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ADVANCED HELICOPTER STRUCTURAL DESIGN INVESTIGATION

Volume I - Investigation of Advanced Structural Component Design Concepts

Boeing Vertol Company
P. O. Box 16858
Philadelphia, Pa. 19142

March 1976

Final Report for Period June 1974 - May 1975

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Prepared for

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
Fort Eustis, Va. 23604

EUSTIS DIRECTORATE POSITION STATEMENT

This effort is one of two parallel contractual studies to define advanced structural configurations, advanced materials, and fabrication technology to satisfy requirements for a complete helicopter. The associated study program under the same title was conducted by Sikorsky Aircraft under the terms of Contract DAAJ02-74-C-0061.

Numerous design concepts, materials, and manufacturing techniques were investigated for various helicopter components (i.e., body group, main rotor, transmission, etc.). The best overall concepts were selected and integrated into a complete advanced helicopter design, with predictions of improved weight, cost, and aircraft performance.

Mr. L. Thomas Mazza, Technology Applications Division, served as project engineer, with Mr. E. Rouzee Givens directing the "Free Planetary Transmission Drive" study portion of the program.

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A preliminary design study for a complete helicopter, incorporating advanced structural design concepts and advanced material, was conducted by the Boeing Vertol Company. The purpose was to evaluate the impact of advanced concepts and materials on the payload and configuration of a medium utility transport helicopter when compared to a baseline helicopter of conventional design. Major structural systems were identified with assessment of structural efficiency, cost/producibility, fail-safety, safety, reliability,

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20. ABSTRACT

maintainability, survivability, and crashworthiness. In addition, risk and feasibility assessment of the advanced structural systems was conducted.

A

SUMMARY

This preliminary design study was conducted by the Boeing Vertol Company for the Eustis Directorate, USAAMRDL. The purpose was to evaluate the practical impact of advanced structural concepts and advanced composite materials on a medium-range utility tactical transport helicopter configuration with specific payload, mission, and design requirements typical of modern (1974) procurement standards. Structural efficiency and producibility/cost were emphasized.

A baseline metal helicopter was designed which met the specifications. Sensitivity studies identified major structural systems in which improved structural efficiency would have significant impact on vehicle size and performance. These key systems were studied and conceptual designs traded for structural efficiency, fail-safety, safety, cost/producibility, reliability, maintainability, survivability, crashworthiness, and detection avoidance, as well as general specification conformance.

A free planet transmission concept was studied in some depth for applicability to this vehicle, and detailed results are reported in Volume II. An advanced structure helicopter was developed for the same mission and gross weight as the baseline, utilizing selected system concepts. A resized helicopter was also configured with the same mission and payload as the baseline, but taking advantage of the efficiency of advanced materials systems.

The preliminary design investigation resulted in an advanced configuration with a 15-percent reduction in gross weight, a 15-percent reduction in fuel required, and a 16-percent reduction in rotor disk area compared to the baseline at competitive production acquisition costs, without excessive development risk.

The conclusion from this study is that advanced structural concepts and materials technology are at a stage of development wherein a reasonable level of applied development activity can lead to demonstration and the introduction of significant improvements into U. S. Army helicopter systems.

PREFACE

This document is Volume I of the final report on the results of a preliminary design exercise entitled Advanced Helicopter Structural Design Investigation; Volume II is USAAMRDL Technical Report 75-56B, Design Application Study for Free Planet Transmissions. The program was conducted by the Boeing Vertol Company for the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, under Contract DAAJ02-74-C-0066, from June 1974 through May 1975.

The work includes definition of a state-of-the-art aluminum baseline medium range utility helicopter, redesign in advanced composites with advanced structural subsystems, and resizing of the advanced helicopter to perform the identical mission of the baseline helicopter.

Technical direction was provided by Mr. L. Thomas Mazza, with the free planetary transmission drive study directed by Mr. E. Rouzee Givens, both of the Eustis Directorate, USAAMRDL.

The study was conducted at the Boeing Vertol facility in Ridley Park, a suburb of Philadelphia, Pennsylvania. The principal Boeing contributors were Donald Hoffstedt, Program Manager; Sidney Swatton, Airframe Design; John Mack and William Rumberger, Transmission Design; Erwin Durchlaub, Structural Analysis; Frank Sauter, Cost Engineering; Arling Schmidt, Weights Analysis; Robert Pinckney, Manufacturing Technology; David Harding, R&M, Survivability/Vulnerability; John Schneider, Preliminary Design; and Harold Rosenstein, Performance and Sizing.

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1. INTRODUCTION

1.1 OBJECTIVES

The objectives of this study were to:

- Define advanced structural configurations using the latest analytical, material and fabrication technology to satisfy requirements of structural efficiency, fail-safety, safety and producibility/cost.
- Conduct a risk/feasibility assessment of advanced structural concepts to determine the areas of greatest payoff and to define potential high technical risks.
- Identify supporting research required to achieve the necessary advanced structural technology, as determined from preliminary designs and associated analysis.

For the study, Eustis Directorate, USAAMRDL, supplied a Specification for an Advanced Structures Study (SASS) for a Medium-Range Utility Transport Helicopter (MUT). This specification (Appendix A) formed the basis for preliminary conceptual designs of the aircraft to meet specific mission capabilities.

To meet the stringent standards of the SASS, the initial task included sizing a modern baseline metal aircraft structure and subsystem, and providing a standard against which advanced structures could be evaluated. When the baseline configuration was determined, each of the various subsystems was studied and conceptualized both as components and as systems, to determine candidate components/systems which might decrease the structural weight of the airframe system (fuselage, controls, rotor hub, transmission, landing gear, drive shafts, etc.) while meeting the basic requirements of the SASS. These concepts were screened, evaluated, and reviewed with USAAMRDL.

The second task included a more detailed comparison of structural concepts, rating of viable options, and recommendation to USAAMRDL of the most promising advanced concepts. The selected configuration was further defined, analyzed, and designed insofar as the conceptual nature of the study required.

The third task involved a comparison of weight and mission performance between the baseline and advanced configuration while maintaining common geometry. A further comparison was performed after resizing the advanced structures configuration to perform the identical mission of the baseline configuration.

The final task involved assessing risk and feasibility of the selected advanced structural design and identification of the highest technical risk areas plus additional supporting research requirements necessary to effectively implement advanced structural concepts.

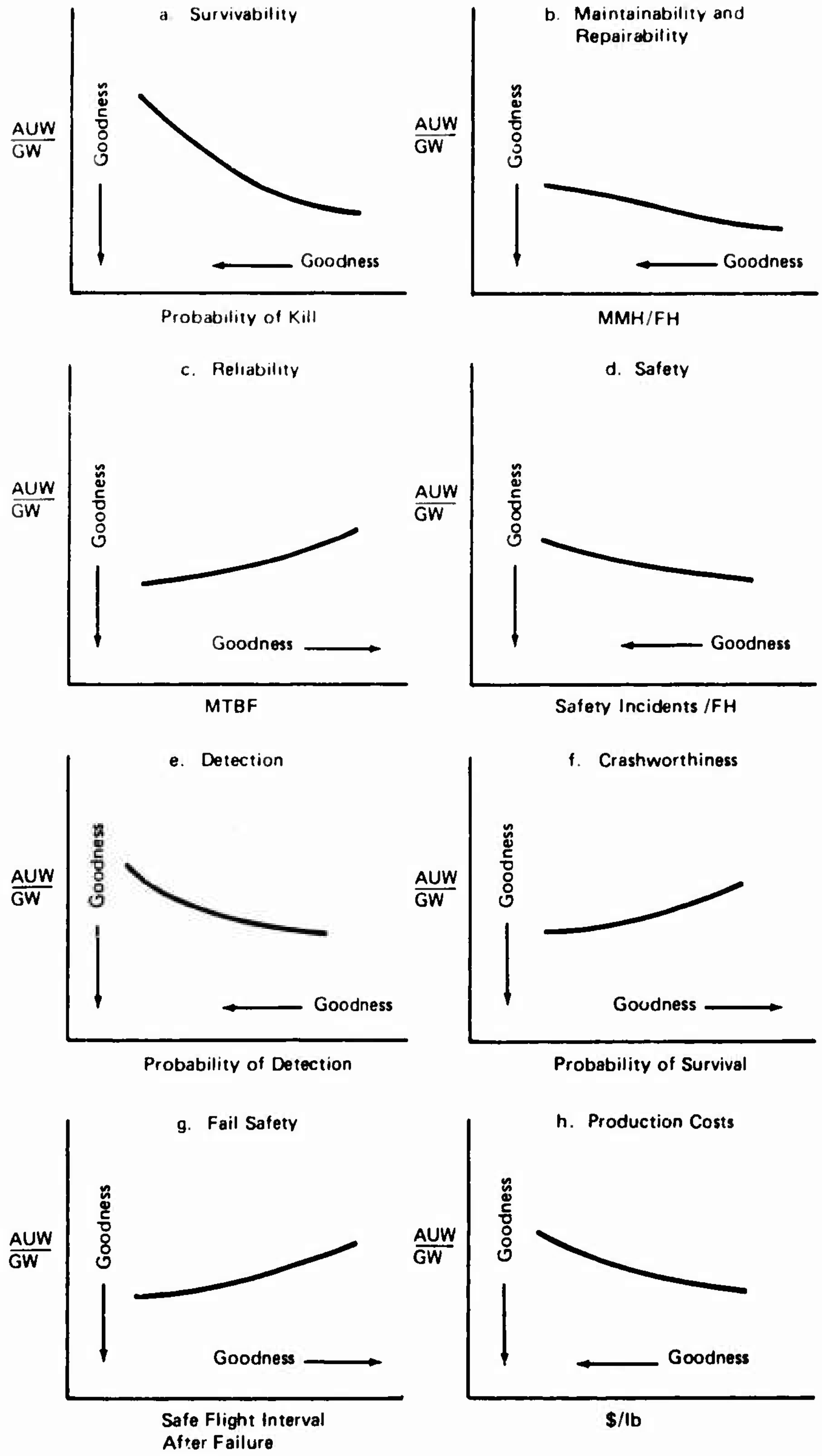
Most of the detailed concept evaluation centered around the fuselage, which was identified as the maximum pay-off structure for improvement of payload/gross-weight ratios. All systems were considered, however, in identifying potential advanced concepts and evaluating their relative impact on the final weight and cost of the aircraft.

1.2 DEVELOPMENT OF PRELIMINARY DESIGN CONCEPTS

The sizing and preliminary design of a baseline modern state-of-the-art helicopter was performed using HESCOMP, the Helicopter Sizing and Performance Computer Program developed for NASA by Boeing Vertol Company under Contract NAS2-6107. The utility designation with litter loading requirement, the requirement for transportability in a C-130 and a C-141, the nap-of-the-earth maneuverability, and the hot-day hover requirement sized the cabin, the rotor height, the tail rotor and the main rotor, respectively. The reliability, maintainability, survivability, and vulnerability requirements, plus the specified maneuver load factor of 3.50, and requirements for fail-safety and design-to-cost, forced a reevaluation of historical weight trend curves upon which HESCOMP is based. The differences between a baseline helicopter derived from traditional weight trends and a baseline helicopter reflecting modern design practices (see SASS, Appendix A), are presented in Table 1. Qualitative impacts of control factors on structural weight are shown in Figure 1.

TABLE 1. MUT BASELINE AIRCRAFT PRELIMINARY DESIGN COMPARISON

Parameter	Historical Design Prediction	Effect of Modern Design Requirements
Gross Weight (lb)	8,477	9,544
Weight Empty (lb)	5,583	6,431
Rotor Diameter (ft)	36.7	38.9
Installed SHP	1,843	2,065
Mission Fuel (lb)	1,437	1,655
Payload (lb)	960	960
V_{CR} (Kt) at S/L Std	150	[illegible]
ROC (fpm) from HOGE, 4,000 ft, 95°	450	450
Mission Endurance (hr)	2.3	2.3



(1) Airframe Unit Weight Definition may be found in MIL-STD-1374.

Figure 1. Control Factor Effects on Airframe Unit Weight[1]/ Gross Weight for Conventional Design (Qualitative).

2. BASELINE HELICOPTER

The HESCOMP sizing results for the MUT baseline aircraft, with weight trend corrections based on Army UTTAS design experience (reflecting procurement requirements similar to Appendix A), are presented in Tables 2 and 3. A baseline design description follows the tabulations.

TABLE 2. BASELINE AIRCRAFT SIZE

Length (Body and Tail Boom)	40.25 ft
Length (Cabin)	6.0 ft
Length (Body)	19.8 ft
Length (Tail Boom) Incl Stinger	20.3 ft
Main Rotor Location	12.3 ft
Cabin Width Outside	8.0 ft
Horizontal Tail	
Aspect Ratio	4.28
Area	21.1 sq ft
Span	9.5 ft
Mean Chord	2.22 ft
Taper Ratio	0.66
Thickness/Chord	0.15
Vertical Tail	
Aspect Ratio	1.722
Area	21.6 sq ft
Span	5.67 ft
Mean Chord	3.3 ft
Taper Ratio	0.43
Thickness/Chord	0.23
Main Rotor	
Diameter	38.9 ft
Solidity	0.100
Disc Loading	8.0 psf
Number of Blades	4
Blade Twist	-12.0 deg
Cut-out/Radius Ratio	0.230
Tip Speed	750 fps
Tail Rotor	
Diameter	7.8 ft
Solidity	0.227
Net Disc Loading	13.8 psf
Number of Blades	4
Blade Twist	-9.0 deg
Blade Cut-out/Radius Ratio	0.250
Tip Speed	700 fps
Main Rotor/Tail Rotor Gap	0.5 ft

TABLE 3. PRELIMINARY DESIGN BASELINE WEIGHTS

Item	Item Weight (lb)	Total Weight (lb)
Propulsion Group		
Main Rotor Group		
Rotor Blades (4)	563	
Rotor Hub	364	
Drive System	1047*	
Primary Engines (2)	468**	
Engine Installation	186	
Fuel System	190	
		2818
Structures Group		
Horizontal Tail	42	
Tail Rotor	56	
Fuselage (includes Vertical Tail)	1067	
Landing Gear, Nose	68	
Landing Gear, Main	217	
Landing Gear, Tail Bumper	11	
Engine Section	135	
		1596
Flight Controls Group		
Cockpit Controls	67	
Main Rotor Controls, Lower	263	
Main Rotor Head Controls	178	
Horizontal Stabilizer Controls	19	
Stability Augmentation System	35	
		562
Weight of Fixed Equipment		1455
	Weight Empty	6431
	Fixed Useful Load	498
	Operating Weight Empty	6929
	Payload	960
	Fuel	1655
	Gross Weight	9544

*Main rotor drive system rating is 1604 hp. Tail rotor drive system rating is 182 hp.

**Power required for baseline helicopter, sized for takeoff at 4000 ft density altitude and 95°F, and 450 fpm vertical rate of climb (at T/W=1.03, both engine operative) is 2065 max standard SL static shaft horsepower.

2.1 MUT BASELINE DESIGN DESCRIPTION

The baseline MUT aircraft, designed to comply with the specification for an advanced structures study for medium-range utility transport helicopter (Appendix A), is shown in Figure 2.

The design features a single main-rotor system employing the hingeless composite rotor blade concept and is powered by twin advanced-technology engines. The aircraft incorporates modern state-of-the-art structure and is supported on a tricycle landing gear with the addition of an attenuating tail bumper.

The pilots' compartment accommodates a crew of two. The cabin width and height is sized for seven passengers and will meet the specified mission requirements of four combat equipped troops and a crew of two. The cabin width is sized for four troops or three litters placed laterally (see Figure 3). The internal cross section dimensions resulting from the litter requirements are the same as those of the current YUH-61A (UTTAS); hence, the crew compartment and the cabin width and height are practically identical.

The aircraft missions include aeromedical evacuations, and the transport of special teams and/or equipment or supplies. An external hook is provided to transport oversized loads up to 2000 pounds (see Figure B-1 in Appendix B).

2.2 GENERAL ARRANGEMENT

The aircraft fuselage consists of three sections: cockpit area, mid-fuselage area, and the tailboom/empennage.

The cockpit arrangement provides the crew with maximum crash protection, good visibility in all directions, normal ingress and egress through hinged jettisonable side doors, emergency egress through an overhead window, shatterproof windshields, and windshield wipers. Flight controls, avionics, and nose landing gear are also located in and around the cockpit area.

The mid-fuselage area contains the troop/cargo compartment, fuel system, and the equipment bays. Floor and ceiling attachments for troop seats, litters, and cargo are provided in the cabin area. This section absorbs loads imposed by the engines, main rotor transmission, and other components of the dynamic rotor system, landing gear, and tailboom. The cabin area is enclosed by two doors on each side. A forward hinged door is used for litter loading and an aft sliding door for troop ingress and egress. With both doors open, a width of 50 inches is provided for loading cargo.

The self-sealing fuel cell is located just aft of the cabin

Figure 2. Medium-Range Utility Transport Helicopter (MUT).

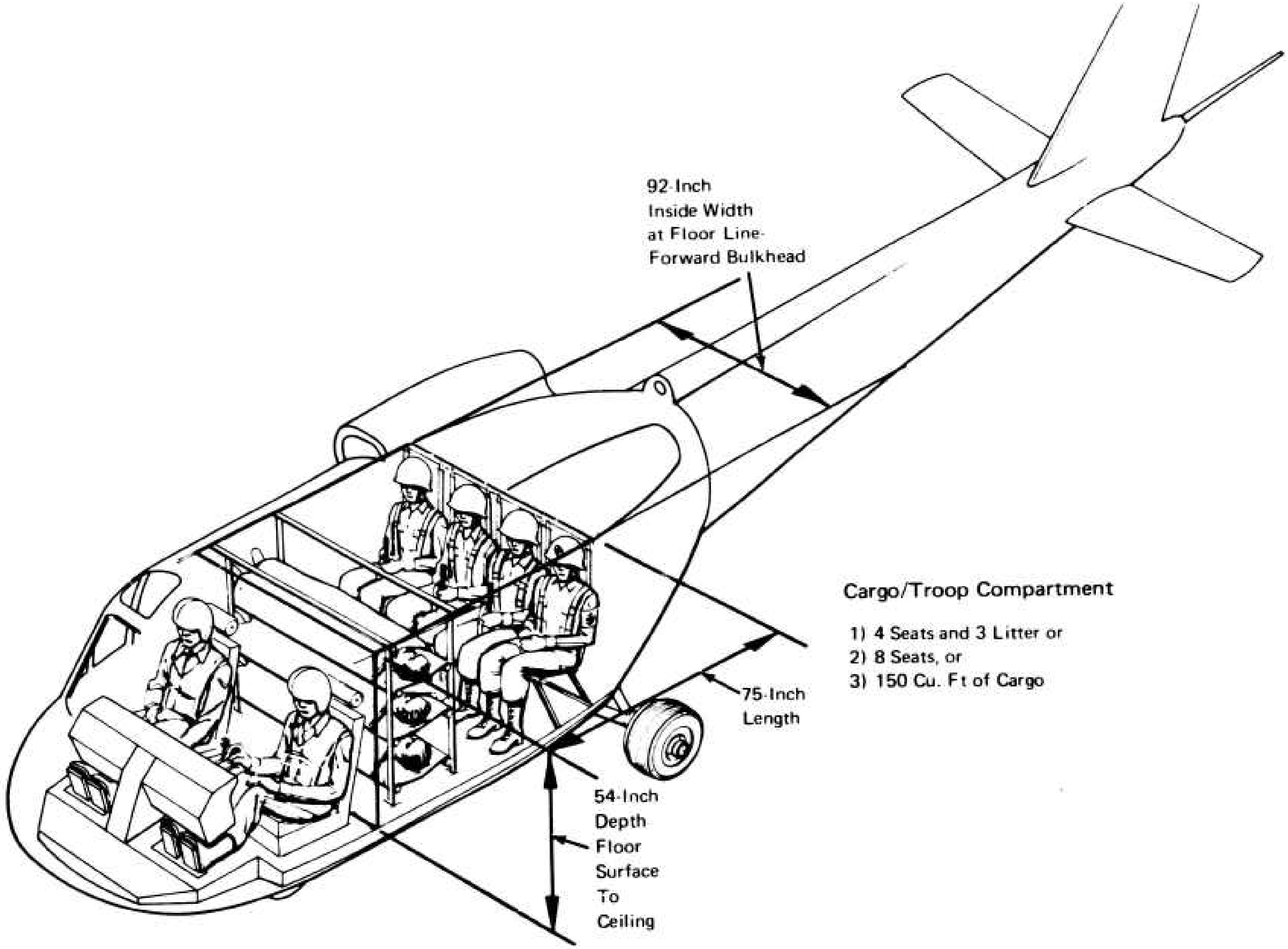


Figure 3. MUT Cockpit/Cabin Compartment Mission Loading Arrangement.

and is completely enclosed by airframe structure. The electronic equipment compartments with large access doors are located on each side of the aircraft, outboard and adjacent to the fuel cell compartment and just aft of the main landing gear well. The engines, main rotor transmission, and other components of the dynamic rotor system are located above the cabin ceiling.

The tailboom supports the tail rotor, shafting, gearboxes, tail bumper, and vertical and horizontal stabilizers (see Figure 4).

2.3 AIRFRAME BASELINE DESIGN

The primary structure is a modified semimonocoque construction consisting basically of aluminum alloy skins, stringers, and frames. Stainless steel and/or titanium is used where feasible (e.g., in firewalls and fittings). Generous use is also made of fiberglass, bonded honeycomb, and other composite materials in secondary structure (doors, fairings, etc.).

The cockpit area structure is arranged to provide good structural continuity with the mid-fuselage section for crash protection. Hard points are provided for attaching the nose gear. The cockpit section is spliced to the mid-fuselage section at sta 78.

The mid-fuselage structure, above the cabin ceiling, consists of buttline longitudinal beams and built-up torque boxes which extend almost the full length. This rugged structure supports the engines, main rotor transmission, and all other components of the dynamic rotor system. Hardpoints are also provided to accommodate the main landing gear and a removable cargo hook in the aft end of the cabin area. The cargo operator's station is adjacent to the hook access opening in the floor. The floor structure consists of lateral floor frames and longitudinal beams, allowing continuity of structure from the fuselage nose to tailboom and accommodating the floor loading requirements.

The tailboom is shaped to provide an effective box structure required for the tail rotor, empennage, and tail bumper. The tailboom and the mid-fuselage section have a field splice between them at sta 239.

The vertical stabilizer supports the tail rotor, gearbox, and shafting. Construction is basically aluminum alloy two-spar, rib, skin, stringer type. The spars extend into the tailboom and are mechanically attached.

The variable incidence horizontal stabilizers are the same type construction as the vertical stabilizer. Each stabilizer

Figure 4. MUT Baseline Structural Arrangement.

has a lug located inboard on the front spar for attachment to a common torque tube through the tailboom (see Figure B-2 in Appendix B).

2.4 LANDING GEAR

The landing gear is a tricycle type with the main gear attached to the aft side of the cabin aft bulkhead. The nose gear is attached beneath the cockpit section.

Each main gear consists of a two-stage oleo having kneeling capability. These features also permit survival in a 95th percentile crash by absorbing energy prior to structural deformation of the airframe.

The nose gear is a single, 360-degree swiveling, nonretractable oleo strut with dual wheels. A viscous shimmy damper and swivel lock are incorporated (see Figure 5).

The attenuating tail bumper is installed to protect the tailboom from structural damage during high-angle flared landings (vertical impact capability of 18 fps).

Ground steering is accomplished by the tail rotor and differential braking of the main wheels.

2.5 PROPULSION SYSTEM

The propulsion system includes two new advanced technology engines with particle separators in the air induction system, exhaust system including infrared suppression, engine compartment cooling, compressor bleed air/pneumatic system, engine mounting with isolation units, propulsion system controls and instrumentation, fuel system (see Figure 6), lubrication system, fire detection and extinguishing system, also the nacelle group, comprising fixed and hinged segments of engine cowling (see Figure 7).

The air-induction subsystem consists of a semiannular inlet and an aerodynamic-shaped shroud which houses the nosebox transmission.

The exhaust subsystem consists of a titanium tailpipe and a titanium ejector shroud to provide ample engine compartment cooling.

The infrared suppression assembly is readily installed upon removal of the tailpipe and ejector shroud.

Engine bleed air provides anti-icing protection, and heating and ventilation for the nacelle inlets.

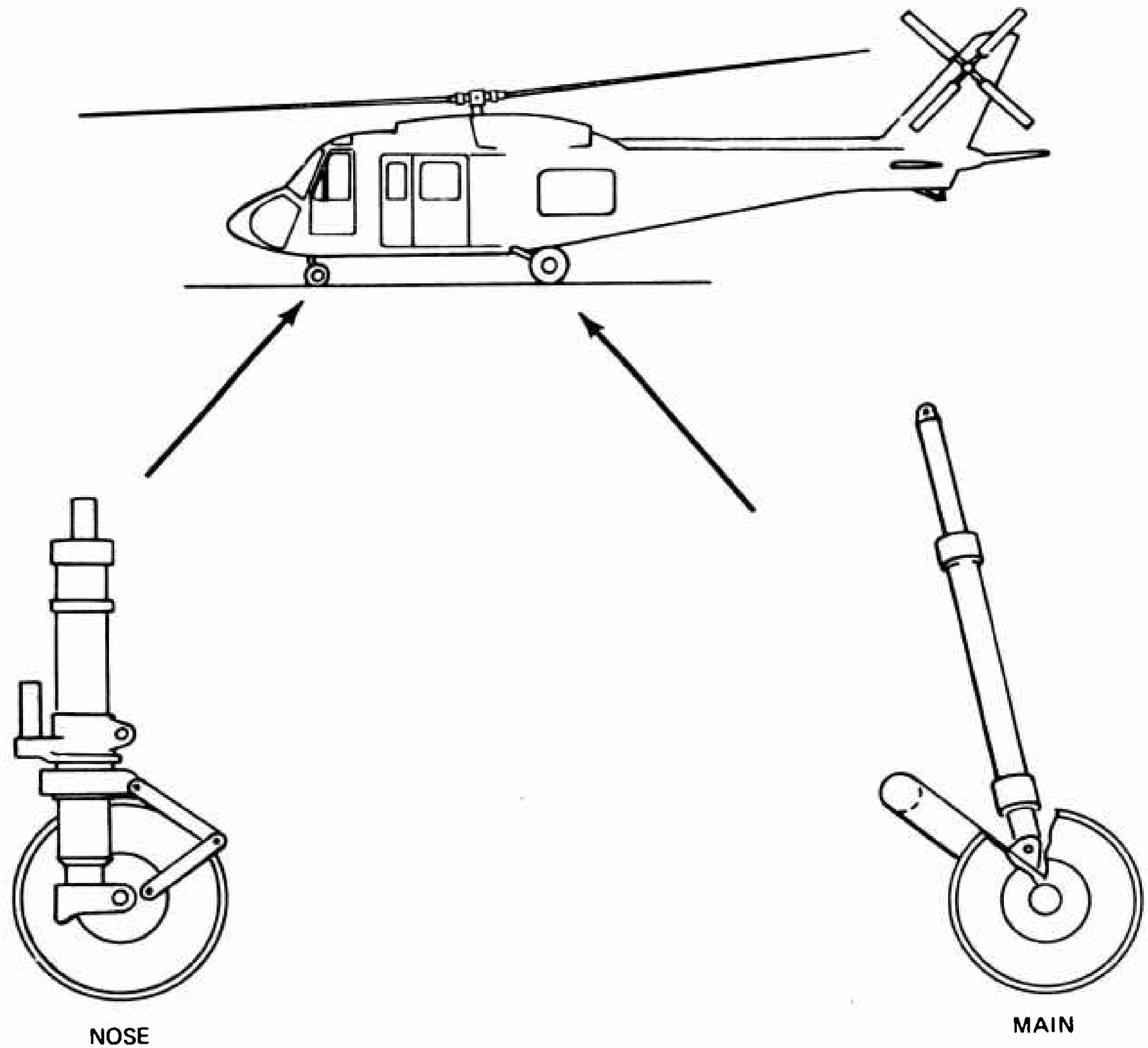


Figure 5. MUT Landing Gear.

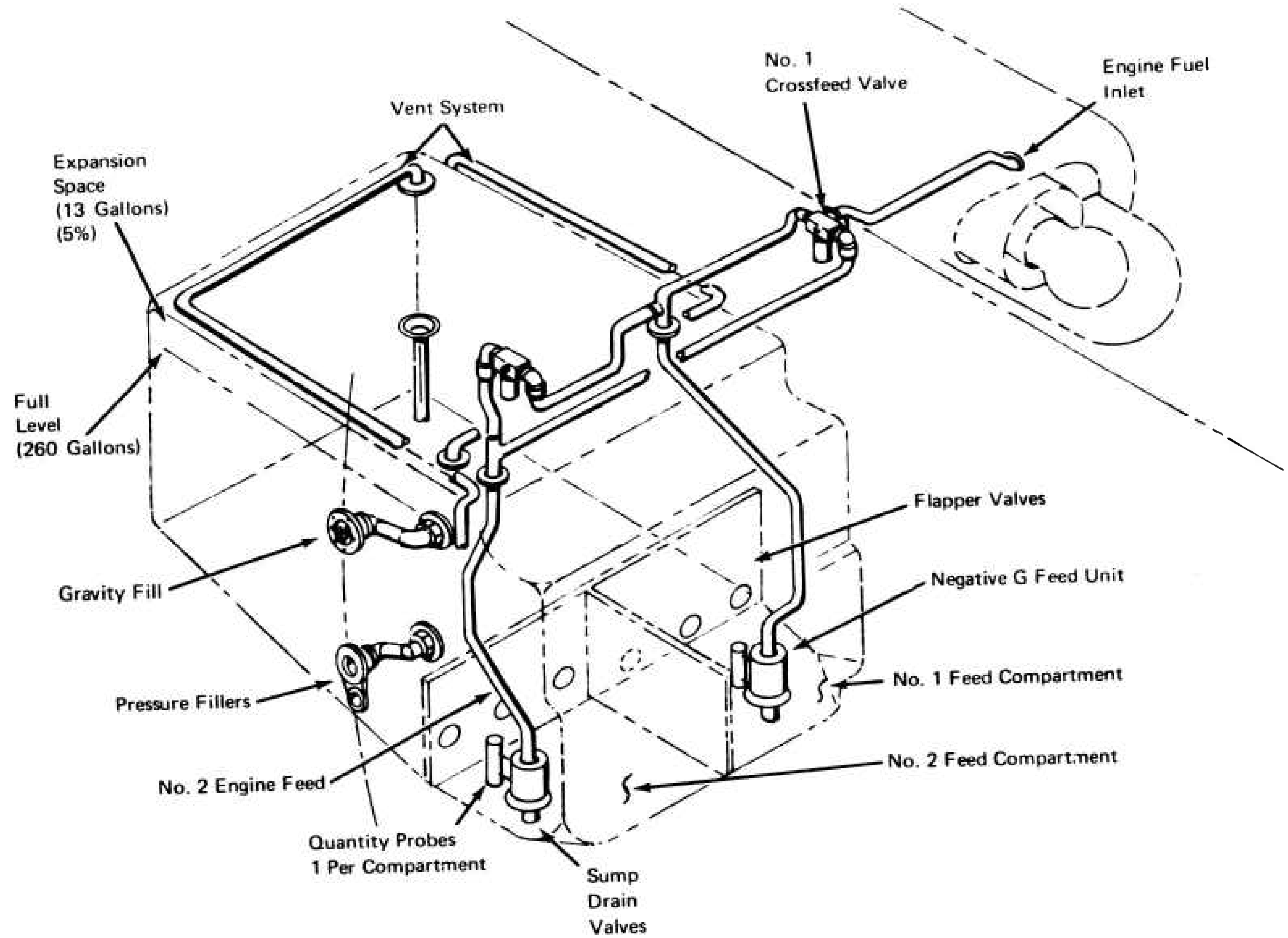


Figure 6. Aircraft Fuel System.

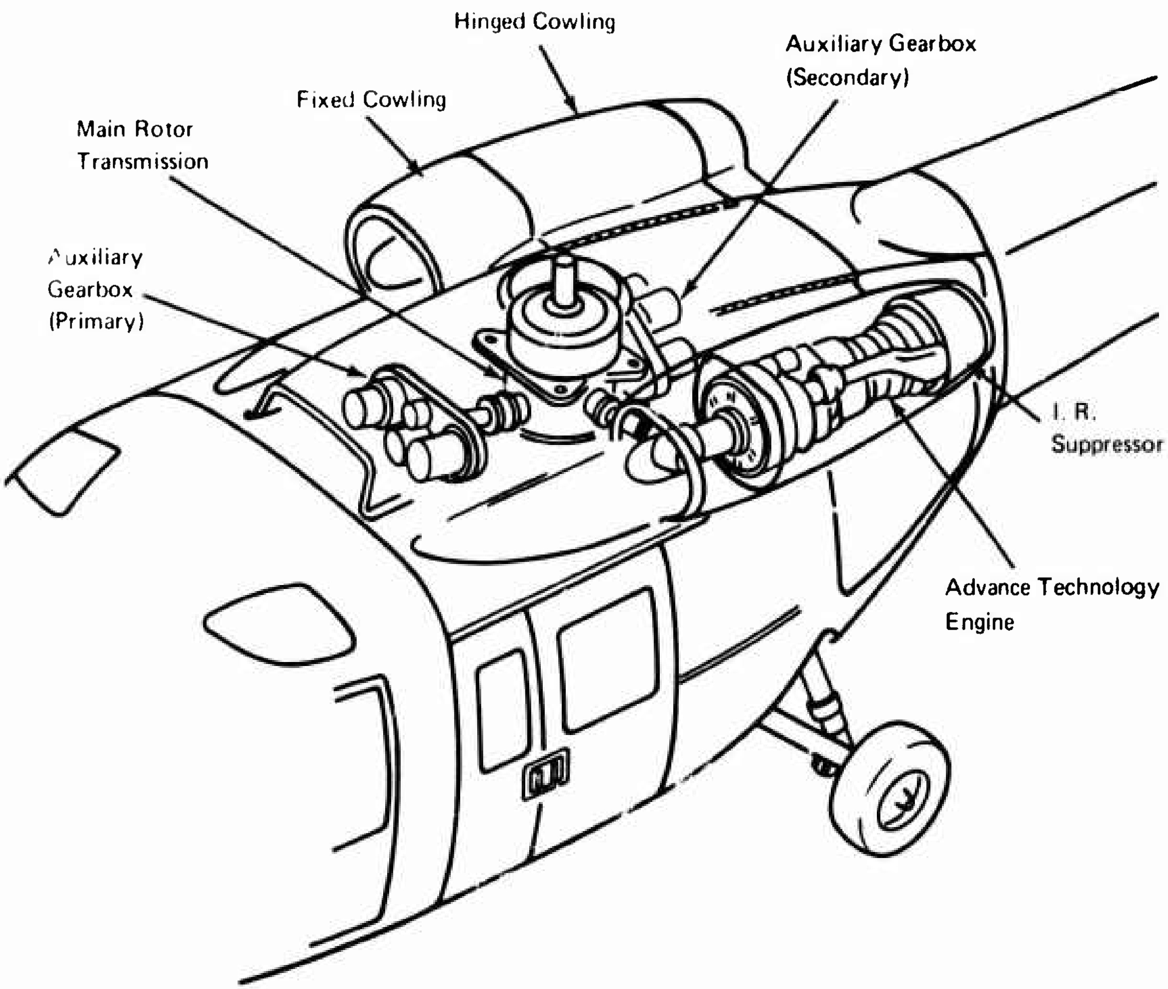


Figure 7. Propulsion and Drive System Arrangement.

The fuel system consists of a single self-sealing, crash survivable fuel tank and supply lines designed for ballistic protection. Independent collector tanks, integral with the main fuel tank, feed each engine. Normal feed and crossfeed are accomplished by suction pumps on each engine. Aircraft refueling is accomplished on the ground from easily accessible locations. Frangible fittings to tank attachments and self-sealing breakaway fittings are typical throughout.

The fire detection and extinguishing system consists of sensing devices in each engine. Two extinguishing bottles are provided with discharge selectivity from the cockpit into the firebox areas.

2.6 DRIVE SYSTEM

The drive system consists of two-engine right-angle-nose, main rotor, intermediate, and tail rotor transmissions; accessory gearboxes; and interconnecting sectionalized shafting. All shafting, except that from the engine to the nose transmission, is aluminum alloy tubing with flexible steel couplings between sections (see Figure 8).

The lubrication system for the main rotor transmission consists of a primary and a backup system. The primary system normally supplies cooling oil to the generator (on the aft accessory gearbox), bearings, gears, and return to reservoir. A sectionalized oil cooler is located on the accessory section of the main rotor transmission. In an emergency, the backup system supplies oil to critical bearing and gear meshes only, thereby limiting flight time. Engine oil is also cooled through the sectionalized cooler on the accessory section. The forward accessory gearbox and the intermediate and tail-rotor transmissions have completely integral air-cooled systems and require no separate coolers. The advanced technology engines may incorporate a completely self-contained lube system.

2.7 FLIGHT CONTROL SYSTEMS

Flight control of the aircraft is accomplished by a redundant mechanical system coupled with inputs from a redundant SCAS (Stability Control Augmentation System) to the hydraulic actuators controlling the main and tail rotors (see Figure 9).

The main rotor actuators impart motion to the nonrotating ring of the swashplate assembly. This motion is transferred to the rotating ring of the swashplate which provides pitch control to the rotor blades through pitch links.

The tail rotor actuator imparts motion to the rotating sliding sleeve on the tail rotor shaft. The sleeve transfers pitch control to the tail rotor blades through pitch links.

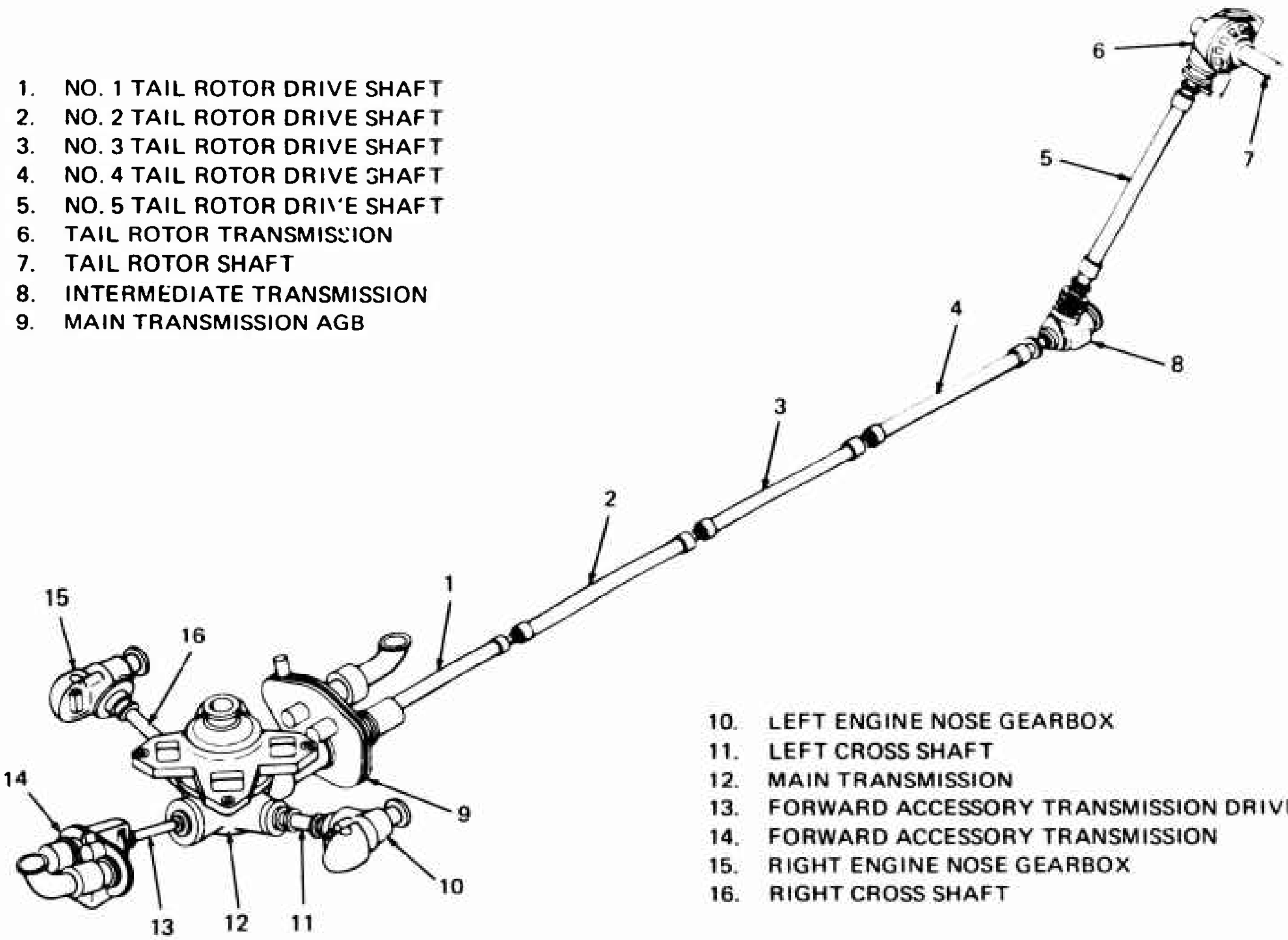


Figure 8. Drive System.

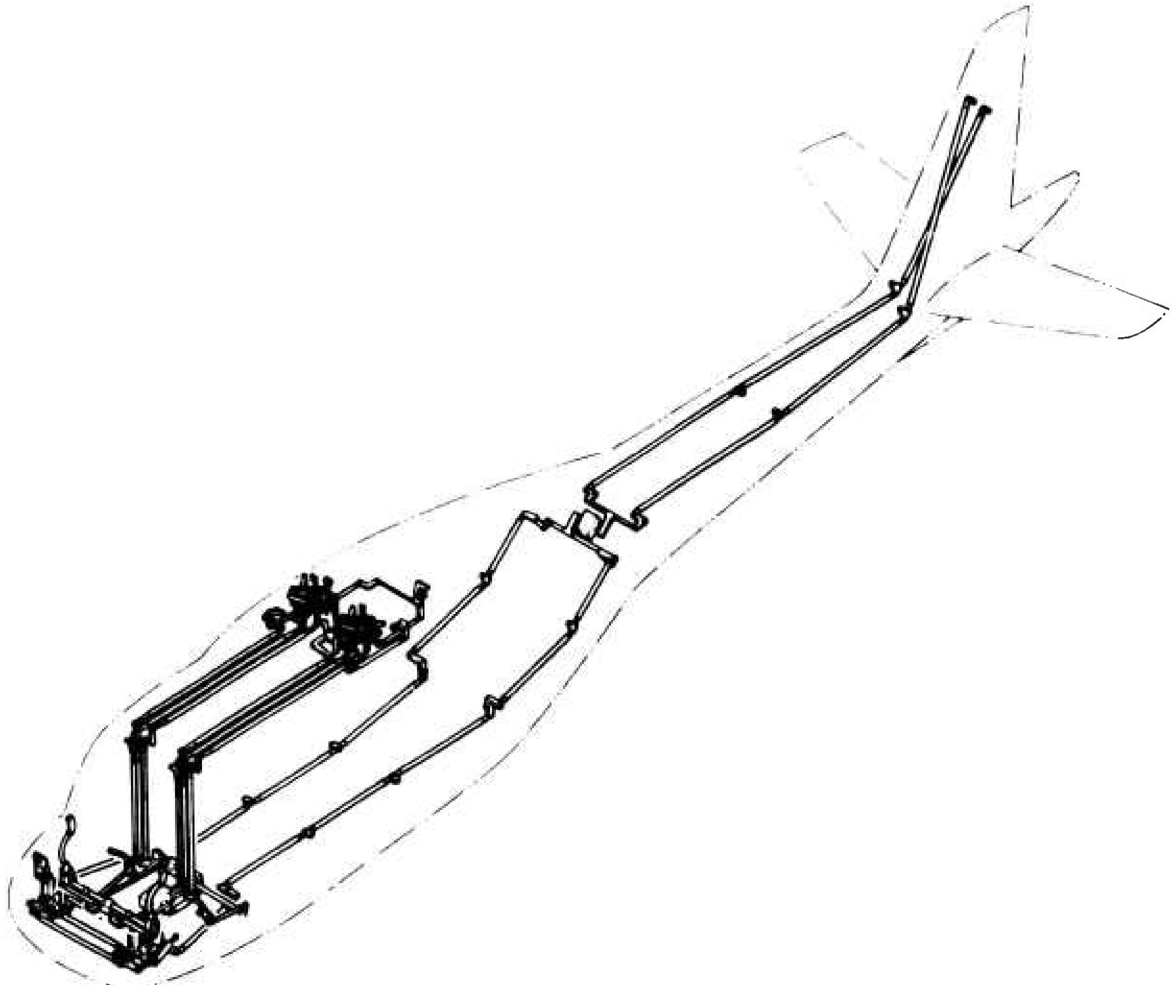


Figure 9. Dual Mechanical Flight Control System.

2.8 ELECTRICAL SYSTEM

The primary electrical power supply is provided by the ac generators, one on each accessory gearbox (AGB). One generator is capable of supplying the entire aircraft electrical power if necessary. The generators also provide for dc power by converting through transformer/rectifiers.

A 28-volt battery (located in the nose avionics compartment) is used for engine starting and is interlocked into the electrical system for emergency use.

2.9 AVIONICS

The avionics equipment for the aircraft provides fixed communications and tailored navigation capabilities to the crew. All avionics equipment is accessible for ease of maintenance. Most of the equipment is located in the fuselage nose and some in the bays in the sides of the fuselage just aft of the main landing gear well (see Figure 10).

The avionics equipment is listed in Table 4.

2.10 HYDRAULIC SYSTEMS

The utility hydraulic system operates at 3000 psi. It supplies power for kneeling/unkneeling of the main landing gear, and serves as an emergency source of hydraulic pressure for the flight control system. The system consists of an accumulator, an ac electric-driven hydraulic pump, a two-stage handpump, plus filters, relief valves, etc.

The flight control hydraulic system consists of two independent systems, with the utility system as an emergency backup in the event of dual system failure. Each system is completely separated from the other and consists of pump-cooler unit, hydraulic component module, accumulator, and associated hydraulic lines.

2.11 MAIN ROTOR BLADES

The aircraft's main rotor system consists of four hingeless blades. The inboard end of each blade is designed to provide the flexibility required for flapping and lead-lag motions. The blades are basically of composite structure, including a fiberglass D-spar, titanium root end fitting and leading edge, and a Nomex honeycomb core. Provisions are incorporated for blade lag damping, erosion and lightning protection, deicing and tuning. The blade design achieves the best balance between weight and strength (load paths and vibratory forces) and incorporates the Integral Spar Inspection System (ISIS) for failsafe operation (see Figure 11).

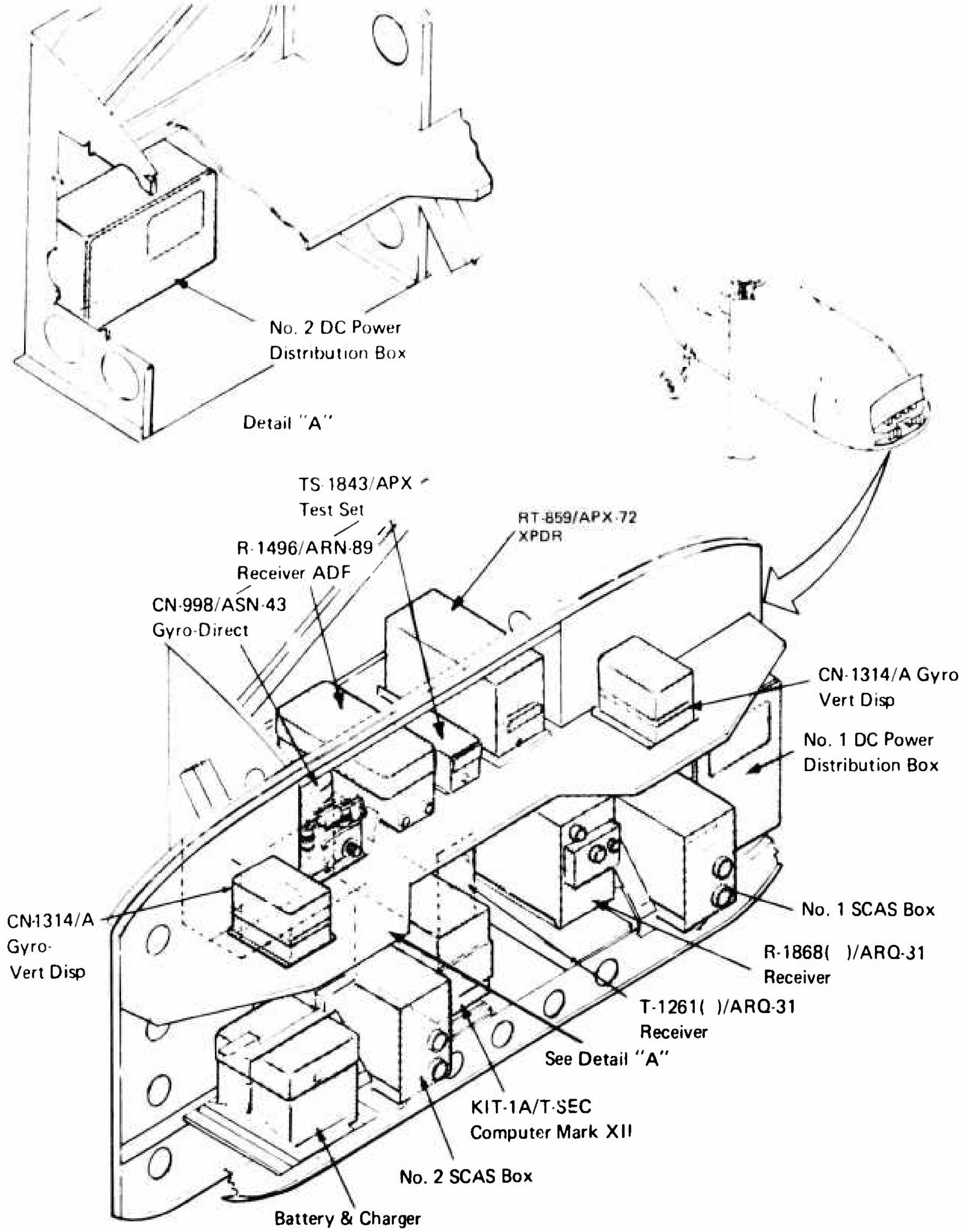


Figure 10. Equipment Installation - Nose Enclosure.

TABLE 4. AVIONICS EQUIPMENT

Qty per Aircraft	Description	Identification	Unit Weight (lb)
Communications			
2	VHF-FM Radio Set	AN/ARC-114	7.0
1	VHF-AM Radio Set	AN/ARC-115	7.2
1	UHF-AM Radio Set	AN/ARC-116	7.5
3	Interphone Control	C-6533/ARC	1.8
Automatic Direction Finder, AN/ARN-89			
1	Receiver, Radio	R-1496( )/ARN-89	6.8
1	Control, Radio Set	C-7392( )/ARN-89	3.1
1	Amplifier Impedance	AM-4959( )/ARN-89	0.2
1	Antenna, Loop	AS-2108( )/ARN-89	2.1
1	ADF Compensation Network		0.2
Gyro Magnetic Compass Set, AN/ASN-43			
1	Gyro, Directional	CN-998( )/ASN-43	5.5
1	Transmitter, Induction Compass	T-611( )/ASN	1.2
1	Compensator, Magnetic Flux	CN-405( )/ASN	0.2
Transponder Set, AN/APX-72			
1	Receiver-Transmitter	RT-859/APX-72	15.3
1	Control	C6280A(P)/APX	3.0
1	Mounting	MT3809/APX-72	1.7
Communication Security Set, TSEC/KY-29			
3	Communication Security Set	TSEC/KY-28	
3	Control Indicator Assembly	C-8157/ARC	
3	Mounting	MT-3802/ARC	
Auxiliary Equipment			
1	Transponder Test Set	TS-1843/APX	2.8
1	Mounting	MT-3513/APX	0.5
1	Computer, Mark XII	KIT-1A/TSEC	14.5
1	Mounting (Vibration Isolated)	MT3949A/U	1.5
VOR Radio Set, AN/ARN-82			
1	Receiver, Radio	R-1388/ARN-82	10.3
1	Control	C-6873/ARN-82	1.2
1	Mount	MT-3600/ARN-82	0.5
1	Tactical Landing System	AN/ARN( )	32.0
1	LORAN C/D Airborne Navigation System	AN/ARN( )	30.0
Glide Slope Market Beacon, AN/ARN-58			
1	Receiver, Radio	R-844/ARN-58	9.0

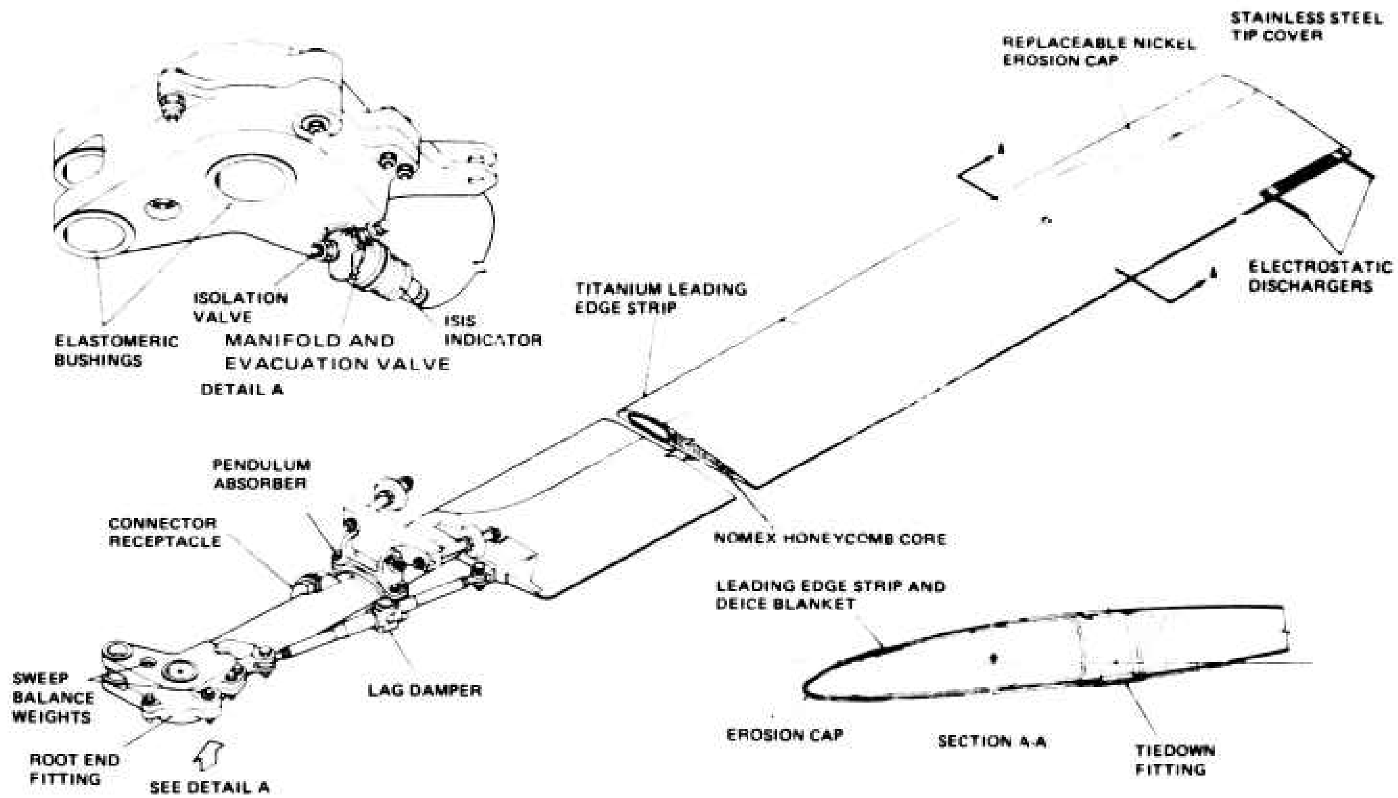


Figure 11. Main Rotor Blade.

2.12 TAIL ROTOR

The tail rotor consists of a tail rotor head, controls, and blades. The tail rotor head is driven by the torque transmitted from the tail rotor drive shaft, through the tail rotor transmission, to the tail rotor shaft (see Figure 8). The hub adapter is splined and fastened to the tail rotor shaft. A rotating sleeve around the tail rotor shaft with pitch links attached, transfers pitch control to the blades. The head consists of two short fiberglass flex straps mounted perpendicular to each end of the straps; the complete assembly is bolted to the hub adapter. The blade design consists of Nomex honeycomb and fiberglass with reinforcements where necessary. Deicing, erosion, and balancing provisions are incorporated (see Figure 12).

2.13 TRANSPORTABILITY

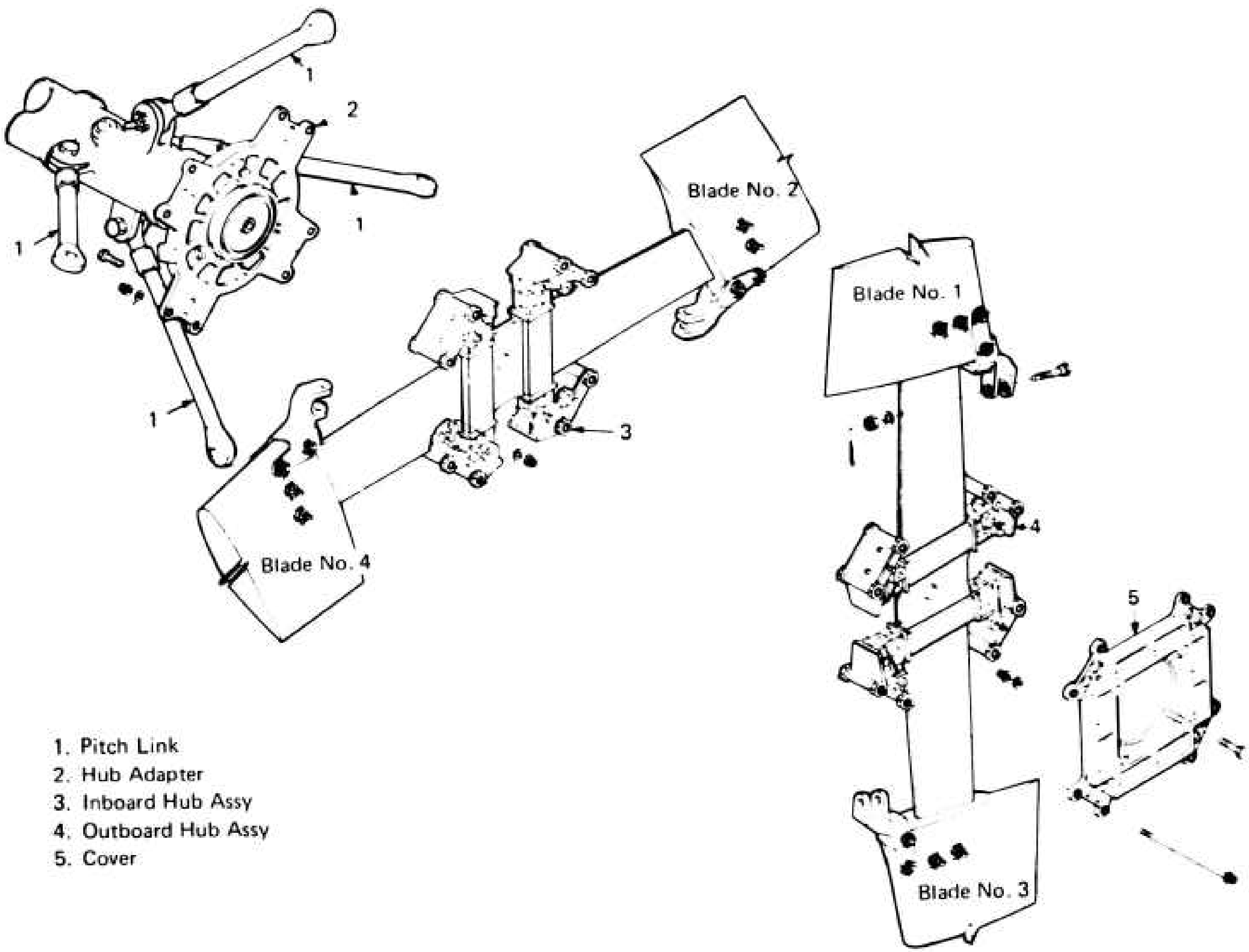
The C-141 aircraft is capable of transporting two MUT helicopters with at least a 6-inch minimum clearance. (See Figure B-6 in Appendix B.) The following tasks are accomplished before loading:

1. Fold main rotor blades.
2. Kneel main landing gear.
3. Remove tail rotor blades from the aft helicopter and reposition tail rotor blades on the forward helicopter.
4. Remove tips from the vertical and horizontal stabilizers.
5. Remove tailcones.
6. Remove tail bumper and fairing from aft helicopter only.

The C-130 aircraft is capable of transporting one MUT helicopter when the following tasks are accomplished:

1. Fold main rotor blades.
2. Kneel main landing gear.
3. Reposition tail rotor blades.
4. Remove tips from the vertical and horizontal stabilizers.

The C-5 aircraft is capable of transporting six MUT helicopters simply by folding the main rotor blades and kneeling the main landing gear.



1. Pitch Link
2. Hub Adapter
3. Inboard Hub Assy
4. Outboard Hub Assy
5. Cover

Figure 12. Tail Rotor.

3. ADVANCED CONCEPT DEVELOPMENT

3.1 AIRFRAME - ADVANCED STRUCTURAL CONCEPT CANDIDATES

Using the MUT baseline as the configuration, a sensitivity trade was performed to identify the maximum payoff areas for weight reduction. The results of this analysis are shown in Figures 13 through 17. Each major subsystem was examined by assuming weight reductions of 0, 10, 20, 30 and 40 percent with respect to the baseline. The "cascading" or multiplying effect by the single subsystem weight reduction on other baseline subsystems was accounted for in the exercise.

Fuselage structure and drive system weight reductions are significantly more rewarding than the other major areas. Since much of the weight in the drive system is gear steel and bearings, the fuselage was identified as the major payoff area for advanced materials.

The conceptual design effort on application of advanced materials to the MUT centers largely around the airframe structure. Improved structural efficiency of the drive system was approached through study of an advanced concept free planetary drive system (discussed later in this section). Other major subsystems were considered and a number of promising configurations were evaluated.

The candidates for each system were evaluated for conformance to design objectives on a point rating system as better than or poorer than the metal baseline design. For example, fuselage structural efficiency rating was approached as follows:

Strength/Weight (Structural Efficiency)

Specific factors in comparing primary structure concepts were:

- Continuity of load paths
- Number and location of joints
- Complexity of fittings
- Dual function load paths
- Minimum gauge inefficiencies
- Structural element size
- Number of cutouts
- Material structural efficiency
- Construction efficiency (special cases)

The results of the screening/selection process are presented in tabular form in the discussion of each major subsystem. Comparative rankings among advanced concepts were used in selecting the least cost approach, based on a common material system. Thus, the selections were reduced to such factors as: the number and complexity of structural components, the number of fasteners and assembly attachments, adaptability/utilization of production processes (such as filament winding, tape winding, broadgoods wrapping, automated tape layup, pultrusion), etc.

3.1.1 STRUCTURAL BREAKDOWN

For the purposes of this investigation, the fuselage was divided into major components or elements producible singly or in combination. Factors such as tooling, materials and process compatibility, field repair, transportabil..ty, replacement of damaged structure, etc., influenced major assemblies. The number of joints were kept to a minimum consistent with these factors.

Table 5 lists the components which are included in the advanced concept designs as well as the structural category of each component (primary or secondary).

TABLE 5. STRUCTURAL CATEGORY OF AIRFRAME MAJOR ASSEMBLIES

	Assembly	Type of Structure
1.	Cockpit enclosure	P
2.	Forward box frame assembly	P
3.	Upper deck and buttline beam assembly	P
4.	Floor panel and underflow structure	P
5.	Bulkhead and rear box frame assembly	P
6.	Fuel and electronics bay structure	P
7.	Tailboom assembly	P
8.	Vertical stabilizer assembly	P
9.	Horizontal stabilizer assembly	P
10.	Pilots door	S
11.	Side sliding door	S
12.	Side hinged door	S
13.	Engine fairing	S
14.	Tailboom drive shaft fairing	S
15.	Nose electronics door	S
16.	Side electronics door	S
17.	Tailcone	S
18.	Access panel, upper deck	S
19.	Tail skid	S
20.	Tail skid fairing	S
21.	Vertical stabilizer tip	S
22.	Transmission cooling duct	S
23.	Leading-edge fairing (vertical stabilizer)	S
24.	Transmission fairing	S
25.	Intermediate transmission box fairing	S
26.	Door track	S

P = Primary structure

S = Secondary structure

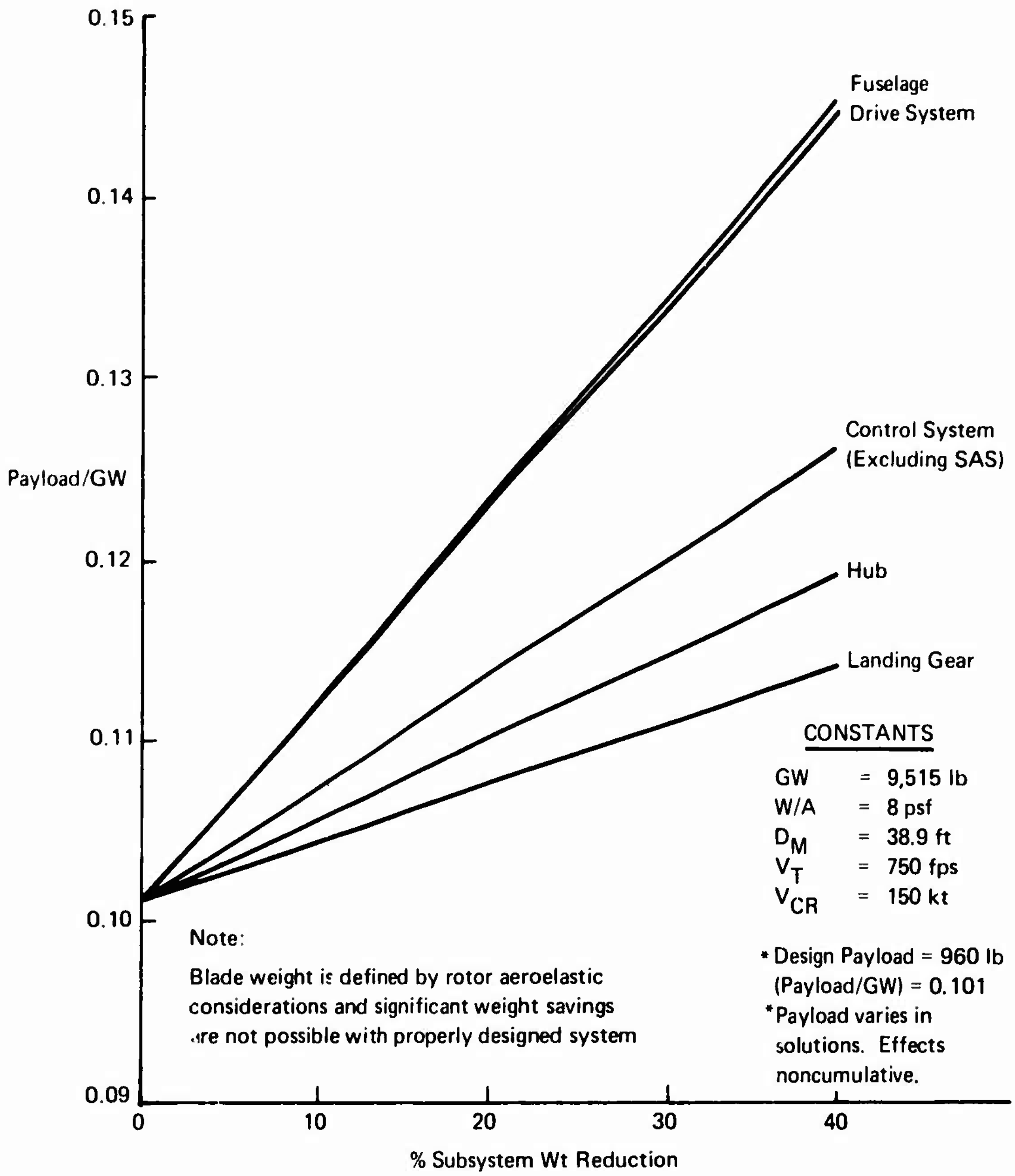


Figure 13. Payload/Gross Weight Variation With Subsystem Weight Reduction, No Resizing.

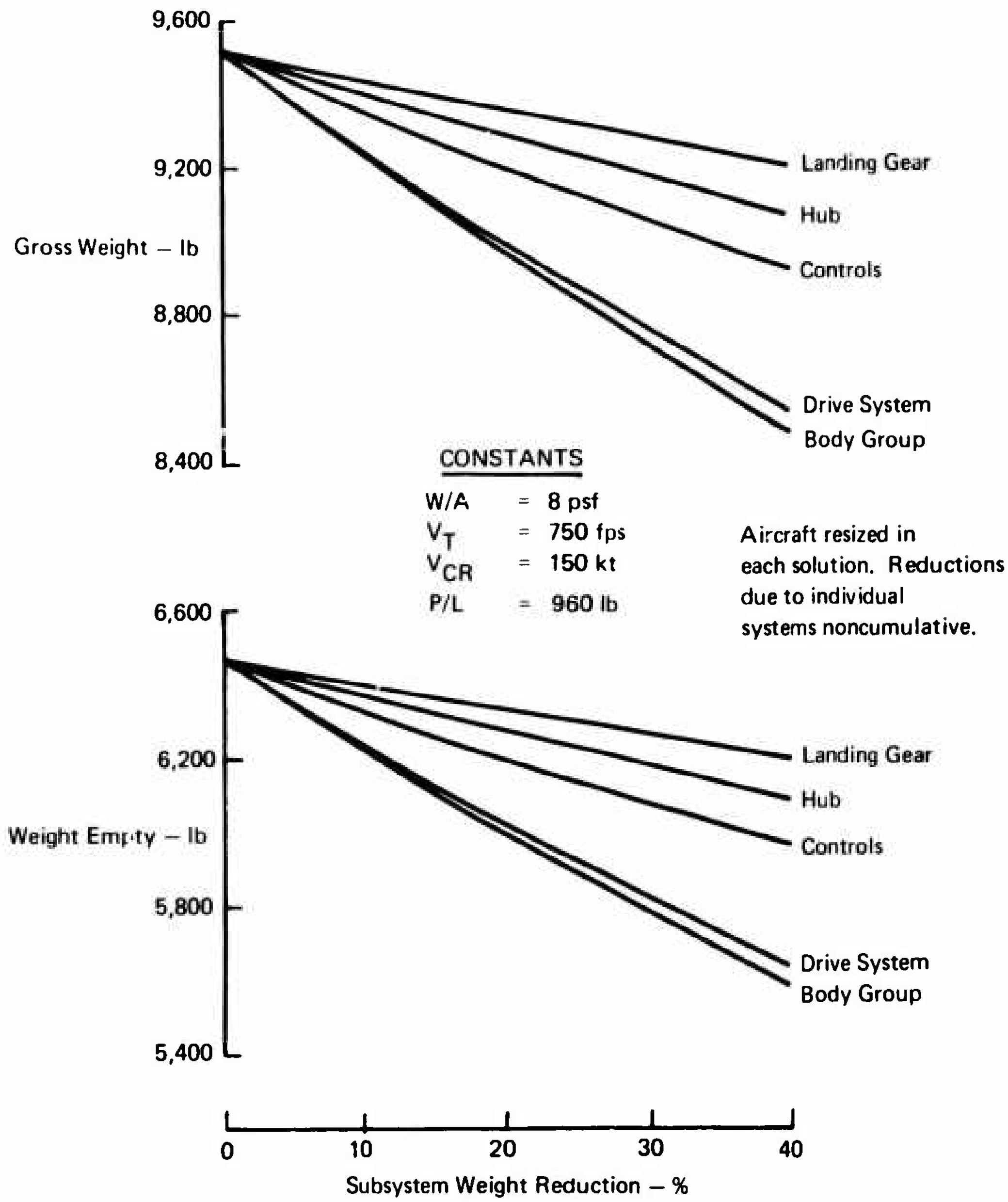


Figure 14. Effect of Subsystem Weight Reductions.

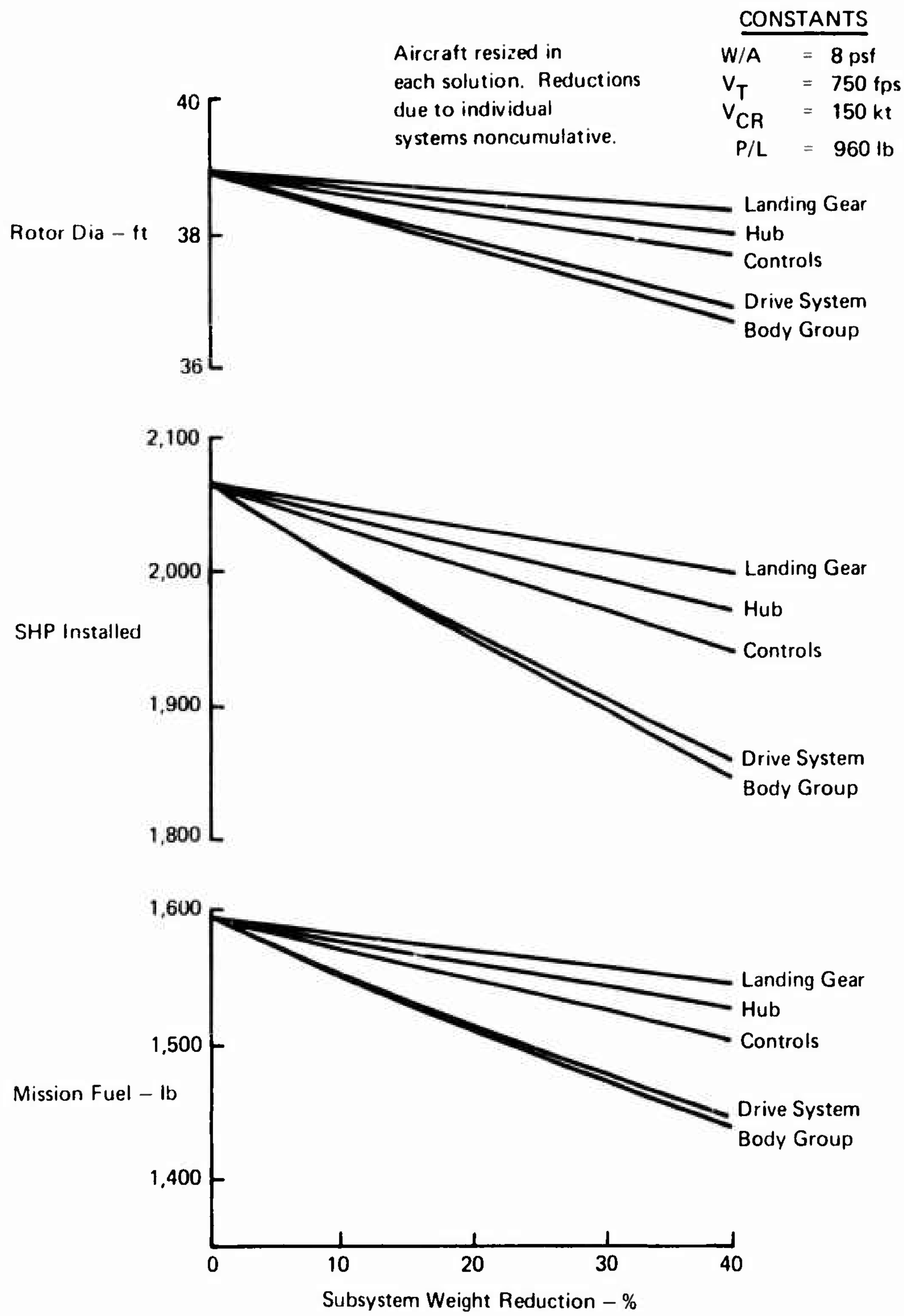


Figure 15. Effect of Subsystem Weight Reductions.

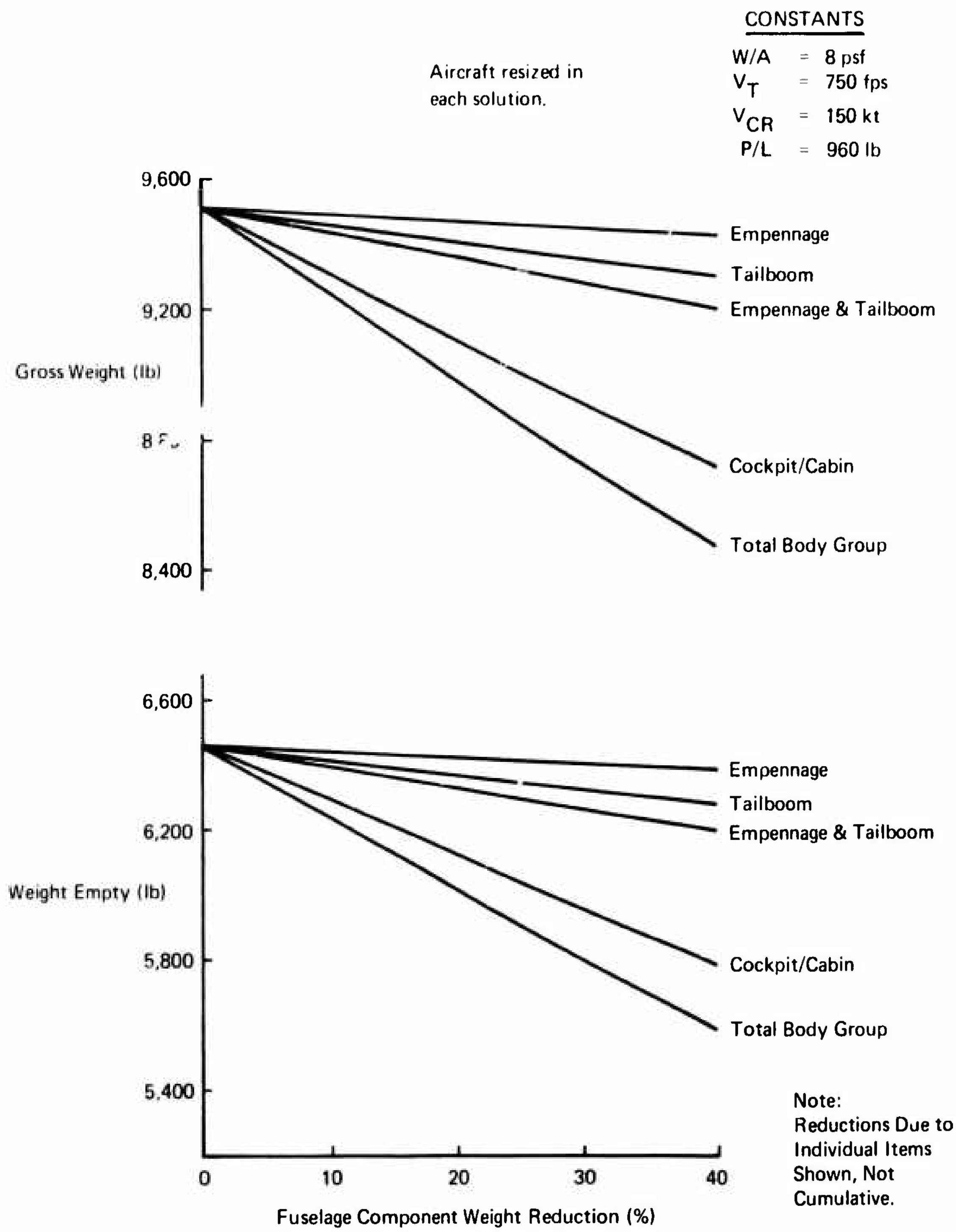


Figure 16. Effect of Fuselage Component Weight Reductions.

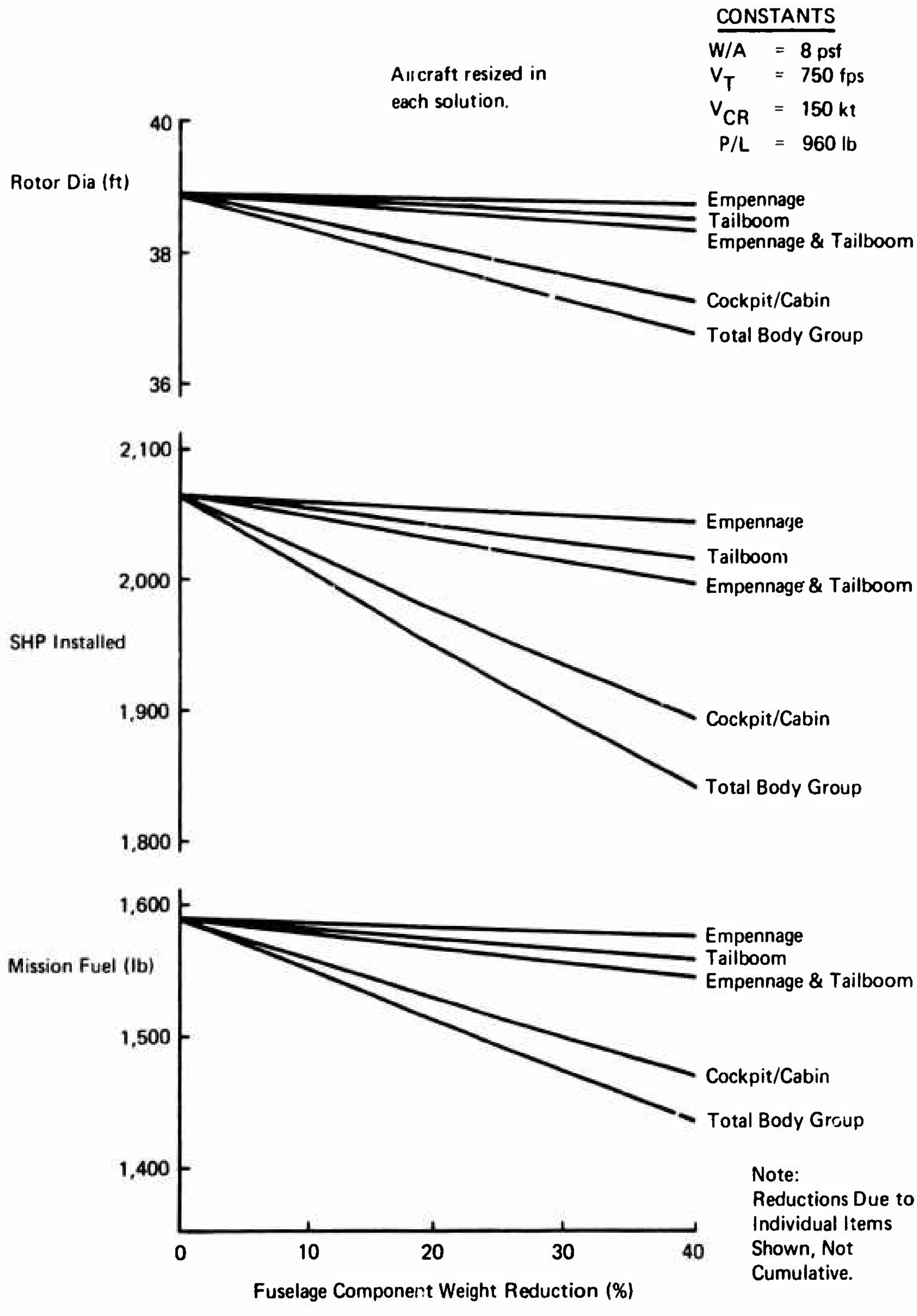


Figure 17. Effect of Fuselage Component Weight Reductions.

3.1.2 CONCEPTUAL DESIGN STUDIES

The helicopter airframe has a relatively light loading intensity compared to high-performance military fixed-wing aircraft. Thin laminates for skin and support structure will be required when designing in composites to achieve lower airframe weight than conventional aluminum construction. The relatively light loading spectrum for MUT inevitably leads to gauge limiting problems (minimum practical manufacturing ply layup is in excess of calculated design thickness required). This constraint adversely impacts airframe structural efficiency and limits the weight gain of composites over aluminum alloy.

Three main composite design categories comprising different structural approaches were considered to improve structural efficiency:

- Category 1 - Honeycomb sandwich hybrid
- Category 2 - Skin/stringer/frame panel molding with selected use of stabilizing foam
- Category 3 - Filament wound, or geodesic, or mix of categories 1, 2 and 3

Within each category, alternative structural arrangements of certain components were considered.

A series of structural design studies was conducted to provide numerous competitive approaches for selective evaluation; these studies are presented as reference drawings in Appendix B. The candidate structures were then rated against the primary and secondary selection factors; these ratings are presented in Table 6 (category 1 concepts shown on sheet 1, category 2 on sheet 2, and category 3 on sheet 3). Isometric sketches illustrating each configuration are presented in Figures 18 through 31. All ratings were in respect to the baseline metal configuration which was defined to meet the SASS requirements for each selection parameter. The comparative production costs of each concept are shown in Figure 32.

3.1.3 RELATIVE COST

The relative costs of each candidate were estimated assuming complete utilization of graphite (AS)/epoxy at a 1974 production quantity price of $50/lb. (The comparable 1974 price of sheet aluminum is $6/lb.)

