THE APPLICATION OF ADVANCED COMPOSITES TO MILITARY AIRCRAFT*

R.N. Hadcock

Director of Advanced Development

Grumman Aerospace Corporation

Bethpage, New York

Abstract

During the past decade, advanced composites have matured from high cost laboratory curiosities to fully qualified and accepted materials for lightweight, safety-of-flight aircraft structures. Many of these structures can now be produced at costs competitive with those of their metal counterparts.

This paper describes the evolutionary development of advanced composite technology which began at Grumman ten years ago, was committed to the use of boron/epoxy for the horizontal stabilizer of the F-14A in 1968, and is currently concerned with the application of mixed fiber (hybrid) composites to large, complex structures. Projected benefits of the extensive application of advanced composites to new aircraft systems are also described.

I. Introduction

The introduction of a new material and design concept to airframe structures has usually been a long and involved process, Figure 1. Early aircraft were made with wood frames, piano wire braces, and fabric covers; and that type of construction was still being used for some light aircraft in the 1940's. Welded steel framework began to replace wood framing by the early 20's. But wooden stressed skin, in the form of molded plywood, continued to be used for some parts well into the 50's.(1)

Although Dr. Hugo Junker's J1 all metal (predominantly aluminum) monoplane with cantilevered wings first flew in December 1915, it was not until the early 1930's that use of load bearing aluminum sheet became fully accepted as a means to provide light monocoque structure.

Since that time, the mechanical properties of the aluminum alloys have been improved, and integrally machined plates have replaced many built up sheet stiffened parts; however, many of the aluminum structures being produced today are generally similar to those built in the 30's. Though the use of titanium for selected airframe parts was introduced in the 1950's, the relatively high material and fabrication costs have limited widespread application to high performance aircraft structures.

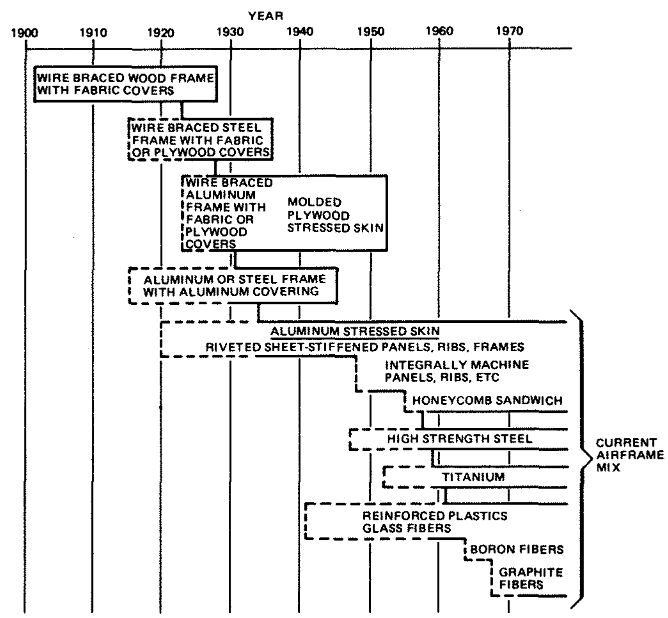

Fig. 1 Airframe Construction (Military Aircraft), 1903-1976

Fiber reinforced plastic airframe structures were first designed and built during the 40's. A Spitfire fuselage made from jute-fiber-reinforced plastic was successfully tested in England in

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1940; and a fiberglass reinforced plastic fuselage was first flown at Wright Patterson Air Force Base in March, 1944. Fiberglass reinforced plastic wings were flown on an AT-6 in 1953. Since that time, fiberglass reinforced plastics have been used extensively for radomes, rudders, flaps, and fairings. However, reliability and quality control considerations and the low stiffness and compression strength of those materials have generally limited their use to secondary structural applications for military aircraft, though they have been used successfully for the complete airframe of two light civil aircraft.(2)

The development, in the 1960's, of a process for manufacturing boron and graphite fibers, which are light in weight and have high stiffness as well as strength, presented the opportunity to develop airframe structures which were 15 to 35 percent lighter than their metal counterparts. This development, though, required that a very close interface be maintained between the various engineering disciplines and technologies - design, analysis, testing, materials, processing, manufacturing, and quality control - since the resulting advanced composite parts could not be used as "one-for-one," substitutes for parts made from metal.

II. Advanced Composites Development

The application of advanced composite materials to military aircraft structures was first studied at Grumman early in 1966, with the establishment of some mechanical properties and a design approach for boron/epoxy composite material. In order to generate the mechanical properties data, new test specimen configurations and testing techniques had to be developed since the unidirectional composite material is highly anisotropic; it possesses exceptionally high strength and stiffness in the fiber direction, but very low strength and stiffness when loaded transversely, or normal to the fibers. The high variability in both tension and compression strength which was characteristic of initial test specimen configurations eventually was minimized by changing the specimen configurations and by exercising care in finishing the free edges.(3)

The design approach that was established minimized the contribution of the epoxy matrix to in-plane strength and structural behavior.(4) The approach utilizes laminates in which the layers are generally oriented in each of four directions (0, 45, 90, and 135 degrees relative to the laminate axis). Such laminates have structural behavior which is essentially fiber controlled under uniaxial and combined loading conditions. This approach has proved to be very practical, since it allows the number of layers in each direction to be tailored individually to local load and stiffness requirements. The approach has also been very successful; no premature failures have been experienced during tests of more than 40 composite components and major subcomponents.

