ADVANCED MATERIAL APPLICATIONS TO SUBSONIC TRANSPORT AVIATION

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ABSTRACT

The application of advanced materials to subsonic transport airframes is identified as the technological area offering great potential for aircraft system improvements. The role of new emerging materials for subsonic transport aircraft structures is defined and evaluated. The effects of the application of these materials to commercial and military aircraft systems in terms of system economics and vehicle performance are quantitatively examined, including the impact on contributing engineering and other technical disiciplines. The benefits to be realized through the integration of the materials into a vehicle system are assessed in terms of system sensitivity to discrete selective usage. A plan for incorporation of advanced materials in subsonic transport systems is discussed. A series of R&D programs designed to develop the technology and demonstrate the inservice life characteristics is outlined.

INTRODUCTION

In the past 24 years, revenue passenger miles in world commercial aviation have increased by a factor of 26. During this period the flight safety of U.S. carriers has improved to a level where fatalities per million passenger miles have decreased to one-tenth the original rate.

Prognostications of air travel trends have been made in which increases of 47 percent are predicted by the year 1980. The improvements in the airplane responsible for these gains have been provided through the application of advanced technologies. Advanced technologies remain the key to the development of efficient and more productive aircraft in the future. Technology improvements are usually evolutionary in nature but in some instances provide quantum jumps in technology, such as provided by the advent of the jet engine. A similar potential for vehicle improvement is rapidly evolving within the structure's technology through the application of advanced material systems. The application of high-modulus fibrous composites and advanced metallic materials has demonstrated that vehicle and system benefits can accrue through usage of these unique materials.

The high strength and stiffness of the composite material combined with a density of 50 to 75 percent of that of aluminum provide the designer with the flexibility to tailor the structures to more efficiently satisfy the vehicle strength and dynamic requirements. Weight savings between 15 to 30 percent have been demonstrated on a wide variety of structural components with reductions up to 50 percent on some selective components.

Emerging metallic materials offer improvements in fracture toughness and corrosion resistance with minimum reduction in basic material physical properties.

The greatest contribution the structural designer can contribute to vehicle performance in the absolute sense is weight reduction.

In addition to performance benefits resulting from component and airplane weight savings, economy gains can be realized through the reduced acquisition cost of component fabrication. Existing and emerging manufacturing techniques offer potential cost reductions over conventional structure and include the utilization of large assemblies, automatic fabrication techniques, and reduced numbers of assembly elements.

The reduction in structural weight fraction initiates a cascading weight reduction in significant vehicle weight elements such as engines and fuel that generate increased economies in vehicle operation thus further lowering the vehicle operating cost.

Indirect benefits beyond those associated with the vehicle itself could include reduced runway, footprint, and length requirements, reduced consumption of fuel, and noise reduction.

The potential and application feasibility of the new material system have been analytically and experimentally demonstrated by technology development programs. Components employing advanced material systems are currently being introduced into man-rated vehicles.

The widespread acceptance of the new material systems will come about as the confidence level of the industry develops.

The application of new materials to an aircraft structure is best approached through the interaction of innovative design, efficient manufacturing methods, and emerging material systems.

The sequence followed in developing this thesis is through

1. A critique of some available emerging materials,
2. A consideration of the applications of emerging materials from the discipline involvement point of view,
3. An overview of four system studies in which the benefits of new material applications are assessed,
4. A discussion of the incorporation of advanced materials into man-rated aircraft structures, and
5. Some suggested areas of research to aid the development and ultimate employment of this technology.

EMERGING MATERIALS

Emerging metallic alloys and advanced fibrous reinforced composite materials are becoming increasingly available for subsonic aircraft design applications. The mechanical properties of these materials are such that they are capable of operating continuously at moderate elevated temperatures up to 250°F. In addition to these mechanical properties, certain physical properties are also required. Included among them are resistance to creep and corrosion, fracture toughness, crack propagation, and manufacturing properties allowing for low-cost forming, fabrication, and assembly techniques necessary for economic manufacture of aircraft components.

Among the metals, the advanced 7000 series aluminum alloys represent those which have shown marked improvement in some significant mechanical and physical properties over current alloys. While used much less extensively in subsonic vehicles, titanium alloys and steel alloys do offer some significant improvements for specific applications. In the range of 700 to 800°F, the titanium alloys are more efficient than stainless steel and aluminum alloys as a group. Aluminum alloys are clearly inferior to both the titanium alloys and Ph stainless steel at temperatures exceeding 300°F. The failure modes considered to be the most important are (1) tension, (2) high compression, (3) compression buckling, (4) shear, (5) fatigue, (6) damage tolerance (fracture toughness and fatigue crack growth rate), and (7) corrosion and stress corrosion (corrosion resistance and threshold stress).

Composite materials are among the oldest and the newest of structural materials. There is no all-inclusive accepted definition of composite materials. The generally accepted definition refers to the concept of something made of a variety of elements. If this general definition is applied to the entire structure hierarchy of materials, one ends in encompassing almost all materials. To be meaningful then, the definition of composites must be confined to the macro-level where one deals with constituents such as glass fibers, metal particles, and matrices.

There are two major reasons for the current interest in composites: (1) the demand for materials that will outperform traditional monolithic metallic materials, and (2) the flexibility that composites offer engineers to design totally new materials with the precise combination of properties needed for a specific task. Major constituents used in composites are fibers, particles, lamina, flakes, fillers, and matrices. The matrix is the body constituent that gives the composite its bulk form. The other five are the constituents that determine the character of the material's internal structure.

The three most widely studied systems are boron-epoxy, boron-aluminum, and graphite-epoxy. Their combination of strength, stiffness, and light weight exceeds that of any monolithic material. In addition to the high specific strengths and specific stiffnesses attainable with composite structures, composites contain a high level of intrinsic fatigue capabilities.

The excellent strength and stiffness properties of the advanced composites are accompanied by certain problems. The designer must contend with a material which is brittle, heterogeneous, nonisotropic, and which probably lacks sufficient mechanical properties data. Each of these problems introduces effects which must be accounted for and result in the stress analysis procedures being more complex for composite materials than for conventional metals.

APPLICATION OF EMERGING MATERIALS

The application of emerging materials clearly indicates a potential for weight saving and improved performance. The transformation of the materials into structural concepts involves a complex tradeoff between confidence levels, economics, and performance gains.

The standard measure of the work of different potential structural concepts is to evaluate the impact of each candidate design on the system direct operating cost (DOC). The direct operating costs reflect the overall costs of purchasing and using the aircraft.

Structural weight saving is therefore seen to be only one factor in the evaluation of new material and structural concepts. A total evaluation of material and manufacturing costs, engineering costs, weight saving, and confidence levels is required.

Since through the combined use of new materials, manufacturing methods, and design finesse, a synergistic effect can be obtained through which the benefit of the combination of the ingredients is greater than the sum of each individual contributor.

Benefits due to increased structural efficiency can accrue in the form of reduced component or vehicle absolute cost, or in system cost effectiveness. The only meaningful measure of the efficiency of a given system, however, is the effect on the total system over its entire operating period. Thus, a component or vehicle with lower acquisition cost is not necessarily a true yardstick of the economics of the air vehicle system.

The effect of application of new materials systems on contributing structural disciplines varies with the materials system itself. With most new metallic systems a minimum of procedural change will be required to perform the various tasks necessary by each discipline during the total hardware development process. In the matter of advanced composite systems, however, this is not true. The form of raw material, the intrinsic brittle nature of the composite material system itself, and other material physical and mechanical properties dictate several changes within the major disciplines.

DESIGN AND ANALYSIS

The advanced metallic materials will require little departure from normal design procedures. The greatest difference will be due to new fabrication techniques available for use with the new metallic systems. In the application of composite materials in structural designs, the designer must however become involved in a more sophisticated approach to the design process. The designer has two basic problems dealing with composite structures, one of designing the basic material itself and one of designing the element of hardware. The basic form of the raw material is such that the designer must compound the layup with the correct filament orientation and sequencing to provide the correct directional functional strengths and stiffnesses for the specific element's application. The reasons for this are quite simple if one understands that the designer is in fact creating a material family variance in the macro form. Thus, an additional step enters the trade study in that the designer, to properly formulate the load geometry constraints for individual elements, must now determine from the monolayer properties of the composite material the compound laminate materials which best meet the requirements of the final geometric hardware. It has been demonstrated that this additional step will increase design time by 25 percent.

The analysis associated with the application of new materials is highly dependent upon the basic nature of the material. The analysis associated with homogeneous isotropic metallic materials remains essentially unchanged. For composite structures, however, not only are standard analysis techniques more complex, but the material, part element interface requires additional analytical treatment.

A more sophisticated stress and stability analysis is necessary because the response characteristic of the materials is different than that of isotropic materials and is dependent upon the particular form of laminate pattern in question. Most laminates used exhibit either orthotropic, anisotropic, or pseudo-orthotropic characteristics, Figure 1.

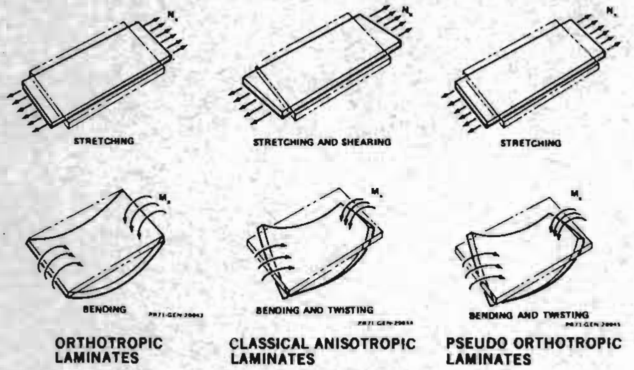


FIGURE 1. COMMON LAMINATE PLY SEQUENCING

The analysis of joints in composite structures requires considerably more analytical attention than metal structures since the brittle nature of the materials and the poor interlaminar strengths can introduce failure modes which were not previously considered. Strength and stiffness analysis is a new field which results from the fact that the materials are laminated from a series of orthotropic sheets which may be oriented in several directions within the plane of the laminate. The area of laminate pattern optimization has also been developed to enable the appropriate selection of orientations of each orthotropic sheet in the laminate for given load and stiffness requirements.

MANUFACTURING

Manufacturing as defined within this text includes the subdisciplines of tooling, fabrication, assembly, and quality control. The changes required to accommodate new material forms in the manufacturing discipline are greater than all other contributing disciplines. Although many new techniques for forming and joining metallic parts have been developed and utilized, the introduction of the new aluminum and titanium alloys has not demanded significant changes in personnel training or machine tool and forming equipment from previous requirements. It is with the introduction of composite structure that the greatest call for new manufacturing requirements has resulted. The form of the composite raw material is such that part fabrication is "built up" to the finished element. Each composite element is fabricated in a building block process through a selected fabrication technique until the completed element is formed. This is in stark opposition to that of metallic structure in which the raw stock is provided in its maximum dimensional form, and through the manufacturing process is reduced in size and weight to the required element geometry. This basic change in the fabrication process results in different material utilization factors. The average transport structure weight is 30 percent of the weight of the raw material stock from which the various elements were fabricated. Composite material utilization is 1.3 to 1.6 times the weight of the finished composite elements including inprocess quality control, scrappage, and resin loss.

The greatest impact on any manufacturing subdiscipline will be that of tooling in which experience to date has indicated a reduction in the number of tools required for a given composite component. The most significant manufacturing change to the aircraft fabricator will be in manufacturing facilities. Machine tools such as lathes, presses, and milling machines will still be needed; however, large automatic machine tools will give way to facility specialties such as large filament winding machines, automatic tape layup machines, and quickcuring devices for ultrasonic and infrared curing processes. Quality control and assurance processes will be required on a more frequent basis throughout the fabrication and assembly process, and will be of a more sophisticated nature and capability than those available today if rapid and economical production processes are to be realized.

SYSTEM STUDIES

To provide the tangible economic and performance benefits due to the application of advanced materials (both metallic and nonmetallic) to subsonic transport systems, studies were conducted on commercial (Reference 3) and military transport vehicles (References 1, 2, and 4).