3.1.4 MATERIAL STRUCTURAL EFFICIENCY

A number of material systems were screened against typical helicopter loading conditions. These systems, the mechanical strength properties used for screening, and their relative structural performance under each of the loading environments are presented in Table 7.

3.1.5 OTHER FACTORS

Other considerations such as safety, fail-safety, and SASS conformance elements of detection avoidance, crashworthiness, repairability, maintainability, and survivability were rated in comparison to the baseline configuration.

3.1.6 SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION-AIRFRAME

Table 6 shows that two concepts, namely F and C, both similar honeycomb-sandwich designs, emerged as joint winners, each with a rating total of 5.

However, the conformance element ratings in the same table reveal a low rating for concept C due to the lack of a field-splice joint for removal of the tailboom and empennage, whereas concept F incorporated a field-splice joint.

Concept F (Figure 23) was selected for its general superiority over the other concepts, but the final selection for further refinement also included the feature of an external mechanical-attachment tailboom-splice joint as shown in the circled view of concept E (Figure 22, circled view on right-hand side).

An alternative method for joining the tailboom to the cabin structure in the fuel bay, stations 163 to 239, via a horizontal joint at WL175 and a vertical joint at station 163 was investigated (see Figure B-19). Despite the redeeming features of this method such as good fail-safety and joint-attachment access, the arrangement was considered cost-prohibitive due mainly to problems associated with fabricating long field-splice joints which are required to align simultaneously in two planes (vertical and horizontal) and the number of splice-attachment bolts involved.

Table 6. Concept Screening Analysis – Aircraft Primary Structure (Sheet 1)

Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1–	
–2	MUCH POORER

CONFORMANCE FACTORS:–
BETTER + } REFERRED
SAME 0 } TO
POORER – } BASELINE

- OBJECTIVE: To select best honeycomb sandwich concept
- COSTS: Based on 1,000 units
- MATERIAL: Composite hybrid graphite AS/Kevlar 49

		CONCEPT A	CONCEPT B	CONCEPT C	CONCEPT D	CONCEPT E	CONCEPT F
		As Figure B-3 (buttline beam concept) but with 1. Reduced section tail boom per Figure B-12 2. Detachable tailboom with four-point attach joint at sta 239 3. Detachable vertical stabilizer. 4. Horizontal stabilizer behind tailboom 5. Cockpit enclosure 1B per Figure B-11 6. Underfloor structure per Figure B-12.	As Figure B-3 and items 1 thru 5 as Concept A But No buttline beams full depth in tank bay (sta 163 239) But with Variation A per Figure B-5	As Figure B-3 and items 1, 3, 4, 5 and 6 as Concept A But no buttline beams full depth in tank bay and no four-point tailboom attach But with extended tailboom (molded clamshell) structural deck with longitudinal beams, also secondary structure avionics, outer assy per Figure B-7	As Figure B-10 (main view) bathtub and lid and integral fuel tank, but with items 1, 3, 4, 5 and 6 as Concept A	Same as Concept D (main view) except for the following differences 1. "Bathtub and lid" concept with mechanical attach joint tailboom to fuselage as shown on Figure B-7	Same as Concept E except for 1. Integral (built in) vertical stabilizer 2. Basic tailboom section changed from slab side to egg shape section (see RH inset view on Figure B-10
RATING ELEMENTS	STRUCTURAL EFFICIENCY	1	0	2	2	2	2
	FAIL SAFETY	1	–1	2	1	1	2
	SAFETY	0	0	0	0	0	0
	PRODUCIBILITY/COST	1	2	1	0	0	1
	RATING TOTAL	3	1	5	3	3	5
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	+	+	+	+	+	+
	CRASHWORTHINESS	+	+	+	+	+	+
	REPAIRABILITY	+	+	0	0	+	0
	MAINTAINABILITY	+	–	–	–	+	0
	SURVIVABILITY	0	–	+	+	0	0
	GENERAL SAS CONFORMANCE	0	–	0	0	0	0
	A/F CONCEPT RANKING	2	5	1	2	2	1

Table 6. Concept Screening Analysis – Aircraft Primary Structure (Sheet 2)

Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1	
–2	MUCH POORER

CONFORMANCE FACTORS:–
BETTER + } REFERRED
SAME 0 } TO
POORER – } BASELINE

- OBJECTIVE: To select best skin/stringer concept
- COSTS: Based on 1,000 units
- MATERIAL: Composite hybrid graphite AS/Kevlar 49

		CONCEPT G	CONCEPT H	CONCEPT J	CONCEPT K	CONCEPT L	CONCEPT M
		As Figure B–4 (buttline beam) but with 1. Horiz stabilizer behind tailboom 2. Cockpit enclosure Concept 1B per Figure B–11 3. Horiz stab. as Figure B–10 4. Underfloor structure as Figure B–12 (including variation D) 5. Detachable vertical stabilizer (pins and lugs as Concept A)	As Figure B–4 (buttline beam) but with 1. Horiz stabilizer behind tailboom 2. Cockpit enclosure Concept 1A per Figure B–11 3. Horiz stabilizer as Figure B–10 inset view (top r/h side of sketch) 4. Underfloor structure per Figure B–12 (including variation D)	Same as Concept G except tank bay section as Variation A per Figure B–5 (No buttline beams)	Same as Concept H except tank bay section as Variation A per Figure B–5 (No buttline beams)	Same as Concept G except tetracore buttline beam construction and sta 163 bulkhead and underfloor beams (see alternative tetracore beam section E–E on Figure B–12	As Figure B–4 (buttline beam) but with 1 Skin stringer extended (full depth) tailboom with secondary side structure for avionics stowage per variation B of Figure B–5 2 Horiz stabilizer behind tailboom 3 Cockpit enclosure Concept 1B per Figure B–11 4 Underfloor structure per Figure B–12 (including variation D)
RATING ELEMENTS	STRUCTURAL EFFICIENCY	1	2	–1	0	1	1
	FAIL SAFETY	1	1	–1	–1	2	1
	SAFETY	0	0	0	0	0	0
	PRODUCIBILITY/COST	–1+	–1	0	0	–2	–2
	RATING TOTAL	1+	2	–2	–1	1	0
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	+	+	+	+	+	+
	CRASHWORTHINESS	+	+	+	+	+	+
	REPAIRABILITY	–	–	–	–	–	–
	MAINTAINABILITY	0	0	–	–	0	+
	SURVIVABILITY	0	0	–	–	+	0
	GENERAL SAS CONFORMANCE	0	0	0	0	0	0
	A/F CONCEPT RANKING	4	3	9	8	5	7

Table 6. Concept Screening Analysis – Aircraft Primary Structure (Sheet 3)

Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1–	
–2	MUCH POORER

CONFORMANCE FACTORS: –
BETTER + } REFERRED
SAME 0 } TO
POORER – } BASELINE

- OBJECTIVE. To select best mixed concept sandwich panel, skin/stringer, geodesic
- COSTS: Based on 1,000 units
- MATERIAL Combinations of thermosetting materials, composite hybrid graphite AS/Kevlar 49

		CONCEPT N	CONCEPT P	CONCEPT Q	CONCEPT R	CONCEPT S	CONCEPT T
		As Figure B–10 but with wound tailboom concept as shown on inset view (top LH side) also with wound spar torque box on vertical stabilizer. Also horiz. stabilizer per inset view (top RH) and cockpit enclosure per Concept 1B of Figure B–11	As Figure B–4 skin/stringer but with 1. Variation B per Figure B–5 skin stronger primary tailboom structure and honeycomb panel secondary fairing structure. 2. Cockpit enclosure per Concept 1B of Figure B–11 3. Horiz. stabilizer behind tailboom. 4. Underfloor structure per Figure B–12 (including variation d).	As Figure B–3 honeycomb sandwich but with 1. Reduced section detachable tailboom 2. Thin honeycomb shell tailboom with four stringers 3. Honeycomb sandwich upper deck structure and beams 4. Cockpit enclosure per 1B of Figure B–11. 5. Horiz. stabilizer behind tailboom 6. Under floor structure per Figure B–12.	As Figure B–7 except for geodesic woven tailboom and buttline beams and detachable vertical stabilizer (See Figure B–13.) Also items 2 and 3 of Concept P	As Concept C but with 1. Tailboom detachable from fuselage by mechanical joint along deck line (horizontal) and around bulkhead STA 163 (vertical) 2. Wound tailboom as Concept N 3. Internal vertical stabilizer as Concept F 4. Egg-shaped tailboom section	As Concept B except built-in vertical stabilizer (as Concept F). This is the optimal 'cost' concept
RATING ELEMENTS	STRUCTURAL EFFICIENCY	2	1	1	1	2	0
	FAIL SAFETY	2	1	1	1	2	–1+
	SAFETY	0	0	0	0	0	0
	PRODUCIBILITY/COST	–2	–1	–1	–1–	–1	2
	RATING TOTAL	2	1	1	1–	3	1+
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	+	+	+	+	+	+
	CRASHWORTHINESS	+	+	+	+	+	+
	REPAIRABILITY	0	0	–	–	0	+
	MAINTAINABILITY	–	0	0	0	+	–
	SURVIVABILITY	+	+	0	+	+	0
	GENERAL SAS CONFORMANCE	0	0	0	0	0	0
	A/F CONCEPT RANKING	3	5	5	6	2	4

Table 7. Comparison of Material Systems and Helicopter Loading Conditions

Preliminary Design Material Selection for Maximum Structural Efficiency		1 GRAPHITE AS/EPOXY	2 KEVLAR 49/ EPOXY	3 S GLASS/ EPOXY	4 E GLASS/ EPOXY (181 STYLE)	5 GRAPHITE AS/THERMO-PLASTIC	6 HYBRID, GR K49/ EPOXY (50 50)	7 7075 T6 (REF)
MATERIAL PROPERTIES								
DENSITY (ρ)	PCI	0.055	0.050	0.066	0.065	0.058	0.0525	0.101
STRENGTH								
Tension 0°	KSI	186	200	219	60.4	173	147	73
Compression 0°	KSI	165	40	110	64.8	132	94	65
Shear $\pm 45^{\circ}$	KSI	50.2	28	36.4	14.3	45.6	28.6	47
Fatigue (R = 0.1) 0°	KSI	43.2	43.2	21.5	7.8	43.2	43.2	8
MODULUS								
Axial 0°	MSI	18.5	11.0	6.3	3.8	16.8	11.2	10.3
Shear $\pm 45^{\circ}$	MSI	4.5	3.0	1.7	0.8	3.6	3.1	3.9
SPECIFIC PROPERTIES								
F_{tu}/ρ 0°	KSI/PCI	3380	4000	3320	929	2982	2800	782
F_{cu}/ρ 0°	KSI/PCI	3000	800	1670	997	2276	1790	733
F_{su}/ρ $\pm 45^{\circ}$	KSI/PCI	913	560	552	220	786	545	465
σ_e/ρ (R = 0.1) 0°	KSI/PCI	785	864	326	121	745	824	79
E/ρ 0°	MSI/PCI	336	220	95	58	289	213	102
G/ρ $\pm 45^{\circ}$	MSI/PCI	82	61	26	12.5	62	59	38.5
COMPARATIVE SPECIFIC PROPERTIES								
STRENGTH	With respect to Column 1 as a base equal to 1.0							
Tension		1.0	1.183	0.982	0.275	0.882	0.828	0.231
Compression		1.0	0.267	0.557	0.332	0.759	0.597	0.244
Shear		1.0	0.613	0.605	0.241	0.861	0.597	0.504
Fatigue		1.0	1.101	0.415	0.154	0.449	1.050	0.101
MODULUS								
Axial		1.0	0.655	0.283	0.173	0.860	0.634	0.304
Shear		1.0	0.741	0.317	0.152	0.756	0.718	0.471
WEIGHT (Reciprocal)		1.0	1.100	0.833	0.846	0.948	1.048	0.545
RANKING Distribution of Material for various critical loadings and structural element configuration								
(A) Lesser of Tension or Compression		1	6	4	5	2	3	7
(B) 70% Tension or Compression 30% Shear		1	5	4	7	2	3	6
(C) 60% (B) 40% Fatigue		1	4	5	7	2	3	6
(D) Sandwich Minimum Gage 80% Weight 20% (C)		2	1	5	6	4	3	7
(E) Skin/Stringer Minimum Gage 30% Weight 70% (C)		1	4	5	7	2	3	6
(F) Sandwich and Skin/Stringer 50% (D) 50% (E)		1	4	6	5	3	2	7
(G) 60% Tension or Compression 25% Shear 10% E Modulus 5% G Modulus		1	5	4	7	2	3	6
(H) Sandwich Minimum Gage & Modulus 80% Weight 10% (C) 5% E Modulus 5% G Modulus		1	2	5	6	4	3	7
(I) Skin/Stringer Minimum Gage & Modulus 30% Weight 50% (C) 12% E Modulus 8% G Modulus		1	4	5	6	2	3	6
(J) Hybrid 50% (H) 50% (I)		1	2	5	6	4	3	7
OVERALL RANKING		1	4	5	6	2	3	7

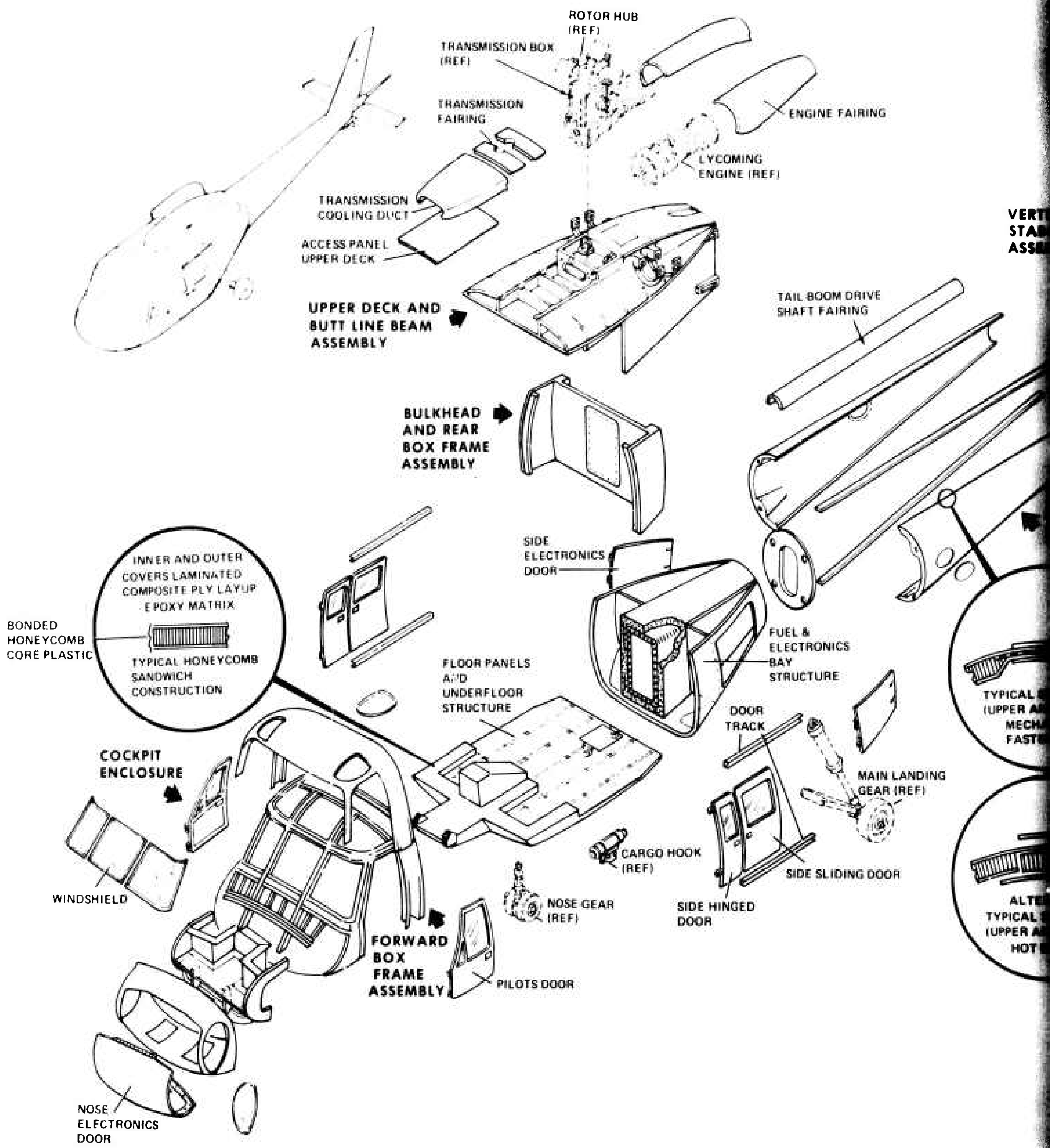
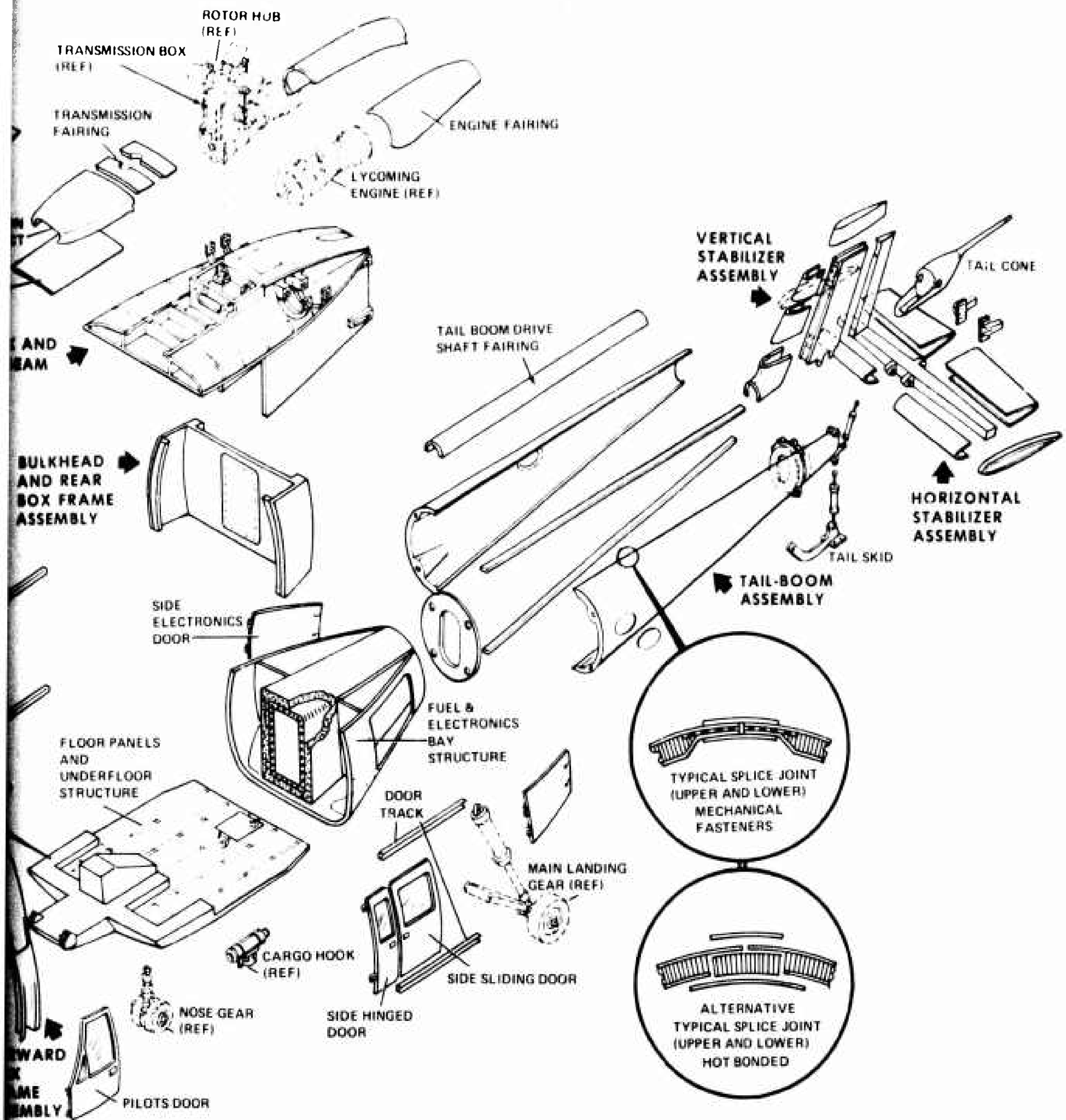


Figure 18. Concept A - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



phite AS/Kevlar Epoxy Honeycomb Sandwich.

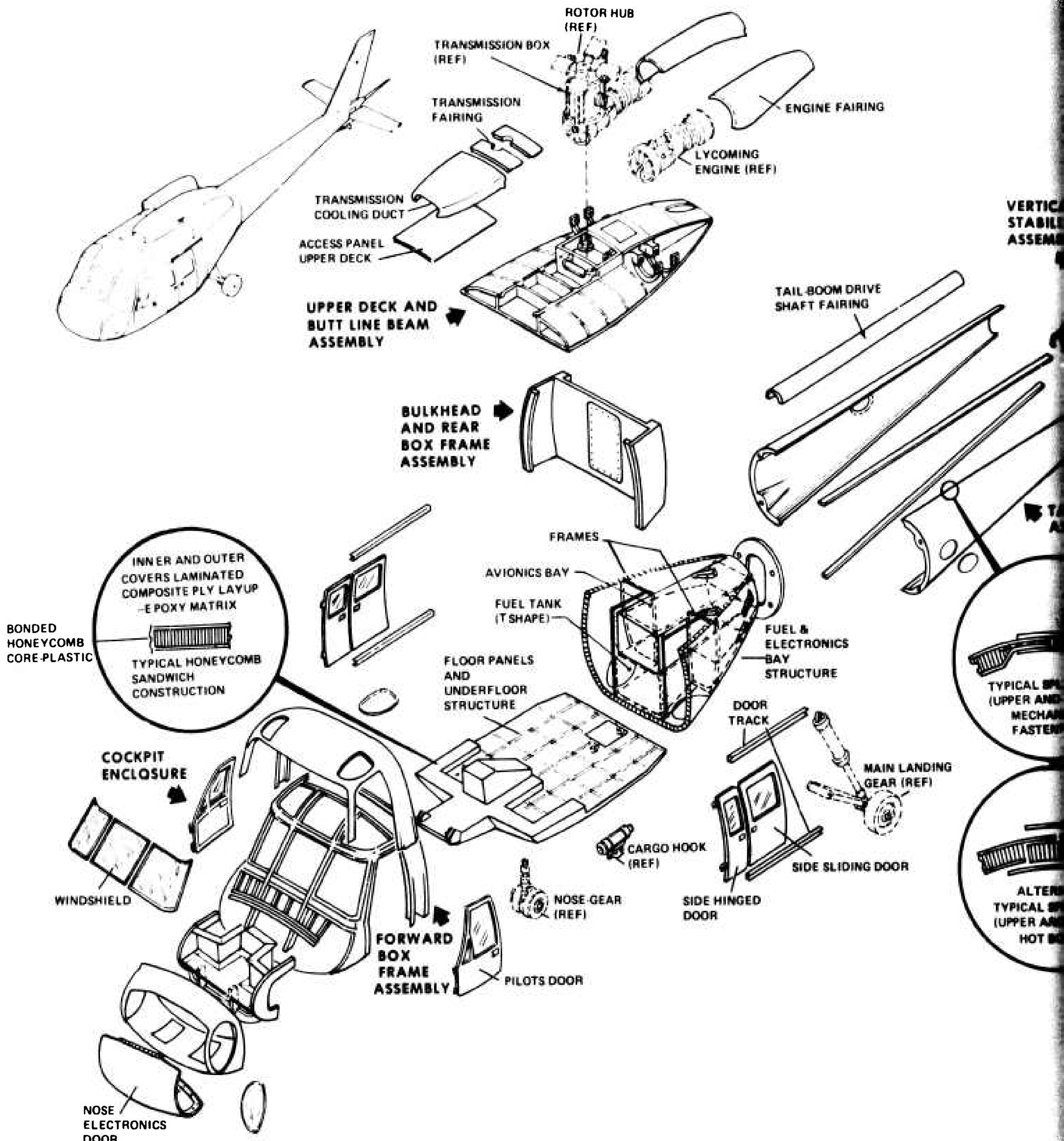
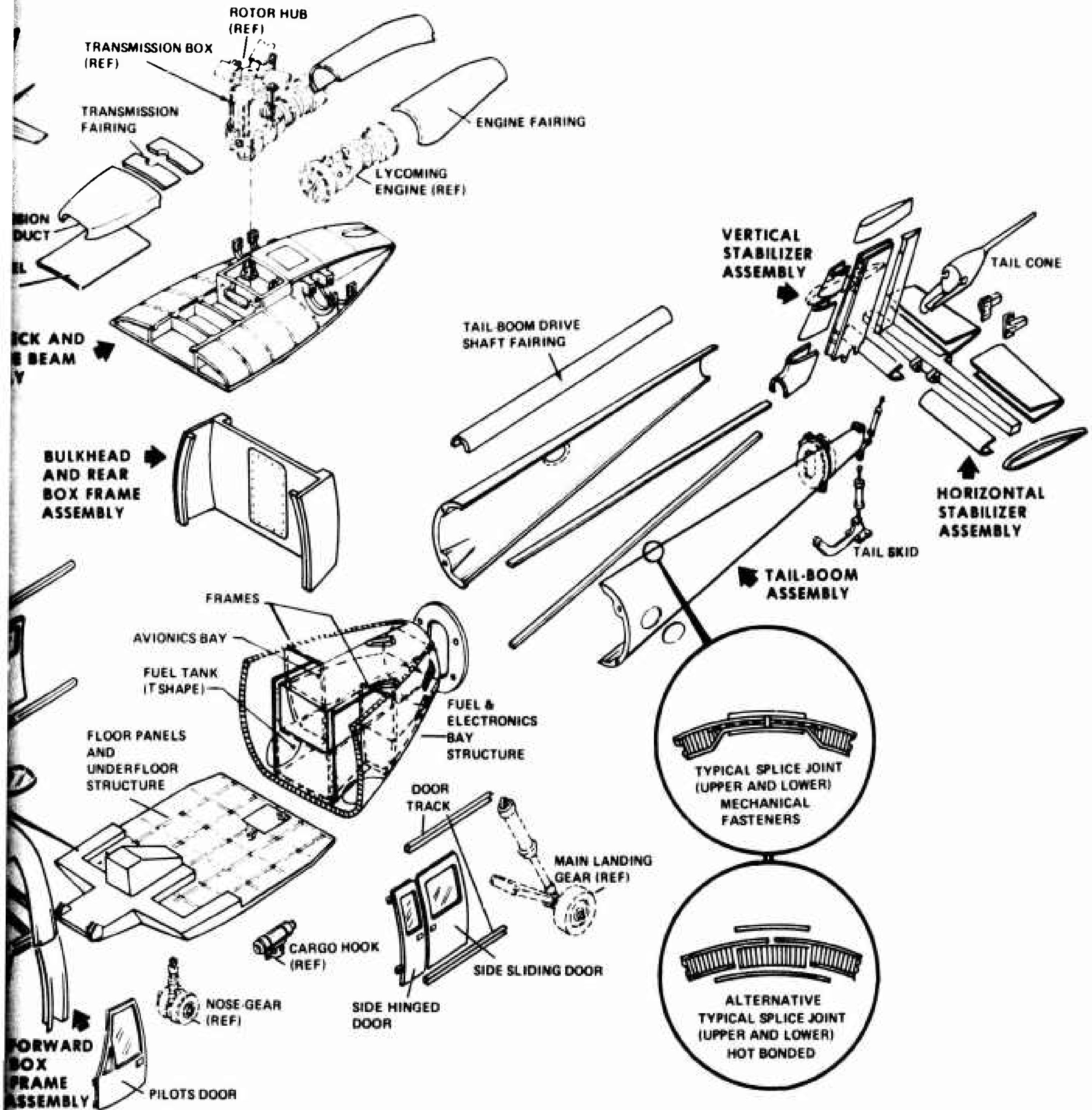


Figure 19. Concept B - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



raphite AS/Kevlar Epoxy Honeycomb Sandwich.

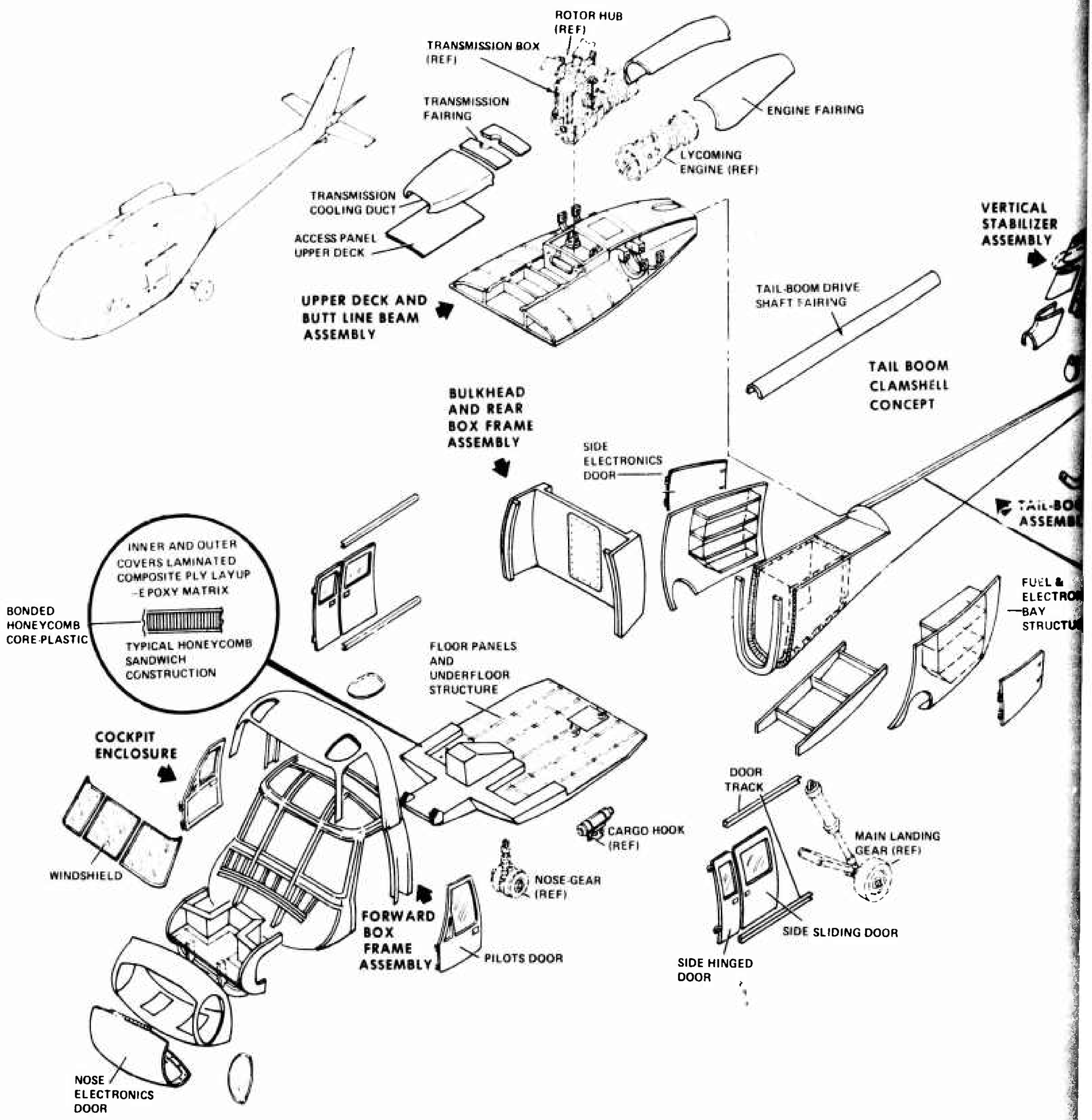
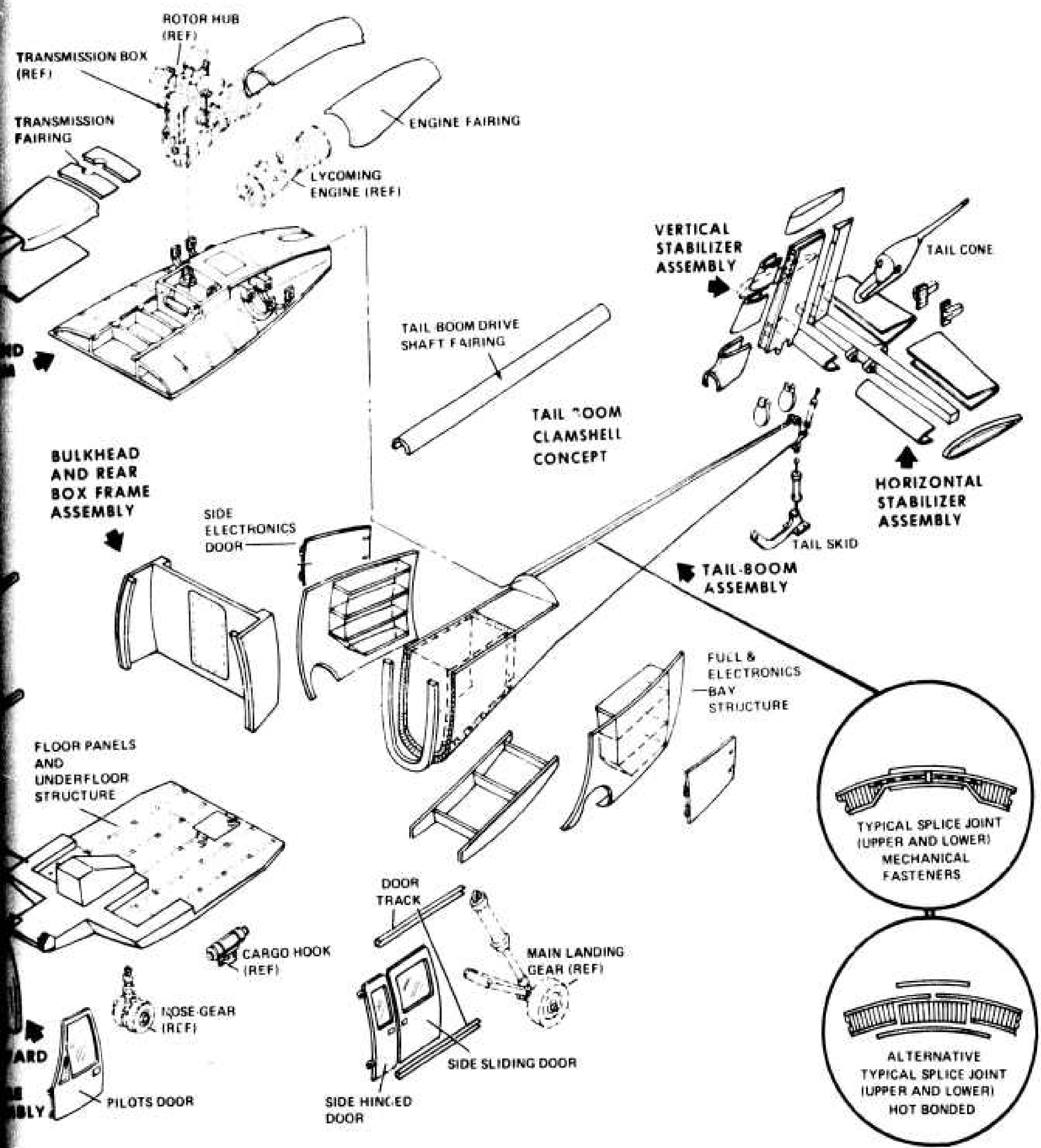


Figure 20. Concept C - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



aphite AS/Kevlar Epoxy Honeycomb Sandwich.

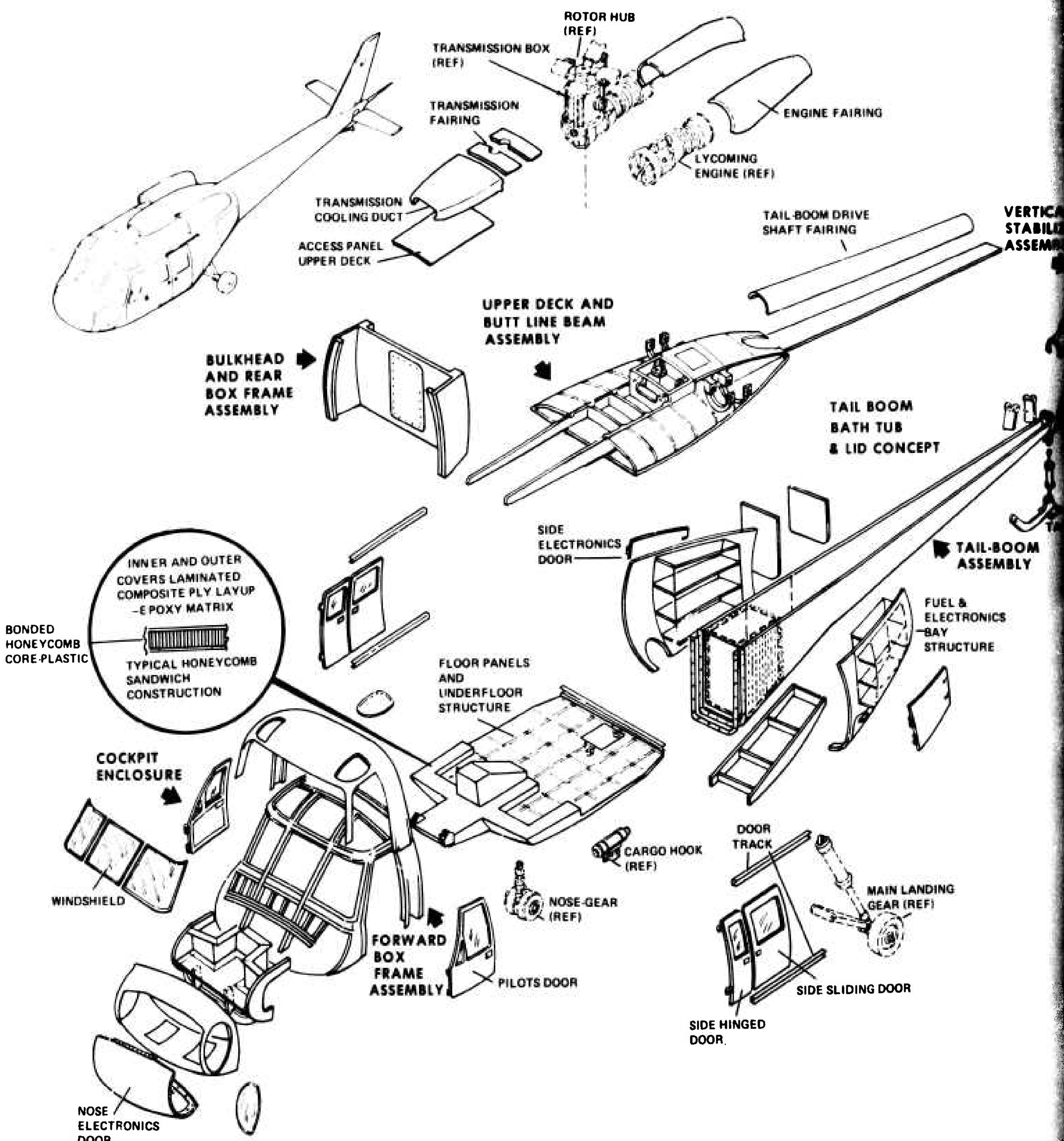
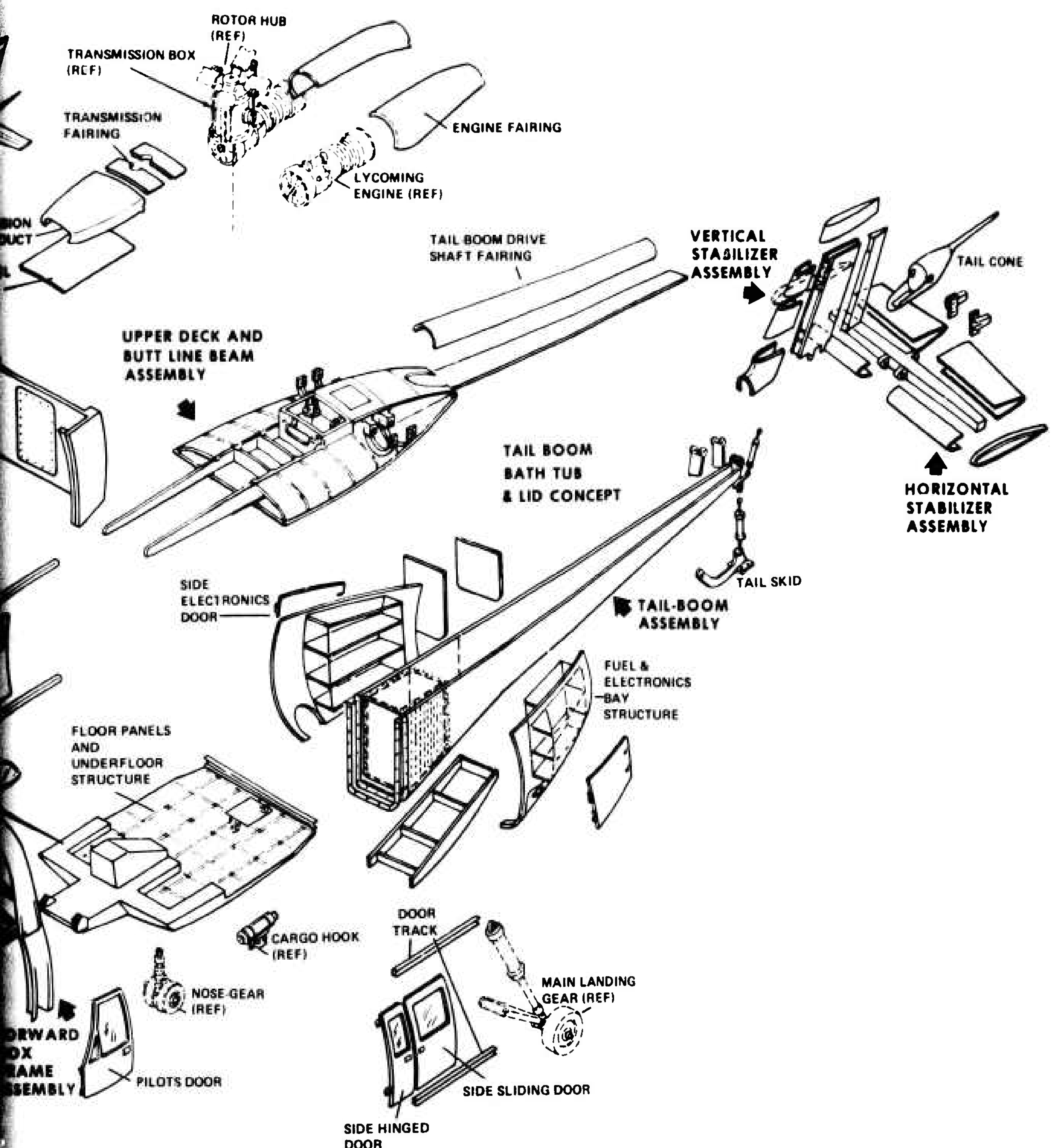


Figure 21. Concept D - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



raphite AS/Kevlar Epoxy Honeycomb Sandwich.

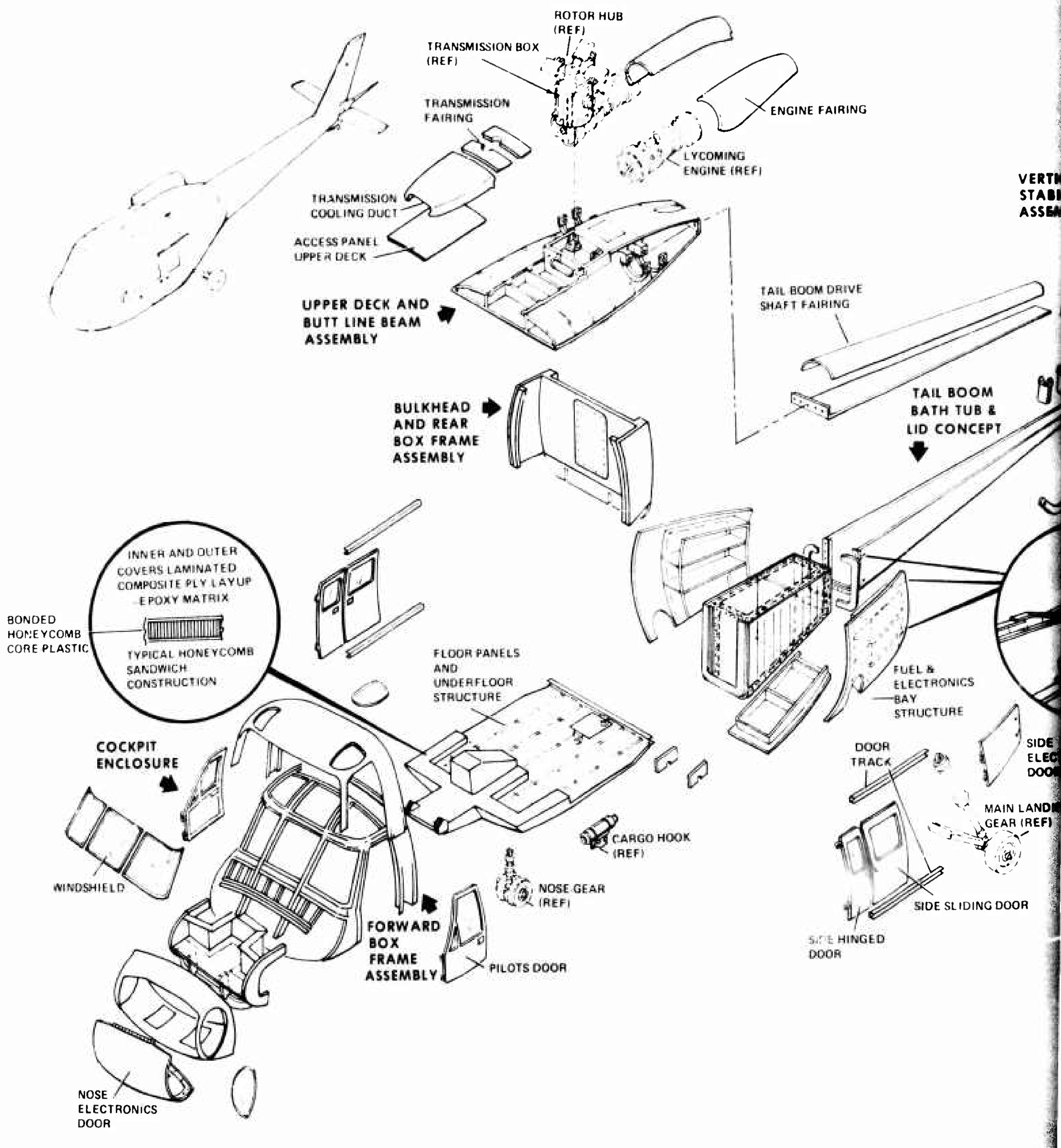
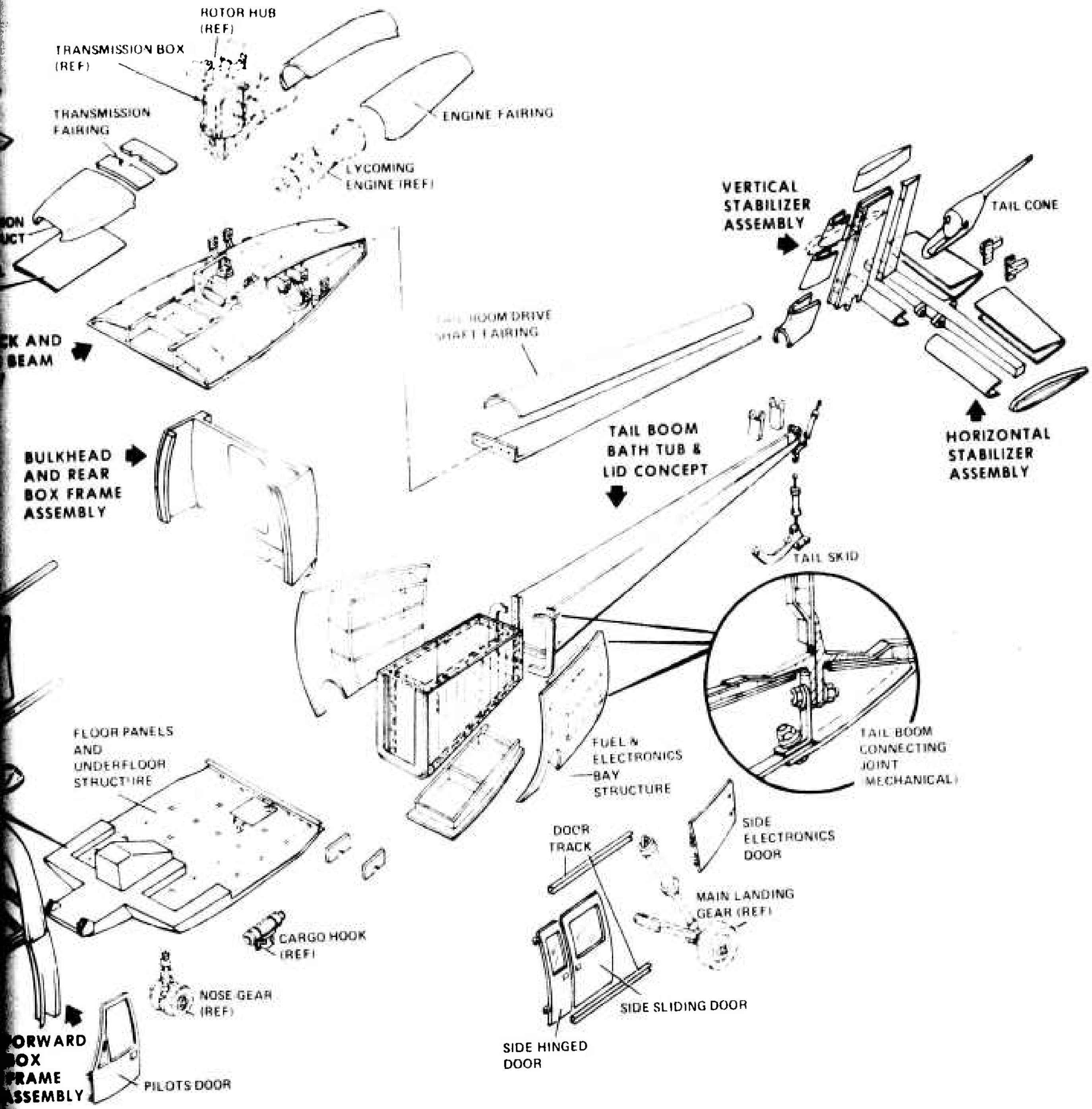


Figure 22. Concept E - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



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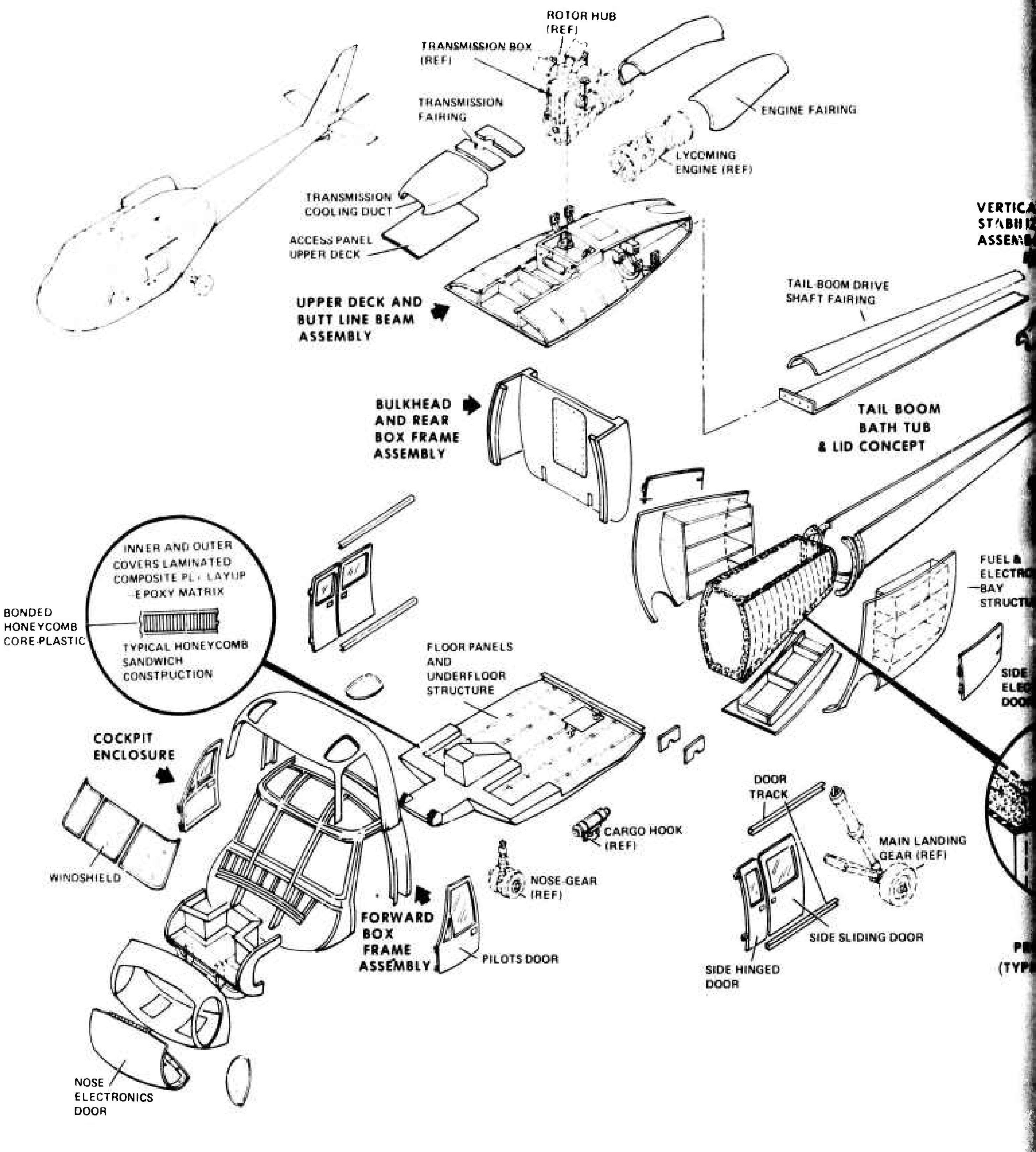
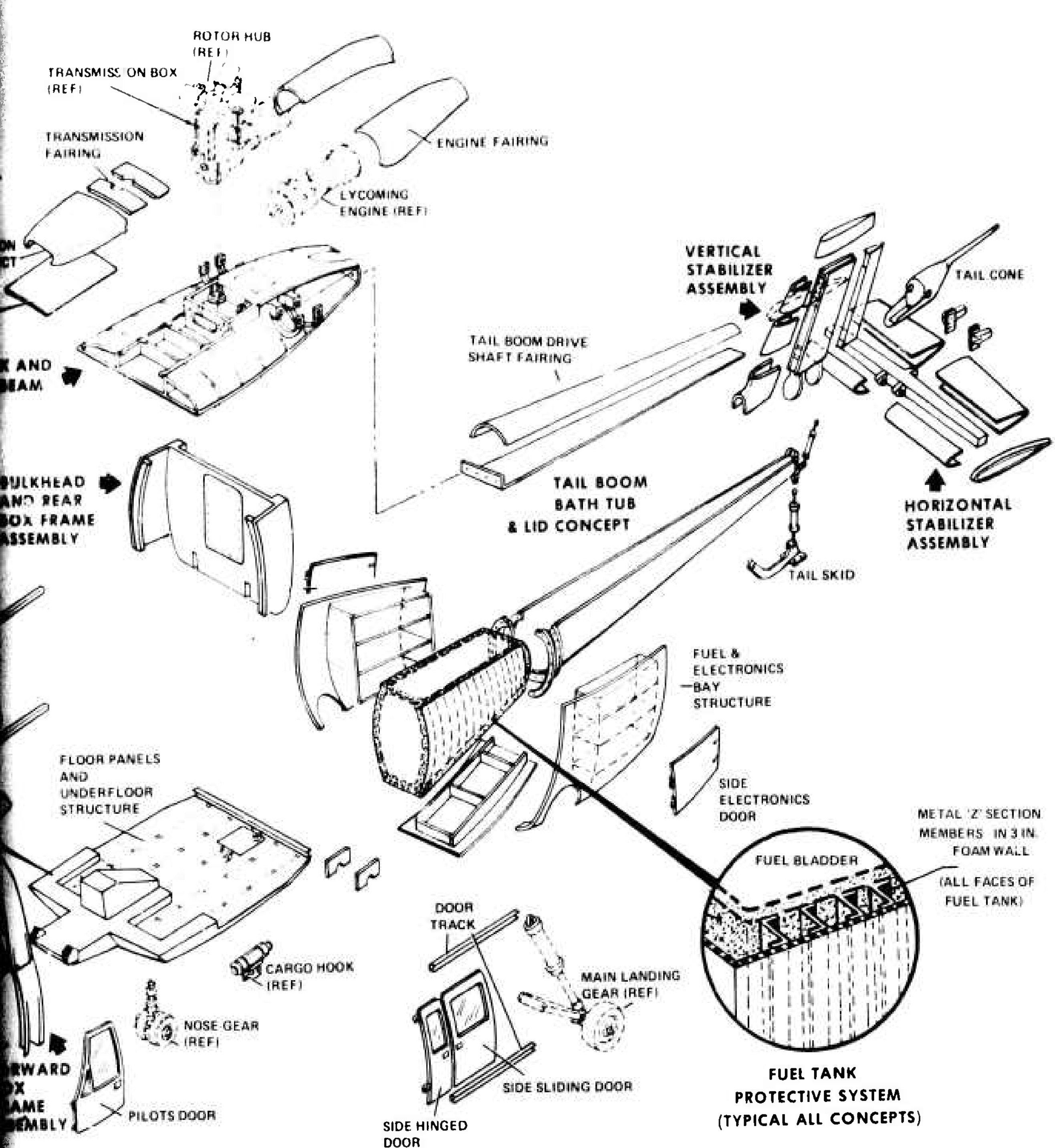


Figure 23. Concept F - Graphite AS/Kevlar Epoxy Honeycomb Sandwich.



phite AS/Kevlar Epoxy Honeycomb Sandwich.

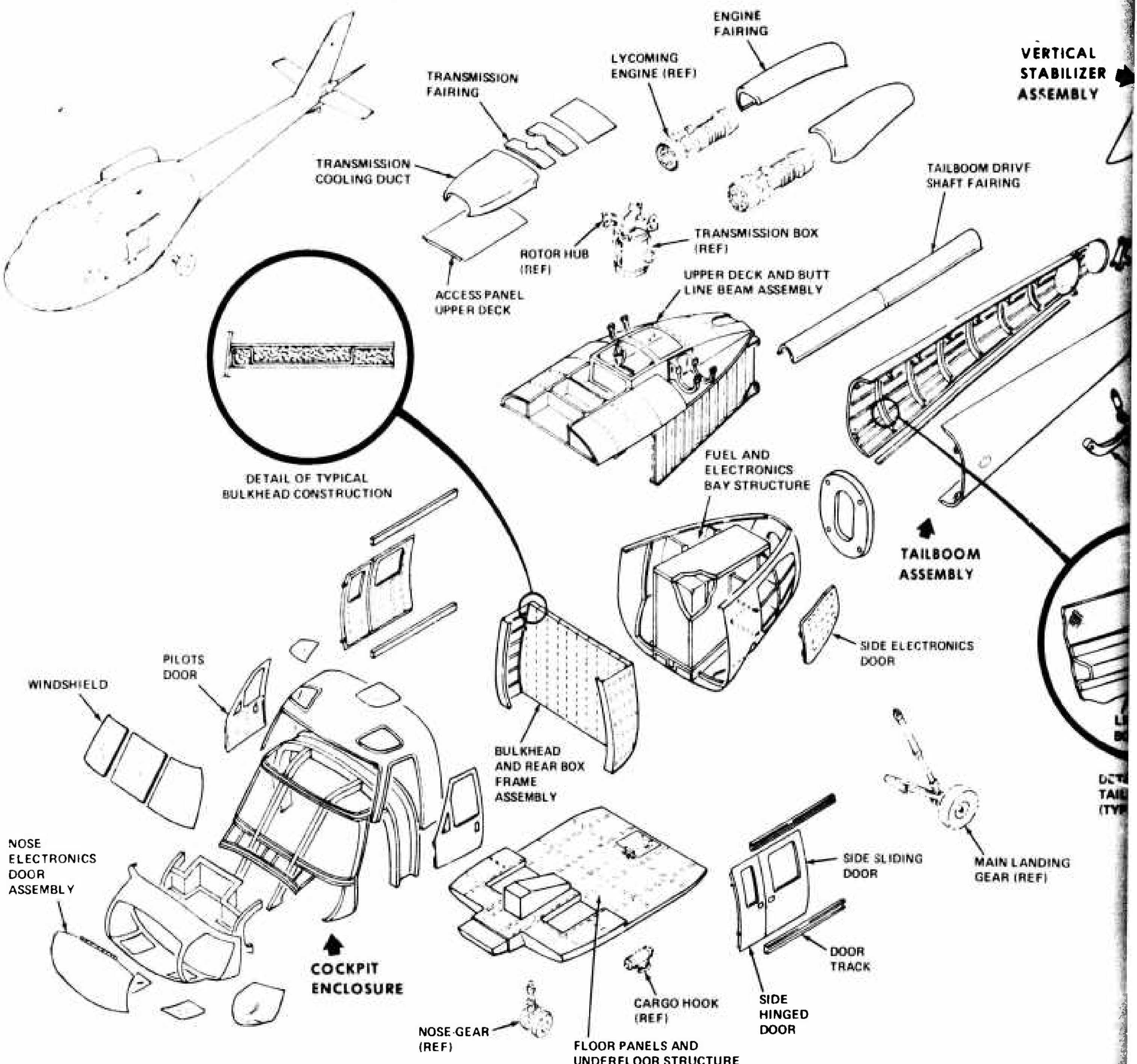
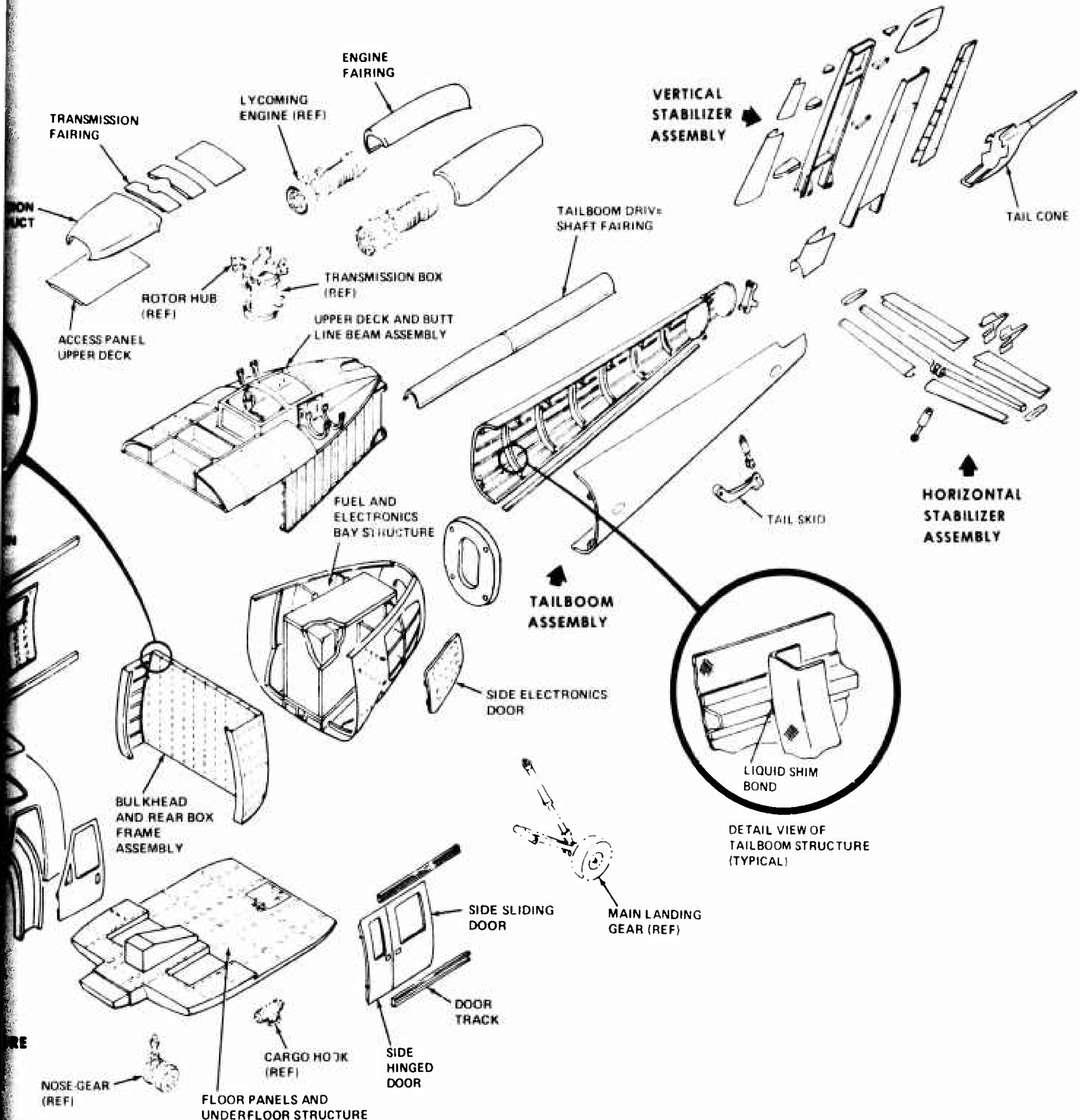


Figure 24. Concept G - Graphite AS/Kevlar Epoxy Skin/Stringer.



raphite AS/Kevlar Epoxy Skin/Stringer.

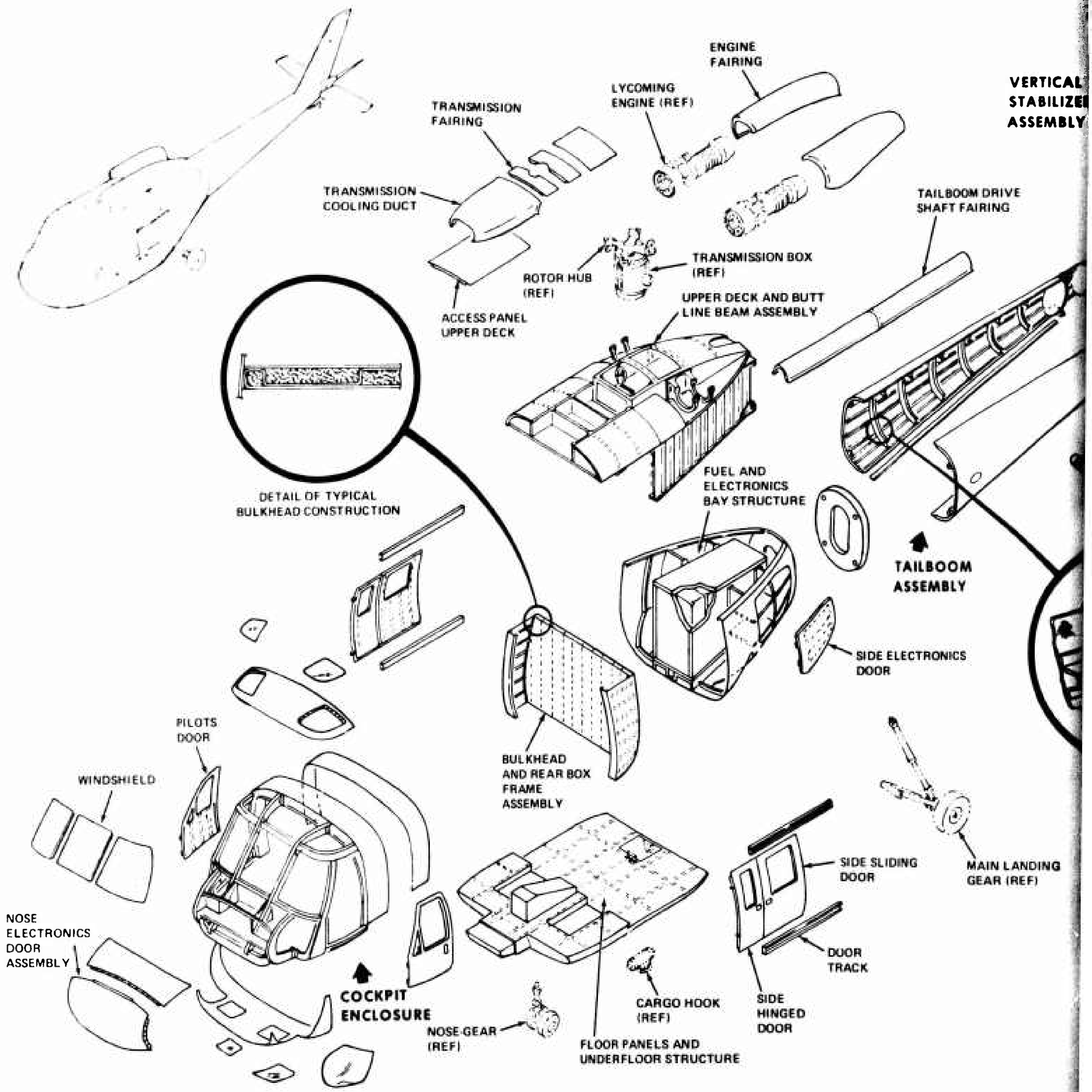
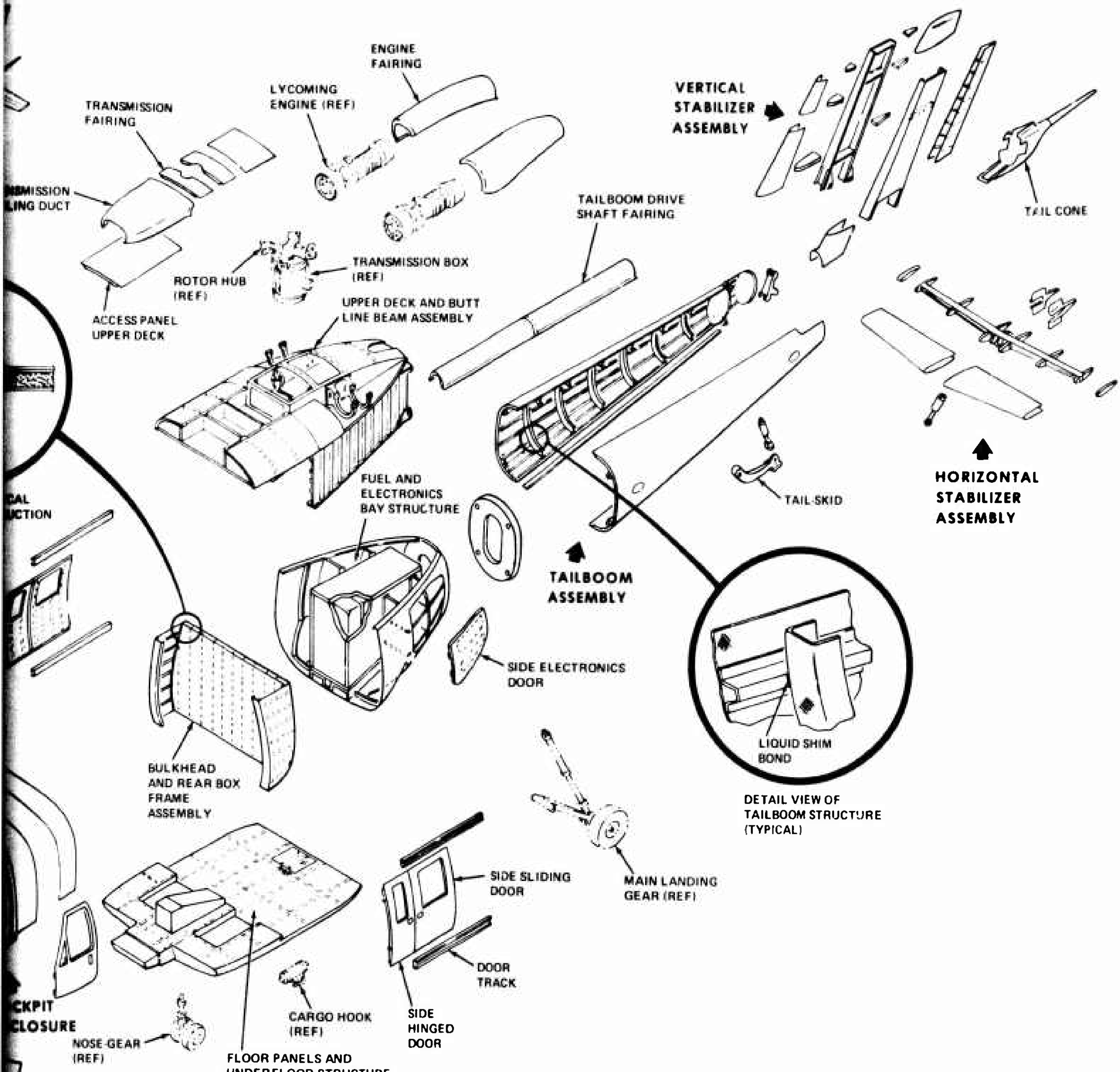


Figure 25. Concept H - Graphite AS/Kevlar Epoxy Skin/Stringer.



aphite AS/Kevlar Epoxy Skin/Stringer.

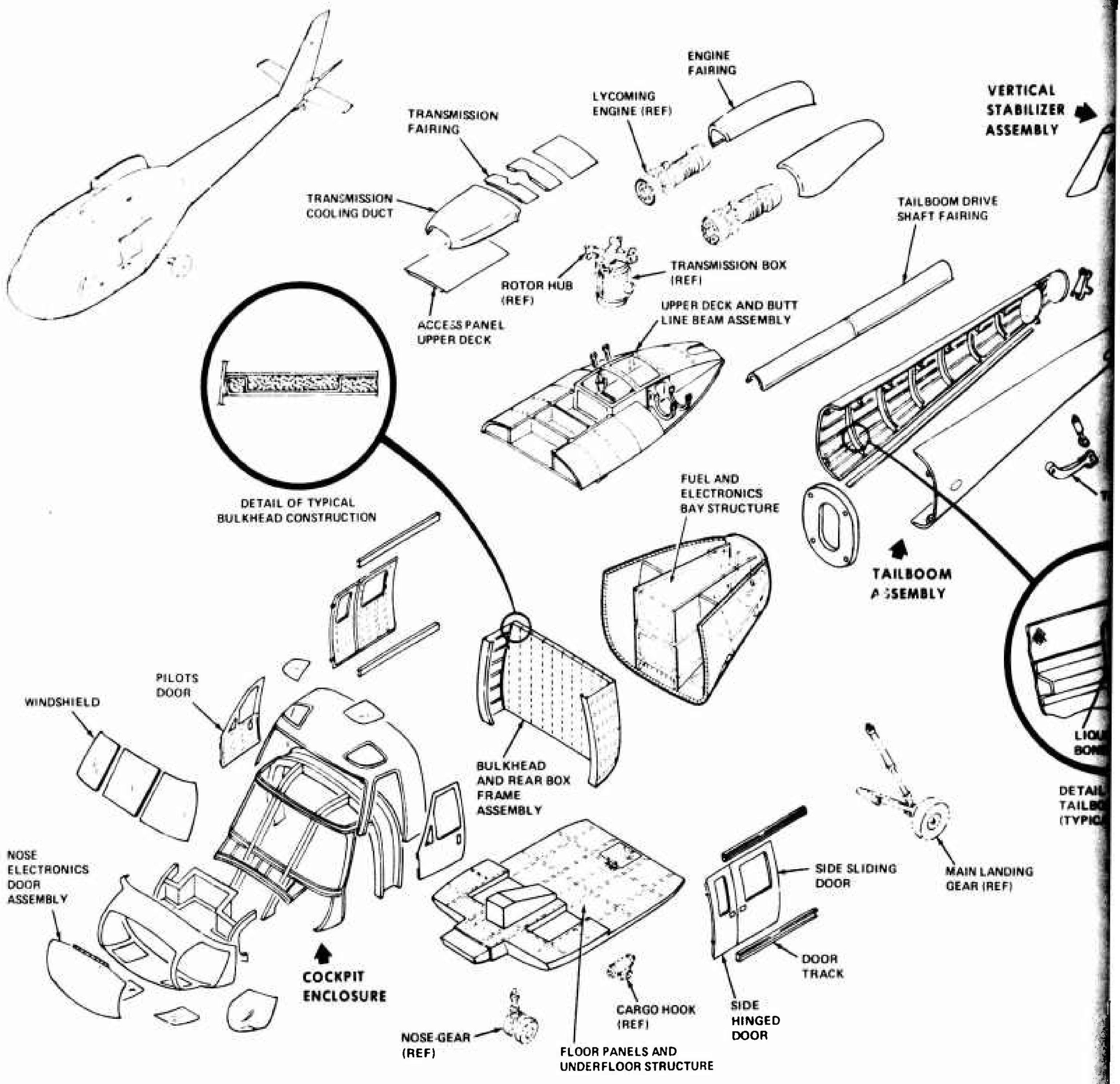
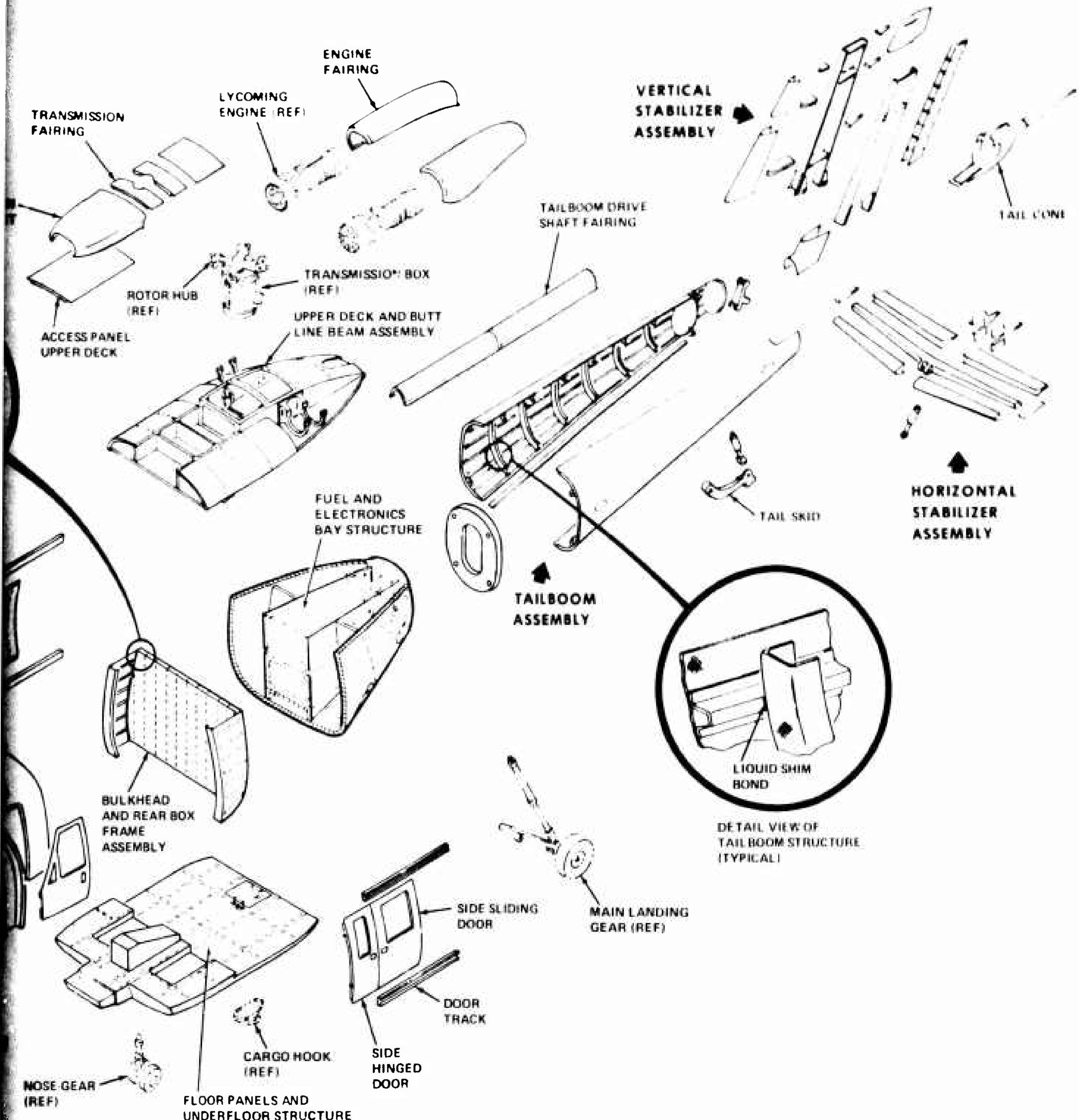


Figure 26. Concept J - Graphite AS/Kevlar Epoxy Skin/Stringer.



phite AS/Kevlar Epoxy Skin/Stringer.

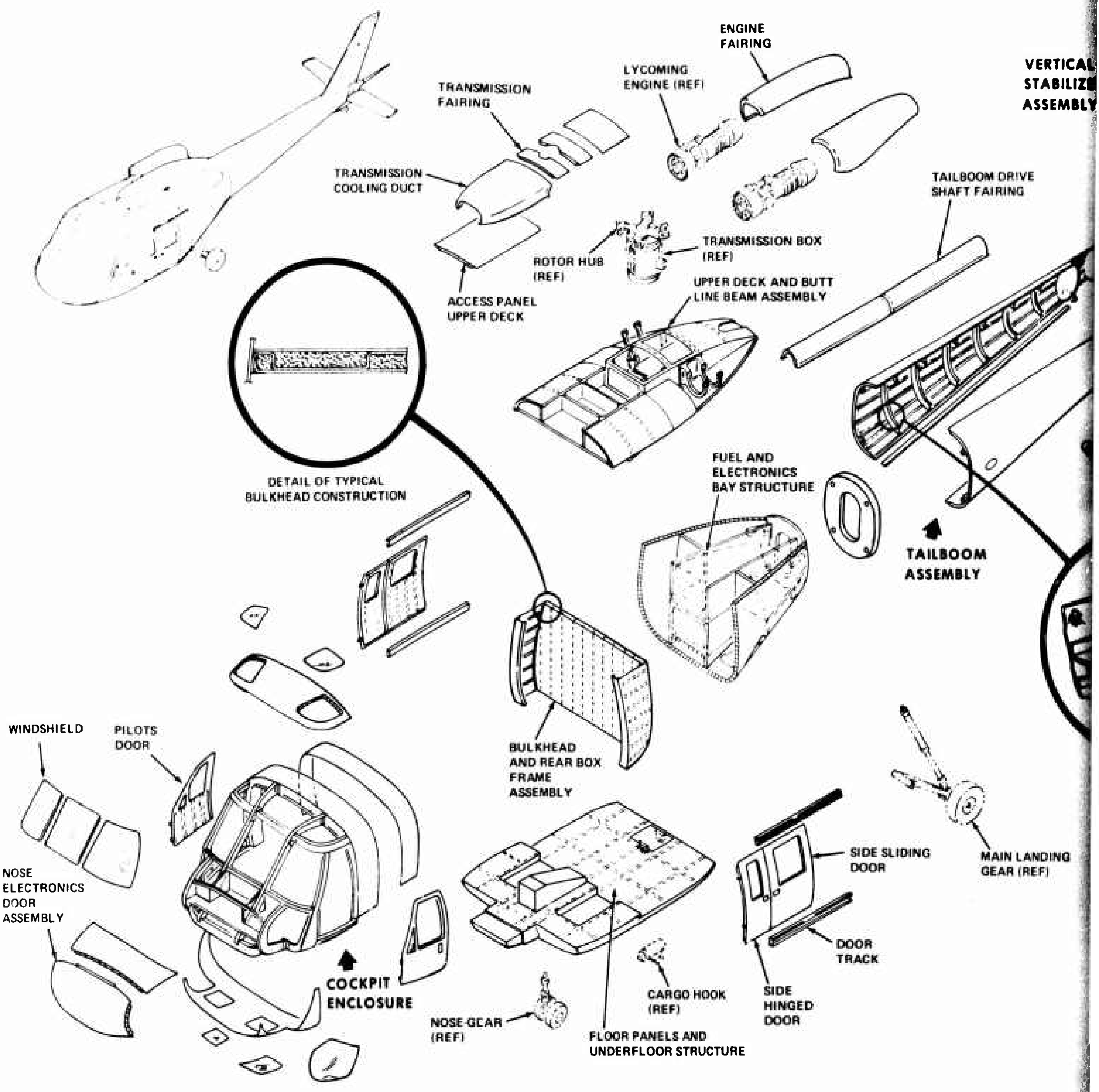
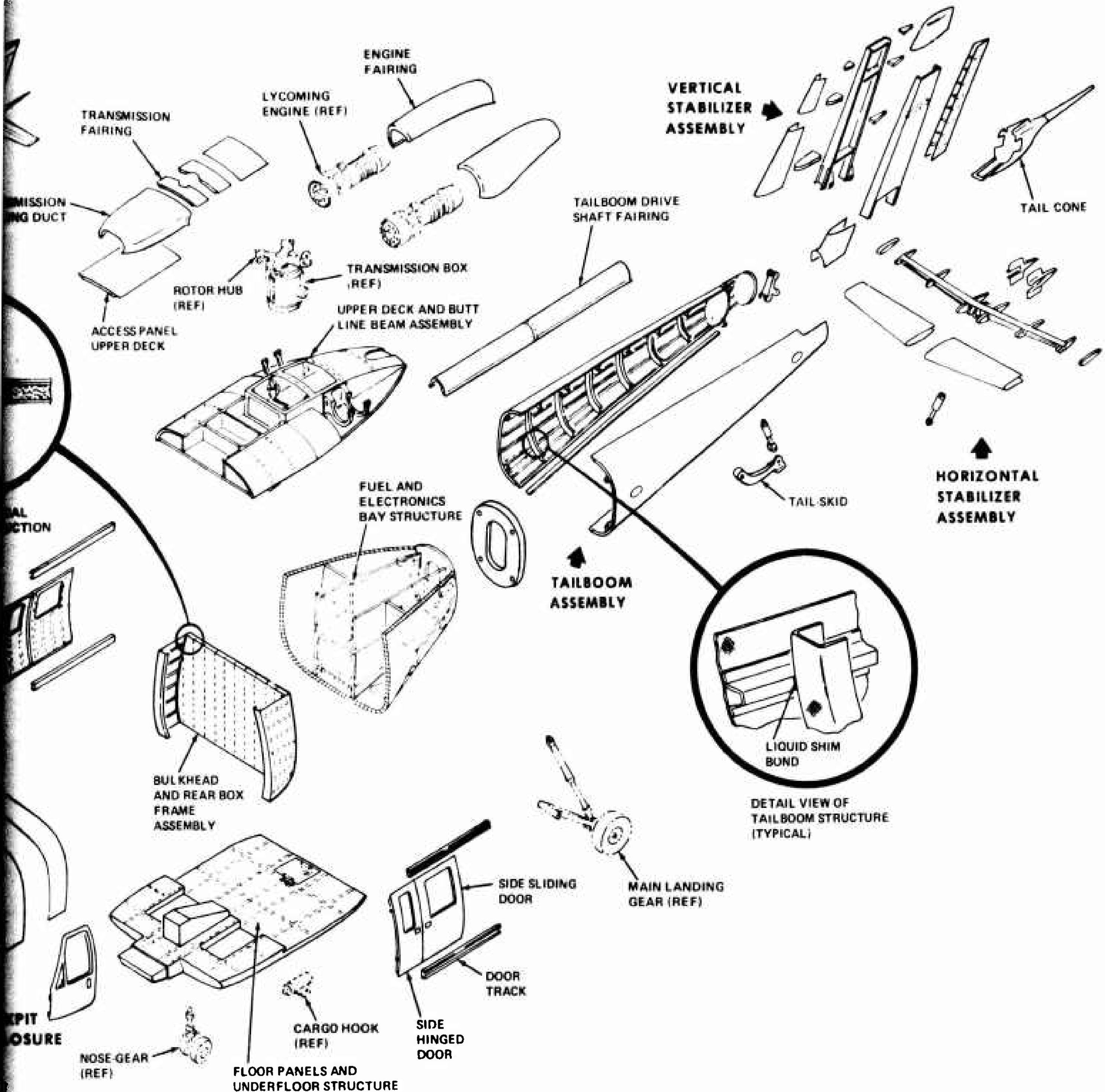


Figure 27. Concept K - Graphite AS/Kevlar Epoxy Skin/Stringer.