Design and manufacturing development has been performed in an evolutionary manner over the past ten years, Figures 2 and 3. The first major hardware program was the FB-111 Wing Box Extension Study in 1967.(5) Data generated in this program led to the 1968 decision to utilize boron/epoxy for the horizontal stabilizer of the Navy's F-14A Tomcat. The resulting F-14A composite stabilizer, Figure 4, is 180 pounds lighter than its metal counterpart.

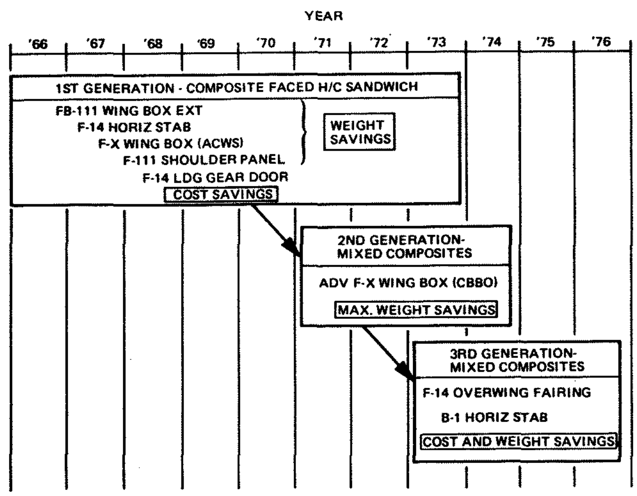


Fig. 2 Grumman Advanced Composite Development, 1966-1976

PROGRAM	MATL	FAIL LOAD, % DUL	MAX. DESIGN TEMP, ^0F	WT. SAVINGS, %	REMARKS
FB-111 WING BOX EXT (IR&D)	B/EP	120	260	17	RESIDUAL STRENGTH AFTER 4 LIVES OF FATIGUE TEST
A-6A WING FENCE (IR&D)	B/EP	100	160	--	FLIGHT TEST 1968
F-14 HORIZONTAL STABILIZER (NAVY)	B/EP	109	330	18	1ST FLIGHT 12/70 - IN PRODUCTION
F-X WING BOX (ACWS) (AFML)	B/EP	123.5	350	17	2ND BOX SUCCESSFULLY FATIGUE TESTED (4 LIVES)
F-111 SHOULDER PANEL, (IR&D)	GR/EP	--	260	20	MANUFACTURING DEVELOPMENT PROGRAM
F-14 MAIN LDG GEAR DOOR (IR&D/NAVY)	GR/EP	250	345	16	9 SETS WILL BE EVALUATED IN SERVICE
ADVANCED F-X WING BOX (CBBO) (AFML)	MIX	110*	350	28	B/AL UPPER COVER,B/GR/EP BEAMS, GR/EP SUBSTRUCTURE
F-14 OVERWING FAIRING (NAVY)	MIX	117	300	25	GR/GL/EP PANELS, GR-B/EP BEAMS, 2ND BOX SUCCESSFULLY FATIGUE TESTED, 5 SETS WILL BE EVALUATED IN SERVICE
B-1 HORIZONTAL STABILIZER (AFFDL)	MIX	132	260	15	GR-B/EP COVER, GR/EP SUBSTRUCTURE, 2ND STABILIZER SUCCESSFULLY FATIGUE TESTED.

*Percent of Predicted Failure Load

Fig. 3 Grumman Advanced Composite Programs

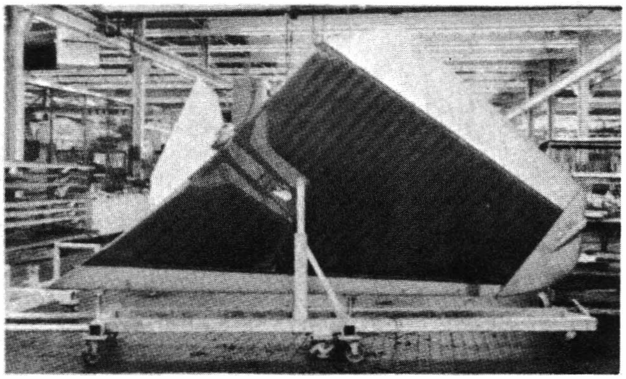


Fig. 4 F-14A Horizontal Stabilizer

The F-14A composite stabilizer was commited to production in 1969 after successful static and fatigue testing, and was first flown in December 1970.[6] These two events represent significant milestones for advanced composite safety-of-flight components - the first production commitment and the first flight testing. Currently, more than 230 aircraft sets of stabilizers have been produced; manufacturing costs have been lower than anticipated, and no problems have been experienced in service.

Evolution of boron/epoxy structures continued with award, by the Air Force, of the Advanced Composite Wing Structures Program in 1968. That program was successfully completed in 1971, which was about the time that the first significant advances were being made in the development of graphite/epoxy materials. One of the major problems associated with boron/epoxy was the relatively large fiber diameter (0.004 inch) which prohibited the use of the material for parts with sharp corner radii (e.g., channel section beams). The much smaller diameter of the graphite fiber (0.0008) posed no such problem, and permitted parts with complicated shapes to be made relatively easily. The formability of graphite/epoxy was used to effect a significant reduction in part count on the F-14A main landing gear door, Figure 5, with an associated reduction in fabrication and tooling costs. The Navy has now funded a program to obtain in-service experience on nine sets of these graphite/epoxy doors.