ADVANCED CAPABILITY TANKER

An advanced capability tanker (Reference 2) version of the Douglas DC-10 was used as the baseline for a study to devise new design concepts having geometric arrangements with metallic material not previously used on aircraft of this class (Figure 2).

FIGURE 2. DC-10 ADVANCED TANKER VERSION

Evaluation of the new design concepts included manufacturing methods, applicability of nondestructive inspection (NDI), and production costs, in addition to weight and aircraft potential payoffs. Representative stations on the wing and fuselage were selected to size the structural elements of the new design concepts. The baseline tanker aircraft design criteria were applied with the addition of the latest Air Force damage tolerance criteria. Materials were selected from an extensive list of metallic candidates including weldable and nonweldable aluminum alloys, high-strength steel alloys, and titanium alloys. Filamentary composites were considered for local reinforcement of metallic structures. Manufacturing method evaluations were conducted for each new component design concept, including metal processing and fabrication. Selection of manufacturing methods was based on the estimated relative production costs.

Wing Box

Three basic wing box substructure concepts were evaluated: (1) the multirib, (2) truss-web, and (3) multiweb.

Multirib Box – Integrally stiffened titanium Beta C alloy was selected for the upper cover panel because of its peak material specific compressive yield strength and to attain a high torsional stiffness for the required compression strength. Aluminum alloy X7475-T6151 was selected for the lower panel because of its high tensile strength and fracture toughness value (similar to the baseline 2024-T351 skin alloy material). The upper panel is stiffened by integral L-section stringers (Figures 3 and 4).

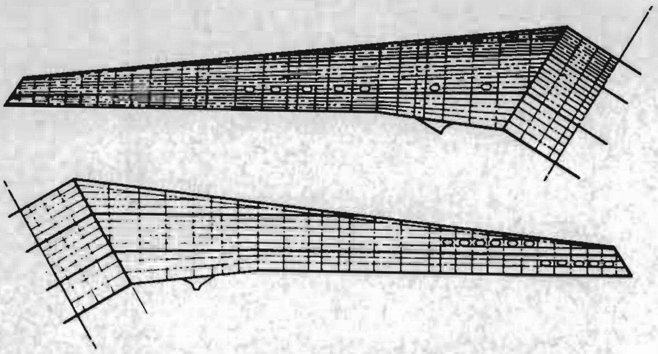


FIGURE 3. MULTIRIB CONCEPT WING BOX COVER PANELS

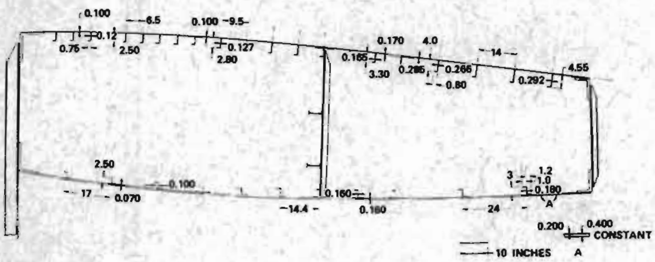


FIGURE 4. MULTIRIB CONCEPT WING BOX CENTERLINE STATION

The lower panel concept was selected as a stringer-stiffened adhesive-bonded honeycomb sandwich to (1) reduce the skin material thickness for increased material toughness, (2) reduce the number of stringers required, (3) reduce the number of fasteners and the associated "hole out" factor, and (4) allow an increased rib spacing while maintaining the required torsional stiffness of the box. The front and rear spar-webs were sized for shear strength and stiffness and incorporate a cross-truss geodetic stiffening geometry for the outboard wing box. The new rib concept utilizes the cross-truss arrangement formed with 7475-T61 aluminum alloy sheet trusses with the caps reinforced with graphite-epoxy. The upper and lower bulkhead caps are fusion- or flash-welded assemblies of Z-section titanium Beta C welded to Beta C forged fittings. A 7475-T61 sheet web is used at the forward end because of the higher shears. Chem-milling is employed for thickness variation with mechanical fasteners joining the parts.

Truss-Web and Multiweb Box Concepts – Truss-web and multiweb box concepts are similar in that both utilize spanwise members between the upper and lower cover panels to stabilize the panels. The concepts differ only in the orientation of the webs with the first of these forming a truss in chordwise section and the second having vertical webs. Preliminary comparison of the two web concepts indicated an estimated 7-percent weight advantage for the multiweb concept. The difference was primarily due to fewer spanwise joints and shorter posts (Figures 5 and 6).

Titanium Beta III alloy and 7475-T6151 aluminum alloy were selected for the upper and lower cover panels, respectively, to produce a direct evaluation comparison between the multirib and multiweb. Honeycomb sandwich was selected for the panels for its high compression load-carrying efficiency, to locate a maximum portion of the required box bending material in the skins, and to minimize fasteners. Spar concepts are the same as those used for the multirib box concept.

The web post honeycomb spanwise assembly is adhesive bonded and mechanically fastened along the lower tee and locally at the upper tee. Materials are 7075-T6 or aluminum alloy web post

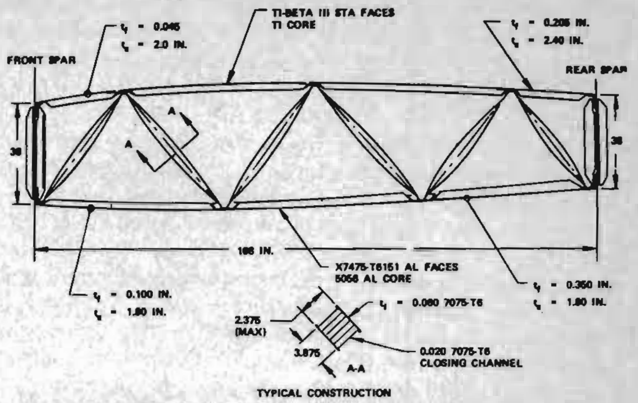


FIGURE 5. TRUSS-WEB CONCEPT WING BOX TYPICAL CROSS SECTION

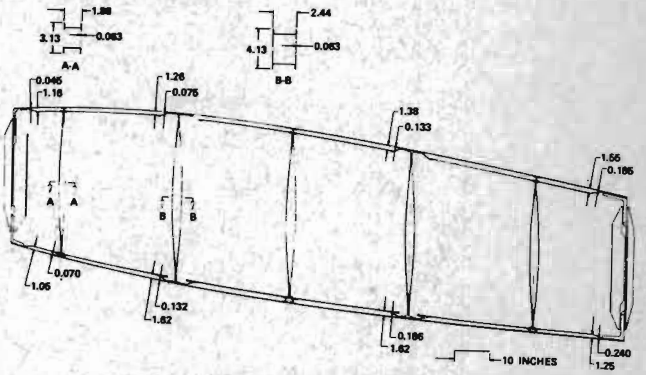


FIGURE 6. MULTIWEB CONCEPT WING BOX

skins, 7050-T6 lower extruded tee, and titanium Beta III upper extruded tee.

Bulkheads

The typical bulkhead is a combined truss and stiffened shear web. The higher shear areas at the front and rear spar are no-draft forgings of 7075-T73 aluminum alloy. Truss members between web posts are adhesive-bonded honeycomb assemblies of 7075-T6 face sheets and edge members, and 7050-T73 clevis end fittings.

Geodetic-Stiffened Wing Panel

Preliminary studies were conducted on a truss geometry-stiffened cover panel for the wing box. The concept is composed of a one-piece stiffener truss machined from a thick plate of aluminum alloy which is mechanically fastened to the skin. Applicability was for the wing box from the engine pylon inboard to the fuselage where a minimal torsional stiffness requirement applies. The concept does result in weight reduction of the basic panel structure. However, consideration of machining costs, a low material utilization factor, the requirement of an extra chordwise splice in each panel at the pylon, and the weight penalty for the attachment of bulkhead shear clips resulted in elimination of the concept for future effort.

Fuselage Shell Concept

Fuselage shell elements considered were the skin, longerons, and frames for the typical areas of the baseline fuselage having continuous hat section longerons through the frame and shear clip combinations. These areas also have titanium tear stoppers located at the frame.

Along with the weight efficiency for the hoop tension, longitudinal tension and compression, and shear loading, a prime design consideration for fuselage shell structure has been corrosion. Another important design consideration is that in large diameter fuselages, much of the shell is loaded at ultimate load intensities below 2000 pounds per inch in compression.

The design concept shown in Figure 7 was designed to economically satisfy the requirements but with a probable upper limit on weight efficiency. The skin and longeron panel consist of a weld-bonded assembly of three pieces. The Z-section longeron is formed from a single sheet and varies in height from the sheet thickness at the frames to a maximum between frames. A single piece doubler trimmed out between longerons and frames is located next to the skin. Materials for the three parts were selected as clad 7475-T61 for its combination of high compression yield strength, toughness, and corrosion protection. Weld bonding was selected to eliminate the rivet "hole out" factor and for its possible higher fatigue strengths for the skin in the hoop direction. In addition, the adhesive bond and the frame surfaces offer corrosion protection.

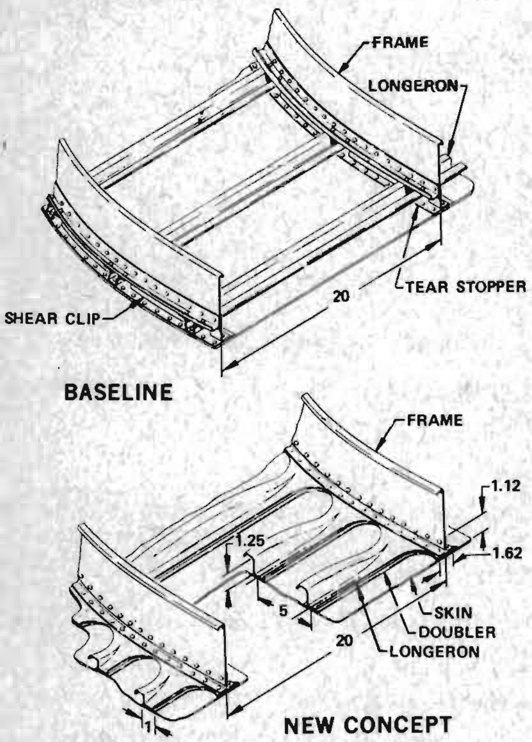


FIGURE 7. FUSELAGE SHELL BASELINE AND NEW CONCEPT

Aircraft Performance Payoffs

Performance payoffs for the tanker class aircraft are derived in two ways based on the weight reductions resulting from the incorporation of the new structural design concepts for the baseline wing box and fuselage shell. The first of these assumes that the aircraft size and takeoff gross weights are fixed, and therefore structural weight reduction results in either an increased fuel offload, increased range, or reduced takeoff field length. The second derivation assumes that resizing is possible to a constant mission and utilizes the growth factors which were derived during the preliminary design of the DC-10 aircraft. Weight reductions are 4205 pounds for the wing box new multirib concept, 3732 pounds for the new multiweb concept, and 1249 pounds for the new fuselage shell concept.

Payoffs Without Resizing

1. Fuel Offload Increase
   (Fuel offload increases include a tankage weight penalty of 0.35 pound per gallon of fuel added.)