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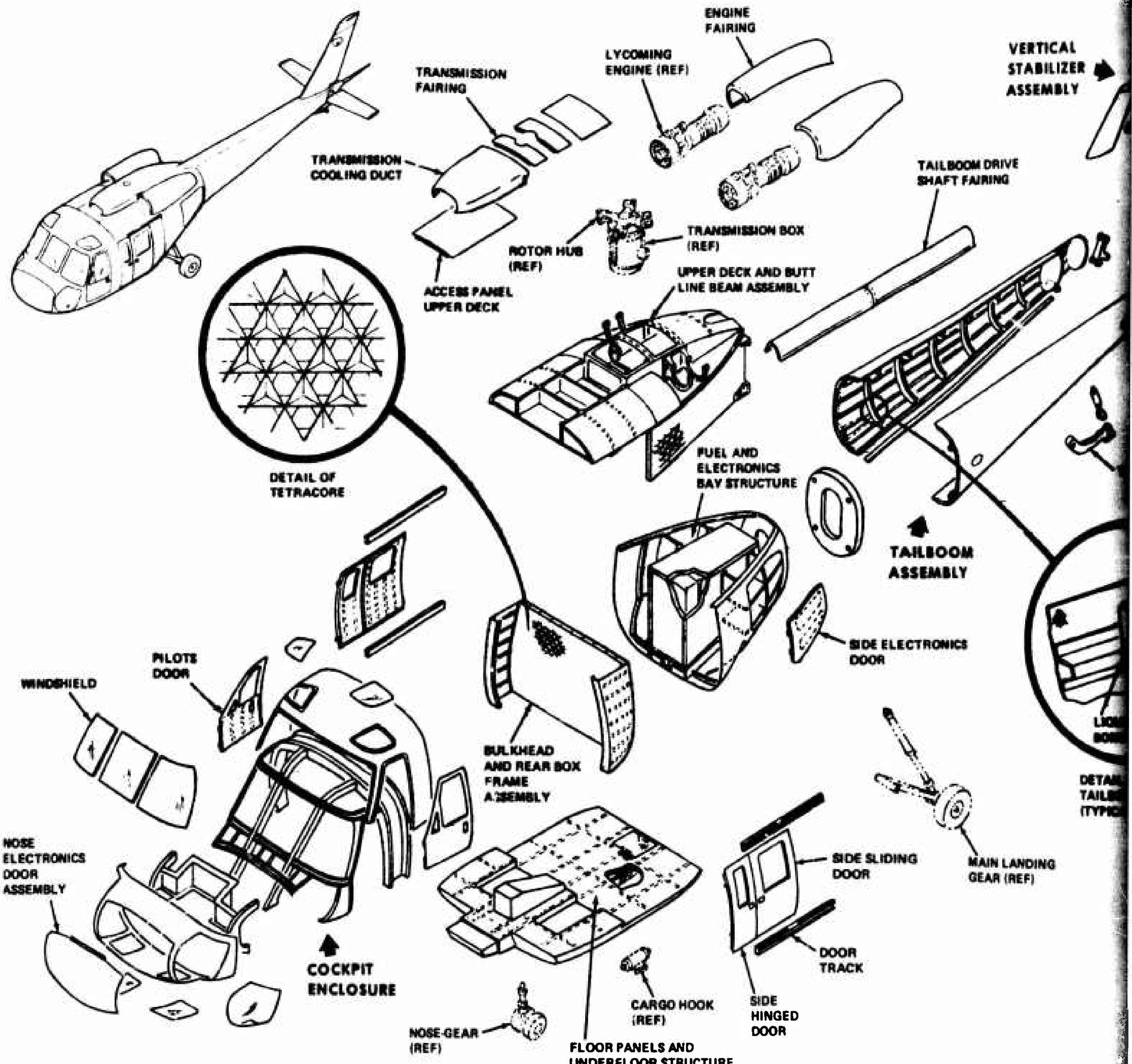
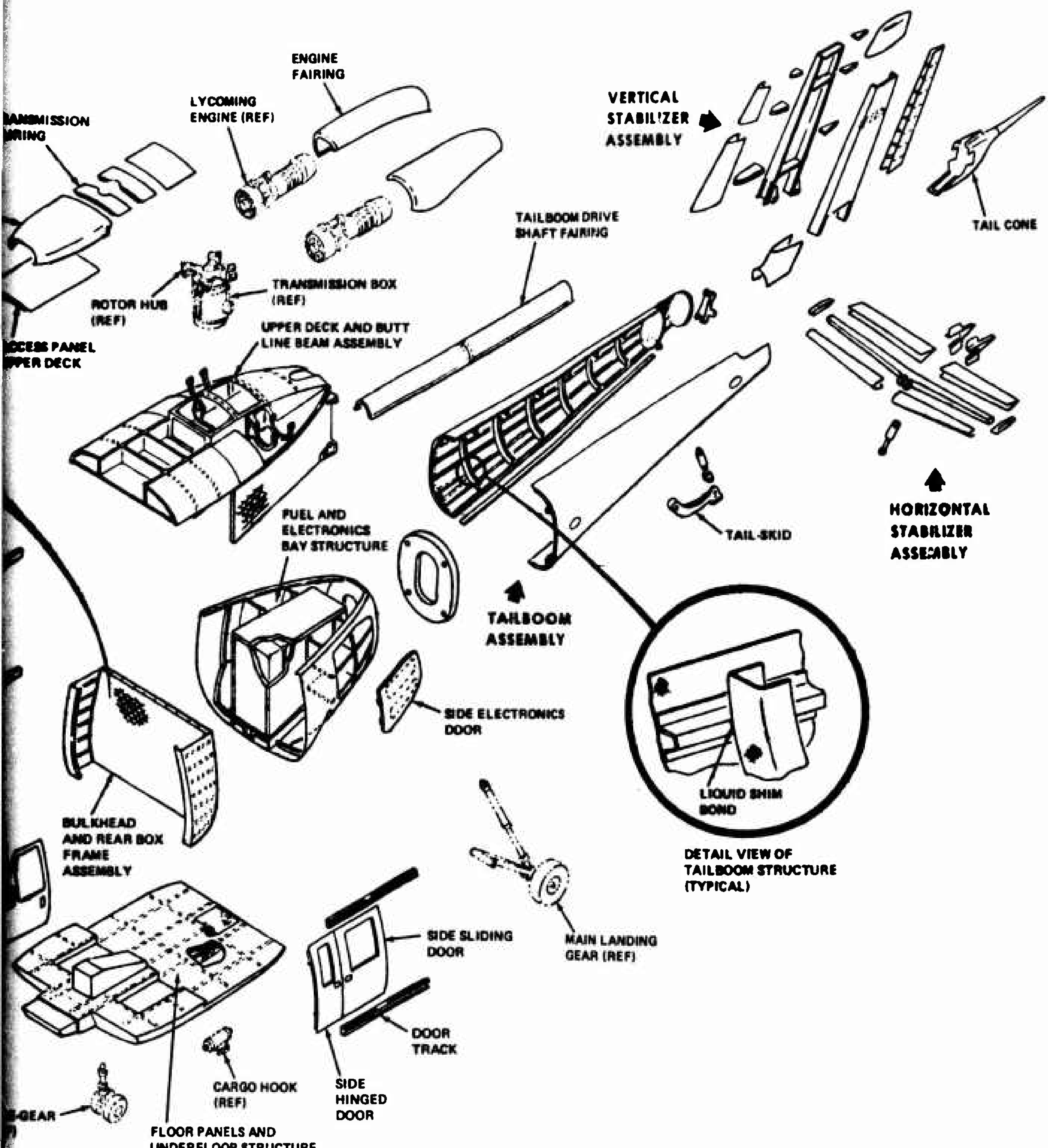


Figure 28. Concept L - Graphite AS/Kevlar Epoxy Skin/Stringer.



e AS/Kevlar Epoxy Skin/Stringer.

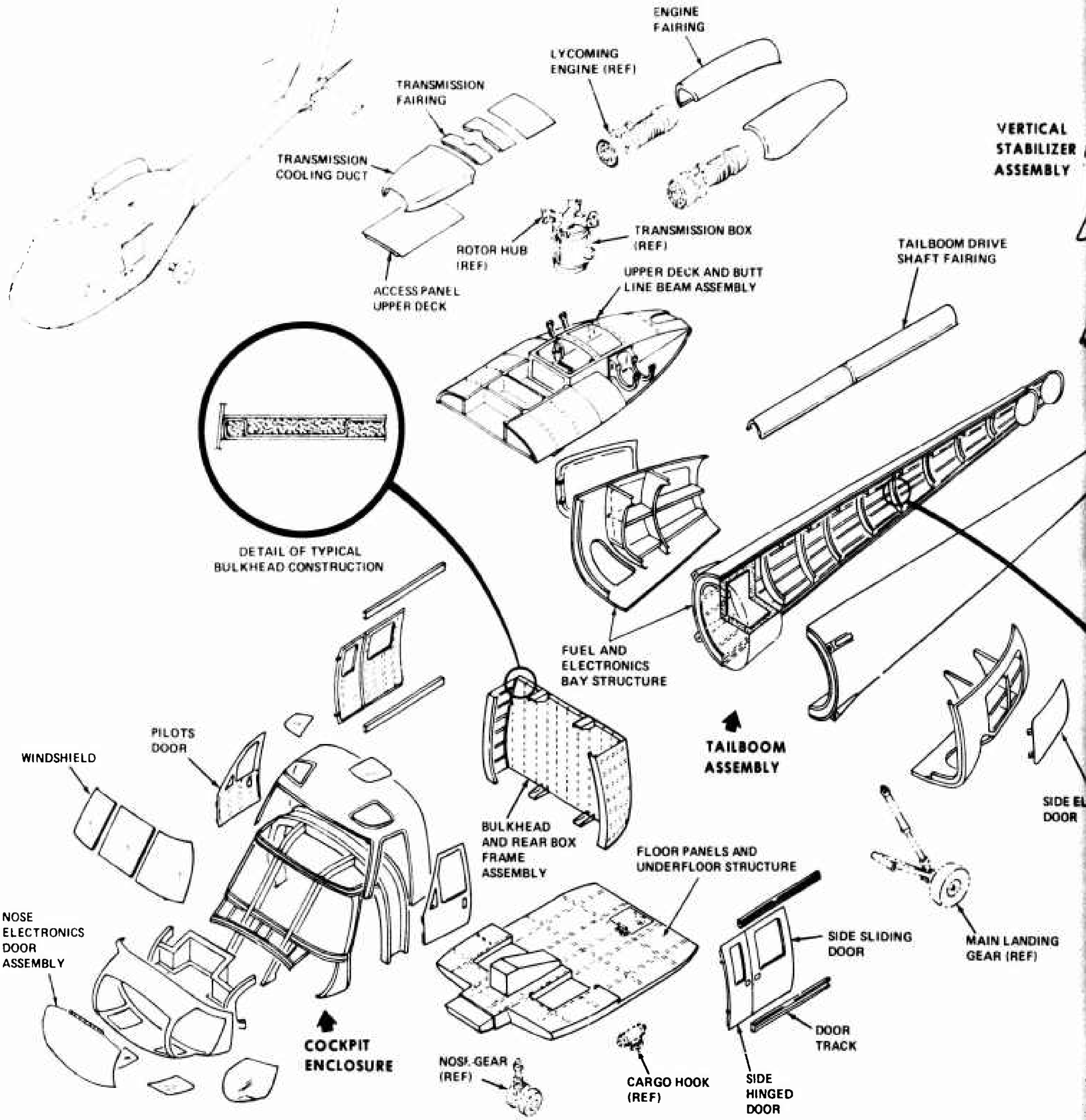
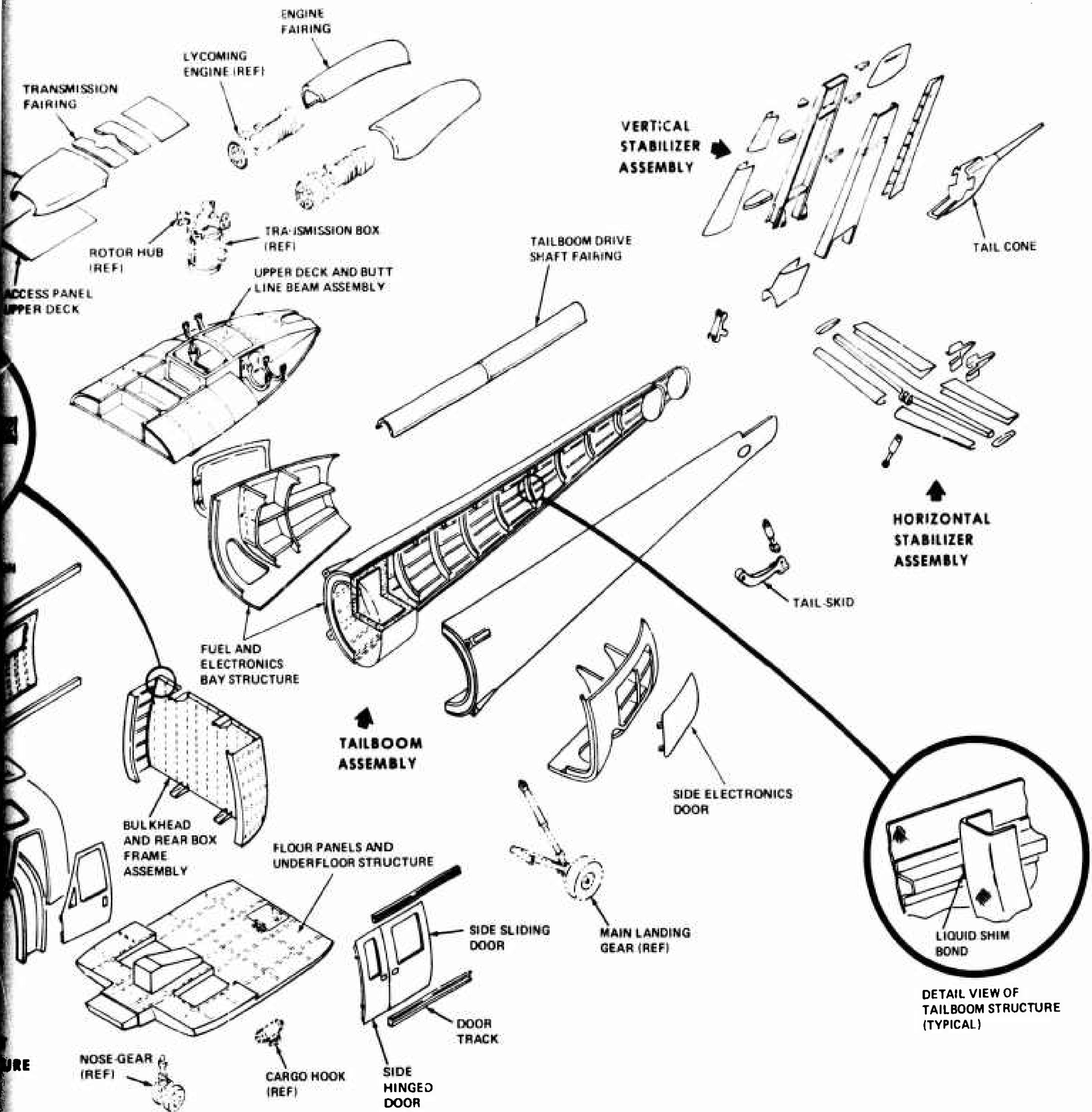


Figure 29. Concept M - Graphite AS/Kevlar Epoxy Skin/Stringer.



raphite AS/Kevlar Epoxy Skin/Stringer.

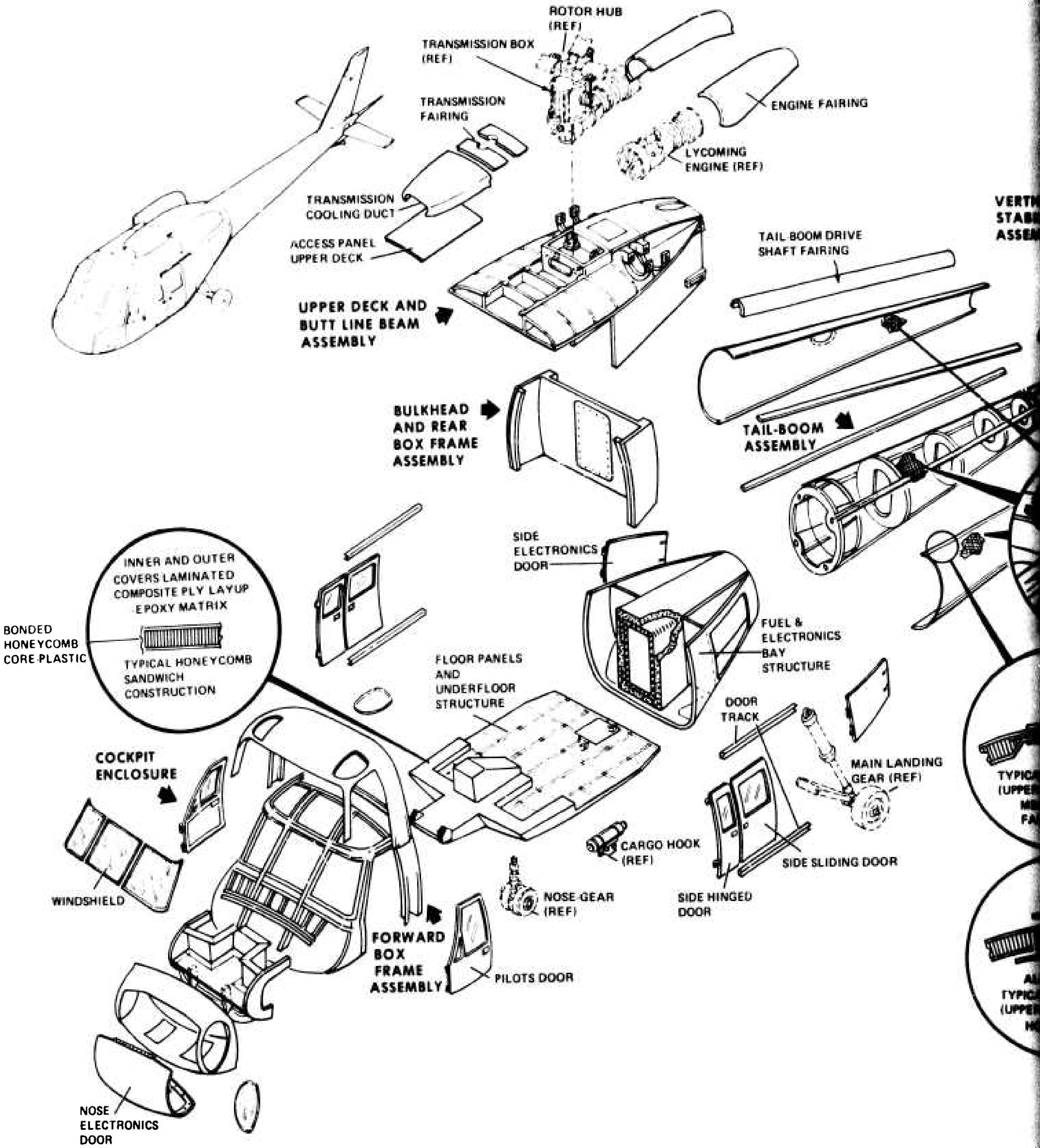
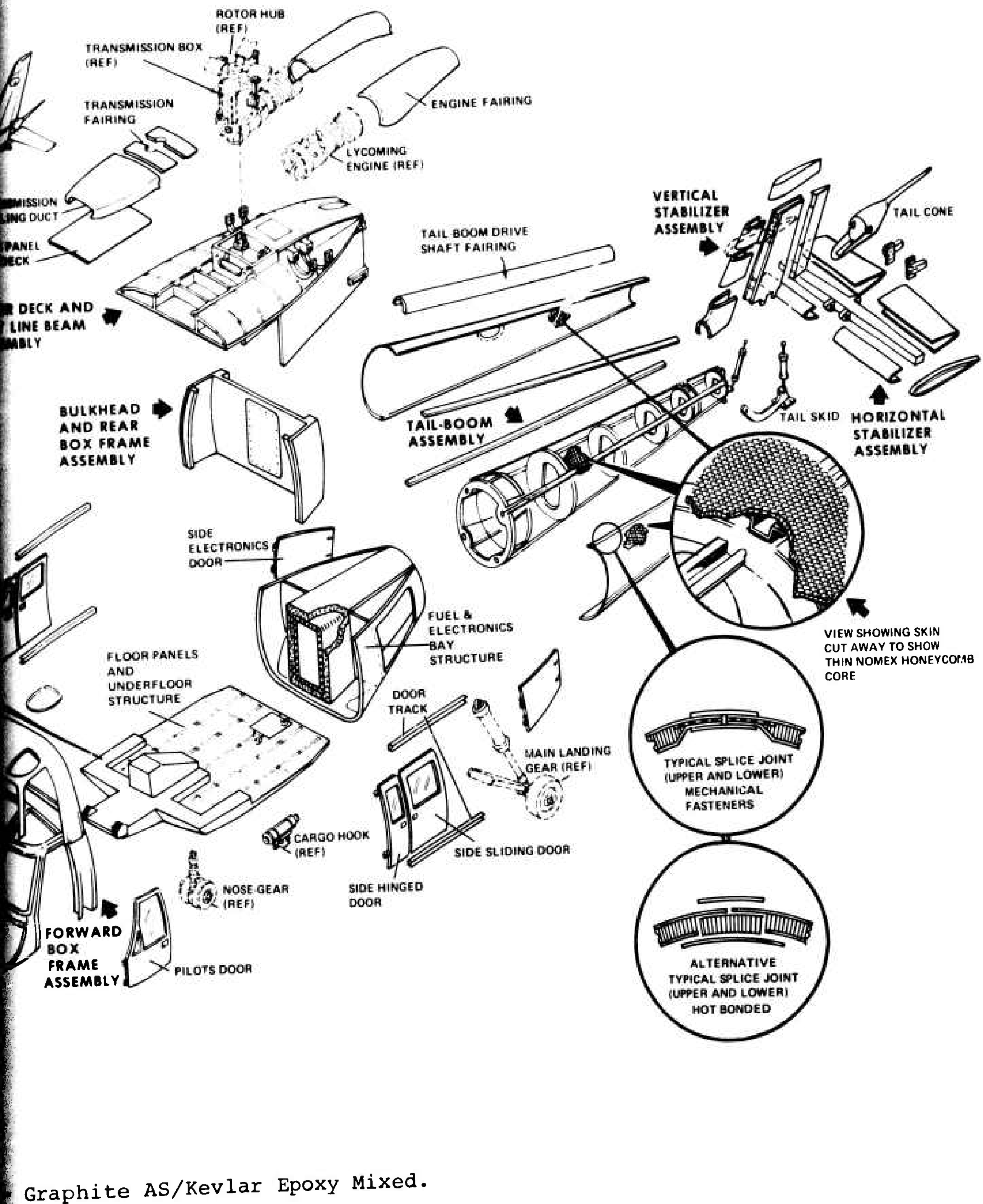


Figure 30. Concept Q - Graphite AS/Kevlar Epoxy Mixed.



Graphite AS/Kevlar Epoxy Mixed.

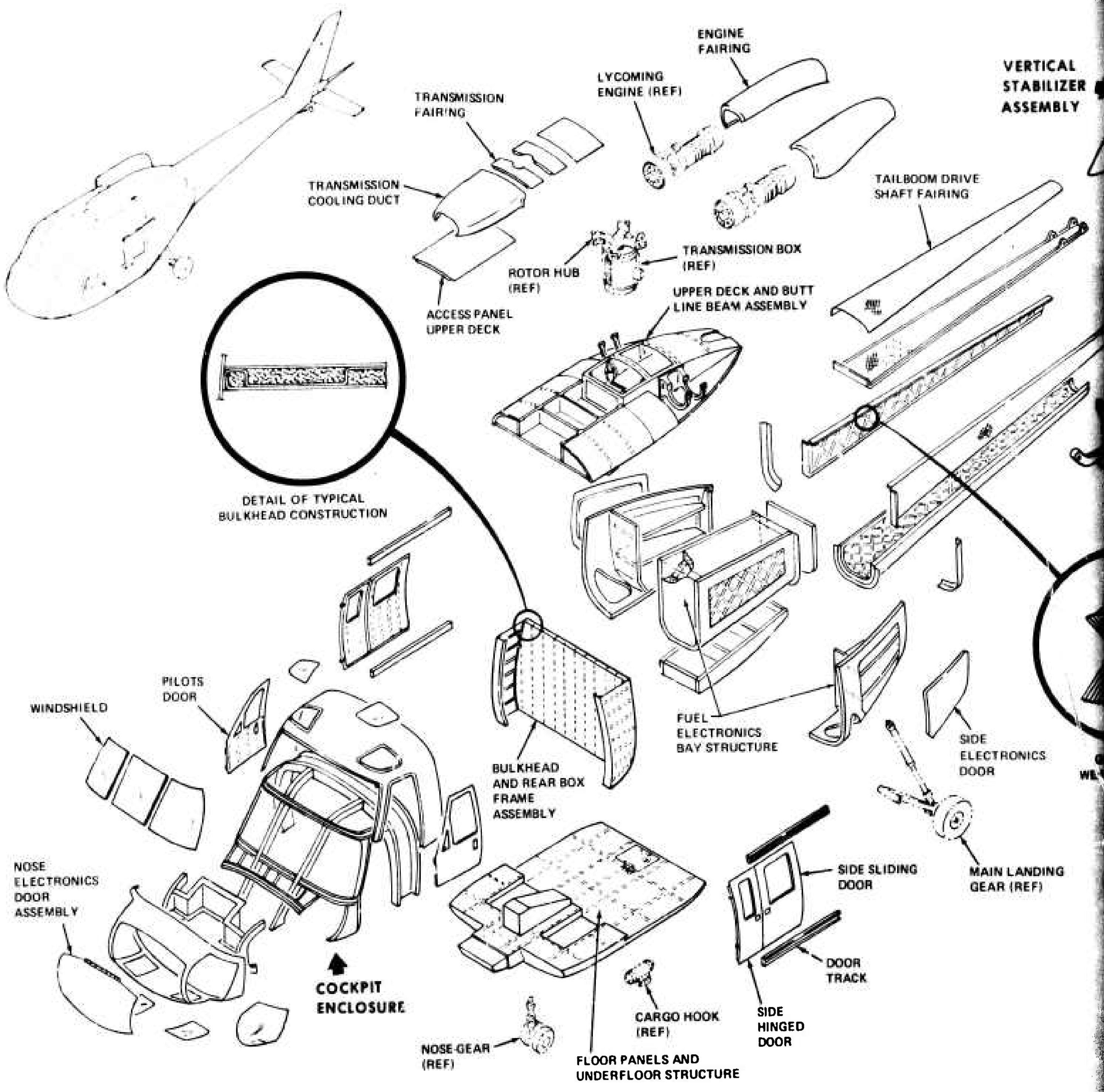
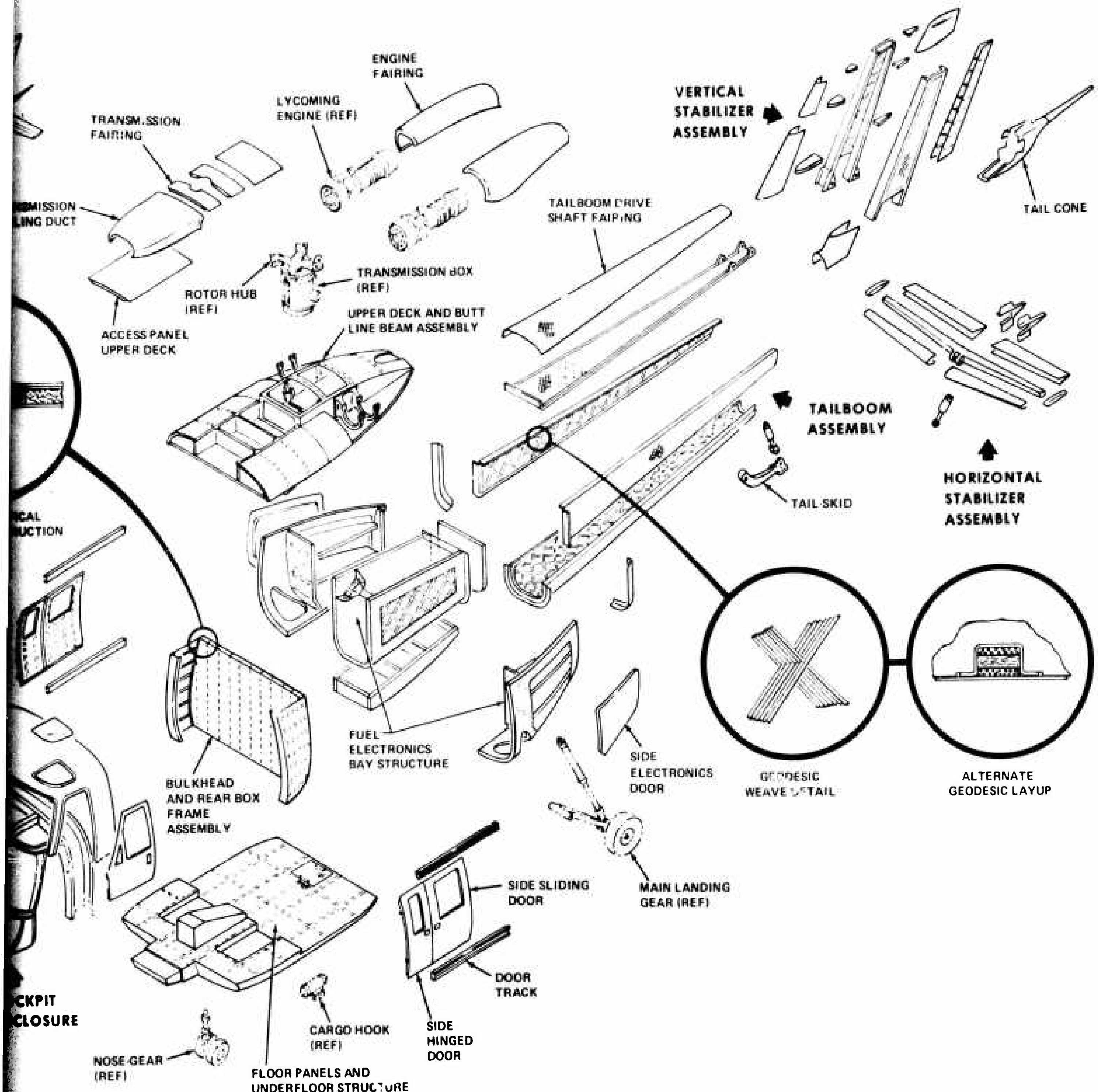


Figure 31. Concept R - Graphite AS/Kevlar Epoxy Mixed.



Graphite AS/Kevlar Epoxy Mixed.

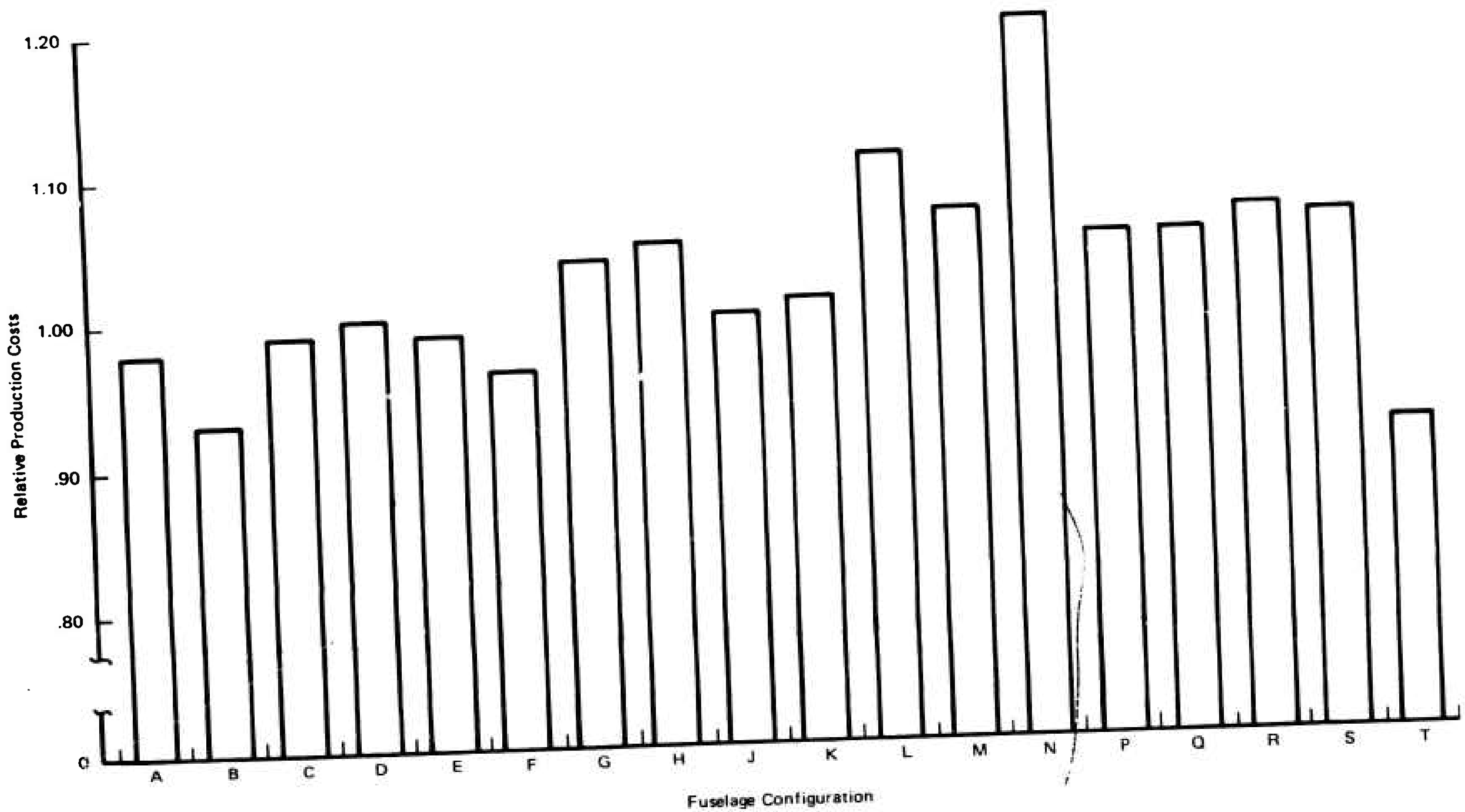


Figure 32. Relative Cost of Fuselage Structural Arrangements.

3.2 DRIVE SYSTEM

Drive system structural efficiency was addressed principally through a mechanical design study of the main transmission; the concepts and ratings are shown in Table 8. Interconnect shafting was rated separately in Section 3.3.

USAAMRDL authorized a separate study of a Curtiss-Wright free planetary drive transmission, sized for the MUT configuration, as part of this contract. This design study and the conclusions drawn are described in USAAMRDL Technical Report 75-56B, Design Application Study for Free Planet Transmissions.

3.2.1 BOEING VERTOL ADVANCED TRANSMISSION DESIGN

Boeing Vertol has conceived an advanced transmission design which is believed to be lighter and less expensive than the baseline state-of-the-art configuration. This system contains features considered proprietary at this time, and the description is reported separately. Comparison factors are presented in Volume II.

Table 8 compares the baseline transmission to the Boeing Vertol advanced transmission and the Curtiss-Wright free planetary drive advanced transmission.

3.2.2 SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION - TRANSMISSION

Upon completion of the evaluation of the free planet system application to the MUT and a comparison with the Boeing Vertol advanced transmission concept, it was decided to recommend the latter, since weight reduction is closer to 20 percent than the 10 percent realized with the free planet drive, and the parts count and associated cost are reduced.

Table 8. Concept Screening Analysis – Main Rotor Transmission

- OBJECTIVE: Select best transmission concept for MUT
- COSTS: Based on 1,000 units

		BASELINE	CONCEPT A	CONCEPT B
Rating factors will be comparative to baseline structure counterpart 2 = MUCH BETTER 1 = BETTER 0 = SAME –1 = POORER –2 = MUCH POORER CONFORMANCE FACTORS:– BETTER (+) REFERRED SAME (0) TO POORER (–) BASELINE		CONVENTIONAL TRANSMISSION	BOEING VERTOL ADVANCED CONCEPT TRANSMISSION • Stiff fail-safe housing • Reduced weight • Reduced cost • Reduced number of bearings • Close coupled hub • Structural monitoring system • Load-balanced planet carrier	CURTIS-WRIGHT FREE PLANETARY CONCEPT TRANSMISSION USAAMRDL Tech Report 74-27 defines general configuration. (See TR 75-56B, Design Application Study for Free Planet Transmissions)
RATING ELEMENTS	STRUCTURAL EFFICIENCY	0	2	1
	FAIL SAFETY	0	0+	0+
	SAFETY	0	0+	0+
	PRODUCIBILITY/COST	0	1+	0–
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	0	0	0
	CRASHWORTHINESS	0	+	0
	REPAIRABILITY	0	0	+
	MAINTAINABILITY	0	+	+
	SURVIVABILITY	0	+	+
	GENERAL SAS CONFORMANCE	0	0	0
A/F CONCEPT RANKING		(3)	(1)	(2)

3.3 DRIVE SHAFTING CONCEPTS (Refer 9)

The advanced system candidates considered for drive shafting evolved from an evaluation of the best means of incorporating graphite/epoxy materials in place of aluminum.

Previous studies on the HLH program, as well as other studies by industry and U.S. Army investigators, established that boron/epoxy, combinations of fiberglass, Kevlar, and graphite, and all graphite shaft concepts are principally advantageous in increasing the length of individual shaft segments without going into supercritical shafting frequency ratios at operating speeds. The weight savings in the tubes is compounded by elimination of shaft elements, adapters, bearings and flexible couplings (see Figure 33).

AS type graphite properties have been selected as representative of the most structurally efficient and cost competitive material. It offers relatively low risk when filament wound with a properly compounded epoxy resin, where low impact velocity and ballistic impact damage are considered.

This study evaluated straight shafting with adapters (Concept A); shafting with integral flanges (Concept B), which eliminate adapters (see Figure 34); and shafting with integral flanges and integral flexures (Concept C, Figure 35), which eliminate both adapters and flexible couplings, allowing tail rotor shaft elements to be combined and mounted in split bearings at intervals determined by shaft geometry elastic properties and operating speed.

In applications where length is dictated by drive system components, such as the high-speed engine shafts and the tail rotor shaft between the intermediate and tail rotor boxes, advantages of composite shafting are not enhanced by capability to go to longer shaft elements and to eliminate intermediate supports. In these applications, judgments must be made on the basis of shaft weight reduction, and elimination or simplification of end attachments and flexures.

SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION - SHAFTING

Concept B (Table 9, Figure 33) was selected because of lack of demonstration of integral flexure in graphite shafting. Use of filament winding, elimination of end adapters, and a reduction in shaft segments offer improved structural efficiency and the best opportunity for cost-competitiveness without undue developmental risk.

Table 9. Concept Screening Analysis – Tail Rotor/Engine Drive Shafting

Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1–	
–2	MUCH POORER

CONFORMANCE FACTORS:–

BETTER + } REFERRED
SAME 0 } TO
POORER – } BASELINE

- OBJECTIVE: Selection of best shaft concept for MUT
- COSTS: Based on 1,000 units

		BASELINE	CONCEPT A	CONCEPT B	CONCEPT C		
		SCALED DOWN UTTAS	COMPOSITE SHAFT	COMPOSITE SHAFT WITH INTEGRAL FLANGES	ONE PIECE COMP SHAFT WITH SPLIT BRGS		
		• Riveted aluminum • Flexible steel couplings between sections Bearing hangers and supports Bearings	Same as baseline. Substitute composite for aluminum. Decrease number of shafts	Same as A but eliminates weight, cost, and failure modes of riveted adapter fittings	Same as B but shaft assembly made in one long unit. Flanges are eliminated and split bearings are installed adjacent to flexures at intervals established for subcritical operation		
RATING ELEMENTS	STRUCTURAL EFFICIENCY	0	1	1	2		
	FAIL SAFETY	0	1–	1	2		
	SAFETY	0	0	0	0		
	PRODUCIBILITY/COST	0	–1	1	2		
	RATING TOTAL		1–	3	6		
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	0	+	+	+		
	CRASHWORTHINESS	0	+	+	+		
	REPAIRABILITY	0	0	0	–		
	MAINTAINABILITY	0	+	+	+		
	SURVIVABILITY	0	+	+	+		
	GENERAL SAS CONFORMANCE	0	0	0	0		
	CONCEPT RANKING		3	2	1		

General Note: Composite shaft design expected to be graphite/epoxy method of construction under investigation by USAAMRDL EUSTIS

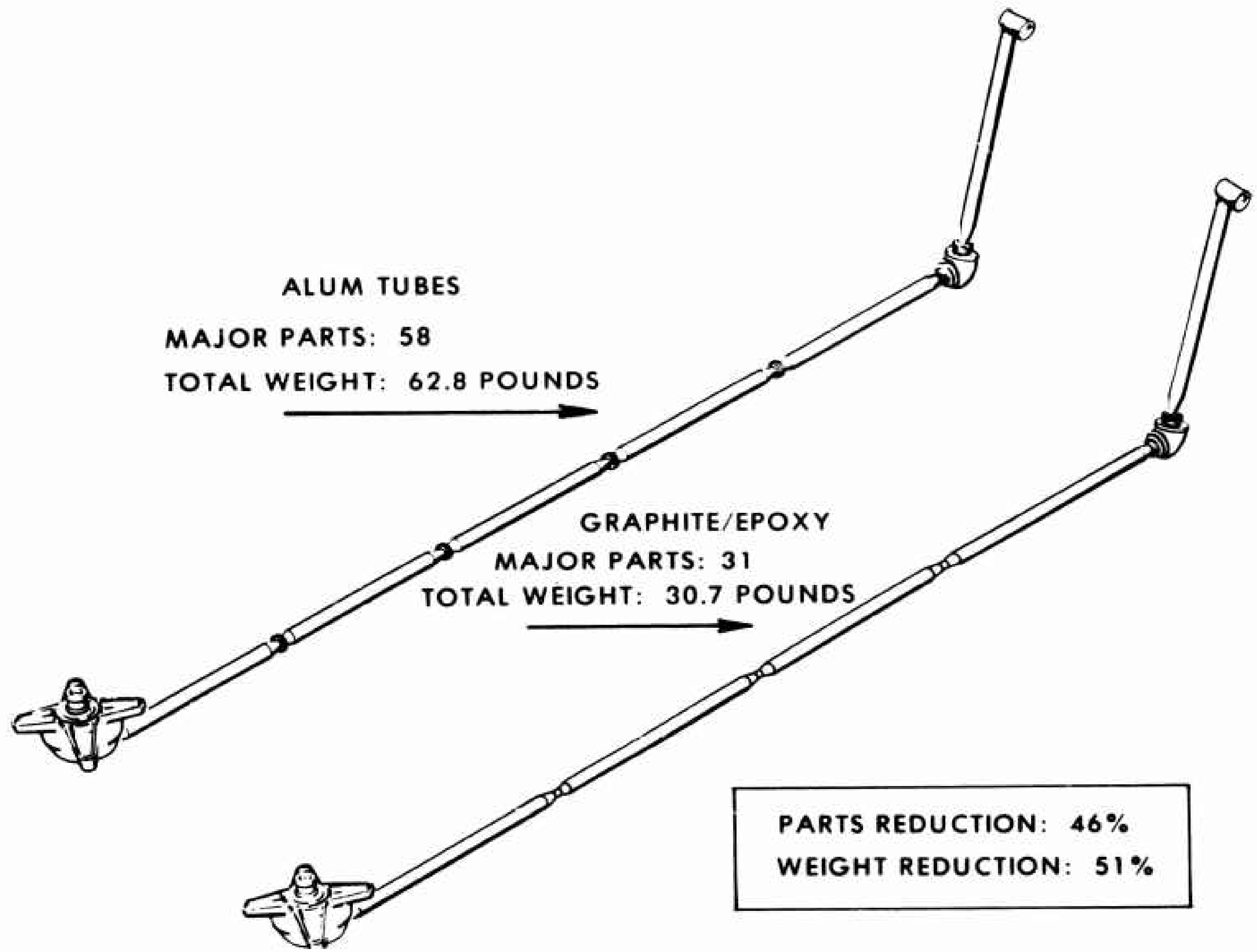


Figure 33. Tail Rotor Shaft Improvements (UTTAS Sized).

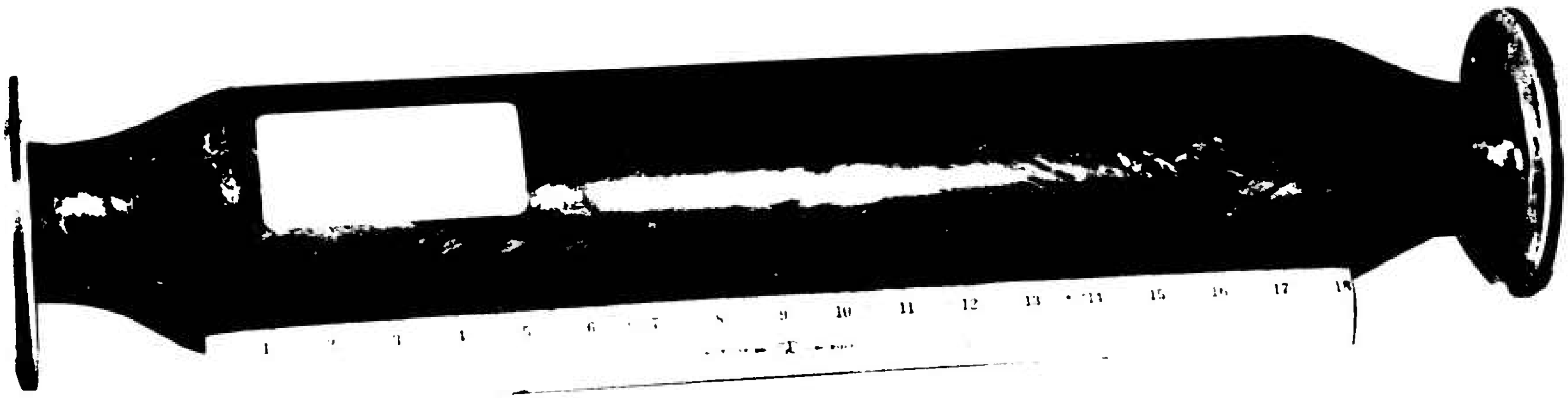


Figure 34. Concept B - Graphite-Epoxy Integral Flange Drive Shaft Test Specimen.

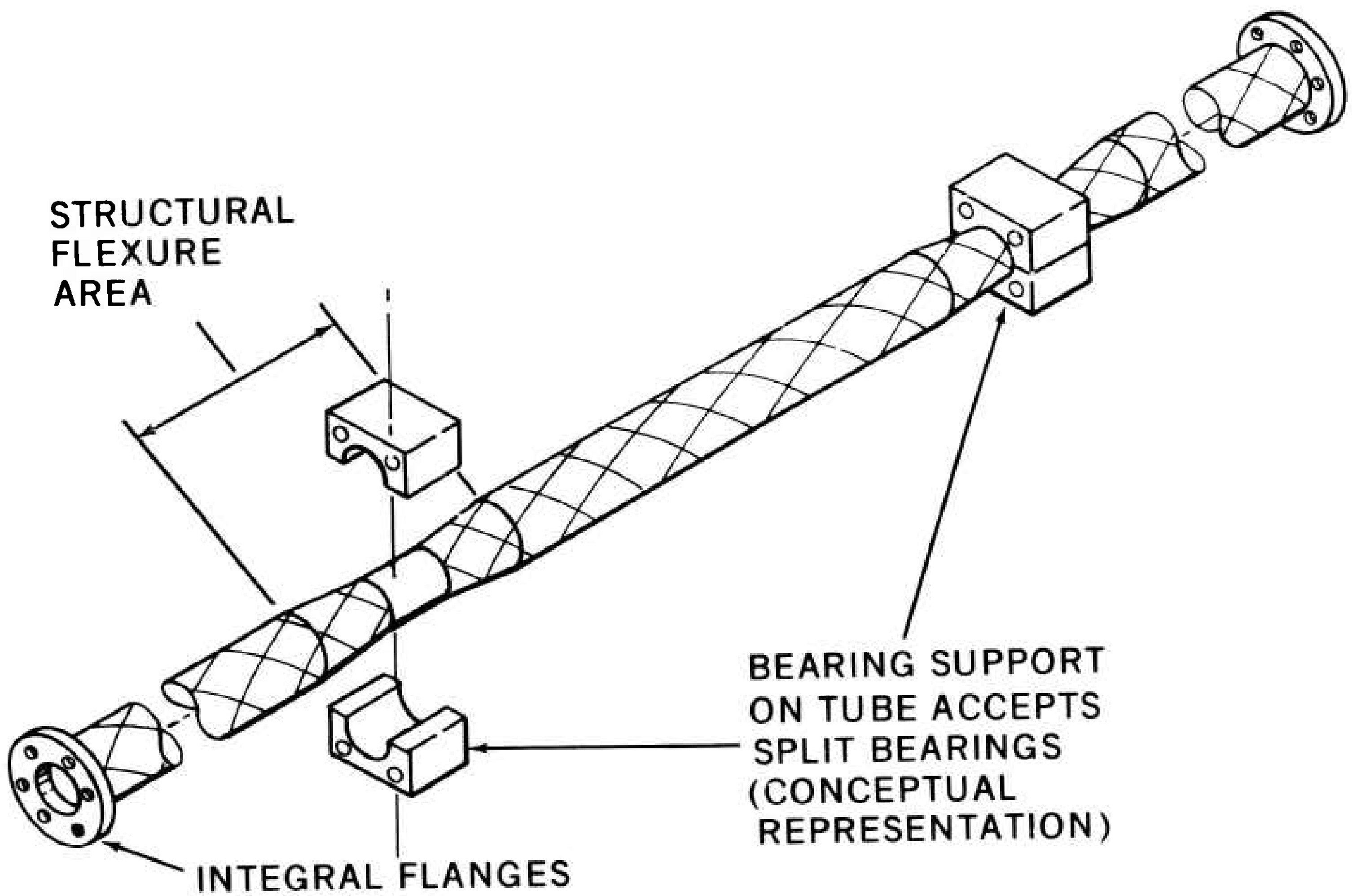


Figure 35. Concept C - Graphite-Epoxy Shaft with Tailored Laminates.

3.4 CONTROL SYSTEM CONCEPTS (Refer to Table 10)

The baseline aircraft employs a dual mechanical flight control system similar to the YUH-61A, with metal push rods, torque tubes, and bellcranks (see Figure 9). Concept A is configured in a similar manner but uses selected composite components.

Composite components have been developed for survivability improvement under U.S. Army auspices (see Figure 36). One of the problem areas is bearing vulnerability, and concepts have been developed which permit operation after ballistic impact on bearings.

Accordingly, an advanced candidate system using a single mechanical system was considered (Concept B, Figure 37).

Another concept was considered (Concept C) wherein the longitudinal, lateral, and collective control systems were retained as dual mechanical systems, while the directional system, considered less critical, used selected composite components.

Concept D (Figure 38) proposes a single mechanical system employing composite components, which is backed up for safety, fail-safety, and survivability improvement by a single electrical fly-by-wire system developed for the Boeing Vertol 347 experimental helicopter.

Concept E is similar to Concept D except that it employs a single metal mechanical system with fly-by-wire backup.

Concept F eliminates the mechanical control system and replaces it with a triply redundant fly-by-wire system similar to that developed for the U.S. Army HLH.

Some of the advantages cited from previous Army studies on composite control system components are listed in Table 11.

SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION - CONTROL SYSTEM

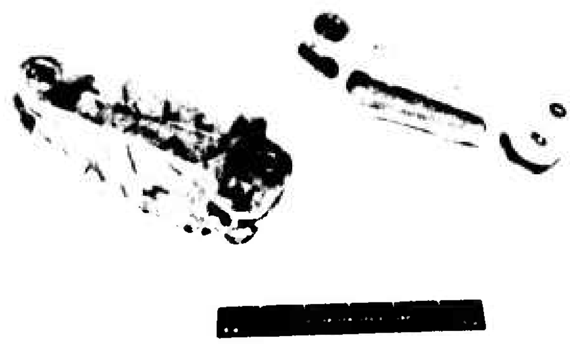
The single mechanical system with fly-by-wire backup, Concept D (Table 7, Figure 38), was selected. This system will use composite bellcranks and torque tubes, but the push-pull tubes will remain metal. A number of considerations were debated in selecting this system for preliminary design. The fly-by-wire backup protects the system against mechanical "opens," but the electrical actuator cannot override a jam in the mechanical system, since the input spool or valve to the hydraulic actuators will be immobilized. However, Boeing Vertol field experience has not revealed a single case of mechanical jam, and CH-47 system tests with deliberate disconnections of push-pull tubes have failed to produce jams at critical locations. This problem can be handled in detail design.

Another discussion centered around recommendation of an all-composite single system, Concept B, versus a system with fly-by-wire backup. Safety, fail-safety, survivability, mission reliability, and operational reliability are improved in Concept D, while maintenance reliability may be somewhat reduced due to the electrical components. Composite push-pull rods were not judged to be structurally efficient or cost-competitive with metal rods; and increasing rod diameter in metal can improve survivability. The weight and cost of the fly-by-wire backup is substantial, however, and might trade off against a single survivable all-composite system. More flight control system work is recommended to fully evaluate all trade possibilities.

Concept D was finally selected as a compromise approach with limited associated risk.

Table 10. Concept Screening Analysis – Flight Controls

Rating factors will be comparative to baseline structure counterpart 2 MUCH BETTER 1+ 1 BETTER 1– 0 SAME –1+ –1 POORER –1– –2 MUCH POORER CONFORMANCE FACTORS:– BETTER + } REFERRED SAME 0 } TO POORER – } BASELINE	• OBJECTIVE: To select best flying control system • COSTS: Based on 1,000 units • BASELINE CONTROL SYSTEM: Dual mechanical UTTAS type all metal system					
	CONCEPT A	CONCEPT B	CONCEPT C	CONCEPT D	CONCEPT E	CONCEPT F
	DUAL MECHANICAL COMPOSITE SYSTEM Arrangement per Figure 9 with control tubes as Figure B–17. Concept 3 and chopped strand molded idler bellcranks as Figure 36. (Illustrations on left-hand side of the page)	SINGLE MECHANICAL COMPOSITE (FAIL-SAFE AND BALLISTIC TOLERANT) Arrangement per Figure B-22. Control tubes as Figure B-17 Concept 3	HYBRID DUAL MECHANICAL SINGLE DIRECTIONAL CONTROLS SELECTED COMPOSITE COMPONENTS Hybrid control tube design as Concept 1 of Figure B–17 (steel fork end fittings composite tubes) Selected composite bellcranks (chopped strand molded with peripheral Uni windings).	SINGLE COMPOSITE MECHANICAL AND BACKUP FLY-BY WIRE SYSTEM Single mechanical as Figure B-22 but with composite control tube arrangement per Figure B-17. Regular control tube design (top sketch) and bellcranks composite epoxy/fiberglass sheet and square tube stock Nema grade per USAAMRDL report 73-20, pages 7 and 9 Fly by wire back up	SINGLE METAL MECHANICAL AND BACKUP FLY BY WIRE SYSTEM As Figure B-22 but with metal control tube design per Figure B-17 regular control tube design (upper picture and bellcranks etc. Fly by Wire backup	DUAL FLY BY WIRE SYSTEM Fly by wire system as per HLH triplex self monitored on line
RATING ELEMENTS – STRUCTURAL EFFICIENCY	0	2	1–	1+	1	2
RATING ELEMENTS – FAIL SAFETY	1	–1	0	0	0	1
RATING ELEMENTS – SAFETY	0	–1	0	0	0	0
RATING ELEMENTS – PRODUCIBILITY/COST	–1	1	1–	0	0	-2
RATING TOTAL	0	1	1	1+	1	1
CONFORMANCE ELEMENTS – DETECTION AVOIDANCE	+	+	0	+	+	+
CONFORMANCE ELEMENTS – CRASHWORTHINESS	0	0	0	0	0	0
CONFORMANCE ELEMENTS – REPAIRABILITY	0	0	0	0	0	0
CONFORMANCE ELEMENTS – MAINTAINABILITY	0	+	+	0	0	+
CONFORMANCE ELEMENTS – SURVIVABILITY	+	–	0	+	+	+
CONFORMANCE ELEMENTS – GENERAL SAS CONFORMANCE	0	0	0	0	0	0
CONCEPT RANKING	3	2	2	1	2	2



A. Idler Bell Cranks

B. Aft Bell Cranks

C. Quadrant Assemblies

Figure 36. Comparison of Ballistic-Damage-Tolerant Flight Control Components with Their Metal Counterparts.

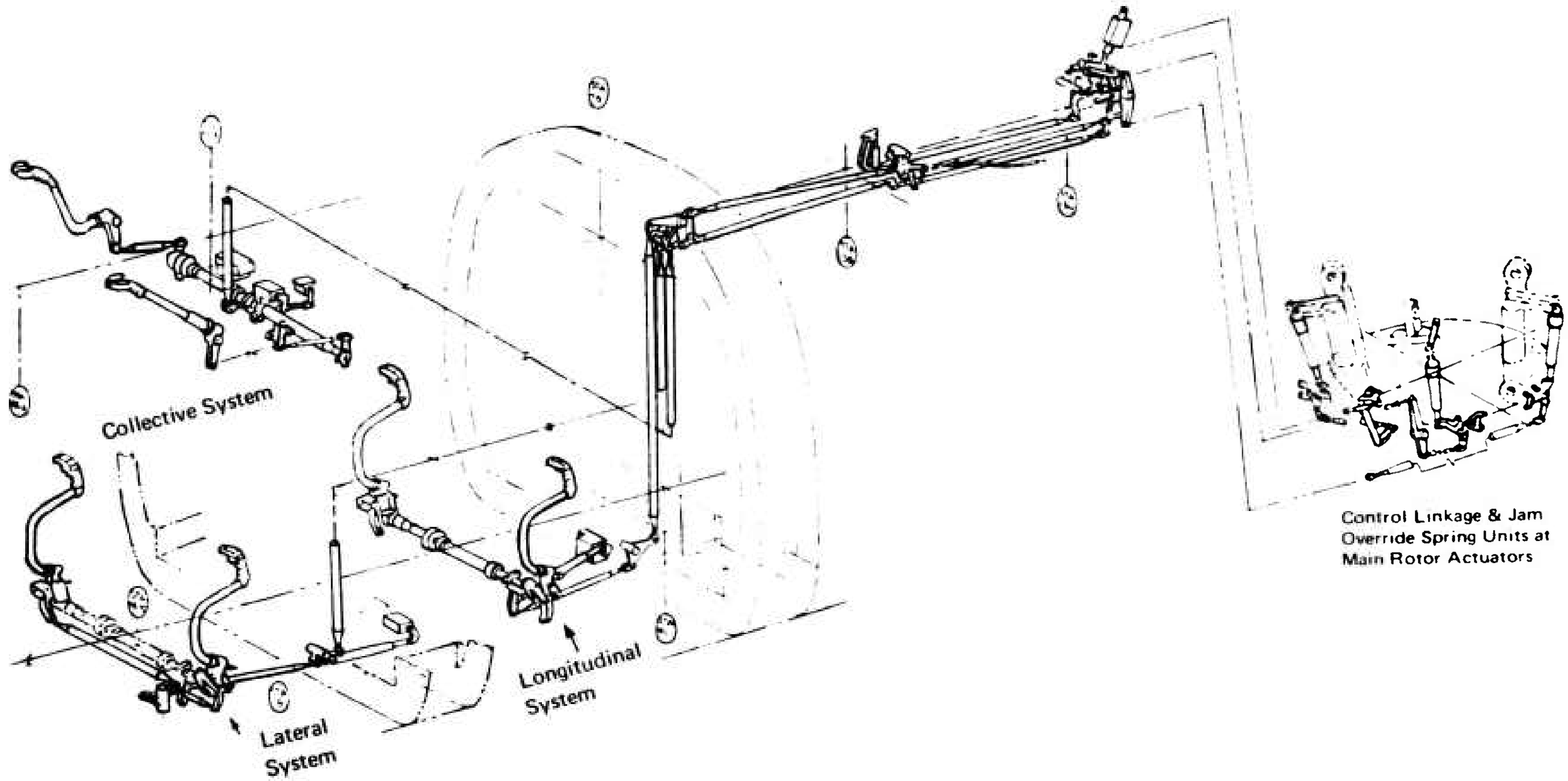


Figure 37. Single Mechanical Control System (Sheet 1 of 2).

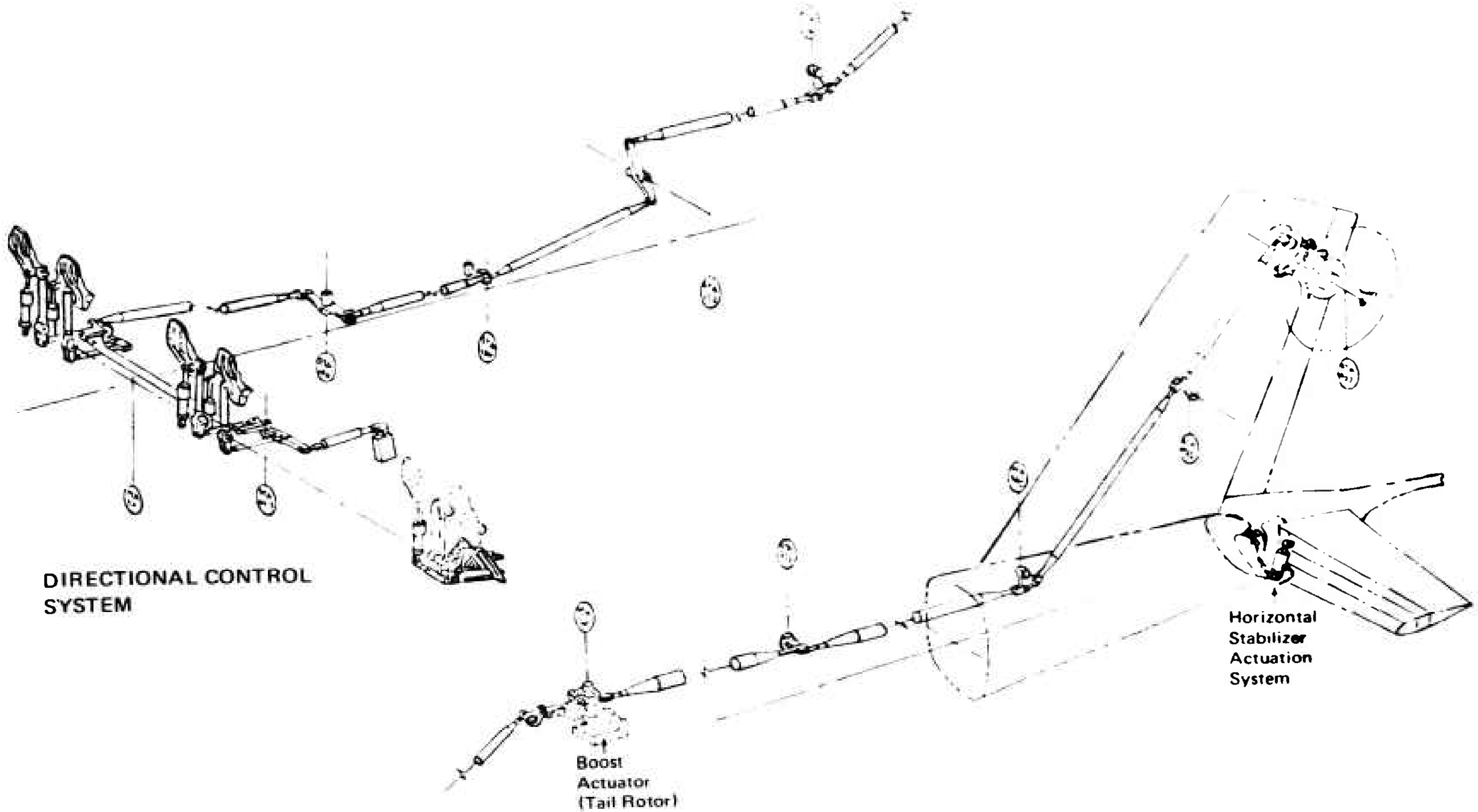


Figure 37. Single Mechanical Control System (Sheet 2 of 2).

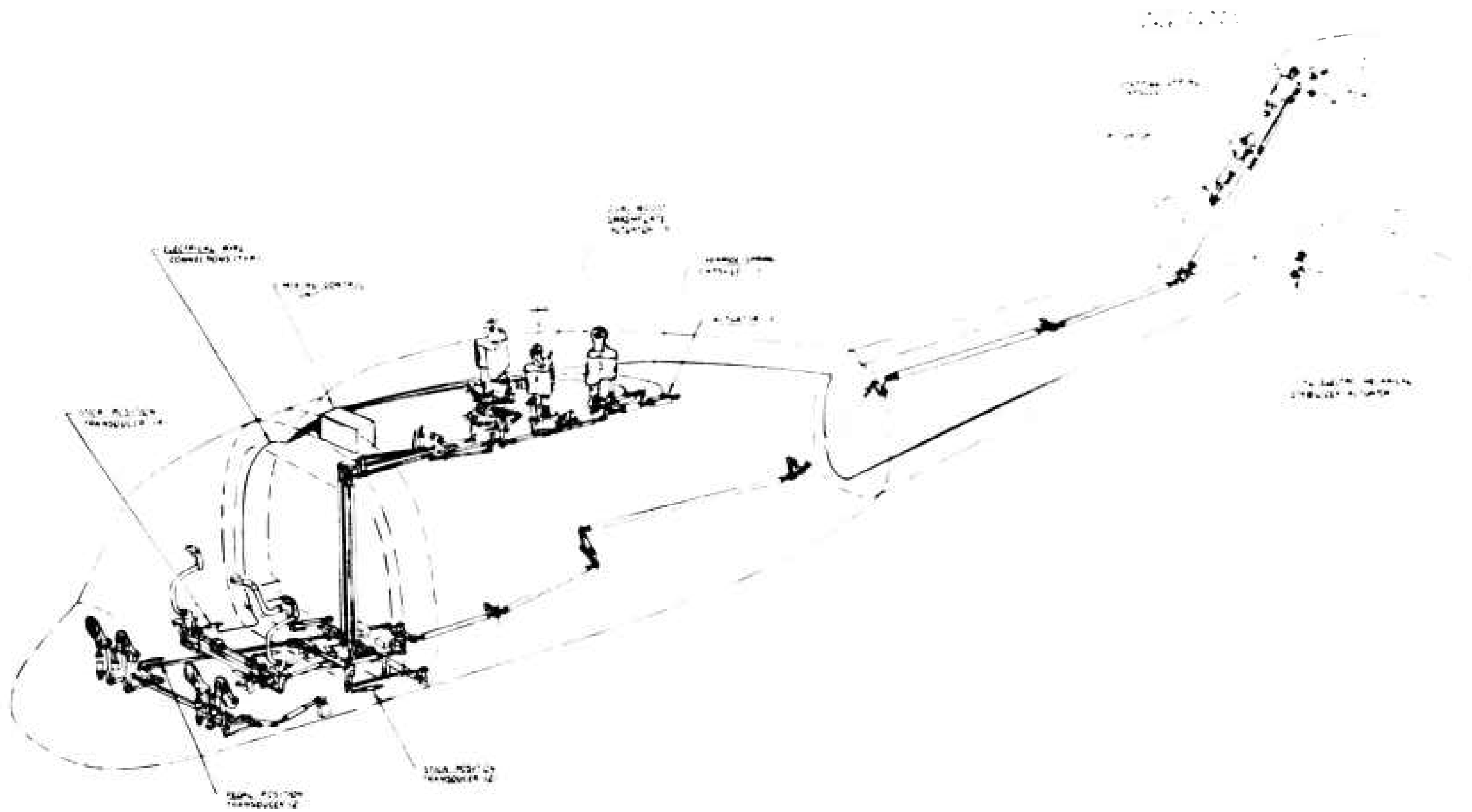


Figure 38. MUT Flight Control System (Single Mechanical With Single Electrical Backup).

TABLE 11. ADVANTAGES AND DISADVANTAGES FROM DESIGN OF FLYING CONTROL BELLCRANKS AND IDLER ARMS IN COMPOSITE MATERIALS

PROPOSED DESIGN: Chopped strand composite fibers in epoxy matrix with continuous uniperipheral wrap when required.

FABRICATION METHOD FOR PRODUCTION: Matched metal die mold.

ADVANTAGES (over baseline 7075 T73 aluminum alloy items)

1. Lighter weight items (12 to 25 percent) for similar strength and stiffness to 7075 items.

2. Cheaper to produce in quantity.

   Higher composite material costs offset by the various final machining operations necessary from basic metal forging plus extra corrosion protective treatments and heat treatment to T73 to alleviate stress corrosion also final finish treatments.

3. Lower environmental corrosion threshold.

4. No stress corrosion problems.

5. Increased ballistic tolerance.

6. Increased fail-safety: less chance of crack propagation, etc.

7. Easier procurement (quality aluminum alloy currently in short supply).

8. Shorter production time to produce items.

DISADVANTAGES

Initial tooling cost can be higher (heated match metal dies).

NOTE: The above is summarized from existing studies on composite designed flight control members in USAAMRDL technical reports.

3.5 ROTOR BLADE AND HUB COMPARISONS (Refer to Table 12)

The soft-in-plane hingeless system concept selected for the baseline configuration was retained.

3.5.1 ADVANCED BLADE CONCEPTS

The basic blade configurations considered were as follows:

Blade A

Baseline blade shown in Figure 11; fiberglass composite with advanced state-of-the-art features including load-sharing titanium leading edge.

Blade B

Scaled-up version of the BO-105 foam-filled fiberglass blade with a light protective titanium leading edge (Figure 39).

It was judged that a direct scale-up of the BO-105 blade concept resulted in a MUT blade which did not compare favorably in weight with the baseline (blade A), and the time required for layup of the larger MUT blade made the processing technology questionable from a materials shelf-life point of view.

Blade C

Multiple tube spar filament-wound composite blades based on the technology under U.S. Army development at Fiber Sciences, Inc.

Two blade design concepts for filament-wound multitubular rotor blades for the MUT helicopter were considered (see Figure 40). These, in combination with the concepts already fabricated by Fiber Sciences (see Figure 41), formed the base for establishing a design concept for an alternate blade to be considered in conjunction with the LTF advanced composite hub.

Two distinct spars with root end configurations have been identified which might satisfy the requirements of fail-safe structural properties, safety, survivability, and cost (see Figure 42).

It next became apparent that the hub configuration selection and the blade configuration selection could not be performed separately because of rating system incompatibilities.

3.5.2 ADVANCED HUB CONCEPTS

The following hub configurations were considered:

Hub A - Baseline Hingeless Hub (Figure 43)

This titanium hub, similar to the YUH-61A, is an articulated hub; it retains tension torsion straps and pitch bearings. The flap and lag hinges are eliminated, and effective hinging is established through flexibility of the inboard portion of the configuration A composite rotor blade.

Hub B - Co-Planar Lag-Torsion-Flexure Composite Hub (Figure 44)

This Boeing Vertol design eliminates all mechanical hinges and bearings. When used with the baseline transmission, the hub plate ties to the splined rotor shaft. The blade root end flexure requirements are eliminated. (This exemplifies why blade and hub must be considered as a system.) The blade attachment is made at about 20 percent radius, and a two-pin (moment transfer capability) attachment is required. Blade folding will be manual about one of the attachment pins. No lag dampers are required.

Hub C - Cross-Strap Lag Torsion Flexure Composite Rotor Hub (Figure 45)

This concept is similar to hub B (LTF), but the hub straps for pairs of opposing blades are continuous and cross at 90 degrees in separate planes. This design can be considered with detachable blades, or the hub can be integrated into the blade and the blades manufactured and assembled as tip-to-tip units comprising two blades with integrated carry-through hub flexures. Blades must be scissored rather than folded, and would require removal for transport.

3.5.3 RATING THE CONCEPTS

With the above blade and hub basic concepts defined, it is possible to rate the most attractive combinations against each other.

In preparing to rate the rotor system candidates, it became apparent that the selection of the transmission concept affected the choice of the hub concept. Hub Concepts A and B could adapt to a splined rotor drive shaft, but Concept C presented more difficulty, and the increased thicknesses due to the crossed straps was also a consideration in overall aircraft height and hub drag.

The rotor systems were grouped and rated as shown in Table 12. This table ranks the more attractive hub and blade combinations. Final configuration decision considers the application as a function of transmission/rotor shaft concept.

3.5.4 SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION - ROTOR SYSTEM

Concept E (Table 12, Figures 45 and 46) was selected, combining the composite lag-torsion-flexure hub with a simplified two-pin automated-tape-layup fiberglass rotor blade with titanium leading-edge protection.

Selection of this hub configuration for an advanced structural system was made in spite of lack of demonstration. Development work is in process or is planned by government and industry in the U.S. and in France. The dynamics of the soft-in-plane rotor and the hinge sequence and characteristics are understood. The system remains to be developed and demonstrated, however. The simplification and cost and weight reduction in the hub, and resulting weight reduction and simplification in the blade root end fabrication make the system a clear choice for an advanced structural system.

The LTF hub concept integrated into a tip-to-tip rotor blade configuration (Concept F, Table 12) shows even more promise, but this system entails higher risk and makes incorporation of filament-wound blade technology extremely unlikely. The filament-wound blade fabrication process under development by the U.S. Army may result in reduced rotor blade costs, but no adequate means of leading-edge protection is available for production application which will meet maintenance man-hour objectives. When this problem is resolved and sufficient data on interchangeability and flightworthiness is demonstrated, this system may be reevaluated for use with the LTF hub. The filament-wound blade concept is not believed viable with a soft-in-plane rotor hub of the YUH-61A or BO-105 type because of the laminate tailoring requirements in the blade root end to achieve effective flap and lag hinge elastic characteristics, although intensive design and development might evolve a suitable approach.

Table 12. Concept Screening Analysis — Rotor System (Hub Plus Blades)

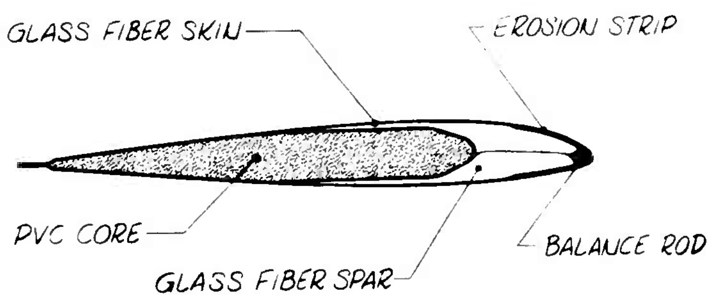
Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1–	
–2	MUCH POORER

CONFORMANCE FACTORS:– BETTER +, SAME 0, POORER – } REFERRED TO BASELINE

- OBJECTIVE: Select best rotor system concept
- COSTS: Based on 1,000 units

		CONCEPT A	CONCEPT B	CONCEPT C	CONCEPT D	CONCEPT E	CONCEPT F
		HUB A PLUS BLADE A Baseline hub plus Baseline blade	HUB A PLUS BLADE C Baseline hub plus Mts filament-wound blade (Difficulty in attaining flexural properties in blade root end for soft-in-plane rotor).	HUB A PLUS BLADE B Baseline hub plus Glass/epoxy and foam core blade scaled up from BO 105 process (No ISIS)	HUB B PLUS BLADE C Bearingless, hingeless, co planar composite hub with detachable blades. Mts filament wound blades. Flexibility in hub-blade is suited for mts concept two-pin attachment	HUB B PLUS BLADE A Bearingless, hingeless, co planar composite hub with detachable blades. Automated version of baseline tape layup blade with two pin attachment and no swan neck requirement. Pin wrap technology similar to HLH glass fiber blade. No metal cuff needed	HUB C PLUS BLADE A Crossed-strap bearingless hingeless composite hub with tip to tip rotor blades Blade is effectively produced version of baseline using automated tape layup Mts blade is not considered adaptable to this configuration
RATING ELEMENTS	STRUCTURAL EFFICIENCY	0	–1–	–1	1	1	2
	FAIL SAFETY	0	0	1+	0	0	0
	SAFETY	0	0	0	0	0	0
	PRODUCIBILITY/COST	0	–1	0	1	1	2
	RATING TOTAL	0	–2	–2+	2	2	4
CONFORMANCE ELEMENTS	DETECTION AVOIDANCE	0	0	0	+	+	+
	CRASHWORTHINESS	0	0	0	0	0	0
	REPAIRABILITY	0	0	0	–	–	0
	MAINTAINABILITY	0	0	0	++	++	++
	SURVIVABILITY	0	0	0	+	+	+
	GENERAL SAS CONFORMANCE	0	0	0	0	0	–
	CONCEPT RANKING	3	5	4	2	2	1



Cross-Section of Main Rotor Blade

Attachment of Main Rotor

Figure 39. Boelkow BO-105 Blade.

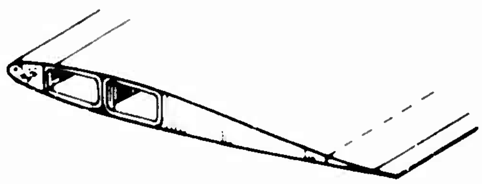
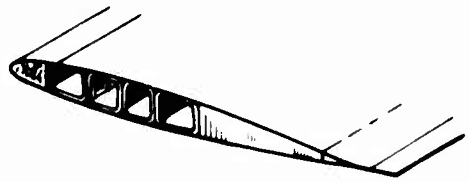
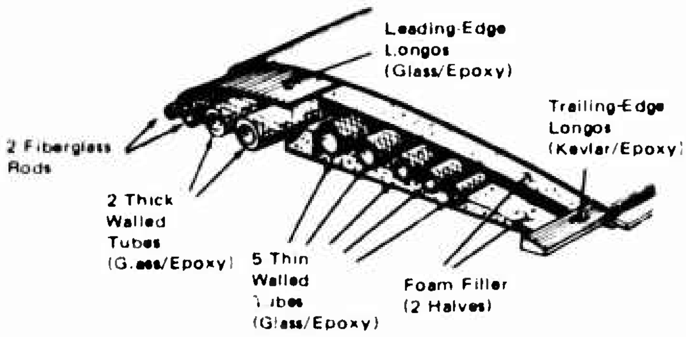
 CONCEPT I Two Tubes Side-by-Side (Dissimilar or Identical)	 CONCEPT II Four Tubes Side-by-Side (Dissimilar or Identical)
Low Fabrication Cost and Minimum Fabrication Complexity Good Ballistic Tolerance Provides the Most Efficient Use of Structural Material Compatible with the Greatest Number of Root End Concepts and Tip Concepts	Higher Fabrication Cost, More Tooling and Handling Complexity Very Good Ballistic Tolerance Less Efficient Structurally Because of Extra Webs Root End Concepts Require Structural Bridge

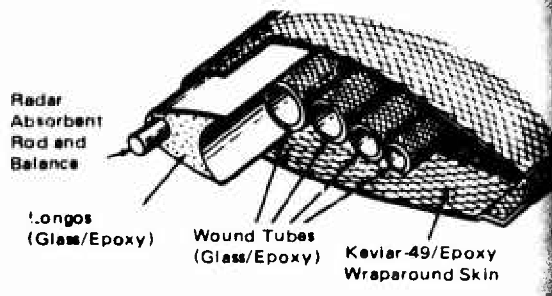
Figure 40. MTS Blade Concepts.

UH-1D

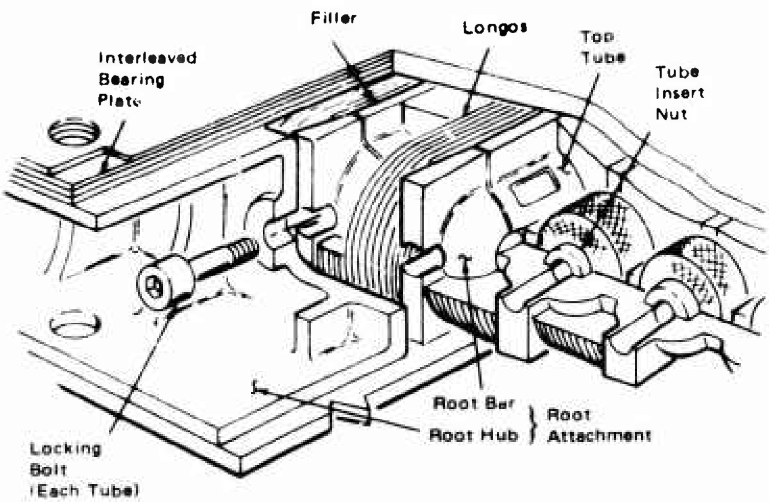


Typical Blade Cross Section

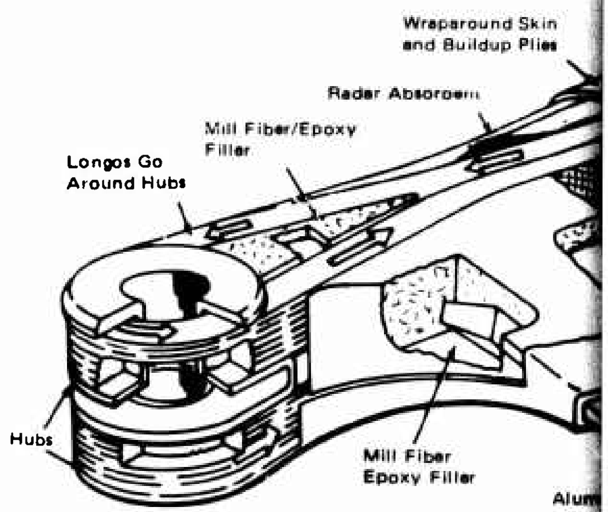
OH-6A



Typical Blade Cross Section



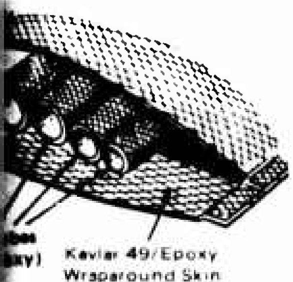
Root End Concept



Root End Concept

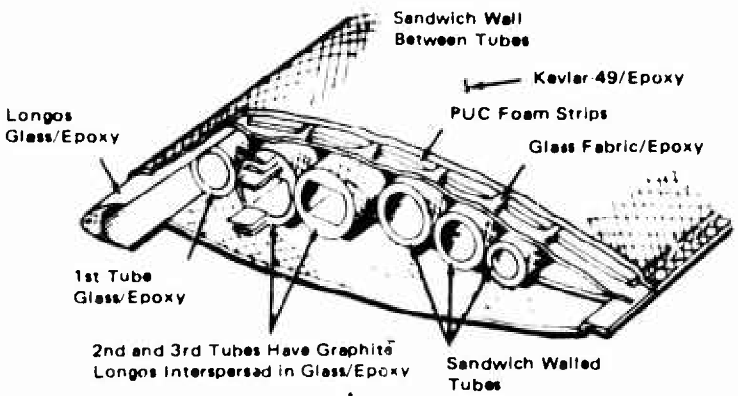
Figure 41. Blades Fabricated by Fiber Sciences.

