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A Study of the Utilization of Advanced Composites in Fuselage Structures of Commercial Aircraft

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Final Report

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Douglas Aircraft Company
Long Beach, California 90846

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January 1985

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16. Abstract

A study was conducted to define the technology and data needed to support the introduction of advanced composites in the future production of fuselage structure in large transport aircraft. Fuselage structures of six candidate airplanes were evaluated for the baseline component. The MD-100 was selected on the basis of its representation of 1990s fuselage structure, an available data base, its impact on the schedule and cost of the development program, and its availability and suitability for flight service evaluation. Acceptance criteria were defined, technology issues were identified, and a composite fuselage technology development plan, including full-scale tests, was identified. The plan was based on composite materials to be available in the mid to late 1980s. Program resources required to develop composite fuselage technology are estimated at a rough order of magnitude to be 877 man-years exclusive of the bird strike and impact dynamic test components.

A conceptual composite fuselage was designed, retaining the basic MD-100 structural arrangement for doors, windows, wing, wheel wells, cockpit enclosure, major bulkheads, and interfaces with existing aircraft systems and cabin interior arrangements. A 32-percent weight savings from the existing MD-100 design was realized for this design.

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PREFACE

This final report was prepared by Douglas Aircraft Company, McDonnell Douglas Corporation, under contract NAS1-17416, "Study of Utilization of Advanced Composites in Fuselage Structures of Large Transports." The study was conducted for the Aircraft Composite Structures Technology (ACST) program which is part of the NASA Aircraft Energy Efficiency (ACEE) program. The program was partially funded by the Air Force Wright Aeronautical Laboratory to ensure that the study would be applicable to large military transport aircraft.

The study program was monitored by John Pyle, ACEE Composites Project Office, Langley Research Center, NASA. James Mullineaux, ADPO-AFWAL, was the Air Force Project Manager. D. J. Watts was the Douglas Project Manager.

In addition to the authors, Douglas contributors to this project included M. P. Amason, electromagnetic effects; M. M. Platte, cost analysis; and R. L. Oswald, program administration.

GLOSSARY

A	area
a	half-crack length
(a)	thermal coefficient of expansion
A_1	area under carbon-epoxy stress-strain curve
A_2	area under aluminum stress-strain curve
ADF	automatic direction finder
ADH	adhesive
AMB	ambient
ASSY	assembly
ATC	Air Traffic Control
ATP	authority to proceed
B	bending
BTU	British thermal units
C	compression
c	characteristic length
℄	centerline
$℄_F$	frame centerline
$℄_L$	longeron centerline
Conf	configuration
CRT	cathode ray tube
dB	decibel
DBLR	doubler
DCB	double cantilever beam
deg	degrees
DME	distance measuring equipment
E	elastic modulus
EMI	electromagnetic impulse
ENGG	engineering
FAB	fabricate
FT	feet
FLEX	flexible
FPS	feet per second
FUS	fuselage
FWD	forward
°F	degrees Fahrenheit
G	acceleration
GH_z	gigaHertz
HC	honeycomb
HF	high frequency
HYD	hydraulic
Hz	Hertz
ILS	instrument landing system

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IN.	inch
INS	inertial navigation sytem
INSTL	install or installation
K	stress intensity factor
k	thousand
(k)	thermal conductivity
kPa	1,000 Pascals
KSI	thousands of pounds per square inch
L	longeron
L&R	left and right
l	length
LB	pound
LN_2	liquid nitrogen
MEK	methyl ethyl ketone
MH_z	MegaHertz
MO	months
NDE	nondestructive evaluation
NDI	nondestructive inspection
NDT	nondestructive test
OMEGA	VLF worldwide navigation
P	pressure
P_a	applied load
PLM	plastic laminating mold
Prep	preparation
PROT	protection
PSI	pounds per square inch
R	radius
RH	relative humidity
RT	room temperature
S	shear
SATCOM	satellite communication
SEC	seconds
SHF	super high frequency
SPEC	specimen
STA	station
STRUCT	structure
T	tension
$\bar{t}$	thickness, smeared area
tan	tangent
TBD	to be determined
T/CAS	threat-alert collision avoidance
TEMP	temperature
TYP	typical

U	deflection
UHF	ultra-high frequency
V	velocity
V_c	cruise speed
VHF	very high frequency
VLF	very low frequency
W	panel width
e_1	failure strain of carbon-epoxy
e_2	failure strain of aluminum
μ	micro
π	pi
σ	stress

SUBSCRIPTS

∞	infinity
K_{tc}	shear concentration factor
MAX	maximum

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ILLUSTRATIONS
(Continued)

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SUMMARY

A study was conducted to define the technology and data needed to support the introduction of advanced composite structures in the future production of fuselage structure in large transport aircraft.

The basic structural integrity requirements for the study were taken from Federal Aviation Regulations Part 25, FAA Advisory Circular 20-107, "Composite Aircraft Structure," and the MIL-H-8860 series and other military specifications. Inputs were obtained from the Douglas Aircraft Product Support division which is in direct contact with several hundred airlines throughout the world. Douglas' management has endorsed the study conclusions and recommendations.

The study accomplished the following tasks:

- Defined the acceptance criteria.
- Identified the technology issues.
- Evaluated six candidate fuselages for a study baseline.
- Assessed three fuselage options for the ground test article and three panel options for the flight service evaluation.
- Defined a composite fuselage technology development plan.
- Identified full-scale tests.
- Prepared a composite fuselage conceptual design.
- Estimated program costs, and the facilities and equipment needed to accomplish the development plan.
- Identified critical technologies for timely program planning.

A comprehensive list of acceptance criteria was formulated for the manufacturer, airline operators, military operators, and the FAA, based on the experience of Douglas.

A set of 13 issues was derived from the acceptance criteria to form the basis for a technology assessment. Each issue was examined to determine the technological, economic, or programmatic problems to be resolved by a composite fuselage technology program. Recognition was given to probable contributions to the technology by other composite programs in Government and industry so that they need not be repeated in the fuselage technology program.

Seven technologies were cited as requiring early development because of their urgency of resolution or their effect on the design integration within a limited design development schedule:

- Damage tolerance
- Durability
- Impact dynamics

- Manufacturing methods
- Large cutouts and joints
- Acoustics
- Electromagnetic effects.

Early development of manufacturing methods for a composite fuselage should be funded by the manufacturer because of its dependence on its own facilities, equipment, and experience base in establishing a methodology.

The electromagnetics issue should be addressed by a parallel program involving avionics and electrical experts to investigate the effects of a low-conductivity composite fuselage shell on the design of avionics and electrical systems, particularly with respect to planned future technological improvements in that discipline.

Fuselage structures of six candidate airplanes were evaluated for the baseline component. The MD-100 was selected on the basis of its representation of 1990s fuselage structure, an available data base, its impact on the schedule and cost of the development program, and its availability and suitability for flight service evaluation. The MD-100 is a derivative of the DC-10 aircraft and has structural commonality with the Air Force KC-10 tanker/cargo airplane.

A three-phase development plan was established to generate the required technology and data:

Phase I — Design Development
Phase II — Structural Verification
Phase III — Flight Service Evaluation

The Phase I plan contained no program option. All tasks defined for the program were considered essential.

The following full-scale tests were specified for Phase II: static ultimate, durability and damage tolerance, bird strikes, and impact dynamics. The center fuselage was selected as the test article for the Phase II static, durability, and damage tolerance tests as a preferred option over a forward or aft fuselage section to cover a more comprehensive range of technology that encompasses the cutout for the wing structure and the keel and main wheel well, in addition to the basic fuselage shell structure. Major repairs will be made on the test article, and these will be subjected to static and repeated loads to the extent practical.

A forward lower fuselage panel was selected for flight service evaluation in Phase III. The panel contains typical skin-longeron-shear tee elements, a cargo door and door jamb structure, and longitudinal and transverse panel joints. The panel would be exposed to damage from runway debris and to abuse during cargo handling.

A conceptual composite fuselage was designed, retaining the basic MD-100 structural arrangement for doors, windows, wing, wheel wells, cockpit enclosure, major bulkheads, and interfaces with existing aircraft systems and cabin interior arrangements. A 32-percent weight saving from the existing MD-100 design was realized for this design.

The study concludes that it is highly improbable that a commitment to manufacture a composite fuselage will be made until sufficient data and technology are available to resolve the economic, programmatic, and technological risks. A comprehensive composite fuselage development program is needed to resolve these issues.

Program resources required to develop composite fuselage technology are estimated at a rough order of magnitude to be 877 man-years exclusive of the bird strike and impact dynamic test components.

Approximately 125,000 square feet of manufacturing facilities will be required to fabricate the full-scale fuselage barrel test article for the static ultimate, durability, and damage tolerance structural verification tests.

SECTION 1
INTRODUCTION

The NASA Aircraft Energy Efficiency (ACEE) composites program has provided the aircraft manufacturer, the FAA, and the airlines with the experience and confidence needed for extensive use of composites in secondary and medium primary structure in future transport aircraft. Secondary and control surface structures made of composites are already in airline service on a production basis, and composite medium primary structures have been introduced for flight service evaluation. Studies to determine the requirements to achieve technological readiness for composite primary wing structures have already been completed under the ACEE wing studies program (References 1 to 3) and key technology issues are currently being addressed under separate contracts.

The composite fuselage structure has significantly different design criteria and structural features from composite wing structures. The wing study findings do not necessarily apply with respect to weight savings, cost, and the programmatic and technical issues involved. The fuselage comprises about 33 percent of the structural weight of a transport aircraft, and weight savings of 25 percent would result in significant benefits in some or all of the following: specific fuel consumption, range, landing field distance, and increased payload.

The objectives of the composite fuselage study are to (1) define the technology and data needed to support an aircraft manufacturer's commitment to utilize composite fuselage structure in future large transport aircraft, and to (2) develop plans for a composite fuselage development program which will supply the needed technology and data. Without the data and a demonstrated technological readiness, commercial and military aircraft operators would be unlikely to accept composite structure for the fuselage.

Two factors strongly influence the amount of technology and data that will be needed to support a commitment to composite fuselage structure:

- Technology for the design and manufacture of conventional fuselage structure has been developed over the past 50 years by a large industry which has invested heavily in test programs, facilities, and equipment, and is supported by the service experience of thousands of aircraft. Regulations have evolved that demand the high level of safety provided by these structures. It is understood that composite fuselage structures will, indeed, require a high level of technology and a proven data base to compete with this mature technology.
- This study is targeted for a 1990s date for a commitment to utilize composites in the fuselage structure. By this time, conventional fuselage construction will be advanced through improvements achieved in aluminum alloys and better manufacturing methods such as adhesive bonding of structure. These advancements do not require a technological breakthrough and are more adaptive to existing facilities and equipment. Corresponding improvements must be attained in the development of the composite fuselage for it to compare favorably with competing systems.

The study was organized to define the issues, assess the state of the art for technology gaps, create a baseline conceptual design, and define composite fuselage technology which will provide the required state of technical readiness. A flow chart for the study tasks is shown in Figure 1-1 and the study schedule is given in Figure 1-2.

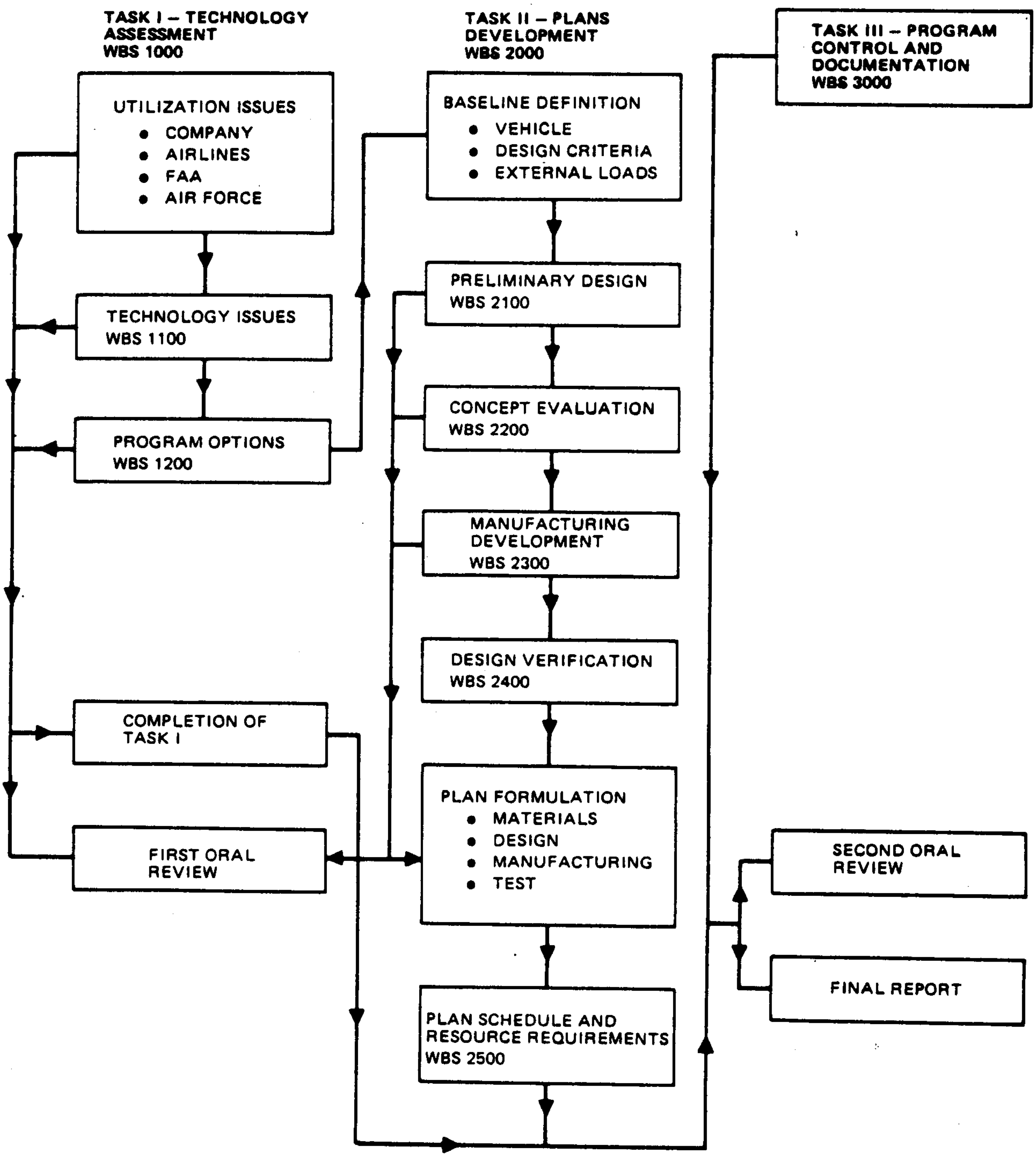

FIGURE 1-1. PROGRAM FLOW CHART

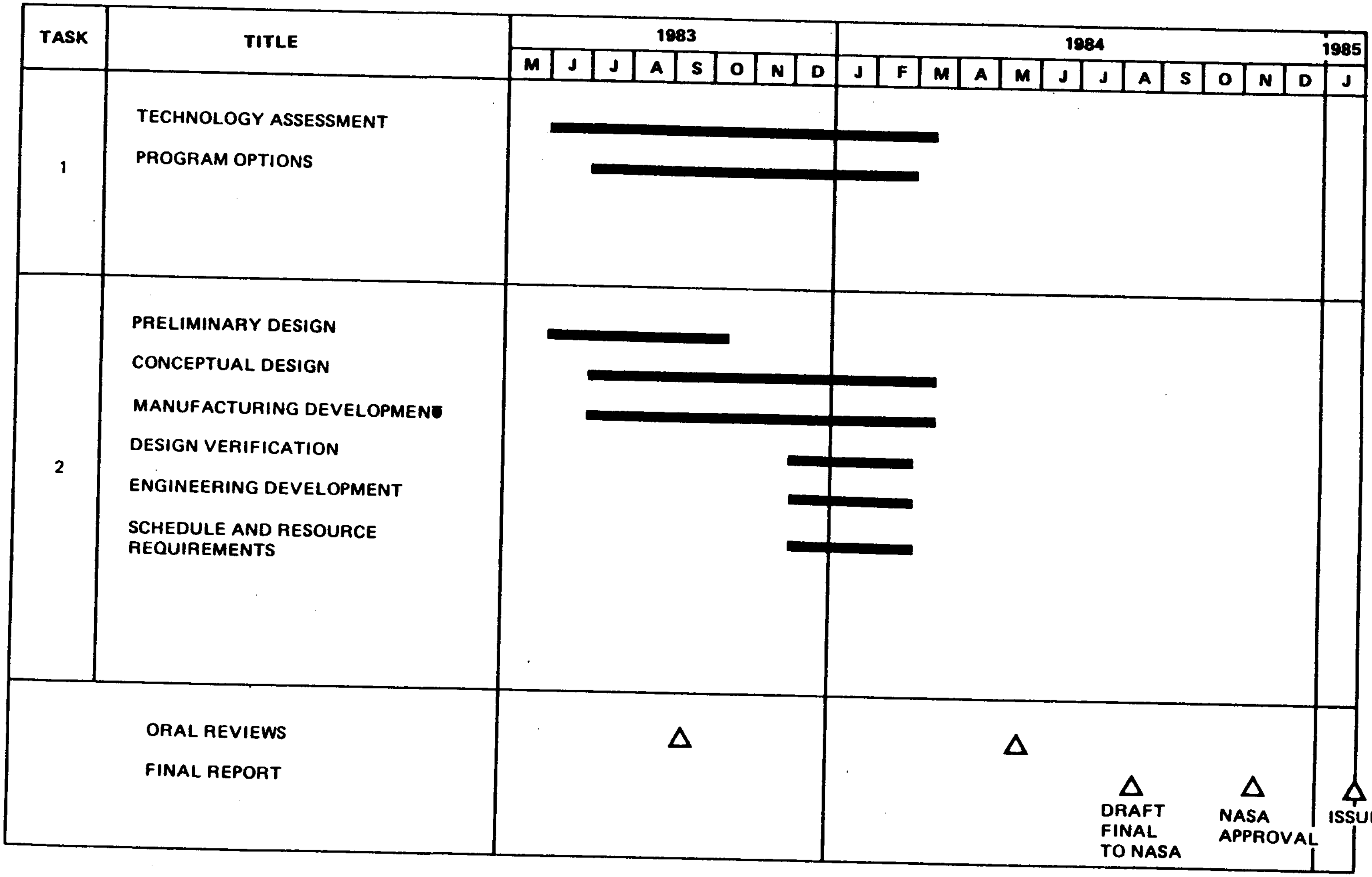

FIGURE 1-2. COMPOSITE FUSELAGE STUDY SCHEDULE

SECTION 2
COMMITMENT TO PRODUCTION

New, large transport aircraft designs are established on the basis of the manufacturer's technology base and the needs of the using commercial airline or military airlift operation. It is not likely that a manufacturer would undertake a major design change such as a composite fuselage structure without a consensus from the airlines or military users. Therefore, in a practical sense, a commitment to production of composite fuselage structure by an aircraft manufacturer is dependent upon its acceptability to the airlines and military users.

Acceptability can be examined on the basis of the benefits to be derived from the change versus the risks encountered in introducing new technology. Potential benefits can be divided into the following areas: (1) reduced manufacturing costs, (2) reduced maintenance costs, (3) longer durability, and (4) improved aircraft performance in terms of range, payload, landing field lengths, and specific fuel consumption.

The risks involved reflect the uncertainties which arise with the introduction of new technology in attaining a high level of structural integrity, achieving projected cost and weight savings, and being able to establish realistic schedules. The seriousness of failure is high; therefore, the probability of failure must be quite low. Table 2-1 summarizes those issues for which the manufacturers, users, and the regulatory agency must have demonstrable evidence of low risk before a production commitment can be made. To put things in the proper perspective, we are talking about decisions affecting the success of a multibillion dollar program. Obviously, these issues will be carefully considered at the highest level of civil and military management.

TABLE 2-1
ACCEPTANCE SUMMARY

	MANUFACTURER	AIRLINES	FAA	MILITARY
STRUCTURAL INTEGRITY FACTORS:				
MATERIAL AND FABRICATION	X	X	X	X
STATIC STRENGTH	X	X	X	X
FATIGUE/DAMAGE TOLERANCE	X	X	X	X
CRASHWORTHINESS	X	X	X	X
FLAMMABILITY	X	X	X	X
LIGHTNING PROTECTION	X	X	X	X
PROTECTION OF STRUCTURE	X	X	X	X
QUALITY CONTROL	X	X	X	X
REPAIR	X	X	X	X
FABRICATION METHODS	X	X	X	X
MILITARY THREATS	X			X

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TABLE 2-1
ACCEPTANCE SUMMARY
(CONTINUED)

	MANUFACTURER	AIRLINES	FAA	MILITARY
OPERATIONAL FACTORS:				
RELIABILITY		X		X
MAINTAINABILITY		X		X
INSPECTABILITY		X		X
REPAIRABILITY		X		X
ECONOMIC FACTORS:				
ACQUISITION COSTS		X		X
LIFE–CYCLE COSTS		X		X
WARRANTIES		X		
FACILITIES	X	X		X
EQUIPMENT	X	X		X
PROGRAM RISK FACTORS:				
DESIGN DATA	X			X
PRODUCIBILITY DATA	X			X
SCHEDULE DATA	X	X		X
COST DATA	X	X		X
STAFF EXPERIENCE	X	X		X
AIRLINE ACCEPTANCE	X	X		
FAA ACCEPTANCE	X	X		
MILITARY ACCEPTANCE	X			X

EXISTING EXPERIENCE BASE

A rapidly growing technology base for composite aircraft structure has emerged during the past few years, although it is still insignificant compared with the technology base for conventional aircraft structure. Table 2-2 lists a number of composite applications cited in DoD/NASA Advanced Composites Design Guide. Some of the more significant applications are the control surface and medium primary structural components developed by the NASA ACEE programs, the Boeing 767/757 secondary structure and control surface applications derived from NASA ACEE experience, the Lear Fan all-composite airplane, the Navy AV-8B Harrier wing, and numerous Air Force-sponsored military aircraft programs. Unfortunately, many of the issues related to production of composite fuselage structure for a large transport aircraft still remain unresolved.

TECHNICAL ISSUES

Section 3 of this study is devoted to an assessment of the technical issues. These issues address flight safety design requirements integrated into a durable and producible low-cost design with significant weight savings as an incentive for the commitment to production to be made.

TABLE 2-2
SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES

COMPONENT/ APPLICATION	SOURCE	MATERIAL SYSTEM
WING COMPONENTS		
737 SPOILERS	BOEING	CARBON-EPOXY
757 AND 767 SPOILERS	BOEING	CARBON-EPOXY
747 AILERON	BOEING	CARBON-EPOXY
757 AND 767 AILERONS	BOEING	CARBON-EPOXY
757 AND 767 FLAP	BOEING	CARBON-EPOXY
A-7 OUTER WING	VOUGHT	CARBON-EPOXY
L-1011 INBOARD AILERON	LOCKHEED	CARBON-EPOXY
DC-10 AILERON ACCESS DOOR	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 WING SKINS	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 WING SLATS	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 FLAPS	MCDONNELL DOUGLAS	CARBON-EPOXY
AV-8B WING	MCDONNELL DOUGLAS	CARBON-EPOXY
AV-8B FLAPS	MCDONNELL DOUGLAS	CARBON-EPOXY
AV-8B AILERONS	MCDONNELL DOUGLAS	CARBON-EPOXY
B-1 SLAT	ROCKWELL	CARBON-EPOXY
B-1 FLAP	ROCKWELL	CARBON-EPOXY
HiMAT WING AND CANARD	ROCKWELL	CARBON-EPOXY
F-100 WING SKINS	ROCKWELL	BORON-EPOXY
F-111B WING SKIN	VOUGHT	CARBON-EPOXY
LEAR FAN 2100 WING, FLAPS, AILERONS	LEAR-AVIA	CARBON-EPOXY
XFV-12A WING SKIN	ROCKWELL	CARBON-EPOXY
A-10 SLATS, WING LEADING EDGE	FAIRCHILD	CARBON-EPOXY
F-16 WING LOWER SKIN	VOUGHT	CARBON-EPOXY
EMPENNAGE COMPONENTS		
B-1 HORIZ STABILIZER	GRUMMAN	CARBON-EPOXY
A-4 HORIZ STABILIZER	MCDONNELL DOUGLAS	CARBON-EPOXY
F-5 HORIZ STABILIZER	NORTHROP	CARBON-EPOXY
737 HORIZ STABILIZER	BOEING	CARBON-EPOXY
727 ELEVATOR	BOEING	CARBON-EPOXY
T-38 HORIZ STABILIZER	NORTHROP	CARBON-EPOXY
L2100 HS AND VS	LEAR-AVIA	CARBON-EPOXY
AV-8B HS	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 HS AND VS	MCDONNELL DOUGLAS	CARBON-EPOXY
B-1 VERT STABILIZER	ROCKWELL	CARBON-EPOXY
DC-10 UPPER RUDDER	MCDONNELL DOUGLAS	CARBON-EPOXY
DC-10 VERT STABILIZER	MCDONNELL DOUGLAS	CARBON-EPOXY
L-1011 VERT STABILIZER	LOCKHEED	CARBON-EPOXY
LEAR 2100 HORIZ AND VERT STABILIZER	LEAR-AVIA	CARBON-EPOXY
HiMAT STABILIZER	ROCKWELL	CARBON-EPOXY
F-16 HORIZ AND VERT STABILIZER	GEN DYNAMICS	CARBON-EPOXY
A-10 HORIZ STABILIZER	FAIRCHILD	CARBON-EPOXY
757 RUDDER AND ELEVATORS	BOEING	CARBON-EPOXY
DC-9 RUDDER TAB	MCDONNELL DOUGLAS	CARBON-EPOXY

TABLE 2-2
SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES (CONTINUED)

COMPONENT/ APPLICATION	SOURCE	MATERIAL SYSTEM
FUSELAGE COMPONENTS		
FUTURE FIGHTER FUSELAGE FRAME	BOEING	CARBON-EPOXY
757 AND 767 LANDING GEAR DOORS	BOEING	CARBON-EPOXY
DC-10 NOSE LANDING GEAR DOOR	MCDONNELL DOUGLAS	CARBON-EPOXY
F-15 SPEEDBRAKE	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 SPEEDBRAKE	MCDONNELL DOUGLAS	CARBON-EPOXY
F-18 AVIONICS AND LANDING GEAR DOORS	MCDONNELL DOUGLAS	CARBON-EPOXY
AV-8B FORWARD FUSELAGE	MCDONNELL DOUGLAS	CARBON-EPOXY
AV-8B FUSELAGE CENTER PANEL	MCDONNELL DOUGLAS	CARBON-EPOXY
DC-10 FLOOR BEAMS	MCDONNELL DOUGLAS	CARBON-EPOXY
F-5 SPEEDBRAKE	NORTHROP	CARBON-EPOXY MOLDED
FUSELAGE/WING COMP	NORTHROP	CARBON-EPOXY
B-1 ELECTRONICS BAY DOORS	ROCKWELL	CARBON-EPOXY
B-1 WEAPONS BAY DOORS	ROCKWELL	CARBON-EPOXY
B-1 STRUCTURAL MODE CONTROL VANES	ROCKWELL	CARBON-EPOXY
HiMAT FUSELAGE PANELS	ROCKWELL	CARBON-EPOXY
F-5 FORWARD FUSELAGE	GEN DYNAMICS	CARBON-EPOXY
F-16 FORWARD FUSELAGE	GEN DYNAMICS	CARBON-EPOXY
F-14 MAIN LANDING GEAR DOOR	GRUMMAN	CARBON-EPOXY
A-7 SPEEDBRAKE	VOUGHT AERO	CARBON-EPOXY
LEAR FAN 2100 FUSELAGE	LEAR-AVIA	CARBON-EPOXY
AFT FUSELAGE	VOUGHT	KEV-EPOXY, CARBON-EPOXY

SOURCE: DoD-NASA ADVANCED COMPOSITES DESIGN GUIDE

The regulatory requirements and means of compliance must be defined at the start of a production program to assure a certifiable product and to assess the program certification costs. In general, the basic military specifications and Federal Aviation Regulations apply to the design of composite structures. The Air Force is currently preparing a new damage tolerance specification for composite structure to be used in lieu of the metal structure called for in MIL-A-83444. The FAA has published guidelines for acceptable means of showing compliance with certification requirements for civil aircraft composite structures. The guidelines have recently been revised to reflect the advances made in composite technology, and periodic revisions are expected as the technology matures.

OPERATIONAL ISSUES

The operational issues deal with keeping aircraft in service and are of concern to the airlines and military users. The design features provided by the manufacturer which satisfy the following operational requirements are included in the technical assessment.

- Reliability — Unscheduled time out of service is an extremely high cost factor because of lost revenue and higher capital investment for reserve aircraft. Fleet readiness for military operations is of vital importance to the military user. Data must be provided to ensure that dispatch reliability is equivalent to that achieved with conventional structural materials.

- Maintainability — Composite fuselage structures must be as maintainable as conventional fuselages. The special maintenance requirements for composites must be assessed such as the level of effort required, the equipment needed to perform the maintenance, and special training for personnel. The manufacturer must identify inspection methods and FAA-approved inspection intervals and formulate realistic accept/reject criteria to support the assessment.

- Durability — Durability in a service environment must be proven. Of particular concern is damage to the fuselage shell from hail, lightning, runway debris, bird strike, and abuse by personnel or equipment. Laboratory tests should be supplemented with flight service experience to provide a credible resolution to this issue.

- Repairability — The operators will not accept composite fuselage structure unless practical repair schemes have been demonstrated. Repair of major damage is the foremost concern. Facilities and equipment must be available at a major repair depot and field methods must be described to effect temporary repairs which will allow ferry flight to the depot. The amount and frequency of damage to composite structure and the ease with which it can be repaired should be comparable to the damage and repairability of conventional aluminum fuselage structures since out-of-service time is very costly. This is a major risk item for the aircraft operator with respect to cost and aircraft fleet readiness.

ECONOMIC ISSUES

The cost differentials between the composite structure and a conventional fuselage must be identified since the costs in conjunction with technical and programmatic risks form the basis for tradeoffs against the benefits realized by the reduced structural weight.

- Acquisition Costs — The recurring and nonrecurring design and production costs must be reasonable to allow the aircraft to be competitively priced.

- Life-Cycle Costs — Higher original equipment costs can be offset by lower maintenance costs and increased durability. A flight service evaluation program of representative composite structure exposed to realistic operational usage could provide a credible basis for predicting life-cycle costs. The operator will also face higher insurance costs, a factor in life-cycle costs, unless evidence is provided to insurance companies that their risk is not increased. Their concern will be that composites not be more susceptible to damage or cost more to repair than conventional structures.

- Facilities and Equipment — The methods for the original manufacture, maintenance, inspection, and repair are strongly influenced by the facilities and equipment available to do the job. These methods affect the design as well and are important drivers for life-cycle costs.

PROGRAMMATIC RISK ISSUES

This set of issues deals with the probability of success or failure to derive the benefits (weight savings) from composite fuselage structures within the cost and schedule framework established to accomplish the task. The design, manufacture, and test of large-scale representative composite fuselage structures and the subsequent design, manufacture, and flight service evaluation of large composite fuselage panels should provide visibility of acceptable risk levels.

Data Base

Technical and programmatic risks are affected by the size and quality of the design data base available for the new composite fuselage design. A data base for in-house design of conventional fuselage structure for large transport aircraft has been accumulated over the years from development and qualification testing of many aircraft models including the DC-6/7, DC-8, DC-9/MD-80, DC-10/KC-10, and other military aircraft, and supplemented by data from NASA and various industry and government sources. The data base includes test data correlated with analytical predictions, data on the development of analytical methods, manufacturing and processing technology, FAA-approved design allowables, and a library of technical manuals and standards.

A new model usually requires an expansion of the data base to account for new technology, variation in design features, and regulatory changes. The data base expansion represents only a modest investment in time and money compared to the generation of new data required for a representative composite fuselage construction subjected to the large transport criteria and loads. The composite fuselage data base will be started from generic data accrued from NASA ACEE and other government-funded composite structure programs and from in-house composite development activities.

The DC-10 can be used to illustrate the application of a data base for a large transport aircraft (see Figure 2-1). The preliminary design configurations were evaluated using DC-8/9 data from more than 2,000 static, fatigue, and fail-safe tests. Extensive fatigue and fail-safe tests were then conducted on specimens with representative DC-10 structural features and loads, as shown in Figure 2-2. Full-scale static proof load tests were conducted on the second flight article, and the fourth production airframe was dedicated for fatigue and fail-safe verification tests.

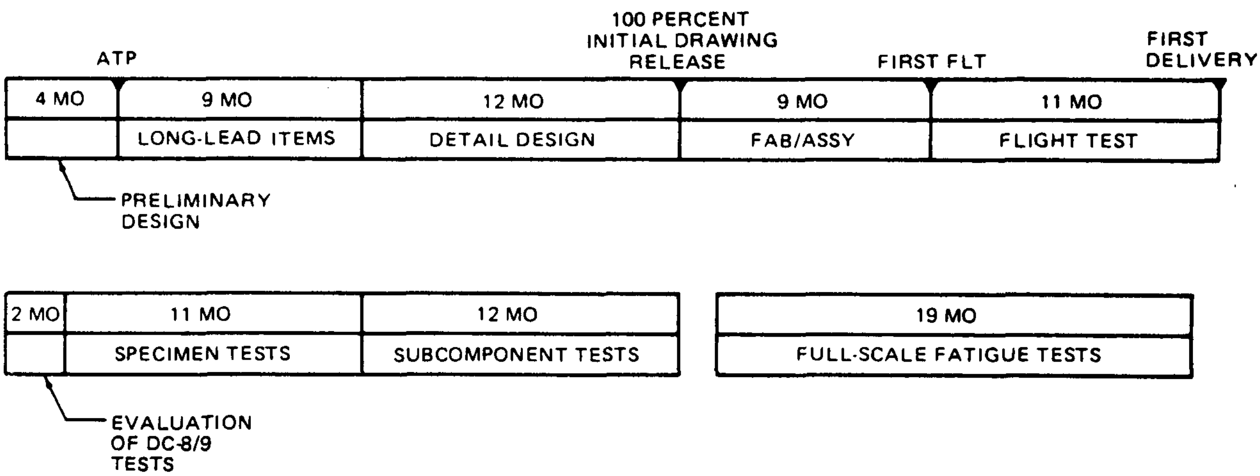


FIGURE 2-1. DC-10 STRUCTURAL DESIGN AND TEST SCHEDULE

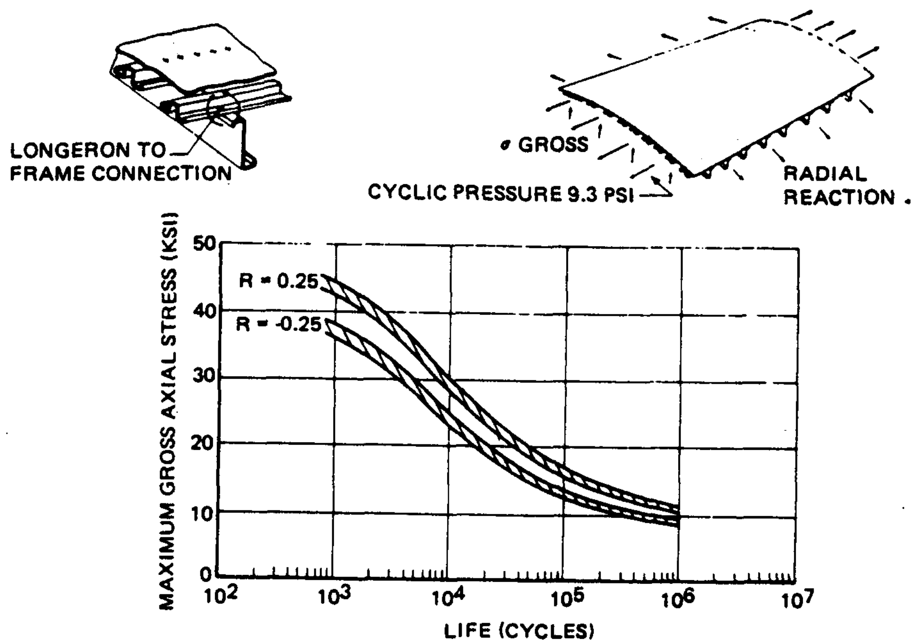


FIGURE 2-2. FUSELAGE DEVELOPMENT COMPONENT TEST RESULTS

Similarly, a data base that includes representative large transport composite fuselage structure must be available at the onset of a new production program to serve the following purposes:

1. To support the initial commitment by the manufacturer and operator during the specification definition phase.

2. To have data immediately available for the design integration which precedes the detail design. Otherwise, typical development schedules must be extended to allow for time to conduct development tests.

Weight Estimates

The attainment of estimated weight savings based on a conceptual design is a risk element. More data are required to establish that predicted weight savings for large transport composite fuselage structure are valid. The weight savings will be adversely affected by the following design considerations.

- Design strain levels may need to be reduced to integrate the durability, damage tolerance, lightning and crashworthiness protection, and acoustic attenuation features.
- Weight tradeoffs may be needed to establish cost-effective production methods for composite fuselage structure.
- Balanced layups and practical orthotropic ply orientations may add weight to the design.

Conversely, advancements in toughened resin systems, improved analysis methods, and an expanded technology data base may result in higher weight savings than conservatively predicted on the basis of a limited data base and experience. There is a risk that the benefits of composite fuselage structure could be underestimated and unjustifiably rejected for production.

Sufficient data are available to confirm the validity of the weight-estimating methodology for composite structure. The variance of structural weight will result from variables in the design integration rather than from weight-estimating techniques.

Schedules

A low risk must be associated with composite fuselage production schedules. Contracts for new airplanes include late delivery clauses to the effect that airlines must be recompensed to offset the cost of providing alternate airlift capability and the loss of revenue generated by late delivery.

Production schedules are set by competitive market forces and have little room to accommodate unforeseen problems. Figure 2-3 shows a typical schedule for the development of a large transport aircraft. The fuselage assembly must be completed and be ready to be joined to the wing structure 19 months after the decision to go ahead is made. Unless a very high confidence level exists, one would also need to carry forward a conventional design to safeguard against the risk of encountering unacceptable schedule delays. This approach would increase development costs and preclude the down-sizing of lifting surfaces and engine thrust for enhanced weight savings.

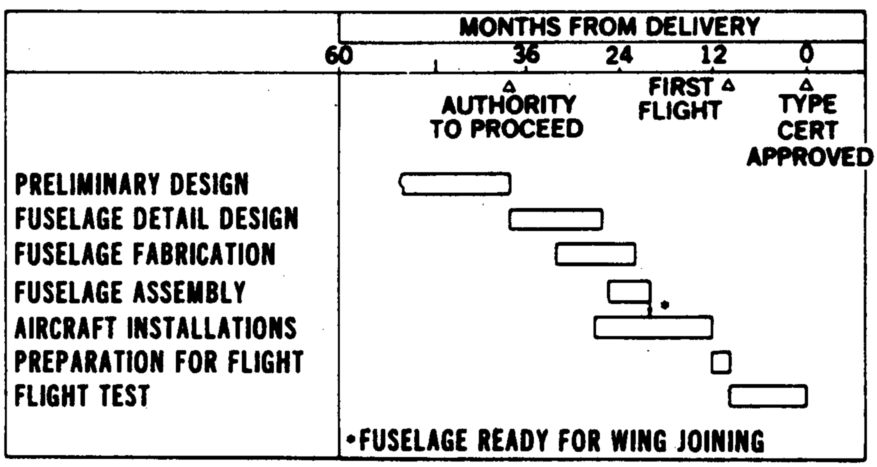


FIGURE 2-3. TYPICAL SCHEDULE DATA FOR A NEW PRODUCTION AIRPLANE

Cost Estimates

The cost estimates for composite fuselage structures for large transport aircraft are presently based on conceptual design studies. The production costs will vary significantly with design features as well as with well-conceived fabrication and assembly methods which utilize automation to avoid labor-intensive cost centers.

The uncertainty of the cost estimates is a high-risk concern. A composite fuselage technology program which includes the design, manufacture, and test of full-scale composite fuselage structures of large transport aircraft is essential to improve confidence in cost predictions.

SECTION 3
TECHNOLOGY ASSESSMENT

The design, manufacture, and flight service of a large composite fuselage structure requires a more mature technology base than is presently available for the industry to risk the investment of capital and resources needed for such a large project. This section identifies the technology issues that will influence a decision to make the commitment to manufacture composite fuselage structure, assesses the state of the art for each issue, identifies the areas of technology that appear to be lacking, and suggests a course of action to resolve the technology gaps which are not expected to be resolved by other research and development programs.

The technical issues that must be addressed are listed in Table 3-1. A priority has been assigned to each issue based on a selection criterion that evaluates when the technology is needed, the length of time needed to develop the technology, the complexity of problem resolution, and the extent to which the technology affects cost, structural weight, and structural integrity.

The following key technical issues will require lengthy development activity to supply data needed at an early date for technology integration during the preliminary design: damage tolerance, durability, impact dynamics, manufacturing methods, and large cutouts and joints. These technologies strongly influence the materials to be selected, the structural design features, and the manufacturing methods to be employed. Structural integrity, weight savings, and cost depend strongly on how successfully these technologies are developed and integrated into the design.

Acoustics has been added to the key issue list, even though it has a low priority score in Table 3-1. The attenuation of cabin noise is directly related to the mass of the fuselage shell, and higher noise levels within the fuselage cabin would not be acceptable to the airlines. The weight savings achieved with the composite materials could be nullified by the acoustic treatment added to maintain the low cabin noise levels and the primary driver for utilizing composite materials for the fuselage structure would cease to exist. Cargo transport aircraft would be less affected by the acoustics issue.

The electromagnetic effects issue is assessed in Table 3-1 on the basis of its influence on the structural design. It would score higher as a key issue if the effects on electrical and avionics systems were included in the assessment. A parallel development program is recommended to determine the weight penalties for shielding and avionics/electrical system designs which account for the lower conductivity of composite fuselage shell structure.

DAMAGE TOLERANCE

The design of transport aircraft fuselage structure must be damage-tolerant to ensure structural integrity and passenger safety. For civil transport aircraft, the level of damage tolerance required by the FAA is specified in FAR 25.571. The foremost damage tolerance issue for conventional metallic structure is the detection and the slow growth or arrestment of fatigue cracks to assure fail-safe residual strength between inspection intervals. The fatigue cracks generally initiate at fastener holes or other points of stress concentration in the thin fuselage skins. Current design practice makes extensive use of empirical data and experience to develop analytical methods and identify fatigue-sensitive areas. The importance of test data and service usage is emphasized by FAA Advisory Circular 20-107A.

Conventional metallic structure is often designed on the basis of slow damage growth so that an intrinsic flaw is prevented from reaching a critical size within specified inspection intervals using prescribed inspection procedures. Although this approach works well in general, relatively small initial flaws have precipitated sudden catastrophic failures in some of the more brittle, high-heat-treated metal alloys. This same brittle behavior is exhibited by conventional composite materials with rapid damage growth characteristics for which the slow-growth inspection approach is unacceptable.

The customary approach for present-day composite parts is to maintain a limit on design strain levels so that damage will not propagate. This methodology has been generally successful when combined with a fail-safe structural arrangement, but such restrictions could nullify the potential weight savings of composite structures, particularly for strength-critical applications. Further, the design complexity of primary fuselage structure is such that premature failures may occur due to secondary out-of-plane loads which produce critical interlaminar stresses, sometimes at unexpectedly low load levels. This inherent weakness of composite materials is a primary concern when the materials are subjected to out-of-plane or interlaminar forces.

The development of a minimum-weight damage tolerant design for composite fuselage structure will require the successful completion of several tasks. First, the types of damage or flaws that may result from manufacturing deviations or in-service use must be identified as well as the likelihood that a given type and size of damage may occur throughout the life of the aircraft. This is accomplished by a complete evaluation of the potential damage sources that are present from the initial stages of fabrication through the lifetime of the aircraft.

Having established these damage probabilities, each category of damage and flaw must be evaluated regarding its level of detectability during manufacture and in service. The inspection intervals required to detect a specific flaw and flaw size must be determined based on the capability of the specified nondestructive inspection equipment. Finally, a structural test program will be required to establish a no-growth behavior or to establish damage growth rates, the residual strength of the damaged structure, and the inspection intervals required for continued safe flight. The test data should be correlated to verify and improve the accuracy of analysis methods.

The task of identifying the types of damage to be considered for a composite fuselage shell is based primarily on the past experience of the manufacturers and operators for both conventional fuselage structure and existing components made with advanced composites. Table 3-2 presents a list of potential flaws or damage grouped under material and manufacturing flaws, in-service damage, and battle damage for military aircraft.

The potential sources of damage during manufacture include flaws inherent in the material processing and fabrication procedures through those occurring during the assembly process. The key parameter to

TABLE 3-2
DAMAGE AND FLAW IDENTIFICATION

MATERIAL AND MANUFACTURING FLAWS
• RESIN OR FIBER-STARVED AREAS
• MISCOLLIMATION
• DELAMINATION
• FIBER BREAKS
• PLY GAPS
• VOIDS
• IMPROPER CURE
• WRONG STACKING SEQUENCE
• WRONG PLY COUNT
• SURFACE DEFECTS/DAMAGE
• WRINKLES AND WAVINESS
• CONTAMINATION
• IMPROPER SURFACE PREPARATION
• ADHESIVE-STARVED AREAS
• IMPROPER FASTENER INSTALLATION
• MISCURED ADHESIVE
• IMPROPERLY DRILLED HOLES
• HEAT-DAMAGED MACHINED EDGES
IN-SERVICE DAMAGE
• LOW-ENERGY IMPACT CAUSED BY GROUND HANDLING, MAINTENANCE OPERATION, STORAGE, TAXIING, FLIGHT OPERATION, ETC.
• LIGHTNING
• MOISTURE AND HEAT
• FATIGUE DELAMINATION
• CORROSION
• THROUGH-THICKNESS CRACKS (HIGH-ENERGY IMPACT)
• DISCRETE SOURCE DAMAGE
BATTLE DAMAGE
• SINGLE AND MULTIPLE PENETRATORS
• PENETRATORS AND BLAST
• STRUCTURAL HEATING
• THERMAL (BURN-THROUGH)

be determined is the maximum flaw size that remains undetected throughout the production process utilizing the prescribed visual and nondestructive inspection methods. These undetected flaws must be assessed for their criticality to the structure by use of established analytical procedures and mechanical test data, the results of which must be accounted for in the structural design.

Of the various types of in-service flaws that have been identified, impact damage is of the most concern. Sources of impact damage include such natural phenomena as hail, bird strike, and gravel or debris along the runway. In addition, damage may occur as a result of maintenance mishaps such as dropped tools or impact from a service vehicle. Damage tolerance assessments have not been required for low-energy impact on aluminum structures because of the material's inherent tolerance to such hazards. However, composites are sensitive to impact damage and subsequent interlaminar failure, the design should be tolerant of nonvisual damage, and damage limits and repair procedures should be established for visual damage.

The same level of safety regarding bird strike as defined in FAR 25 will be required for a composite fuselage shell. This topic is discussed later as a separate issue.

TABLE 3-1
TECHNICAL ISSUES FOR COMPOSITE FUSELAGE UTILIZATION

TECHNICAL ISSUE	CRITERIA FOR SCORING: TIMING — NEED DATE (WEIGHTING FACTOR 4)	TIMING — TIME REQUIRED TO DEVELOP (4)	URGENCY OF RESOLUTION — COMPLEXITY AND RISK OF FAILURE (8)	URGENCY OF RESOLUTION — EFFECT ON STRUCTURAL INTEGRITY (3)	URGENCY OF RESOLUTION — EFFECT ON STRUCTURAL WEIGHT (3)	URGENCY OF RESOLUTION — EFFECT ON MANUFACTURING COSTS (3)	URGENCY OF RESOLUTION — EFFECT ON OPERATIONAL COSTS (FLIGHT SERVICE) (3)	TOTAL WEIGHTED SCORE	FINAL RANKING
(WEIGHTING FACTOR → / SCORE → / WEIGHTED SCORE →)									
DAMAGE TOLERANCE	10 PHASE I [40]	10 >2 YEARS [40]	10 INTEGRATES WITH MATERIAL DEVELOPMENT, DESIGN CONCEPTS AND SIZING, NDI, ACCEPT/REJECT CRITERIA, AND MANUFACTURING METHODS. MANY TRADEOFFS [80]	10 MUST SATISFY FAR 25.571 OR MILITARY DAMAGE TOLERANCE REQUIREMENTS. [30]	10 HYBRID DESIGNS AND LOW STRAIN LEVELS COULD INCREASE WEIGHT. [30]	10 COULD IMPOSE MANUFACTURING METHODS THAT ARE MORE LABOR-INTENSIVE. [30]	10 COULD INCREASE INSPECTION COSTS AND TIME OUT OF SERVICE. [30]	280	1
DURABILITY	8 PHASE I [32]	10 >2 YEARS [40]	10 MUST PROVIDE DURABILITY FOR LIGHTNING STRIKE, FOD, HAIL, PERSONNEL AND EQUIPMENT ABUSE. INTERLAMINAR TENSION AND SHEAR FORCES ARE DESIGN PROBLEMS. [80]	10 COULD REDUCE RESIDUAL STRENGTH. [30]	10 SUSCEPTIBILITY TO DAMAGE COULD REQUIRE LOWER DESIGN STRAIN LEVELS AND PROTECTIVE COATINGS. [30]	2 MORE MECHANICAL ATTACHMENTS MAY BE REQUIRED TO MINIMIZE PEEL FORCES. [6]	10 RECURRING DAMAGE WILL BE COSTLY TO INSPECT AND REPAIR. [30]	248	2
IMPACT DYNAMICS	9 PHASE I [36]	10 > 2 YEARS [40]	10 ANALYTICAL PREDICTIONS NOT RELIABLE. REQUIRES EVENTUAL TEST WITH SIMULATED MASS, VELOCITY, AND CRASH SITE. ADEQUATE LEVEL MAY BE DIFFICULT TO ACHIEVE. [80]	10 BRITTLE BEHAVIOR COULD CAUSE PROTECTIVE SHELL TO SHATTER UPON IMPACT DURING OFF-RUNWAY INCIDENT. [30]	5 DESIGN SOLUTIONS FOR CRASHWORTHINESS MAY DICTATE INEFFICIENT DESIGN CONCEPTS. [15]	8 DESIGN SOLUTIONS (SUCH AS STITCHING) MAY PRECLUDE LOW-COST MANUFACTURING METHODS. [24]	5 INSURANCE COSTS COULD BE HIGHER. REPAIR COSTS COULD BE HIGHER. [15]	240	3
MANUFACTURING METHODS	8 PHASE I [32]	10 > 2 YEARS [40]	10 INTEGRATES WITH DESIGN, MATERIALS AND PROCESSES, FACILITIES, AND EQUIPMENT. LOW-COST, HIGH-QUALITY GOALS. MANY TRADEOFFS. [80]	8 MUST HAVE METHODS THAT ENSURE REPEATABLE PRODUCTION OF HIGH-QUALITY STRUCTURE. [24]	7 LACK OF MANUFACTURING TECHNOLOGY COULD INCREASE WEIGHT TRADEOFFS. [21]	10 LACK OF TECHNOLOGY COULD RESULT IN SELECTION OF MORE EXPENSIVE METHODS. [30]	2 CHOICE OF MANUFACTURING METHOD COULD INCREASE REPAIR COSTS. [3]	230	4
ELECTROMAGNETIC EFFECTS	8 PHASE I [32]	5 < 1 YEAR (USE CONTEMPORARY CONDUCTIVE MATERIALS) [20]	10 INTEGRATES WITH BASIC MATERIAL SYSTEMS, DAMAGE TOLERANCE, DURABILITY, PAINTING, AND MANUFACTURING METHODS. NEEDS PARALLEL PROGRAM STARTED FOR ELECTRICAL/AVIONICS DESIGN AND PROTECTION. [80]	10 LIGHTNING STRIKE DAMAGE COULD BE WIDESPREAD AT VISUAL AND NONVISUAL LEVELS OF DELAMINATION. FLIGHT SYSTEMS COULD BE UNSAFE. [30]	5 MODERATE WEIGHT PENALTIES EXPECTED FOR LIGHTNING PROTECTION. SHIELDING WEIGHT PENALTIES NOT ESTABLISHED BY CFDP. [15]	3 VERY LITTLE IF CONDUCTIVE OUTER PLY IS DESIGN CONCEPT. SURFACE PREPARATION FOR PAINTING COULD BE MORE COSTLY. [9]	4 HIGHER INSPECTION AND REPAIR COSTS IF LOW-COST REFURBISH PROTECTION METHOD USED AFTER STRIKE. [12]	198	6
LARGE CUTOUTS AND JOINTS	9 PHASE I [36]	10 > 2 YEARS [40]	8 NEED TO PREDICT INELASTIC LOAD REDISTRIBUTION AROUND CURVED REINFORCED CUTOUTS AND IN JOINTS. EMPIRICAL ANALYSIS METHODS TO BE DERIVED AND CORRELATED. [64]	10 UNDERSTRENGTH DESIGN COULD RESULT IN FAILURE OF PRESSURIZED FUSELAGE. [30]	4 SMALL PERCENTAGE OF STRUCTURAL WEIGHT IS AFFECTED. [12]	4 DESIGN REINFORCEMENT MAKES LAYUP AND ASSEMBLY MORE COMPLEX IN LOCAL AREAS. [12]	2 DESIGN REINFORCEMENT MAKES INSPECTION AND REPAIR MORE DIFFICULT IN LOCAL AREAS. [6]	200	5
NONDESTRUCTIVE INSPECTION	5 PHASE II [20]	4 DURING CFDP PHASE 2 [20]	8 DEPENDS ON GENERIC NDI EQUIPMENT DEVELOPMENT. MUST BE PROVEN TO BE EFFECTIVE DURING CFDP. [64]	10 UNDETECTED FLAWS/DAMAGE COULD PRECIPITATE STRUCTURAL FAILURE. [30]	8 INSENSITIVE NDI METHODS TO FIND AND QUANTIFY FLAWS WILL REDUCE DESIGN STRAIN LEVELS. [24]	3 NDI IS LABOR-INTENSIVE AND CAN INFLUENCE MANUFACTURING METHODS. [9]	5 IMPROVED METHODS WILL REDUCE INSPECTION COSTS. [15]	182	7
BIRD-STRIKE RESPONSE	4 PHASE II [16]	4 DURING CFDP PHASE 2 [16]	4 DIFFICULT TO DEVELOP CREDIBLE ANALYTICAL METHODS. VERY LITTLE DESIGN DATA AVAILABLE. WILL REQUIRE BIRD SHOT TESTS USING EXISTING FACILITIES. [32]	10 CREW AND FLIGHT-CRITICAL SYSTEMS MUST BE PROTECTED FROM BIRD PENETRATION. MUST HAVE RESIDUAL STRENGTH TO RESIST DISCRETE DAMAGE. [30]	3 THE FRONTAL AREA OF THE FUSELAGE MAY NEED REINFORCEMENT OR ALUMINUM STRUCTURE. [9]	1 COST OF HEAVIER STRUCTURE IN FORWARD FUSELAGE. [3]	2 HIGH REPAIR COSTS BUT OCCURRENCE INFREQUENT. [6]	112	12
FLAW/DAMAGE ASSESSMENT AND DISPOSITION	5 PHASE II [20]	5 DURING CFDP [20]	5 DEPENDENT UPON DESIGN STRAIN LEVELS, DAMAGE TOLERANCE, AND MANUFACTURING METHODS. [40]	8 ACCEPTABLE FLAWS MUST NOT GROW IN SERVICE AND MUST HAVE RESIDUAL STRENGTH FOR ULTIMATE DESIGN LOADS. [24]	3 TRADEOFF BETWEEN ACCEPTANCE OF FLAWS AND DESIGN STRAIN LEVELS. [9]	3 LACK OF TECHNOLOGY INCREASES REJECTION RATE AND REWORK. [9]	9 DATA NEEDED TO ASSURE STRUCTURE IS FLIGHTWORTHY AND TO FACILITATE NECESSARY REPAIRS. [27]	149	8
POSTBUCKLING	8 PHASE I [32]	5 1 YEAR [20]	5 SEMIEMPIRICAL ANALYSIS METHODS NEED TO BE ADOPTED AND CORRELATED FOR COMPOSITE FUSELAGE STRUCTURE. [40]	5 REPEATED BUCKLING COULD CAUSE DELAMINATION AND LOSS OF RESIDUAL STRENGTH. [15]	3 QUANTITATIVE DATA TO ALLOW POSTBUCKLED STRUCTURE WILL ALLOW MORE WEIGHT-EFFICIENT DESIGN. [9]	0 NEGLIGIBLE. [0]	1 HIGHER REPAIR COSTS IF DELAMINATIONS OCCUR. [3]	119	11
ACOUSTICS	8 PHASE I [32]	6 1-2 YEARS [24]	6 INFLUENCED BY DESIGN FEATURES AND STRUCTURAL MASS PLUS SOUND-ABSORPTION MATERIAL. DECISIONS WILL BE BASED ON TEST DATA AND PHASE 2 FULL-SCALE MEASUREMENTS. [48]	0 PASSENGER-COMFORT ISSUE [0]	10 SHOW-STOPPER (1). IF MASS IS THE CRITICAL PARAMETER FOR NOISE ATTENUATION, THEN WEIGHT SAVINGS WILL BE LOST. [30]	2 MINOR INFLUENCE ON MANUFACTURING COSTS. [6]	0 SHOULD NOT AFFECT MAINTAINABILITY. [0]	140	9 (1)
MAJOR REPAIRS	3 PHASE III [12]	4 DURING CFDP PHASE 2 [16]	4 DEPENDENT UPON DAMAGE TOLERANCE AND JOINT TECHNOLOGY DEVELOPMENT; ALSO CONFIGURATION-DEPENDENT. [32]	8 REPAIRS MUST SATISFY DAMAGE TOLERANCE AND ULTIMATE STRENGTH REQUIREMENTS. [24]	1 ASSUMES DESIGN STRAIN LEVELS SET FOR DAMAGE TOLERANCE CRITERIA WILL PERMIT BOLT BEARING/BY-PASS STRESSES OF REPAIR. [3]	2 REPAIRABILITY MAY EXCLUDE SOME LOW-COST METHODS. [6]	10 BRITTLE FAILURES COULD CAUSE MORE WIDESPREAD DAMAGE THAN DUCTILE MATERIALS. TIME-OUT-OF-SERVICE COSTS COULD BE HIGH. [30]	123	10
THERMAL COMPATIBILITY	3 PHASE III [12]	3 DURING CFDP PHASE 2 [12]	5 DESIGN AND SIZING FOR LARGE FLIGHT SERVICE EVALUATION PANEL WILL BE BY ANALYSIS. IN-FLIGHT STRESSES AT −65°F WILL BE MEASURED AND CORRELATED WITH ANALYSIS. [40]	8 COULD CAUSE FAILURE IF THERMAL STRESSES ARE NOT ACCOUNTED FOR IN DESIGN. [24]	1 LOCAL REINFORCEMENT. [3]	0 NEGLIGIBLE. [0]	3 THERMALLY COMPATIBLE MATERIALS MUST BE USED OR THERMAL STRESSES ACCOUNTED FOR IN REPAIR DESIGN. [9]	100	13

NOTE (1) MUST BE GIVEN HIGHER PRIORITY TO ASSURE WEIGHT SAVINGS ARE NOT NULLIFIED.

The potential for encountering a hailstorm with multiple impacts of varying mass and velocity must be considered. Hail damage to transport aircraft is a common occurrence, and the relatively low impact resistance of composite materials will require that a hail impact criterion and damage limits be established. The impact energy levels for an aircraft on the ground during a hailstorm are much lower than in flight. The composite fuselage shell design criteria include provisions that inspection and repair are not required for ground hail impact. The possibility of hail damage while in flight is a serious damage tolerance concern. Conventional structures have suffered extensive in-flight hail damage, particularly in forward-facing regions, and a minimum criterion for composite structure should require that structural failure of the pressurized cockpit enclosure does not occur to ensure the safety of the occupants. The probability and severity of these hail encounters can be estimated from historical data (Reference 4).

The potential damage from runway debris during takeoff and landings or service incidents such as tool drops can be statistically predicted using historical data and typical values for parameters such as object size, weight, velocity, and drop height. The potential damage associated with discrete sources such as engine fragments or shrapnel from tire failures can be estimated in a similar fashion.

The effects of temperature, humidity, and other environmental factors that degrade material properties must be considered in the damage tolerance assessment. The potential for structural damage due to lightning strike must be included in the design phase. Recent testing of large wing-type composite structure has shown that substantial damage can occur for structure much thicker than fuselage panels (Reference 5) and the recurrent incidents involving lightning strike on large transport aircraft will require that a criterion be established.

For cases of undetectable damage, the structure must have the residual strength to carry ultimate loads. The reduction in strength caused by such flaws must therefore be included in the basic evaluation of static strength, and a further criterion requires that this damage will not grow under repeated service loads. Detectable damage must be categorized as a function of the likelihood of occurrence, criticality to the structure, and the level of inspection required to detect the damage. Damage resulting from an obvious discrete source will demand that the aircraft withstand so-called get-home loads for safety of flight.

Improvements in state-of-the-art damage tolerance technology have prompted modifications to the requirements of regulatory agencies, such as the new FAA Composities Advisory Circular (No. 20-107A.) An Air Force-sponsored committee is working on definitions of inspectable and noninspectable flaw/damage assumptions and inspection schedules based on the degree of inspectability, typical inspection intervals, and load magnification factors in its draft of new military specifications for damage tolerance of composites.

The development of effective methods for damage tolerance analysis is required for the application of composite materials to transport fuselage structure. The anisotropic and nonhomogeneous characteristics of composite materials do not allow direct application of classical fracture mechanics to damage tolerance and fatigue life analysis of flawed laminates. While theoretical methods involving the prediction of infinitesimal microfailure from micromechanics analysis have been considered, the establishment of empirical formulas guided by classical fracture mechanics seems a very practical alternative.

Such an approach was used to provide the analytical prediction of Figure 3-1 for through-crack stress intensities under tension load (Reference 6). More importantly, work must be done to develop predictive capabilities for compression failure modes, particularly when the failures are dominated by critical interlaminar stresses as in delamination failures or induced transverse tension failures.

Without accurate analytical methods, extensive structural testing will be required to demonstrate the damage tolerance capabilities of selected design concepts. The effectiveness of enhanced damage-tolerant designs using softening strips, zebra cloth, stitching, and the like have been examined in several research and development programs. These and other concepts must be tested for application to fuselage structure.

Material system manufacturers are attempting to develop new, toughened resin systems in order to intrinsically increase the interlaminar strength of cross-ply laminates for improved damage tolerance.

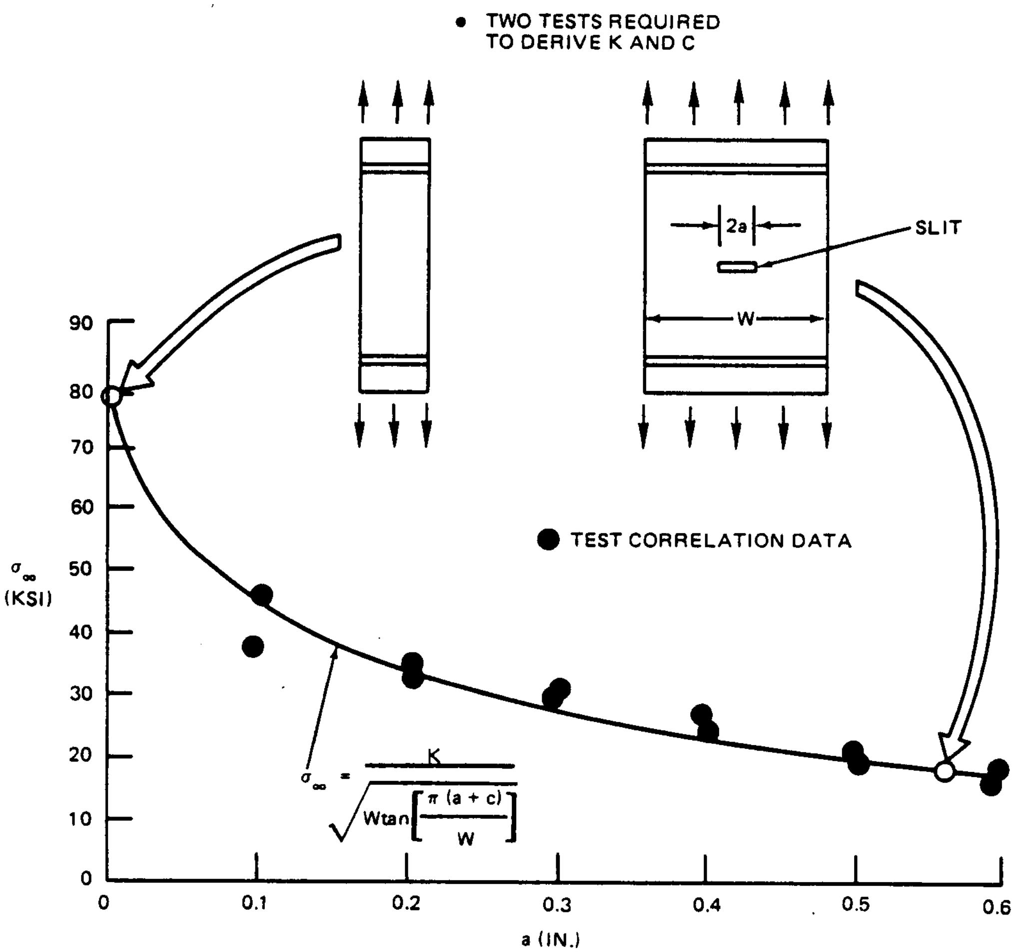


FIGURE 3-1. DERIVATION AND VALIDATION OF THEORETICAL CURVE FOR TENSION FAILURE

DURABILITY

The durability of fuselage structure may be defined as the ability to maintain structural integrity throughout the intended service life of the aircraft with reasonable maintenance costs.

With conventional metal structures, durability concerns usually include maintaining adequate fatigue strength and providing sufficient resistance to the various forms of corrosion. Many years of flight service evaluation have provided the manufacturers with an extensive data base on durability. Appropriate working stress levels have been established for aluminum alloys and adequate fatigue life can be maintained in these materials by minimizing points of high stress intensity in structural design. New alloy forms, processing methods, and protective surface coatings have improved resistance to metallic corrosion.

