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Study of Utilization of Advanced Composites in Fuselage Structures of Large Transports

Final Report

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LOCKHEED-CALIFORNIA COMPANY
BURBANK, CALIFORNIA

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FOREWORD

This is the final report of the program completed by the Lockheed-California Company and the Lockheed-Georgia Company, "Study of Utilization of Advanced Composites in Fuselage Structures of Large Transports". This program was conducted from June 1983 through May 1984. This work was sponsored by the National Aeronautics and Space Administration (NASA) Langley Research Center and the Air Force Wright Aeronautical Laboratories (AFWAL). The engineering manager for Lockheed was Mr. Anthony C. Jackson. Mr. Herman L. Bohon was the project manager for NASA Langley. The technical representative for NASA Langley was Mr. Jon Pyle and the technical representative for AFWAL USAF was Mr. James L. Mullineaux.

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IN FUSELAGE STRUCTURES OF LARGE TRANSPORTS

FINAL REPORT

A.C. Jackson, M.C. Campion, and G. Pei

SUMMARY

A study was performed to plan the effort required by the transport aircraft manufacturers to support the introduction of advanced composite materials into the fuselage structure of future commercial and military transport aircraft. The study identified the technology issues which must be resolved, assessed the potential benefits to military life-cycle costs and commercial operating costs, and defined a plan to develop the technology and confidence needed to commit to production of composite fuselages for large transport aircraft in the 1990's.

The study program consisted of three overlapping phases: (1) Technology Assessment, (2) Plans Development and (3) Program Schedule and Resource Requirements.

The objectives of the Technology Assessment phase were threefold. The first objective was to assess the state-of-the-art in composites technology as applicable to fuselage structures and to identify and prioritize the technology issues to be resolved. The second objective was to determine the extent of verification testing required to provide confidence that the technology is at hand to support a decision to commit to the production of composite fuselages. The verification test options ranged from fuselage panel tests through full-scale fuselage testing. The third objective was to identify the major cost benefits to be derived from the application of composites to the fuselage structures of both military and commercial large transport airplanes.

The key technology issues were defined and assessed. The most urgent issues are: impact dynamics, acoustic transmission, pressure containment and damage tolerance, post buckling, cut-outs and joints and splices.

The assessment of the program options identified the most cost-effective minimum risk program to be to provide demonstration with the ground test of a full-scale fuselage barrel section.

Analyses were performed to determine the commercial operating costs and the military life-cycle costs for the composite airplane compared with a conventional aluminum airplane and an advanced aluminum airplane. The commercial

composite airplane showed a reduction of 8.1 percent in direct operating costs compared with the conventional aluminum baseline and 6.0 percent compared with the advanced aluminum airplane. The military composite airplane showed a 10.2 percent reduction in life-cycle costs compared with the conventional aluminum baseline and 8.0 percent compared with the advanced aluminum airplane.

The procurement cost for the all composite airplane was approximately 5 percent less than the conventional aluminum for both military and commercial. However when the effects of automated fabrication were included this savings increased to approximately 11 percent. Fuel savings of approximately 14 percent were shown for both the military and commercial composite airplanes compared with the conventional aluminum airplanes.

The composite fuselage alone provides a fuel saving of between 4.5 and 5.3 percent compared to the conventional aluminum baselines.

The primary objective of the Plan Development phase was to develop a program plan which, if implemented, would develop the engineering and manufacturing technology required to provide confidence in the use of advanced composite structures for fuselages of future transport aircraft.

This program consists of six technical phases culminating the fabrication and ground test of a full-scale fuselage barrel section. The phases are: detail design, manufacturing development, design development testing, tool design and fabrication, barrel fabrication and validation testing of the barrel section.

Preliminary design and concept evaluation trade studies were performed to define the probable structural configurations of a composite fuselage for the commercial and military airplanes.. The results of the trade studies indicated a discretely stiffened skin, using blade or jay type stiffeners, or an orthogrid design to be the most structurally and cost efficient concepts.

The Program Schedule and Resource Requirements phase had the objectives of determining the schedule and resources for the proposed program.

The overall program required to develop the technology and data needed to support the introduction of advanced composite materials in the fuselage structure of future commercial and military transport airplanes has been defined and the relationship of the on-going and planned programs has been identified. The Fuselage Technology Demonstration program extends from approximately 1987 to 1992. The estimated cost of the engineering/manufacturing effort is approximately 278 man years including program management.

INTRODUCTION

The National Aeronautics and Space Administration (NASA) Langely Research Center, through the Aircraft Energy Efficiency (ACEE) composites program, has provided the transport aircraft manufacturers, the FAA, and the airlines with the experience and confidence in advanced composite structures needed for extensive use in secondary and medium primary components of future large commercial and military aircraft.

In 1981 NASA embarked on a program to develop the key technologies needed to lead to the introduction of advanced composite wing primary structures for large transport aircraft. Major drivers, both technical and programmatic, which challenge the application of composites to fuselage primary structures are significantly different from those for wing structures. The potential benefits must be assessed since the fuselage comprises about 33 percent of the structural weight of transport aircraft; a weight savings of 20 to 25 percent over current metal designs could significantly improve fuel efficiency and the range capability of the aircraft. To evaluate the merits of commercial and military transports, NASA and the USAF have supported studies by the three major manufacturers, Lockheed, Boeing, and McDonnell Douglas, to assess the state-of-the-art and to evolve a technology development plan to accomplish the transition from current construction materials and practices to the extensive use of composites in fuselages of aircraft by 1990.

The duration of the program was 11-1/2 months. The master schedule is shown in Figure 1.

This study has defined the technology issues which must be addressed, evaluated the program options, and defined a plan leading to the introduction of advanced composites in the fuselage structure of large transport aircraft in the 1990s. The schedule and the resource requirements to achieve these ends have been identified. The study also defined the expected benefits of applying advanced composites to both military and commercial transport fuselages.

ABBREVIATIONS AND SYMBOLS

A/C	Aircraft
ACEE	Aircraft Energy Efficiency
ACMA	Advanced Civil/Military Aircraft
ACSDT	Advanced Composite Structures Design Technology
AEHP	Atmospheric Electricity Hazards Program
AFWAL	Air Force Wright Aeronautical Laboratories
$A\ell$	Aluminum
ASSET	Advanced Systems Synthesis and Evaluation Technique
CIM	Computer Integrated Manufacturing
dBA	Decibels ("A" weighted)
DOC	Direct Operating Costs
DoD	Department of Defense
DoT	Department of Transportation
f	Stress
F	Fahrenheit
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations
FS	Fuselage Station
g	Gravity
G&A	General and Administrative
GASP	General Aircraft Sizing Program

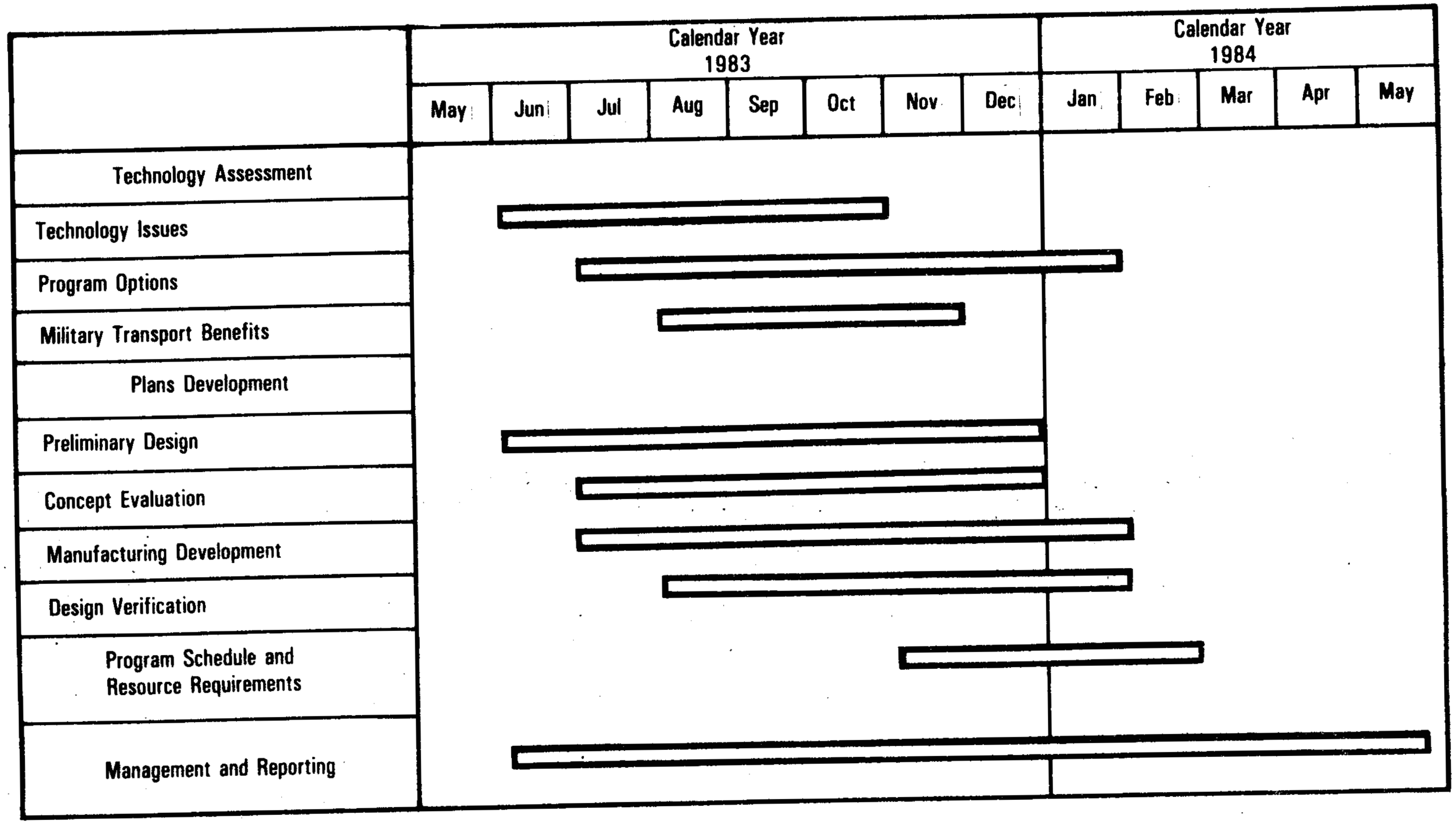

Figure 1. - Program master schedule.

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Gr/EP	Graphite/Epoxy
Gt	Shear stiffness (Shear modulus x thickness)
HPLC	High Pressure Liquid Chromatography
IOC	Indirect Operating Costs
IR	Infra-Red
KEAS	Knots Equivalent Air Speed
LCC	Life Cycle Costs
LOI	Limiting Oxygen Index
L&R	Left and Right
M	Mach Number
M&P	Materials and Processes
NASA	National Aeronautics and Space Administration
N	Load/inch
NBS	National Bureau of Standards
NDE/NDI	Non-Destructive Evaluation/Non-Destructive Inspection
NDT	Non-Destructive Test
N.M	Nautical Miles
O and S	Operation and Support (Cost)
P	Pressure
PEEK	Polyetheretherketone
PSI	Pounds per Square Inch
q	Shear Flow
QA	Quality Assurance
R	Radius
RIM	Resin Injection Molding
ROI	Return on Investment
RRIM	Reinforced Reaction Injection Molding
R and T	Research and Test
RDT&E	Research Development Test and Evaluation
RTD	Room Temperature, Dry
RTM	Resin Transfer Molding
TOC	Total Operating Costs
UF	Ultimate Factor
USAF	United States Air Force
WL	Water Line
ZFW	Zero Fuel Weight
3-D	Three Dimensional
$M	Million Dollars

SUBSCRIPTS

cr	critical
ult	ultimate
x	longitudinal (fore and aft) direction
y	hoop direction
xy	shear direction

1. TECHNOLOGY ASSESSMENT

This phase of the program consisted of three tasks. The first task, Technology Issues, involved an assessment of the state-of-the-art in composites technology as applicable to fuselage structures of large commercial and military transport aircraft. The second task, Program Options, involved definition and evaluation of the various options available to achieve production readiness by 1990. The third task in this phase was Military Transport Benefits. Analyses were performed to identify both the military and the commercial benefits to be derived from the application of composites to large transport aircraft fuselages.

1.1 Technology Issues

A list of the technology issues was assembled from the inputs of specialists in the various disciplines within Engineering, Manufacturing, and Quality Assurance. The list is shown in Table 1. A parallel review of the specifically military technology issues was performed. Battle damage and repair, service damage in the rigorous military environment, and the problems of high load input from such sources as landing gear and cargo drop doors were identified as significant issues for military aircraft. The military issues are discussed in Section 1.1.13.

A literature search was performed to determine the state-of-the-art for the issues identified. The data files searched included: Defense Technical Information Center, NASA, National Technical Information Services, Transportation Research Information Exchange (DoT), SCI Search, Smithsonian Science Information Exchange, Frost and Sullivan Defense Market, and Compendex (Engineering Index, N.Y.).

TABLE 1. - TECHNOLOGY ISSUES

General

- Impact dynamics
- Acoustic transmission
- Joints and splices
- Pressure containment
- Post-buckling
- Shell cutouts
- Automated manufacturing
- Processing science
- Electromagnetic effects
- Repair
- NDE/NDI
- Flame/smoke

Unique to military aircraft

- Battle damage
- Concentrated high loads from
  - Fuselage-mounted landing gear
  - Cargo drop doors

The results of the assessment of the state-of-the-art and the indicated technology voids are shown in Table 2.

The issues were reviewed and ranked according to the urgency of their resolution and the complexity of the solution. The results are shown in Table 3. The number 1 indicates the highest urgency and the most complex.

The urgency ranking is based on factors which must be resolved before a production design commitment can be made and on the relationship of the particular technology issue to other issues. The complexity ranking is based on the amount of effort which may be required to find a cost-effective solution.

Impact dynamics ranks as the most urgent because its resolution may affect the basic structural concepts of the lower fuselage shell. The complexity ranking is based on the possible need for full-scale demonstration. It is a significant issue because of its possible impact on cost and weight. Tests have shown that Gr/Ep systems, being brittle, cannot absorb energy to the same extent as aluminum. Hybridizing with fiberglass or Kevlar 49 shows improvement but indications are that the structure must be designed to be energy absorbent in the manner in which it crushes and deforms. This approach implies weight and cost increases.

Acoustic transmission is considered urgent because the magnitude of the problem must still be defined. The complexity rating is based on the assumption that the problem is major and would require solutions beyond simple interior treatments. The significance of this issue is the relationship between the interior noise level and the mass of the fuselage shell. Unless an effective solution can be found, it may be necessary to add back the weight saved in the fuselage shell as interior acoustic treatment, thus negating any benefits from the use of advanced composites.

Joints and splices are urgent from the point of view of the frame-to-skin joints and the question of whether some kind of mechanical attachment is required along with bonding. The complexity issue here pertains to the reduction of manufacturing costs. Large fuselage shells will not be fabricated in one piece but in large cocured assemblies which must then be joined together to form the complete shell. The significance of this issue involves the fact that many of the joints will be carrying high multidirectional loads and out-of-plane loads from pressurization effects. Efficient, reliable joining techniques and analytical capabilities are essential.

Pressure containment is ranked fourth on the urgency list on the basis of fail-safe design aspects rather than basic shell design or damage tolerance, which are broadly included in this category. Damage tolerance is already receiving much attention on other programs. Fail-safe design is complex, particularly from the manufacturing cost aspects. The main significance of this issue relates to maintaining the pressure integrity of the fuselage from the safety and supportability viewpoints. Thus fail-safe designs must be verified and damage tolerance must be such as to minimize supportability requirements.

TABLE 2. - SUMMARY OF TECHNOLOGY ASSESSMENT

State Of The Art	Technology Voids
Impact Dynamics	
Material characterization and crushing tests have demonstrated that advanced composite materials cannot absorb as much energy as metals. Helicopters have been built with special energy absorbent lower fuselages. Hybrid designs using fiberglass and Kevlar along with graphite are able to increase the enrgy absorption capability of the graphite.	Development of structural concepts for large aircraft that are capable of absorbing energy and efficiently carrying structural loads. Predictive methodology for impact dynamics of composite structures. Test data for large composite structures.
Acoustic Transmission	
Methods are available for the prediction of passenger compartment noise in metallic fuselages caused by prop fans and turbo fans Interior treatements are available to reduce passenger compartment noise at a weight penalty. Limited test data are available for composite panels; no data exist for cylindrical composite structures.	Analytical methods for the prediction of noise levels inside composite fuselages, particularly due to boundary layer noise. Effective treatments to reduce interior noise in a composite fuselage which would not negate most or all of the weight savings of composites over metals.
Joints and Splices	
Analysis methods are available for bolted and bonded joints under uniaxial loads although biaxial analysis capability is limited. Design concepts for highly loaded joints in wings have been developed. Lightly loaded joints are designed with high safety factor.	Analysis methods for joints under biaxial loads and pressure. Joint optimization techniques.
Pressure Containment	
Methods are available based on metals technology for: Mechanical fasteners skin-to-frame Plug doors Fail-safe straps A large amount of work has been accomplished in damage tolerance of composite structures but not regarding the effect of damage on pressure containment.	Effects of pressure on skin-to-frame interfaces. Damage tolerance under pressure cycling. Fail-safe criteria and design concepts for composite fuselage.

TABLE 2. - SUMMARY OF TECHNOLOGY ASSESSMENT (CONTINUED)

State Of The Art	Technology Voids
Postbuckling	
Shear panels have been designed and tested to $q/q_{cr} > 5.0$ for flat and curved panels. Analysis methods are under development. Flat and curved panels have been built and tested to evaluate compression postbuckling for large diameter shells. Stiffener configurations for postbuckled design have been established.	Effects of repeated buckling on skin to stringer and skin-to-frame interfaces. Cost-effective design approaches to prevent separation of skin and stringers. Effects of buckling on damage tolerance.
Shell Cutouts	
Cutout design and reinforcement is handled on an individual case-by-case basis.	Systematic approach to reinforcement design for large cutouts and window belts. Effects of interlaminar stresses at edges under complex loads. Edge reinforcement techniques to minimize or prevent damage around doors and other cutouts.
Automated Manufacturing	
Primarily manual methods are used at present. Islands of automation exist with single function machines – primarily mechanized systems. Limited automated material handling capabilities exist. Computer-aided cure monitoring is available. Degree of automation varies from manufacturer to manufacturer.	Processing of feedback. Automated cure control. Flexible manufacturing systems are needed, i.e., Distributed numerical control Automated material handling Grouping technology
Processing Science	
Chemical analysis techniques are available for standard resin systems. – HPLC, IR Test procedures and mathematical models are available for viscosity/flow characteristics. In-process cure cycle monitoring through dielectric measurement is under development.	Specific procedures and requirements for new resins. Implementation of closed-loop processing systems using dielectric monitoring.
Electromagnetic Effects	
Lightning protection methods for structural components are well defined and well tested. Electromagnetic interference shielding has been demonstrated for military airplanes (fighters).	Effects of lightning strikes on digital electronic systems and fly-by-wire systems in composite shells.

TABLE 2. - SUMMARY OF TECHNOLOGY ASSESSMENT (CONTINUED)

State Of The Art	Technology Voids
Repair	
Repairs have been developed for specific components in military service and for commercial secondary structures, control surfaces and empennage. Repair guide is available. Field level and depot level repairs are being developed. Most repairs for lightly loaded structures are both bolted and bonded.	Bonding techniques for reliable high strength repairs. Repairs of large diameter pressure shells.
Non-Destructive Evaluation/Inspection	
The available inspection methods include: Ultrasonics – computer controlled contour following and data acquisition. Two types of systems – Signal enhancing systems and imaging systems – are available. Acousto-sonics – combination of acoustic emission and ultrasonics Radiography Laser-sonics – being developed for field repair.	Determination of bondline strength. Defect evaluation.
Flame and Smoke	
Hazards have been assessed using airplane statistics on accidents and fires. Simulation capabilities include: – C-133 cabin fire simulator – McDonnell Douglas Test Chamber Measurement capabilities Include: – FAR 25.853 flammability requirements – Limiting oxygen index (LOI) tests – NBS smoke chamber tests – Animal toxicity tests	Accurate hazard determination model. Accepted test procedure for flammability of exterior materials. Simulation or experience with composite fuselages in the areas of fire start and spread through fuselage and smoke and toxin generation and spread.

TABLE 3. - RANKING OF URGENCY AND COMPLEXITY OF TECHNOLOGY ISSUES

	Urgency	Complexity
Impact dynamics	1	1
Acoustic transmission	1	1
Joints and splices	1	2
Pressure containment	1	2
Post buckling	1	2
Shell cutouts	1	2
Automated manufacturing	2	1
Processing science	2	3
Electromagnetic effects	3	4
Repair	3	3
NDE/NDI	3	2
Flame/smoke	3	4

Post-buckling impacts other issues such as pressure containment and joints and splices. The signifiance of this issue relates in particular to fuselages. Aluminum fuselage structures are designed to operate in the post-buckled range. Advanced composite structures can only be weight effective if they are also designed to operate in the post-buckled range.

Shell cutouts and reinforcements can affect the basic shell design, particularly for passenger aircraft, and may impact fabrication methods. The main significance of this issue is the impact on cost and weight of finding an efficient reliable approach to reinforcing structure around the multiplicity of cutouts typical of transport aircraft fuselages.

Automated manufacturing is primarily a question of flexible automation. This essentially necessitates multifunctional machines which can be rapidly changed from one operation to another rather than hard system robotics, which are only cost-effective for high-volume production lines. Most transport airplane production is on the order of two to four craft per month. Because the need date is some years away, the solution, while complex, is not urgent. The solution of this issue rests mainly on the commercial availability of the necessary machinery.

Processing science in the form of computer controlled processing is being developed under existing programs and should be relatively well established by the need dates for fuselages. The significance of this issue relates to improved reliability and much reduced scrap due to processing problems.

Electromagnetic effects are being addressed under other programs, most notably a multiagency program with Boeing. This is a significant issue in that unless effective shielding techniques can be developed which are cost and weight effective, then the use of advanced composites may be strictly limited in airplanes which incorporate digital avionics and fly-by-wire systems.

Repair will not be needed until the late eighties and is tied to NDE/NDI and processing science to some extent. The main significance of this issue relates to the development of verifiable and durable repair techniques which can be easily performed overnight during routine maintenance.

NDE/NDI is primarily needed for ascertaining bondline strength. As with manufacturing, the need date is some years away, but the solution will be complex. This is a significant issue because current procedures do not permit reliance on a bond in critical structural components. Current NDE/NDI techniques can only generally verify that a bond exists, its strength is questionable.

Flame/smoke/toxicity is peculiar to a given material system and is not regarded as a major concern for exterior structures at this time. Its significance is primarily in interior trim and furnishings to provide survivability for occupants in case of fire.

A more detailed look at some of the issues follows.

1.1.1 Impact dynamics.- A primary technological problem associated with transport airplanes designed with advanced composite materials is to achieve energy absorption and load-carrying capabilities comparable to those of current metal designs. This ultimately reduces to a need to develop fuselage designs using advanced materials that will protect the occupants of the aircraft during a survivable crash as well as, or better than, current metal designs.

The accepted measures of occupant protection are:

1. Maintaining loads at or below human tolerance levels.
2. Providing for a protective shell around the occupants.
3. Providing for safe egress.
4. Preventing lethal blows.

More specifically, the current crash design requirements for transport airplanes which affect fuselage design are stated in FAR-Part 25, paragraph 25.561 as follows:

Emergency Landing Conditions

25.561 General

(a) The airplane, although it may be damaged in emergency landing conditions on land or water, must be designed as prescribed in this section to protect each occupant under those conditions.

(b) The structure must be designed to give each occupant every reasonable chance of escaping serious injury in a minor crash landing when:

  (1) Proper use is made of seats, belts, and all other safety design provisions,

  (2) The wheels are retracted (where applicable), and

  (3) The occupant experiences the following ultimate inertia forces acting separately relative to the surrounding structure.

    (i) Upward - 2.0g

    (ii) Forward - 9.0g

    (iii) Sideward - 1.5g

    (iv) Downward - 4.5g, or any lesser force that will not be exceeded when the airplane absorbs the landing loads resulting from impact with an ultimate descent velocity of five fps at design landing weight.

(c) The supporting structure must be designed to restrain, under all loads up to those specified in paragraph (b) (3) of this section, each item of mass that could injure an occupant if it came loose in a minor crash landing.

The most crucial aspect of determining structural dynamic response for a crash condition is the ability to adquately represent the lower fuselage crush behavior. Typical wide-body load-deflection characteristic data for current metal bulkheads and frames, while available, are limited. Efforts are underway both at the FAA and NASA to perform tests to obtain load-deflection behavior characteristics for current metal narrow- and wide-body aircraft structure. Analysis of the structure is planned in addition to the testing. Presumably, if advanced materials were used in lieu of, or in conjunction with, metal structure and if strength and load-deflection characteristics were comparable, then one could anticipate that the potential for occupant survivability would be equal. Data on the behavior of advanced composite materials under crash conditions are needed before such comparisons are possible.

Most of the progress made so far in the study of advanced composite materials under crash conditions has been in the automobile industry. Within the aircraft industry, the most significant advances have been based on helicopter designs. Research in the area of impact strength of composite materials has concentrated on local impacts (tool drop, etc.), and research relating to compression failures has been directed toward the prediction of static design allowables rather than energy absorption.

As a result, although a few general observations can be made regarding materials that would be applicable to transport airplane fuselages, much of the data acquired to date is not relevant to large diameter fuselages because of the design and scale factors involved. Composite materials are generally less energy-absorbent than aluminum although they are able to resist higher peak loads than equivalent-weight aluminum designs. The energy absorption capabilities of advanced composite materials are affected somewhat by changes in layup and can be increased by using hybrid designs incorporating combinations of different materials. Because transport aircraft are much deeper from passenger floor to ground contact point than helicopters are, they are not as well-suited to the use of the relatively short columns, tubes, and beams that are sufficient to provide adequate energy absorption for helicopters and general aviation aircraft. The ability of the lower fuselage to sustain slide-out loads is dependent on the structural design of the fuselage as well as the materials used, and this is another area where helicopter technology is not easily transferred to large transport aircraft.

It is important to quantify advanced composite material behavior for crash impact conditions at different levels of design. For example, data obtained from element tests are desirable for an assessment of the relationship between material design properties and load-carrying capability. Section test data will allow for an evaluation of the effects of design restraints, multiple element interaction, and combined loading on crash performance. Airframe tests provide the opportunity to verify the level of occupant protection that may be realized from a particular design.

Another aspect of the impact dynamics issue is the ability to analyze fuselage structures to determine their capabilities under crash conditions. Because of the extremely high cost of fabricating and testing large structures, as new designs are developed and new materials are used, the ability to predict behavior analytically for a wide range of structural concepts and/or variations will become an economic necessity. As a minimum it may be necessary to obtain basic data experimentally and utilize analytical procedures to determine the sensitivity of response to variations in design or load.

The most feasible approach to analyzing the crash behavior of aircraft would use experimental substructure data and approximate large fuselage structures with simple representations. Thus, the development of load-deflection data for substructural elements is highly desirable. Unfortunately, there has been relatively little effort directed toward the analysis of the load-deflection characteristics of substructure. An understanding of the crash behavior characteristics of the substructure is critical to understanding the response of the floor structure and subsequently of the occupants.

1.1.2 Acoustic transmission.- The use of advanced composite materials for aircraft structures offers the promise of significant weight savings and lower fabrication costs when compared to aluminum structures. Realization of this promise may be inhibited by the requirement that interior noise levels for composite aircraft be comparable to current wide-body aluminum aircraft. This level has generally been identified as 80 dBA. Current wide-body aluminum aircraft have required noise control measures to achieve this noise level and it will not be an easy task to match these levels with a lighter weight composite fuselage.

Lower density material and lower structural damping properties are typical of graphite/epoxy structures. Such properties normally result in higher acoustic transmission. Noise transmission paths leading to the cabin interior are broadly characterized as airborne and structure borne. The airborne paths are associated with propeller noise, engine noise, and turbulent boundary layer noise. The turbulent boundary layer is the most significant airborne noise source in modern turbofan transport aircraft. Structure-borne noise is the noise transmitted into the cabin by wing and empennage vibrations and certain mechanical systems.

Airborne noise is transmitted into the cabin interior via flexural waves that are excited in the fuselage sidewall. The latter consists of traveling waves which move along the fuselage wall with the external pressure fluctuations and standing waves which are due to partial reflections of traveling waves at structural discontinuities (viz., frames). Standing waves are structural resonances which may build up due to repeated reflections between structural discontinuities.

A flexural wave will radiate sound most efficiently when its wave length is greater than that of the sound wave it radiates. A wave that satisfies this condition is said to be "acoustically fast." Conversely, when the wave length of a flexural wave is shorter than that of the sound wave radiated, it is said to be "acoustically slow." Even acoustically slow waves can be relatively efficient sound radiators when structural discontinuities interrupt the normally sinusoidal mode shapes that are typical of flexural oscillations in perfectly uniform structures.

Although traveling waves due to an acoustical field are always acoustically fast, they may or may not be "well coupled" to the excitation field. Below the ring frequency (the frequency of the fundamental extensional or "breathing" resonant mode of the shell in which the entire shell expands and contracts circumferentailly, as a whole about its neutral position) of a cylinder, axially traveling waves have a natural (or "free") speed of propagation that is greater than the speed of sound in air. There exists an angle of incidence which will make the trace wave length of the impinging sound wave on the fuselage wall equal to the wave length of a freely traveling flexural wave. This condition, called "coincidence", represents the maximum degree of flexural wave excitation for a given impinging sound level.

The first order of priority in a program addressing acoustic transmission through composite shells is to determine the magnitude of the problem which exists. This would be accomplished by analysis and test, with the testing initially used to verify the analytical techniques. As boundary conditions have a major effect on acoustic transmission, the use of stiffened cylinders is required for test. Methods for sound attenuation must then be developed which will produce lightweight, cost-effective solutions.

1.1.3 Joints and splices.- Much work has been performed in developing analytical methods for basic airplane-type joints and splices, most notably by Douglas under NASA and DoD funding. Much of the work in design and test has been oriented toward highly loaded wing-type joints where loading is primarily in one direction.

The major fuselage-type joint concerns to be addressed are the fuselage frame-to-skin joint, and the problem of high biaxial loads and shear coupled with pressure and buckling which affect fuselage longitudinal and girth splices.

A second type of joint, which has so far received little attention, is the major attachment joint (e.g., of wing-to-fuselage, of landing gear-to-structure, etc.). The reduced tolerance to local stress concentration, which is characteristic of the nonmetallic composites, makes the load diffusion from the attachment bolt or pin into the adjacent shell structure much more difficult. The necessity for very thick laminates, or for metallic inserts, introduces additional grounds for caution, due to thermal gradients, to "invisible" delaminations or voids, to corrosion, and to load diffusion and load transfer processes.

1.1.4 Pressure containment.- Pressure containment also involves to a large extent the issues of damage tolerance and fail safety, so they are considered jointly here.

The main areas of concern are the skin-to-frame interfaces, the effects of pressure on damage tolerance, and the definition of damage threats and fail-safe criteria for composite fuselages.

The skin-to-frame interface has already been discussed under joints and splices. A primary concern is the effect of damage which causes separation of the bond between the skin and the frame and delamination in the skin locally which could grow rapidly under the flight-by-flight pressure cycle.