OH-6A

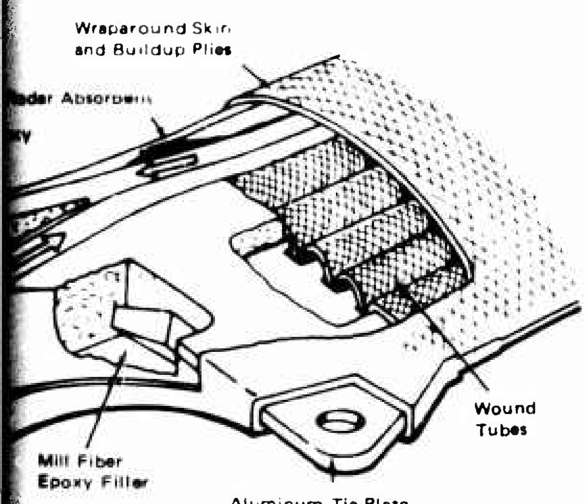


de Cross Section

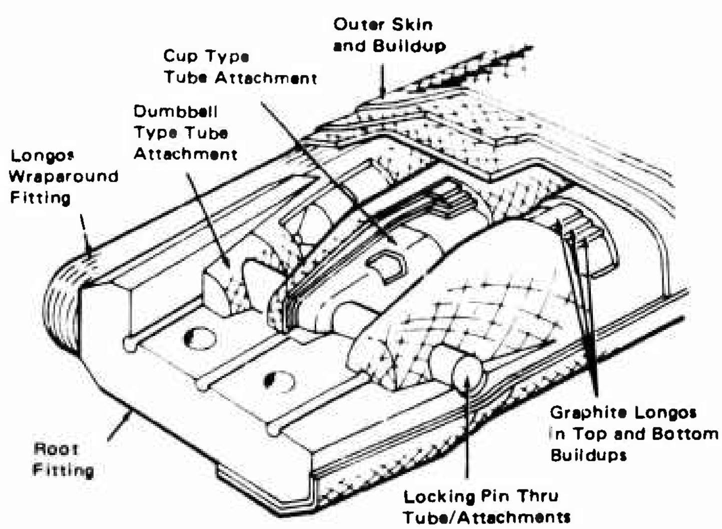
HLH



Typical Blade Cross Section



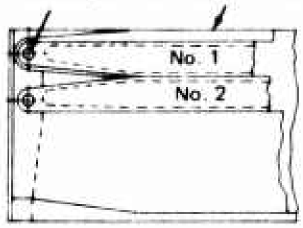
Root End Concept



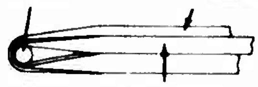
Root End Concept

1) Root Ends for the Two-Tube Spar Concept

a) Tube Wound Around Terminal Concept



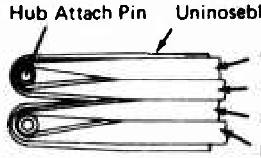
b) Full Tube Wraparound (Cane)



Root ends for the two-tube spars can be either wound terminal or cane type constructions.

- For cure of the spar tubes as a total spar, or within the cured outer skin and doublers, the wound terminal tubes are simply laid into heavy foam wedges and positioned on proper centers.
- The cane construction requires that the No. 1 tube be folded and wrapped around the pin in a counter clockwise direction; the No. 2 tube is laid next to the No. 1 tube and wrapped clockwise around the pin thereby capturing the free end of the No. 1 tube. The leading edge uni is then brought around the pin in a counter-clockwise direction, thereby capturing the free end of No. 2 tube. Additional fiberglass unidoubler pack is added to the cane arrangements before the No. 1 tube is wrapped (not shown for clarity). It must be wound around the terminal and be a part of the tube prior to assembly. This fiberglass material around the pin provides confidence to permit PRD-49 or Thornel 300 to be wrapped around the pin as retention material.

2) Root End for 4-Tube Spar Concept



4-Tube Spar

There is little difference in fabricating the pin root end for a four-tube spar. Four tubes would result in four vertical terminals for the wound ended tubes and in two vertical terminals (shown at left) for the cane concept. In each case a structural fitting is required to bridge the loads properly between pins and assure adequate load sharing. This fitting can be made of precured composite.

Figure 42. Root End Concepts.

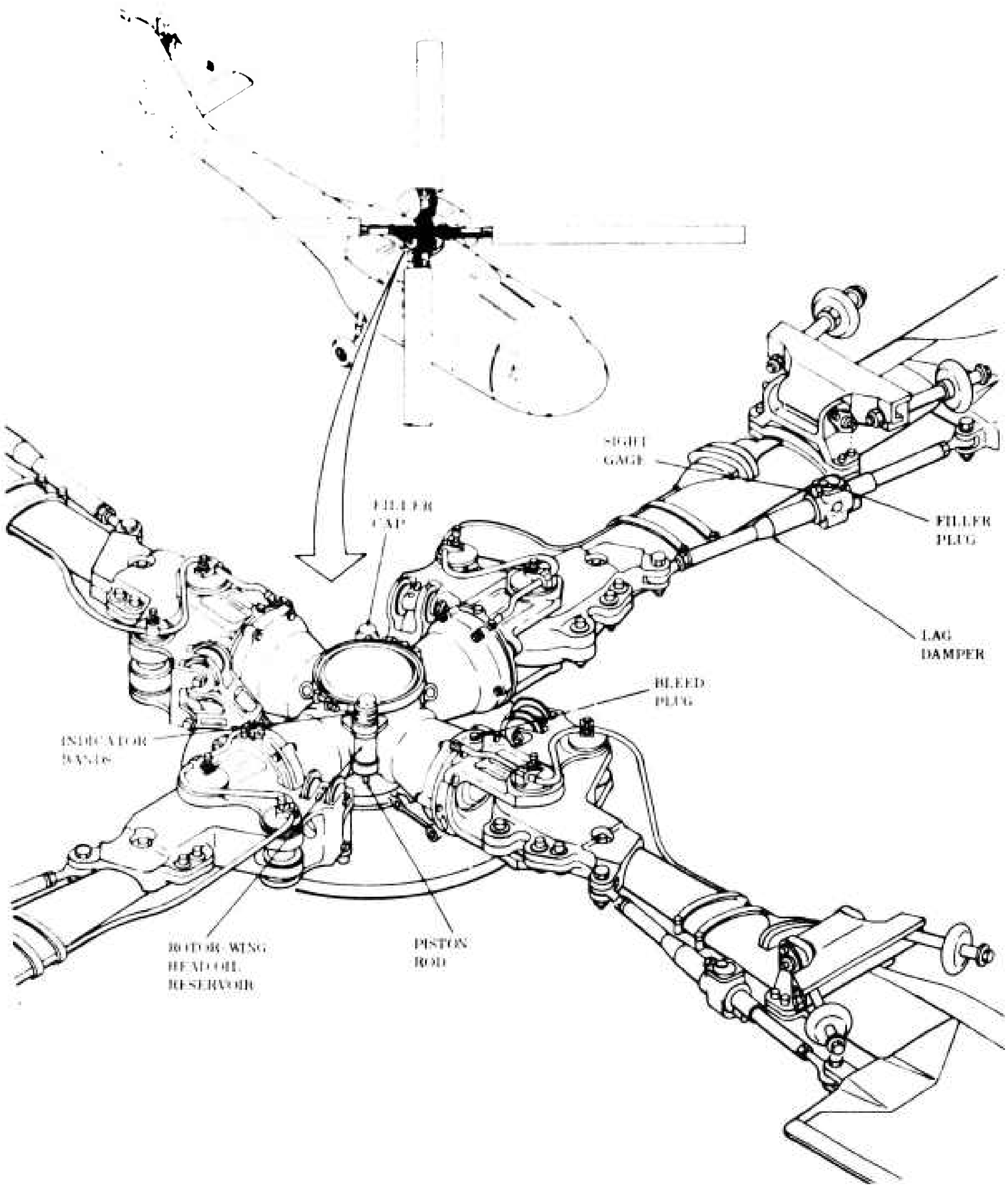


Figure 43. Hub A (Baseline, Scaled Down YUH-61A).

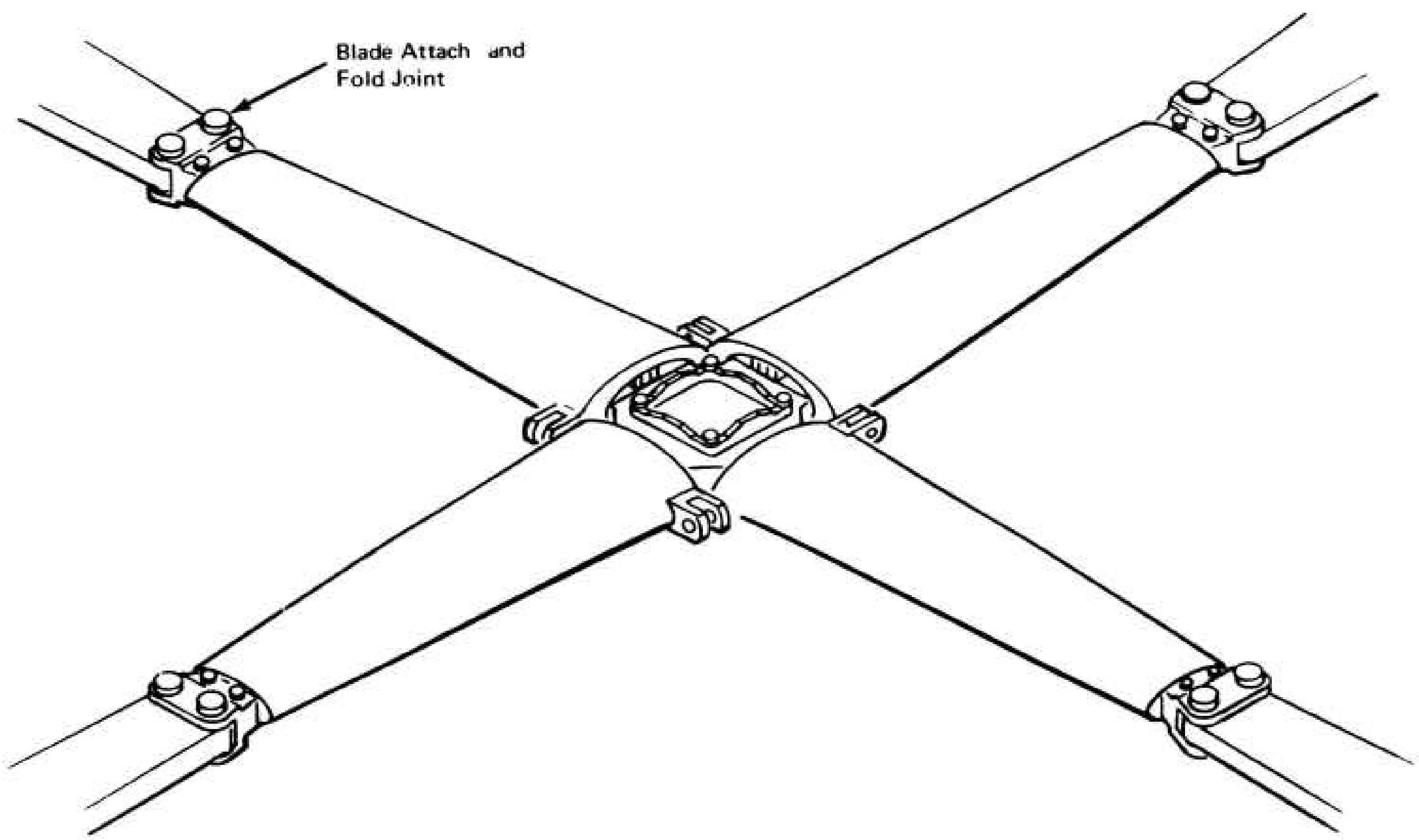


Figure 44. Hub B (One-Piece LTF).

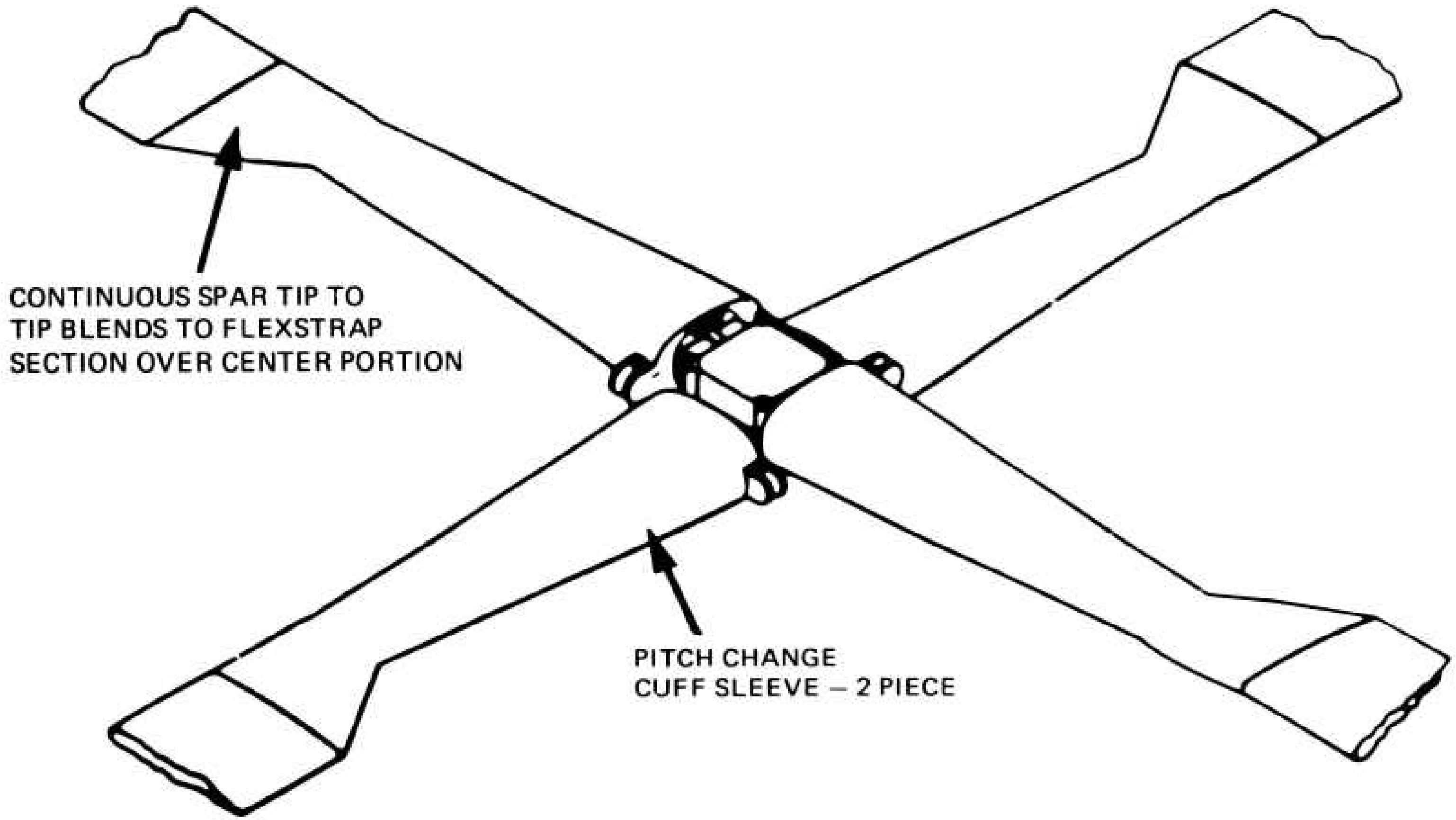


Figure 45. Hub C (LTF Tip-to-Tip Blades).

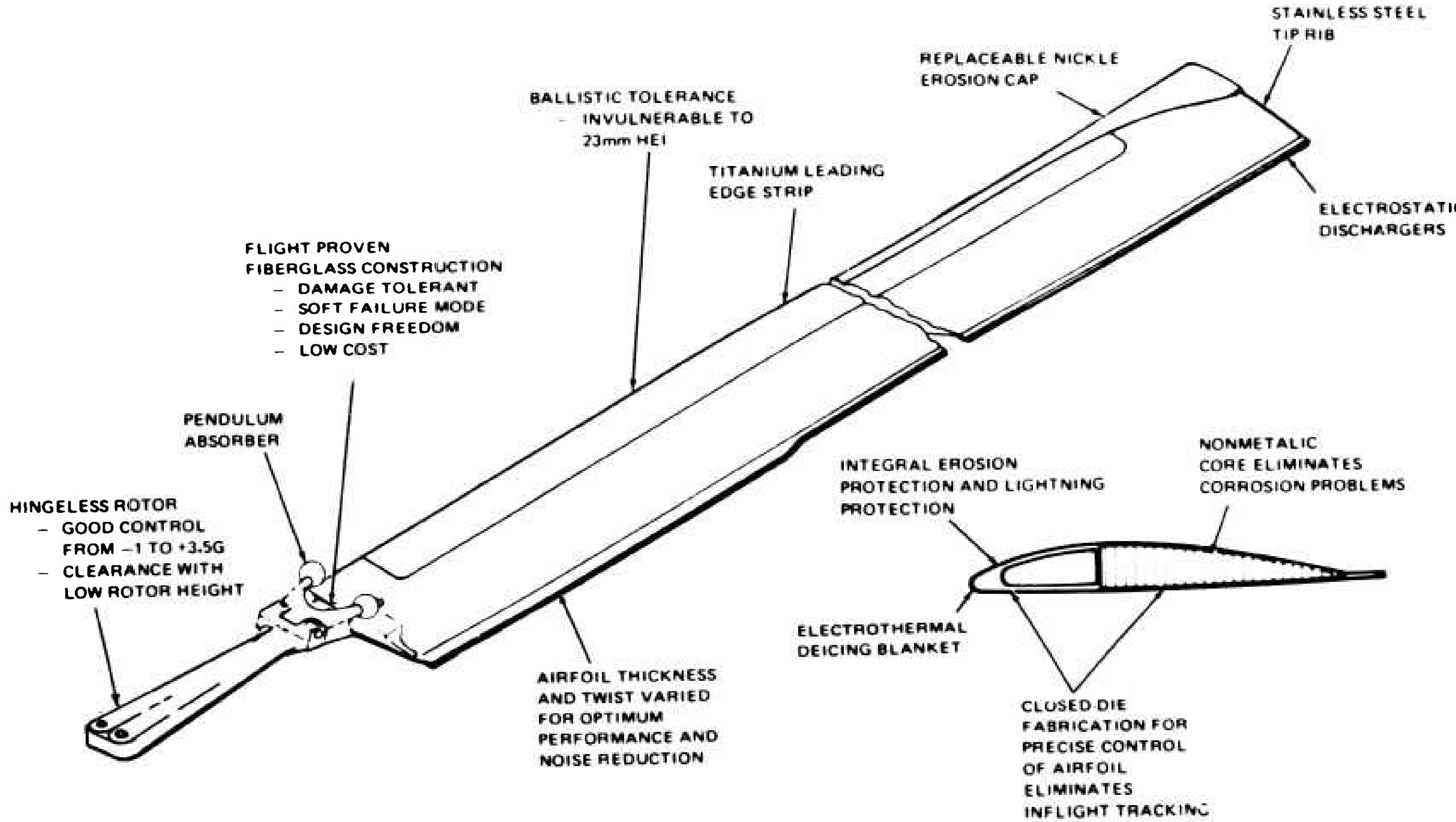


Figure 46. MUT Baseline Main Rotor Blade Modified to Fit LTF Hub.

3.6 LANDING GEAR CONCEPTS (Refer to Table 13)

The baseline landing gear system, Concept A, is illustrated in Figure 47 and is similar to the YUH-61A, designed for 10-foot-per-second sink speeds plus 20-foot-per-second hard landing and crash impact loads without gear yielding. Transport of the MUT in a C-130 or C-141 requires kneeling the gear, and the baseline system provides for integral kneeling, using the system illustrated in Figure 48.

The system was reconfigured in Concept B to a single-piston design (Figure 49) more suited to composite application while preserving the integral kneeling capability and strength criteria. Shock strut assemblies for Concepts A and B are compared in Figure 50.

The desire for further simplification led to Concept C, which eliminates integral kneeling in favor of ground support equipment to bleed and refill the oleos for transport.

Application of composite structure to the gear is shown in the oleo assembly (Figure 51). A composite version of Concept B using graphite epoxy in the shock strut and trailing arm is designated Concept D, maintaining integral kneeling.

Concept E is similar to Concept D but uses boron aluminum in the shock strut assembly and graphite epoxy in the trailing arm.

Concept F combines the composite system of Concept E with the GSE kneeling system of Concept C.

SELECTION AND LOGIC FOR FINAL STUDY CONFIGURATION - LANDING GEAR

Concept F (Table 13) was selected, employing a single-piston two-stage shock strut design with a graphite/epoxy trailing arm and GSE kneeling system. It was decided, however, to recommend retention of aluminum design since production quantities of the MUT would make die forgings cost-effective over boron aluminum or graphite (based on 1974 material costs). This conclusion should be reevaluated if material costs are reduced with time.

All other systems were retained as defined in the baseline configuration.

Table 13. Concept Screening Analysis – Main Landing Gear

- OBJECTIVE: To select best landing gear concept
- COSTS: Based on 1,000 units

Rating factors will be comparative to baseline structure counterpart

2	MUCH BETTER
1+	
1	BETTER
1–	
0	SAME
–1+	
–1	POORER
–1–	
–2	MUCH POORER

CONFORMANCE FACTORS: –
BETTER + / SAME 0 / POORER – } REFERRED TO BASELINE

	CONCEPT A	CONCEPT B	CONCEPT C	CONCEPT D	CONCEPT E	CONCEPT F[…]
	METAL, 2-STAGE INTEGRAL KNEELING BASELINE	METAL, 2-STAGE INTEGRAL KNEELING SYSTEM	METAL, 2-STAGE KNEELING PROVISIONS	HYBRID COMPOSITE, 2-STAGE INTEGRAL KNEELING SYSTEM	HYBRID COMPOSITE, 2-STAGE INTEGRAL KNEELING SYSTEM	HYBRID COMPOS[…] 2-STAGE KNEELING PROVISION[…]
	All metal design 2-stage shock strut (double piston design) comprising Shock strut, Trailing arm, Trunnion pin, Torque arms, Brake unit, Wheel, Axle integrated kneeling as in Concept B. See Figure 65.	Single piston design (see fig 67). Also with integrated kneeling system. See Figure 66 comprising 1 Provisions in oleo for oil transfer (annulus) 2 Solenoid operated valves (1 per oleo) 3 Individual hand control valves (1 per oleo) 4 Pipe and connectors to transfer cylinder 5 Kneeling transfer cylinder with internal pistons	Single piston design as Concept B but with items 2 to 5 inclusive not included in aircraft equipment. (These items would comprise a GSE package, including a portable hydraulic pump rig)	Single piston design as Concept B but with main shock strut piston, cylinder and trailing arm unit in graphite/epoxy composite (Light inner aluminum cylinder liner for wear purposes only.) Also thin outer steel case left on piston O/D)	Single piston design as Concept B but with main oleo piston and cylinder only in boron aluminum trailing arm in graphite epoxy (Light inner aluminum cylinder liner for wear purposes only) Also thin outer steel case left on piston (O/D)	Single piston design as Concept D but with kneeling provisions not included in aircraft equipment (GSE package)
RATING ELEMENTS						
STRUCTURAL EFFICIENCY	0	1–	1	2	2	2
FAIL SAFETY	0	0	1–	1	1	1+
SAFETY	0	0	0	0	0	0
PRODUCIBILITY/COST	0	1	2	–1	–1	0
RATING TOTAL	0	2–	4–	2	2	3+
CONFORMANCE ELEMENTS						
DETECTION AVOIDANCE	0	0	0	0	0	0
CRASHWORTHINESS	0	0	0	0	0	0
REPAIRABILITY	0	0	+	0	0	+
MAINTAINABILITY	0	+	+	+	+	+
SURVIVABILITY	0	0	+	+	+	+
GENERAL SAS CONFORMANCE	0	0	–	0	0	–
CONCEPT RANKING	5	4	1	3	3	2

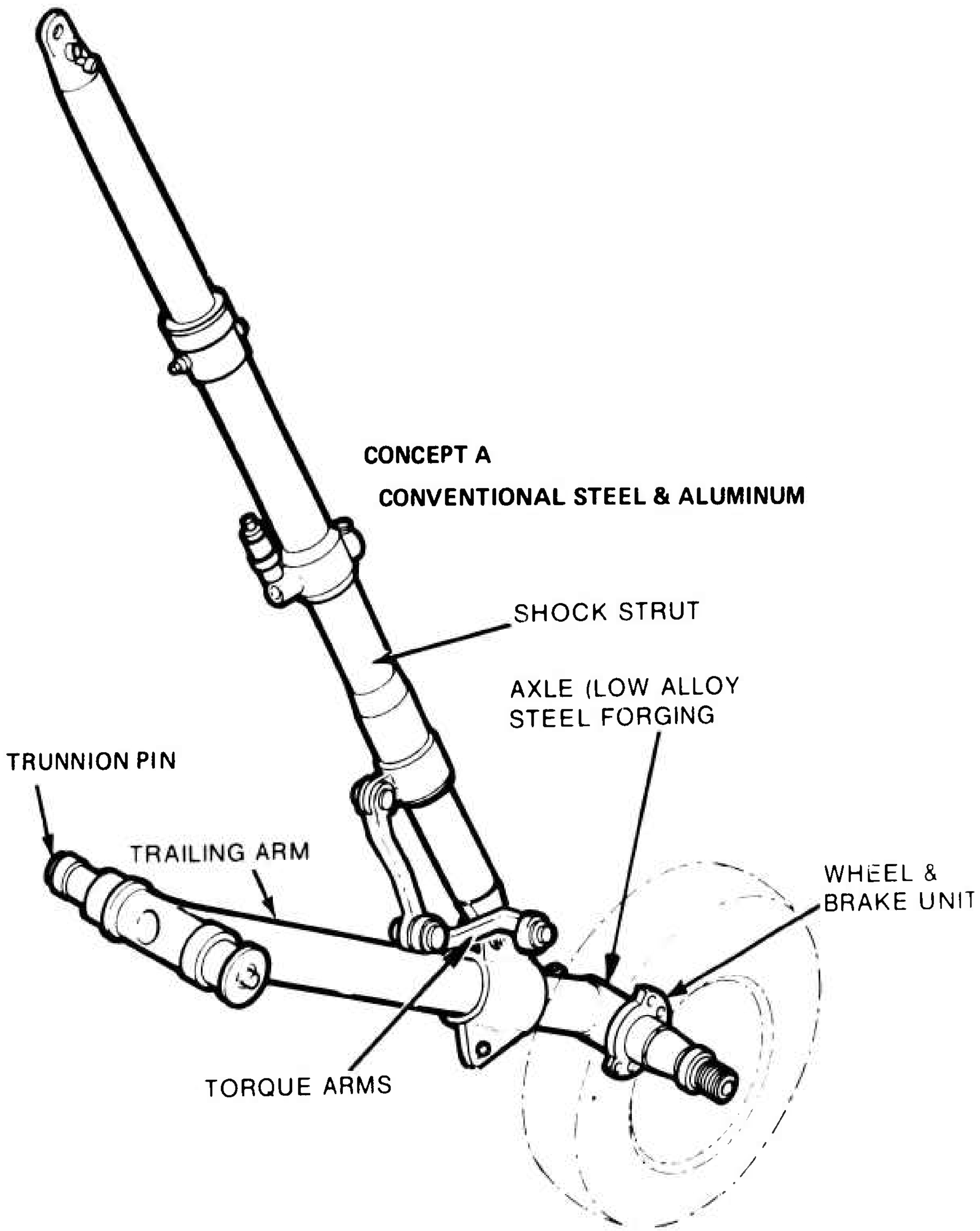


Figure 47. MUT Main Landing Gear (Lever Suspension; Double Piston Kneeling Type; 2 Stage).

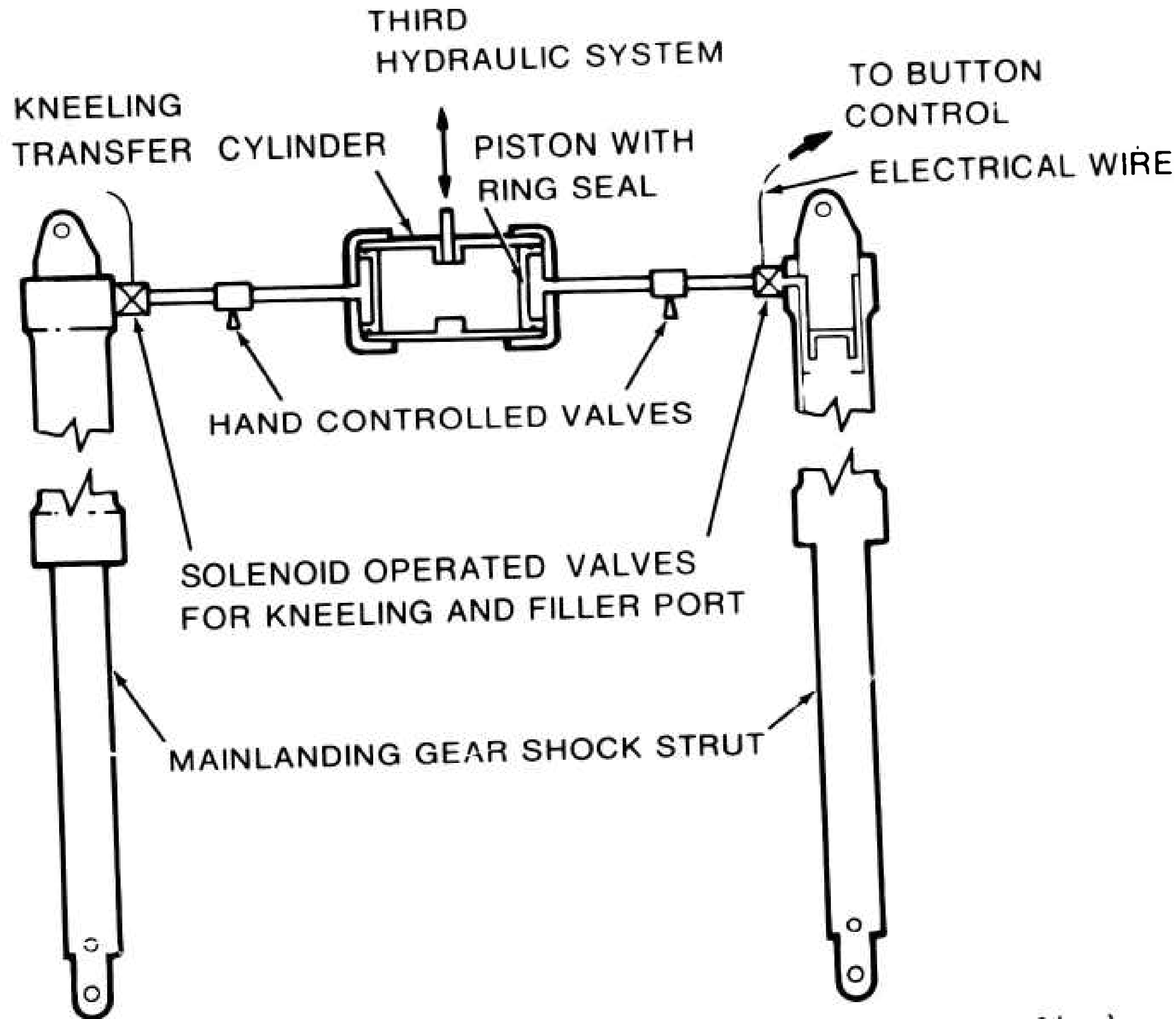


Figure 48. Main Landing Gear Kneeling System Schematic (Baseline).

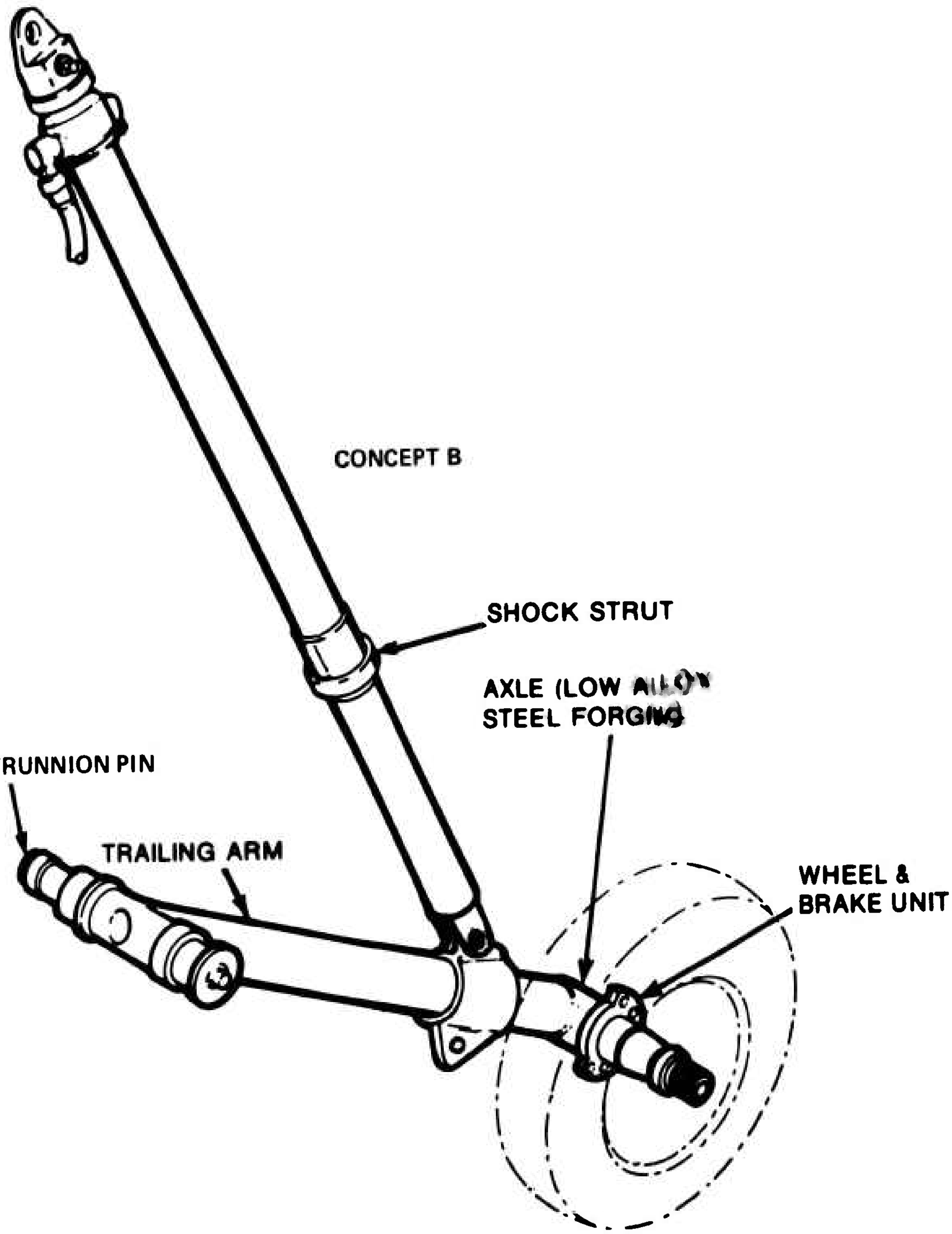


Figure 49. MUT Main Landing Gear (Lever Suspension; Single-Piston Kneeling Type; 2 Stage).

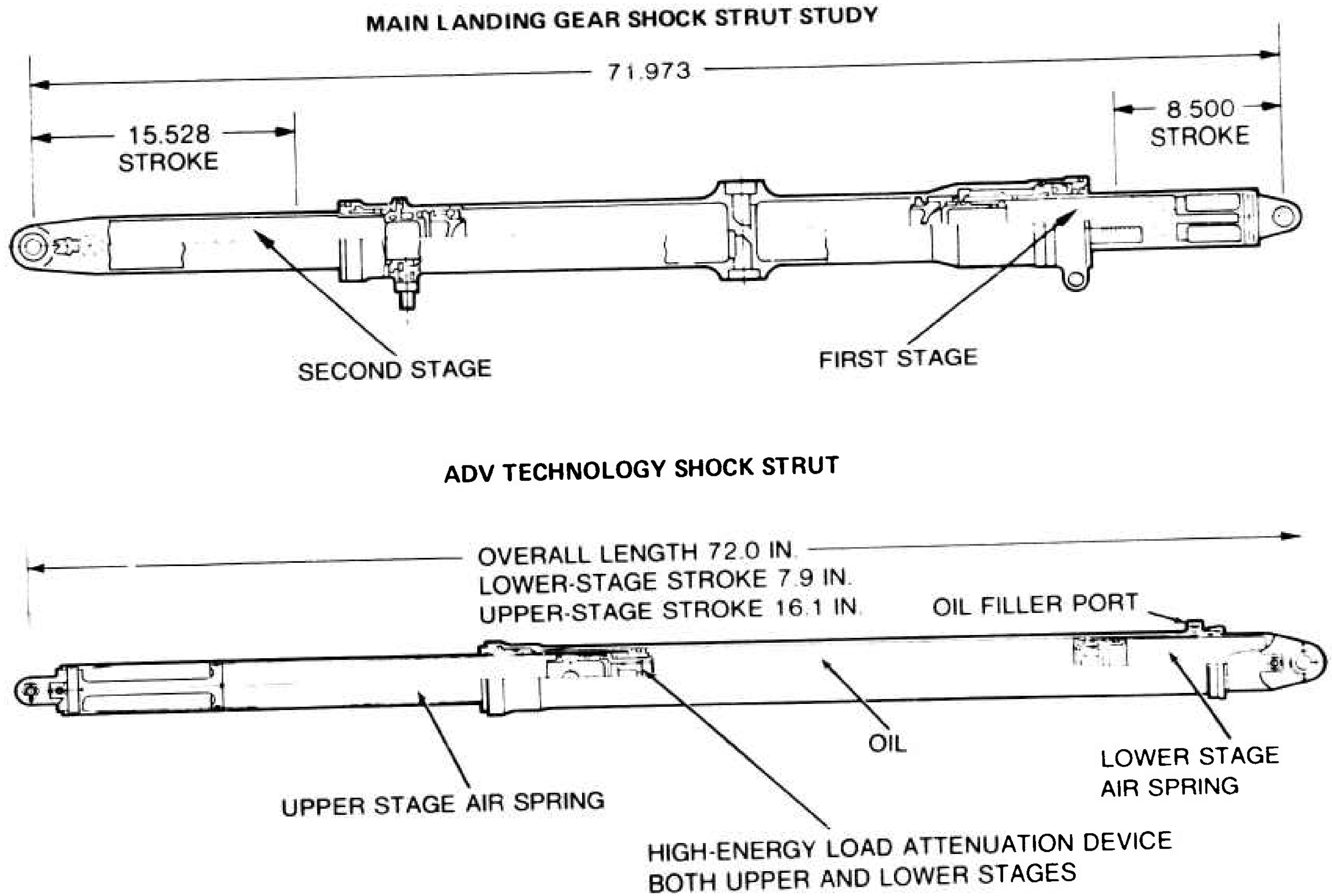
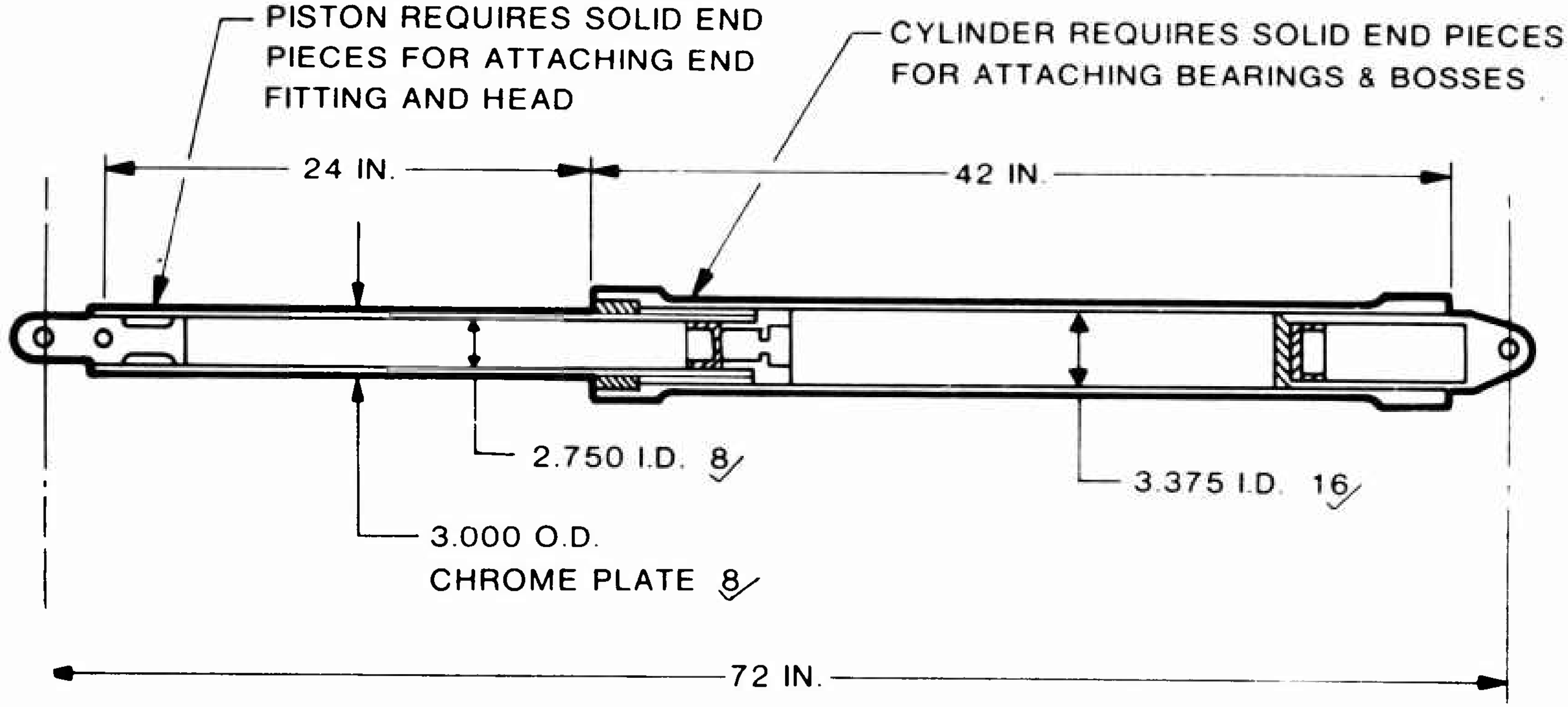


Figure 50. Shock Strut Comparison.



FINAL AREAS AND WEIGHTS

MATERIAL	DENSITY (LB/IN³)	AREA (IN²)	WEIGHT/INCH	% SAVING
G/E	.056	1.13	.0635	44%
B/E	.071	.85	.0605	46%
B/A1	.094	.595	.0560	51%
K49/E	.048	2.60	.1250	–
ALUMINUM	.101	1.13	.1140	–
HYBRID B/A1 AND G/E		.566	.0627	45%

Figure 51. Preliminary Sketch of Advanced Material Shock Strut.

3.7 FINAL STUDY CONFIGURATION

The results of the screening exercises on the major subsystems were reviewed at USAAMRDL with Eustis Directorate personnel, prior to selecting the systems to be incorporated into the MUT advanced configuration. Figure 52 and Table 14 illustrate the features. The refinement process on the airframe was conducted in some depth and is reported in Section 4.

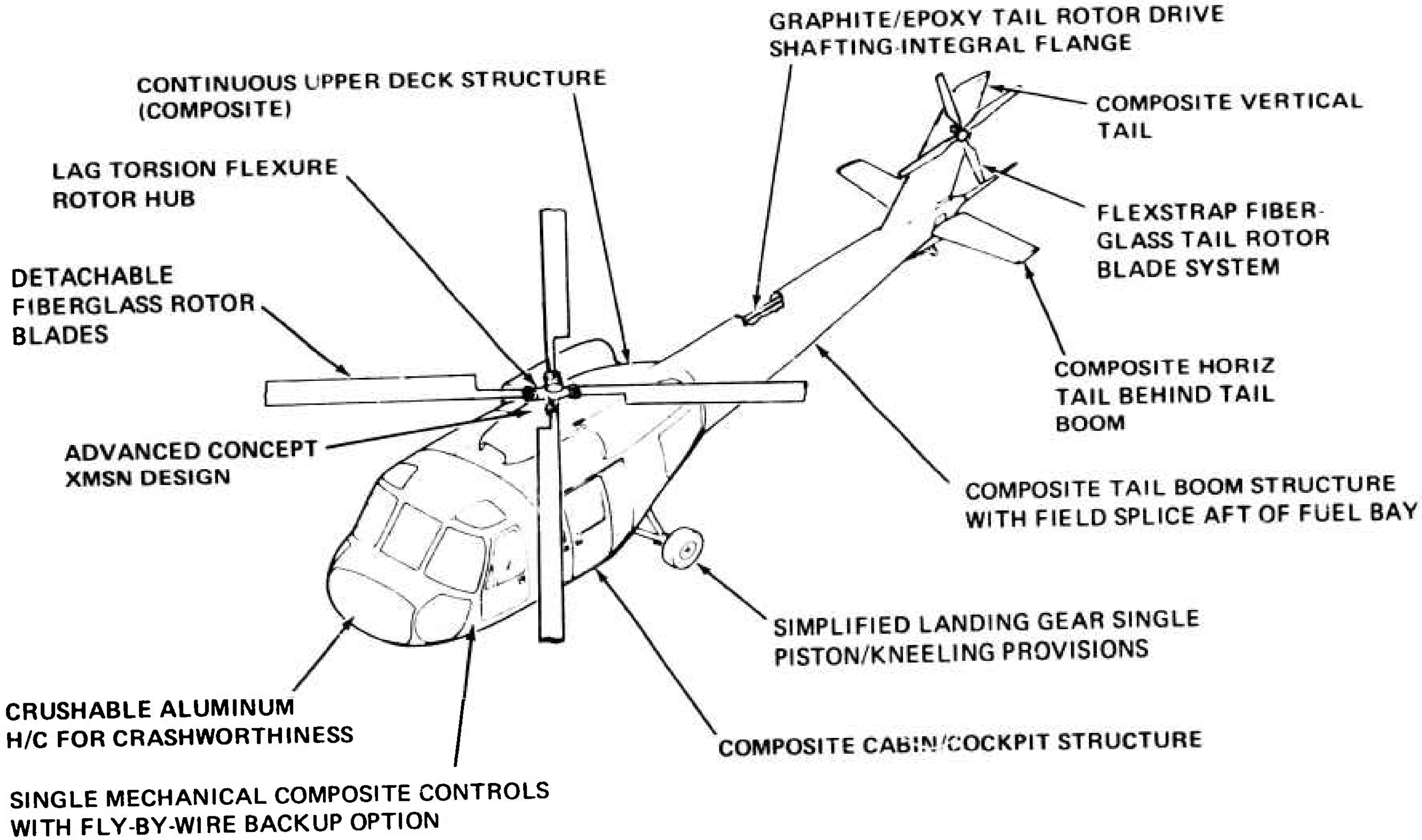


Figure 52. Recommended Final Study Configuration.

TABLE 14. FINAL STUDY CONFIGURATION SELECTION SUMMARY OF SELECTED CONCEPTS

Component	Concept	Description
Airframe	F	Composite Honeycomb Sandwich Hybrid
Main Transmission	A	Boeing Vertol Advanced Concept (See Volume II)
Drive Shafts	B	Composite Shaft with Integral Flanges
Flight Control System	D	Single Composite Mechanical with Backup Fly-By-Wire System
Rotor System	E	Soft-in-Plane Bearingless, Hingeless System
Landing Gear	F	Hybrid Composite 2 Stage (Single Piston) with Kneeling Provisions

4. DEVELOPMENT OF SELECTED AIRFRAME DESIGN

4.1 CONCEPT REFINEMENT

In the refinement stage, the various design features outlined in Section 3 were integrated into the Concept F configuration together with specified Army requirements (crashworthiness and safety features, etc.). The final preliminary design configuration is shown in Figure B-20 in Appendix B.

Supporting the general structural arrangement drawing is an isometric exploded sketch (Figure 56) of the final airframe concept together with isometric sketches of all eleven airframe modules (Figures 63 through 79). Further isometric sketches illustrate the crashworthiness features in the airframe, the identification and location of all metal primary fittings in the structure, the method of mechanically attaching all modules together, and finally, a picture of the fully assembled helicopter.

The overall vehicle envelope remains the same as the baseline (total length, width and height), but individual components have in some cases been resized and/or repositioned. Design guidelines are reviewed in Table 15, and a listing of the principal geometric and construction differences from the baseline design is given below. This is followed by a detailed discussion of design considerations during finalization of the preliminary design, and a definition of the modularized airframe assembly.

The principal differences between the baseline metal arrangement and the advanced composite design are as follows:

1. Honeycomb sandwich construction primarily replaces baseline skin/stringer design.

2. Cockpit section spliced to cabin section at sta 91 (baseline sta 78 is splice position).

3. Cabin height in composite arrangement is 57 in. (baseline ≈54 in.) (affords more space between litter stack in cabin).

4. Tailboom section area reduced considerably at forward end on composite design. Lines more streamlined (tear drop shape).

Table 15. Guidelines

PRIMARY	
1. STRUCTURAL EFFICIENCY	
2. FAIL-SAFETY	
3. SAFETY	(OPERATING 15,000 HOURS AIRFRAME LIFE – SAFE LIFE DESIGN REQUIRED TO SATISFY FATIGUE LIFE)
4. PRODUCIBILITY AND COST	
5. SATISFY MISSION REQUIREMENTS	
SECONDARY	
1. REDUCTION OF DETECTION	(REDUCED SIGNATURE FOR
2. REPAIRABILITY	A. NOISE B. RADAR C. INFRARED TO LEVELS AS PER PAGE 1 OR APPENDIX II)
3. MAINTAINABILITY	
4. VULNERABILITY	(REDUCE VULNERABILITY OF CREW CRITICAL SUBSYSTEMS AND COMPONENTS)
5. CRASHWORTHINESS	(TO COMPLY WITH MIL STD 1290 AV ALSO TR 71-22) 95TH PERCENTILE SURVIVABLE
6. USE OF STANDARD PARTS	(FASTENERS, TUBE, TIE RODS CABLES, FITTINGS, ACCESSORIES)

MATERIAL PROPERTIES – MAJOR CONSIDERATIONS (MIL-HDBK 5, 17, AND 23)

1. FRACTURE TOUGHNESS
2. ENVIRONMENTAL PROTECTION
   (AGAINST A) HUMIDITY, B) CORROSION, C) RAIN AND SAND EROSION, D) SUNLIGHT AND FUNGUS)

OTHER REQUIREMENTS

1. AVIONICS	– USE UTTAS AVIONICS PACKAGE (350 POUNDS); ALL ITEMS AS STATED ON PAGE 12 OF RFP.
2. CARGO HANDLING	– CARGO HOOK AND CONTROLS REQUIRED TO TRANSPORT AND RELEASE EXTERNAL LOADS UP TO 2,000 POUNDS.

5. Horizontal stabilizer moved back of tailboom structure - minimizes cutouts in narrow tailboom section; permits simpler, lighter attachment to tailboom; and allows horizontal stabilizer to be made in one piece (for lighter weight and a cost reduction through reduced parts count).

6. Redesigned landing gear oleo; kneeling operation on advanced concept is by ground equipment hydraulics.

7. Field splice joint, tailboom to fuselage, incorporated at sta 239.

8. Extension of tailboom structure into the fuel bay up to sta 163 bulkhead replaces the buttline beam arrangement of baseline structure.

9. Composite materials generally replace aluminum alloy except in specified locations.

4.1.1 COST CONSIDERATIONS

Considerable cost data has already been published for evaluation of highly loaded primary aircraft structure using high modulus and high-tension graphites and boron. However, limited costing is available to date on helicopter primary structure designed for relatively low stress intensity and a high degree of vibratory loading, into which category the MUT airframe falls.

Preliminary information is available on the recently developed AH-1G composite-wound tailboom and also on the Boeing UTTAS horizontal stabilizer, both of which are helicopter primary structure components fabricated with lower modulus composite materials.

It has been established that reduced parts count is a driving factor for cost reduction; and using this principle as a major design objective, the final configuration airframe design has been investigated and developed to an unusual degree of detail for a preliminary design study specifically to establish accurate and realistic parts counts as a means of generating reliable costs data (an alternative to trend curves).

The in-depth airframe study also affords visibility into most primary and secondary structure areas of the helicopter, thus assisting the cost engineer in determining possible complexity factor allowances to be applied to fabrication costs. Similarly, it gives the weights engineer a method of checking the factored composite/metal weight trend used, for by dividing the parts count realized in the study in depth by the weight of each module, the parts per pound figure is found, and it is directly comparable to existing composite and baseline metal structures.

4.1.2 DESIGN CONSIDERATIONS

The primary airframe structure for a contemporary heliccpter design usually comprises a form of skin stringer construction mainly in aluminum alloy material with some steel components and certain secondary structure areas in nonmetallic material such as fiberglass.

Aluminum alloy costs, although rising, are still considerably lower than advanced composite materials such as boron, graphite, or Kevlar, which are projected to lower price levels over the next few years. When contemplating an all-composite or near-all-composite airframe design, ways must be found of offsetting the higher materials costs by utilizing the many inherent advantages of reinforced plastics. However, it must be remembered that low-cost materials is only one of the primary considerations for the specified helicopter. Other considerations are reduced fabrication cost methods, high structural efficiency, producibility, safety and fail-safety, and general aircraft specification requirements. Further, secondary conformal requirements are reduction of detection, crashworthiness protection to 95th percentile, maintainability, repairability, and reduced vulnerability.

With all the above considerations in mind, a combination of innovative design features and new material applications are proposed in the final configuration to demonstrate an all around superior composite airframe design over the previously described baseline metal concept.

4.1.3 DESIGN OBJECTIVE

The objective is to design an advanced concept airframe and landing gear system that meets all Army specifications previously outlined while realizing acquisition cost equivalence and weight savings over the metal baseline concept.

4.2 DESIGN FEATURES

The advanced airframe has the following design features:

1. Resizing of components due to superior structural efficiency of the selected advanced composite materials.
2. Modular assembly concept.
3. Honeycomb sandwich construction.
4. Dual purpose structure.
5. Minimization of highly loaded joints and complex fittings.
6. Minimum parts count.
7. Significant reduction of mechanical fasteners by replacing with bonded joints within module assemblies.
8. Graphite AS (or equivalent graphite material) and Kevlar material selection.
9. Hybrid application to skins and fittings.
10. Reinforced thermoplastic materials application.
11. Low-temperature positive-pressure cure systems and minimal cure cycles.
12. Automated processes.
13. Laminate tailoring.
14. Reduced radar signature.

4.2.1 COMPONENT RESIZING

The higher strength-to-weight and modulus-to-weight ratios attainable with advanced composite materials allow components to be designed lighter and/or smaller for the same load- and stiffness-carrying capability.

However, due to the limitations imposed by the design specification requirements for the MUT cockpit and cabin sizes, it is impossible to attain maximum benefit from this inherent material advantage. Within these limitations it has been possible to resize the tailboom, horizontal stabilizer (slightly), and the overhead structural support system in the main cabin, which confers further benefit in that an extra three inches of internal height become available in the cabin to allow suitable clearance between three litters (the baseline did not quite meet the requirements for 18-inch spacing between litters using standard Army equipment because of dynamic system space requirements). An example of the resizing limitation problem as it affects the cockpit enclosure design is shown in Section 4.2.3.2.

4.2.2 MODULAR ASSEMBLY CONCEPT

The advantages of making monolithic moldings of whole segments of airframe structures are obvious, and an idealized structure would be two half-total airframe moldings bonded together on assembly (see the illustration of an extreme example, Figure 53). However, practical considerations in tooling, automated processing, and aircraft system installations preclude at this time such an extensive molding method. Another limitation is the number of subassemblies fulfilling special functions that are required in numerous locations (that is, crash attenuation structure, removable floor panels, ballistic protected fuel tank, etc.).

A careful evaluation of all these specialized design areas has indicated that a reasonable arrangement for the largest practical MUT module sizes commensurate with meeting all functional requirements is as follows:

Module 1: cockpit enclosure (including pilots doors)

Module 2: floor structure (including nose gear support)

Module 3: upper deck assembly and fairings

Module 4: bulkhead sta 163 and side panel assembly (including main gear support)

Module 5: tank support and side avionics structure

Module 6: tailboom and vertical stabilizer torque box

Module 7: tail cone fairing

Module 8: tailbumper and absorber assembly

Module 9: horizontal stabilizer and actuator

Module 10: vertical stabilizer; tip and leading- and trailing-edge fairings

Module 11: side hinged and sliding doors

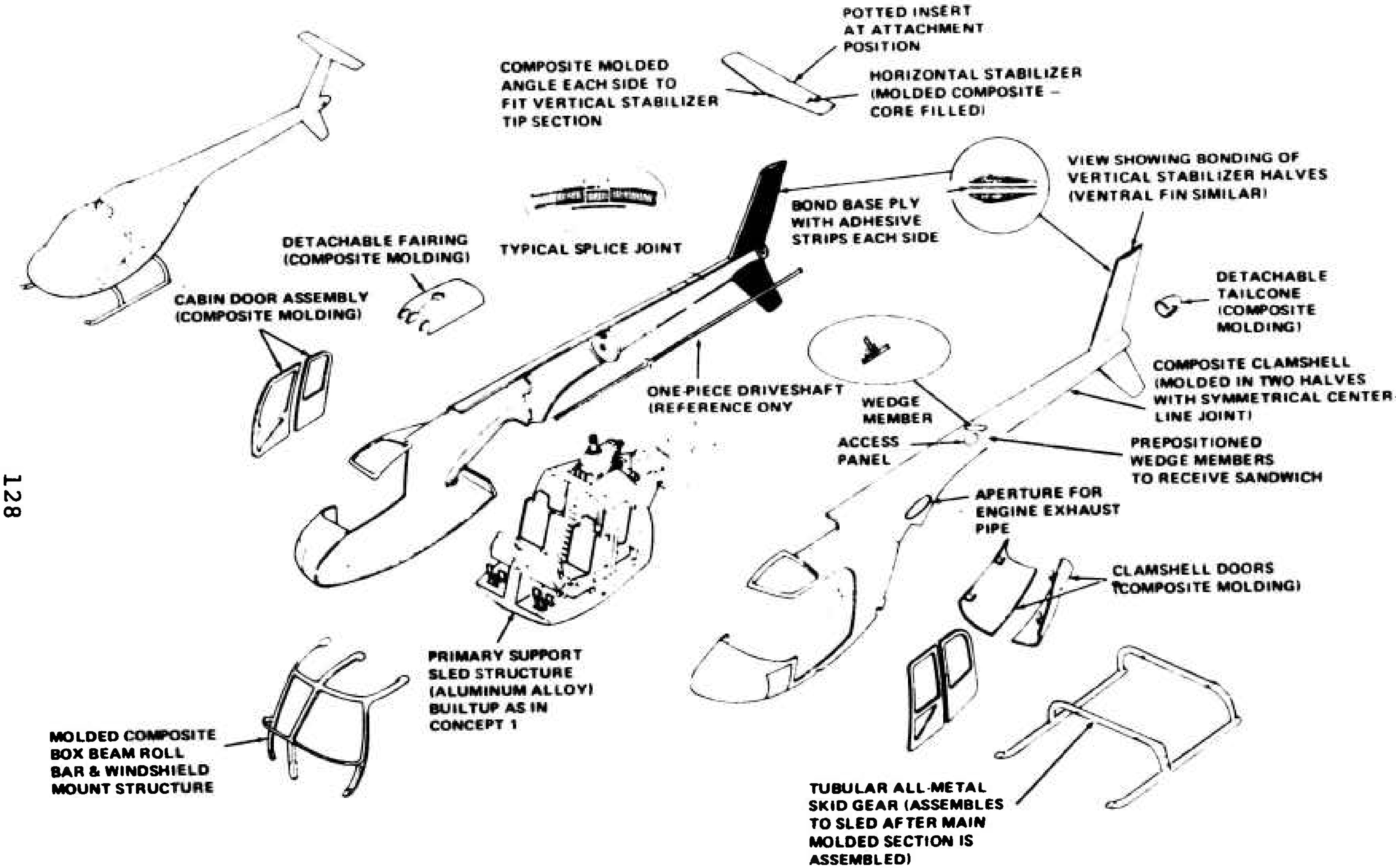


Figure 53. Fuselage Modularization (Extreme Example).

Each of the above modules is designed in the most suitable material and configuration for the particular loading spectrum and environmental conditions to be met, and they represent the largest separate assembly units; however, smaller subassemblies may be made up in each module. In the final assembly of the complete airframe, all modules are attached to each other by mechanical means only.

4.2.3 HONEYCOMB SANDWICH CONSTRUCTION

Analysis of honeycomb sandwich versus skin/stringer design in composite materials indicates that, for the MUT airframe application, despite certain disadvantages, the sandwich method is the better approach for much of the airframe. Accordingly, after preliminary design studies of various constructions, a honeycomb sandwich, Concept F, was chosen as the final configuration for further detail design and analysis. Some of the main advantages and disadvantages of this construction follow:

4.2.3.1 Advantages of Honeycomb Sandwich

1. Honeycomb composite structure panel design for airframe applications is regarded as low risk. Various composite flight articles in this construction have already proven suitable, and fabrication problems have mainly been resolved. By opting for refinement of this system, the follow-on development can be concentrated mainly on joints and some fittings which comprise a much smaller percentage of the airframe parts.

2. Sandwich construction skin and frames show between 10 and 15 percent weight saving over conventional aluminum alloy skin stringer.[1] If the composite skin stringer design is to be competitive, it must be loaded into the tension field regime (high σ/σ_{cr}) where little data is available for composite materials - hence this is considered higher risk.

3. Fail safety is increased. If one skin of honeycomb panel is damaged, loads are redistributed in the damaged laminate and are carried locally by the other skin while the honeycomb core supports the skin in the area of the damage and slows the propagation rate.[1] Whereas, should a single structural element of a skin stringer system sustain a crack, unrestrained propagation could cause structural failure.

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1 Bert, C. W., and Berger, H. K., Structural Cost Effectiveness of Composites, Society of Automotive Engineers No. 730338, April 1973.

4. Fewer parts are required than in multiple stringers and cleating systems (with intersecting frames). Also, the honeycomb sandwich system requires fewer frames due to increased panel stiffness. The number of fasteners and the associated expense is drastically reduced.

5. The skin/stringer system exhibits reduced stiffness due to skin buckling, so stiffening members must be stronger and heavier. Also, stringers tend to twist during buckling, producing peel stresses.

6. The honeycomb panel construction confers improved aerodynamic smoothness and less skin wrinkling under in-flight loading. The drag penalties are reduced, in addition to the asthetic improvement.

7. Sonic insulation and fatigue resistance are improved with honeycomb panels.

8. Improved panel impact damage results with thin laminates fully supported (this fact was demonstrated convincingly by drop tests on floor panels).

4.2.3.2 Disadvantages of Honeycomb Sandwich

1. The weight of the adhesive necessary to attain bond fillets to honeycomb core at inner and outer skin surfaces is inefficient unless proper processes are employed.

2. The panel weight advantage can easily be dissipated by carelessly designed boundary/joint members.

3. Care must be taken to insure long-term protection against water infiltration into the core cells; aluminum core can corrode if improperly selected or processed.

4. Inspection of panels requires more sophisticated equipment than inspection of skin/stringer structure. (Conversely, rivet and rivet hole inspections are eliminated, and costs may actually be reduced.)

5. Honeycomb structure might be heavier in very lightly loaded applications.

4.2.4 DUAL PURPOSE STRUCTURE

This is an advantageous design arrangement wherein a single structural element performs two or more functions not necessarily simultaneously. For example, on the proposed composite floor structure module, the underfloor beams perform a triple function: cargo support structure, fuselage bending and load carrying (compression and tension), and backup structure for the crash attenuation box. Another example is the bulkhead at Station 163, which has a multiple function. It reacts surge loads from the fuel tank, forms part of the support structure for the main transmission and hub, supports the main landing gear, and redistributes vertical and torsional shears from the tailboom.

By careful incorporation of such multifunctional arrangements within the airframe, considerable savings in cost and weight can be realized without compromising fail-safety features.

4.2.5 MINIMIZATION OF HIGHLY LOADED JOINTS AND COMPLEX FITTINGS

Careful design visualization of the MUT airframe concepts from the earliest preliminary design stages and implementation of the modular assembly concepts were methods employed to reduce the number of complex and costly joint fittings normally found in conventional constructions.

Minimization of the occurrence and intensity of concentrated load points was a prime objective throughout; it led to utilization of simple overlap attachments. An example of this method is the jointing of the forward tailboom/tank support structure to the deck module, where the deep longitudinal beams over the deck were arranged to be in alignment with the walls of the tank support structure underneath, such that the primary tailboom and empennage bending loads are diffused gradually over the complete tank bay length on each side from the tank support structure into the deck and beam structure. In this instance simple bolted cap-to-cap attachment with minimal stress concentration replaces one or more potentially sophisticated and expensive concentrated load joint fittings.

However, there are certain parts of the structure where the use of a relatively complex fitting is unavoidable, such as at the transmission attachment points, engine support attachments, landing gear locations, etc. The design of these high load intensity fittings in composite materials represents a risk factor which, coupled with the elevated cost factor for the fitting material and fabrication phase, renders the use of composite fittings doubtful in this application. For this reason it was decided that aluminum alloy fittings be utilized at all positions indicated in Figure 54. Fittings of 7075-T-73 were tentatively selected. However, improved aluminum forging alloys such as 7175, 7049 and 7050 (in -T66, -T736 and -T73 condition) should be considered. These alloys offer improved mechanical properties and/or improved fracture toughness and stress corrosion cracking properties over 7075-T73.

It should be noted that there are several ongoing study contracts where composite and composite/aluminum fittings for various aircraft applications are under development. These may well afford an improved technical base for reappraisal of the proposed MUT metal fittings. It is estimated that approximately 150 pounds of metal fittings and supports on the MUT are potential composite candidates.

4.2.6 MINIMAL PARTS COUNT

Recent industry and in-house cost studies of component assemblies have identified a positive relationship between parts count and fabrication costs for both metal and composite construction, and the fact has been established that reduced parts count is a driving factor leading to lower manufacturing costs.

Within each module assembly the separate parts count has been reduced to a minimum, commensurate with mandatory design requirements. As an example of parts count reduction, consider the comparison of the proposed MUT tailboom composite structure against the tailboom structure of a similar size utility helicopter currently in the U.S. Army inventory. (Both are bare tailboom structures, primary and secondary, not including any mechanical attachments such as bolts, rivets, etc.)

- Existing production metal tailboom - 450 parts
- MUT composite tailboom - 40 parts

NOTE: The existing production tailboom is of metal skin/stringer construction; the parts were counted from parts lists on drawings.

4.2.7 SIGNIFICANT REDUCTION OF MECHANICAL FASTENERS

Construction within each module is arranged so that all frames, ribs, and fittings are bonded together either on assembly or as a secondary bonding operation if necessary. The only exceptions are where it is extremely difficult to gain adequate access to apply curing pressure, and where a replaceable item may be reinstalled. All joints will be primary bonded, and no extra fasteners are required except in limited applications where there is the possibility of peeling.

All identified metal fittings are considered detachable and will therefore be mechanically attached. Modules will be mechanically attached to each other on final assembly. The attachment details are shown in Figure 55.

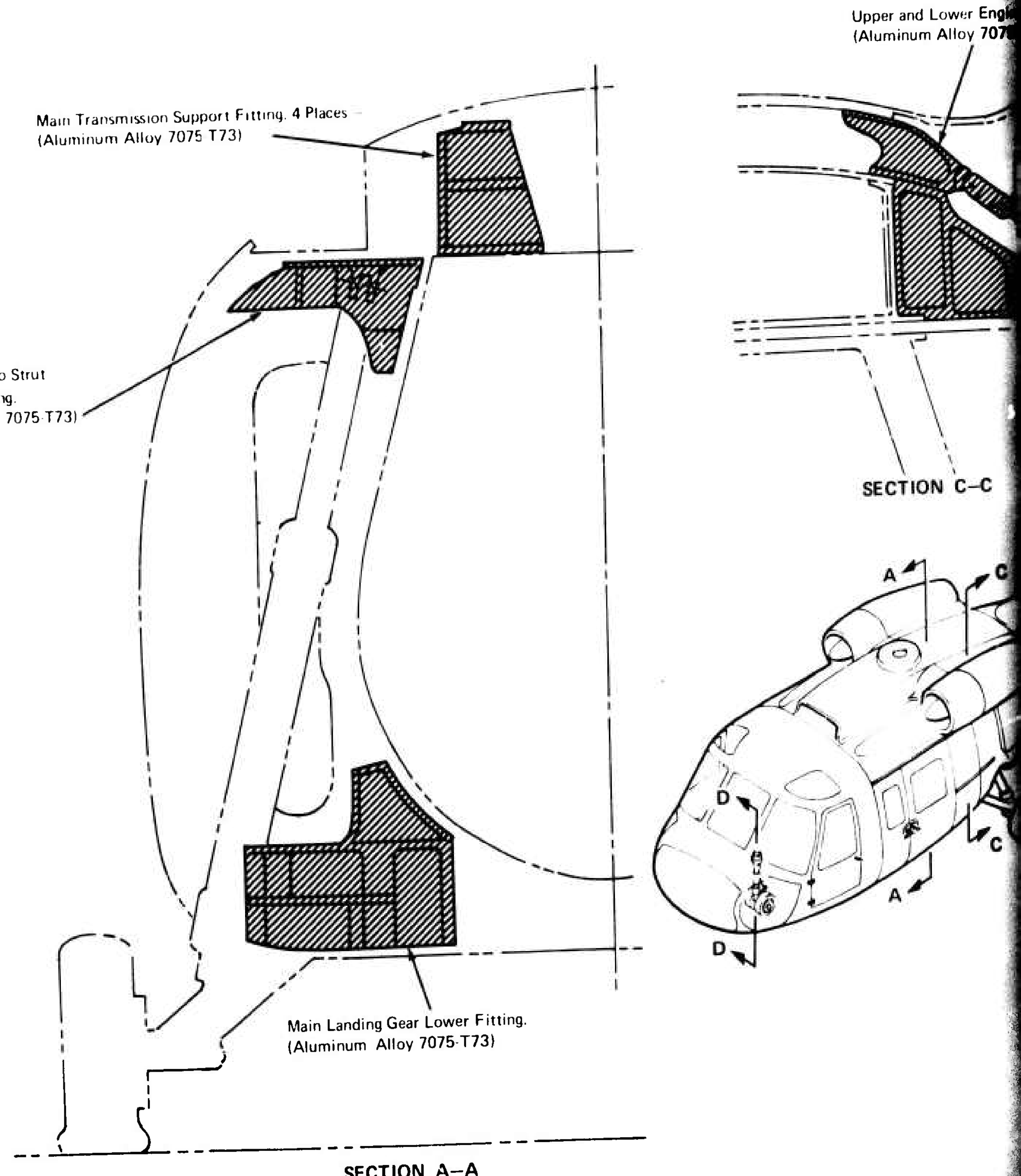
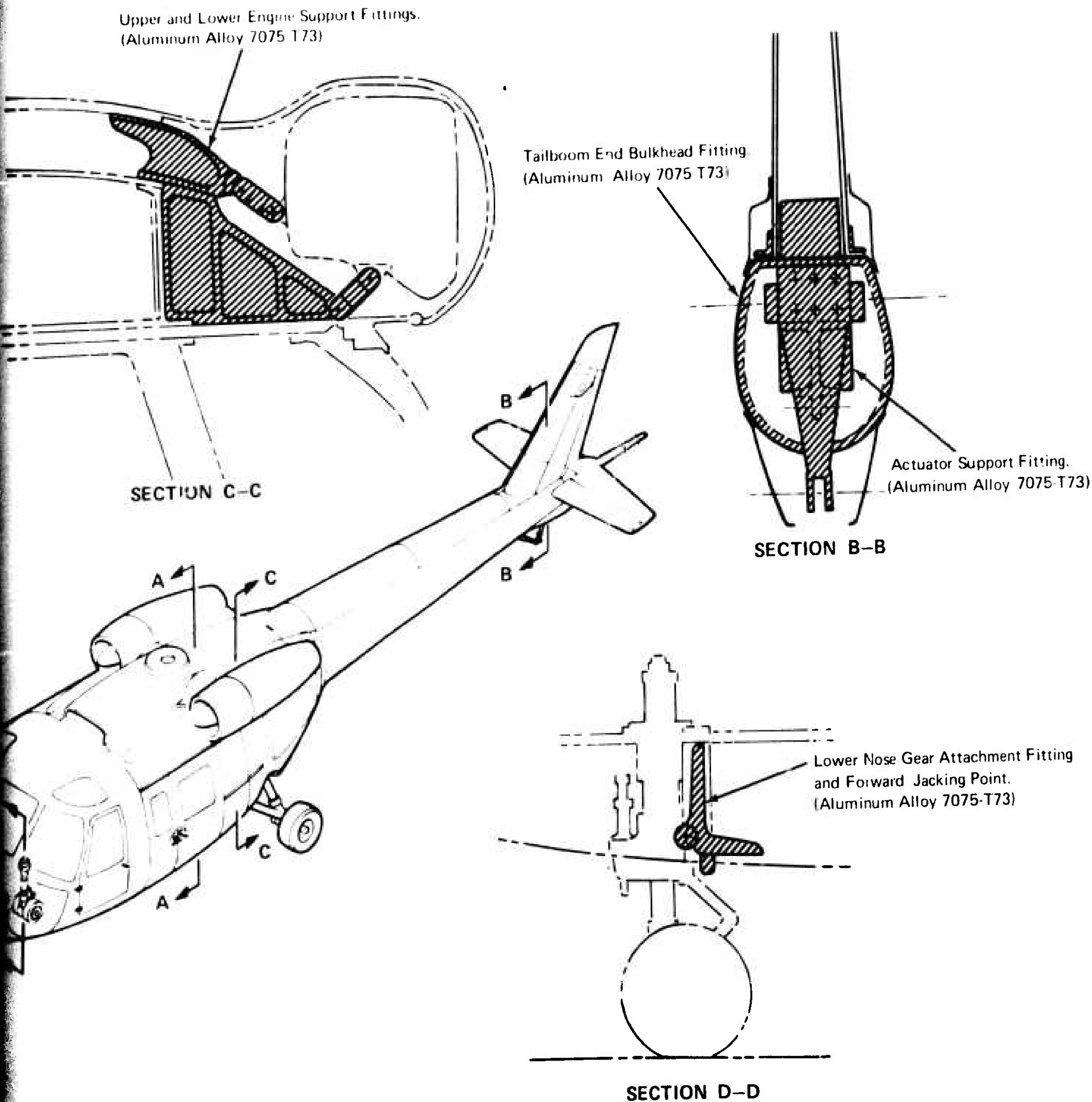


Figure 54. MUT Airframe - Identification of Metal Primary Load Fittings in Airframe.


Upper and Lower Engine Support Fittings.
(Aluminum Alloy 7075 T73)
SECTION C–C
Tailboom End Bulkhead Fitting
(Aluminum Alloy 7075 T73)
Actuator Support Fitting.
(Aluminum Alloy 7075-T73)
SECTION B–B
B
B
A
C
C
A
Lower Nose Gear Attachment Fitting
and Forward Jacking Point.
(Aluminum Alloy 7075-T73)
SECTION D–D

oad

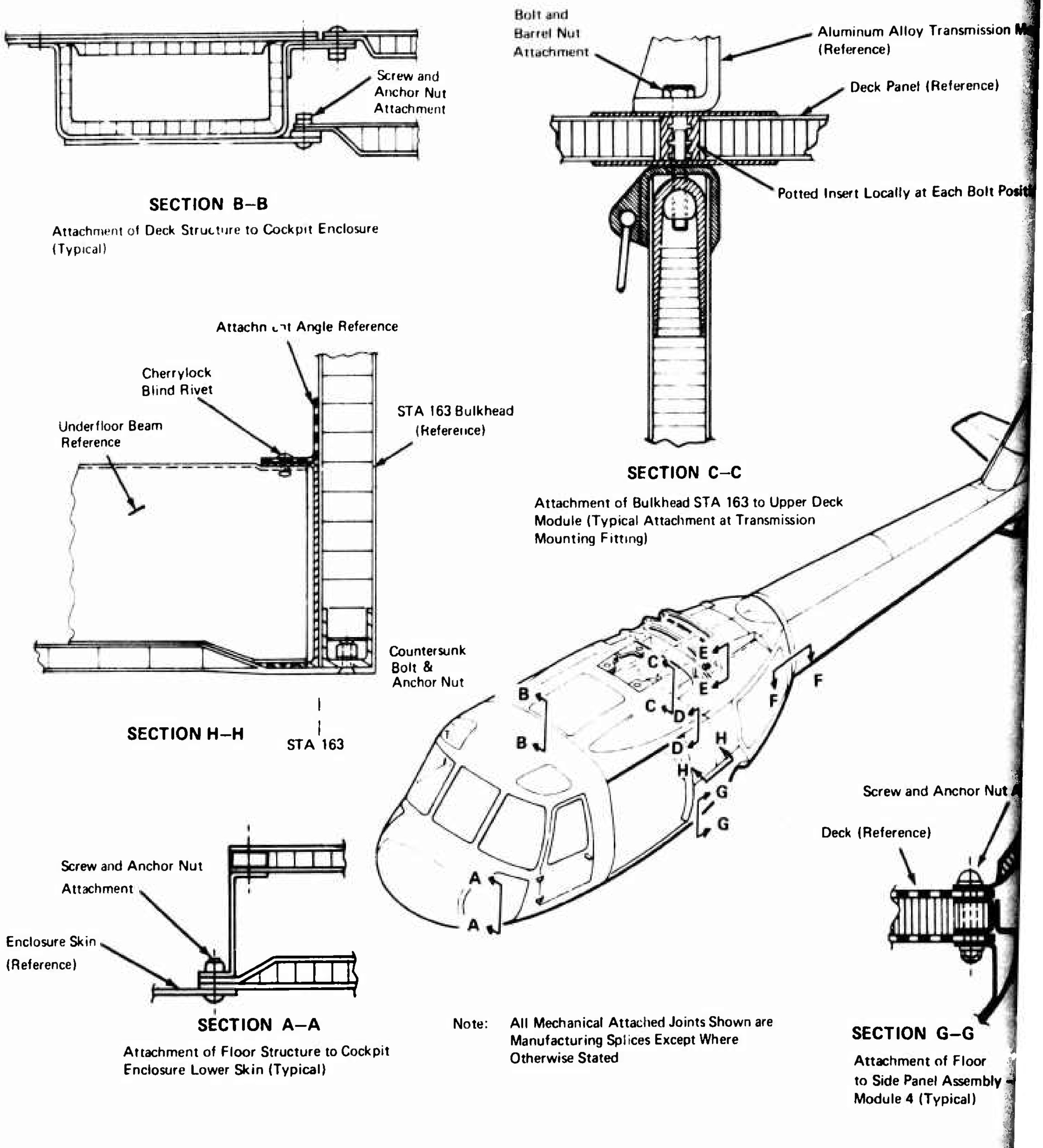
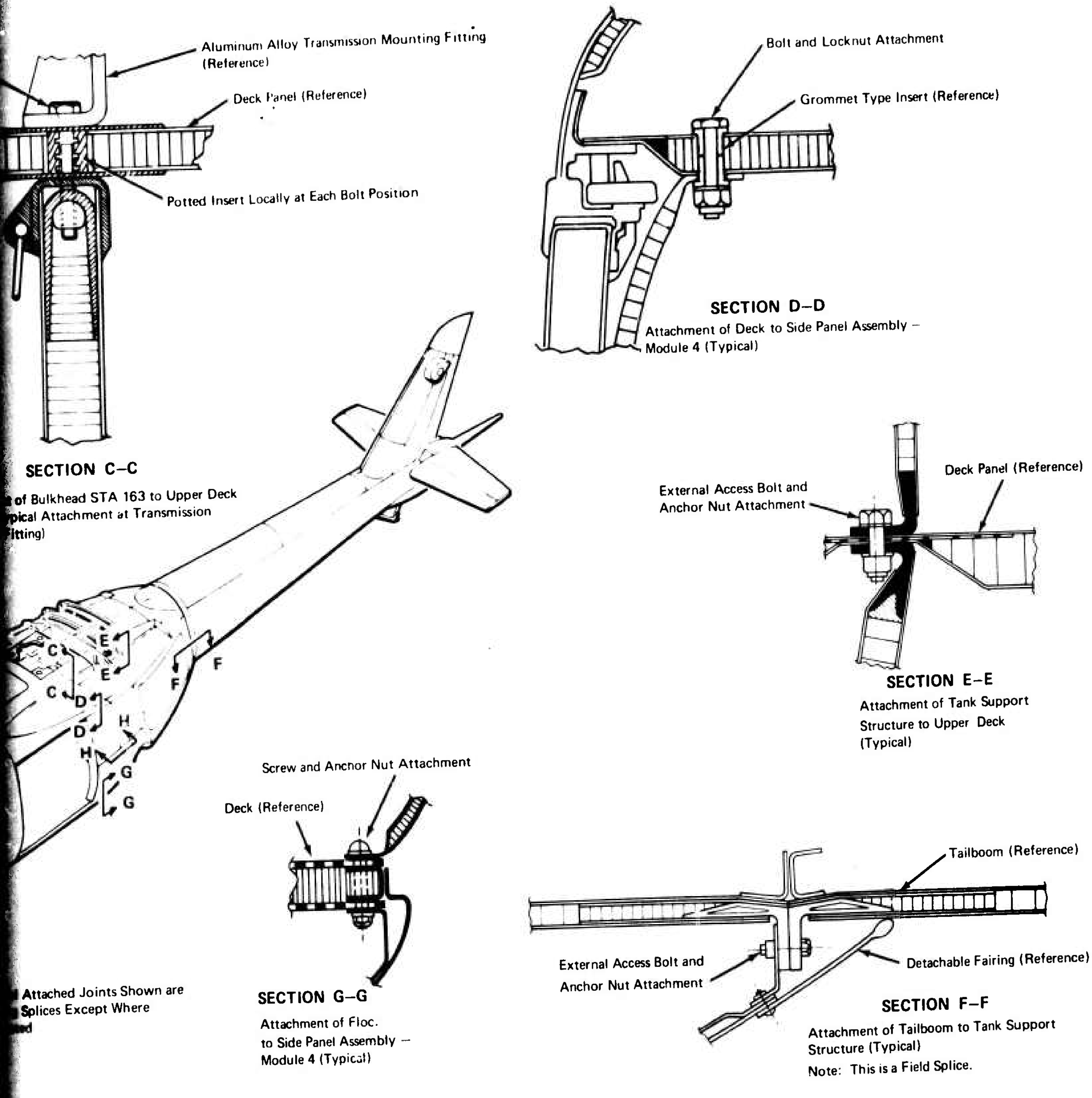


Figure 55. Assembly of Airframe Modules (All Mechanical Attachments).



Aluminum Alloy Transmission Mounting Fitting (Reference)
Deck Panel (Reference)
Potted Insert Locally at Each Bolt Position
SECTION C–C
of Bulkhead STA 163 to Upper Deck
pical Attachment at Transmission
itting)
Bolt and Locknut Attachment
Grommet Type Insert (Reference)
SECTION D–D
Attachment of Deck to Side Panel Assembly – Module 4 (Typical)
C
C
E
E
F
F
D
D
H
H
G
G
External Access Bolt and Anchor Nut Attachment
Deck Panel (Reference)
SECTION E–E
Attachment of Tank Support Structure to Upper Deck (Typical)
Screw and Anchor Nut Attachment
Deck (Reference)
SECTION G–G
Attachment of Floc. to Side Panel Assembly – Module 4 (Typical)
Attached Joints Shown are
Splices Except Where
ed
Tailboom (Reference)
External Access Bolt and Anchor Nut Attachment
Detachable Fairing (Reference)
SECTION F–F
Attachment of Tailboom to Tank Support Structure (Typical)
Note: This is a Field Splice.

anical

It should be noted that many cost analysts are of the opinion that the cost of alignment, drilling, deburring, hole inspection, rivet installation, and rivet inspection are equal to or greater than costs attributable to parts count.

4.2.8 GRAPHITE AND KEVLAR SELECTION

High strength and high modulus materials such as boron, graphite HT, and graphite HM provide the highest structural efficiency in primary airframe structure, but material costs are high. Further, as discussed in Section 4.2.9, the relatively low stress intensity airframe primary structure is largely gauge limited; therefore, the selection of very high strength or stiffness materials at extra cost was not justifiable.