Composite materials present a different set of durability concerns. While composite structures have proven to be highly resistant to in-plane fatigue loading and are not susceptible to the corrosion problems of metals, the failure modes listed in Table 3-1 and an inherent lack of resistance to impact damage suggest that a new set of durability criteria will need to be established.

Every reasonable effort should be made to qualify structural components under the damage tolerance provisions. However, where single-load path or inaccessible and uninspectable blind areas exist and suitable fail-safe damage tolerance cannot practically be provided, the structure should be shown to comply with the established durability (safe-life) requirements to ensure its continued airworthiness. Where possible, the structure should be designed for safety through damage tolerance and for economy through durability.

This approach implies that two overall objectives must be met. First, the structure must be designed so that when subjected to the conditions of normal operation, maintenance, inspection, and repair, the life of the fuselage shell should not be limited by degradation of structural integrity. Second, the day-to-day costs of maintenance and repair including ground service time required should not exceed those of present-day aircraft.

To demonstrate the retention of material strength and integrity after the structure is subjected to the repeated loads and long-term environmental exposure, a comprehensive series of element, subcomponent, and component tests must be conducted to establish fatigue scatter and the effects of environmental exposure for representative fuselage structure. These must be followed by full-scale fatigue tests which include the appropriate spectrum of axial load, shear, and cabin pressure together with environmental effects.

The primary durability concern for a composite fuselage shell is the vulnerability of the basic material to impact damage or any damage source which may create delaminations. Other concerns include the ability to resist out-of-plane forces, the long-term effects of the environment, and the imperfections resulting from deviations in material processing and manufacturing.

The impact damage issue is critical because of the countless sources of impact that may be encountered throughout the life of the fuselage shell. Potential sources of damage include hailstones, runway debris, tire or wheel fragments, personnel and equipment abuse, and other sources of foreign object damage. Impact phenomena data that are evaluated to establish the necessary criteria for damage tolerance must also be considered from a durability or economic standpoint.

The frequency of occurrence of the various types of impact damage must be established as a function of location on the fuselage shell, possibly requiring some type of flight service evaluation. The degree of damage suffered by the composite structure must be determined for the anticipated damage sources and associated impact energies. This will require extensive structural testing in the absence of accurate methods for analytical predictions. The residual strength of the impacted structure must be determined, as previously discussed with regard to damage tolerance.

Experimental results indicate that composite laminates can be sensitive in fatigue to tension-compression or compression-compression cyclic loading (Reference 7), with the primary failure mechanism being progressive delamination leading to fiber buckling and eventual laminate failure. Further, some of the more complex design features of fuselage structure may induce out-of-plane forces that are not readily identified by present analytical techniques. These phenomena, when combined, can cause critical levels of interlaminar stresses, resulting in unexpected fatigue failures. Thus, the designer must be careful to avoid designs that induce loads that are normal to the plane of the laminate. The inherent interlaminar strength of the basic material system must be determined by test for representative layup sequences when the interlaminar forces cannot be avoided.

The effects of long-term exposure to the environment on the strength of composite structure can be determined only through a controlled evaluation of in-service performance over an extended time period. In current programs, production composite components have shown good retention of structural integrity in their service to date (Reference 8), but such results are limited. Fatigue tests should be conducted to simulate these effects on representative fuselage structure.

Large transport aircraft often encounter storms in flight or on the ground which may result in the structure being struck by lightning. The resistance of composite fuselage structure to damage must be determined under these conditions and included in the durability assessment.

Defects can also be introduced in the composite structure during the manufacturing process and degrade the durability of the structure. Data should be developed to establish durability as a function of the product quality level. Other data should be developed to establish the relationship between quality and manufacturing cost. Together, the data will permit a tradeoff between structural weight and manufacturing costs since, for a given durability criterion, the design strain levels are influenced by the structural quality of the product. Additionally, the data will assist in the engineering disposition of manufacturing deviations, as discussed later.

The durability technology gap for composite structures will be partially closed by the results of other composite technology programs, particularly the question of strength degradation due to long-term environmental exposure and the reduction of stiffness due to cyclic loads. Some contribution will be made to the interlaminar shear fatigue strength data base and the effects of defects on composite fuselage durability. Plans must be made to resolve the impact damage issue. The composite fuselage technology program will require provisions for acquiring durability test data, improving the ability to design durable structure, and demonstrating durable qualities by means of a full-scale fatigue test and an in-service flight evaluation.

IMPACT DYNAMICS

Transport aircraft are designed to ensure that occupants have every reasonable chance of escaping serious injury under realistic and survivable impact conditions. In designing for this criterion, structural evaluations are usually made by analysis supported by test evidence. Structural tests can range from the element level to full-scale, or test evidence may be provided by related service experience. Analytical comparisons with conventional structure may be used if applicable. Beyond the issue of occupant safety, the level of damage suffered by the shell itself under survivable conditions is an economic concern involving the extent of repairs that will be required to return the aircraft to service.

An impact-survivable accident may be defined as an accident in which none of the occupants receives serious injuries as a result of impact forces imposed during the crash. There are three types of impact scenarios in which passenger safety is a concern; (1) when the aircraft is descending on approach before reaching the airport, (2) when the aircraft touches down on or near the runway and overruns or veers off after touchdown, and (3) when the aircraft is nearing rotation or after liftoff before the landing gear or flaps are retracted. The last scenario usually includes a tire or engine failure.

A substantial data base has been gathered on past accidents involving transport aircraft (Reference 9). While these incidents are confined to conventional metallic fuselage structure, a great many parameters — airspeed or ground speed, impact locations, impacted objects, sliding/skidding distance, and drop distance, as well as type of aircraft, gross weight, and number of passengers — have been documented for most cases and are invaluable for the crashworthiness evaluation of composite fuselage structure.

The concern over impact dynamics for a composite fuselage structure is focused on the nonductile or low strain-to-failure characteristics of the basic material. Composite materials do not possess the elastic-plastic stress-strain properties of the relatively ductile aluminum alloys presently used in fuselage structure. This difference in performance results in the dramatic reduction in specific energy absorption from aluminum alloy to carbon-epoxy, as shown in Figure 3-2. This nonductile behavior gives rise to the brittle modes of failure typically associated with composite structures. These phenomena suggest several specific concerns about the performance of a composite fuselage shell under impact conditions.

If the lower energy absorption characteristics of composites as compared to aluminum translates into similar relative performance for full-scale structure, higher load factors on the occupants and other mass items may result for a given impact condition. The plastic deformation behavior of aluminum structure provides a "softening" effect, resulting in a lower rate of deceleration for a given mass and impact velocity than would be the case for a linear-elastic structure of equal stiffness. For a composite fuselage shell to exhibit similar characteristics, innovative design concepts must be developed to allow progressive failures despite unforgiving material properties.

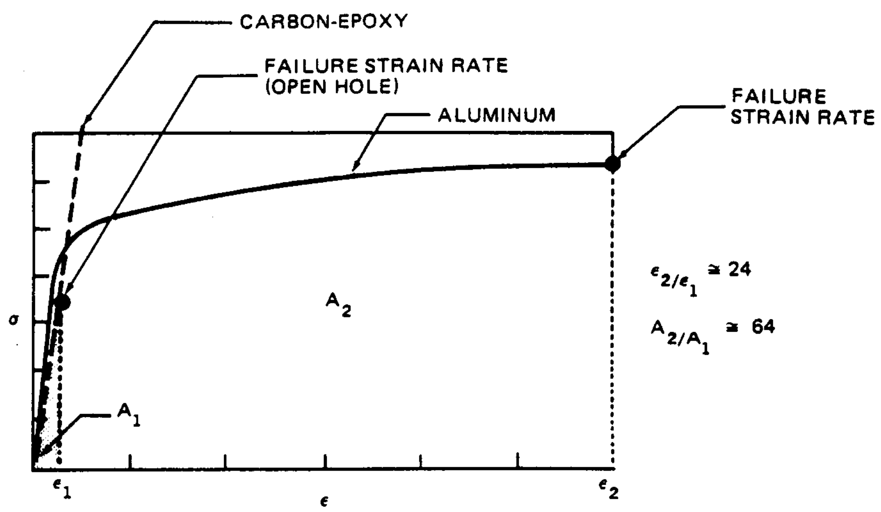


FIGURE 3-2. STRESS-STRAIN RELATIONSHIP OF CARBON-EPOXY AND ALUMINUM MATERIALS

The lack of ductile material properties presents another problem related to the failure modes typically exhibited by composite structure. When a conventional metallic fuselage shell is subjected to high impact forces during a survivable incident, the structure has the ability to plastically deform rather than fracture — sometimes quite severely with large indentations of the structure — without losing the overall integrity and general shape of the shell. By preserving the continuity of the protective shell and floor structures, occupants are more likely to be safe in the event of an impact. However, the brittle failure modes of composite materials raise the question of how an all-composite shell would perform under these conditions. Impact scenarios can include forces normal to the shell due to vertical drop or impact of an obstruction as well as sliding and dragging along a runway or unprepared surface with initial velocities at or near takeoff or landing speeds. If the initial failure of a composite shell were a clean fracture as opposed to plastic deformation, it is possible that the ensuing ground impacts or collisions with obstacles could continue to cause brittle failure of the structure, progressively destroying the protective fuselage shell. This issue could be critical to the level of occupant safety afforded by a composite fuselage and, with the existing technology, can be accurately investigated only by test.

This same potential for widespread or extensive damage under survivable impact conditions suggests the possibility of a significant economic issue relating to impact dynamics. Simply stated, if a composite fuselage suffers more extensive damage than conventional structure in a given impact, it would likely entail increased repair costs, longer time out-of-service, and possibly higher insurance costs.

The issue of impact dynamics must be investigated primarily by structural tests. Present analytical methods for predicting the dynamic response of an aircraft under impact conditions lack the sophistication to perform a credible analysis of a large-transport incident. Development of such methods is warranted, and should include structural evaluation, material characterization, and failure analysis. To properly characterize the energy absorption of the structure, the effects of postbuckled behavior, joint flexibilities, and the complexity of composite laminate failure modes must also be considered.

Design concepts aimed at maximizing the ability of the shell to retain structural integrity under impact conditions should be developed. The use of new damage tolerance design methods and "tougher" material systems should be examined in an effort to minimize brittle failure modes. Special attention should be given to the design details of the lower fuselage for impact resistance, such as the keel structure configuration, and the potential use of metals in these areas.

To evaluate these concepts, structural tests must be performed from the element level to the subcomponent, component, and full-scale structure. Tests should be conducted that are designed to simulate some aspect of an actual accident and evaluate a material system for impact applications, with candidate tests including horizontal impact, vertical drop, abrasion, and sparking. A full-scale test progam is recommended in subsequent sections of this report.

LARGE CUTOUTS

Contemporary transport aircraft contain numerous cutouts throughout the fuselage shell in a variety of configurations. The term "large cutouts" refers to major discontinuities in the structure such as the wing carry-through structure, wheel well, cargo doors, windows, cockpit enclosure, and other openings. The design and analysis of these cutouts presents a challenging task even with conventional metallic structures. The transfer of load around door or window cutouts produces large areas of stress concentration with peak stresses occurring at the corner radii of the cutout. Fastener holes or damage in the corners of these discontinuities superimpose local stress concentration fields on the wide-area field produced by the larger cutout.

In typical fuselage designs, the passenger door cutouts in the shell are reinforced with increased area frames, header beams, and skin doublers. The fatigue strength of the structure around the cutout can still be a problem because the superimposed stress concentrations and the presence of local bending and out-of-plane loads are difficult to account for by analysis and coupon test data. Full-scale test experience has proven that the concern is justified. The general design solution is to add more doubler material in the critical area, but such an approach can sometimes attract more load to the area without substantially reducing the stress levels.

The application of advanced composites to transport fuselage structure can be expected to greatly reduce the likelihood of in-plane fatigue failures under these conditions because of the materials' inherent resistance to fatigue damage, both in general and at fastener holes. However, the nearly linear-elastic behavior of composite materials will require that the stress concentration effects discussed above are accounted for in the static strength assessment of the cutout region.

While these effects have caused fatigue problems in conventional structure, the ductile properties of aluminum allow for a substantial amount of load redistribution prior to failure, facilitating the task of providing adequate static strength. Bolted joints in composite structures have shown substantial stress concentration relief from linear-elastic theory (Reference 10), but this characteristic has not yet been demonstrated for large cutouts. In addition to these effects, local bending of the fuselage skin material in the corner of the cutout further intensifies the stress field and produces out-of-plane forces that are undesirable for laminated structure. Further technology development is required for large cutouts in composite fuselage structure to ensure that each of the chosen design concepts is durable and damage-tolerant, with enough static strength to support design ultimate loads in an efficient and producible manner.

The design arrangement of these cutouts is constrained only by the size and location of each door or window, and the general configuration of the surrounding structure. Various concepts for structural design of the cutout region must be considered to efficiently minimize the peak stresses from load transfer. Since the stress concentration due to the cutout itself is a function of the corner radii, these may need to be increased (within reasonable limits) to reduce the stress intensity. Fastener holes or other discontinuities in the cutout area should be omitted to the extent possible to minimize superimposed stress concentrations. Composite materials seem appropriate for this application with capability for co-cured fabrication or bonded assembly of cutout reinforcement using gradual runouts and no fasteners whatsoever.

Such concepts may still not make up for the lack of plasticity in the basic material and the ductile properties of metals may provide the most effective means of local reinforcement. The ability to yield plastically under high stress levels and the relative insensitivity to minor impact damage are advantageous characteristics which support the use of metals in this application. Yet while the "softening" provided by metals is desirable, it produces an additional set of concerns. As noted previously, metallic structures have frequently exhibited fatigue problems at the very locations that require reinforcement and the use of metals could reinstate fatigue as the critical failure mode. Further, the combination of composite and metallic structure will introduce thermal incompatibility problems where the two materials are joined.

Whatever the design concept, the overall structural integrity of the cutout region must be verified by established analytical methods and, where necessary, a comprehensive structural test program. Previous work in this area has been limited to small-scale structure with flat panels, but results have been encouraging: the structural efficiency of cutout concepts and the ability to predict ultimate strengths, failure modes, and stress/strain distributions have been demonstrated.

Some valuable experience with cutouts in composites was gained during the DC-10 composite vertical stabilizer ground test program where a premature failure of the full-scale ground test article occurred in the region of a cutout (Reference 11). The failure occurred when a structural door was attached with loose-fit fasteners which provided insufficient load transfer around the cutout. This event further establishes the need for careful attention to detail for analysis of composite structures where the basic material is linear-elastic to failure, with no plastic relief as with ductile aluminum alloys. A detailed finite-element analysis of this cutout in its actual form provided excellent correlation with test results and suggests that existing analysis methods are effective.

Finite-element methods can be successfully applied to the analysis of cutouts for all types of structure. Contemporary analysis programs equipped with anisotropic element capabilities can be combined with appropriate failure criteria to provide accurate strength predictions for large cutouts in composite structures, particularly for in-plane stresses. Out-of-plane deformations can also be calculated through iterative solutions, but the use of such analyses may be severely restricted by high computing costs. That which is most lacking in predictive methods is the ability to calculate critical interlaminar stresses caused by unexpected out-of-plane loads or edge effects leading to premature delamination failures. These concerns are sufficient to warrant a series of structural tests to verify design concepts.

The required tests may include anything from flat panel tests of unreinforced cutouts to full-scale barrel tests that account for all the effects contained in actual structure. Initial tests are required to develop conceptual designs and verify stress distributions as analytically predicted. Full-scale structural tests demonstrate the level of technology while ensuring that no unanticipated failure modes will prove to be critical. These tests must be accompanied by detailed strength analyses of the cutout area.

JOINTS AND SPLICES

Conventional transport fuselage structure containts numerous joint and splice configurations. These joints are designed to have adequate static strength, durability, and damage tolerance with a minimum-weight design.

The baseline MD-100 fuselage is assembled primarily with mechanically fastened joints, with limited use of bonded connections on interior panels or secondary structure. A large number of longitudinal and transverse splices are used throughout the fuselage for subassembly and assembly of panels and barrel sections. Examples of these joints in the MD-100 fuselage are shown in Figure 3-3.

The hoop tension loads from cabin pressure are critical for the longitudinal splice. Cabin pressure loads acting in the longitudinal direction combined with fuselage bending stresses provide the critical load conditions for the transverse splice.

The conventional fuselage shell skin gage is sized primarily for design fatigue loads resulting from cabin pressure combined with flight shear loads since much of the joint structure is fatigue-critical rather than static strength-critical. Design strengths are based on net-section and bearing allowables for joint members, shear allowables for fasteners, and allowable fatigue loading developed from test data and

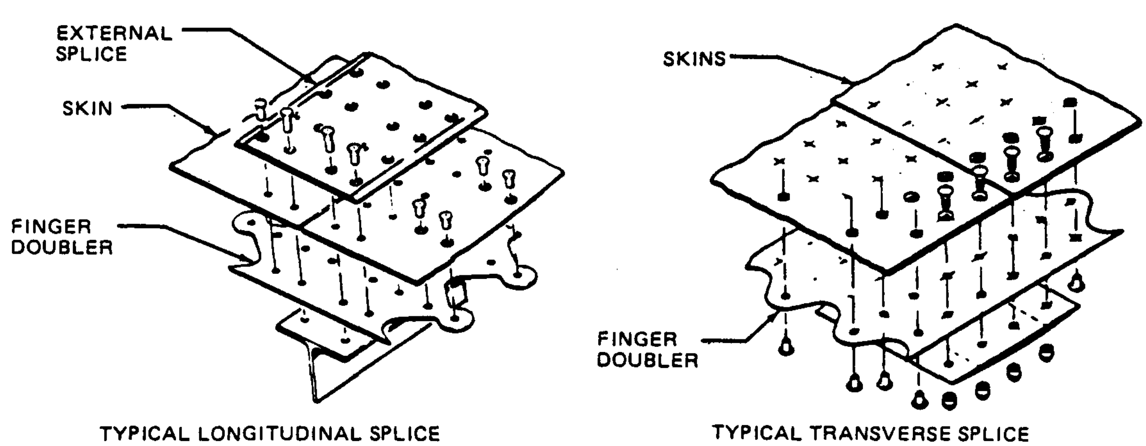


FIGURE 3-3. FUSELAGE SPLICES

service experience. As a design goal, the static and fatigue strength of the joint should be slightly higher than for the adjoining basic panel structure. Composite joints have been investigated on numerous research and development programs including some recent efforts conducted at Douglas Aircraft Company in Long Beach, California (References 11 and 12). The state of the art in design and analysis methods for composite joints has progressed rapidly in recent years, but several critical issues in bonded and bolted composite joint technology warrant further investigation.

The analysis of bolted joints in composites requires the determination of stress concentration factors associated with loaded and unloaded holes. While analytical methodology for single-axis loading has been proven effective, advances must be made in the present capabilities to analyze biaxial loads that often occur in fuselage joints and splices. Structural tests will be required to verify the methodology. Analysis methods and failure criteria for biaxially loaded bonded joints are also in need of development for adhesive bonding assembly throughout the fuselage.

The pressurization of the fuselage shell and the associated "pillowing" effect between frames impose substantial transverse tension-loads between the frame shear tee and skin interface. For a bonded joint in this application, the shear tee may be subjected to critical interlaminar tension stresses. When this phenomenon is combined with the effects of postbuckling, a condition is produced that could develop critical peel forces at the tip of the bond line or at the first ply of a thickness transition. This phenomenon has been investigated (Reference 13), but no substantial effort has been made to quantify these out-of-plane forces. In the case of bolted joints, countersunk fasteners may be subjected to fastener pull-through forces of critical magnitude.

The behavior of countersunk fasteners in thin composite panels will need to be investigated. The analysis of multirow bolted joints requires an accurate determination of the fastener load distributions, and is achieved through the use of fastener load-deflection characteristics. These parameters must be measured for countersunk fasteners in thin composite skins for both single- and double-shear applications, and correlated with available predictive methods. Load-deflection curves for single-shear composite joints have been shown to be highly nonlinear, which may require modifications to existing analysis methods.

Additional concerns include the behavior of composite panels at free edges, particularly when subjected to combined loads and out-of-plane deflections. For bolted splices, areas of combined loading where bearing stresses interact with transverse far-field stresses may become critical at a free edge. Thermal compatibility is an issue for both bonded and bolted joints where other materials are joined to composite panels, and is discussed later in this section.

State-of-the-art analysis methods typically use an approach that combines classical solutions with semi-empirical methods. The analysis methodology development that will be required to account for these phenomena must be supported by a comprehensive structural test program.

Element tests are required to develop single-hole properties for bolted joint concepts unique to composite fuselage structure. A series of tests should be conducted to measure the strength of skin-to-shear tee bonded connections that must resist out-of-plane forces. Once a sufficient data base is generated to support the analytical techniques, a series of subcomponent and component tests should be conducted to demonstrate the design concepts and validate the analytical methods.

POSTBUCKLING

The postbuckling range of conventional fuselage panels in transport aircraft has been established primarily for fatigue strength and to prevent permanent set below limit load. This criterion is generally established by specifying the load intensities at which initial skin buckling is allowed to occur.

Present criteria for the baseline MD-100 fuselage require that no buckling will occur below 1 g flight loads with or without normal pressure loads. As a general rule, panels are allowed to buckle in shear at roughly 50 percent of limit load, while compression buckling is governed by ensuring that a local instability will precede a general instability failure. In order to achieve the weight savings objectives associated with the use of composites, a composite fuselage shell should have similar postbuckled structural design criteria.

The postbuckling behavior of composite stiffened panels has been widely investigated throughout the industry. In several programs on this subject, tests indicate that for shear loading or compression loading, composite stiffened panels display exceptional fatigue strength with failure loads of four to five times the initial buckling load (Reference 14). In several cases, it has been shown that no degradation will occur in residual strength until applied loads are substantially above design limit load levels.

Despite these promising results, further investigation of postbuckled composite fuselage structure is warranted in several areas. The inability of composite materials to yield plastically, permitting redistribution of stresses, requires that local stress concentrations be accurately accounted for in static strength assessments.

Restrictions may be required on postbuckling to limit out-of-plane deflections that could result in severe secondary bending stresses or unanticipated interlaminar failures. Such deformations can lead to high forces inducing the separation of skin and stiffening elements, such as critical induced peel stresses for bonded assembly, or fastener pull-through forces for mechanically fastened joints.

Beyond these issues, the overriding concern associated with postbuckling of composite fuselage panels is the durability or residual strength of panels subjected to impact damage. Figure 3-4 presents the results of extensive work performed by NASA showing the structural response at initial buckling, failure load, and failure load after impact for a variety of stiffened panel configurations loaded in compression (Reference 15). Test results indicate that significant reductions in postbuckled strength may occur when stiffened panels damaged by impact are loaded in compression. The specific strength loss as compared to that of an undamaged panel is a function of several parameters including skin thickness, stiffener spacing, and assembly methods as well as impact energy and location of the damage. It has been shown that impact damage at a skin-stiffener interface is far more critical than impact in a central skin location, as the expected separation mode of failure is initiated.

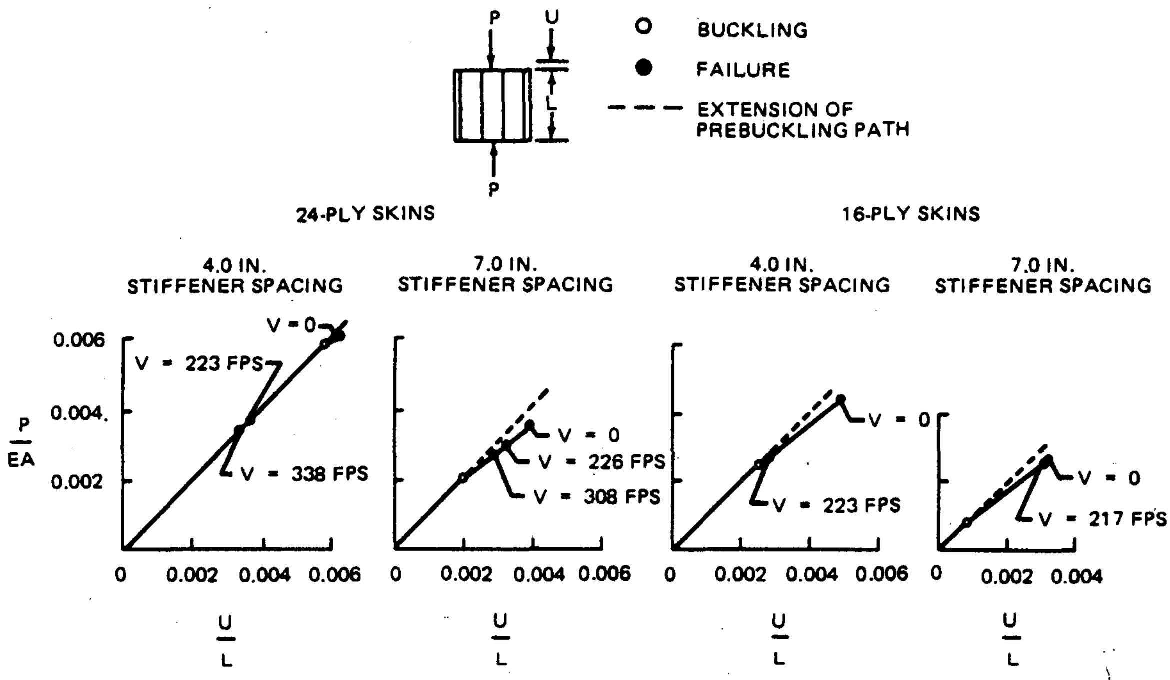


FIGURE 3-4. EFFECT OF IMPACT DAMAGE ON THE POSTBUCKLED STRENGTH OF STIFFENED CARBON-EPOXY PANELS

This issue did not exist for conventional structure, and its resolution is essential for determining the extent to which composite fuselage structure will be permitted to operate in the postbuckled regime. In some cases, the postbuckling behavior of composite panels may govern the impact criteria for damage tolerance.

Classical equations for predicting the postbuckled strength of aluminum structure include empirical constants so that theoretical solutions correlate with test data. In this approach, allowances are made for the ductile, inelastic behavior of aluminum at stress concentrations to predict overall panel failure. These empirical equations do not provide the methodology for predicting the local stress concentrations on the panel for a nonductile material which will precipitate panel failure at much lower panel loads.

Finite-element analysis methods presently used to predict the postbuckling behavior of composite stiffened panels are generally capable of accurate solutions but require more effort and much more computer time than the classical approach. Methodology exists for predicting the onset of panel buckling, the buckling mode, and the postbuckling behavior (Reference 16). Predictions can be made of surface strains, stress concentrations for high in-plane stresses, and in-plane failure modes. Analytic capabilities are lacking, however, for the assessment of two critical phenomena. Present methods cannot predict skin-stiffener separation for co-cured or bonded composite panels subject to postbuckling deformations, which is a common mode of failure. Methods are also lacking for predicting the residual strength of damaged panels. Developments in damage tolerance analysis methods and postbuckling methodology will be needed to address this problem.

Until these methods have sufficiently progressed, structural integrity must be substantiated by structural tests. Curved stiffened panels representative of composite fuselage structure should be tested under compression, shear, and combined loads for static strength, fatigue life, and residual strength after impact. Methods to improve postbuckling performance should be evaluated, including concepts for increased strength at the skin-stiffener interface and improved techniques for impact resistance.

BIRD STRIKE

Bird strikes occur often enough to require design features that protect the occupants and assure continued safe flight for expected flight conditions. In particular, FAR 25.571 specifies a discrete source damage tolerance protection and FAR 25.775 contains a more specific requirement for the windshield and its supporting structure to protect the occupants from a 4-pound bird strike at the airplane design cruise speed (V_c) at sea level.

The capability of the composite fuselage shell structure to absorb a 4-pound bird strike at 350 knots is questionable in light of the low-energy absorption characteristics, as shown in Figure 3-2. A glancing blow that deflects the bird will absorb only enough energy to generate a rebound force. The shell must then deform without fracturing since the rebound force is lost when the fracture occurs. Penetration of the fuselage will then occur with very little energy absorption. The penetration mode must be evaluated for injury to occupants, for residual strength for continued safe flight, and for repairability of the structure.

While penetration could be acceptable in some cases, the obvious design goal should be to not allow the bird to penetrate the shell. Further study is required to determine if composite designs can satisfy the bird strike criterion without undue weight penalty to avoid penetration. The more ductile aluminum shell structure may still be the best material for the frontal area of the fuselage.

Resistance to penetrations from a bird strike is a complex analysis problem and there is no test base from which to develop valid analytical methods for composite materials. Based on the state of the art, the resolution of the bird strike will require a test of the full-scale structure with a 4-pound bird strike at the appropriate design speeds.

ASSESSMENT AND DISPOSITION OF FLAWS AND DAMAGE

Deviations from the engineering drawing and process specification are bound to occur during the manufacture of composite fuselage structure. These deviations will be detected and recorded by the inspection personnel. One of the following engineering dispositions will then be prescribed:

1. The parts may be rejected as unacceptable for strength, fit and function, or quality.
2. The parts may be accepted for use when proven to have adequate structural integrity.
3. The parts may be reworked to meet engineering requirements.

Accept/reject criteria can be included in the process specification that will govern the engineeing acceptance of minor imperfections such as those listed in Table 3-2. The usefulness of accept/reject criteria depends on the capability to predict the type of flaws that will occur, the characterization of the flaws, and how far the acceptance limits can be extended without compromising the durability and damage tolerance of the structure.

These acceptance limits are a function of the design strain levels. As design strain levels are increased, the structure becomes more weight-efficient, but this also results in an increased sensitivity to flaws or deviations. The tradeoff between higher manufacturing costs due to increasing rejection of parts and the desire for high design strain levels to reduce weight is performed during the design integration process.

The development and production of composite structure throughout the industry will increase the data base and experience level for flaw or damage assessment and disposition. The manufacture of development and test hardware for the composite fuselage development program will improve the capability to predict the types of flaws and damage that will occur, will allow characterization of the flaws, and will demonstrate the capability to make disposition of the flaws and damage in a cost-effective manner.

REPAIR OF MAJOR DAMAGE

Experience has proven that in-service aircraft are prone to damage from sources that cannot reliably be safeguarded against by the design. When the damage occurs, an expedient means of repair must be available to restore the aircraft to a flightworthy status. The loss of revenue while the aircraft is being repaired is a very-high-cost item to the aircraft operator. The acquisition of a larger fleet, to avoid loss of revenue during the aircraft out-of-service repair time, is not an economical alternative.

Damage can be incurred as a result of repeated service loads and exposure to the environment or from discrete sources such as those encountered during off-runway incidents. Damage from runway debris, lightning, hail, bird strike, and tool/equipment impact is usually classed as a discrete source when the damage is such that repairs are required before further flight.

Structural repair manuals are published to provide instruction for the repair of local-area damage in aircraft. These repair methods are considered to be within the state of the art for flightworthy composite structure since a design data base is obviously available for use in the repair design. The structural repair manual is submitted to the governing regulatory agency for approval. Subsequent repairs by the operator in accordance with the manual instructions are then acceptable without further review.

Repair of larger damage is usually configuration-dependent. Specific repair designs are usually prepared by the manufacturer and require regulatory approval on a case-by-case basis.

The specific repair design must provide for restoring structural integrity to the same design criteria and loads as the original design. The repair design features are constrained by the undamaged surrounding structure and by the materials, facilities, and equipment that are available to make the repair. Temporary field repairs are often made for a placarded ferry flight to a major repair depot where facilities are available for making permanent repairs.

The analysis methods and test data used to develop the original design are applicable to the design of major repairs. The design technology needed is to devise repair concepts that can be effected using facilities and equipment available to the aircraft operators.

THERMAL COMPATIBILITY

Whether thermal compatibility emerges as a technology issue depends on whether metal structure is needed to resolve other technology issues. Variations in flight service temperatures in conjunction with the large differences in coefficients of thermal expansion between composites and metals will create significant internal loads at the bolted or bonded interface of the two materials.

The ductile properties of metals may prove to be more preferable than composites for the nose fuselage area to provide the necessary resistance to bird strike. Portions of the fuselage may feature metallic structure to provide additional crashworthiness, or perhaps metals will be used for reinforcement around large cutouts. In any case, if it is decided that these applications of metallics are required, then the associated thermal compatibility problems will have to be addressed.

Once a particular design concept using metallic structure is to be examined, it should not be exceedingly difficult to determine the severity of the thermal incompatibility. State-of-the-art analytical methods for bonded and bolted joints can accurately account for the effects of temperature variations. The prediction of interlaminar stresses resulting from thermally induced internal loads is somewhat more complicated, but these and any other effects that are beyond the scope of our analytical methodology can be examined through a suitable series of structural tests.

These tests and analyses will be less frequently required as a greater percentage of the aircraft structure is converted to composites. In the case of fuselage structure, if the wing or empennage structure were made of composites rather than aluminum, the thermal compatibility problems would be substantially reduced at their junction.

A dilemma will occur when the resolution of one technology issue absolutely requires the use of metallics, yet this application is unacceptable from a thermal compatibility standpoint. Such circumstances could necessitate minor or major changes in the planned fuselage configuration and may lead to solutions that could result in weight penalties.

ELECTROMAGNETIC EFFECTS

A number of problems are encountered when low-conductivity composite fuselage structure is used in large transport aircraft. They include structural damage from lightning strike, electromagnetic interference between onboard systems, effects on and between antenna systems, and effects of electrical hazards on personnel.

Present lightning protection methods and techniques, including use of the basic structure itself and the application of conductive materials to either or both sides of the structure, appear to be ineffective for large transport aircraft. Other techniques that involve local shielding and direct system protection appear to be necessary in order to obtain a low-cost, lightweight construction.

The low conductivity or nonconductivity of composites affects the inter- and intrasystem wire coupling far more than with traditional all-metal construction. Structural grounding approaches are also heavily impacted. All of these changes associated with composite structure affect the electromagnetic compatibility of electrical and electronic systems which are housed within the composite structure.

The introduction of composite materials in fuselage structures creates undefined design and operational problems for the electrical distribution system. Conventional electrical power systems utilize the aircraft structure as a ground plane for ac and dc circuits. Also, the extensive use of composite materials in large fuselages creates significant grounding problems. Composites are poor conductors; therefore, new grounding schemes must be evaluated before composite technology can be utilized in large airframes. The previous use of composite primary structures, primarily for small military and commercial aircraft, has not provided data directly applicable to transport aircraft technology.

The electrical performance of antennas installed on composite structure is different from antenna installed on metal structure. The different performance characteristics result from changes in antenna impedance, radiation patterns, and relative gain. Isolation between low-frequency antenna systems and other flight systems on large transport aircraft has not been evaluated.

On conventional aircraft, metal structure is used for the electrical ground system for all electrical and electronic circuits. The use of low-conductivity composite material can cause high voltage to appear within the fuselage that may result in adverse safety conditions for personnel.

A number of solutions must be evaluated in order to achieve a balanced rationale for the electromagnetic problems. Each solution should be assessed for technical parameters such as attenuation, thickness, and placement of shielding. But additionally, the reliability of the solution, cost of installation, cost of maintenance, and the additional weight must be thoroughly quantified.

ACOUSTIC TRANSMISSION

The application of advanced composites to transport fuselage structure promises substantial weight savings over conventional structure. The issue in acoustic effects is whether or not the reduction in mass will increase interior noise to an unsatisfactory level.

Several parameters are involved in determining acceptable interior noise levels. Passenger comfort must be considered, including noise levels at frequencies that may interfere with speech. Interference with speech is an issue for the crew as well, and is related to the interference with work performance associated with excessive noise levels. These parameters are all considered in the noise specifications prepared for the commercial airlines. Beyond these requirements, maintaining relatively low interior noise levels is beneficial to the manufacturer from a competitive standpoint.

A comparison of speech interference levels throughout the cabin in several commercial aircraft is presented in Figure 3-5. These curves are measured noise levels and are plotted along with an analytical prediction of the increase in noise resulting from the loss of mass associated with an all-composite fuselage. Speech levels are typically in the range of 1,000 to 4,000 Hz. The speech interference level is an arithmetic average of three frequency bands of 1,000, 2,000, and 4,000 Hz. Figure 3-5 shows an increase in noise of approximately 3 dB for the composite fuselage compared to the MD-100 fuselage, a sufficient rise to warrant a serious evaluation of acoustic effects.

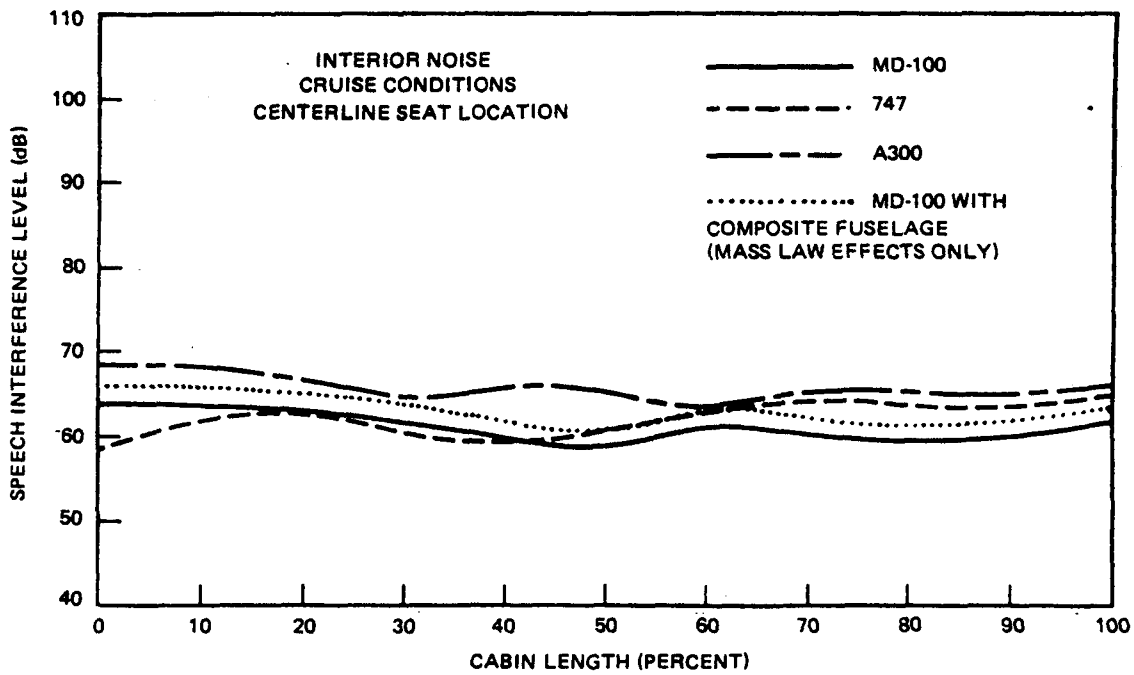


FIGURE 3-5. ACOUSTIC TRANSMISSION

Sound transmission loss is generally a function of structural damping, fuselage wall depth, and structural mass and stiffness. A data base must be developed to determine the relative influence these parameters have on acoustic transmission characteristics for a composite fuselage shell. These data may be compared with existing properties of conventional structure to form a reasonable basis for design concepts.

If increases in the noise level prove to be significant, a number of methods may be employed to resolve the problem, including (1) changing fuselage wall stiffness, (2) increasing structural damping, and (3) increasing cabin sound absorption. The spacing between the fuselage skin and interior trim panels might also be increased, but this would result in either an increase in fuselage diameter or a decrease in the interior room, both of which are undesirable. Increasing sidewall surface density is another potential solution, but is the least desirable since it involves adding mass back to the structure and negating the intended weight savings. In any case, this issue appears to be potentially critical and a major effort to examine the problem is warranted.

To reduce the technical risk associated with acoustic effects for composite fuselage structure, noise reduction properties should be determined from test results as opposed to theoretical prediction techniques. The higher level of confidence gained from an acoustic test program will result in a more efficient design, or more competitive interior noise guarantees to the aircraft operators, or both. An acoustic test program should evaluate material and structural damping over the ranges of mass and stiffness representative of actual structure. Test specimens could range from flat panels to full-scale barrel sections, with transmission loss and structural damping examined.

NONDESTRUCTIVE TESTS

Nondestructive testing of carbon-epoxy composite structure presents many challenges when a large structure such as a composite fuselage is considered. The detail parts and assemblies must be inspected for defects (Reference 17). In addition, the defects need to be defined — by type, location, orientation, and size. This information is used to make accept/reject decisions. However, there are problems that need to be addressed in inspecting and defining such defects in fuselage structure. Some of these problems are:

- Reliability of flaw detection.
- Difficulty of inspecting thin, composite laminates of complex geometry.
- Need for development of computerized, contour-following, automated ultrasonic inspection equipment.
- Constructive or destructive ultrasonic wave interference due to variations in laminate or adhesive bonded layer thickness.
- Need for equipment calibration standards.
- Need for development of accept/reject criteria for fuselage structure.
- Need for quantitative nondestructive inspection methods for measuring resin and void content.
- Need for in-service nondestructive inspection methods to assess damage and quality of repairs.
- Artifacts affecting test results.

Possible solutions to these problems are given in the following text.

Reliability

To assure test reliability, personnel must be qualified, equipment standardized, signal-to-noise ratio controlled, artifacts eliminated, detailed test procedures written and approved, and computerized signal processing and automated test equipment used appropriately.

Inspection Difficulty for Thin Laminates

Fuselage structure is composed of thin, complex geometry details that require the use of low-kilovoltage radiography or high-resolution pulser/receiver ultrasonic equipment that measures in nanoseconds. The complex geometry of skins, longerons, and frames generally requires the use of manual inspection and, hence, can become labor-intensive. Careful attention will need to be paid to assembly operations and when nondestructive inspection is to be conducted to avoid labor-intensive inspections.

Automated Test Equipment

Composite structure, because of its anisotropy and variability, needs to be inspected by nondestructive means. This can be done in a timely and cost-effective manner by automated computerized test equipment. Computerized signal processing is necessary to assist in interpreting test results. Automated x-ray and ultrasonic equipment is presently available, but must be programmed to scan parts with complex geometry. Further development is needed to improve signal processing techniques.

Constructive/Destructive Wave Interference

Experience with thin composites and bonded joints dictates a need to eliminate interference effects from variations in laminate and adhesive bondline thicknesses. This is best overcome by using broad-band (multifrequency) ultrasonic transducers. A computer program is available to predict the performance of search units prior to manufacturing.

Calibration Standards

Experience indicates that calibration standards are necessary for ultrasonic inspection. These standards must be similar to the parts being inspected and must contain simulated flaws. A wide variety of standards has been developed, and the technology transfer is possible for composite fuselage inspections.

Accept/Reject Criteria

For inspections to be meaningful, accept/reject criteria must be developed before conducting inspections. Structural mechanics engineers, familiar with composites, must develop realistic criteria as a basis for establishing quality standards for composite fuselage structure.

Resin and Void Content Measurement

Destructive methods used to determine resin and void content are time-consuming and results are limited to a sample location. The feasibility of ultrasonic attenuation, velocity, and backscattering techniques in measuring resin and void content by nondestructive means has been shown. Standard procedures need to be developed to prove the reliability of these techniques to measure resin and void content quantitatively in production parts.

In-Service NDI Methods

Various ultrasonic techniques are capable of detecting impact damage and delaminations that can occur in service. The applications and limitations of these techniques need to be determined so that applicable procedures can be added to the Nondestructive Testing Manual used by operators. Some work has been done to solve this problem, as indicated in Reference 18. Additional work needs to be done in determining the quality of bonded repairs. This is one of the most difficult problems confronting inspectors utilizing nondestructive means. Various ultrasonic and bond testing techniques have been partially successful. However, the variation in repairs can produce erratic inspection results. Test specimens must be damaged, repaired, and inspected by nondestructive methods. Correlations must be determined between nondestructive inspection results and quality of the repairs.

Artifacts

Conditions such as surface porosity, waviness, wrinkles, and surface roughness are known as artifacts. They can have considerable influence on inspection results, especially in ultrasonic testing. Specimens having artifacts need to be fabricated to determine their influence on inspection results and material properties. Methods to eliminate them, or at least reduce their occurrence during manufacturing, need to be studied.

MATERIAL AND PROCESS TECHNOLOGY

The use of carbon-epoxy in fuselage structure for transports presents new considerations regarding material selection, product repeatability, and adhesive bonding.

Material Selection

Material evaluation criteria include the following parameters: toughness, mechanical strength in the actual environment, processibility, and smoke and burn. Recently, many new toughened resin/fiber systems have been introduced by manufacturers of preimpregnated materials which match high-elongation fibers with toughened ductile epoxy or bismaleimide resins. Their product is a composite laminate that shows a marked increase in resistance to impact damage.

Extensive testing must be conducted to evaluate candidate materials. Under a NASA contract, results of several impact tests were evaluated using a variety of brittle and toughened systems (Reference 19). Environmental tests conducted at Douglas include hot-wet environment, jet fuel, hot hydraulic fluid, temperature soak (+200°F and −65°F), and solvent resistance. Short beam shear and flexural strength tests were run for all environments. In addition, compression tests were run in hot-wet soaks.

Fabrication of large fuselage panels presents several processing difficulties that can be minimized with the right material system. A straight-up cure cycle, in which full autoclave pressure is applied at the beginning of the cure, and the dynamic viscosity of the resin need to be considered for future production. Additionally, good tack and long out-life are necessary in automated tape-laying or hand layup. Enhanced processibility is a property being developed by the preimpregnated material manufacturers.

The material selection process will include the FAR 25 vertical burn tests as well as smoke-generating tests. With other properties being equal, the system chosen would have lower smoke and burn properties.

Product Repeatability

In fabricating large composite parts, there is a great need for specifications that define the processes and controls which ensure consistently high quality; i.e., repeatability. The parameters must be controlled by adequate instructions in the material and processing specifications. Resin rheology versus one-time profiles must be developed. A chemical characterization profile should be made to describe a particular resin system and predict its cured laminate properties. The rate of heat rise must be defined to ensure that the difference in temperature between any two points of the tool is within permissible limits. Premature curing in some areas can cause volatile entrapment due to blocked pathways.

A process specification must deal with the different tooling methods used to fabricate parts. Although a single resin system may be processed by trapped rubber, press, or autoclave, the layup materials and processing cycles will vary for different curing methods.

Adhesive Bonding

Secondary bonding or one-step co-curing is suitable for attaching longerons and shear tees to the fuselage skin. The selection criteria for adhesives will rely on mechanical properties obtained from double-lap shear tension, peel, and creep specimens.

There are gaps in the technology of adhesive bonding in the following areas: (1) room temperature versus high temperature curing; (2) optimization of environmental resistance and ductility; (3) surface preparation and verification; (4) vacuum pressure processing; (5) bond strength verification by nondestructive inspection, and (6) chemical characterization for quality control. Environmental resistance and ductility are two properties that are difficult to obtain in the same adhesive system. Often, the chemical modifications that provide low modulus and high elongation have deleterious effects on solvent resistance and other environmental properties.

Secondary bonding in large fuselage sections could be accomplished without the use of an autoclave. These lower pressures could simplify the bonding operation.

MANUFACTURING TECHNOLOGY

To meet production requirements at a competitive cost, there are several issues that must be resolved and an orderly transition made from a metallic to a composite fuselage. These issues include tooling needs, fabrication and process analysis, assembly plans, facility and equipment requirements, and an automation utilization study.

Tooling

Tooling for a large composite fuselage may be considered in terms of tooling materials and tool design and fabrication. The materials are of concern because dimensional stability and durability are required together with a predictable tool life to establish acceptable recurring and overall costs. The design and fabrication need investigation to provide a better match of the tool to the details and processes for improved quality of the final parts.

Fabrication and Process

Fabrication and process analysis considers the size and complexity of the composite parts. The basic fuselage skin panels may have precured longerons and shear tees bonded to the skin. The loft surface, dimensions, straightness, and waviness requirements are all critical for the detail parts. An analysis of the available fabrication methods such as press curing, autoclave, filament-winding, and the pultrusion process must be critically appraised for manufacture of quality detail parts. In addition, fabrication of thermoplastic composite components requires an assessment of forming technology.

Assembly

Assembly and subsassembly operations rely heavily upon accurate details and correct placement of stiffeners, longerons, and shear tees. For example, the details must be spliced near panel edges to adjoining parts during subassembly.

The joining of panel assemblies to form fuselage barrel sections highlights the need for maintaining frame stations to avoid prestressing the panel at the splice regions. The joining of barrel sections to complete the fuselage poses a challenge to provide matching cross sections at the transverse manufacturing joint without excessive shimming of the skins or longerons.

Facilities and Equipment

An analysis of the production facility and equipment requirements for manufacture of composite fuselage indicates a need for a facility of sufficient size to house freezers for material storage, a controlled environment for cutting and layup of composite materials, and an autoclave station housing the necessary quantity and sizes of autoclaves. Ovens, machining equipment, nondestructive inspection equipment, and subassembly and final assembly tooling are needed as well as floor space for manufacture and production control, offices, and other support activities.

Automation

With large panel sizes, up to 20 by 40 feet, a high degree of automation should be achieved at all levels of manufacturing. Automation must be given high priority, from computerized nesting of parts for cutting to layup of the plies using robotics. Automation could also be utilized to control the movement of the composite material between production processes. An automated work station to assemble fuselage barrel sections should include material handling, shimming/splice and backup alignment, robotic drilling and hole inspection, installation of fasteners, frame splice installation, and assembly inspection.

SECTION 4
MANUFACTURING DEVELOPMENT

This section discusses nondestructive testing of carbon-epoxy composite structure, materials and processes and manufacturing activities. The technology issues related to each of these areas are included in Section 3.

NONDESTRUCTIVE TESTS

The discussion of the development of nondestructive test methods for advanced composite structures includes an assessment of the various defects to be encountered, inspection techniques to be utilized, reference standards needed, and accept/reject criteria. Automated inspection is also discussed.

Examples of Manufacturing Defects

Carbon-epoxy laminates may contain one or more of the defects illustrated in Figure 4-1. The cured laminate may contain delaminations, interlaminar voids or porosity, foreign objects, and delaminations at corner radii. If pressure is lost during the curing process, the most likely defects are interlaminar porosity or voids. Cured laminates are usually joined by adhesive bonding, which can result in voids, porosity, and lack of bonding.

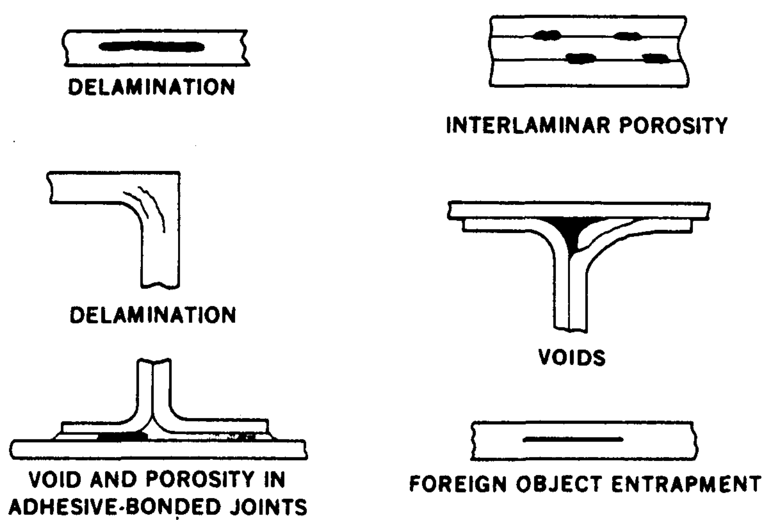


FIGURE 4-1. EXAMPLES OF MANUFACTURING DEFECTS

Ultrasonic Inspection

Various ultrasonic techniques are used for the different shapes of the composite assemblies. These techniques include the squirter through-transmission C-scan, the immersed through-transmission C-scan, the immersed reflector-plate C-scan, contact through-transmission, contact and immersed pulse-echo, ultrasonic thickness gage, and resonance impedance. The various ultrasonic techniques are illustrated in Figure 4-2.

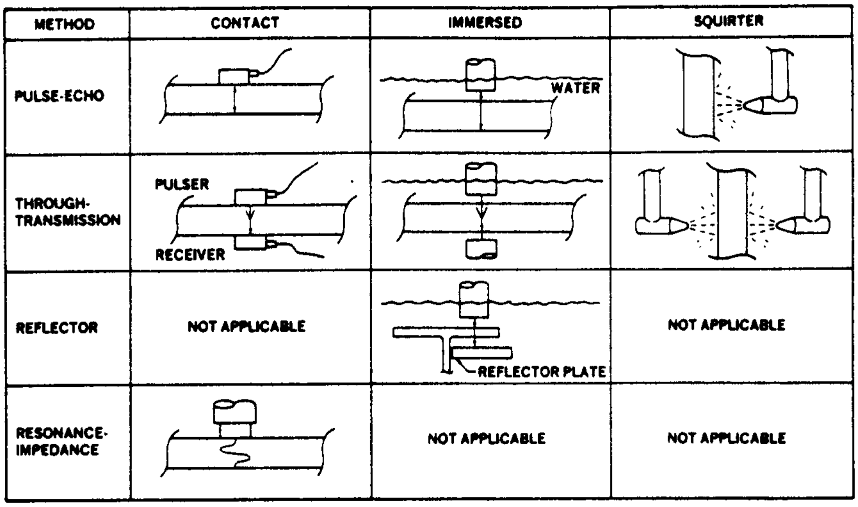


FIGURE 4-2. NONDESTRUCTIVE INSPECTION METHODS – ULTRASONIC

Accept/Reject Criteria

Before inspecting production parts, the appropriate accept/reject criteria need to be developed. The maximum allowable defect size must be defined along with the frequency and severity of smaller defects. The engineering drawing may be zoned to require different quality levels at different areas of the same part. Usually, the highest quality will be required at the area of maximum stress or strain. To assure that the maximum detectable flaw size can be detected, the built-in defects in the reference standards must be of the appropriate size. Additional standards must be established for foreign objects and porosity. Realistic accept/reject criteria can be established only after an evaluation is made of the effects of the various defects on the mechanical or fatigue properties of the structure. Considerable time and money can be expended to arrive at realistic criteria.

Service-Induced Defects

Composite structure may develop the following defects during service on operational aircraft: impact damage, delamination of plies, disbonds, cracks, fastener hole damage, water entrapment in honeycomb, lightning strike, and burning or overheating.

The service inspection philosophy may be summarized in the following concepts:

1. Visual inspection is the principal method of damage detection.
2. Directed visual inspection and nondestructive inspection are to be made of specific components or areas of specific components.
3. Nondestructive inspection is to be made to determine the extent of visual damage.
4. Nondestructive inspection is to be made for postrepair inspection.

Periodic nondestructive inspection may be required of critical areas of the fuselage. When required, these procedures would be contained in the nondestructive testing manual using techniques indicated in Reference 18.

Automated Ultrasonic Inspection of Fuselage Structure During Fabrication

It is envisioned that large composite skin panels, longerons, and frames will be fabricated, cured, and then inspected automatically. For high production rates, separate automated, computer-controlled ultrasonic systems will be required to inspect the skins, longerons, and frames separately. Because the skins are thin, the reflector plate technique (Figure 4-2) appears to be most applicable for inspection. Multiple array search units can be used to cover more area of the skin, resulting in reduced inspection time. Either the reflector plate or through-transmission techniques appear most feasible for inspection of the frames and longerons.

To automatically scan a typical longeron-to-skin bond joint would require four multiple scans using one probe or one scan using four probes in one common holder, as illustrated in Figure 4-3. The quality requirements under probes 1 and 4 are different than under probes 2 and 3. Hence, a multiplexer is required to fire 1 and 4 and then 2 and 3 from separate bond testers. Bond tester 1 is set to detect smaller flaws than bond tester 2. By having a multiple array of search units and instruments, more than one longeron-to-skin bond joint may be inspected in one scan, as illustrated in Figure 4-4. A similar ap-

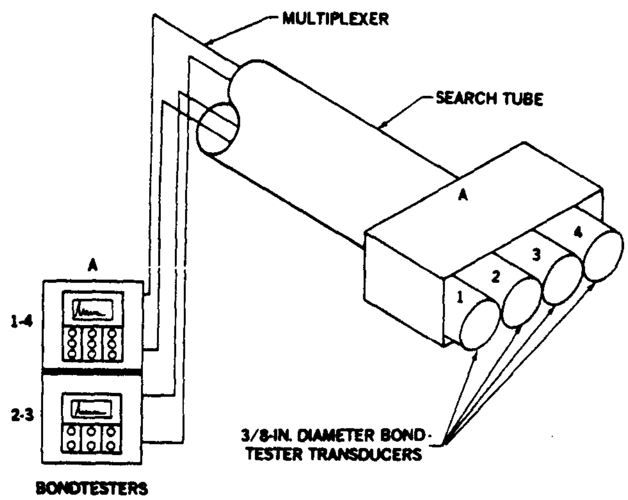


FIGURE 4-3. PROBE HOLDER FOR SEARCH UNITS AND MULTIPLEXER

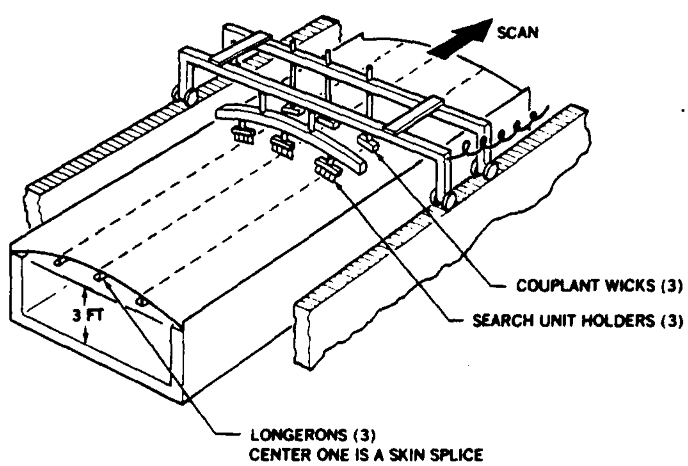
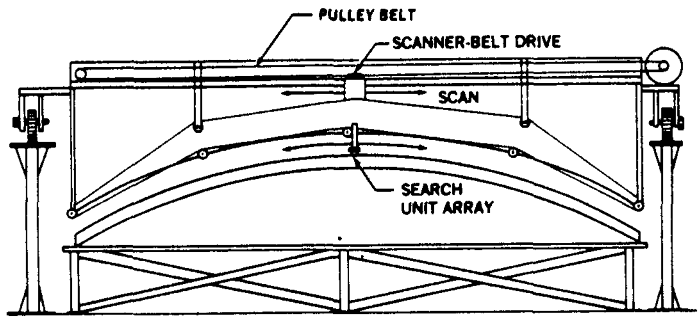


FIGURE 4-4. AUTOMATED ULTRASONIC LONGERON INSPECTION SYSTEM CONCEPT

proach can be used to inspect the frame-to-skin bond joints, as illustrated in Figure 4-5. Three or more 4-probe arrays could be used to reduce inspection time, as illustrated in Figure 4-6. For nonsymmetrical fuselage structure, one 4-probe array, operated by a microprocessor to compensate for the variation in contour, would be most easily applied. Various computer programs would need to be established for each frame location.

These concepts would need further study to establish the detailed requirements for automated ultrasonic inspection of the composite fuselage structure during fabrication.



NOTE: ONLY ONE FRAME INSPECTED PER SCAN. BRIDGE IS MOVED TO EACH FRAME POSITION

FIGURE 4-5. AUTOMATED ULTRASONIC INSPECTION SYSTEM FOR FRAME-TO-SKIN BOND JOINTS

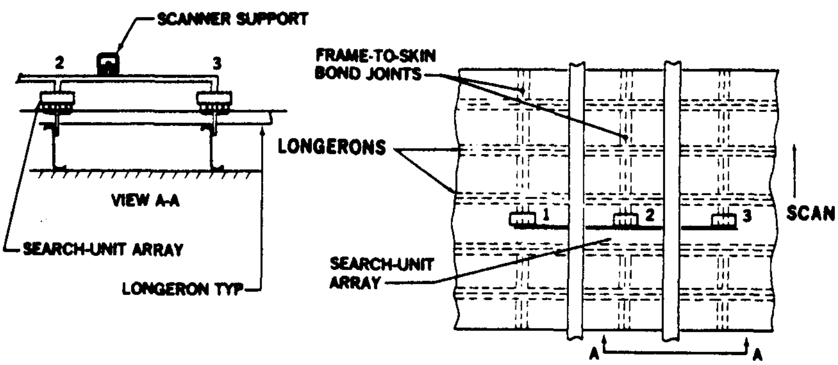


FIGURE 4-6. AUTOMATED ULTRASONIC INSPECTION SYSTEM FOR FRAME-TO-SKIN BOND JOINTS

MATERIALS AND PROCESSES

This section discusses materials, a rating system, each of the properties evaluated in the selection process, and adhesive bonding. A typical rating system for candidate fuselage materials is shown in Table 4-1. Some of the current material systems on the market are shown in Table 4-2.

TABLE 4-1
COMPOSITE MATERIALS RATING
(4 RATING IS BEST)

MATERIAL (MODIFICATION)	TOUGHNESS	ENVIRONMENTAL RESISTANCE	MECHANICAL STRENGTH	PROCESSIBILITY	SMOKE AND BURN
ICI PEEK (NONE)	4	4	2	1	4
DUPONT K-RESIN (NONE)	4	3	2	1	4
CIBA 2566/CHS (NONE)	4	2	3	4	1
HERCULES 3501-6 (NONE)	2	3	4	2	3
CIEA 914/T300 (NONE)	2	2	3	4	1
CIBA 914/T300 (>3 PERCENT VOID CONTENT)	4	1	2	4	1
NARMCO 5208/T300 (NONE)	1	4	4	3	3
NARMCO 5208/T300 (0.25-IN. KEVLAR STITCH)	2	4	2	3	2

TABLE 4-2
STANDARD TEST DATA
PER NASA REFERENCE PUBLICATION 1092

PROPERTIES

MATERIAL	ST-1	ST-2	ST-3	ST-4	ST-5
HEXCEL HX1504/CHS	31.8	0.96	52.5	33.9	
HERCULES 2220-3/AS6	26.4	0.52	53.6	31	1.25
NARMCO 5245/CHS	30.6	1.24	64	35.7	1.26
CIBA LSC 2566/CHS	35.9		61.8	38.8	2.19

ST-1 (COMPRESSION AFTER IMPACT, KSI)
ST-2 (EDGE DELAMINATION TENSION, LB/IN.) 8-PLY
ST-3 (OPEN-HOLE TENSION, KSI)
ST-4 (OPEN-HOLE COMPRESSION, KSI)
ST-5 (DOUBLE CANTILEVER BEAM, LB/IN.)

Dynamic impact test machines will also be used to evaluate candidate materials. Under a NASA contract, several impact tests were evaluated using a variety of brittle and toughened systems (Reference 19). Test data showed good correlation between the Gardner and Rheometrics machines. The Gardner uses a free-falling weight that causes damage, and the area of damage can be measured by C-scan. The disadvantage of this method is that damage is evaluated only in two dimensions, and a very brittle material and a tough material can both show the same damage under C-scan. The Rheometrics machine measures the force required to drive a shaft through a composite laminate as load versus deflection. This shows a laminate's resistance to complete penetration. With the Rheometrics machine, test data are more easily quantified than with Gardner data.

A third through-impact test uses an Instron test machine to drive a 5/8-inch-diameter hemispherical head through an eight-ply isotropic panel.

The technology for toughness evaluation is being developed under separate Company-sponsored and contracted programs and should be adequate for selecting material systems for the composite fuselage development program.

Environmental - Environmental testing is especially important when toughened resins are being considered for fabrication of parts. Table 4-3 lists environmental tests used to screen materials at Douglas.

TABLE 4-3
TYPES OF ENVIRONMENTAL TESTS

ENVIRONMENT	DURATION (DAYS)	TEST TEMPERATURE
HOT, WET (140°F/95 PERCENT RH)	30	RT/200°F
JP-4 JET FUEL	30	RT
HOT SKYDROL (HYDRAULIC FLUID) (140°F)	30	RT
TEMPERATURE SOAK (200°F)	30	200°F
(−65°F)	30	−65°F
SOLVENT RESISTANCE (MEK)	30	RT

Short beam shear and flexural strength tests are run for all environments; in addition, compression tests are run in hot-wet soaks. Toughened resins seem very susceptible to the hot-wet environment. Brittle resins like Narmco 5208 show little effect. Current programs will provide much greater insight into potential long-term problems and will better define accelerated test methods.

Processibility — Large fuselage panel fabrication presents several processing difficulties that can be minimized if the right material system is chosen. The properties noted in the following paragraphs will be considered when choosing a fuselage material system.

A straight-up cure cycle means that full autoclave pressure is applied at the beginning of the cure. This procedure has several advantages. The vacuum bag integrity is checked before cure begins, the operator does not have to wait for a "viscosity window" in the cure cycle before applying pressure, and the 100-psi pressure during heat-up provides more efficient debulking and densification than vacuum alone (Reference 20).

The dynamic viscosity of a resin capable of this cure cycle is much higher than past systems such as Narmco 5208 or Hercules 3502. Complete fiber wetting and volatile elimination are two areas of concern when the tough resins are processed.

Good tack and long out-life are necessary in automated tape-laying technology or hand layup. Total fabrication times for large stiffened skins could be as long as 4 to 5 weeks, and material must remain pliable. Related to these properties is long-term storage at 40°F. Conventional 0°F storage materials waste valuable time because they must reach ambient temperatures before they can be used. At 40°F, materials will have less condensation problems and reach room temperature much faster. Proper tack also minimizes air pockets during layup and decreases the chances of unacceptable panels.

Low-flow, net resin preimpregnated materials are usually cured by the straight-up pressurization cycle. Net resin preimpregnated materials are ordered with wet resin contents within a few percent of the cured resin content. Besides being more economical, they enable edge dams and excess bleeder material to be eliminated. Problems like resin-starved areas are minimized because of high viscosity and faster gel times.

Enhanced processibility is a property being developed by the preimpregnated material manufacturers. The present state-of-the-art materials are considered adequate for fuselage fabrication.

Smoke and Burn — The material selection process will include the FAR 25 vertical burn test as well as smoke generation tests. New tough resins often use plasticizers and extenders that generate smoke when ignited. With other properties being equal, the system with the lower smoke and burn properties would be chosen.