New Concept	Increase (lb)	Percent of Baseline
(1) Wing multirib	4,275	1.55
(2) Wing multiweb	3,790	1.37
(3) Fuselage shell	1,305	0.47
(1) + (3)	5,580	2.02
(2) + (3)	5,095	1.85

2. Range Increase

New Concept	Increase (n mi)	Percent of Baseline
(1) Wing multirib	86	7.61
(2) Wing multiweb	76	6.72
(3) Fuselage Shell	26	2.30
(1) + (3)	112	9.91
(2) + (3)	102	9.02

3. Takeoff Field Length Reduction

New Concept	Reduction (lb)	Percent of Baseline
(1) Wing multirib	463	5.72
(2) Wing multiweb	411	5.21
(3) Fuselage shell	137	4.41
(1) + (3)	600	3.90
(2) + (3)	548	1.31

Payoffs With Resizing –

1. Takeoff Gross Weight Reduction

New Concept	Reduction (lb)	Percent of Baseline
(1) Wing multirib	9,250	1.55
(2) Wing multiweb	8,210	1.38
(3) Fuselage shell	2,750	0.46
(1) + (3)	12,000	2.01
(2) + (3)	10,960	1.84

2. Operating Empty Weight Reduction

New Concept	Reduction (lb)	Percent of Baseline
(1) Wing multirib	6,520	3.01
(2) Wing multiweb	5,820	2.69
(3) Fuselage shell	1,950	0.90
(1) + (3)	8,470	3.91
(2) + (3)	7,770	3.59

3. Wing Area Reduction

New Concept	Reduction (sq ft)	Percent of Baseline
(1) Wing multirib	84	0.94
(2) Wing multiweb	74	0.83
(3) Fuselage shell	25	0.28
(1) + (3)	109	1.22
(2) + (3)	100	1.11

4. Sea Level Static Thrust Reduction

New Concept	Reduction (lb)	Percent of Baseline
(1) Wing multirib	590	0.38
(2) Wing multiweb	520	0.33
(3) Fuselage shell	175	0.11
(1) + (3)	765	0.49
(2) + (3)	695	0.44

5. Takeoff Field Length Reduction

New Concept	Reduction (ft)	Percent of Baseline
(1) Wing multirib	34	0.32
(2) Wing multiweb	30	0.28
(3) Fuselage shell	10	0.10
(1) + (3)	44	0.42
(2) + (3)	40	0.38

Production Costs – The foundation for the production man-hour estimates for the new concepts is the accumulated Douglas experience and cost data in the manufacture of the DC-10 and previous transport aircraft. This experience includes the incorporation of titanium, adhesive bonding, and fatigue qualified fasteners in production. Manufacturing experience also includes the production use of large wing box skins (which are machine tapered from plate and shot peen formed), machining of long tapered section stringers from extrusions, and their forming by stretch wrapping, as well as normal miscellaneous sheet forming methods such as roll, brake, and hydroform.

Development of the man-hour estimates for the new concepts and materials was derived by using existing basic manufacturing method and material experience cost indices which were modified by the pooled judgments of experts within the manufacturing and process areas. These experts, in many cases, have experience in smaller scale experimental fabrication of laboratory type test components constructed by similar methods including atmospheric pressure diffusion bonding and weld bonding.

The estimating process and development of cost factors consisted of the following steps:

1. Subcomponent and element costs were developed (utilizing DC-10 baseline costs) and modified as described above. Man-hours per linear foot were developed for stringers and man-hours per square foot were developed for skins and sheet metal components.

2. Production learning curves were developed to obtain the cumulative average manufacturing costs.

The following production man-hours and dollar costs are estimated for the new design concepts:

a. Wing Box New Multirib Concept

	Hours	Dollars
Manufacturing	143,100	
Planning	10,700	
Tooling	10,400	
Inspection	15,700	
Materials		997,500
Total	179,900	

b. Wing Box New Multiweb Concept

	Hours	Dollars
Manufacturing	177,100	
Planning	13,300	
Tooling	12,900	
Inspection	19,655	
Materials		1,200,500
Total	222,955	

c. Fuselage Shell New Concept

	Hours	Dollars
Manufacturing	14,900	
Planning	1,100	
Tooling	700	
Inspection	1,500	
Materials		7,610
Total	18,200	

The production man-hours and costs for the baseline components are the following:

d. Wing Box

	Hours	Dollars
Manufacturing	122,500	
Planning	9,200	
Tooling	6,100	
Inspection	12,000	
Materials		350,000
Total	149,800	

e. Fuselage

	Hours	Dollars
Manufacturing	17,100	
Planning	1,300	
Tooling	900	
Inspection	1,700	
Materials		9,070
Total	21,000	

All of the above estimates are derived based on the following:

(1) The numbers are a cumulative average for 200-aircraft production run, using an adjusted learning curve for the new design concepts.

(2) The estimates exclude Design, Development, Test and Engineering, Production Tooling, and Facilities.

MEDIUM STOL TRANSPORT

A study was conducted on the advanced medium STOL transport (AMST) prototype (Reference 1) to examine possibilities for a reduction in structural weight fraction to enhance the productivity of the aircraft in that more payload and/or range be obtained for the same size vehicle. A structural weight reduction can additionally be the basis for resizing the aircraft to reduce production and life-cycle costs. An isometric view of the baseline metallic, externally blown flap (EBF) aircraft is illustrated in Figure 8. The configuration is characterized by a high wing, 4JT8D-17 engines and a T-tail, and features a large cross section fuselage, rear end cargo loading, and high flotation landing gears. A supercritical wing is utilized to provide reasonable cruise speeds and sufficient fuel volume for the ferry mission. The wing and horizontal stabilizer have straight leading and trailing edges, and rear spars which are normal to the aircraft centerline. The vertical stabilizer is a constant chord, constant thickness surface.

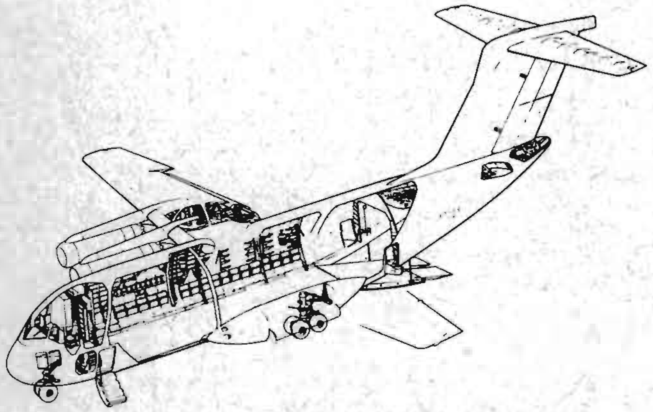


FIGURE 8. ADVANCED MEDIUM STOL TRANSPORT (BASELINE AIRPLANE)

The structural concepts of the baseline airplane were used for comparison in the study. This structure represents the current "state of the art" for design and manufacturing for the major components of the wing, fuselage shell, and empennage. The materials for all primary structures are aluminum alloys. The utilization of a particular alloy for a specific component has been determined by loading conditions or expected environmental use.

The study approach was to determine the capabilities and costs of the baseline structural concept and improve these concepts by integrating new structural geometries, new materials, and manufacturing advances.

Innovative Wing Panel Concepts

Computer-aided parametric studies were conducted to evaluate (1) weight efficiencies of the baseline concept and (2) selected new design concepts for a load environment representative of the AMST. Emphasis was placed on the wing upper and lower cover panels. Design concepts investigated were stiffened panels, honeycomb panels, corrugated unidirectional core sandwich, integrally machined sandwich, beryllium egg crate sandwich, and selective reinforced skin panels.

Stiffened Panel Concepts – Integrally (flanged) and Z-stiffened skin panels were selected as the most efficient stiffened panel concepts in compression (Figure 9). The skin was allowed to buckle to maintain the minimum stringer spacing. Fully effective skin was also evaluated to determine the effect of constraining the stringer spacing. The integrally (flanged) stiffened and Z-stiffened design concepts offer further possibilities for increased weight saving. Weight saving on the lower surface may be realized using adhesive bonding or weld bonding techniques that eliminate attachments. The absence of holes in the lower surface reduces the K_t factor for joints and allows the skin panels to be worked to a higher gross area stress level and still meet the required fatigue life.

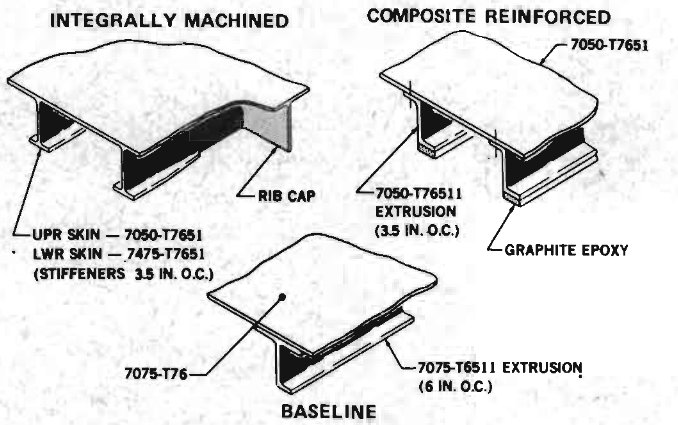


FIGURE 9. WING BOX COVER SKIN DESIGN CONCEPTS

Honeycomb Panel Concepts – Computer-aided analysis of honeycomb sandwich panels considered (1) adhesive system for the face-to-core joining of aluminum material, (2) dense core edge strips, and (3) mechanical fasteners along the edges. The use of honeycomb panels with the geometry and load intensities required in this application were nonoptimum as failure modes did not occur simultaneously (Figure 10).

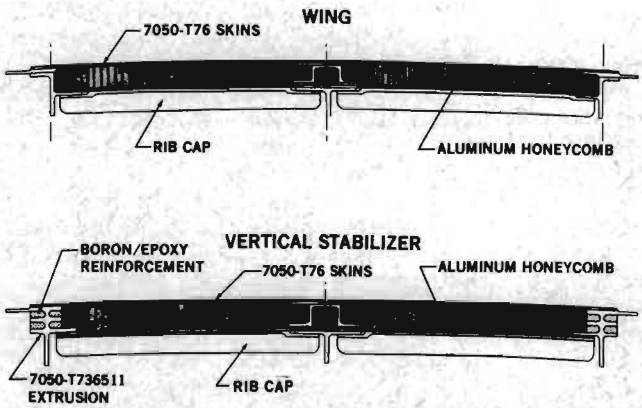


FIGURE 10. HONEYCOMB COVER SKIN DESIGN CONCEPTS

Corrugated Unidirectional Core Concepts – This concept permits the core as well as the face sheets to resist the uniaxial loads in the panel. Weld or spot welding techniques could be employed with any of the candidate materials (Figure 11).

Integrally Machined Sandwich – The sandwich panel consists of machined upper and lower skins spotwelded or bonded together. The inner skin has bulkhead caps integrally machined or bonded in place. Spanwise stiffeners were provided to make the skin material fully effective for the compressive load. Chordwise gussets were added for shear stability of the panels (Figure 12).

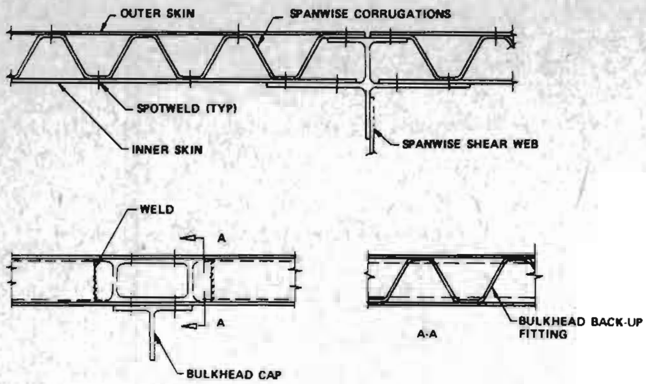


FIGURE 11. CORRUGATED CORE SANDWICH

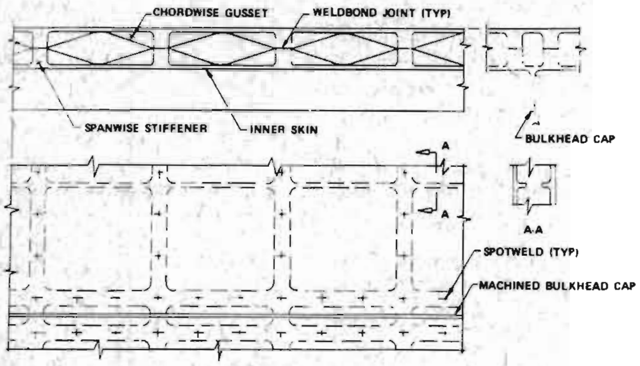


FIGURE 12. INTEGRALLY MACHINED SANDWICH

Beryllium Eggcrate Sandwich Concept – The all-beryllium sandwich design features spanwise and chordwise stiffeners that are intermeshed through a series of machine cuts in the stiffeners joined by adhesive bonding at stiffener intersections. The spanwise stiffeners are spaced to make the outer interface sheets fully effective for the compressive load. Chordwise stiffeners are provided for shear stability of the panels (Figure 13).

