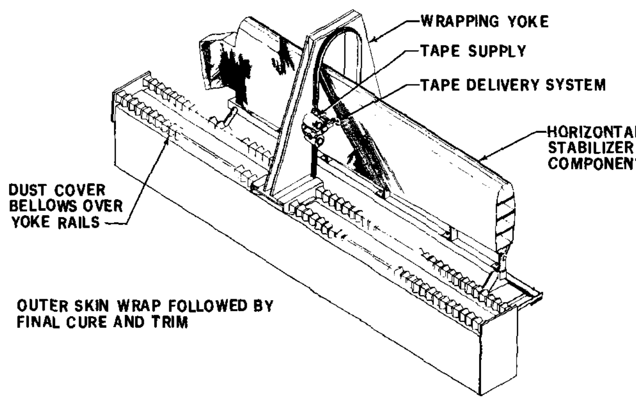

FIG. 5–*Five-axis numerical control winding machine.*

FIG. 6–*Conrac automatic tape layup machine [2].*

Recently, a sophisticated numerically controlled tape laying machine was developed by the CONRAC Corporation [*2*] under contract to General Dynamics, Fort Worth, and the Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio, which is a good example of the ultimate production machine being contemplated. Designated the CONRAC TLM-100 (Fig. 6), this automatic machine was designed to apply filament tapes and foils of various sizes and types, such as boron, fiberglass, and graphite, on an open surfaced part along a preprogrammed path. In addition to the tape laying function, the machine automatically cuts the tape or foil to a predetermined length and angle at the end of a laying course. The dimensions of the work table are 6 by 30 ft. Appropriate mechanical devices are provided in the head assembly to hold the tape reel, tension the tape, feed the tape into the application area, and separate and contain the tape backing paper. The reel handles tape in widths from 1/3 in. to 3 in., lengths of up to 5000 ft, and has a hub diameter and outside diameter of 8 in. and 26 in., respectively. The machine is equipped with a numerical control system for controlling all movements and rotations, and is arranged for operation in either the manual, semi-automatic, or full numerical control modes. It is not unreasonable to expect that future aerospace manufacturing needs associated with composite structures will be handled by such machines.

Continuous Filament Winding Process

In a simplified form, the filament winding technique for composite materials consists of unwinding a band of parallel resin impregnated filaments (prepreg tape) from a storage spool and winding it under programmed tension upon a surface of revolution (mandrel). The mandrel (surface of revolution) upon which the prepreg tape is wrapped, supplies the outline for winding and subsequently must be removed. The tension imparted to the tape during winding packages the filaments densely and provides the necessary interfilament pressure during curing. When the winding operation is complete, the wound structure and mandrel are wrapped in a vacuum bag and heated in an autoclave for the required temperature, pressure, and time.

Boron Filament Wound Cylinders

Perhaps the simplest and most common geometrical shape which has been filament wound for highly loaded composite material applications is a cylinder. In a recent Air Force/Douglas program [*3*], this technique was applied to the production of boron filament wound epoxy prototype landing gear strut structures. The struts measured 3 in. inside diameter, 26 inches long, and 0.135 in. wall thickness. In order to resist the combined loading on this strut (namely, bending, torsion, and compression), the orientations of the boron filaments were to be primarily longitudinal, but with a few layers of circumferential and helical (± 45 deg) interspersed among them. Since the boron filaments are 0.005 in. in diameter, a total of 27 layers of filaments in the form of prepreg tape ribbons