For primary structure purposes the use of Kevlar 49/epoxy alone is unacceptable due to the low compressive strength of the material but could be advantageously combined with intermediate strength graphite, identified as graphite AS on the structural illustrations. (The nomenclature AS infers use of a Hercules AS graphite product. This is tentative only; further characterization of graphite intermediate range materials could result in the use of a similar grade produced by another company.)

Although the density of both Kevlar 49 and graphite is low, at approximately 0.05 pound per cubic inch the prepreg price varies considerably:

- Graphite AS = \$50/lb (any quantity)
- Kevlar 49 = \$20/lb (large quantity)

based on 60-percent fiber volume and 1974 material cost

Since both of the above materials have near compatible coefficients of linear expansion, it was decided that a discrete blend of the two materials be used for essentially all primary structure items, and the Kevlar 49 alone be used for secondary structure.

4.2.9 HYBRID APPLICATION TO SKINS AND FITTINGS

The blending or hybridizing of the Kevlar 49 and graphite can be effected using an epoxy or thermoplastic matrix. A compromise strength and stiffness level must be accepted in the mix; this is discussed in the stress analysis materials selection discussion (Section 4.4.3).

Preliminary stress analysis has revealed that the major portion of the airframe primary structure is relatively low stress intensity and in much of the area is gauge limited in that the theoretical number of plies required for strength and stiffness is less than that considered a practical minimum for field service conditions.

By using hybrid laminates of Kevlar 49 and graphite AS such that Kevlar is used in approximately a 2:1 proportion over the more expensive graphite, a lower cost laminate is obtained which possesses the required strength and stiffness criteria.

Kevlar 49 contributes further to the laminate characteristics by affording increased impact resistance, reduced radar signature, and increased damping. The mechanical effects of the proposed hybridization are discussed in the stress section.

4.2.10 REINFORCED THERMOPLASTIC APPLICATIONS

Until recently the use of thermoplastics in aircraft structures was confined strictly to internal furnishing such as trim moldings and duct work. However, with the emergence of improved matrices, which exhibit strength characteristics approximating the existing epoxy systems, and with the techniques now developed for combining these matrices with reinforcing fibers, such as fiberglass, graphite, and Kevlar 49, a whole new fabrication process for composites is available.

With suitable analysis and development, reinforced thermoplastics materials seem well suited for low stress intensity primary structures such as the MUT airframe.

The Boeing Company has played a leading role in developing these newly emerging systems and has completed two Navy contracts to investigate their potential applications and to characterize various material combinations and fabrication methods[2].

The principal advantages of thermoplastics over conventional thermosetting systems is in reduced fabrication and material control costs where considerable savings are indicated. Reinforced thermoplastic sheets can be laminated in the flat or held as pre-pregs and indefinitely stored in a manner similar to metal sheets, whereas uncured or B-staged thermoset materials must be placed in refrigeration until approximate usage time and open or room temperature exposure time carefully controlled.

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2. Hoggatt, John T., Investigation of Reinforced Thermoplastics for Naval Aircraft Structural Applications, Contract N00019-72-C-0526, D180-17531-1, May 1973; and NASC Report D180-12884-1, May 1971.

Another advantage is that all trim or scrapped reinforced thermoplastic material can potentially be sold off to an active chopped strand molding composites market, whereas there is no market for raw or cured epoxy trim and scrap.

In-process scrappage will be reduced since reinforced thermoplastics can be recycled in the event of a break in cure cycle or dimensional discrepancy.

The fabrication cost savings alluded to earlier are effected mainly because of the suitability of thermoplastics to post-forming processes with short heat/pressure cycles. Reinforced thermoplastic sheets can be made up in the flat with any selected ply orientation and ply thickness as broadgoods, or it can be suitably modified by a numerically controlled tape layup machine, then laminated and stored.

Match metal mold dies or autoclave mold/bags are used to post form the sheets under heat and pressure to straight line element shapes (for example, flanges, channels, Z sections, etc.) and to single curvature shapes. Discrete compound curvature forms are also attainable.

Laminating and post-forming times average minutes, as against hours for an epoxy molded item, thus saving expensive autoclave or tool usage time. In a large production run, this not only substantially reduces costs but also relieves the demand on autoclave and tool capacity, and therefore facilitates rapid production.

Two promising thermoplastic matrices identified as suitable for use with composite reinforcement for airframe applications are polysulfone and phenoxy. The former is superior for 250°F useful temperature environment but is more expensive and requires higher fabrication temperatures than the latter. Phenoxy is a suitable matrix material for the operating temperature range of most MUT airframe applications, from -65°F to +160°F.

4.2.11 LOW TEMPERATURE CURE SYSTEMS AND MINIMAL CURE CYCLES

Composite thermosetting matrix systems curing at 350°F and more recent systems curing at 250°F have similar strength and stiffness characteristics, but the 250°F system minimizes thermal mismatch problems when bonding together materials having different thermal expansion coefficients, and it is more tolerant of part/tool thermal mismatch. Thus, a 250°F system might permit utilization of lighter weight aluminum tooling with faster heat-up rates and free-machining characteristics.

A known technique for minimizing the number of cure cycles on a fabricated component is by designing for "cocuring", whereby all parts in the assembly are located together and simultaneously cured in one operation. For example, a honeycomb panel could be made and cured this way with inner skin, outer skin, honeycomb core, and Z-edge members all prefitted, as opposed to the method where skins and edge members are laid up and cured independently, and then core, skins, and edges are joined in an assembly bonding operation. The latter method requires increased fabrication man-hours, longer cure cycle time, and usually more tooling fixtures.

4.2.12 AUTOMATED PROCESSES

Wherever possible, fabricated assemblies are designed to facilitate use of automated processes. The predominantly straight line element primary structure honeycomb panel inner and outer skins may be laid up on a numerically controlled tape layup machine such as ATLAS, which has three-dimensional layup capabilities. Some large panels lend themselves to drape-forming of widegoods laid up flat by machine and transferred to the mold tool. Another more limited manufacturing cost-reduction tool is the pultrusion process where unidirectional or some combinations of uni- and angle-ply filaments may be pulled through a die of the required cross section to form a beam cap, a channel, or other member of constant cross section. After passing through the resin bath and the forming die, the stiffener sections can also be pulled onto a mold which will form them lengthwise to any reasonable single curvature shape before curing takes place.

The use of rapid heating and cooling match metal die tools for efficient mold forming, compacting, and curing of various panels and fittings facilitates high production rates; it also confers other advantages, primarily those of close tolerance control, dimensional repeatability, and excellent finish.

Another automated process considered is thermoplastic postforming, where high rates of production can be attained by bumping out identical moldings from prelaminated flat panels under heat and pressure. Postforming capability of flat graphite skin/HRH core honeycomb panels has already been demonstrated for simple curvature.[2]

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2 Hoggatt, John T., Investigation of Reinforced Thermoplastics for Naval Aircraft Structural Applications, Contract N00019-72-C-0526, D180-17531-1, May 1973; and NASC Report D180-12884-1, May 1971.

4.2.13 LAMINATE TAILORING

Here advantage is taken of making up shell structures by successive layers of prepreg tape, where the numbers and orientation of plies can be controlled exactly and tailored to fit the precise strength or stiffness requirements at each section of the structure. Longitudinal or lateral skin reinforcements can easily be added or dropped off at will using a programmed tape on the layup machine to form components having high structural efficiency.

An example of efficient laminate tailoring on the MUT airframe is the tailboom honeycomb shell where extra longitudinal laminates may be added for tuning, if necessary, and circumferential laminates are added at the forward and rear ends for local reinforcing where joint fittings are attached.

This discourse highlights a shortcoming in the use of filament winding for construction of a tailboom, in that continuous winding applies the same volume of wetted filaments along the total length of the boom despite a usual decrease in sectional area of the structure due to taper. This results in significant thickness increase at the small end resulting in a non-optimum tailboom. One possible solution is to continuously wind on sufficient material for the tail end stress condition and then to have a second layup operation limited to the forward end of the boom as required for the higher bending moment at that end. When the geometry, loads, and minimum gauge material allow, continuous nonlinear geodesic paths may be found to minimize this effect. In this case, lower costs may result, which trade off against a reduction in structural efficiency.

4.2.14 REDUCED RADAR SIGNATURE

The general replacement of metal by composite structure on the MUT final configuration will reduce the overall radar cross section in two ways. First, the honeycomb sandwich skin panels with Nomex core provide a favorable absorption method. Second, the low dielectric properties of the hybrid composite skin cladding all over further contribute to reduced signature. These methods augmented by the use of electronic countermeasure techniques and the minimization of reflective angles associated with the airframe contour can effectively reduce radar reflectivity.

4.3 CONFIGURATION DEFINITION - ADVANCED COMPOSITE AIRFRAME AND LANDING GEAR

The airframe structure is comprised of a series of interconnecting modules (as shown in Figure 56) which are mechanically attached to form the complete airframe (as shown in Figure 57).

The airframe is a semimonocoque design; although the tailboom is truly monocoque, the main fuselage structure is essentially composed of longitudinal beams plus vertical frames carrying the horizontal and vertical loads and moments in lieu of skin panels. This arrangement is due to the numerous large cutouts in the external skins for cabin doors, hatches, avionics doors, and the like.

The principal construction method for the airframe is honeycomb sandwich using all composite materials, although some assemblies such as the cockpit enclosure and avionics nose and side doors, all of compound curvature, are constructed of molded channel section support members bonded to skin moldings to form box section components.

4.3.1 HONEYCOMB CORE SELECTION

Nomex nonmetallic core is commonly used. However, special energy-absorbing metallic core is used in crash attenuation applications, namely, the crash box in the forward underfloor structure area, and the main forward rollover frame.

While aluminum honeycomb is relatively low cost and is extensively used, it lacks resiliency and dents easily, especially at low density, and cannot be used in structural applications where low dielectric properties and radar transparency are required. Aluminum also corrodes in the presence of moisture, which leads to debonding of the face and thus requires frequent and costly inservice examinations. In addition, aluminum is a poor thermal and acoustical insulator. In spite of these shortcomings, aluminum honeycomb is the most commonly used type in structural parts, because no other commercial honeycombs have as high a shear modulus and compressive modulus at equivalent weight.

Nomex honeycomb core HRH10 is a high temperature fiber/phenolic resin honeycomb available in various densities and core sizes, which is not subject to corrosion and is readily bonded to graphite and Kevlar 49 materials forming a chemically inert system.

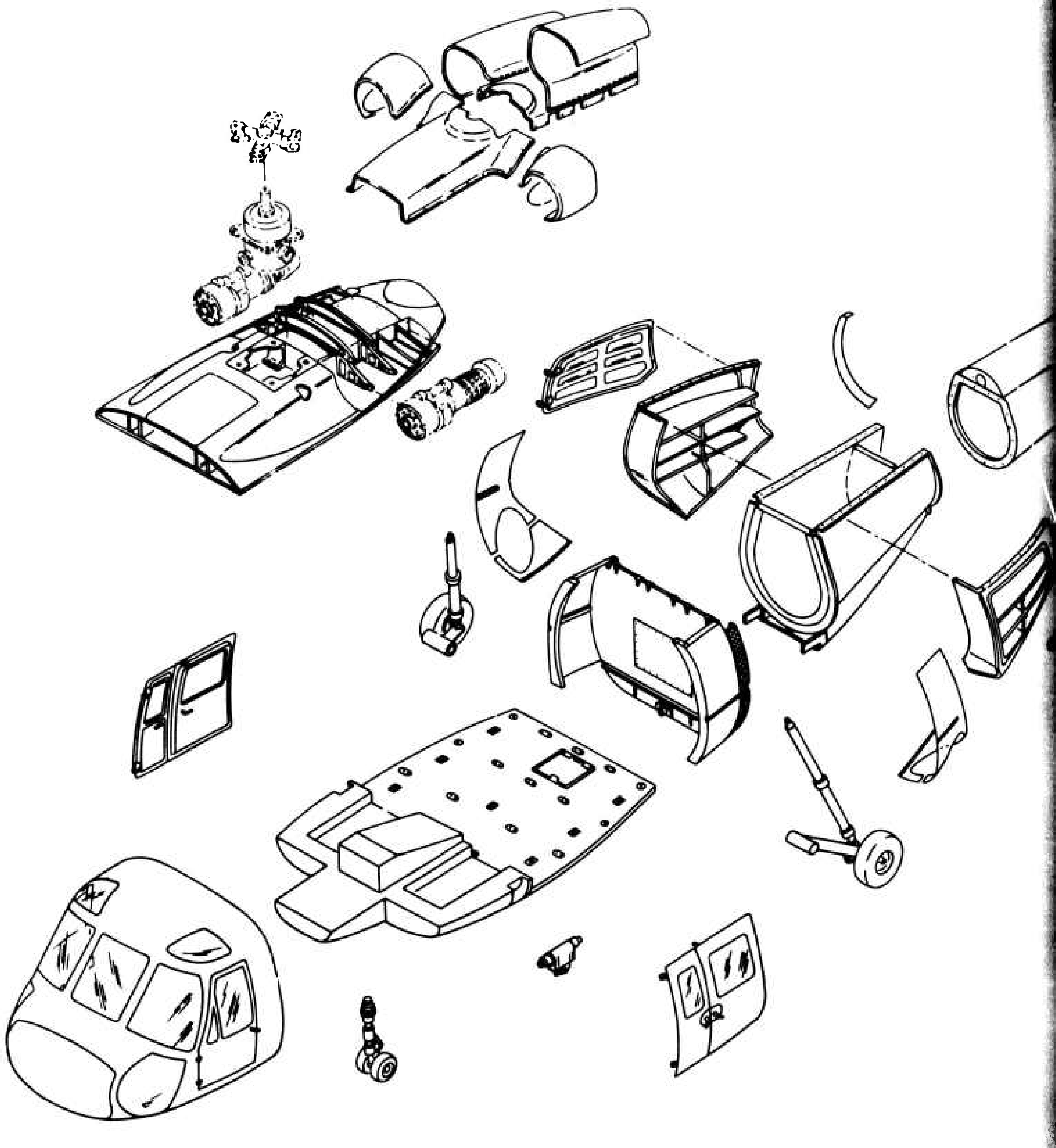
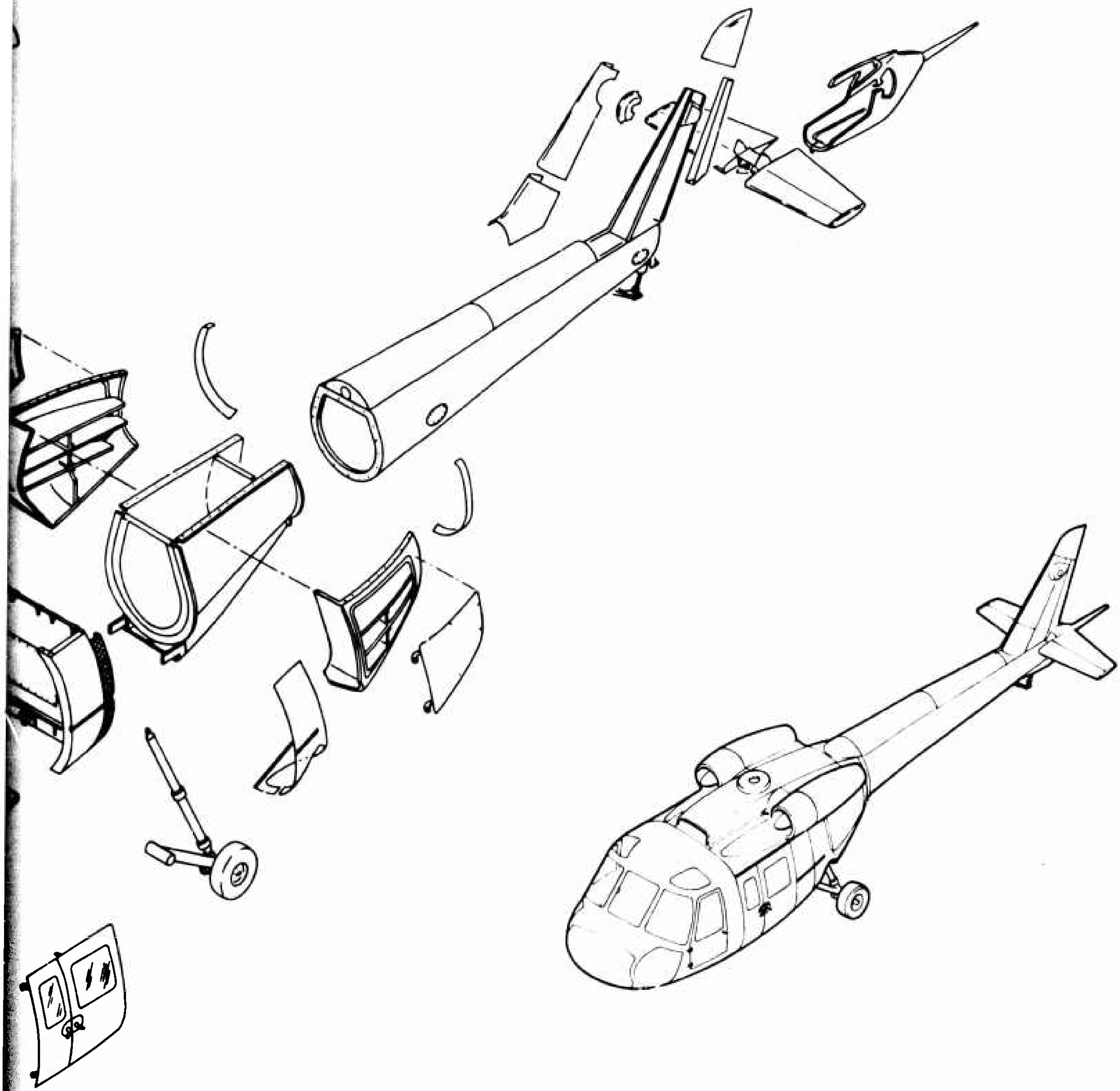


Figure 56. MUT Airframe - Final Configuration Exploded View of Module and Subassembly Components.



ded View of

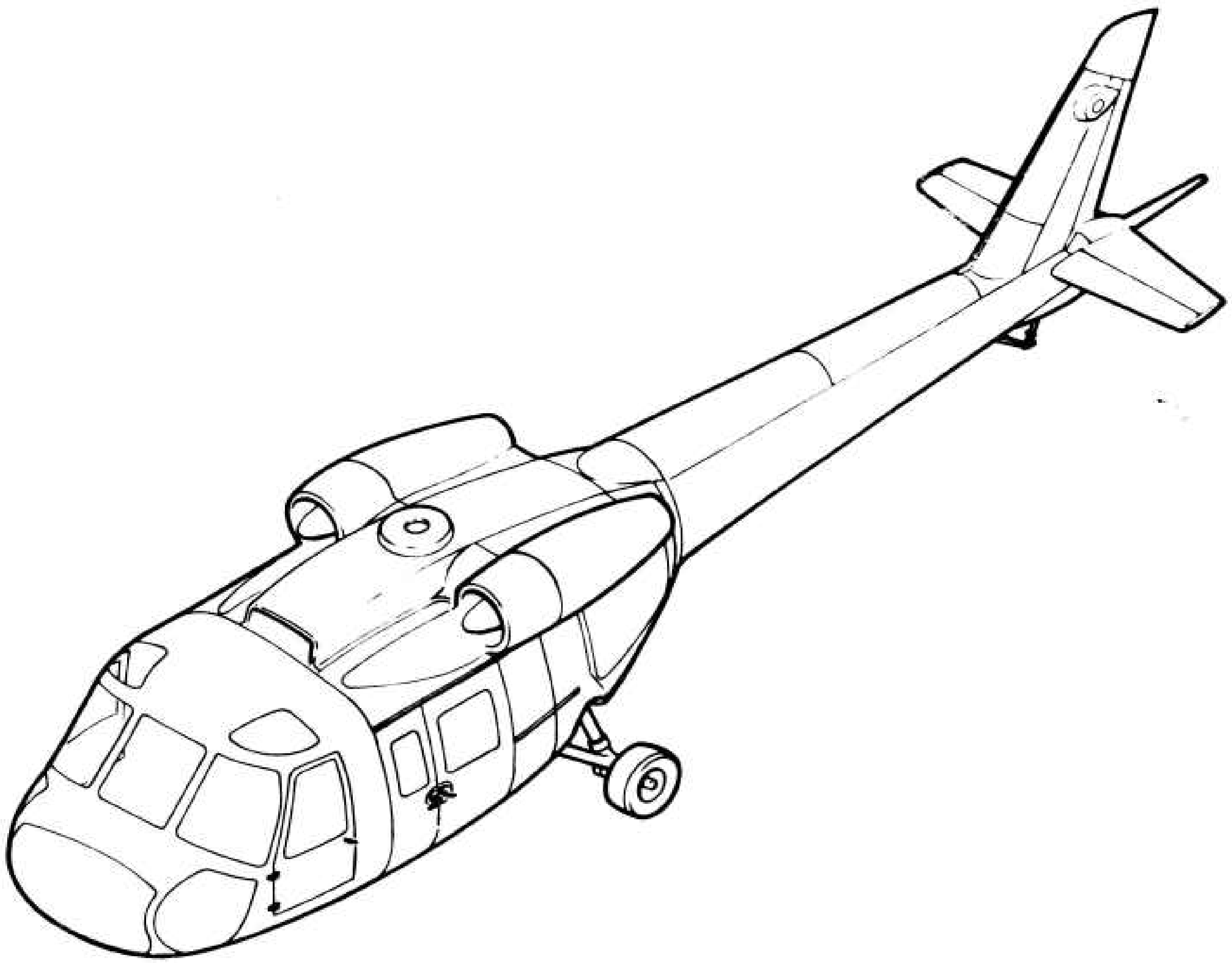


Figure 57. MUT Airframe Total Modular Assembly.

Although the shear modulus of Nomex is lower than aluminum honeycomb, its allowables are adequate for the relatively light stress intensity airframe structure of the MUT in most areas; however, in localities where honeycomb core is required for specialized purposes such as crash attenuation, then aluminum honeycomb with high energy-absorbing characteristics is used.

When, due to tolerance variations, an aluminum honeycomb sandwich panel, which is overly thick, is subjected to high pressure and made to conform to the controlled thickness of a metal mold die, a core crippling mode is induced with accompanying deterioration of strength characteristics. Nomex, however, under similar circumstances, will deform along the core to skin line and will not progress into the crippling mode when panel thickness tolerance does not exceed 4 percent (approximately 0.015 in. per surface on a 1.0-in.-thick panel).

4.3.2 MODULAR ASSEMBLY

The airframe structure consisting of eleven modules of primary and/or secondary structure is joined on final assembly, together with the nose and main landing gear, by mechanical attachment (see Figure 55). The various module attachment joints fall into five categories: manufacturing joints, field splice joints, quick-release joints, mechanical entrapment (sliding door rollers and guider in tracks), and pivot/bolt joints.

There is a field splice joint at station 239 which facilitates rapid removal of the tailboom and empennage from the main fuselage. The horizontal stabilizer is connected to the tailboom by a pivot/bolt joint and is easily removed after first detaching the tailcone fairing by cover plate removal and release of quarter-turn fasteners. The manufacturing joints are bolted attachments, with either locking type anchor nuts or ordinary locknuts and suitable core material stabilization at the joint. The modular airframe construction method affords high repairability turnaround times, in that individual modules may be relatively easily replaced if seriously damaged. Within each module, localized damage is repaired in situ by overlap splice and similar secondary bonding schemes.

The various fairings contained on certain of the modules are detachable by quick-release quarter-turn fasteners.

A general description of each module is included in this section. Isometric sketches indicating design details and materials for each are snown with the description of each module. The capacity of the crash-resistant fuel tank is 260 gallons, not including 5-percent void content; the space available in the cargo bay is 145 cubic feet, leaving 6 inches of clearance around the sides and at the top.

An aluminum wire mesh (adhesive bonded) may be added on all exterior skin panels of modules with connecting bonding straps for lightning protection if necessary. (The weight penalty for this is 50 pounds total.) Other suitable protection schemes are under investigation.

4.3.3 SECONDARY STRUCTURE

This category of structure requires stiffness, high impact resistance, and resistance to erosion and corrosion. Kevlar 49 meets all these requirements and will be used exclusively for all designated secondary structure items. It has similar fabrication characteristics to fiberglass; any operations which can be completed in S or E glass can be performed with Kevlar 49. (The drilling and cutting of Kevlar 49 requires slightly different tools and procedures which have been established.)

Studies to date indicate that the most cost-effective secondary structure design commensurate with low weight is honeycomb sandwich construction. However, many of the secondary structure panels required are in areas of compound curvature where it is not feasible to use regular hexagonal core and where broadgoods layup skin panels will require the introduction of material darts and similar time-consuming hand operations to ensure a wrinkle-free compound contour.

One solution suitable for production quantities is the use of fiber-reinforced thermoplastic design, whereby a suitable sandwich panel in the flat is fabricated using selected composite woven fiber broadgoods with a thermoplastic matrix as upper and lower skins sandwiching a nonmetallic reinforced plastic "Flex Core" (Hexcel). The Flex Core cell configuration provides for excellent formability into compound curvatures with reduced antielastic curvature and controlled buckling of the cell walls. HRH-10 Nomex is the selected material for the Flex-Core configuration.

Secondary paneling of this construction may be laid up and cocured in the flat, then subsequently postformed under heat and pressure in a mold to form any reasonable compound-curved shape.

4.3.4 AIRFRAME CRASHWORTHINESS CAPABILITY

The cockpit and cabin structure was specially designed to incorporate all crashworthiness requirements to the 95th percentile to comply with MIL-STD-1290 (AV), and a tail bumper with energy absorber unit is incorporated to protect tailboom during high impact flared landings (see Figure 58). Sections A-A and B-B of Figure 58 illustrate the pilot and troop crash attenuating seats, respectively. Figure 59 shows further details of pilot seat, including armor protection, and Figure 60 shows additional troop seat details.

The completed airframe design was reviewed for anticipated capability to meet specification requirements. The results are summarized in Table 16.

4.3.5 TYPICAL PANEL CONSTRUCTION DETAILS

Figures 61 and 62 illustrate design details of two typical composite honeycomb sandwich frames, and a similar construction typical skin panel which would be subject to light to medium stress intensities with a relatively high vibration environment. The primary structure honeycomb sandwich components of the airframe are designed as thermosetting moldings using an epoxy matrix, while the secondary components are thermoplastic post formings using a phenoxy matrix. The objective was to minimize cost by generating a simple low-weight design with minimum parts count, no special machining of the core, positively sealed core edges, minimal fabrication operations, minimum curing cycles and time required in an autoclave, and also to meet load and fatigue criteria.

4.3.5.1 Frame

The frame inner and outer skins are molded together around the edges to form an integral flange with no separate edge members required. The higher loaded frames will have extra ply reinforcements added around the flange cap locality. The Nomex core is supplied with sufficient tolerance control on its face surfaces that no machining is necessary, and therefore trimming of the edges to the required shape is the only operation required on the core. The bonded preformed C.C.A. foam wedge strip serves as an efficient core seal and forms a ramp for bringing the inner skin down onto the outer skin; it also prevents collapse of the core edge during cure under pressure. The manufacturing method for a production run on this item is a one-shot cocure mold operation using match metal mold die tooling. The inner and outer skins are prepreg tape layups using an automatic layup machine. The correct shape skins are laid up on a flexible carrier for handling into the molds, and an adhesive sheet is interposed between the skin and core on each side. The semiautomatic fabrication and minimal cycle curing combined with low parts count for each component should result in a low-weight cost-effective airframe.

4.3.5.2 Skin Panel

The design of this component is similar to that of the frame in that the inner and outer skins are brought together and bonded around edges without the requirement of separate edge members; however, in this case, the skins are not flanged over at 90 degrees to the surface as on the frames. The corner treatment of the skin panel presents a minor problem in that with anything but very thin core the inner skin will not drape neatly over the corners, as material must be lost or skin wrinkles will result. A solution is to miter-slot the corners of the inner skin to prevent puckers, then, in a separate operation, bond a thermoplastic molding onto the inner panel surface at each corner.

4.3.6 AIRFRAME CONSTRUCTION

The following notes apply to the descriptions of the different airframe modules and the sketches that accompany them.

1. Graphite AS - This is an intermediate strength, relatively low-cost graphite material.

2. Kevlar 49 - This is an organic fiber made by the DuPont Company and supplied as a filament, a prepreg tape, or in woven form by various companies.

3. C.C.A. - This is Cellular Cellulose Acetate, a strux foam, precured with a density of 6 pounds per cubic foot.

4. Where fiberglass is called out on sketches, assume that it is E-glass/epoxy unless otherwise stated.

5. Where graphite and Kevlar 49 are called out, assume that an epoxy matrix is used unless otherwise stated.

6. Assume that all joints are bonded unless otherwise stated, in both primary and secondary structure.

7. The isometric sketches are pictorial only and do not represent any particular scale; also the thicknesses of skins, panels, etc., may be exaggerated.

8. Isometric module sketches do not necessarily illustrate all detail structure, but they do identify basic primary and secondary structure required (for example, support brackets for controls and equipment, etc., are not shown).

Allowance is made for these items in the 20-percent contingency added to each module parts count).

9. NSRP = Neutral Seat Reference Point

4.3.6.1 Cockpit Enclosure - Module 1 (See Figure 63)

This is the structure housing the flight crew forward of the fuselage splice joint at sta 91. The cockpit enclosure structure does not include any floor structure which extends into the cockpit, being part of the underfloor structure module.

The pilot and copilot load attenuating seats are located side by side, separated by a center control console which connects at its forward end to a YUH-61A configuration instrument panel.

Ingress and egress to the cockpit is by a hinged door at each side, which is jettisonable in an emergency. A further emergency exit is available at each side through knockout acrylic windows located in the ceiling above each crew member.

Lightweight windshields are of glass/acrylic nonscratch sandwich construction with the capability of resisting a 4-pound bird impact at 150 knots. Ceiling and door windows are stretched acrylic. Windshield and window sizes and locations essentially meet the external vision plot requirements of MIL-STD-850B for helicopters with side-by-side seating (see Figure 64).

In the extreme nose section of the cockpit, an external upward-opening door provides access to avionics units mounted on two shelves. A telescoping-tube fitting supports the door in the open position. The nose avionics units are isolated from the cockpit area by longitudinal and lateral diaphragms extending from the floor to the upper nose skin, which assist in stiffening the nose structure.

The pilot and copilot seats are of the energy-absorbing type, which attenuate in vertical, longitudinal, and lateral directions. Special wells in the underfloor structure allow for vertical stroking of the seats, which are of similar design to YUH-61A seats (see Figure 59). The seats are adjustable up and down and fore and aft.

The cockpit structure comprises a skeletal structure of windshield posts, door posts, a horizontal eyebrow arch, windshield support moldings, and a large box frame with vertical posts; the latter provides the main tie-in structure to the floor and upper deck modules.

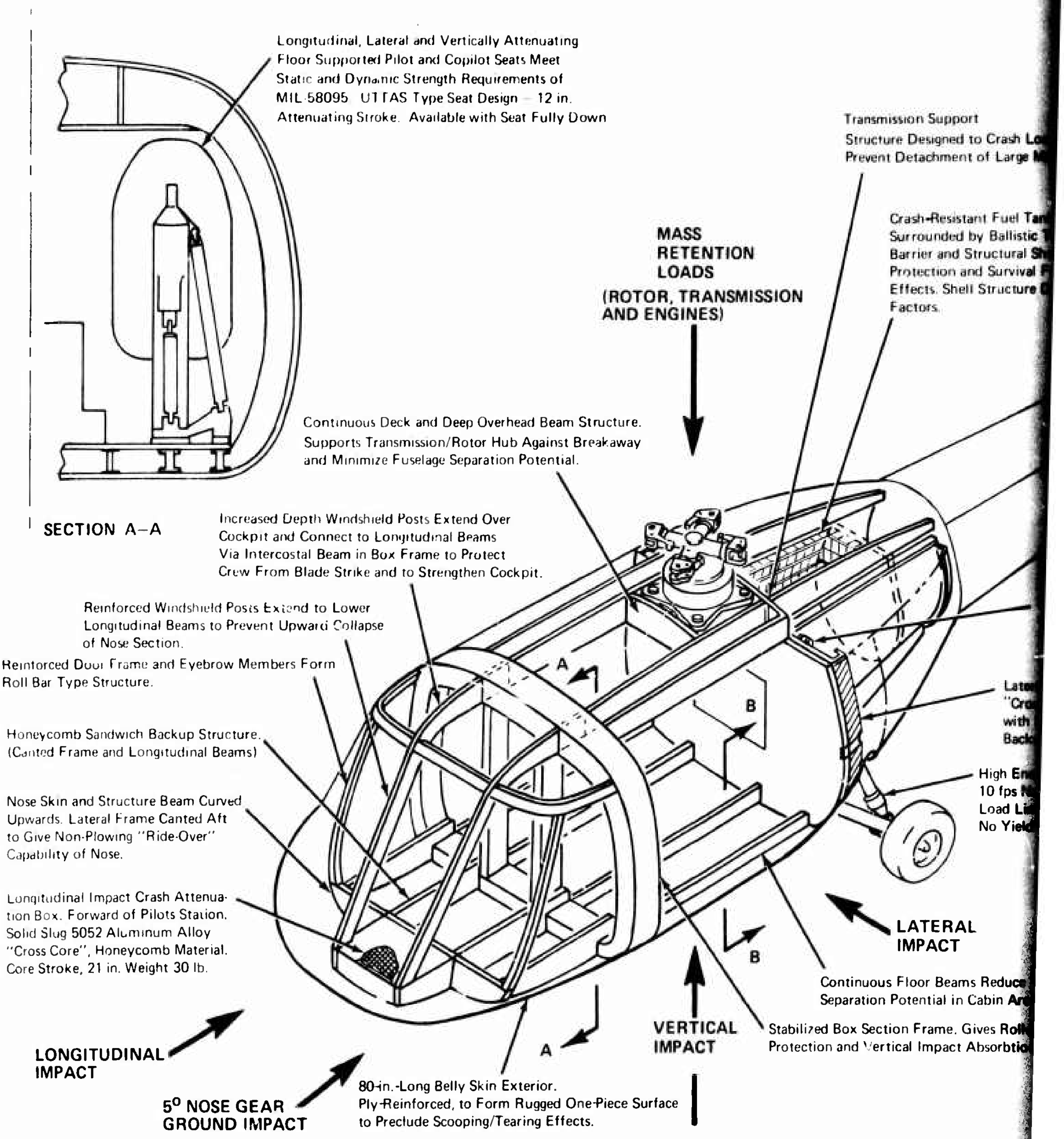
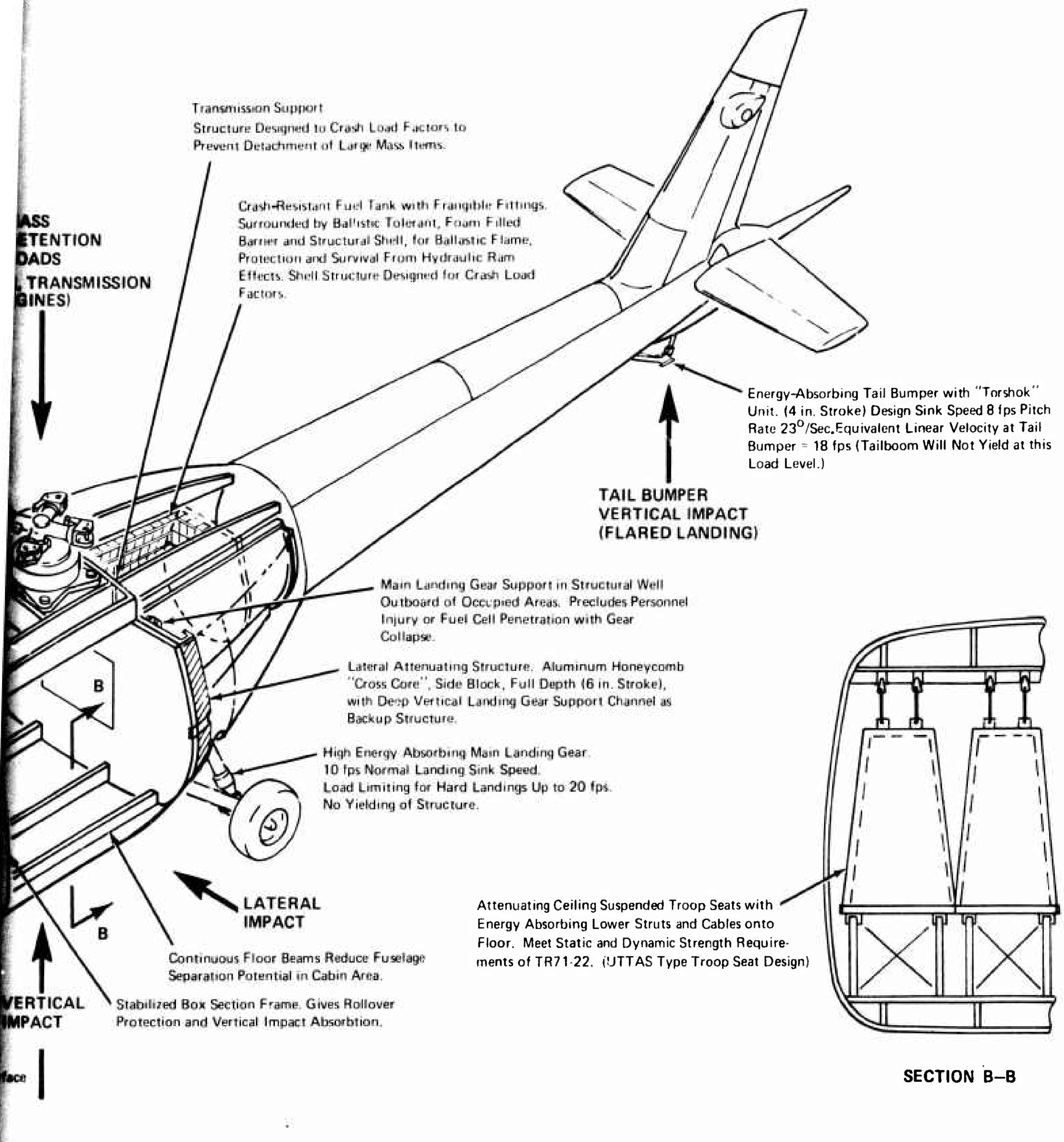


Figure 58. MUT Airframe Crashworthy Features.



Transmission Support
Structure Designed to Crash Load Factors to Prevent Detachment of Large Mass Items.
Crash-Resistant Fuel Tank with Frangible Fittings. Surrounded by Ballistic Tolerant, Foam Filled Barrier and Structural Shell, for Ballastic Flame, Protection and Survival From Hydraulic Ram Effects. Shell Structure Designed for Crash Load Factors.
ASS
ETENTION
DADS
TRANSMISSION
GINES)
Energy-Absorbing Tail Bumper with "Torshok" Unit. (4 in. Stroke) Design Sink Speed 8 fps Pitch Rate 23°/Sec.Equivalent Linear Velocity at Tail Bumper = 18 fps (Tailboom Will Not Yield at this Load Level.)
TAIL BUMPER
VERTICAL IMPACT
(FLARED LANDING)
Main Landing Gear Support in Structural Well Outboard of Occupied Areas. Precludes Personnel Injury or Fuel Cell Penetration with Gear Collapse.
Lateral Attenuating Structure. Aluminum Honeycomb "Cross Core", Side Block, Full Depth (6 in. Stroke), with Deep Vertical Landing Gear Support Channel as Backup Structure.
B
High Energy Absorbing Main Landing Gear.
10 fps Normal Landing Sink Speed.
Load Limiting for Hard Landings Up to 20 fps.
No Yielding of Structure.
LATERAL
IMPACT
B
Attenuating Ceiling Suspended Troop Seats with Energy Absorbing Lower Struts and Cables onto Floor. Meet Static and Dynamic Strength Requirements of TR71-22. (UTTAS Type Troop Seat Design)
Continuous Floor Beams Reduce Fuselage Separation Potential in Cabin Area.
ERTICAL
IMPACT
Stabilized Box Section Frame. Gives Rollover Protection and Vertical Impact Absorbtion.
face
SECTION B–B

Vertical Seat Adjustment = 5.5 in. (Up 3 in., Down 2.5 in.)

Longitudinal Seat Adjustment = 6.0 in. (Fwd 1.5 in., Aft 4.5 in.)

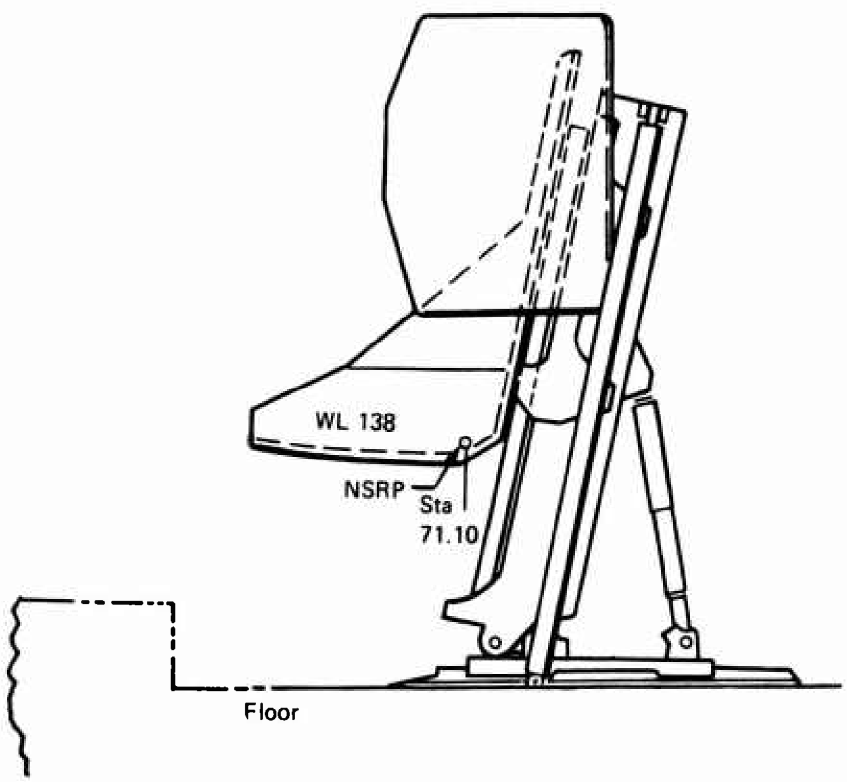


Figure 59. Attenuating Pilots Seat.

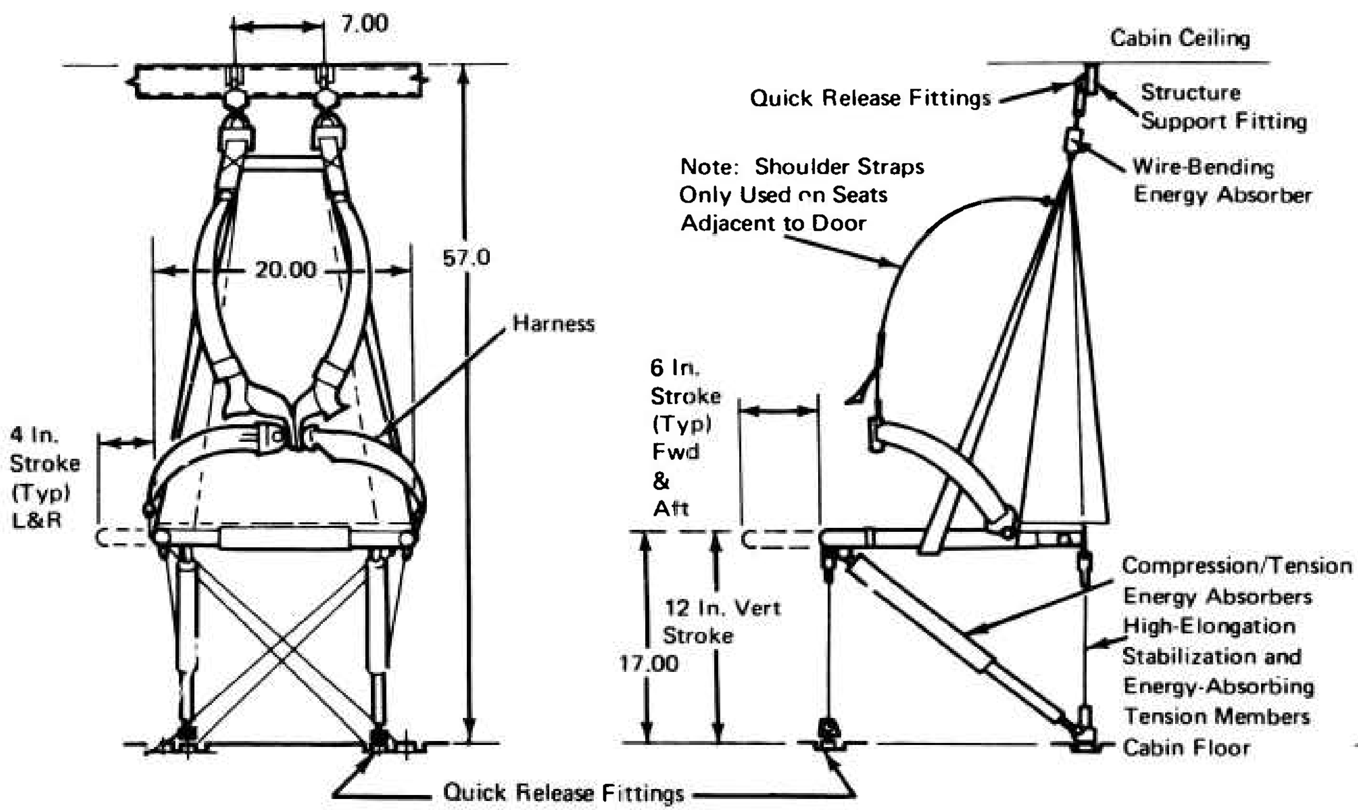


Figure 60. Attenuating Troop Seat.

Table 16. MUT Airframe Crashworthy Capability

IMPACT CONDITION	SPEC REQUIREMENT	MUT CAPABILITY ESTIMATE
LONGITUDINAL	20 FPS INTO RIGID WALL SAFE EVACUATION OF CREW	MEETS REQUIREMENTS
	40 FPS INTO RIGID WALL TROOP COMPARTMENT REDUCTION NO MORE THAN 15%	MEETS REQUIREMENTS
	60 FPS AT 15° NOSE DOWN – REDUCTION OF COCKPIT OR TROOP COMPARTMENT LIVING SPACE NO MORE THAN 5%	MEETS REQUIREMENTS
VERTICAL	42 FPS – LIVING SPACE REDUCTION NO MORE THAN 15%	PARTLY MET *
LATERAL	30 FPS – REDUCTION IN COMPARTMENT LIVING SPACE NO MORE THAN 15%	MEETS REQUIREMENTS
TURN OVER STRUCTURE	AIRCRAFT RESTING ON GROUND 4 W PERPENDICULAR TO WL 4 W LONGITUDINALLY PARALLEL TO WL 2 W LATERALLY	MEETS REQUIREMENTS
	GROUND IMPACT AT 100 FPS AT 5° ANGLE-PASSENGER OCCUPIED VOLUME REDUCTION NO MORE THAN 15%	MEETS REQUIREMENTS
NOSE PLOWING	FORWARD 25% FUSELAGE UNIFORMLY LOADED 10g UP AND 4g AFT (10g BASED ON EFFECTIVE MASS) PRECLUDE SCOOPING	MEETS REQUIREMENTS
TAIL BUMPER	MIL-A-008862A	MEETS REQUIREMENTS
	10 FPS SINK SPEED AND PITCH ATTITUDE CORRESPONDING TO IGE HOVER IN 60 KNOT TAIL WIND	MEETS REQUIREMENTS
BLADE STRIKE	ROTOR MAST SHALL NOT FAIL TRANSMISSION SHALL NOT BE DISPLACED INTO OCCUPIABLE SECTION WHEN MAIN ROTOR BLADES IMPACT INTO A RIGID 8-IN.-DIAMETER OBJECT IN THE OUTER 10% BLADE RADIUS AT OPERATIONAL ROTOR SPEED	MEETS REQUIREMENTS

* In the event of a 42-fps vertical impact stroking of the main landing gear with 20 fps capability leaves residue of 36 fps to be absorbed by the fuselage structure. Further energy is absorbed by the gear metal attachment fittings passing through the plastic stage before fracture, and also by composite structure deflections.

Lateral Attenuation – Predicated on main landing gear striking obstruction first and absorbing some energy before airframe side core is attenuated

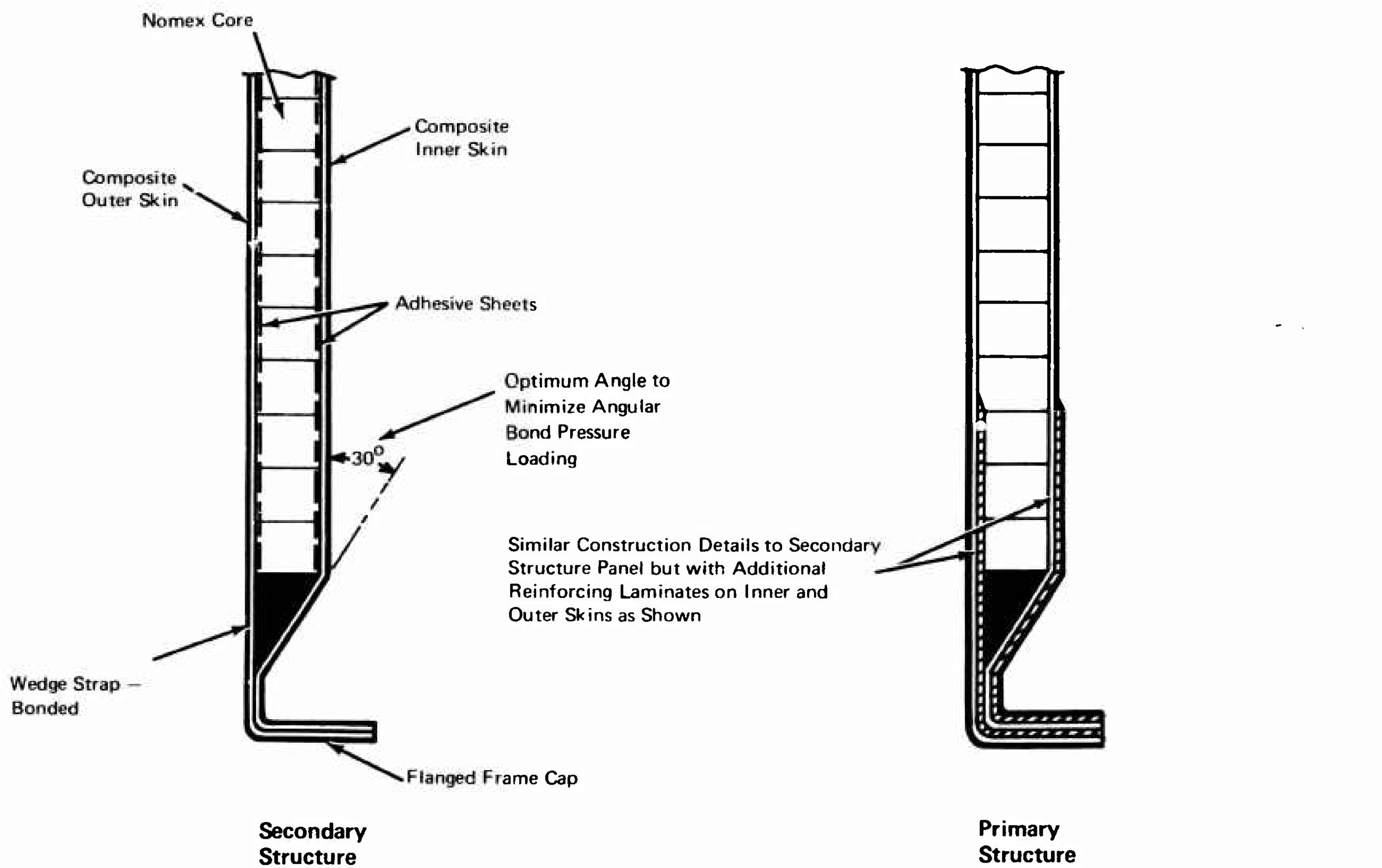


Figure 61. Typical Sandwich Frame Construction.

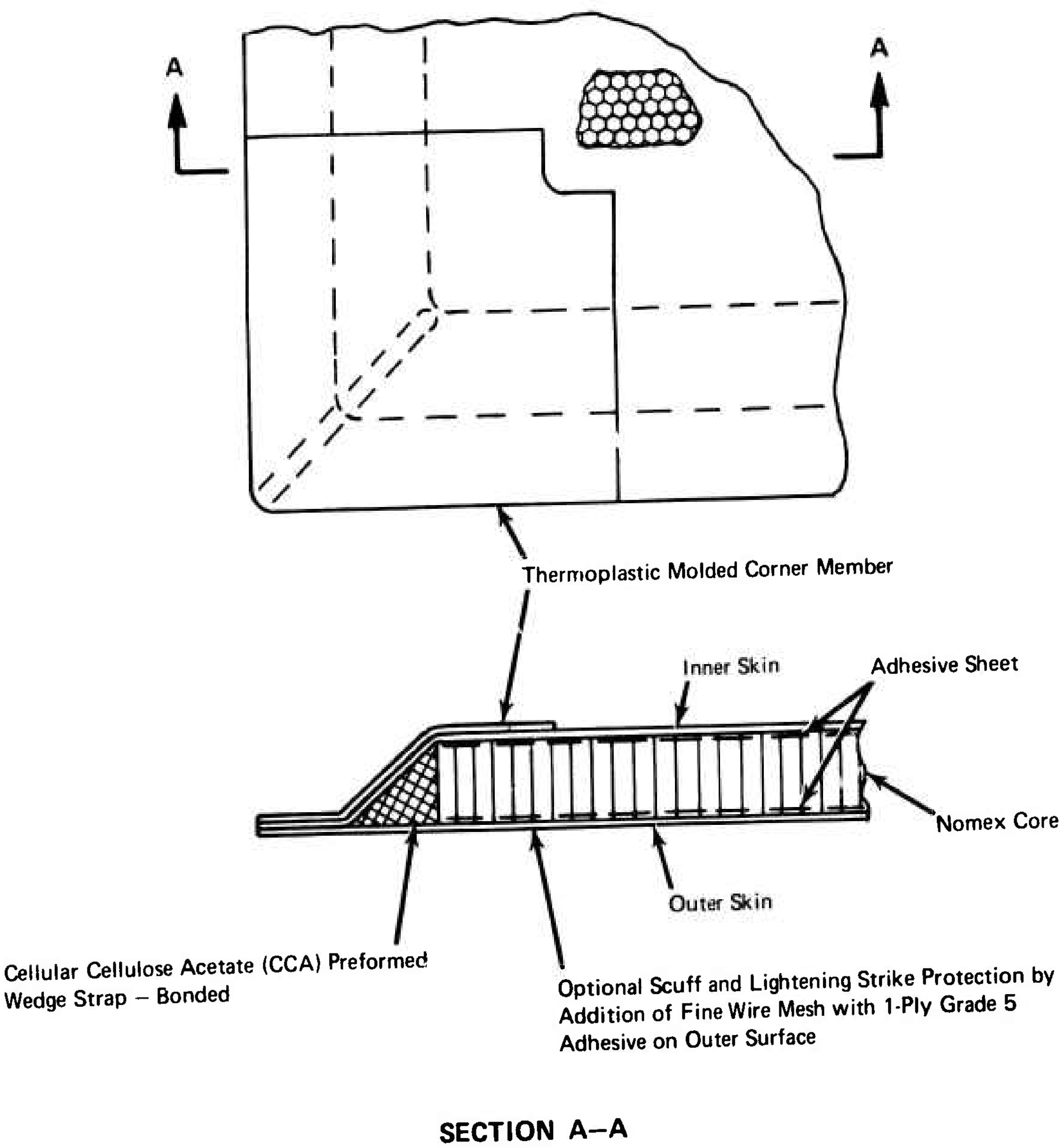


Figure 62. Typical Composite Skin Panel.

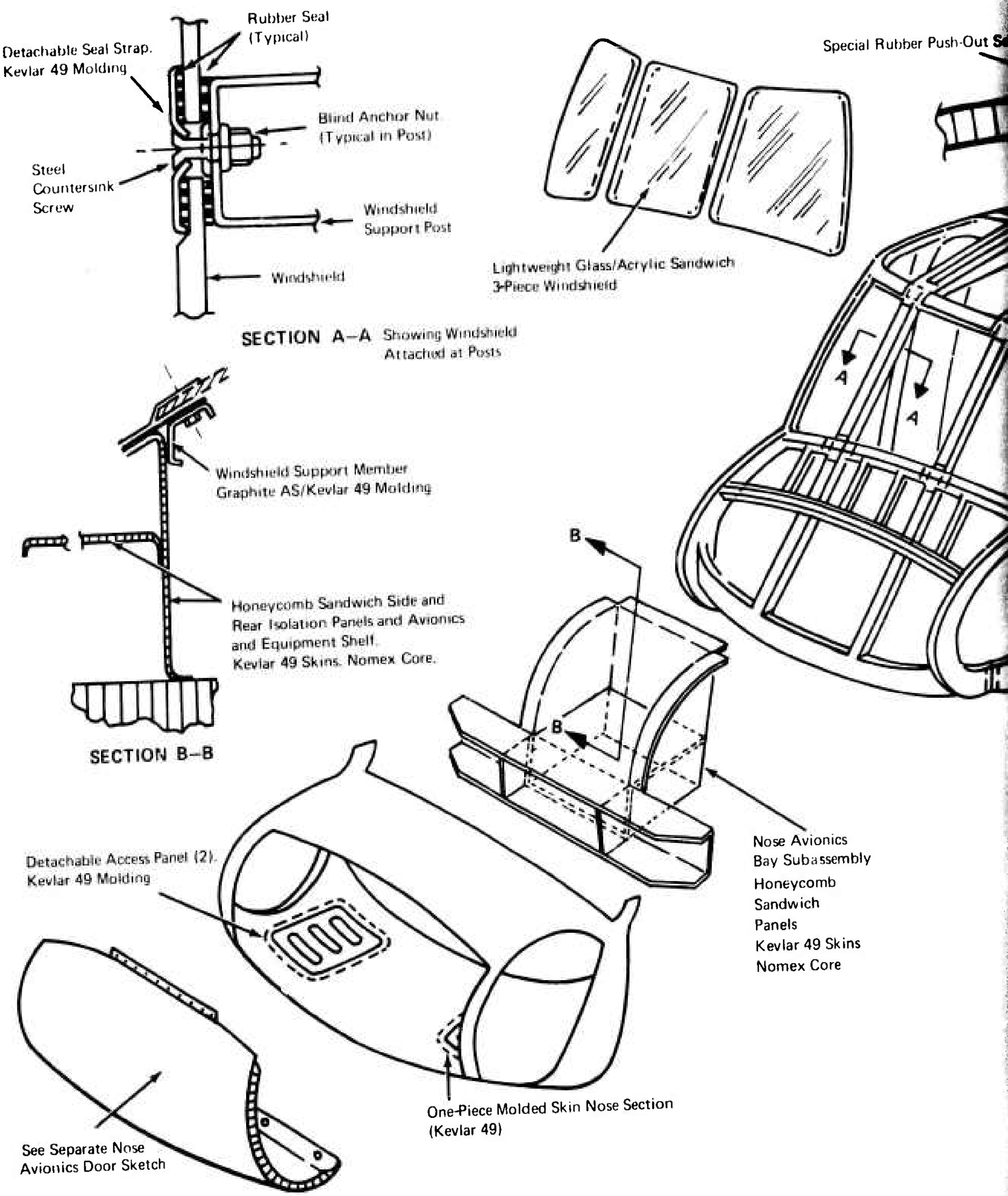
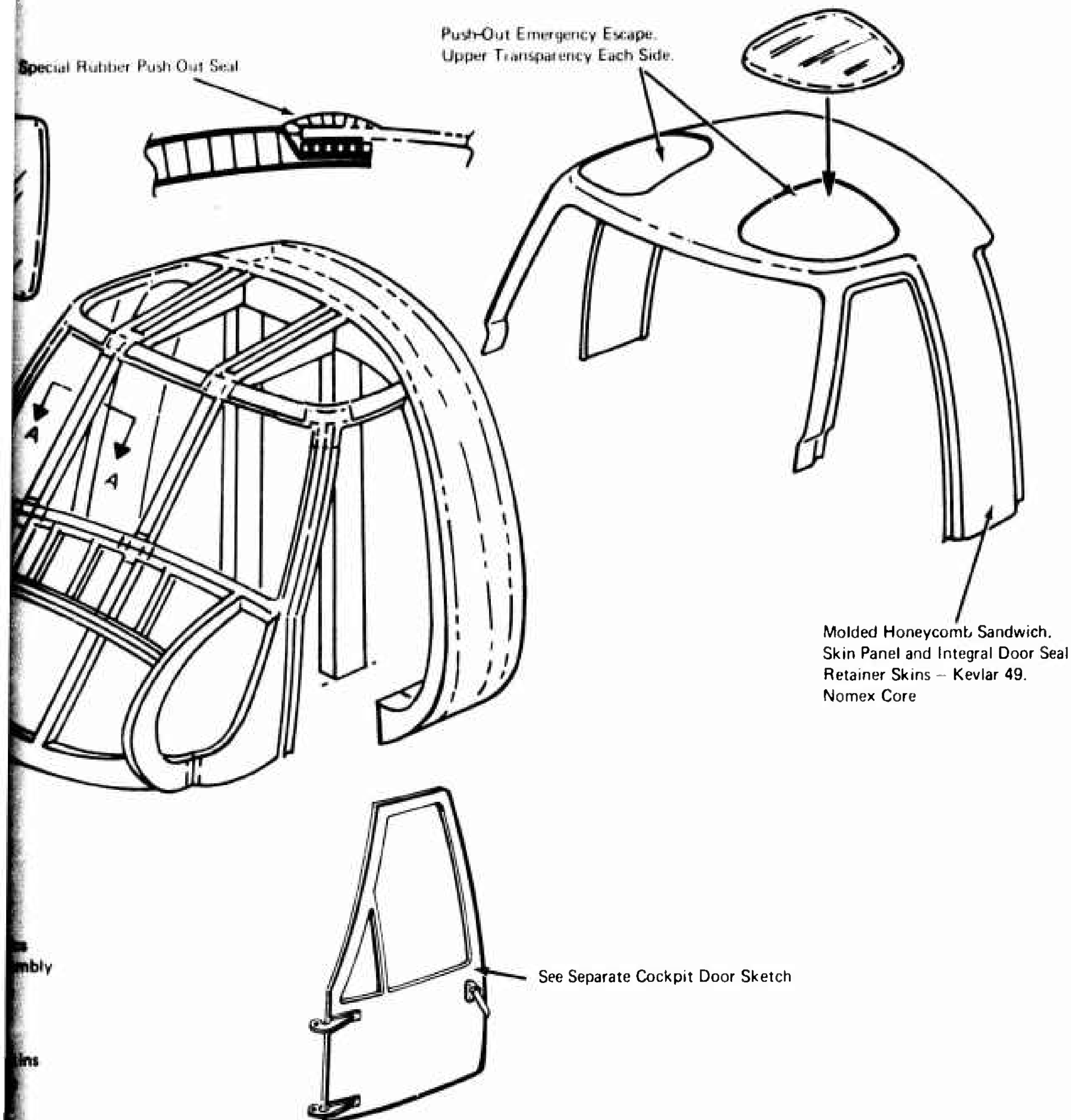


Figure 63. Airframe Module 1 - Canopy Enclosure Skins, Transparencies and Nose Avionics Bay Subassembly. (Sheet 1 of 4)

p.158 Blank


Special Rubber Push Out Seal
Push-Out Emergency Escape.
Upper Transparency Each Side.
A
A
Molded Honeycomb Sandwich.
Skin Panel and Integral Door Seal
Retainer Skins – Kevlar 49.
Nomex Core
See Separate Cockpit Door Sketch
mbly
ins

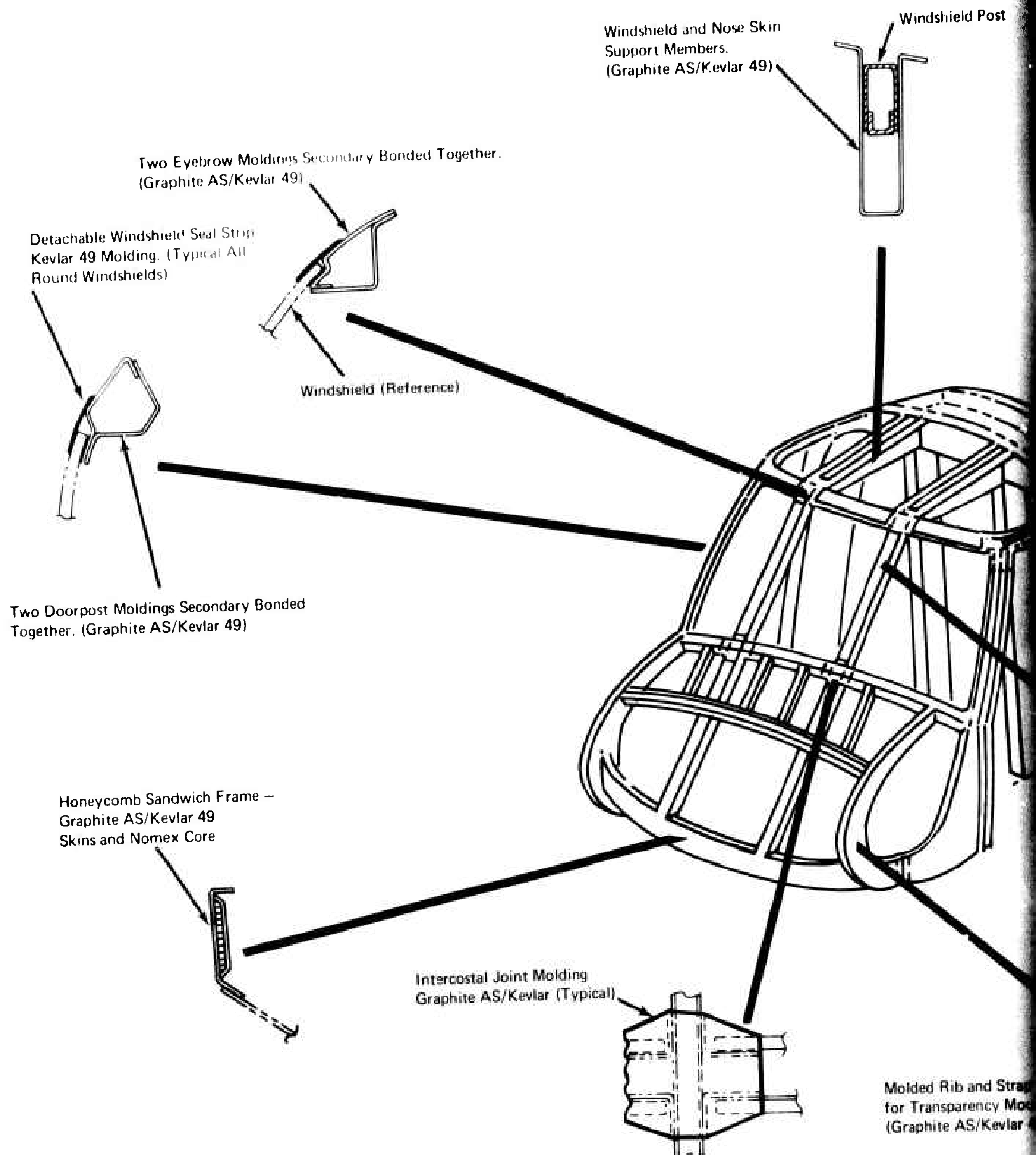
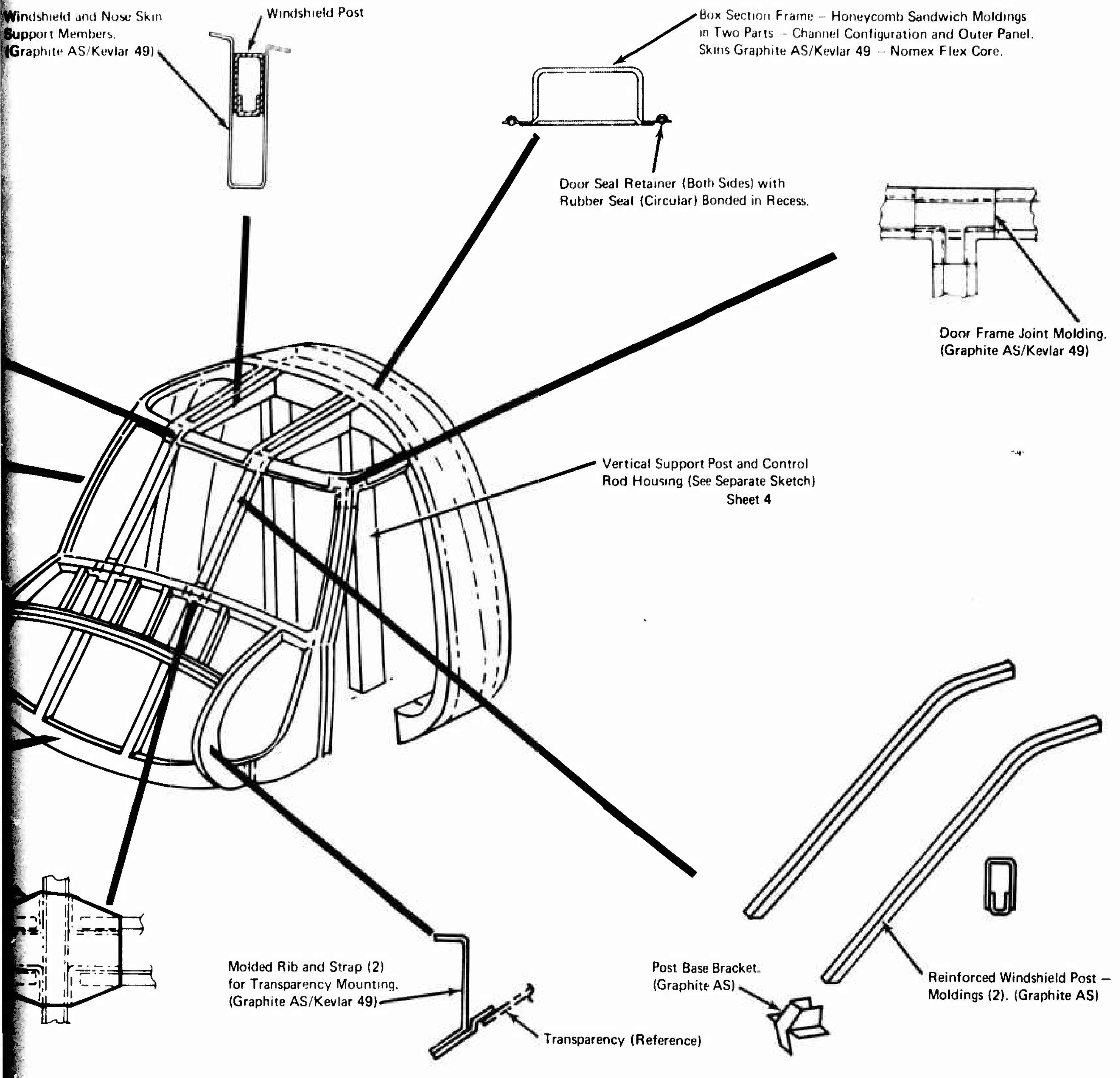


Figure 63. Airframe Module 1 - Canopy Enclosure - Skeletal Subassembly. (Sheet 2 of 4)



Windshield and Nose Skin Support Members. (Graphite AS/Kevlar 49)
Windshield Post
Box Section Frame – Honeycomb Sandwich Moldings in Two Parts – Channel Configuration and Outer Panel. Skins Graphite AS/Kevlar 49 – Nomex Flex Core.
Door Seal Retainer (Both Sides) with Rubber Seal (Circular) Bonded in Recess.
Door Frame Joint Molding. (Graphite AS/Kevlar 49)
Vertical Support Post and Control Rod Housing (See Separate Sketch) Sheet 4
Molded Rib and Strap (2) for Transparency Mounting. (Graphite AS/Kevlar 49)
Transparency (Reference)
Post Base Bracket. (Graphite AS)
Reinforced Windshield Post – Moldings (2). (Graphite AS)

eletal

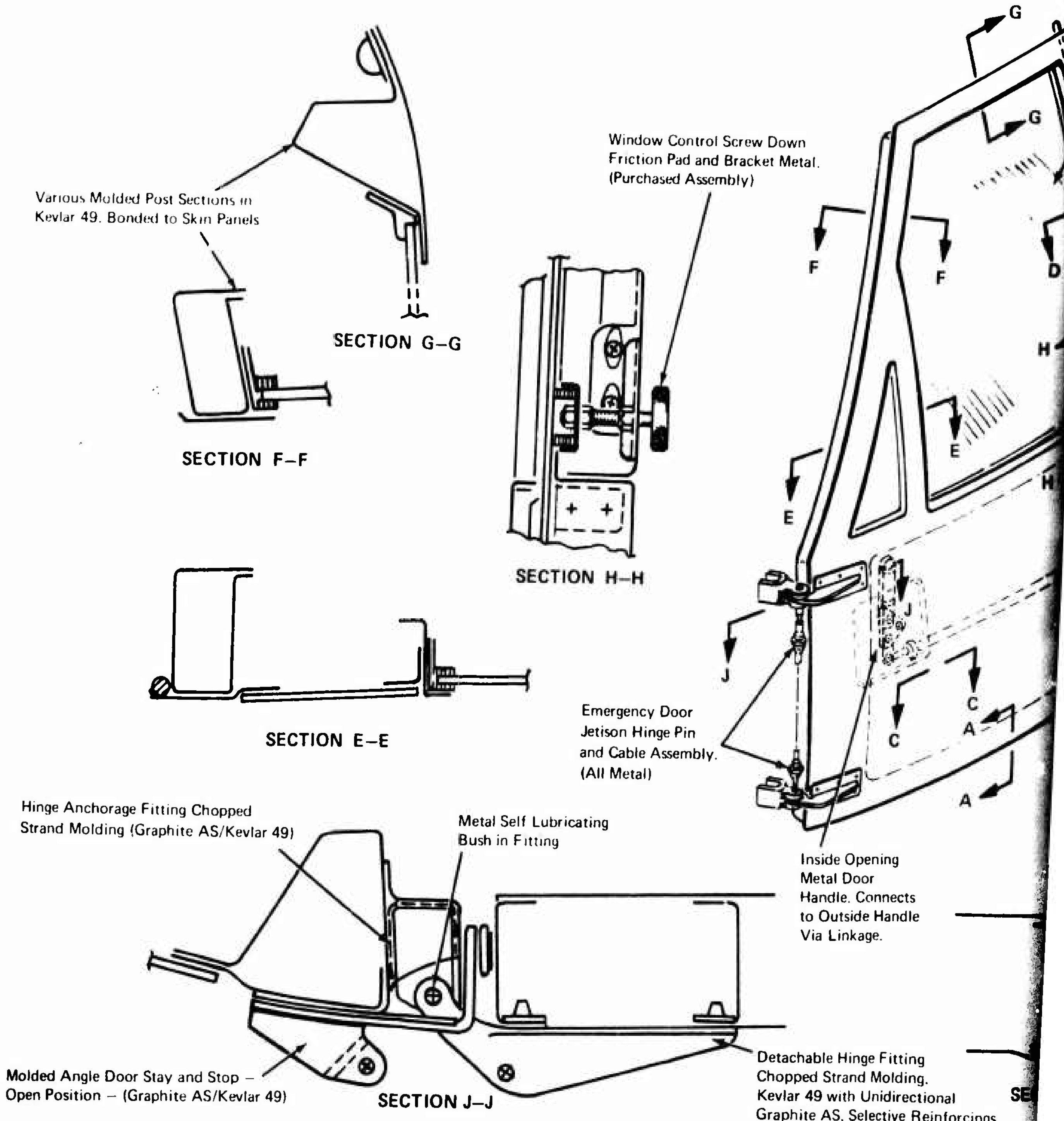
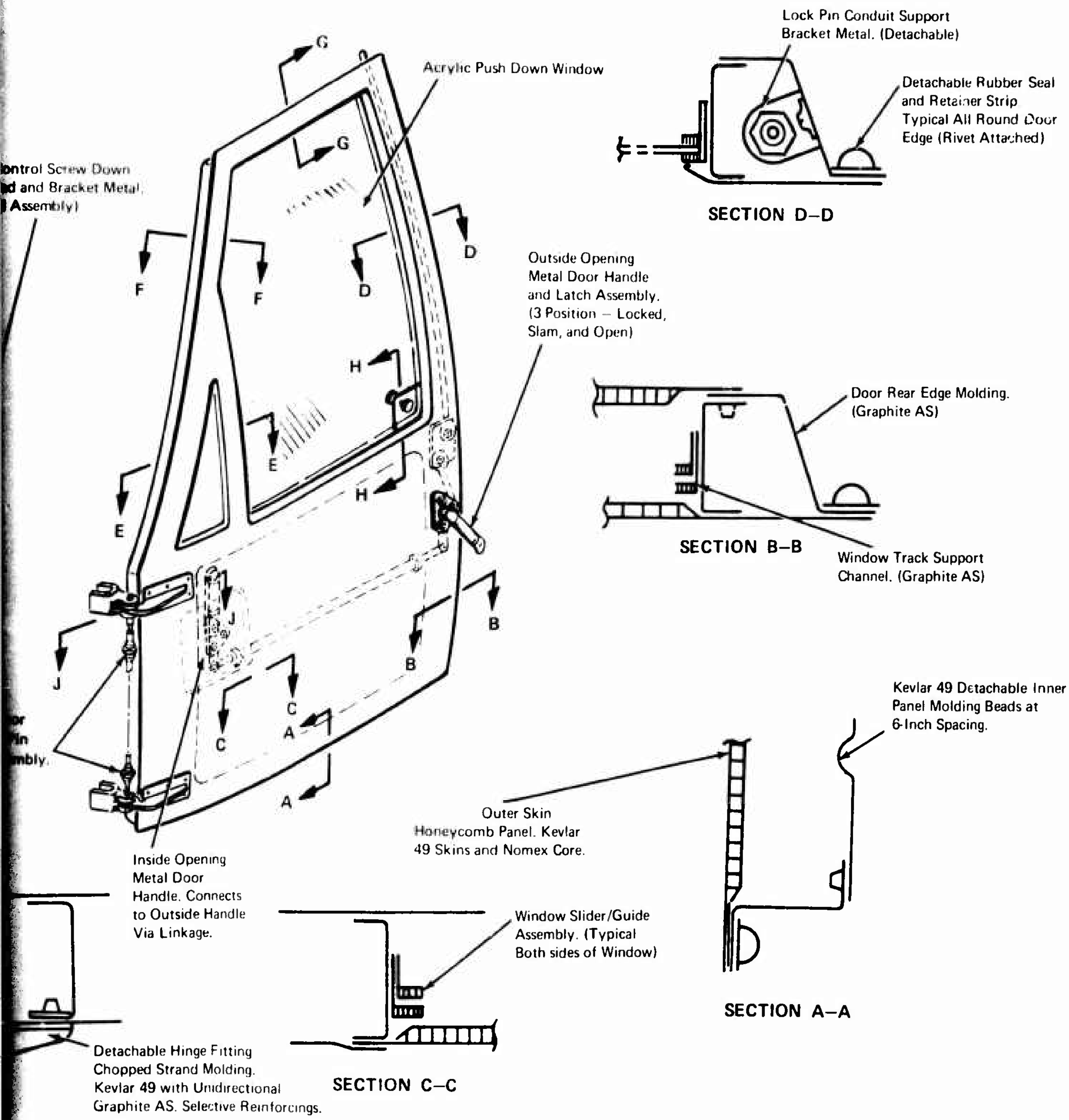


Figure 63. Airframe Module 1 - Canopy Enclosure - Cockpit Door. (Sheet 3 of 4)



Lock Pin Conduit Support
Bracket Metal. (Detachable)
Detachable Rubber Seal
and Retainer Strip
Typical All Round Door
Edge (Rivet Attached)
SECTION D–D
Acrylic Push Down Window
ontrol Screw Down
d and Bracket Metal.
Assembly)
Outside Opening
Metal Door Handle
and Latch Assembly.
(3 Position – Locked,
Slam, and Open)
Door Rear Edge Molding.
(Graphite AS)
SECTION B–B
Window Track Support
Channel. (Graphite AS)
Kevlar 49 Detachable Inner
Panel Molding Beads at
6-Inch Spacing.
Outer Skin
Honeycomb Panel. Kevlar
49 Skins and Nomex Core.
Inside Opening
Metal Door
Handle. Connects
to Outside Handle
Via Linkage.
Window Slider/Guide
Assembly. (Typical
Both sides of Window)
SECTION A–A
Detachable Hinge Fitting
Chopped Strand Molding.
Kevlar 49 with Unidirectional
Graphite AS. Selective Reinforcings.
SECTION C–C

it Door.