Delamination-type damage can occur in any part of the skin and the effects of the flight-by-flight pressure cycle in combination with other likely loads must be investigated so that slow (or no) growth and damage containment concepts can be developed. Damage threats must be defined so that the types and magnitude of damage likely to occur can be properly investigated. The locations of the principal threats are summarized in Figure 2.

Fail-safety criteria need to be developed. The classic metals approach of crack arresters does not apply to composites in the same sense as metals because delamination or impact damage is interlaminar in nature. Conversely, major damage of the type expected from engine fan blades can be dealt with in a gross sense in the same manner as metals, i.e., alternative load paths and load redistribution.

1.1.5 Post-buckling.- Many metallic structures are designed to buckle at load levels considerably below design limit loads. A q/q_{cr} of 4 to 5 is not uncommon. If composite structure were designed to operate only in the unbuckled state, then the potential weight savings offered by the use of advanced composite materials would not be realized.

NASA and DoD programs have already shown that it is possible to operate composite structures in the post-buckled range. Work has also been accomplished on analysis methods and on optimization of skin-stiffener joints.

There are essentially three major voids in the technology today. The first is quantification of the effects of repeated buckling on the skin-to-stringer and frame interfaces. The second is the development of cost effective methods to prevent separation of the skin and stiffener or frame. The frame interface with the skin is critical and is the most likely trouble spot because the pressure inside the fuselage combines with the buckle to cause separation. The third issue is the effect of buckling on damage tolerance. In particular, this refers to disbonding or delamination growth which may occur due to repeated buckling. Reliable methods are needed to provide adequate interlaminar strength at frame/skin and stiffener/skin interfaces.

1.1.6 Shell cutouts.- Joggles introduce undesirable out-of-plane stresses in laminated composite stiffeners and frames. New methods for reinforcing the structure around a cutout need to be developed which will eliminate or minimize these effects. The effects of interlaminar stresses in the edges of the cutouts could be much more severe under the complex loads in a fuselage than in a structure designed to carry load primarily in one direction. Edges of cutouts are prone to damage. Cargo containers, ground handling equipment, and general wear and tear all contribute to service problems. In metals this does not generally amount to a significant problem and can be accommodated, but, in composites, such damage could be a major maintainability problem. Edge reinforcement techniques need to be developed to minimize or prevent such damage. The large number of cut-outs required on commercial transport airplanes for functional reasons is illustrated in Figure 3.

1.1.7 Automated manufacturing.- Automation falls into two basic categories - robotics and flexible automation. Robotics applies to high volume repetitive work and has applications primary at the detail level. Flexible automation on the other hand has application on the larger scale, low volume level where machines must be rapidly changed from one function or set-up to another.

Robotics development has proceeded under USAF programs for the "factory of the future". The large capital investment required can only be justified when large production programs are planned. Flexible automation, on the other hand, can be phased into the manufacturing facilities. The development needed is in the area of computer integrated manufacturing in all areas of composite parts fabrication and assembly, including quality control.

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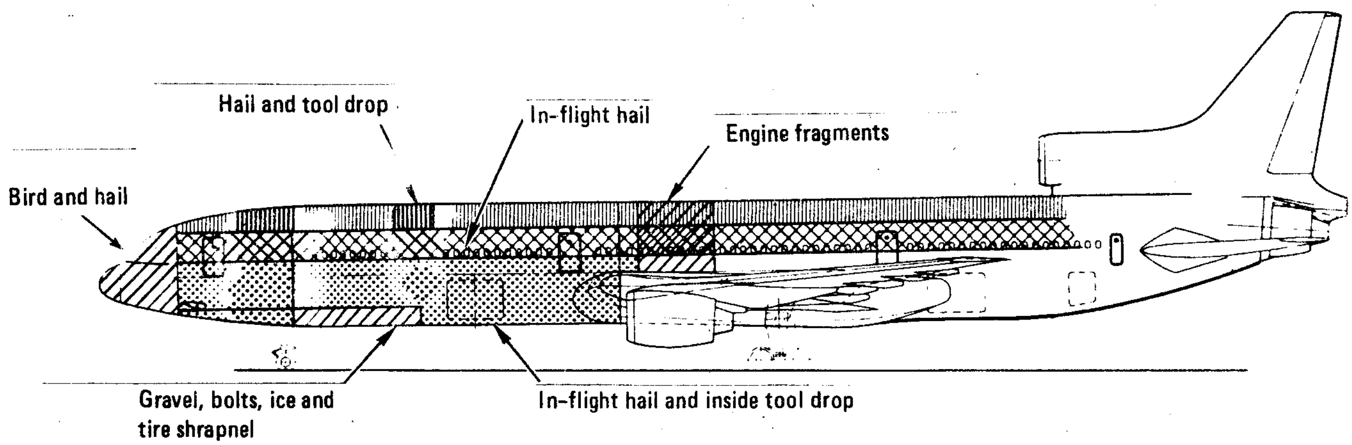

Figure 2. - Locations of principal damage threats.

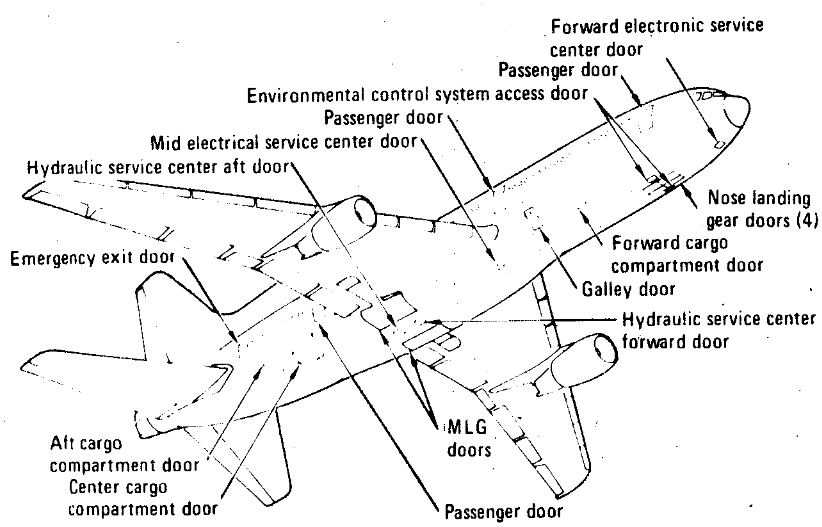

Figure 3. - Typical fuselage cut-outs in commercial transport aircraft.

1.1.8 Processing science.- The technology issue of processing science is not identified as a potential showstopper in the manufacture of an advanced composite fuselage. In fact, a fuselage could be built with current processing methods. However, the advances in processing will have a great impact on the overall performance of the fuselage material and on the fuselage cost.

Performance improvements can be divided into three areas: high strain graphite fibers, toughened epoxies, and thermoplastic matrices. The high strain fibers currently available are characterized by strains-to-failure of about 1.7% or 1.8% compared with current production-type fiber strain values of 1.5%. Eventually, strain-to-failure values are expected to reach 2%. Laminates prepared with these high strain fibers should exhibit better toughness, which will translate into higher design allowables and increased weight savings.

A second method of building tougher laminates is through modification of the epoxy matrix. Toughened epoxies are currently available from several sources and generally are used with high strain fibers. Development of even tougher epoxies is anticipated.

Thermoplastic matrices, principally PEEK-type matrices, have been promoted as superior to epoxy matrices in the areas of processability and toughness. PEEK-type composites should be especially suited to use for stringers and frames on a composite fuselage. Because of the significant processing differences between PEEK-type systems and epoxy systems, evaluation of PEEK-type composites will have to be more extensive than for new epoxies, but should still be completed well before a 1990's composite fuselage production date.

The cost of manufactured composite parts can be reduced through improved processing methods. Among these are dielectric monitoring of the cure processes and closed-loop processing. Investigations are underway to determine the value of dielectric control in improving the processing of composites. Whether or not dielectric control is determined to be of value, closed-loop computer control and monitoring of the cure process should be incorporated into any large-scale composite program. Both of these processing improvements are technologically feasible at the current time and require only the investment of time and money by the various manufacturers to incorporate them into the production process.

Alternatives to autoclave or press cures are being investigated. At present, laminates manufactured by these methods are inferior to autoclave-cured parts, but the potential cost reductions from using these alternate methods are great.

Alternative methods of manufacture offer the greatest potential for cost saving. Many other methods besides hand layup exist, such as pultrusion, filament winding, molding, automated tape laying, etc., are continually being improved.

1.1.9 Electromagnetic effects.- Electromagnetic pulses in airplanes are generated by lightning strike. A transport fuselage is classified as Zone 2A except at the nose and is thus a swept stroke zone with low probability of flash hang-on. Graphite/epoxy conducts electricity but has a much lower conductivity than aluminum. As Zone 2A, a composite fuselage structure requires only minimal protection as graphite/epoxy will dissipate the current. Protection against structural damage is generally only required in areas subject to direct strike.

Protection of digital systems and fly-by-wire systems, however, is more of a problem in composite shells than in metal shells. This problem is being addressed specifically in the current "Atmosphere Electricity Hazards Protection" (AEHP) program administered by USAF AFWAL and sponsored by FAA, NASA, and the armed services, with Boeing as prime contractor. The effects of an electromagnetic pulse can be to reverse stored logic, erase computer memories or damage electrical joints. If a system is critical to flight safety, it should be protected regardless of whether the shell is metal or composite.

The USAF/Boeing AEHP program is structured to investigate the protection concepts for four different classes of flight vehicles. One of these is the large transport/bomber class. It is expected that the evaluation testing will be applicable to all the four classes being considered. Pending the results of this program, it does not appear that any significant new programs in this field are necessary.

1.1.10 Repairs.- Much work has been performed regarding the repair of advanced composite structures both in the field and at depot level. Most repairs to primary structures involve the use of mechanical fasteners because bonding often proves to be difficult. Use of flush mechanical fasteners means that repairs often have to be thicker than may otherwise be required so as not to create feather edges due to full depth countersinking. This can create aerodynamic problems. Also, lack of back side access means that blind fasteners must be used with the associated problems of poor clamp-up and loose holes.

Bonding techniques and reliable adhesives need to be developed for repairs which will provide the quality of bond required for primary structure repairs. In particular, high peel strength is desirable for fuselage repairs because of pressurization effects. New NDI/NDE techniques must also be developed to provide for determination of bondline strength.

1.1.11 NDE/NDI.- The major technology issues in the area of NDE/NDI are:

- Defect evaluation - to determine size and type reliably, and to determine the effect of detectable defects on product integrity.

- Bond line evaluation - developing a reliable NDT method capable of determining bond strength and laminate strength through direct readings.

A testing program must be developed to determine what effect particular NDT responses have relative to selected materials and cross sections. This program would develop a detailed NDT data base for graphite/epoxy laminates using ultrasonic, radiographic, and eddy current methods. Data currently generated from NDT inspections do not characterize the discontinuities within the laminates because of technique limitations and a lack of physical testing of laminates containing anomalies. Correlation of NDT data and physical testing must be accomplished by evaluating selected defective laminates in detail. This will provide a baseline document for reference and modification.

A parallel program should be initiated to study the feasibility of developing a reliable NDT method to determine bond strength. Acoustosonic and ultrasonic bond testing methods have been regarded as successes in measuring bond strength; however, these methods rely on data obtained from destructive test programs.

1.1.12 Flame, smoke, and toxicity.- The use of a flammable material for the fuselage of a commercial passenger airplane may pose additional hazards in the event of a crash or fire. The extent of these hazards is unknown but is expected to be small. Two cases should be considered--interior fires and exterior fires. In an interior fire, because such materials as carpets, seat cushions, and seat coverings burn and emit smoke much more readily than graphite/epoxy composites, flame and smoke from the fuselage structure would probably be negligible. In an exterior fire, a composite fuselage may be a benefit because graphite composite has anisotropic heat transfer characteristics. Heat will be conducted over the surface of the fuselage, decreasing the local temperature and increasing the time until the epoxy reaches combustion temperature. Heat transfers less readily through the thickness of the composite, keeping the interior relatively cool and allowing passengers time to exit.

Early in the development of a composite fuselage, the flame, smoke, and toxicity hazards should be examined in detail and the hazards quantified. As the program progresses, developments in aircraft fire safety should be monitored to insure that questions of composite fuselage fire safety can be properly addressed.

1.1.13 Military technology issues.- The technology issues which are important from a military cargo/transport viewpoint are generally the same as the commercial issues. Table 4 shows the military technology issues listed under four technology areas each in rough order of importance. Most of these issues have already been discussed and the differences between military and commercial requirements have an insignificant effect on the fuselage technology development. The main issue which is military peculiar is the battle damage/service damage/repair issue. This issue is being addressed by current and planned DoD programs.

The emergency landing requirements for military aircraft are more severe than for commercial aircraft as can be seen in Table 5. This table compares inertia load factors to be used for the design of seats and their attachment

TABLE 4. - MILITARY AREAS OF CONCERN AND ASSOCIATED TECHNOLOGIES

Logistics (Supportability)	Pressure Integrity	Manufacturing Technology	Design Technology
Splices and Joints Delamination Damage Tolerance Battle Damage Service Damage NDE/NDI Repairability	Splices and Joints Delamination Damage Tolerance Battle Damage Service Damage Durability	Splices and Joints NDE/NDI Main Attachments Handling Processing Tooling Auto. Manuf. Cost Prediction	Splices and Joints Damage Tolerance Battle Damage Service Damage Main Attachments Allowables Optimization Cut-Outs Impact Dynamics Acoustic Response Cost Prediction

TABLE 5. - MILITARY/COMMERCIAL EMERGENCY LANDING COMPARISON OF INERTIA LOAD FACTORS

	Up	Forward	Side	Down
MIL-A-8865 (3.31)	4.0	16.0 (3.0 Aft)	5.5	16.0
FAR 25.261	2.0	9.0	1.5	4.5

structure for the two types of aircraft. These higher 'g' loads may make it much more difficult to build the floor support structure using advanced composite materials in the military configurations than in the commercial.

1.1.14 Generic research.- The major influence of generic research in the development of advanced composite structures is well recognized. This research is traditionally performed by NASA and DoD both in house and under contract to industry and academia. It is most important that this research continues. Some of the areas which will greatly assist in the development of advanced composite fuselage structures are listed below:

- Improved materials and standardization of resin systems to ensure multisource procurement.
- Continued development of standard test methods for chemical, physical, and mechanical properties so that test data can be pooled to provide stronger allowables data bases.
- Understanding of failure mechanisms on the micro and macro levels will greatly assist the designers in selecting structural concepts.
- Effects of defects and damage to build confidence for disposition of rejected parts and defining acceptable limits.

- Damage arrest and containment techniques for fail-safe structures.

- Energy absorption characteristics of materials and structural configurations to provide data bases for impact dynamics design and analysis.

- Development of a data base which will permit correlation of short-term testing with real-time service in adverse environments.

- Bonded repair techniques including better adhesives and processes.

- Quantitative NDE/NDI techniques and rapid large area scanning techniques.

- Design technology for stiffened curved panels with particular attention to interfaces between skin and stiffener and skin and frame under post-buckled conditions and under internal pressure, and the effects of repeated buckling.

- Continued development of damage tolerant designs and definition of realistic threat scenarios.

1.2 Program Options

The objective of this task was to determine the extent of verification testing required to provide confidence that the technology is at hand to support a decision to commit to the production of composite fuselages. The verification test options ranged from fuselage panel tests through full-scale fuselage testing. The relative costs and technical risks of each option were evaluated.

1.2.1 Option 1 - Validation by component testing.- This option would commence with concept evaluation testing. These tests would be structured to evaluate various concepts for the major structural elements. These concepts would be based on the results of the Fuselage Critical Technology programs and would include consideration of producibility and scale-up to full-scale structures. These tests would narrow down the concepts to one or two for each element. The remaining concepts would then be tested in combination to evaluate their compatability as far as is possible at the component level.

Fuselage structures are subject to relatively high loads in the longitudinal direction due to bending and in the hoop direction due to pressurization, as well as high shear loads. These high combined loads have significant effects at all the cutouts in the shell. These effects are peculiar to fuselages; wing and empennage structures generally have significant spanwise loading but much lower chordwise loads. The ground-air-ground pressurization cycle has significant effects at frame-to-skin and floor interfaces. Panel testing introduces edge effects which make it difficult to simulate and evaluate truly the effects of combined loads and pressure with respect to the full-scale barrel section.

Tests would include testing of panels to assess the effects of buckling and pressure on panels which contain cutouts and splices to determine the effects of the induced interlaminar stresses. Static and fatigue tests would be performed to evaluate the damage tolerance and fail-safe characteristics.

Once the overall design concept is established, a number of large curved panels of the order of 10 feet by 6 feet would be tested to verify the concept.

From the military point of view, the concept evaluation plan would be extended to include determination of susceptibility to battle damage and service damage. The former would include projectiles of various types and the latter would include physical contact from cargo items (vehicles, pallets, ammunition boxes, etc.) as well as from ground debris.

Further extensions to this program would include repair assessment in its broadest aspects. The necessity for repair, the performance of the chosen repair, and the acceptance of the repair should all be studied in preliminary programs of the Option 1 level.

Option 1 is the lowest cost approach; it also, however, has the highest technical risk. As described above, panel testing cannot properly simulate the effect of interlaminar stresses which is one of the major problems with advanced composites today. The effects of high local load inputs can be evaluated to a limited extent only in this option.

Thus there are risks that, when a full-scale fuselage design and production program is embarked upon, problems will arise during the fabrication and/or the full-scale ground testing which would have a major cost and schedule impact on the program. This makes it unlikely that any manufacturer would embark on a production program based on the results of Option 1.

1.2.2 Option 2 - Validation by barrel section test.- This option would include all the concept evaluation tests under Option 1 but the verification testing would be accomplished using a full-scale barrel section 20 feet long. A major feature of this option is that full-scale tools and the appropriate processes would be used. This is important because when large scale panels or full barrels are being processed, self-contained tooling may be used so that parts can be fabricated which are too large for autoclaves. The appropriate assembly procedures would be used and thus properly demonstrated. Nondestructive inspection (NDI) techniques could be evaluated on a full-scale basis especially those aimed at field inspection for battle damage assessment.

The barrel section could be used to verify the noise attenuation concepts incorporated in the structure with interior treatments included.

A full structural test program would follow. This test program would consist of the following elements: a limit combined loads test, an ultimate pressure test, one lifetime of fatigue testing, damage tolerance tests for another lifetime, fail-safe tests, repair, and final static tests to failure. A second barrel section might be necessary for impact dynamics verification in a dynamic drop test. The large curved verification test panels discussed under Option 1 would not be necessary under this option.

The barrel components envisaged for this option will provide the ability to verify realistic three-dimensional loading behavior, which is not possible with panels. The interactions between shell and main frames can also be studied to give greater confidence in the behavior of thick or metal-reinforced frames under high local load inputs (gear loads, hinge loads, etc.).

Measurements of overall deflections with suitably large barrel sections would enable stiffness predictions to be confirmed, thus adding to confidence in estimates of dynamic behavior without recourse to ground vibration tests or flight tests.

Option 2 is a higher cost approach than Option 1 but much of the technical risk is eliminated. A possible variation of this option involved design, fabrication, and test of subscale barrel sections. This was eliminated because the requirements for balanced laminates make it difficult if not impossible to scale the structure down realistically and because, in general, fabrication procedures are not scalable. A laminate stacking sequence must be balanced about its midplane, so that a 45° ply, three plies in from one surface, is balanced by a 45° ply three plies in from the other surface. Each 45° ply also requires a 135° ply in the laminate to balance it. So 45° and 135° plies are always in multiples of 4 plies. Similarly, 0° and 90° plies must each be balanced relative to the midplane though not necessarily relative to each other. Unbalanced laminates not only warp and bend during cure but when load is applied, a bending-twisting coupling occurs causing out-of-plane deformations. This is the anisotropic effect. While a full-scale barrel section is expensive, it represents a cost-effective option because of the confidence level achieved to proceed to production. This option fully exercises the design and analysis methodology and the manufacturing and inspection procedures. It also provides a good cost data base for the projection of production costs.

1.2.3 Option 3 - Validation by full-scale fuselage ground test. - This option would provide the most realistic ground test and would fully check out the structure. It is, however, a very much more expensive option than Option 2. This added expense is not justified by the additional data and confidence achieved. The full-scale barrel test will supply all the manufacturing data and confidence required to commit to production. The additional test data from a full-scale fuselage is not necessary for a design commitment for production to be made. A full-scale fuselage test would be performed as a natural occurrence in a production program at a later date.

Option 3 is an expensive option and does little to change what technical risk exists in Option 2. The additional confidence achieved does not justify the cost.

1.2.4 Option 4 - Validation by full-scale fuselage flight test.- Lockheed has long maintained that flight testing of composite structures is not required prior to a production commitment. Analysis and full-scale ground tests are adequate if properly conducted and accomplished.

Flight test does not provide the long-term real-world operational scenario which would be needed. A prototype test airplane cannot simulate the day-by-day service environment and the normal airline handling and maintenance procedures.

Option 4 is the most expensive option and it is doubtful that the technical risk will be reduced significantly. Flight test could consist of a barrel section installed in an existing metal airplane or a complete advanced composite fuselage. The barrel section introduces the additional problem of splicing to metallic barrel sections and thus severely limits the design options for the barrel. The differences in thermal expansion between the metal and the composite will introduce thermal stresses which will significantly effect the design of the joint and any difference in modulus in the hoop direction between the metal and composite barrels will introduce additional stresses due to differential expansion under pressure. The interface effects from both loads and environmental conditions can overshadow the objectives of the flight testing. A full fuselage would be prohibitive in cost for a "one-off" article. The only benefit of flight test would be short term. The airplane would explore the flight envelope but could not provide service life experience in a time frame that would be acceptable. It is also extremely unlikely that any airline would be willing to accept a unique "one-off" airplane. Flying an experimental airplane would never simulate the day to day operation in airline service. Thus the cost would be extremely high and the technical risk would not be significantly reduced from Option 2.

1.2.5 Selection option.- Option 2 is considered as the minimum technical risk option and the most cost effective. Option 1 will not tie the technologies together and will thus leave too many unanswered questions on overall fuselage structural behavior. No manufacturing and inspection experience in full-scale barrel fabrication will be gained, thus little confidence and no costing data base will be achieved.

Option 3 is many times more expensive because of all the tooling which would be required for a complete fuselage as well as labor and materials for fabrication of one article.

Option 4 is unnecessary and would be unlikely to provide any benefit in a realistic time frame.

1.3 Benefits Studies

The benefit studies were accomplished using the Lockheed-California Company aircraft concept evaluation program, Advanced Systems Synthesis and Evaluation Technique (ASSET), and the Lockheed-Georgia Company General Aircraft Sizing Program (GASP).

These programs evaluate the weight, performance and cost of acquiring and operating a particular airplane configuration and optimize the configuration for either minimum acquisition cost, minimum fuel requirements or minimum life cycle/operating costs.

The weight and cost inputs to these programs are based on accumulated historic data and on the results of this study as available at that time. The weight savings for particular items are input to a computer program which then calculates the overall weight savings based on the weight distribution between skin, stringers, frames, etc. The weight savings estimates are considered to be generally conservative and easily realizable within producibility and cost constraints. The skins and stringers in the military and commercial transport are based on the Jay stiffened concept. The frames are based on the concept of Gr/Ep Z frames with Gr/Ep angle shear ites to the skin. These concepts are discussed in Section 2.2.

1.3.1 Military Benefits.- Any assessment of the quantitative benefits expected to result from the application of composite materials to a transport aircraft structure depends on a number of assumptions. The "conventional" aircraft, against which the "innovative" aircraft is compared, must be defined by the same set of ground rules if a credible evaluation is to be made.

The normal procedure in establishing the configuration of a transport aircraft is first to define a payload-range design point, cruising speed and an altitude. Constraints on field length and climb gradient are then chosen, together with engine characteristics, most notably the specific fuel consumption. An optimum design can then be found by iterative methods, incorporating appropriate information on the materials to be used.

If cost is to be reflected in the optimization, the labor and material costs are only two of the additional data items. Acquisition costs depend on the length of the production run (fixed costs must be proportionately allocated, and the recurring costs will decrease along the learning curve), hence the fleet size must be defined. Operational costs are largely dependent on the fuel used, which requires knowledge of the number of flights (or the hours flown). Life-cycle costs combine these, and also necessitate definition of the life in years.

For military accounting, the separation of acquisition from operation leads to different results from those obtained in evaluation of civil direct operating costs; these reflect the interest paid on the purchase cost and the depreciation of the aircraft. The different utilization rates (typical values are 4000 hours/year for commercial airliners and 900 hours/year for military cargo transports) also influence the relative importance of the various cost factors, as do the different fleet sizes (perhaps 500 civil aircraft compared with 80 military airlifters).

The final decision to be made is the choice of optimization parameter. The three principal alternatives are:

(a) acquisition cost,

(b) block fuel,

(c) life-cycle cost: this combines the effects of the other two choices and is frequently regarded as the most logical basis for design optimization; it is also the most difficult to justify, as it implies predictions of cost and utilization for perhaps twenty years into the future.

The implications of the choice of optimization parameter have been examined and will be discussed later.

To provide a meaningful comparison, the conventional and advanced aircraft must be designed to realistic ground rules. In order to take advantage of an extensive data bank which already existed, a Lockheed-Georgia concept for the Advanced Civil/Military Aircraft (ACMA) (reference 2) was selected as the baseline for the present study. The basic design parameters are:

Payload	331,000 lb
Range	4,000 n.mi
Cruise speed	0.8 M
Cruise altitude (minimum)	31,000 ft
Approach speed (half fuel)	142 KEAS
Fleet size	100 units
Fuel price	$2/gallon
Annual utilization	900 hours
Design life	20 years
Optimization parameter	life-cycle cost
Material	aluminum
Cost basis	1983 dollars

The baseline configuration which resulted from the use of Lockheed-Georgia's in-house advanced design General Aircraft Sizing Program (GASP) is shown in Figure 4. The fuselage width and height are designed to accommodate three rows of containers, side-by-side, or the majority of the army heavy equipment items.

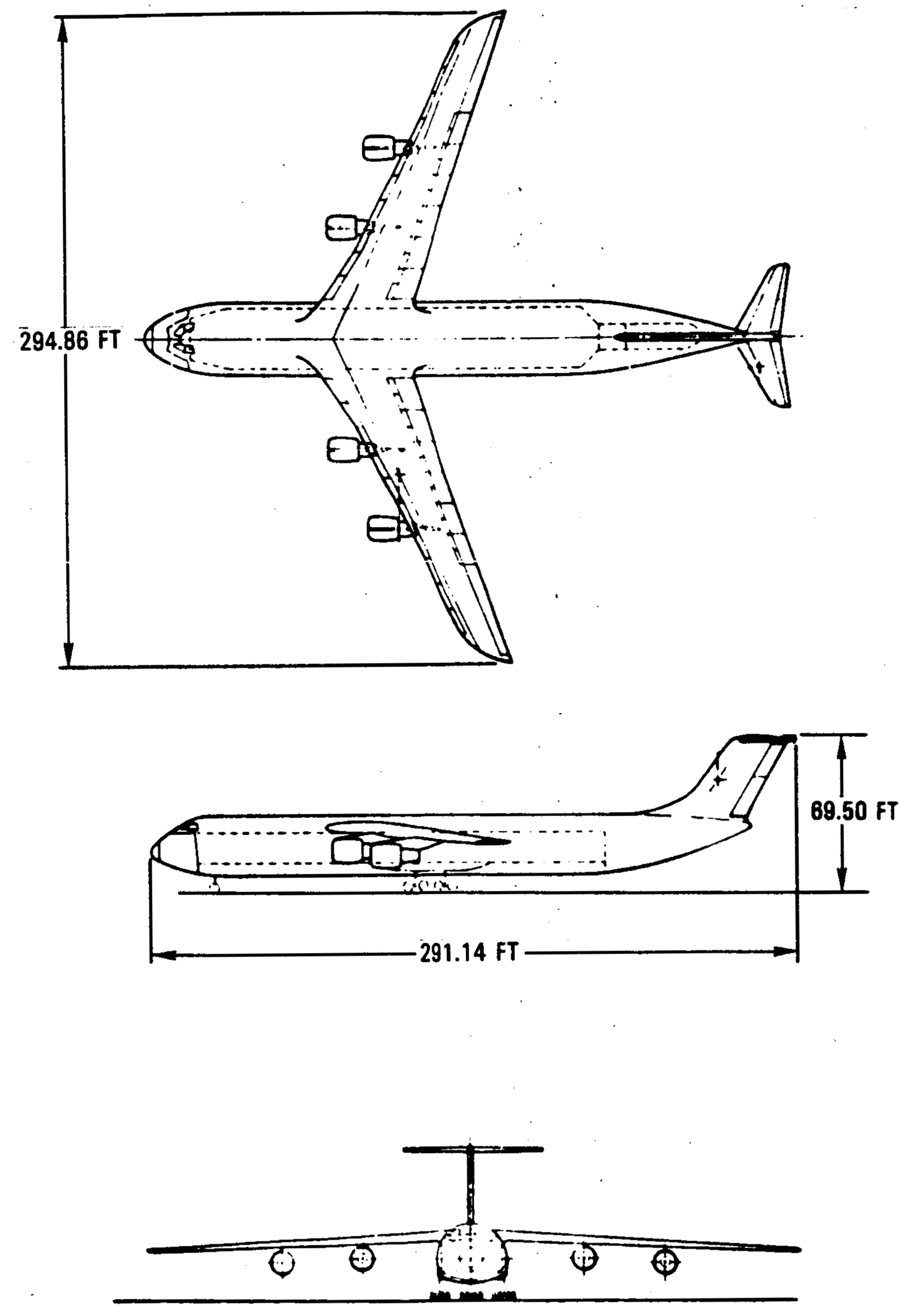

Figure 4. - Military baseline configuration.

Four configurations were analyzed. These configurations are:

1. Baseline conventional aluminum airplane.- The fuselage was sized to accommodate 331,000 pounds of cargo consisting of either the largest likely army vehicle or of three side-by-side rows of 8-foot wide containers. The basic price of the raw materials was assumed to be \$3/lb for sheet and \$6/lb for shaped stock.

2. Advanced aluminum airplane.- This configuration was derived from the baseline by assuming an across-the-board substitution of aluminum-lithium with similar mechanical properties but a density of 94 percent of that of the basic aluminum alloys. With the high fuel price assumed to apply in the mid 1990s, the savings in life-cycle costs were judged to be greater using a lower density aluminum alloy than using a stiffer aluminum. Advantage was taken of the weight saving to resize the aircraft, resulting in a smaller aircraft, thus using smaller engines and less fuel. Payload and fuselage volume remained unchanged. In the absence of any definitive price projections for 1995 it was assumed that sheet stock would cost \$9/lb and shaped raw stock \$18/lb.

3. Advanced technology composite fuselage with conventional aluminum for the remainder of the structure.- This configuration assumed a predominately graphite/epoxy fuselage structure in which the maximum practical use was made of advanced composite materials. Conventional aluminum structure was retained where no clear advantage could be foreseen for composite conversion. In particular, the pilots' cabin, highly loaded main frames, and the cargo floor were retained in aluminum. Thus 70 percent of the fuselage structure was converted to graphite/epoxy. The airplane was resized to take advantage of the weight saving. The graphite/epoxy material was assumed to cost \$32/lb and a low degree of automation was assumed.