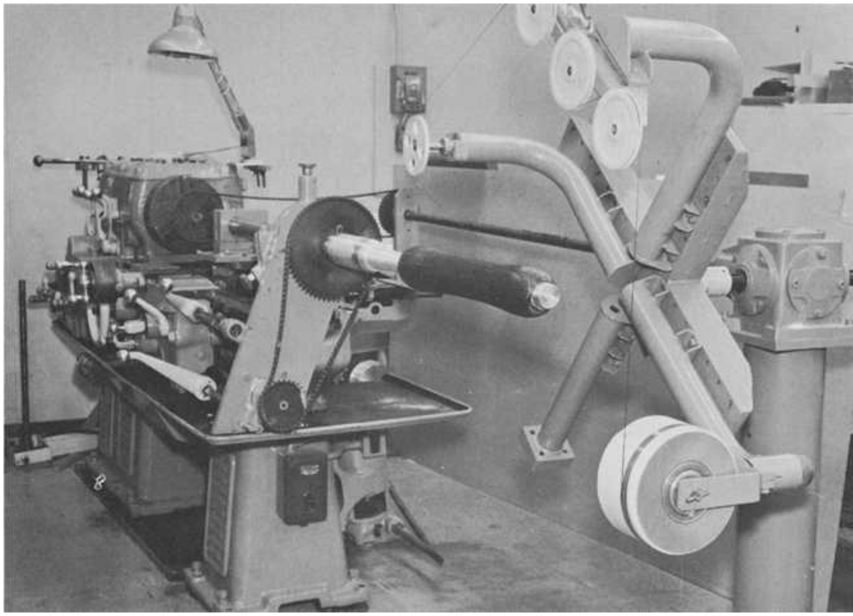

FIG. 7–*Polar winding [3].*

was required to layup the total 0.135-in. wall thickness.

It was originally intended that all of the longitudinally oriented filaments of the strut specimens would be filament wound by means of a polar filament winding machine modified to utilize 1/8-in. wide boron filament-epoxy prepreg tape, Fig. 7. The 45-deg helical wraps were to have been laid by hand using 3-in. wide boron-epoxy prepreg tape, Fig. 8. Finally, at the required intervals of the laminate buildup, the circumferential wraps should have been applied with a lathe type winder modified to utilize 1/8-in. wide boron-epoxy prepreg tape, Fig. 9.

However, when the first attempt to filament wind the longitudinal layers by

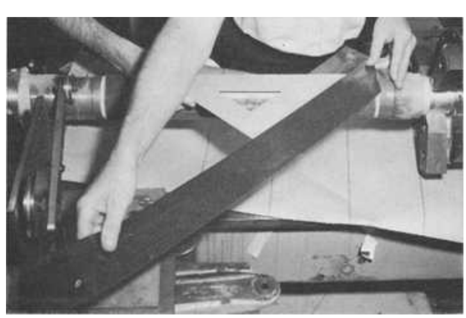

FIG. 8–*45-deg helical winding [3].*

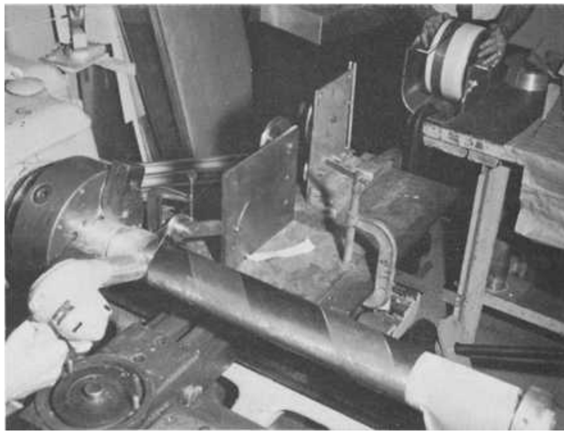

FIG. 9—*Circumferential filament winding [3].*

means of the polar winding machine were made, it quickly became apparent that this technique was impractical for boron filaments on 3-in. diameter mandrels. The problem is that because the boron filaments are so large and stiff (for example, boron filaments are 60 000 times stiffer than glass filaments), they resist bending onto the domed ends of the 3-in. cylinders. Thereafter, by a combination of slipping and straightening, they began to lose contact with the mandrel. Eventually, the cylindrical layup assumed a shape similar to a dumbbell, Fig. 10. Although the filaments could temporarily be pressed by hand to lie flat against the mandrel, the tackiness of the resin binder was not sufficient to retain them there, and they repeatedly sprang out again. It was concluded that the polar winding of boron filaments would be feasible only if: (*a*) filaments were much smaller in diameter, (*b*) mandrel and end dome diameters were much greater than 3 in., or (*c*) filaments were immediately anchored as they were wound by continuous curing of the resin or interspersing of a cylindrically wound layer over each pair of polar wound layers.