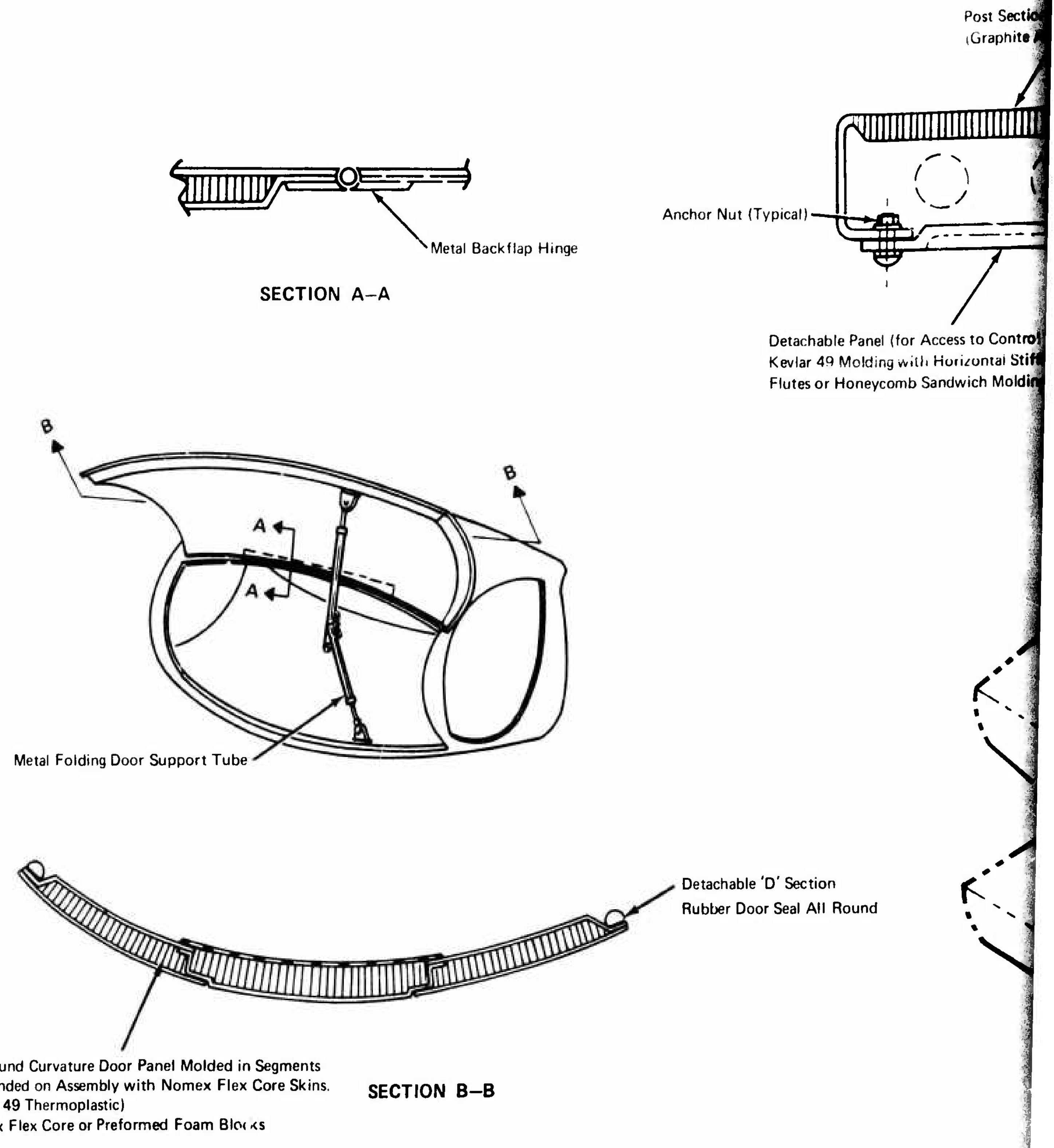
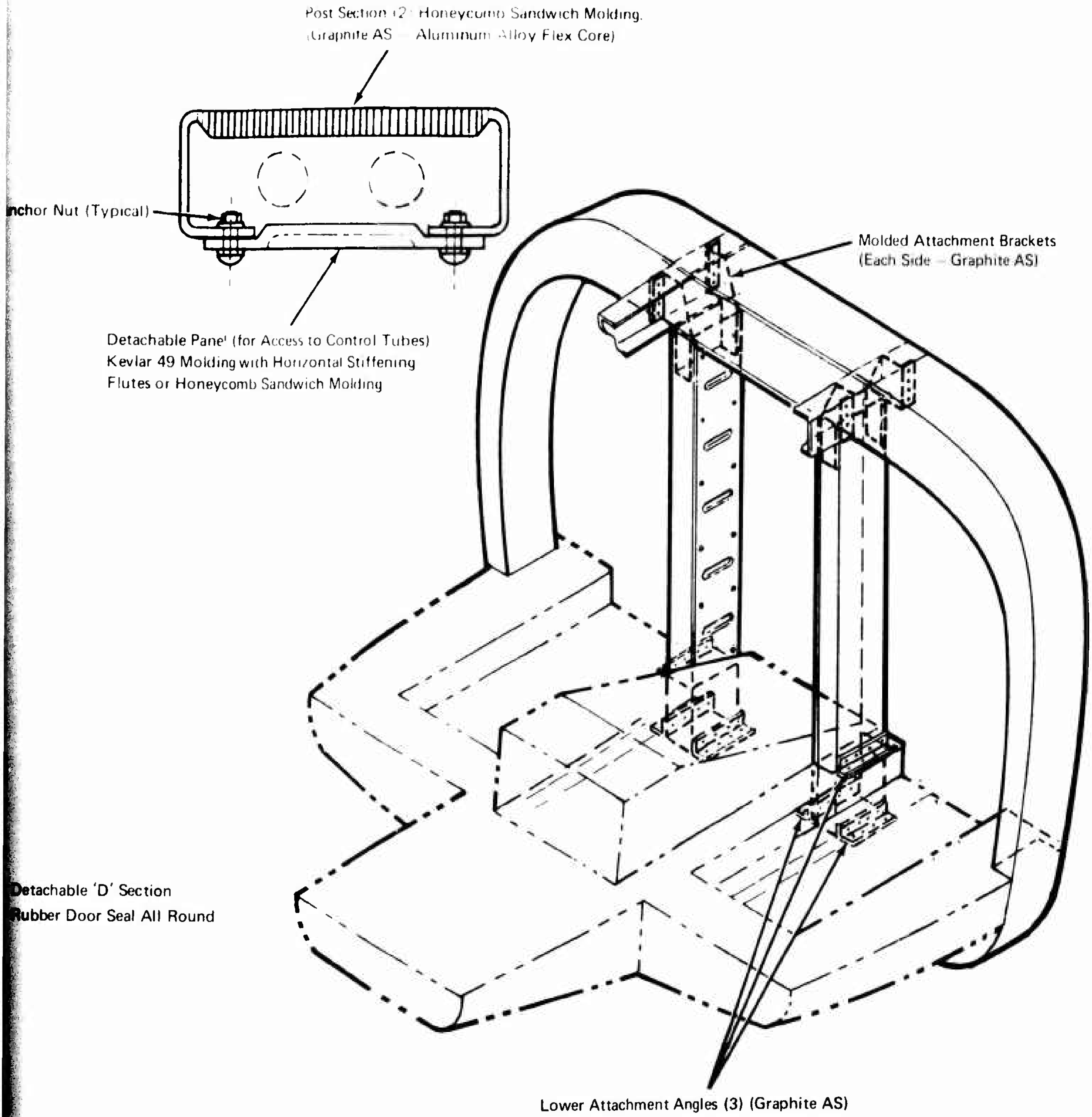
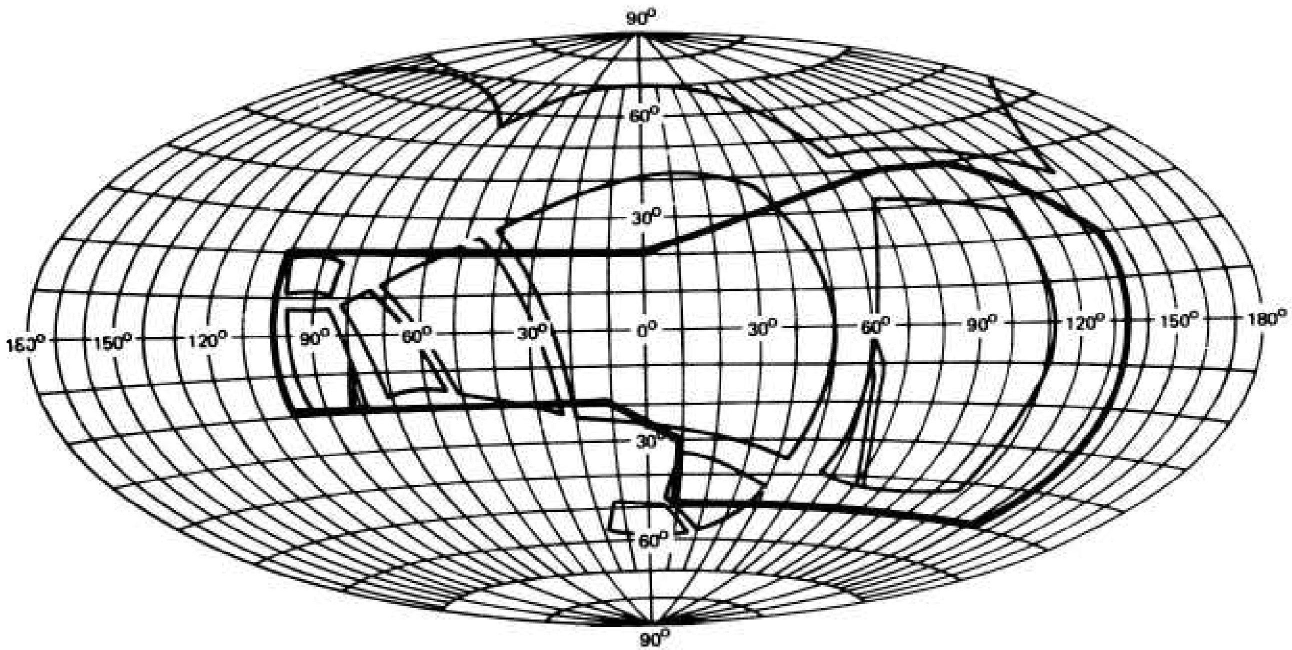


Figure 63. Airframe Module 1 - Canopy Enclosure - Subassembly, Avionics Door and Vertical Posts. (Sheet 4 of 4)


Post Section (2) Honeycomb Sandwich Molding.
(Graphite AS — Aluminum Alloy Flex Core)
nchor Nut (Typical)
Detachable Panel (for Access to Control Tubes)
Kevlar 49 Molding with Horizontal Stiffening
Flutes or Honeycomb Sandwich Molding
Molded Attachment Brackets
(Each Side — Graphite AS)
Detachable 'D' Section
Rubber Door Seal All Round
Lower Attachment Angles (3) (Graphite AS)

ssembly,
of 4)

Aitoff's Equal Area Projection of the Sphere



Key:

- Area Bounded by Thick Line Indicates MIL-STD-850B Requirement for Helicopter with Side-by-Side Seating
- Areas Bounded by Thin Black Lines Indicate Actual MUT Transparencies

Figure 64. MUT External Vision Plot (From Aircrew Stations) Side-by-Side Seating.

Apart from the two windshield posts, which are two-piece channel moldings bonded to form a box section, these members are primarily moldings of channel section arranged so that the external canopy skin moldings can be bonded over them to form continuous box section members without the use of mechanical fasteners. The vertical members provide protection for the crew by preventing the inward collapse of the structure into the cockpit living space. The side posts and eyebrow structure provide protection in the event of rollover, and the windshield posts prevent the upward collapse of the floor structure in the event of a low-angle impact.

The box section frame (sta 81 to 91) is a large hat section structure formed in sandwich construction with a sandwich skin panel bonded on to form a box. Two separately attached vertical posts are part of this frame, which, in addition to being primary load-carrying members, provide a cavity for the flight control system run from the lower section to the upper deck area.

<u>Cockpit Loads</u> - The cockpit enclosure panels and glassed areas are designed for aerodynamic pressure loads, but the governing design conditions for the support posts and rollover frame are to meet the crashworthiness requirements of MIL-STD-1290 (AV) for nose plowing, rollover, longitudinal, and lateral impact accidents. Local loads which may design some details include airloads, bird impact, and personnel induced loads.

The nose gear is supported directly by the floor module, which in turn is partially supported by the cockpit structure. Some load is reacted by the forward windshield post, but the main reaction for the gear loads occurs at the box section frame and main posts at sta 81 to 91. The posts provide the primary load path between the floor and upper deck modules and are sized by large bending and compression forces for both landing and flight conditions.

The box frame is designed primarily by rollover loads of four times the aircraft weight. Additionally, the box frame redistributes side and torsion loads to the upper and lower cabin structure.

<u>MUT Cockpit Enclosure and Structural Arrangement Design Constraints</u> - This section presents an example of composite resizing design limitations. Figure 65 shows the initial consideration which is the accommodation of pilot and copilot in side-by-side crash attenuating seats with sufficient spacing for lateral and vertical stroking. The applicable military specifications are listed in the figure. Figure 66 indicates the next considerations affecting size and shape

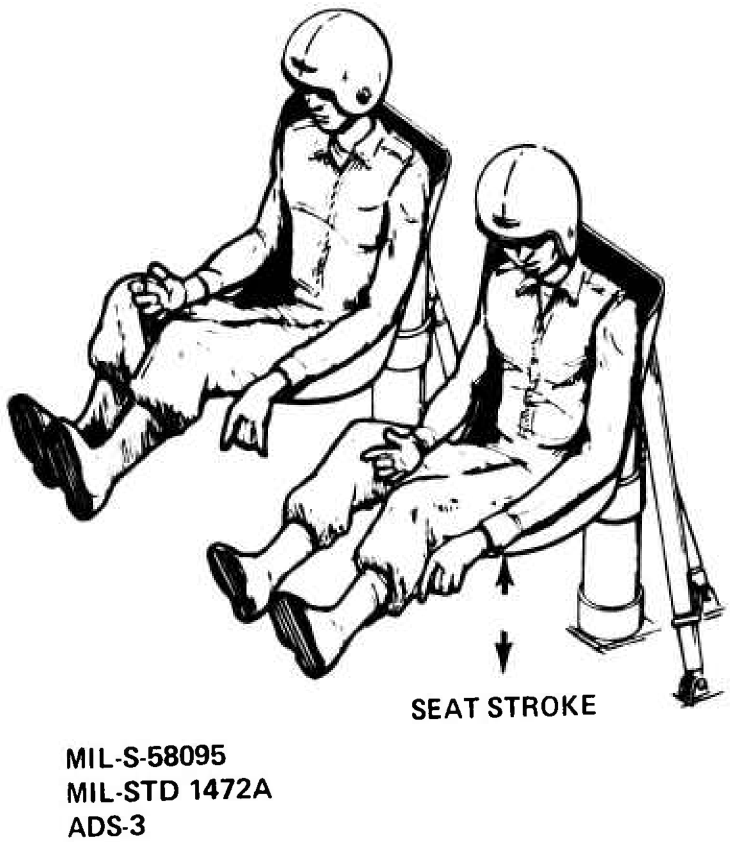
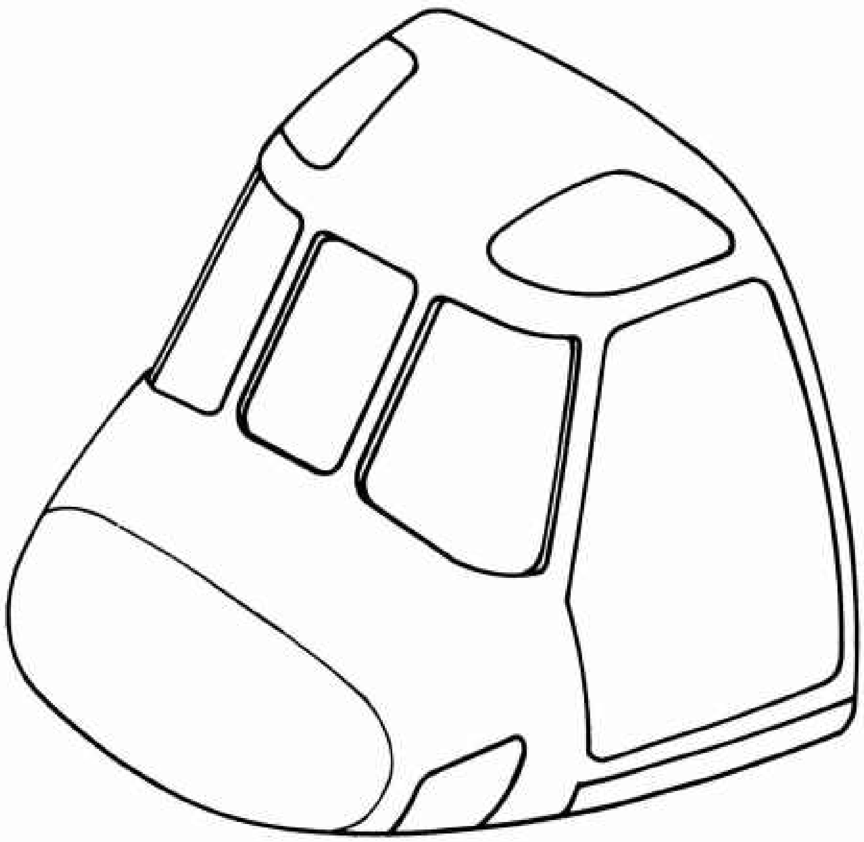


Figure 65. Side-by-Side Pilot and Copilot in Crashworthy Seats.



1) CANOPY INTERNAL CLEARANCE DIMENSIONS AROUND CREW MEMBERS
MS 33575 MS 33573
2) FORWARD, SIDE, UP AND DOWN VISION ANGLE REQUIREMENTS
MIL STD 850 B
3) WINDSHIELD, WINDOW AND DOOR LOCATION – EMERGENCY EGRESS FROM DOORS AND UPPER WINDOWS AT EACH SIDE
4) AERODYNAMIC FAIRING FOR MINIMAL CROSS SECTION

Figure 66. MUT Cockpit Enclosure Design Consideration.

of cockpit enclosure together with applicable specifications. Note that from the vision plot shown in Figure 64 it can be seen that these requirements virtually control the outside dimensions of the enclosure.

Figure 67 spotlights the impact of the crashworthiness requirements from MIL-STD-1290 (AV) on the cockpit area. Figure 68 illustrates all other necessary internal control and equipment items in their required positions which further influence the final size of a modern military helicopter cockpit enclosure.

Figure 69 is a summation diagram with the conclusive message that it is not possible to substantially reduce size of this module despite use of stronger and stiffer composite materials.

These constraints illustrate the effect of military specifications on resizing. Similar constraints exist in the cabin area and those structures required to mount and house standard items of fixed equipment, optional avionics, etc.

4.3.6.2 Floor Structure - Module 2 (See Figure 70)

The floor structure extends from sta 11 to sta 163, actually extending into cockpit enclosure area from sta 11 to sta 91. The most forward floor section consists of a skinned box filled with aluminum alloy cross-core honeycomb core which serves as the primary attenuation means for longitudinal impact loads. The reactive backup structure for the crash box comprises a lateral canted honeycomb sandwich underfloor frame with support from longitudinal beams.

The cabin floor is made up of forward light-duty and rear heavy-duty detachable floor panels supported by an egg-crate-type grid of continuous longitudinal beams with intercostal lateral frames, all of sandwich construction. Five honeycomb sandwich design longitudinal beams are continuous over total floor length, with the two BL 14 beams extending into and forming the side walls of the forward crash box described previously. The center (BL O) beam forms an anchorage for the nosewheel gear (lower attachment). The anchorage fitting also incorporates a jacking point. The rear end of this same beam is also used to support the forward pintle of the rotating cargo hook axle.

A center console box structure located between the pilot seats forms a support platform for controls and instrumentation and also is the mounting position at its forward end for the instrument panel.

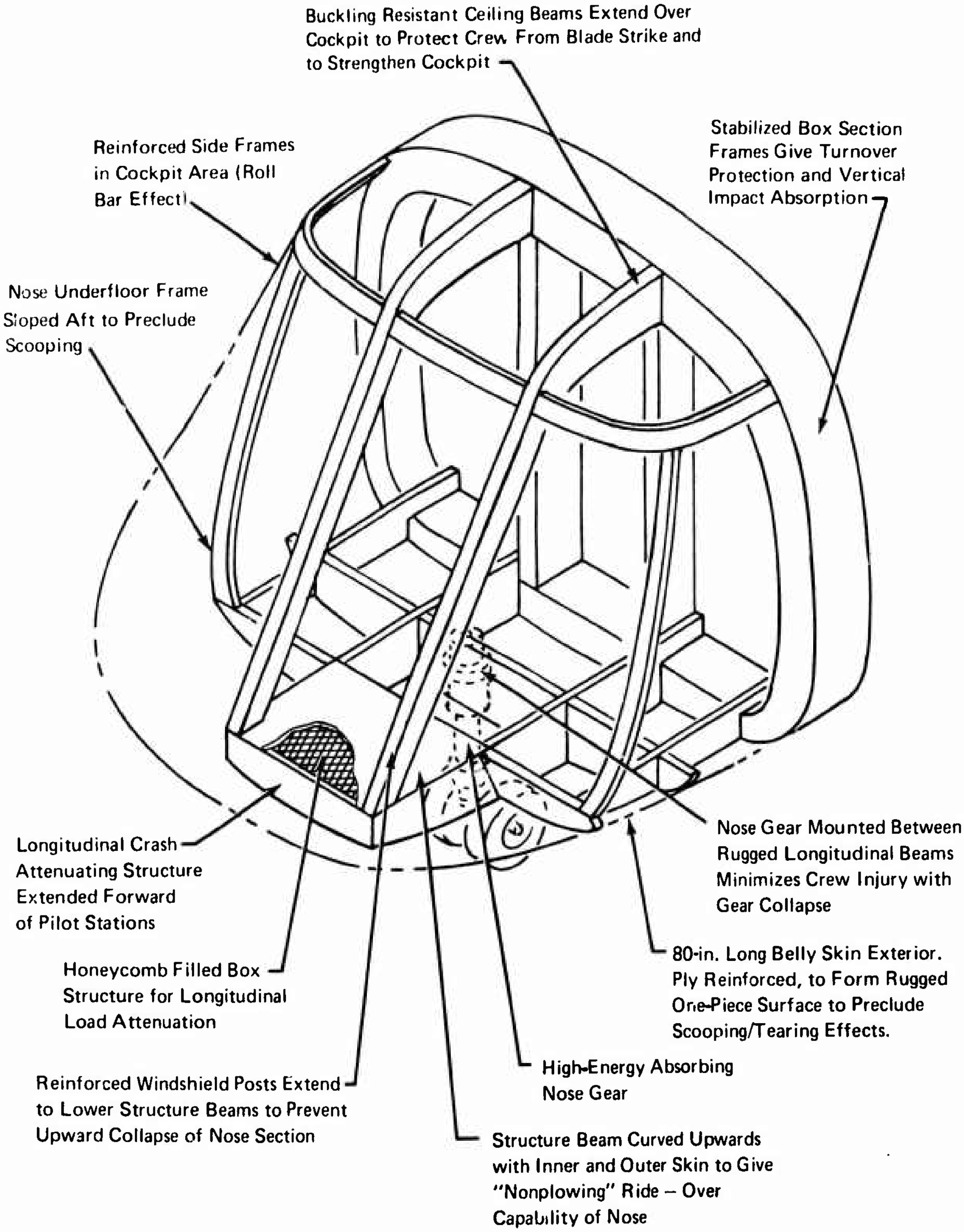
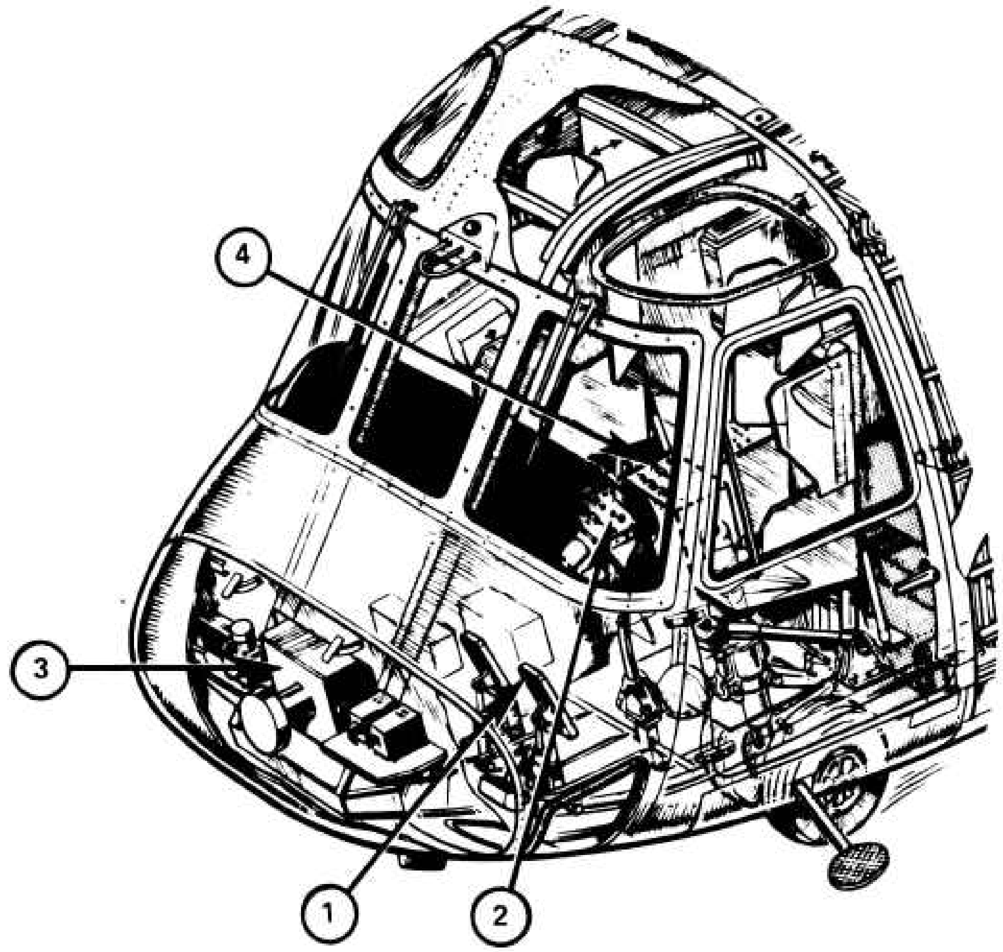
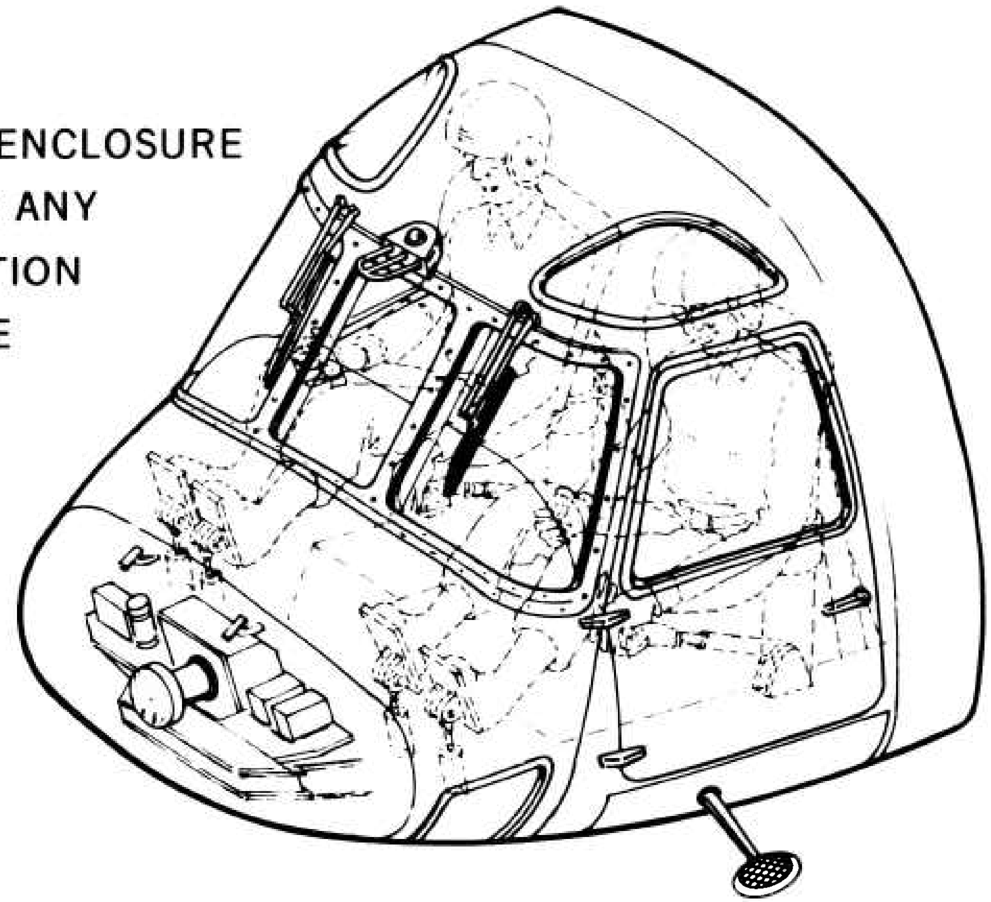


Figure 67. Crashworthy Features in Cockpit Enclosure.



1) CONTROL PEDAL, STICK, AND COLLECTIVE LEVER POSITIONING
2) CENTER CONSOLE & INSTRUMENT PANEL LOCATION MIL-STD-250C
3) FORWARD AVIONICS & BATTERY BAY LOCATION
4) PROTECTIVE HOUSING FOR NOSE LANDING GEAR OLEO

Figure 68 . MUT Cockpit Enclosure Design Considerations.



FULLY EQUIPPED COCKPIT ENCLOSURE REQUIREMENTS PRECLUDE ANY RE-SIZING DUE TO UTILIZATION OF COMPOSITE STRUCTURE SPECIFICATION

NOTES:

1) COCKPIT ENCLOSURE STRUCTURE IS DESIGNED BY CRASHWORTHINESS REQUIREMENTS
2) COCKPIT ENCLOSURE DOES NOT INCLUDE CREW FLOOR STRUCTURE WHICH IS PART OF MAIN FLOOR MODULE
3) OVERALL WIDTH 96 IN.; OVERALL HEIGHT 74.0 IN.; OVERALL LENGTH 89.75 IN. — MINIMUM FAIRED SIZE TO MEET ALL COCKPIT SPECIFICATIONS

Figure 69. MUT Cockpit Enclosure Design Considerations.

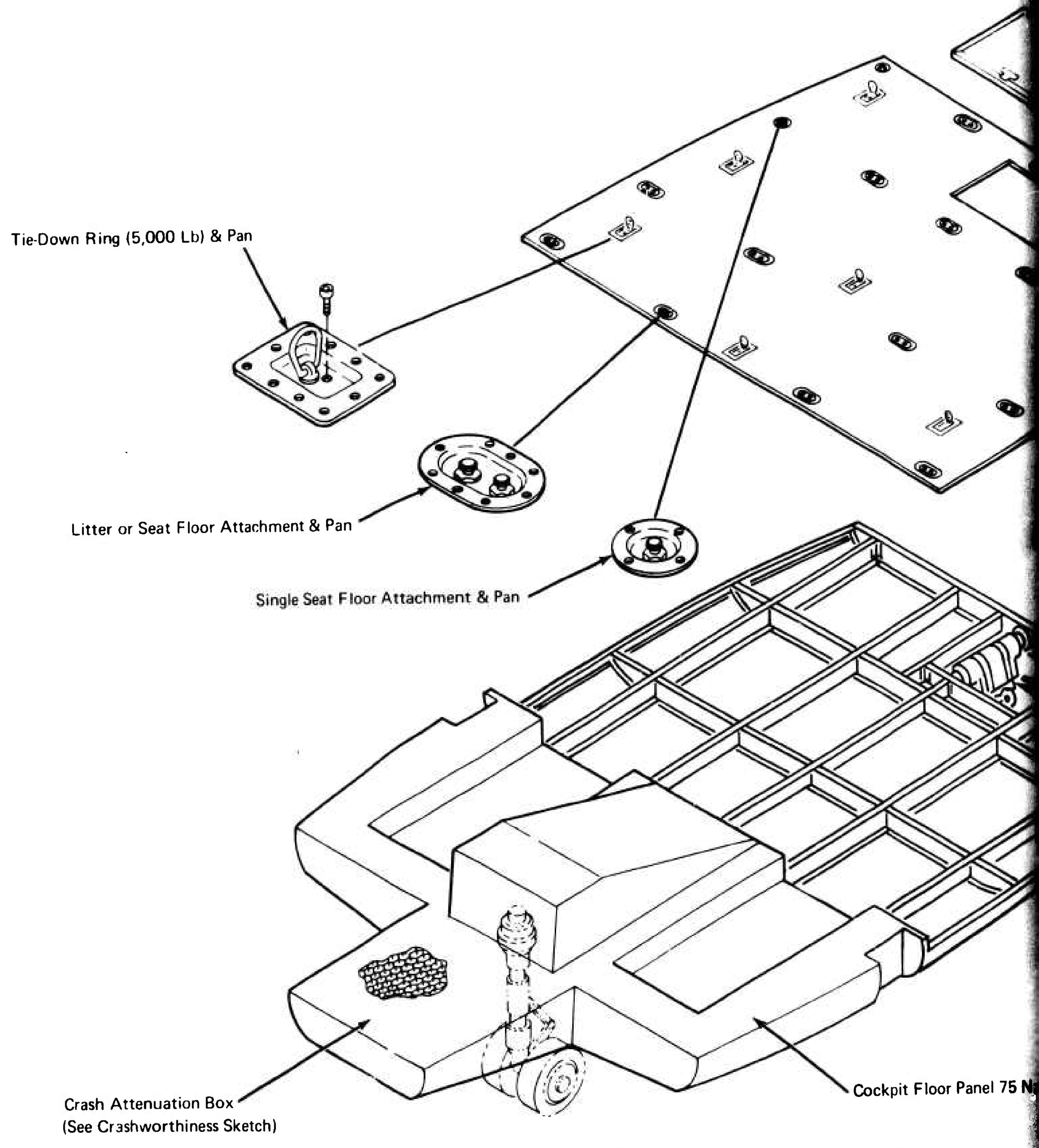
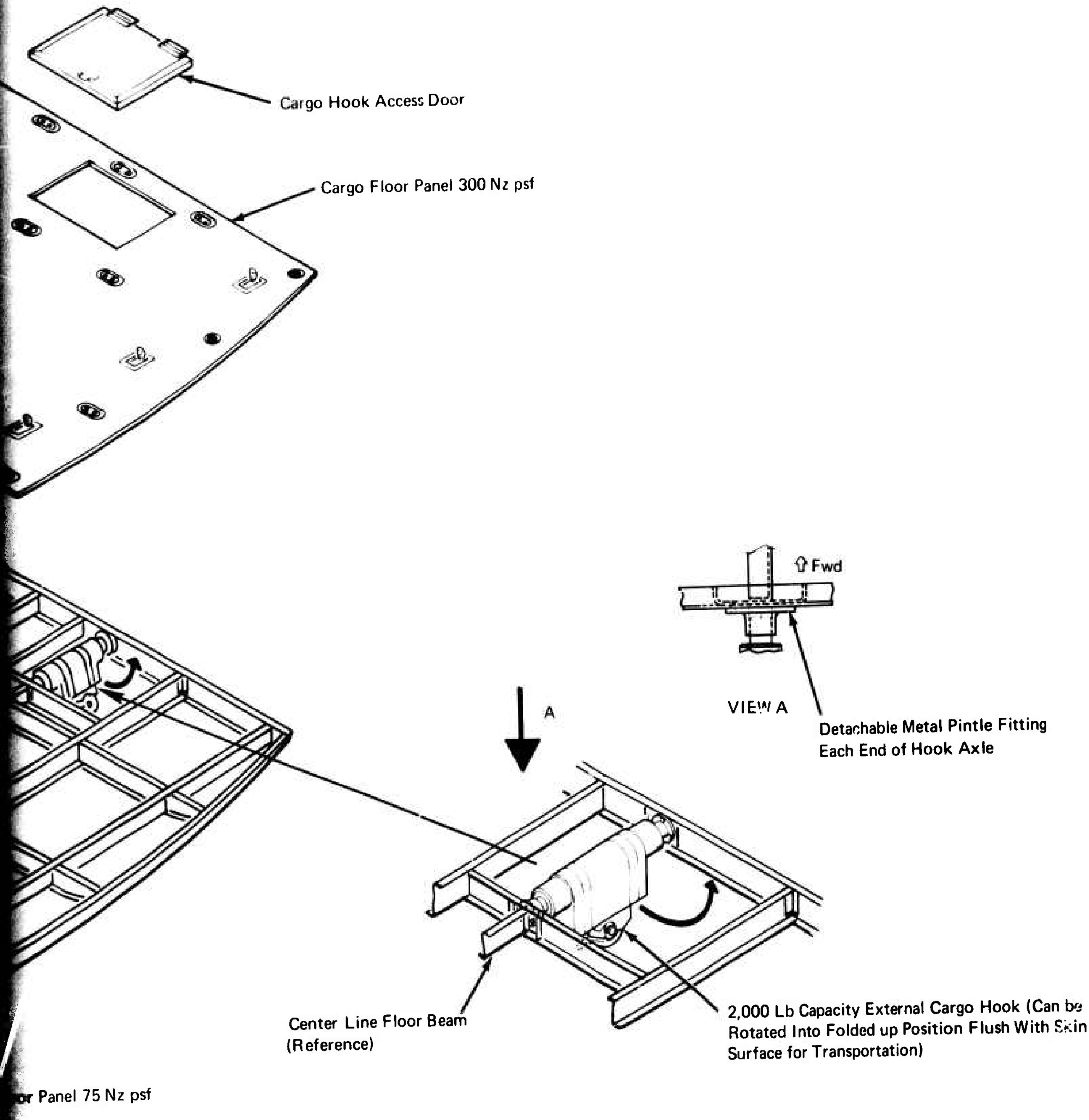


Figure 70. Airframe Module 2 - Floor Structure (Sheet 1 of 2)


Cargo Hook Access Door
Cargo Floor Panel 300 Nz psf
Fwd
VIEW A
A
Detachable Metal Pintle Fitting
Each End of Hook Axle
Center Line Floor Beam
(Reference)
2,000 Lb Capacity External Cargo Hook (Can be
Rotated Into Folded up Position Flush With Skin
Surface for Transportation)
or Panel 75 Nz psf

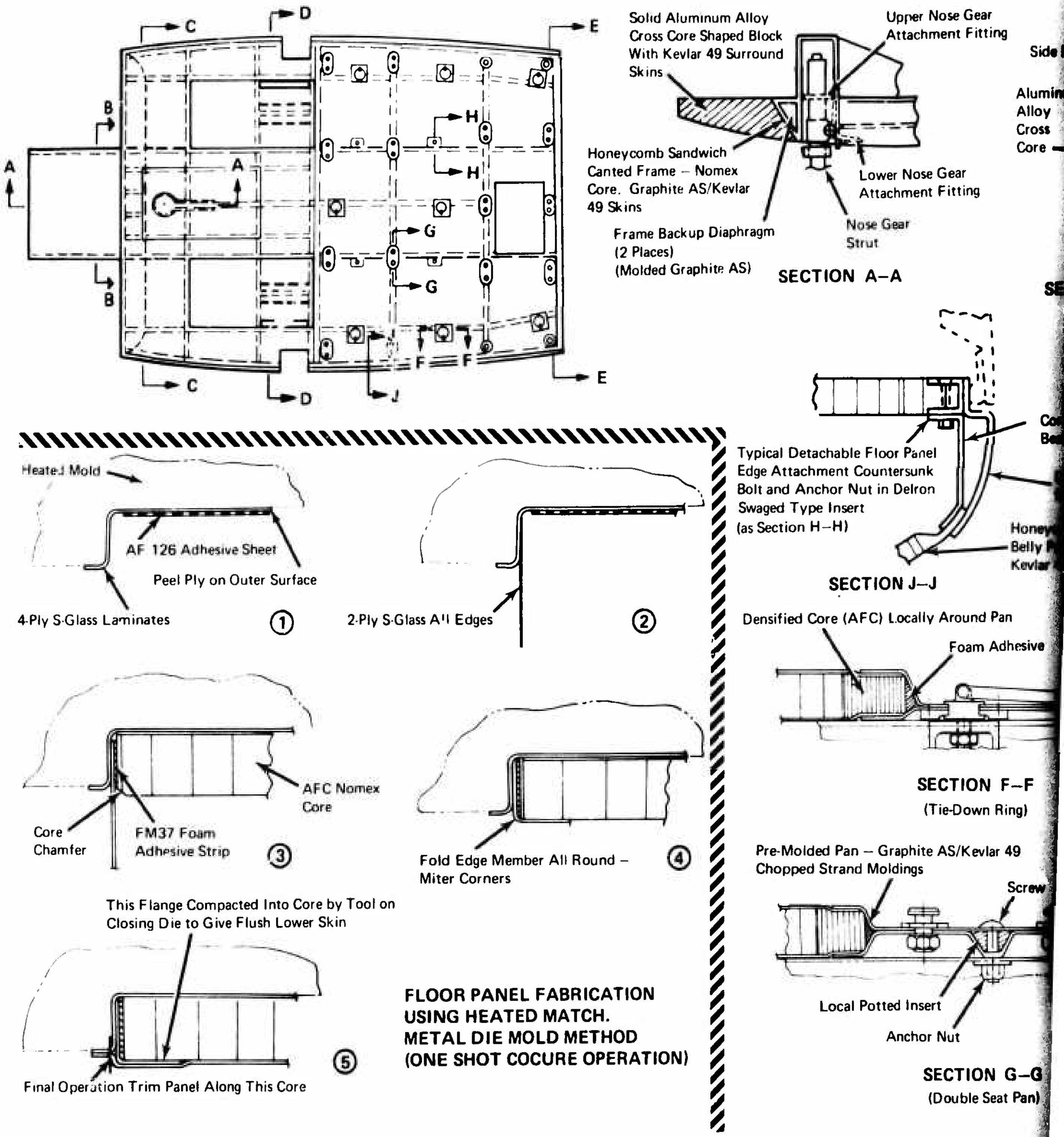
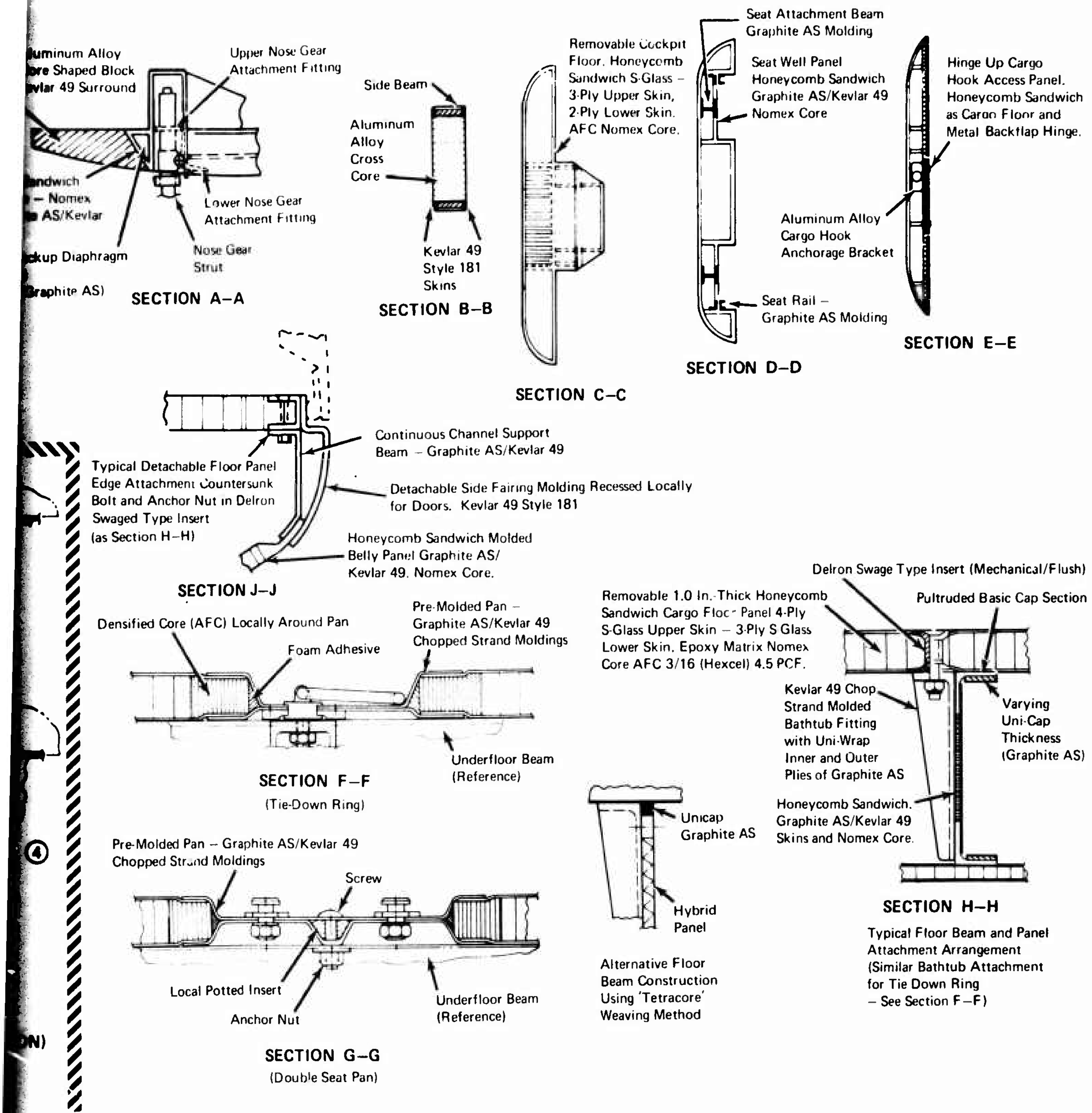


Figure 70. Airframe Module 2 - Floor Structure - Panel and Underfloor Structure Details. (Sheet 2 of 2)


uminum Alloy
ore Shaped Block
vlar 49 Surround
Upper Nose Gear Attachment Fitting
ndwich
– Nomex
AS/Kevlar
Lower Nose Gear Attachment Fitting
kup Diaphragm
Nose Gear Strut
raphite AS)
SECTION A–A
Side Beam
Aluminum Alloy Cross Core
Kevlar 49 Style 181 Skins
SECTION B–B
Removable Cockpit Floor. Honeycomb Sandwich S-Glass – 3-Ply Upper Skin, 2-Ply Lower Skin. AFC Nomex Core.
SECTION C–C
Seat Attachment Beam Graphite AS Molding
Seat Well Panel Honeycomb Sandwich Graphite AS/Kevlar 49 Nomex Core
Seat Rail – Graphite AS Molding
SECTION D–D
Hinge Up Cargo Hook Access Panel. Honeycomb Sandwich as Cargo Floor and Metal Backflap Hinge.
Aluminum Alloy Cargo Hook Anchorage Bracket
SECTION E–E
Continuous Channel Support Beam – Graphite AS/Kevlar 49
Typical Detachable Floor Panel Edge Attachment Countersunk Bolt and Anchor Nut in Delron Swaged Type Insert (as Section H–H)
Detachable Side Fairing Molding Recessed Locally for Doors. Kevlar 49 Style 181
Honeycomb Sandwich Molded Belly Panel Graphite AS/ Kevlar 49. Nomex Core.
SECTION J–J
Densified Core (AFC) Locally Around Pan
Foam Adhesive
Pre-Molded Pan – Graphite AS/Kevlar 49 Chopped Strand Moldings
Underfloor Beam (Reference)
SECTION F–F
(Tie-Down Ring)
Pre-Molded Pan – Graphite AS/Kevlar 49 Chopped Strand Moldings
Screw
Local Potted Insert
Anchor Nut
Underfloor Beam (Reference)
SECTION G–G
(Double Seat Pan)
Unicap Graphite AS
Hybrid Panel
Alternative Floor Beam Construction Using 'Tetracore' Weaving Method
Delron Swage Type Insert (Mechanical/Flush)
Removable 1.0 In.-Thick Honeycomb Sandwich Cargo Floor Panel 4-Ply S-Glass Upper Skin – 3-Ply S Glass Lower Skin. Epoxy Matrix Nomex Core AFC 3/16 (Hexcel) 4.5 PCF.
Pultruded Basic Cap Section
Kevlar 49 Chop Strand Molded Bathtub Fitting with Uni-Wrap Inner and Outer Plies of Graphite AS
Varying Uni-Cap Thickness (Graphite AS)
Honeycomb Sandwich. Graphite AS/Kevlar 49 Skins and Nomex Core.
SECTION H–H
Typical Floor Beam and Panel Attachment Arrangement (Similar Bathtub Attachment for Tie Down Ring – See Section F–F)
④
N)

nel and
of 2)

Crew floor panels are located from sta 44 to sta 91, with a well recess on each side for seat displacement. These floor panels are removable and are designed for 75 Nz per square foot. The cargo floor detachable panels extending from sta 91 to sta 163 are designed for heavier loading up to 300 Nz per square foot and meet Army specifications for a 200-pound box dropped on one corner of radius 0.50 inch from a height of 15 inches (see Figure 65). Three types of floor attachments are integrated into the panels: 5000-pound tie-down rings for cargo, double seat pan for seat leg locating or litter leg anchorage, and single seat pan for single seat leg attachment.

A one-piece molded honeycomb sandwich panel forms the outer belly skin and when bonded to the beams and frames forms a series of torque boxes to stiffen the underfloor structure. Extra thick skin laminates at the forward section outside skin surface are added to preclude scooping/tearing effects when a nose plowing crash condition occurs.

The underfloor structure module is designed to fulfill three principal functions: to augment fuselage bending stiffness (approximately 20 percent by differential bending), to support cargo and troop seat loads, and to perform as an energy absorption system for longitudinal impact condition.

The underfloor beam system has a multiple function, supporting the floor panels, reacting cargo ring point loads, and redistributing nose gear loads forward to the windshield posts and aft to the support posts at sta 91, as well as forming a rugged backup structure in the event of a longitudinal impact crash and also reacting longitudinal seat loads. Should a longitudinal nose-down crash occur, the upper support for the gear is designed to fail and let the gear leg and wheels rotate rearwards to lay under the fuselage to prevent the possibility of the leg bursting into the cockpit and injuring the crew. (The nose gear itself will absorb considerable energy before designed failure.)

4.3.6.3 Upper Deck Assembly and Fairings - Module 3 (Figure 71)

Primary Structure - The upper deck module, extending from sta 91 to 236, is comprised mainly of a flat sandwich panel deck at waterline 78, the full width of the fuselage, and two buttline beams (BL 15), also extending the full length of the module. In the cabin section, the BL 15 beams are augmented with additional buttline beams (BL 22) and outboard external skin panels, which with the WL deck form torque boxes between stations 91 and 163. Primary ribs are positioned to back up local load introductions at the main transmission and engine locations. Intermediate ribs are positioned in the forward and aft bays to stabilize the panels.

The WL 78 deck is fabricated as a sandwich panel using Nomex core. The section forward of sta 163 provides the ceiling for the cabin section and the lateral shear material for the loads in the upper fuselage section. It has a structural upper face sheet of a unidirectional graphite AS and Kevlar 49 angle ply hybrid laminate, and a lower face sheet which is a perforated multi-ply Kevlar 49 laminate, which, together with the rest of the sandwich, acts as an acoustic barrier. This lower face is also structural to the extent that it and the core stabilize the structural upper face sheet. The center panel between the BL 15 beams contains structural hinged panels for access to the underside of the transmission and other equipment in this area. The aft section (sta 163 to 239) provides the upper panel for the fuel cell, the upper closure for the avionics compartment, and the upper closure for the structural shell in the fuel bay.

The main buttline beams are fabricated of angle cap members with one leg buried in and bonded to a thin sandwich shear web. The caps are molded graphite AS laminates, primarily of unidirectional plies with sufficient angle plies to provide shear, crippling, and fastener bearing strengths where necessary. The face sheets for the sandwich webs are a hybrid of a pair of $\pm 45°$ graphite AS plies for strength and a layer of woven Kevlar 49 cloth to provide a minimum gauge and to add damage resistance especially for the inner face, which is exposed to an area where maintenance tasks are periodically performed. The auxiliary beams outboard of the main beams are fabricated in a similar manner. The center section between the buttline beams is covered with a removable panel, both in the forward and aft sections of this module.

The basic structure supports the transmission through four metal fittings, which are attached mechanically at the corners of a rectangular box formed by the buttline beams, an auxiliary forward rib, and a rib atop the sta 163 bulkhead. Each engine is supported by metal truss-like fittings and link tubes in two locations (sta 173 and sta 187). Separate upper and lower fittings are used in each position to afford a degree of fail safety. Fittings and link tubes are detachable, with the lower fitting attaching to the deck and beam while the upper fitting attaches to the lateral support beam and also to the lower fitting.

The complete module is attached to its adjoining modules with mechanical fasteners. The attachment to the fuel cell side walls is made through the beam cap members and the WL deck. The attachment to the sta 163 bulkhead is made into barrel nuts contained in the bulkhead core section. The buttline

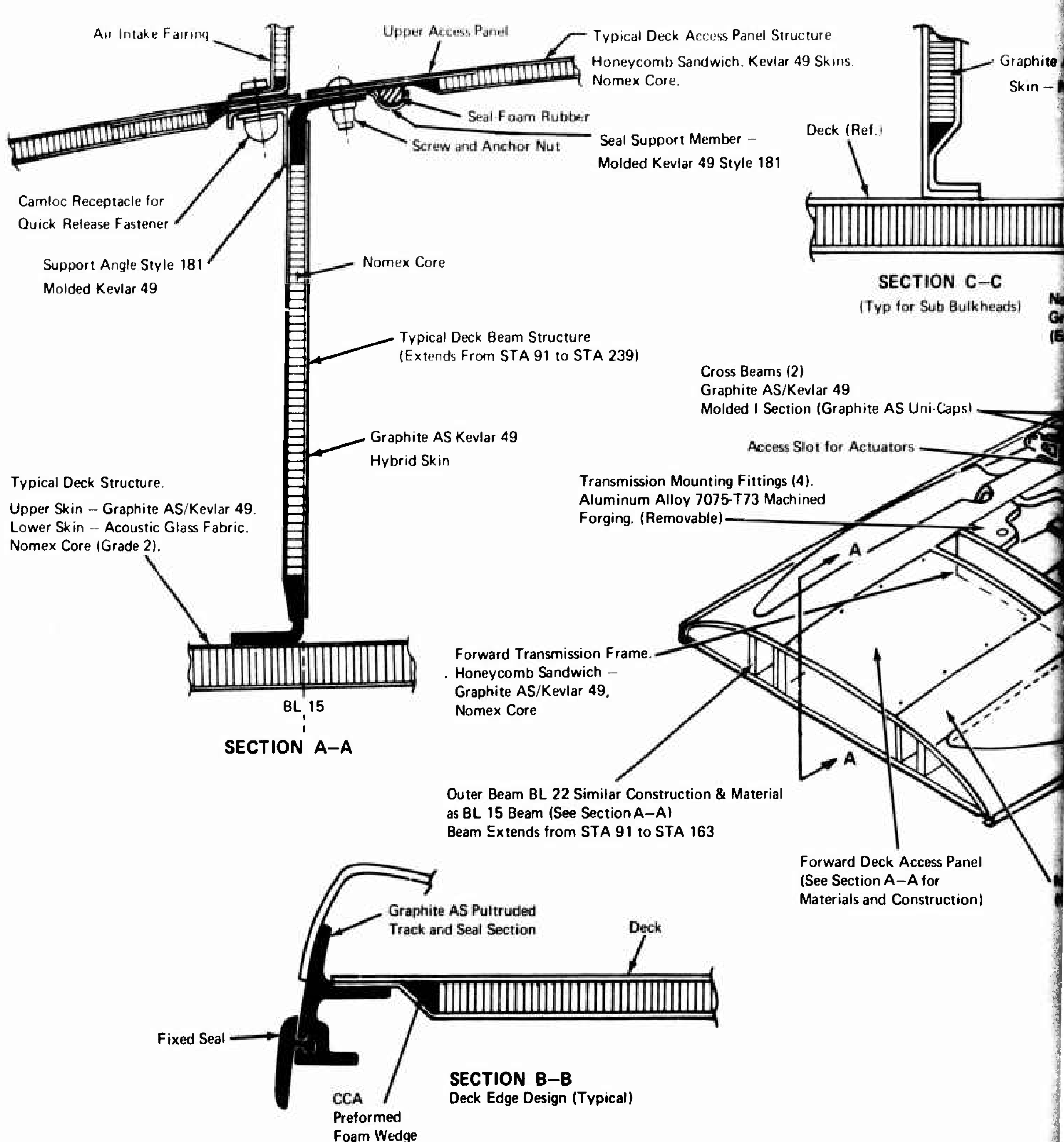
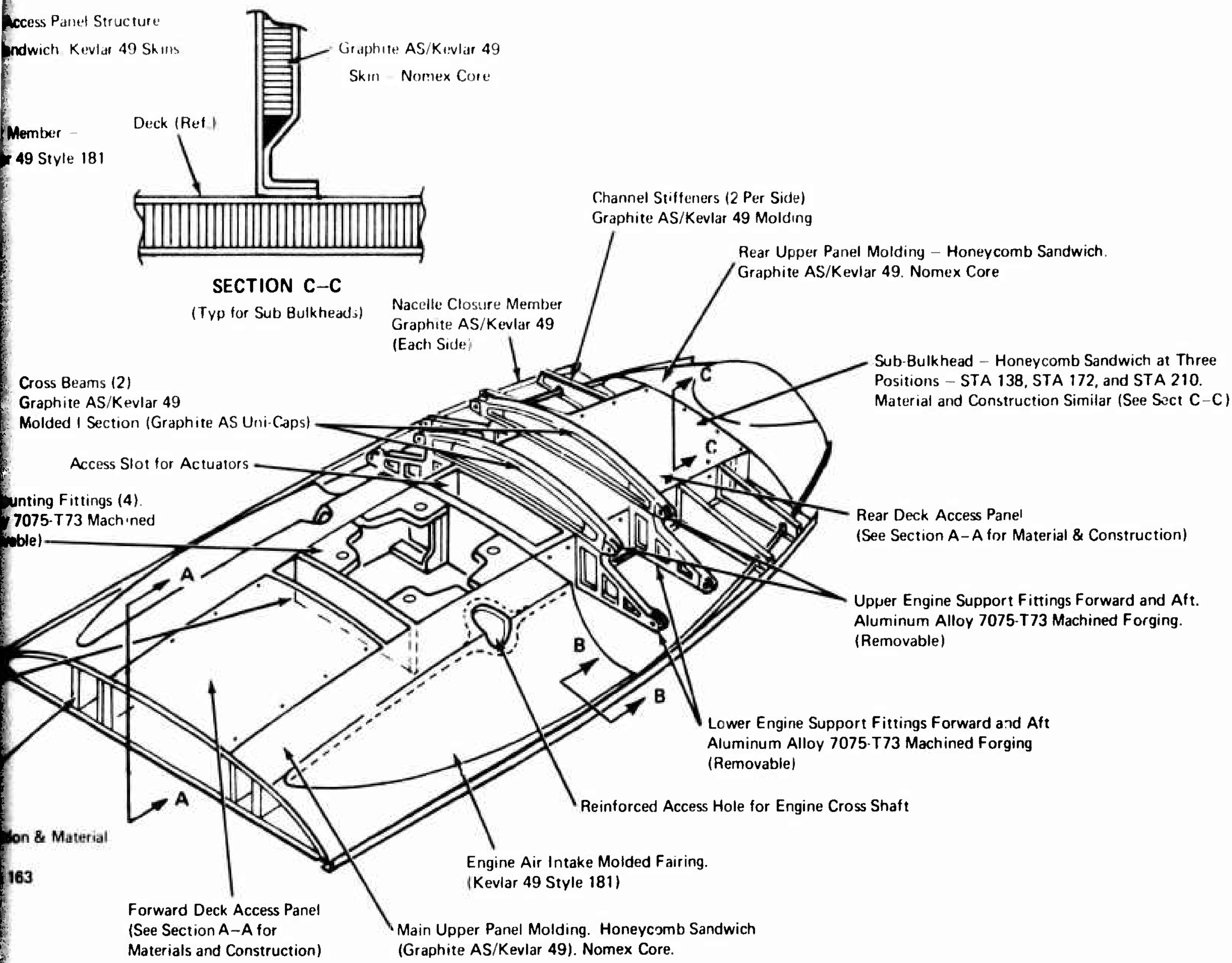


Figure 71. Airframe Module 3 - Upper Deck Assembly and Fairings - Deck Structure. (Sheet 1 of 2)

ccess Panel Structure
ndwich Kevlar 49 Skins

Member –
r 49 Style 181



on & Material

163

and Fairings -

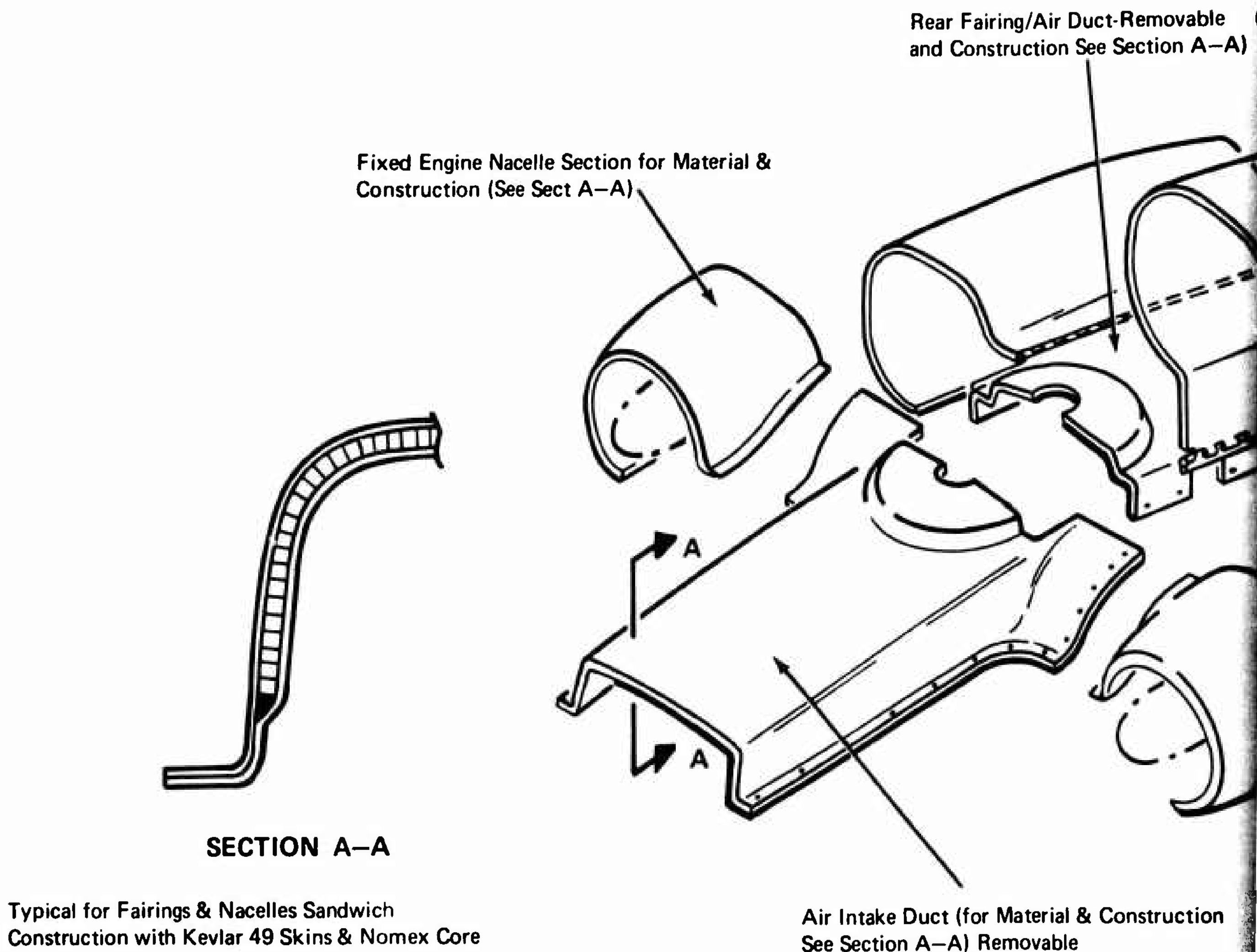
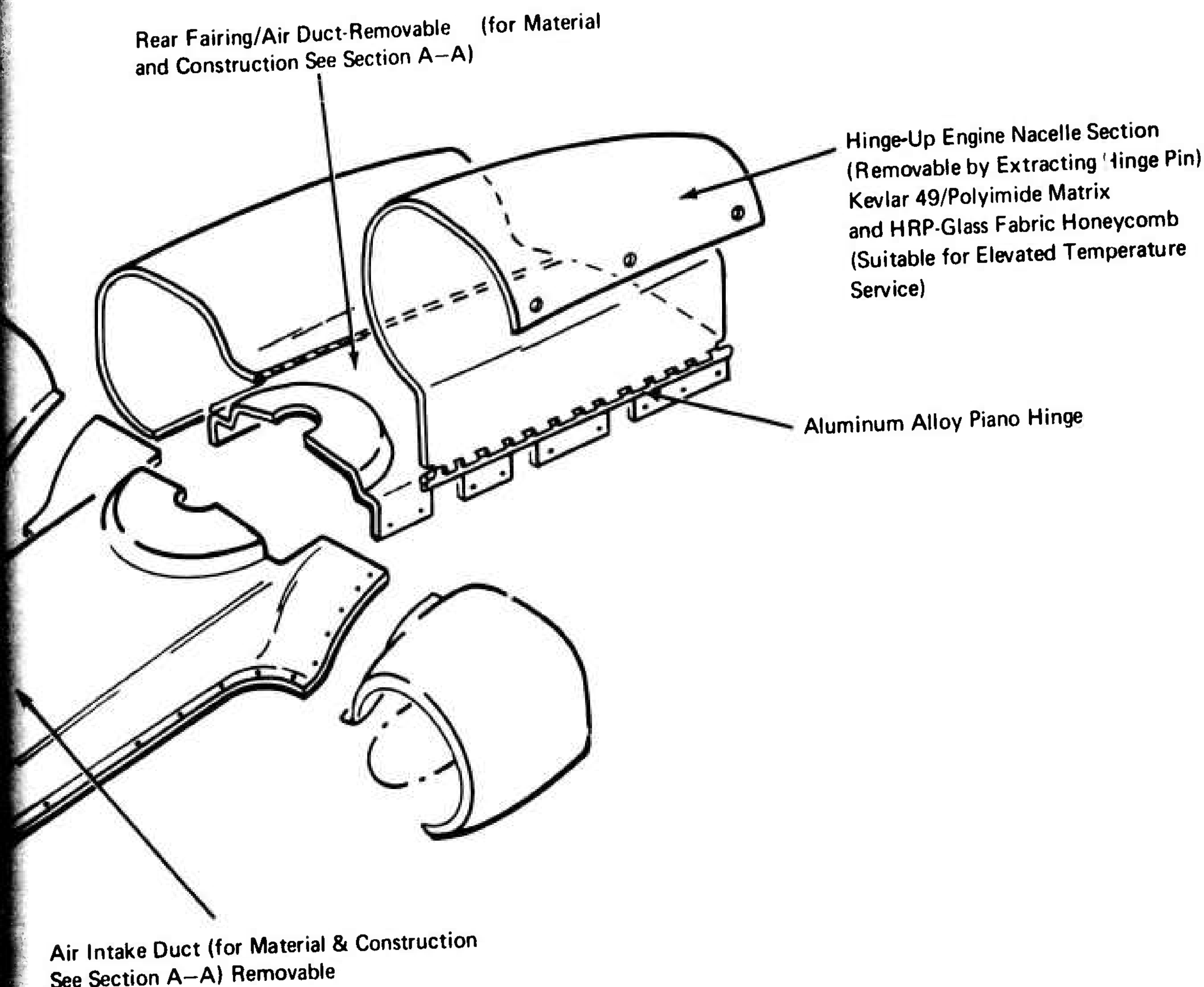


Figure 71. Airframe Module 3 - Upper Deck Assembly and Fairings - Fairing Details. (Sheet 2 of 2)


Rear Fairing/Air Duct-Removable (for Material and Construction See Section A–A)
Hinge-Up Engine Nacelle Section (Removable by Extracting Hinge Pin) Kevlar 49/Polyimide Matrix and HRP-Glass Fabric Honeycomb (Suitable for Elevated Temperature Service)
Aluminum Alloy Piano Hinge
Air Intake Duct (for Material & Construction See Section A–A) Removable

nd Fairings -

beams are joined to the cockpit box frame and post structure in a joint designed to carry shear, axial, and bending loads. A shear tie is effected at the aft closure frame at sta 239. The secondary structures (cabin side panels, cabin door slides, engine cowling support hinges, avionics bay panels, etc.) are mechanically fastened at their interfaces.

Engine Bay Fire Protection - The buttline beam and WL 78 deck panel in the engine bay, sta 163 to sta 217, are protected by a one-piece stainless steel firewall 0.015 in. thick totally covering all composite primary structure web and caps. The composite layups in this area are designed using a high-temperature polyimide matrix for protection against elevated temperatures conducted into webs, etc., via the firewall in the event of an engine fire. The aluminum alloy engine support fittings are covered on assembly by a special ablative paint system similar to that used on YUH-61A fittings.

The upper deck module is the backbone of the aircraft, distributing the rotor and engine loads forward and rearwards and also into the main bulkhead at sta 163. The predominant critical loads result from crash conditions where the large mass items (rotor, transmission, and engines) must be restrained from breaking away from their mountings and entering the liveable areas to endanger the occupants.

Portions of this structure are also critical for fuel pressure loads, landing loads, and in-flight maneuver loads, both symmetrical and asymmetrical. Because of the large side cutouts for the cabin doors, the forward structure section carries the major portion of the fuselage shear, bending, and torsional load reactions supporting the cockpit.

The attachment to the fuel cell side walls and the bulkhead at sta 163 is the primary joint to the entire aft fuselage section.

Secondary Structure - Located essentially above the primary deck structure are a series of fairings and engine nacelle segments, all of which are of thin honeycomb sandwich construction with Nomex core. This method affords maximum fairing stiffness and minimum internal protuberances commensurate with low parts count assemblies. The air intake duct extends back from sta 91 to the transmission shaft at sta 149 and is easily detachable by quick-release fasteners. Directly behind this is the rear fairing/air duct, which is similarly removable.

Each engine nacelle comprises a fixed and a hinge-up segment. The forward smaller segment, which is the air intake portion of the nacelle, is located adjacent to the forward intake duct and is attached by screws and anchor nuts. The rear fairing which envelopes the engine is hinge-attached to the rear fairing and rotates up, where a folding spring strut is used to hold it open for maintenance purposes. This fairing is removable by withdrawing the one-piece pin of the backflap fairing hinge. With the hinge-up nacelle arrangement, it is necessary to position the rotor blades for the fairing to swing up. An alternate but heavier and more complex design, similar to the YUH-61A, is for the nacelle/fairing to fold down to form an engine work platform; however, the smaller MUT engines and consequently smaller matching nacelle result in a relatively short, narrow platform which hardly seems to justify the extra cost and weight involved. The rear fairing, extending from sta 163 to sta 218, forms an air duct, and has an exit slot at its rear end to evacuate the heated air from the main transmission unit and from the engine bay.

Located directly under the forward air intake duct and also the rear fairing are thin honeycomb sandwich access panels with edge seals all round for access into the forward and aft equipment bays, the sides of which are formed by the deck structure longitudinal beams and lateral frames. The doors are attached by the screw and anchor nut method and contribute to overall deck structure stiffness. The removal of either or both of the fairing ducts allows ready access to the upper controls swashplate and main actuators, the latter by slots in the main deck at each side of the transmission bay.

4.3.6.4 Bulkhead (Sta 163) and Side Panel Assembly Including Main Landing Gear - Module 4 (See Figure 72)

This module is an all-primary-structure assembly comprising the main landing gear/fuel bay bulkhead and an integrated skin panel assembly on each side forward of the bulkhead. The bulkhead is of honeycomb sandwich construction with hybrid composite skins and Nomex core; it acts as a redistribution member transferring loads from the upper deck structure and underfloor structure into the fuel bay support structure aft of it. The main landing gear fitting vertical and lateral loads are reacted by the bulkhead at the shock strut and lower trunnion attachment points, which are then redistributed to the adjacent modules.

A molded channel section frame cap is bonded between face skins around the side and bottom periphery of the bulkhead. Along the top surface, which locates against the deck at WL 178, a deeper near-channel section molded composite loop fitting, also bonded between the skins, extends the full bulkhead width and entraps barrel nuts at regular intervals. A structural joint of the deck assembly to the bulkhead is made by bolting down from the upper deck sta 163 frame, through the continuous deck, picking up the barrel nuts to make a tension/shear joint. Access slots are made in the upper bulkhead web to torque the bolts.

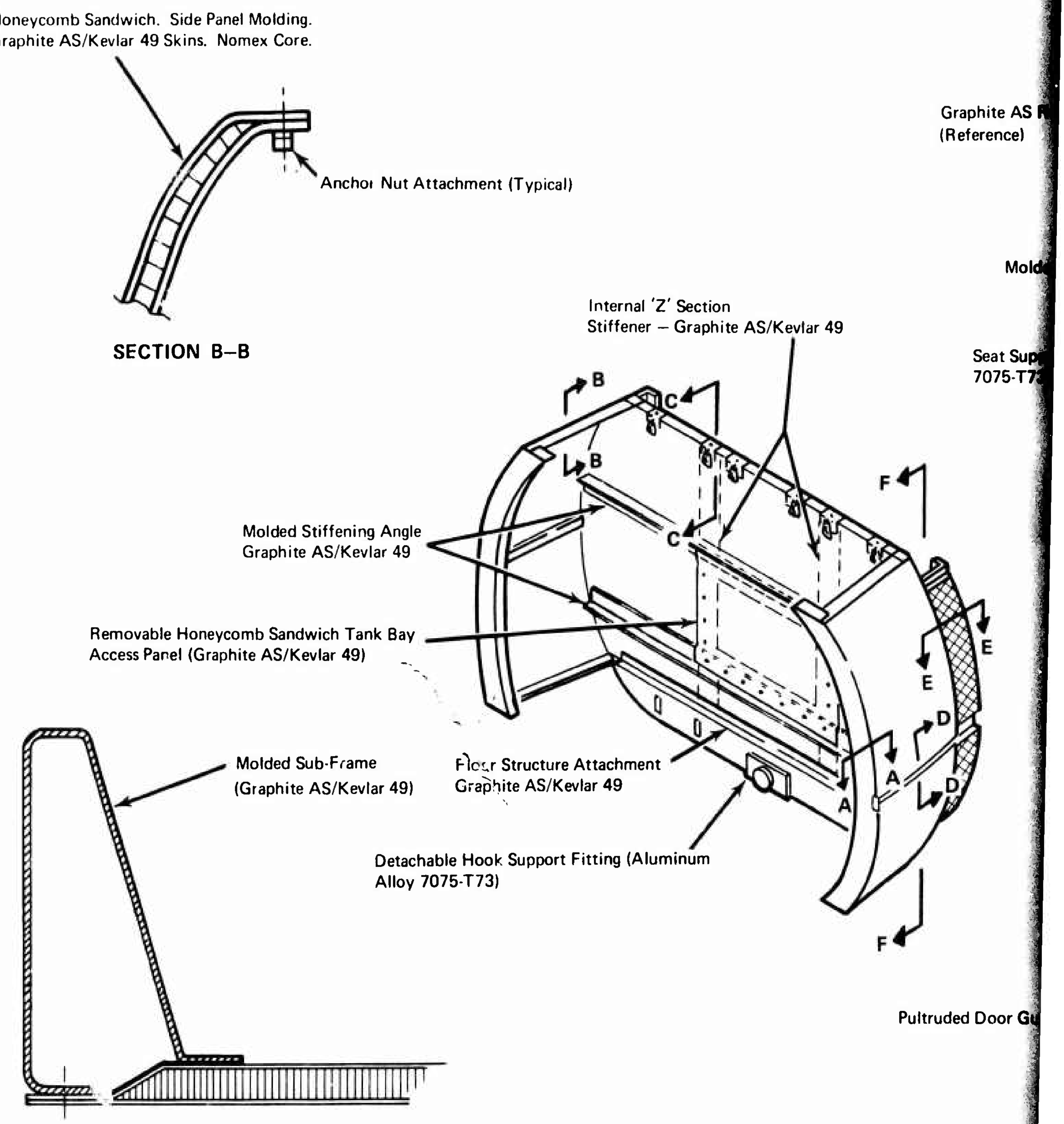
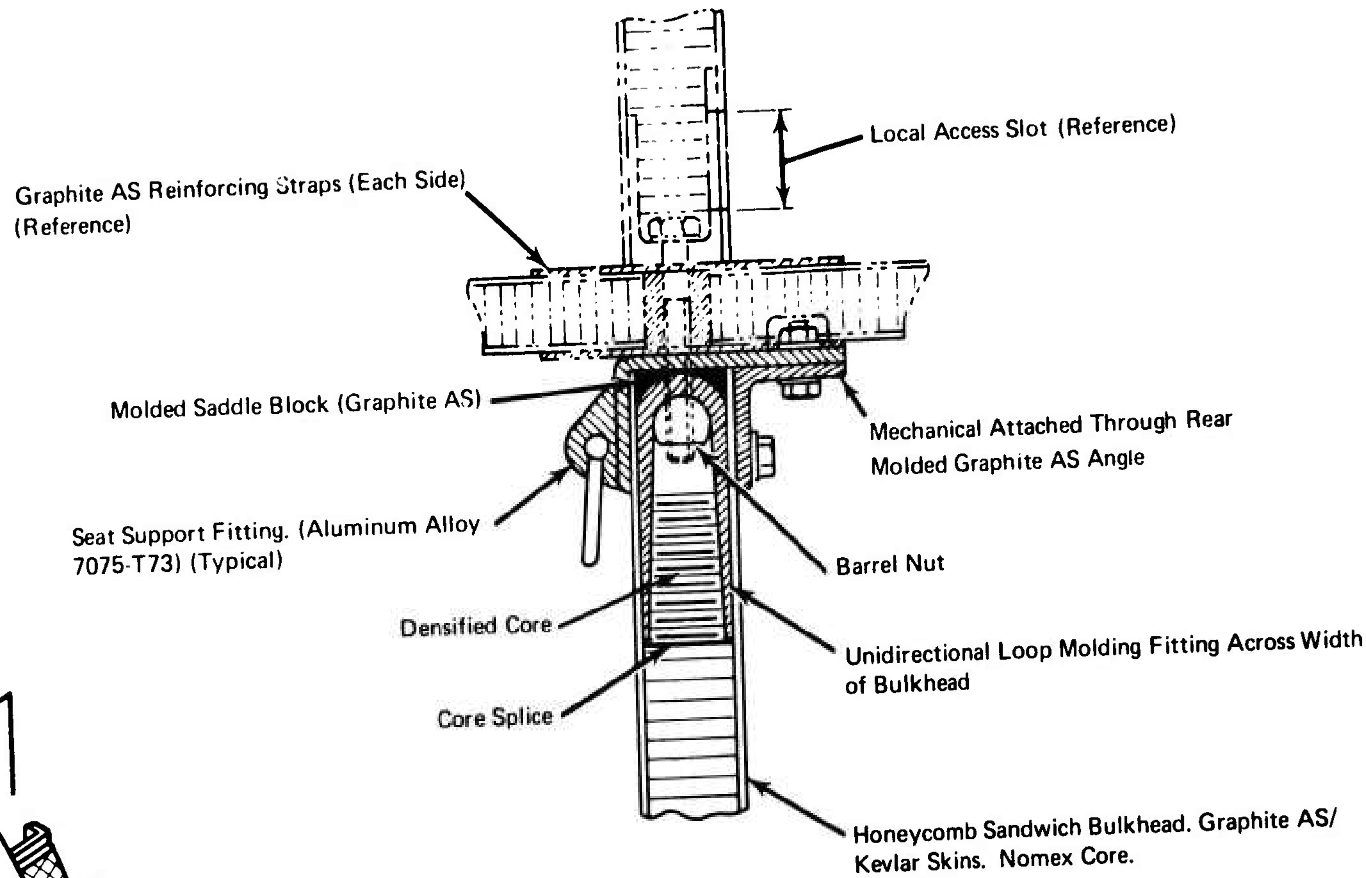
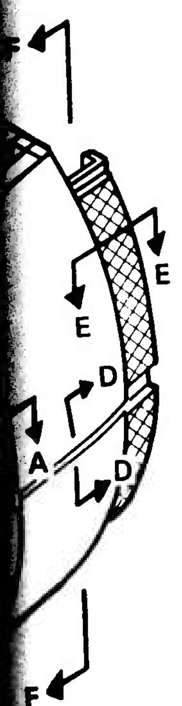
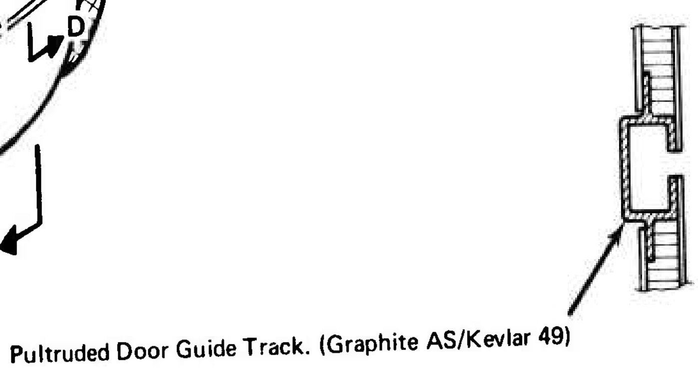


Figure 72. Airframe Module 4 - Bulkhead Sta 163 and Side Panel Assembly - Side Panel Details. (Sheet 1 of 2)

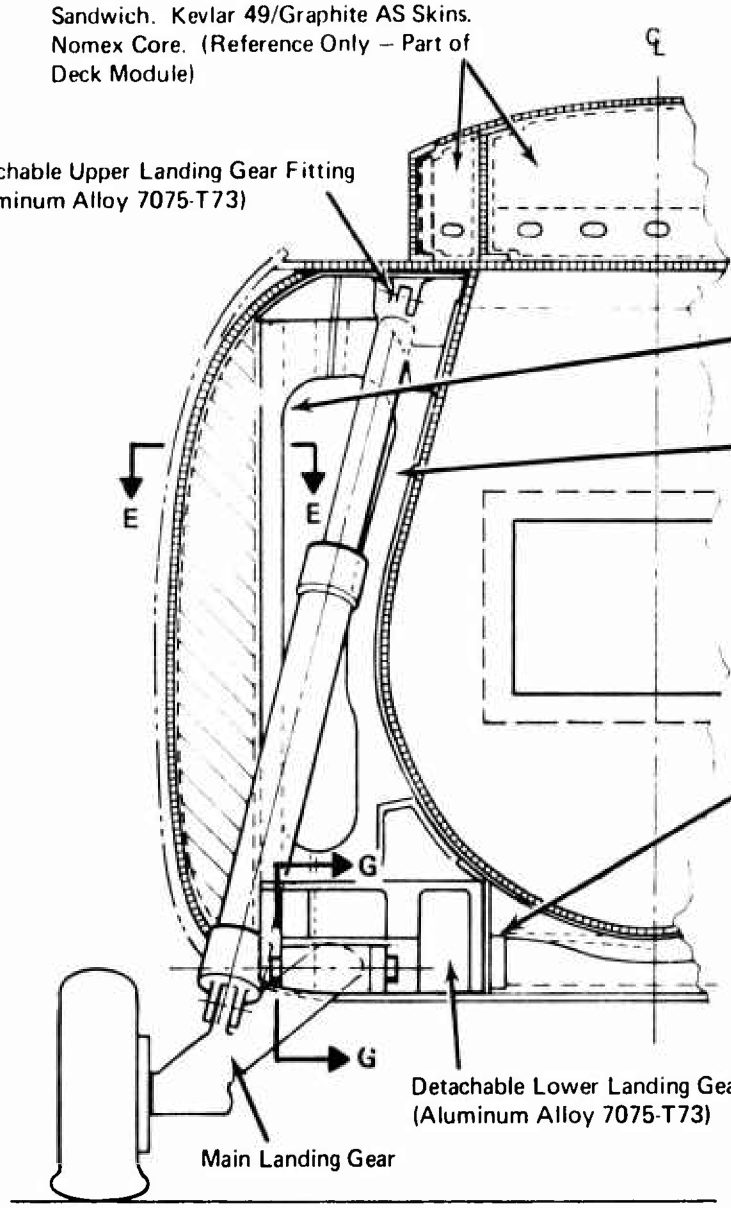


SECTION C–C

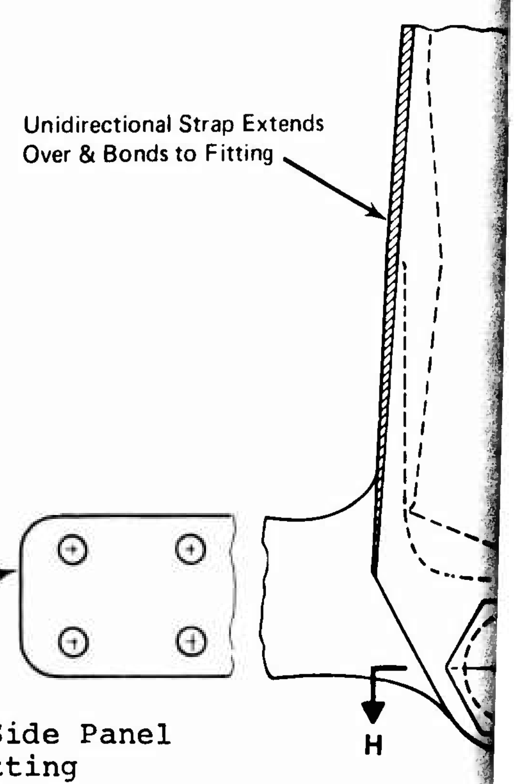




SECTION D–D

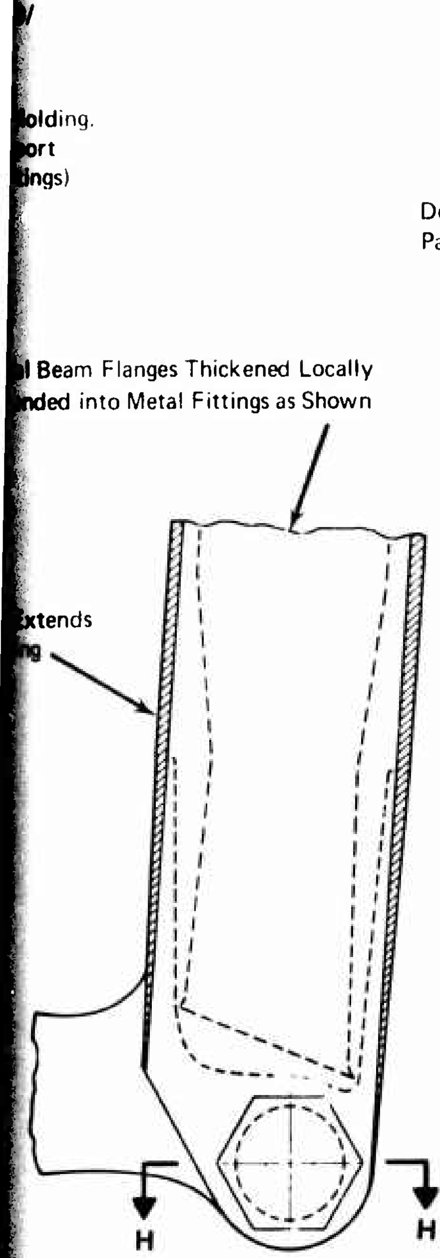


SECTION F–F

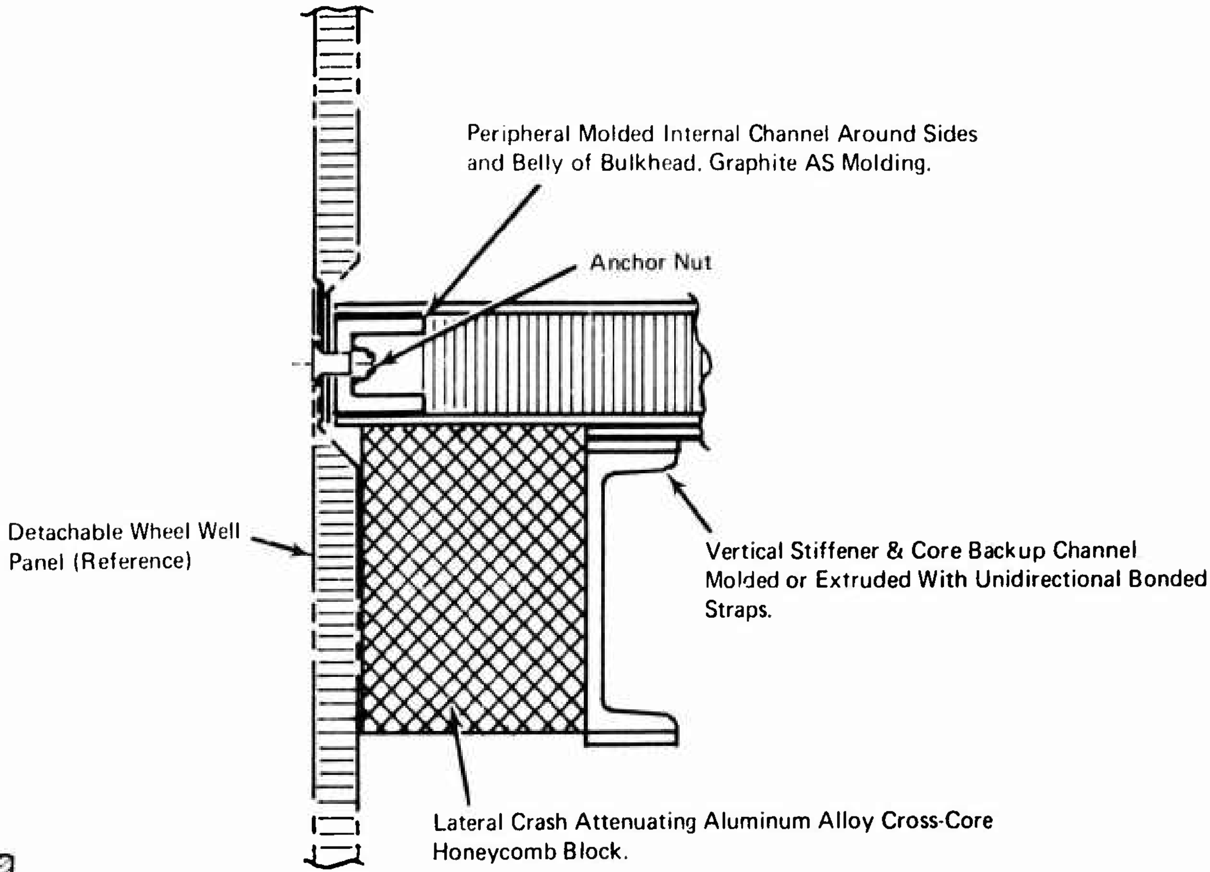


SECTION G–G

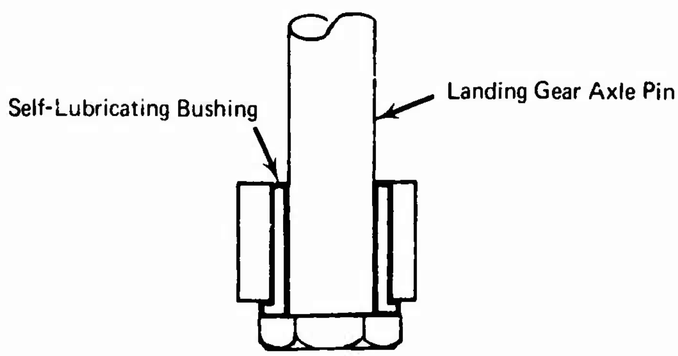
Figure 72. Airframe Module 4 - Bulkhead Sta 163 and Side Panel Assembly - Landing Gear Attachment and Fitting Details. (Sheet 2 of 2)



SECTION G–G



SECTION E–E



SECTION H–H

Small access slots for replacing the barrel nuts are designed in the skin core and loop fitting locally adjacent to the attachment bolt positions. Also located along the top surface of the bulkhead are eight detachable metal troop seat support fittings mechanically fastened. A large oblong cutout in the center of the bulkhead with a mechanically attached honeycomb sandwich panel affords access into the fuel tank and allows bladder replacement. Vertical Z section pultruded members are sandwiched between bulkhead skins at BL 15 on each side, and lateral angles are bonded to the skin forward face at WL 154 and WL 131. These vertical and horizontal overlapping members adequately frame around the bulkhead cutout.