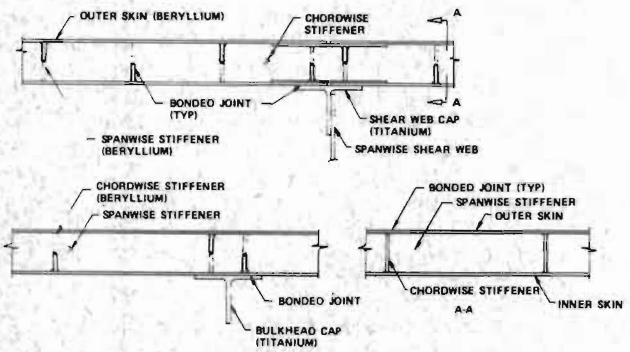


FIGURE 13. BERYLLIUM SANDWICH

Selective Reinforced Panel Concepts – Two concepts under study utilize skin and stringer construction. One concept consists of an integrally machined skin and stringer panel with selective reinforcement of graphite-epoxy tape added to the stringers. The second concept utilizes basic skin and stringer construction with a skin consisting of two face sheets of aluminum or titanium with a layer of graphite-epoxy tape bonded between.

Fuselage Panel Concepts

Five innovative fuselage structural shell concepts were investigated to evaluate relative panel weights as a function of material and geometry.

Stiffened Panel Concepts – The first new concept was simply an increase in the spacing of the longerons around the circumference of the shell and consisted of a combination of 7075-T761 skins and 7075-T6511 longerons.

Isogrid Panel Concepts – A second concept employed a simple integral isogrid scheme (Figure 14). Three different materials were evaluated for use in this application. Various geometric dimensional values were evaluated in combination with the materials. Shell weights are based on the assumption that intermediate frames are not required.

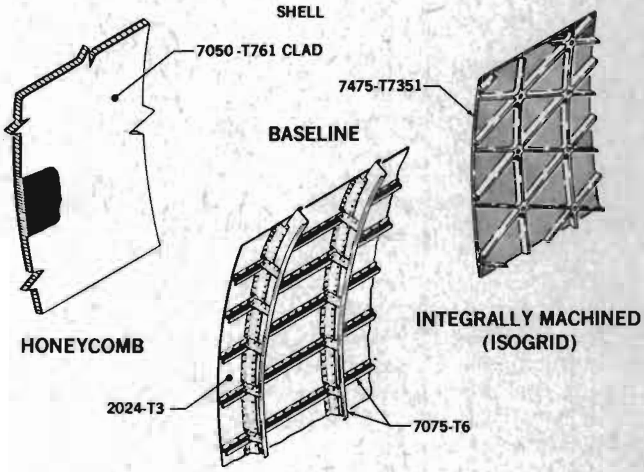


FIGURE 14. FUSELAGE DESIGN CONCEPTS

Honeycomb Sandwich Panel Concept – A third new concept is the honeycomb sandwich (Figure 14). Aluminum 7050-T76 and titanium 6Al-4VA were evaluated and compared. The results indicate that for the low load intensities experienced by the fuselage shell the aluminum sandwich panel exhibits a weight saving (20 to 28 percent) while the titanium sandwich imposes a weight penalty up to 12 percent. A face sheet thickness of 0.030 inch minimum was assumed in recognition of the practical considerations of damage tolerance and fatigue associated with primary structure as it relates to pressurized shell design.

Integrally Stiffened Panel Concept – The fourth concept is an integrally stiffened panel utilizing 7475-T761 plate incorporating J-section longerons spaced at 10, 15, 20, and 25 inches. Weight saving ranged from less than 7 percent to 12 percent.

Selectively Reinforced Panel Concept for Cargo Floor

Boron-epoxy composite reinforcement infiltrated into the hollow openings of 7075-T6511 extruded planks and channels is utilized in this design concept. The epoxy matrix is room temperature cured and post-cured at 250°F. The result is a composite reinforced aluminum member with no measurable distortion or locked-in residual stresses attributable to thermal mismatch. The average weight saving for the boron-epoxy reinforced floor structure is 9 percent (Figure 15).

Horizontal Stabilizer Structure

A honeycomb skin panel concept was evaluated that employed 7050-T76 chem-milled tapered skin panels with an aluminum honeycomb core of 3.8 pounds per cubic foot density. The panel width was reduced by the addition of a lightweight center spar. This method eliminated the need for stringers and 10 intermediate ribs per horizontal stabilizer. The resulting weight was 1.26 pounds per square foot, thus providing a weight saving of 17 percent over an integrally stiffened skin plank concept.

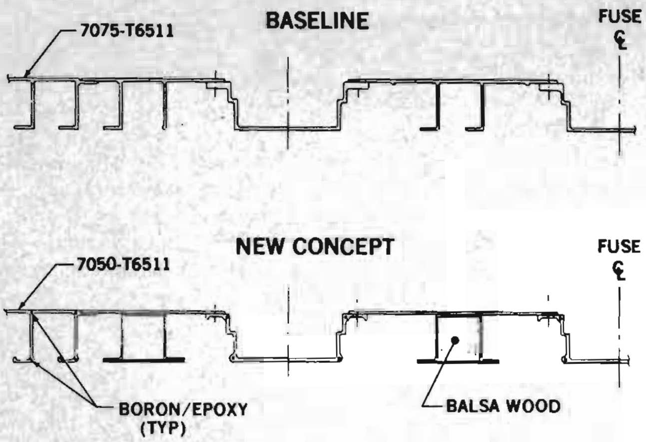


FIGURE 15. CARGO FLOOR DESIGN CONCEPTS

Spar Caps – The machined spar caps are made from 7050-T76511 aluminum extrusion and are bonded to the honeycomb skin planks during the same curing cycle.

Ribs – The only ribs in the horizontal stabilizer resist either hinge or actuator loads. Tension field ribs met the criteria more satisfactorily than other concepts evaluated. The ribs are fabricated from machined integrally stiffened 7075-T736511 aluminum plate, Figure 16. To keep web thickness to a minimum necessitated chem-milling after machining to reduce distortion and minimize residual stresses.

Spar Webs – The spar webs were evaluated as an open isogrid concept utilizing 7075-T73651 aluminum plate. On comparison with an integrally machined tension field spar web, the tension field method exhibited a weight saving and therefore was adopted (Figure 16).

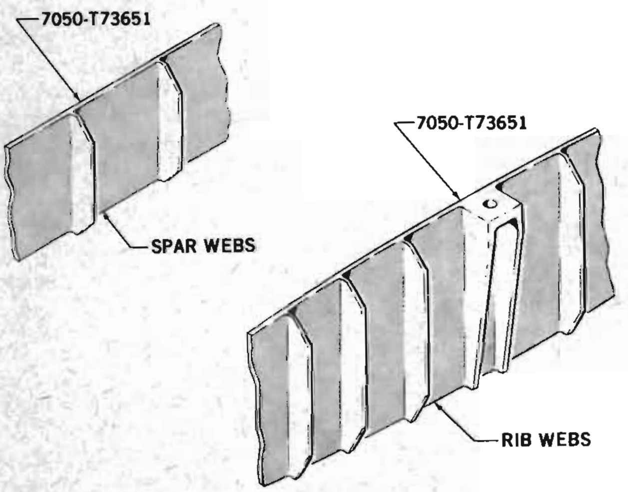


FIGURE 16. EMPENNAGE SUBSTRUCTURE

Vertical Stabilizer Structure

The vertical stabilizer structure (Figure 10) is essentially the same as the horizontal structure with the exception of the forward and rear spar caps which are reinforced with boron-epoxy to provide additional stiffness for meeting flutter requirements.

Performance Analysis

The improvements in aircraft performance of two AMST vehicles featuring new design concepts with attendant reduced structural weights relative to the base metal aircraft were defined. The performance analysis was consistent with the methods used in defining the base metal aircraft. Empennage areas were sized based on limiting stability and control requirements.

The structural concepts employed in the two vehicles are shown below:

Vehicle A.	Honeycomb Sandwich Fuselage Shell Integrally machined wing cover panels Honeycomb sandwich empennage cover panels Composite reinforced floor
Vehicle B.	Isogrid Fuselage Shell Integrally machined wing cover panels Honeycomb sandwich empennage cover panels Composite reinforced floor

Performance (Vehicle A) – The performance improvement options for payload capability, radius capability, and reduced field length for Vehicle A, which provides a structural weight reduction of 1850 pounds, are shown in Table 1.

TABLE 1
PERFORMANCE IMPROVEMENT OPTIONS (VEHICLE A)

	MIDPOINT WEIGHT (LB)	PAYLOAD CAPABILITY (LB)	RADIUS CAPABILITY (N MI)	FIELD LENGTH MIDPOINT (SL 103°F) (FT)
BASE METAL AIRCRAFT	150,000	27,000	400	2000
OPTION 1	150,000	$\boxed{28,000}$	400	2000
OPTION 2	150,000	27,000	$\boxed{458}$	2000
OPTION 3	147,990	27,000	400	$\boxed{1958}$

Performance Vehicle A (Resized and with Fixed Engine) – The resized vehicle configuration was sized to minimize weight and cost consistent with the base metal aircraft performance requirements. The wing and empennage area and engine size were reduced to match the field-length requirement (2000 feet, SL 103°F) and mission capability (400-nautical-mile radius with 27,000-pound payload). A structural weight saving of 3390 pounds was attained when completely resizing the aircraft. However, a loss in wing fuel volume and, hence, ferry range capability, results with the resizing.

The fixed engine size concept was sized to minimize weight and cost by reducing wing and empennage area. This allows a greater reduction in wing and empennage area relative to the completely resized concept, when sizing to consistent field-length and mission requirements. A structural weight saving of 3150 pounds was obtained, but a further decrease in ferry-range capability resulted from reduced wing area.

Performance (Vehicle B) – The performance improvement options for payload capability, radius capability, and reduced field length for Vehicle B, which provides a structural weight reduction of 1080 pounds, are shown in Table 2.

Performance Vehicle B (Resized and with Fixed Engine) – The resized vehicle concept was sized to minimize weight and cost, consistent with the base metal aircraft performance require-

TABLE 2
PERFORMANCE IMPROVEMENT OPTIONS (VEHICLE B)

	MIDPOINT WEIGHT (LB)	PAYLOAD CAPABILITY (LB)	RADIUS CAPABILITY (N MI)	FIELD LENGTH MIDPOINT (SL 103°F) (FT)
BASE METAL AIRCRAFT	150,000	27,000	400	2000
OPTION 1	150,000	$\boxed{28,080}$	400	2000
OPTION 2	150,000	27,000	$\boxed{458}$	2000
OPTION 3	148,830	27,000	400	$\boxed{1975}$

ments. The wing and empennage area, and engine size were reduced to match the field-length requirement (2000 feet, SL 103°F), and mission capability (400-nautical-mile radius with 27,000-pound payload). A structural weight saving of 1970 pounds was attained with the completely resized aircraft. However, a loss in wing fuel volume and, hence, ferry-range capability, results from resizing.

The fixed engine size concept was sized to minimize weight and cost by reducing wing and empennage area. This allows a greater reduction in wing and empennage area relative to the completely resized concept, when sizing to a consistent field length and mission requirements. A structural weight saving of 1850 pounds was obtained with a decrease in ferry-range capability due to reduced wing area.

COMPOSITE MEDIUM STOL TRANSPORT

A program (Reference 4) was conducted to develop advanced composite airframe design concepts offering high payoff in terms of vehicle performance improvements, increased reliability, reduced cost, and to develop airframe concepts from the standpoint of manufacturing cost reduction. The baseline airframe for this effort was the AMST.