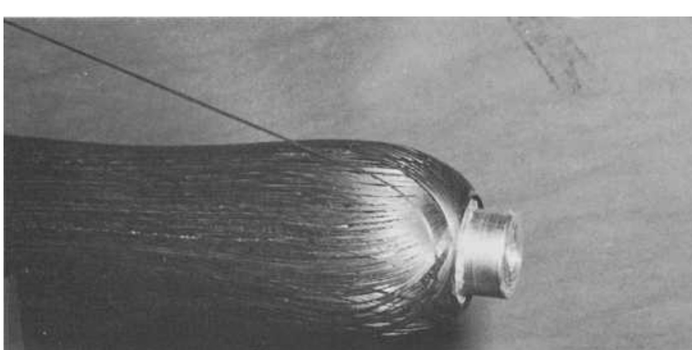

FIG. 10—*Effect of filament slippage [3].*

Because these approaches were not feasible for this application, the concept of polar winding was abandoned and replaced by a hand layup procedure for the placement of longitudinal filaments. To forestall possible wrinkling of the outside of the cylindrical specimen after curing under heat and pressure, a "densification" step was introduced after approximately each four layers were applied. This densification step consisted of placing the specimen in a vacuum bag and pressurizing it in an autoclave at 100 psi and 200 F for 1 h. After the final circumferential ply was wrapped, the strut specimen was again placed in a vacuum bag and cured in an autoclave at 100 psi and 350 F for 90 min.

Unfortunately, in spite of the densification steps which had been performed on the strut specimen, when it was removed from the autoclave after curing it had a large wrinkle along its entire outer length and was unacceptable for test. X-ray examination showed that approximately one third of the wall thickness was damaged by the wrinkle. It was decided, therefore, to salvage the specimen by removing the affected number of wrinkled plies. This required a very difficult grinding operation using a carborundum wheel under constant water flow. Extreme care had to be exercised to avoid damage to the metal step laps. After 12 plies were removed, the metal fittings were grit blasted and prepared for rebuilding.

This time the densification step, which took place after each four plies, consisted of vacuum bagging and curing in an autoclave at 100 psi and 300 F for 2 h. This almost complete cure after each four layers completely densified the sequential laminate buildups and produced solid foundations for the succeeding plies. After the last cure step, the specimen was visually inspected and X-rayed, and its appearance was good.

Filament Wound Box Beams

The basic cross sectional geometry of an aircraft wing or the horizontal or vertical stabilizer represents a rectangular box with two or more cells, Fig. 11.

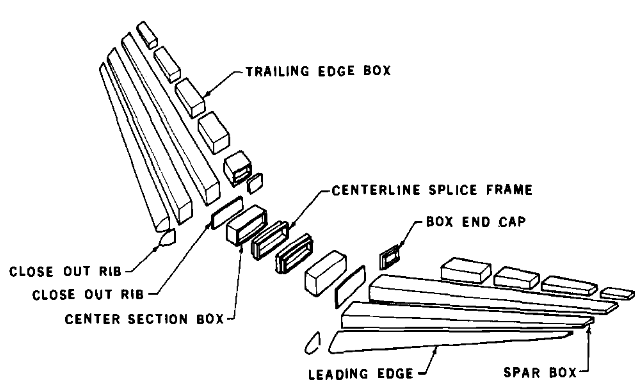

FIG. 11–*Box beam horizontal stabilizer construction.*

The filament wound box concept involves winding a separate box for each cell by high speed helical, polar, or circumferential winding, or by hand layup of biased broadgoods, Fig. 12. Reinforcing can be added to the sides of each cell during winding to provide more effective shear webs. The cells are then cured and assembled side by side, and the skins are either prefabricated and bonded to the boxes or wound on in a manner similar to the winding of the cells. To form even more rigid internal structures, the spar webs can be separated by honeycomb cores. Similarly, the skin panels may either be solid laminates or honeycomb sandwich covers depending on the rigidity desired.