Later programs have focused on the use of mixtures of graphite/epoxy, boron/epoxy, and glass/epoxy to achieve both weight and cost savings. The first of these, the Air Force Composite Box Beam Optimization Program,[7] used a mixed composite (boron/epoxy and graphite/epoxy) for the lower tension cover, boron/aluminum for the upper compression cover, and graphite/epoxy for the substructure. The resulting box, Figure 6, was 28 percent lighter than its titanium counterpart. Both the lower cover and substructure had projected production cost savings; the upper, boron/aluminum cover was 50 percent lighter, but significantly more expensive, than the titanium cover.

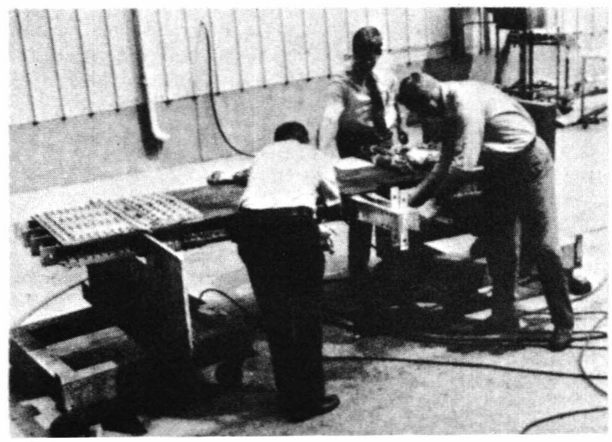


Fig. 6 Advanced F-X Mixed Composite Wing Box (CBBO)

The Navy F-14 Composite Overwing Fairing Program utilized graphite/, boron/, and glass/epoxy materials to reduce part count and cost.[8,9] The fairing, Figure 7, is 25 percent lighter

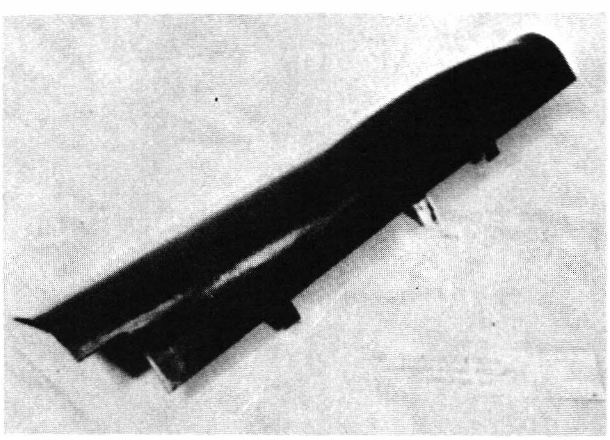


Fig. 5 F-14A Main Landing Gear Door

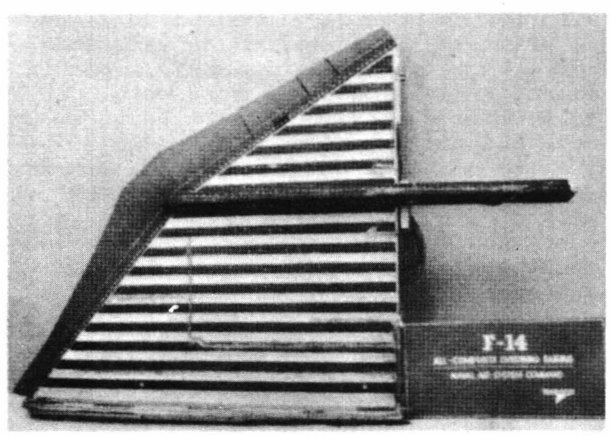


Fig. 7 F-14A Overwing Fairing

and 40 percent less expensive (on a part-by-part comparison) than its metal counterpart. Design conditions for the fairing, which blends the moving wing into the fuselage, are complex: The fairing must be sufficiently flexible to deflect 4½ inches when the wing is loaded in the fully swept condition. It must also be sufficiently stiff and strong for the trailing edge to maintain contact with the wing while it is carrying almost 20,000 pounds of aerodynamic suction loading. The tailorability of the mixed composite materials made them ideal for the overwing fairing application. Two fairings were built for static and fatigue testing; these tests were successfully accomplished at Naval Air Development Center. Five additional sets of fairings are currently being manufactured, and will be installed on aircraft to obtain in-service experience on mixed composite structures.

Composites technology developed under the Air Force Composite Box Beam Optimization Program and the Navy F-14 Overwing Fairing Program led to a concept for a mixed composite B-1 Horizontal Stabilizer; this concept was proposed to the Air Force in 1973 in response to their request for proposal on a program for Advanced Development of Conceptual Hardware.

III. B-1 Composite Horizontal Stabilizer

The USAF Materials Laboratory initiated a program with Grumman, in July 1973, to demonstrate that weight savings, reduced costs, improved damage tolerance, and vehicle performance improvements could be achieved by applying composite design concepts to the B-1 Horizontal Stabilizer.[10] The design of the stabilizer which was developed under this 42-month program is shown in Figure 8. The stabilizer has an area of 240 square feet, with a root chord of 17 feet and a length of 30 feet. Depth at the root is approximately 14 inches. The stabilizer pivots about a shaft which is mounted low on the Vertical Stabilizer, and is activated by two hydraulic cylinders.

The Horizontal Stabilizer composite torque box, Figure 9, was designed to minimize the number of detail parts, and thus reduce assembly costs. Bonded assembly sandwich and bonded metal splice plate construction techniques were eliminated, since both entail excessive costs on a structure of such a large size. Instead, the torque box is assembled by drilling the composite parts and fastening with A 286 stainless steel Jo-Bolts or titanium Hi-Lok fasteners.