Repeatability — One of the principal needs in fabricating large composite parts are specifications that define the processes and controls which ensure consistently high quality. Following is a discussion of material and manufacturing deviations that must be controlled by adequate instructions in the material and processing specifications:

Resin rheology versus out-time profiles must also be developed. Out-time and relative humidity will change the viscosity profile as temperature is applied. The cure cycle must be modified to reflect this new viscosity profile in order to prevent excessive or insufficient flow during the cure. This kind of flexibility means computer-controlled equipment that can interpret dynamic viscosity data.

The chemical characterization profile is a series of tests that will describe a particular resin system and predict its cured laminate properties. Once a material has been qualified, it is important that the formula not be changed. Mechanical tests for quality control of incoming material are not sensitive enough to pick up differences that could show up after long-term environmental testing. Besides being more accurate, chemical analysis of the quality of incoming material is far less expensive.

The rate of heat rise must be defined to ensure that the difference in temperature between any two points of the tool is not greater than the maximum allowable. Premature curing in some areas can cause volatile entrapment because of blocked pathways.

A process specification must deal with the different tooling methods used to fabricate parts. Although a single resin system may be processed by trapped rubber, press or autoclave, the layup materials and processing cycles will be very different. Repeatability is especially critical when using trapped rubber. The silicone rubber does not continue to apply exactly the same expansion pressure cure after cure. The result is variations in ply thickness and resin content from part to part.

Adhesive Bonding

Current state-of-the-art adhesives being evaluated are modified epoxies with room temperature or elevated temperature curing. The selection criteria will rely on mechanical properties such as double-lap shear tension, peel, and creep specimens.

The most widely used adhesive systems are modified epoxies that can be cured from 200°F to 350°F. These systems have chemical formulations very similar to epoxy preimpregnated formulations which allow them to be co-cured without compatibility problems. They are low-modulus, high-peel-strength materials whose ductility more closely matches new high-strain composites. AF3113-2 and FXM 250 are examples of new film adhesives with 250°F cure requirements. The adhesive chosen will undergo testing to demonstrate compatibility with the composite laminates. Minimum strength requirements will be defined as well as an assessment made of surface preparation techniques.

Adhesives like FM 73 and FM 300 must rely on autoclave or mechanical pressure to cure without excessive foaming. Vacuum pressure causes the gases entrapped in the material to expand, which has detrimental effects on the bond strength. New adhesive systems like Hysol EA 9628 NW have shown lap shear properties, using 14-psi vacuum pressure, equivalent to high-pressure autoclave cures (Reference 21). Secondary bonding in large fuselage sections could be accomplished without the use of an autoclave. These lower pressures would also prevent the distortion or collapse of I- or J-section longerons during bonding operations.

These tests will assure exact repeatability from a formula constituent viewpoint and will be useful in determining the compatibility of the preimpregnated material and the adhesive. New materials are continually being evaluated on supporting programs throughout the industry. At the time of fabrication, the system showing the best combination of properties will be chosen.

MANUFACTURING

Manufacturing a large primary structure such as a fuselage from advanced composite materials depends on a complete understanding of the materials and manufacturing methods and development of a quality composite fuselage design integrated with a cost-effective production development plan. It has been shown in many studies that the fabrication and assembly of both metallic and nonmetallic fuselage structures account for 40 percent to 45 percent of the total airframe cost. With the cost of composite raw materials being five times higher, there is a clear need to thoroughly analyze the assembly and subassembly operations.

At present, most composite assembly is done in a very labor-intensive and inefficient manner. The technology does exist to automate many of the processes of producing composite structures, but the manufacturing feasibility study must first be integrated with the conceptual design.

In considering the manufacture and assembly of a large transport aircraft, an integrated "sectional" structure for assembly was selected. The size and complexity of these sections are limited by access requirements, repair, dimensional tolerance control (during bonding and curing), handling, and factory equipment capability. In manufacturing this design, all substructures must be interfaced with skins, longerons, frames, and cutouts to achieve structural integrity of a total pressurized fuselage. A tradeoff of handling, tooling, fabrication, and assembly requirements versus structural design was then made to determine the location and number of longitudinal and circumferential splices.

In order to meet the production requirements at a competitive cost, the manufacturing technology issues and facility needs for an orderly transition from a metallic to a composite fuselage must be analyzed. These requirements include a fabrication and process analysis, tooling requirements, subassembly and assembly plans, facility needs, equipment requirements, and an automation utilization study.

Tooling

Tooling for a large composite fuselage is divided into two separate issues, tooling materials and tooling fabrication. Tooling materials are of concern because close dimensional tolerances are required and a predictable tool life is needed to establish the recurring and overall costs to produce large composite fuselages.

The materials utilized to fabricate these composite components must be dimensionally stable and durable, such as those chosen for the plastic laminating mold shown in Figure 4-7. (The plastic laminating mold for the skin would typically be constructed of an eggcrate substructure fitted with adjustable stud attachments that permit a loft surface to be rough-formed and set to proper loft by locally adjusting the heights of the studs supporting the surface. The surface and substructure would be defined on the Computer-Aided Design Tooling system.

Dimensional and thermal stability of the surface tool materials utilized can be assessed by thermal-mechanical analysis which measures the thermal coefficient of expansion and dimensional changes over a range of thermal cycles. (Evaluation of composite tooling resin has shown that many composite tooling resins do not stabilize until 8 to 10 thermal cycles). In contrast, a metallic tool surface, which has predictable expansion, must be designed to account for the thermal mismatch between the composite component fabricated and the metallic tool surface.

Durability of the composite tooling material system affects the surface finish of the part, vacuum tightness for efficient bagging, and recurring tooling costs associated with surface refinishing or replacement. As epoxy resin systems undergo repeated thermal cycling (ambient to 350°F), the polymer

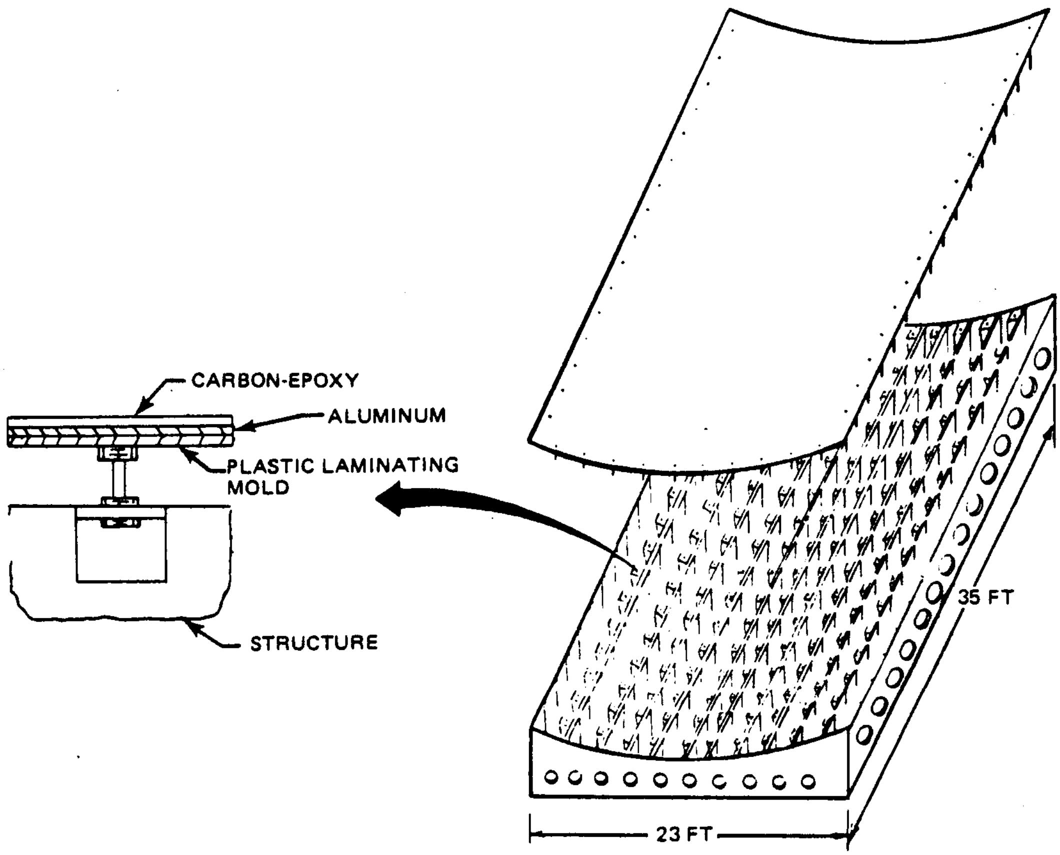


FIGURE 4-7. SKIN TOOL WITH ADJUSTABLE STUDS FOR CONTOUR CONTROL

molecules link closer together to craze the epoxy and ultimately cause a physical degradation of the tool. Techniques have been developed for resurfacing degraded tools, but eventually the tools will require replacement. Metallic tools normally withstand the shop handling environment and the repeated cure cycles much better than composite tools.

Tool Fabrication — Currently, tools are fabricated in a time-honored method, with a plaster master formed by hand from guide templates. This is followed by a series of intermediate tools to eventually produce the final cure tools. Innovative concepts of tool fabrication are greatly needed to produce cost-effective tooling for this fuselage.

Some areas of tool fabrication that could be investigated to improve this process include a thermally and dimensionally stable master tooling material that could be readily machined for direct layup of carbon-epoxy tooling material. It could be utilized in the electroforming nickel process and as a metal-sprayable tooling base. Thermal and residual stress analysis might also be integrated into tool design, and predetermined tool nesting data might be based on thermal mass analysis.

The potential benefits of improving tool fabrication technology and tooling material selection include: (1) improved quality of final parts due to better matching of the tool to the part and process, (2) improved repeatability and reliability of the process due to early design analysis, (3) reduction of the overall cost due to the reduction in fabrication operation, and (4) increased tool life and higher temperature resistance if proper tooling materials are selected and utilized.

Tool Material Comparisons — The tooling materials used to fabricate the stiffened fuselage skin tool (20 by 40 feet) must have good dimensional stability and be very durable. In order to hold tolerances on the skin to 0.020 inch, the tool itself must hold tolerances of ±0.010 to be acceptable. Materials being investigated are noted in Table 4-4.

TABLE 4-4
THERMAL PROPERTIES OF CANDIDATE MATERIALS

MATERIAL	THERMAL COEFFICIENT OF EXPANSION (α)*	THERMAL CONDUCTIVITY (k)**
ALUMINUM: 6061-T6	12.5×10^{-6}	1.53
STEEL: 1020	6.1×10^{-6}	0.33
ELECTROFORMED NICKEL	5.6×10^{-6}	0.24
CARBON-EPOXY: FIBERITE MXG-7620	1.6×10^{-6}	0.15
CARBON/BISMALEIMIDE (BMI)	1.7×10^{-6}	0.16
GLASS/HIGH-TEMP RESIN: REN 4015	4.4×10^{-6}	–
CERAMIC: RESCO RS-17A	5.0×10^{-5}	0.06

*UNITS FOR α: IN./IN./°F
**UNITS FOR k: Btu IN./FT^2.h.°F

A matrix was developed to assess the overall ratings of the candidate tooling materials (see Table 4-5). Based on overall ratings, the materials which appear to be the best in this preliminary evaluation are carbon-bismaleimide, carbon-epoxy, glass/high-temperature epoxy, and steel.

TABLE 4-5
PRELIMINARY COMPARATIVE RATING OF TOOLING MATERIALS

TOOLING MATERIAL	DIMENSIONAL STABILITY[1]	HEAT TRANSFER[2]	DURABILITY (SURFACE DEGRADATION)[3]	COSTS		HANDLING	RATING
				FABRICATION[4]	MATERIAL		
ALUMINUM	5	1	2	3	3	3	2.8
STEEL	3	2	1	4	3	3	2.7
ELECTROFORMED NICKEL	3	2	1	5	5	4	3.3
CARBON-EPOXY	1	3	4	2	4	1	2.5
CARBON-BMI	1	3	3	2	4	1	2.3
GLASS-EPOXY	2	3	4	2	2	2	2.5
CERAMIC	1	5	5	3	1	5	3.3

[1]DIMENSIONAL STABILITY IS DIRECTLY RELATED TO COEFFICIENT OF THERMAL EXPANSION.
[2]HEAT TRANSFER IS RELATED TO THERMAL CONDUCTIVITY.
[3]LOWEST RATING NUMBER INDICATES THE BEST MATERIAL FOR DURABILITY CONSIDERATIONS AND FINAL OVERALL RATING
[4]WEIGHTING FACTORS ARE NOT INCLUDED.

On the basis of dimensional stability, carbon-bismaleimide and carbon-epoxy tooling materials look good because they have approximately the same thermal coefficient of expansion as the manufactured skin, but from a durability standpoint carbon-epoxy tools generally tend to degrade and therefore lose vacuum after being thermally cycled. The carbon-bismaleimide tools may be subjected to thermal gravimetric analysis to determine surface degradation since no long-term experience with this tooling material is available. Steel tools, on the other hand, have excellent durability but have a higher thermal coefficient of expansion that would have to be accounted for in the tool design. If a fuselage panel were to be fabricated today, steel tooling would be selected for the skin tool because of durability and possibly carbon-bismaleimide tooling for subassembly tooling because of its ease of fabrication for complex geometries. The rating method used does not include a weighting factor which affects the final selection of materials.

Part Fabrication

Skin Panel — These low-curvature panels will be approximately 20 by 30 feet long and 0.07 inch thick with localized doublers and a picture frame reinforcement. Because of their size, hand layup would be impractical; thus, the layup must be automated utilizing a tape-laying machine or equivalent. Material control, issuing and dispensing, should be automated so that material is utilized on a first-in, first-out basis. Actual skin layup could be accomplished on a low-curvature metal or composite tool, or possibly a flat tool with curvature being set at the time of final assembly. The latter may or may not be feasible due to interlaminar stresses induced during skin forming from the flat to the proper curvature. This would have to be evaluated in a separate study. Localized doublers and picture frame reinforcing could be plied up manually and positioned on the plied-up skin for co-curing.

Stiffened Fuselage Skin Panels — The basic fuselage skin panels will have longerons and shear tees attached to the skin by integrally curing the longerons and secondarily bonding precured shear tees. Precise longeron spacing must be controlled by the curing and bonding tools to assure fit-up into the assembly fixtures, minimize shimming requirements, and control alignment of longerons between barrel sections. Assuming a nominal panel size of 20 by 30 feet, the tooling must provide loft line control and longeron alignment. Techniques have been developed to hold longeron positions within 0.060 inch over a 5-foot length. Improvements must be made to hold the same tolerance over a 30-foot length.

One method of achieving this control is to reverse normal tooling approaches by using male tooling to the inside skin loft dimensions and providing machined grooves in the tool to hold longeron positions (Figure 4-8). Precured longerons can be placed into the tool. The skin would be laminated over these members with a layer of film adhesive to bond the longerons to the skin during the skin cure cycle. This procedure would assure a perfect match between skin and longerons. A thin caul sheet placed over the skin during cure has proven successful in providing a smooth aerodynamic exterior surface in subscale tests at Douglas. One minor drawback of this concept is the possibility of small tolerance errors in skin thickness moving to the exterior surface. For skins approximately 0.070-inch thick, this error could accumulate to 0.008 inch, an acceptable value.

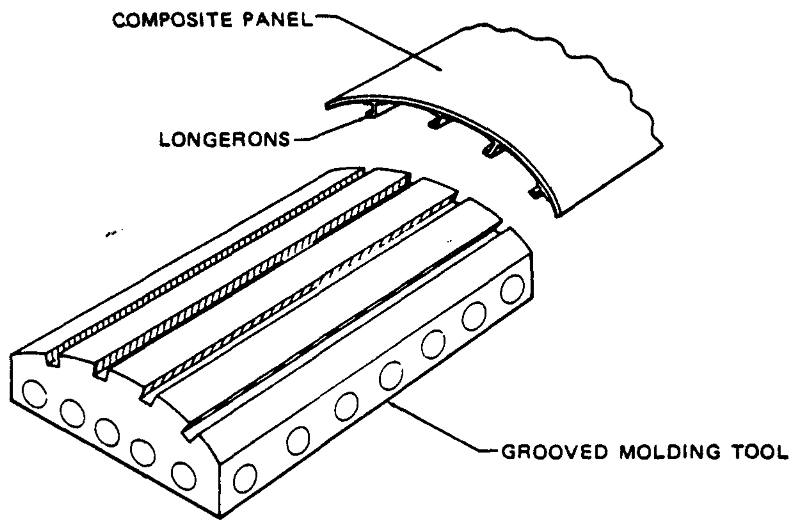


FIGURE 4-8. MALE MOLDING TOOL WITH LONGERON-POSITIONING GROOVES

The shear tees could also be positioned and co-cured to the skin if the cutouts for the continuous longerons were eliminated. Because these cutouts now present an unsupported discontinuity under the skin, it is not possible to compact the skin at these junctions (Figure 4-9). If the shear tees cannot be redesigned, then the cured skin and cured shear tees must be secondarily bonded on a male tool. The shear tees are over four orders of magnitude stiffer than the skin (3 inches versus 0.080 inch). By supporting the stiff shear tees in the tool, the relatively compliant skin will readily form to the shear tee curvature. Traditional bonding methods, employing female tools supporting the skin, attempt to force the stiff members into the curvature of the skin. This approach inevitably causes mismatch of the skin and shear tee flange which requires shimming. There appears to be distinct advantages in using male tooling to the inside skin loft line.

In the production of the AV-8B composite wing, the curing tools for the wing skin were tooled to the inside surface. Substructure spars and ribs were tooled to the same inside surface to assure a close match of the components. The AV-8B wing skin was then mechanically fastened to the substructure.

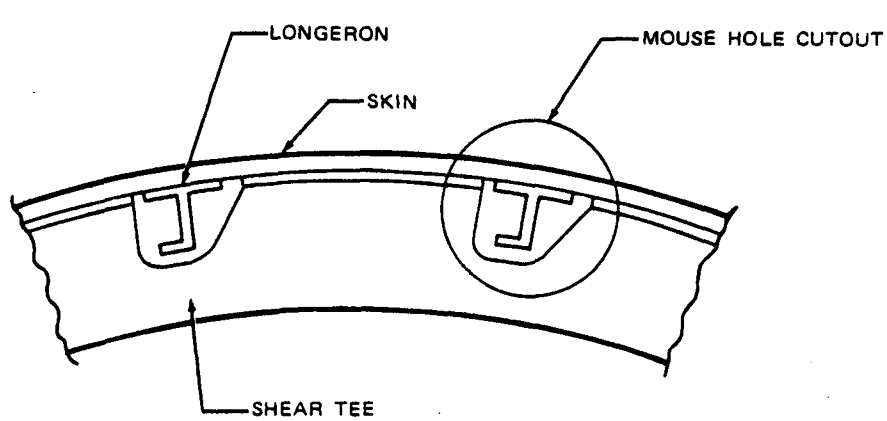


FIGURE 4-9. CUTOUT AREAS OF SHEAR TEE

Fabrication of Detailed Parts — Structural members such as floor beams, panel stiffeners, and straight splice doublers are required in large numbers. Because of the constant cross sections of most of the details, several manufacturing processes could be used to rapidly produce parts and eliminate expensive hand layup. Typical detail quantities and part sizes for a wide-bodied aircraft are shown in Table 4-6.

TABLE 4-6
TYPICAL PARTS FOR WIDE-BODIED AIRCRAFT

ITEM	SIZE	QUANTITY/AIRCRAFT
FLOOR BEAMS	10 x 240 IN.	100
FUSELAGE PANEL LONGERONS	2 x 360 IN.	400
FUSELAGE CIRCUMFERENTIAL SHEAR TEES	3 x 80 IN.	300

The feasibility of press curing floor beams has been investigated utilizing the filament-winding process to lay up the beams in high volume. Carbon-epoxy floor beams 48 inches long were successfully fabricated and press cured to demonstrate the process. During the test, the need for the following became apparent:

1. A fast-cure resin system with adequate mechanical properties and staging to permit handling and storage until ready for cure.
2. A die configuration permitting rapid throughput and adequate pressure distribution on all part surfaces, resulting in a void-free cross section.
3. Filament-winding methods that enable multiple parts to be fabricated from one winding mandrel.

Each of these issues requires validation on a scale large enough to prove that press curing is a reliable manufacturing process. Special equipment required includes a filament-winder (McClean-Anderson W-60) with a 25-foot bed length, hot pressure (325°F), 400-ton capacity, and a size of 260 by 30 inches.

The pultrusion process is an alternative to press curing stiffeners and other composite details of constant cross sections. In this process, oriented fibers and matrix resin are pulled through zone-heated dies. The continuous output from the shaped die is similar to extended metallic shapes, such as angles, tees, and channels. Pultrusion has been used to produce fiberglass/polyester shapes for electrical conduits and for nonstructural molding applications. The normal fiber feed into the die was primarily 0-degree orientation, combined with glass mat or veil materials. Resin pickup was not critically controlled, using a dip tank impregnator to apply polyester or vinylesters. These resin systems were selected because of their quick-curing, low-exothermic properties. Epoxy resin systems have not been successful for pultrusion applications because of long cure cycles and the evolution of volatiles causing porosity. Quick-curing modified epoxies such as U.S. Polymeric E7K7 have the potential for use in pultrusion. Some reduction in mechanical properties is incurred (10 to 15 percent).

The forces and temperature distribution within the die are critical to the quality of the resulting lamination. For shapes such as J- or tee-sections, compressive forces are required normal to all surfaces. This has been controlled by movable die sections or by tapering the cavity section to gradually induce compressive forces as the material travels through the die. Further work must be accomplished in die geometry, design, and rheological process control to quantitatively determine methods of providing pressure.

Fiber orientation for the pultrusion process can most easily accommodate 0-degree direction as this is also the pulling direction. When higher angle fibers (± 45 degrees) are included to carry shear loads in the part, the tensile forces from the pultrusion process tend to rotate the fibers toward 0 degree. Several techniques have been employed to reduce this effect by plying the 45-degree fibers within 0-degree plies so the predominant tensile loads are carried by the 0-degree fibers.

Because pultrusion can occur at 3 to 12 inches per minute, the potential for continuous production is very attractive. The indicated problem areas must be solved to allow the process to produce quality parts. Pultrusion companies around the country can be placed under contract to develop equipment and processing techniques for carbon fiber parts.

Assembly

Out-of-Autoclave Bonding

In the area of fabrication and process, dimensional tolerance control is a major concern. Another concern involves secondarily bonding structural details such as the longerons, frame shear tees, and clips onto the skins. Close tolerance must be maintained to minimize shimming and splice rework during assembly of stiffened panels. A large assembly/bonding tool (Figure 4-10) is recommended to locate frames and longerons accurately, and then by utilizing pneumatic pressure and heat, secondarily bond the details to a precured skin in an out-of-autoclave tool. Some advantages seen with this concept include reduced energy and labor costs, elimination of expensive, time-consuming autoclave operation, and reduced risk of scrapping an assembly (bag blowout, etc.).

Accurate fabrication of the circumferential stiffeners (shear tees) is a critical issue controlling bonding line thickness. Variable skin thickness as in doublers and the buildup area for window belt and doors are of main concern because shear tees must be formed with joggles in these areas. When considering the hundreds of shear tees needed to fabricate a composite fuselage, tooling for each tee or group of tees could be very costly. A thermoplastic composite shear tee could be used to eliminate the problems caused by variable skin thickness. The thermoplastic composite shear tees could be locally heated at the flange area, then positioned and bonded in the out-of-autoclave tool. The heated flange of the shear tees would conform to any variable in skin thickness and eliminate any excess bond-line gap. Technology to adhesively bond the thermoplastic must be developed to support this approach.

Automation of Production

The most labor-intensive operation in a composite assembly is the actual layup of the composite parts. Panel assemblies that are installed into a fuselage barrel would be approximately 20 by 40 feet. This poses a very difficult handling problem. The ply-by-ply layup of a fuselage skin would most likely be

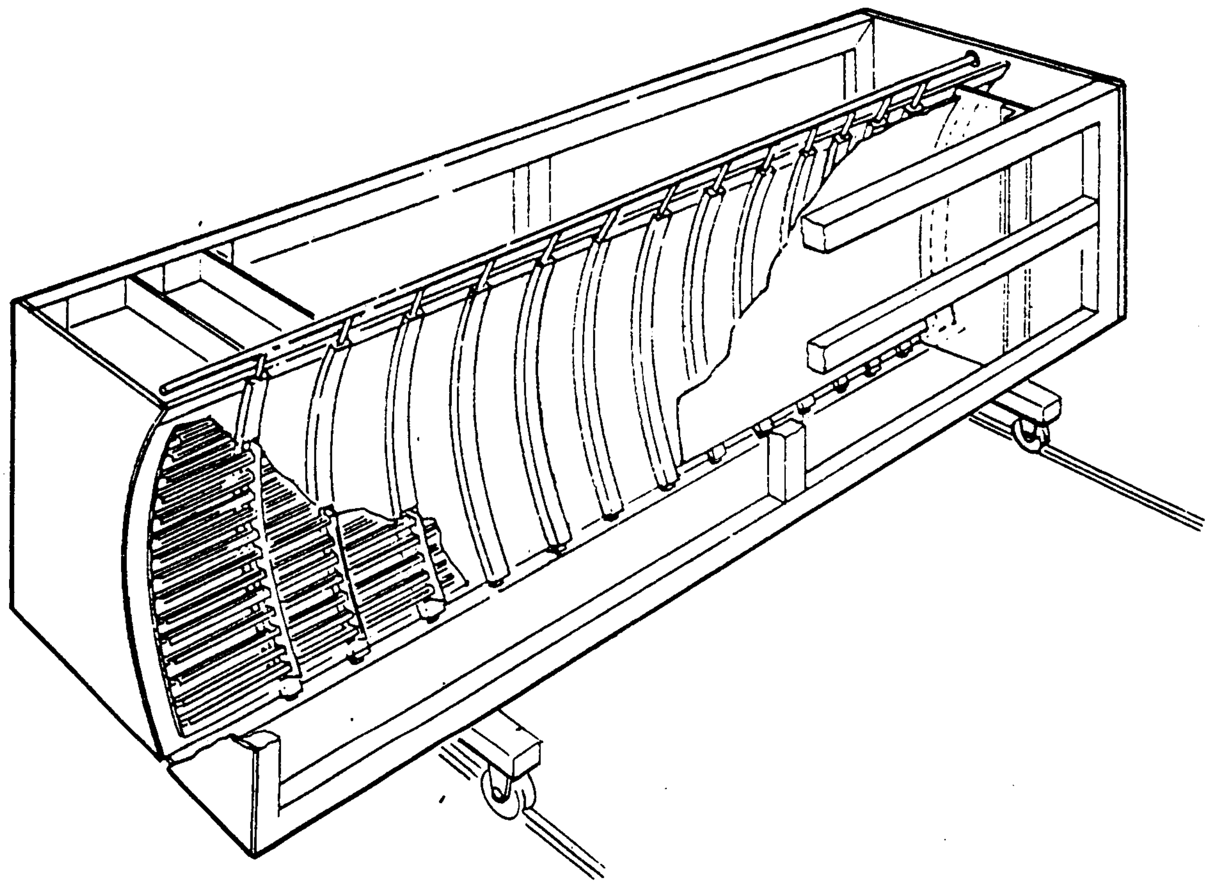


FIGURE 4-10. ADVANCED BONDING TOOL

automated by a computer-controlled tape laying machine, with the details being fabricated on broadgoods cutting and layup equipment. The tape-laying machine dispenses the composite material from reels which normally have widths of 1 inch, 3 inches, or 6 inches. The material is cut to the correct lengths by shear or diagonal cutters and laid up on flat or contoured up to ± 15-degree sloped surfaces. This contour limitation is in the development stage at Vought Corporation, which has been developing a 7-axis tape-laying machine with Cincinnati Milicron Corporation (Figure 4-11). When the limitations of today's tape-laying machines are overcome, one could feasibly lay up shims, doublers, and buildups for cutouts directly into the large-radius fuselage skin tool. This would eliminate the labor-intensive step of transferring a large, flat layup to a contoured tool.

When making smaller details (longerons, shear tees, splices, intercostals, and the like), the material is cut and prepared for layup. A computer software package on a graphics terminal is used to nest the parts for cutting, with each resulting cut piece having the correct fiber direction. The same software package then generates numerical control instructions for cutting out the pieces on a numerical control saw or water-jet cutting device. The cut-out pieces of material would then be transferred to a layup area via an automated material handling system. Here, robotics could be utilized to lay up the plies, "kit" the laminates, and transfer to the curing tool. Each of the processes described would be automated, with computerized numerical control being used on the tape-laying machine, cutters, and robotics.

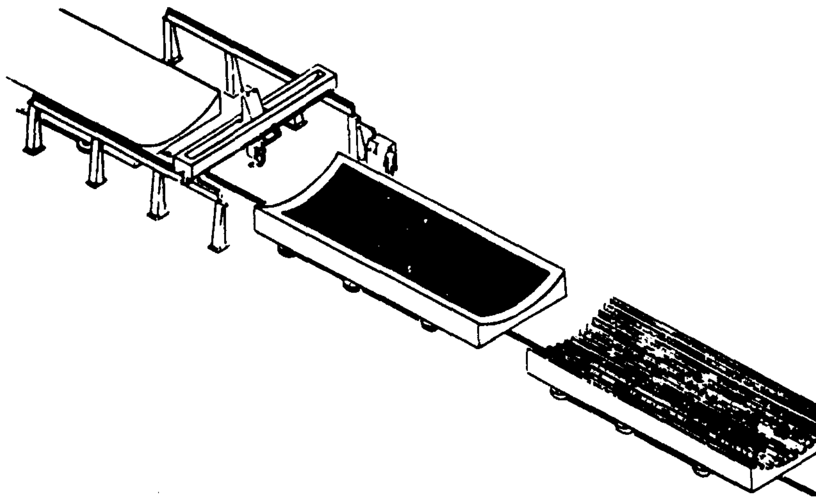


FIGURE 4-11. 7-AXIS TAPE LAYING MACHINE

A distributed numerical control system delivers data files to the computerized numerical control machines and collects information from other machines. The system could also be utilized to control the automated material handling equipment used to move the composite material between the processes.

Automation could also be applied to the postcure operations of cutting, trimming, and drilling. Robots could be used to load and unload the parts, with computer-controlled cutting done by water-jet, reciprocating knife, saw, or router. Since these operations are both physically and environmentally hazardous, automation is easy to justify in these areas.

Automation of assembly could also have great impact on the reliability of production flow and also decrease costs by reducing labor, improving material handling and flow requirements, and decreasing assembly rework. An automated work station to assemble fuselage barrel sections should include material handling, shimming/splice and backup alignment, robotic drilling, robotic hole inspection, installation of fasteners, frame splice installation, and assembly inspection.

The high arch is an automated assembly cell concept where panels are loaded into a large assembly jig and robotically fastened together and inspected (Figure 4-12). This will avoid the labor needed to join two 30-foot-long panels and the problems associated with drilling and joining composite panels (health and machine hazards from dust generated, hole alignment and breakout, and others).

Under the concept, a five-axis robotic gantry system moves on expandable tracks and has the capability to assess shimming and dispense liquid or hard shim for cured panels. This cell optically aligns backup structure for drilling and inspecting holes, and installing fasteners.

After the panels are placed in the assembly jig and trimmed, a profiling device automatically assesses the amount of shimming needed. After shimming, automatic hole drilling and hole inspection (panels and splices) occur. Fasteners are then installed automatically and inspected for flushness. This two-thirds barrel section is rotated on the assembly jig for installation of the final one-third of the upper barrel.

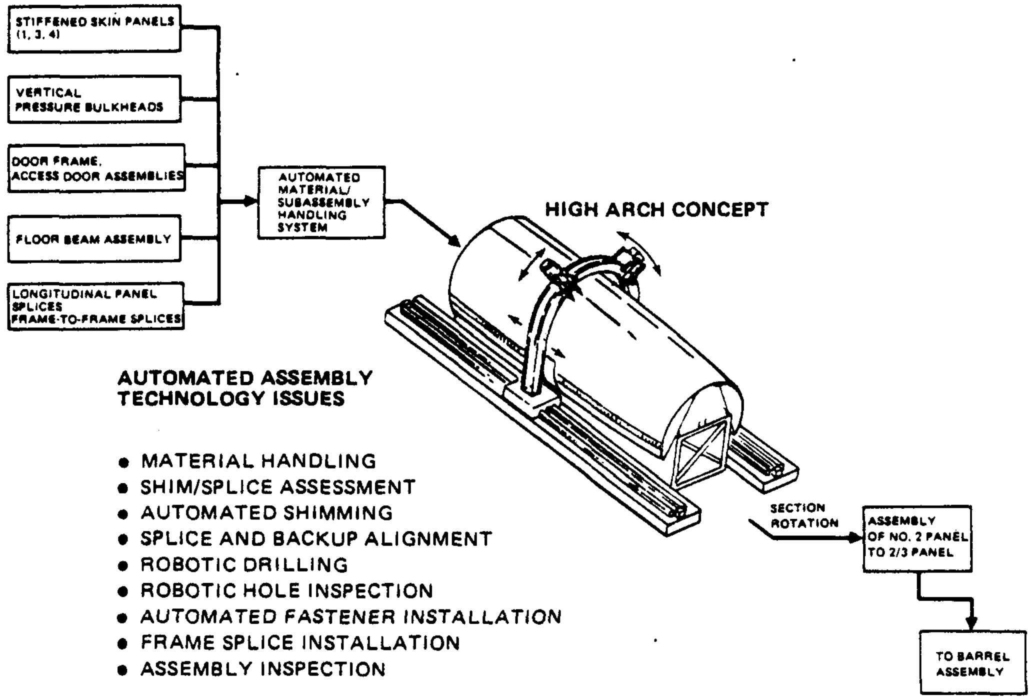


FIGURE 4-12. TECHNOLOGY ISSUE – AUTOMATED ASSEMBLY

Thermoplastic Composite Materials

Thermoplastic matrix materials have long been recognized as an improved candidate for composite material applications in aerospace structures. Now that problems such as solvent degradation and limited temperature capability have been solved with materials such as polyetheretherketone (PEEK), polyphenylene sulfide (PPS), and polyether sulfone (PES), manufacturing and fabrication issues must be addressed in order to apply this technology.

Some of the advantages of carbon-thermoplastic matrix systems include lower fabrication cost because of apparently unlimited room temperature storage, elimination of clean room production facilities, and very short process cycles. Other advantages include less susceptibility to microcracking at fastener holes, less sensitivity to moisture absorption, and delamination which can be re-fused rather than filled. These advantages, combined with the systems' increased toughness and improved burn and smoke characteristics, make them very advantageous for use in a composite fuselage.

There are, however, many manufacturing issues that must be resolved in order to utilize this material. The dimensional stability of tooling material is a critical issue because of the thermal states encountered due to the high forming temperature and quick cool-down of parts (quenching rates). PEEK material, for example, must be heated to 725°F, then formed and quenched at a rate of 40° to 80°F per minute. This is done to stabilize the crystalline formations in the matrix in order to obtain optimum properties from this composite system.

Automated tape layup and filament-winding of thermoplastic matrix composites are also technology issues that must be resolved before these materials can be utilized in large structural parts. Localized heat and pressure must be applied at the point of contact of the preimpregnated material to the part being laid up. This could be done by utilizing highly localized infrared heat or a controlled laser, and roller pressure from a gantry arm.

Optimally, a fuselage could therefore be laid up, consolidated, and cured directly on a layup tool or cured during filament-winding.

Forming — In fabrication of thermoplastic composite components, forming technology is a critical issue in itself. Some of the forming processes which need to be assessed are: (1) placing heated material into matched warm molds in press (transfer molding), (2) vacuum forming, (3) hot compression molding, (4) cold stamping/cold braking, (5) roll forming, (6) comoforming — placing heated resin into preform material, (7) filament-winding, and (8) continuous pultrusion.

The promise of high formability for carbon-PEEK must be substantiated in light of the low elongation to failure of the carbon fiber (1.0 to 1.7 percent). Development forming work on woven carbon fiber impregnated with PPS has indicated that hat sections and simple curved sheet can be formed by preheating the workpiece to 550°F and forming on 350°F dies. Deformed PPS sheet was restraightened by heating under low pressure without any apparent effect on the quality of the sheet.

Fastening — Drilling and machining of thermoplastics could cause unique problems not shared with thermosetting epoxies. If the hole-drilling process produces sufficient heat to cause a local transition above the material's glass transition (Tg) or melting temperature (Tm), then the subsequent cool-down on the hole inside diameter will control the crystalline form of the matrix. Rapid cool-down would produce a highly amorphous structure with low strength.

SECTION 5
PROGRAM OPTIONS

Objectives of this study program were to define a course of action that will resolve the technology gaps and provide the data and experience to support a commitment to produce composite fuselage structure. This section examines various options that are available, compares them, and selects a preferred set of alternatives for the composite fuselage development program.

A work breakdown structure is presented in Table 5-1 that reflects an overall concept of the development program which would provide the necessary data, experience, and demonstration of a capability to support a commitment by the manufacturers, airlines, and DOD. The program is divided into the following three phases: Phase I — Design Development, Phase II — Structural Verification, and Phase III — Flight Service Evaluation. The program options to be evaluated for each phase are shown in Table 5-2.

SELECTION OF BASELINE AIRPLANE

A baseline aircraft was selected for the fuselage study in order to compare the composite fuselage with conventional aluminum structure, to provide a data base for the conceptual design, to form a basis for the technology assessment, to define the scope of the development program, and to determine the facilities and equipment requirements for manufacture of composite fuselage structure of a large transport aircraft.

In the study, only McDonnell Douglas aircraft were considered as candidates for the baseline in order to have immediate access to the airplane data base. Both commercial and military models were included in the evaluation. A set of criteria was compiled to describe the desirable characteristics that the baseline aircraft should possess to best serve study purposes:

- It should be a large transport aircraft with an available data base for the composite fuselage structural arrangement and conceptual design.
- The design criteria, loads, and features of the baseline structure are sufficiently representative to identify the technical issues associated with the design and manufcture of a next-generation composite fuselage.
- A data base exists to compare the cost and weight of composite fuselage structure with conventional structure.
- Civil or military specifications exist to regulate the design.
- A flightworthy aircraft should be available to conduct a flight service evaluation of composite structure.

Both civil and military transport models were evaluated on the basis of design loads and criteria, functional design features, and in-service mission profiles.

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TABLE 5-1
WORK BREAKDOWN STRUCTURE

WBS 1000 DESIGN DEVELOPMENT

1100 ADVANCED ENGINEERING

1110 BASELINE VEHICLE SELECTION
1120 DESIGN DATA, LOADS, CRITERIA
1130 INTERFACE CONTROL REQUIREMENTS
1140 STRUCTURAL ARRANGEMENT

1200 ENGINEERING TECHNOLOGY DEVELOPMENT

1210 ANALYSIS METHODS
1220 CONCEPTUAL DESIGN
1230 MATERIALS AND PROCESS DEVELOPMENT
1240 CRITICAL TECHNOLOGY DEVELOPMENT
1250 BIRD-STRIKE TECHNOLOGY DEVELOPMENT
1260 EME TECHNOLOGY DEVELOPMENT

1300 MANUFACTURING TECHNOLOGY DEVELOPMENT

1310 TOOLING METHODS
1320 FABRICATION METHODS
1330 ASSEMBLY METHODS

1400 PRELIMINARY DESIGN

1410 STRUCTURAL OPTIMIZATION
1420 TRADE STUDIES
1430 DESIGN LAYOUTS
1440 STRESS ANALYSIS
1450 DESIGN ASSESSMENT
- PRODUCIBILITY
- INSPECTABILITY
- MAINTAINABILITY AND REPAIRABILITY
- WEIGHT EFFICIENCY
- COST

1500 DEVELOPMENT TESTS

1510 TEST PLANS
1520 TEST SPECIMEN DESIGN
1530 DESIGN AND FABRICATION OF TEST SETUP
1540 INSTRUMENTATION AND INSTALLATION
1550 CONDUCT TESTS
1560 REPORT TEST RESULTS

1600 PANEL TESTS

1610 TEST PLANS
1620 DESIGN TEST SPECIMENS
1630 DESIGN AND FABRICATION TEST SETUP
1640 INSTRUMENTATION AND INSTALLATION
1650 CONDUCT TESTS
1660 REPORT TEST RESULTS

WBS 2000 DESIGN VERIFICATION

2100 VERIFICATION TEST PLANS

2110 STATIC, DURABILITY AND DAMAGE TOLERANCE TESTS
2120 BIRD STRIKE TEST
2130 ACOUSTICS TESTS
2140 IMPACT DYNAMICS TESTS
2150 ELECTROMAGNETIC EFFECTS TESTS

TABLE 5-1
WORK BREAKDOWN STRUCTURE (CONTINUED)

WBS 2000 DESIGN VERIFICATION (CONTINUED)

2200 TEST SPECIMEN DESIGN

- 2210 DETAIL DRAWINGS
- 2220 STRESS ANALYSIS
- 2230 MATERIAL AND PROCESS SPECIFICATIONS
- 2240 TASK ASSIGNMENT DRAWING PREPARATION

2300 MANUFACTURING

- 2310 MANUFACTURING PLAN
- 2320 TESTS
- 2330 PRODUCIBILITY VERIFICATION
- 2340 TOOL DESIGN
- 2350 TOOL FABRICATION
- 2360 FABRICATION OF COMPOSITE DETAILS
- 2370 FABRICATION OF METAL DETAILS
- 2380 SPECIMEN ASSEMBLY

2400 TEST

- 2410 DESIGN OF TEST SETUP
- 2420 FABRICATION OF TEST SETUP HARDWARE
- 2430 INSTRUMENTATION AND INSTALLATION OF SPECIMEN INTO THE TEST SETUP
- 2440 CONDUCT VERIFICATION TESTS
- 2450 REPORT TEST RESULTS

2500 DESIGN EVALUATION

WBS 3000 FLIGHT SERVICE EVALUATION

3100 FLIGHT SERVICE CERTIFICATION PLAN

3200 ACQUISITION OF AIRCRAFT

3300 ENGINEERING DESIGN

- 3310 COMPOSITE STRUCTURE DRAWINGS
- 3320 AIRPLANE REWORK DRAWINGS
- 3330 INSTALLATION DRAWINGS
- 3340 STRESS ANALYSIS

3400 TOOL DESIGN AND FABRICATION

- 3410 COMPOSITE STRUCTURE FABRICATION
- 3420 METAL STRUCTURE FABRICATION
- 3430 ASSEMBLY TOOLS

3500 MANUFACTURING FABRICATION

- 3510 COMPOSITE STRUCTURE FABRICATION
- 3520 METAL STRUCTURES FABRICATION

3600 AIRCRAFT INSTALLATIONS

3700 TEST PROGRAMS

- 3710 STRUCTURAL SUBCOMPONENT TESTS
- 3720 AIRPLANE SYSTEMS CHECKOUT
- 3730 FLIGHT TEST

3800 FAA DOCUMENTATION

TABLE 5-1
WORK BREAKDOWN STRUCTURE (CONCLUDED)

WBS	3000	FLIGHT SERVICE EVALUATION (CONTINUED)
	3900	FLIGHT SERVICE EVALUATION
	3910	PLANS
	3911	INSPECTION PLANS
	3912	MAINTENANCE PLANS
	3913	STRUCTURAL REPAIR MANUAL
	3920	FLIGHT SERVICE
WBS	4000	PROGRAM MANAGEMENT
	4100	PROJECT MANAGEMENT
	4200	PROJECT ADMINISTRATION AND CONTROL
	4300	DOCUMENTATION
	4400	REVIEWS
	4500	TECHNOLOGY TRANSFER

TABLE 5-2
PROGRAM OPTIONS

PHASE	TASKS	OPTIONS
	DEFINE THE SCOPE OF A COMPOSITE FUSELAGE DEVELOPMENT PLAN	• WHICH AIRCRAFT TO USE FOR THE BASE-LINE VEHICLE • PHASE I ONLY • PHASE I PLUS PHASE II • PHASE I PLUS PHASE II PLUS PHASE III
I DESIGN DEVELOPMENT	• ADVANCED ENGINEERING • ENGINEERING TECHNOLOGY DEVELOPMENT • MANUFACTURING TECHNOLOGY DEVELOPMENT • PRELIMINARY FUSELAGE DESIGN • DEVELOPMENT TESTS • PANEL TESTS	NONE
II STRUCTURAL VERIFICATION	• TEST PLANS • TEST DRAWINGS • MANUFACTURING • TEST • DESIGN EVALUATION	• FUSELAGE SECTIONS TO BE USED FOR THE GROUND TEST ARTICLE • IMPACT DYNAMICS (TBD)
III FLIGHT SERVICE EVALUATION	• CERTIFICATION PLAN • PANEL DESIGN AND FABRICATION • AIRCRAFT INSTALLATIONS • TEST PROGRAMS • FAA DOCUMENTATION • FLIGHT SERVICE PLANS • FLIGHT SERVICE	• LOCATION AND SIZE OF FUSELAGE PANEL

In-flight loadings would be comparable. Both aircraft are designed to 2.5 g limit maneuver load, and gust loads are dependent upon the airplane configuration, weight, altitude, and speed since the gust environment is a natural phenomenon. Low-level or terrain-following design criteria would impose a more severe repeated loads criterion for the military version. Ground loads criteria are more severe for a military transport assuming STOL operations and unimproved runways. Flutter speed margins are slightly higher for the civil transport.

Damage tolerance criteria have the same objective: continued safe flight in the presence of damaged structure — and the structure will probably be damaged at some time in the life of the aircraft. An airplane designed to meet FAR 25 damage tolerance criteria should nearly satisfy Air Force damage tolerance specifications. A civil transport option will not address survivability criteria for projectiles, nuclear blasts, and other military threats, although lightning protection and damage tolerance criteria would provide a certain level of survivability. Other military programs will adequately address the technology needed for designing composite structure to meet survivability criteria.

Functional differences between military and civil aircraft are found in the following components:

	Military	Civil
Wing	Top of Fuselage	Mid-Fuselage
Main Landing Gear	Fuselage-Mounted	Wing-Mounted
Floor	Lower and Rugged for Special Military Cargo	Mid-Fuselage for Passenger Seating and Lower Cargo Bay

The three civil and three military transport aircraft which were evaluated for the baseline aircraft are shown in Figure 5-1. The MD-100 and the KC-10 are derivative models of the DC-10, and the MD-80 is a derivative of the DC-9. A number of the Air Force KC-10 tanker/cargo aircraft are in service. Two prototype YC-15 aircraft were built and flown, and are currently stored in Arizona. The C-17 is currently being developed for the Air Force. The D-3300 is an advanced engineering commercial design.

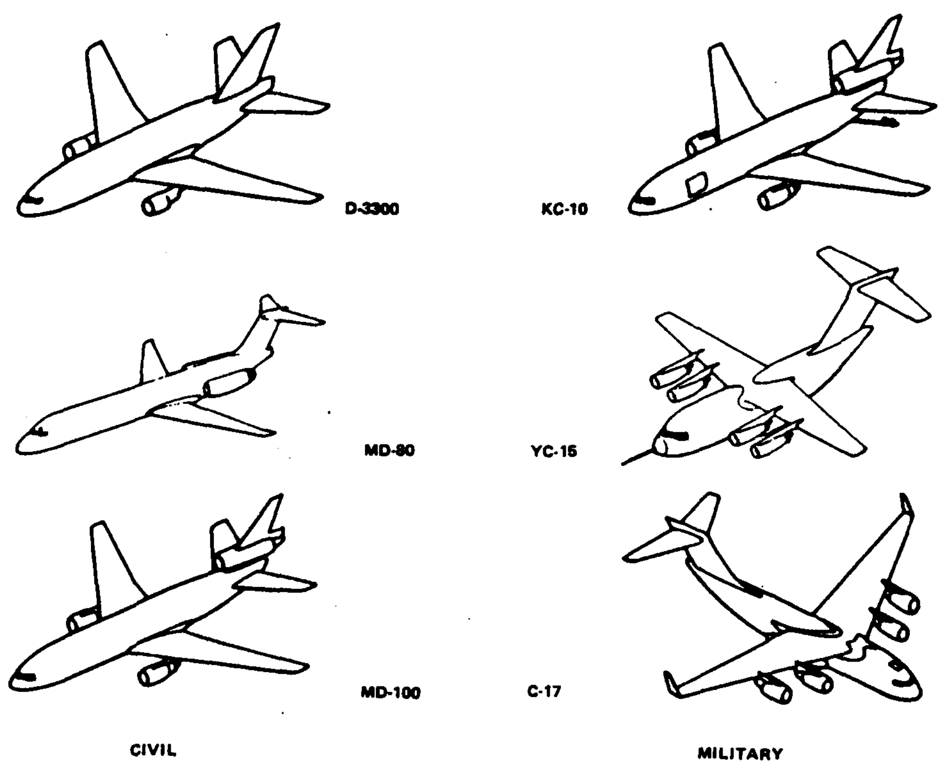


FIGURE 5-1. CANDIDATE BASELINE AIRCRAFT

The six candidate aircraft were evaluated on the basis of how well the program objectives would be achieved with each model. The results of the evaluation are shown in Table 5-3. The MD-100/KC-10 versions scored the highest for the baseline aircraft. The flight service evaluation can be performed on either a DC-10 or a KC-10 aircraft.

TABLE 5-3
SELECTION OF BASELINE AIRCRAFT

DECISION: TO SELECT A LARGE TRANSPORT AIRCRAFT AS A BASELINE VEHICLE FOR A CONTRACTUAL COMPOSITE FUSELAGE DEVELOPMENT PLAN

	WEIGHTING FACTOR	ALTERNATE CHOICES						BASIS FOR HIGHEST RATING (SCORE / WEIGHTED SCORE)
		CIVIL			MILITARY			
		MD-80	MD-100	D-3300	KC-10	YC-15	C-17	
1. CONDUCT A PRELIMINARY DESIGN OF A COMPOSITE FUSELAGE	10	10 / 100	10 / 100	7 / 70	10 / 100	7 / 70	8 / 80	• MOST COMPLETE AND AVAILABLE DESIGN DATA BASE
2. COMPARE PRODUCIBILITY AND COST TO CONVENTIONAL STRUCTURE	5	10 / 50	10 / 50	5 / 25	10 / 50	5 / 25	5 / 25	• MOST AIRCRAFT BUILT INCLUDING DERIVATIVE MODELS
3. ESTIMATE FUSELAGE WEIGHT SAVINGS COMPARED TO CONVENTIONAL DESIGN	5	10 / 50	10 / 50	7 / 35	10 / 50	8 / 40	7 / 35	• ACTUAL WEIGHTS
4. CONDUCT LARGE-SCALE DEMONSTRATION TESTS	8	10 / 80	10 / 80	9 / 72	10 / 80	9 / 72	7 / 56	• LOWEST COST TO BUILD SPECIMEN AND CONDUCT TEST
5. ACQUIRE DESIGN DATA BASE	8	9 / 72	9 / 72	10 / 80	9 / 72	7 / 56	10 / 80	• HAS STRUCTURAL FEATURE OF A 1990s AIRCRAFT
6. ACQUIRE MANUFACTURING EXPERIENCE AND DATA BASE	8	9 / 72	9 / 72	10 / 80	9 / 72	7 / 56	10 / 80	• HAS STRUCTURAL FEATURE OF A 1990s AIRCRAFT
7. TIMELY EXECUTION OF PROGRAM	7	7 / 49	10 / 70	10 / 70	10 / 70	10 / 70	7 / 49	• SHORTEST SCHEDULE
8. CONDUCT A FLIGHT SERVICE EVALUATION	10	10 / 100	10 / 100	0 / 0	10 / 100	5 / 50	0 / 0	• MOST REPRESENTATIVE UTILIZATION
*FLIGHT SERVICE EVALUATION		Σ 573	594		594	439		
*NO FLIGHT SERVICE EVALUATION		Σ 473	494	432	494	389	406	

PHASE I — DESIGN DEVELOPMENT

There are no program options defined for Phase I. The baseline aircraft should be consistent throughout all three phases to minimize repetitive work and to form a basis for comparison and correlation. Phase I tasks are planned to resolve the technology gaps, to design a composite fuselage structure, and to prepare for accomplishing later tasks in Phases II and III. The activity is considered essential for all program activities.

PHASE II — STRUCTURAL VERIFICATION

The technology developed in Phase I is extended and verified by testing large-scale structures in Phase II. The program options are focused on the configuration of the fuselage barrel section for the ground test articles for the static ultimate, durability, and damage tolerance tests. A full length fuselage option was excluded on the basis of high cost and a lengthy schedule. The barrel options are shown in Figure 5-2 and the results of the selection evaluation are shown in Table 5-4. The merit functions for the value analysis are the verification of structural integrity, the validation of manufacturing technology, the effect on program costs and schedules, and for a demonstration program which will best impart confidence to the manufacturer and customer to make the commitment to production and utilization.

No options were defined for the bird strike test. The test article must include the frontal area of the fuselage vulnerable to bird strikes which would affect continued safe flight.

One large-scale test article for the impact dynamics test was defined for illustrative and planning purposes. A range of options should be developed after the development test of impact dynamics in Phase I gives a better understanding of this vital issue.

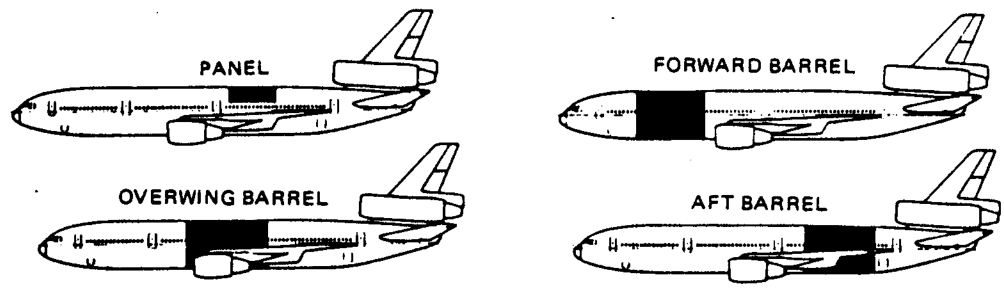


FIGURE 5-2. STRUCTURAL BARREL OPTIONS

TABLE 5-4
EVALUATION OF STRUCTURAL BARREL OPTIONS

		GENERIC		FORWARD		CENTER		AFT		
MERIT FUNCTION	WEIGHT	R	WR	R	WR	R	WR	R	WR	BASIS FOR HIGH RATING
LARGE CUTOUTS	10	8	80	8	80	10	100	8	80	MOST LARGE CUTOUTS - HIGH LOADS
DAMAGE TOLERANCE	10	8	80	9	90	10	90	9	90	HIGH LOADS - MOST DIVERSE STRUCTURE
DURABILITY	7	7	49	9	63	10	70	8	56	HIGH LOADS - MOST DIVERSE STRUCTURE
HIGH LOAD TRANSFER	8	8	64	9	72	10	80	8	64	MOST REPRESENTATIVE JOINTS - HIGH LOADS
ACOUSTICS	4	10	40	10	40	10	40	10	40	MOST REPRESENTATIVE JOINTS - HIGH LOADS
POSTBUCKLING	4	10	40	7	28	8	32	10	40	HAS STRUCTURE DESIGNED FOR POSTBUCKLING
EFFECTS OF DEFECTS	5	7	35	10	50	10	50	8	40	DIVERSITY OF STRUCTURE
REPAIR	5	7	35	10	50	10	50	8	40	DIVERSITY OF STRUCTURE
THERMAL COMPATIBILITY	10	0	0	0	0	0	0	0	0	ENVIRONMENTAL EXPOSURE
NONDESTRUCTIVE INSPECTION	6	8	48	8	48	10	60	8	80	DIVERSITY OF STRUCTURE
MANUFACTURING METHODS	10	6	60	9	90	10	100	8	80	DIVERSITY OF STRUCTURE AND SIZE
INFLUENCE ON PROGRAM COST	10	10	100	7	70	6	60	8	80	LOW COST
INFLUENCE ON PROGRAM SCHEDULE	5	10	50	8	40	6	30	8	40	SHORT SCHEDULE
			681		721		762		698	

LEGEND: R = RATING ON SCALE OF 1 TO 10
WR = WEIGHTED RATING

PHASE III — FLIGHT SERVICE EVALUATION

The Phase III flight service evaluation makes the following contributions to achievement of program objectives.

- Provides the manufacturer and airline with actual operational data on the durability and maintainability of composite fuselage structure.
- Provides training and experience with inspection techniques, equipment, evaluation of findings, man-hours, and schedules.
- Adds realisim to the total program. Flightworthy hardware must be designed and fabricated, installed, and substantiated for structural integrity to the FAR 25 requirements. Technical problems which affect structural integrity cannot be deferred or passed to some other program for resolution.

Three panel options have been defined for a flight service evaluation program, as shown in Table 5-5. It would be desirable to include the following panel features in a flight service program:

- A fairly large panel with design features most representative of typical fuselage panel construction.
- Exposure to service and maintenance traffic with a risk of inflicted damage from hand-held objects or vehicles.
- Exposure to damage from foreign objects on runways during taxi, takeoff, and landing.
- The panel should be from the Phase II structural test article to provide a data base for FAA structural substantiation.
- Easy removal of the aluminum structure and reinstallation of the composite panel.
- The panel must be accessible for periodic nondestructive inspection.

The three flight service options have been evaluated on the basis of the desired panel features being weighted merit functions. The evaluation indicates that a panel underneath the fuselage exposed to foreign object damage and including a large cutout for exposure to service and maintenance abuse is a clear-cut choice. See Table 5-6.

Two deficiencies are encountered in using this panel for flight service evaluation with respect to substantiating structural integrity in accordance with FAR 25.

- The Phase II ground test article, as defined, does not include a cargo door cutout.
- The installation of a composite panel into an otherwise aluminum fuselage will superimpose thermal stresses upon stresses due to cabin pressure and flight loads. The present definition of the Phase II ground test does not include provisions for temperature control or for thermally incompatible panels.

TABLE 5-5

CANDIDATE PANEL OPTIONS FOR PHASE III FLIGHT SERVICE EVALUATION

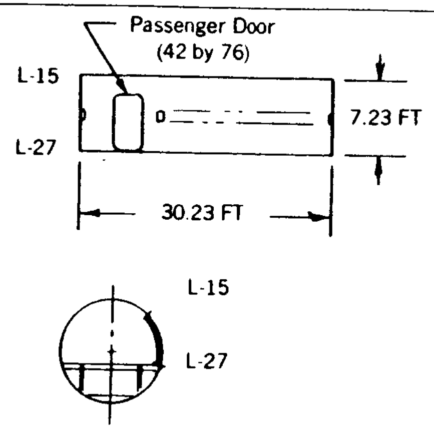
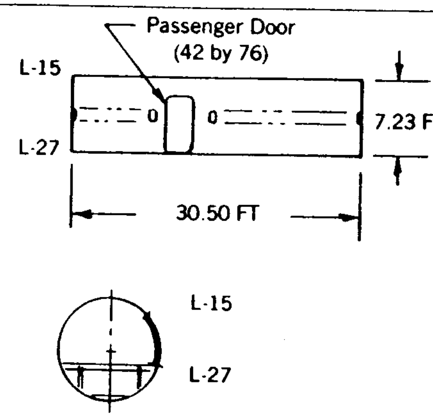
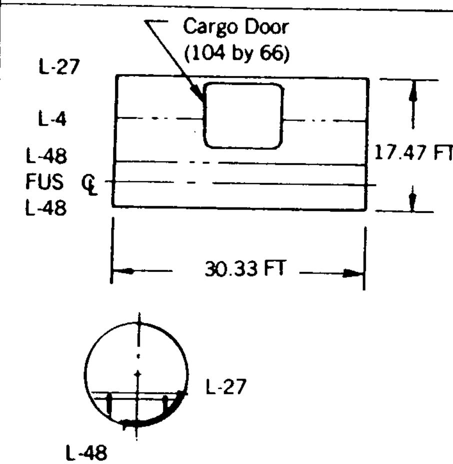
	SIDE PANELS		LOWER PANEL
	FORWARD FUSELAGE	CENTER FUSELAGE	FORWARD FUSELAGE
			
ADVANTAGES	– Passenger door cutout – Window belt – Existing production longitudinal and transverse splices	– Passenger door cutout – Window belt – Existing production longitudinal and transverse splices	– Cargo door cutout – Underside region included for foreign object damage evaluation – Existing production longitudinal and transverse splices
DISADVANTAGES	– Somewhat difficult to remove metal panel and install composite panel – Some new tooling may have to be provided	– Somewhat difficult to remove metal panel and install composite panel – Some new installation tooling may have to be provided	– Moderately difficult to remove metal panels and install composite panels – Some new tooling may have to be provided – More of substructure affected – Largest panel assembly

TABLE 5-6
EVALUATION OF PHASE III FLIGHT SERVICE
PANEL OPTIONS

MERIT FUNCTION	WEIGHT	FORWARD SIDE FUSELAGE	CENTER SIDE FUSELAGE	LOWER FORWARD FUSELAGE
SCORE — TYPICAL FUSELAGE CONSTRUCTION WEIGHTED SCORE —	8	10 • PASSENGER DOOR • WINDOW BELT • SKIN • LONGERONS • SHEAR TEES • TRANSVERSE SPLICE • LONGITUDINAL SPLICE 80	9 72	8 • CARGO DOOR 64
LOW-COST PANEL FABRICATION AND INSTALLATION	6	8 • SOME NEW FABRICATION TOOLS • LIGHTER CONSTRUCTION 48	10 • HEAVIEST CONSTRUCTION • USE PHASE III FABRICATION TOOLS 60	5 • SOME NEW FABRICATION TOOLS • LARGEST PANEL SIZE • CARGO DOOR AREA IS NEW CONSTRUCTION 30
LOW-COST SUBSTANTIATION EFFORT FOR FAA CERTIFICATION	7	8 • USE PHASE I AND PHASE II TEST DATA WHERE APPLICABLE • MUST ACCOUNT FOR THERMAL STRESSES 56	10 70	5 • NO CARGO DOOR TEST DATA FROM PHASE II 35
ACCESSIBILITY FOR PERIODIC NONDESTRUCTIVE INSPECTION	10	7 GOOD ACCESSIBILITY FOR NONDESTRUCTIVE INSPECTION 70	7 70	10 • INNER SURFACE OF PANEL IS MORE ACCESSIBLE FROM THE CARGO COMPARTMENT 100
EXPOSURE TO FOREIGN OBJECT DAMAGE	10	0 0	0 0	10 YES 100
EXPOSURE TO SERVICE AND MAINTENANCE ABUSE	10	8 AROUND PASSENGER ENTRY DOOR IN MID-FORWARD FUSELAGE LOCATION 80	6 AROUND OVERWING PASSENGER DOOR AND ACCESS TO UPPER WING FOR MAINTENANCE 60	10 • CARGO LOADING/OFF-LOADING • VEHICLE TRAFFIC • WATER, TOILET SERVICE • PERSONNEL TRAFFIC 100
TOTAL WEIGHTED SCORE		334	332	429
FINAL RANKING		2	3	1

RECOMMENDED PROGRAM

The following program options are recommended as the baseline for the study tasks and for defining a composite fuselage development program.

- MD-100 civil transport as the baseline vehicle.
- Three program phases: Phase I — Design Development
  Phase II — Structural Verification
  Phase III — Flight Service Evaluation
- The center fuselage barrel section option is selected for Phase II.
- The forward lower panel option is selected for Phase III.

The flight service evaluation can be conducted on either a commercial DC-10 transport aircraft or an Air Force KC-10 tanker/cargo aircraft. Both airplanes have a structural commonality with the MD-100 and both are FAA-certified for compliance with Federal Aviation Regulations Part 25 Airworthiness Standards (FAR 25).

SECTION 6
CONCEPTUAL DESIGN

BASELINE AIRCRAFT

The MD-100 was selected as the baseline aircraft for technology development, manufacturing, demonstration, and structural verification tests. A general arrangement of the MD-100 is shown in Figure 6-1. This aircraft is of sufficient size and complexity to represent the principal design and manufacturing processes associated with an all-composite fuselage.

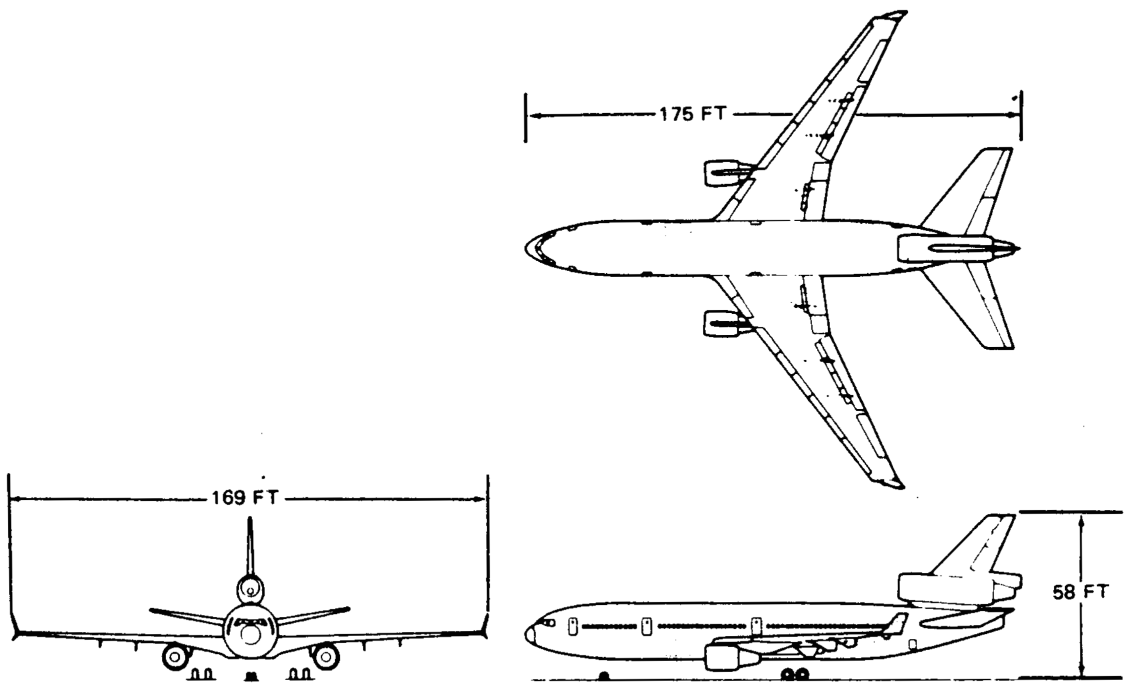


FIGURE 6-1. GENERAL ARRANGEMENT – MD-100 BASELINE AIRCRAFT

Some of the advantages of using the MD-100 as the study baseline aircraft are that an adequate data base for criteria, loads, and weights exists; commercial and military (KC-10) versions are available, and a composite demonstration panel (110 by 168 in.) can be accommodated in an existing test fixture in a straightforward manner.