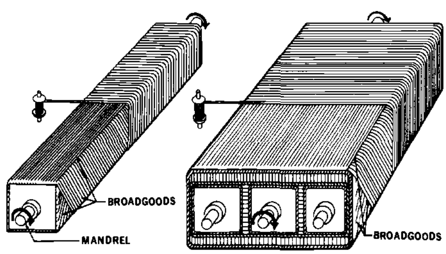

FIG. 12–*Filament wound box beam concept.*

Filament Wound Rib Stiffeners

In a typical box beam structure, the cells are divided into fuel cell compartments by means of ribs which also serve to stiffen the skins. The usual technique for integrating these rib stiffeners into the box beam structure is to filament wind and then bond them to the inside surface of the separately fabricated skins. The manufacturing process recommended for the ribs is similar to that described for producing spanwise box cells, Fig. 12. The majority of the reinforcing fibers are at approximately ±45 deg to the lengthwise direction of the rib. These reinforcements, cut from prepared unidirectional plies of broadgoods, are laid over the male tool. A final filament winding operation

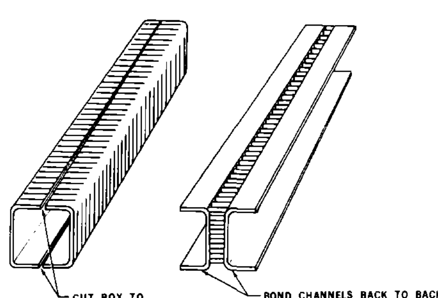

FIG. 13–*I-beam rib stiffeners.*

applies a layer of circumferential "B"-staged composite tape. Lengthwise fibers, again in the form of broadgoods, can be laminated in place if desired. When all the tape layers have been applied, metal pressure plates are located over the four outer surfaces, and the part is bagged and cured in an autoclave. The cured box is then cut from the mandrel in the form of two equal channels. These channels are bonded back to back to form an I-beam cross sectioned rib, Fig. 13. If desired, the channel backs can be separated by and bonded to a honeycomb core. In either case, tooling is required to hold the flanges of the I-beam rib in the proper relative location during bonding.

The technique of winding or wrapping box cells, cutting them, and then bonding them together in another way can be repeated in many variations to yield other familiar cross sectional shapes. Figure 14 shows, for example, the familiar angle, T, Z, J, and hat section. When a sufficient quantity of rib stiffeners such as these have been prefabricated, they can then be bonded to flat or singly curved laminate skins to provide the desired stiffening.

Molding Plus Secondary Bonding

The automatic tape layup machine and fabrication technique are suitable for producing flat or slightly curved laminates, and the filament winding machine and technique are suitable for cylindrical or spherical structures. However, in their present form, neither of these approaches lend themselves to the production of the many other intricate and complex shapes which are needed in aircraft manufacture. For such constructions as box beams, rib stiffeners, and molded rib fittings, special variations of present fabricating techniques, or perhaps some new ones, are needed. In any case the basic steps are similar, namely:

1. Assemble a composite prepreg layup to the desired shape and size by laminating successive layers of composite prepreg tapes with the filaments

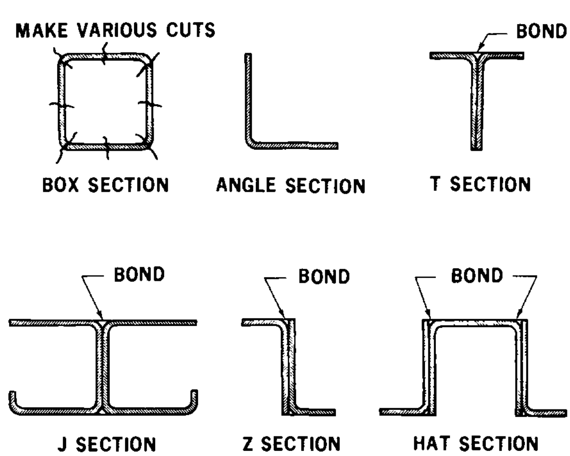

FIG. 14–*Rib stiffener shapes.*

oriented in prescribed directions and designed to yield a laminate of the desired strength and stiffness.

2. Cure the prepreg layup under heat and pressure on a mandrel or in a mold under the heat and pressure of an autoclave or a press.

3. Perform secondary machining and bonding, joining, or laminating operations as required to assemble the laminates into the final structural component.