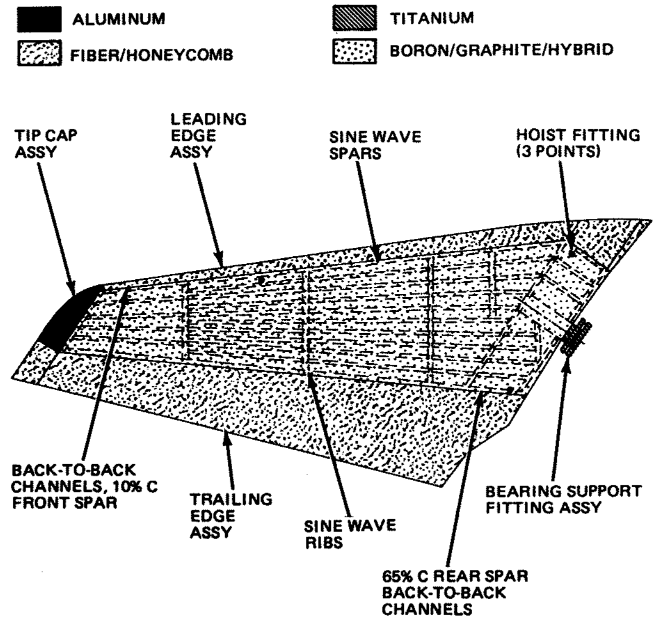


Fig. 8 B-1 Composite Horizontal Stabilizer

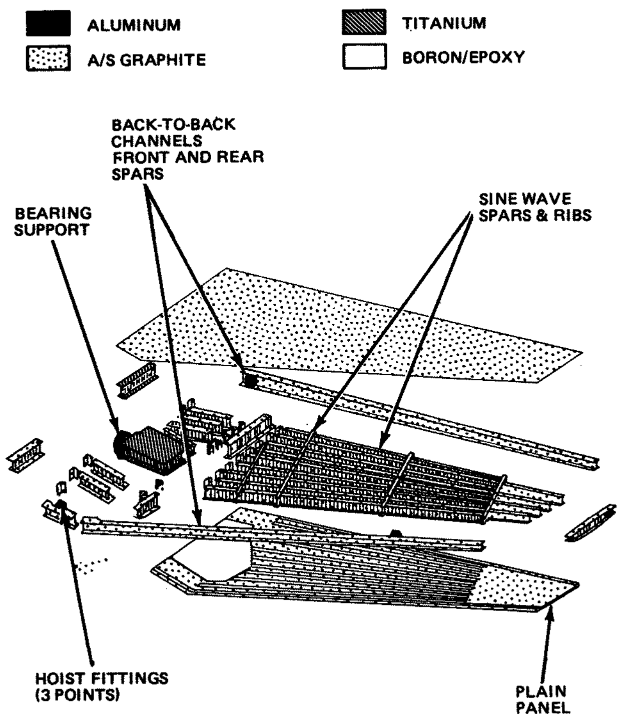


Fig. 9 B-1 Horizontal Stabilizer Composite Torque Box

The covers, which are the largest composite parts of the stabilizer assembly, are made from a mixture of 3501/AS graphite epoxy and Avco 5505 boron/epoxy. The basic skin, which is 0.19 inch thick, is composed of 36 graphite/epoxy layers. Boron/epoxy strips, submerged in the outer skin over the intermediate spars, provide longitudinal bending strength and stiffness; boron/epoxy plies laid into the inboard skin aid in load transfer from the covers into the bearing support fitting. A typical section through the outer portion of the cover is shown in Figure 10. Cover-to-spar attachment holes are drilled through graphite/epoxy strips which are interspersed between pairs of boron/epoxy strips, thereby eliminating the structural penalties associated with holes drilled in the boron/epoxy and reducing drilling costs. Approximately 90 percent of the cover attachment holes are drilled through graphite; only at the root are holes drilled through the 92-ply (0.5 inch thick) mixture of boron/ and graphite/epoxy.

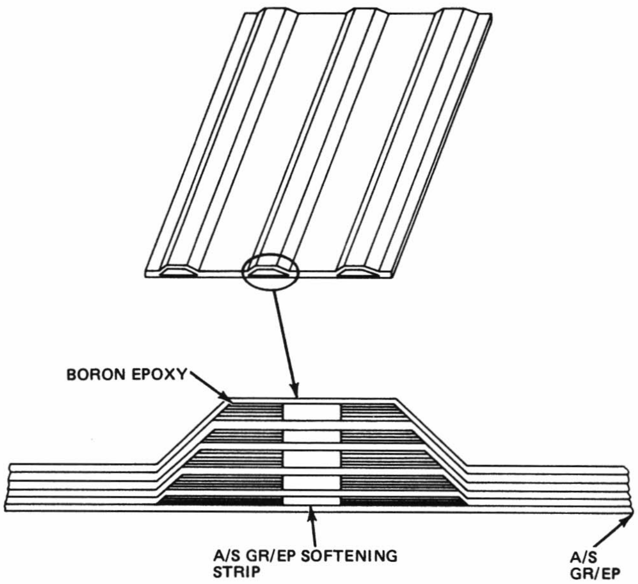


Fig. 10 B-1 Composite Horizontal Stabilizer Cover Details (Outboard)

The stabilizer substructure is composed of graphite/epoxy sine wave intermediate spars and ribs with flat-web front and rear spars. The sine wave intermediate spar and outer ribs, which generally have webs only 6 plies (0.03 inch) thick, are made in a single operation on split matched tooling. The front and rear spars, and some of the inboard ribs, are made by adhesively bonding pairs of channel sections and upper and lower cap strips together to form I-section components.