The fuselage of the MD-100 airplane is of conventional semimonocoque construction. The fuselage pressure shell is 1,732 inches long and the constant-diameter section is 237 inches in diameter, or approximately 144 feet long by 20 feet in diameter. The MD-100 fuselage design is shown in Figure 6-2.

CONCEPTUAL DESIGN GENERATION

Conceptual designs were generated based on the general layout of the MD-100 baseline fuselage. The concepts take into consideration basic design criteria. These criteria and the design loads define the con-

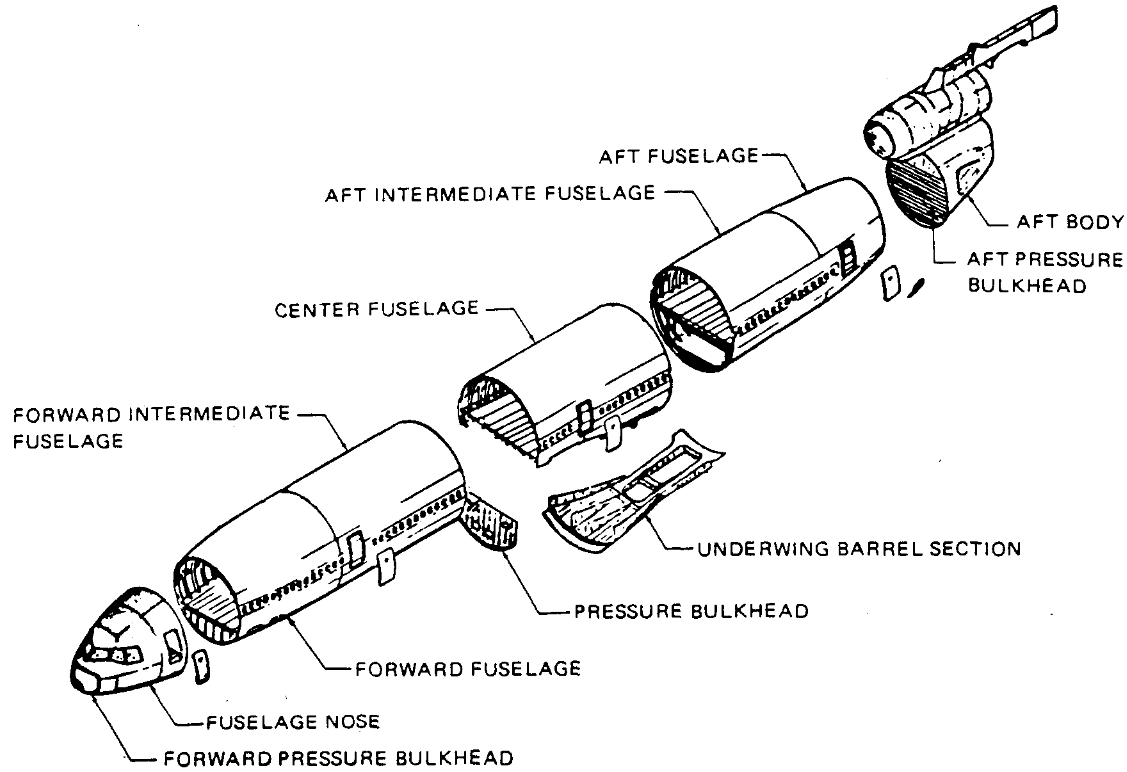


FIGURE 6-2. FUSELAGE STRUCTURAL ARRANGEMENT

ditions that must be satisfied by the design. The conceptual designs aid in the evaluation of various concepts through trade studies, which serve to select the best design concept and manufacturing method. The selected designs are then integrated into a complete fuselage concept. The resulting conceptual design may then be used for the generation of weight, cost, and performance estimates. The conceptual designs and weight estimates are for the portion of the fuselage between the forward pressure bulkhead and the rear pressure bulkhead.

General Layout

The general layout of the composite design closely follows the layout of the metal MD-100 baseline. Trade studies were performed to determine the basic geometry of the fuselage shell.

Skin Panels — A nomograph illustrating the effect of cabin pressure, fuselage diameter, and skin layup on the minimum required skin thickness to resist pressure loading is shown in Figure 6-3. An example is shown of an ultimate pressure loading of 18.2 psi with a pseudo-isotropic skin layup and a strain limit of 0.0045 in./in. (ultimate). The minimum required skin thickness in this case is approximately 0.06 inch.

Frame Spacing — Trade studies were undertaken to evaluate the effect of frame spacing on the total weight of the stiffened skin, frame, and floor beams. The results of the studies are shown in Figure 6-4. An increase in frame spacing increases the effective column length of the stiffened panel. This results in a significant increase in panel weight to sustain the same compression loading. In addition, the

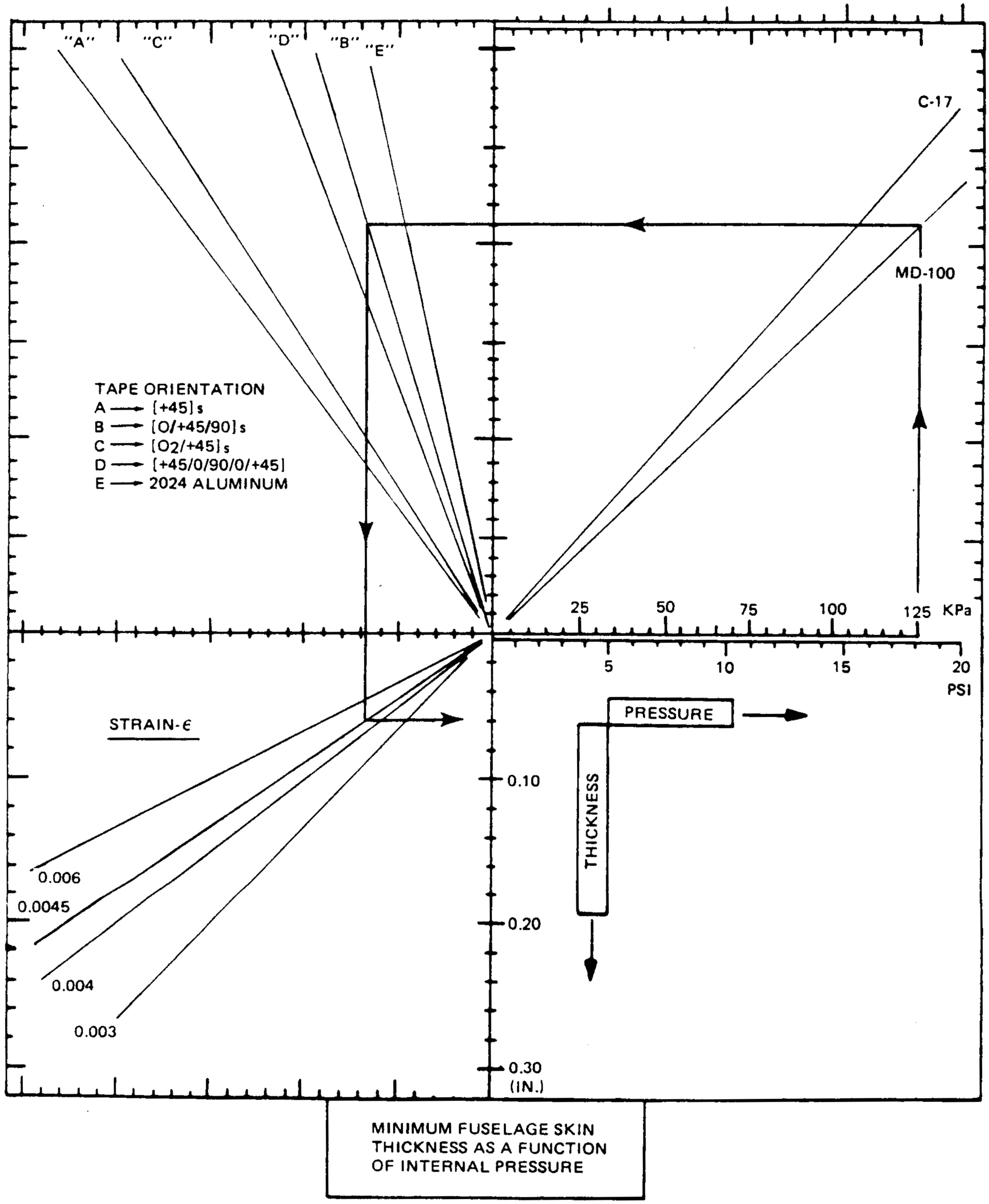


FIGURE 6-3. MINIMUM FUSELAGE SKIN THICKNESS REQUIREMENTS

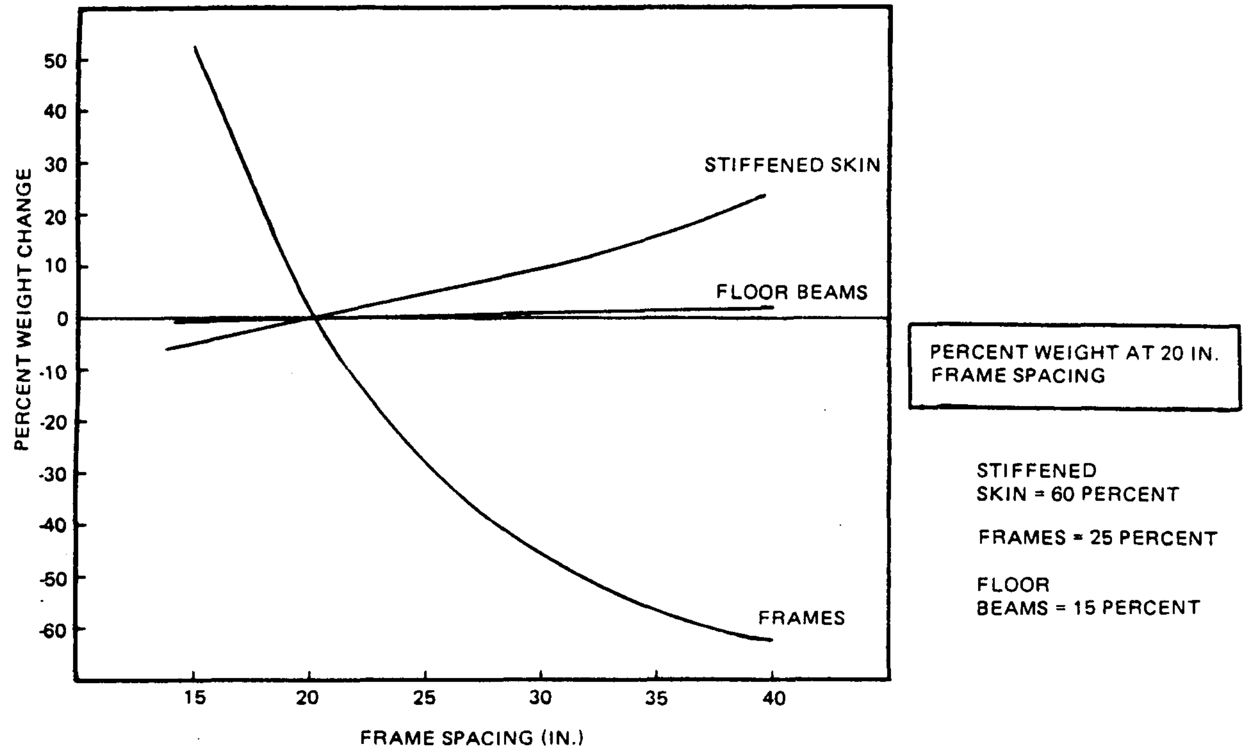


FIGURE 6-4. EFFECT OF FRAME SPACING ON FUSELAGE WEIGHT

total floor beam weight increases slightly. However, the total frame weight will decrease significantly since there are fewer frames. In general, it was found that the overall fuselage weight will increase as frame spacing is increased past a 20-inch spacing. An important consideration, however, is that the cost of the fuselage will decrease significantly as the part count is decreased. A cost-performance tradeoff exists between the weight-efficient closely spaced concept and the cost-efficient widely spaced concept. A frame spacing of 20 inches has been selected as the best compromise between the constraints.

Longeron Spacing — Stiffened panel trade studies were conducted to determine the effect of longeron spacing on panel weight based on J-sections and hat sections. For a typical skin thickness of 0.088 inch and a compression allowable of 2,000 lb/in., it appears that a desirable longeron spacing of 6 to 8 inches is indicated for J-section stiffened panels (Figure 6-5). Similar conclusions can be reached for the design of shear panels (Figure 6-6).

The hat-stiffened panels are similar in performance to the J-section panels with a slightly greater longeron spacing for the same conditions of 2,000 lb/in. load intensity and 0.088-inch thickness skin. (See Figure 6-7 and Figure 6-8.) The limits imposed on the panels include a frame spacing of 20 inches, pseudo-isotropic layups, a longeron height of 1.0 inch, and a buckling limit of 50 percent of limit load.

DESIGN REQUIREMENTS AND CRITERIA

The fuselage concepts are based on MD-100 design criteria which encompass both Douglas and FAA requirements. The criteria used for the development of the design concepts are shown in Table 6-1 and Figures 6-9 and 6-10, and are described below.

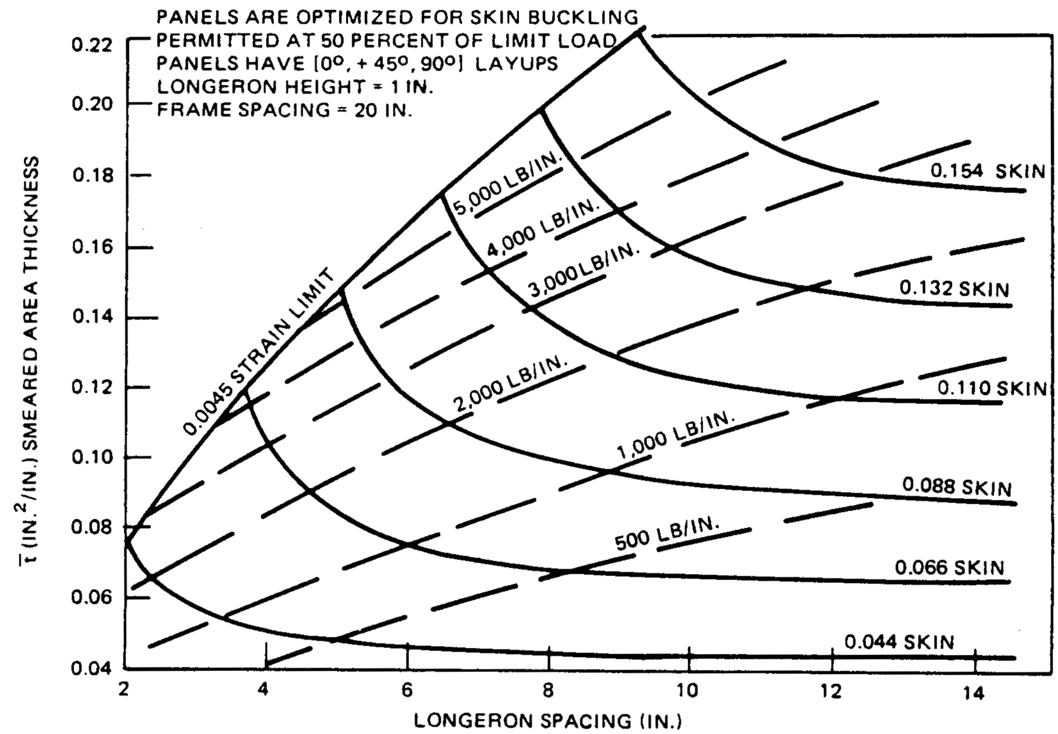


FIGURE 6-5. COMPRESSION ALLOWABLES FOR INTEGRAL COMPOSITE J-STIFFENED PANELS

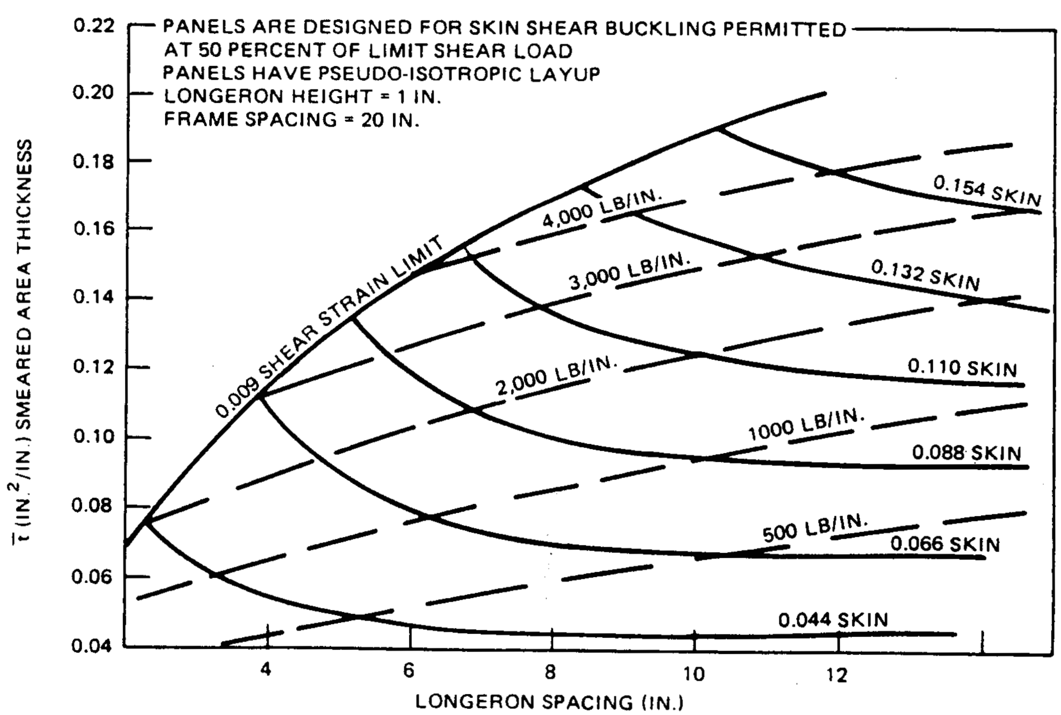


FIGURE 6-6. SHEAR ALLOWABLES FOR INTEGRAL COMPOSITE J-STIFFENED PANELS

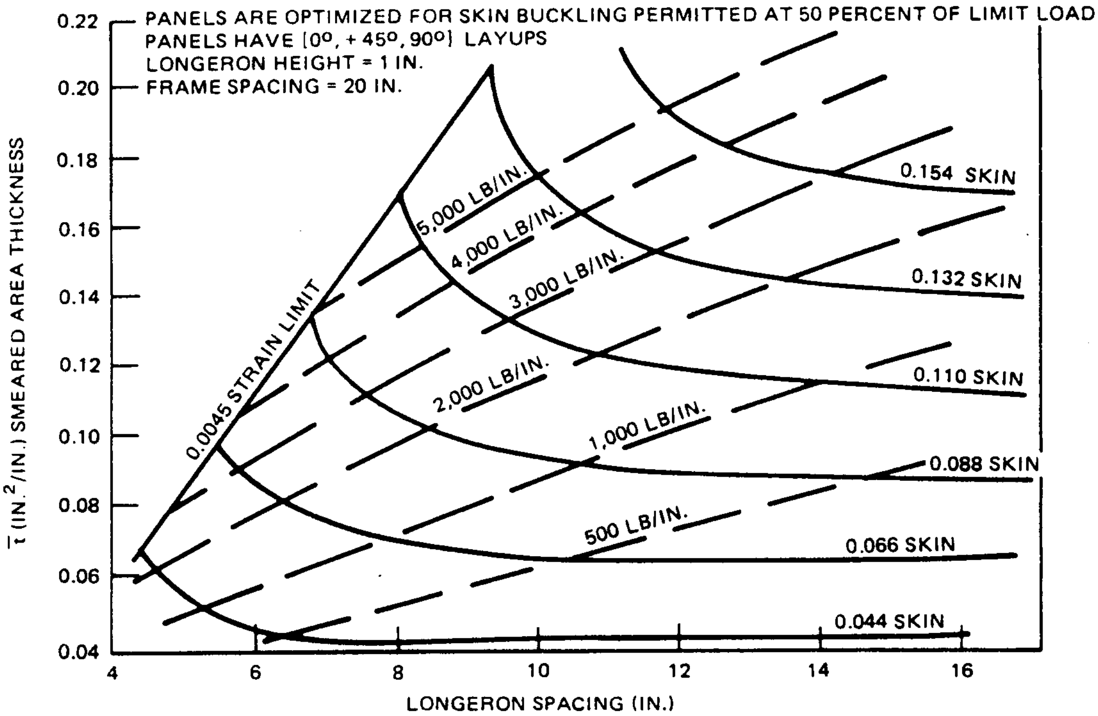


FIGURE 6-7. COMPRESSION ALLOWABLES FOR COMPOSITE HAT-STIFFENED PANELS

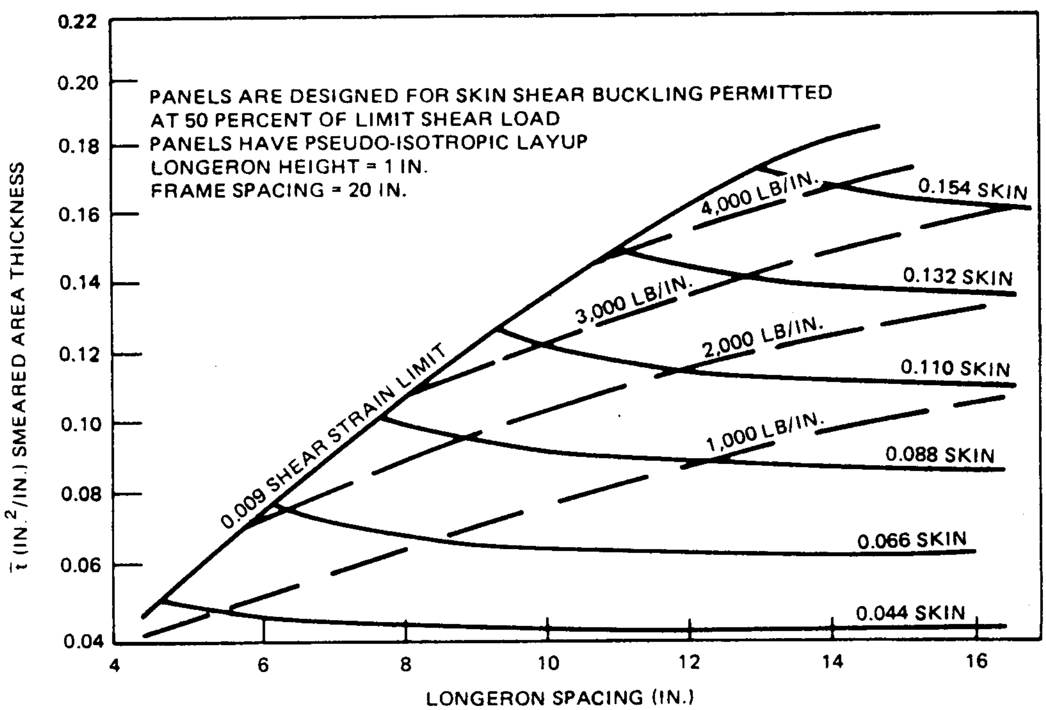


FIGURE 6-8. SHEAR ALLOWABLES FOR COMPOSITE HAT-STIFFENED PANELS

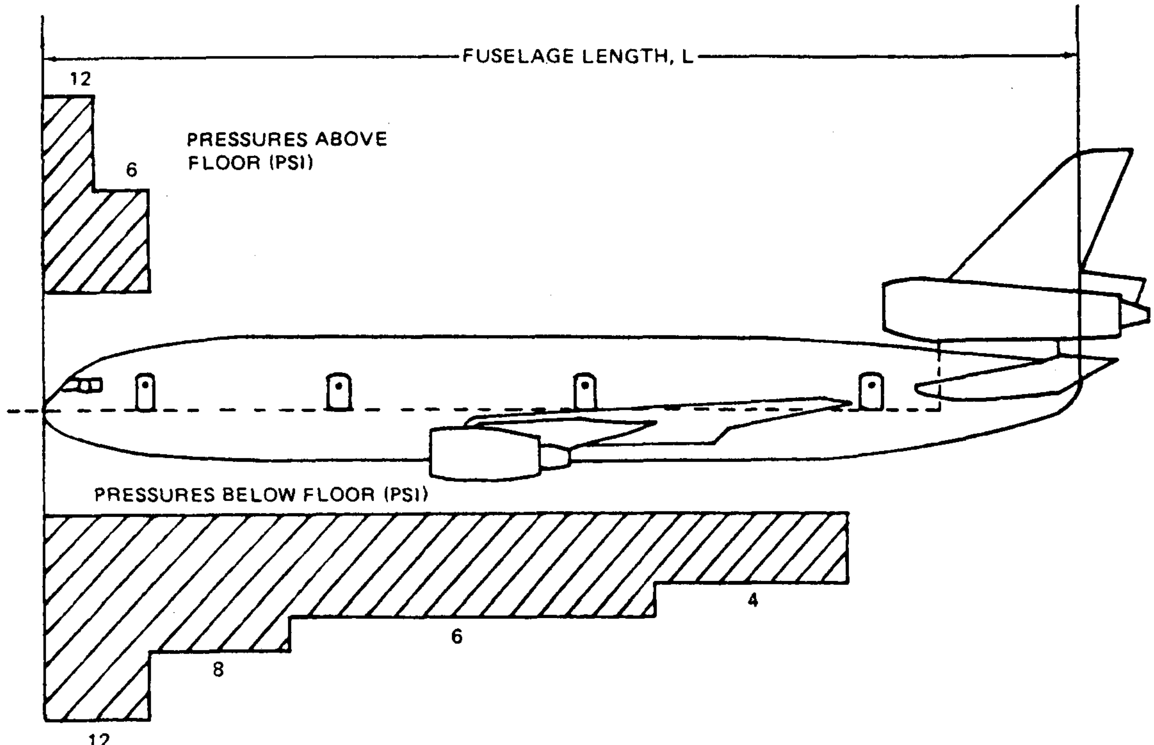


FIGURE 6-9. MD-100 DITCHING PRESSURE LOADS FOR FUSELAGE SHELL

Buckling

The fuselage skin is designed to operate in the postbuckled range. The allowable skin buckling load plays an important role in the final shell weight. A method of increasing the buckling load is to increase the skin thickness. However, increasing the skin thickness results in increased strains due to panel curvature when the skin buckles. Two possibilities are available to the designer, either thin, postbuckled skin or thick, heavier completely unbuckled structure. An alternative method of increasing the buckling load of a stiffened panel would be to decrease the spacing of the substructural stiffening elements. This would, of course, result in an increase in weight and complexity. The added complexity would drive the cost of the design up and make operational inspection and repair more difficult.

The skin buckling criterion allows no buckling at 1g loads. Buckling is permitted at 50 percent of limit load. This criterion ensures a smooth aerodynamic surface during normal flight conditions, but does not impose undue weight penalties which would result from lower buckling limits.

Minimum Gage

A minimum skin thickness of 0.06 inch is adequate for strength requirements. However, because of the uncertainty of actual service requirements and the lack of experience with composite primary structure, a higher minimum gage of 0.070 inch was specified for the conceptual design and weight studies.

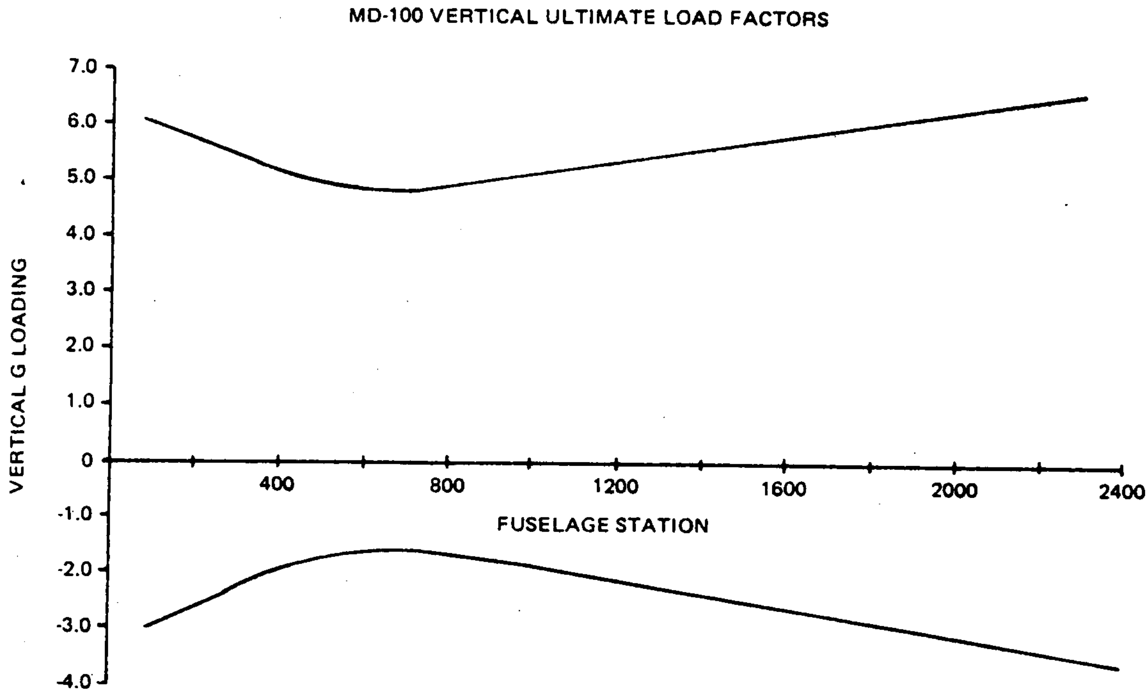
Damage Tolerance

Two design techniques for increasing the level of tolerance to inflicted damage is to design to lower strain allowables or use damage containment elements such as glass softening strips. Reduced strain allowables must be incorporated in the preliminary design stage to ensure adequate sizing. Softening strips may be inserted in a conceptual design with no major changes since the glass simply replaces carbon fibers. For this reason, the conceptual designs are based on a 0.0045 in./in. strain allowable.

TABLE 6-1

DESIGN REQUIREMENTS AND CRITERIA

DESIGN STRAIN LEVELS (μIN./IN.)	
• ULTIMATE LOAD	±4,500
• LIMIT LOAD	±3,000
LIMIT FLIGHT LOAD FACTORS (G)	
FLAPS UP MANEUVER	+2.5, -1.0
FLAPS DOWN MANEUVER	+2.0
DESIGN GROSS WEIGHTS* (LB) (MD-100 BASELINE)	
TAXI WEIGHT	503,000
TAKEOFF WEIGHT	500,000
LANDING WEIGHT	370,000
ZERO FUEL WEIGHT	345,000
*MAXIMUM	
FUSELAGE PRESSURE (PSI)	
MAXIMUM NORMAL OPERATING	8.6
MAXIMUM RELIEF VALVE SETTING	9.1
DESIGN LIMIT LOAD	12.1
DESIGN ULTIMATE LOAD	18.2
SKIN PANEL BUCKLING	
NO BUCKLING	1g WITH PRESSURE 1g (NO PRESSURE)
ALLOWABLE SHEAR BUCKLING	50 PERCENT LIMIT LOAD
ALLOWABLE COMPRESSION BUCKLING	CONFIGURATION-DEPENDENT
ACOUSTIC TRANSMISSION (dB)	
GENERAL:	
AVERAGE OVERALL NOISE LEVEL	86
AVERAGE SPEECH INTERFERENCE LEVEL	64
MAXIMUM (SEAT POSITION):	
AVERAGE OVERALL NOISE LEVEL	93
AVERAGE SPEECH INTERFERENCE LEVEL	67
REPEATED LOAD REQUIREMENT	2 LIFETIMES 60,000 FLIGHTS/LIFETIME
EMERGENCY LANDING LOAD FACTORS PASSENGER SEATS AND RETENTION STRUCTURE	4.0 G UP 9.0 G FORWARD 1.5 G LATERAL 7.0 G DOWN
DITCHING LOADS	SEE FIGURE 6-9
MAXIMUM VERTICAL AND LATERAL LOAD FACTORS	SEE FIGURE 6-10
FLOOR LOAD (MAXIMUM) (LB/IN.)	
UPPER FLOOR (10-ABREAST SEATING)	78
UPPER FLOOR (FREIGHTER VERSION)	100
LOWER FLOOR (CARGO COMPARTMENT)	117



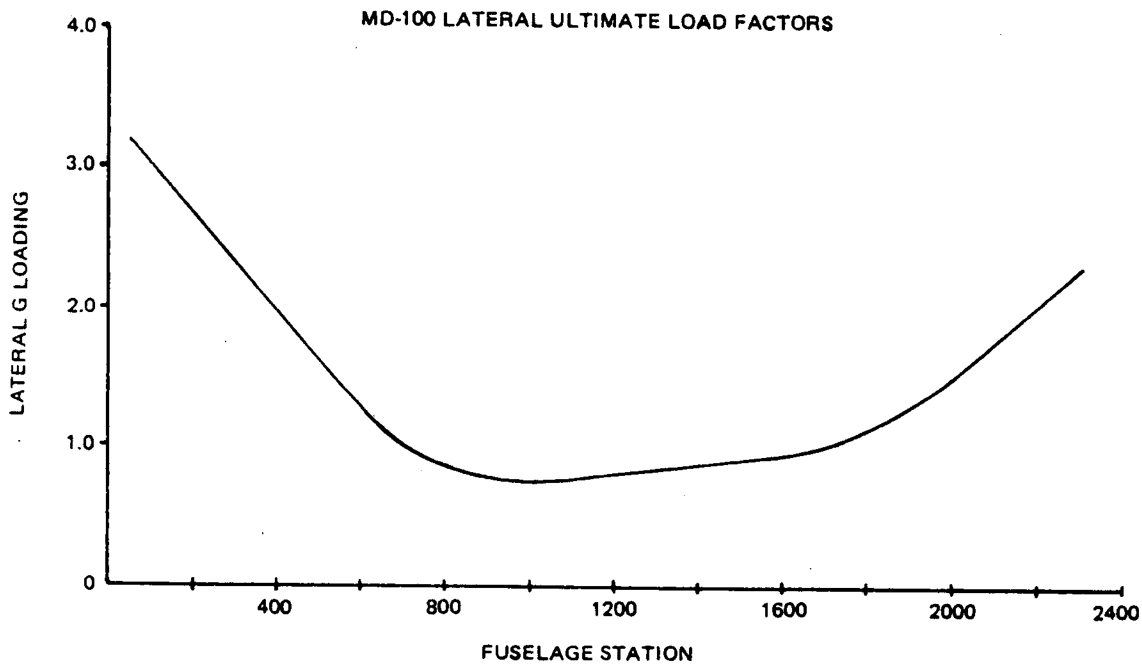


FIGURE 6-10. ULTIMATE VERTICAL AND LATERAL LOAD FACTORS

Crashworthiness

A design criterion that may have a large impact on the design of an all-composite fuselage is crashworthiness. A concept for a fuselage keel capable of absorbing energy and protecting the aircraft's occupants during a crash is shown in Figure 6-11. So little information exists, however, that any design at this time must be viewed as very tentative.

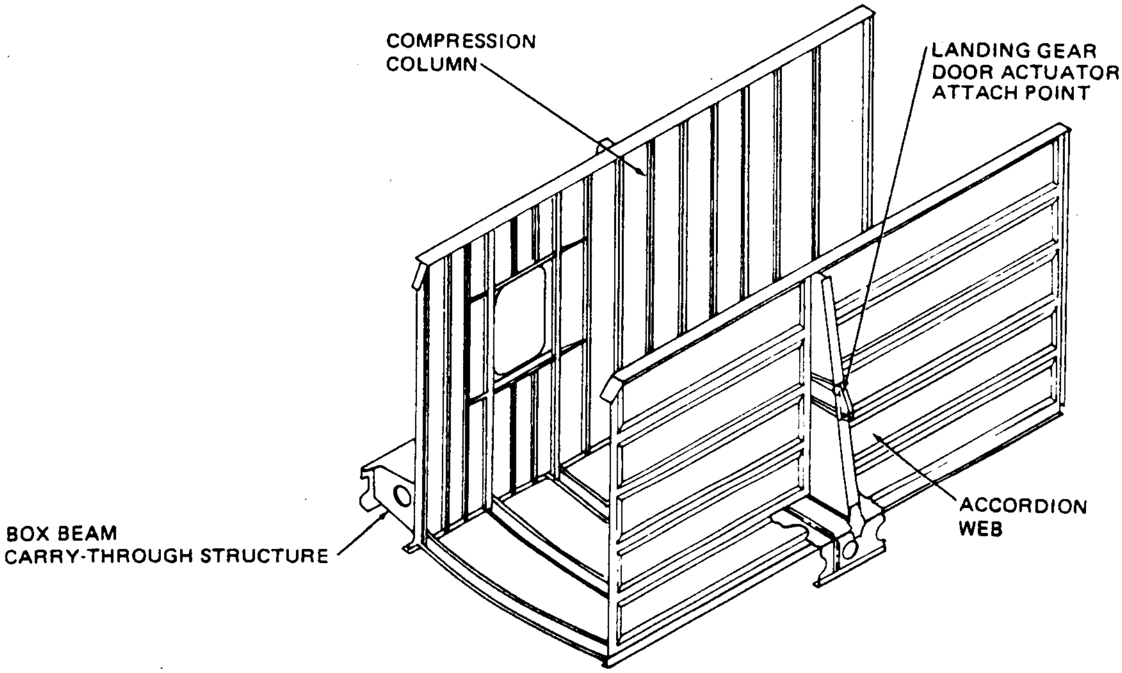


FIGURE 6-11. PRELIMINARY CRASHWORTHY KEEL DESIGN

Design Loads

The design loads criteria do not differ significantly from a conventional metal structure. The ultimate MD-100 shell loads used in the conceptual design, shown in Figures 6-12 and 6-13, are the axial and shear loads resulting in the lowest margins of safety for the MD-100. The loads were used for the fuselage shell sizing.

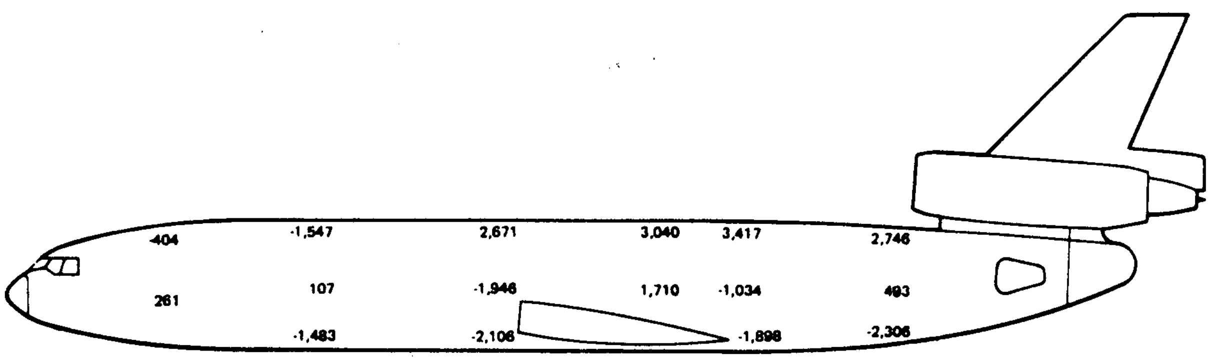


FIGURE 6-12. MAXIMUM AND MINIMUM AXIAL LOAD INTENSITIES (LB/IN.)

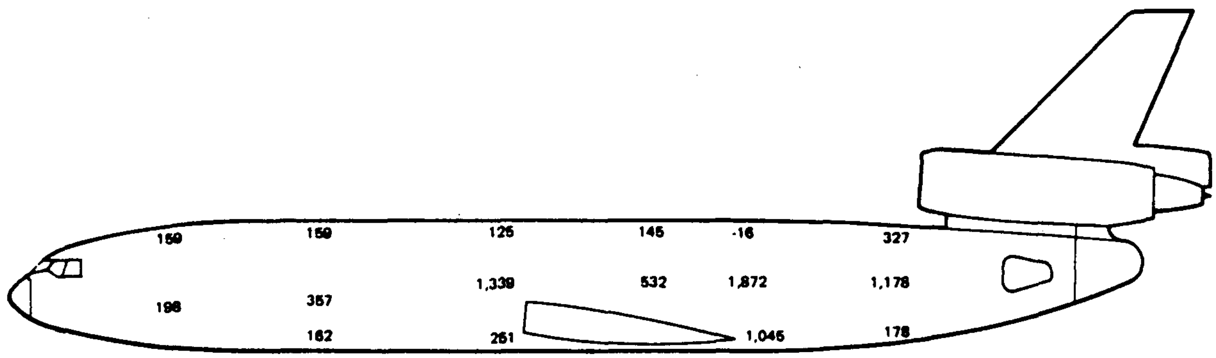


FIGURE 6-13. MAXIMUM AND MINIMUM SHEARS (LB/IN.)

DESIGN CONCEPTS

Design concepts have been developed to aid in the evaluation of the feasibility of an all-composite fuselage and to obtain weight estimates for the design. The concepts developed are for the skin panel, skin splices, longeron and frame splices, floor beam, pressure bulkheads, wing joint and side panels, nose section structure, cabin window structure, door jamb structure, and keel. These concepts are explained in the following sections.

Skin Panel Design

Trade studies were conducted to determine the optimum stiffened skin panel design based on weight, durability, and ease of manufacture, inspection, and repair. Two concepts, J-section and hat section discretely stiffened skins, were compared with sandwich stiffened skins. The J-stiffened concept was found to be marginally better than the hat-stiffened concept in terms of ease of manufacture, assembly, and inspection. The results of a trade study of J-stiffened skin versus sandwich panel are shown in Table 6-2. The sandwich panel concept offers great potential for weight and cost reduction. This may be achieved by eliminating many of the longerons and frames to reduce the overall part count of the structure. The resulting structure would be more efficient and cheaper to produce. However, what is gained in efficiency of the basic shell may be lost in the inefficiency of necessary variations from the simple shell such as floor beam-to-fuselage intersections, window installations, and system and equipment installations.

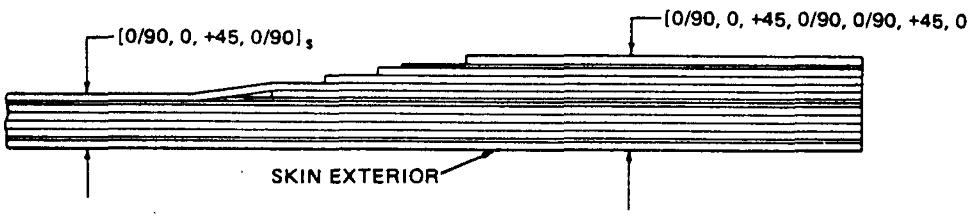
This concern and others such as damage tolerance characteristics, water absorption, and inspection uncertainties were sufficient to eliminate honeycomb-stiffened skin from consideration. This judgment is by no means final; the potential manufacturing and cost benefits of honeycomb-stiffened skin are worthy of future development effort.

TABLE 6-2

J-STIFFENED VERSUS HONEYCOMB SANDWICH FUSELAGE PANELS

	J-STIFFENED	HONEYCOMB
EFFICIENCY		X
EASE OF MANUFACTURE		X
INSPECTABILITY	X	
REPAIRABILITY	X	
FRAME INTERSECTION	X	
EASE OF SPLICING	X	
REDUCTION OF JOINTS		X
FLOOR STRUCTURE INTERFACE	X	

The skin concept which was selected is an integral J-stiffened postbuckled design attached to frames by bonded shear tees. The minimum gage skin panel layup, shown in Figure 6-14, is $(0/90, \pm 45, 0/90, \overline{0/90})_S$ cloth over most of the fuselage length. Areas requiring greater than minimum gage structure are reinforced as required. The portion of the fuselage where minimum gage is acceptable is shown for three different maximum strain limits in Figure 6-15. A typical view of the skin/substructure interface is shown in Figure 6-16. This view shows the skin, frame, shear tee, longeron, and frame to longeron clip as they are assembled to form the fuselage structure.



SKIN PANEL EDGE BUILDUP

FIGURE 6-14. SKIN PANEL DESIGN

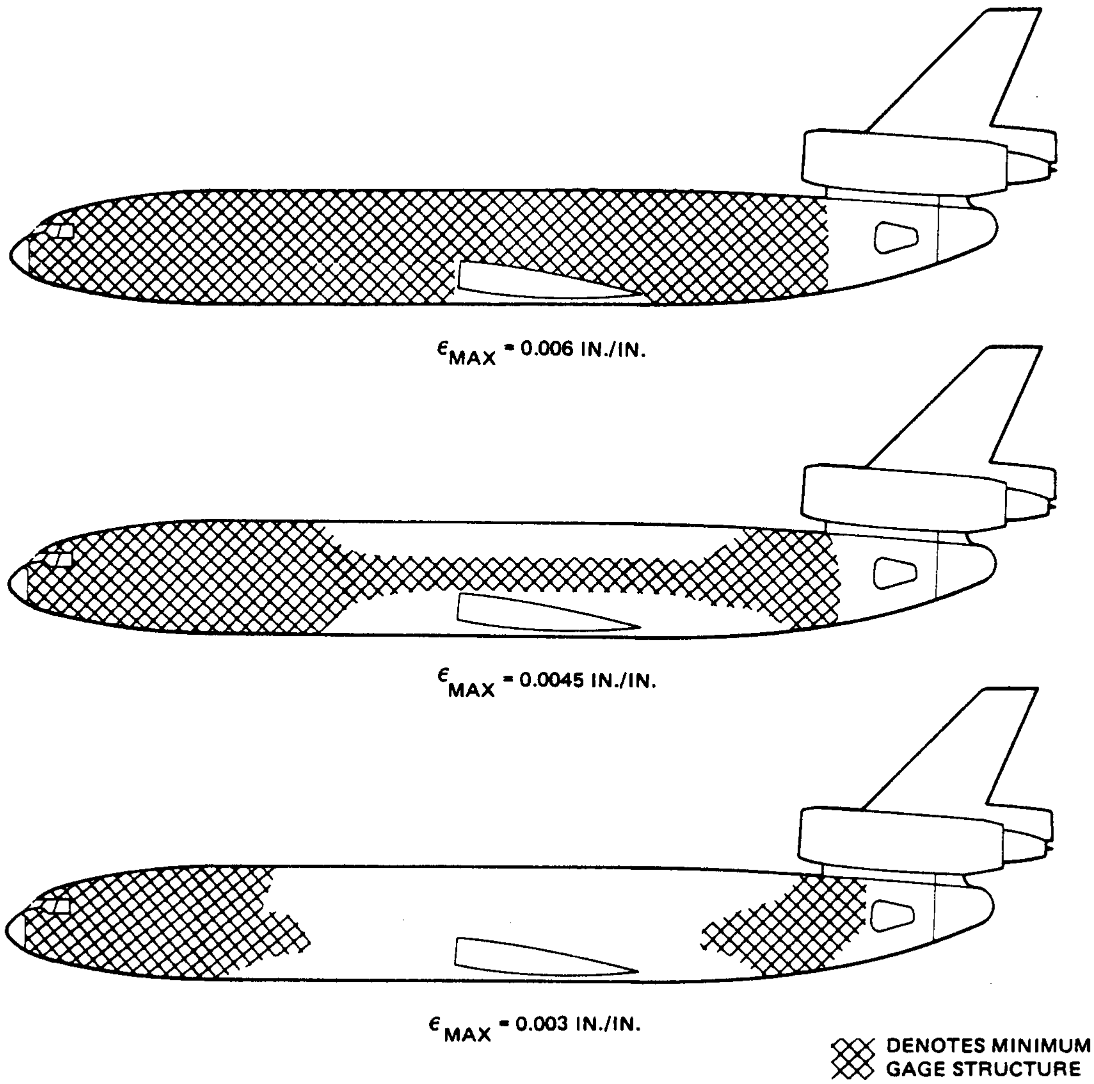


FIGURE 6-15. MINIMUM GAGE REGIONS FOR DIFFERENT STRAIN LIMITS

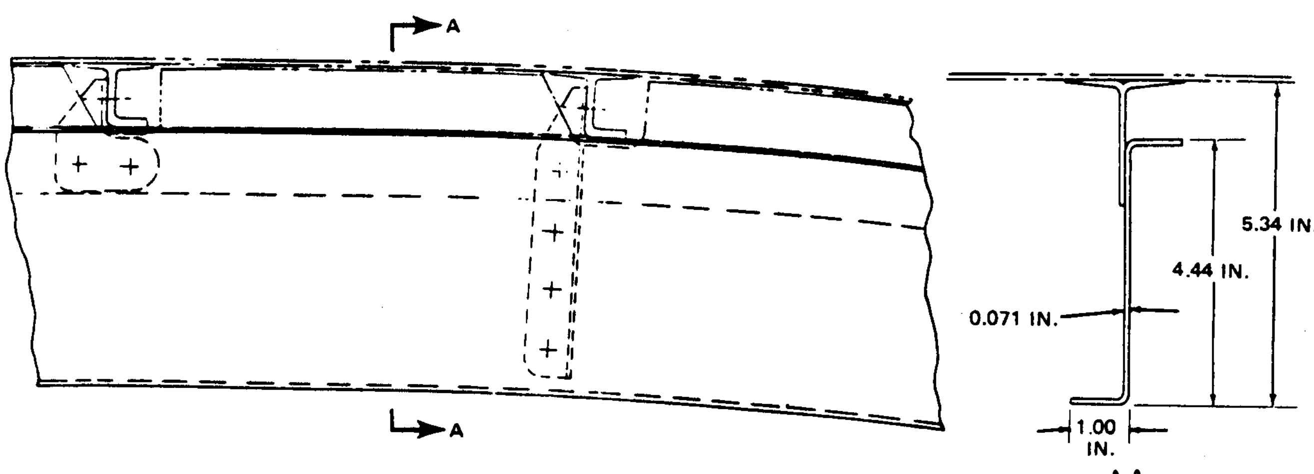


FIGURE 6-16. FRAME/LONGERON INTERSECTION

Skin Splices

The number of manufacturing joints was changed from the baseline MD-100, as shown in Figures 6-17 and 6-18. The transverse joint locations from the baseline are suitable and appropriate for the composite fuselage. The longitudinal skin splices, however, were reduced from 10 splices in the baseline to 4 in the composite fuselage.

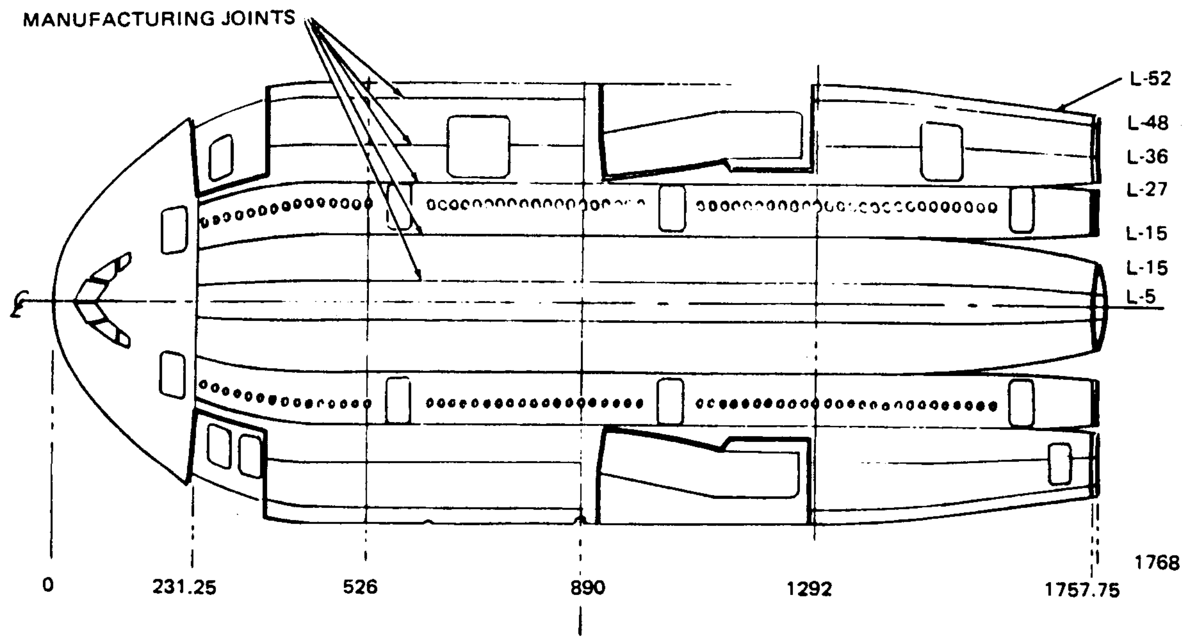


FIGURE 6-17. ALUMINUM SKIN PANEL SPLICES

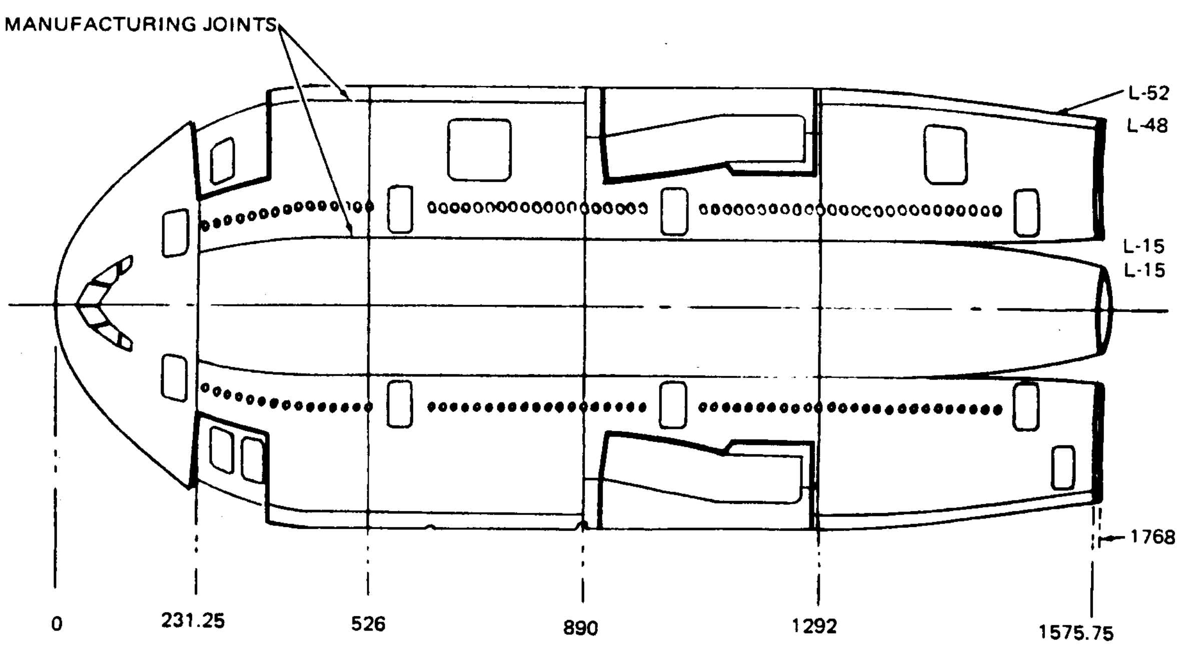


FIGURE 6-18. COMPOSITE SKIN PANEL SPLICES

In developing design concepts for the longitudinal and transverse skin splices, both double-strap butt splices and lap splice concepts were compared for structural efficiency and ease of assembly. The double-strap butt splice concept was chosen for further development because it reduces the load path eccentricity within the joint.

Longitudinal Skin Splice — The MD-100 longitudinal skin splice was designed in enough detail to size the elements and verify the concept by analysis. The splice at longeron 15 in the region of stations 1131 to 1529 was selected. A loads survey was made and it was determined that the maximum shear in this area is 2,235 lb/in. An initial joint concept was then designed utilizing a double shear butt joint. The joint concept was then further developed by use of the computer program BOLTJ, an interactive program that aids in tailoring the joint geometry for a specific strength. This sizing procedure resulted in the final joint design of Figure 6-19.

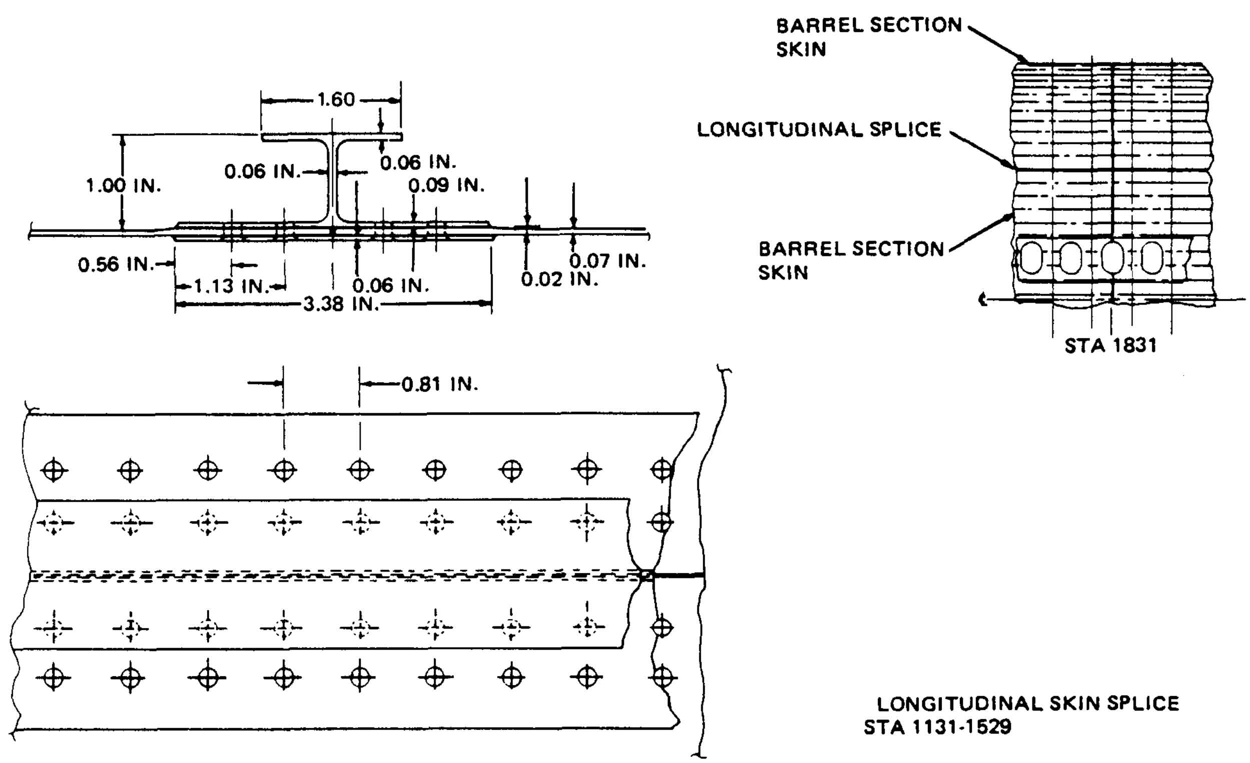


FIGURE 6-19. LONGITUDINAL SKIN SPLICE

The splice is a double shear joint using an external splice strap and an I-section member which serves as a combination longeron and splice strap. The base flange of the longeron functions as an internal splice strap. The typical longeron, however, remains a J-section. Four rows of 3/16-inch-diameter countersunk titanium bolts are used.

Transverse Skin Splice — A transverse skin splice was designed in enough detail to size the elements and verify the concept by analysis. The splice of the upper portion of the fuselage in the region of station 1531 was selected for consideration. The splice in this area will consist of an internal single-shear splice strap.

The maximum skin load in this area was found to be 3,400 lb/in. The computer program BOLTJ was used to develop the geometry of the joint. It was determined that the skin thickness had to be increased in the region of the joint to prevent through-the-hole tension failures. The splice strap is stepped to maximize load-sharing. Four rows of 3/16 inch-diameter-countersunk titanium bolts are used in the splice.

Four-Way Skin Splice — In the region where four skin panels are joined together, the joint is similar to the basic splices. The longeron at the splice location acts as an internal longitudinal splice strap. This means that the base of the longeron must be able to withstand tension and shear loads across its entire width. This is done by making two-thirds of the material in the attaching flange continuous across the flange width of the longeron. One-third of the flange material becomes part of the longeron web. The skin splice strap itself is spliced where necessary outside the four-way skin splice region for simplicity. The four-way skin splice concept is shown in Figure 6-20. The external longitudinal skin splice is interrupted outside the four-way skin splice area.

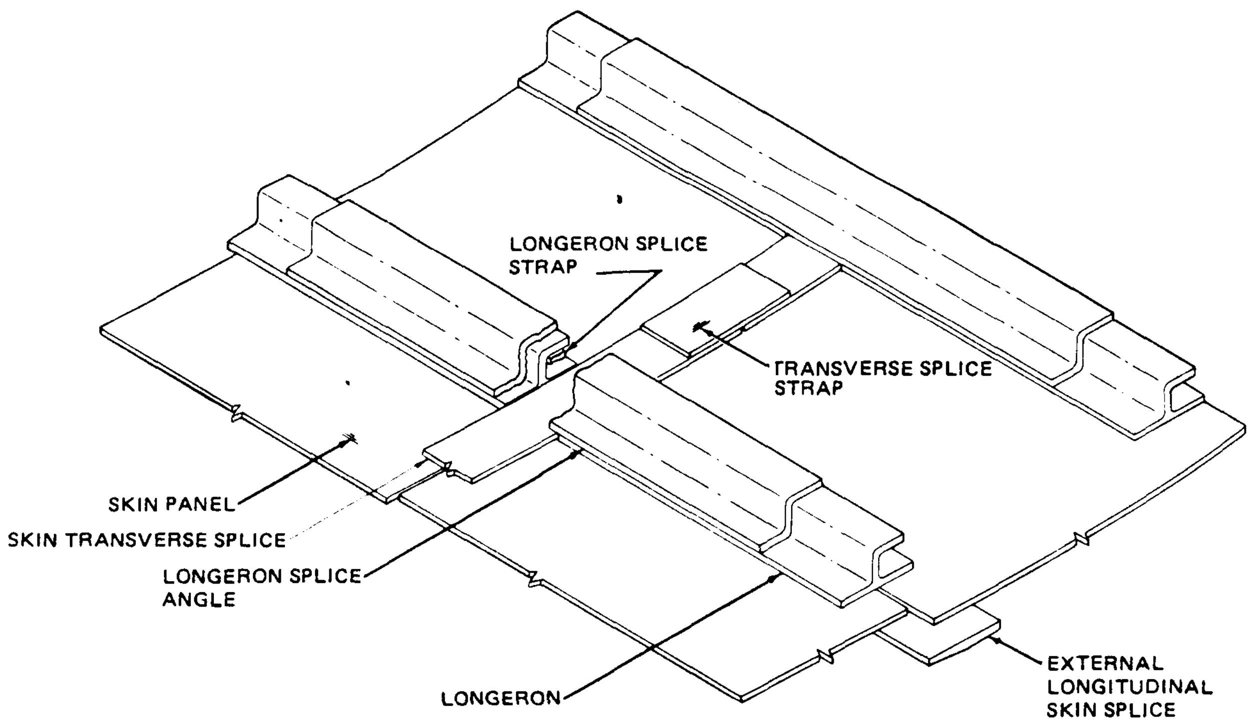


FIGURE 6-20. FOUR-WAY SKIN SPLICE

Longeron Splice

A longeron splice has been selected from the baseline aircraft for detailed design. The selected splice is at longeron 1, station 1531. This is at the top of the fuselage immediately aft of the wing. The total ultimate load transfer for the longeron at this location is 25,000 pounds. The joint design program BOLTJ was used to tailor the joint geometry to this load. The longeron joint was designed in enough detail to prove the concept.

C-5

The basic longeron is stopped just short of the skin buildup at the edge of the panel. A Z-section splice plate and a flat splice plate are used to join the longerons. The thickness of the longeron flange is selected so that the splice plates span the transverse skin splice plate. The joint is attached by 3/16-inch-diameter countersunk titanium fasteners. The fasteners are installed through the longeron splice plates, the skin splice straps, and the skin. The stepped portion of the skin splice strap must be shimmed flush with the longeron splice plates. Shims also must be used to eliminate fit problems within the joint. Figure 6-21 shows the joint design.