Actually, this approach is not exactly new, since it has been employed extensively in the past to fabricate fiberglass or bonded metal structures. However, some of the special techniques which are unique to the application of advanced composites involve some new ideas, and a few typical examples are described in the following sections.

Filament Wound Broadgoods Material

An essential intermediate material form in the technique of molded composite structures is a long wide prepreg band of collimated filaments known as broadgoods. Although similar to woven cloth prepregs in its application, the broadgoods material offers considerably more promise, because its filaments are perfectly straight and undamaged instead of being twisted and woven as in the case of cloths. In the application of the high modulus, high strength, or large diameter advanced filamentary reinforcements, this is an important consideration if the potential of such filaments is to be realized. The broadgoods materials are fairly simple to make, or they can be purchased with the desired size and resin binder from prepreg suppliers.

An example of the fabrication of a piece of boron filament broadgood [*4*] is

FIG. 15–*Boron filament broadgoods.*

shown in Fig. 15. Collimated boron filaments in the form of a 1/8-in. wide prepreg tape are wound under tension to the desired width on a 32-in. diameter drum using a feed rate of 0.104 in. per revolution. The surface of the drum is covered with a 3-mil Mylar parting film prior to winding. Heat from a portable electric heat gun is used to warm the tape during the winding operation and improve the resin "tack". When finished, the wide bands are cut from the drum and laid out into flat sheets of broadgoods as in Fig. 15. The broadgoods can then be packaged and stored at 0 F until needed. When it is ready to be laminated into sheets or shapes, the broadgoods can be cut to size with scissors or shears and handled quite readily because of cohesiveness provided by its 3-mil Mylar film. For those applications where angle-ply orientations are required, some broadgoods waste can be eliminated by helically cutting the original wide bands from the drum so as to create diamond shaped sheets rather than rectangular sheets.

Preoriented Prepreg Matched Die Molding

The application of the technique of preoriented preform matched die molding to the fabrication of advanced fibrous composite structural parts is just getting started, but the potential for its increasing use is extremely promising. In contrast to the simple tape layup or filament wound type technique previously described, this method readily lends itself to the fabrication of an unlimited variety of high strength complicated or unusual shapes. The three essential elements of this process are a heated press, a die, and the preform. The first two are standard items similar to ones currently in use for fabricating fiberglass molded parts, but the preform is the item which underwent the greatest advancement when it was applied to the advanced composite molded part.

In the fiberglass matched die molding process, the preform is an arrangement of chopped glass fibers in the exact shape of the part to be molded. The basic approach is to deposit the chopped fiber (normally approximately 1.5 in. long) uniformly over the surface of a screen that has been formed to the shape of the final molded part. As this fiber is being deposited, a small amount of binder resin is applied. Deposition proceeds until sufficient preform thickness is obtained to provide the optimum glass-resin ratio in the molded part. While still on the screen, the preform is then heated until the binder resin cures. This provides the preform with sufficient integrity to be stripped from the screen and transported to the press for final molding.

As it is placed into the die, the additional resin is added to the preform. Closing the male and female halves of the matched die not only distributes the resin throughout the fibers but also trims the preform to the required dimensions. To perform this operation most efficiently, the press has dual closing rates—one fast to bring the two parts of the mold rapidly together, and an infinitely variable slow rate to effect final closing. Although manual control can be used for such work, more consistent quality can be obtained in presses which have automatically controlled closing, as well as temperatures, pressures, and time intervals. This process, then, is ideally suited for automation.

The improvement being introduced in the preform for molding advanced fibrous composites, such as boron or graphite composites, is in the form of the reinforcement and the application of the resin. Essentially, the process consists of producing the preform over the male half of the die by a careful layup of strips of collimated prepreg tape similar to that used in the automatic tape layup process. The directions in which the fibers are oriented coincide with the directions of the stresses to which the structural part is subjected in service, and the number of plies which are applied are proportional to the magnitude of the loads which have to be supported. To determine the amount of reinforcement needed and the orientations of the fibers, a computer program assisted micromechanical stress analysis is usually necessary. If the procedure is followed carefully, the fiber directions and number of layers are perfectly predetermined rather than arrived at in random fashion. In this way, the superior properties of the advanced fibers can be utilized to their best advantage with no waste of material or excess weight in the molded part.