Design criteria for the stabilizer, developed from past experience in designing with composite materials, included use of fiber controlled laminates, laminate stacking sequences, ply drop-off rates, and unbuckled structure up to ultimate load. Design allowable properties were either taken from existing documents or were generated by analysis of the results of over 1000 element tests performed under the program. These properties, in conjunction with internal loads obtained by finite element analysis and flutter analyses, were used to size the covers and substructure. Several large structural elements were fabricated and tested under a variety of static, fatigue, and environmental conditions to verify design details. All of the tests, including acoustic fatigue, 200,000 amp lightning strike discharge protection (6 mil flame sprayed aluminum covering 50 percent of the covers), and fatigue tests of panels with simulated foreign object damage, were successful, with the exception of some tests made on sine wave spars. Local design and manufacturing changes have resulted in subsequent successful tests of these parts.

The first composite stabilizer was statically tested in February, 1976, Figure 11. After sustaining ultimate load in the three critical conditions, the stabilizer was loaded to failure; the upper compression cover failed at a location about 6 feet outboard of the root rib, at 132 percent of design ultimate load. In June, 1976, the second stabilizer completed a two lifetime flight-by-flight spectrum fatigue test, followed by limit load tests to both the positive and negative design conditions. No change in structural behavior was observed during or after the fatigue test.

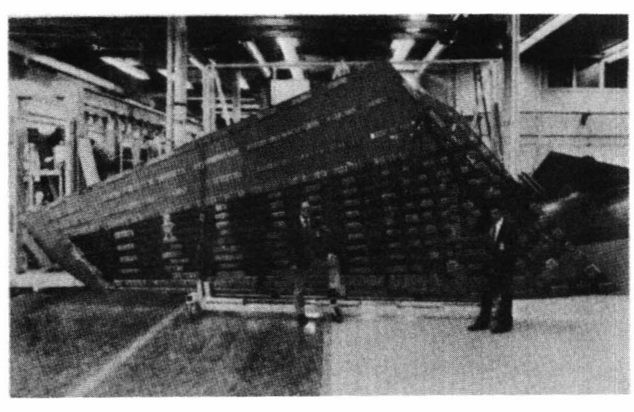


Fig. 11 B-1 Horizontal Test Stabilizer

Actual weights of the composite test stabilizer were 501 pounds less than that of the current metal configuration, Figure 12. This weight saving represents an additional 42 pounds over the goal established by the Air Force in February, 1974. Projected production costs of the metal and composite stabilizers are shown in Figure 13. As shown in the figure, the increased cost of the composite covers are more than offset by the reduction in substructure and assembly costs due to the far lower part count.

WEIGHT (LB/SHIPSET)		
COMPONENT	METAL	COMPOSITE
COVERS (INCLUDING SURFACE PROTECTION AND FASTENERS)	1687.6	1267.7
FRONT AND REAR SPARS	192.0	131.9
INTERMEDIATE SPARS	244.4	217.1
RIBS	153.1	124.5
MISC (SHIM AND HOIST FITTING)	.....	46.3
TOTAL BOX	2277.1	1787.5
LEADING EDGE, TRAILING EDGE, TIP	575.5	575.5
SEAL (INBOARD)	46.3	15.4
FINISH	40.3	40.3
BEARING SUPPORT FITTING	376.0	395.0
TOTAL	3315.2	2813.7*
GOAL		2856
SAVING OVER METAL DESIGN		501.5 (15%)
* ACTUAL WEIGHT OF STATIC (1400 LB) AND FATIGUE (1413.7 LB) ARTICLES		

Fig. 12 B-1 Horizontal Stabilizer Weight Comparison

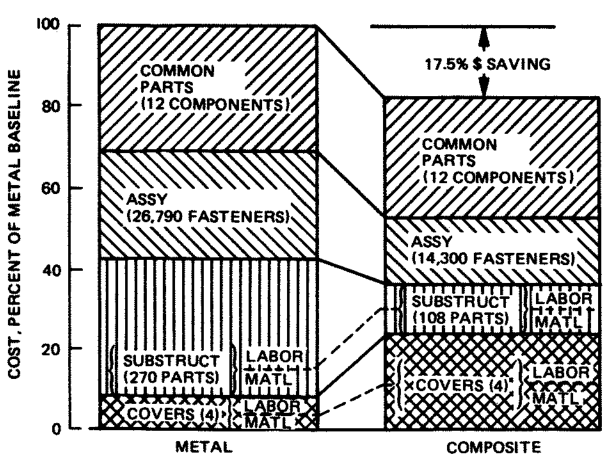


Fig. 13 B-1 Horizontal Stabilizer Production Cost Comparison

IV. Future Projections

Advanced composites are now accepted materials for the empennage of new fighter aircraft; the F-14, F-15, F-16, and F-18 all use composites to reduce empennage weight. Composite horizontal and vertical stabilizers are in strong contention for implementation in B-1 production. A number of different composite wing torque boxes have been made, and the use of graphite/epoxy is being considered for the covers of the F-18 wing box and for the complete torque box of the Advanced Harrier wing. Composite fuselage technology has been developed to the point where both feasibility and effectiveness have been demonstrated. The technology programs are providing the confidence needed for commitment to widespread utilization of composites on the next generation of high performance military aircraft.