4. Advanced technology composite airplane.- The fuselage assumptions were the same as for configuration 3. In this configuration the wing and empennage structures were also converted to graphite/epoxy where a clear advantage would be shown. Thus 80 percent of the wing and 60 percent of the empennage were converted. This configuration was also resized to take advantage of the weight saving. The payload and fuselage volume again remained unchanged.

Each of the above configurations was optimized for minimum life-cycle cost. Table 6 shows a numerical comparison of the weights of each configuration along with the wing area, aspect ratio, and rated thrust/engine.

It should be noted that for the composite fuselage conventional aluminum airplane the fuselage weight changes by only a small amount after resizing. The volume and the payload must remain the same, but because the rest of the aircraft is scaled down, there is some reduction in fuselage loads from the wing, empennage, and gear which results in a small weight saving. Figure 5 shows a graphical weight comparison.

TABLE 6. - MILITARY CONFIGURATIONS WEIGHT COMPARISONS

Quantity/Configuration	Units	Baseline	Advanced Aluminum	Composite Fuselage	Composite Airplane
Wing Area	sq ft	9,487	8,949	8,760	7,818
Aspect Ratio	—	9.14	9.20	9.14	9.63
Rated Thrust/Engine	lb	78,387	75,207	74,283	67,736
Structure Weight					
Wing	lb	178,767	159,344	163,276	112,332
Fuselage	lb	168,882	158,129	132,100	131,107
Empennage	lb	14,454	13,084	13,693	10,290
Other	lb	74,526	66,846	69,747	54,828
Total	lb	436,629	397,403	378,816	308,557
Propulsion	lb	90,230	86,728	85,661	77,527
Systems & Equipment	lb	48,006	47,006	46,897	44,325
Operational Wt. Empty	lb	574,865	531,137	511,374	430,409
Cargo	lb	331,000	331,000	331,000	331,000
Zero Fuel Weight	lb	905,865	862,137	842,374	761,409
Block Fuel	lb	289,716	277,854	274,322	247,108
Reserve Fuel	lb	52,577	50,380	49,677	44,702
Airplane Gross Weight	lb	1,248,158	1,190,371	1,166,373	1,053,219

Table 7 summarizes the cost data for the four configurations. The structure costs include material and labor, with the exception of engineering and QA. These costs are average costs for the last 95 of a total of 100 aircraft produced. The first 5 are included in the RDT&E cost. Other structure include pylons, nacelles and landing gear. Production support covers engineering and QA and includes warranty cost. The profit is assumed to be 20 percent of airframe less engines. Fleet recurring cost is unit recurring cost times 95 planes. RDT&E includes design, structural test articles, testing, etc. and five aircraft for flight test with cost of refurbishing prior to delivery. Thus fleet total cost is for 100 aircraft.

The unit acquisition costs are compared graphically in Figure 6 and unit life-cycle costs are compared graphically in Figure 7.

Table 8 summarizes the weight, fuel and cost savings.

Figures 8, 9 and 10 compare graphically the airplane gross weight savings, the fuel cost savings and the life-cycle cost savings.

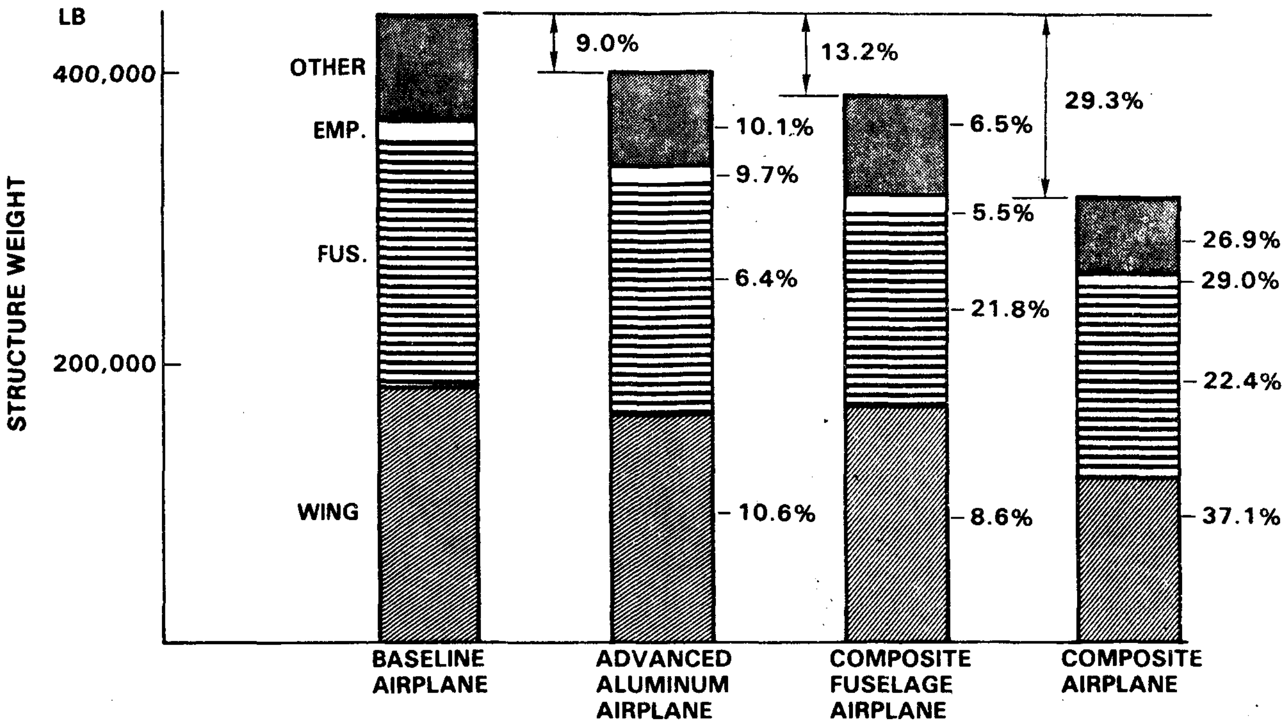

Figure 5. - Military configurations weight comparisons.

The above comparisons are affected by a number of choices that must be made during the optimization process. The effects of two of the more influential of these decisions are discussed below.

Influence of choice of optimization parameter: The configurations described previously were obtained by optimizing the total life-cycle costs over 20 years, with 100 aircraft flying 900 hours a year and with fuel priced at \$2/gallon. Since the fuel cost represents approximately one-third of the total life-cycle cost, while acquisition cost represents roughly one-half, it can be expected that changing the optimization parameter will result in a major change to the configuration. The effects of two other optimization parameters were examined during the study:

(a) acquisition cost, which represents the immediate short-term benefit

(b) block fuel weight, which reflects the most energy efficient design

The optimum solutions for both the conventional aluminum and the graphite/epoxy airplanes were obtained using the same ground rules as were used for the life-cycle cost optimization. Since acquisition cost and block fuel weight both ignore fuel price, the utilization rate and fuel price have no influence on the solutions. The minimum acquisition cost solution becomes simply the cheapest to build, which (for a constant fuselage) translates into the smallest wing and thus into a low aspect ratio configuration. Table 9 summarizes the results of the study. It can be seen that both the minimum acquisition cost

TABLE 7. - MILITARY COST COMPARISONS

Element	Unit	Baseline	Advanced Aluminum	Composite Fuselage	Composite Airplane
Wing Structure	$M	24.25	24.39	22.36	24.92
Fuselage Structure	$M	28.23	29.40	30.92	30.70
Tail Structure	$M	3.07	3.07	2.94	2.97
Other Structure	$M	8.29	8.81	7.80	8.05
Total Structure	$M	63.84	65.67	64.02	66.64
Propulsion	$M	31.88	31.14	30.93	29.91
Systems and Equipment	$M	15.71	15.05	14.76	13.76
Prod Support and Fee	$M	68.71	67.67	64.76	60.77
Spares and Support	$M	23.95	24.00	23.58	23.94
Unit Recurring Cost	$M	204.09	203.53	198.05	196.14
Fleet Recurring Cost	$M	19388.00	19336.00	18815.00	18633.00
RDT and E Total	$M	8598.00	8535.00	8224.00	7780.00
Fleet Total Cost	$M	27986.00	27871.00	27039.00	26413.00
Avg Unit Acquisition Cost	$M	279.86	278.71	270.39	264.13
Unit Lifetime Fuel Cost	$M	175.85	168.60	166.45	149.50
Other Unit Cost/Lifetime	$M	124.18	118.60	117.09	107.20
Unit Life Cycle Cost	$M	579.89	565.90	553.93	520.80

TABLE 8. - MILITARY WEIGHT AND COST SAVINGS

Element	Unit	Baseline	Advanced Aluminum	Composite Fuselage	Composite Airplane
Fuselage Structure Wt. Saving	%	–	6.4	21.8	22.4
Airplane Structure Wt. Saving	%	–	9.0	13.2	29.3
Airplane Gross Weight Saving	%	–	4.6	6.6	15.6
Acquisition Cost Saving	%	–	0.4	3.4	5.6
Fuel Cost Saving	%	–	4.1	5.3	15.0
Life-Cycle-Cost Saving	%	–	2.4	4.5	10.2

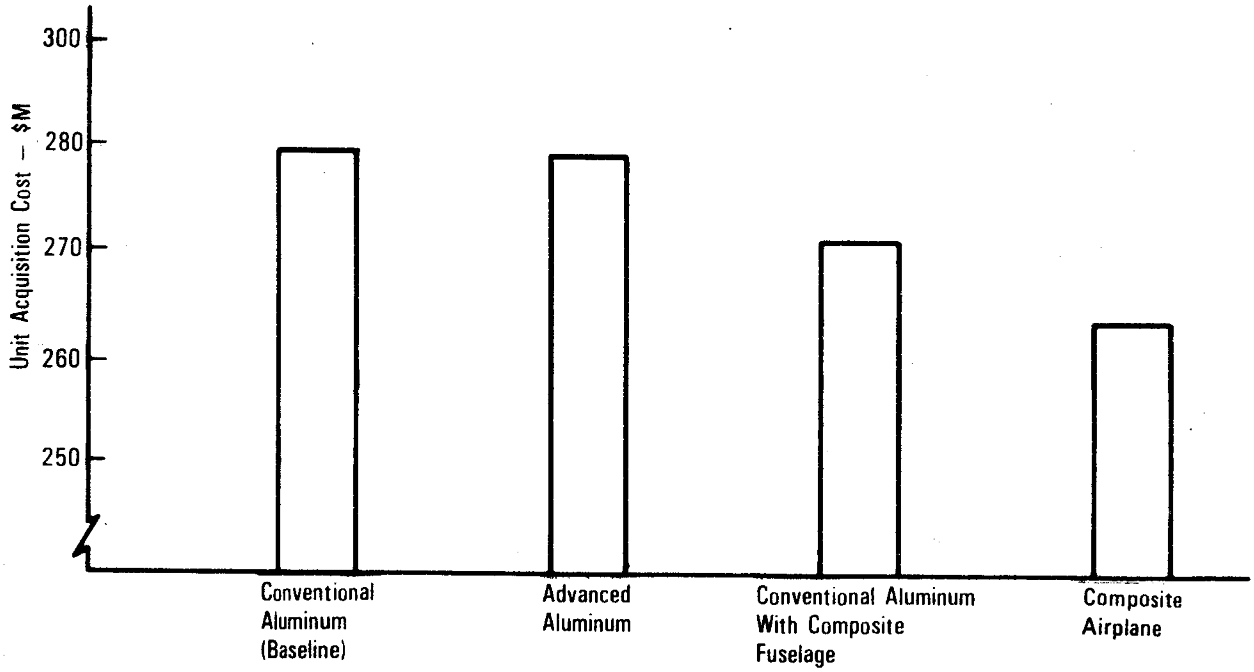

Figure 6. - Unit acquisition costs for various military aircraft configurations.

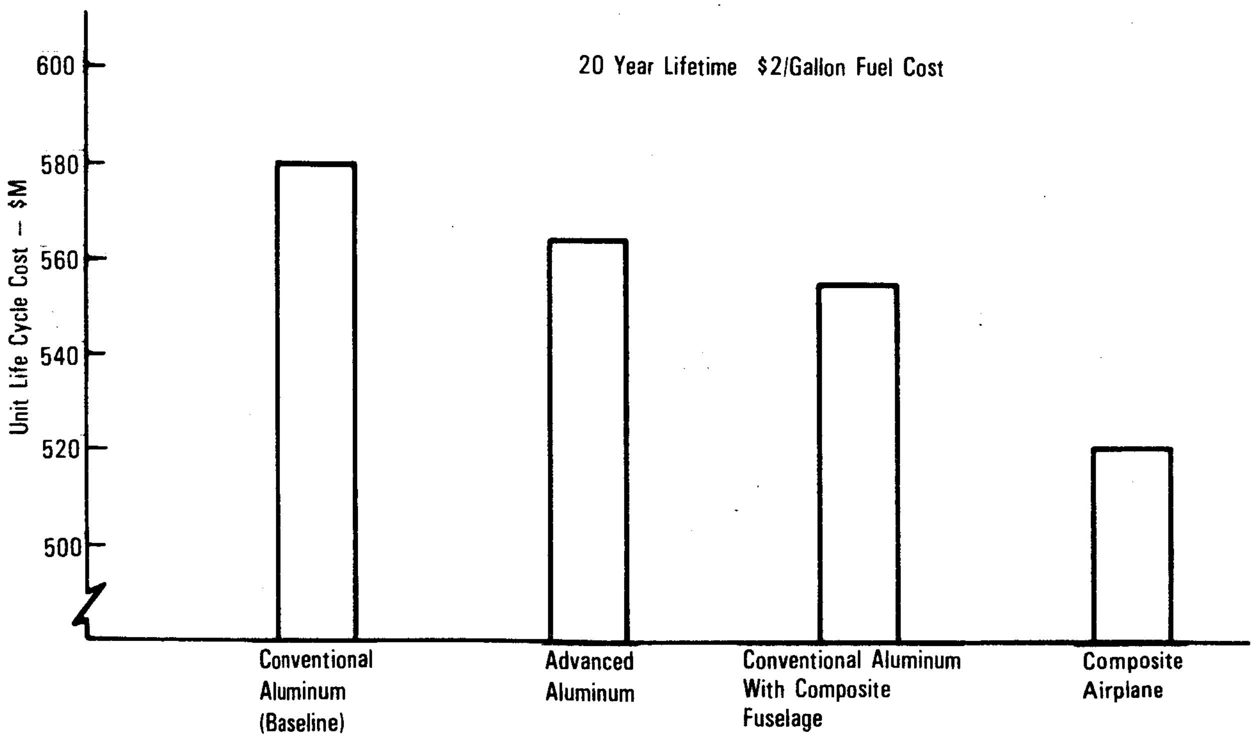

Figure 7. - Unit life cycle costs for various military aircraft configurations.

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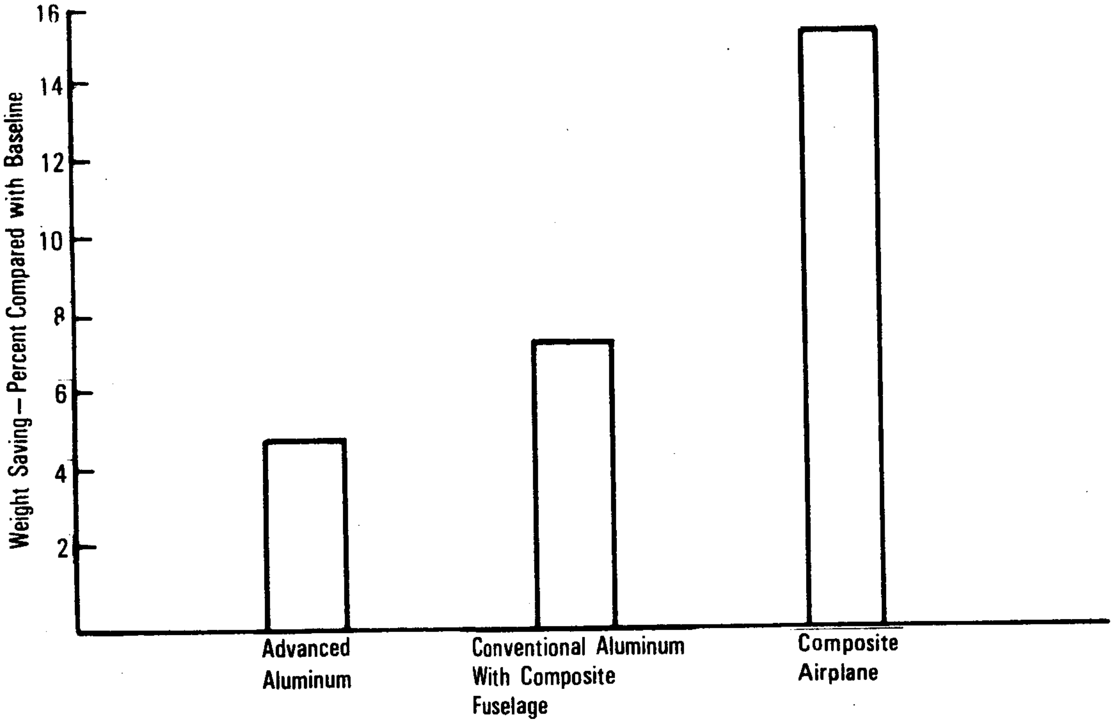

Figure 8. - Aircraft gross weight savings for various military aircraft configurations.

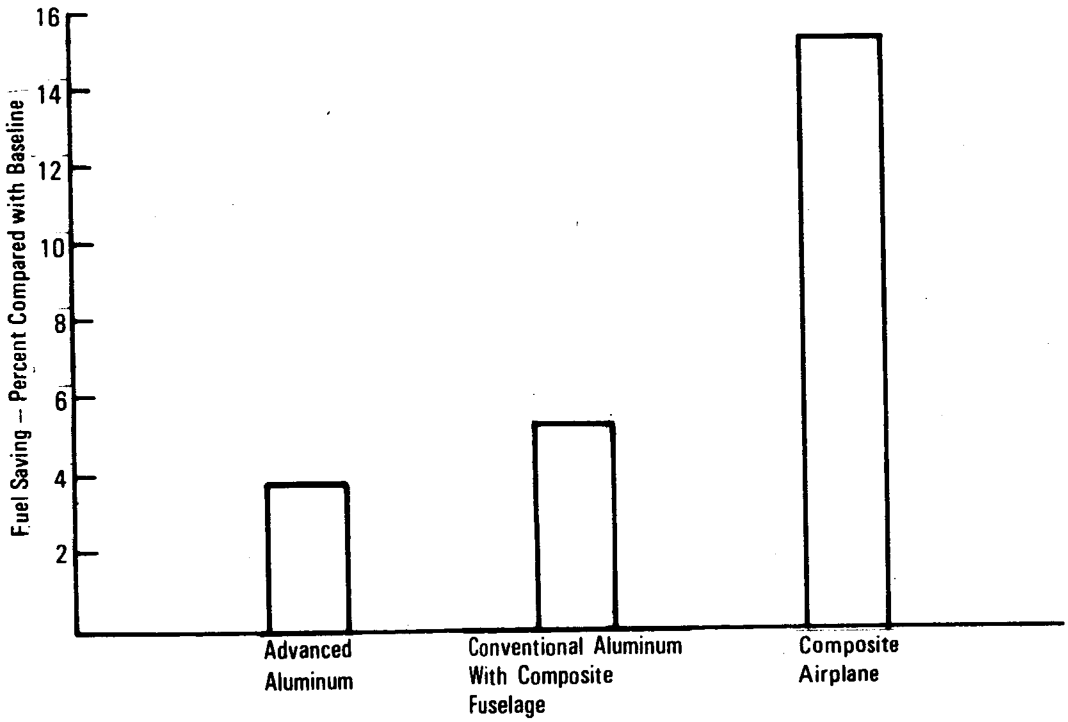

Figure 9. - Savings in fuel costs for various military aircraft configurations.

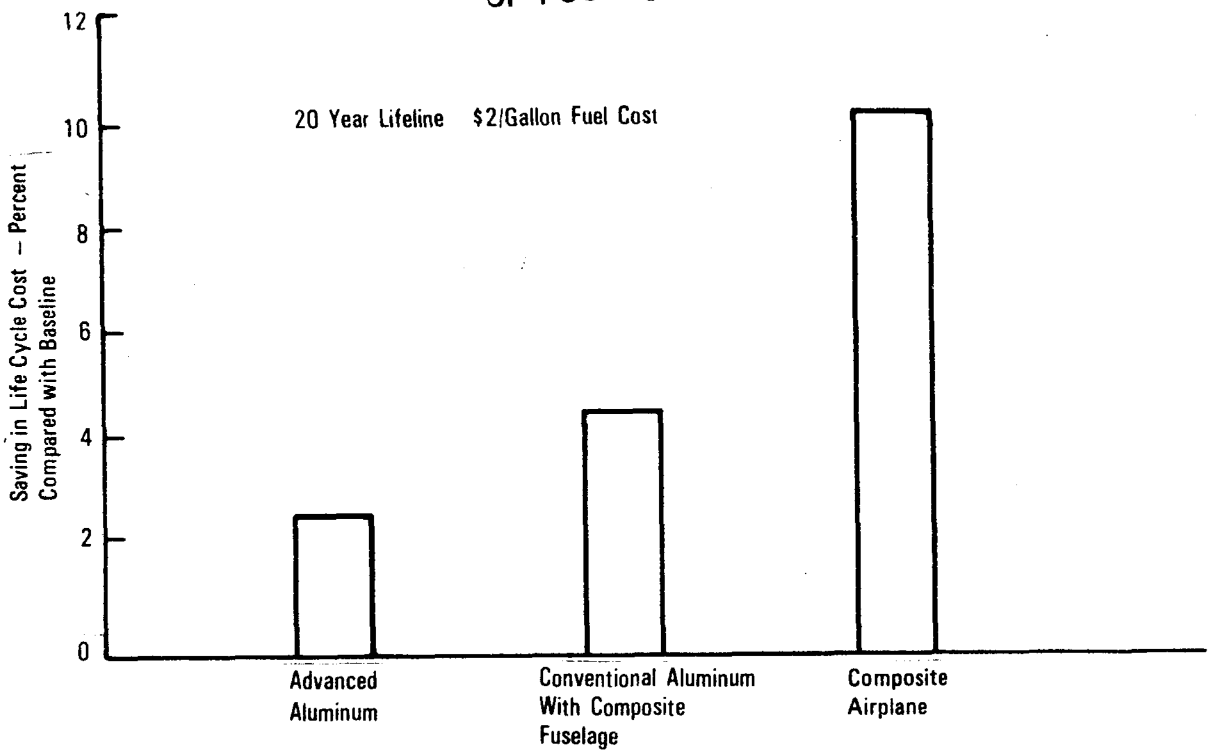

Figure 10. - Savings in life cycle costs for various military aircraft configurations.

and minimum block fuel solutions have higher life-cycle costs than the solution obtained by minimizing the life-cycle cost directly. For the graphite/epoxy airplane, in the first case the cheaper buy is some 17 percent more expensive in fuel, while in the other case the most fuel-efficient design costs some 17 percent more to buy. Regardless of the optimization parameter selected, the graphite/epoxy airplane is less expensive than the conventional aluminum airplane.

Influence of fuel price: The large contribution of fuel costs to the total life-cycle cost (approximately one-third), combined with the uncertainty attached to predictions of fuel cost in the mid-1990's (since fuel prices are now more a result of political manipulation than of normal supply-and-demand processes), suggested that a study of the sensitivity of the predicted benefits to variations in fuel price would be worthwhile.

The configuration optimization, using total life-cycle cost, was therefore repeated for fuel prices of $1 and $3 per gallon, and the results compared with the original $2/gallon values. The conventional aluminum and graphite/epoxy airframes were both examined for the trends, which are summarized in Table 10 and shown graphically in Figure 11.

From these results, it can be estimated that fuel contributes one half of the total O & S cost if the fuel price reaches $1.50 per gallon, and that the total O & S cost and acquisition cost each contribute 50 percent to total life-cycle cost. At higher fuel prices, fuel costs begin to dominate, until at

TABLE 9. - EFFECT OF CHANGING OPTIMIZATION PARAMETER

Optmn. Param. / Quantity	Conventional Aluminum Airplane			Composite Airplane			Units
	Acq. Cost	LCC	Fuel Wt.	Acq. Cost	LCC	Fuel Wt.	
Wing area	9,594	9,487	10,393	8,102	7,818	9,329	Sq. ft.
Aspect ratio	6.29	9.14	12.35	6.65	9.63	13.84	—
Rated thrust/engine	92,585	78,387	79,724	80,259	67,736	69,891	lb.
Struct. wt., Wing	132,523	178,767	274,534	84,416	112,332	188,837	lb.
Fus.	168,853	168,882	170,112	131,258	131,107	131,855	lb.
Other	92,383	88,980	95,347	68,545	65,118	69,616	lb.
Total	393,759	436,629	539,993	284,219	308,557	390,308	lb.
Prop., sys., & eqpt.	154,830	138,236	140,845	137,706	121,852	126,778	lb.
Operating weight	548,589	574,865	680,838	421,925	430,409	517,086	lb.
Payload	331,000	331,000	331,000	331,000	331,000	331,000	lb.
Zero fuel weight	879,589	905,865	1,011,838	752,925	761,409	848,086	lb.
Block fuel	331,596	289,716	270,252	287,372	247,108	223,922	lb.
Res. fuel	58,854	52,577	49,431	50,930	44,702	40,810	lb.
Gross wt.	1,270,039	1,248,158	1,331,521	1,091,227	1,053,219	1,112,818	lb.
Unit acq.cost (1)	271.5	279.9	317.2	258.2	264.1	309.1	$M
Fuel/a-c/yr	10.11	8.79	8.07	8.75	7.48	6.64	$M
Other costs/a-c/yr (2)	6.68	6.21	6.66	5.71	5.35	5.75	$M
Total fuel/a-c (3)	202.1	175.8	161.3	175.0	149.5	132.7	$M
Other cost/a-c (3)	133.7	124.2	133.3	114.1	107.2	115.2	$M
Unit LCC	607.3	579.9	611.8	547.3	520.8	557.0	$M

(1) Includes RDT&E and spares
(2) Does not include fuel
(3) 20 years

TABLE 10. - INFLUENCE OF FUEL PRICE ON MINIMUM LCC DESIGN

	Conventional Aluminum Airplane			Composite Airplane			Units
Quantity \ Fuel Price	1	2	3	1	2	3	$/gal
Wing area	9,450	9,487	9,371	7,730	7,818	7,830	Sq. ft.
Aspect Ratio	8.63	9.14	9.39	9.04	9.63	10.29	—
Rated thrust/engine	79,828	78,387	77,680	68,700	67,736	67,160	lb.
Str. wt. - Wing	168,324	178,767	184,411	106,258	112,332	121,934	lb.
Fus.	168,793	168,882	168,925	131,091	131,107	131,157	lb.
Other	88,869	88,980	88,931	65,226	65,118	65,194	lb.
Total	425,986	436,629	442,267	302,575	308,557	318,285	lb.
Prop., sys., eqpt.	139,819	138,236	137,369	122,706	121,852	120,946	lb.
Operating weight	565,805	574,865	579,636	425,281	430,409	439,231	lb.
Payload	331,000	331,000	331,000	331,000	331,000	331,000	lb.
Zero fuel weight	896,805	905,865	910,636	756,281	761,409	770,231	lb.
Block fuel	293,674	289,716	287,198	253,161	247,108	240,360	lb.
Reserve fuel	53,137	52,577	52,200	45,692	44,702	43,592	lb.
Gross wt.	1,243,616	1,248,158	1,250,034	1,055,134	1,053,219	1,054,165	lb.
Unit acq. cost (1)	277.2	279.9	281.9	261.2	264.1	268.7	$M
Fuel/a-c/year	4.46	8.79	13.07	3.84	7.48	10.87	$M
Other costs/a-c/year (2)	6.22	6.21	6.20	5.36	5.35	5.37	$M
Total fuel/a-c (3)	89.2	175.8	261.4	76.8	149.5	217.4	$M
Other cost/a-c (3)	124.5	124.2	124.0	107.2	107.2	107.4	$M
Unit LCC	490.9	579.9	667.3	445.2	520.8	593.5	$M

(1) Includes RDT&E and spares
(2) Does not include fuel
(3) 20 years

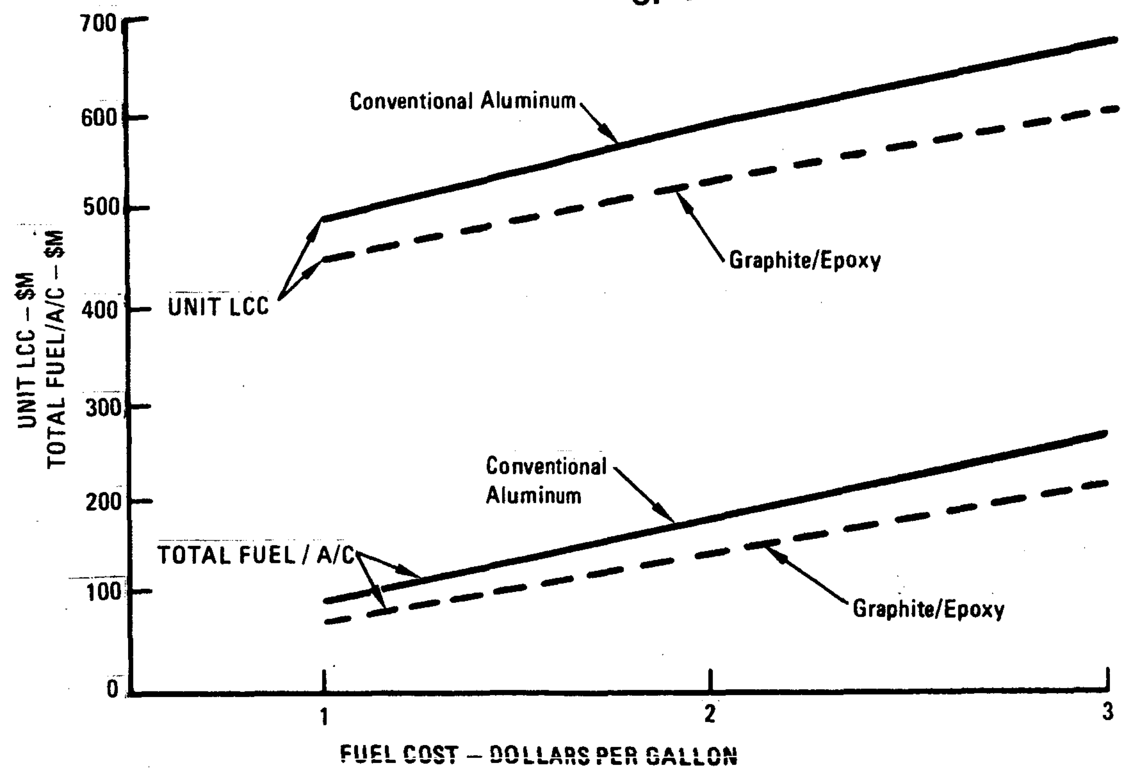

Figure 11. - Influence of fuel price on LCC.

$3.00 per gallon, they almost equal acquisition cost, each providing about 40 percent of the total life-cycle cost.