After the preform is layed up on the male lower half, the female upper half is lowered over it by means of the press platens to which the mating parts are attached. This tends to minimize dragging or creasing of the preform as the mold is closed. To obtain optimum reproducible parts, the preform is made as consistent as possible. Molds are coated with mold-release compound just as in all other processes. In selecting the prepreg tape resin, a combination of temperature, catalyst concentration, and resin formulation is used to cause gelation in a time long enough for complete resin flow and short enough to be commercially feasible. The final molding is done at pressures up to 3000 psi and is very similar to normal compression molding.

Because of the large diameter and brittle nature of boron filaments, they are not suitable for molding parts where the radii of curvature are very sharp or where the wall thicknesses are extremely thin. This limitation does not apply, however, to the much smaller diameter and less brittle graphite fibers. Although the use of both of these materials is still developmental, they appear to be highly promising for a variety of molding applications where complex shapes are encountered, yet maximum strength and rigidity are mandatory.

Molded Graphite Rib Fitting

The possibility of making complex fittings of molded graphite has always appeared attractive. Boron is of limited usefulness in this respect because of the severe restrictions on allowable filament bend radius. Graphite, because of its very small individual filament diameter, has no such limitation and can easily follow the contours of a complex part. The actuator rib fitting, Figs. 16 and 17, of a graphite composite A-4 landing flap, Fig. 18, was selected as being a suitable part for this application and showed promise of a significant weight saving [*4*]. In the original all-metal version of this flap, the actuator rib fitting was made of forged aluminum and was quite heavy.

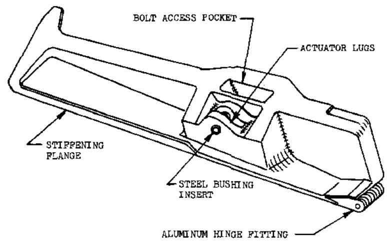

FIG. 16–*Molded graphite rib fitting concept.*

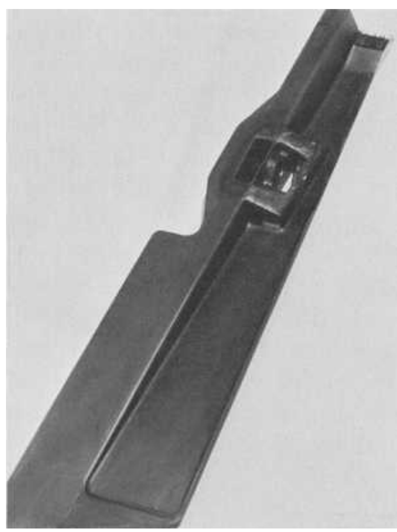

FIG. 17–*Molded graphite rib fitting.*

Initially, it was intended to use chopped fibers, but it soon became apparent that it would be more economical to place tapes within the mold cavity, since in this way, the desired strength could be achieved with smaller wall thicknesses. The fabrication technique involved an intermediate preform operation to overcome the "bulk-factor" problem. This refers to the reduction in laminate thickness during the application of temperature and pressure in the cure cycle.

The basic channel section of the aluminum fitting was changed to a Z-section utilizing an essentially constant wall thickness. The original channel tended to become a solid section at the smaller rib depths, which was not only wasteful of weight but also undesirable from a molding point of view. Furthermore, by turning the lower flange of the Z-section inboard to the edge of the flap, some support was given to the overhanging portion of the lower flap skin. This lower flange was, in fact, taken to the edge of the skin and was provided with a vertical

FIG. 18–*A-4 composite landing flap.*

stiffening flange, 0.25 in. high, so that no other support members were needed. At its trailing edge corner, the lower flange occupies the full depth, and bending stiffness was provided by running filaments at 90 deg to the fitting web at the top and bottom of this flange. The graphite fitting extended to the flap trailing edge, whereas the aluminum forging stopped several inches short and required an extension filler member to complete the structure.

At the leading edge of the fittings a short length of piano hinge was provided to transfer the fitting loads to the aircraft support structure. Limitations of space and uncertainties about the ability of the graphite material to take repeated loads without wear necessitated the use of an aluminum hinge insert. This insert was bonded in place between the layers of the vertical and lower flanges of the graphite fitting. The upper flange of the rib fitting was folded over to make a good shear connection with the flap leading edge member.