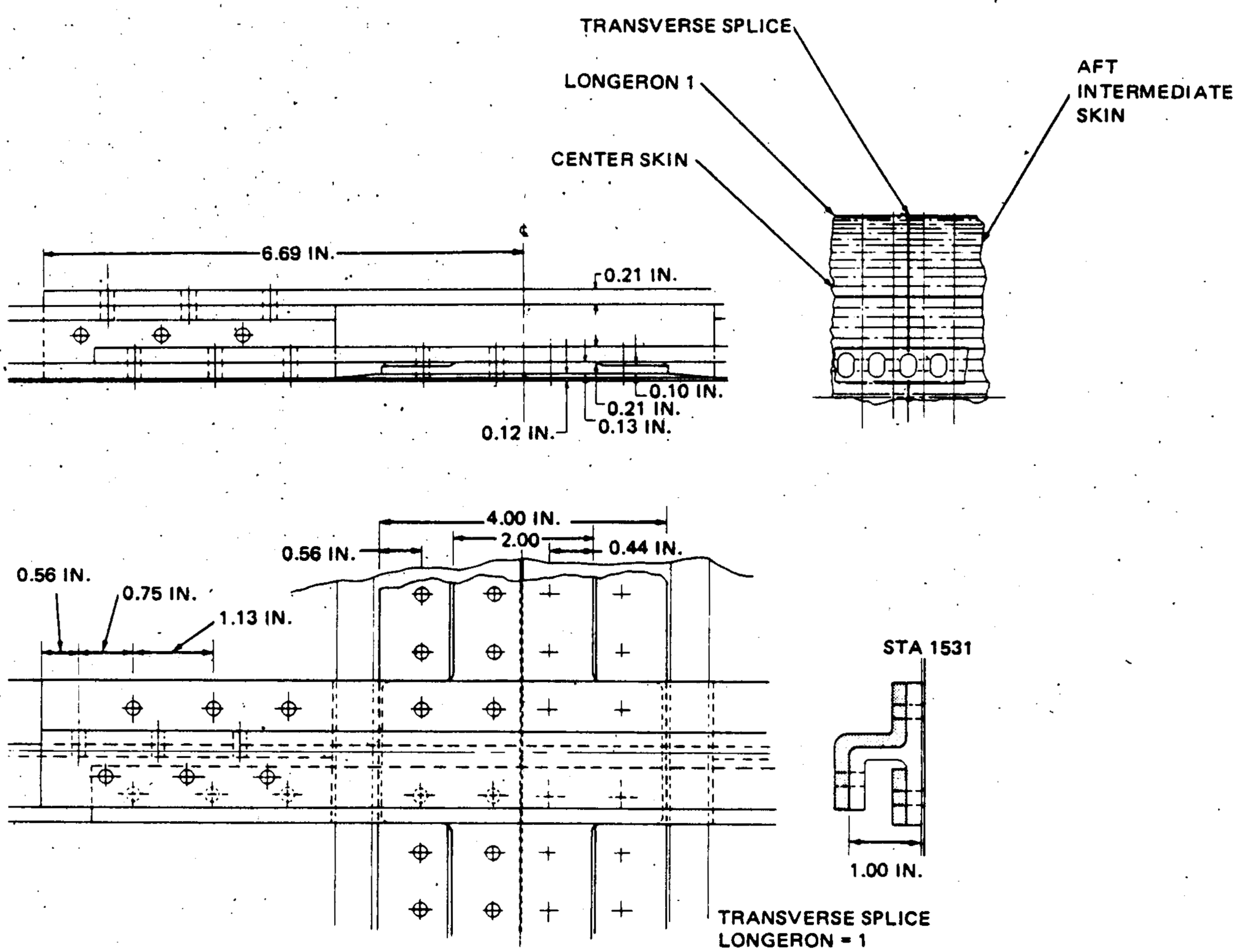


FIGURE 6-21. TRANSVERSE SKIN/LONGERON SPLICE

It should be noted that the longeron joint is quite compact and that a greater longeron depth may be desired for ease of manufacture. This may be made possible since the most highly loaded longerons are located at the top and bottom of the fuselage and are in an area where depth is not critical. An alternative method would be to decrease the longeron load by decreasing the longeron spacing. This approach is probably not desirable, however, since increased part count invariably tends to raise manufacturing costs.

Frame Splice

The frame splice utilizes two L-section splice plates. The plates are bolted over the frame joint through the web and caps, as shown in Figure 6-22.

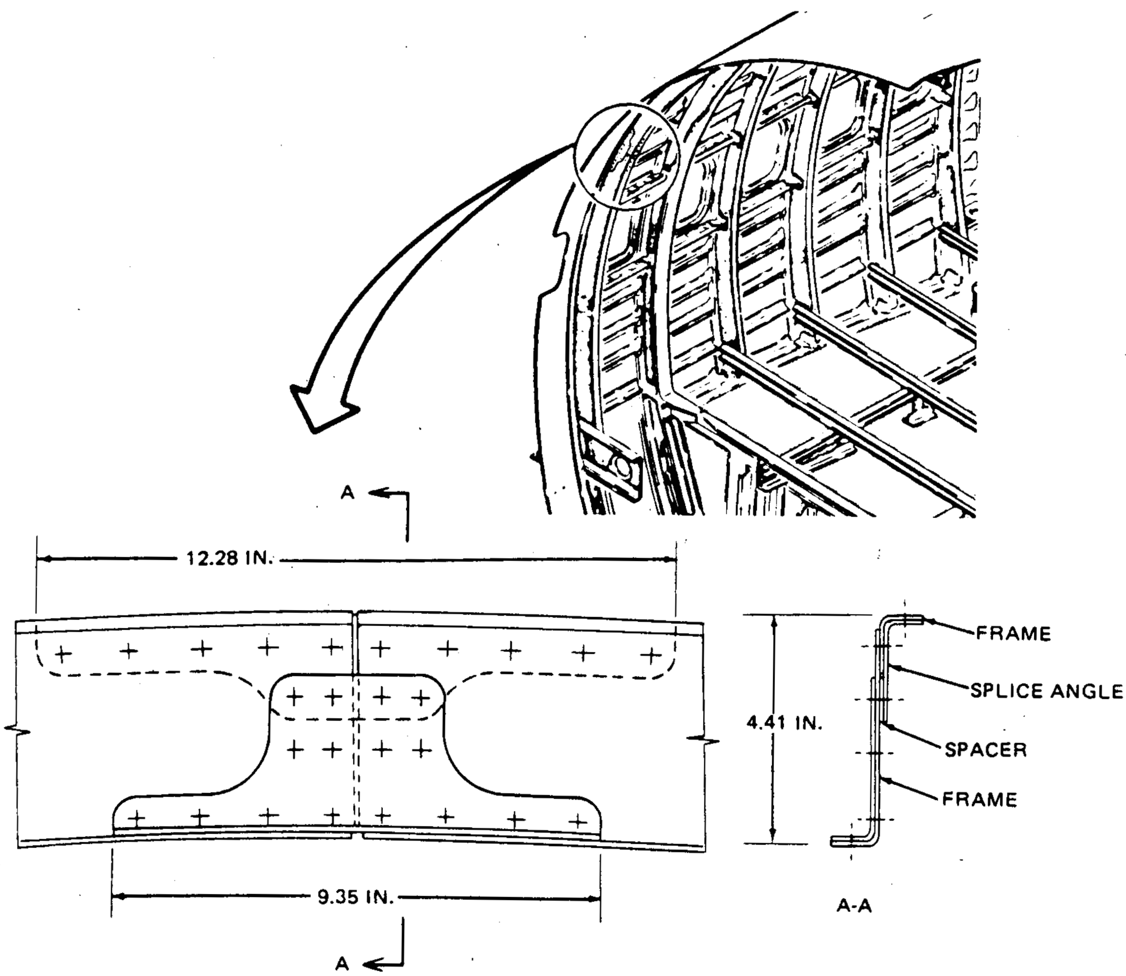


FIGURE 6-22. FRAME SPLICE

Floor Beams

A typical constant-section floor beam and side support strut arrangement suitable for the MD-100 was based on existing DC-10 data. (See Figure 6-23.) The overall length is 231 inches, with a depth of 10 inches. The beam is supported at each end by the fuselage frames and at two intermediate locations by vertical side struts. The beam top flange is stabilized by seat tracks running fore and aft, while the lower flange is stabilized by lower cargo ceiling liner supports.

The floor beam web is basically a pseudo-isotropic layup of 0/45/90/-45 plies with large, flanged access hole cutouts. The basic web thickness is 16 plies (0.08) with a thickness buildup to 28 plies (0.140) at the beam ends. Additional 0 degree plies are added to the upper and lower beam flanges for the required local flange strength and beam stiffness.

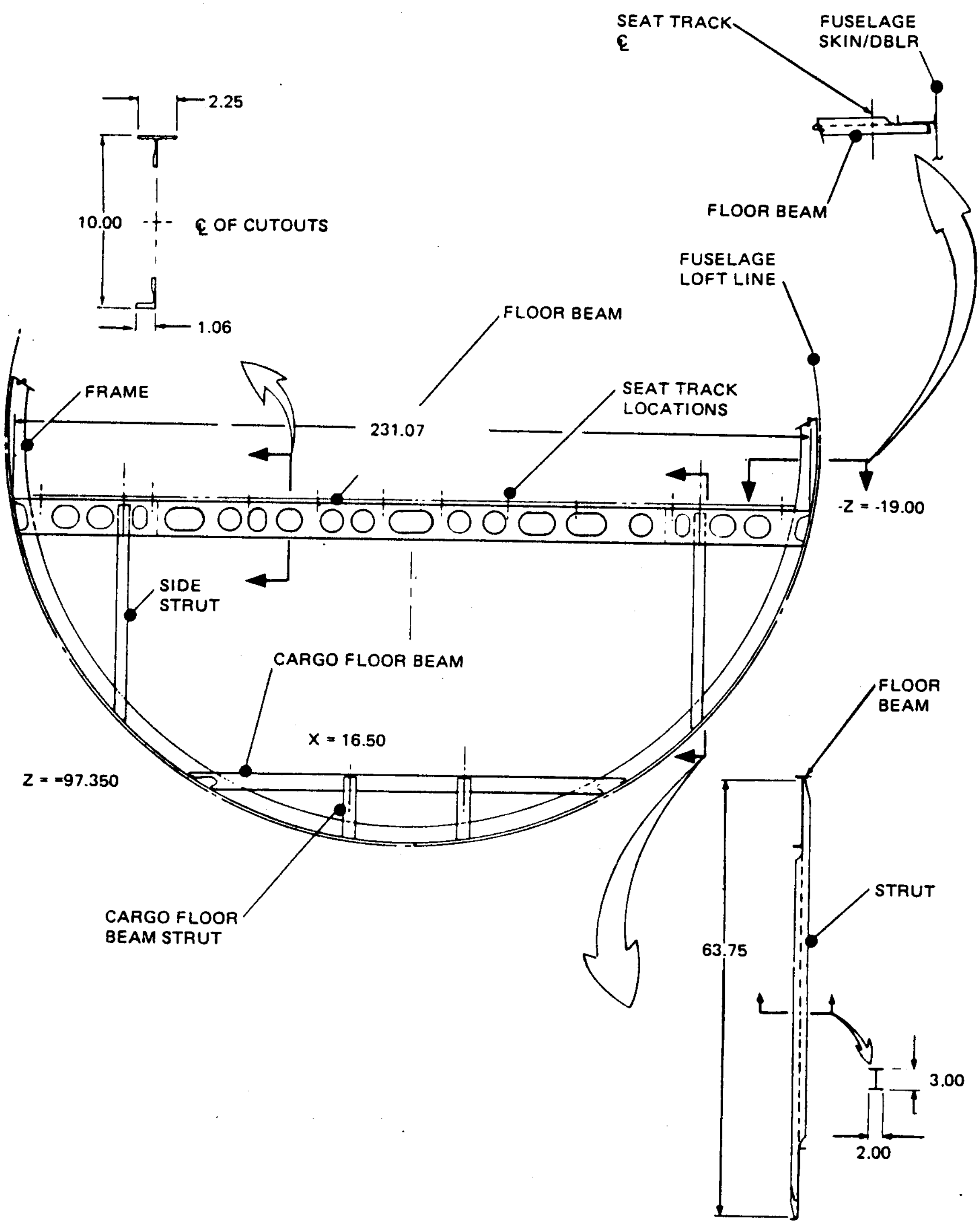


FIGURE 6-23. GENERAL ARRANGEMENT OF FLOOR BEAM AND STRUT

This composite floor beam and support strut configuration have been designed, fabricated, and FAA-certified, and in-service data are presently being accumulated on the configuration. This floor beam is shown installed in a DC-10 fuselage in Figure 6-24. A weight saving of 26 percent from the aluminum baseline was achieved with this carbon-epoxy design.

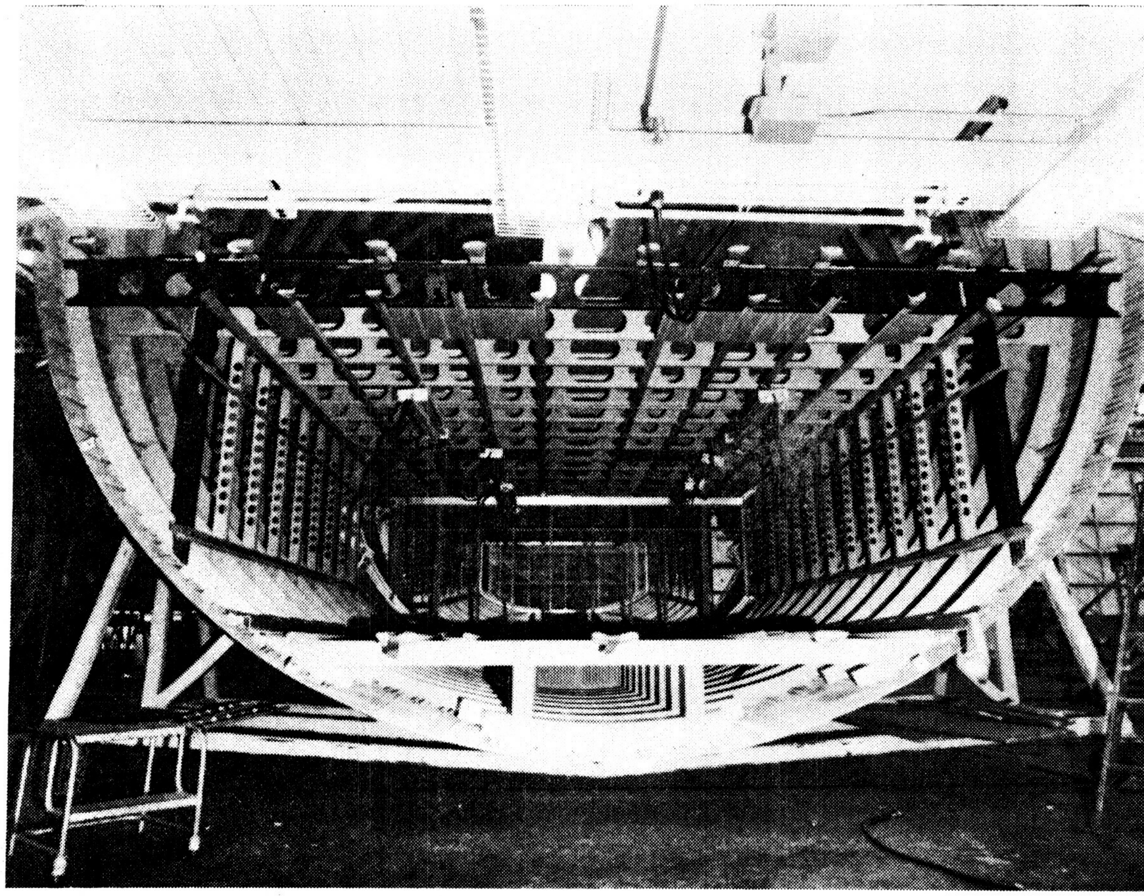


FIGURE 6-24. COMPOSITE FLOOR BEAM

Pressure Bulkhead and Pressure Panel Concepts

The main landing gear wheel well has flat pressure panels above and aft of the cavity, as shown in Figure 6-25. The conceptual design of the aft vertical bulkhead web is a thin (0.080-in.) solid laminate in a pseudo-isotropic layup pattern, as shown in Figure 6-26. The horizontal stiffeners are a sandwich type construction integral with the web. The core may be honeycomb material such as Nomex or a suitable foam. The vertical beams are J-section members in a solid laminate configuration with a beam depth of nearly 8 inches. The vertical beams are on the forward side (wheel well area) of the bulkhead while the horizontal stiffeners are on the aft side of the web.

The horizontal pressure panel forms the pressure boundary between the passenger cabin and the main gear wheel well. The panel is located at the top of the wheel well between the slant panel attached to the wing rear spar and the vertical pressure bulkhead. (See Figure 6-27.) The panel is basically in five segments; i.e., the panel follows the contour of the floor beam lower cap. The center segment and the two outer segments near the outer shell are horizontal. The remaining segments are slightly inclined and complete the pressure panel between the center and outer segments.

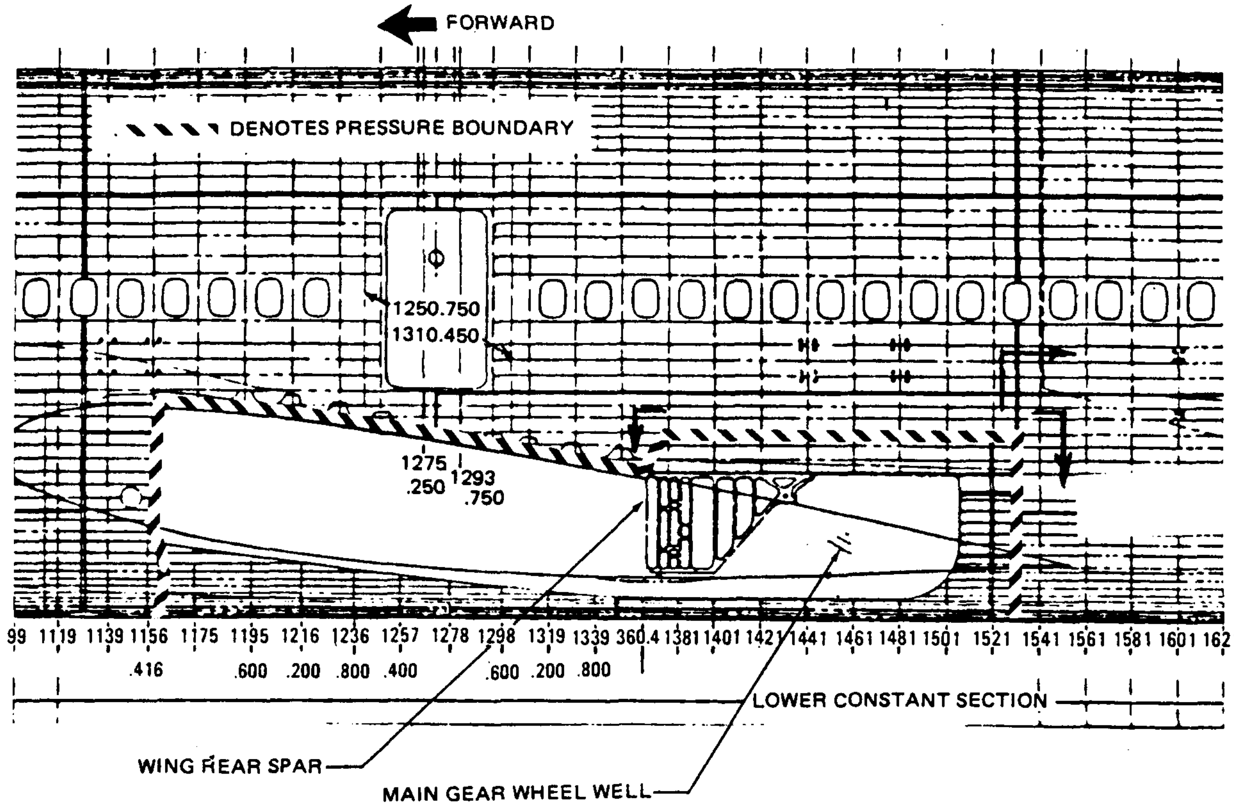


FIGURE 6-25. PRESSURE BULKHEAD AND PANELS

The pressure panel is 0.080 inch carbon-epoxy in a pseudo-isotropic solid laminate with integral hat stiffeners. The stiffeners are located on the lower surface of the panel and are oriented fore and aft with a 7.5- to 8.5-inch spacing. The stiffeners are a sandwich type construction approximately 1 inch deep. The core may be honeycomb material such as Nomex or a suitable foam.

Wing-to-Fuselage Joint

Three different wing joint concepts were considered; pin-joined, shear tee-joined, and a joint through a metal buffer zone. The fuselage was assumed to be attached to a composite wing. This is a reasonable assumption since composite wing technology probably leads fuselage technology at this time. This eliminated the metal buffer zone concept from further consideration since thermal compatibility is not a driving issue with a composite wing. The pin joint concept was eliminated because of the difficulty in handling the very large loads inherent in this type of design. The shear tee concept was selected.

In the shear tee concept, the fuselage loads are reacted by the wing through a titanium shear tee. This concept is shown in Figure 6-28. The shear tee is bolted to the fuselage skin. The bottom of the tee is attached to the wing splice plate by two rows of mechanical fasteners. The joint in this area does not need to be aerodynamically flush because it is covered by the wing-to-fuselage fairing. The frames supporting the sidewall skin in the overwing region of the fuselage are full depth in this area. The frames are

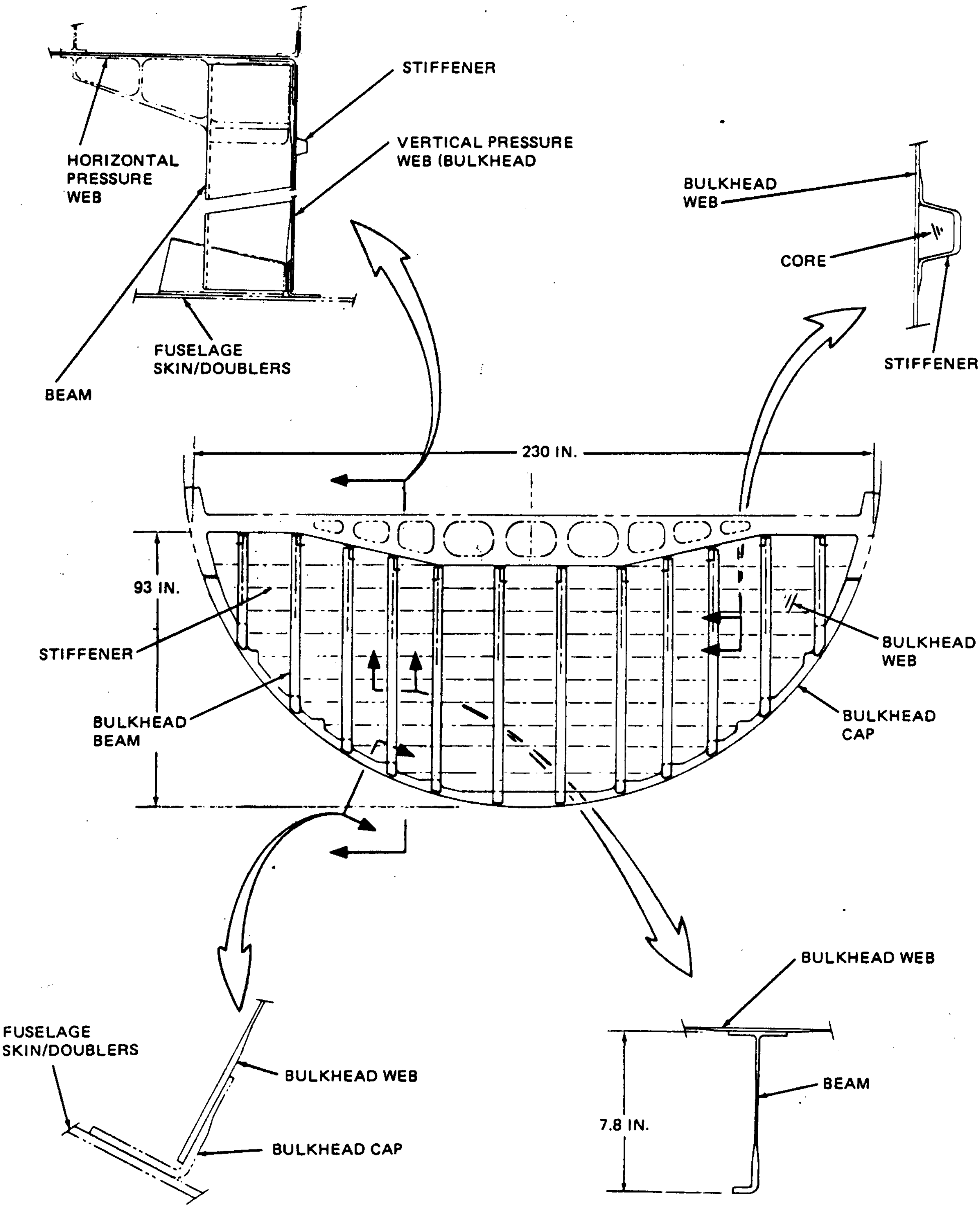


FIGURE 6-26. WHEEL WELL VERTICAL PRESSURE BULKHEAD

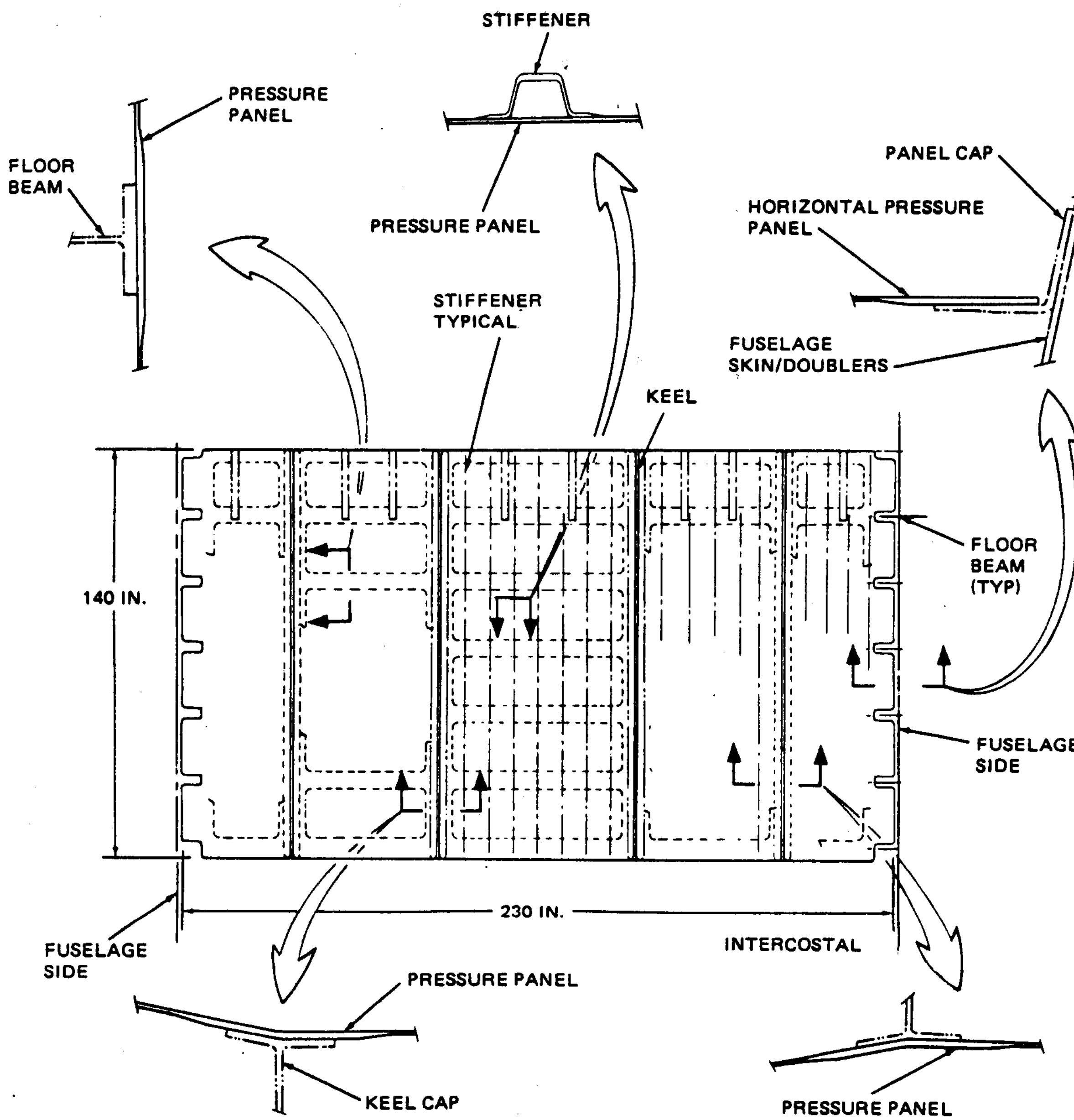


FIGURE 6-27. HORIZONTAL PRESSURE PANEL

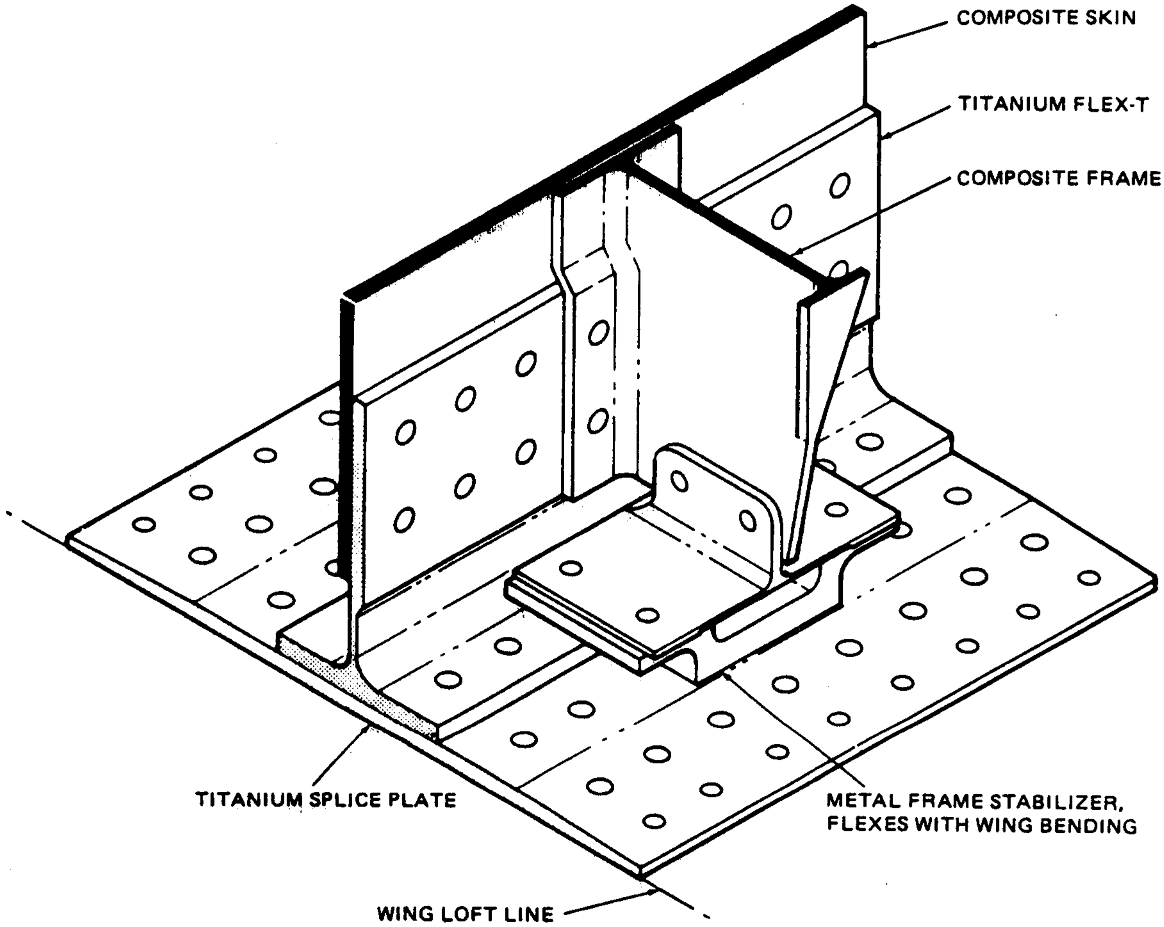


FIGURE 6-28. FUSELAGE SKIN AND FRAME-TO-WING INTERSECTION

attached to the wing through flexible fittings to prevent frame bending moments from being introduced into the wing joint. The fittings are designed in such a way, however, as to allow shear transfer between the wing and frame. A detail of this area is shown in Figure 6-29.

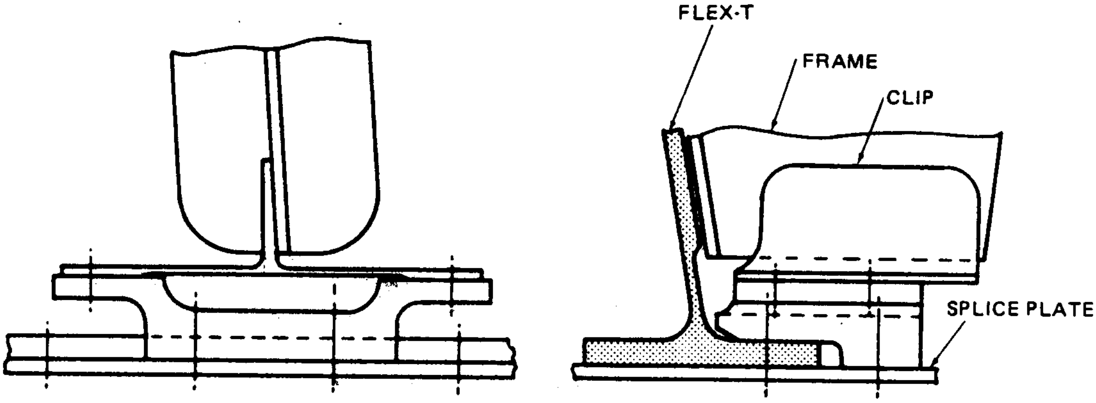


FIGURE 6-29. DETAIL OF FUSELAGE-TO-WING INTERSECTION

Interrupted Longeron Design

The fuselage sidewall area of the overwing region is an area of high shear. The frames in this area are full depth to the skin. The longeron design selected for this area utilizes short segments of blade-section longerons between each frame. This interrupted longeron design has the advantage of not requiring "mouse" holes through the frames. The longerons are attached to each other by three identical fittings and one extended fitting. The fitting and joint design are shown in Figure 6-30. The extended fitting is used for frame stability. A thin fiberglass corrosion barrier is installed between the aluminum fitting and the carbon-epoxy longerons. The fittings are attached to the longeron only; there is no attachment made through the skin. This eliminates the need for shimming of the skin or frame.

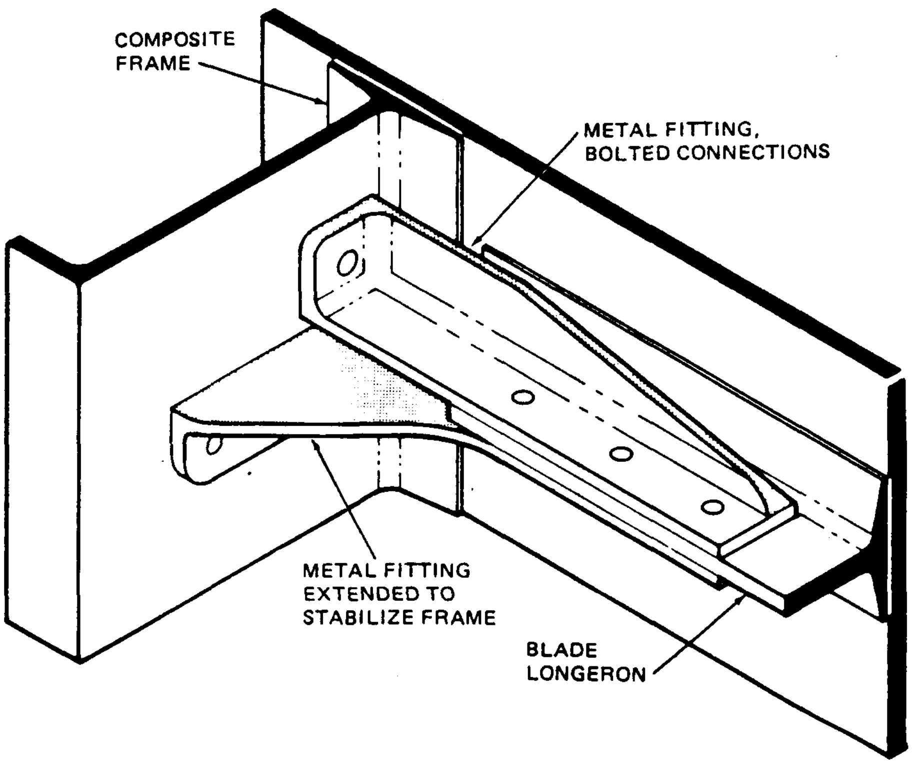


FIGURE 6-30. FULL DEPTH FRAME/INTERRUPTED LONGERON JOINT

Cabin Window Cutouts

A cabin window cutout structural concept has been developed that avoids interlaminar stresses from load introduction. The passenger window concept uses a pressure-sealed window in a lightly loaded frame. The frame simply holds the window in place; i.e., pressure loading is not transmitted from the window to the frame (see Figure 6-31).

Passenger Door Jamb Structure

The passenger door cutout is 78 inches high by 40.5 inches wide. The door cutout lies between two partial (stub) frames and is surrounded by a jamb frame on each side and header beams at the top and bottom. The jamb frames are connected to the stub frames by intercostals which also support the door

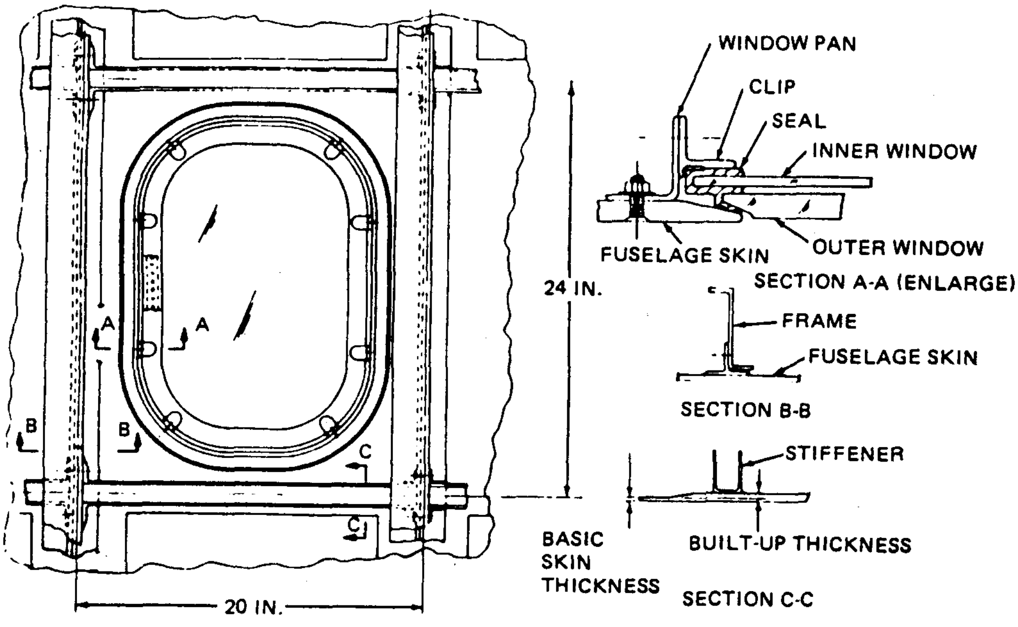


FIGURE 6-31. CABIN WINDOW STRUCTURAL CONCEPT

stops. The header beams are each composed of two continuous members connected by intercostals. The door itself is a plug type which is supported by seven stop fittings mounted on each jamb frame. This system isolates the door from shell loads. The function of the door frame is to redistribute the shell loads around the door cutout and to distribute the door loads due to cabin pressurization into the shell structure. The door frame concept is shown in Figure 6-32.

Door Jamb Frames — The composite jamb frame assists in carrying the internal fuselage shell pressure and flight loads around the side of the passenger door cutout. In addition, the door stop fitting loads are reacted by this frame. The jamb frame distributes these stop loads into the shell structure. The composite stub frame serves a dual purpose; it reacts passenger door jamb intercostal loads, and it provides a fail-safe load path for bending moments in the event of failure of the jamb frame. The jamb and stub frames are full depth integrally molded J-sections bonded and bolted to the skin. The frames are connected to each other through intercostals and a bonded inner skin.

Header Beams — The header beam assembly is a major composite structural component which reinforces the fuselage shell above the passenger door. It is designed to carry the longitudinal loads and bending moments in the region above the door opening. The header beam is composed of two parallel full depth integrally molded J-sections bonded and bolted to the outer skin. The beam elements are connected to each other with closely spaced intercostals and a bonded inner skin.

Intercostal Design — The intercostals provide support for the door stops, thus preventing the door stops from twisting the jamb frames due to pressurization loads. A secondary purpose is to provide fail-safe features in the door jamb structure.

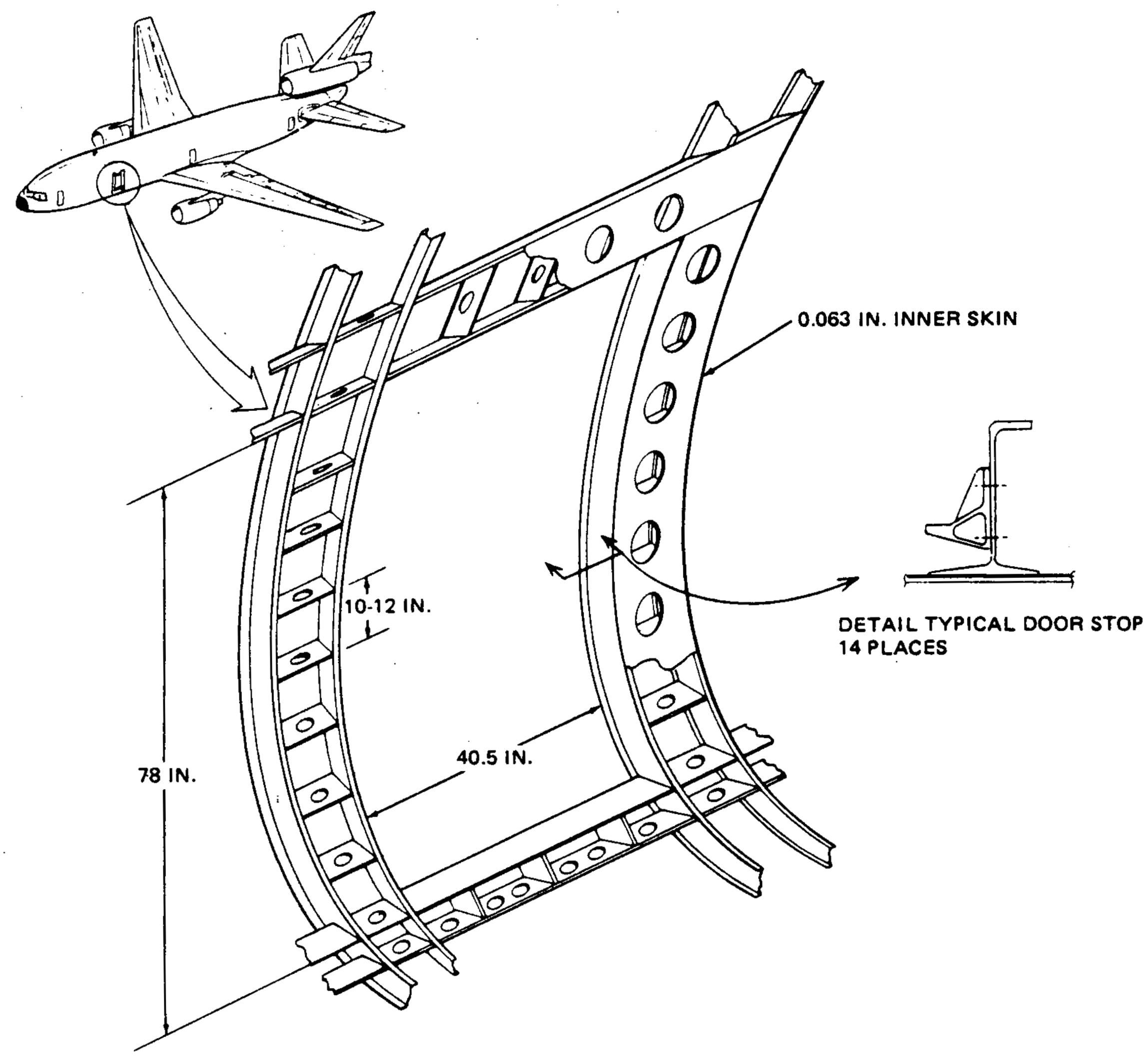


FIGURE 6-32. PASSENGER DOOR FRAME

Two concepts have been defined for the intercostal design. The first uses shear clips to attach the intercostal web to the frame webs and inner skin. This design is relatively insensitive to tolerance problems since the clips can be adjusted to fit. A disadvantage, however, is the number of detail parts required and the time and tooling effort necessary to assemble the individual pieces together.

The second concept utilizing a single-piece molded intercostal with integral flanges appears to have greater potential for production. The flanges are designed with a 6- to 8-degree open angle relative to the matching structure. This enables the intercostal to be installed without clips and yet still tolerate structural variation. The flanges are preloaded on assembly to fit. The reason this concept has not been applied in the past is because preloaded metal structure is prone to stress-corrosion cracking. This limitation does not apply to composite structure. Figure 6-33 shows two intercostal details utilizing both design concepts.

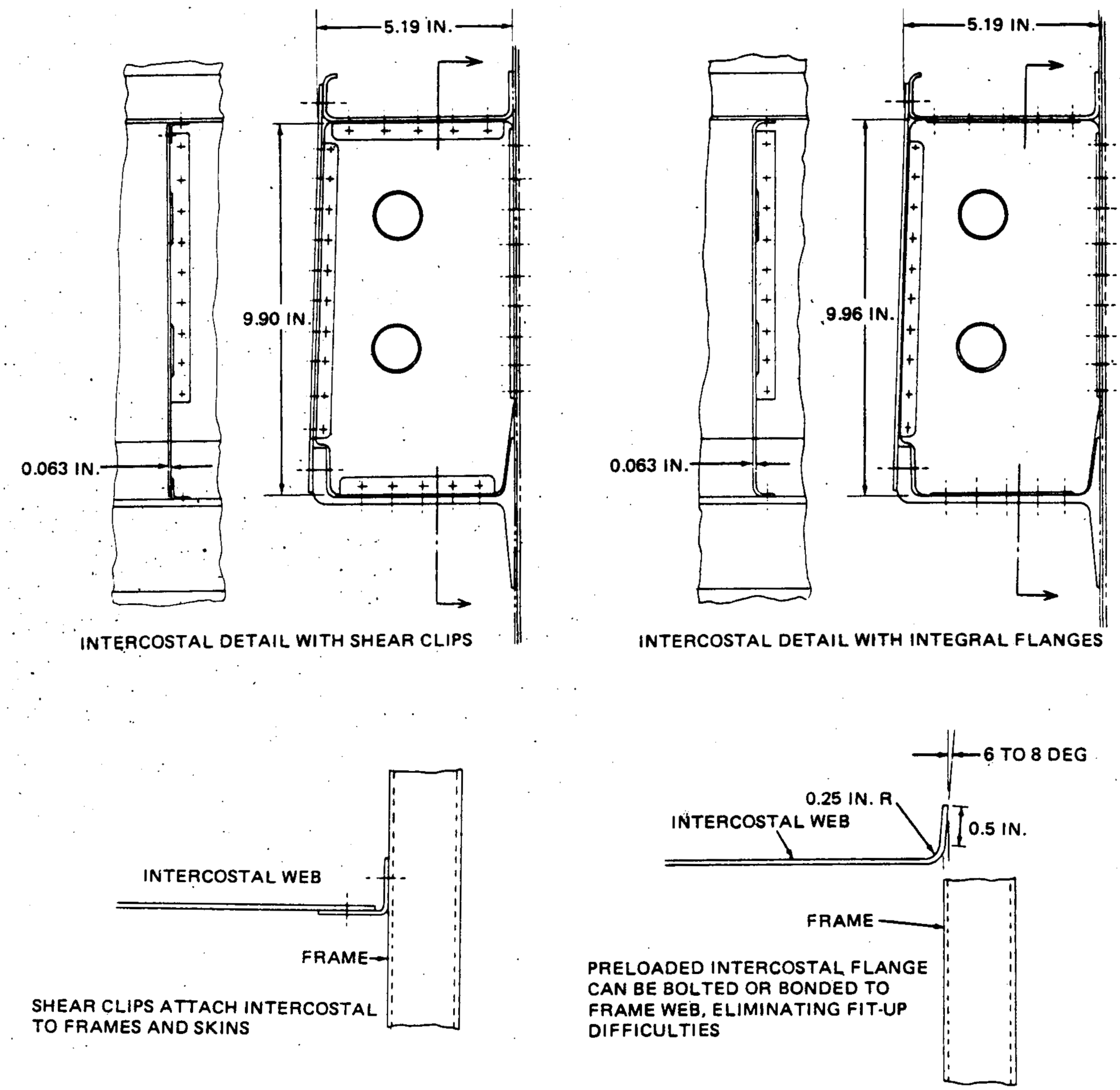


FIGURE 6-33. INTERCOSTAL ARRANGEMENT

Skin Panels — The fuselage panel structure above and below the door opening carries the redistributed shell shears and longitudinal loads resulting from the cutout. The fuselage skin is reinforced in these areas to accommodate the increased loads. The skin reinforcement is shown in Figure 6-34. An alternative reinforcement scheme which may be beneficial uses low-modulus fibers such as fiberglass as a reinforcement in the ±45-degree direction. This may increase the strength of the region around the cutout without a corresponding increase in stiffness and the resulting higher loads which are normally attracted to doublers and reinforcements.

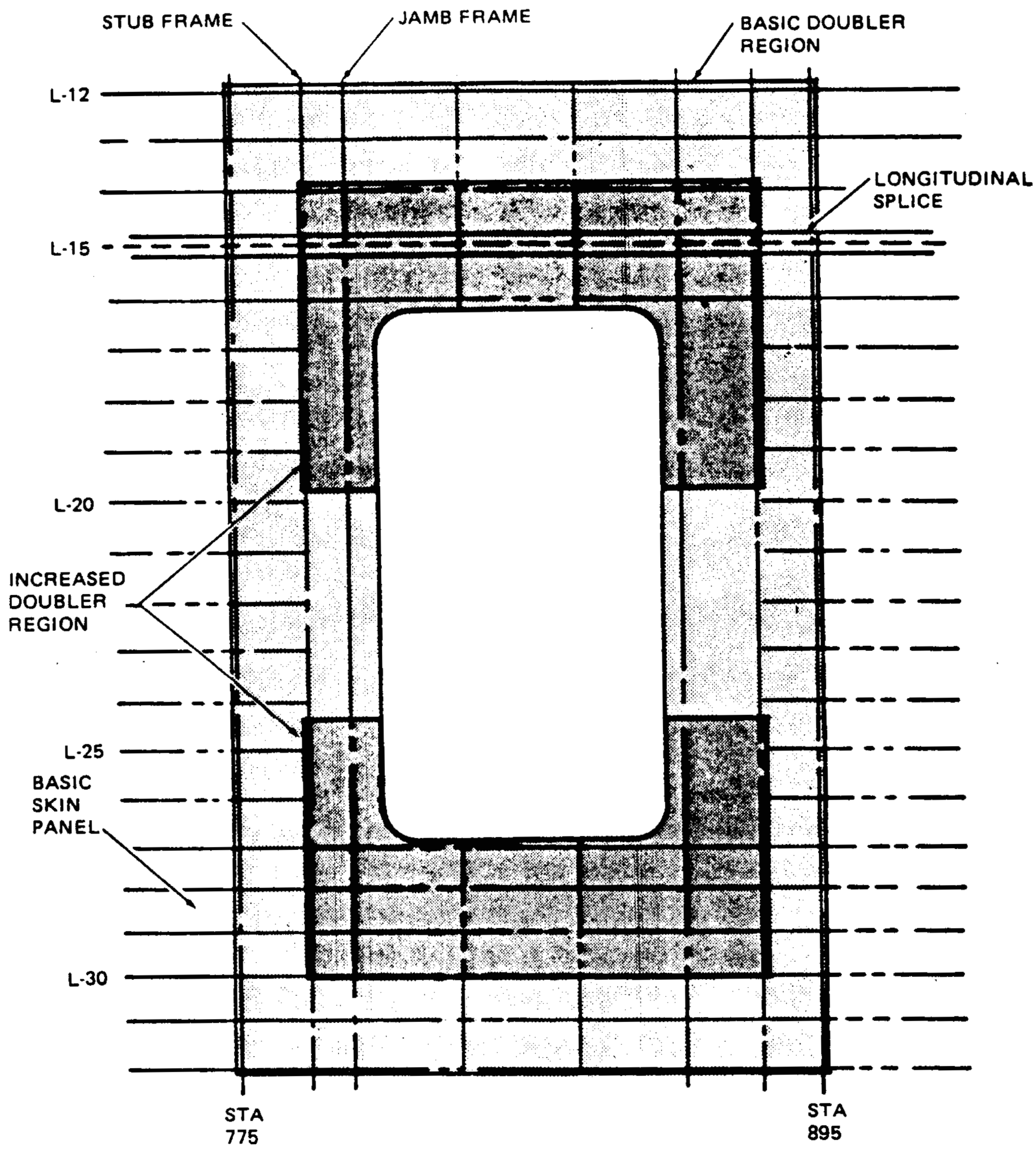


FIGURE 6-34. SKIN REINFORCEMENT AREAS

Nose Section Design

The design philosophy for the nose section is different from the philosophy for fuselage constant-section because of its unique geometry and loading conditions. The nose section is in a complicated compound curvature area which would make the interface between the external skin and the substructure very difficult if a conventional design consisting of built-up structure were used. This is especially true with composite structure since its brittle nature makes fit-up difficult. Although the nose section structure is in an area of relatively low flight loads, it is still subjected to pressure loading and must be capable of supporting the windshield frame, cockpit floor structure and a variety of electric, control system, and instrument installations.

The nose section concept outlined in Figure 6-35 utilizes the moldability of composites to manufacture a complex compound curvature section. The upper nose section is designed as a three-piece structure. Each piece is an integral cocured element consisting of an outer skin, partial inner skin, and I-section frames, and intercostals supporting the two. The upper nose section elements are the two sidewall panels and the flight deck roof. These elements are attached to each other by mechanical fasteners, as shown in Figure 6-36. A titanium windshield frame is attached, as shown in Figure 6-37. A titanium frame is used for thermal compatibility. The structure around the windshield must be able to withstand the windshield pressure loads and anti-icing.

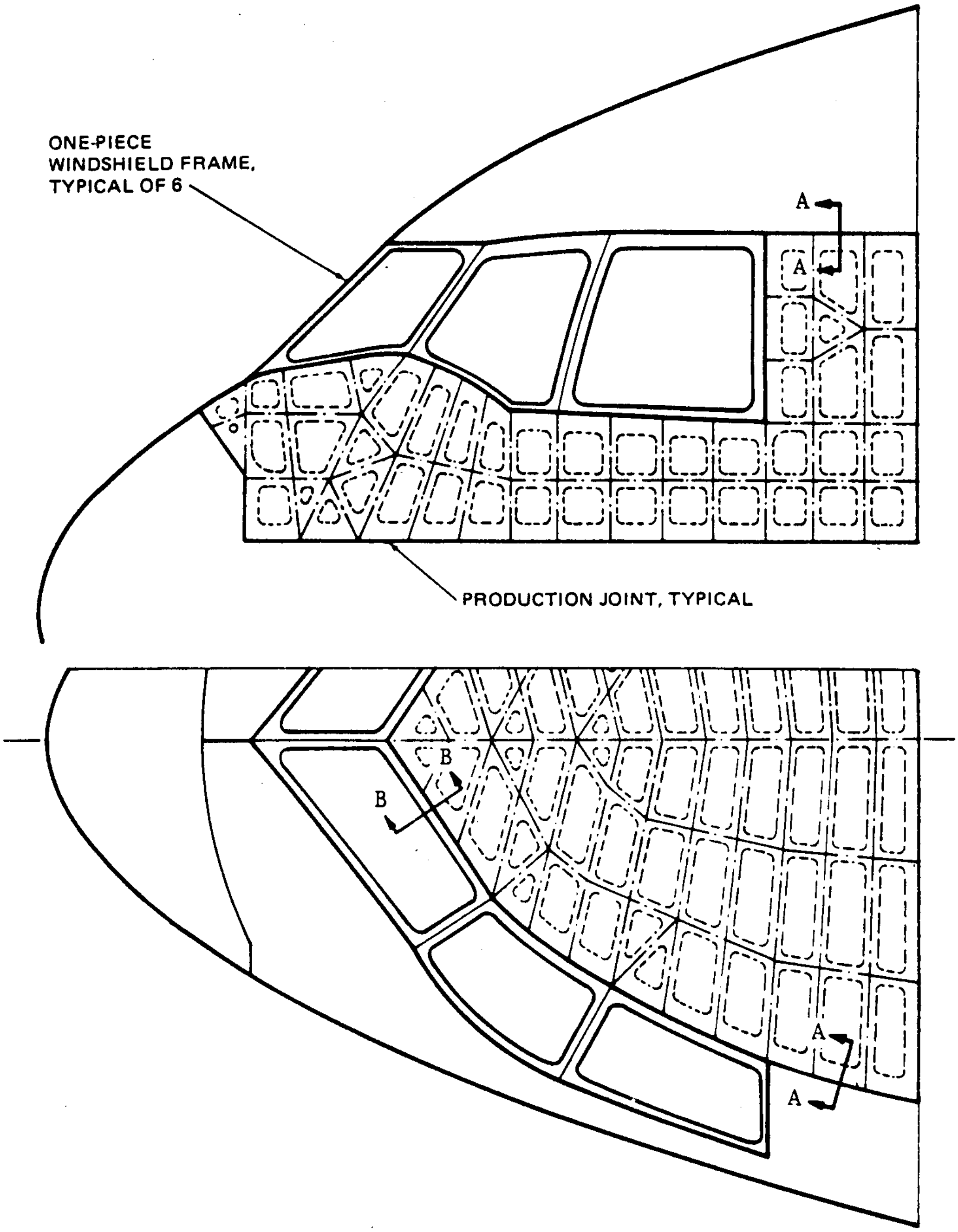


FIGURE 6-35. CO-CURED NOSE SECTION STRUCTURE

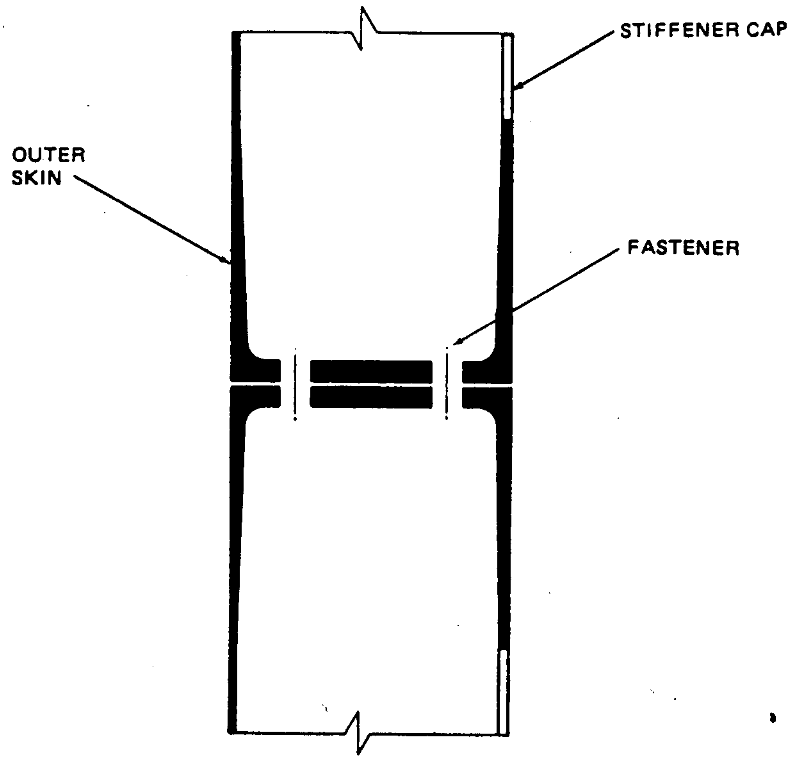


FIGURE 6-36. PRODUCTION JOINT FOR CO-CURED NOSE SECTION

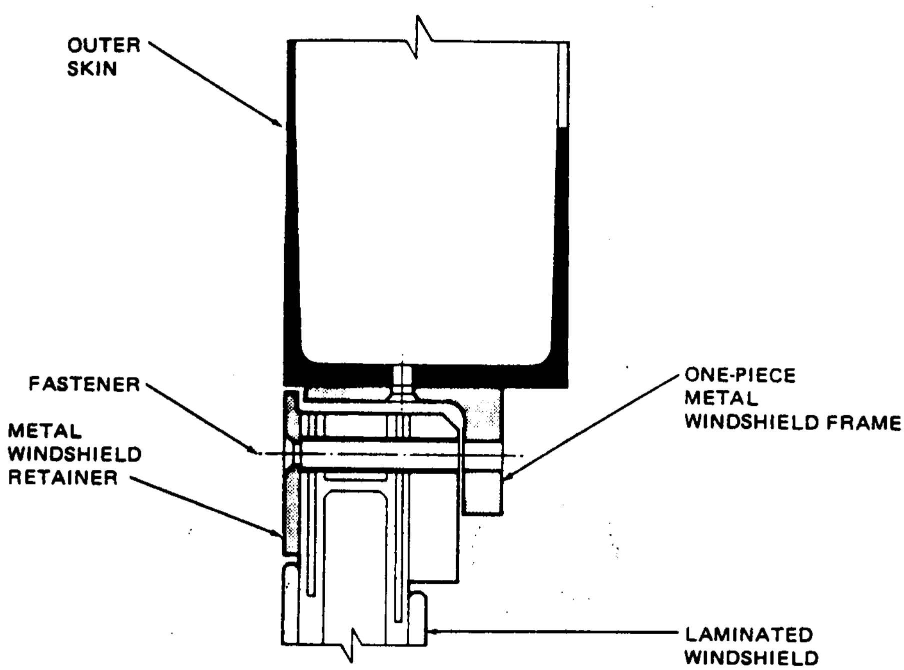


FIGURE 6-37. WINDSHIELD TO NOSE SECTION JOINT

Composite Fuselage Keel Design

The fuselage keel is located aft of the wing box and is primarily configured to provide space for main landing gear. The keel structure is sized to transfer high fuselage bending loads through the wheel well cutout. The fuselage bending loads, which are primarily compressive in this area, are carried past the wheel well cutouts by box beam compression columns. The compression columns are attached to stabilized shear webs. The webs are required to support many secondary loads such as landing gear door actuators and up-locks, mechanisms, and hydraulic components.

In the event of a wheels-up landing or similar accident, the aircraft structure must absorb the vertical impact energy while protecting the passengers or payload. In addition, the wing fuel tank must be protected from scraping on the landing surface. The keel carry-through structure must provide a significant portion of this protection. If the accident results in very high vertical impact energy, the composite keel structure is designed to absorb this energy by the progressive failure of structural elements. The shear web is composed of accordion failure regions and is backed up by vertical stiffeners, sized and attached to the web in such a manner as to provide for this progressive failure. However, very little room is available for vertical deflections in the keel area since the retracted landing gear must not be allowed to penetrate the passenger compartment.

Figure 6-38 shows the tradeoff relationship between clearance and structural thickness for minimum adequate protection. The curve is based on empirical data collected from actual crash experience for metal aircraft. Figure 6-11 shows a preliminary design concept of the keel structure incorporating structural elements for crash protection. These elements are the box beam compression columns which are designed for stability during a crash, the accordion webs, and the stabilized vertical stiffeners which are designed for progressive failure.

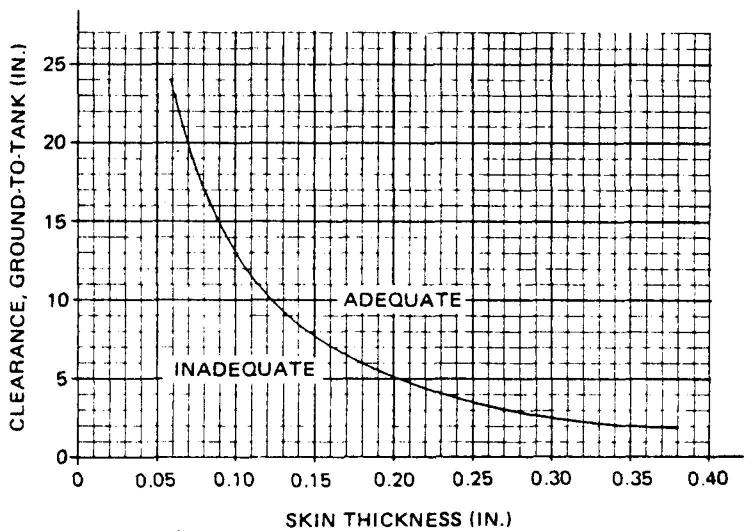


FIGURE 6-38. CLEARANCE VERSUS SKIN THICKNESS REQUIREMENTS FOR CRASH PROTECTION

WEIGHT ANALYSIS

A complete fuselage weight analysis has been made. The assumptions used for this analysis are as follows:

- Frames are assumed to be stiffness-critical; therefore, the bending stiffness of the composite frames was matched to the baseline.
- Longerons are assumed to be critical under axial load; therefore, the axial stiffness of the composite longerons was matched to the baseline.
- Minimum gage skin was assumed to be 0.07 inch. Strength-critical skin gages up to 0.11 inch were used where required.
- Floor beam weight is the same as on the DC-10 composite floor beam flight evaluation article.

No weight penalties were assigned for uncertain requirements such as bird strike or crashworthiness.

Table 6-3 presents a baseline MD-100 weight breakdown. This shows that 80 percent of the fuselage weight is accounted for by seven items. Particular emphasis was placed on the design concepts for these items to obtain an accurate weight savings estimate.

TABLE 6-3

METAL BASELINE WEIGHT BREAKDOWN BY ITEM

ITEM	WEIGHT (LB)	PERCENT OF TOTAL
FRAMES	4,518	11
SKIN PANELS	14,480	35
FLOOR BEAMS	3,948	9
FLOORS	2,049	5
BULKHEADS	2,929	7
DOORS	1,954	5
DOOR JAMBS	2,887	7

The shell weight savings achieved at a somewhat typical strain limit of 0.0045 in./in. was compared with the savings possible at a conservative 0.003 in./in. strain limit and an ambitious 0.006 in./in. limit. A chart showing the incremental shell weights for the three different limits is shown in Figure 6-39. The chart is based on a constant composite longeron height. The skin thickness and layup pattern are allowed to vary with strain limit and load. The chart illustrates the major weight savings that may be achieved over the baseline metal structure.