Definition of the benefits and ramifications of unrestrained application of composites to a completely new supersonic interdiction fighter aircraft is the subject of a current Grumman study on an Advanced Design Composite Aircraft (ADCA), performed for the Air Force Flight Dynamics Laboratory.[11] The specific object of the study is to develop an airframe based on maximum use of currently available composite materials, and thereby reduce size, weight, and cost; such an aircraft would perform the same mission as its metal counterpart, but at a lower life cycle cost.

Configurations capable of performing the desired ADCA mission were evaluated using computer sizing programs. Primary input data included weight reduction factors for advanced composite materials, performance characteristics of candidate engines, mission requirement constants, and gross aerodynamic planform characteristics. The data shown in Figure 14

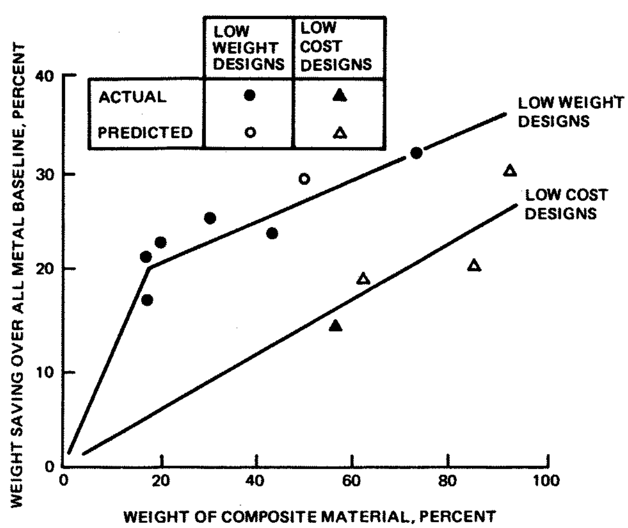


Fig. 14 Advanced Composite Empenneage Weight Savings

indicate a relationship between weight savings and the percentage, by weight, of composite material used in a component. Evaluation of such data results in the weight reduction factors shown in Figure 15. These factors are associated with low cost approaches and the overall goal of employing 75 percent composites, by weight, in the airframe. For completeness, Figure 15 also presents projected reduction for a 1980 advanced metal aircraft composed of 85 percent advanced metallic materials and 15 percent advanced composite materials.[12]

COMPONENT	WEIGHT SAVINGS, PERCENT	
	COMPOSITES	ADVANCED METAL AIRCRAFT
FIXED WING-NO TWIST CONT	28	9
-WITH TWIST Λ=20°	26.5	
40°	23	N/A
60°	17	
-DOUBLE DELTA	23.5	
SWING WING-NO TWIST CONT	20	9
-WITH TWIST CONT	17	N/A
TAILS AND CANARDS-SLAB	23	9
-FIXED	30	9
BODY	22	3
AIR INDUCTION-FIXED INLET	20	5
-VAR INLET	20	5
LANDING GEAR	16	10

Fig. 15 Composite Weight Reduction Factors

ADCA engine candidates were limited to those presently available or readily derivable by 1980. Since this requirement effectively narrowed engine choice to the F100, F101, and F404, the performance characteristics of these engines, suitably modified for installation effects, were used for the screening process.

Various aerodynamic configurations with different planforms were selected for screening. The delta planform exhibited the best supersonic characteristics, whereas the more conventional transonic swept wing configuration had the best transonic maneuver capability. Variable sweep was included since it enables the configuration to be matched to the various flight regimes. A trisonic wing configuration, which was an optimized combination of the delta and transonic configurations, was also included in the study.

The results of the configuration screening showed that a trisonic canard configuration and a transonic configuration with an aft tail, each with either one F101 or two F404 engines, were the lightest candidates. A more detailed study that was made of these aircraft showed that the trisonic canard configuration would be approximately 800 pounds lighter than the transonic configuration. The trisonic configuration thus was selected, and a further study made to determine whether to employ one F101 engine or two F404 engines. The configuration finally selected was the one F101 engine version shown in Figure 16.

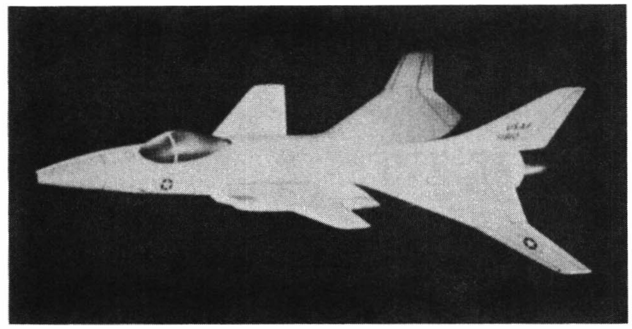


Fig. 16 Advanced Design Composite Aircraft (ADCA)

The general arrangement of the ADCA is shown in Figure 17. Subsystems and avionics, weapons, and crew systems are located in the forward section to provide easy access and maintenance. Flaperons and leading edge slats are used for both active camber and subsonic roll control. Slab canards provide pitch and supersonic roll control.

The ADCA airframe structural assembly is shown in Figure 18. This assembly maximizes use of large integrally molded components to minimize fabrication and assembly costs.