The most difficult part of the rib to design and fabricate was the region around the actuator attachment bolt. Here, the actuator applies its operating load through a short link member that swings through an arc when the flap is deflected. Adequate clearance had to be provided for the operating mechanism, and this largely influenced the geometry of the rib. The link load is transferred through the bolt to two lugs, only one of which has a direct load path into the shear web of the rib. The other lug is remote from the web, and the load has to be transferred across by transverse beams on either side of the link. These beams have flanges composed of filament at 90 deg to the fitting web and transfer their loads directly into the flap skins. It was necessary that his load path be very efficient because the alternative path was not good. To take more than 50 percent of the load through the lug adjacent to the web requires that the actuator bolt must transfer load by bending, and it has only a small capability of doing this.

The resulting structure in this region was much more bulky than the equivalent metal design and showed one disadvantage of composite structures. Metals have the ability to transfer loads in all directions, but composites only have good strength along the lines of the filaments. In regions where the load system is complex, filaments with different directions are often competing to occupy the same space, and it is not always easy to devise an adequate transfer of loads.

Access was provided to the bolt at the face of the rib by means of a recessed pocket. This enabled the nut to be properly locked, and it replaced the fiberglass dish and access door design that was used on the boron flap. In the upper flange the 0-deg layers that provide cap bending strength had to be diverted around this pocket region, since cutting the filaments would cause a reduction in strength. The basic laminate pattern to which the 0 and 90-deg layers were added was composed of layers at ±45 deg. Thermal balance of the pattern was maintained at all points on the rib to avoid distortion during the cure. The exact layup sequence was described in a processing specification and involved a systematic layup of oriented strips of graphite prepreg tape in prescribed directions and numbers of layers in the female cavity of a two piece mold. At appropriate stages of this layup procedure, debulked preform sections of the rib consisting of B-staged epoxy resin impregnated chopped graphite fibers were inserted between the oriented tape layers. The aluminum hinge fitting insert and steel actuator pin bushings were coated with an adhesive and also inserted between the layups at the appropriate times. When the prescribed number of tape layers and inserts had been introduced into the mold, the cavity was filled to excess. During the heating and pressurization cycle within a heated high pressure press this excess was partially taken up by compaction of the composite and partially by flow-out of some of the excess resin. After cure this resin flash was trimmed off, and the molded rib was ready for inspection, testing, and secondary bonding into the flap.

The successful molding of a complicated structure such as this A-4 flap rib fitting predicts the possibility of high pressure molding of many similar other fittings, ribs, spars, beams, columns, etc., for various structural applications. The composite materials could include mixtures of either chopped fibers or continuous tapes and could also involve mixtures of graphite and glass filaments where each reinforcement could be included as required by the loading and environmental conditions associated with the application. Metallic inserts or external plates could either be molded or secondarily bonded to provide means for joining or attaching the molded part to other structures or to improve resistance to bearing and wear if relative motions are involved. The combination of composites and metals by molding or adhesive bonding techniques produces a family of new materials which are known as hybrids, and the number of possible substitutions of such strong, stiff, and light combinations for the current all-metal structures is unlimited.

Injection Molding

This high production process was primarily designed for use with thermoplastic materials and is still largely neglected by the reinforced plastics industry. Of course, until quite recently the softening points of most thermoplastics were so low that they were seldom considered for structural applications, and hence, there was no point in reinforcing such materials. However, the introduction of new high softening point thermoplastics has opened new application areas, and interest in reinforcing such materials with glass or other fibers is increasing. Polystyrene, polypropylene, polycarbonate, nylon, polymethylmethecrylate, and Teflon are all being injection molded with fiberglass reinforcement and are finding new markets because of increased dimensional stability, impact resistance, and low temperature properties. Recently, the development of rapidly curing phenolic resins has led to their combination with chopped graphite fibers in the production of small motor case nozzles for the missile industry. In the injection molding process the filaments and liquid resins are heated and softened in a heating chamber before being injected into a cold mold. The mold then cools and solidifies the compound to the shape of the mold. When the mold is opened, the molded parts plus the connecting runners are positively ejected by hydraulically embedded knockout pins. The high production rates, which are available with the injection molding process, should lead to its becoming one of the most rapidly growing fabrication techniques in the composites industry.