Similarly, the benefits of replacing aluminum with graphite/epoxy composite material increase as fuel costs increase. For fuel at $1 per gallon the life cycle cost is reduced by 9.3 percent but by 11.1 percent when fuel is $3 per gallon. The total fuel cost over 20 years is lowered by 13.9 percent when fuel is at $1 per gallon but by 16.8 percent when fuel is at $3 per gallon. Since fuel price is no longer a normally predictable economic factor but more subject to international political pressures reducing fuel usage is an important consideration.

Adoption of the minimum acquisition cost design will achieve less total saving because of the greater fuel consumption of the low aspect ratio wing. Selection of the minimum block fuel design, although gaining a little more benefit in fuel cost, will cost more to buy, and the life-cycle cost will be greater.

1.3.2 Commercial benefits.- Commercial benefit analyses were performed using several versions of the ATX-350I airplane shown in Figure 12. These configurations were:

1. Baseline conventional aluminum airplane

2. Advanced aluminum airplane

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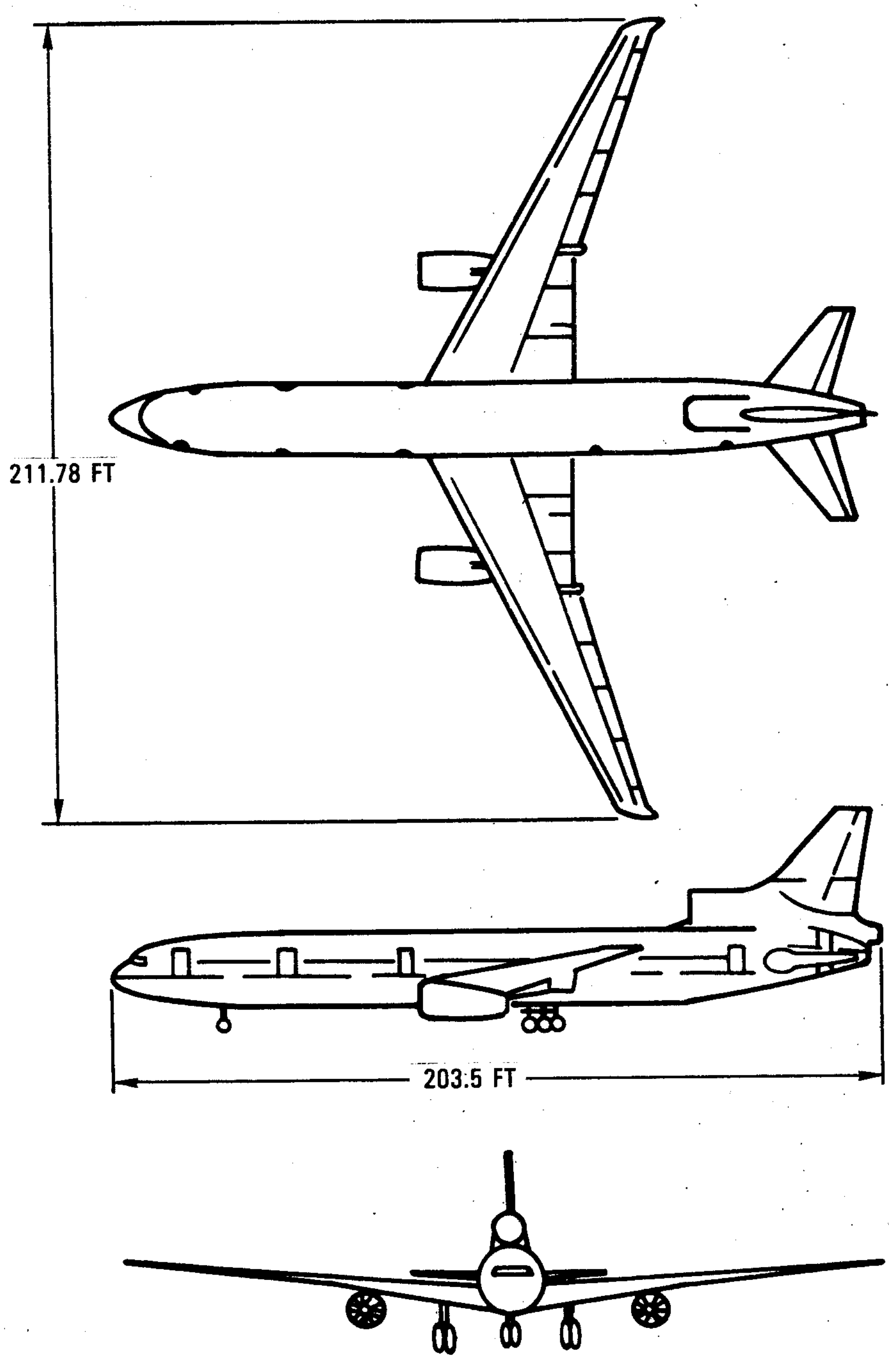

Figure 12. - ATX-350I general arrangement.

3. Advanced technology composites fuselage with conventional aluminum for remainder of airframe structure.

4. Advanced technology composites airplane with conventional aluminum for remaining aluminum structure.

Each configuration makes use of advanced systems and propulsion and all but the baseline include advanced structural concepts. A more detailed description of each configuration is presented later in this section.

The weight breakdown of the baseline fuselage is shown in Table 11. This weight breakdown was reviewed to define the percent conversion to composites and the anticipated percent weight savings for input to ASSET. These are also shown in Table 11.

The cost model in ASSET includes three routines for development, procurement, and operation and support (direct and indirect operating costs). The aircraft manufacturing section in the procurement routine operates at a subsystem level (wing, tail, electrical, etc.) for manufacturing labor and for material. The structural elements (wing, tail, body, etc.) are further subdivided by material type to provide for the various structural material mixes.

TABLE 11. - ESTIMATED COMPOSITE FUSELAGE WEIGHT SAVINGS

	Orig Struc Wt	% Conv to Comp	% Wt Saved	Wt Orig Mat	Wt Comp Wt	Total Struc Wt	% Comp After Conv
Fuselage	100.0	75.0	15.2	25.0	59.8	84.8	70.5
Skins	29.4	100.0	22.0	0.0	22.9	22.9	100.0
Stringers	7.1	100.0	26.0	0.0	5.3	5.3	100.0
Splices	1.5	80.0	18.0	0.3	0.9	1.2	75.6
Bulkheads	11.0	66.4	15.0	3.7	5.7	9.4	60.5
Frames	8.8	100.0	15.0	0.0	7.5	7.5	100.0
Pressure deck	7.2	93.2	15.0	0.5	5.6	6.1	92.0
Floor support	9.9	100.0	18.0	0.0	8.1	8.1	100.0
Floor & fairings	6.9	0.0	0.0	6.9	0.0	6.9	0.0
Doors	5.3	87.0	15.0	0.7	3.8	4.5	84.7
Windows	3.7	0.0	0.0	3.7	0.0	3.7	0.0
Misc	9.2	0.0	0.0	9.2	0.0	9.2	0.0
Total Weight Savings = 15.2%							

The basic design paremters for the ATX-350I configurations used in these analyses are:

Payload	73,500 lb
Range	4,600 n.mi
Cruise speed	0.8 M
Cruise altitude	36,000 ft

Average stage length	2,500 n.mi
Block time	5.99 hr
Flight time	5.66 hr
Utilization	4,142 hr/year
Fare	$265.10
Cost basis	1980 dollars
Total procurement	300 airplanes

A more detailed description of the ASSET program in general, as well as the procedures followed, data used and configurations analyzed in this study, is presented on the following pages.

The ASSET program resizes each configuration from the baseline to take advantage of the weight saved.

The fuselage was broken into representative constituent parts based on percent of total fuselage weight. Each part was then examined for application to advanced composites. The percentage converted to composites and the percentage savings for each part were estimated, resulting in post-conversion weight of original material and of advanced composite, as well as the total post-conversion weight.

Table 11 presents the estimated material conversion and weight savings factors resulting from aggressive application of advanced composite materials in the fuselage.

The study indicated a potential weight savings of 15 percent on a fuselage with 75 percent of the material converted to composites. These estimates are based on design ultimate strain allowables of 0.0060 in/in tension, 0.0045 in/in compression and 0.0048 in/in shear. The compression strain cutoff is a minor influence in the fuselage because structure which is compression critical tends to be designed by buckling criteria and usually buckles at strains of 0.0020 to 0.0030 in/in. Local post-buckled strains were not considered a limiting factor. Increasing compression strain allowables would possibly permit more conversion of metal structure but in the fuselage this would have a minor effect on the weight saved.

The final configurations output from ASSET have been resized to take advantage of the weight saving. A reduction in weight causes a reduction in the wing and tail loads. The wing and tail can now be reduced in size so that the loading is increased back to the optimum. This in turn reduces the weight of the wing and tail and a cascading weight saving results. The fuselage must remain a constant volume to satisfy the payload requirements so planform area remains constant. The reduced wing and tail loads however do reduce the fuselage shears and bending moments to some extent and thus permit some reduction in overall fuselage weight. Thus the final resized configuration of an airplane with a composite fuselage will show a fuselage weight savings a little higher than the calculated input value of 15.2 percent shown in Table 11. The standard format for commercial configuration weights is discussed below.

The weight of the structure for wing, fuselage, empennage, landing gear, and nacelles is totaled to give total structure weight. The propulsion and systems are then added to give the manufactured empty weight (MEW). This includes furnishings and other items of equipment that are an integral part of the aircraft. It also includes any fluids that are contained in closed systems.

The standard items and operational items are now added in to obtain the operational empty weight (OEW). The standard items consist of equipment and fluids that are not an integral part of the airplane but do not vary between aircraft of the same type. Normally included are:

Unusable fuel
Engine unusable oil
Engine usable oil
Chemical lavatory fluids
Basic emergency and oxygen system equipment
Supplemental electronic equipment

Operational items consist of personnel, equipment, furnishings and supplies necessary for particular flight operations, unless some of these items have already been included in the empty weight. Normally included are:

Crew, cabin attendants and their baggage
Manuals and navigational equipment
Cabin service
Food and beverages
Usable washing and drinking water
Overwater emergency equipment
Cargo containers

The payload is now added in to give the zero fuel weight (ZFW). Payload is passengers and cargo.

Finally, the available fuel is added to give the aircraft gross weight.

Table 12 shows a comparison of the resized weights of the four configurations and Figure 13 shows a graphical comparison of the gross weights and Figure 14 shows the structure weight comparisons. Immediately following is a detailed description of each configuration and a breakdown of the major components into their constituent materials.

Configuration 1-baseline conventional aluminum airplane: The material mix in the various parts of the airplane was defined based on the L-1011. The materials are broken down into the following categories: aluminum, titanium, steel, composite, and other (see Table 13). For the baseline configuration, composite consists of fiberglass and Kevlar 49.

TABLE 12. - COMMERCIAL CONFIGURATIONS WEIGHT COMPARISONS

Quantity/Configuration	Units	Baseline	Advanced Aluminum	Composite Fuselage	Composite Airplane
Wing Area	sq ft	4,708	4,510	4,466	4,029
Aspect Ratio	—	12.00	12.00	12.00	12.00
Rated Thrust/Engine	lb	44,120	42,226	41,855	37,755
Structure Weight					
Wing	lb	84,026	76,105	79,037	53,741
Fuselage	lb	66,478	62,576	55,608	54,761
Empennage	lb	8,758	7,802	8,042	5,313
Land Gear	lb	21,976	20,758	20,848	18,806
Nacelle	lb	11,355	10,558	10,700	9,171
Total Structure Weight	lb	192,593	177,799	174,235	141,792
Propulsion	lb	33,444	32,057	31,755	28,689
Systems	lb	59,911	59,607	59,540	58,873
Manufactured Empty Wt.	lb	285,949	269,464	265,530	229,354
Operational Items	lb	10,345	10,335	10,333	10,313
Standard Items	lb	9,577	9,569	9,568	9,550
Operational Empty Wt.	lb	305,870	289,368	285,431	249,218
Payload	lb	73,500	73,500	73,500	73,500
Zero Fuel Weight	lb	379,370	362,868	358,931	322,718
Available Fuel	lb	158,678	152,568	151,493	137,709
Aircraft Gross Weight	lb	538,048	515,436	510,424	460,426

TABLE 13. - PERCENTS OF MATERIALS BY WEIGHT (CONFIGURATION 1)

	Aluminum	Titanium	Steel	Composite	Other
Wing	89	3	4	2	2
Tail	75	8	5	0	12
Fuselage	84	6	1	2	7
Landing Gear	23	0	31	0	46
Nacelles	21	55	13	10	1
Air Induction	96	4	0	0	0

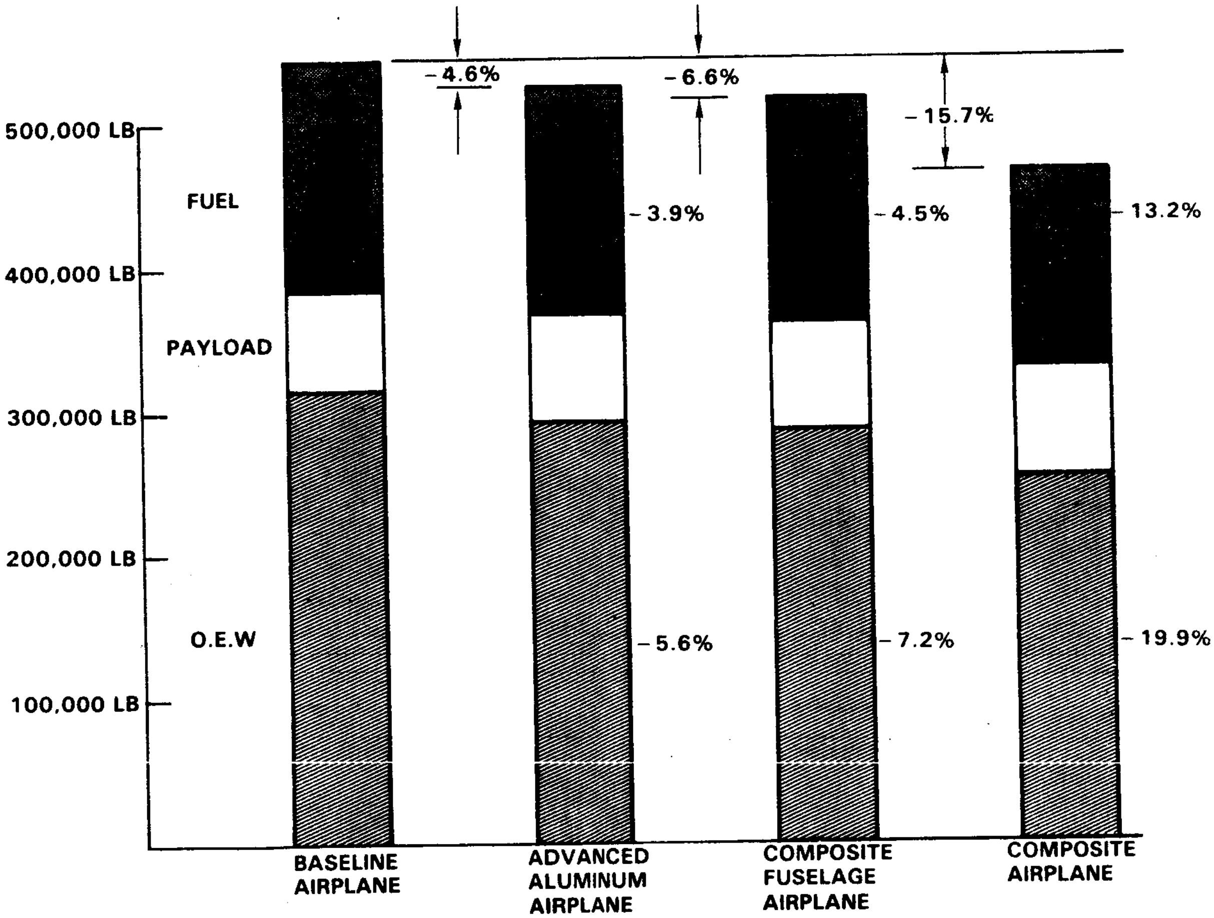

Figure 13. - Commercial aircraft gross weight savings.

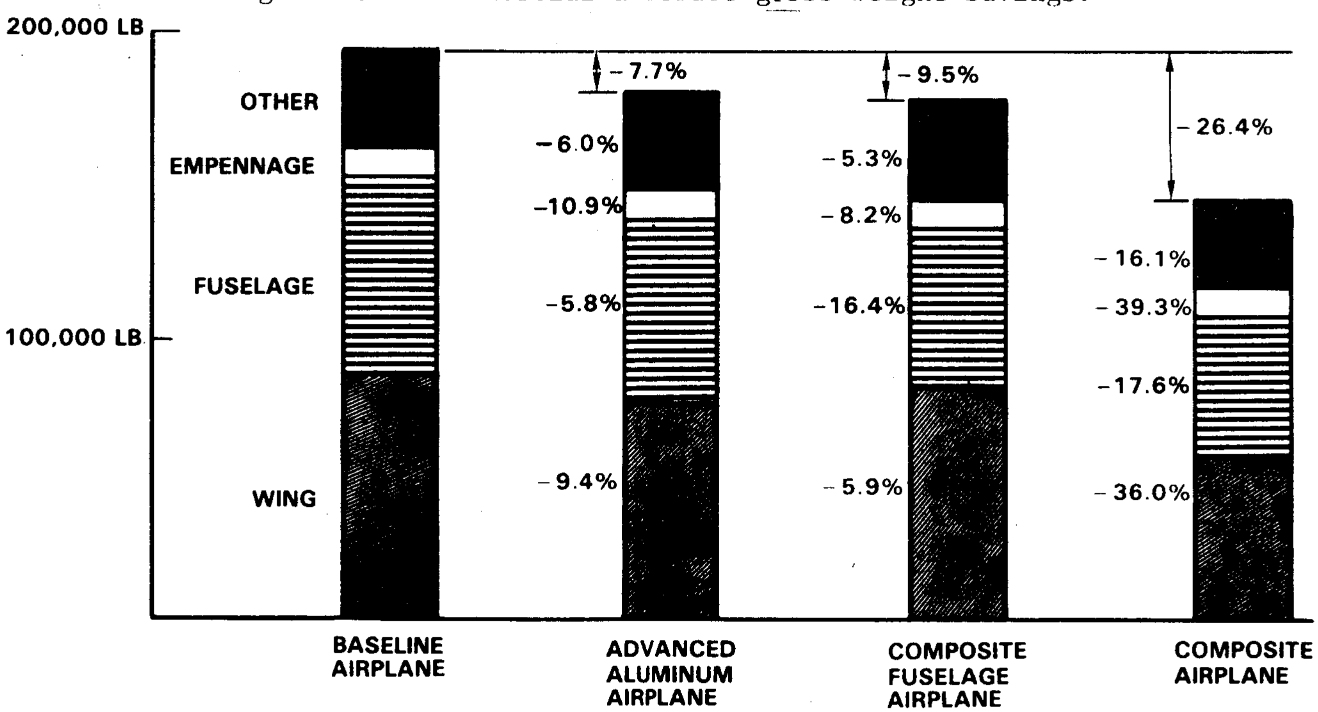

Figure 14. - Commercial aircraft structure weight savings.

The data shown in Table 13 were input to the ASSET program. The program outputs the total weight of each material in each component after sizing the airplane (or resizing in the later configurations). These weights for the baseline airplane are shown in Table 14.

Configuration 2 advanced aluminum airplane: This configuration is based on the same assumption as the military configuration, that is that all the aluminum is replaced by aluminum-lithium alloys of the same strength and stiffness but with a density of 94 percent of the conventional aluminum alloys. The material mix of the major components is shown in Table 15.

The data shown in Table 15 were input to the ASSET program. The weight matrix presented in Table 16 shows the weights output by the program after resizing.

Configuation 3 composite fuselage: This configuration assumed the composite conversion for the fuselage shown in Table 12. The material mix in the airplane is shown in Table 17. It should be noted that the percent composite in Table 17 for the fuselage is higher than that shown in Table 12 because of the inclusion of fiberglass.

Table 17 data were input to ASSET. The final weights of the various materials in each component after being resized by ASSET are shown in Table 18.

Configuration 4 all-composite airplane: The all-composite airplane is based on maximum use of composites but still retains metal in areas where it would not be cost-effective to use composites. The weight savings estimated for the wing is 25 percent and for the tail 22 percent. The same 15 percent for the fuselage is used as with configuration 3. The mix of materials by weight percent is shown in Table 19. These data were input to ASSET.

The final weight of the various materials in each component after resizing by ASSET is shown in Table 20.

TABLE 14. - CONFIGURATION 1 MATERIAL-WEIGHTS MATRIX

	Aluminum	Titanium	Steel	Composite	Other	Total
Wing	74,783	2,521	3,361	1,681	1,681	84,026
Tail	6,569	701	438	0	1,051	8,758
Fuselage	55,842	3,989	665	1,330	4,653	66,478
Landing Gear	5,055	0	6,813	0	10,109	21,976
Nacelle	1,798	4,710	1,113	856	86	8,564
Air Induction	2,680	112	0	0	0	2,791
Totals	146,726	12,032	12,390	3,866	17,580	192,593

TABLE 15. - PERCENTS OF MATERIALS BY WEIGHT (CONFIGURATION 2)

	Aluminum	Titanium	Steel	Composite	Other
Wing	88	3	5	2	2
Tail	74	8	5	0	13
Fuselage	83	6	1	2	8
Landing Gear	22	0	31	0	47
Nacelle	20	56	13	10	1
Air Induction	96	4	0	0	0

TABLE 16. - CONFIGURATION 2 MATERIAL-WEIGHTS MATRIX

	Aluminum	Titanium	Steel	Composite	Other	Total
Wing	67,201	2,435	3,273	1,598	1,598	76,105
Tail	5,758	655	406	0	983	7,802
Fuselage	52,000	3,942	688	1,314	4,631	62,576
Landing Gear	4,546	0	6,518	0	9,694	20,758
Nacelle	1,610	4,485	1,063	813	81	8,052
Air Induction	2,401	105	0	0	0	2,506
Totals	133,516	11,623	11,947	3,726	16,986	177,799

TABLE 17. - PERCENTS OF MATERIALS BY WEIGHT (CONFIGURATION 3)

	Aluminum	Titanium	Steel	Composite	Other
Wing	89	3	4	2	2
Tail	75	8	5	0	12
Fuselage	11	7	1	73	8
Landing Gear	23	0	31	0	46
Nacelle	21	55	13	10	1
Air Induction	96	4	0	0	0

TABLE 18. -- CONFIGURATION 3 MATERIAL-WEIGHTS MATRIX

	Aluminum	Titanium	Steel	Composite	Other	Total
Wing	70,343	2,371	3,161	1,581	1,581	79,037
Tail	6,032	643	402	0	965	8,042
Fuselage	6,117	3,893	556	40,594	4,449	55,608
Landing Gear	4,795	0	6,463	0	9,590	20,848
Nacelle	1,694	4,438	1,049	807	81	8,069
Air Induction	2,526	105	0	0	0	2,631
Totals	91,507	11,450	11,631	42,982	16,665	174,235

TABLE 19. - PERCENTS OF MATERIALS BY WEIGHT (CONFIGURATION 4)

	Aluminum	Titanium	Steel	Composite	Other
Wing	11	4	5	77	3
Tail	10	10	6	59	15
Fuselage	11	7	1	73	8
Landing Gear	23	0	31	0	46
Nacelle	11	57	13	18	1
Air Induction	72	4	0	24	0

TABLE 20. - CONFIGURATION 4 MATERIAL-WEIGHTS MATRIX

	Aluminum	Titanium	Steel	Composite	Other	Total
Wing	5,912	2,150	2,687	41,380	1,612	53,741
Tail	531	531	319	3,135	797	5,313
Fuselage	6,024	3,833	548	39,976	4,381	54,761
Landing Gear	4,325	0	5,830	0	8,651	18,806
Nacelle	774	4,012	915	1,267	70	7,038
Air Induction	1,536	85	0	512	0	2,133
Totals	19,102	10,611	10,298	86,270	15,511	141,792

The ASSET program requires inputs for cost in the form of material cost factors and labor cost factors. These factors are based on the airplane type, general configuration and historical data. Plenty of historical data exists on metal airplanes. The mix of sheet, plate, extrusion, and forging for a particular configuration can be determined with reasonable accuracy as can the mix of aluminum, titanium, steel, fiberglass, etc. Historical data also exist for many configurations which indicate the "fly weight" of material relative to the "buy weight," or scrap factors. For aluminum in the ATX-350I configuration these scrap factors vary from 1.8 for sheet to 3.3 for forgings. Based on the mix of aluminum forms in the structure, the cost/pound fly weight of conventional aluminum is $7.66 in the wing, $7.26 in the tail, and $6.71 in the fuselage. The corresponding raw material costs for advanced aluminum are $19.10, $17.80, and $16.20. To these costs must be added the costs of vendor machining and other costs not attributable to direct labor. Thus the material cost factors input to ASSET include all nonlabor costs. The material cost factors used in this study are shown in Table 21 and are in $/lb of material by fly weight. There are little historic data for advanced composites and the raw material costs are not as well defined as for metals. The material cost input factor for composites is assumed to be $65.50/lb. This figure has been used in various studies previously and is considered to be a good average all up cost assuming current manufacturing procedures.

The labor costs for the advanced composite configurations are based on current fabrication techniques with minimum automation. The all composite configuration was also analyzed for moderate automation and major automation. These labor cost factors were reduced by 25 percent and 40 percent respectively for the composites.

The costs for each aircraft configuration were based on the following guidelines:

- The development and production costs were determined from cost estimating relations (CERs) developed from total Lockheed experience. The development cost is amortized into production cost for determining depreciation expense.

TABLE 21. - INPUT MATERIAL COST FACTORS ($/LB)

Component \ Material	Conventional Aluminum	Advanced Aluminum	Titanium	Steel	Composite	Other
Wing	29.42	40.17	115.80	41.60	65.50	29.42
Tail	22.47	32.22	123.60	31.20	65.50	22.47
Fuselage	14.53	23.07	73.60	23.40	65.50	14.53
Landing Gear	15.24	30.48	50.70	50.70	—	15.24
Nacelle	33.65	—	98.80	98.80	65.50	33.65
Air Induction	6.76	—	9.20	9.10	65.50	6.76

- The development and production costs were Lockheed's actual January 1980 levels for direct, overhead, and general and administrative rates, plus profit factor.

- The operating costs were determined from 1967 Air Transport Association (ATA) equations with coefficients updated from January 1980 experience.

- Passenger load factors of 60 percent at average stage length and 100 percent at design range.

- Fuel prices of $1, $2, and $3 per gallon.

- Crew of three.

Three cost components are used in defining advanced technology aircraft costs. These are: development, production, and operation.

For development costs, basic program elements are identified within each of the phases. These basic elements were selected at a component or function level where significant cost variations may occur. This is a level where configuration and program variations can be directly reflected in cost and yet at a level compatible with conceptual design analysis. Cost-significant configuration and program parameters were identified and combined into cost estimating relationships (CER) for each basic element. These CERs are programmed within the cost module of the Lockheed ASSET computer program for calculation of investment cost, operating expenses, and return on investment.

The CERs for the development and production costs are formulated from a comprehensive analysis of Lockheed aircraft. Tooling and engine CERs are provided by a RAND Corporation analysis (reference 3) augmented by data from the engine manufacturers. The Lockheed database includes 14 prototypes and 16 production programs.

The outputs of the development and production CERs are, for the most part, in the form of labor hours and material dollars. Hours are translated to dollars, using Lockheed's actual January 1980 direct, overhead and general and administrative rates plus a profit factor of 15 percent.

Development costs include all the costs necessary to design, develop, and demonstrate that the aircraft meets its requirements culminating in FAA certification.

Operational expenses include both direct operating costs (DOC) and indirect operating costs (IOC). The 1967 Air Transportation Association (ATA) equations with coefficients updated to January 1980 experience are used to calculate all elements of DOC. Indirect operating costs are based on a Lockheed-Boeing method of coefficients and factors. The factors were extracted from U.S. Civil Aeronautics Board (CAB 41) data reflecting inputs through 1978.

Economic data for the reference aircraft consist of a cost summary which details total RDT&E program costs, aircraft production cost, and procurement (flyaway) cost per aircraft. A summary of the aircraft operational costs (both direct and indirect) and rate of return on investment was determined for a hypothetical airline operator.

Direct operating costs include:

1. Flight crew

2. Fuel and oil

3. Insurance

4. Depreciation

5. Maintenance

Indirect operating costs include:

1. Ground property and equipment expense -- local and system

   Maintenance
   Maintenance burden
   Depreciation
   Landing fees
   Aircraft servicing
   Servicing administration

2. Aircraft control and communications

3. Cabin attendant expense

4. Food and beverage expense

5. Passenger handling
   Reservations and sales

6. Baggage and cargo handling

7. Passenger service - other expense
   Passenger agency commissions
   Passenger advertising and publicity

8. Freight commissions
   Freight advertising and publicity

9. General and administrative expense

Table 22 and Figures 15 and 16 show a comparison of the airplane costs. The operating costs and return on investment (ROI) analyses are for a 2500 mile flight using fuel costs of $1.00/gallon. The all-composite configuration was analyzed using three different labor cost assumptions. Subconfiguration A represents current fabrication techniques (minimum automation). In Subconfigurations B and C, the labor cost factors were reduced by 25 percent and 40 percent, respectively, to represent moderate and major amounts of automation.

Rate of return on investment analysis assumed an airline operating over a 16-year period purchasing 8 airplanes in year 1, and adding 8 in year 2 and 7 in year 3 for total fleet of 23.

Table 23 shows a summary of the cost and weight savings compared to the conventional aluminum baseline. These data are also presented graphically in Figures 17 and 18.