Of the various wing box structural configurations studied, the one-piece, through-wing design shown in Figure 18 appeared to be the lightest and least expensive. This design is based upon the use of plain graphite/epoxy covers supported by a multi-spar substructure. The wing box structure, shown in Figure 19, is optimized for strength and flutter. In addition, the spanwise cover layers in the outer portion of the wing are oriented forward at 15 degrees to the outer wing axis to provide a measure of aeroelastic tailoring. These off-axis plies couple bending and twist deformations to provide some additional, though small, aerodynamic benefits. The nose down tip twist induced by the wings bending upward results in a modest improvement in sustained maneuver capability. More significant benefits were found in a study of the effects of aeroelastically tailoring of the vertical stabilizer, Figure 20; that study showed that aeroelastic tailoring could improve both flutter speed and fin effectiveness, thereby enabling a smaller and lighter fin to be used.[13]

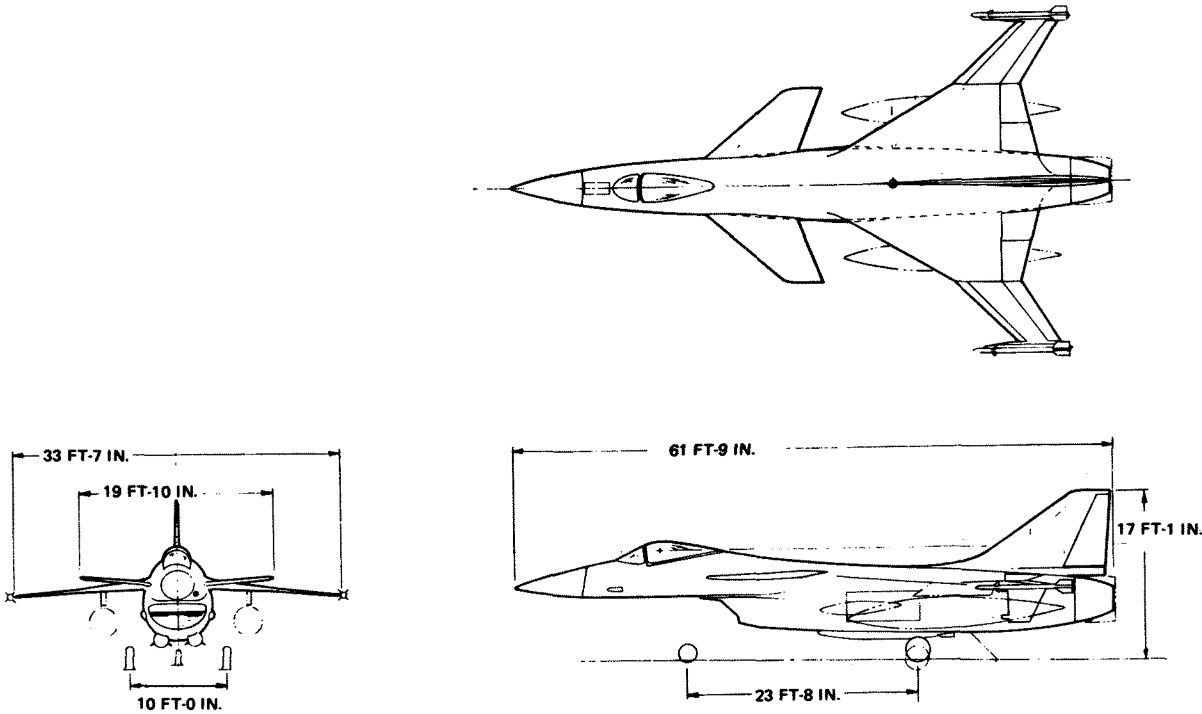


Fig. 17 ADCA Configuration

Various forms of construction were selected for the fuselage: The forward fuselage would be constructed of graphite-faced thin honeycomb sandwich panels with stiffening rings; the center fuselage, which forms a major fuel tank, would be constructed of a thicker honeycomb sandwich, supported by bulkheads and frames. Longerons would consist of a mixture of graphite/ and boron/epoxy. Access doors would be constructed similarly to the center fuselage, with edge protection added in the form of integrally molded metal foil or woven fiberglass/epoxy to minimize handling damage.

POINT	NUMBER OF PLIES					
	0°	-30°	-45°	-90°	-135°	TOTAL
1	29	–	8	5	8	50
2	46	–	10	5	10	71
3	43	–	8	5	8	64
4	22	–	11	6	11	50
5	38	–	21	6	21	86
6	41	–	14	6	14	75
7	12	–	15	6	15	48
8	30	–	25	6	25	86
9	35	–	18	6	18	77
10	30	60	–	19	6	115
11	28	76	–	19	6	129
12	28	50	–	19	6	105
13	23	30	–	23	4	80
14	20	2	–	20	2	44