Another pultruded angle is bonded onto the forward bulkhead face at WL 120 for the mechanical attachment of the floor structure module. On the aircraft centerline at WL 122 is located the aft end of the cargo hook support fitting, which mechanically attaches to the bulkhead web using potted inserts with local reinforcing laminates on the web skin at both sides.

Attached to the rear face of the bulkhead at BL 35 is a deep molded channel section beam which spans between upper and lower metal landing gear attachment fittings; and as well as performing as a vertical load carrying and stiffening element, it forms a rugged backup structure for the lateral crash attenuating core slab, which is located on and bonds to the outer face of the beam. A reinforcing plate bonded to the rear bulkhead skin matches the contour of the adjoining fuel bay support structure shell and is also grown out locally top and bottom to extend under the upper and lower landing gear fittings.

The upper landing gear attachment fitting has a short upper flange extending aft where a double lug is grown out to pick up the upper end of the shock strut. In line with the fitting, above, and extending forward, is the BL 22 deck beam, to which is attached a fingerplate extending forward from the fitting to react vertical and longitudinal loads.

The lower fitting has two lugs extending aft that contain bushings to support the main landing gear attachment, which is in the form of a trunnion assembly. A steel axle pin passes through both lugs and the trailing arm of the gear and is detachable by withdrawing outboard. The pin resembles a large bolt, and a locknut holds it in position. The outer lug of the fitting is extended upwards and machined to fit onto the end of the vertical channel beam. Another lug of this fitting extends forward, through a slot in the bulkhead, to locate and bolt onto the web of the BL 35.0 underfloor beam to react the lower drag loads.

The upper and lower fittings are designed with generous web areas in order to spread the local loads emanating from the bolt attachment into the reinforcings and bulkhead honeycomb sandwich where potted inserts are used at each connection point.

The side panel assemblies are located between sta 141 and sta 163 and are primarily of molded honeycomb sandwich construction with a bonded door edge stiffening frame vertically placed on the forward edge. Light mechanical fasteners attach through upper and lower panel flanges attaching to the upper deck module and floor structure respectively. The skin panels are split on each side, and a door track is bonded in flush to accommodate a door slider fitting.

The main landing gear is described in Section 4.3.7.

Loads - This bulkhead is the main structure joining the deck module, floor module, and fuel bay module; also, it is the primary structure for receiving the vertical and lateral load components from the main landing gear. Critical loadings for the various structural elements (webs, caps, joints, etc.) arise from each major loading condition; flight maneuvers, landing, and crash loadings involving mass and fuel retention, seat restraint, and the preservation of livable cabin space during rollover.

4.3.6.5 Tank Support and Side Avionics Structure - Module 5 (See Figure 73)

Tank Support Structure (Primary) - The tank support structure extends from sta 163 to sta 239 and is essentially a honeycomb sandwich truncated egg section shell with three longitudinal sandwich beams bonded to the underside of the section. A lower panel of sandwich construction covers the beams and forms the belly skin. At the forward end the shell skins are reinforced and brought together to form an angle type flange, to which anchor nuts are attached to make the mechanical joint to the adjoining sta 163 bulkhead (module 4). The rear end vertical joint is made with a separate molded joint angle which is buried within the shell thickness all round. The skins, inner and outer, in the joint vicinity have additional reinforcing laminates. The upper edges of the shell present a flat panel for attachment to the deck module at WL 178. This joint transfers the major portion of the loads from the tailboom and empennage into the deck structure; it consists of a molded cap section comprising uni and cross-ply layup that is sandwiched between the reinforced inner and outer shell skins, displacing the Nomex core and flanged over to make the flat mechanical joint necessary to attach this module to the deck module. Anchor nuts are affixed to the underside of the flanges for simple bolt attachment.

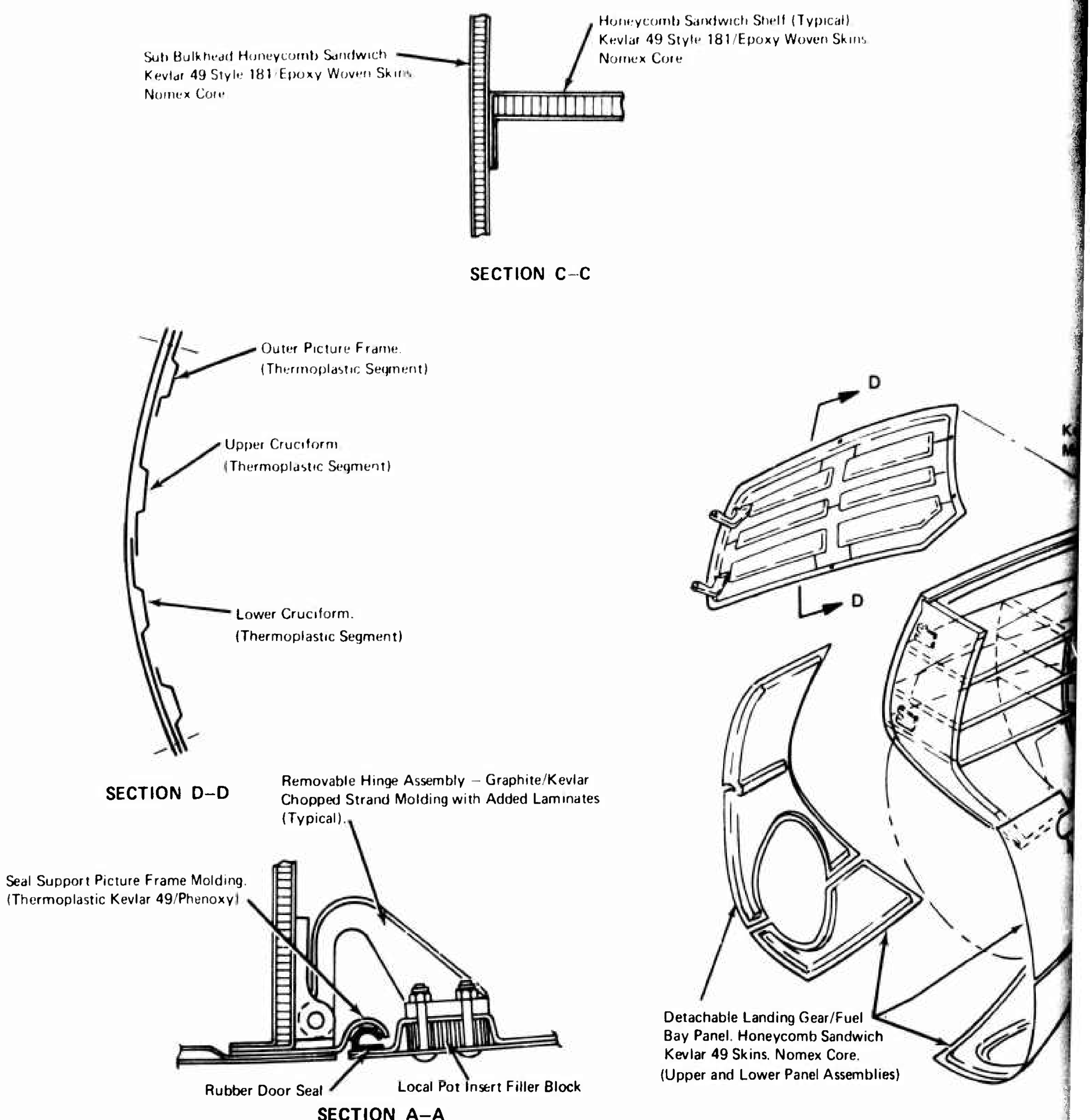
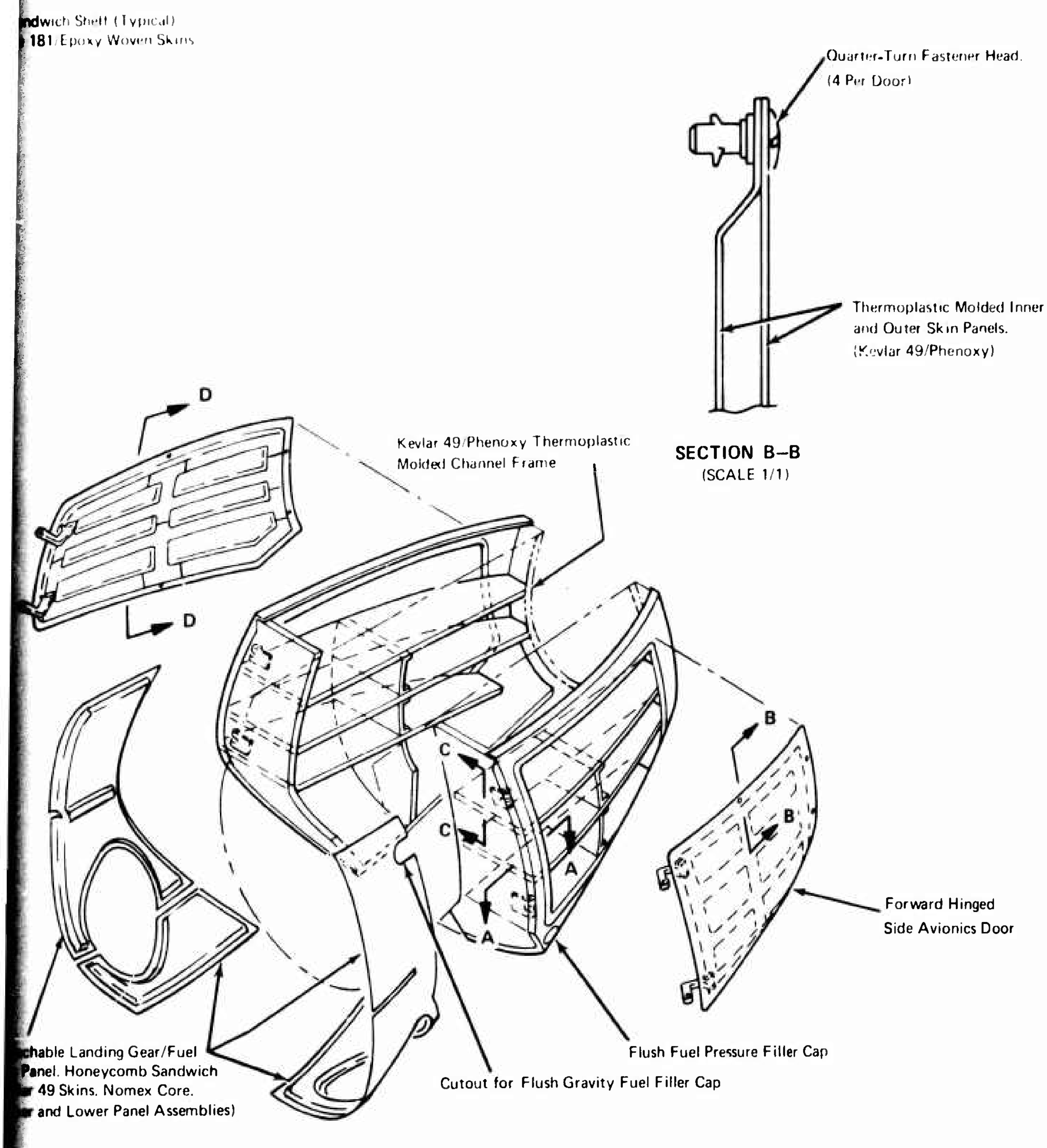


Figure 73. Airframe Module 5 - Tank Support and Side Avionics Structure - Subassembly Side Avionics Compartment (Sheet 1 of 2)



de Avionics
ompartment

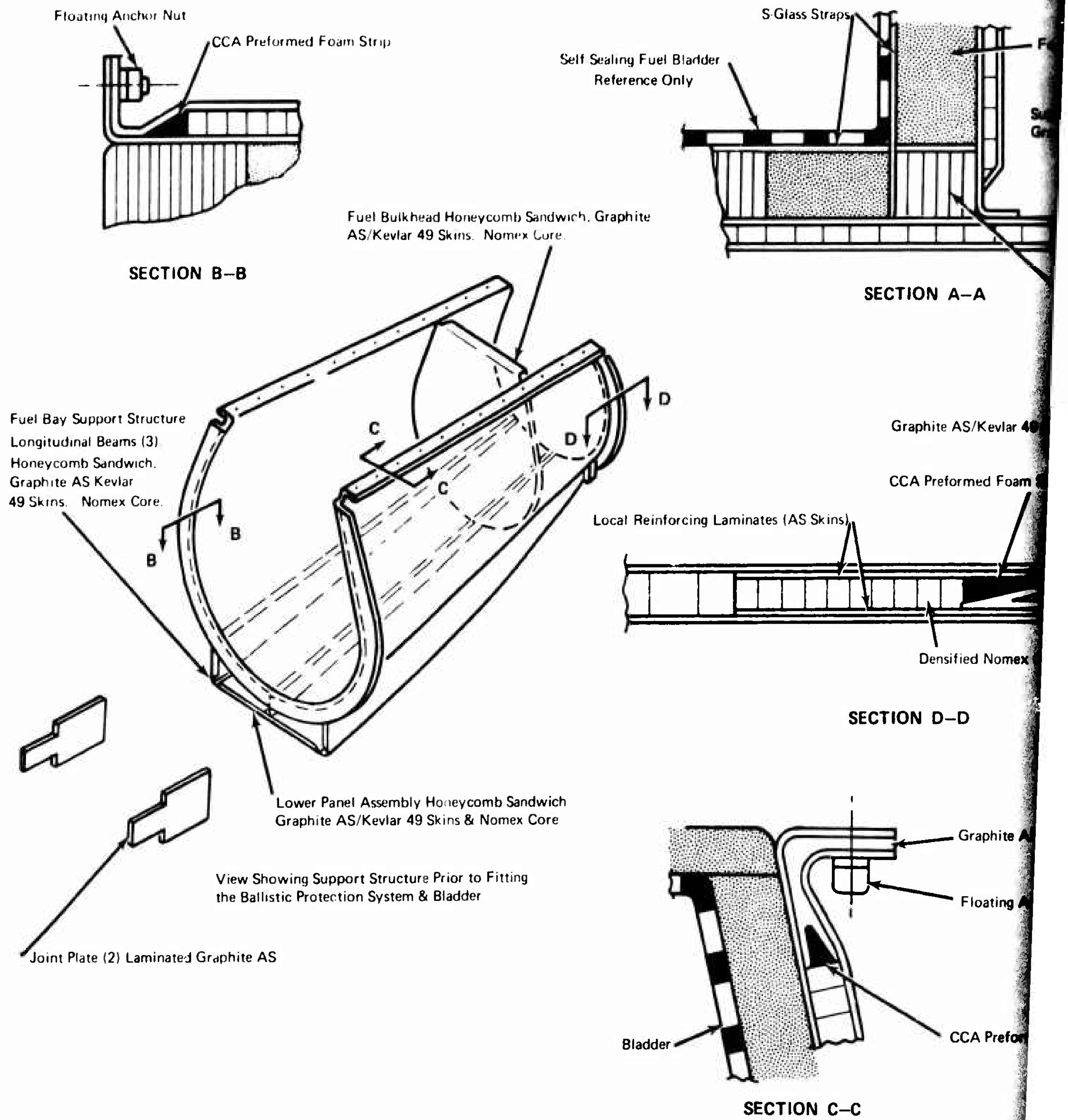
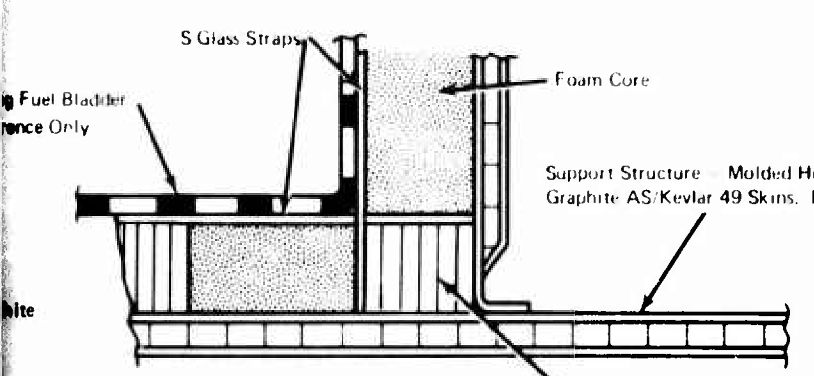
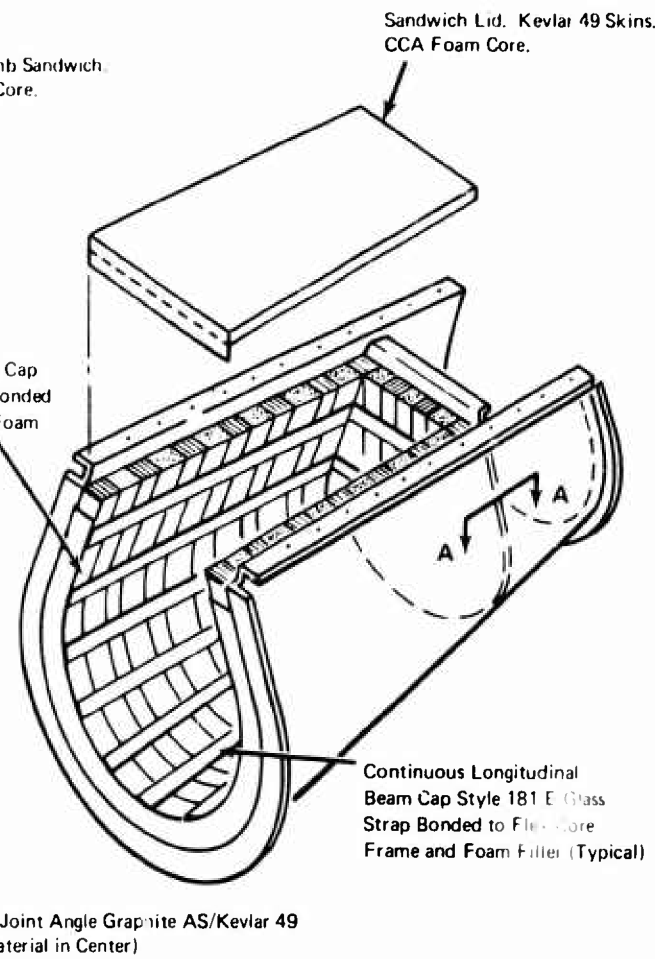
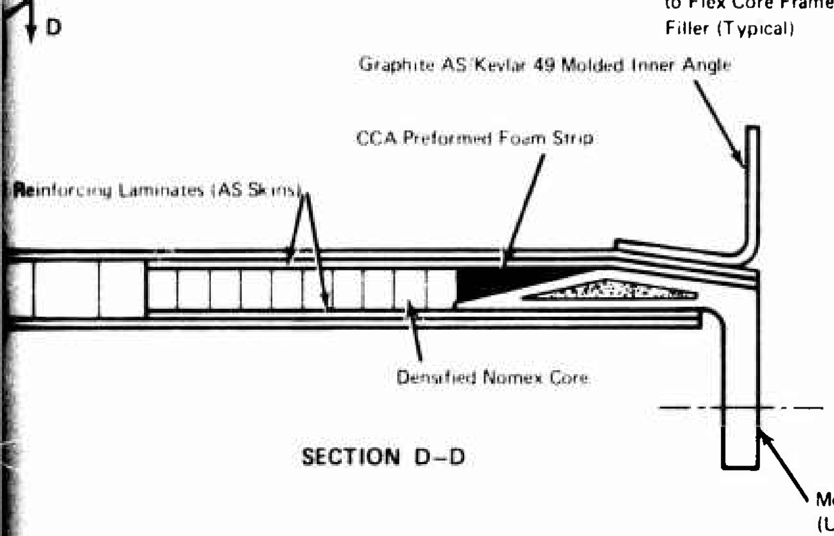


Figure 73. Airframe Module 5 - Tank Support and Side Avionics Structure - Fuel Tank Support Structure and Ballistic Protection System. (Sheet 2 of 2)



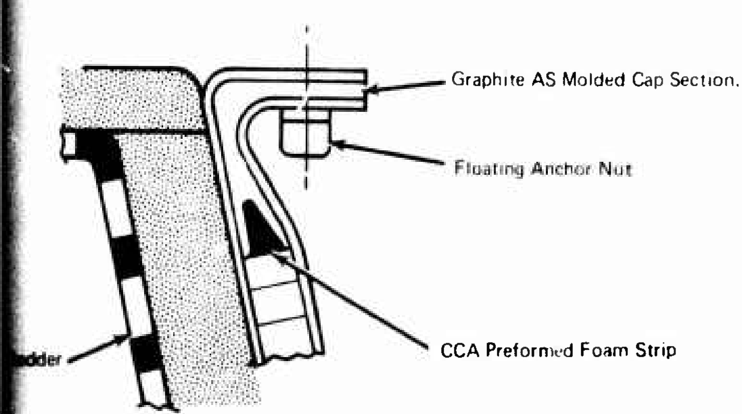
SECTION A–A





SECTION D–D

View Showing Ballistic Protection System Located in Shell Support Structure Prior to Fitting the Self-Sealing Fuel Bladder



SECTION C–C

Avionics
nd Ballistic

In order to attain continuity of the longitudinal load path to module 2, joint plates attach to the outer beams and protrude on assembly through a slot in the sta 163 bulkhead and mechanically attach to matching BL 15 underfloor beams. The basic shell is all straight line element, but the beam lower edges follow the aircraft lower contour curvature.

The tank bay itself is shorter than the tank support structure shell, extending from sta 163 to sta 218, and is closed off at the rear end by a honeycomb sandwich bulkhead. Inside the shell, over this area, is fitted the ballistic protection system comprising a series of closely spaced 3-inch-square radial frames made up of foam-filled Nomex core with unidirectional cross-ply inner cap straps. The space between each frame is filled with preformed C.C.A. or similar type foam blocks, also running radially all round. Narrow fiberglass straps are bonded longitudinally at regular intervals across the frames and foam inserts.

The semiflexible crash-resistant fuel bladder locates against the protective system, and a preformed foam lid with composite skins is fitted to completely cover the top of the tank.

<u>Side Avionics Structure (See Figure 73)</u> - This secondary structure subassembly comprises side panels, semibulkhead, shelves, and door, and is mechanically attached at each side to the tank support shell at the lower edge and to the upper deck module along the upper edge.

The fixed side panel, sta 178 to sta 239, is of honeycomb sandwich construction and follows the required aircraft double curvature contour. The skin is bounded at its forward end by a semibulkhead also of honeycomb sandwich design, which seals off the avionics bay from the landing gear/fuel accessory bay situated immediately forward. At the rear end, the bay tapers down to 3 inches wide, where a single skin molded frame forms the bay close off. Three honeycomb sandwich shelves span the avionics bay and are supported at their centers by intercostal honeycomb sandwich diaphragms.

The compound curvature avionics bay door is hinged from its forward end and is locked closed by quarter-turn fasteners. The door is designed to facilitate economical production and repeatability as an assembly of thermoplastic moldings comprising an outer skin and segmented inner dished panels which are fuse bonded together on final assembly. A detachable seal is located around the inner door periphery which, when the door is closed, compresses onto the seal surround molding positioned around the skin cutout.

The hinges and also the hinge anchorage fittings, which are mechanically attached to the subbulkhead, are designed as chopped strand moldings. The ... fitting has a T-section gooseneck, which is strengthened by unidirectional laminates bonded to the outer flange.

Forward of the avionics door is the landing gear/fuel accessory bay, which consists of a well at each side from sta 163 to sta 178 (growing wider below mean half breadth) that is covered by a detachable honeycomb sandwich panel slotted at its lower extremity to allow protrusion of the shock strut. In the left side well (at WL 171, sta 175) is the gravity fuel filler cap and below (at sta 189, WL 134) is the pressure fuel filler cap.

The attachment of the tank support Module 5 to the tailboom is by a multibolt field-splice joint, described elsewhere in this text.

The primary loading conditions for the fuel bay are those for fuel containment and fuselage shear, bending, and torsion loads arising from flight and landing conditions. The maximum fuel pressure loads occur during crash conditions with a specification load factor of 20g's with a half-full tank. These pressures design the frames in bending. The tailboom (fuselage) loads are introduced at the sta 239 splice, carried through the shell, and distributed to the WL 178 deck, sta 163 bulkhead, and the floor module. The aft end of this bay is critical for the tail bumper impact condition, and the forward end is critical for the vertical takeoff condition. These conditions size the sandwich shell and attachments to the deck and bulkhead.

The secondary structure is sized to carry local airloads, equipment loads, and personnel handling loads for both strength and stiffness.

4.3.6.6 Tailboom and Vertical Stabilizer Torque Box - Module 6 (See Figure 74)

Primary Structure - The tailboom is a honeycomb sandwich monocoque structure which extends from sta 239 to sta 423 and supports the empennage.

At sta 423 the forward attachment of the fuel bay structure is in the form of a field splice joint, and the complete tailboom and empennage is quickly removable by disconnecting an external attachment bolt system after first removing a small fairing covering the joint.

The torque box portion of the vertical stabilizer is built as a subassembly and integrated into the tailboom molded shell.

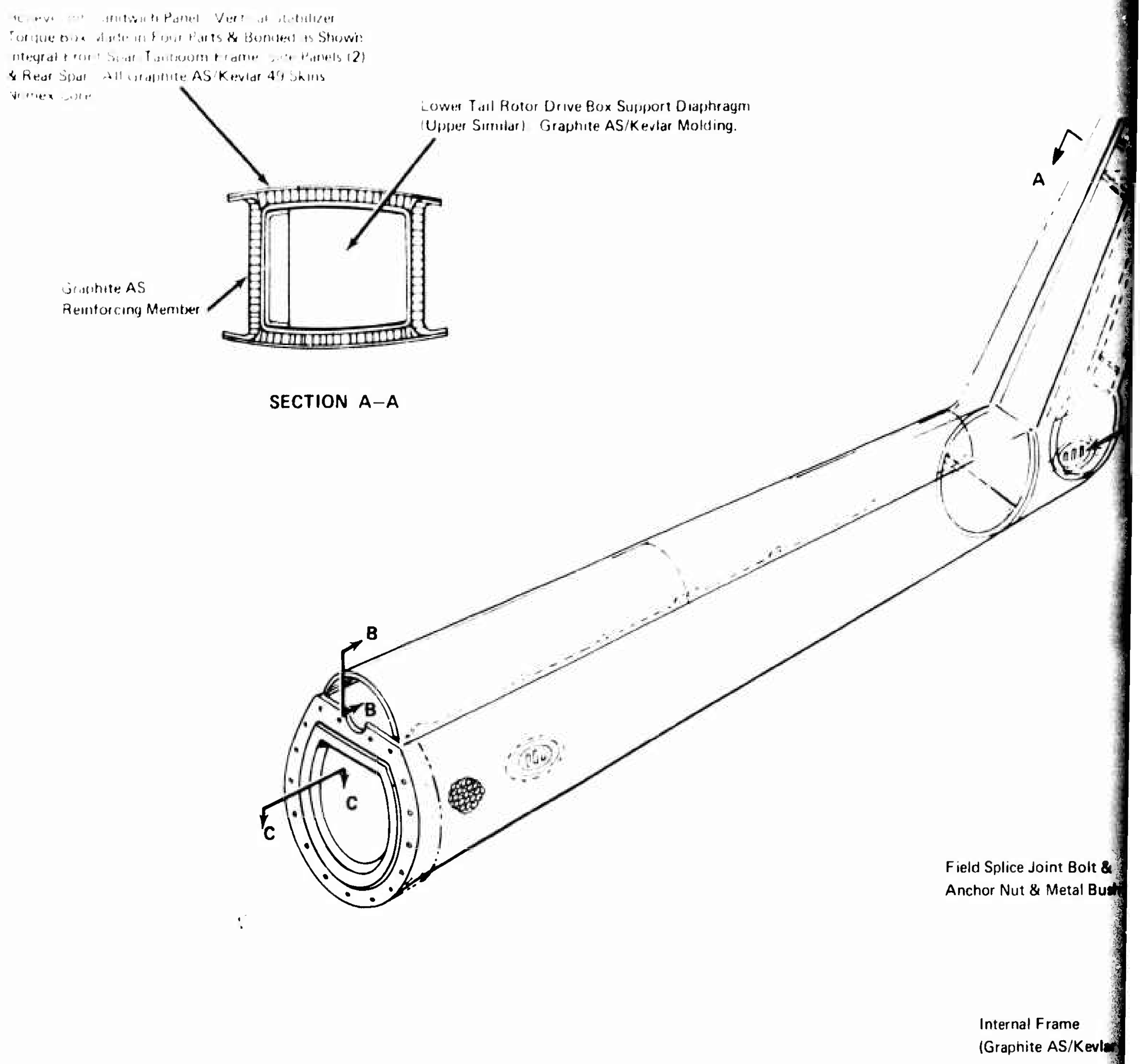
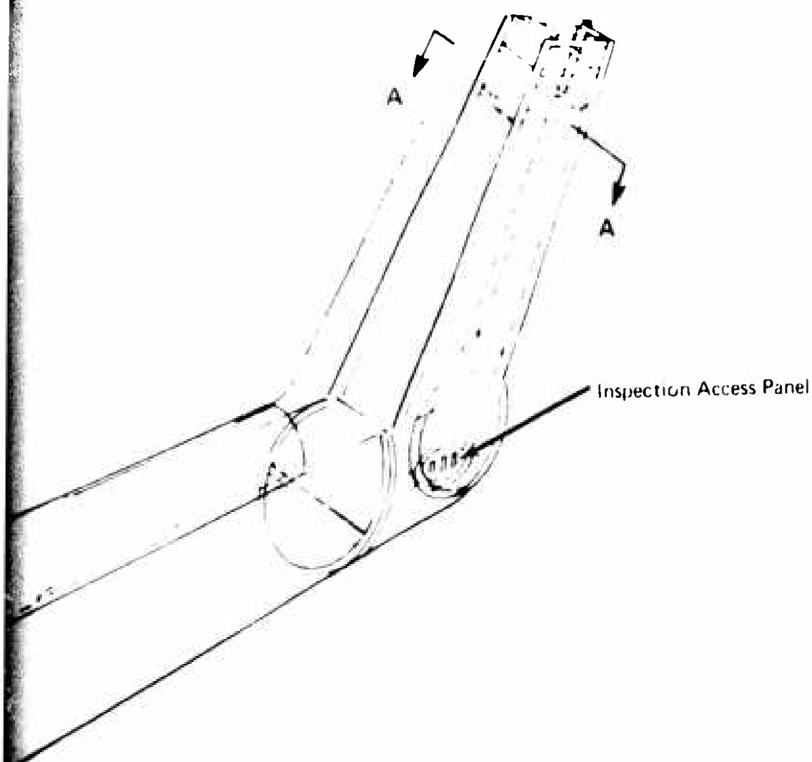
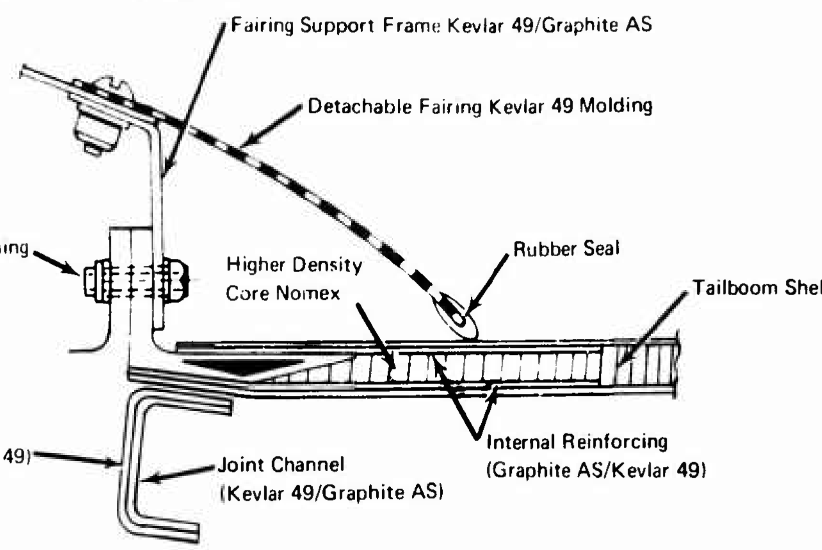


Figure 74. Airframe Module 6 - Tailboom and Vertical Stabilizer Torque Box - Main Assembly and Field Splice Joint Details. (Sheet 1 of 2)





SECTION C–C

l Stabilizer
ice Joint

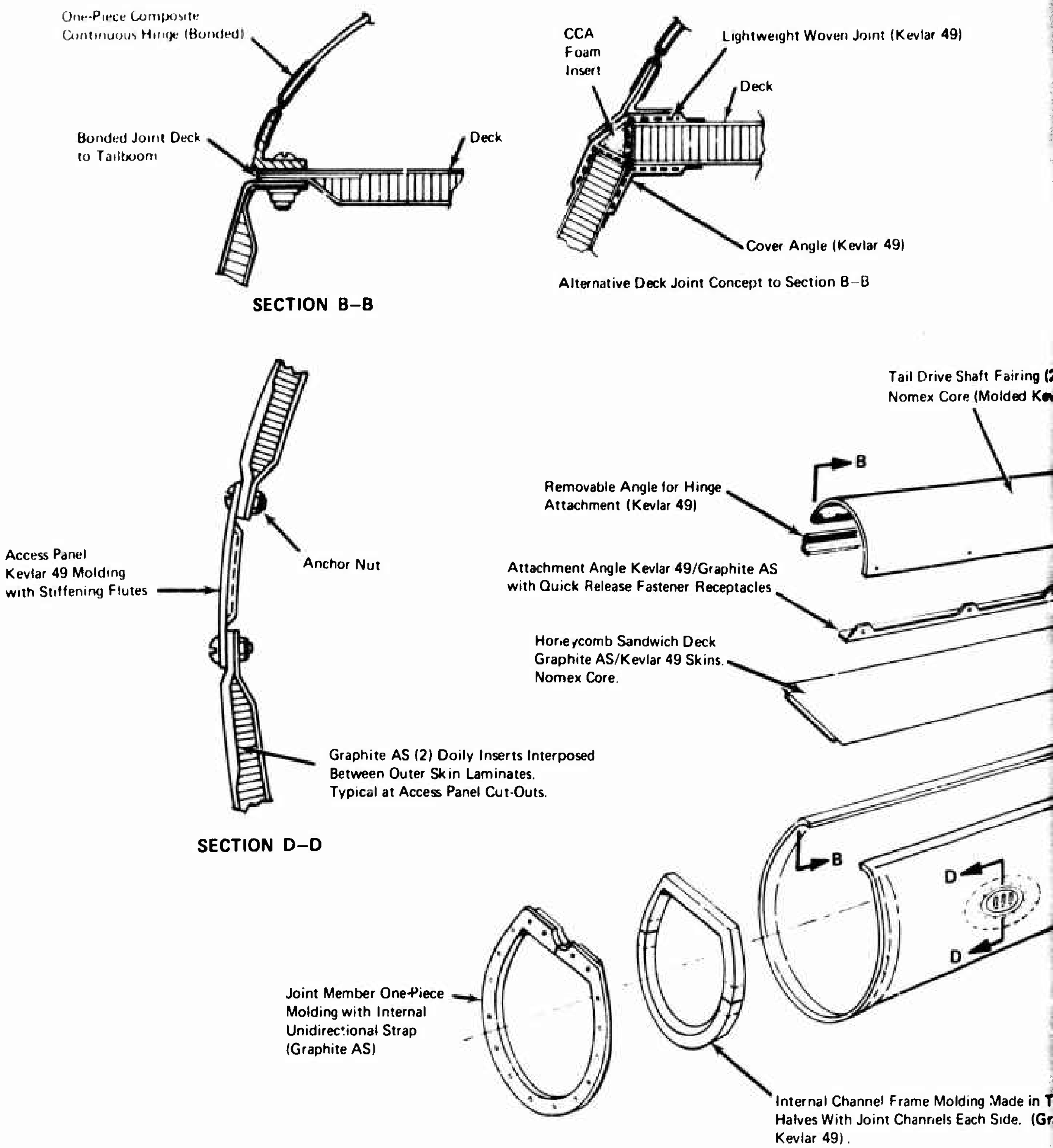
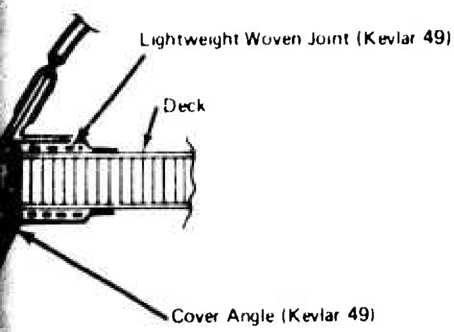
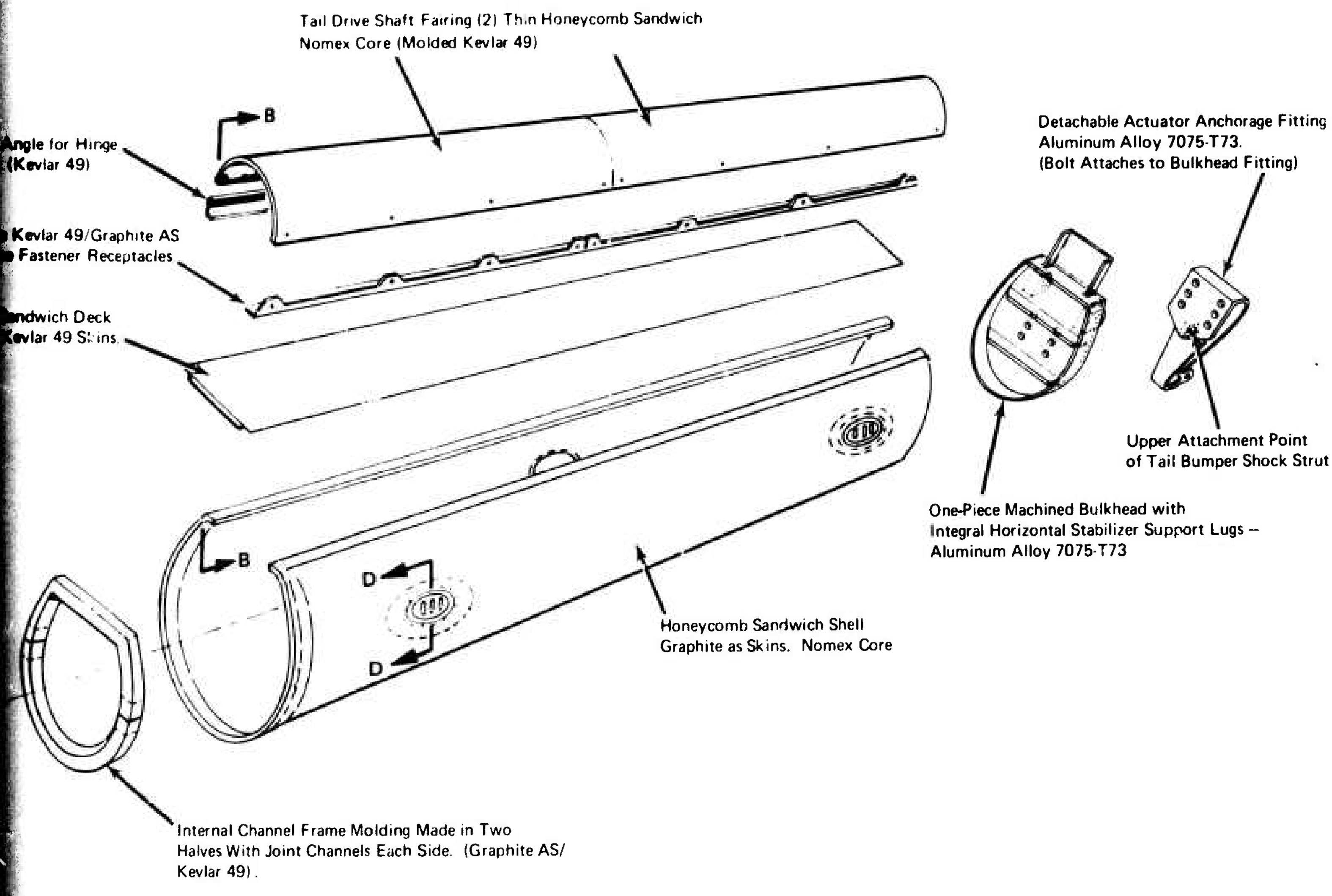


Figure 74. Airframe Module 6 - Tailboom and Vertical Stabilizer Torque Box - Structure and Tail Drive Shaft Fairing Details. (Sheet 2 of 2)



Deck Joint Concept to Section B-B



cal Stabilizer
Shaft Fairing

The main shell of the tailboom is in one piece and of truncated egg section. A separate flat deck, also of honeycomb sandwich construction with planwise taper matching the shell taper, is bonded onto each side of the shell top to form the monocoque.

The attachment structure at the forward end consists of a molded angle with a special tapered flange that fits within the shell thickness circumferentially, displacing the Nomex core. The addition of a continuous interior molded channel frame forms a mechanical lock of the inserted flange in addition to the bonded joint. Local reinforcing straps on inner and outer skins strengthen skin in the general joint area.

The vertical stabilizer torque box is made up of four honeycomb sandwich panel structures of composite skins and Nomex honeycomb core, which are bonded together enclosing two rugged support diaphragms located at the tail rotor drive box attachment position. The forward structure is a one-piece front spar and bulkhead combined, which resembles a banjo shape. Unidirectional cap material extends around the periphery of this member, which is subjected to lateral bending.

The rear spar member extends down only to the tailboom where it overlaps and attaches to the end bulkhead of the tailboom. This bulkhead is all metal with a flange extending forward into the tailboom sandwich all round and with two integral lugs extending rearwards to constitute the fulcrum for supporting the horizontal tail.

A second metal fitting bolts onto the bulkhead and extends down and to the rear to form an anchorage position of the stabilizer actuator and also the upper attachment point of the tailbumper energy absorber unit. Access into the boom for inspection purposes is through molded detachable access panels.

<u>Secondary Structure</u> - A hinge-up cover for the tail driveshaft is fabricated in two segments consisting of thin honeycomb sandwich moldings with Nomex core. The fairings fold over from left to right on a continuous all composite hinge and are fastened to a separate molded retention angle by quick-release (quarter-turn) Camloc fasteners (receptacles are blind riveted to angles for detachability).

4.3.6.7 <u>Tailcone Fairing - Module 7</u> (See Figure 75)

This secondary structure assembly attaches to the tailboom to form a fairing for both tailboom and vertical stabilizer, as well as a ventral fairing for the tailbumper unit, which is stowed within the fairing for transportation when disconnected from the energy absorbing actuator (see Figure 75). The long stinger part of the fairing extends rearward past the tail rotor tip radius essentially to prevent personnel from accidentally walking into the rotating tail rotor blades. A further function of the fairing is to form a flat sliding surface on each side in line with the horizontal stabilizer through structure and cover plates to make a suitable aerodynamic seal. The tailcone fairing must be removed first before the horizontal stabilizer can be disconnected by its forward pivot bolt attachments. This is accomplished by removing the quick-release fasteners connecting the cone to the tailboom and also removing a pivot cover plate on each side and sliding the cone rearward to detach it.

For ease of fabrication, the tailcone fairing is made in two halves with a vertical centerline joint. The half segments are molded with a stiffening angle added on each side in a cocure operation. After curing, the two parts are affixed together in a secondary bond operation, and cover plates and quarter-turn fastener studs are added. The material used for complete assembly is Kevlar 49/epoxy style 181 fabric. For production quantities, this item could also be post-formed Kevlar 49/thermoplastic.

4.3.6.8 Tail Bumper and Absorber Assembly - Module 8
(See Figure 76)

A tailbumper is provided at the aft end of the tailboom to absorb energy due to sink speeds up to 18 fps in a hard tail-down landing. No yield is allowed in the tailbumper system or the airframe structure at that sink speed. Energy is absorbed by a Torshock unit. An alternative attenuating device would be a shock strut with a compressed silicone elastomer flowing through an orifice.

The tailbumper assembly is composed of 8 parts. The main beam is of composite design and is made in matching molded halves with vertical flanges to facilitate efficient bonding together on assembly. A stiff T-section extrusion located at the rear end is trapped between the halves and bonded with them on assembly. This section supports a replaceable wear pad of stainless steel which is mechanically attached to the flat of the T section. At the forward end of the bumper assembly U section, composite molded fittings bond to each side of the halves to provide the pivot anchorage for the beam. Chopped strand molded fittings, bonded to the front spar bulkhead, extend down through the tailboom lower skin and support the tail-bumper forward attachments. The energy-absorbing shock strut lower anchorage is made onto the vertical flange of the T section by a bushed hole.

4.3.6.9 Horizontal Stabilizer and Actuator - Module 9
(See Figure 77)

In order to optimize flying qualities and minimize rotor hub

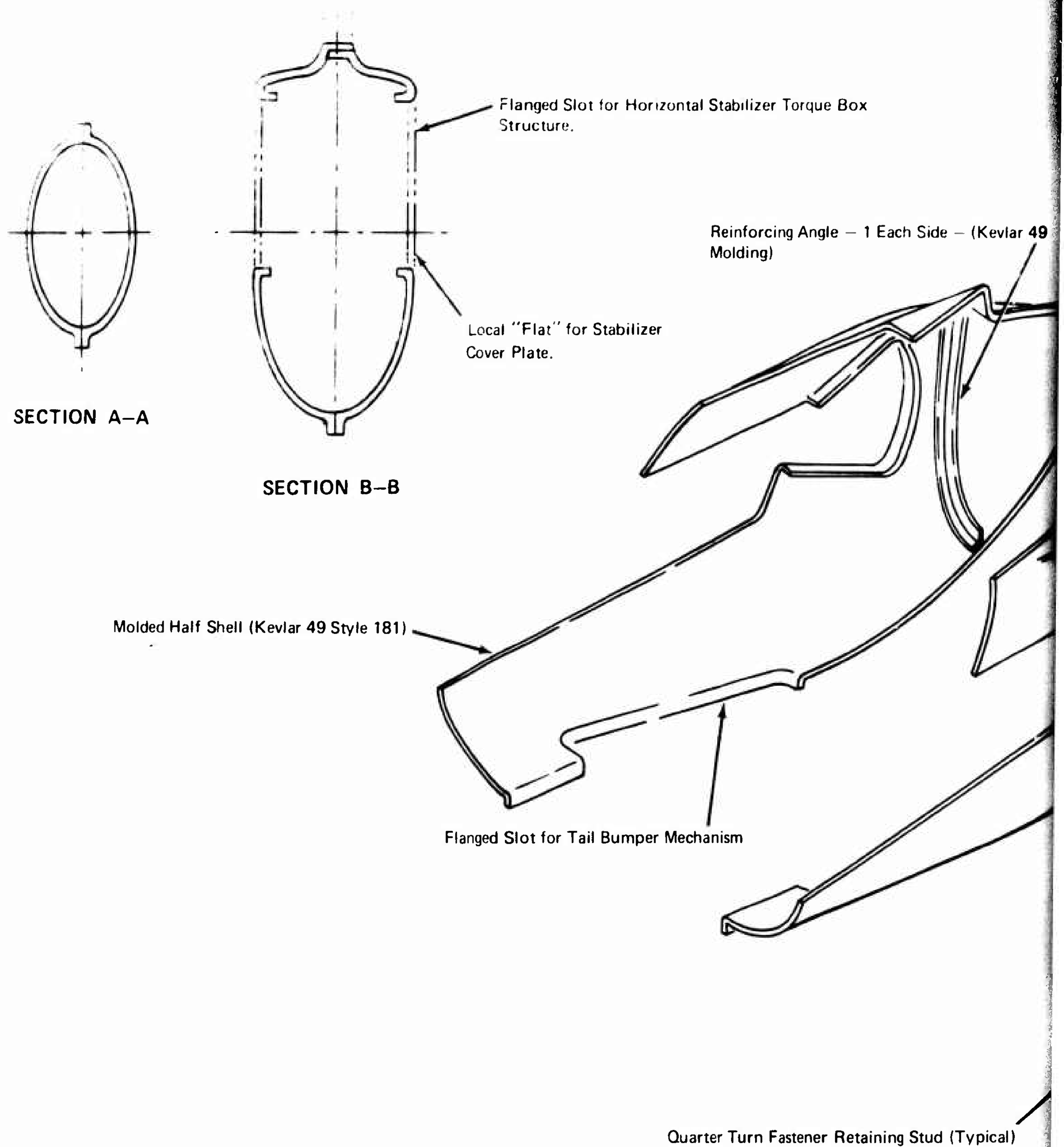
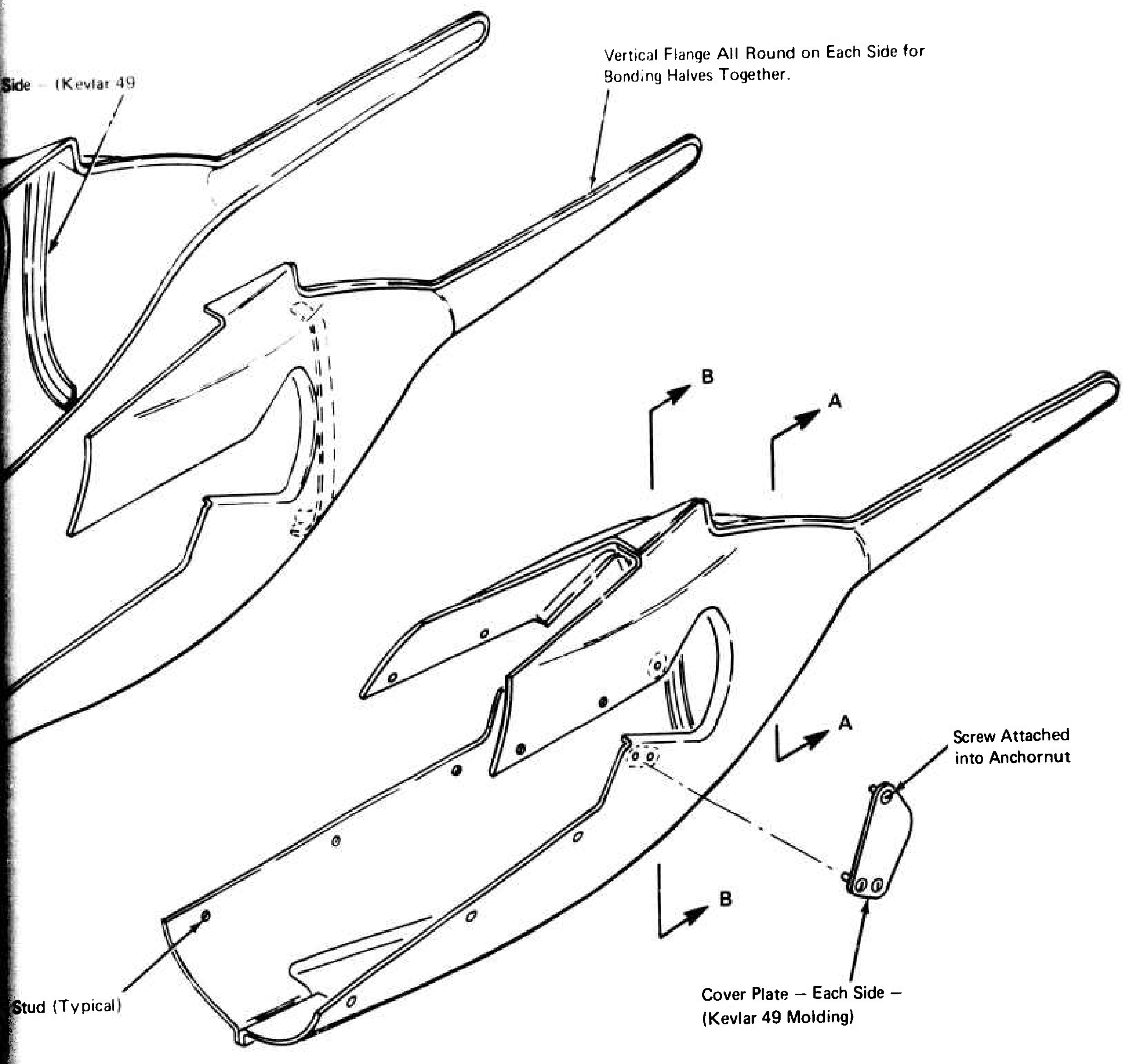
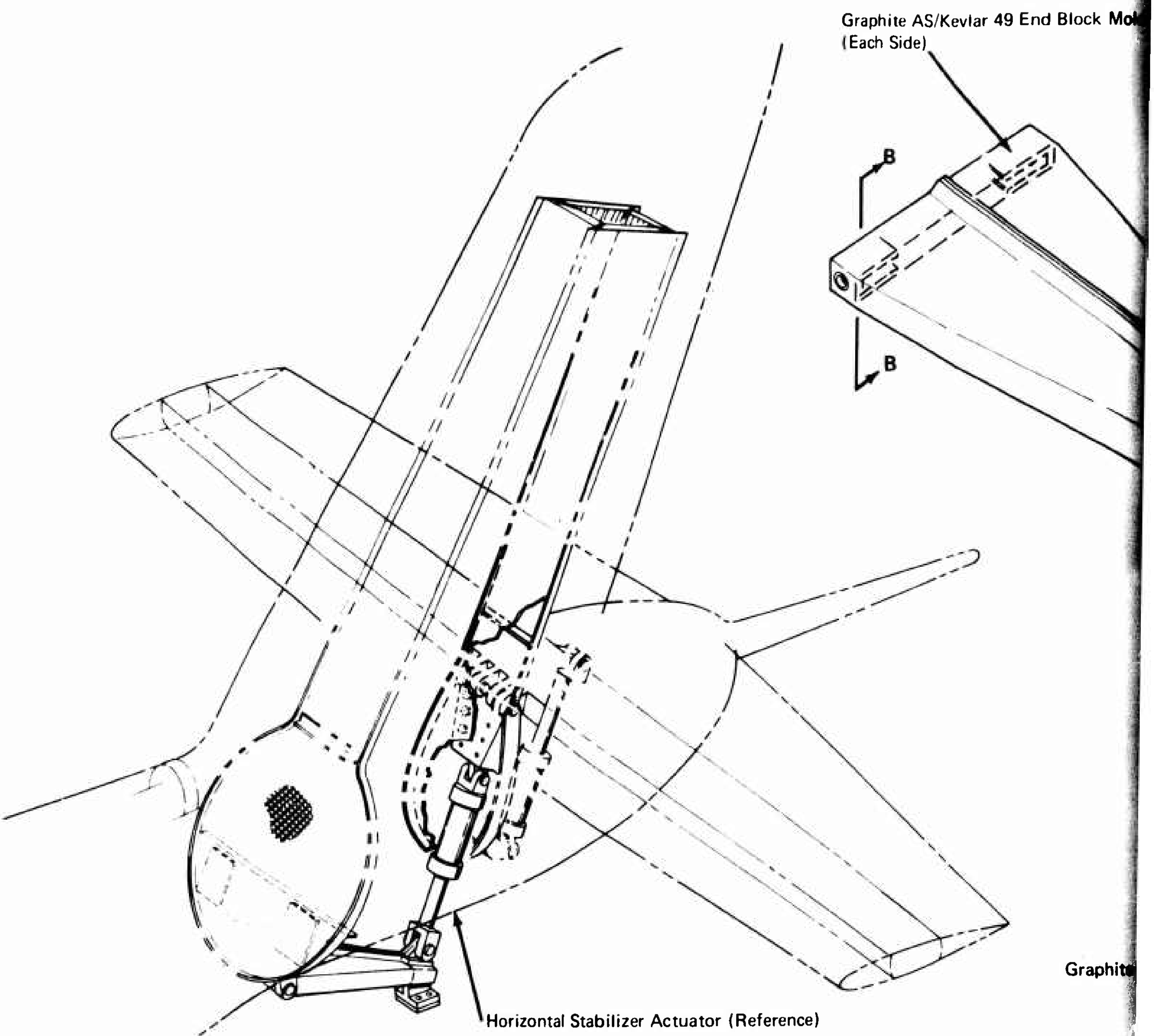


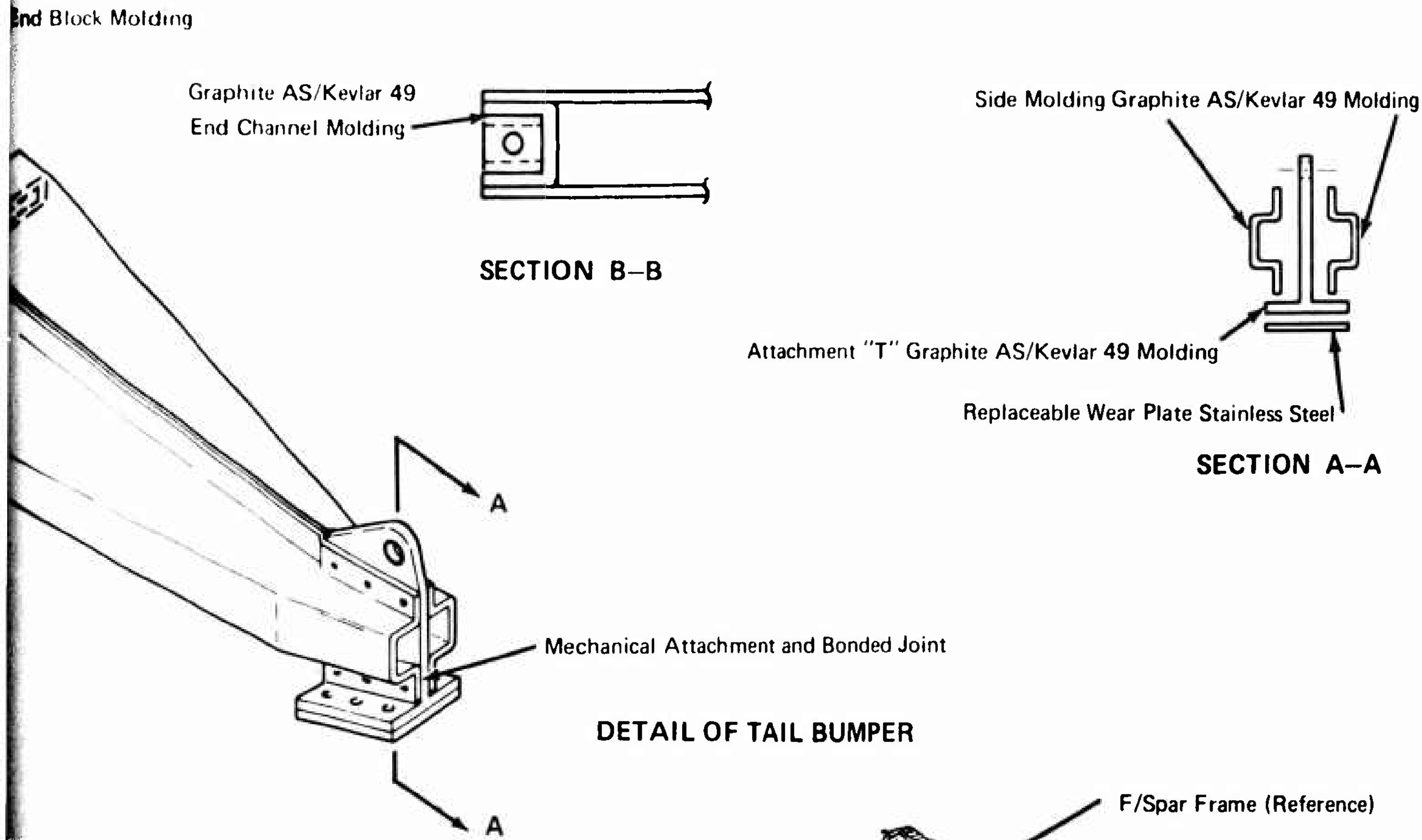
Figure 75. Airframe Module 7 - Tailcone Fairing Assembly.


Side – (Kevlar 49
Vertical Flange All Round on Each Side for Bonding Halves Together.
B
A
A
B
Screw Attached into Anchornut
Stud (Typical)
Cover Plate – Each Side – (Kevlar 49 Molding)

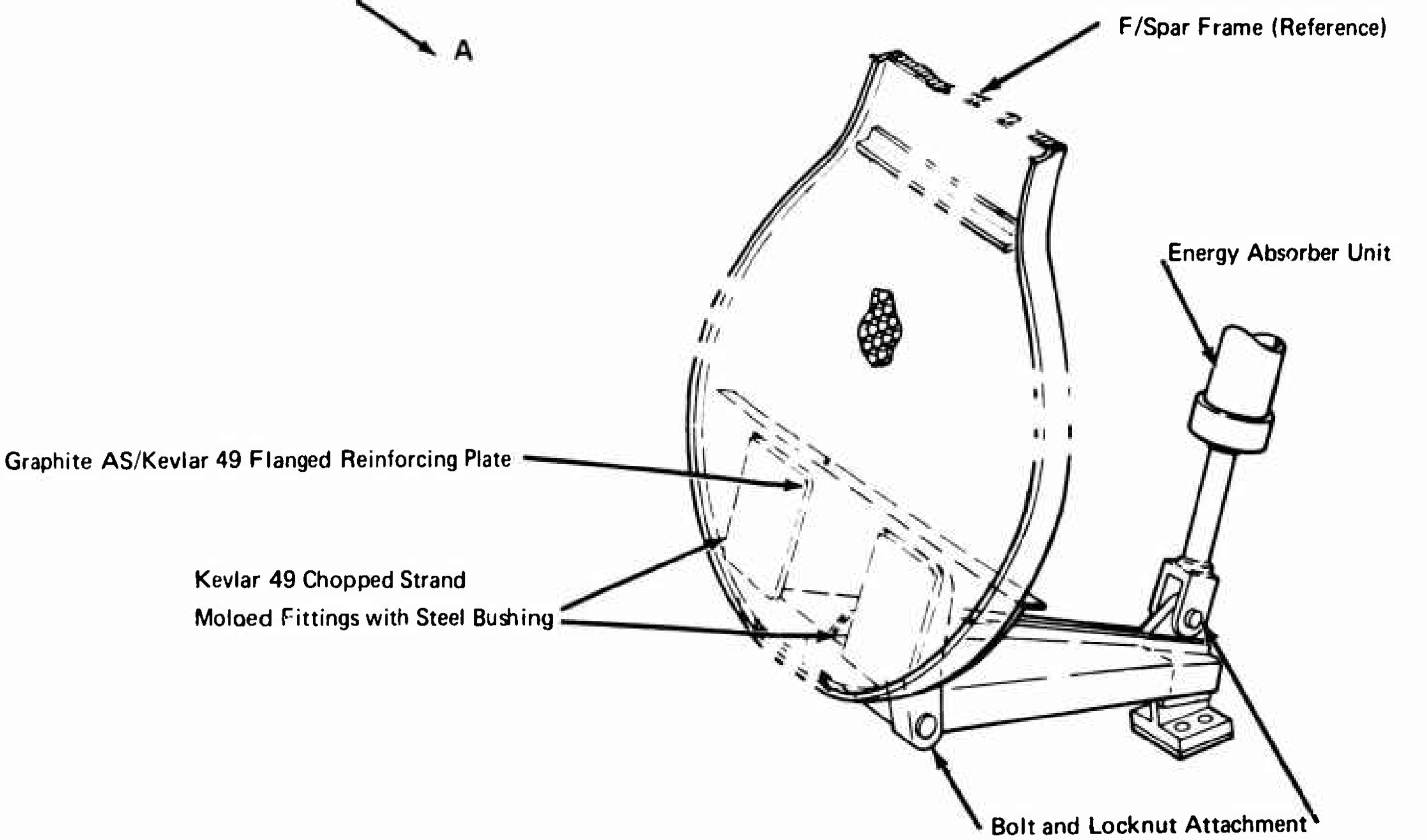


VIEW OF BUMPER ARM AND ENERGY ABSORBER ASSEMBLED

Figure 76. Airframe Module 8 - Tailbumper and Absorber Assembly.



DETAIL OF TAIL BUMPER



VIEW OF TAIL BUMPER AND ENERGY ABSORBER UNIT

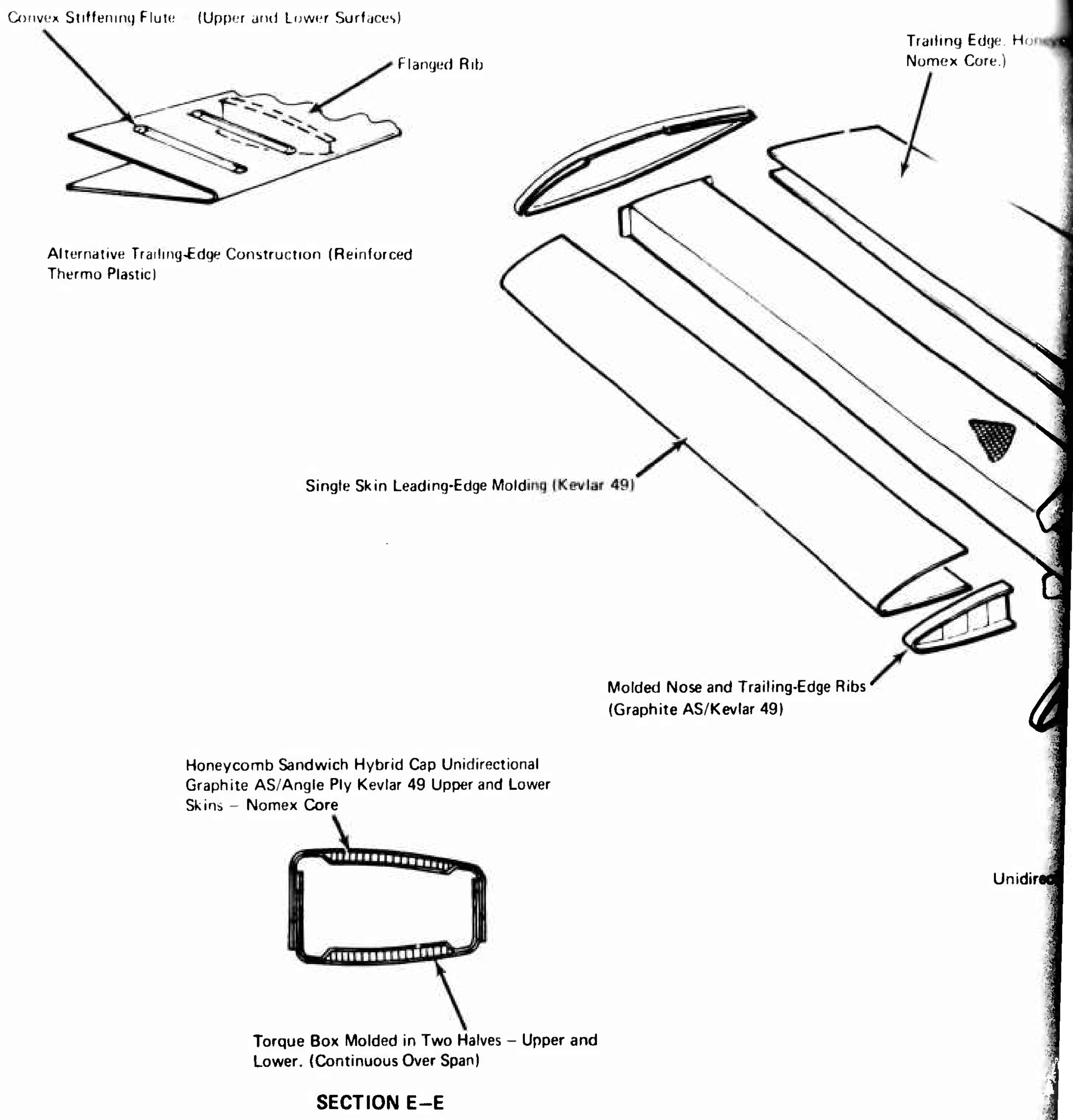
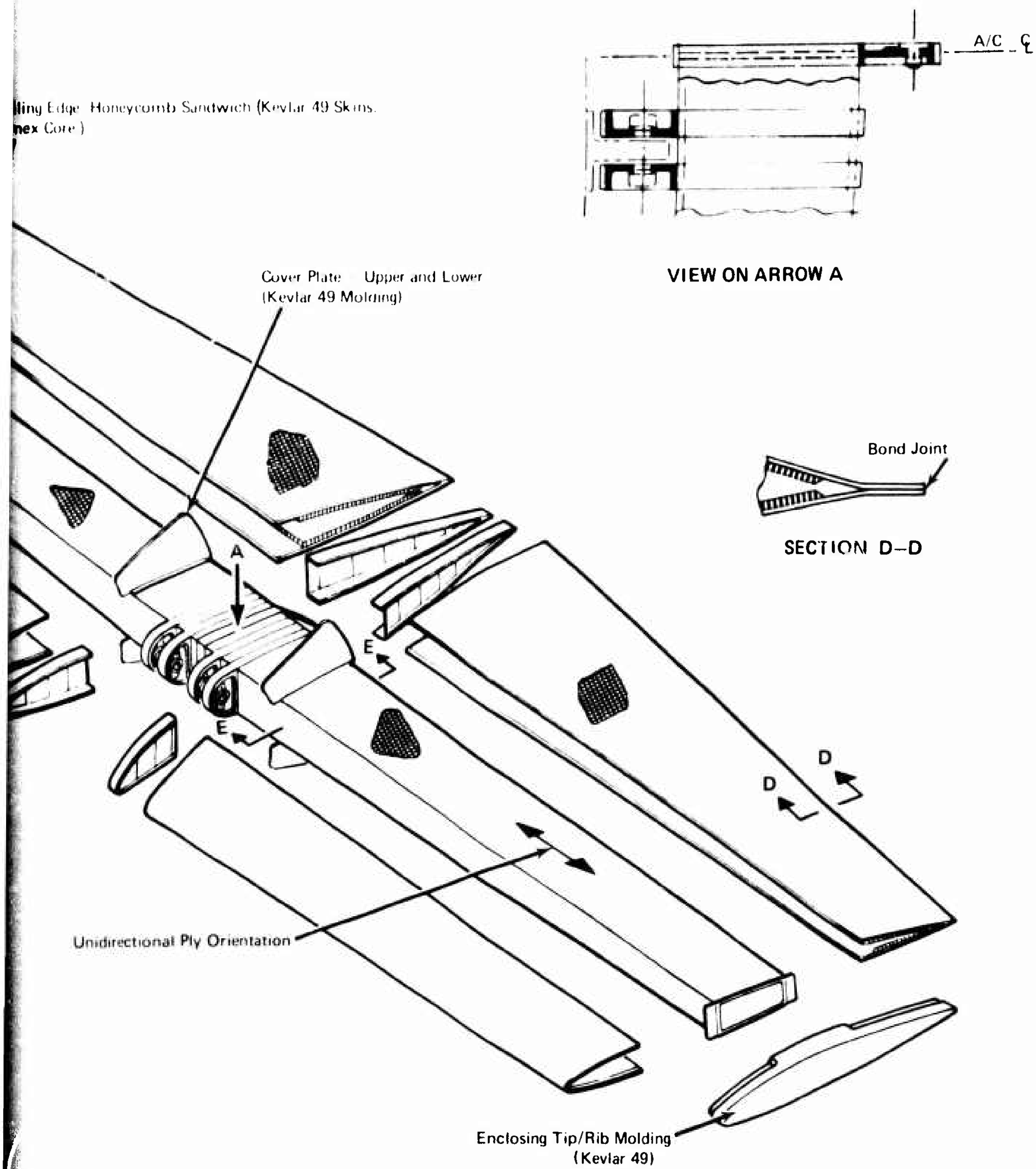
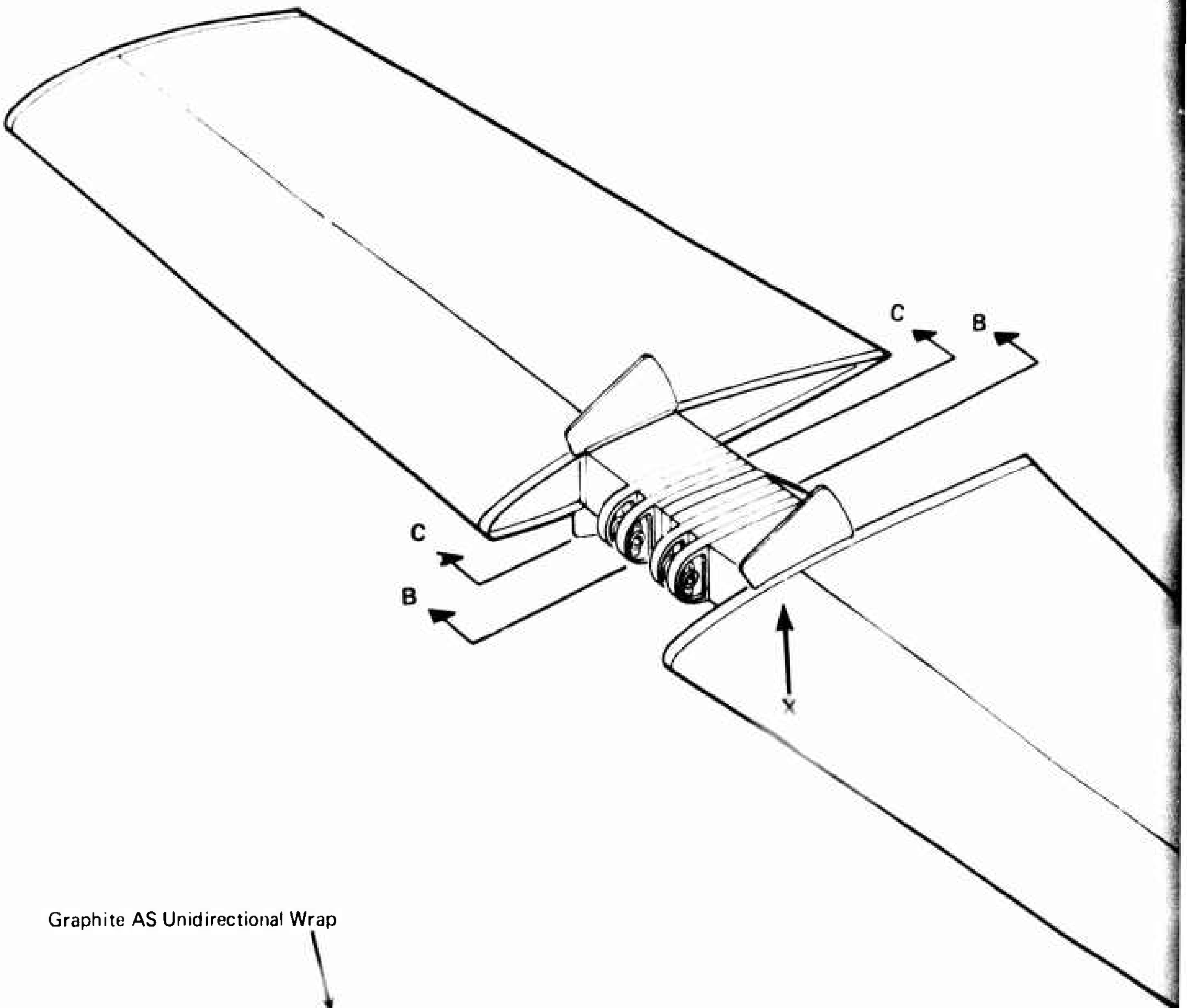
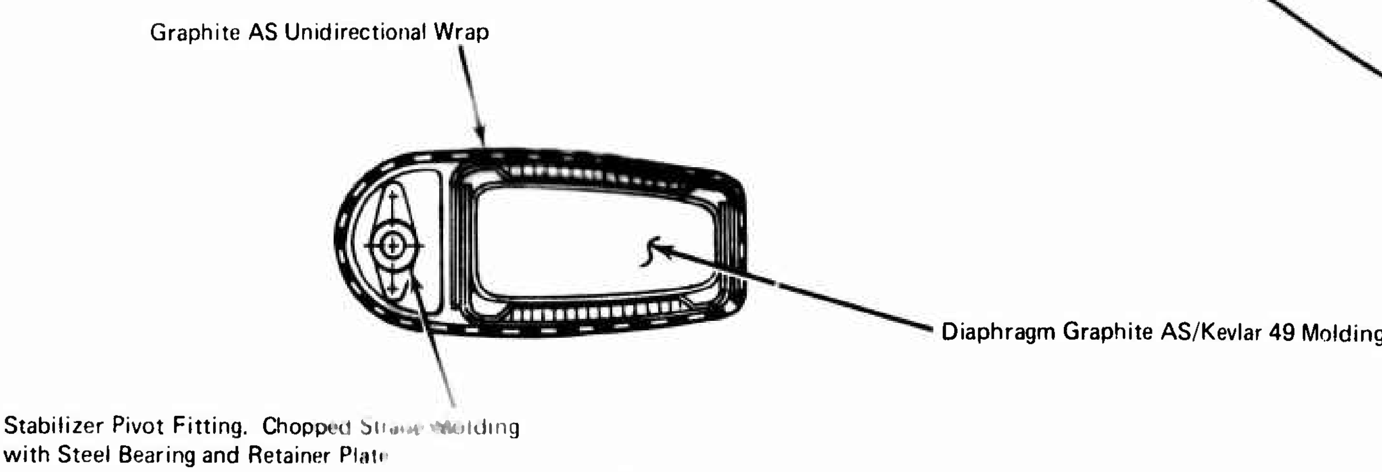


Figure 77. Airframe Module 9 - Horizontal Stabilizer and Actuator Exploded View. (Sheet 1 of 2)

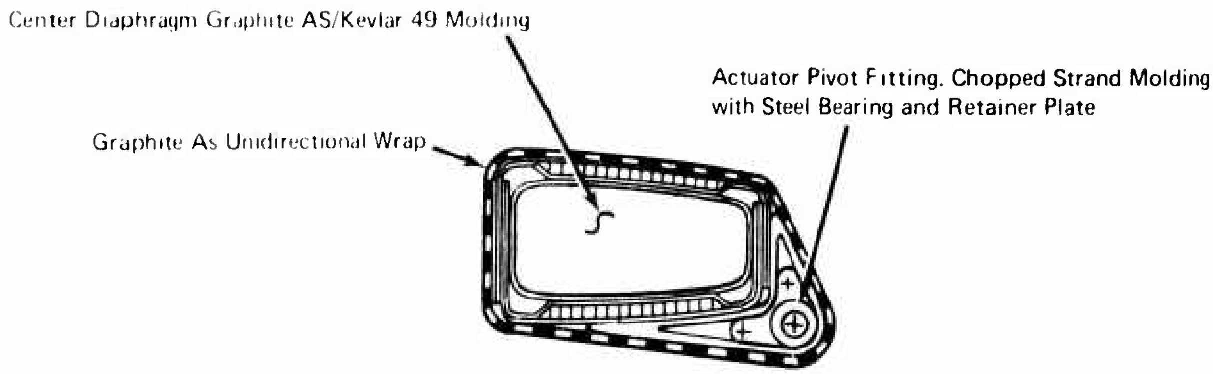

ling Edge Honeycomb Sandwich (Kevlar 49 Skins.
nex Core.)
A/C ℄
VIEW ON ARROW A
Cover Plate Upper and Lower
(Kevlar 49 Molding)
Bond Joint
SECTION D–D
A
E
E
D
D
Unidirectional Ply Orientation
Enclosing Tip/Rib Molding
(Kevlar 49)



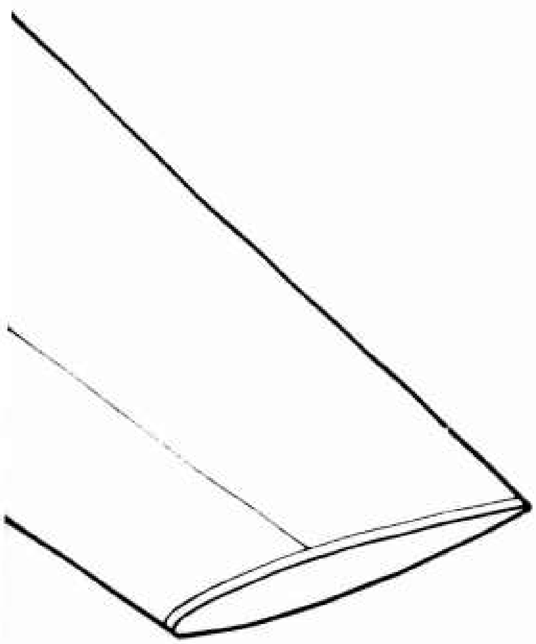


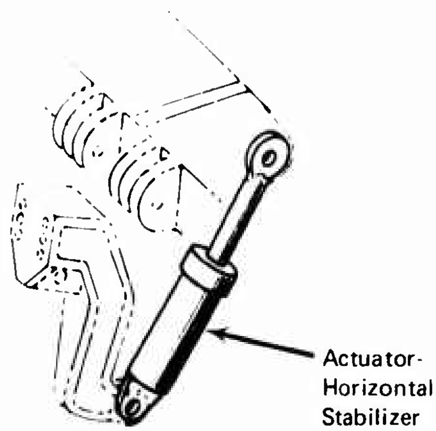
SECTION C–C

Figure 77. Airframe Module 9 - Horizontal Stabilizer - Assembled View. (Sheet 2 of 2)



SECTION B–B





DETA!L X

ng

loads, a variable incidence horizontal stabilizer is incorporated. It is operated by an electromechanical actuator which is controlled by airspeed, longitudinal, and collective stick positions. Angular displacement of the stabilizer is +5° nose down to +45° nose up.

The one-piece full-span tailplane consists of a single-skin leading-edge section, a box spar assembly, a trailing-edge section, and separate tip ribs.

The box spar spans from tip to tip, tapering down in cross section both in depth and width from the aircraft centerline. It is constructed in upper and lower halves, each half in the form of a channel section with sandwich webs providing the bending material. The flanges of each half overlap to form a shear resistant web and are bonded together at assembly. In the root area where the shear loads are highest, the flanges overlap the full depth of the spar, while outboard the overlap is reduced to that required for the shear load transfer in the overlap bond. The sandwich face sheets are laminated of zero-degree graphite epoxy and angle ply Kevlar to form a hybrid. The channel flanges which form the spar shear material are angle-ply Kevlar laminates extending out of the sandwich face sheet laminates.

The nose section is a single-skin layup comprising style 181 plies, and is supported by ribs where the cutout is made at center for clearance about the tailboom. The nose skin overlaps and is riveted to the spar using blind cherry bulb (non-expanding shank) rivets, thus allowing a degree of detachability.