TABLE 22. - COMMERCIAL COST COMPARISON

Element	Units	Baseline	Advanced Aluminum	Composite Fuselage	Composite Airplane [1] A	Composite Airplane [1] B	Composite Airplane [1] C
Wing Structure	$M	7.773	7.884	7.361	6.866	6.173	5.840
Fuselage Structure	$M	7.868	7.971	8.934	8.917	7.927	7.382
Tail Structure	$M	1.041	1.001	0.963	0.831	0.759	0.723
Other	$M	3.496	3.393	3.319	3.026	2.985	2.960
Total Structure	$M	20.178	20.249	20.577	19.640	17.844	16.905
Propulsion	$M	13.910	13.426	13.320	12.234	12.234	12.234
Systems	$M	13.239	13.127	13.145	12.923	12.923	12.923
Other	$M	11.329	11.201	9.625	10.975	10.332	9.995
Total Production	$M	58.656	58.003	58.667	55.772	53.333	52.057
Support	$M	3.149	3.118	3.153	3.013	2.898	2.838
Spares	$M	8.623	8.459	8.514	8.024	7.758	7.619
Prod. Dev	$M	0.737	0.726	0.745	0.708	0.680	0.665
Total Procurement	$M	71.165	70.306	71.079	67.517	64.669	63.179
DOC	c/Seat Mile	2.957	2.893	2.893	2.717	2.686	2.669
IOC	c/Seat Mile	2.866	2.849	2.845	2.809	2.809	2.808
TOC	c/Seat Mile	5.823	5.742	5.738	5.526	5.495	5.477
ROI	%	8.290	9.100	9.000	11.270	11.930	12.290

(1) A – Manufactured using current techniques (minimum automation)
B – Moderate automation
C – Major automation

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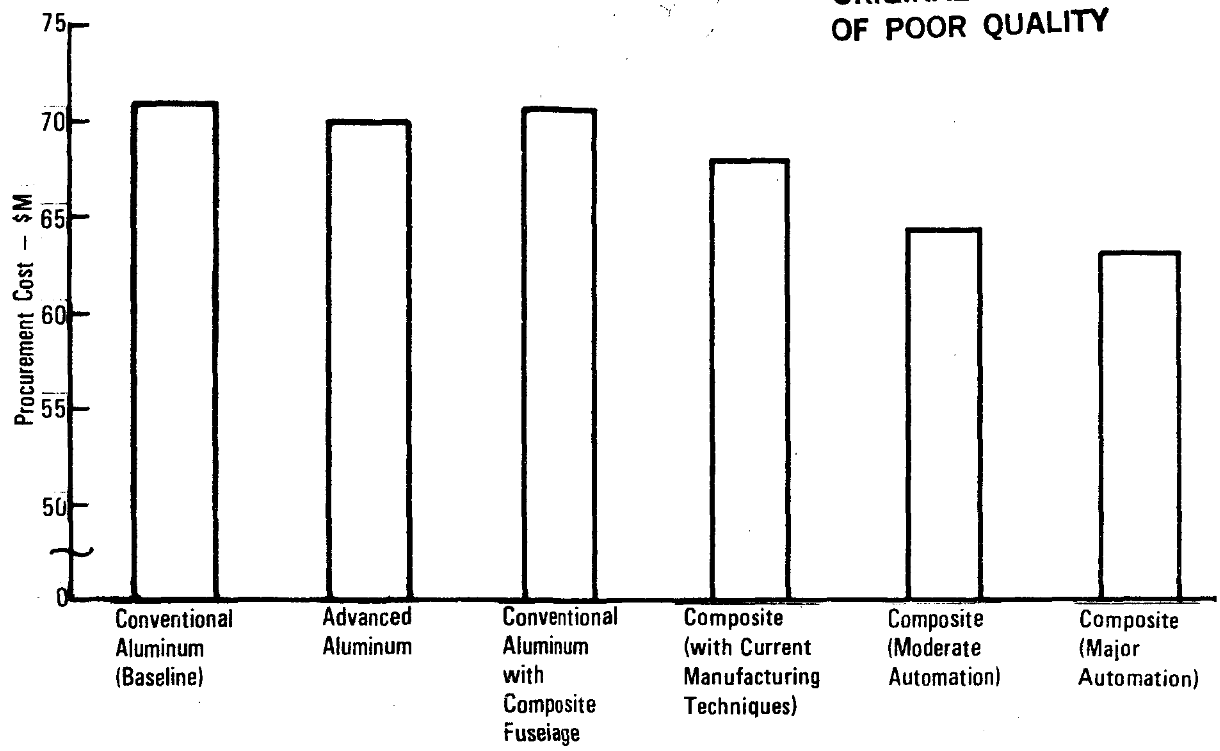


Figure 15. - Procurement costs for various commercial aircraft configurations.

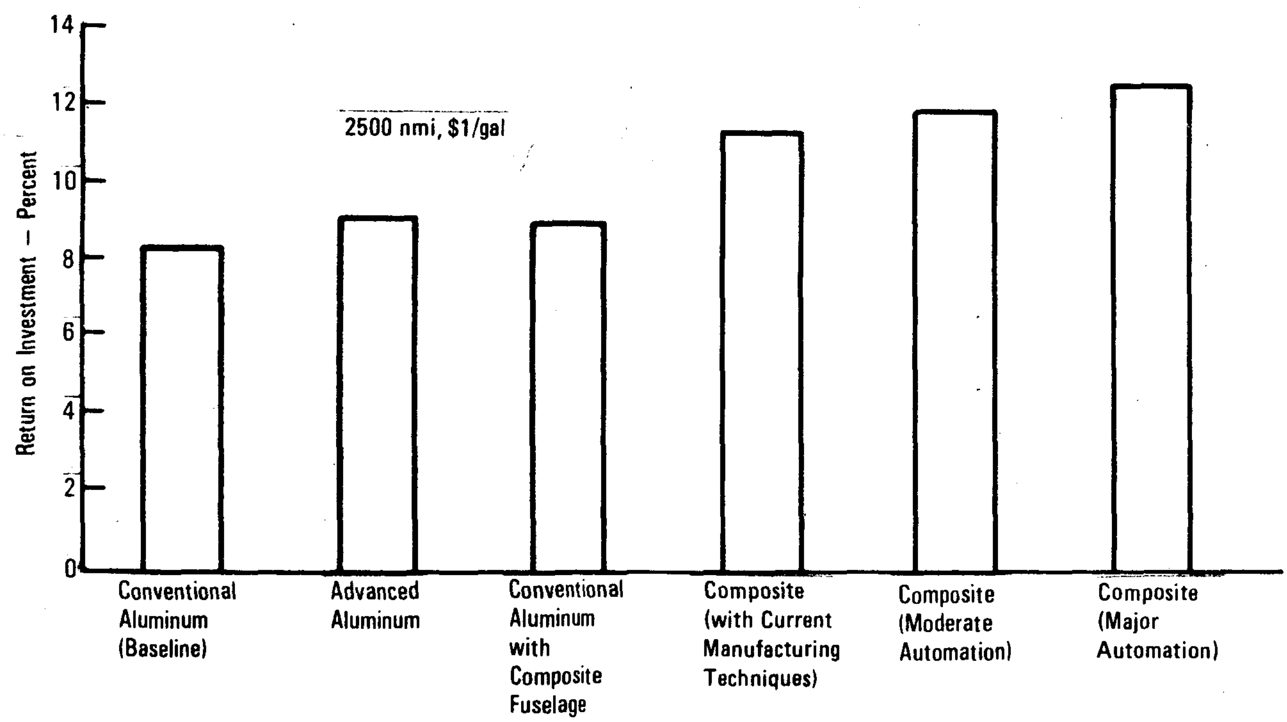


Figure 16. - Return on investment for various commercial aircraft configurations.

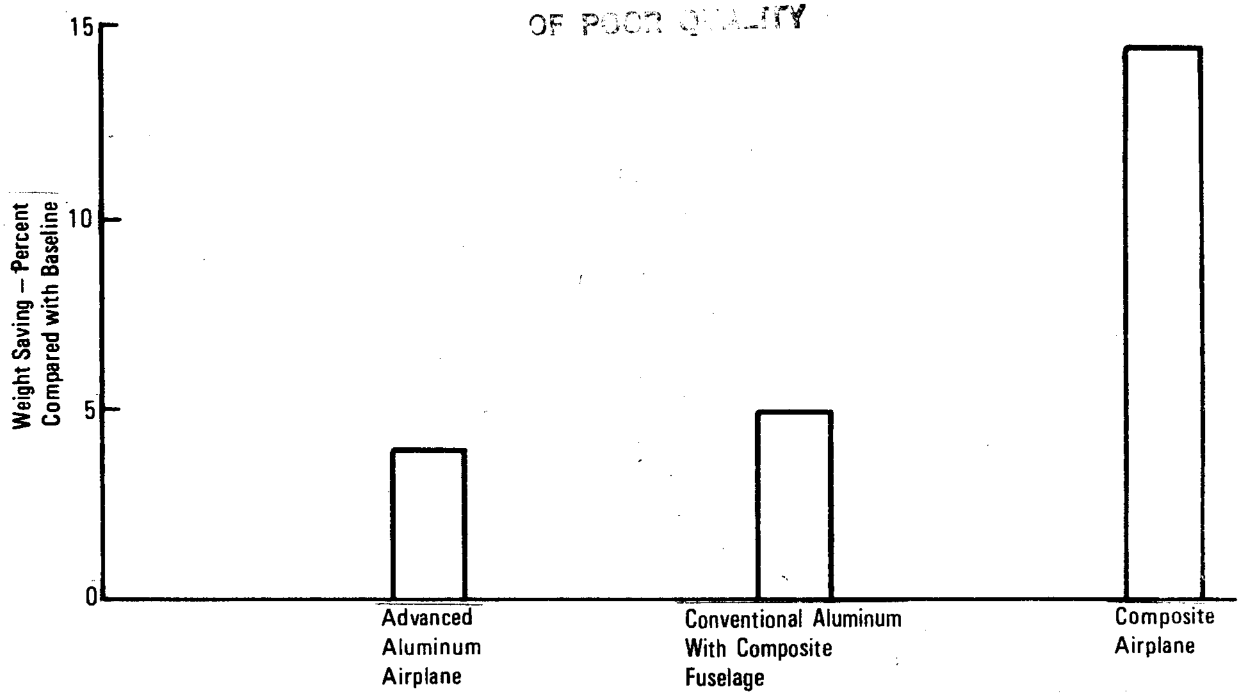


Figure 17. - Aircraft structural weight savings for various commercial aircraft configurations.

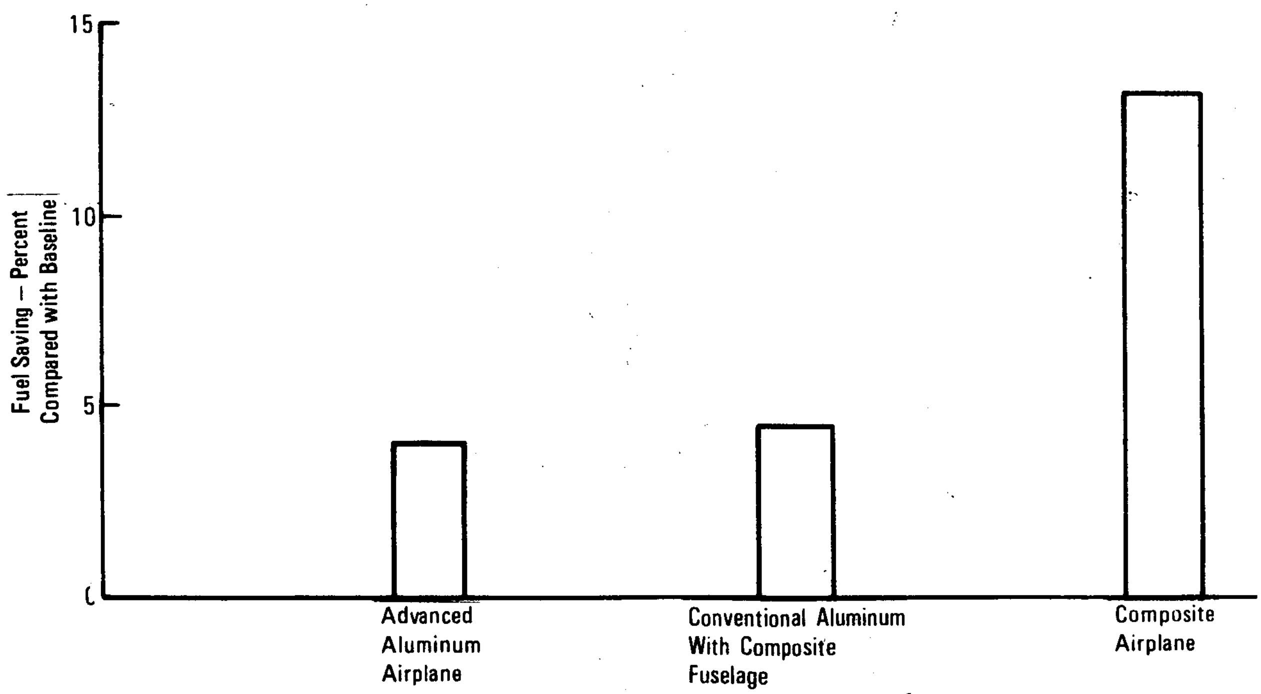


Figure 18. - Fuel weight savings for various commercial aircraft configurations.

TABLE 23. - COMMERCIAL FUEL AND COST SAVINGS

	Advanced Aluminum	Composite Fuselage	Composite Airplane (1) A	B	C
Fuselage Structure					
Weight Saving %	5.8	16.4	17.6	17.6	17.6
Cost Saving %	-1.3	-13.5	-13.3	-0.8	6.2
Airplane Structure					
Weight Saving %	7.7	9.5	26.4	26.4	26.4
Cost Saving %	-0.4	-2.0	2.7	11.6	16.2
Total Airplane					
Fuel Weight Saving %	3.9	4.5	13.2	13.2	13.2
DOC Saving %	2.2	2.2	8.1	9.2	9.7
IOC Saving %	0.6	0.7	2.0	2.0	2.0
TOC Saving %	1.4	1.5	5.1	5.6	5.9
ROI Improvement %	9.8	8.6	35.9	43.9	48.3
Procurement Cost %	1.2	0.1	5.1	9.1	11.2

(1) A Manufactured using current techniques (minimum automation)
B Moderate automation
C Major automation

The sensitivity of DOC and IOC to fuel prices is shown in Table 24 and Figures 19 and 20. IOC is affected because fuel price for ground transportation influences the G&A expense.

1.3.3 Benefits study summary.- The studies show distinct economic benefits for transport airplanes with advanced composite fuselages compared with both conventional and advanced aluminums. The military transport with a composite fuselage and conventional aluminum wing and tail structure has a lower acquisition cost than either the conventional or the advanced aluminum configurations and the life cycle costs are less as well. These benefits are even greater for the all-composite airplane, which shows over 10 percent reduction in life cycle costs compared to the conventional aluminum baseline. These figures were derived assuming minimum use of automation so higher benefits can be anticipated as automation is introduced.

Much the same picture exists for the commercial transport. In this case the composite fuselage with the conventional aluminum wing and tail structure resulted in only a small reduction in procurement cost compared to the conventional aluminum baseline and was slightly more expensive than the advanced aluminum airplane when minimum benefit from automation was assumed. The commercial transport contains many more cut-outs than the military aircraft - windows, passenger doors and below deck cargo doors in particular. This leads to a more complex shell structure and increased labor costs. The all-composite

TABLE 24. - SENSITIVITY TO FUEL PRICES

Configuration	Direct Operating Cost c/Seat Mile			Indirect Operating Cost c/Seat Mile			Overall Operating Cost c/Seat Mile		
	$1/Gallon	$2/Gallon	$3/Gallon	$1/Gallon	$2/Gallon	$3/Gallon	$1/Gallon	$2/Gallon	$3/Gallon
Baseline	2.957	4.125	5.286	2.866	2.909	2.953	5.823	7.034	8.239
Advanced Aluminum	2.893	4.014	5.129	2.849	2.891	2.933	5.742	6.905	8.062
Conventional Aluminum with Composite Fuselage	2.893	4.003	5.106	2.845	2.886	2.928	5.738	6.889	8.034
Composite (with Current Manufacturing Techniques)	2.717	3.723	4.722	2.809	2.846	2.884	5.526	6.569	7.606
Composite (Moderate Automation)	2.686	3.691	4.691	2.809	2.846	2.883	5.495	6.537	7.574
Composite (Major Automation)	2.669	3.675	4.674	2.808	2.846	2.883	5.477	6.521	7.557

configuration resulted in a lower procurement cost than either of the all-aluminum configurations and with increasing automation, the procurement cost saving more than doubles. Operating costs are less for the composite fuselage/conventional aluminum wing and tail airplane when compared with both all-aluminum configurations. The all-composite airplane shows a significant improvement in ROI, with a range of improvement from 36 to 48 percent depending on the degree of automation used in fabrication.

1.4 Technology Assessment Summary

The studies show that the benefits of converting from aluminum to advanced composites exist across the board. The composite fuselage will provide lower acquisition costs and lower operating costs. Combined with composite wings and empennage across the board benefits will result for both commercial and military airplanes. However, the technologies which exist today are not developed sufficiently to commit to any large scale use of composites in the fuselage structure of large transport aircraft. A concerted effort to develop and prove those technologies must be implemented. When these technologies are developed, significant savings in acquisition costs, fuel consumption and overall operating and life cycle cost will be achieved. Discussion of the Plan Development and Schedule and Resource Requirements to fulfill these aims follows in the subsequent sections.

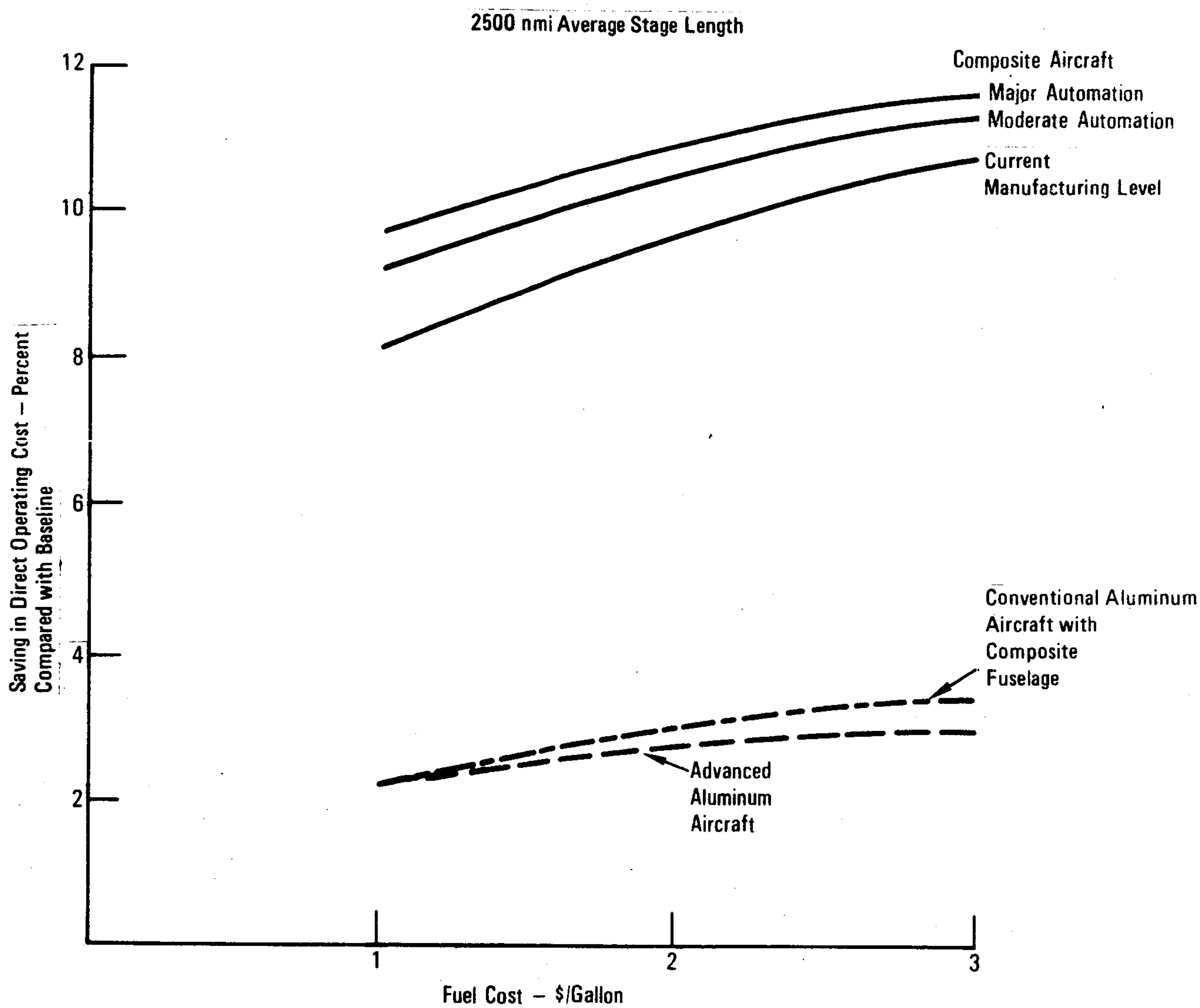


Figure 19. - Effect of fuel prices on direct operating costs for various commercial aircraft configurations.

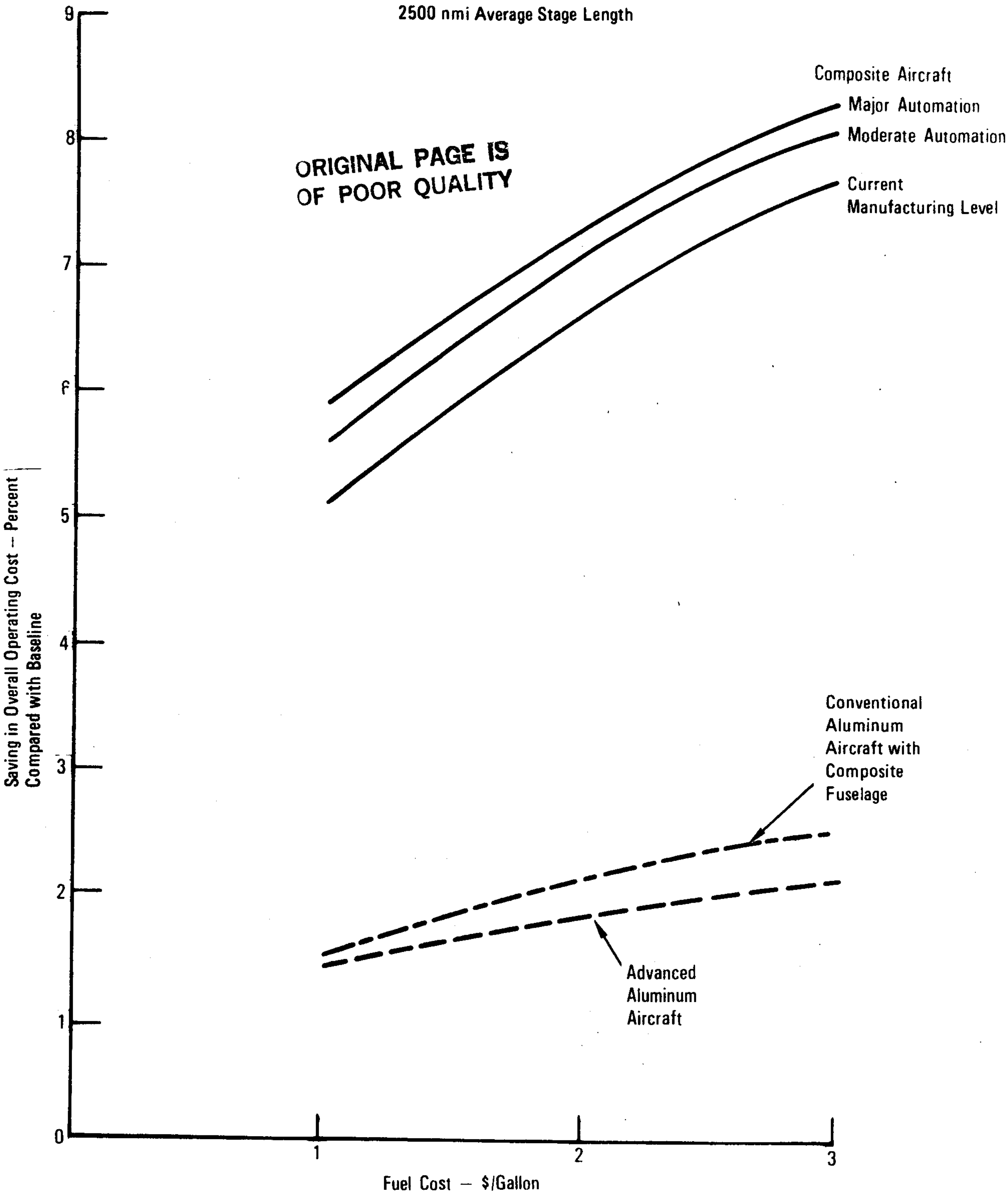


Figure 20. - Effect of fuel prices on overall operating costs for various commercial aircraft configurations.

2. PLAN DEVELOPMENT

This phase of the program consisted of four tasks: Preliminary Design, Concept Evaluation, Manufacturing Development, and Design Verification.

The objective of these tasks was to develop the basis for defining the plan to support the extensive use of composites in fuselage structure of commercial and military transport aircraft in the 1990 time period.

In order to develop a plan, certain ground work had to be performed. A preliminary design of both a commercial and a military baseline aircraft was necessary to identify all the design criteria and requirements. An evaluation of possible design concepts was performed in order to establish concepts that are producible as well as structurally efficient, and to identify concepts which with manufacturing development could provide weight and/or costs advantages. To support these efforts a manufacturing development study was performed and a verification test plan was developed.

2.1 Preliminary Design

2.1.1 Baseline airplanes.- Two baseline airplanes were selected for this study. The commercial baseline airplane was the ATX-350I which is shown in Figure 12. This airplane is an advanced technology configuration based on the L-1011. This baseline has been used in previous study programs, in particular, contract NAS1-16273, "Integrated Technology Wing Design Study" (Reference. 4). The basic ASSET input data files were thus available. The ATX-350I has a large diameter fuselage (236 inches) and, being a passenger-carrying airplane, it has numerous cut-outs for windows and doors.

The military baseline airplane was the "Advanced Civil/Military Aircraft," which is shown in Figure 4. This airplane has been used in various studies by the Lockheed-Georgia Company, including reference 2. It has a large diameter fuselage (upper lobe 388 inches in diameter and lower lobe 522 inches in diameter), nose and tail loading doors, and military payloads.

Based on the technology issues defined, the constant diameter barrel section of the ATX-350I just aft of the wing and main gear wheel wells was selected for study. Figure 21 shows a schematic of the barrel section. It is similar to the barrel section forward of the wing but is a little more highly loaded.

For the military baseline the barrel section forward of the wing was selected. On a military cargo airplane the aft fuselage is heavily influenced by configuration. The load levels for the forward fuselage are known with reasonable confidence since the critical design conditions are the steady flight maneuvers or the braked taxi cases. The aft fuselage loads will vary with the degree of active controls which will tend to drop these loads to levels comparable with the forward fuselage. The forward fuselage is shown in Figure 22.

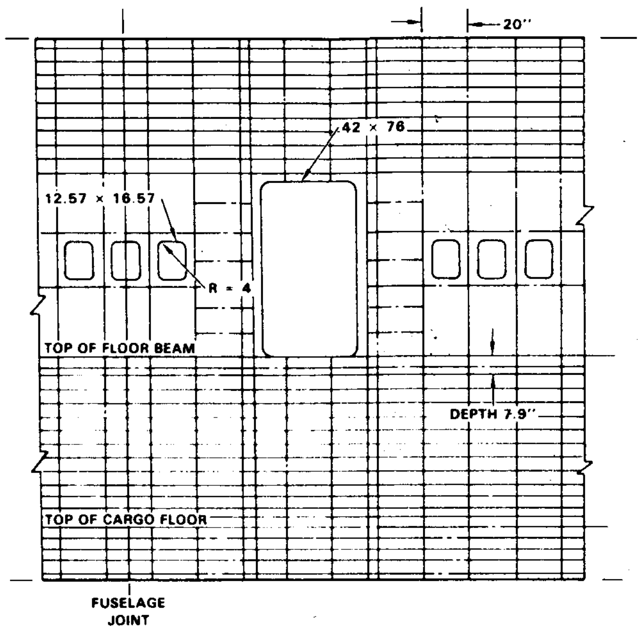


Figure 21. - Commercial baseline study barrel section.

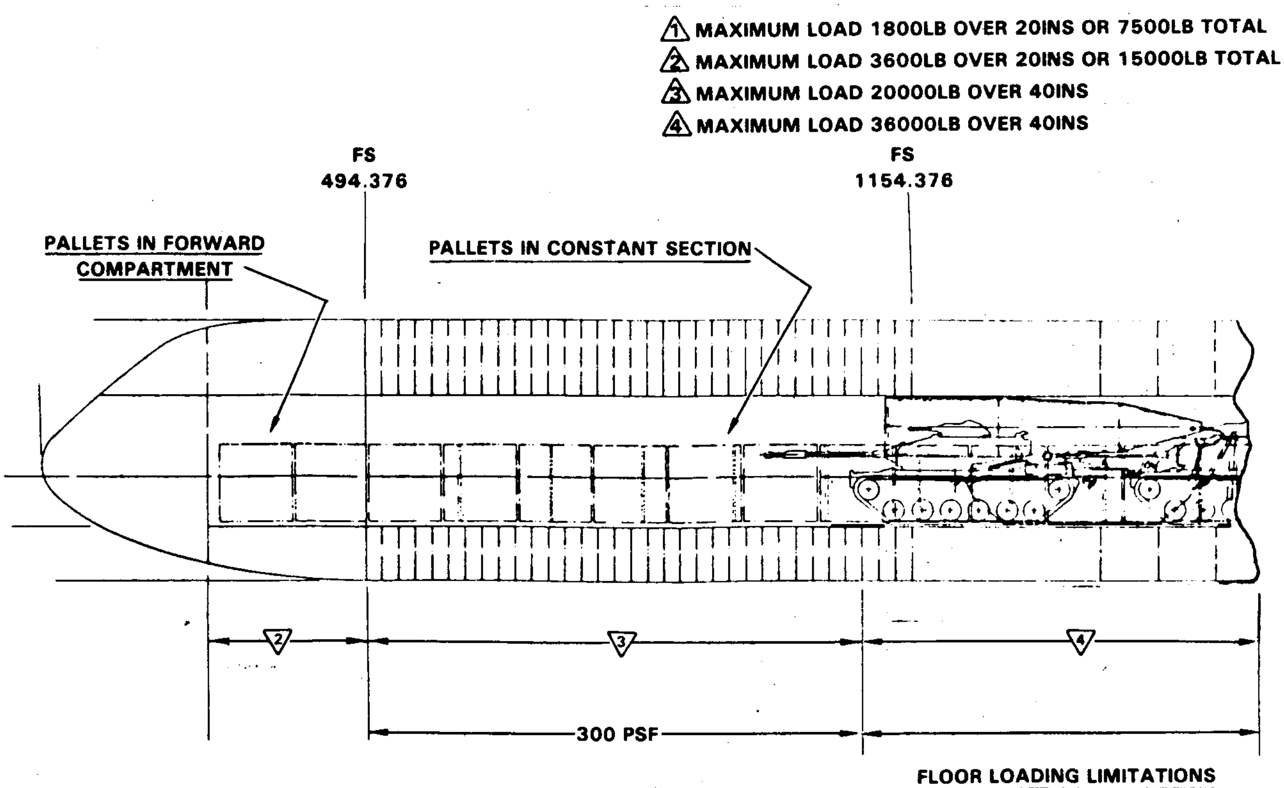


Figure 22. - Military baseline study barrel section.

2.1.2 Criteria and allowables. - The general performance criteria for both baselines are defined in Section 1.3, Benefit Studies.

The structural design criteria used in this study for the commercial baseline were based on the L-1011 criteria. The many structural considerations must adhere to the requirements defined in the Federal Aviation Regulation, Part 25 (reference 1) and pertinent advisory circular (reference 5). Essentially the only changes were those which related specifically to the materials. These criteria were originally documented under the ACSDT contract NAS1-15949, Task Assignment No. 1, reference 6, Some of the more pertinent criteria are summarized below:

- For one-g static, three-wheel ground and one-g level flight conditions, the structure shall be designed for $f/f_{cr} \leq 1.15$. Pressure structure shall not buckle under one-g level flight loads in combination with normal pressure loads. Tension-field webs shall also be designed for $q_{ult}/q_{cr} \leq 5.0$ at ultimate flight and ground load conditions.

- Minimum structure temperature -65°F. Maximum structure temperature of 223°F can occur on the upper crown of the fuselage while the aircraft is sitting on the ground. However, rapid cooling occurs during taxi and take-off roll. The maximum temperature for flight loads will be < 200°F and will be assumed to be 180°F in combination with maximum flight loads.

- Strength requirements for metal airframes are governed by fatigue and fail-safe requirements. Composite materials are generally not fatigue limited but are governed by stress concentrations due to holes and impact damage.