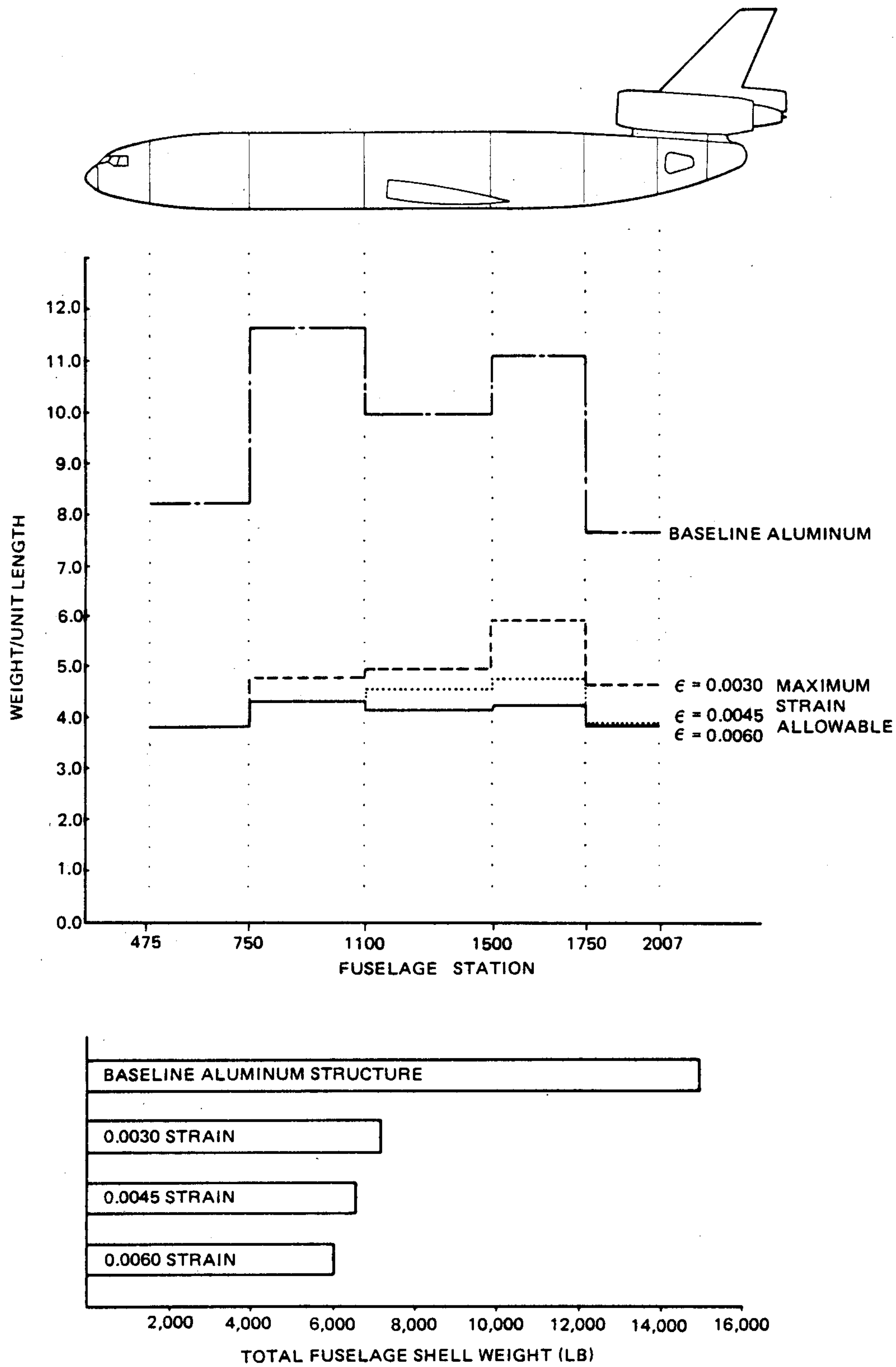


FIGURE 6-39. COMPOSITE FUSELAGE SHELL WEIGHT

Table 6-4 presents detailed weights savings for the total fuselage at a design strain limit of 0.0045 in./in. The weight savings for each type of structure ranged from 56 percent to 10 percent. The overall weights savings for this strain level is 13,249 pounds, or 32 percent of the baseline. Table 6-5 shows a comparison of the weight savings for the total fuselage designed to the three strain limits. It appears that a reduction of strain allowables for the fuselage shell can be prescribed without an undue weight penalty.

TABLE 6-4

COMPOSITE FUSELAGE

ϵ_{MAX} (SKIN) = 0.0045 IN./IN.

ITEM	BASE WEIGHT (LB)	COMPOSITE WEIGHT	Δ WEIGHT	PERCENT CHANGE
FRAMES	4,518	2,530	1,988	44
SKIN PANELS	14,480	6,300	8,180	56
FLOOR BEAMS	3,948	2,921	1,027	26
FLOORS	2,049	1,618	431	21
BULKHEADS	2,929	2,284	645	22
DOORS	1,954	1,368	586	30
DOOR JAMBS	2,887	2,165	722	25
PAINT & LIGHTNING PROTECTION	0	1,230	-1,230	-
OTHER	9,000	8,100	900	10
	41,765 LB	28,516 LB	13,249 LB	32 %

TABLE 6-5

TOTAL FUSELAGE WEIGHT SAVINGS VERSUS DESIGN ULTIMATE

CONFIGURATION	WEIGHT (LB)	WEIGHT SAVINGS (LB)	PERCENT CHANGE
BASELINE METAL	41,765	0	0
0.003 IN./IN. STRAIN	29,316	12,449	29.8
0.0045 IN./IN. STRAIN	28,516	13,249	31.7
0.006 IN./IN. STRAIN	28,241	13,524	32.4

SECTION 7
THE DEVELOPMENT PLAN

The statement of work for the development plan has been scheduled in three phases, as shown in Figure 7-1. Table 7-1 summarizes the tasks to be performed in each phase by departmental functions. Cost, schedule, and technical performance can be monitored and evaluated, and program redirection given as downstream developments deviate from estimates. Each phase can be separately funded to allow a reallocation of funds to support the redirection. This will tend to minimize the programmatic risk associated with creative endeavors.

The development plan contains the following provisions:

- A comprehensive technology development program.
- Design of a 1990s composite fuselage based on the conceptual design.
- Design and construction of large tools for composite parts.
- Test verification to meet FAA structural integrity requirements.
- Monitoring and evaluating the performance of the composite fuselage panel for 5 years while in revenue service.

The development plan is composed of an engineering plan, materials and process plan, manufacturing plan, and test plan.

ENGINEERING PLAN

The Engineering Plan consists of design development in Phase I, structural verification in Phase II, flight service evaluation in Phase III, and engineering support throughout the entire program. Substantiation reports for compliance with Federal Aviation Regulations will be prepared and submitted as required for FAA certification of the aircraft with the composite panel installed for flight service evaluation.

Phase I — Design Development

Engineering activity in the design development phase will be devoted to advanced engineering, engineering technology development, preliminary design, development tests, and panel tests. Design criteria and loads will be used in conjunction with structural arrangements for the structural optimization and design layouts. The design at this point is expanded to include such items as design allowables, candidate materials, a safety program plan, and a verification plan. These designs will provide the basis for a cost/weight evaluation and should have a risk level comparable to that for conventional designs.

The engineering design development effort involves the design integration process shown in Figure 7-2. This process is an iterative one which will parallel and interface with the manufacturing, development

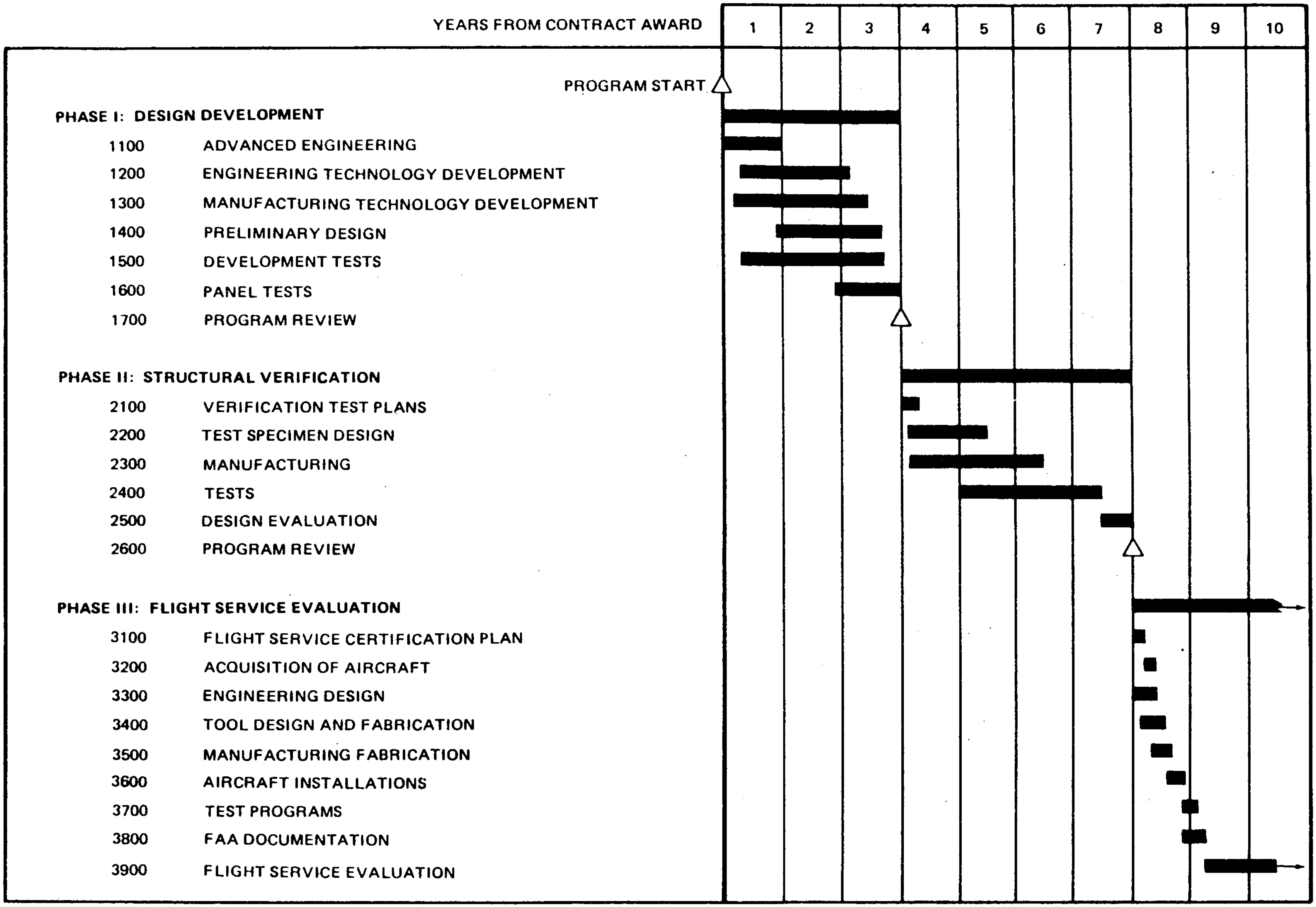


FIGURE 7-1. THE DEVELOPMENT PLAN SCHEDULE

TABLE 7-1

THE DEVELOPMENT PLAN

	ENGINEERING PLAN	MATERIAL AND PROCESSES PLAN	MANUFACTURING PLAN	QUALITY ASSURANCE PLAN	TEST PLAN
PHASE I DESIGN DEVELOPMENT	- ADVANCED ENGINEERING - ENGINEERING TECH DEVT - PRELIMINARY DESIGN	- TECHNOLOGY DEVELOPMENT - MATERIALS SELECTION	- MANUFACTURING TECHNOLOGY DEVELOPMENT	- CONCEPT REVIEW	- DEVELOP-MENT TESTS - PANEL TESTS
PHASE II STRUCTURAL VERIFICATION	- TEST SPECIMEN DESIGN - DESIGN EVALUATION	- MATERIAL AND PRODUCI-BILITY STANDARDS - NDI STANDARDS	- MANUFACTURING	- SPECIFICATIONS	- VERIFICA-TION TEST PLANS - TEST
PHASE III FLIGHT SERVICE EVALUATION	- FLIGHT SERVICE CERT PLAN - ACQUISITION OF AIRCRAFT - ENGINEERING DESIGN - STANDARD REPAIR MANUAL - ACCEPT/REJECT CRITERIA	- SUPPORT - NDI METHODS - NDI STANDARDS	- TOOL DESIGN AND FABRICATION - MANUFACTURING FABRICATION - AIRCRAFT INSTALLATIONS	- MATERIAL PROCESSES - FABRICATION AND ASSEMBLY	- TEST PROGRAM - FAA DOCU-MENTATION - FLIGHT SERVICE EVALUATION

test, and maintenance and repair activities. The initial evaluation of structural candidates will determine which concepts are to be designated for development and test. Data from these evaluations will be fed back to the layout effort for design refinement and to the trade studies for reevaluation and elimination of the less efficient concepts. This process will result in the preliminary design of the concept selected for detail design and fabrication of a full-size composite fuselage barrel.

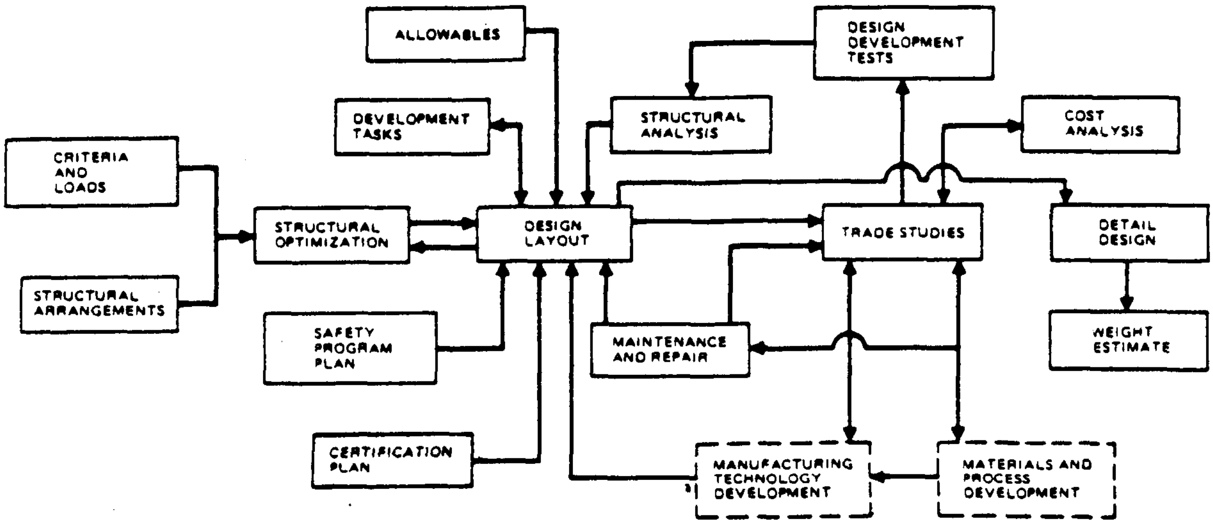


FIGURE 7-2. DESIGN INTEGRATION PROCESS

Structural Design Criteria and Loads — The basic criterion to be followed throughout this program is that the composite fuselage must be comparable to the aluminum fuselage in all areas of structural integrity, flight systems performance, ground handling, and maintenance. To achieve this, the composite fuselage will be designed to satisfy all Federal Aviation Regulations applicable at the start of Phase II. Compliance with applicable FARs will be demonstrated in accordance with FAA Advisory Circular 20-107, "Composite Aircraft Structure." A summary of FAA criteria with the source of each requirement is presented in Table 7-2. In the military transport case, the design integration process differs from the commercial in only one way: the aircraft structural integrity plan replaces the certification plan. (See Figure 7-3.) A design requirements list comparable to the certification guidelines is shown in Table 7-3.

Both civil and military criteria are intended to achieve the same objective: assuring a high level of structural integrity throughout the operational life of the aircraft. The significant operational differences between the MD-100 civil transport and a large military transport are for STOL operations on unimproved runways. These ground operations would result in higher loads for the landing gear support structure and the floor support structure of a military transport. Foreign object damage to the lower fuselage skin panels would also be more severe. Actual operational loads during flight are comparable since high airplane maneuvering load factors for large transports occur only during avoidance situations. The criterion for calculating gust load factors is slightly different for military and civil aircraft, but the actual airplane response varies only with respect to difference in aerodynamics, weight, speed, altitude, and other factors.

TABLE 7-2

CERTIFICATION GUIDELINES FOR CIVIL COMPOSITE FUSELAGE AIRCRAFT STRUCTURES

• MATERIAL AND FABRICATION DEVELOPMENT	- FAR 25.603, 25.613 AND 25.615
• PROOF OF STRUCTURE - STATIC	- FAR 25.305 AND 25.307 (A)
• PROOF OF STRUCTURE - FATIGUE/DAMAGE TOLERANCE	- FAR 25.571
• PROOF OF STRUCTURE - FLUTTER	- FAR 25.629
• IMPACT DYNAMICS	- FAR 25.561, 25.601, 25.721, 25.783(C)(G), 25.785, 25.787(A)(B), 25.789, 25.801, 25.809, AND 25.963(D)
• FLAMMABILITY	- FAR 25.609(A), 25.853, 25.855, 25.859, 25.863, 25.865, 25.867, 25.903(C), 25.967(E), 25.1121(C), 25.1181, 251182, 25.1183, 25.1185, 25.1189(A)(2), 25.1191, AND 25.1193(C)(D)(E)
• LIGHTNING PROTECTION	- FAR 25.581 AND 25.609
• PROTECTION OF STRUCTURE	- FAR 25.609
• QUALITY CONTROL	- FAR 21.143
• REPAIR	- FAR 121.67(A) AND 43.13(A)
• FABRICATION METHODS	- FAR 25.603 AND 25:605

The MD-100 transport program will provide composite technology which applies equally to military transports except for the higher ground loads and for special military criteria such as battle damage, survivability, and nuclear weapons effects.

Although the ground loads are higher and there is more severe exposure to foreign object damage, the MD-100 technology should provide an adequate data base for the military transport structure influenced by these parameters. Composite technology issues related to military threats are being addressed by ongoing and future Air Force/Navy fighter aircraft programs which are expected to contribute to the data base.

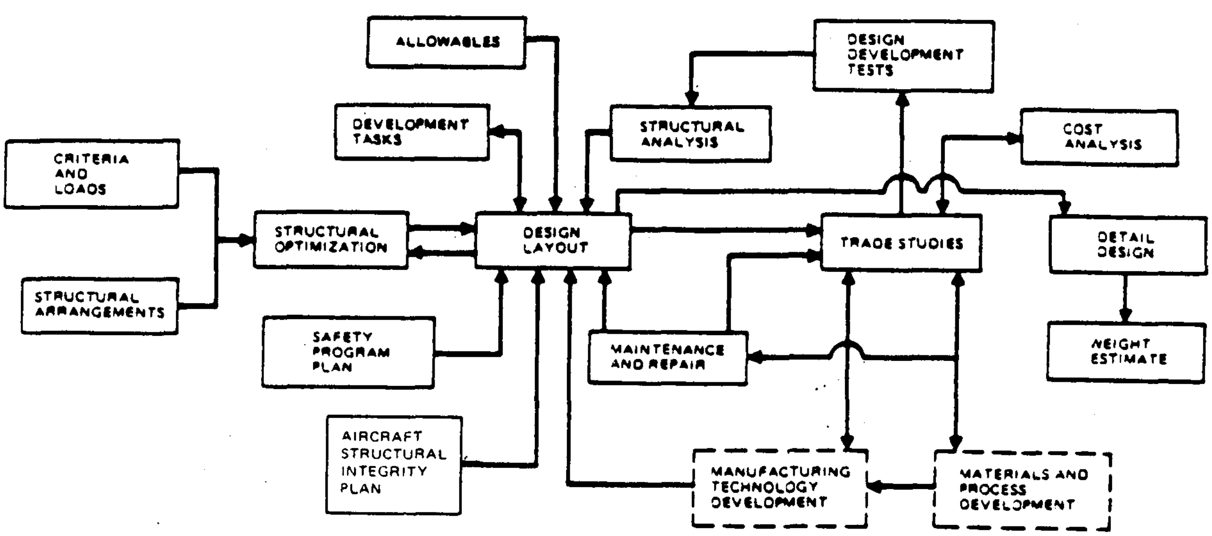


FIGURE 7-3. DESIGN INTEGRATION PROCESS

TABLE 7-3

AIR VEHICLE DESIGN REQUIREMENTS FOR A MILITARY TRANSPORT FUSELAGE STRUCTURE

CRITERIA	SPECIFICATION
• AIRCRAFT STRUCTURAL INTEGRITY	- MIL-STD-1530 AND RELATED SPECIFICATIONS. - AFSC DH 2-1, DESIGN NOTE 3A6 - AIRCRAFT STRUCTURAL INTEGRITY CONTRACTUAL PLAN
• DURABILITY AND DAMAGE TOLERANCE	- MIL-STD-1530
• MATERIALS AND PROCESSES SELECTION AND CORROSION PREVENTION, AND CONTROL	- MIL-STD-1568 AND MIL-STD-1587
• MANUFACTURING AND PROCESSES CASTINGS AND FORGINGS	- MIL-STD-1587
• QUALITY ASSURANCE PROVISIONS	- MIL-STD-8860
• OPERATION IN AMBIENT ATMOSPHERE	- MIL-STD-210
• CASTING: CLASSIFICATION AND INSPECTION	- MIL-C-6021
• MINIMUM FLYING WEIGHT AND MAXIMUM DESIGN WEIGHT	- MIL-W-25140, SECTION 6
• STRUCTURAL INTEGRITY OF COMPOSITES	- MIL-A-8860 WITH ADDITIONS TO ESTABLISH: A. EXPECTED ABSORPTION RATE AND SATURATION LEVEL OF MOISTURE IN THE COMPOSITE MATRIX. B. RESULTANT STRENGTH/MODULUS AND FATIGUE LIFE DEGRADATION ASSOCIATED WITH THIS MOISTURE CONTENT AND TEMPERATURE EXTREMES. C. DESIGN ALLOWABLES REFLECTING THE WORST-CASE CONDITION. D. A STATISTICAL DESCRIPTION OF COMPOSITE FAILURE PARAMETERS E. VALIDITY OF FATIGUE/ENVIRONMENT INTERACTION EFFECTS FROM COUPON TESTS BY TESTS OF REPRESENTATIVE SUBCOMPONENT STRUCTURE. F. REDUCTION IN RESIDUAL STRENGTH CAPABILITY AS A RESULT OF EXPOSURE TO FATIGUE LOADS WITH THERMAL AND HUMIDITY ENVIRONMENT.
• AIRPLANE STRENGTH AND RIGIDITY FLIGHT LOADS	- MIL-A-8861 AND MIL-A-83444
• LANDING AND GROUND HANDLING LOADS	- MIL-A-8862
• MISCELLANEOUS LOADS	- MIL-A-8865
• RELIABILITY REQUIREMENTS, REPEATED LOADS AND FATIGUE	- MIL-A-8866
• GROUND TESTS	- MIL-A-8867
• NUCLEAR WEAPONS EFFECTS	- MIL-A-8869 SHALL NOT APPLY
• VIBRATION, FLUTTER AND DIVERGENCE	- MIL-A-8870
• AIRPLANE TESTS, STRENGTH AND RIGIDITY FLIGHT AND GROUND OPERATIONS	- MIL-A-8871

TABLE 7-3

AIR VEHICLE DESIGN REQUIREMENTS FOR A MILITARY TRANSPORT FUSELAGE STRUCTURE (CONTINUED)

CRITERIA	SPECIFICATION
• AIRPLANE STRENGTH AND RIGIDITY, VIBRATION	· MIL-A-8892
• AIRPLANE STRENGTH AND RIGIDITY, SONIC FATIGUE	· MIL-A-8893
• AIRPLANE DAMAGE TOLERANCE REQUIREMENTS	· MIL-A-83444

NOTE: SOME OF THE ABOVE SPECIFICATIONS MAY BE MODIFIED FOR SPECIFIC PROGRAMS. PORTIONS OF APPLICABLE MILITARY SPECIFICATIONS ARE GRANTED EXCEPTIONS, MODIFICATIONS, AND/OR REPLACED WITH SPECIAL REQUIREMENTS.

Candidate Concepts — Preliminary layouts and concepts will be based upon the MD-100 baseline aircraft (Figure 7-4). There are two basic differences in primary structure between commercial and military transports. First, the commercial airplane is designed to transport people and some cargo, while the military transport is designed to carry large, heavy vehicles and weapons that can be quickly loaded and unloaded. The latter requirement results in a large cargo door and cargo ramp which are absent from the typical civil transport (see Figure 7-5). The second difference is in structural arrangement; e.g., the MD-100 has a low wing versus a high wing for a typical military transport.

The differences in detail design between the civil and military transports are such that FAA certification requirements for civil transport basically establish the confidence that the design requirements for the military transport can be met. For example, the large cargo door (224 by 384 inches) in the lower aft fuse-

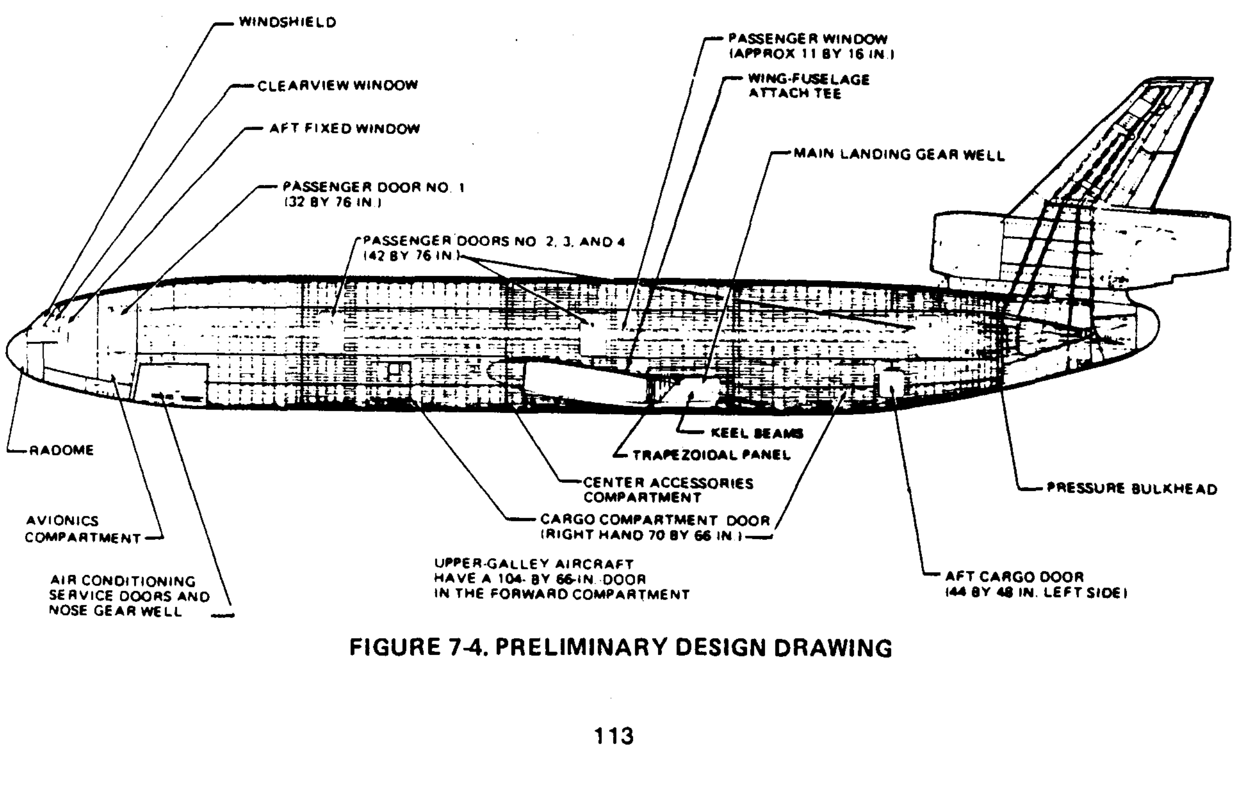


FIGURE 7-4. PRELIMINARY DESIGN DRAWING

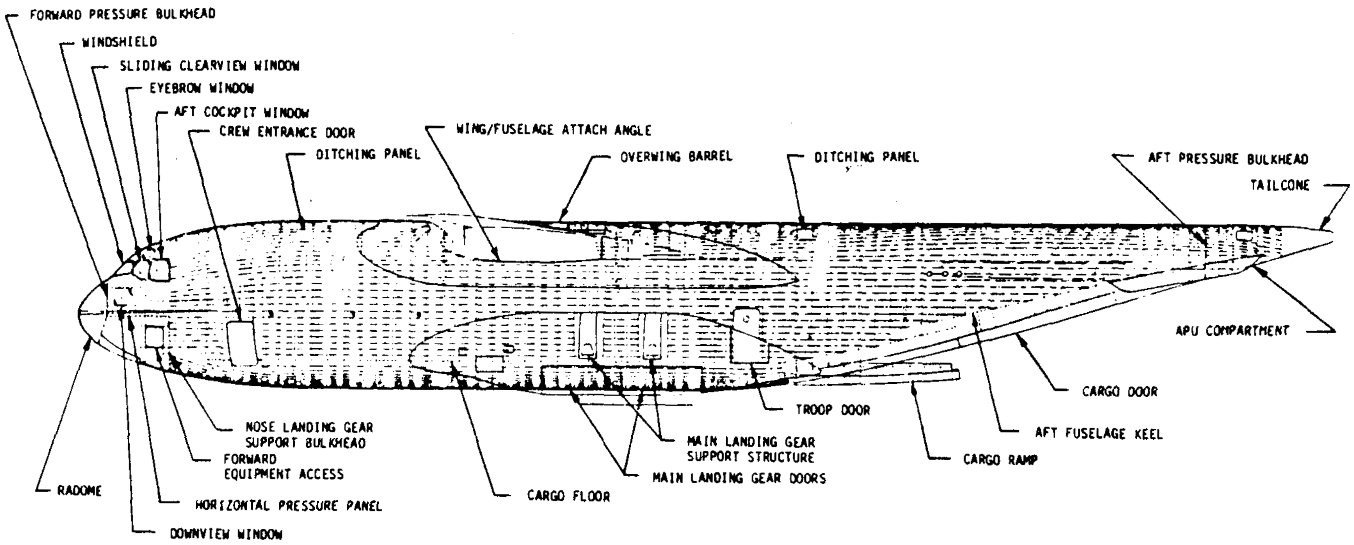


FIGURE 7-5. ADVANCED MILITARY TRANSPORT

lage of the typical military transport is actually a larger, heavier version of the cargo door in the lower forward fuselage of the MD-100. In other words, the resolution of all technology issues for the civil transport should provide the design understanding and data, and the analytical methodology and capability needed to accomplish the engineering on the larger aft cargo door for the military transport.

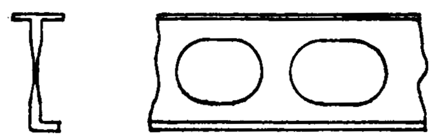
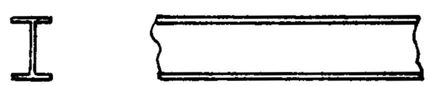
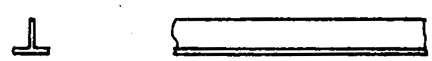
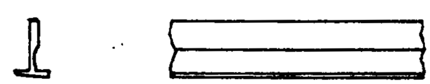
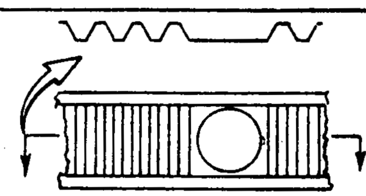
The differences in structural arrangement are a relatively simple matter. The high wing on the military transport can also be found on a relatively large number of production commuter aircraft. The technology issue is the design of the cutout in the pressurized fuselage where the wing structural box penetrates the shell. The very high loads, strains, and deflections that occur at the wing/fuselage intersection are essentially the same regardless of the vertical position of the wing relative to the fuselage.

Structural concepts are grouped into three categories: (1) basic structure such as skin-longeron elements, (2) reinforcement of basic structure around cutouts for doors and windows, and (3) joints and fittings to transfer loads through discontinuities in the structural elements.

Discrete longitudinal stiffeners (longerons) and frames at standard spacing have been selected for the composite fuselage. Other arrangements including full depth honeycomb have been considered in previous studies that have verified the closely spaced longerons and closely spaced frames as the best overall configuration from the standpoint of structural efficiency, maintainability, and repair.

Candidate subcomponent concepts and joints to be included in design integration are presented in Table 7-4. For skin panel assembly concepts, the longeron configuration has demonstrated efficient application to composite designs in previous efforts. Investigations to date have tended to indicate J-section stiffened panels as the most cost- and weight-effective in this application because of the required stiffness constraints. However, these studies have been preliminary. All concepts presented will be considered candidates until eliminated by a more thorough investigation. Candidate joints and fittings are the standard ones generally considered for composite applications.

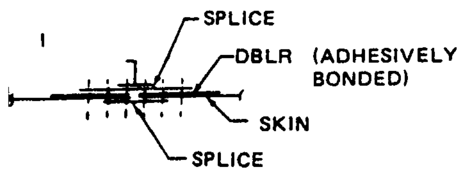
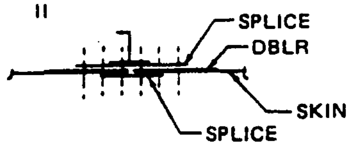
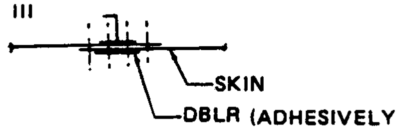
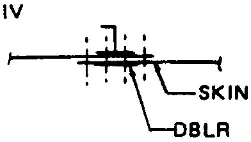
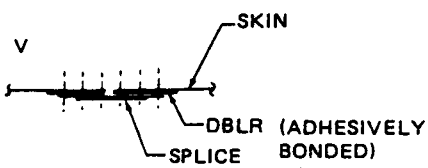
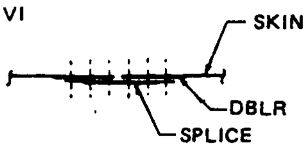
TABLE 7-4
CANDIDATE CONCEPTS

TYPE	CONFIGURATION	APPLICATION*
STIFFENED SHEET	J 	A, B
	HAT 	B
SANDWICH	HC CORE 	E
J-SECTION		C, D
H-SECTION		D
T-SECTION		F
		G
CORRUGATION		C

NOTE:

*A - SKIN PANEL, B - BULKHEAD, C - FLOOR BEAM, D - FLOOR BEAM STRUT, E - KEEL, F - SHEAR TEE, G - WING/FUSELAGE ATTACH TEE

TABLE 7-4
CANDIDATE CONCEPTS (CONTINUED)

TYPE	CONFIGURATION	APPLICATION
JOINTS	I  DOUBLE STRAP BUTT SPLICE II  DOUBLE STRAP BUTT SPLICE III  LAP SPLICE IV  LAP SPLICE	LONGITUDINAL SPLICE
	V  SINGLE STRAP BUTT SPLICE VI  SINGLE STRAP BUTT SPLICE	TRANSVERSE SPLICE

Structural Optimization — Structural optimization, the initial concept evaluation effort, serves a twofold purpose. First, it narrows the field of candidate component concepts to a manageable number for design development. Second, it provides preliminary structural sizing and weight estimates for remaining concepts. The optimization process entails determining the structural arrangement, section geometry, and element sizes which result in the least weight for each candidate. The relative weights of the candidates are then compared and those demonstrating the greatest structural efficiency without indicating a potential for excessively high cost or risk are retained for further study.

Lightning Protection Features — The low-conductivity characteristic of carbon-epoxy materials relative to aluminum must be considered in the structural design of the composite fuselage. A study assumption was made that conductive materials will be developed by either making the carbon fiber more conductive or by interweaving other conductive fibers with the carbon fiber. The task of developing special conductive material is not included in this program, although the application of the conductive material is included. The increased conductivity materials must be tested to establish their strength, durability, and damage tolerance properties. These materials should then be utilized in the construction of large demonstration test articles unless it can be adequately shown by ancillary tests that the treatment to make the composite material more conductive has a negligible effect on strength, durability, and damage tolerance properties.

Changes in the aircraft avionics/electrical systems caused by the use of less conductive composite material are outside the scope of this development program. Total aircraft cost and weight may be increased to satisfy electromagnetic effects criteria such as rerouting of wire bundles, shielding, and the selection of less efficient systems due to lowered fuselage conductivity.

Design Layout — The design layout effort will establish initial structural arrangements of candidate concepts for further optimization studies. The design layouts will also be used to incorporate these concepts into an integrated fuselage design which reflects the compromises that must be made.

Preliminary layouts of the structural candidates will be prepared in enough detail to determine limitations on element size and spacing for optimization studies. These will include advanced design of joints between skin panels, frames, and bulkheads; rough layouts of interfacing structure and systems; and laminate patterns in areas of low loading.

Those concepts selected as a result of the optimization studies will then be further developed. The layouts will define the major structural and manufacturing aspects of design integration into a complete fuselage structure. This effort will proceed along the same lines as described for the conceptual design. Layouts will be made of major structural members and typical substructure, joints, and interface structure. The basic sections of the skin panels and frames will be designed at a number of stations along the fuselage length and sized for minimum weight.

The internal substructure will be defined by preliminary layouts of a typical frame, floor beam, floor beam support, and cargo floor. Supporting frames and structure will be designed at the wing-to-fuselage attachment and the landing gear support.

The design layouts will define each candidate concept to the extent required for trade study evaluation and determination of development test requirements and specimen design. The layouts will be continually updated as more complete strength analyses refine component sizing, and manufacturing, maintenance, and test data inputs indicate the need for design changes. The layouts will be periodically reevaluated by trade studies during the design integration iteration.

Trade Studies — Trade studies will be the second evaluation effort after structural optimization. These studies will compare the candidate concepts as defined by design layout in terms of weight, cost, and risk. The result is the selection of the concept designated for detail design and fabrication.

Initial trade studies will narrow the field of candidates down to a number that can reasonably be carried through the development and test efforts while permitting the program to remain within budget. The structural arrangement of the longeron and frames of the baseline MD-100 fuselage will be retained. Skin panel, frame, and bulkhead candidate concepts will be narrowed down to two or three by initial trade studies.

The trade studies will keep abreast of all development efforts. As the layouts are revised by application and analysis of new data, the trade studies will be updated. Candidates will be evaluated until one concept is clearly established as the most efficient, considering all areas of design, fabrication, maintenance, and repair.

Structural Analysis — The structural analysis effort involves methods development and structural sizing. The approach includes theoretical analysis, definition of development test plan requirements, and interpretation of results.

Plate and shell analysis methods are used in the design integration phase. Composite structural analysis is based on orthotropic analysis techniques which have been developed at Douglas during the past few years on both in-house and contracted programs. Both design charts and computer programs are available for composite structural analysis. Computer programs are more versatile and generally provide the more complete analysis. Existing programs at Douglas that can be used to optimize and analyze basic structure, joints, and other discontinuities are presented in Table 7-5. The analysis task includes the development and verification of methodology to provide fast and reliable assessment of structural integrity.

The strength of skin panels, frames, and bulkheads under basic fuselage bending, shear, and torque will be considered in the structural optimization. Additional strength analysis of these components will include critical combinations of loadings. All modes of failure will be investigated.

Special attention will be given to joints, cutouts, and supporting structure. Analysis of strength of mechanical attachments and local areas in the vicinity of fittings will require analysis of stress distribution, theoretical strength prediction, and interpretation of data as they become available from development tests.

Structural assessment of basic structure, joints, cutouts, and assemblies will include damage tolerance, durability, and repair procedures based on both analysis and test results.

TABLE 7-5

AVAILABLE COMPUTER PROGRAMS

NASTRAN	GENERAL-PURPOSE STRUCTURAL ANALYSIS PROGRAM SOLVING PROBLEMS BY THE FINITE-ELEMENT METHOD
CASD	COMPUTER AIDED STRUCTURAL DESIGN, AN IN-HOUSE STRUCTURAL ANALYSIS PROGRAM USING THE FINITE-ELEMENT METHOD
A4EI	ANALYSIS PROGRAM FOR ADHESIVE-BONDED JOINTS HAVING AN ELASTIC-PLASTIC ADHESIVE AND LINEARLY ELASTIC ADHERENDS
A4EJ	ANALYSIS PROGRAM FOR MULTIROW BOLTED JOINTS
A4EK	ANALYSIS PROGRAM FOR COMBINED BONDED-BOLTED JOINTS
A4EP	PRESSURE PILLOWING ANALYSIS PROGRAM
BJSFM	BOLTED JOINT STRESS FIELD MODEL
BREPAIR	BOLTED REPAIR COMPUTER ANALYSIS PROGRAM
ALLOW	BUCKLING/CRIPPLING ANALYSIS OF THIN PLATES
BUCKHAT	HAT-STIFFENED PANEL BENDING ANALYSIS
BUCKJ	J-STIFFENED PANEL BENDING ANALYSIS
BUCKPLATE	BENDING/BUCKLING ANALYSIS OF ORTHOTROPIC PLATES
BUCKPSF	BENDING/BUCKLING ANALYSIS OF ANISOTROPIC LAMINATE
BUCKSINE	BUCKLING OF A SIMPLY SUPPORTED CORRUGATED ORTHOTROPIC PANEL
BUSHFIT	STRESSES IN INTERFERENCE-FIT CIRCULAR ASSEMBLIES
COMPOSITE	ANALYSIS OF BOLTED JOINTS
CURVEBEAM	ELASTICITY ANALYSIS OF A CURVED BEAM
DABEAMS	DYNAMIC ANALYSIS OF UNIFORM BEAMS
DCB	ANALYSIS OF A DOUBLE CANTILEVER BEAM
DIAGONAL	DIAGONAL TENSION ANALYSIS OF A SHEAR PANEL
EFFDEFF	ALLOWABLE STRESS ANALYSIS OF A DEFECTIVE LAMINATE
ELASTPROP	ELASTIC PROPERTIES OF AN ANISOTROPIC LAMINATE
ERTFLD	STIFFNESS BUCKLING ANALYSIS OF SHEAR WEB AND TENSION FIELD BEAM
FASTENER	LOAD ANALYSIS OF AN ARBITRARY FASTENER PATTERN
FINELEMNT	FINITE-ELEMENT ANALYSIS (TRIANGULAR ELEMENTS AND LINEAR INTERPOLATION FUNCTION)
FREQUENCY	FIRST TWO NODAL FREQUENCIES AND SHAPES OF A STRUCTURE (SODOLA'S POWER METHOD)
JNTLAP	ANALYSIS OF A DOUBLE LAPPED JOINT
JNTSCARF	ANALYSIS OF A SCARFED JOINT
MGNSAFE	MARGIN-OF-SAFETY ANALYSIS OF AN ANISOTROPIC LAMINATE
MINSTRUDL	ANALYSIS OF CONTINUOUS BEAM/FRAME FOR JOINT DISPLACEMENT AND FINAL MEMBER FORCES
PLASBEND	ELASTIC-ELASTOPLASTIC-PLASTIC BENDING ANALYSIS

TABLE 7-5

AVAILABLE COMPUTER PROGRAMS (CONTINUED)

PLATES	BENDING ANALYSIS OF LONG RECTANGULAR PLATES
SANDBUCK	BUCKLING OF NONSYMMETRICAL ORTHOTROPIC SANDWICH PANELS
SANDPRES	ANALYSIS OF A SANDWICH PANEL UNDER PRESSURE
SANDSIZE	SIZING OF NONSYMMETRICAL ORTHOTROPIC SANDWICH PANELS
SBEAM	CONTINUOUS BEAM - COLUMN ANALYSIS
SECTION	PROPERTIES OF AN IRREGULAR CROSS SECTION
SHEARLAG	SHEAR LAG BETWEEN TWO PARALLEL LOAD-CARRYING MEMBERS
STRENGTH	STRENGTH PROPERTIES OF AN ANISOTROPIC LAMINATE

Sustaining Engineering — Engineering support will be provided throughout this phase of the program.

Phase II — Structural Verification

Phase II will commence with the preparation of test plans for a technology demonstration program of large-scale fuselage structure. The following structures will be tested:

- A ground test article (GTA) composed of a 40.2-foot-long center fuselage barrel section for a durability, damage tolerance, and static strength test program.
- A cockpit enclosure section for bird strike and hail damage tests.
- A large generic fuselage shell structure for an impact dynamics test.
- Large panels for acoustics and lightning strike tests.

To reduce program risk, the test plans will also include structural element test specimens based on the detail design to verify the readiness for the large GTA test program.

These test plans will involve FAA participation since much of the test data will be used for structural substantiation of the large Phase III flight service panels to the FAR 25 requirements.

Test drawings will be prepared for the GTA based on the Phase I material system section and preliminary design, and the MD-100 design criteria and loads. The GTA design will be supported by a rigorous strength analysis and durability and damage tolerance assessment.

Engineering will prepare task assignment drawings which transmit engineering requirements to the Test Laboratory for the test configuration, instrumentation, test setup, test procedures, data acquisition, and reports of test results.

The test preparation and the test itself will be monitored and supported by engineering personnel. Test results will be reviewed and a final report prepared to document the findings. A separate report will be prepared to correlate the GTA strength analysis with measured strain and deflection gage data. The correlation will be used to improve the accuracy of the analysis methods.

A weight estimate will be prepared for the GTA and will be progressively updated with actual weights as parts are fabricated. The test article will be weighed and a weight report prepared that compares actual weights throughout the program with the initial weight estimates.

Engineering will support the preparation of tools and the fabrication and assembly of the large test specimens by revising drawings to facilitate manufacture when it is possible to do so without compromising program objectives. Manufacturing discrepancies will be reviewed and dispositioned according to standard engineering practices. Accept/reject criteria will be prepared to establish the minimum flaw sizes to be reported by inspectors. Significant manufacturing rework will become part of the test specimen definition for the repair data base. Similar engineering activities will be conducted in the other structural tests.

Phase III — Flight Service Evaluation

Detail drawings will be prepared for fabrication, assembly, and installation of the large panel to be placed on a DC-10 aircraft for a flight service evaluation. This panel requires FAA approval for compliance with the applicable FAR 25 requirements. A certification plan will be submitted to the FAA that outlines the proposed method of showing compliance by analysis supported by test evidence. The analysis methods and test data from Phase II will be used for the strength substantiation except for the presence of thermal strains due to the different coefficients of expansion between the composite panel and the aluminum shell. This difference will be accounted for by analysis supported by flight test strain gage data recorded at various temperatures. Although the specific panel with a cargo door will not have been tested in Phase II, the data and experience gained in Phase II should be sufficient to validate the analysis for FAA approval.

Engineering will prepare a weight report comparing estimated weights with actual weights as was done in Phase II.

The lightning strike protection must be substantiated for the structure to satisfy static strength and damage tolerance requirements. The electrical/avionics systems in the airplane will be shielded as required to maintain existing levels of system performance, and these will be confirmed by a flight test system checkout.

Documentation to substantiate airworthiness requirements for the reconfigured airplane must be prepared and submitted for FAA approval.

MATERIALS AND PROCESS PLAN

The Materials and Producibility Engineering department will support the Engineering design section during design development (Phase I), structural verification (Phase II), and flight service evaluation (Phase III). This support will include the selection of materials and assessment of the producibility of the design.

The material systems to be used for the fuselage structure will be selected at the time of the actual program. The materials selected will have proven handling and processing characteristics and acceptable mechanical, environmental, and impact toughness properties.

Design data specimens will be fabricated, conditioned, and tested as prescribed by Structural Engineering, using manufacturing techniques proposed for fabrication of the large fuselage structure (time, temperature, pressure, and methods).

Design Development Phase

Technology development by Materials and Producibility Engineering is recommended for nondestructive testing in Phase I.

Nondestructive Testing — Ultrasonic velocity variations and neutron gaging techniques appear as viable methods for quantitatively measuring resin content in carbon-epoxy composite structures. Panels containing variations in resin content will be fabricated, analyzed for resin content by nondestructive testing techniques, and checked for resin content by chemical digestion as a reference. The panels will be cut and tested for flexural strength and short beam shear strength to verify their mechanical quality. An analysis will be conducted to correlate the relationships for nondestructive testing to measure and establish the laminate resin content.

Ultrasonic attenuation appears to be a viable method of quantitatively measuring void content. Studies will be made on carbon-epoxy composite laminate specimens of typical thickness to determine the optimum ultrasonic test frequency, test methods (e.g., pulse-echo or through-transmission), and search-unit size. Various void content reference standards will be fabricated and tested to arrive at a relationship between void content and ultrasonic attenuation. All specimens will be mechanically tested to establish the relationship between void content and strength.

If carbon-epoxy composites are to be used on primary structure for commercial or military aircraft, nondestructive testing methods will be desirable to determine the degradation of the structure as related to strength and durability. Development of quick, low-cost, and reliable nondestructive testing techniques to determine a change in structural characteristics is being investigated in the industry and the latest technology available will be utilized to assess aging and environmental effects.

Structural Verification Phase

A materials specification will be prepared in Phase II to identify the basic material handling, physical, and material properties of composite laminate structure. The specification will document purchasing instructions, quality control test procedures for incoming material, and acceptance requirements, storage conditions, and requalification procedures for B-stage materials and cured laminates.

Process specifications will be prepared that will prescribe the materials and the detailed, step-by-step manufacturing process for the fuselage structure. The process specifications will include provisions for quality assurance and accept/reject requirements and procedures.

A nondestructive test specification will be prepared to prescribe the detailed nondestructive testing methods and acceptance criteria to be used for the fuselage structure.

Materials and Producibility Engineering will assist and support Manufacturing during fabrication of the Structural Verification specimens in Phase II. Their efforts will include surveillance of manufacturing operations, procedural techniques, quality cotnrol and inspection records, and participation in any rework that may be necessary.

Flight Service Evaluation Phase

The material and process specification developed during Phases I and II will be used to fabricate the Phase III flight service panel. Specifications and data will be prepared and submitted to the FAA for approval. Materials and Process Engineers will support the fabrication and inspection of the flight-worthy structure.

MANUFACTURING PLAN

The perceived manufacturing problems associated with producing a composite fuselage, as discussed in the technology assessment, are based upon the experience we have gained thus far. In any major program that extends technical capabilities, unanticipated problems arise during the development effort. A study program can only address the predictable problems and propose paths for their solution. An innovative program that extends the limits of existing technology requires more effort to support the advanced concepts. The manufacturing concepts presented herein for producing high-quality composite fuselages are also intended to achieve cost parity with an aluminum fuselage structure. Thus, the scope of the fuselage manufacturing development program is not limited to satisfying the immediate need for producing a prototype, but includes the resolution of low-cost manufacturing technology issues so that in the long term, production of composite fuselages will become a reality.

Phase I — Technology Development

Phase I tasks identify the technical problems, producibility risks, and overall requirements for manufacturing a composite fuselage. The Phase I development test specimens will be produced and a series of stiffened panels made using several manufacturing methods. Evaluation of the fabrication of the basic element of the fuselage assembly, a longitudinally and circumferentially stiffened panel with cutouts, will generate experience and foster reliability with a lower risk of loss than if a whole barrel section were fabricated. When integrated with the preliminary design, alternative manufacturing methods can be evaluated with reliable, realistic data.

Stiffened Panel Evaluation — Manufacturing feasibility can be evaluated by producing stiffened panels by three different methods (see Table 7-6). The evaluation will be based upon several criteria which will include tooling requirements, nondestructive inspection requirements, labor involved, repeatability (for quality acceptance), and the overall producibility risk for final evaluation. A 9- by 14-foot discretely stiffened panel will be made by secondary bonding, co-curing, and/or filament winding (see Figure 7-6). These panels, after inspection, will have bonded circumferential stiffeners (shear-tees) and door details mechanically attached. The shear-tees will be secondarily bonded to permit co-curing the continuous longerons.

TABLE 7-6
STIFFENED SKIN PANELS
MANUFACTURING PROCESS EVALUATION

PANEL FABRICATION METHODS	TOOLING	FABRICATION	MANUFACTURING CONCERNS
1. SECONDARY BONDING OF LONGERONS	REQUIRES SKIN TOOL, LONGER TOOLS, BONDING FIXTURE, AND FEMALE TOOLS.	SKIN AND LONGERONS SEPARATELY CURED, SECONDARILY BONDED.	LOW RISK MULTIPLE CURE CYCLES (3), FIT-UP OF LONGERONS TO SKIN DOUBLERS, PROVIDES EASY NDE WITH SEPARATE DETAILS, MOST LABOR REQUIRED.
2. CO-CURING LONGERONS/ SKIN	USES MALE GROOVED TOOL FOR CURING SKIN AND PRECURED LONGERONS. ELIMINATES BONDING FIXTURE.	LONGERONS PRECURED, INSPECTED PRIOR TO CO-CURING TO SKIN	GOOD FIT-UP OF LONGERONS TO SKIN, REDUCES CURE CYCLES (2), LONGERONS PREINSPECTED. NDE VERIFIES SKIN SOUNDNESS PLUS BOND AREA QUALITY MEDIUM RISK.
3. FILAMENT WINDING SKINS WITH PREPLACED LONGERONS	REQUIRES A WINDING MANDREL WHICH ACCOMMODATES LONGERONS; MAY REQUIRE EXPANSION AND COLLAPSIBILITY; MAY REQUIRE FEMALE CURING MOLD. BUILDUPS ON SKIN COMPOUNDS TOOLING DIFFICULTY.	FAST MATERIAL LAYDOWN POSSIBLE TO FAB SEVERAL SKINS/OPERATION, POTENTIAL FOR GROWTH, WET OR PREPREG WIND, LONGERONS PRECURED, OVERWIND SKIN.	AVAILABILITY OF WINDING EQUIPMENT MANDREL/MOLD DEVELOPMENT, RESIN/ VOID CONTROL, EXTERNAL SURFACE FINISH, DIMENSIONAL CONTROL, THIN SKIN HANDLING DAMAGE, HIGHEST RISK AT PRESENT, HIGHEST POTENTIAL FOR COST REDUCTION.

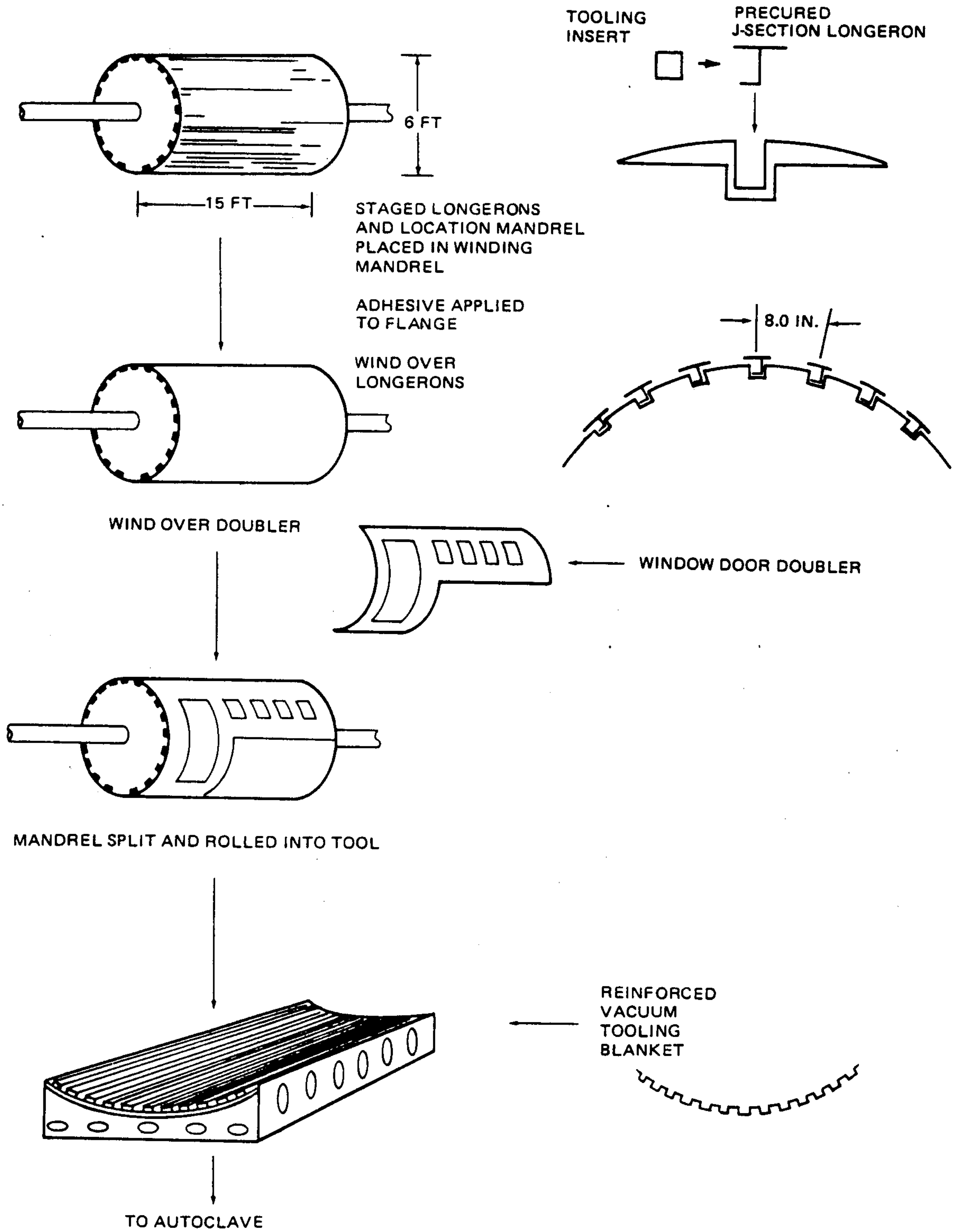


FIGURE 7-6. FILAMENT – WOUND PANELS

A further evaluation will be made in Phase II to secondarily bond the shear-tees in an advanced bonding tool which utilizes localized pneumatic pressure and heat in a computer-controlled bonding jig. This method of producing a stiffened fuselage panel has been demonstrated at Douglas.

Three methods of construction are to be evaluated:

1. Conventional Precured Details — Secondary Bonding Operation

   Skin with buildup and doubler, longerons, and shear-tees would be separately cured in an autoclave. The details would be subjected to nondestructive inspection and secondarily bonded either in a bonding jig or on a skin curing tool with a reinforced vacuum bag or other stiffener tooling aid (Figure 7-7). No significant advances in integral curing technology will be obtained by this approach, but it should have the lowest producibility risk.

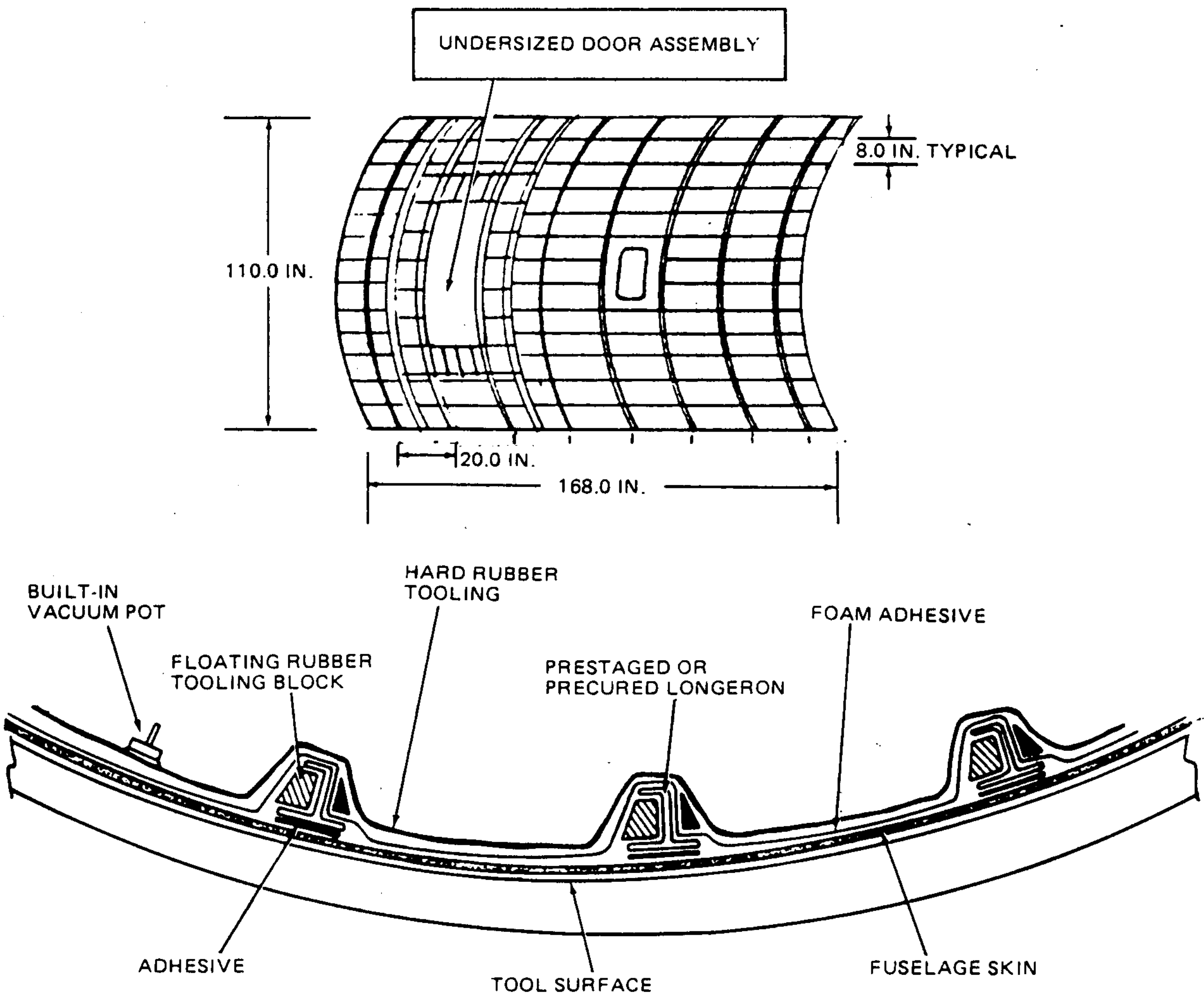


FIGURE 7-7. PHASE I – MANUFACTURING DEMONSTRATION PANEL

2. Co-Cured Longeron Stiffened Panel

   The fuselage skins with doublers would be a hand layup operation — in production, this would be done by a tape laying machine. The longerons would be precured on curing mandrels and then located on the skin with a picture frame or a reinforced vacuum bag tooling aid. The stiffened panel would then be trimmed and subjected to nondestructive inspection. An alternative method would be to B-stage the longerons (250°F cure for one hour) and trim and nondestructively inspect the longerons before final cure. The B-stage longerons would not be fully cured and therefore, when heated to 350°F in final cure, adequate resin flow would occur and the longerons could still conform around buildups on the skin.

3. Filament-Wound Stiffener Panels

   Precured or B-stage longerons would be placed in a mandrel. Adhesive would be applied to the back of the stiffeners and the skin wound over. Door and window doublers would be prekitted and densified, placed on a mandrel, and wound over to desired skin thickness. The mandrel would then be transferred to a curing tool and the wound skin split and rolled into the tool. The part would then be autoclave cured. Another option would be to use B-staged or uncured longerons and stitch the base of the longeron to the skin while still on the winding mandrel.

After construction, the longeron-stiffened panels would be transferred to a bonding tool (autoclave bonding) or a bonding jig (nonautoclave bonding) where the shear-tees would be secondarily bonded to the panel. After all bond lines are evaluated by nondestructive inspection, the panels would be located in an assembly tool where the passenger door jamb assembly would be mechanically attached to the skin panel. The panel assembly flow is shown in Figure 7-8.

Passenger Door Jamb — The cutout for the passenger door is structurally reinforced to transmit pressure and flight loads around the discontinuity. Two 6-inch-deep frames on either side of the cutout are separated by intercostals at 12-inch spacing. The two longerons above and below the cutout are built up into header beams. An inner skin attached to the frames and header beams completes a torque box to stabilize the shell at the edge of the cutout. Details of this baseline arrangement are described in Section 6.

Separate tools are required to lay up and cure each of the frames and header beams. The intercostals can be cured as one long channel and then sliced into individual parts. The joint structure can be mechanically attached or adhesively bonded to the skin and to other members. The preferred method will be established as a Phase I task.

The overwing frames attach directly to the skins along the sides of the fuselage, which results in the longerons being interrupted at each frame station in the region. The fabrication and assembly of these parts will be labor-intensive whether the longerons are secondarily bonded or mechanically attached. The preferred method will be selected during Phase I unless recommended design studies can establish that the longerons are not required along the sides where the longitudinal loads are low.

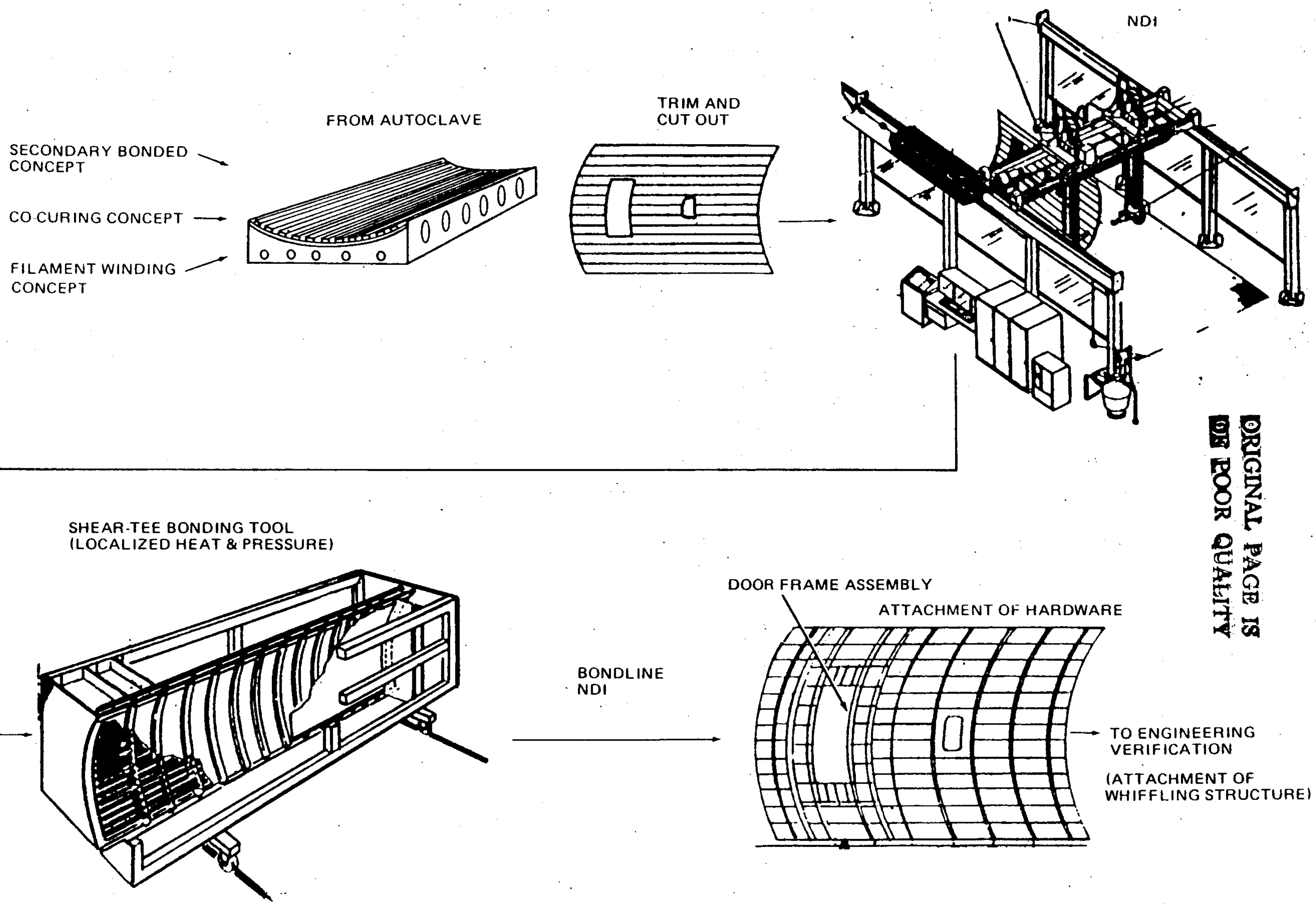


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FIGURE 7-8. PHASE I – EVALUATION CONCEPTUAL FLOW

Phase II — Ground Test Article Assembly

The ground test article (GTA) is a very complex fuselage structure because of the wing/fuselage joints and the landing gear wheel well. The GTA is 40.2 feet long and has a constant diameter of 19.2 feet except for that portion over the wing and main landing gear area and below longeron L-27, the passenger floor level. This area flares out starting near the wing front spar and returns to a true diameter at frame station 1521. The lower side skin, bounded by station 1281, L-27, and station 1531, L-34½, is a separate detail. The GTA has a manufacturing break at each end so that the center barrel is 33.5 feet long. This approach includes the technology of joining large composite fuselage barrel sections together as established by the design.