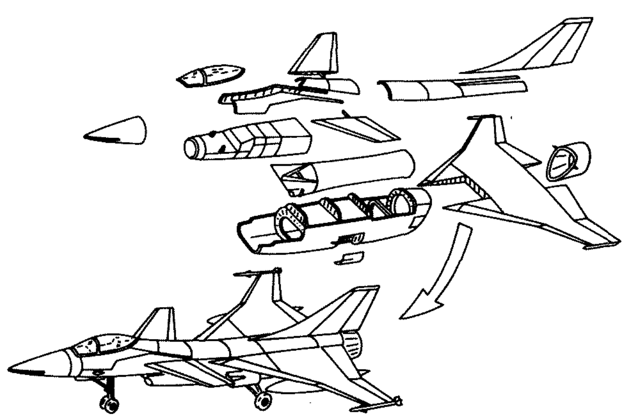


Fig. 18 ADCA Structural Breakdown

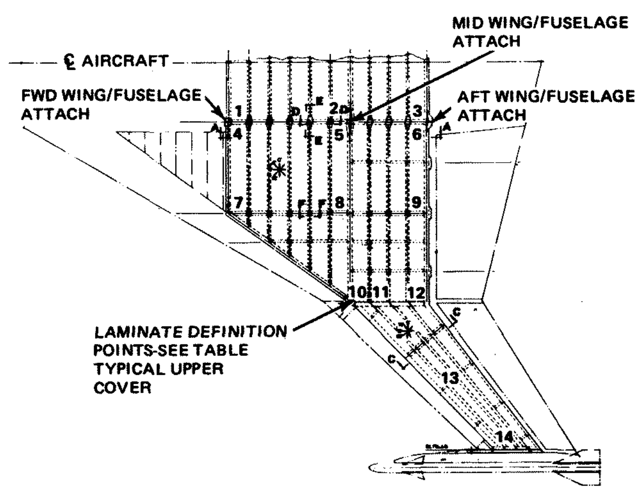
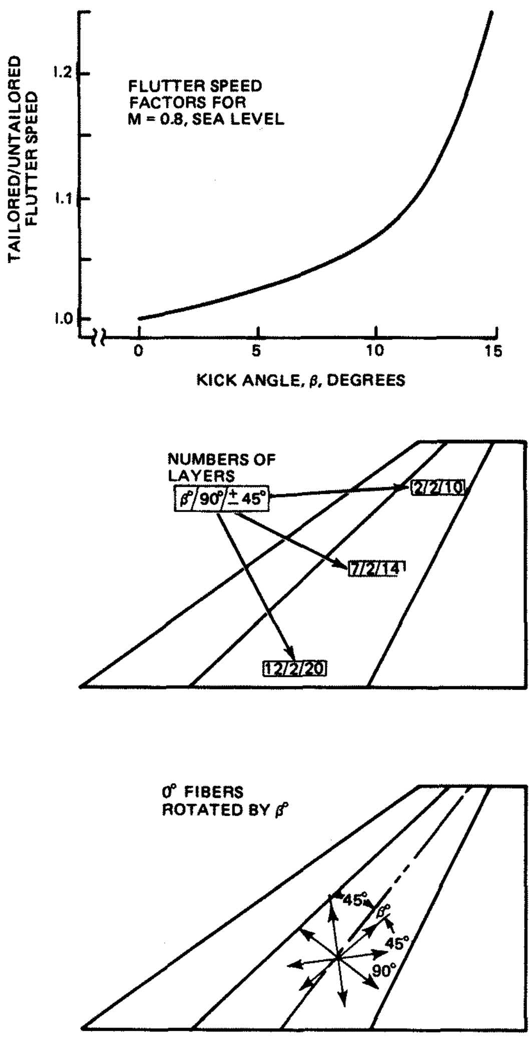


Fig. 19 ADCA Wing



NOTE: COVER LAYUP IS ALL GRAPHITE FOR STRENGTH AND STIFFNESS

Fig. 20 ADCA Vertical Stabilizer Flutter Speed Increase Through Stiffness Tailoring

As part of the ADCA study, a 1980 advanced metal aircraft capable of performing the same mission as the ADCA was sized using the projected weight reductions shown in Figure 15. Weights of the selected ADCA are compared to weights of the advanced metal aircraft in Figure 21. The advanced metal aircraft would be considerably heavier than the ADCA, and also larger; in fact, three F404 engines would be required to enable it to perform the same mission. An additional study was made to determine the weight of the advanced metal aircraft if composites were substituted for metal, with no change in aircraft size. This substitution aircraft has a 13 percent reduction in structural weight and a 6 percent reduction in takeoff gross weight. In contrast, the smaller ADCA has a 35 percent reduction in structural weight and a 26 percent reduction in takeoff gross weight. This analysis indicates that major benefits can by gained only by designing with composites from the outset.

A preliminary cost evaluation indicates that production costs for the ADCA should be 21 percent lower than those for the larger advanced metal aircraft. Moreover, the projected fuel savings of 30 percent may be very significant for aircraft operating in the 1980's.

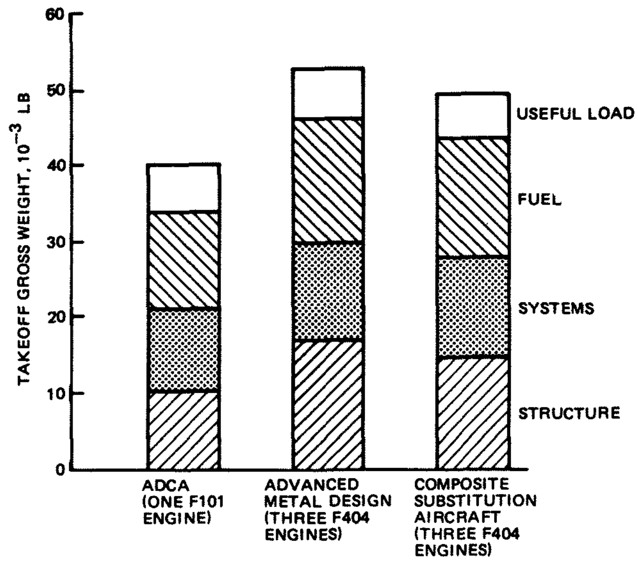


Fig. 21 Comparative Aircraft Weights

V. Conclusions

The weight savings that result from the application of composite materials to aircraft structures have clearly been demonstrated. A rapidly growing number of composite structural components thus will be entering service during the next five years. These components will provide the technical and cost data, as well as the confidence, needed for commitment to widespread application of composites to the next generation of military aircraft.

As described in this paper, the predicted weight, cost, and performance benefits of fully integrated and extensive application of composites to future high performance aircraft are truly significant. In fact, the application of composites may be critical to the performance of more demanding future mission requirements within any reasonable cost constraints.

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