The preliminary design loads are shown in Figure 23.

For the military transport configuration the general criteria must conform with the MIL-A-series specifications for cargo transport aircraft. One difference for the military design is the internal pressure requirements.

The cabin pressure differentials are:

p = 8.2 psi (8000 ft cabin altitude at 40,000 ft pressure altitude). Relief valve settings associated with flight cases, -1.0 psi and +9.0 psi, and with landing cases, -1.0 psi to +1.5 psi.

A set of typical design load conditions was prepared at three locations on the forward fuselage, representing:

- ground burst pressure: two times the maximum internal pressure, acting alone

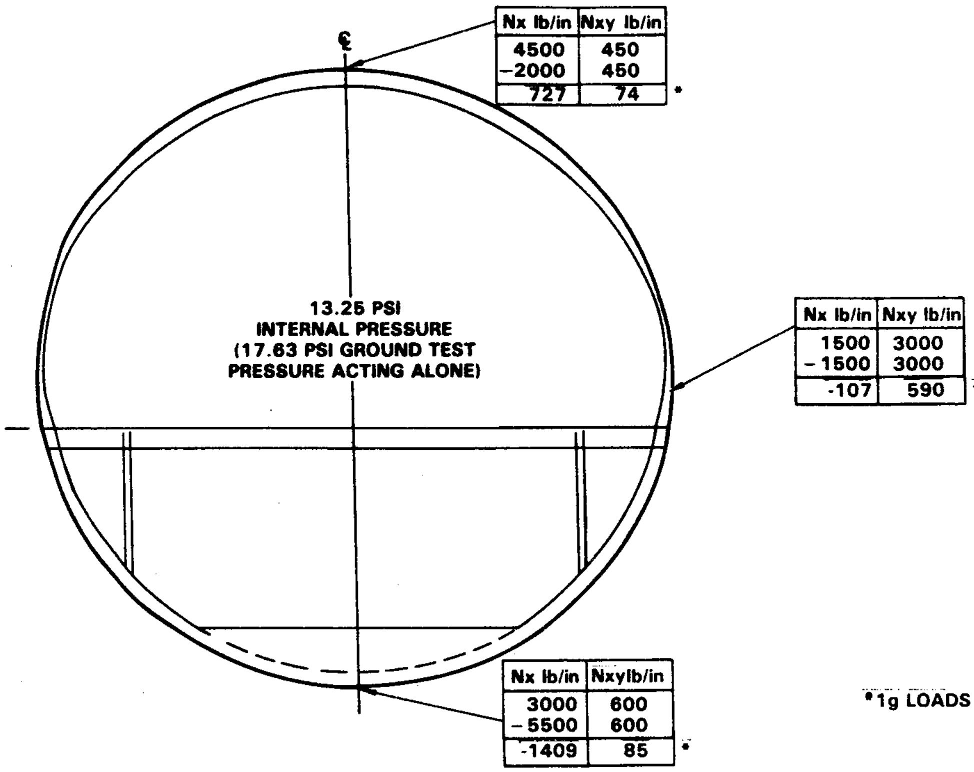


Figure 23. - ATX-350I preliminary design loads.

- maximum compression load for stability: 1.5 times the combination of maximum bending load with minimum internal pressure

- maximum tensile repeated load, for durability: 1.5 times the combination of maximum bending load with maximum internal pressure.

Values at the top, side and bottom of the fuselage are shown in Figure 24.

The candidate materials for the program were those considered in the Wing Key Technology Program, NAS1-16856, "Fuel Containment and Damage Tolerance in Large Composite Primary Wing Structures." These materials were:

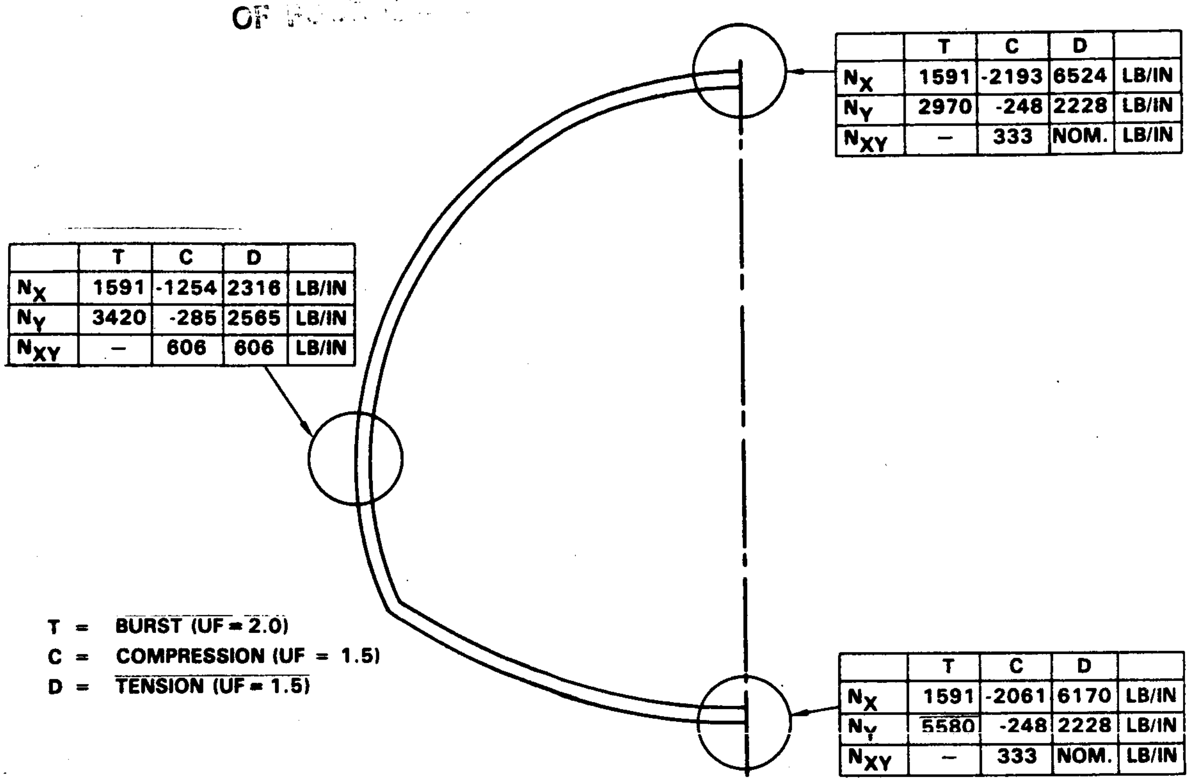
Hercules AS4/3502

Hercules AS4/2220-1

Narmco Celion/5245

Hexcel Celion/1504

American Cyanamid Celion/982



	T	C	D	
N_X	1591	-2193	6524	LB/IN
N_Y	2970	-248	2228	LB/IN
N_{XY}	–	333	NOM.	LB/IN

	T	C	D	
N_X	1591	-1254	2316	LB/IN
N_Y	3420	-285	2565	LB/IN
N_{XY}	–	606	606	LB/IN

	T	C	D	
N_X	1591	-2061	6170	LB/IN
N_Y	5580	-248	2228	LB/IN
N_{XY}	–	333	NOM.	LB/IN

Figure 24. - Preliminary design loads at FS 1154 (military baseline.)

The material selected was Hercules AS4/2220-1. This selection was based on the data available in July 1983 with respect to processing and mechanical properties. The allowables are comparable to T300/5208 except that the allowable tension strain has been raised from 4750 μ in/in to 6000 μ in/in. The T300/5208 allowables were developed for the Advanced Composite Fin and Aileron programs and are FAA approved (reference 7).

2.2 Concept Evaluation

A matrix of composite skin/stiffener and frame configurations was assembled from inputs provided by the various Engineering and Manufacturing disciplines. The configurations for skin/stiffener are shown in Figure 25 and the frame configurations are shown in Figure 26.

These concepts were then evaluated by Stress, Producibility, Materials and Processes, Manufacturing, and Quality Assurance personnel. The parameters considered in the evaluation are listed below:

Structure efficiency

Joints

Frame interface

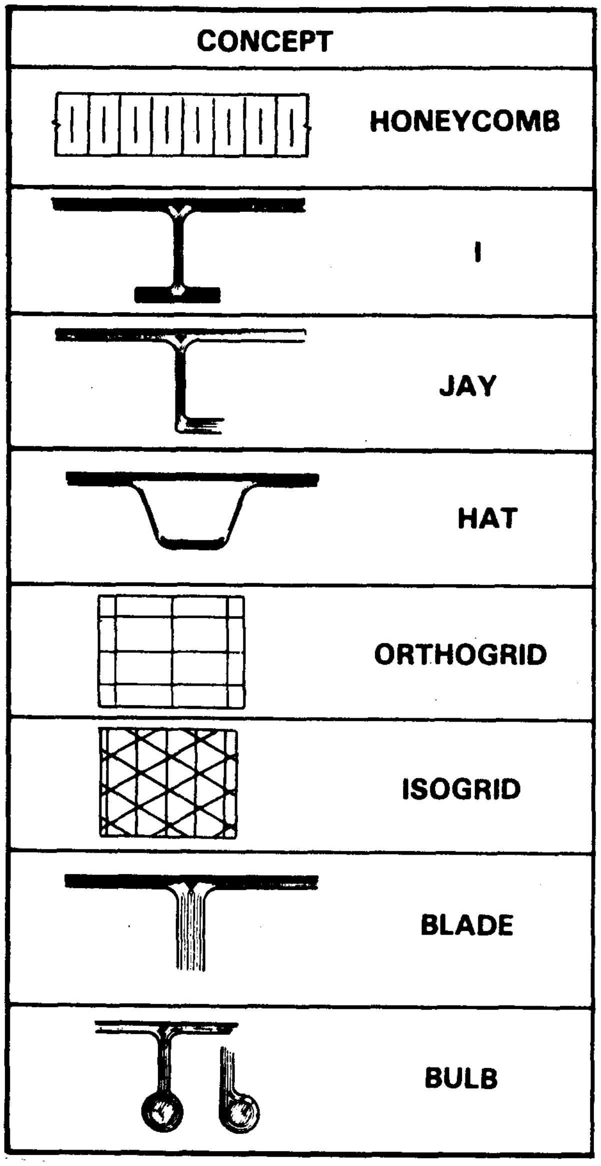


Figure 25. - Candidate skin/stiffener configurations.

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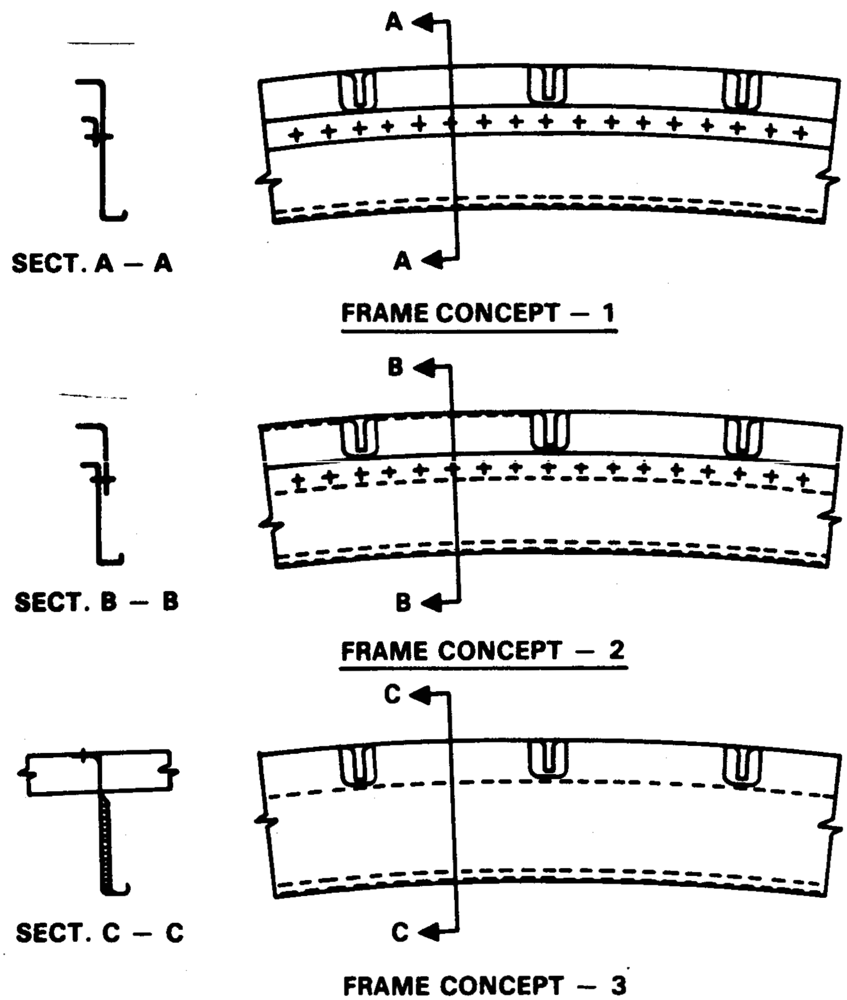


Figure 26. - Candidate concepts for fuselage frames (Sheet 1 of 2).

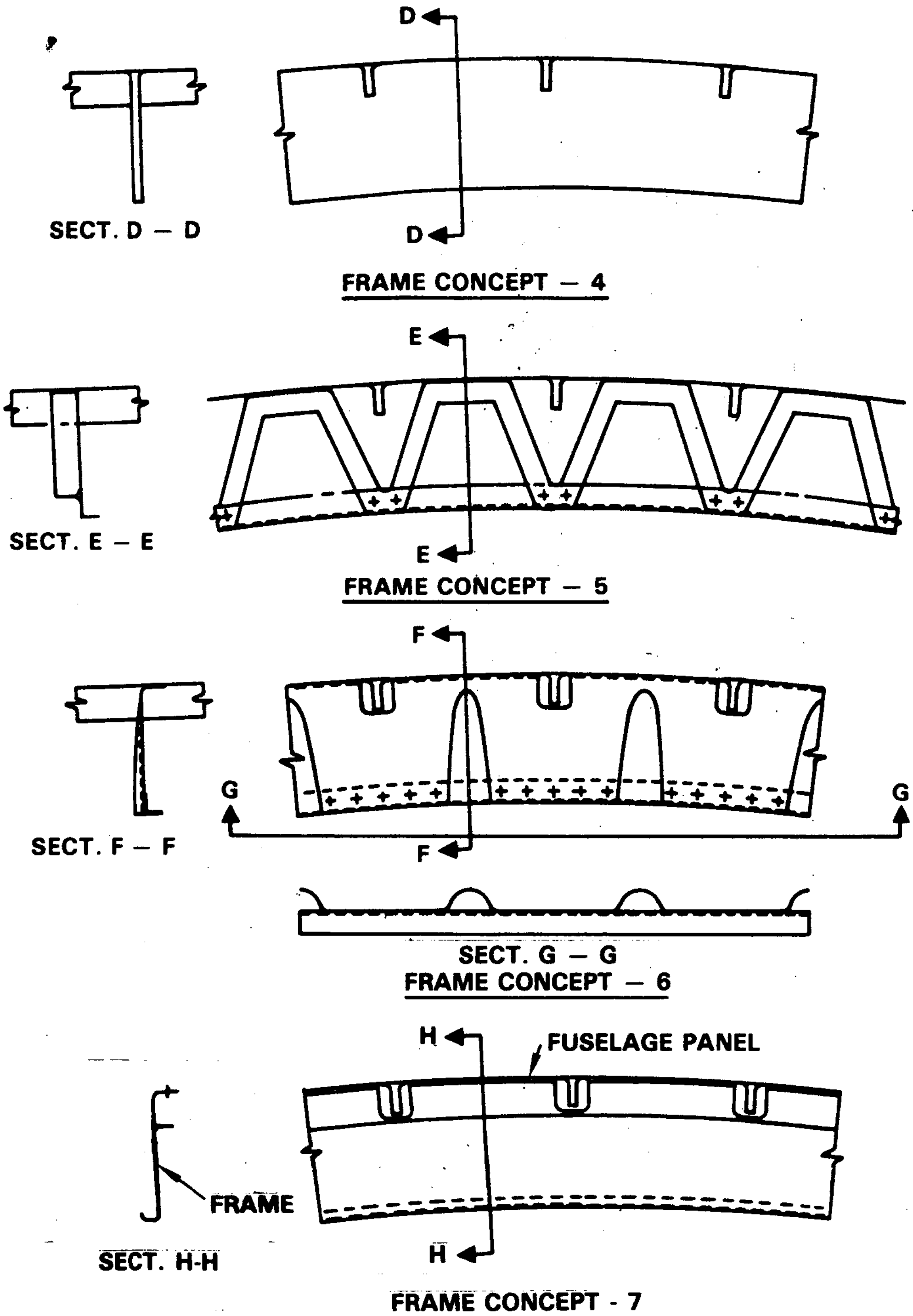


Figure 26. - Candidate concepts for fuselage frames (Sheet 2 of 2).

Repairability

Manufacturing suitability

Tool complexity

Manufacturing readiness

Equipment investment

Accessibility

Supportability

Proven reliability

Costs - recurring

- nonrecurring

Automation potential

Manual NDI

Automated scanning

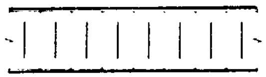
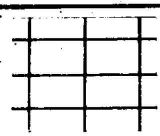
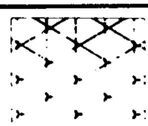
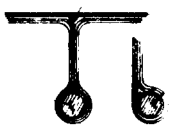
Acoustic transmission is not considered an influence in concept selection and impact dynamics is primarily a lower fuselage problem so neither was considered in these initial ratings. Also, weight was considered to be a minor influence and was not considered until later in the concept evaluation.

2.2.1 Skin/stiffener concept evaluation. - Of the skin/stiffener concepts, two were eliminated immediately. These were the honeycomb and the isogrid. The honeycomb was eliminated because of supportability problems and the general dislike of honeycomb by the airlines. Honeycomb cores are prone to moisture entrapment and require a considerable amount of in-service inspection and repair. The need for solid inserts at frames and around doors and cutouts also complicates the design and fabrication. The isogrid was felt to have more application to military fuselages than commercial because of its possible tolerance to battle damage. The window belt design would be complex with isogrid, and isogrid offers no real advantages for commercial transports when compared to orthogrid.

The remaining six concepts were then evaluated and rated. The results are summarized in Table 25.

The I-section was then eliminated as it is more difficult to fabricate than the J-section and it is much more difficult to install shear clips between the I-stiffener and the frame. Because of the narrow free flanges, splicing is more complex. The splicing problem also led to the elimination of the

TABLE 25. - CANDIDATE FUSELAGE SKIN/STRINGER CONCEPTS EVALUATION

CONCEPT		STRESS	PROD	M&P	QA	MANUF	Σ
	HONEYCOMB	ELIMINATED PRIOR TO EVALUATION					
	I	3	6	6	3	5	6
	JAY	1	5	2	2	4	2
	HAT	6	4	3	5	3	4
	ORTHOGRID	4	3	4	6	2	3
	ISOGRID	ELIMINATED PRIOR TO EVALUATION					
	BLADE	2	1	1	1	1	1
	BULB	5	2	5	4	6	5

* 1 IS BEST

bulb section. The hat section was eliminated because of supportability problems in the lower fuselage caused by entrapment of bilge fluids and because mechanical splicing would require the use of blind fasteners. This left three concepts: the J, the orthogrid and the blade. The final evaluation of these three concepts based on joining requirements is summarized in Table 26.

All skin/stiffener concepts except the honeycomb were analyzed to derive sizing based on the loads shown in Figure 23. The shear stiffness property (Gt) of the skin was designed to be no less than that of the aluminum baseline. The sized composite skin/stiffener designs for the upper fuselage segment are shown in Figure 27. The aluminum baseline structural configuration is shown in Figure 28. A comparison of the weights is presented in Table 27.

2.2.2 Frame Concept Evaluation.- Initially six frame concepts were formulated. These are shown in Figure 26. Concept #1 is a full depth Z-frame keyholed at the skin stiffeners. A separate angle is mechanically fastened and/or bonded to stabilize the frame inboard of the keyholes. Concept #2 is a floating Z-frame with a separate angle shear tie to the skin which is mechanically fastened and/or bonded to the Z-frame. Concepts 1 and 2 are essentially "metal replacement" designs. Concept #3 has a honeycomb stiffened web which apart from being the lightest design had little merit. Concept #4 is an orthogrid concept. Concept #5 is a molded truss concept which was considered to evaluate the possible use of molded chopped fiber components. Concept #6 is beaded to help provide better fiber alignment around the curvature. The inboard circumference of the frame is less than the outboard circumference. When material is laid up to follow the curvature the excess material toward the inboard edge is taken up periodically as a bead. This bead would provide some stiffening effect for the frames. For large diameter fuselages the differences between the inboard and outboard circumferences is small and will result in a shallow and inefficient bead.

The ranking of these six frame concepts is shown in Table 28. Concept #3 was eliminated because of the delamination potential in the core runout and general maintenance and repair problems. Concept #5 was eliminated because of high local bending moments in the trusswork. Also the frame depth is too shallow to permit a truss system to be designed without a weight penalty. Producibility indicated the concept was not cost-effective. Concept #6 was eliminated also. The free edge of the bead would be prone to damage and the web would be relatively thick because of stability requirements for the free-edge bead.

Concept #4 was retained because of the potential structural and manufacturing benefits of the overall orthogrid concept which has significant automation potential. Concepts #1 and #2 were also retained.

The most pertinent result of the evaluation was that innovative fabrication techniques must be developed for frames so that fiber alignment will always follow the curvature of the shell. A study then ensued which resulted in some possible alternative approaches to frame design and construction.

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TABLE 26. - STRINGER JOINTS EVALUATION

Concept	Stress	Manufacturing	Q.A.	Producibility	Overall
Blade	1	1	1	1	1
Jay	3	3	2	3	3
Orthogrid	2	2	3	2	2

TABLE 27. - WEIGHTS OF STRINGER CONCEPTS

Configuration	Pounds/Inch	Percent Increase Over Minimum
Blade	6.75	--
Orthogrid	6.82	+1.0
Hat	7.10	+5.2
Bulb	7.25	+7.4
J-Section	7.48	+10.8
I-Section	7.52	+11.4
Isogrid	8.48	+25.6

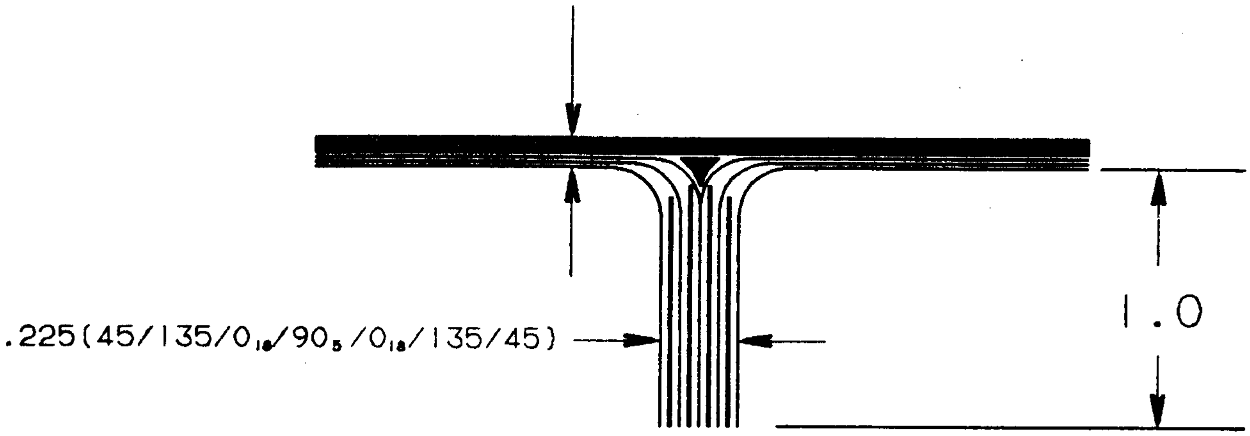
TABLE 28. - CANDIDATE FRAME CONCEPTS EVALUATON

Concept	Stress	Manufacturing	Producibility	Weights	Total	Rank
#1 Zee	2	2	2	3	9	1
#2 Zee	1	1	3	5	10	2
#3 Honeycomb	6	5	5	1	17	5
#4 Orthogrid	4	4	4	2	14	4
#5 Truss	5	6	6	4	21	6
#6 Beaded	3	3	1	6	13	3

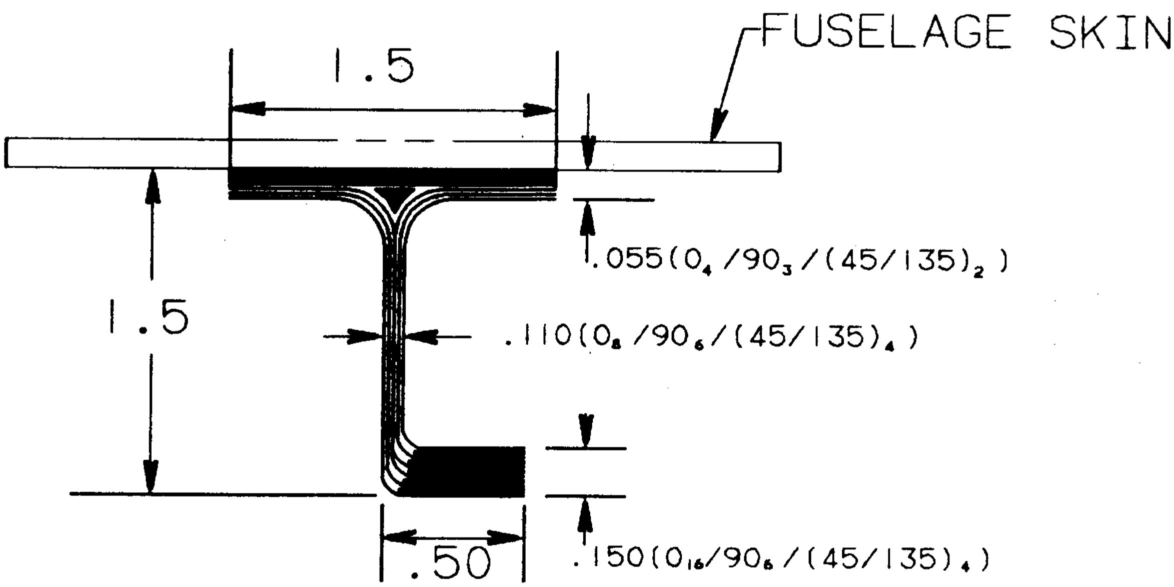
TABLE 29. - FRAME WEIGHTS

Concept	Optimum Weight lb/in/frame	Nonoptimum Factor	Component Wt lb/in/frame	Component Wt Saving %	Fastener Wt Saving %	Overall Wt* Saving %
#4 Orthogrid	0.064	1.14	0.073	4.6	40.0	10.2
#7 Filament wound	0.049	1.18	0.057	25.5	0.0	20.7

*Relative to the aluminum baseline

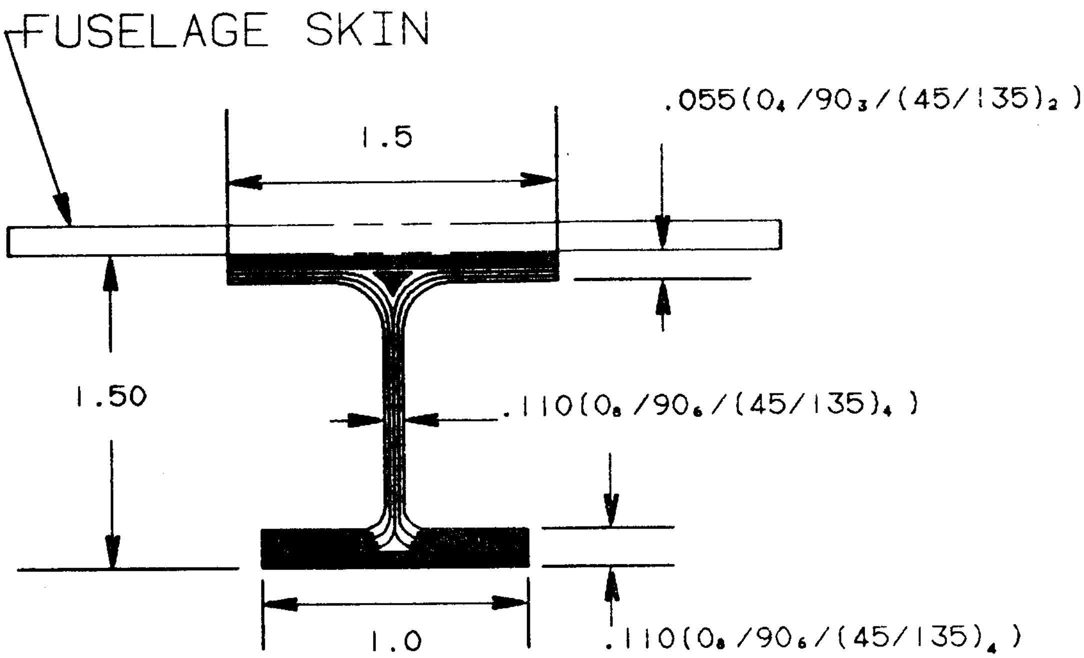


A. BLADE DESIGN

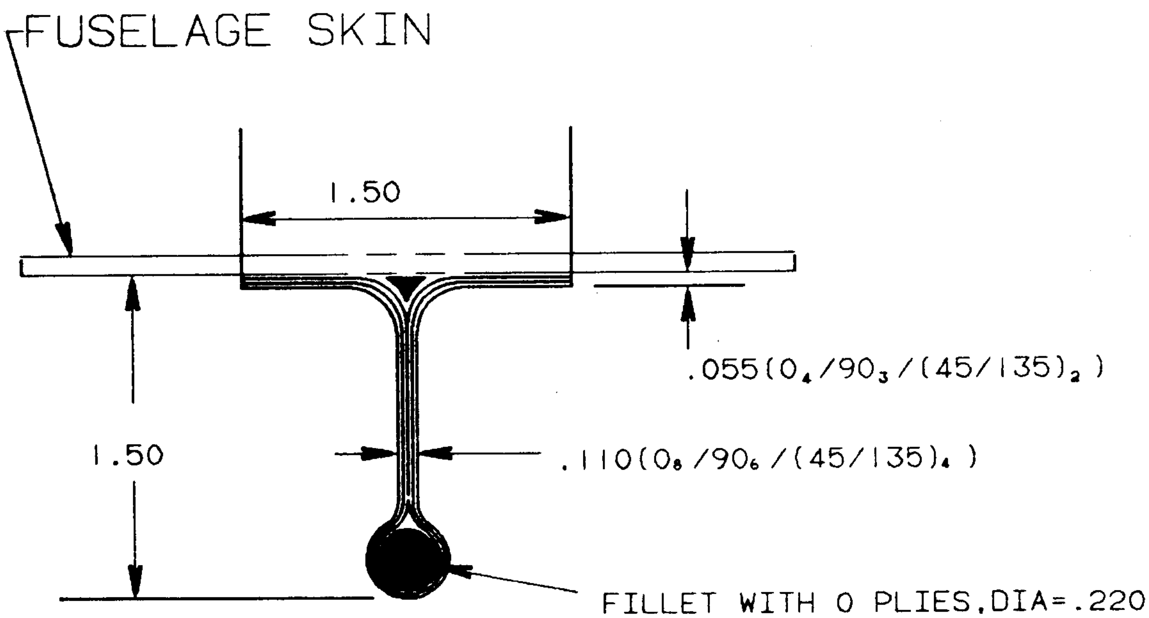


B. J SECTION DESIGN

Figure 27. - Sized skin/stiffener designs for upper fuselage (Sheet 1 of 5).