The recommended assembly breakdown is shown in Figure 7-9 and the sequence of the flow in Figure 7-10. The center barrel section is broken down into four subassemblies; (1) floor and pressure panel assembly, (2) main gear wheel well, (3) top panel assembly, and (4) side/window panel assembly.

The floor beams over the main gear wheel well are recommended for press curing fabrication. They have a depth of 17 inches across the center section and taper to a depth of 6 inches at the ends. These are integrated and mechanically fastened with all other lateral and longitudinal beams (stations 1129 to 1531) and pressure barrier panels (intermediate, center, and outboard) above the main gear wheel well in a large assembly jig.

The main gear wheel well assembly is further broken down into a keel beam assembly, torque box assembly, bottom and side panel assembly (L-34½ to L-44½), and pressure bulkhead assembly, and then assembled in a large jig. Vertical picture frame fixtures are used to assemble the bulkhead keel beam and torque box. This is then tied onto the bottom panel in the subassembly jig. The basic construction of the bulkhead consists of a co-cured hat-stiffened panel with vertical J-beams secondarily bonded to the web.

The recommended assembly of these subassemblies would be in a large, multiple-station assembly jig (Figure 7-11), where the main gear wheel well assembly is positioned first and then located by the landing gear door actuator points. The passenger floor is then located in the large assembly jig and mechanically fastened to the wheel well assembly.

The side panels are located by external contour tooling fittings integrated with longeron index fittings. The top one-third panel is then brought to the assembly fixture by overhead cranes and located by longeron indexing fittings, and intermediate frame station location points.

The stiffened side skin panels (interrupted longeron design) will be positioned in the assembly jig where the door frame assembly will be attached. The one-third top panel will be transferred from a vertical picture frame structure to an overhead transfer frame structure to be lowered for assembly.

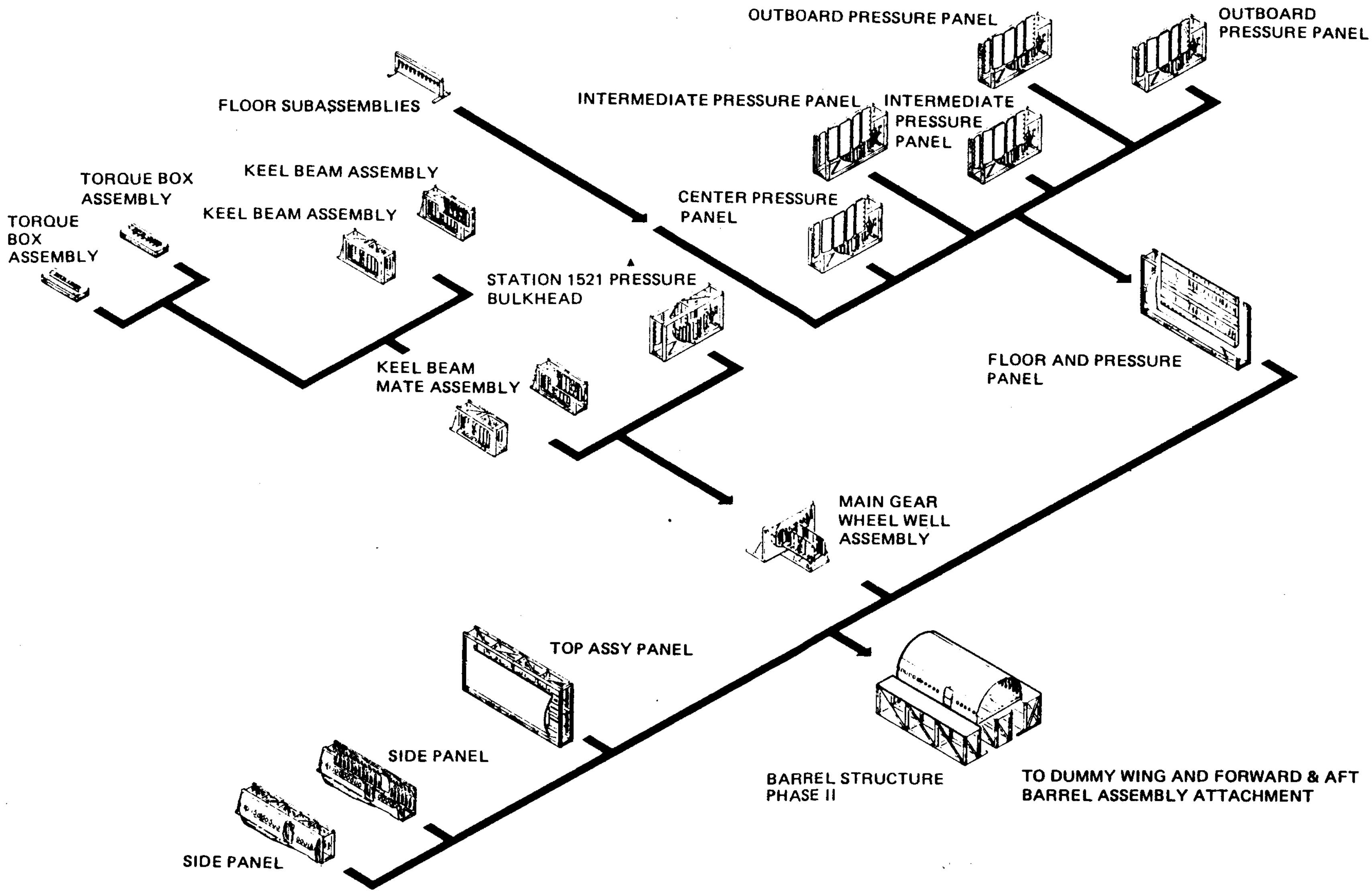


FIGURE 7-9. PHASE II - GROUND TEST ARTICLE ASSEMBLY FLOW

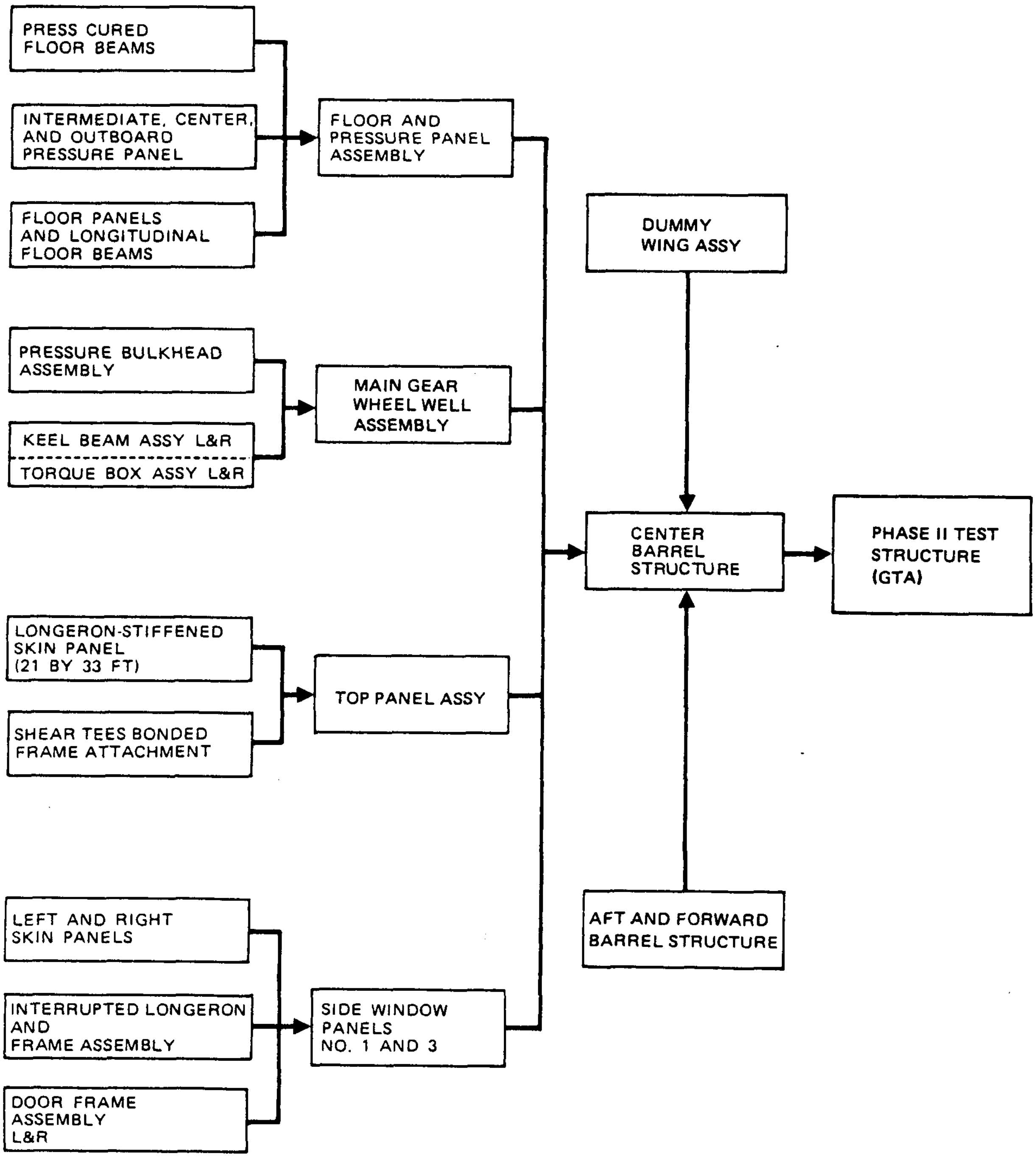


FIGURE 7-10. CENTER-BARREL ASSEMBLY - FLOW

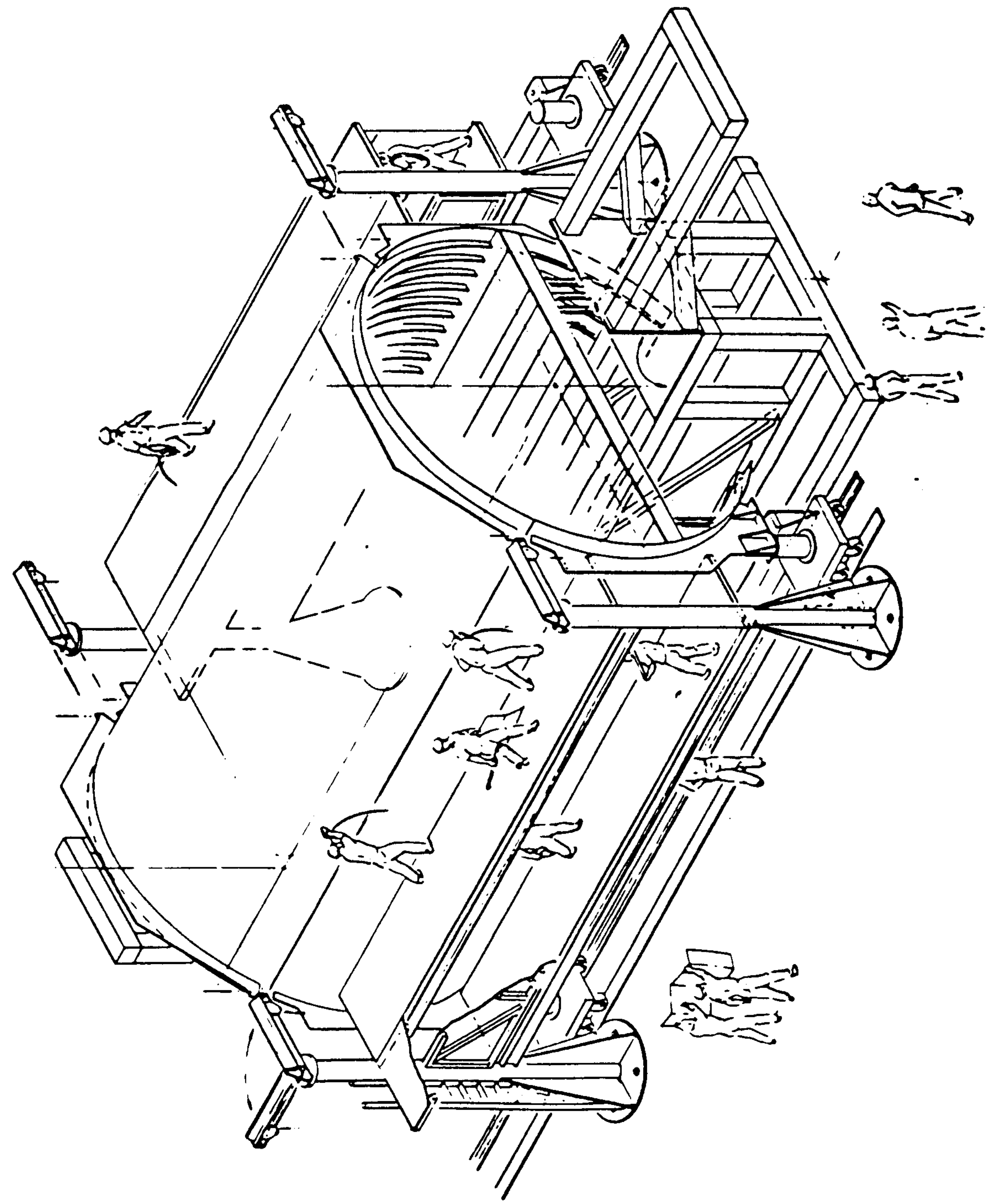


FIGURE 7-11. MANUFACTURING DEVELOPMENT CENTER-BARREL ASSEMBLY JIG

The short barrel sections forward and aft of the center barrel are constant-diameter with the standard skin, longeron, shear-tee, frame, and window belt structural arrangement. The alignment of the structural elements with corresponding elements in the center barrel sections must be carefully controlled to minimize shimming, rework, and the manufacture of special details to facilitate the joining operation. The end portions of these barrels will be reinforced for mechanical attachment to the large steel load introduction barrels. This latter operation does not involve the production process and should not require the development of special technology.

Phase III — Flight Evaluation Panel

Manufacturing will fabricate and assemble the flightworthy fuselage panel for the flight service evaluation program in accordance with FAR 21 quality system requirements. Fabrication tools used to manufacture the Phase II GTA will be available, but the cargo door and cargo door jamb structure were not included in the GTA and new tooling will be required for this structure. Methods of construction and assembly are expected to duplicate those of Phase II, and new technology development is not required.

Manufacturing will rework the aircraft to remove the aluminum panel and install the composite panel in its place. The composite panel will be located and drilled to match existing fastener holes on the aircraft and will be trimmed to fit with adjoining panel edges. Actual drill and trim operations will be performed away from the aircraft to avoid deposits of corrosive carbon fiber dust particles on the aluminum structure. Hydraulic, electrical, and other aircraft systems that must be partially dismantled to complete the installation will be restored in accordance with production procedures.

The Planning department will prepare cost-effective, one-of-a-kind fabrication and assembly outlines to produce the tooling and to manufacture and install the flightworthy panel. Cost data will be tracked for input into the cost data base.

TEST PLAN

The overall test program and task relationships from design requirements for tests through FAA flight certification are shown in Figure 7-12.

Some technical development for test purposes is anticipated for this program as a result of the use of composite materials and composite design and production processes. The very large full-scale fuselage barrel section, if tested during the warm summer months, could result in temperatures above 130°F and high relative humidities (90 percent) inside the fuselage section during the extended simulated ground-air-ground cycling using compressed ambient air. The flight test and flight service evaluation phase with a large composite panel mechanically attached to a portion of a predominately aluminum fuselage could result in relatively high thermally induced compressive stresses in the longitudinal direction plus induced stresses in the transverse direction during flight at cruise altitudes. It is anticipated that some of the specimens, test articles, and transition areas between the carbon-epoxy and metal test structure will require special consideration and analytical treatment.

Material allowables tests and design verification tests will be performed to demonstrate compliance with applicable requirements of Federal Aviation Regulations, Part 25, and the current FAA Advisory Cir-

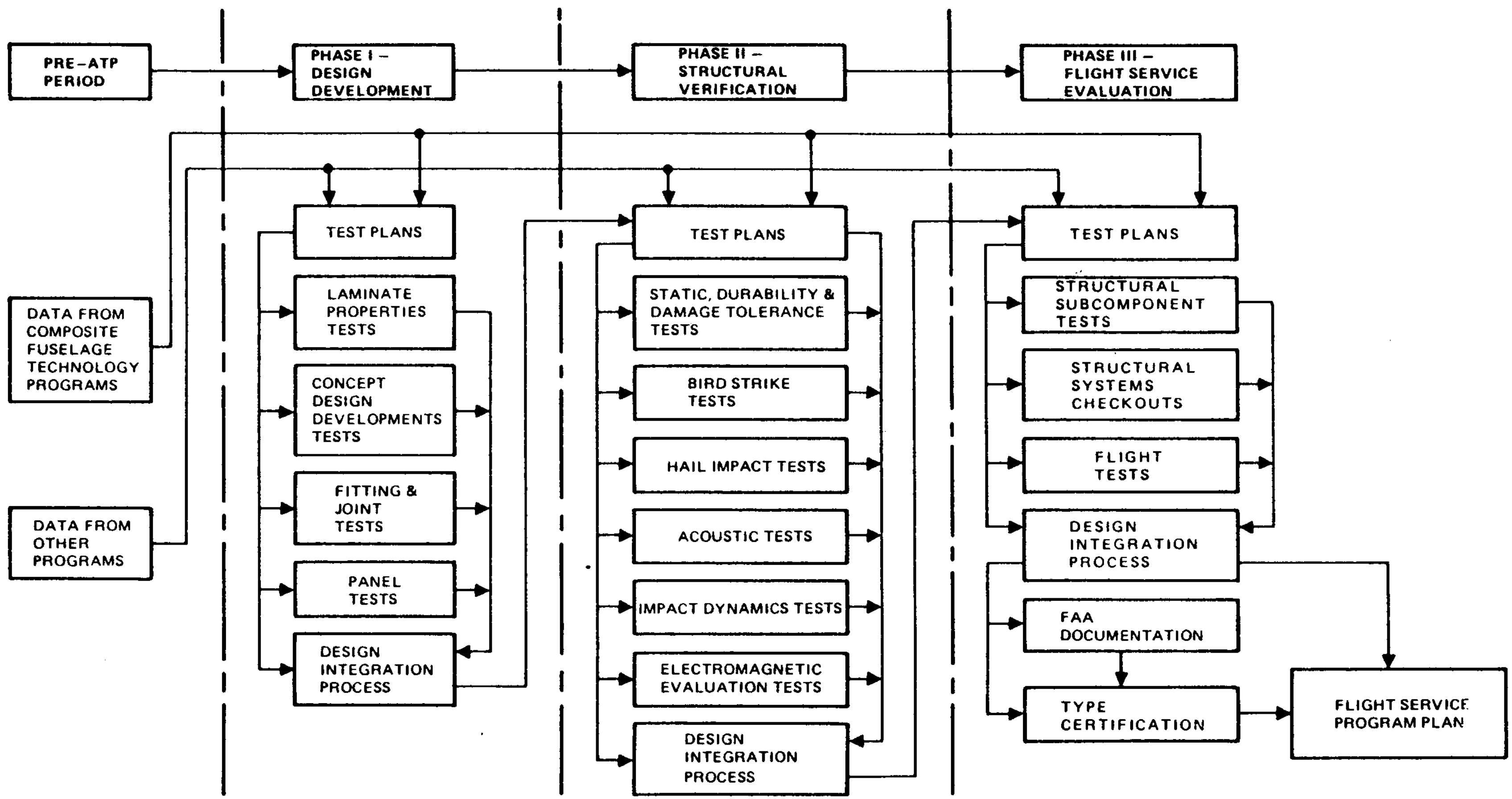


FIGURE 7-12. TEST FLOW DIAGRAM

cular on Composite Aircraft Structure. The FAA will approve the test articles and test setup for design conformity, approve the test plans including load conditions, witness the test, and approve the final test report.

Final reports will be prepared of the test results and, as appropriate, of their correlation with the predicted values.

Design Development Tests

A design development program will be conducted in Phase I to determine composite material properties and structural component performance that cannot be found in published documents or other approved sources and to develop design concepts that will meet strength, damage tolerance, durability, and electromagnetic effects criteria.

The development test program will be determined on the basis of available data for composite material from in-house composite programs, industry sources, and government agencies. A preliminary structural development test program is presented in the following paragraphs.

Structural Design Development Tests — A full spectrum of tests will be developed to substantiate the selected design concepts and demonstrate the required degree of technological readiness and integration. The first series of tests will be conducted to provide a basis for material selection. These tests will include those outlined in Reference 20. Two candidate materials will be compared with each other as well as with data published in the literature. Basic material properties, stress concentration behavior at hole boundaries, interlaminar fracture toughness, compression, and delamination behavior will be evaluated. A material system will be selected based on the results of these tests. A material selection test plan is outlined in Table 7-7.

Once a structural material system has been selected, structural configuration testing will begin. The tests will include electromagnetic interference/lightning protection system evaluation. These tests will be conducted on basic panels to establish a baseline for evaluating the effectiveness of candidate protective systems. Panels will be fabricated with several alternative methods of protection such as widely spaced fine wires and metallized carbon fibers woven into fabric. Tests will evaluate the strength of each candidate material to determine the load-carrying capability of metallized carbon fibers and the lightning strike behavior of the material in the Douglas lightning test facility. An electromagnetic interference/lightning protection system will then be selected based on the results of these tests.

A series of tests will be conducted to determine the best substructural concept. Two candidate structural configurations of each type will be tested. These will include shear tee pull-off tests, longitudinal and transverse skin splices, and both longeron and frame splices. The basic structural configurations will be selected as a result of these tests. This series of tests is outlined in Table 7-8. Further structural configuration tests will include loaded hole tests to obtain k_{tc} values for bolted joint analysis.

Critical structural elements of the composite fuselage are to be selected for design development testing. Typical test specimens and conditions for concept evaluation of fuselage skin panels, joints and splices,

TABLE 7-7
LAMINATE PROPERTIES DESIGN DEVELOPMENT TESTS

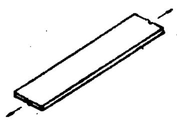
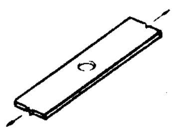
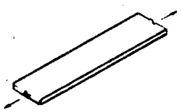
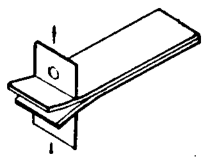
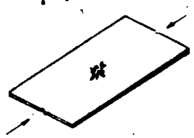
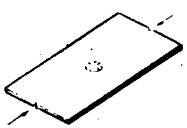
	SPECIMEN SIZE	PURPOSE	LOAD	TEST TYPE	PRETEST CONDITIONING TEMP	PRETEST CONDITIONING RH	NO. OF SPECIMENS TEST TEMP −65°	NO. OF SPECIMENS TEST TEMP RT	NO. OF SPECIMENS TEST TEMP 180°	NO. OF SPECIMENS TOTAL	REMARKS
	2 by 4 (Tension) 3 by 9 (Compression)	Basic Material Prop	Tension and Compression	Material A	180 AMB	95 AMB	6 6	6 6	6 6		
				Material B	180 AMB	95 AMB	6 6	6 6	6 6	72	
	2 by 12 (Tension) 5 by 12.5 (Compression)	Open Hole Behavior Tension & Compression	Tension and Compression	Material A ST-3 & Typ Fus Struc	180 AMB	95 AMB	12 12	12 12	12 12		NASA Standard Tests ST-3 (Tension) ST-4 (Compression)
				Material B	180	95	12	12	12	108	Plus tests based on typical fuselage sizes
	1.5 by 10	Edge Delamination	Tension	Material A	180 AMB	95 AMB	3 3	3 3	3 3		NASA Standard Test ST-2 Plus tests based on typical fuselage sizes
				Material B	180 AMB	95 AMB	3 3	3 3	3 3	36	
	1.5 x 9	Hinged DCB	Tension	Material A	180 AMB	95 AMB	3 3	3 3	3 3		NASA Standard Test ST-5 Plus tests based on typical fuselage sizes
				Material B	180 AMB	95 AMB	3 3	3 3	3 3	36	
	7 by 12.5	Compression After Impact	Compression	Material A ST-1 & Typ Fus Struc	180 AMB	95 AMB	6 6	6 6	6 6		NASA Standard Test ST-1 Plus tests based on typical fuselage sizes
				Material B ST-1 & Typ Fus Struc	180 AMB	95 AMB	6 6	6 6	6 6	72	
	3 by 9	Bonded Delaminations	Compression	Material A	180 AMB	95 AMB	3 3	3 3	3 3		
				Material B	180 AMB	95 AMB	3 3	3 3	3 3	36	

TABLE 7-8
CONCEPT DESIGN DEVELOPMENT TESTS

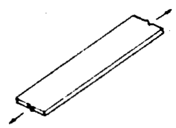
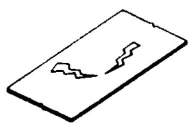
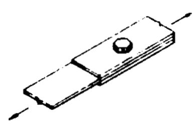
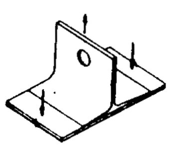
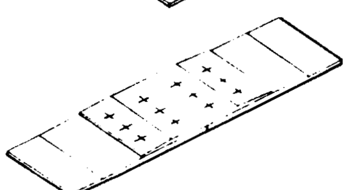
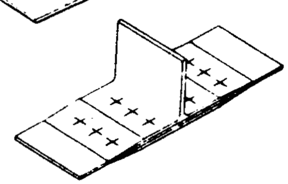
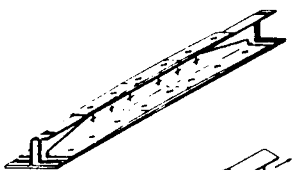
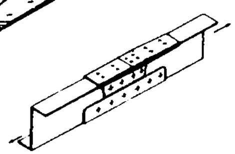
	SPECIMEN SIZE	PURPOSE	LOAD	TEST TYPE	PRETEST CONDITIONING TEMP	PRETEST CONDITIONING RH	NO. OF SPECIMENS TEST TEMP −65°	TEST TEMP RT	TEST TEMP 180°	TOTAL	REMARKS
	2 by 4	EMI/Lightning Protec System effects on strength	Tension	Material A	180	95	3	3	3		
					AMB	AMB	3	3	3		
				Material B	180	95	3	3	3		
					AMB	AMB	3	3	3	36	
	3 by 9	Lightning Strike		Material A	AMB	AMB		3			
				Material B	AMB	AMB		3		6	
	2 by 4	Loaded Hole	Tension	Conf 1 (1/8-in. hole)	180	95	3	3	3		
					AMB	AMB	3	3	3		
				Conf 2 (1/4-in. hole)	180	95	3	3	3		
					AMB	AMB	3	3	3		
				Conf 3 (1/2-in. hole)	180	95	3	3	3		
					AMB	AMB	3	3	3	54	
	6 by 3	Shear Tee Pull-Off	Tension	Conf 1	180	95	3	3	3		
					AMB	AMB	3	3	3		
				Conf 2	180	95	3	3	3		
					AMB	AMB	3	3	3	36	
	12 by 4	Longitudinal Skin Splice	Tension and Fatigue	Conf 1	180	95	6	6	6		
					AMB	AMB		12			
				Conf 2	180	95	6	6	6		
					AMB	AMB		12		60	
	12 by 4	Transverse Skin Splice	Tension, Compression, and Fatigue	Conf 1	180	95	9	9	9		
					AMB	AMB		18			
				Conf 2	180	95	9	9	9		
					AMB	AMB		18		90	

TABLE 7-8
CONCEPT DESIGN DEVELOPMENT TESTS
(CONTINUED)





				PRETEST CONDITIONING		NO. OF SPECIMENS TEST TEMP				
SPECIMEN SIZE	PURPOSE	LOAD	TEST TYPE	TEMP	RH	−65°	RT	180°	TOTAL	REMARKS
8 by 3	Longeron Splice	Tension and Compression	Conf 1	180	95	3	3	3		
				AMB	AMB	3	3	3		
			Conf 2	180	95	3	3	3		
				AMB	AMB	3	3	3	36	
8 by 3	Frame Splice	Tension	Conf 1	180	95	3	3	3		
				AMB	AMB	3	3	3		
			Conf 2	180	95	3	3	3		
				AMB	AMB	3	3	3	36	

cutouts, and fittings are shown in Tables 7-9 through 7-11 and Figures 7-13 through 7-16. A summary of development panel tests is shown in Figure 7-17.

The shear panel test fixture is shown in Figure 7-18. A typical test setup is presented in Figure 7-19, and a schematic of a typical test setup with instrumentation is shown in Figure 7-20.

TABLE 7-9

SKIN PANEL CONCEPT DESIGN DEVELOPMENT TESTS

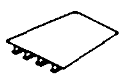
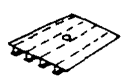
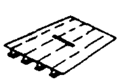
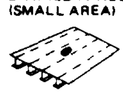
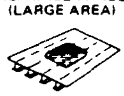
TEST NO.	TEST SPECIMENS	SPEC SIZE (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP (°F) (−65)	AMB	(180)	TOTAL
1	BASIC PANELS	48 x 24	TENSION STRENGTH AND STIFFNESS	LONGITUDINAL TENSION	6	(180) AMB	95 AMB	2	7	2	11
2		48 x 90	COMPRESSION STRENGTH AND STIFFNESS	LONGITUDINAL COMPRESSION	6	(180) AMB	95 AMB	2	7	2	11
3			SHEAR STRENGTH AND STIFFNESS	IN-PLANE SHEAR	6	(180) AMB	95 AMB	2	7	2	11
4		48 x 60	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	3	AMB	AMB		9		9
5			STRENGTH UNDER COMBINED LOADING	COMPRESSION AND SHEAR	3	AMB	AMB		9		9
6		14 x 24	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	3	AMB	AMB		3		3
7	WINDOW BELT	48 x 90	TENSILE STRENGTH	LONGITUDINAL TENSION	2	AMB	AMB		2		2
8			COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	AMB	AMB		2		2
9			SHEAR STRENGTH	IN-PLANE SHEAR	2	AMB	AMB		2		2
10			COMBINED COMPRESSION AND SHEAR	LONGITUDINAL COMPRESSION AND IN-PLANE SHEAR	2	AMB	AMB		6		6
11			FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
12	DAMAGED PANEL (SMALL AREA)	14 x 24	POSTDAMAGE TENSION STRENGTH	LONGITUDINAL TENSION	2	(180) AMB	95 AMB	2	3	2	7
13		48 x 90	POSTDAMAGE COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	(180) AMB	95 AMB	2	3	2	7
14		14 x 24	POSTDAMAGE FATIGUE	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
15	DAMAGED PANEL (LARGE AREA)	14 x 24	POSTDAMAGE TENSION STRENGTH	LONGITUDINAL TENSION	2	(180) AMB	95 AMB	2	3	2	7
16		48 x 90	POSTDAMAGE COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	1	(180) AMB	95 AMB	2	3	2	7
17		14 x 24	POSTDAMAGE FATIGUE	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
18	REPAIRED PANEL (SMALL AREA)	14 x 24	TENSILE STRENGTH	LONGITUDINAL TENSION	2	(180) AMB	95 AMB	2	3	2	7
19		48 x 90	COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	(180) AMB	95 AMB	2	3	2	7
20		14 x 24	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
21	REPAIRED PANEL (LARGE AREA)	14 x 24	TENSILE STRENGTH	LONGITUDINAL TENSION	2	(180) AMB	95 AMB	2	3	2	7
22		48 x 90	COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	(180) AMB	95 AMB	2	3	2	7
23		14 x 24	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2

TABLE 7-10

FITTING CONCEPT DESIGN DEVELOPMENT TEST

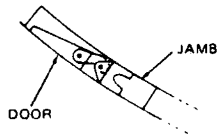
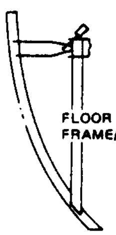
TEST NO.	TEST SPECIMENS	SPEC. SIZE (IN by IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING		NO. OF SPECIMENS			
						TEMP -F	% RH	TEST TEMP (°F) -65	AMB	180	TOTAL
1	TRAPEZOIDAL PANEL INSTL 	(24 by 40)	STATIC STRENGTH	MAX COMBINED LOADS	2	AMB	AMB		2		2
2	CARGO DOOR LATCH 	5 by 30	STATIC	MAX COMBINED LOADS	2	AMB	AMB		2		2

TABLE 7-11

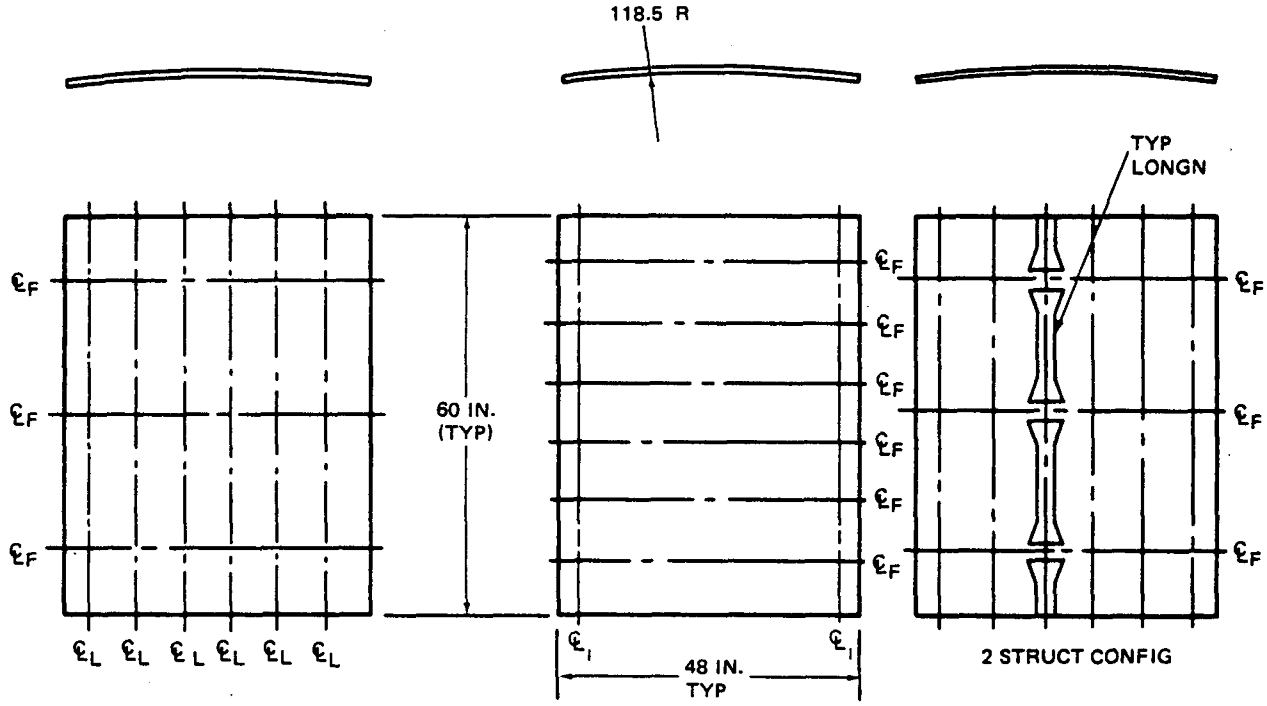
JOINT CONCEPT DESIGN DEVELOPMENT TEST

TEST NO.	TEST SPECIMENS	SPEC SIZE (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP (°F)	PERCENT RH	NO. OF SPECIMENS TEST TEMP (°F) -65	AMB	180	TOTAL
1	FUS-TO-WING JOINT 	12 x 24	SHEAR STRENGTH	STATIC SHEAR	3	180 AMB	95 AMB	2	4	2	8
2		10 x 30	SHEAR STRENGTH AXIAL STRENGTH BENDING	STATIC SHEAR	6	180 AMB	95 AMB	2	7	2	11
				FATIGUE	2						2

The large pressure panel test fixture and typical pressure panel setup are shown in Figures 7-21 through 7-23.

The 897 specimens illustrated with 15 different types of design detail sections are considered representative of a concept design development program for an MD-100 composite fuselage structure. More than one configuration, as noted in Tables 7-9 through 7-11 and Figures 7-13 through 7-15, would be tested for a given detail section. Typical differences in configuration might be found in structural shape, in variations in lightning protection, and in combinations of tape and broadgoods.

The results of these design development tests and previous composite fuselage programs will be utilized in the detail design of the full-scale composite fuselage. Additionally, these data will provide the necessary data base and confidence for proceeding into the Phase II tests.



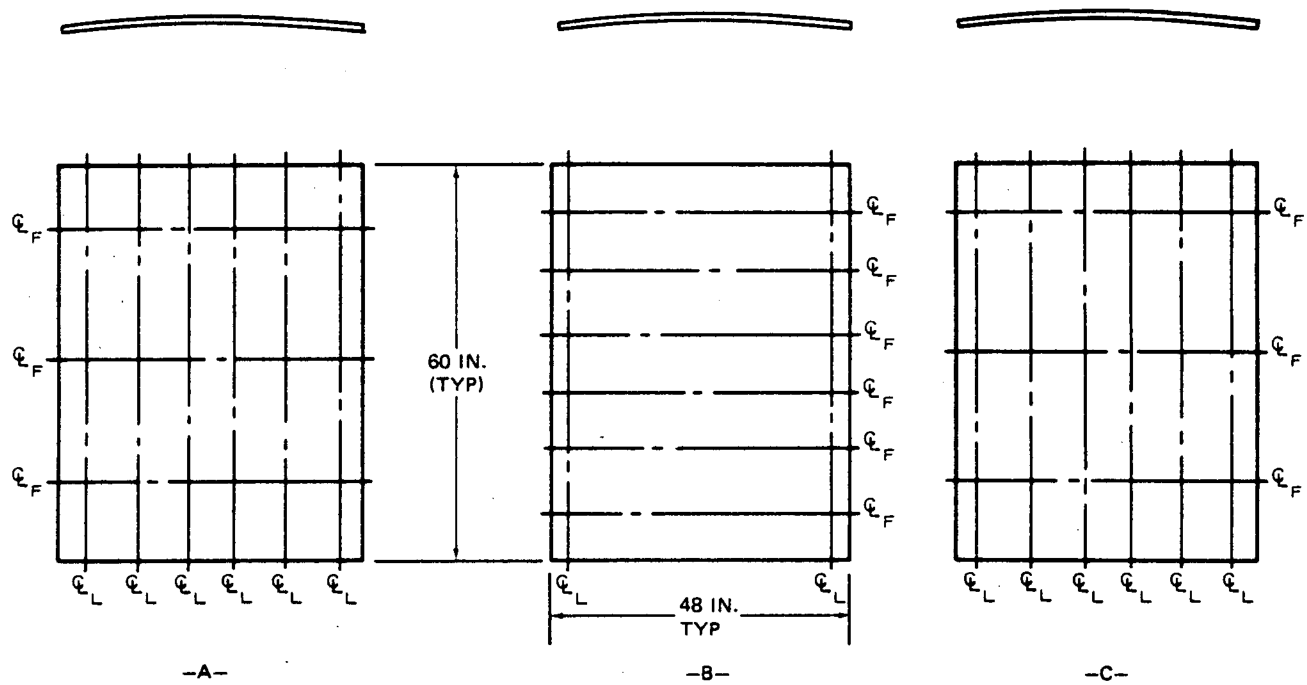
	–A–	–B–	–C–
CONFIGURATION:	TYP SKIN PANEL S&C, S&T WITH AND WITHOUT LIGHT PROT. 1 AND MULTILAYERS ADH 2 STRUCT CONF	FWD FUS SHEAR ONLY WITH AND WITHOUT LIGHT PROT. 1 AND MULTILAYERS ADH	ABOVE WING SHEAR ONLY WITHOUT LIGHT PROT. MULTILAYERS ADH
TEST TEMPERATURE:	ROOM TEMPERATURE		
CARBON-EPOXY:	SELECTED CARBON-EPOXY		
ADHESIVE:	SELECTED ADHESIVE		
SKIN:	A (THIN AND THICK)	B ONE t(NOMINAL)	C ONE t(NOMINAL)
TEST:	COMP, SHEAR, S+C, S+T		

QUANTITY:	8	4	4	TOTAL = 16

REPLICATES: ONE

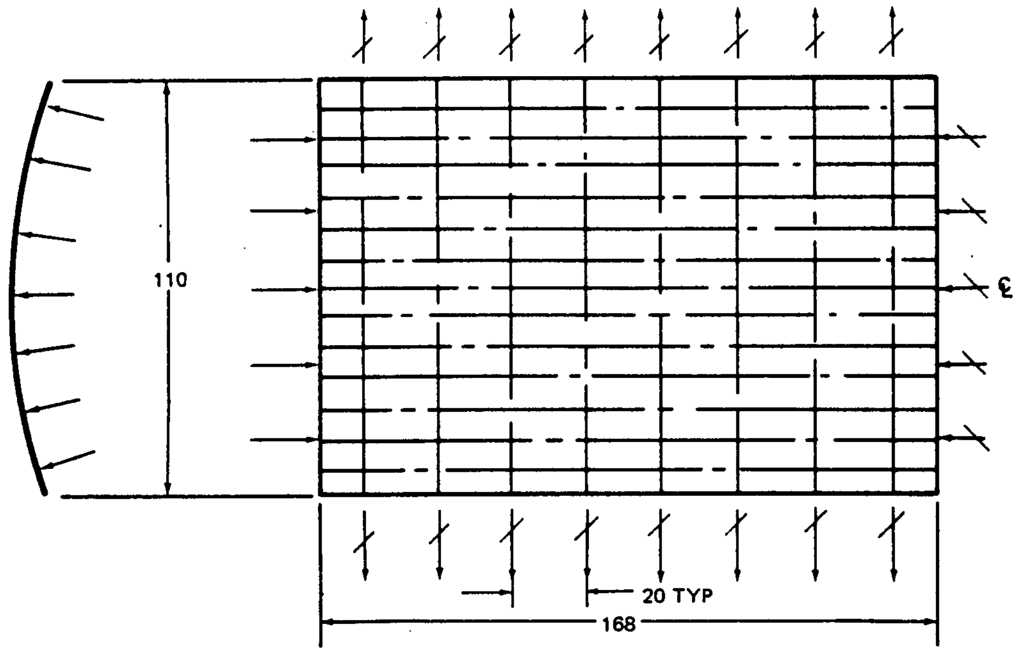
PURPOSE: DETERMINE COMPRESSION, SHEAR, COMBINED SHEAR AND COMPRESSION, AND SHEAR AND TENSION INTERACTIONS ALLOWABLE FOR CONFIGURATION A PANELS. MORE THAN ONE INTERACTION DATA POINT MAY BE OBTAINED WITH EACH PANEL. DETERMINE SHEAR FOR B AND C PANELS.

FIGURE 7-13. SHEAR AND SHEAR INTERACTION TEST PANELS



CONFIGURATION:	BEST OF STATIC SPECIMENS		
TEST ENVIRONMENT:	PHASED FATIGUE SPECTRUM OF SHEAR, COMPRESSION,AND CABIN PRESSURE LOADING, AMBIENT ENVIRONMENT		
CARBON-EPOXY:	BEST OF CARBON-EPOXY SYSTEMS		
ADHESIVE:	BEST OF ADHESIVE SYSTEMS TESTED		
SKIN:	THIN AND THICK	NOMINAL	NOMINAL
TEST:	SHEAR, COMPRESSION, AND CABIN PRESSURE		
QUANTITY:	FOUR SPECIMENS		
PURPOSE:	DETERMINE THE ACCUMULATED DAMAGE OF DURABILITY GROWTH (TBD FLAWS) AND ADHESIVE JOINT DURABILITY UNDER THE COMBINED EFFECTS OF SKIN WRINKLING, AND HIGH TEMPERATURE AND HUMIDITY		

FIGURE 7-14. DURABILITY TESTS OF PANELS



CONFIGURATION	THREE (A, B, AND C)
TEST TEMPERATURE:	ROOM TEMPERATURE
CARBON EXPOXY:	BEST CARBON-EPOXY SYSTEM
ADHESIVE:	BEST ADHESIVE SYSTEM
SKIN:	NOMINAL THICKNESS
TEST:	COMPRESSION AND PRESSURE

FIGURE 7-15. COMPRESSION AND PRESSURE TESTS

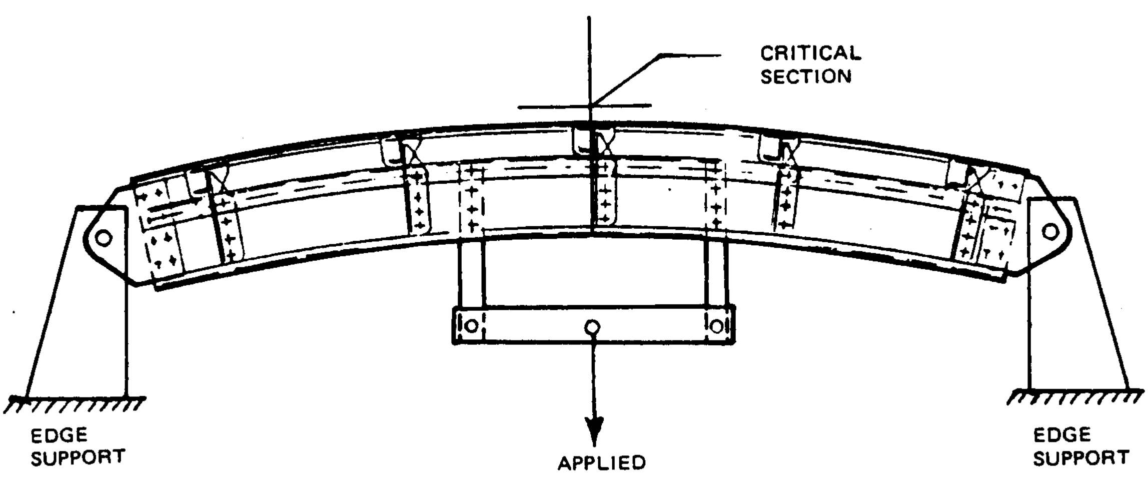
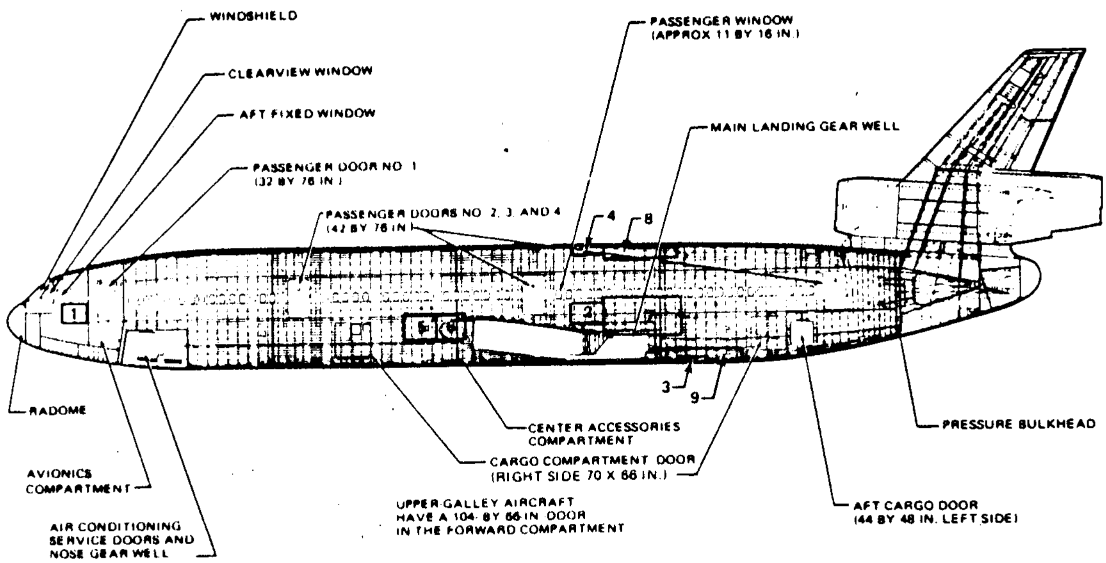


FIGURE 7-16. FUSELAGE FRAME BENDING TEST SCHEMATIC



SPECIMEN IDENTIFICATION		1 CURVED PANEL MEDIUM	2 CURVED PANEL MEDIUM	3 CURVED PANEL MEDIUM	4 CURVED PANEL MEDIUM	5 CURVED PANEL MEDIUM	6 FRAME BENDING	7 CURVED PANEL-LARGE	8 CURVED PANEL-LARGE	9 CURVED PANEL-LARGE
SPECIMEN SIZE (IN.)		48 by 61.5	48 by 61.5	48 by 61.5	48 by 61.5	48 by 61.5	36 by 44	110 by 168	110 by 168	110 by 168
LOADING		S, S+P	S, S+P	S & C, S, C+P	S & T, S, T+P	S, S+P	B	C & P	T & P	C & P
TESTS	STATIC	2	2	2	2	2	1	1	1	1
	FATIGUE	1	1	1	1	1		1	1	1
	DAMAGE TOLERANCE	1	1	1	1	1		1	1	1
	CRITICAL MODE (TBD)	1	1	1	1	1				
TOTAL SPECIMENS		4	4	4	4	4	1	1	1	1

NOTE: T= TENSION, C= COMPRESSION, S=SHEAR, B= BENDING, P= PRESSURE

FIGURE 7-17. SUMMARY OF DEVELOPMENT PANEL TESTS

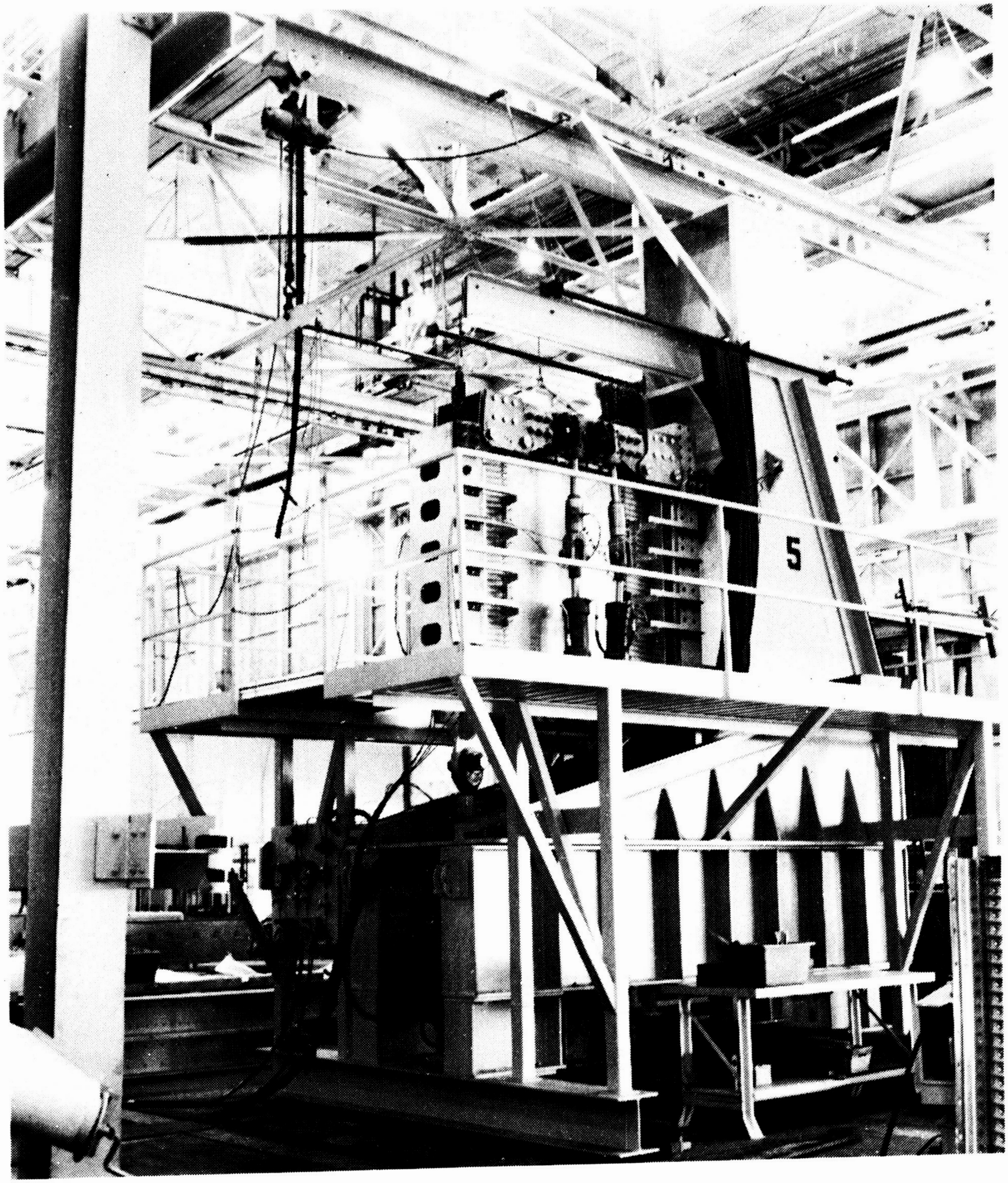


FIGURE 7-18. SHEAR PANEL TEST FIXTURE

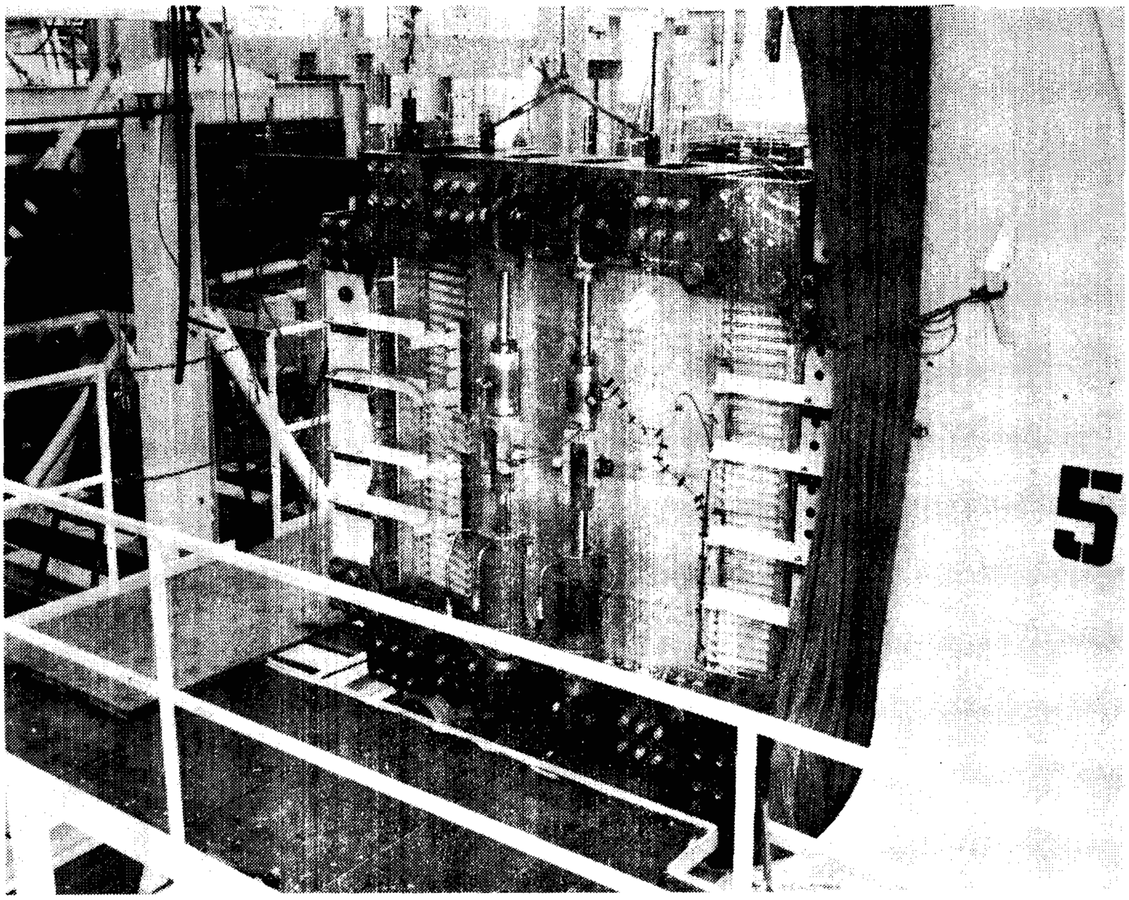


FIGURE 7-19. TYPICAL TEST SETUP FOR SHEAR PLUS AXIALLY LOADED PANELS

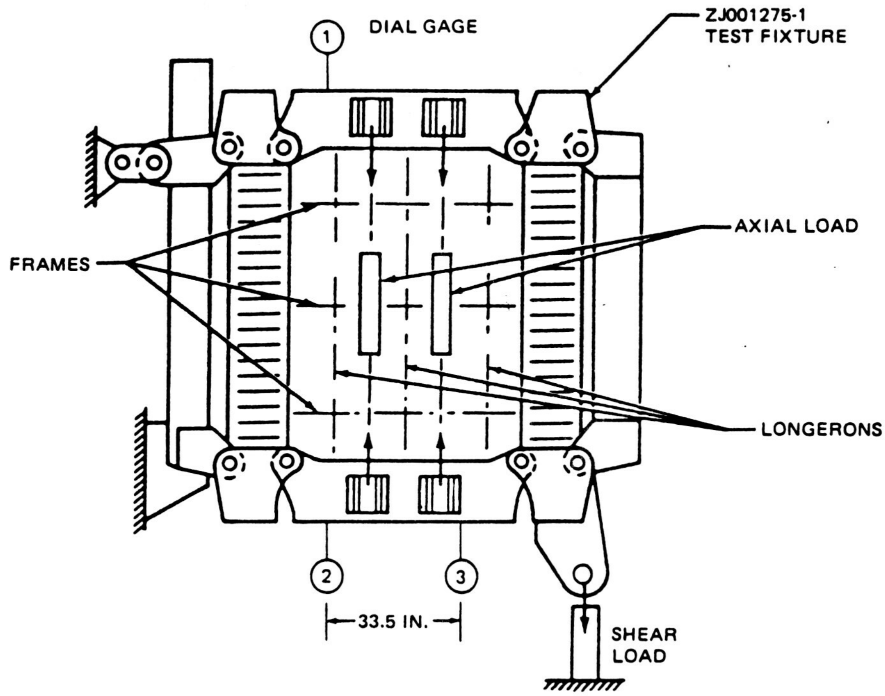


FIGURE 7-20. SCHEMATIC OF TYPICAL TEST SETUP WITH INSTRUMENTATION

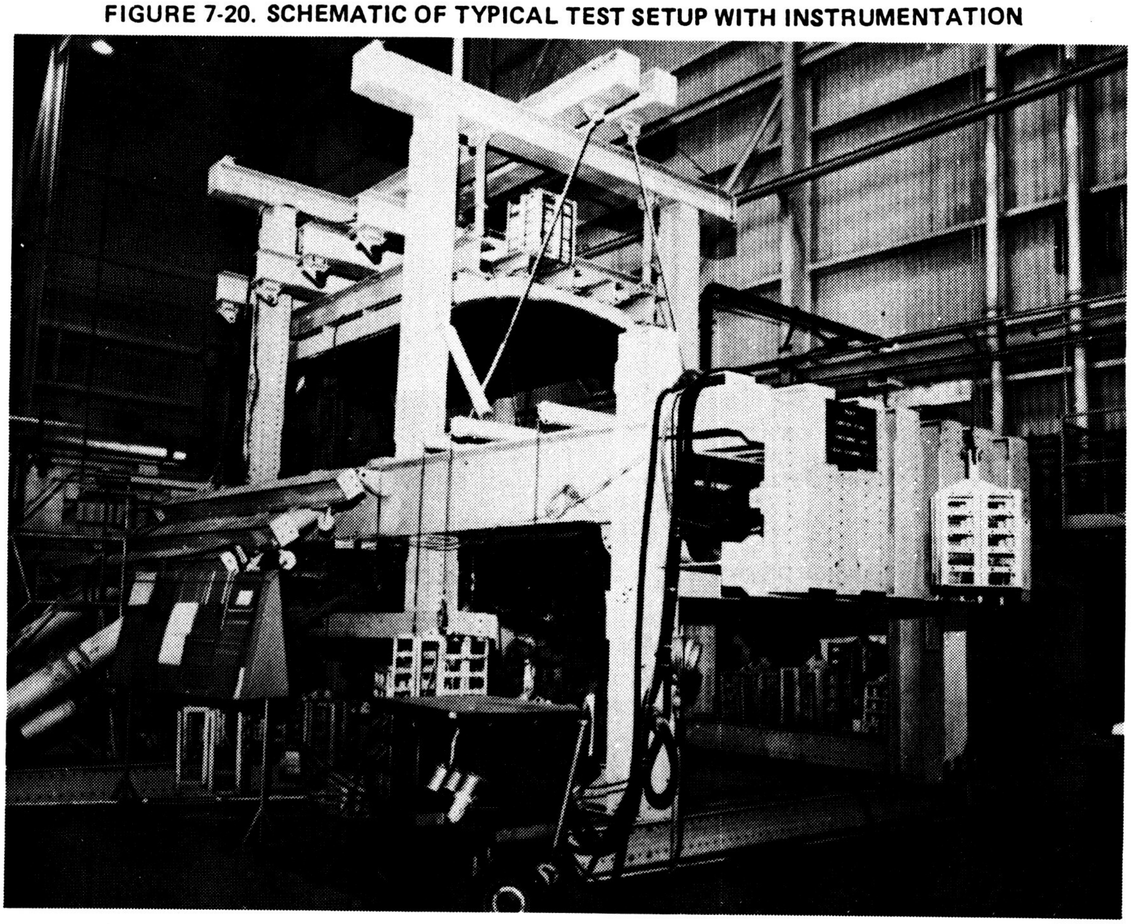
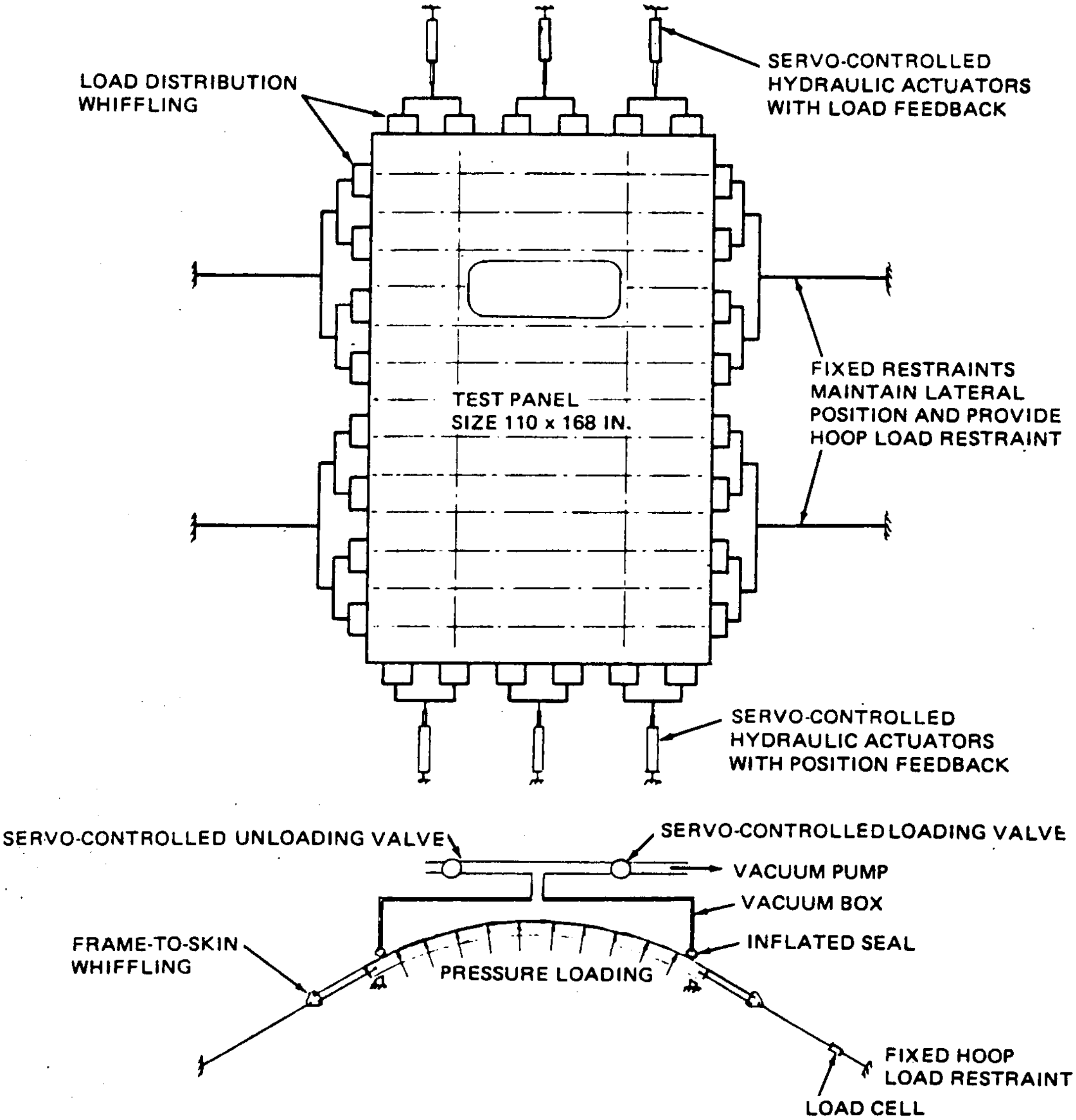


FIGURE 7-21. TYPICAL PRESSURE PANEL TEST FIXTURE



CONCEPT FEATURES:

REALISTIC AIR LOADING FOR CRACK GROWTH

REALISTIC PANEL EDGE LOADING CONDITIONS

INSIDE OF PANEL AVAILABLE FOR INSPECTION UNDER LOAD

COUNTERBALANCED VACUUM BOX LIFTS FOR
OUTER PANEL SURFACE INSPECTION

FIGURE 7-22. LARGE FUSELAGE PANEL TEST

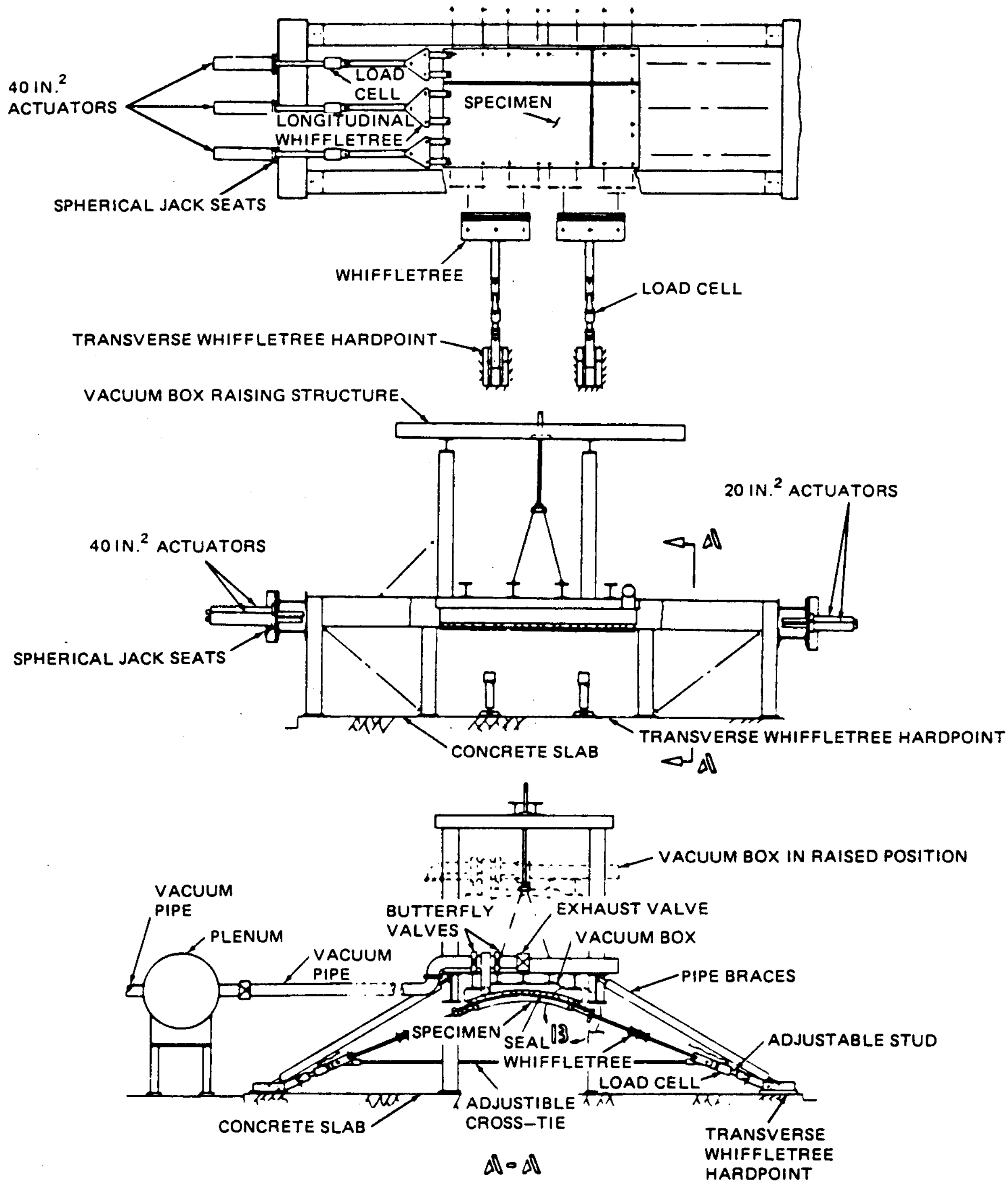


FIGURE 7-23. LARGE FUSELAGE PANEL TEST FIXTURE

Fuselage Frame Bending Test — Local instability at frame discontinuities under high design bending loads will be assessed by applying bending loads to the frames with the panel supported along the longitudinal sides. Bending stresses will be determined from strain gages as the loading is incrementally increased until failure. A constant bending moment across the critical structure will be obtained by loading the panel as a simply supported beam, as sketched in Figure 7-16.