A preliminary composite aircraft configuration was required to scale the loads from the metallic baseline airplane. Aircraft resizing was performed based on a 12-percent reduction in manufacturer's empty weight, while maintaining the same field-length and design mission performance as the metallic aircraft. The resized aircraft has an 8-1/3 percent lower takeoff gross weight, wing area, and engine size than the metal baseline as shown in Table 3.

High-strength graphite epoxy (Thornel 300/5208) was the primary composite material selected for use in this program on the basis of relatively low cost and adequate performance.

Wing and Empennage Concept Selection

Initial design activity evolved the design of a variety of composite structure elements for evaluation by engineering and production disciplines. The concepts included primarily wing box and empennage box geometry arrangements falling into four main categories: truss-web variations, truss-rib, truss-spar and multirib designs, Figures 17, 18, and 19. Accompanying the box concepts was a series of cover stiffening and substructure stiffening concepts.

TABLE 3
BASELINE METAL AND INITIAL COMPOSITE AIRPLANE COMPARISON

	BASELINE	COMPOSITE
WING AREA (FT²)	1,740	1,596
THRUST/ENGINE (SLS, LB) (JT8D-17 TYPE)	16,000	13,660
TOGW (MIDPOINT) (LB)	150,000	137,400
W/S (MIDPOINT)	86	86
T/W (MIDPOINT)	0.40	0.40
PAYLOAD (LB)	27,000	27,000
DESIGN RADIUS (N MI)	400	400
AT MACH NO.	0.70	0.70

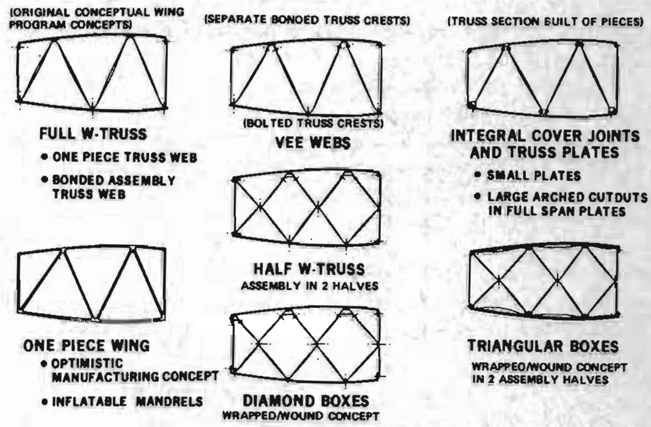


FIGURE 17. TRUSS-WEB ALTERNATES

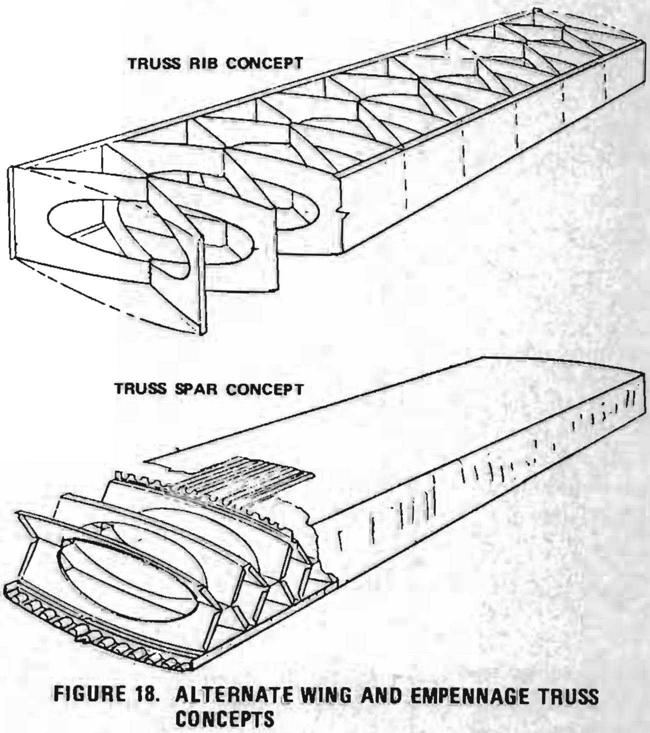


FIGURE 18. ALTERNATE WING AND EMPENNAGE TRUSS CONCEPTS

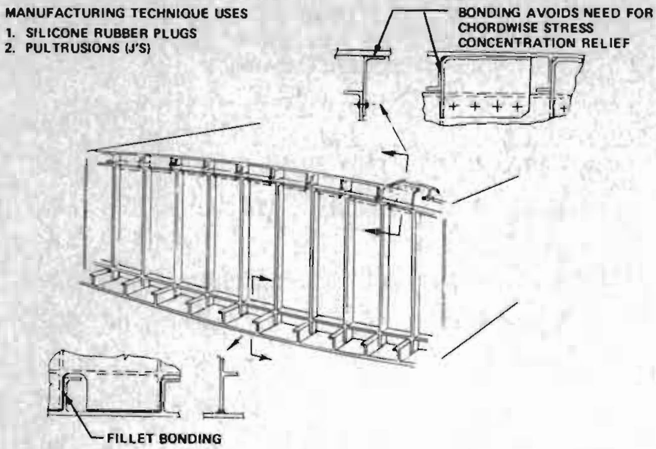
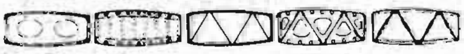


FIGURE 19. MULTIRIB SOLID LAMINATE WING CONCEPT

The preliminary evaluation of box and panel stiffening concepts was aimed at choosing combinations that presented the greatest potential for low-cost construction without going into detailed estimates. Long-term cost items (maintainability and reliability) were also rated. Preliminary concept selection criteria were used as follows: (1) fabrication, (2) assembly and tooling costs, (3) structural reliability (ease of incorporating inherent failsafe, and fatigue characteristics), (4) maintainability and repairability, (5) environmental vulnerability, (6) manufacturing feasibility, and (7) component weight, and fuel volume in the case of wing components.

The relative weights of the five final selections are shown on Table 4.

TABLE 4
RELATIVE WEIGHTS – WING BOX CONCEPT FINAL SELECTIONS



	A	B	C	D	E
COVERS	1.00	1.13	0.96	1.20	1.89
SUBSTRUCTURE	1.00	1.13	1.09	1.14	1.12
BOX	1.00	1.13	1.00	1.18	1.63

A solid laminate multirib with conventionally shaped hat stiffeners costs approximately 7 percent less than the sandwich multirib, primarily because of the lower fabrication labor cost and in spite of an increased composite material cost for the solid laminate design. The truss web, denoted C, was 16 percent less cost than the D truss web and 28 percent less cost than the competing solid laminate multirib. The C truss web was thus selected for potentials of low cost and equivalent weight. The proposed long-term savings in maintenance and repair costs through use of solid laminate construction over honeycomb construction did not appear quantifiable in the initial evaluation. Therefore, the short-term cost saving design selection rationale was adopted.

The truss web adapts well to the eccentric elevator hinging requirements on the horizontal tail. The vertical stabilizer will be detailed as a sandwich panel multirib design since it has symmetrical rudder hinge requirements, and the use of honeycomb panels in the substructure offers less problem where there is no fuel.

In spite of somewhat increased composite materials costs for the solid laminate construction, derby-hat-stiffened solid laminate shear webs are to be used for all wing and empennage box structures because of the many subsystem attachments to these webs, and because of the hinge, flap, and slat loads introduced through them.

Wing Box Fabrication Concept

The wing box skins are of a sandwich design that incorporates a constant tapered, corrosion-resistant aluminum honeycomb core bonded to inner and outer graphite-epoxy skins. The aluminum honeycomb core is rough machined with constant taper from inboard to outboard in the HOBE (honeycomb before expansion) condition and the core is electrical discharge machine (EDM) finished and prepared for bonding.

The inner and outer wing box skins use basically the same ply-orientations throughout, tapering in thickness from inboard to outboard. The solid laminate portions of the full-length skins are automatically tape laid on plastic laminated molds at the rate of 60 feet per minute on a computerized tape machine. A time interval of 15 seconds is required for acceleration, deceleration, cutting, and indexing of 1-foot-wide individual tapes. The skins are densified and staged prior to transferring from the plastic laminating mold (PLM) to a trim fixture by a conveyor belt built into the PLM.

The front and rear spars are automatically tape laid in the same manner as the wing box upper and lower skins. The spar webs are stiffened with integral derby-hat stiffeners. The inner hat stiffener plies are automatically laid in a flat pattern. The hat stiffener layup is densified, staged, heat formed to configuration, and finally cocured and bonded to the spars flat laminate section.

Fuselage Concept Selection

Three concepts were candidates for fuselage construction: (1) the arch frame, (2) the thick honeycomb, and (3) the isogrid. As with the wing boxes, fabrication man-hour estimates were obtained for the three fuselage shell sections. Corresponding weight estimates for the sections, including one circumferential end joint, were obtained and are shown in Figures 20, 21, and 22. Relative weight and labor and material costs for 100 units are shown in Table 5. Tooling cost estimates were not obtained so the comparison was made on a basis of factored labor and material costs ($20.00 per pound composite) only. The indicated costs of the honeycomb shell and isogrid are so similar that other factors such as weight and environmental resistance (which are apparent long-term cost factors) must be considered in the selection. The isogrid concept was the recommended selection for the fuselage shell construction.

Isogrid Fuselage Fabrication Concept – The isogrid fabrication concept employs multiple banding heads that automatically lay a complete layer of ±30-degree legs of the equilateral triangles in the grooves of the inflatable mandrel. Individual 90-degree hoops (the third leg of the isogrid triangles) will be tape wound, once for each layer of ±30-degree pattern. After densification of the isogrid pattern, the outer skin is automatically wound prior to installation of the segmented exterior mold. The tape wound fuselage is vacuum sealed to the exterior mold. As the temperature and pressure are increased, the inflatable mandrel is vented to autoclave pressure. The skin and isogrid are cocured and bonded at 350°F and 100-psig pressure. The exterior mold is removed and the fuselage section is placed in an assembly fixture. The mandrel is deflated and removed (Figure 23).

Aerodynamic Trade Studies

The effect of aspect ratio on the aircraft weight is illustrated in Figure 24. As shown, there is essentially no difference in gross

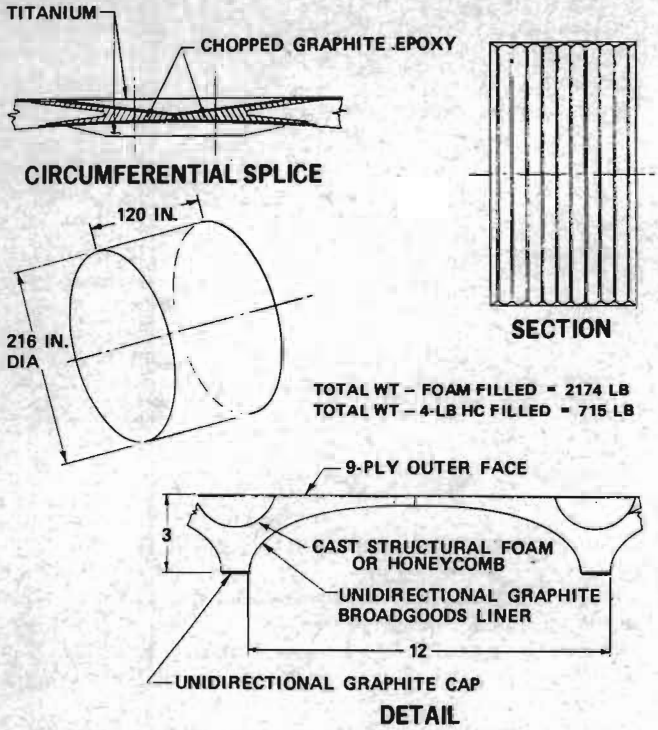


FIGURE 20. FUSELAGE COST COMPARISON SECTION – ARCH FRAME

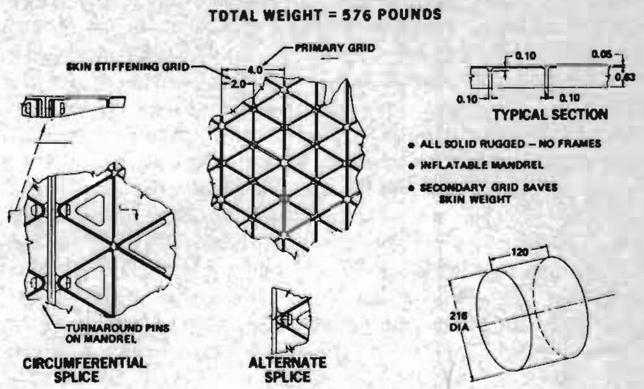


FIGURE 22. FUSELAGE COST COMPARISON SECTION – ISOGRID

TABLE 5
RELATIVE WEIGHT AND COST EVALUATION

FUSELAGE CONCEPTS	RELATIVE WEIGHT	COSTS (100 UNITS) MATERIALS*	LABOR	TOTAL
HONEYCOMB SANDWICH SHELL WITH FRAMES	1.35	1.16	1.00	1.06
ARCH FRAME, 4 PCF HONEYCOMB	1.24	1.19	1.26	1.17
ISOGRID (ALL GRAPHITE)	1.00	1.00	1.10	1.00
ISOGRID (GLASS/GRAPHITE)	1.07	0.35	1.10	0.62

*GRAPHITE AT $20/LB PLUS HONEYCOMB, GLASS, ADHESIVE, METAL, ETC.