C. I SECTION DESIGN



D. BULB SECTION DESIGN

Figure 27. - Sized skin/stiffener designs for upper fuselage (Sheet 2 of 5).

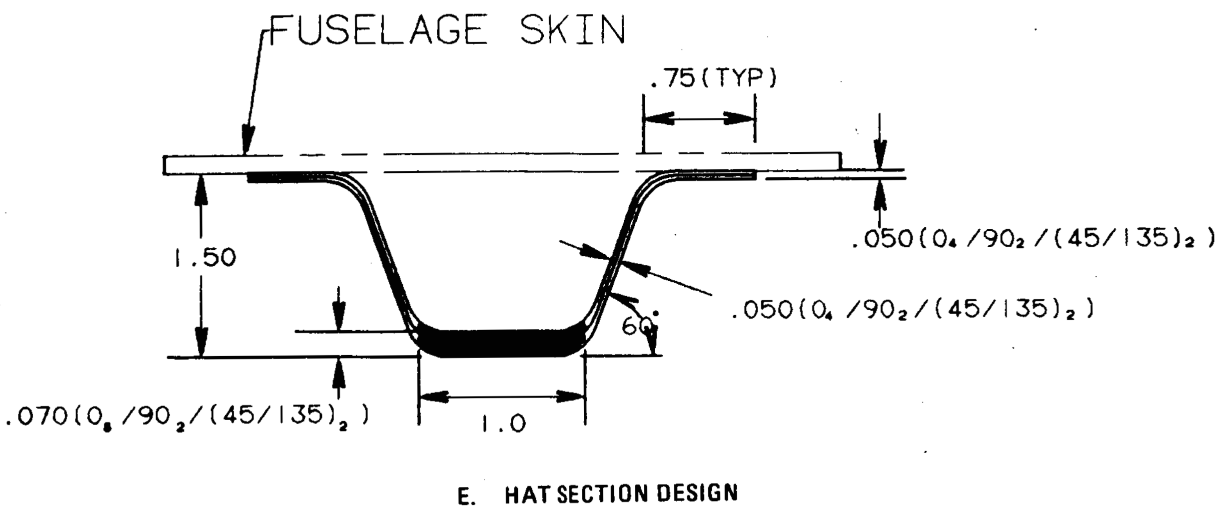


Figure 27. - Sized skin/stiffener designs for upper fuselage (Sheet 3 of 5).

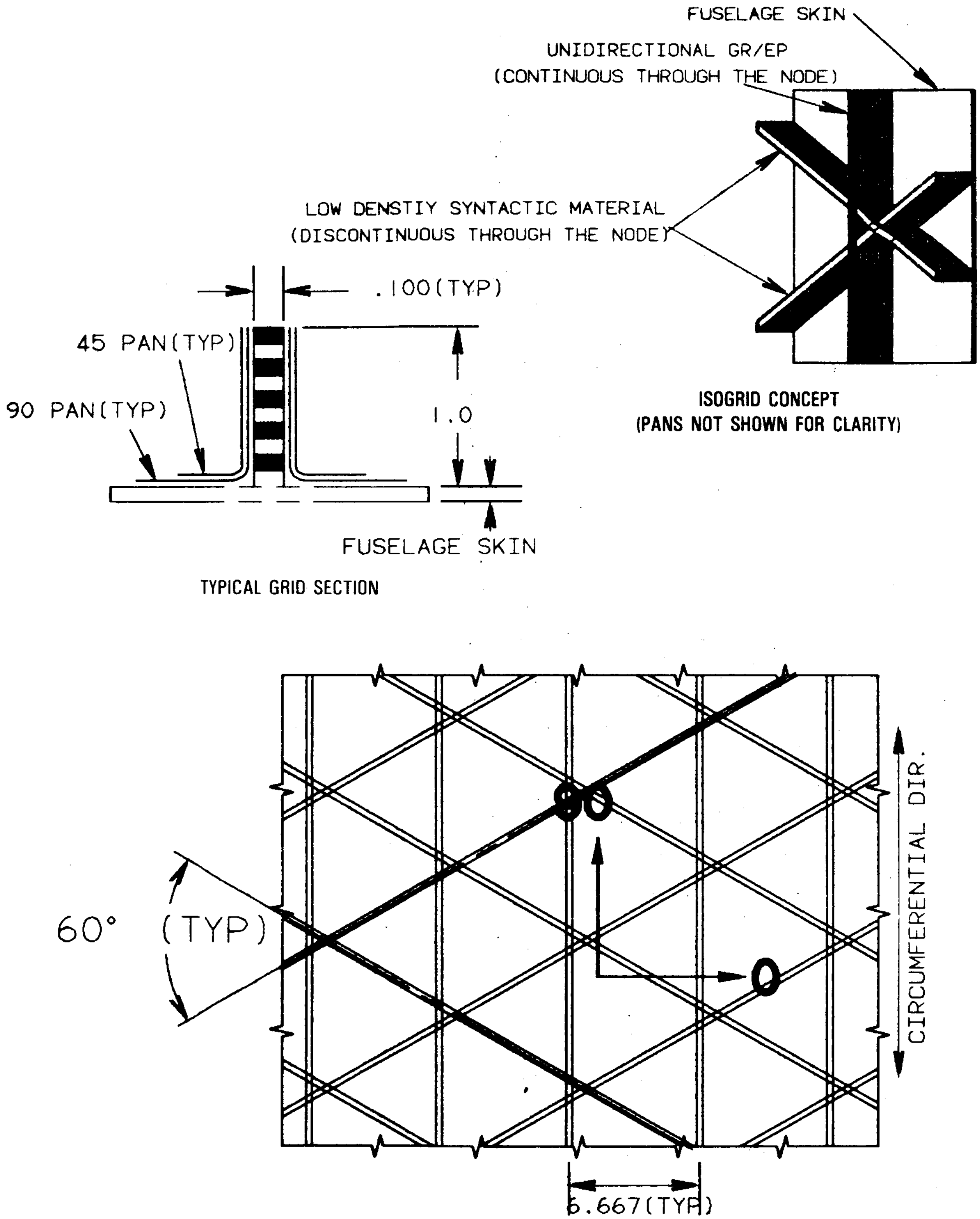


Figure 27. - Sized skin/stiffener designs for upper fuselage (Sheet 4 of 5).

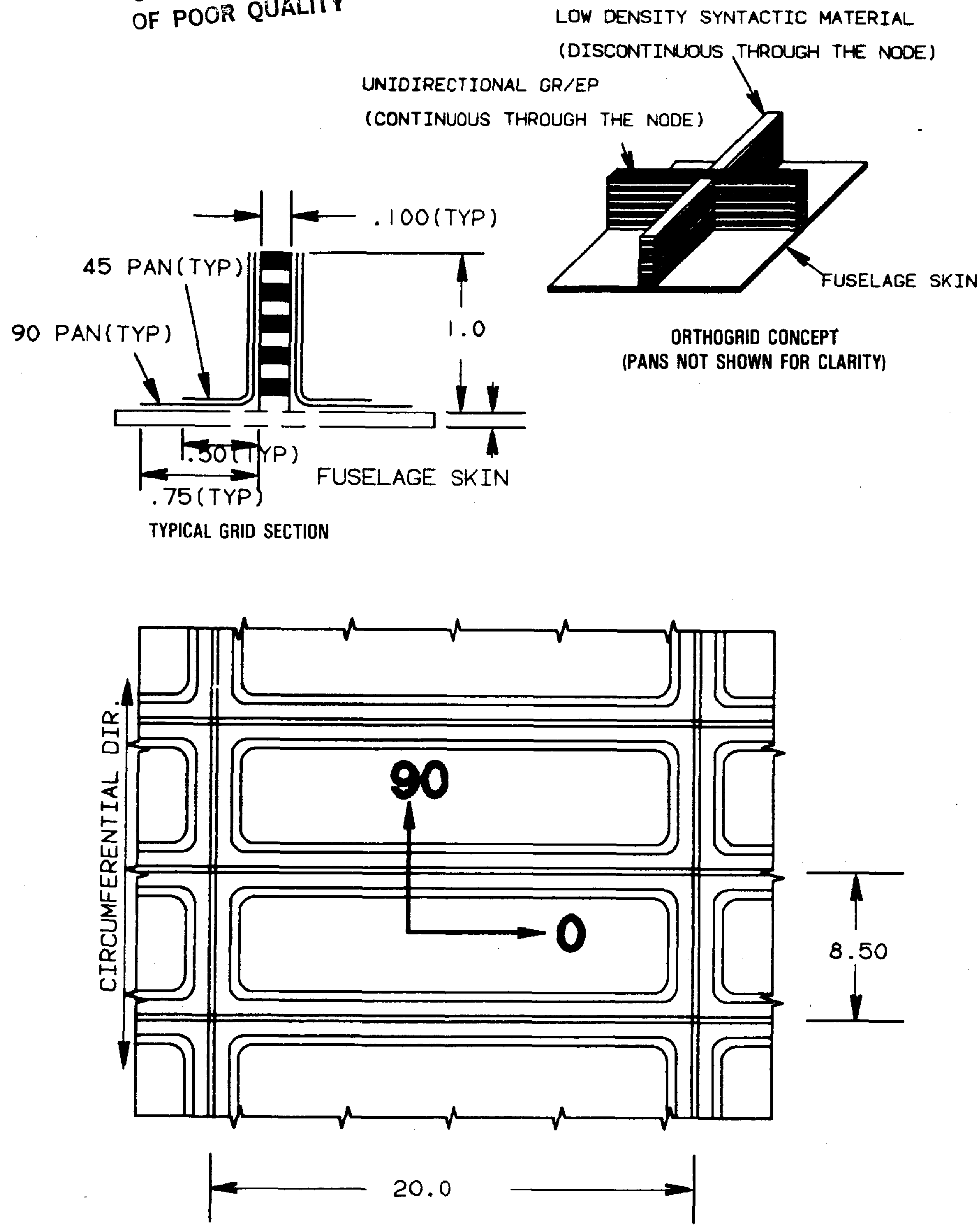


Figure 27. - Sized skin/stiffener designs for upper fuselage (Sheet 5 of 5).

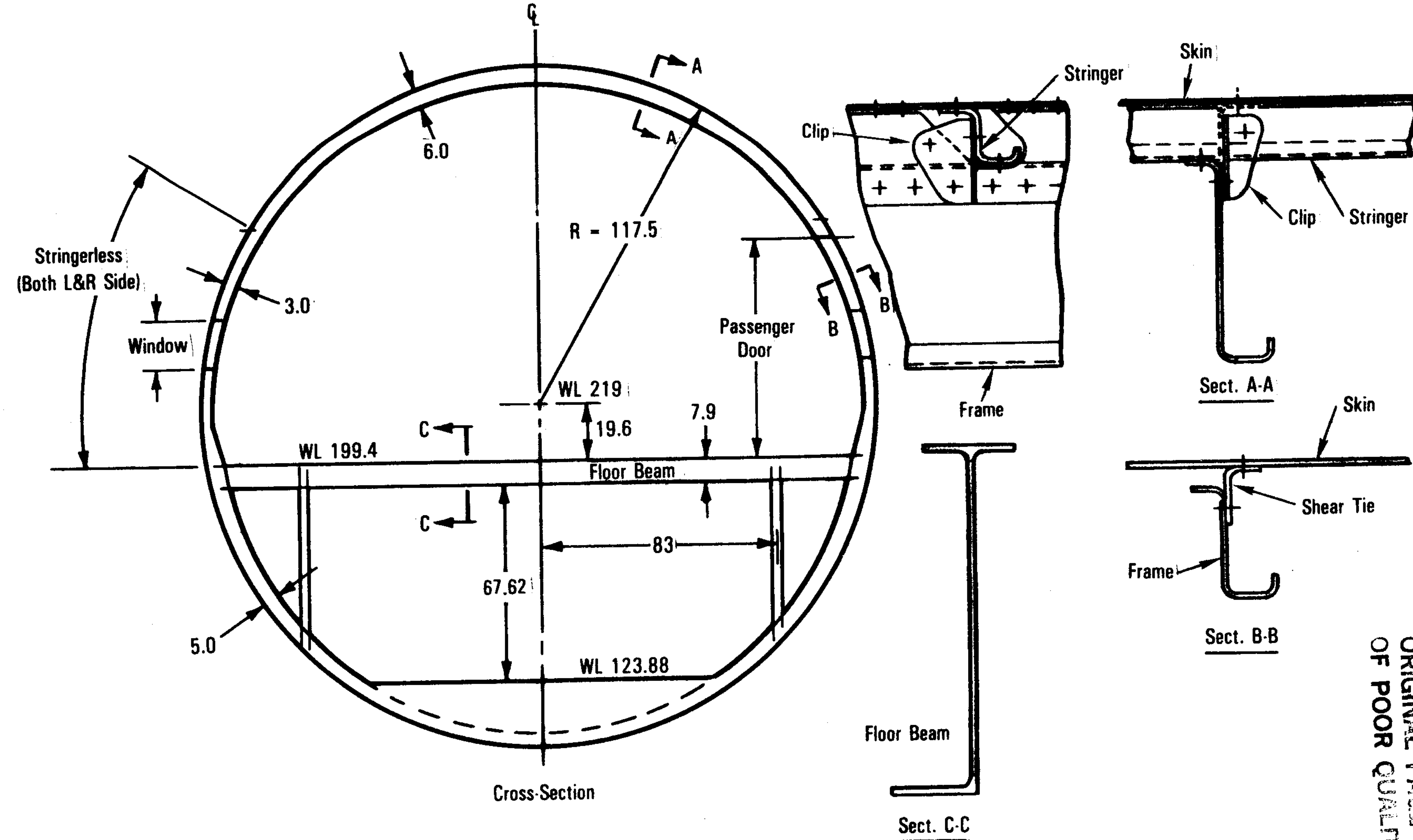


Figure 28. - Baseline aluminum fuselage construction.

A one-piece frame design was developed based on configurations #1 and #2. This configuration, concept #7 would be produced by a filament winding process. It is shown in Figure 29 along with the orthogrid concept. Both these concepts were sized and a weight analysis was performed. The results are shown in Table 29. Concept #7 shows potential for further development.

2.2.3 Overall evaluation.- A design study was made of the various concepts for stringers and frames in combination. Particular attention was paid to interfaces with floor beams and cutouts. The results looked much like metal structures. The conclusion is that these interfaces are essentially nothing more than detail design problems connected in large part with producibility concerns. Reinforcement around cutouts needs detailed study to determine the most efficient methods of interfacing skin reinforcement with stringer and frame details, particularly from the interlaminar stress point of view. The best overall frame, skin, stiffener configuration is the orthogrid. However this concept relies on certain proprietory materials and fabrication techniques which are still under development. Although the blade stiffener rated best it does not lend itself to the use of floating frames in the upper fuselage. Floating frames reduce costs in the upper lobe of the skin and have potential for reducing acoustic transmission. For this reason the Jay section stiffener is selected for the final concept along with the filament wound concept #7 frame and variations of that frame concept.

2.2.4 Military considerations.- These concepts are based on the military configurations which generally have few if any windows and doors in the shell structure other than the nose and tail loading doors and because of the low floor have no cargo space below. The fuselage shell concepts shown in Figure 30 were reviewed by Stress, Design and Manufacturing personnel experienced in military transport structure. The selection of these configurations for

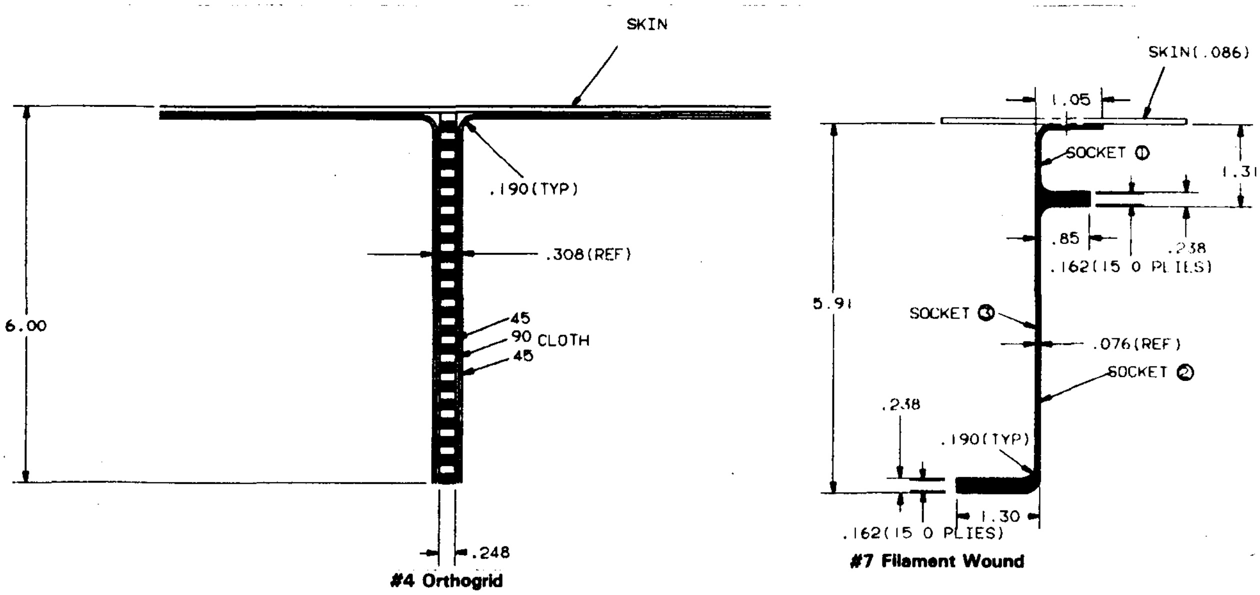


Figure 29. - Final sized frame concepts.

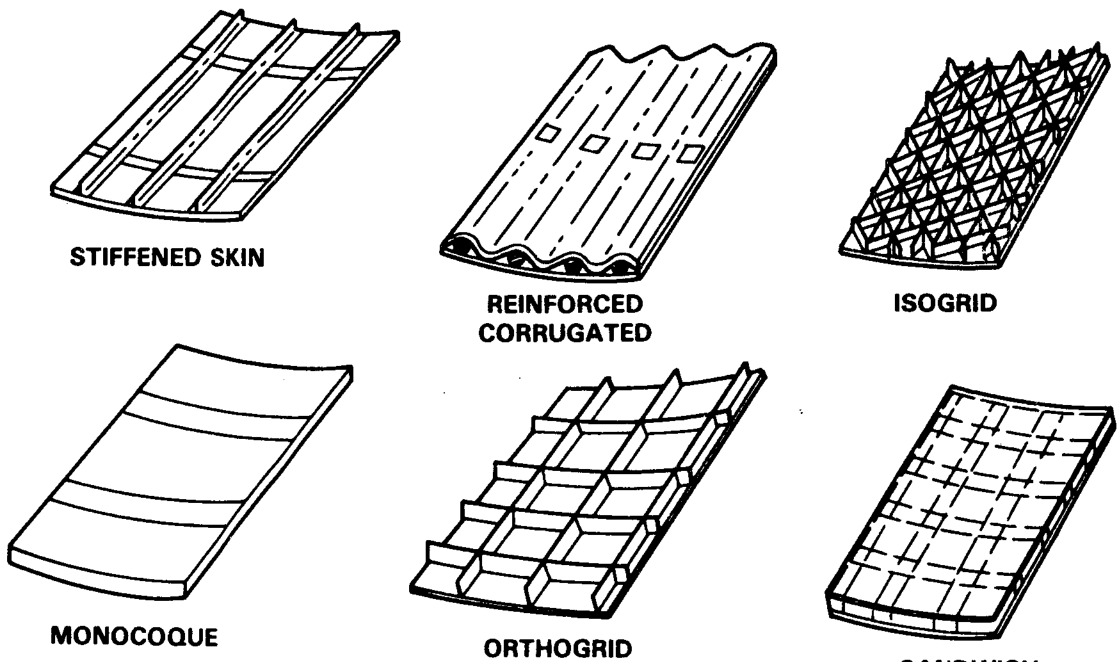


Figure 30. - Fuselage shell concepts for military transports.

consideration was based on the transport of heavy cargo such as tanks and on the need for battle damage tolerance. Four primary areas of concern were included in the reviews: logistics, pressure containment, manufacturing technology, and design technology. The resulting ratings are summarized in Table 30.

The evaluation of various stiffened skin concepts shown in Table 26 applies to the military design as well as the commercial design. The Jay stiffened concept was subjected to structural optimization for both an aluminum design and an advanced composite design. Analyses were performed at the top and side at F.S. 1154 (see Figure 21) by the front spar. A point analysis of the isogrid configuration was also performed at F.S. 1154, top using finite element modeling; the isogrid was included because of its potential tolerance to battle damage. The results of the analyses are summarized in Table 31.

Engineering preferred the stiffened skin approach mainly because the design technology is available while manufacturing preferred the orthogrid approach. The monocoque design was felt to have the best logistics rating. The isogrid, while rated last because of the lack of data, was felt to have the best potential for battle damage tolerance.

2.2.5 Impact of new materials.- As part of the concept evaluation, the impact of new materials was assessed. The primary improvements identified as most likely to occur in a time frame which could benefit a 1990 airframe were the new thermoplastic resins (PEEK type) and 2% strain fibers.

TABLE 30. - MILITARY FUSELAGE SHELL CONCEPTS EVALUATION

	Stress	Design	Manufacturing	Σ
Stiffened Skin	1	1	2	1
Reinforced Corrugated	6	4	4	5
Isogrid	5	6	6	6
Monocoque	2	2	3	2
Orthogrid	4	5	1	3
Sandwich	3	3	5	4

TABLE 31. - SKIN/STIFFENER COMPARISONS

Skin Stiffening	Al J	Gr/Ep J	Gr/Ep Isogrid	Units
Station				
1154, top				
Stringer Spacing	5.28	6.47	17.82	in.
Stringer Height	1.42	2.14	2.05	in.
Skin Thickness	0.095	0.120	0.180	in.
Weight	0.0146	0.0099	0.0119	lb/in^2
1154, side				
Stringer Spacing	4.0	5.26	–	in.
Stringer Height	1.0	1.0	–	in.
Skin Thickness	0.080	0.130	–	in.
Weight	0.0110	0.0096	–	lb/in^2

The thermoplastic resins are those which are processed by heating above the softening temperature, forming and consolidating, and then cooling below the softening temperature. No chemical change occurs, no bleeder or breather is needed, and no vacuum bags are required. Parts can be reformed and scrap can be salvaged and reused. Thermoplastics require no refrigerated storage and have a nearly infinite shelf life.

PEEK, a promising thermoplastic, requires forming at 720°F. While this is high for plastics, it is a very low temperature when compare to metals or ceramics. Thus current metal forming techniques may be adaptable to the processing of thermoplastics. The result would be reduced capital investment and a potential for reduced material costs. Thermoplastics also exhibit much improved toughness and damage tolerance compared to thermosetting resins. PEEK also has much improved resistance to solvents compared to other thermoplastics. The 2% strain fiber along with the improved toughness of the PEEK resin will translate into higher strain allowables. However, the need to maintain balanced laminates and requirements for at least some plies in the 0° and 90° directions tends to reduce the overall potential weight saving. The use of lamina less than 0.005 in. nominal thickness could be considered but indications are that higher material cost and lower consistency could preclude this.

2.3 Manufacturing Development

High-performance requirements of current and future generation of DoD and commercial aircraft programs require the increased application of advanced materials and manufacturing technologies, while budget and funding constraints necessitate that they be cost competitive with their conventional counterparts. The application of primary composite structure is progressing at a slower pace than earlier projections because of high production costs associated with the fabrication of components. The development of lower-cost production processes to fabricate primary composite structural components is essential.

In order to fabricate a large-scale fuselage, a manufacturing development plan must be established. The plan's underlying theme is to present a course of action which includes various options for a specific process and to identify problems that might occur. Under the current philosophy, utilization of structural composites is based upon cost-competitive manufacturing of producible designs; therefore, automated processing and minimization of manufacturing risk are emphasized.

Automated advanced composites manufacturing is at the beginning of its learning curve. Automation in metals manufacturing, by comparison, is years down its learning curve, owing to extensive use of metal since the beginning of the industrial revolution, and in the aerospace industry for the past fifty years. Mistake upon mistake has been made. Voluminous data has been assembled on those methods that do work. Schools have trained large numbers of designers, tool makers, and methods engineers. Complex advanced machines have developed. The learning (improvement) is not as rapid at this point on the slope. Major break-throughs in manufacturing technologies and automation have already been exploited. This does not mean that automation in metals has bottomed out, but most of the major manufacturing developments such as extruding, stretch forming, multi-axis machining, automatic fastener installation, etc., have recently been automated. Automation is already being applied where part quantity justifies the investment cost.

Advanced composites manufacturing is still in its infancy compared to metals manufacturing. Despite considerable development efforts having recently been made, production fabrication and processing remain more an art than a science. Small numbers of people in each discipline are in the process of developing first a workable design and then a workable system of manufacture. This can be easily verified by visiting the major aerospace manufacturers where some tape laying and robotic systems are in their infancy but manual methods continue to be used because efficient automated equipment has not been fully developed.

Early attempts to automate aerospace composites manufacturing have failed mainly because of wrong approaches. Machine designers have simply attempted to mechanize operations that were performed by hand on product designs that were borrowed from conventional sheet metal technology. The correct approach is to develop complete manufacturing systems which incorporate product design, tools, processes, and automated machines that work synergistically. The goal should be to develop manufacturing systems capable of running with the factory lights turned off. Truly great payoff yields will then begin to be realized.

2.3.1 Cost considerations.- Composite fuselage structures must be produced at lower costs than corresponding metal assemblies to be competitive. More efficient methods to process materials must be developed, including the use of 48-inch wide tape, woven fabrics, rovings, crossplied sheets, and narrow tapes (compatible for tape winding or braiding).

Cost-effective manufacturing processes are also considered of primary importance. Since layup and material handling are high-cost and labor-intensive operations, automation must be implemented, including numerically controlled layup, laser and waterjet cutting, robotic stacking, and continuous forming methods. Rapidly installed prefabricated vacuum bags and shorter autoclave and press cure cycles must be used and nonautoclave curing must be developed further.

Cost studies of aircraft assemblies have indicated that mechanical fastening and secondary bonding of precured details are costly operations. It is imperative, therefore, that design/tooling/processing concepts, such as molding to net trim, automated drilling, and mechanized fastener installation be developed to make better use of the unique properties of structural composites.

Ultimately, all structures must be interfaced with skins, clips, doors, ribs, and longerons to permit final assembly. The size and complexity of unitized structures is limited by aircraft access requirements, dimensional tolerance control on the unitized structures, and factory equipment capability. Costs for assembly operations will not fall below 35 to 40 percent of the total airframe cost, unless significant advances are made in manufacturing methods. The tasks to be performed in composite assembly operations will be particulary important because they are generally associated with high value assemblies on which a joining error could result in the loss of the entire structure, thus negating the assembly benefit gained by integral curing. An advanced manufacturing method is needed not only to reduce the high percentage assembly costs, but also to preserve the gains made by integral laminating.

In order to expand the production base being achieved through composite production programs, improvements must be made in assembly methods to reflect the following key criteria for state-of-the-art composite structure assembly:

- Close control of cover-to-substructure drilling parameters to prevent delamination

- Elimination of highly labor intensive drilling operations associated with imprecisely located substructures at the assembly stage

- Drilling and countersinking holes in a single operation using specially designed carbide tools, and automating to drill a large number of holes rapidly.

2.3.2 Manufacturing development plan approach.- The primary objective of the Manufacturing Development plan is two-fold. First, to identify articles and components typical of large-scale aircraft fuselages which could benefit most from the adaptation of high production rate, commercial plastic/composite industry materials and processes. Second, to identify those materials and processing concepts which would satisfy the unique environmental and structural requirements of a fuselage structure.

An effort was undertaken to identify families of metal parts for which substantial cost savings and potential weight savings could be realized through the effective utilization of low-cost plastic/composite materials and manufacturing methods. Five families of parts were identified as being typically expensive metal hardware:

1. High quantity, moderately complex assemblies

2. Parts requiring high energy forming and subsequent welding

3. Chemical etched and machined details

4. Parts requiring drop-hammer forming

5. Investment castings.

From each of these categories, a candidate demonstration article should be selected, with subsequent identification of a suitable advanced composite material and processing method. A cost analysis, therefore, can be made where cost factors such as tooling, material, fabrication, subassembly, reject rate, and material costs are assessed.

The first step in screening the potential material and processing combinations is to establish an efficient design and an effective manufacturing technique for each article. Efficient application of advanced composites can only be accomplished through an effective integration of flexible, innovative design, and cost-effective material and process selection. The screening process, therefore, will be as follows:

1. Establish candidate processes based on structural and environmental requirements for the article.

    a. Injection molding

    b. Resin transfer molding (RTM)

    c. Reaction injection molding (RIM)

    d. Reinforced reaction injection molding (RRIM)

e. Pultrusion

f. Filament winding

g. Compression molding

h. Roll forming

i. Vacuum forming/hydro forming

j. Thermoforming

k. Extrusion

l. Matched die

m. Automated tape placement.

2. Develop conceptual structural configurations and design approaches.

3. Identify which of the selected processes are applicable to each design concept.

4. Itemize advantages and disadvantages for each design from structural and manufacturing points of view.

5. Select most suitable design concepts.

6. Identify potential materials for each concept.

7. Select a concept.

8. Assess automation potential.

The final selection of the materials and processes to be used in a manufacturing plan of a large-scale fuselage will be based on the cost analysis. Per unit costs will be estimated on an anticipated production run of a certain number of shipsets. The following data should be established:

- Material cost
- Tooling cost
- Set—up time
- Material scrap rate
- Rejection rate
- Secondary operations

- Assembly

- Repairability

2.3.3 Fabrication methods.- Potentially, the best way to make major cost reduction in composite manufacturing is to maximize the unique characteristics of composites by applying innovative design/manufacturing approaches. A significant effort is underway to accomplish this by developing the following manufacturing processes.

- Thermoplastic matrix composites appear most promising for manufacturing cost reduction primarily from the fact that these materials can be processed much like sheet aluminum. In-house investigations have identified several solvent resistant thermoplastics that exhibit outstanding material properties such as toughness.

- Pultrusion is the composite equivalent of aluminum extrusion in which continuous fibers are pulled through a mold. This is a well-developed process for industrial fiberglass forms, though existing processes use resins which have limited mechanical properties.

- Braiding of composite fibers provides a highly automated, low-cost composite preform. It is also a solution to the problem of low interlaminar strength present in conventional layups due to the lack of through-thickness fiber orientation.

- Tufting is distinguished from sewing in that sewing requires a mechanical means on both the top and bottom of the workpiece to loop the thread and establish the stitch. Tufting, on the other hand, utilizes only a one-sided stitch, leaving the opposite side free as a "tuft" of thread. The advantage of tufting appears as a practical consideration when considering how one would stitch a very large panel of composites or even a complete fuselage section. Tufting eliminates the requirement for massive machinery and requires only a relatively lightweight head mechanism to drive the needle through the composite material.