Structural Verification Tests

The structural verification tests are to be conducted in Phase II on panels, subcomponent sections, joints, and fittings to verify that design details from the development tests satisfy the design and FAA requirements. These tests are to be completed before fabricating the full-scale composite fuselage subcomponent section for ground test. Subcomponents representative of the final design will be tested to verify the static strength, fatigue, and damage tolerance characteristics of critical design details of the composite fuselage and to demonstrate satisfactory repairability of the composite fuselage. Table 7-12 and Figure 7-24 present typical design verification tests. These tests will be initiated as soon as possible after development tests on a particular design detail are completed.

Acoustic Testing — The test program described below and in Table 7-13 will utilize specimens fabricated for structural tests and inexpensive advanced test techniques. Both panel acoustic tests and barrel tests will be performed.

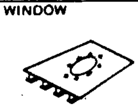
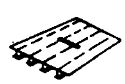
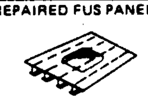
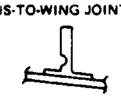
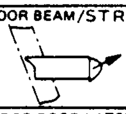
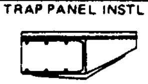
The panel acoustic tests will evaluate the effects of varying mass and stiffness on noise reduction. These tests will be conducted on flat panels over ranges of mass and stiffness which are likely to be encountered in actual fuselage designs. For comparative purposes, one of these panels will be similar to the design used in the fuselage barrel section.

To reduce costs and to provide additional insight into the behavior of the panels, it is recommended that time delay spectrometry be used for the panel tests. This test method eliminates the need for a paired-chamber acoustic laboratory and avoids the problems associated with panel mounting systems. It requires a panel size of approximately 8 by 8 feet to achieve a lower frequency limit of about 100 Hz.

As presently planned, composite panel specimens for structural testing will be fabricated in two parts; one part being the skin, longerons, and shear tees; and the other part being the frames. The two parts will be assembled using conventional fasteners. Four 8- by 8-foot panels will be fabricated in this manner for acoustic testing, and their mass and stiffness varied by progressively bonding additional material plies to the skin or frames. Material bonded to the skin effectively adds mass to the panel without significantly altering the stiffness. Conversely, material bonded to the frames significantly increases the stiffness while having only a minimal effect on panel mass. This approach will reduce the cost of fabricating entire panels for each parametric variation studied.

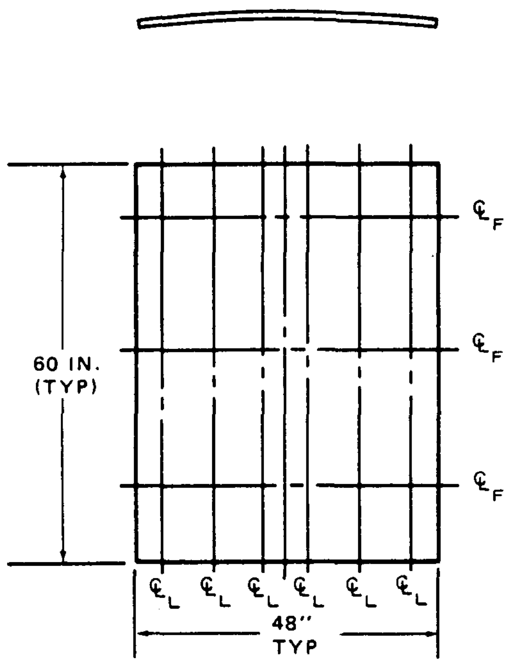
Coincidence effects, which are important to the high-frequency noise transmission behavior of panels, will be investigated using the data obtained from the panel tests. No additional testing requirements are anticipated.

TABLE 7-12
DESIGN VERIFICATION TESTS

TEST NO.	TEST SPECIMENS	SPEC SIZE (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP. (°F) -65	NO. OF SPECIMENS TEST TEMP. (°F) AMB	NO. OF SPECIMENS TEST TEMP. (°F) 180	NO. OF SPECIMENS TOTAL
1	FUS SKIN PANEL	48 x 60	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	180	95		1		2
				COMPRESSION AND SHEAR	1	180	95		2		2
				COMPRESSION AND NORMAL PRESSURE	1	180	95		1		1
2			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	180	95		2		2
3	WINDOW	48 x 60	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	180	95		1		1
				COMPRESSION AND SHEAR	1	180	95		1		1
4			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	180	95		1		1
5	DAMAGED PANEL	48 x 60	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	180	95		1		1
				COMPRESSION AND SHEAR	1	180	95		1		1
			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	180	95		1		1
6	REPAIRED FUS PANEL	48 x 60	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	180	95		1		1
				COMPRESSION AND SHEAR	1	180	95		1		1
			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	180	95		1		1
7	FUS-TO-WING JOINT 	12 x 24	JOINT STRENGTH	FLEXURE	1	AMB	AMB		2		2
8	FLOOR BEAM/STRUT 	24 x 72	JOINT STRENGTH	3-POINT BEAM BENDING	1	AMB	AMB		2		2
9	CARGO DOOR LATCH	5 x 30	STATIC STRENGTH	DESIGN ULTIMATE LOAD	1	AMB	AMB		1		1
			FATIGUE STRENGTH	TBD	1	AMB	AMB		1		1
10	TRAP PANEL INSTL	24 x 40	STATIC STRENGTH	DESIGN ULTIMATE LOAD	1	AMB	AMB		1		1
			FATIGUE STRENGTH	TBD	1	AMB	AMB		1		1
11	FUS SKIN PANEL 	96 x 96	MASS AND STIFFNESS ON NOISE REDUCTION	ACOUSTIC	1	AMB	AMB				1

Material damping measurements can be taken on one of the 8- by 8-foot flat panels that will also be used for transmission loss measurements. The test method is the same as described for the structural damping measurements on the fuselage barrel section. The material damping measurements should be taken before the transmission loss measurements on a panel similar to the design used for the barrel tests. The frame portion of the panel should be removed to eliminate the damping action of the mechanical fasteners.

In the second series of tests, the fuselage barrel discussed under full-scale verification tests will be used for some acoustic testing. Transmission loss of a pressurized composite fuselage barrel section is determined by means of noise reduction and interior absorption measurements. The test specimen and pres-



CONFIGURATION:	TYPICAL CONSTANT SECTION SKIN PANEL ASSEMBLY
TEST ENVIRONMENT:	PHASED SPECTRUM LOADS (SHEAR, COMPRESSION, AND PRESSURE) WITH SPECTRUM ENVIRONMENT (-50°F TO 140°F AND 100 PERCENT RH)
CARBON-EPOXY:	(TBD)
ADHESIVE:	(TBD)
SKIN:	THICK
TEST:	SHEAR, COMPRESSION, AND CABIN PRESSURE
QUANTITY:	ONE SPECIMEN
PURPOSE:	DETERMINE THE ACCUMULATED DAMAGE OF FATIGUE, GROWTH (TBD FLAWS) AND DURABILITY OF ADHESIVE JOINTS UNDER THE COMBINED EFFECTS OF SKIN WRINKLING AND HIGH TEMPERATURE WITH HUMIDITY VARYING TO LOW TEMPERATURE. THE SHEAR-COMPRESSION LOAD AND ENVIRONMENT WILL BE IN A SPECTRUM AMPLITUDE BUT IN A GIVEN PHASE, WITH ONE ANOTHER CONSISTENT WITH THE PREDICTED AIRCRAFT IN SERVICE REAL-TIME FLIGHT EXPOSURE.

FIGURE 7-24. PHASED SPECTRUM LOADS TEST

surization requirements of this test are common to structural testing; therefore, this test will incur only minimal additional cost, yet yield valuable acoustic data that cannot be reproduced by other methods such as panel testing.

Noise attenuation through the fuselage barrel will be measured using a loudspeaker noise source and interior and exterior microphone arrays. Four to six microphones in each array should be sufficient. Absorption in the form of fiberglass blankets or open-cell foam is required in the interior to reduce strong acoustic modal response. The loudspeaker source excites the test section exterior with the desired spec-

TABLE 7-13

ACOUSTIC TEST REQUIREMENTS

TEST OBJECTIVE	SPECIMEN	TEST METHOD	COMMENTS
TRANSMISSION LOSS VERSUS MASS	TWO 8- BY 8-FOOT FLAT PANELS	TIME DELAY SPECTROMETRY	PROGRESSIVE ADDITION OF SKIN PLIES
TRANSMISSION LOSS VERSUS STIFFNESS	TWO 8- BY 8-FOOT FLAT PANELS	TIME DELAY SPECTROMETRY	PROGRESSIVE BUILDUP OF FRAME ELEMENTS
COINCIDENCE EFFECTS	FOUR 8- BY 8-FOOT FLAT PANELS	TIME DELAY SPECTROMETRY	SPECIMENS ARE COMMON TO OTHER ACOUSTIC TEST PANEL REQUIREMENTS. INVESTIGATION OF COINCIDENCE EFFECTS WILL NOT REQUIRE ADDITIONAL TESTING
MATERIAL DAMPING	ONE 8- BY 8-FOOT FLAT PANEL	TAP TESTING	SPECIMEN IS COMMON TO OTHER ACOUSTIC TEST PANELS
TRANSMISSION LOSS OF PRESSURIZED FUSELAGE	FUSELAGE BARREL SECTION	NOISE REDUCTION AND ABSORPTION MEASUREMENTS	PRESSURIZED SPECIMEN IS COMMON TO STRUCTURAL TESTING REQUIREMENTS
STRUCTURAL DAMPING OF PRESSURIZED FUSELAGE	FUSELAGE BARREL SECTION	TAP TESTING	SAME SPECIMEN AS FOR FIFTH TEST LISTED

tra, simulating boundary layer or propulsion system noise. The exterior microphone array will be taped to the fuselage surface and measure noise impinging on the test section exterior. The interior microphones will be positioned at various distances from the sidewall near the area of peak external noise level to obtain a space-averaged interior level. Noise reduction will be obtained by subtracting the interior level from the exterior level. Noise reduction values obtained in this manner can be converted to transmission loss using absorption measurements conducted inside the fuselage. These absorption measurements should be obtained for the same fiberglass blanket or open-cell foam and interior microphone configuration used for the noise reduction measurements. Absorption can be measured by the reverberation time method or the reference noise source method.

Additional acoustic testing of the pressurized fuselage barrel section includes evaluation of structural damping by means of tap testing. A dual-channel analyzer with an instrumented hammer, such as the Hewlett Packard HP5420 system, can be used for these tests. One roving accelerometer will be used as the output transducer.

Full-Scale Verification Tests — This group of tests is also planned for Phase II. All of the data obtained from the previous tests on responses of the different structural concepts and material combinations to the wide range of test loadings and conditions will be utilized. The three large test articles in this group are the bird strike specimen (also used for the hail impact tests), the full-scale barrel section, and the impact dynamics test article.

Bird Strike Tests

Bird strike tests will be conducted on the forward fuselage section of the MD-100 composite structure. The test article will be composed of the structural members located in areas subject to bird impact, as shown in Figure 7-25. The transparencies may be dummy parts, depending on a final determination to be made during the design development.

The pilot, copilot, and any flight systems equipment required for continued safe flight will be simulated during the test to the extent necessary to substantiate flight safety.

The tests will be conducted with a 4-pound bird fired from a pneumatic gun at a speed corresponding to V_C at sea level (350 knots for the MD-100) in accordance with FAR 25. A number of FAA-approved test facilities are available to conduct the tests.

The weight of the bird and the velocity at impact will be verified and documented in accordance with FAA-approved procedures. High-speed movies will be taken of the impact event.

Critical areas to be impacted will be determined during the contract period.

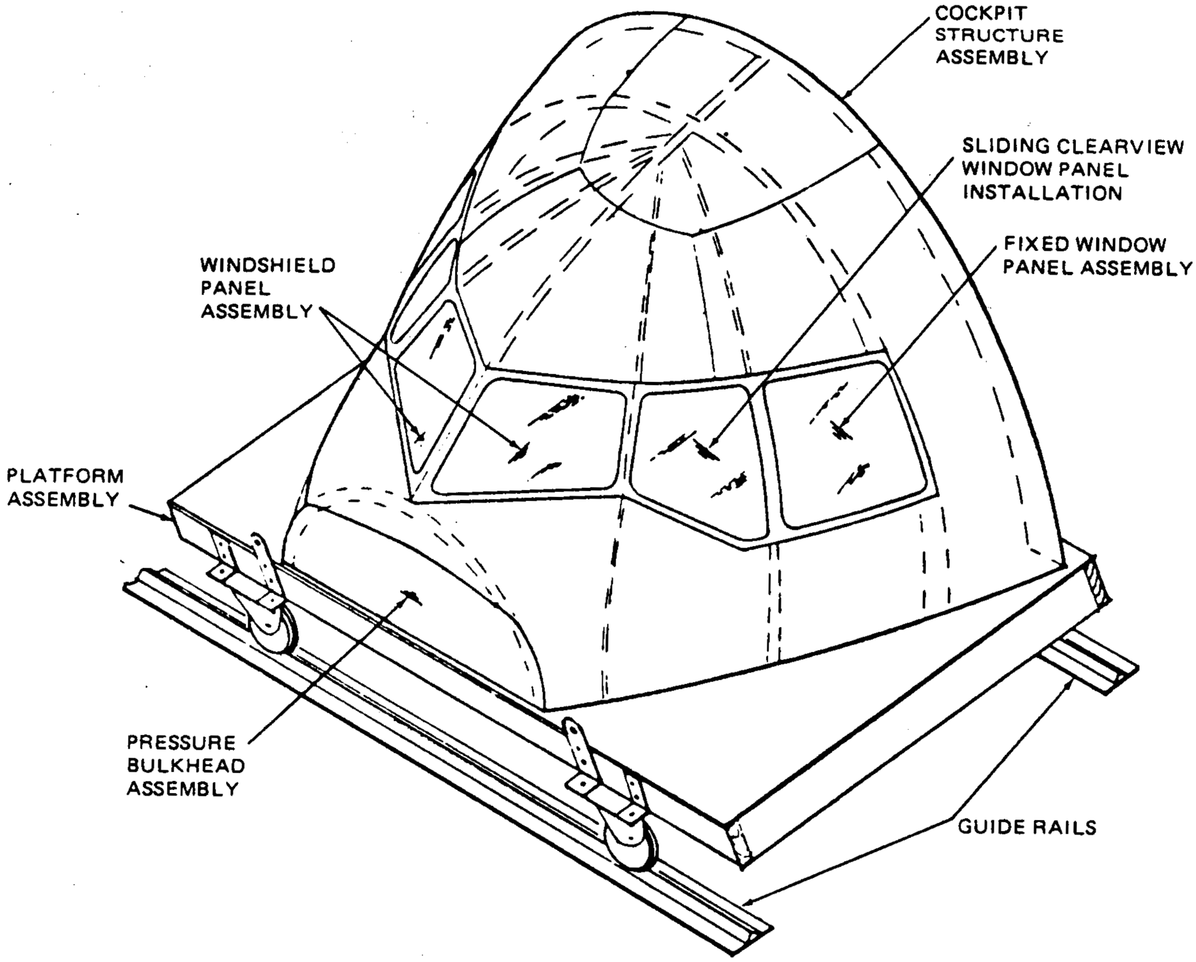


FIGURE 7-25. BIRD IMPACT TEST SPECIMEN

Hail Impact Tests

Hail impact tests will be conducted on selected regions of the bird impact test specimen (Figure 7-25) and on representative curved sections of the composite fuselage. The test panels will be oriented to simulate hail impact during flight or ground operations.

The tests will be conducted with the appropriate size and weight of simulated hailstones fired at the composite panel at selected velocities. Composite fuselage impact criteria and test conditions and procedures will be defined during the contract period based upon the latest available literature and data (e.g., as in Reference 4).

This type of test has often been conducted in the past. Relatively simple facilities are adequate. The weight of the ice pellets and the velocity at impact will be verified and documented in accordance with FAA-approved test plans. High-speed movies will be taken of the impact area.

Critical areas to be impacted and the mass and velocity of the ice pellets will be determined during the contract period.

Full-Scale Barrel Tests

The test article will be a structurally complete composite fuselage barrel approximately 40 feet in length. It will include all structurally significant items such as longitudinal and transverse skin splices, windows (dummy), a door, floor beams, and any additional fittings required for handling and test loading. The composite fuselage will be joined with a metal dummy stub wing. The dummy wing will provide the pressure boundary and the structural interface with the fuselage. It will also function as a test fixture for introduction of wing loads. The composite fuselage will be inspected during fabrication, assembly, test setup, and test, in accordance with FAA conformity inspection procedures.

The fuselage barrel will be used to verify compliance with design requirements and to demonstrate repair procedures. (See Figures 7-26 and 7-27.) Instrumentation will consist of deflection gages, strain gages,

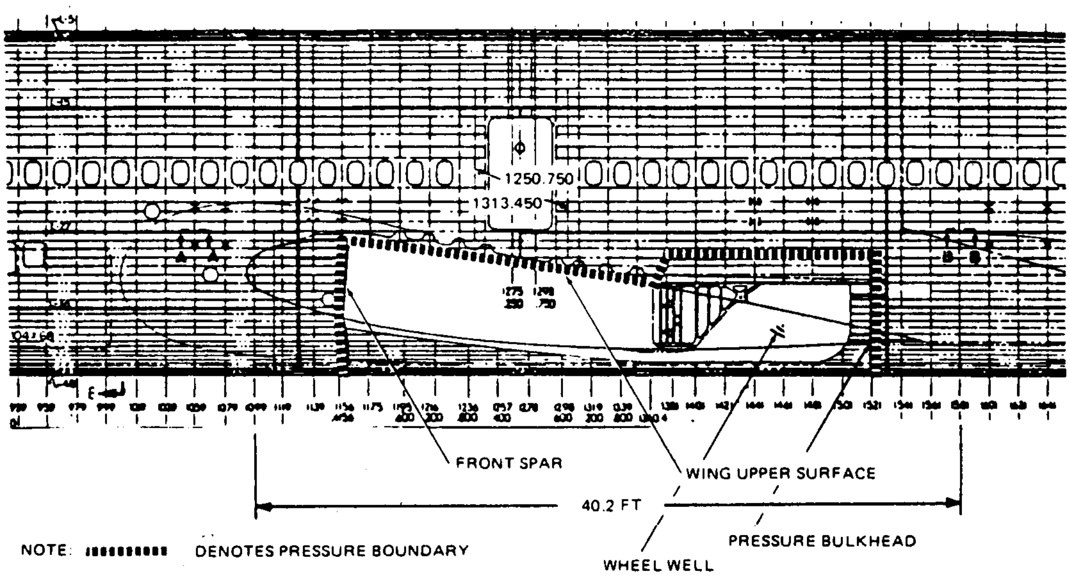


FIGURE 7-26. FULL-SCALE DEMONSTRATION COMPONENT

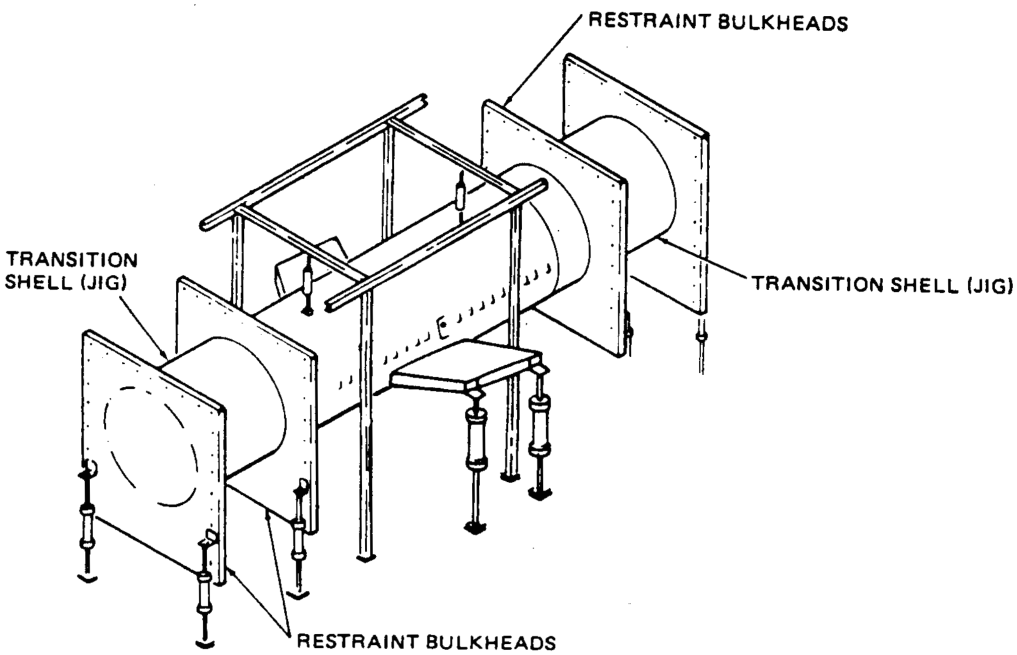


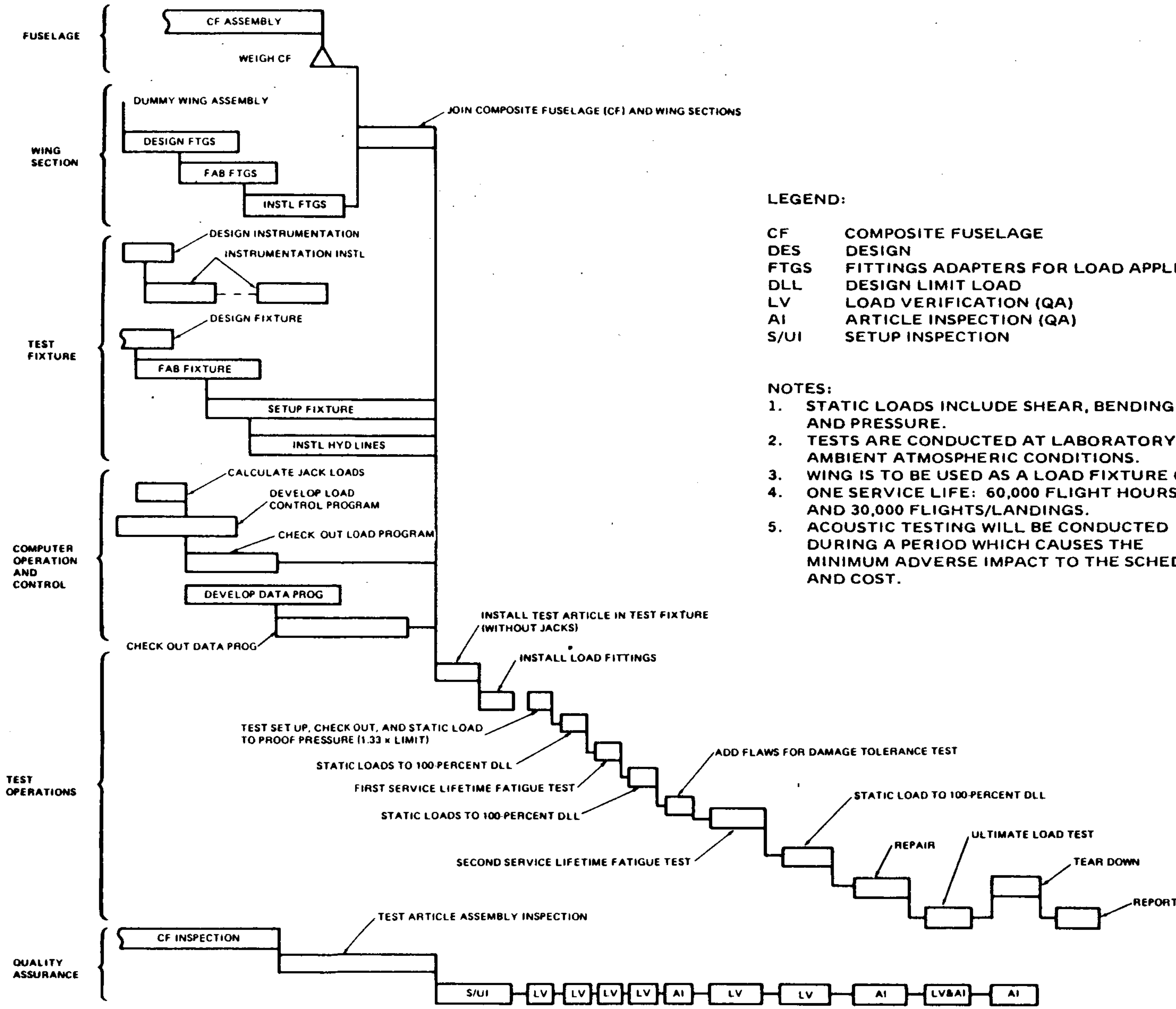
FIGURE 7-27. FUSELAGE BARREL TEST

load cells, pressure transducers, and associated signal conditioning, calibration equipment, power supplies, cabling, computers, and other instruments for load control, protection against overload, and data acquisition. A major design goal for this test is to demonstrate that the composite fuselage possesses the strength, durability, damage tolerance, residual strength, inspectability, and repairability equivalent to or better than the MD-100 metal fuselage. Accordingly, comprehensive acquisition, reduction, and analyses of data are planned to verify that the structural characteristics and strength of the composite fuselage conform to design requirements.

The proof pressure and design limit load tests will probably be conducted first, followed by a design service loads spectrum equivalent to one lifetime (i.e., durability test) and by strength tests to design limit load. Flaws will then be induced in selected locations. Additional tests will include a second service lifetime durability test, design limit strength, and design ultimate strength for the MD-100 critical load conditions. The test article will then be loaded to failure for the most critical condition. Sequencing for the test is shown in Figure 7-28.

Additionally, the full-scale barrel may be subjected to acoustic tests in order to determine transmission loss by means of noise reduction and interior absorption measurements. Further acoustic testing will include evaluation of structural damping by means of tap testing with an instrumented hammer such as the HP5420 system.

The test plan will be approved and the test witnessed by the FAA.



LEGEND:

CF	COMPOSITE FUSELAGE
DES	DESIGN
FTGS	FITTINGS ADAPTERS FOR LOAD APPLICATION
DLL	DESIGN LIMIT LOAD
LV	LOAD VERIFICATION (QA)
AI	ARTICLE INSPECTION (QA)
S/UI	SETUP INSPECTION

NOTES:

1. STATIC LOADS INCLUDE SHEAR, BENDING AND PRESSURE.
2. TESTS ARE CONDUCTED AT LABORATORY AMBIENT ATMOSPHERIC CONDITIONS.
3. WING IS TO BE USED AS A LOAD FIXTURE ONLY.
4. ONE SERVICE LIFE: 60,000 FLIGHT HOURS AND 30,000 FLIGHTS/LANDINGS.
5. ACOUSTIC TESTING WILL BE CONDUCTED DURING A PERIOD WHICH CAUSES THE MINIMUM ADVERSE IMPACT TO THE SCHEDULE AND COST.

FIGURE 7-28. COMPOSITE FUSELAGE BARREL TEST PROGRAM

Impact Dynamics

The test specimen will consist of a full-scale fuselage structure of sufficient size and suitable geometry to provide an acceptable representation of a large transport aircraft. A candidate configuration for such a test specimen is shown in Figure 7-29. The constant-diameter section has common frames, longerons, shear tees, and skin gages to minimize tooling, fabrication, and assembly costs. In this design, the loft-line defines a body of revolution to provide a representative fuselage shape while minimizing fabrication and assembly costs. To achieve further cost savings, the aft section structure is basically identical to the forward nonconstant section. Structural details throughout the shell are to be as generic as possible without sacrificing the quality or validity of test results.

A stub wing will be added to the fuselage to act as an outrigger and provide the fuselage roll stability afforded by the wing, pylon, and nacelle structure. The stub wing may consist of a simple rectangular box construction using low-cost materials such as aluminum or steel. The cross section of the wing structural box could be a straightforward trapezoidal shape with all sides flat.

The interior structure will consist of floor beams and support struts for both the upper and lower floors. Floor panels will be installed with seat tracks and passenger seats in selected locations.

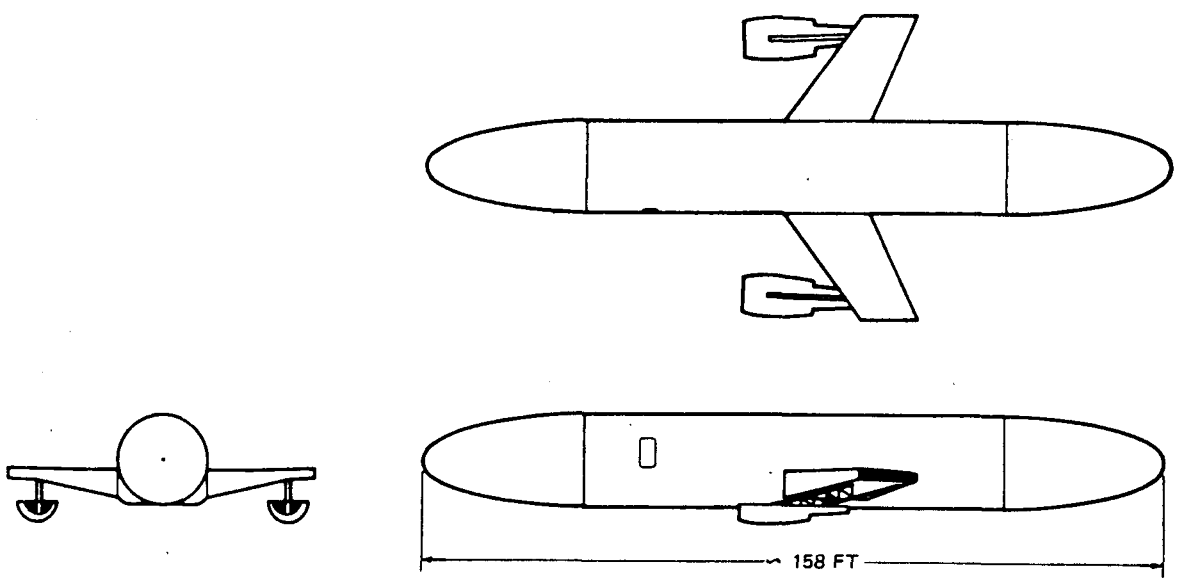


FIGURE 7-29. IMPACT DYNAMICS TEST ARTICLE

Definition of the actual test procedure will follow a careful evaluation of test sites, launching methods, and instrumentation from both a cost and technical standpoint. The fuselage structure must include sufficient payload to produce a representative gross weight and the test specimen must be accelerated to a velocity which is approximately equal to landing speed. Appropriate instrumentation will include strain gages and accelerometers in the structure as well as videotaping equipment to observe the test.

Flight Service Evaluation

The lower skin panel assembly at the forward end of the fuselage constant section will be the test panel to be manufactured, installed in the airplane, flight tested, and placed in airline service for a flight service evaluation program. The panel is bounded by longeron 27 along its upper edge, longeron 48 on the opposite side of the fuselage at the bottom, station 765 at the forward edges, and fuselage station 1129 at the aft end. (See Figure 7-30.) The panel includes a large lower cargo door and jamb installation. The panel is 17.5 feet along the circumference and nearly 30.5 feet in length.

Major test articles, including the full-scale ground test composite fuselage test article, will be damaged, repaired, and tested to develop approved techniques and procedures for manufacture and in-service repairs.

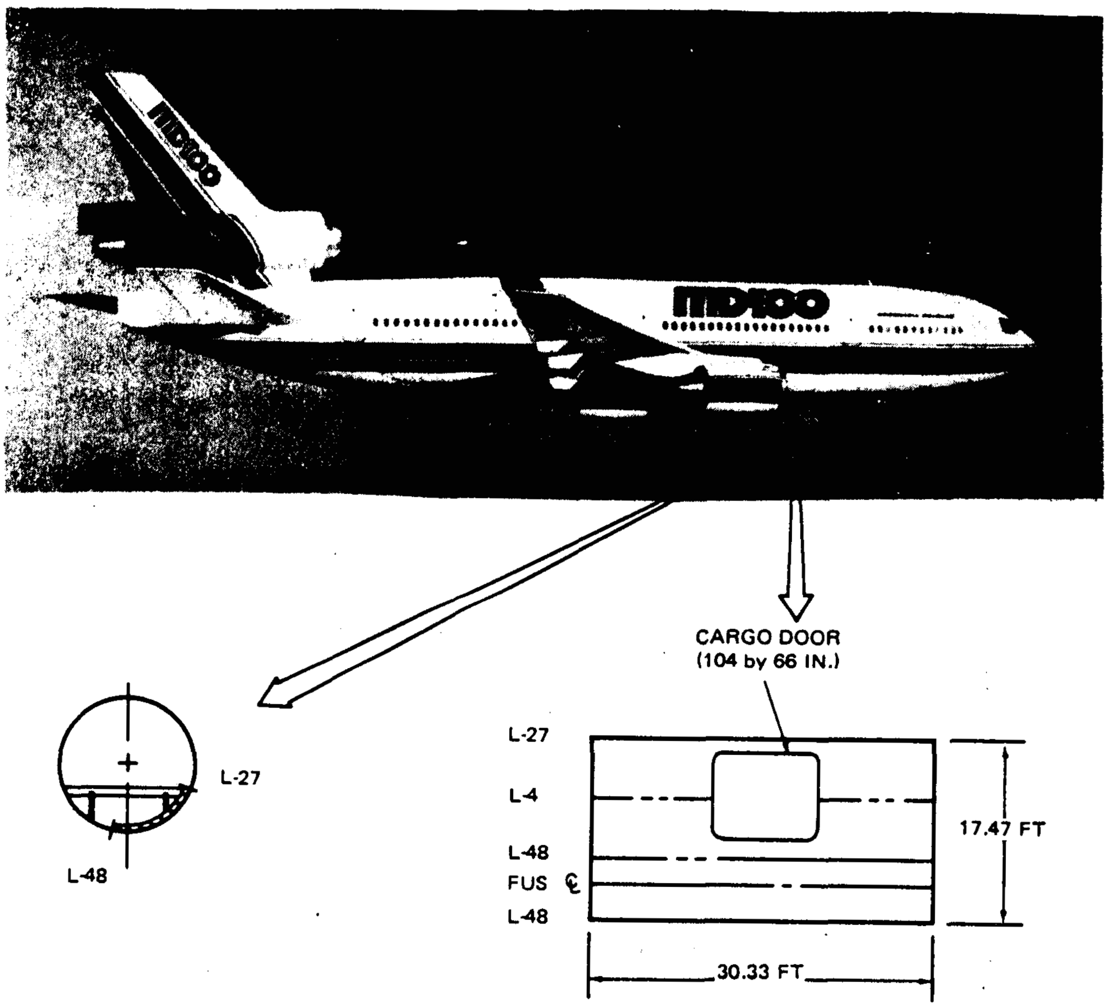


FIGURE 7-30. FLIGHT TEST PANEL

Instrumentation on the panel will be limited to strain gages and thermocouples. The primary concern is the difference in thermal coefficients of expansion between the carbon-epoxy panel assembly and the aluminum fuselage. The thermal incompatibility results in locked-in residual stresses in the composite panel assembly and the adjacent metal fuselage structure. Tensile stresses are induced in the metal and compression in the composite when the aircraft is flying at high altitudes where the ambient air temperature is less than the temperature was on the ground when the panel was installed in the fuselage.

Appropriate inspection procedures will be followed to ensure that the installation of the composite panel in the MD-100 fuselage meets or exceeds the requirements for flightworthy transport aircraft.

A functional ground check will be performed on the installed strain gages and thermocouples. One or more flight tests are planned after all necessary instrumentation has been installed, checked, and approved by the appropriate agencies. Prior to the test flight, the FAA will issue an experimental certificate of airworthiness. Details of the flight test program will be developed in accordance with the requirements discussed in the following text.

The flight test demonstration of a composite fuselage panel on the MD-100 aircraft will be limited to items that could be affected by the installation of the composite fuselage panel. An FAA certification test requirement program will be prepared by the Douglas flight test engineers with the coordination and agreement of FAA personnel to show compliance with FAA regulations.

The extent of the thermal incompatibility will be continuously monitored during flight, and a careful assessment will be made as to the magnitude and location of the residual stresses.

Following the flight test program and FAA certification for airline service, the MD-100 will be refurbished. All test intallations will be removed, the necessary interior items for revenue service reinstalled, and the aircraft returned to the airline to begin in-service flight evaluation.

RESOURCE ALLOCATIONS

The manpower, materials, and other cost items required to conduct the three-phase composite fuselage technology development program have been estimated and allocated in accordance with the program schedule shown in Figure 7-1.

The estimates for materials, computing budget, travel, and other miscellaneous costs are expressed in equivalent man-years to simplify the presentation. The resource estimates do not include man-hours for the design, manufacture, and test of two major test articles: (1) the large generic composite structure for the Phase II impact dynamic test (Figure 7-29), and (2) the forward fuselage bird strike test (Figure 7-25).

The Phase II technology development effort for these two items was described to illustrate that a significant hardware program will probably be required in Phase II to resolve the technical issues of impact dynamics and bird strike. The requirements for the test articles in these two large-scale demonstration tests will be defined in a Phase I task.

The Phase III resource allocation estimates terminate with the delivery of the aircraft to an airline for the flight service evaluation. The cost to install the panels on the aircraft and the FAA certification costs are included in the estimates. The acquisition costs of the airplane and the in-service maintenance costs, evaluation estimates, and assessment of performance are not included in the resource estimates.

The rough-order-of-magnitude estimates of the equivalent man-years needed to conduct the composite fuselage technology development program are shown in Table 7-14. These estimates were made to compare program options and to prepare plans. They are not based on rigorous procedures, and should not be construed as suitable for other purposes.

Figure 7-31 shows the percentages of the resources allocated by phase and by functional department. Figure 7-32 distributes the resources in accordance with the 8-year program schedule.

TABLE 7-14
COMPOSITE FUSELAGE DEVELOPMENT PROGRAM
MAN-YEAR ESTIMATES*

PHASE	TASK	ENGINEERING	MANUFACTURING	TEST	TOTAL	PERCENT
I	DESIGN DEVELOPMENT	153	180	140	473	52
II	STRUCTURAL VERIFICATION	28	250	83	361	43
III	FLIGHT SERVICE EVALUATION	8	23	12	43	5
	TOTAL	189	453	235	877	
	PERCENT	21	52	26		100

*MATERIAL, TRAVEL, AND OTHER MISCELLANEOUS ITEMS ARE INCLUDED IN EQUIVALENT MAN-YEARS

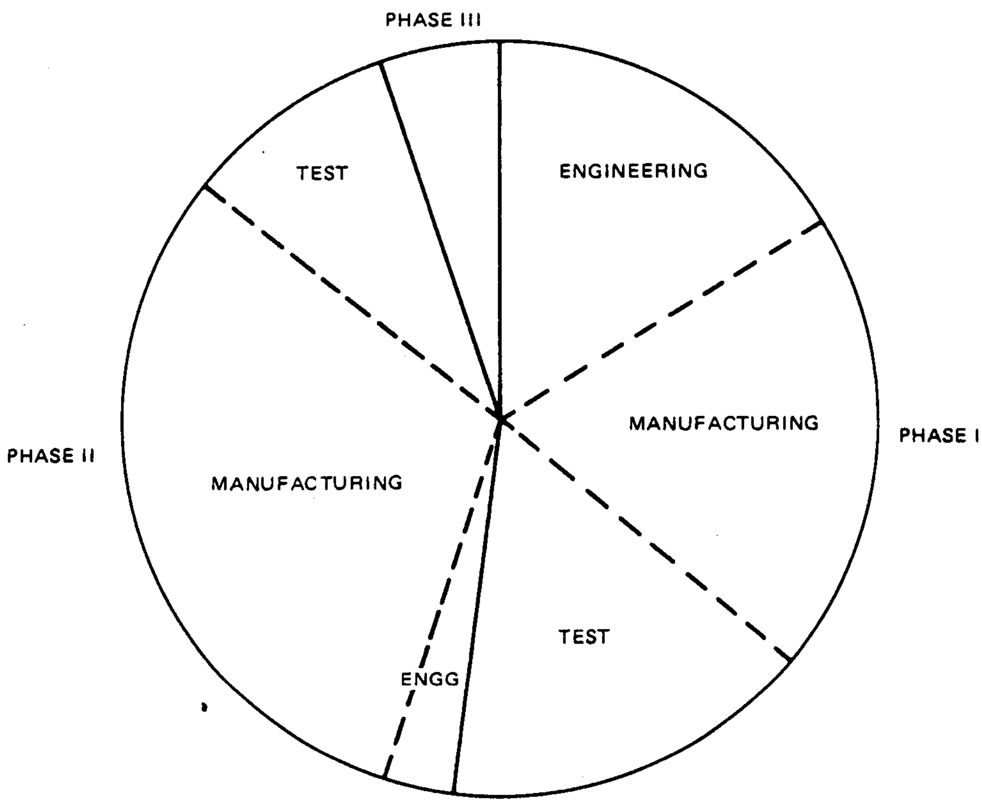


FIGURE 7-31. FUNCTIONAL DISTRIBUTION OF RESOURCES FOR COMPOSITE FUSELAGE DEVELOPMENT PROGRAM

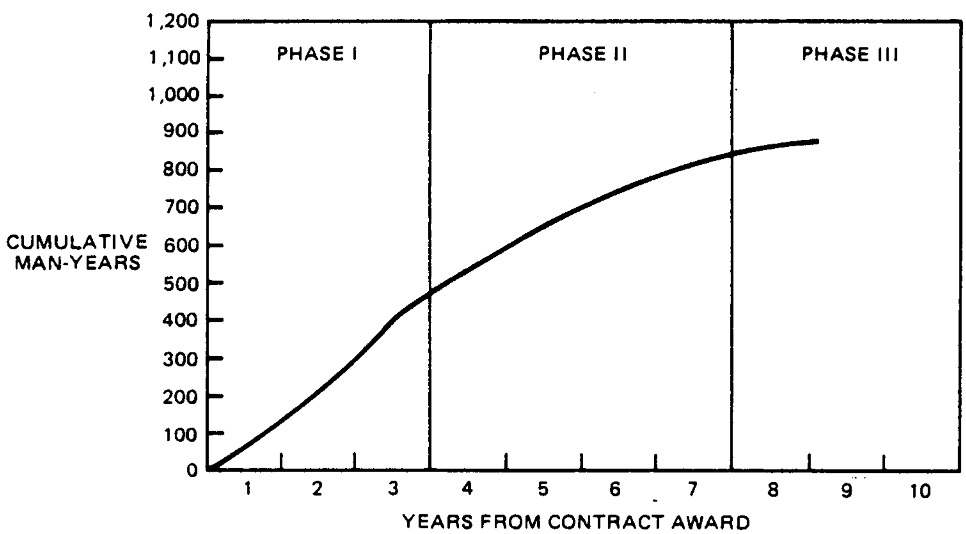


FIGURE 7-32. RESOURCE ALLOCATION FOR THE COMPOSITE FUSELAGE DEVELOPMENT PROGRAM

SECTION 8
FACILITIES AND EQUIPMENT

PHASE I — DEVELOPMENT

Phase I of the Development Plan covers the preparatory effort for construction of the large barrel section. All structural element and panel test specimens will be fabricated in the Manufacturing Research and Development (MR&D) Center, which supports composite activities with facilities and directed development of tooling, curing, and assembly methods (Figure 8-1). Existing environmental layup rooms, material storage systems, autoclave curing facilities, and nondestructive examination systems will be utilized. Details requiring mechanical trimming, drilling, and fastener installation will be assembled in much the same way as in the NASA Composite Vertical Stabilizer program.

A total of 28,000 ft^2 is dedicated to manufacture of composites. Facilities include a computerized material inventory system (MAPPER), two environmentally controlled layup rooms, two autoclaves (10 foot diameter and 5 foot diameter), an automated material cutting system (Camsco), Automation Industries' nondestructive examination squirter (40-foot tank), an assembly area with overhead crane, drills, a dust collection system, and a supporting tool and die shop. Sufficient floor space exists in the center for tooling storage, inspection, and production control of details.

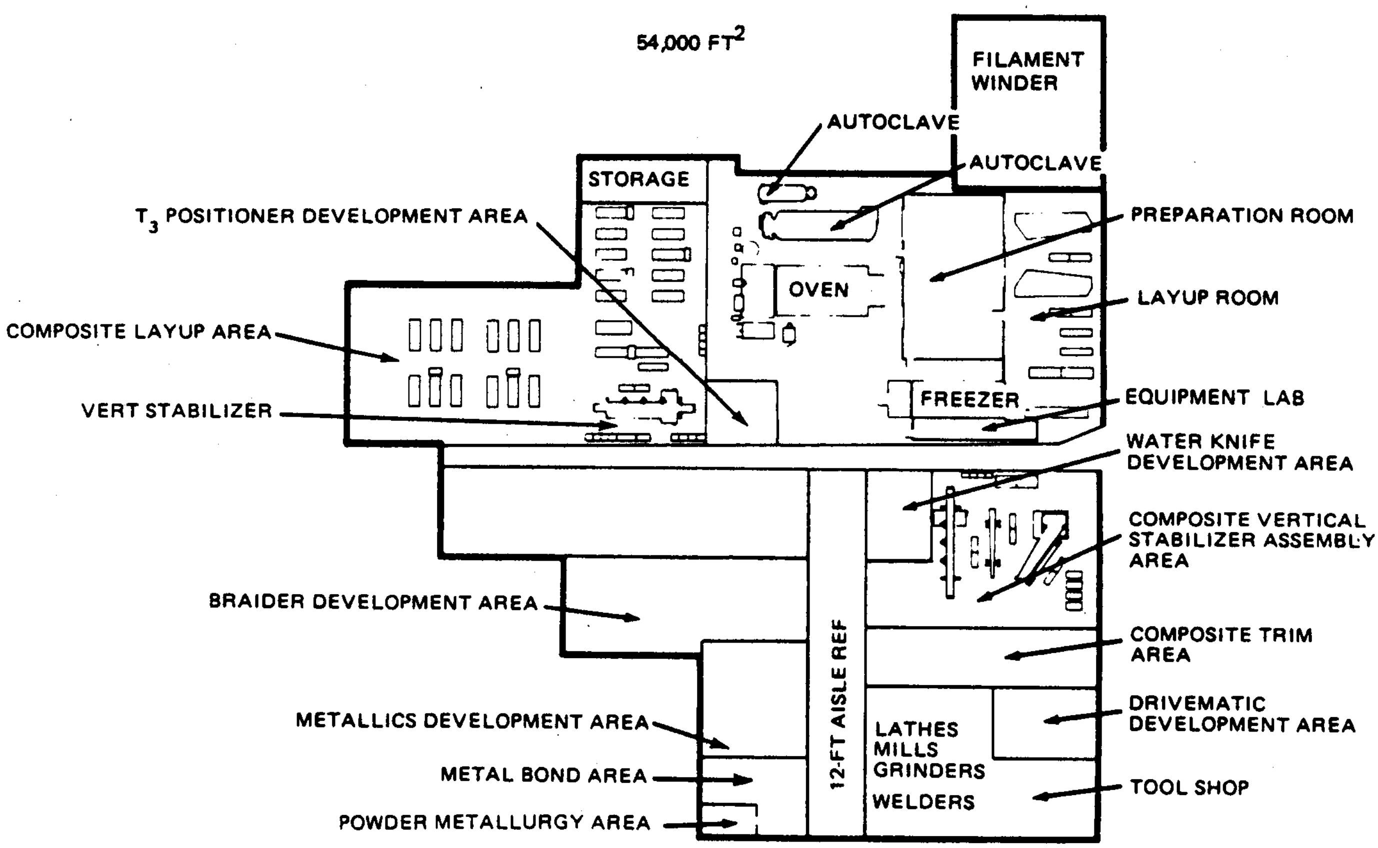


FIGURE 8-1. MANUFACTURING RESEARCH AND DEVELOPMENT CENTER

During Phase I, an investigation will be made to determine how practical the filament-winding process is for panel fabrication. MR&D presently has a McLean-Anderson W-2 winder in operation. Modifications to the winder are underway to provide computerized controls that allow winding any fiber angle with dwell capability on the mandrel poles. The winder will be functional long before the fuselage program need date. This machine is adequate for initial development tasks, but a larger machine will be required if the Phase II ground test article becomes a filament-wound structure.

PHASE II — STRUCTURAL VERIFICATION

The overwing barrel is the most complex of the fuselage sections because of the wing joints and the keel and wheel well structure. Two demonstration barrels are proposed, each representing an actual production fuselage section. The fabrication of these barrels requires layup tooling for all the detail parts, adequate layup area, and curing equipment. Once parts are fabricated, inspected by nondestructive methods, and trimmed, they will be accounted for by standard production control methods. The barrel sections will be assembled in normal production shop buildings fitted with environmental control equipment peculiar to composite drilling and processing.

The Composites Manufacturing Development Facility will be utilized for this effort. Facility plans call for the following equipment and accommodations in approximately 125,000 square feet of floor space. (See Figure 8-2.)

Material Storage

— Freezer, 20 by 30 by 10 feet, with retrieval system

— Cold room, 30 by 40 by 15 feet, 40°F, for temporary holding during layup

Material Preparation

— Clean room layup area, 30,000 ft^2

— Gerber material cutting system, 10 by 40 feet

— Tape layup machine, 30 by 60 feet

Curing Equipment

— Autoclave, 25-foot diameter by 50 feet

— Autoclave, 10-foot diameter by 30 feet

— Oven, 25 by 40 feet

— Oven, 6 by 12 feet

— Heated press, 25 by 6 feet, 325°F

Inspection

— Automated ultrasonic scanning system (NDE)

— In-motion x-ray equipment

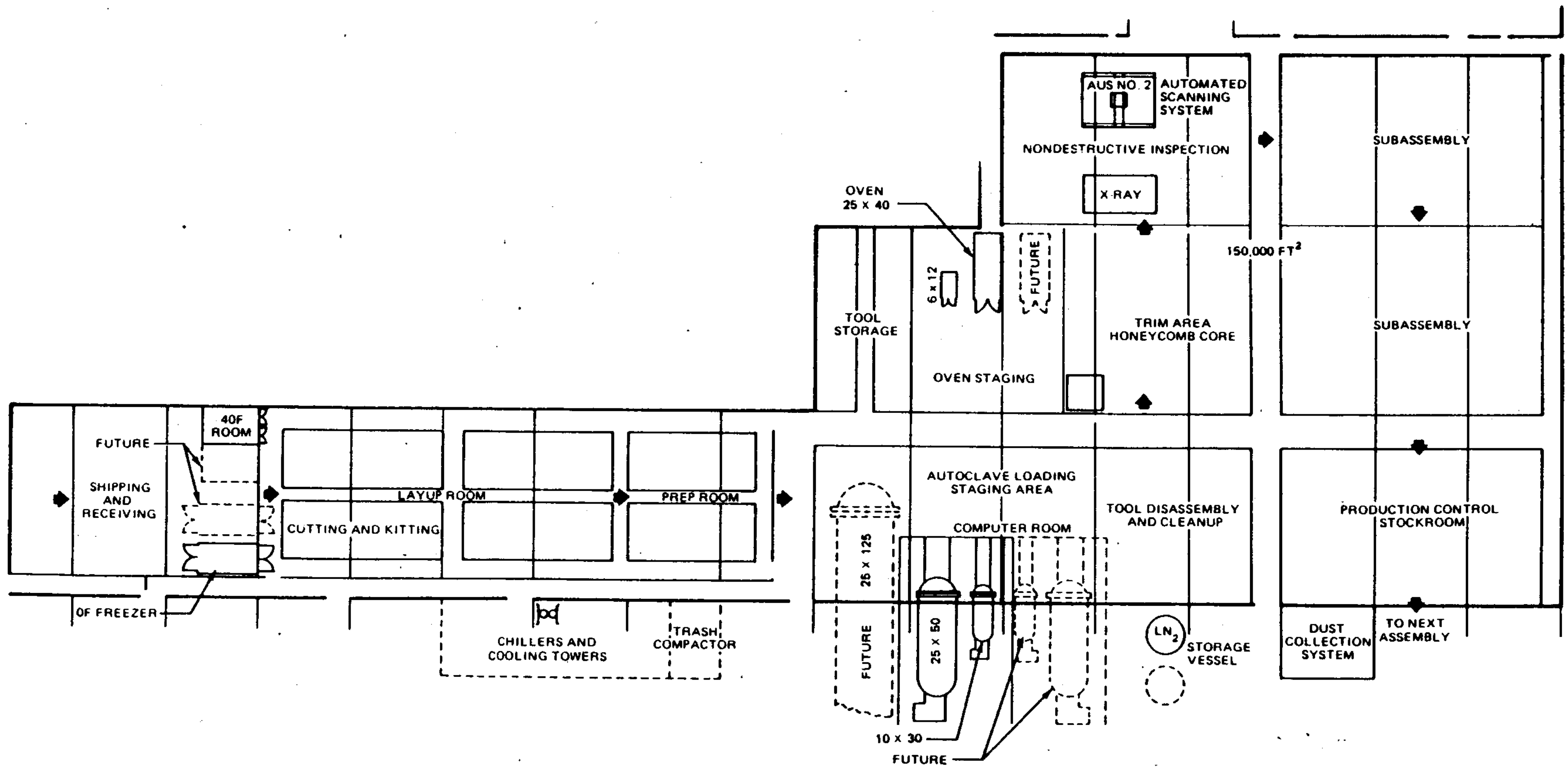


FIGURE 8-2. PHASE II GROUND TEST ARTICLE MANUFACTURING DEVELOPMENT FACILITY

In addition, the development facility will have a machine shop and tool fabrication, staging, and receiving areas.

An estimated 75,000 ft^2 of floor area will be needed to assemble the barrel sections. This area will be located in a high-bay building with subassembly tools positioned to feed into the final assembly fixture. A flow diagram of the assembly area for Phase II was shown in Figure 7-10. Standard tool clearances have been provisioned to allow access for work crews.

PRODUCTION FACILITY

A production facility forecast for the manufacture of MD-100 composite fuselage structures would include consideration of the other components of the MD-100 that are being converted to advanced composites, the MD-100 production rate, and the established policy of subcontracting major portions of the airframe. The advanced composite utilization on other types of aircraft in production at the same time and their production rates and subcontract plans must also be considered.

A serious commitment to the production of composite fuselage structure must be accompanied by a like commitment to a facility of sufficient scale and automation with an efficient layout to permit low-cost production of quality structure.

Initial estimates are that 700,000 to 1 million square feet would be sufficient with proper design and equipment (Table 8-1) and with present subcontracting practices. Periodic reviews of the facilities plan should be made, for such reasons as to include filament-winding methods if proven to be effective during the technology development program.

TABLE 8-1

FACILITY REQUIREMENTS - PRODUCTION

EQUIPMENT	DEVELOPMENT PHASE II EXPANSION PHASE I	PRODUCTION EXPANSION PHASE
MATERIAL STORAGE FREEZER	20 BY 30 BY 10 FT	20 BY 30 BY 10 FT (3)
MATERIAL HANDLING	OVERHEAD CRANES/HOISTS/ GANTRY (BUILDING STANDARD EQUIPMENT)	OVERHEAD GANTRY
CUTTING EQUIPMENT	10 BY 40 FT GERBER CUTTING SYSTEM	10 BY 40 FT CUTTING SYSTEMS (2) WATERJET TRIMMING SYSTEM (1)
MATERIAL LAYOUT EQUIPMENT	30 BY 60 FT 7-AXIS TAPE LAYING MACHINE	30 BY 150 FT 7-AXIS TAPE LAYING MACHINE EXPANDED
	W-60 FILAMENT WINDING MACHINE (DEVELOPMENT)	
CURING EQUIPMENT	25 BY 50 FT AUTOCLAVE 10 BY 30 FT AUTOCLAVE 25 BY 40 FT OVEN 6 BY 12 FT OVEN 25 BY 6 FT PLATEN PRESS	25 BY 125 FT AUTOCLAVE 25 BY 50 FT AUTOCLAVE 10 BY 30 FT AUTOCLAVE (2) 25 BY 40 FT OVEN (2) 25 BY 6 FT PRESS (2)
NONDESTRUCTIVE INSPECTION EQUIPMENT	AUSS 3 SYSTEM (C-SCAN) AMEX SYSTEM (IN MOTION X-RAY)	AUSS 4 SYSTEM (2) AMEX SYSTEMS (2)
TOOLING -		
FABRICATION	SHIPSET (GTA)	SHIPSETS (4)
ASSEMBLY	SHIPSET (GTA)	SHIPSETS (3)

SECTION 9
APPLICATIONS AND BENEFITS

The MD-100, selected as the baseline for the application of composites to the fuselage structure, has also been evaluated for application of composites to empennage structures.

The commitment to utilize composite materials for the fuselage structure of a large transport aircraft will not be made until the technical issues such as damage tolerance, durability, lightning protection, large cutouts, and joints have been resolved. Since composite wing technology development is planned before composite fuselage technology development, it is probable that by the time the aircraft industry is ready to accept a large commercial transport with a composite fuselage, the wing and empennage structures would also be fabricated of composites.

Accordingly, the analyses which were conducted to determine the weight and fuel savings of an MD-100 type aircraft also featured composite wings and empennage structures as well as a composite fuselage. The analyses did not include down-sizing the lifting surfaces or engines to take full advantage of the reduced structural weight. The results of the analyes are shown in Table 9-1. The total weight savings of 33,189 pounds was established from an estimate for each of the components. The fuselage estimate was based upon an allowable ultimate strain level of $\pm 4{,}500$ μin./in. for strength-critical structure and on maintaining fuselage stiffness criteria for shell stability and panel buckling. The wing estimate was based upon current and previous wing studies. The empennage weight reduction is supported by data from the NASA ACEE-funded DC-10 composite rudder and composite vertical stabilizer programs.

TABLE 9-1
ESTIMATED WEIGHT SAVINGS FOR
MD-100 COMPOSITES UTILIZATION

	WEIGHTS		SAVINGS	
	BASELINE (LB)	COMPOSITE (LB)	WEIGHT (LB)	PERCENT
FUSELAGE	41,765	28,516	13,249	32
WING	65,064	48,798	16,266	25
EMPENNAGE AND AFT BODY	18,624	14,950	3,674	20
				AVERAGE
	125,453	92,264	33,189	26

The fuel savings are based upon a typical 2,000-nautical-mile mission with a 55,350-pound payload, which results in a takeoff gross weight of 377,640 pounds. The annual fuel consumed is 6.05 million gallons for the baseline and 5.52 million gallons for the composite fuselage. The resulting performance change is 0.53 million gallons of fuel. The composite benefits are summarized in Table 9-2. The cost savings are shown in terms of known 1983 dollar fuel costs and with an assumed 8-percent annual cost increase. Although world fuel prices have remained fairly stable in the last few years, highly volatile world political forces, artificial economic supply constraints, and a dwindling oil reserve could cause fuel prices to soar and invalidate low-growth cost projections based on recent performance.

TABLE 9-2

BENEFITS OF COMPOSITES ON LARGE COMMERCIAL TRANSPORT

WEIGHT SAVINGS:	33,189 POUNDS
FUEL SAVINGS:	10,600,000 GALLONS
COST SAVINGS:	(BASED ON 20-YEAR LIFE)
• 1983 FUEL COST AT $0.89 PER GALLON:	$ 9,434,000
• PROJECTED 20-YEAR COST SAVINGS AT AN 8-PERCENT ANNUAL INCREASE IN FUEL COST	$43,970,000
<u>OR</u>	
PAYLOAD INCREASE OF 33,189 POUNDS	

SECTION 10
STUDY CONCLUSIONS

The technology base for composite fuselage structure has been developed in recent years with the many advances made in utilization of composite materials in aircraft. Now, the data base should be extended to cover design features unique to large transport fuselage structure, and to perform the design integration to assure structural integrity and function at a lower cost-to-weight ratio than can be provided by competing fuselage structures that are expected to be within the state of the art by the time the advanced composite fuselage is ready for production.

A comprehensive program should be conducted to accomplish the following: (1) develop the engineering and manufacturing data base; (2) resolve the technology gaps; (3) integrate the numerous design requirements in a manner that satisfies structural integrity and functions; (4) demonstrate the design by a convincing full-scale test program; and (5) conduct a flight service evaluation to prove maintainability and durability under realistic operating conditions.

The composite fuselage technology developed for either a civil or military transport aircraft would be applicable to the other. Most of the structure is designed to comparable criteria and loads. Exceptions include criteria and loads for operations from unimproved runways and survivability for hostile threats during military operations. The Air Force has ongoing programs that address the survivability issue.

Optimal conceptual design studies indicated that a 32-percent weight saving was attainable for the study composite fuselage compared to the conventional MD-100 baseline aluminum fuselage. The conceptual design did not fully integrate durability, damage tolerance, impact dynamics, lightning strike, and other technical requirements, but did make some weight allowance for design strain limitations and for lightning strike protection.

Secondary structure, control surfaces, and other primary structures made of composites will be technically ready for production before composite fuselage structure. A realistic facilities and equipment plan should include fuselage production requirements as part of the total composite airframe facilities plan.

The comprehensive program outlined to achieve technology readiness for the application of composites to transport fuselages totals 877 man-years. Cost-benefit studies for an all-composite commercial transport suggests that $44 million cost savings can be achieved in reduced fuel usage over a 20-year service period.

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