Quality Control

Regardless of the fabrication process that is employed for producing composite components, quality control activities must be performed continuously from initiation to completion to ensure reproducible high quality production. This necessity for quality control of composite materials is greater than for homogeneous materials, because the final product is manufactured from various intermediate material components rather than from mechanical operations on a homogeneous material. This means that incoming raw materials must be inspected for conformance to material specifications; in-process quality control must be established for conformance to processing specifications for the fabrication of the composite component; and as a final check, some combination of destructive, nondestructive, and proof testing of the finished article must be scheduled.

Quality Control for Incoming Materials

Quality control tests should be conducted on each separate lot or batch of incoming materials in accordance with prescribed specification requirements, and those materials which do not meet the requirements should either be rejected or returned to their manufacturers for possible salvage and rework. Usually small samples of incoming materials, such as prepregs for example, are checked for their resin and void contents, flexural strengths and moduli, and interlaminar shear strengths. Other items, such as filament alignment, ends per inch, and reinforcement spacing, must also be determined, particularly when the materials are supplied on a carrier material which is retained in the final structure. Whenever possible, it is advisable to use standard specifications and test methods as published by the American Society for Testing and Materials and the Federal government, but if these are inadequate or non-existent, they must be devised, prepared, and approved prior to proceeding further into a fabrication program. Since most composite materials are very expensive, and the labor associated with fabricated components even more so, it is not feasible to accept questionable incoming materials for production.

In-process Inspection

In-process inspection must start with the initial design and proofing of the fabrication tools and continue through the testing of control coupons from the finished composite parts. In-process control procedures include the recertification of materials which are known to change with time and temperature, accomplishing prefit operations prior to bonding of components within an assembly, processing test coupons of identical construction to the part on the same tool which can later be trimmed off for testing, and proofing the production tooling by prior fabrication of a part to the approved process specification and determining its conformance to both contour and tolerance. The part can be nondestructively tested and either statically tested or else cut into test coupons whose properties can be determined by testing. During the specimen cutting, observations should be made of the various sections to determine the fit of individual components and the condition of any adhesive or core material. This procedure should be repeated periodically, particularly after any tool modifications have been made, until the tolerance of consecutive tested parts indicates that complete uniformity of processing has been attained.

Nondestructive Testing

Obtaining and maintaining accurate NDT records is essential for composite structure certification and for diagnostic analysis whenever failure occurs. The knowledge gained is invaluable in the development of improved composite designs and acceptable field repair techniques. For the scrutiny of filament distribution and filament breaks, radiographic methods are preferred. In the case of boron filaments, single filaments even in multilayer composites can generally be observed under routine conditions. The most common technique for detecting debonds and delaminations is the ultrasonic pulse-echo method, in which a beam of ultrasonic energy directed into the material produces an echo when it encounters such defects. Analysis of the time difference between the echos from the far side and the imperfection yields information about the depth of the defect. Alternately, a complete scan of the surface (known as a C-scan) can be recorded on a two-dimensional plot indicating the projected area and location of the imperfection. In cases where delaminations extend to the surface, these can be detected by the use of dye penetrants. In addition to locating specific imperfections, the foregoing NDT methods as well as many others are being used in conjunction with destructive tests of composite structures in order to determine the limits and tolerances to which they can be allowed to remain or to which the selected NDT methods must be responsive.

Concluding Remarks

It has been thoroughly demonstrated that satisfactory composite components can be fabricated by the automatic layup, filament winding, or high pressure molding techniques, and that considerable weight savings can thereby be achieved. The development of new organic or even metal matrices and other fibers will increase the applicability of these fabrication techniques further. The high cost of such materials and processes is inhibiting their widespread acceptance, but progress is being made in reducing the raw material costs and in taking advantage of automatic fabrication of large numbers of identical parts. Gradual introduction of these and related materials and processes appears to be inevitable in both military and commercial applications within the next decade.

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