  Tufting has been found to provide equivalent strenth to a sewn structure when the tufts have been properly embedded in the epoxy matrix. Both sewing and tufting are highly amenable to automated processes and have been well developed by the garment industry. Where appropriate as a replacement for mechanical fasteners, these techniques hold great promise to significantly reduce joining process costs in aircraft composite structure.

- Filament winding hold a tremendous potential for substantial cost reduction of composite structures. Winding machines can wind 100 to 700 pounds of material per hour and are normally operated by two or three workers. At $50 per man-hour and fiber plus resin at $18 per pound, total cost of laying up 700 pounds of composites varies between $18.21 and $19.50 per pound. The comparable cost for manual layup at 1-1/2 pounds per man-hour (withprepreg at $43/pound) is $76.36 per pound. Automated conventional layup at 10 pounds/man-hour translates into $48 per pound. In addition, complete assemblies can be wound, reducing assembly costs.

The cost of tooling has also become a significant factor in determining whether composites are to be used in production hardware. A prerequisite imposed by Production is that tooling be easy to use in order to reduce labor costs, as well as, function reliably to reduce the high cost of rework. Important factors, therefore, in the preliminary development of the tooling and manufacturing concepts are:

- Establishment of coordination points required to maintain the orientation of filaments and to control alignment of details
- Feasibility of using formed or cast molds
- Use of convertible assembly fixtures to make left and right handed parts in one fixture
- Feasibility of segmenting various large structural elements to facilitate cure or final consolidation
- Establishment of tape placement processes for large toroidal shapes and large flat laminates
- Establishment of solid phase pressure forming techniques for advanced composite materials such as thermoplastics
- Establishment of joining and fastening methods
- Development of large scale machinery to wind large diameter cylindrical elements.

Finally, as the design of the fuselage elements progresses, the flow of detail parts into assemblies and subassemblies and the manufacturing methods can be established. Fabrication of each detail part must then be evaluated to finalize a processing plan. The evaluation will include the following:

- Contour and fitup requirements
- Layup
- Integrally molded metal parts, i.e., metal fittings to composites
- Integrally cocured core consisting of syntactic foams or honeycombs
- Bleeder systems
- Edge trim
- Bagging methods

- Autoclave cure or project tool
- Oven post-cure
- Trimming
- Finishing and sealing requirements
- Handling and shipping methods
- End-item inspection.

Establishment of this plan becomes the basis for the development of the quality control plan, tooling concept, and shop work orders. A planning effort is thus required to increase the efficiency and part flow of the manufacturing effort. Above all, the various efforts must be coordinated to ensure effective processing.

2.3.4 Automated fabrication.- Although composite structures are currently being manufactured in production shops, they are often being produced in a very inefficient manner. The industry is in its nascent stage of understanding automated production, production planning, and the associated material and detail flow throughout the production shops. As the airframe industry moves toward the next generation of advanced composite aircraft, dealing with more and larger structures, the manufacturing and assembly procedures currently being used to fabricate the composite structures must change. To expand the current production base and to drive the present and future cost down, progress must be made in the development and validation of automated fabrication and assembly centers.

Recent developments in decreasing costs of graphite based composites, advances in low cost manufacturing techniques and new material forms, coupled with simplified design concepts have generated a potential for reductions in manufacturing costs and for high volume production of composite assemblies. Emphasis has been directed on individual development of specialized equipment for automated dispensing, cutting, and placement of advanced composite material forms. Even with these developments, however, the tasks remain labor intensive.

Extensive development must be made to implement automation, vis a vis computer integrated manufacturing (CIM), in areas of composite parts fabrication, assembly, material distribution, quality assurance, and production control. Emphasis must be placed on flexible manufacturing systems because of the normally low production rates for large transport aircraft.

CIM embraces six areas of computing technology: computer-aided design, group technology (a type of software for coordinating process planning, scheduling, materials requirements, design and manufacturing), manufacturing planning and control, automated materials handling, computer-aided manufacturing, and robotics. Through these means, CIM can provide computer control to all manufacturing and business functions.

CIM is a unique way of organizing a manufacturing business. To implement it in the aerospace industry, a plan of attack is required. Clear goals with definitive step by step plans are necessary. In preparing for automation, it is important to consider what future developments are likely to be available so that one is not locked out of using a new technology.

2.3.5 Summary.- In conclusion, to fabricate structural shapes and assemble them into a large full-scale fuselage in a cost effective manner, several key areas of manufacturing must be addressed. First, new fabrication methods and materials must be implemented to make simplified design concepts. Second, a manufacturing and assembly plan must be developed, and third, CIM methods must be applied to the manufacturing and assembly plan. It is estimated that a cost reduction of approximately 25% can be realized when the above methods are developed and implemented. The technologies are available though methods of implementation have yet to be applied. However, in order to justify conversion to large scale composite fuselage production at this time, extensive expenditures for capital equipment and additional development would be required.

2.4 Design Verification

The complete test program required to provide the confidence to commit to the production of advanced composite fuselages for large-transport aircraft consists of three phases.

The first phase is concept development testing. This phase not only provides data on the various concepts for each component but also provides verification of analytical methods and solutions for technology problems. These tests will involve a wide range of design concepts, each of which satisfy one or more technology issues.

The second phase is concept evaluation where the concepts developed in the first phase are integrated and evaluated with respect to all the design criteria and technology issues. This phase also involves the testing of concepts designed with consideration of advanced manufacturing techniques. The results of this phase will be to narrow concepts into one or two generally viable configurations.

The third and final phase is the fuselage technology demonstration phase. The preferred configuration is now selected and a full-scale barrel section is designed and fabricated for ground test. Some concept verification testing will be performed during the detail design.

The first phase is being addressed in the Fuselage Critical Technology programs. These programs are designed to provide some of the answers for the technology issues which are addressed in Section 1.1. In these Critical Technology programs Lockheed is developing technology for acoustic transmission and impact dynamics, Boeing is developing technology for pressure containment, damage tolerance and post buckling, and Douglas is developing technology for cut-outs, joints and durability.

The second and third phases of the test program are summarized in Table 32.

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TABLE 32. - DESIGN VERIFICATION TESTS

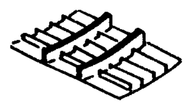
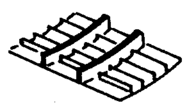
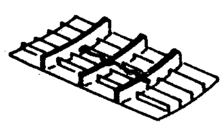
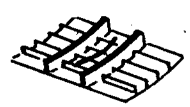
Item No.	Description	Type of Test	Number of Specimens	Test Variables	Purpose of Test
1	Column  5 ft x 3 ft	Static compression	2	2 Designs RTD	Evaluate post buckling strength, skin-to-frame and stiffener interfaces
2	Post-impact Strength  5 ft x 3 ft	Static, combined compression, shear and pressure Fatigue, combined compression shear and pressure	4 2	2 Designs 1 each hot wet 1 each cold dry 2 Designs RTD	Damage tolerance under adverse environments Damage growth
3	Fail Safe  7 ft x 3 ft	Longitudinal cut, axial compression, shear and hoop tension	2	2 Designs cold (-65°F) dry	Evaluate damage containment
4	Fail Safe  5 ft x 3 ft	Circumferential cut, axial compression, shear and hoop tension	2	2 Designs cold (-65°F) dry	Evaluate damage containment

TABLE 32. - DESIGN VERIFICATION TESTS (CONTINUED)

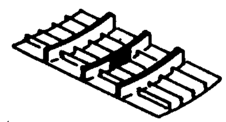
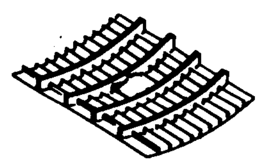
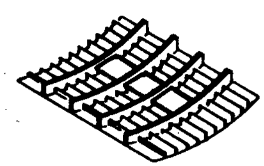
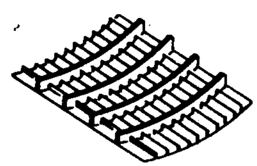
Item No.	Description	Type of Test	Number of Specimens	Test Variables	Purpose of Test
5	Repair  7ft x 3 ft	Combined axial tension, shear and pressure	2	2 Designs cold (-65°F) dry	Evaluate service repairs
6	Cut-out  8 ft x 4 ft	Static, shear and tension	2	2 Designs RTD	Evaluate cut-out reinforcement
		Fatigue, shear and compression, with edge damage	2	2 Designs RTD	Damage tolerance door cutouts
7	Windows  8 ft x 4 ft	Static, combined shear and pressure	1	RTD	Verify design
		Fatigue, combined shear and pressure, impacted	1	Impact damage at edge of window cut-out	Damage tolerance
8	Joints  6 ft x 3 ft	Static, combined shear tension and pressure	2	1 Longitudinal 1 Girth RTD	Verify design
		Fatigue, combined shear, tension and pressure	2	Impact damage in joint 1 Longitudinal 1 Girth	Damage tolerance

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TABLE 32. — DESIGN VERIFICATION TESTS (CONTINUED)

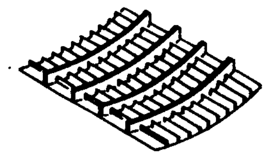
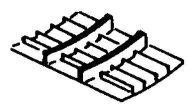
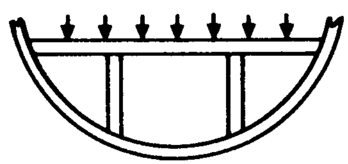
Item No.	Description	Type of Test	Number of Specimens	Test Variables	Purpose of Test
9	Electromagnetic  6 ft x 4 ft	Swept stroke, spliced panel	1	1 Design	Verify protection
		Swept stroke, spliced panel Fatigue, combined loads	1	1 Design	Verify protection damage tolerance
10	Sonic Fatigue  3 ft x 2 ft	Sonic fatigue (primarily military application)	1	1 Design	Verify design
11	 20 ft x 8 ft	Static compression	1	RTD	Floor support design verification

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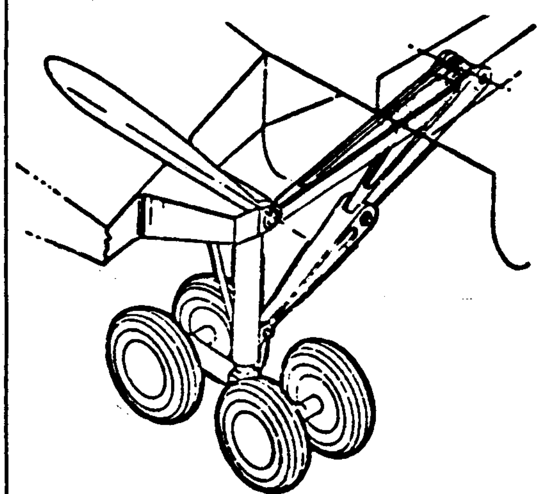
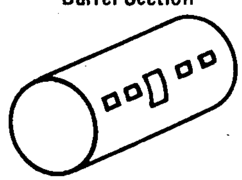
TABLE 32. – DESIGN VERIFICATION TESTS (CONTINUED)

Item No.	Description	Type of Test	Number of Specimens	Test Variables	Purpose of Test
12	Landing Gear Side Stay Support Structure 	Static	1	RTD	Design verification of an area of high local load introduction
13	Barrel Section 	Static and fatigue	1	RTD Limit load Fatigue Damage tolerance Fail safe Static strength	Design validation

The full-scale barrel test program for the fuselage technology demonstration will be performed in a number of phases. A schematic of the test setup is shown in Figure 31. The first phase consists of a static test to limit load. This will involve testing three conditions: maximum down bending, maximum up bending, and maximum torsion. All testing will include the appropriate internal pressure. The results of these tests will be evaluated to determine that the structure is behaving as anticipated.

The second phase will involve a fatigue test for one lifetime to verify the structural durability.

The third phase will involve fail-safe testing with major damage inflicted. One flight of loads will be applied with appropriate factors.

The major damage will then be repaired and small damage due to impact will be inflicted at predetermined critical locations. The fourth phase will then be a second lifetime of fatigue loading to establish that the damage tolerance criteria have been met and to validate the repairs. Inspection will be performed at regular intervals and repairs performed as required.

The fifth phase will involve static testing to Design Ultimate Loads with appropriate factors to account for environmental effects. The same conditions tested in the first phase will be applied. Finally a static test to failure will be performed.

A schedule for the full-scale barrel ground test is shown in Figure 32. The approximate man-year effort is shown with each test.

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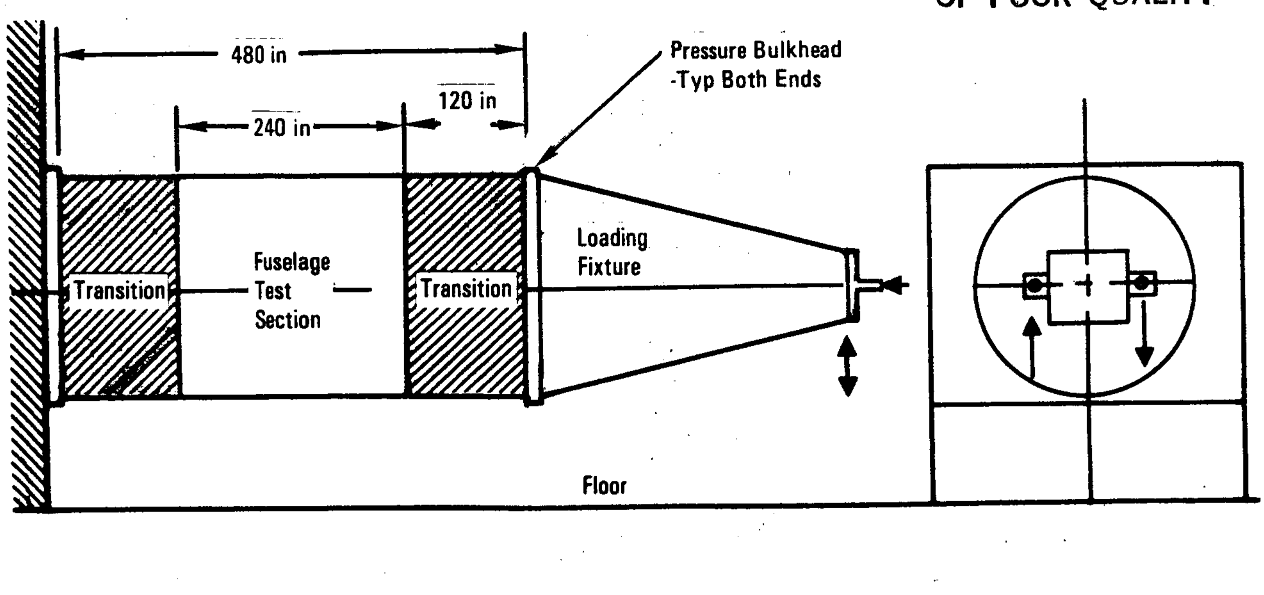

Figure 31. - Schematic of barrel test setup.

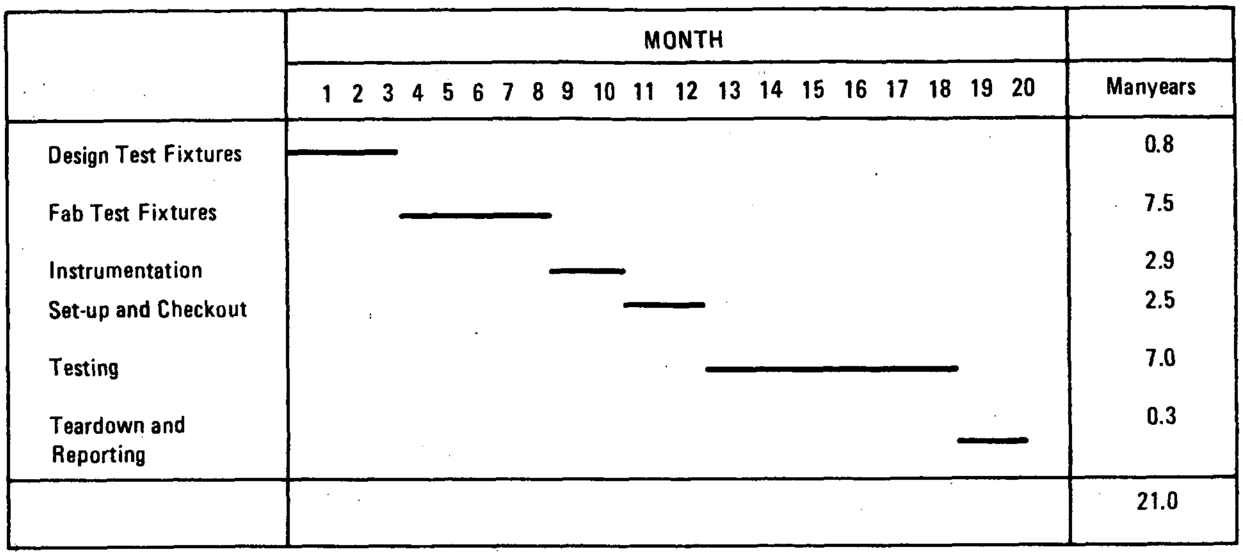

Figure 32. - Full-scale barrel ground test schedule.

3. PROGRAM SCHEDULE AND RESOURCE REQUIREMENTS

A schedule has been developed for the fuselage technology demonstration program and an estimate of the manpower resources to accomplish the program has been made.

The proposed Fuselage Technology Demonstration program will consist of six technical phases: Engineering Development, Manufacturing Development, Design Development Tests, Tool Design and Fabrication, Barrel Fabrication and Validation Testing. A program schedule is depicted in Figure 33 and the engineering/manufacturing man years are shown. Management and reporting will add approximately 20% for a total of 278 man years.

The barrel section designed will be 20 feet in length and 19 feet 7 inches in diameter. It will contain doors and windows and at least one feature inputting a high local load. The design will incorporate all the technologies developed under the earlier programs including full impact dynamics treatments. An indepth analysis will be performed to evaluate fully the 3-D loading effects on the structure. During the detail design phase key areas will be identified for design development testing.

Because of the size of the barrel section, almost 20 ft in diameter, a certain amount of Manufacturing Development will be required. A manufacturing plan detailing the fabrication and assembly procedures will be prepared early in the program. During the Manufacturing Development, a parallel effort will be conducted to develop the nondestructive inspection procedures and methods to ensure structural integrity. Every effort must be made to employ production-type tooling and fabrication methods, particularly automated methods for fabrication of such things as frames, stringers, and floor beams. A number of large panels and other components will be fabricated during this phase. Nonautoclave curing techniques will be exploited.

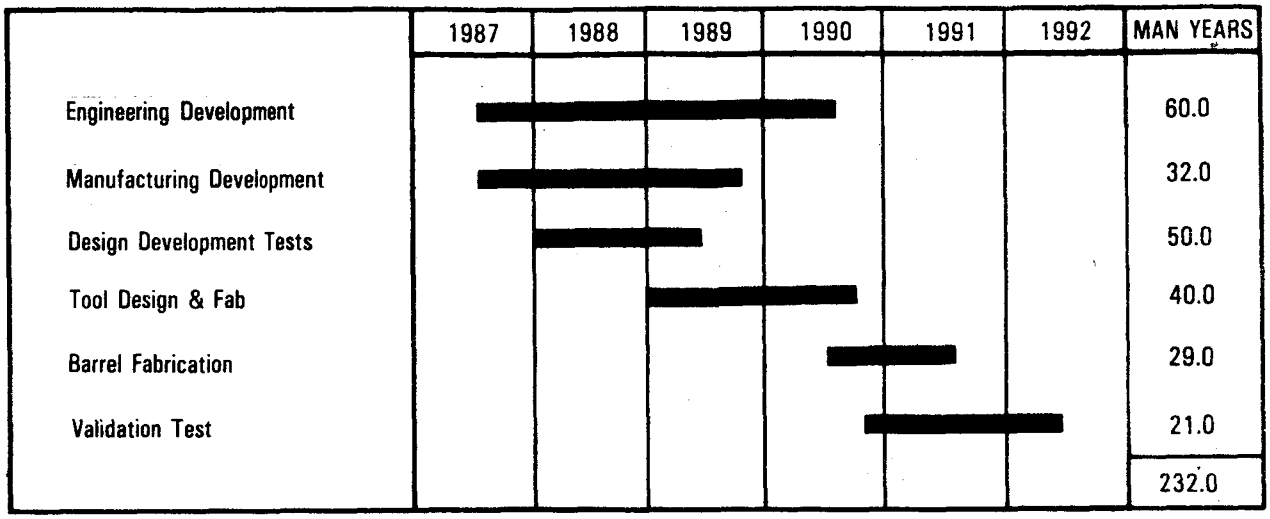

Figure 33. - Fuselage technology demonstration schedule.

The Design Development Test phase will be coordinated with the Manufacturing Development phase as far as is possible so that the test panels will be the panels fabricated during the Manufacturing Development. Tests to be performed during this phase will include, but not necessarily be limited to fail-safe, major load input details, lightning strike, combined loads on environmentally conditioned panels and major joints and splices. The exact test program will be dependent on the results of the related technology development programs.

Tool design and fabrication will be oriented to provide tooling which is representative of production approaches although not necessarily rate tooling. Tooling will be designed to take advantage of automated fabrication techniques available at that time.

Fabrication of the barrel section will be performed by the production plastics shop. Production planning and on-line inspection will be used and every effort will be made to employ automated in process controls.

The validation test program is discussed in Section 2.4, Design Verification.

3.1 Relationship of Other Composites Technology Programs

There are programs currently underway and planned which could provide some of the data and technology needed for a composite fuselage, in particular, the development of new improved materials and processes. The development of improved materials is largely the responsibility of the vendors and because of the potential business, this is indeed occurring. Improved processes and controls of processing are the aim of several existing contracts and will provide increased reliability and reduced cost. No material or basic processing development is included in the proposed program.

The Wing Key Technology Programs are providing a basis for damage tolerance, lightning protection, and the design of major joints. The Fuselage Technology programs will examine the major technology issues associated with composite fuselages: acoustic transmission, impact dynamics, pressure containment, damage tolerance, cut-outs, joints, durability, and post buckling. These programs will include periodic technology transfer workshops to maximize the flow of data among the contractors.

Various NASA and DoD programs, both in-house and contractual in the RD and T areas are also contributing to the technology data bank with new analytical methods and work on specific issues such as post buckling, damage tolerance, and impact dynamics.

The USAF Fuselage Mantech program is based on a 15 foot diameter fuselage shell. One key area where this program may provide data will be in the fabrication of frames. Commercial wide body passenger transports and large military cargo transport airplanes can have fuselage diameters well in excess of 15 feet. Manufacturing technology is not normally scalable. Large fuselages cannot be economically or reliably fabricated in one piece or even in one-piece barrel sections. Large panel assemblies would need special tooling and could not generally be autoclave cured. An assessment of the possible technology transfer from the Fuselage Mantech program cannot be made at this time.

The Fuselage Technology Integration program will provide generic data to the industry. The results of the program will be primarily applicable to specific structural requirements, design approaches and fabrication methods. The effects of this program were thus not included in the development of the resource requirements shown in Figure 32.

An assessment was made of the maximum potential input from other programs in order to establish a lower bound for the Fuselage Technology Demonstration program. The possible relationship between the programs is shown in Figure 34.

4. CONCLUDING REMARKS

This study has defined the technology issues, the military and commercial benefits and a plan for development of the technology readiness to enable a production commitment to be made in the 1990's for an advanced composite fuselage on a large transport aircraft.

The technology issues in need of resolution in order of urgency are: impact dynamics, acoustic transmission, joints and splices, pressure containment, post buckling, shell cutouts, automated manufacturing, processing science, electromagnetic effects, repair, NDE/NDI, and flame, smoke and toxicity. Damage tolerance and fail-safety are included under pressure containment. The major issues are essentially the same for both military and commercial aircraft.

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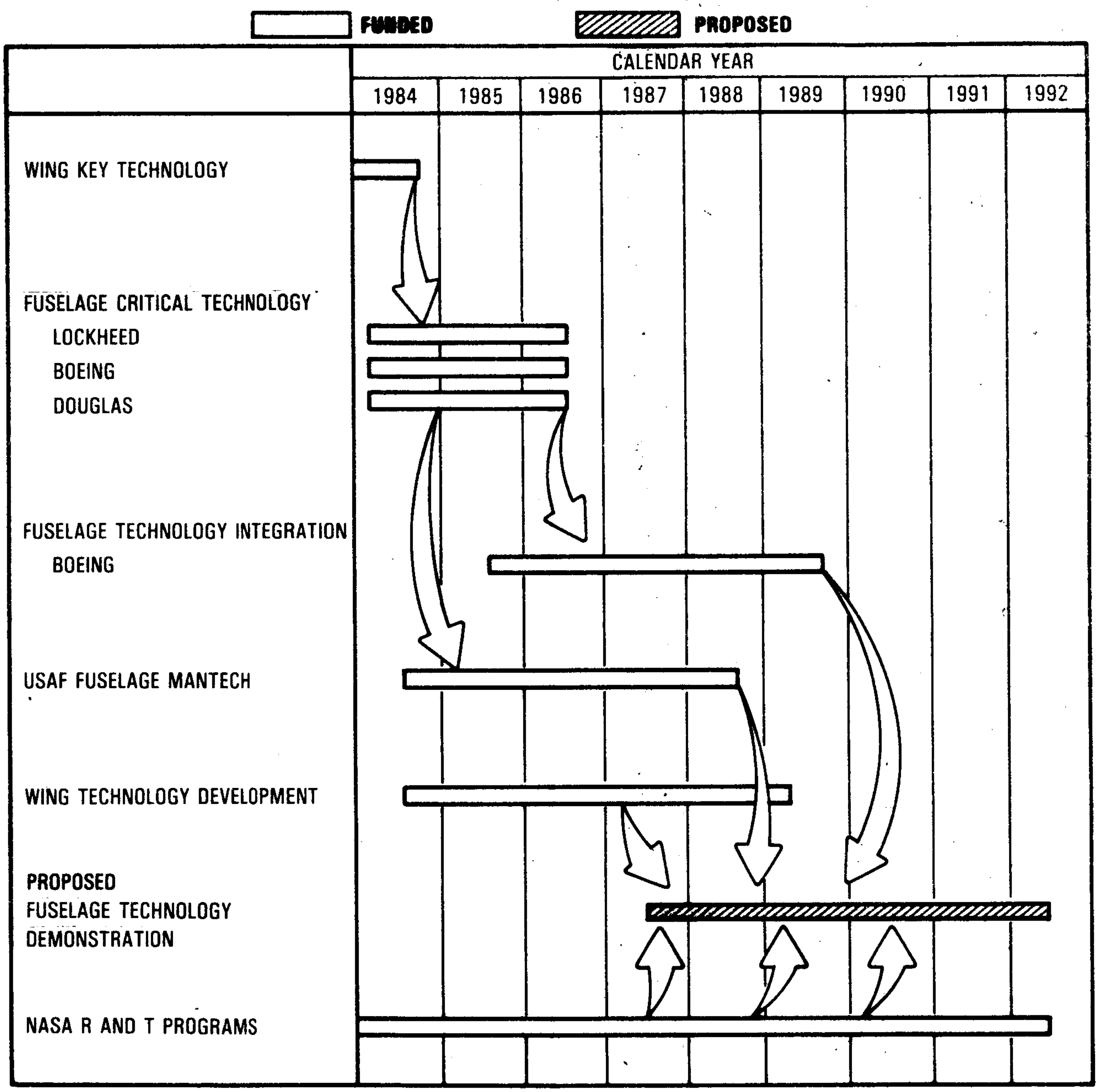

Figure 34. - Relationship of composites technology programs and overall schedule.

The first six items listed are the most important. The impact dynamics issue is urgent because its resolution may affect the basic structural concepts of the lower fuselage shell. Acoustic transmission is urgent because the magnitude of the problem must be defined to determine if the weight of acoustic treatment needed will negate the weight saved by using advanced composites. Joints and splices are urgent from the point of view of the frame-to-skin joints and the question of whether mechanical attachment is required along with bonding. Pressure containment is ranked fourth on the basis of the fail-safe design aspects which must be resolved. Post-buckling has an impact on both pressure containment and joints and splices. Shell cutouts and reinforcement can affect the basic shell design for passenger aircraft and can introduce significant out-of-plane stresses. The resolution of these issues is now the subject of the Fuselage Critical Technology programs. The remaining issues are subordinate to these major issues and will be resolved in the overall fuselage technology development program.

The benefits analyses identified cost and weight savings from the incorporation of advanced composites in fuselage structures and in the overall airplane structure. Structure weight savings of 16 percent and 22 percent are projected for commercial and military fuselages respectively. For the all composite aircraft structure weight savings are 26 percent for commercial and 29 percent for military.

Commercial total operating costs are reduced by 5 to 6 percent for the all composite airplane and return on investment improvements of up to 48 percent were projected. The military life cycle costs were reduced by 10 percent for the all composite airplane. There is a lack of production cost data for large composite structures so analyses are open to interpretation. However, it is likely that automation will have a more significant effect on reducing costs of composite structures than on metallic structures because of the large cocured assemblies and the resultant reduced assembly requirements. Manufacturing cost savings are thus dependent on the degree of automation.

A test plan was developed summarizing the requirements for the planned Technology Integration program and the proposed Technology Demonstration program.

The proposed Fuselate Technology Demonstration program was developed from a review of the program options. The options included component testing, barrel section testing, and full-scale fuselage testing with the additional options of ground test or flight test.

The proposed program involves approximately 278 man-years of effort over a 5-year period. The program consists of six technical phases; Engineering Development, Manufacturing Development, Design Development Tests, Tool Design and Fabrication, Barrel Fabrication and Validation Tests. The program culminates in the design, fabrication and test of a full-scale barrel section. This program offers the most cost-effective approach to providing the technical and manufacturing confidence required. The proposed program is beyond the effort of the current technology development programs.

A fuselage structure is subjected to multi-axial loading induced by bending, torsion and pressure loading. The frames and floor support structure and the large cutouts for passenger and cargo doors induce significant out-of-plane loads in the structure. The interaction of the various structural elements and loads can only be properly simulated by the test of a full-scale barrel section or a complete fuselage. However, testing of a complete fuselage does not offer any significant improvement in technology demonstration when compared with testing a full-scale barrel section and does not justify in any way the significantly increased cost.

Similarly, manufacturing technology is not "scalable," that is, the techniques and processes for fabrication and assembly of a small diameter cylinder are not the same as for a large diameter cylinder. Consequently, fabrication of a full-scale barrel section is necessary to validate the manufacturing technology for a large transport composite fuselage.

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16. Abstract

A study was performed to plan the effort required by the transport aircraft manufacturers to support the introduction of advanced composite materials into the fuselage structure of future commercial and military transport aircraft. The study identified the technology issues which must be resolved, assessed the potential benefits to military life-cycle costs and commercial operating costs, and defined a plan to develop the technology and confidence needed to commit to production of composite fuselages for large transport aircraft in the 1990's.

The most urgent technology issues defined are impact dynamics, acoustic transmission, pressure containment and damage tolerance, post-buckling, cutouts and joints and splices.

A technology demonstration program was defined and a rough cost and schedule identified. This program culminates in the fabrication and test of a full-scale fuselage barrel section. Commercial and military benefits were identified. Fuselage structure weight savings from use of advanced composite were 16.4 percent for the commercial and 21.8 percent for the military. For the all composite airplanes the savings were 26 percent and 29 percent respectively. Commercial/operating costs were reduced by 5 percent for the all composite airplane and military life-cycle costs by 10 percent.

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