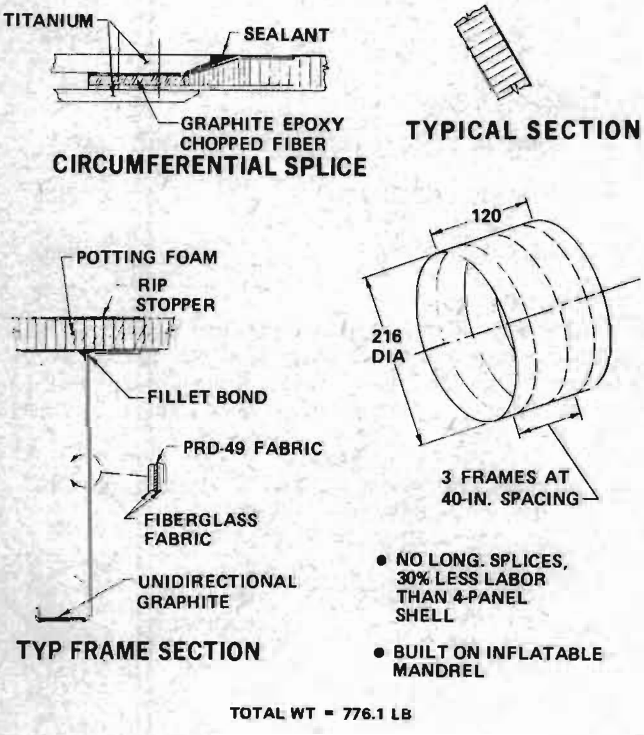


FIGURE 21. FUSELAGE COST COMPARISON SECTION – THICK HONEYCOMB

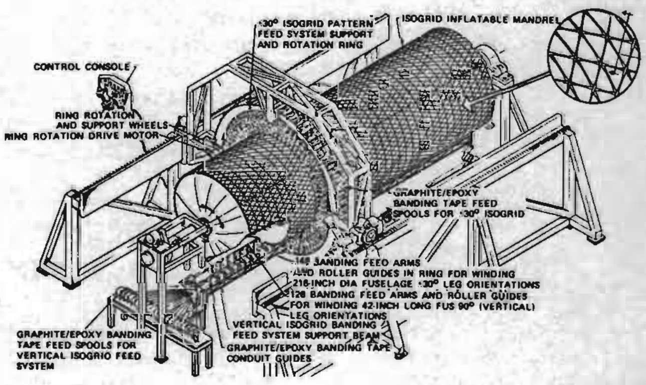


FIGURE 23. ISOGRID FUSELAGE FABRICATION INFLATABLE MANDREL

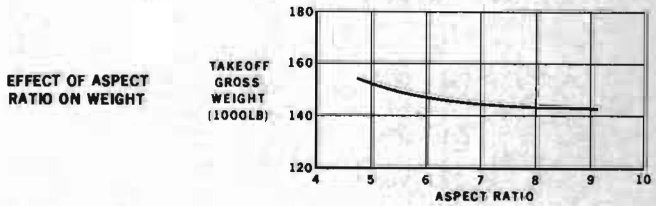


FIGURE 24. EFFECT OF ASPECT RATIO ON WEIGHT

weight for aspect ratios between 7 and 9. Factors such as lateral control response, aeroelastic effects, structural dynamics, and aircraft overall dimensions favor the lower aspect ratios. For these reasons the aspect ratio of 7 was maintained for the composite wing. The wing sweep (Figure 25) of 5.9 degrees provides a straight flat hinge line perpendicular to the airstream. Any weight reduction realized by reducing this sweep angle would be more than offset by the higher flap attachment structural weights and increased structural costs. The thickness ratio (T/C) of 0.139 of the baseline wing provides sufficient fuel volume for the basic mission in the composite aircraft. A higher T/C does not result in an appreciably higher weight saving. Because of the relative insensitivity of baseline aircraft characteristics to weight and cost, significant improvements in aircraft geometry due to the use of composite materials do not exist for the AMST vehicle. The major impact of composites is to reduce wing and tail areas rather than to revise wing geometry. The fuselage geometry and dimensions will remain fixed by cargo-loading volume requirements.

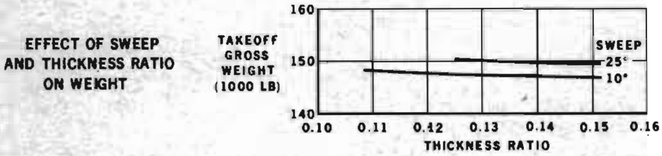


FIGURE 25. EFFECT OF SWEEP AND THICKNESS RATIO ON WEIGHT

Performance Payoffs

Payoff studies included an unresized composite airplane, the resized basic mission airplane, and a resized derivative airplane which has the wing area and fuel volume for the best match with the baseline engine. For the resized basic mission aircraft, a scaled JT8D-17 type engine is used. For the other two aircraft the basic baseline engine is used. These three airplanes offer a full range of performance and cost comparison with the baseline. Three options for performance improvements for the unresized composite aircraft are shown in Table 6.

TABLE 6
UNRESIZED COMPOSITE AIRCRAFT PERFORMANCE IMPROVEMENT OPTIONS

OPTION	PAYLOAD (LB)	MISSION RADIUS (N MI)	MIDPOINT FIELD LENGTH (FT)
1	27,000	400	$\boxed{1880}$
2	27,000	$\boxed{585}$	2000
3	$\boxed{32,560}$	400	2000

Cost Comparisons

Composite weights and costs have been developed for the resized basic mission aircraft. Both initial (flyaway) and life-cycle costs have been developed. Costing methodology is illustrated in Figure 26.

Aircraft production costs are considered in the following level of detail:

Manufacturing labor
Manufacturing material
Engineering
Flight test
Laboratory tests
Development for avionics subsystems.

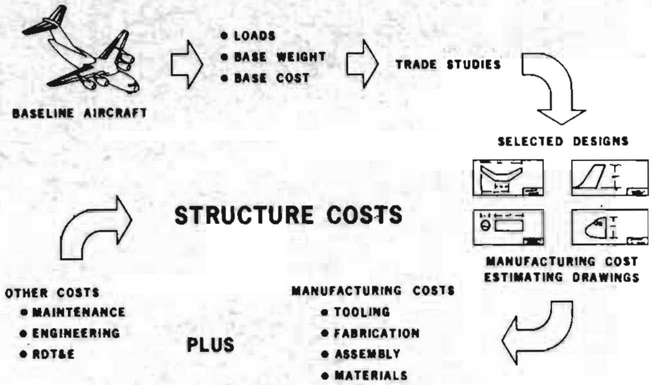


FIGURE 26. STRUCTURE COSTING METHODOLOGY

Costs are divided into development and production portions where appropriate. Program emphasis is on the first two cost components, manufacturing labor and material. Remaining cost items are developed to a level of detail to assure their proper magnitude in relation to total costs (Table 7).

TABLE 7
PRICE COMPARISON OF PRODUCTION METAL AND COMPOSITE AIRCRAFT (1973 DOLLARS – MILLIONS)

AIRCRAFT SUBSYSTEM	METAL BASELINE AIRCRAFT	RESIZED BASIC MISSION AIRCRAFT ★
AIRFRAME	1.00	0.99068
ENGINES	1.00	0.925
AVIONICS	1.00	1.00
TOTAL (295 AIRCRAFT)	1.00	0.98126
CUM AVERAGE PRICE	1.00	0.9809

★ PITCH-BASED FIBER

Life-cycle costs derived for evaluation purposes and developed as the cost criterion for cost effectiveness studies represent the sum of all anticipated dollar expenditures required from acquisition through complete aircraft usage until vehicle retirement (Table 8).

TABLE 8
LIFE CYCLE COST COMPARISON OF METAL BASELINE AND COMPOSITE AIRCRAFT (1973 DOLLARS – MILLIONS)

RESOURCE CATEGORY	METAL BASELINE AIRCRAFT	RESIZED BASIC MISSION ★ AIRCRAFT
DEVELOPMENT	1.00	0.9459
PRODUCTION (INCLUDES SUPPORT SYSTEM)	1.00	0.97975
OPERATING AND SUPPORT — 20 YEARS	1.00	0.98157
LIFE CYCLE COST	1.00	0.97851

★ PITCH-BASED FIBER

CIVIL STOL TRANSPORT

Costs and benefits of applications of advanced composites primary airframe structure were studied to define cost effective applications for a civil STOL aircraft (Figure 27). Applications were studied by comparing costs and weights with a baseline metal airplane which served as a basis of comparison throughout the study.

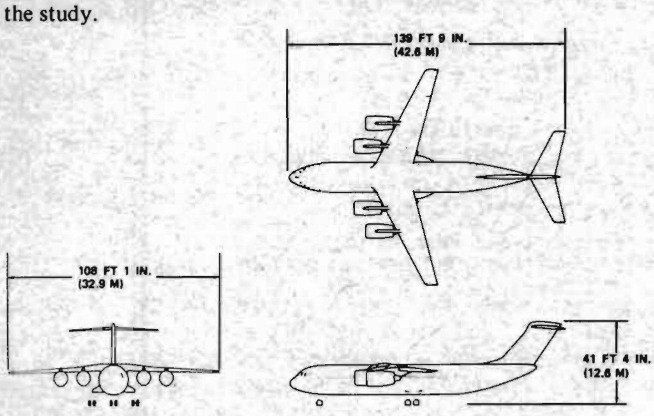


FIGURE 27. METAL BASELINE AIRCRAFT GENERAL CONFIGURATION

The aircraft selected for study has a 150-passenger payload, a 3000-foot (914-meter) design field length, a takeoff gross weight of 149,000 pounds (67,600 kg) and a 0.69-M cruise speed. Composite aircraft resizing procedures assumed the same range, payload, and takeoff field length as the baseline aircraft and was designed for introduction in the mid-1980's.

The baseline structure is a conventional aluminum design. The wing structural box is a stiffened skin, two-spar design with internal ribs and bulkheads located to support the various high-lift devices and control surfaces. The box is an integral fuel tank compartmented with solid bulkheads. The two-segment flaps extend from the fuselage to 75 percent of the wing span.

The fuselage is a semimonocoque shell with conventional skin longeron, frame, and bulkhead arrangements. The vertical and horizontal stabilizers have two-spar stiffened skin boxes. The elevator is built up from a single spar, ribs, skins, and an extruded trailing edge. The wing has 25 degrees of sweep and a supercritical airfoil.

The general arrangement of the composite airplane is identical to the baseline metal aircraft. The structural detail, however, has been developed to take advantage of the specific properties of advanced composite materials while recognizing their high cost relative to metals.

The basic structure is developed from honeycomb sandwich panels with graphite-epoxy face sheets and aluminum honeycomb core. Final assembly joints are mechanically fastened while secondary joints are bonded. All major fittings, such as flap attachment fittings and wing-to-fuselage fittings, are aluminum.

Wing Construction

The wing box utilizes a multirib substructure with sandwich upper and lower skins. Major joints are bolted and bonded. The wing box is an integral fuel tank compartmented as in the baseline design by solid bulkheads. Ribs, bulkheads, and spar webs are sandwich construction. Major bulkheads have extruded aluminum caps, while secondary ribs have formed graphite-epoxy attachment angles. The inboard leading edge, tips, and leading edge slats are conventional aluminum construction because of lightning and environmental requirements. Secondary control surfaces are full-depth honeycomb with aluminum skins identical to the baseline design. The outboard fixed leading edge is a solid graphite-epoxy skin over an aluminum rib substructure. The aft segment of the trailing edge flap is constructed of a solid graphite-epoxy skin over an aluminum honeycomb core with a single graphite-epoxy spar. The forward segment utilizes a graphite-epoxy spar and molded graphite rib substructure with a solid graphite-epoxy skin.

Fuselage Construction

The fuselage is built up from a graphite-epoxy/aluminum honeycomb core sandwich shell supported by composite frames. Major landing gear and attachment frames are aluminum with upper and lower composite segments. The floor is of conventional aluminum construction utilizing some boron-reinforced components. Primary cockpit enclosure structure is conventional aluminum construction.

Empennage Construction

The horizontal and vertical stabilizer structural boxes are essentially the same as for the wing. All major frames are metal. Leading edges and leading edge control surfaces are conventional metal construction where they are directly exposed. Elevators and rudders are graphite-epoxy structures.

Materials

Materials considered for use in this program were graphite-epoxy, boron-epoxy, graphite-polyimid, glass-epoxy, PRD-49-epoxy, and boron-aluminum. The composite structure arrangement is shown in Figure 28.

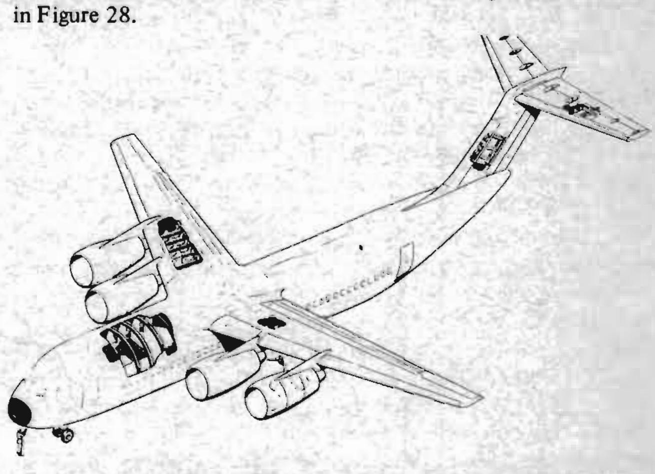


FIGURE 28. COMPOSITE AIRCRAFT STRUCTURAL ARRANGEMENT

Weights

Total weight saving and TOGW is 11.2 percent for the resized composite aircraft. For the airframe itself, taken to consist of the wing, fuselage, empennage, and propulsion groups, the weight saving is 15.6 percent for the unresized aircraft and 22.1 percent for the resized aircraft. While the design considered in the broad sense is an all-composite design, only 40 percent by weight of the material is composite material for the resized version. The weight saved to composite used was 96 percent for the wing box and 52 percent for the fuselage, reflecting more efficient material usage in the wing box where the loads are higher and fewer minimum gauge areas are encountered.

Structure Costing Methodology

An overview of the basic costing methodology procedure was shown in Figure 26. The baseline aircraft definition includes design loads, base weights for all items, and base costs. Through a series of trade studies, competitive structural concepts were evaluated for the most cost-weight competitive designs. Manufacturing cost estimating (MCE) drawings were constructed for the selected concepts that provide the basis on which total airframe manufacturing costs were derived. Manufacturing costs were segregated into categories directly comparable to similar costs for the baseline case. These categories include fabrication and assembly, tooling, quality assurance, planning, and materials. All other costs necessary to completely evaluate the impact of composites are added to these costs and include maintenance, engineering, and an assessment of required research and development.

Aircraft Costs

Aircraft costing methodology is centered around the manufacturing costs described earlier. To these costs are added all other costs required to evaluate a complete system. The buildup of costs and revenues permits cost-effectiveness evaluations by comparisons of operating costs and return on investment. Aircraft production costs were considered to the following level of detail: (1) manufacturing labor, (2) manufacturing material, (3) engineering flight test, (4) laboratory test, (5) development support, and (6) avionic subsystems. Each of these costs was broken into development and sustaining portions when appropriate. Remaining cost items were developed to the level of detail necessary to reflect impact on total costs.

To measure the full impact of composite applications, life-cycle costing was employed. Life-cycle costs derived for evaluation purposes and developed as the cost criterion for cost-effectiveness studies represent the sum of all anticipated dollar expenditures required for acquisition and use through aircraft retirement. The resource categories associated with these expenditures were RDT&E, investment, and total operating costs (TOC), which consists of direct operating costs (DOC), indirect operating costs (IOC), as well as any penalty assessments. These expenditures represent all funds that the aircraft manufacturer and operator would expend for introduction and use of the system.

Principal DOC elements considered are flying costs, depreciation, and maintenance. The DOC computations were based on a modified 1967 Air Transportation Association method.

The total airframe cost for the composite aircraft has increased by 0.7 percent. A basic reason for this increase is the material cost which has increased by $0.26 million, offsetting the $0.16-million decrease in manufacturing labor. Other significant elements are quality assurance and tooling costs. The most important elements are manufacturing and tooling labor, since these costs can more significantly affect airframe costs and DOC.

Benefit Analysis

A benefit analysis was performed by comparing DOC for the composite and baseline airplanes. For conventional aircraft in the same class the trend in the DOC would generally be expected to follow the price pattern. However, the composite airplane operating cost analysis developed an incremental increase in maintenance cost which tended to offset savings derived from potential price advantage and savings in fuel consumption. Total operating costs (TOC) and the elements of DOC and IOC for the principal cases studied are shown in Table 9. The total operating costs vary from the baseline metal airplane over a range of −1.2 to +1.2 percent depending on composite material cost. The DOC change varies over a range of −0.5 to +3.5 percent.

Some significant results of the study can be summarized. Total saving in takeoff gross weight (TOGW) was 16,700 pounds (7580 kg) or 11 percent. The saving in manufacturing empty weight (MEW) was 15,500 pounds (7000 kg) or 16 percent. The total weight saving was 10,700 pounds (4900 kg) or 23 percent. Total

TABLE 9
OPERATING COST SUMMARY

CANDIDATE SYSTEM	OPERATING COST — ¢/ASSM (¢/ASKM)* DOC	IOC	TOC
BASELINE METAL AIRCRAFT	1.99 (1.24)	1.30 (0.808)	3.29 (2.05)
ADVANCED COMPOSITE AIRCRAFT			
LOW CASE ($10/LB COMPOSITE MATERIAL COST)	1.98 (1.23)	1.27 (0.789)	3.25 (2.02)
NOMINAL CASE ($25/LB COMPOSITE MATERIAL COST)	2.03 (1.26)	1.27 (0.789)	3.30 (2.05)
HIGH CASE ($30/LB COMPOSITE MATERIAL COST)	2.06 (1.28)	1.27 (0.789)	3.33 (2.07)

* ASSM — AVAILABLE STATUTE SEAT MILE.
** ASKM — AVAILABLE STATUTE SEAT KILOMETER.

amount of composite material used for the four components listed was 14,200 pounds (6500 kg) or 40 percent of the resized composite airplane. Total material costs were 26 percent of the nominal airframe cost for the composite aircraft compared to 23 percent for the baseline aircraft. For the composite airframe, total manufacturing labor costs decreased by $160,000 or 5 percent, and material costs increased by $260,000 or 17 percent. Tooling was found to decrease by $152,000 or 21 percent and quality assurance increased by $88,000 or 22 percent. Based on the nominal graphite-epoxy price, total maintenance costs increased by 10.1 percent. A range of DOC was developed and varied from a decrease of 3.4 percent to an increase of 3.5 percent, depending on material and maintenance costs. Composite airframe fabrication and assembly labor costs were found to be the most significant production cost that can be influenced by composite material, amounting to 42 percent of total production costs for the composite aircraft.

Sensitivity Analysis

The yardstick of success in the assessment and implementation of new system designs, assuming performance requirements are met, is largely one of economics. The variability of significant parameters provides insight into the impact on a system by the quantitative variance of that parameter. For example, in the civil STOL system study cited, the principal contributions to the DOC increase for the composite airplane are maintenance costs and raw material costs, while the principal decrease was caused by a reduction in manufacturing labor.

The impact of variations in the price of graphite-epoxy material on DOC is shown in Figure 29. The DOC is seen to be moderately sensitive to composite material price for the designs developed in this study where manufacturing only is considered. Composite materials represent only 26.3 percent of total material costs. Maintenance materials are not included in the data of Figure 29 but are included in the maintenance costs discussed below.

Total manufacturing labor cost has a significant impact on DOC as shown in Figure 30. A 10-percent decrease in manufacturing labor cost can change the DOC by 0.9 percent compared to the nominal composite airplane. Since manufacturing labor alone represents 42 percent of the total airframe production costs, it is a key area in which significant impact of composite applications of total system economics can be anticipated.

Reduction in total maintenance costs seems to be significant as shown in Figure 31 and can lower DOC by 0.5 percent if total maintenance costs are the same as the baseline aircraft. If maintenance costs for the composite aircraft were assumed equal to a metal airplane (a conventional metal aircraft with the same

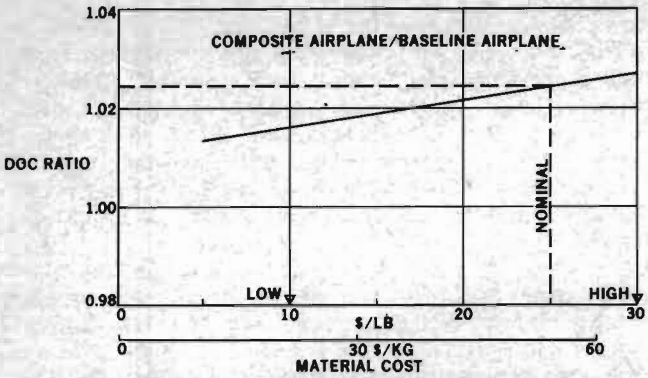


FIGURE 29. EFFECT OF GRAPHITE-EPOXY RAW MATERIAL PRICE ON DOC

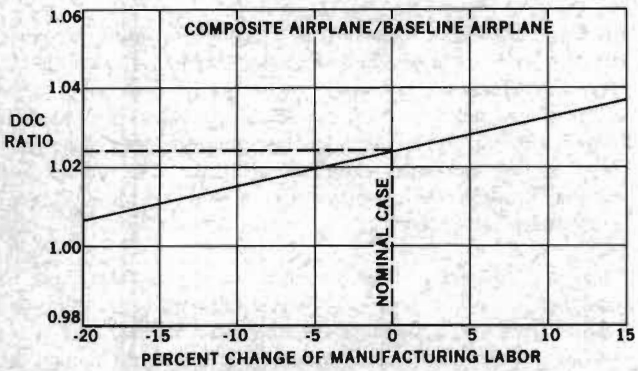


FIGURE 30. EFFECT OF MANUFACTURING LABOR ON DOC FOR THE COMPOSITE AIRPLANE

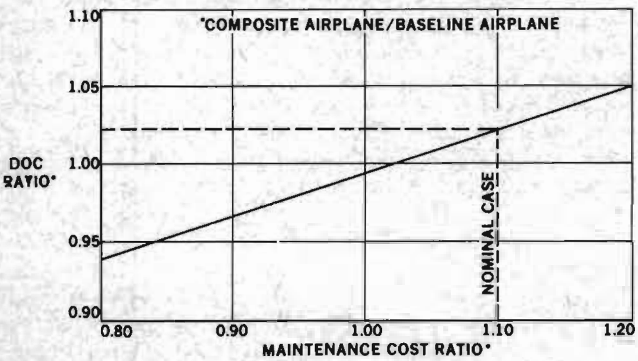


FIGURE 31. EFFECT OF MAINTENANCE COST ON DOC

cost and weight as the composite aircraft), the DOC would be decreased by 2.1 percent. Both changes are relative to the baseline aircraft. Thus, if the low material price were combined with the equivalent metal airplane maintenance cost, the total DOC of 1.92¢ per available statute seat mile (ASSM) (1.20¢ per available statute seat kilometer (ASKM)), a 3.4-percent improvement over the baseline metal cases is obtained.

Table 10 summarizes the impact of material and maintenance cost changes on DOC. The fuselage develops 57 percent of structural material maintenance labor cost and 33 percent of structural maintenance material cost for the composite airplane. Furthermore, for the composite design considered, the fuselage develops 38 percent of total material cost based on the nominal case of which 71 percent is composite materials. Therefore, maintenance and composite material cost changes can be particularly significant for the fuselage.

From material usage in maintenance cost points of view, broad applications of advanced composites appear to have less potential

TABLE 10
DOC COMPARISONS – ¢/ASSM (¢/ASKM)

CONFIGURATION	NOMINAL MAINTENANCE (1)	EQUIVALENT MAINTENANCE (2)
BASELINE AIRCRAFT	1.99 (1.24)	1.99 (1.24)
ALL COMPOSITE AIRCRAFT		
LOW MATERIAL	1.98 (1.23)	1.92 (1.19)
NOMINAL MATERIAL	2.03 (1.26)	1.95 (1.21)
HIGH MATERIAL	2.06 (1.28)	1.96 (1.22)

(1) BASED ON HIGHER (NOMINAL) EQUIVALENT METAL AIRCRAFT MAINTENANCE
(2) BASED ON EQUIVALENT METAL AIRCRAFT MAINTENANCE

for cost effective applications to the fuselage structure than to other components.

INCORPORATION OF EMERGING MATERIALS INTO OPERATING SYSTEMS

Although system studies such as those in the previous section indicate the direction of improved aircraft systems, it is unlikely that a commercial manufacturer would launch an aircraft program on the results of such a study. Further steps are required to develop the confidence level necessary to enable the manufacturer to risk the entire future of the company on the success of the aircraft.

In order to establish the necessary confidence level it is necessary to conduct programs which obtain service experience on selected components. These programs would establish a manufacturing cost data base and verify life and service environmental characteristics.

Two programs are being sponsored by the National Aeronautics and Space Administration to obtain manufacturing experience on transport aircraft control surfaces. The Boeing Company is manufacturing 114 spoilers with graphite-epoxy face sheets for the Boeing 737. Four spoilers (Figure 32) will be installed on each of 27 aircraft representing five major airlines operating in different environmental circumstances. The McDonnell Douglas Corporation is manufacturing 19 DC-10 upper aft graphite-epoxy rudders (Figure 33). Eighteen of these rudders will be placed in commercial service. Thirteen boron-epoxy inboard leading edge slats were manufactured for the C-5A aircraft by the Lockheed Georgia Company (Figure 34). Ten of the slats have been installed on production aircraft for full flight service demonstration. These programs should establish a data base and a confidence level necessary to enable an aircraft system to be produced with advanced composite control surfaces. Other programs will be required, however, to develop the confidence level and experience on other types of structure.

CONCLUSIONS AND RECOMMENDATIONS

The main thesis of this paper is an affirmation of the benefits derived from the synergistic effects of innovative design, emerging materials, and advanced manufacturing techniques. Conclusions and the ensuing recommendations will be directed toward those areas that have the greatest impact upon system improvements and which require additional technological development.

Development of reliable maintenance cost data and maintenance procedures for composite structures is of primary importance.

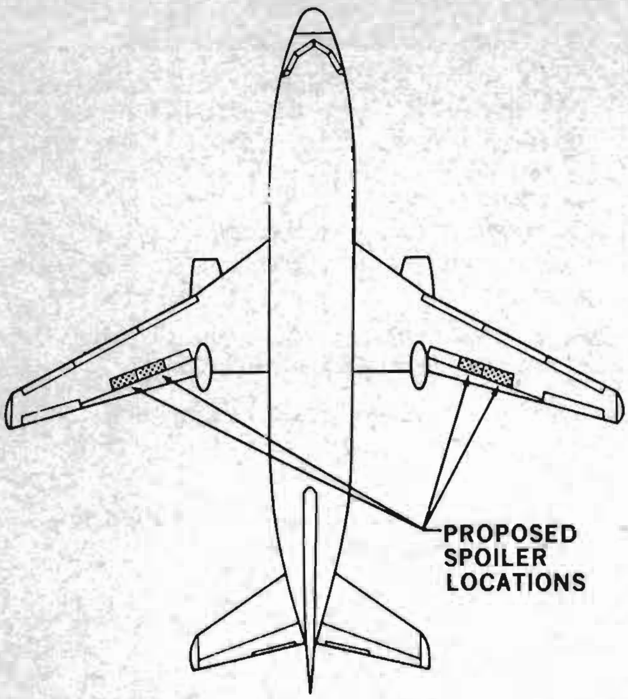


FIGURE 32. PLAN VIEW OF B737 AIRCRAFT

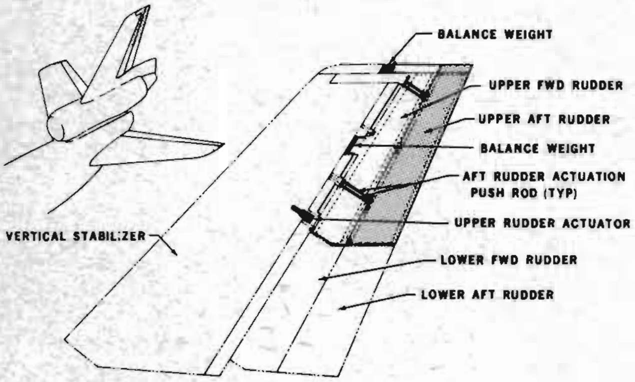


FIGURE 33. DC-10 UPPER AFT RUDDER ASSEMBLY

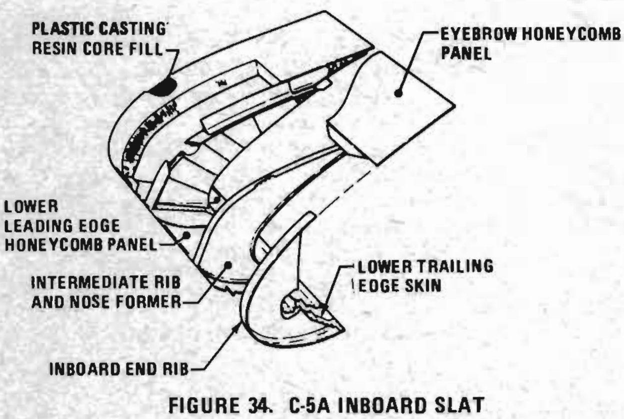


FIGURE 34. C-5A INBOARD SLAT

Although the flight programs that are currently under way are effectively developing relative costs of maintenance of secondary structure and control surfaces, maintenance cost data for large-scale application of composites are nonexistent. In addition, major damage repair procedures for composite structure are relatively undeveloped, although techniques for local damage have been developed and are in use. As the complexity of the aircraft and continuous nature of the structure increases, repairs and repair techniques become more important. Relative to metals, repair of major damage (impact of service trucks, towing accidents, et cetera) can cause excessive aircraft down time and consequent loss of revenue. One large U.S. carrier has reported an average loss of dispatch of one vehicle per day due to structural damage caused by ground support vehicles and maintenance crews.

It is recommended that a program be initiated to develop major damage repair procedures. In addition, a costs analysis associated with implementation of the procedures to airline operators should be developed and an integral part of all flight component programs should be the development and documentation of maintenance cost data.

In complete consort with the development of maintenance costs, the data from flight programs are extremely important; although several programs are under way, none involve large components of the size required for transport aircraft. Flight service programs involving major components are a prerequisite for the development of the necessary confidence on the part of airframe manufacturers and airline operators that will lead to the ultimate use of composite materials in primary structure.

While development of large-scale hardware is important and is a must before large-scale application of composite structures will become a reality, sufficient attention must be paid to upgrade the technology. Analysis techniques must be established that can be utilized by designers and analysts who are not necessarily composite or mathematical specialists. Nondestructive inspection (NDI) techniques must be developed that are quick, accurate, and reliable. The phenomenon of crack propagation and fracture toughness of the material system must be better understood, and realistic criteria developed for its application. And finally, the financial risk of machine tool and fabricating equipment expenditures must be undertaken to establish the techniques and methodology required for low-cost manufacture of composite construction. This involves automatic layup equipment, rapid cure techniques, and the quality control necessary for safe, reliable, high-confidence man-rated vehicles.

The most promising metallic materials for use in future aircraft structures appear to be aluminum alloys X7475 and X7050 for their combination of high tensile ultimate strength, compression yield, fracture toughness, and low cost. Titanium Beta C and Beta III show promise for compression critical structures, but new concepts must be compatible with a high material utilization factor to compensate for their higher costs compared to the aluminum alloys.

Fatigue testing of weld bonding is recommended in basic structure and in skin splices to verify assembly and cleaning procedures and analytical potential. For honeycomb-stiffened sandwich concepts, parametric strength studies and tests are recommended. Variations should include core depth, face thickness, stiffener geometry, and stiffener spacing over a range of rib support spacings. Testing is also recommended in both compression and tension for the fuselage weld bond skin panel concept for correlation with analyses.

Before production of primary structural components utilizing heat-treated titanium alloys and bonded elements, the following NDI method developments are recommended: (1) production method to determine heat treatment of titanium, (2) method to determine bond strength of assemblies, (3) methods to determine surface cleanliness prior to bonding, and (4) methods for measuring residual stresses in assembled structures.

REFERENCES

1. R. E. Adkisson, G. V. Deneff, et al., "Analytical Investigation of Medium STOL Transport Structural Concepts," Douglas Aircraft Company final report, Air Force Contract F33615-73-C-3049, October 1973.

2. G. R. Danielson, G. V. Deneff, et al., "Advanced Military Tanker Wing/Fuselage Structural Concepts Study," Douglas Aircraft Company, Air Force Report AFFDL-TR-72-89, July 1972.

3. T. R. Logan, et al., "A Study of the Costs and Benefits of the Application of Composite Materials to Civil STOL Aircraft," NASA Report CR114700, December 1973.

4. W. D. Nelson, "Conceptual Design Studies of Composite AMST," Douglas Aircraft Company Final Report – Air Force Contract F33615-73-C-5164.

BIBLIOGRAPHY

D. M. Purdy, H. G. Schaeffer, "Stress Analysis of Composite Structures," Douglas Paper 5840, April 1971.

Henry R. Clauser, "Advanced Composite Materials," General Electric Company.

R. E. Black, J. A. Stern, "Creative Advanced Design-A Key to Reduced Life-Cycle Costs," Douglas Paper 6181, October 1973.

C. S. Glasgow, "Improving Aircraft Productivity – We All Have a Part of the Action," Douglas Paper 6219, October 1973.

G. V. Deneff, "Wing/Fuselage Structural Concept Study for a Subsonic Transport Aircraft," Douglas Paper 6167, October 1973.

Paul L. Sandoz, "The Next Careful Steps in Commercial Aircraft Structures," Astronautics and Aeronautics.

H. C. Schjelderup and D. M. Purdy, "Advanced Composites – The Aircraft Material of the Future," Douglas Paper 5883, July 1972.