The trailing-edge segment is made up of two full-span sandwich panels which bond onto the spar at their forward ends and run down to bond together at their trailing edge. An alternative one-piece thermoplastic molded skin with stiffening flutes and separate support ribs is also shown in the stabilizer isometric sketch, the inset view in Figure 77.

A molded style 181 tip rib/fairing closes off the stabilizer at each end, while channel section root rib moldings seal off fairing at the center where a segment is removed each side of the spar for clearance about the tailboom.

Enclosed within the box spar each side of centerline at butt-lines 3 and 5 are molded ribs, while in line with these on the forward side of the spar are smaller molded ribs which hold detachable fittings enclosing steel bearings (two per side), forming the pivot for the stabilizer. These forward ribs are bonded to the spar, and each is also supported by a unidirectional circumventing bonded strap at each bearing position. A similar internal molded rib is located on the centerline with a smaller rib containing the actuator pivot bearing bonded onto the aft surface of the box spar and consolidated by a similar undirectional strap arrangement. Molded cover plates are bonded to upper and lower skin surfaces over box spar in line with the trailing-edge inboard closure ribs (total of four plates).

4.3.6.10 <u>Vertical Stabilizer Leading-Edge and Trailing-Edge Fairing and Tip - Module 10</u> (See Figure 78)

The leading edge of the stabilizer is divided into two secondary structure sections, namely the intermediate gearbox fairing and the driveshaft cover fairing, extending from the tailboom up to the nose support rib. The intermediate gearbox fairing is designed as a thermoplastic molding with integral cooling louvers on each side. It is detachable by quick-release fastener operation for access to the gearbox.

The driveshaft cover fairing is of thin honeycomb sandwich construction and is hinge attached to the stabilizer main torque box (part of the tailboom module) on the right side, while the left side of the fairing contains quick-release fastener studs which attach to receptacles in the torque box. This fairing extends from the nose support rib to the tip support rib, which is attached to the torque box assembly, and has an integrally molded rotor drive box fairing (forming the forward half of the fairing). Another detachable rear section of the rotor drive box fairing attaches via anchor nuts in the torque box.

Forward-extending fairing support ribs attach to the stabilizer torque box and are located between the intermediate gearbox fairing and drive shaft fairing and also between tip and drive shaft fairings.

The trailing-edge section extending from the tailboom to the tip attachment rib is a thermoplastic one-piece molding with horizontal concave stiffening flutes. End and intermediate molded ribs support the fairing, which is a bonded on assembly to the torque box edge.

The fin tip is a thermoplastic molding which, due to the narrow shape, may be made in two segments then fuse bonded together by a joggled overlap skin arrangement. The tip is easily removed by screw attachments into anchor nuts located in the tip support rib.

4.3.6.11 <u>Cabin Side and Hinged Doors - Module 11</u> (Figure 79)

Access to the cabin is effected by opening the hinged forward door and/or sliding back the rear door. Doors are located on both sides of the fuselage and may be opened together (for litter loading, etc.) or individually.

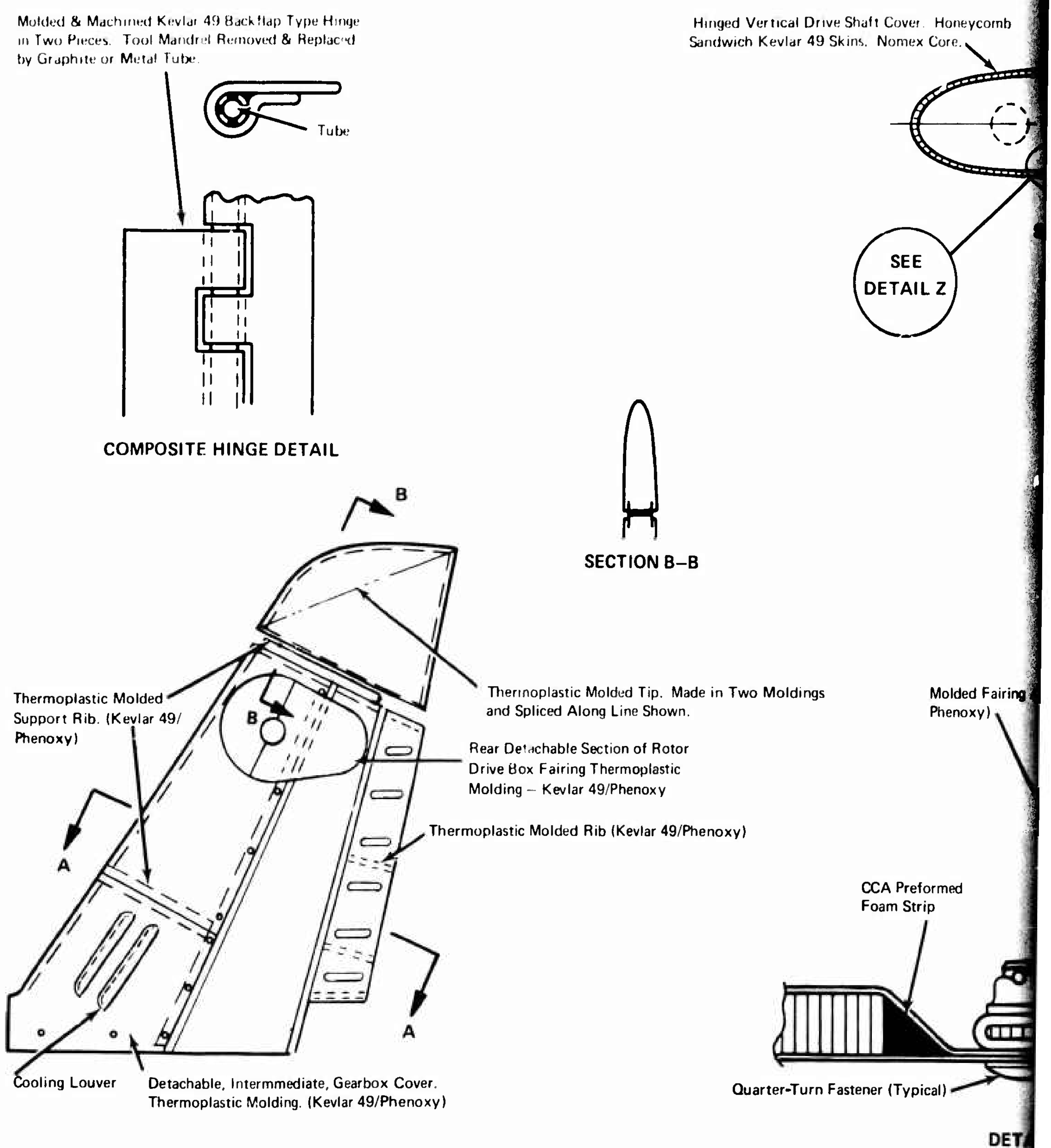
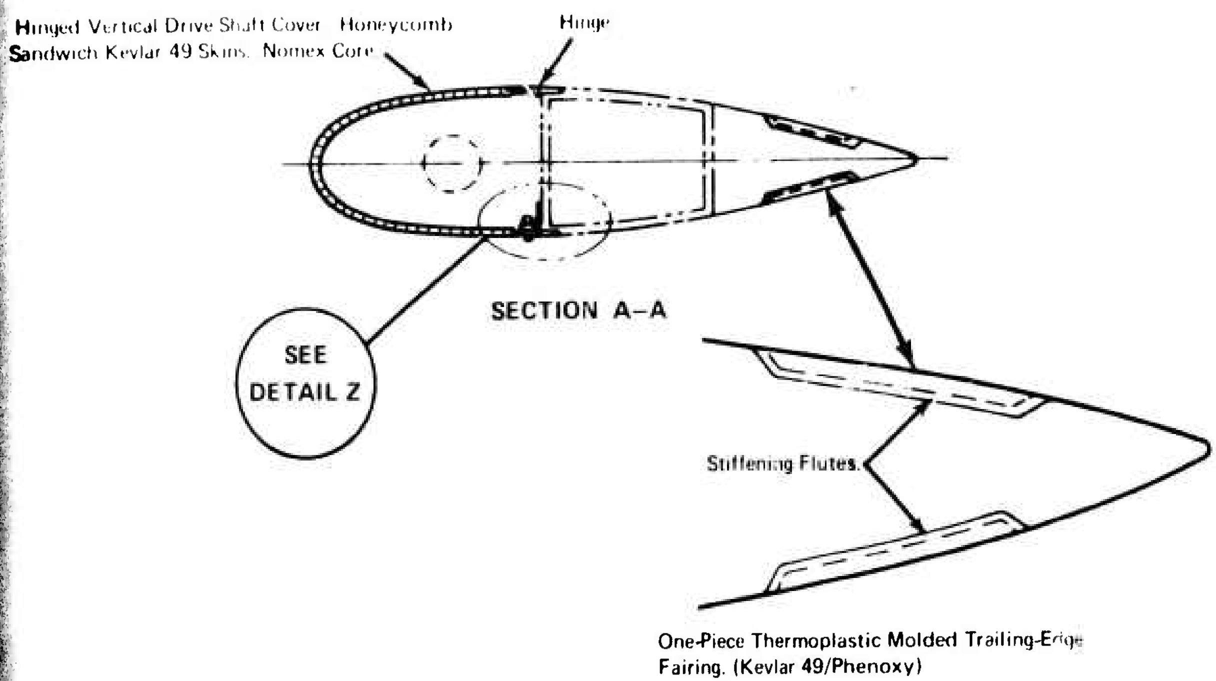
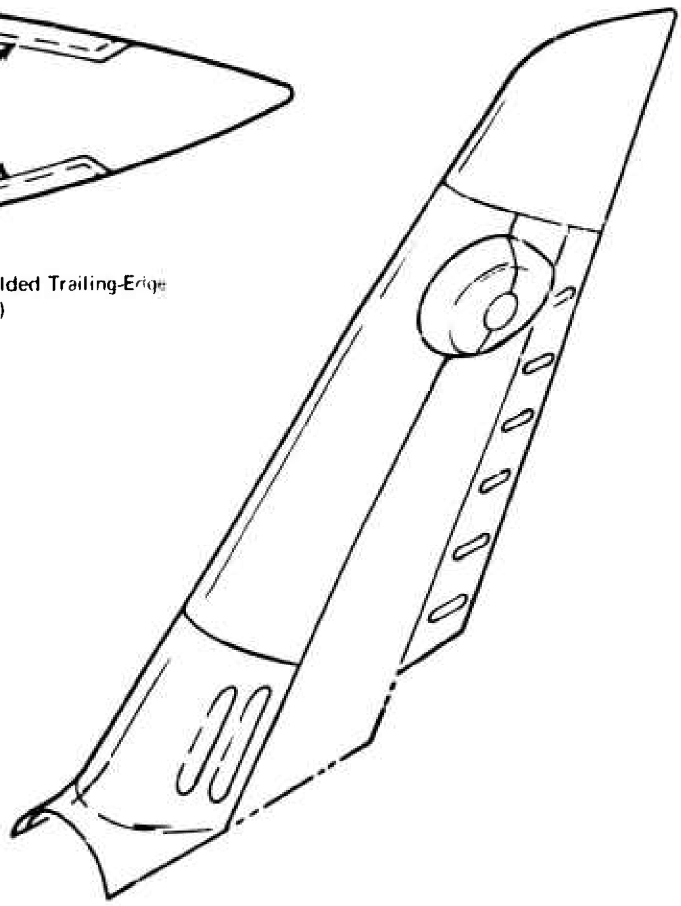


Figure 78. Airframe Module 10 - Vertical Stabilizer - Tip, Leading Edge and Trailing Edge Fairing.



SECTION A–A



-B

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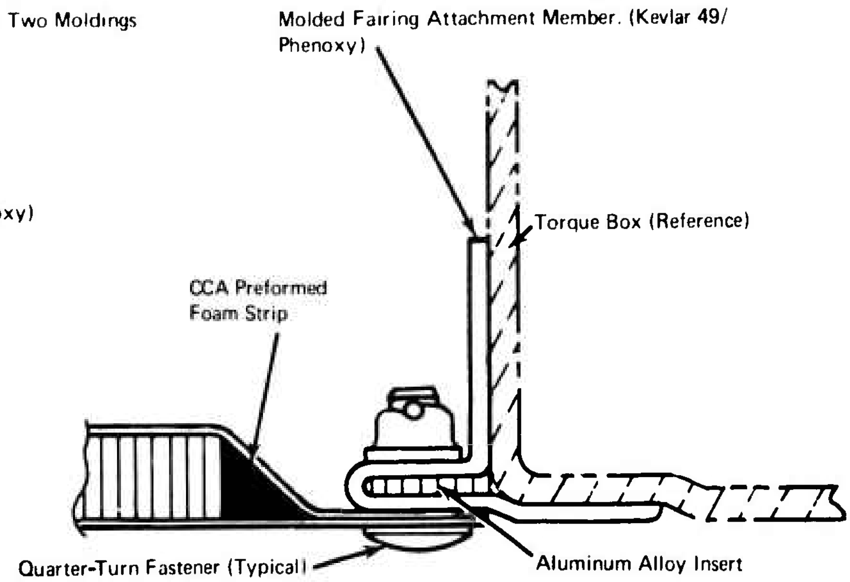
enoxy)



DETAIL Z

Tip,

2

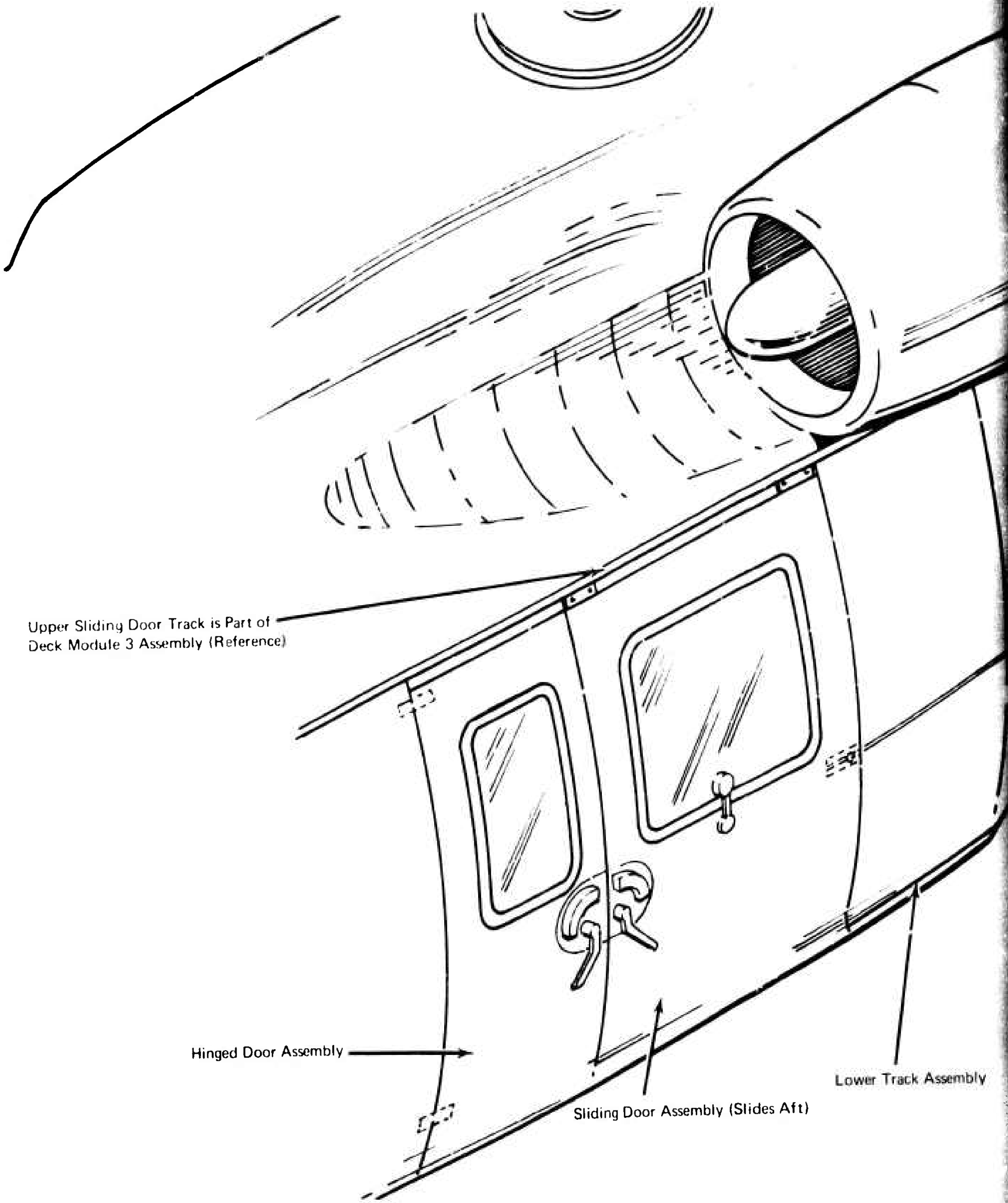
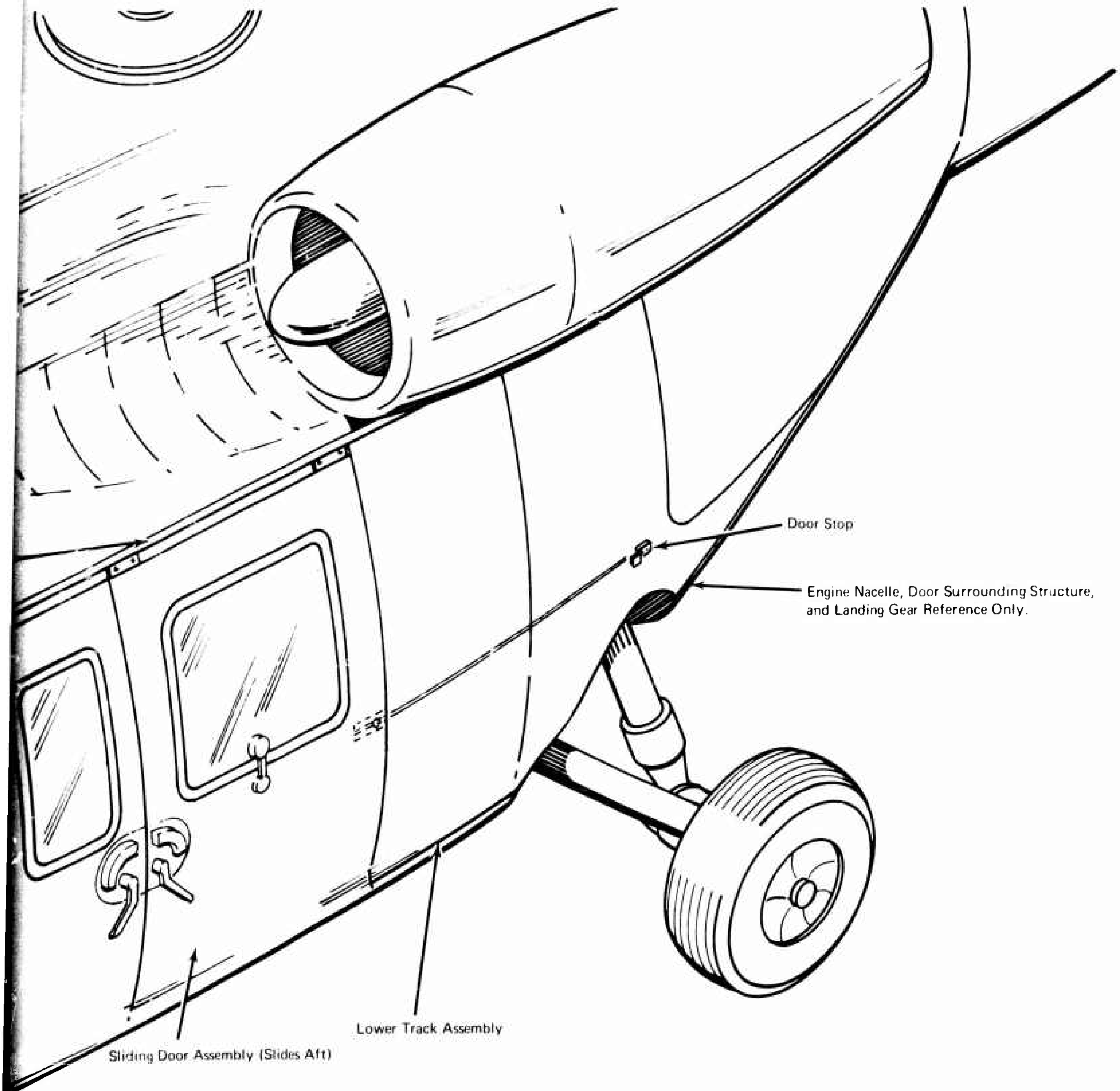


Figure 79. Airframe Module 11 - Cabin Sliding and Hinged Doors - Assembled View. (Sheet 1 of 3)



liding and Hinged Doors -
3)

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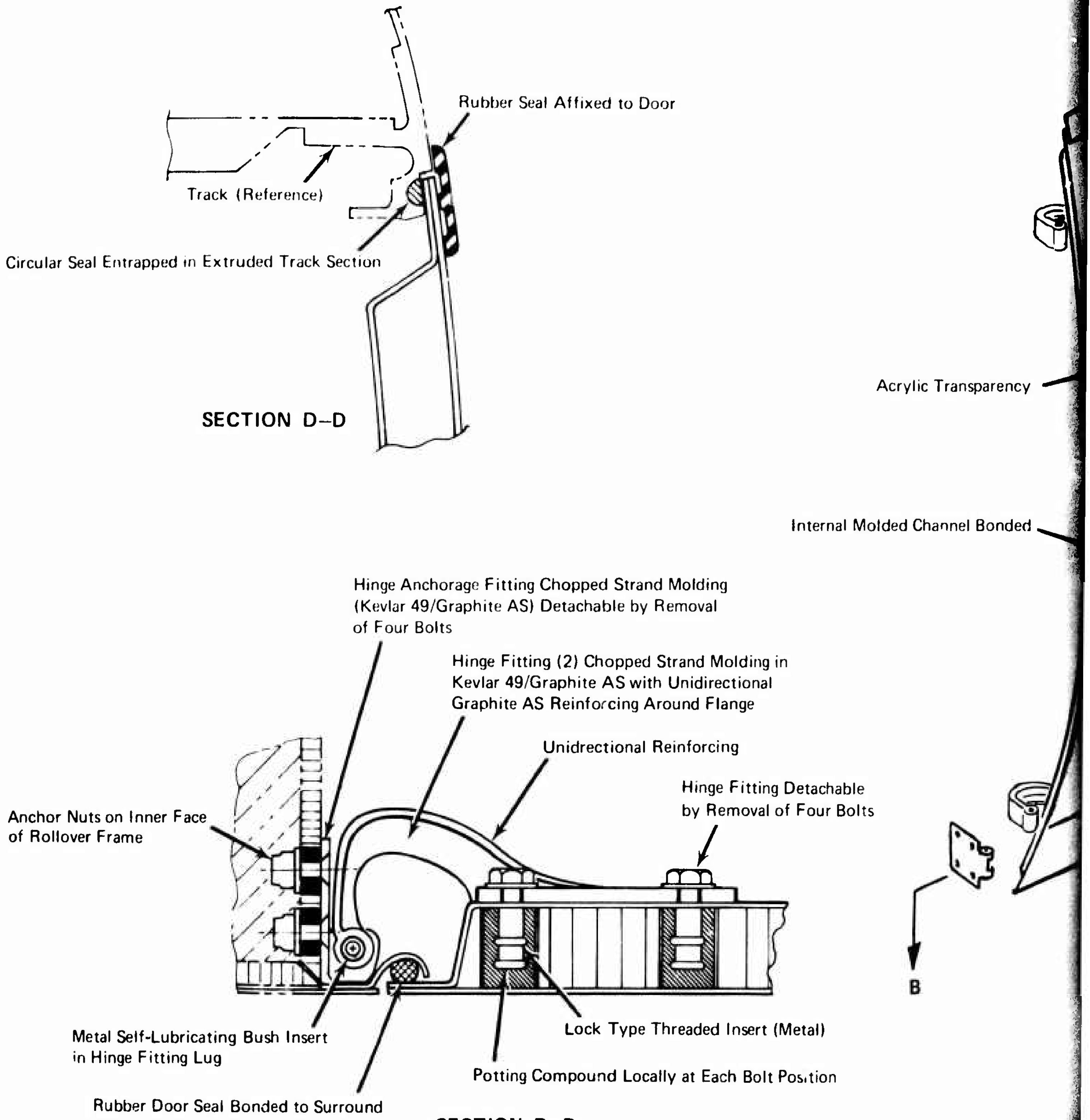
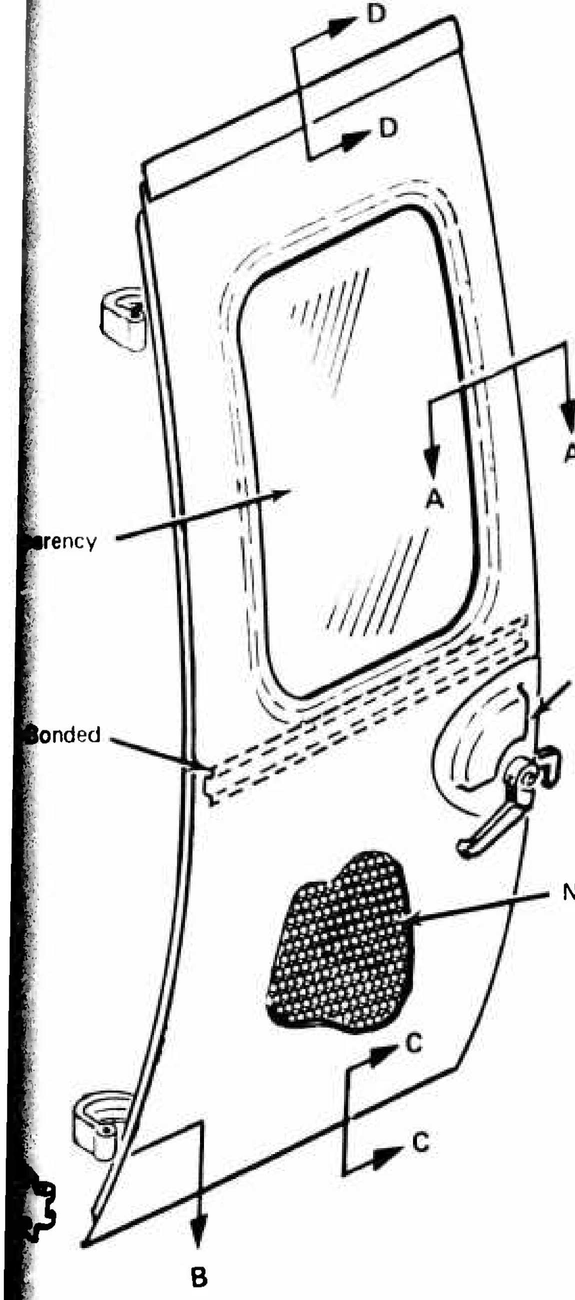
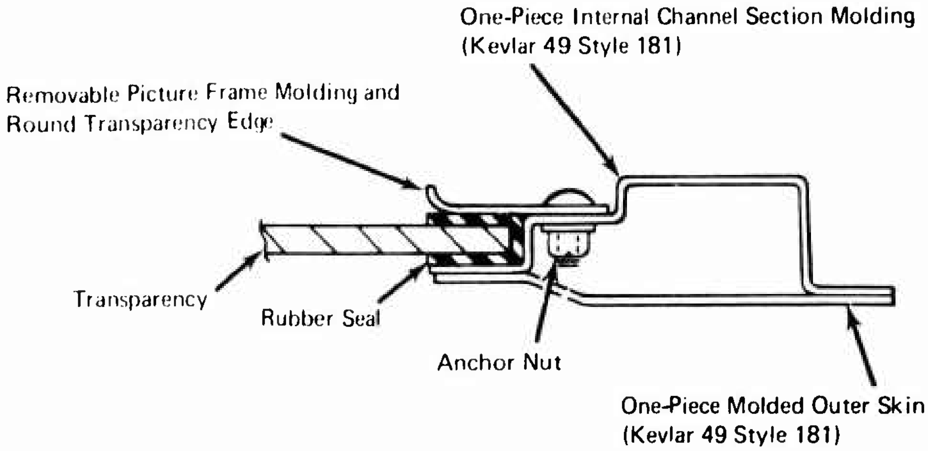
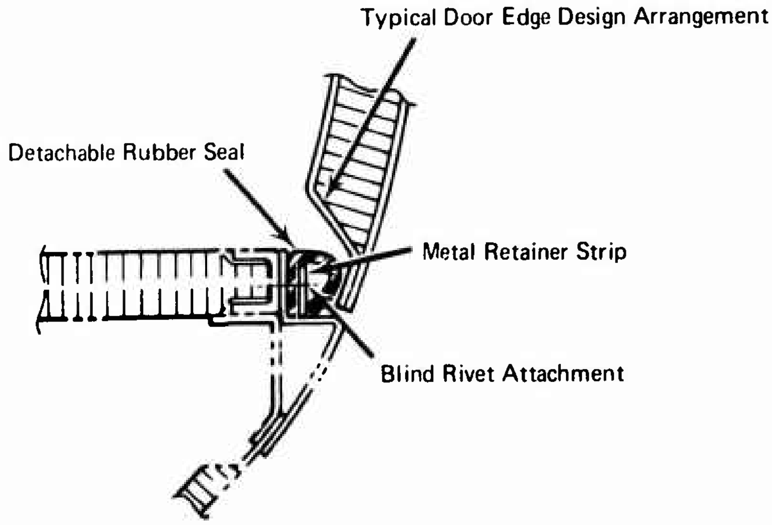


Figure 79. Airframe Module 11 - Cabin Sliding and Hinged Doors - Hinged Cabin Door. (Sheet 2 of 3)





SECTION A–A



SECTION C–C

2

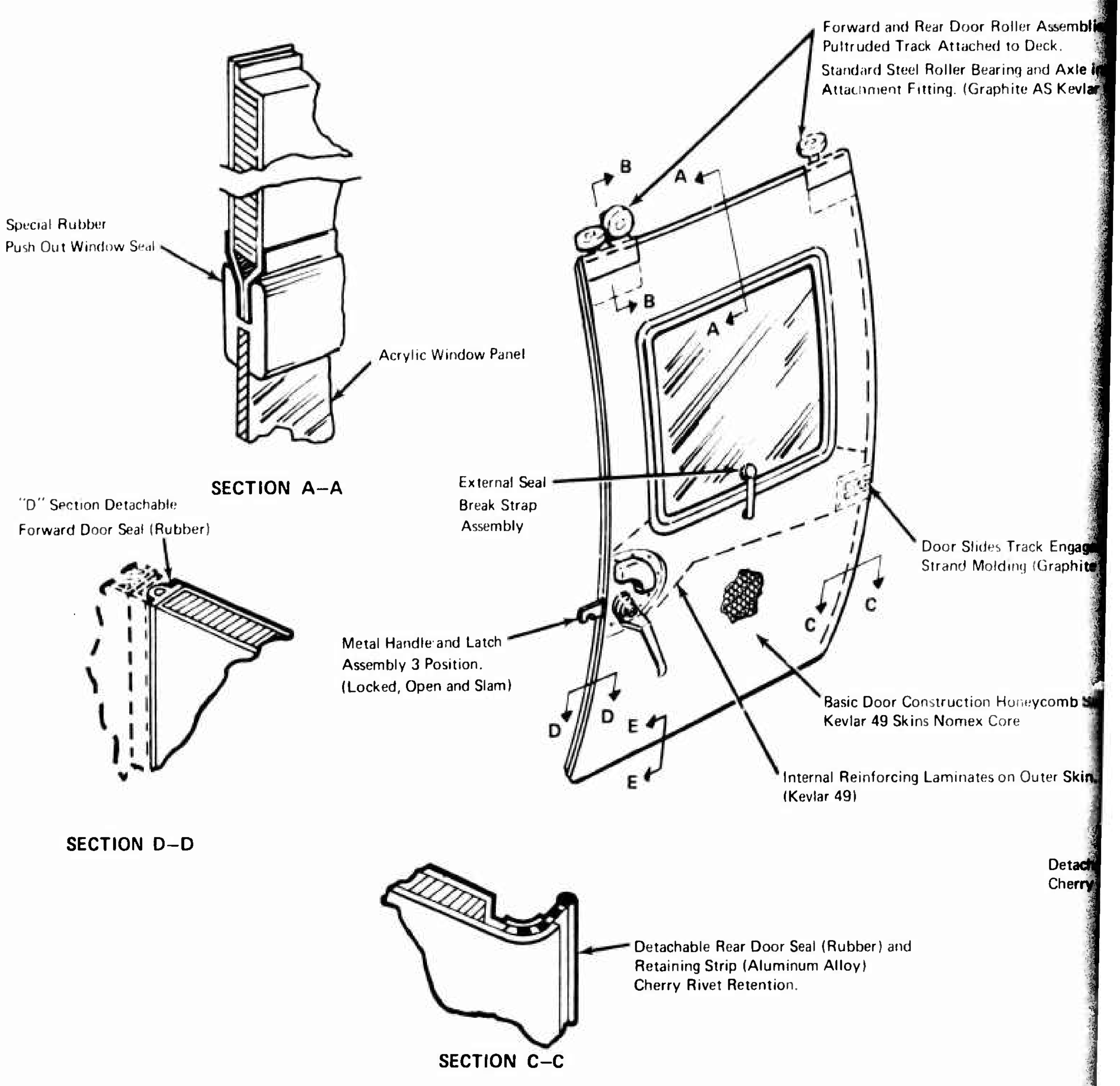
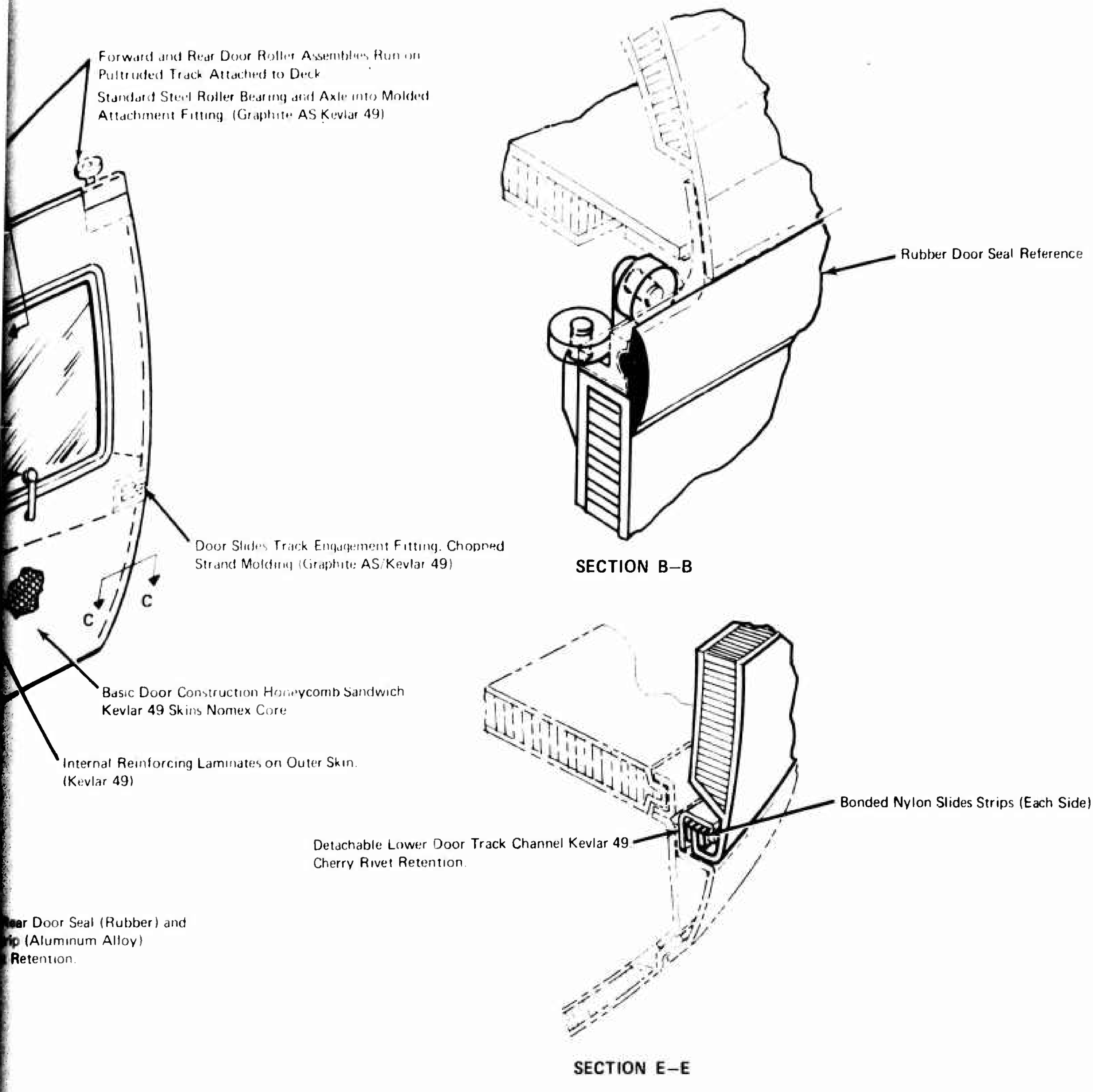


Figure 79. Airframe Module 11 - Cabin Side and Hinged Doors - Sliding Door Assembly. (Sheet 3 of 3)


Forward and Rear Door Roller Assemblies Run on Pultruded Track Attached to Deck
Standard Steel Roller Bearing and Axle into Molded Attachment Fitting. (Graphite AS Kevlar 49)
Rubber Door Seal Reference
Door Slides Track Engagement Fitting. Chopped Strand Molding (Graphite AS/Kevlar 49)
C
C
SECTION B–B
Basic Door Construction Honeycomb Sandwich Kevlar 49 Skins Nomex Core
Internal Reinforcing Laminates on Outer Skin. (Kevlar 49)
Bonded Nylon Slides Strips (Each Side)
Detachable Lower Door Track Channel Kevlar 49 Cherry Rivet Retention.
ear Door Seal (Rubber) and
ip (Aluminum Alloy)
Retention.
SECTION E–E

Doors -

7

The forward door is hinged along its forward edge and is easily detachable by removal of hinge pins. The door is essentially constructed of molded composite inner and outer skins, with Nomex honeycomb core used only in the area below the window, where an internal molded channel bonds to inner and outer skin and forms a close off for upper edge of core. The upper and lower hinges are molded chopped strand with unidirectional reinforcing fibers; they are of gooseneck configuration to confer a flush outer surface. The hinge anchorage fittings are of similar construction to the hinges and are bolt-attached inside aircraft contour onto the rollover frame web.

The removable window is an acrylic transparency and is the fixed type with a rubber seal. A rubber seal strip is attached along the top of the door (side seals are included on other modules), and a detachable seal is located at the lower edge of the door attached to the floor module deck.

The door handle and latch is a metal assembly with a three-position setting, open, slam, and locked. The hinged door locks onto the sliding door.

The sliding door is located directly aft of the hinged cabin door on each side of the helicopter and moves aft on a track and roller system. The door is easily removed by unbolting the rear door stop fitting and sliding the door off its tracks.

The single-curvature door is of simple honeycomb sandwich construction comprising an inner and outer skin bonded to a Nomex core.

A large emergency push-out window with a peripheral rubber seal is incorporated in the upper segment of the door. Internal reinforcing laminates stiffen the door at the handle and latch and slider fitting just below the window.

At the two top corners of the door are located composite roller fittings which are bonded to the door panel. Removable cam-follower-type nylon and metal rollers are attached to these fittings. The metal door slider fitting is bolt-attached to the rear inner face of the door midway up and serves as a combined fore and aft slider and door lateral retainer. The lower door edge is tapered down to a single thick skin which is molded into a channel section (facing inboard) and has nylon slider strips bonded to the upstanding vertical flange of the molding.

Upon assembly of the door to the fuselage, this lower door edge slides along a detachable molded retention channel which is blind-rivet-attached to the floor module while the upper roller system moves in the integral track of the upper deck assembly, Module 3.

Detachable rubber door seals are located at forward and rear vertical door edges. The metal door handle and latch assembly is bolted to door-edge combined-thickness skins and is operable from inside or outside.

4.3.6.12 External Skin Protection

For improved protection of all exterior surfaces of primary structure skin (scratches, erosion, etc.), an additional bonded scuff ply of Style 120 Kevlar 49 could be added, conferring a weight penalty of 12 pounds. No increase in panel strength or stiffness is assumed due to this ply; however, it is anticipated that a small increase in skin impact resistance will result.

This additional skin would also serve as a suitable covering media for the wire-mesh lightning-protection system mentioned elsewhere.

4.3.7 LANDING GEAR

4.3.7.1 Design

The landing gear is a tricycle type with the main gear attached at the aft end of the cabin and the nose gear mounted near the forward end of the cockpit. The structural arrangement of the main landing gear is shown in Figure 80. The main leg of each gear is a two-stage air/oil shock strut. These stages provide for normal sink speed landings of 10 fps and for hard landings up to 20 fps. Both shock struts may be compressed to a kneeling position (for transportation in the C-141) by a valve using an external hydraulic system (ground equipment).

Each main gear is mounted from three hard points on the primary airframe structure. One is the upper shock strut connection at sta 166 and WL 176 where a metal attachment fitting transfers vertical loads into the bulkhead and its stiffening beams, the longitudinal loads into the two deck beams at BL 16 and BL 22, and lateral loads into the honeycomb sandwich deck at WL 179 and the upper bulkhead at sta 163.

The other two lower support points are at BL 23 and BL 33 and WL 118 and sta 166, where a wide metal attachment fitting spans both underfloor (buttline) beam connection points and supports the landing gear trunnion, with cantilever lugs containing heavy-duty bearings which allow rotary motion due to the trailing arm gear movement. Vertical and side loads for the lower connection are taken by the bulkhead and reinforcing beams and angles on its rear face. Lateral shears and torsional loads are redistributed from the bulkhead into the deck floor, fuel cell walls, and torque box structure above, while lower attachment drag loads are taken by a long extension lug of the metal fitting which extends forward and picks up BL 35 underfloor structure beam.

The structural configuration of the nose gear is shown in Figure 81. The gear is a single nonretractable air/oil type shock strut with two wheels. The gear is able to swivel 360° and incorporates a viscous shimmy damper and swivel lock. A tiedown shackle and towing capability, by a lug at the axle, are provided. The strut is mounted at two points; the upper attachment is a flat fitting attachment to the cockpit floor panel at WL 127.5, which is capable of transferring load only in its own plane at WL 127.5. The lower attachment is made to a metal fitting attached to the underfloor frame at the sta 55 web. This attachment is a pin jointed design which takes vertical, lateral, and drag loads. The latter loads are transferred into the underfloor beam at centerline by an integral lug of the metal fitting extending rearwards.

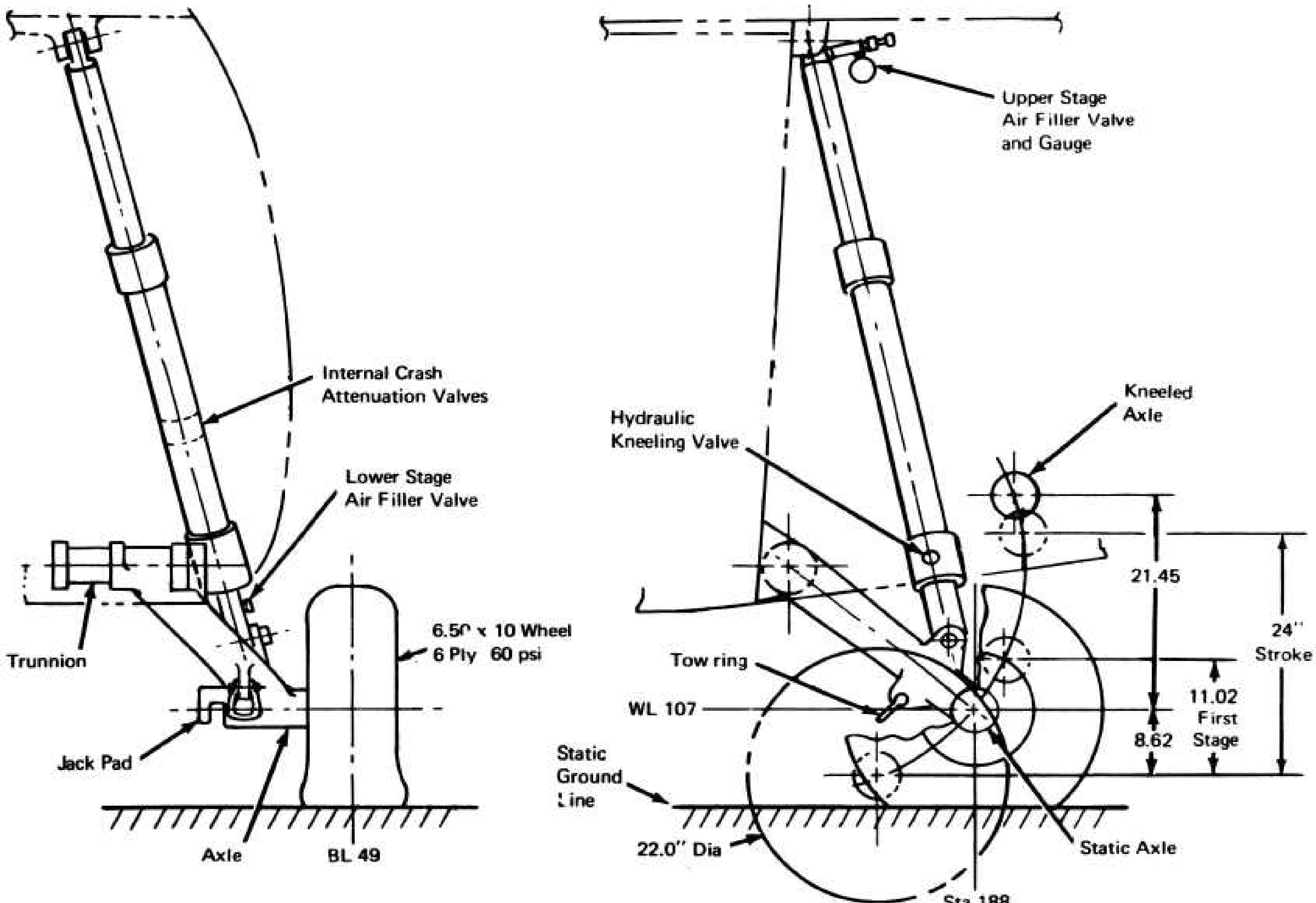


Figure 80. MUT - Main Landing Gear.

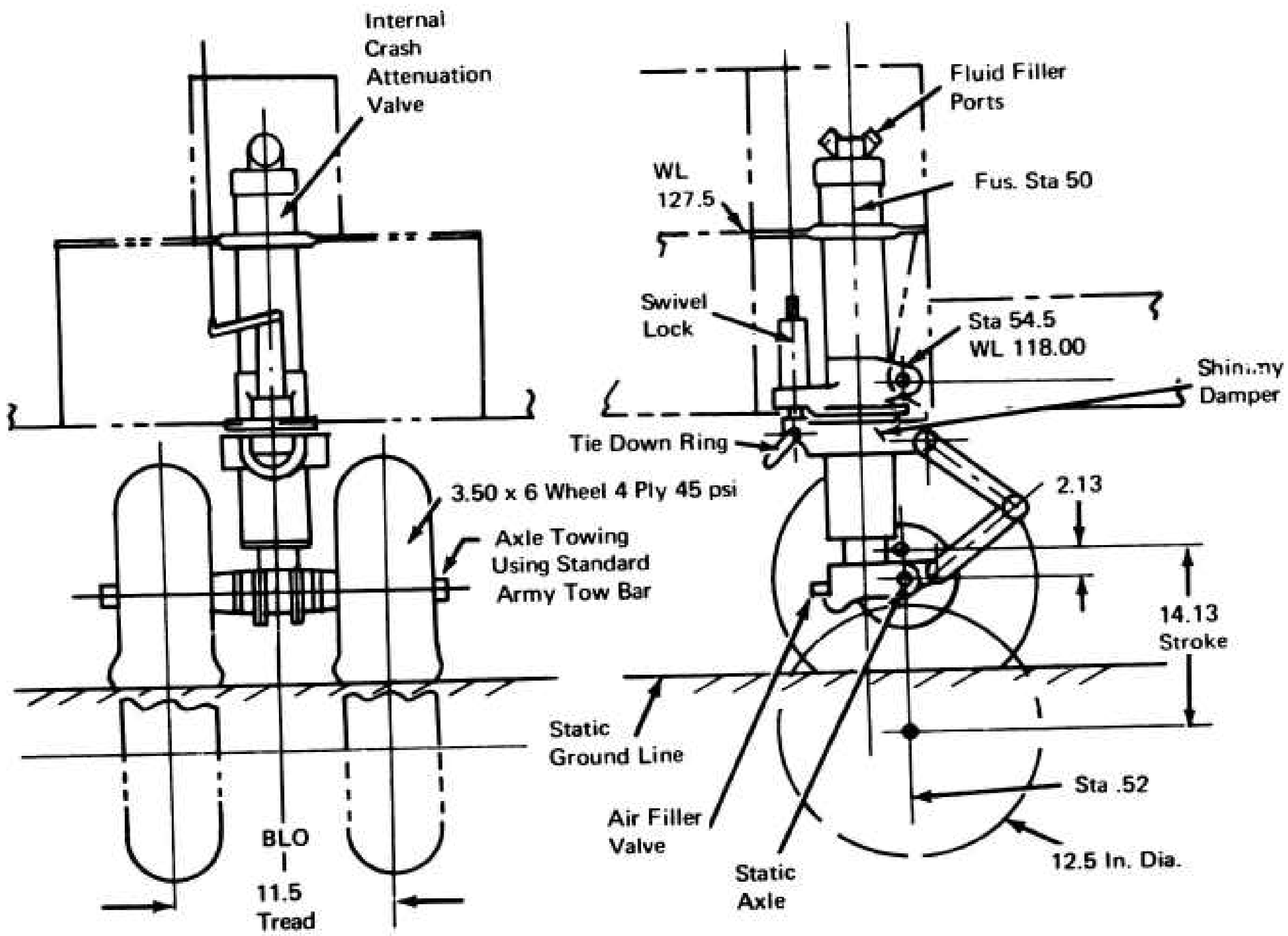


Figure 81. MUT Nose Landing Gear.

4.3.7.2 Aircraft Handling

Towing provisions are designed in accordance with MIL-STD-805 to permit towing under field conditions of CBR 2.5.

4.3.7.3 Construction

Landing gear components including the side braces, torque arms, and drag struts have been fabricated in boron/aluminum, boron/epoxy, and graphite epoxy. These components are not considered high risk. The oleo strut assembly has been fabricated in graphite epoxy with moderate success, but further development is required before the risk is acceptable for implementation on the MUT.

The weight savings available is estimated at 30 percent for a graphite/epoxy trailing arm assembly of the main landing gear (excluding trunnion, bushings, and bearing weight). Shock strut tubular members (excluding internal and external fittings) are estimated to yield a 40-percent weight savings in either boron/aluminum or graphite/epoxy. Axles, wheels, tires, and brakes are not considered in weight trades.

Total landing gear weight on the baseline MUT is estimated at 296 pounds. Of this, the trailing arm assemblies on the main gear weigh 12.2 percent, or 36 pounds, and the tubular members of all shock strut assemblies weigh about 18.5 percent, or 54.75 pounds. Thus, the net savings available with composites are:

Trailing arm assemblies 0.30 x 36	= 10.8 pounds
Shock strut tubular members 0.40 x 54.75	= 21.9 pounds
	32.7 pounds

The recurring cost of these components in a production run may show competitive composite costs at about 500 production units, although the cost data available is sketchy.

From the study materials on landing gear application to the MUT and other aircraft, it would appear that lighter weight composite landing gear can be competitive or even show reduced costs over conventional gear in prototype or small quantity production, where the cost of die forgings is not warranted for metal components. Larger helicopters with larger hogged-out components are good composite applications, with the Army heavy-lift helicopter an extreme example of small quantity payoff in both weight and cost. Shorter lead times also can be obtained with composites.

Composite landing gear components for production quantities of the MUT, while contributing to structural efficiency, are not considered cost competitive, and hence are not recommended at this time.

4.4 AIRFRAME STRUCTURAL ANALYSIS

The analysis of the selected structural configuration consists of the determination of external and internal loads and detail sizing consistent with preliminary design techniques. A few critical loading conditions were examined and applied to selected structural elements so that detail problems (such as minimum gauge and buckling) could be assessed and solutions proposed.

4.4.1 EXTERNAL LOADS

The loading conditions that were used are those typically critical for helicopters of this class and size. No attempt was made to examine all the possible critical conditions for every structural element. The overall loading conditions chosen for analysis, all at a baseline design gross weight of 9515 pounds were:

a. Symmetrical dive and pullout:

$N_z = 3.5g$

$\dot{\alpha} = 3.27$ rad/sec

b. Vertical takeoff:

$N_z = 3.5g$

$\dot{\alpha} = 0.0$ rad/sec

c. Tail bumper:

Sink Speed = 18 ft/sec

$\alpha = 19.5$ deg (nose up)

d. Nose gear impact:

Sink Speed = 15 ft/sec

$P_r = 23{,}850$ lb (effective load at gear)

The overall shear, bending moment, and torsion loads were derived for the first three conditions using Boeing Vertol computer program S-06, the results of which are plotted in Figures 82 through 86.

The loads for mass retention during crash conditions result from the following combination of load factors applied to such items as engines, transmission, rotor, and fuel.

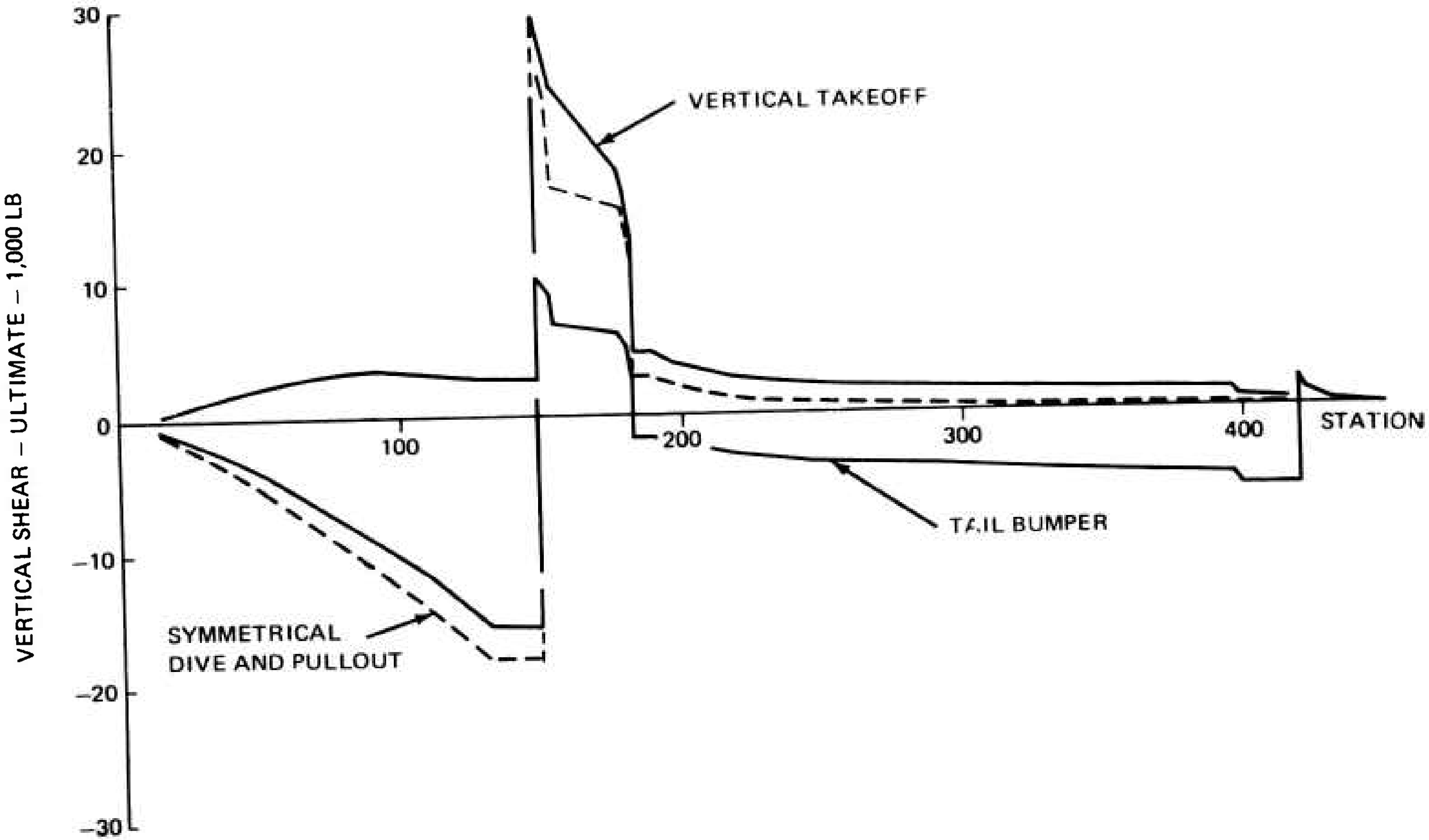


Figure 82. Fuselage Vertical Shear.

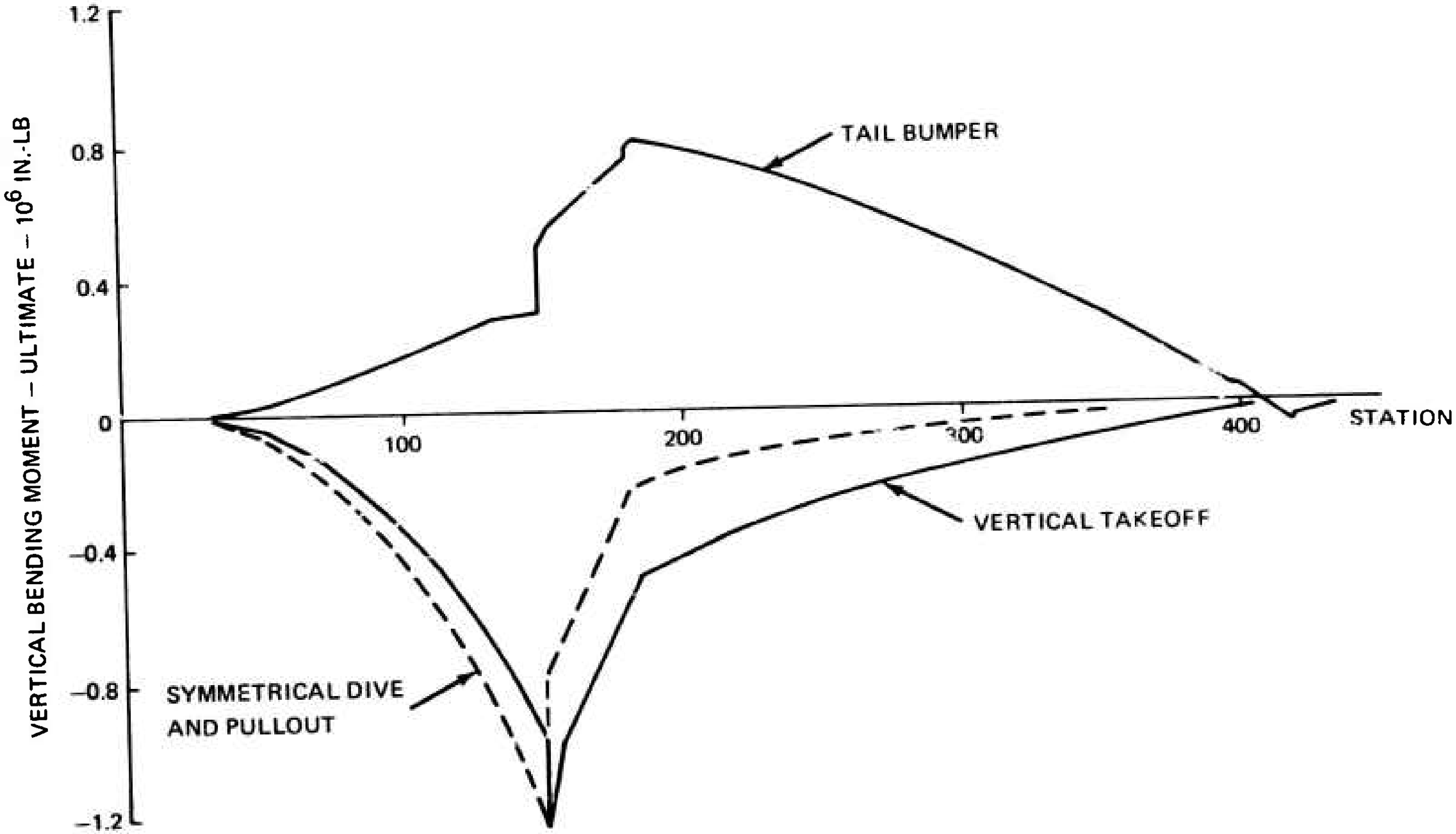


Figure 83. Fuselage Vertical Bending.

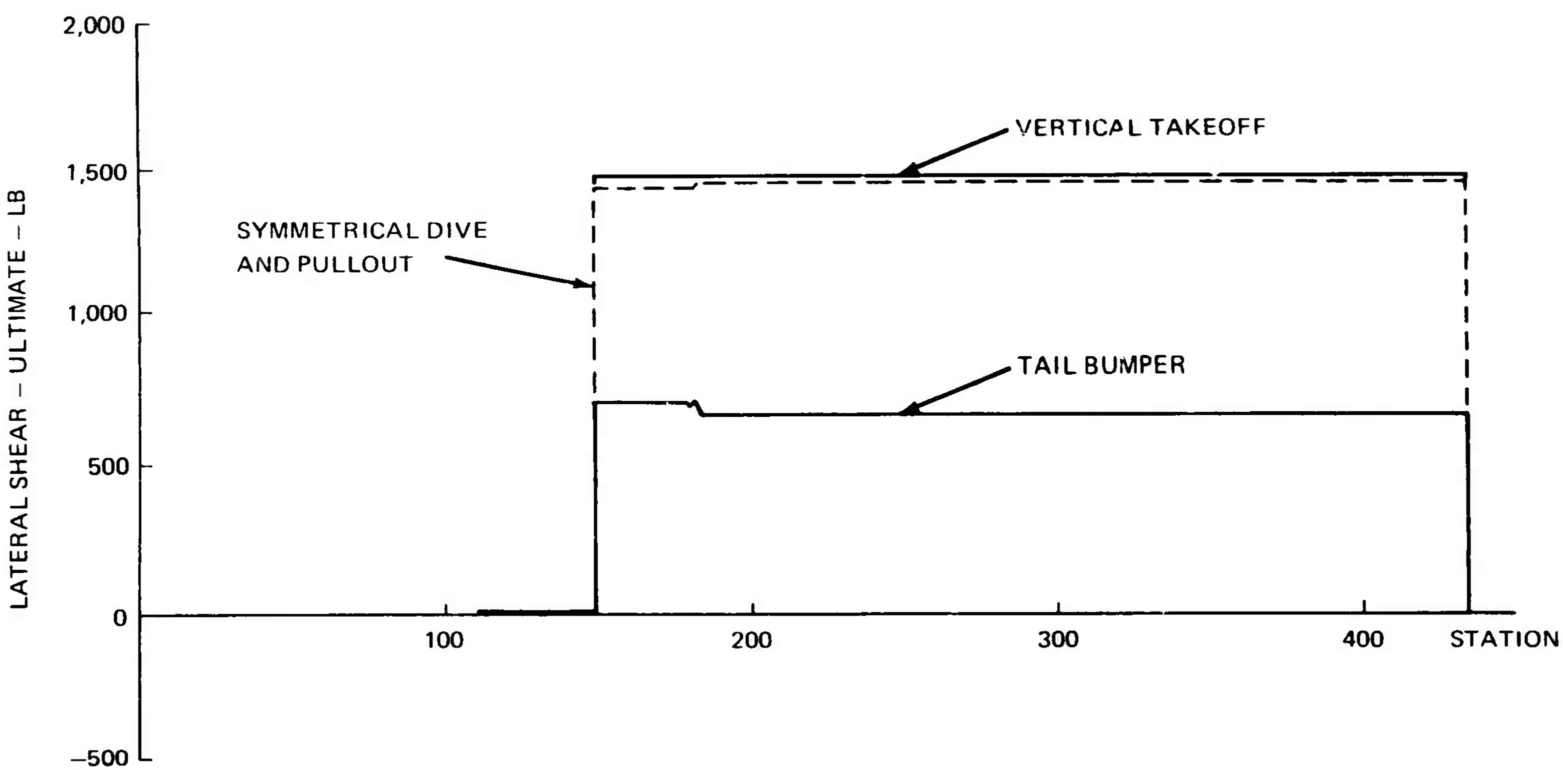


Figure 84. Fuselage Lateral Shear.

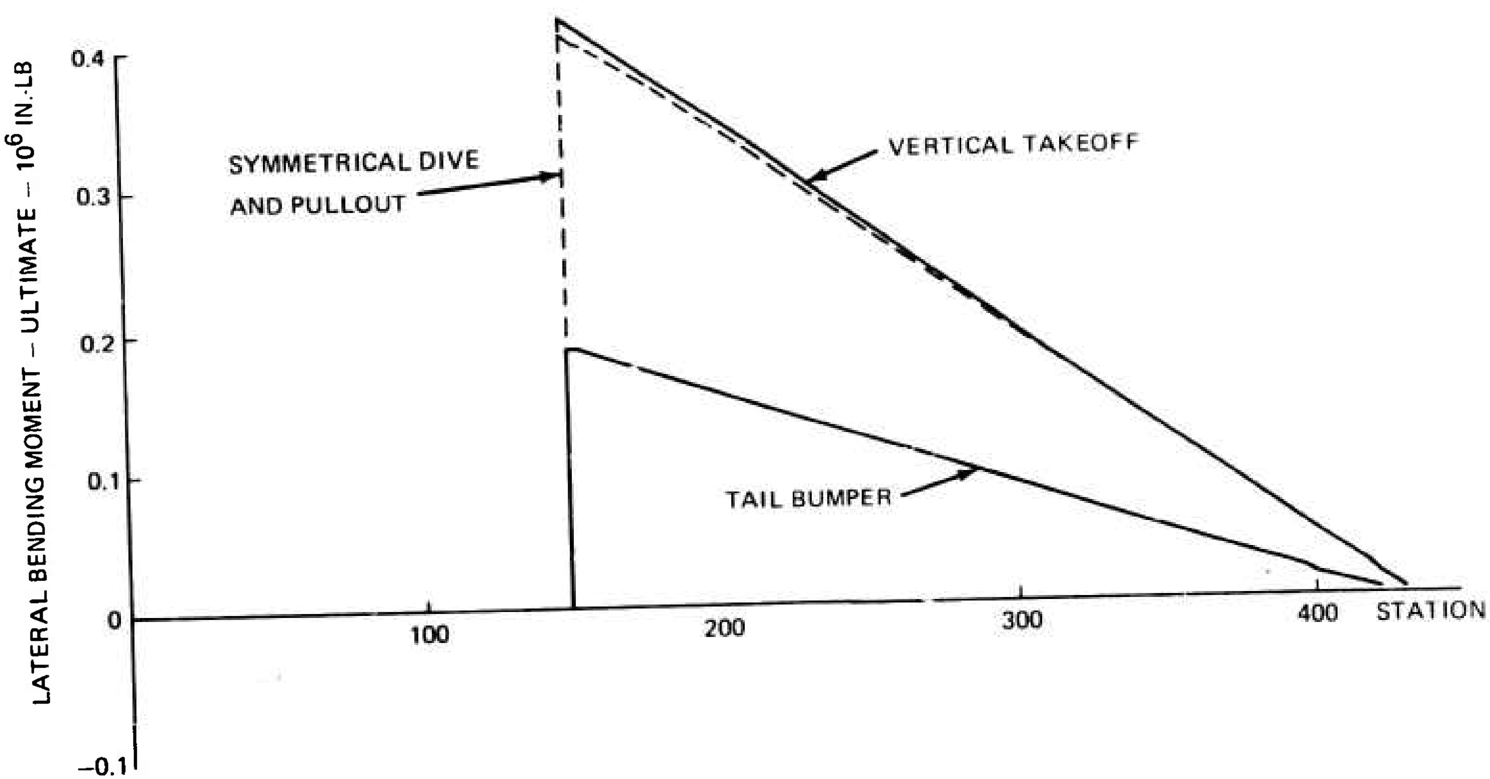


Figure 85. Fuselage Lateral Bending

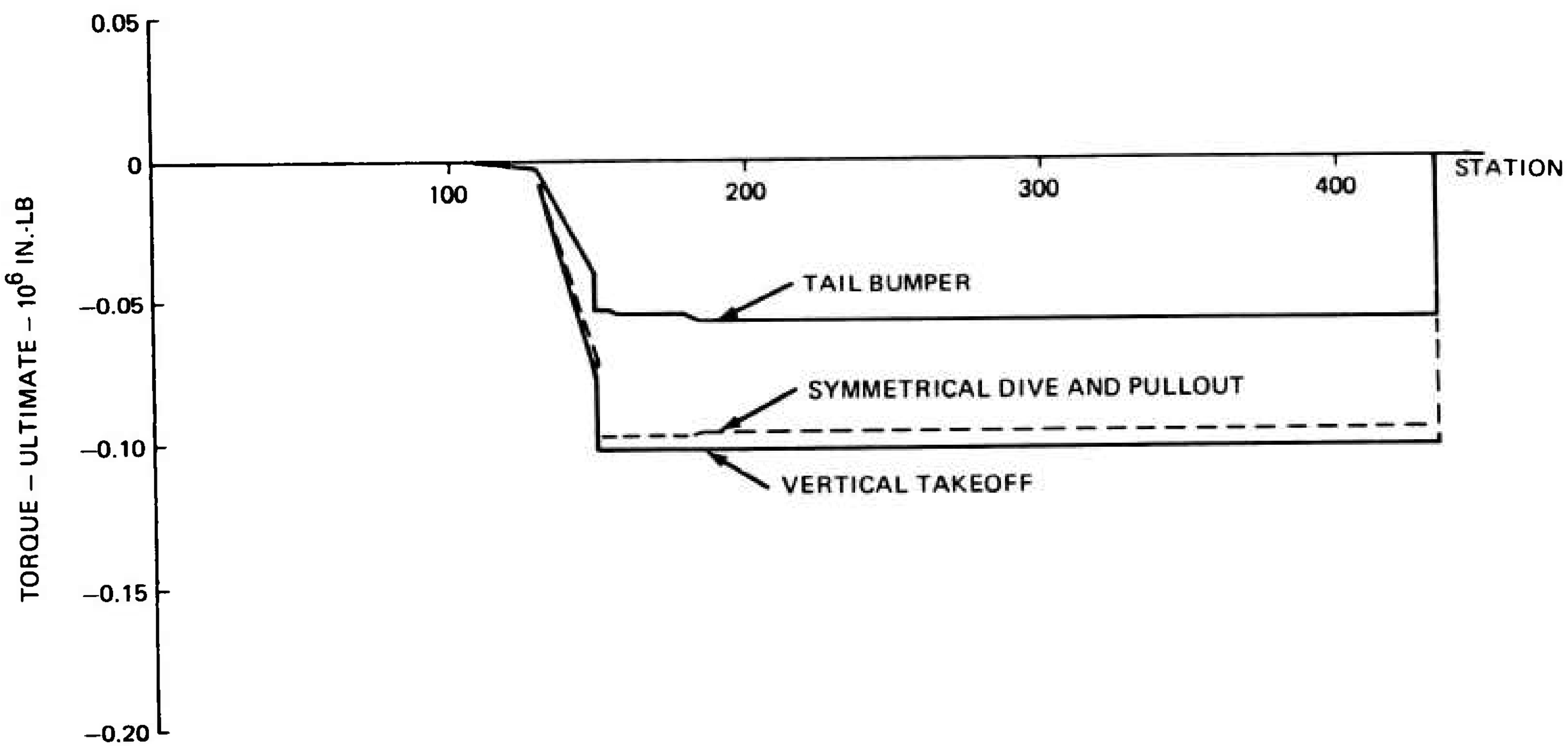


Figure 86. Fuselage Torque.

Direction	Load Factor (g) Separately (a)	Simultaneously (b)	(c)	(d)
Longitudinal	±20	±20	±10	±10
Vertical	+20/-10	+10/-5	+20/-10	+10/-5
Lateral	±18	±9	±9	±18

4.4.2 INTERNAL LOADS

The load curves have limited utility since the cabin section carries loads not in the normal MC/I and VQ/I fashion (as in the tailboom section) but as a fixed-beam truss work, with bending occurring in each of the framing structures. The structure was modeled for a two-dimensional simplified NASTRAN computer program (Boeing Vertol Program S-80) so that load distribution to the crown and floor structure could be determined. The loads applied to the model are those from the S-06 program. The results for conditions (a) through (d) are shown in Figures 87 through 90, which also show the simplified structural model. Each of the 16 general beam elements is capable of carrying all loads in a two-dimensional system (torsion excluded). A shear element was used in the section between sta 163 and sta 239, representing the fuel cell sides.

4.4.3 DETAIL ANALYSIS

4.4.3.1 Structural Analysis Considerations in Configuration Trades

A helicopter of this size and weight produces loads in the shell structures of relatively low intensity. In many areas this results in strength requirements for material below what may judiciously be selected as a minimum gage for both handling and in-service exposure. Therefore, minimum gages (or number of plies) will be sufficient in many areas. The choice of sandwich construction in these areas makes the penalty for minimum gages worse since two faces are needed. Another choice is to use skin/stringer construction. Thin gage skin/stringer aluminum alloy riveted structure has been proven to be efficient and structurally adequate when well past buckling for both shear and compression loads. In laminated fibrous composites, the performance of highly buckled skin panels, either riveted or bonded together, has not been adequately demonstrated, particularly for the vibratory environment experienced in helicopters. Some development test programs for fixed-wing primary structure have demonstrated the adequacy of composites in an elastically buckled state. The maximum stresses in these cases have exceeded the critical buckling stress (σ/σ_{cr}) by

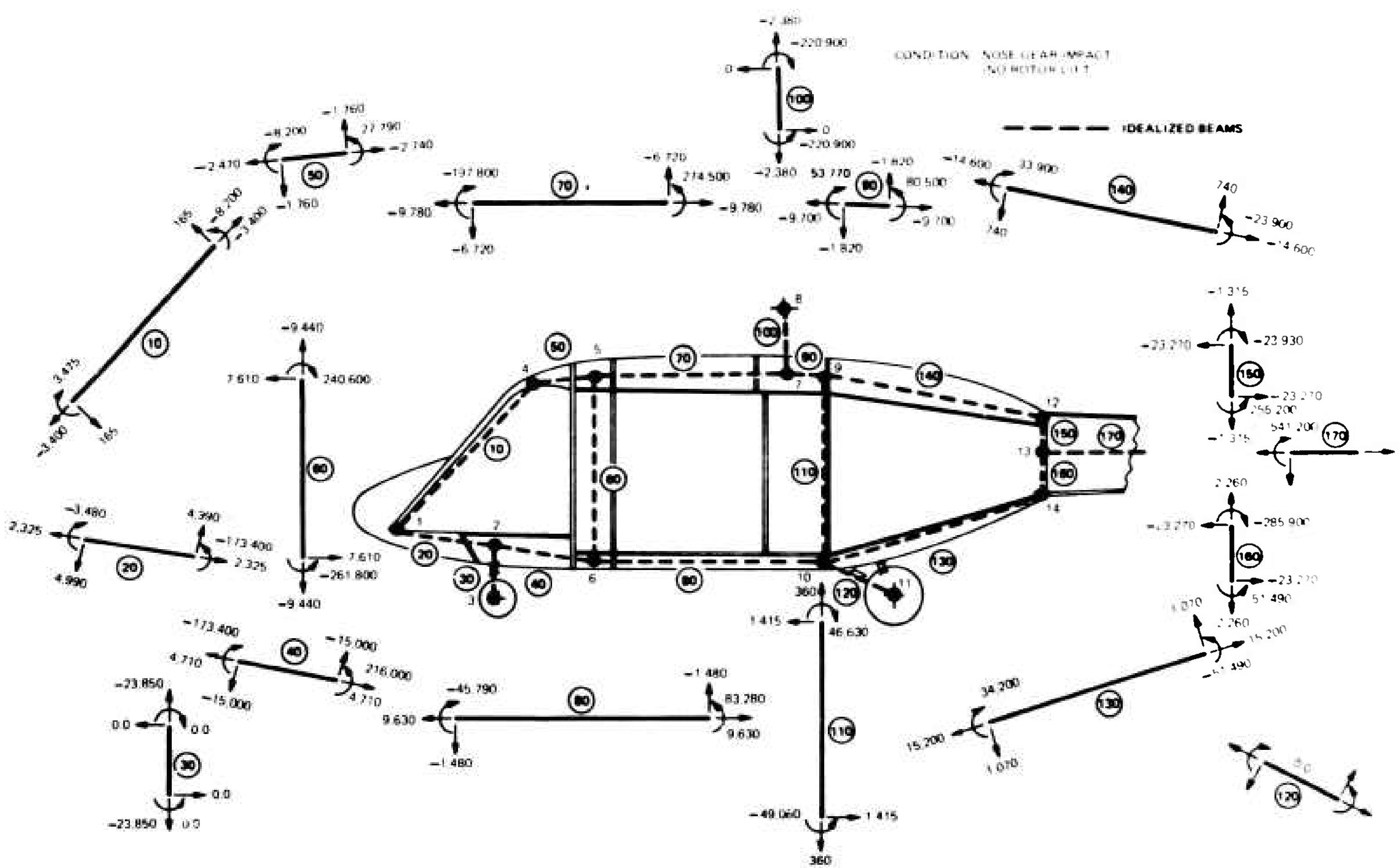


Figure 87. Internal Loads for Nose Gear Impact Condition.

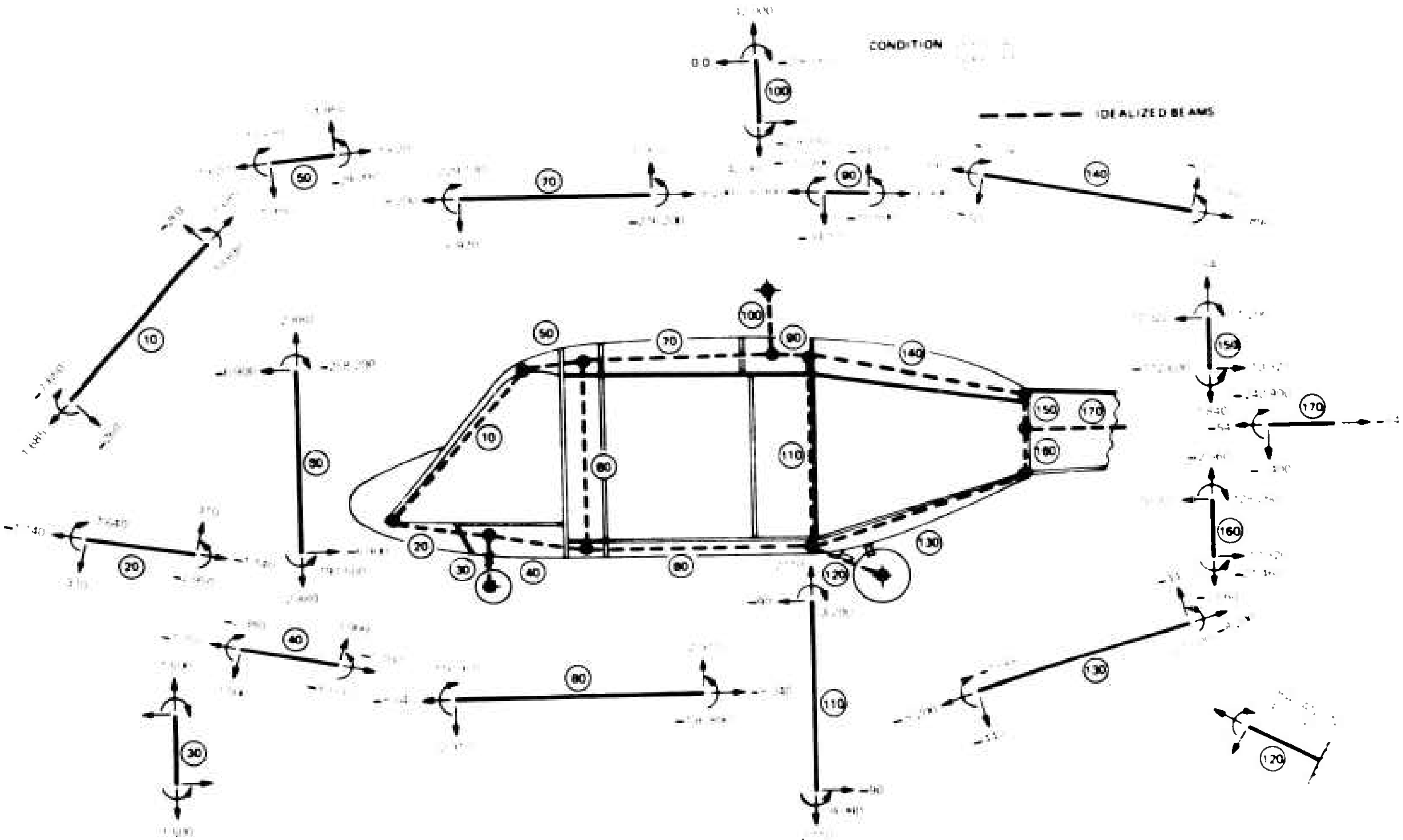


Figure 88. Internal Loads for Vertical Takeoff Condition.

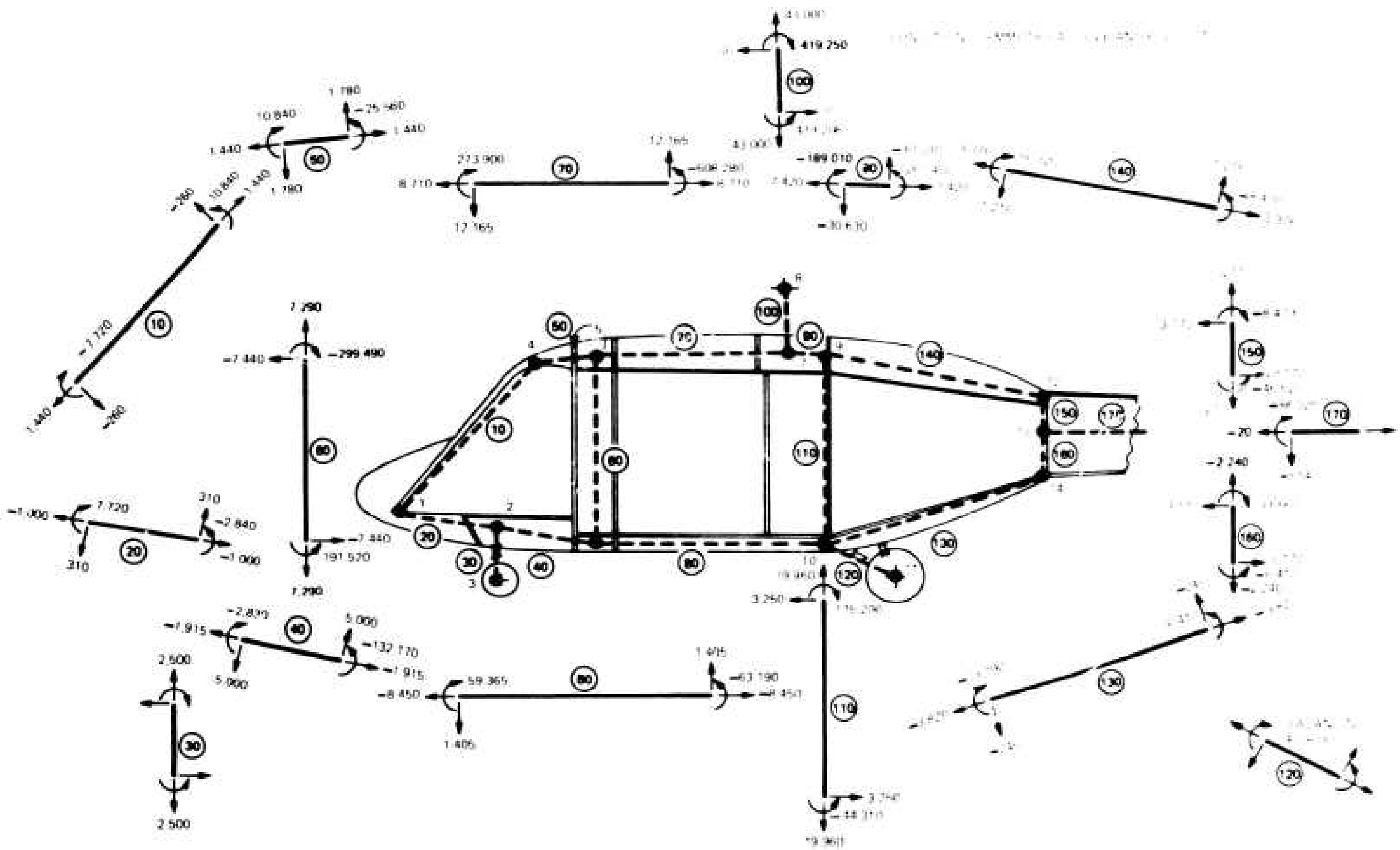


Figure 89. Internal Loads for Symmetrical Dive and Pullout Condition.

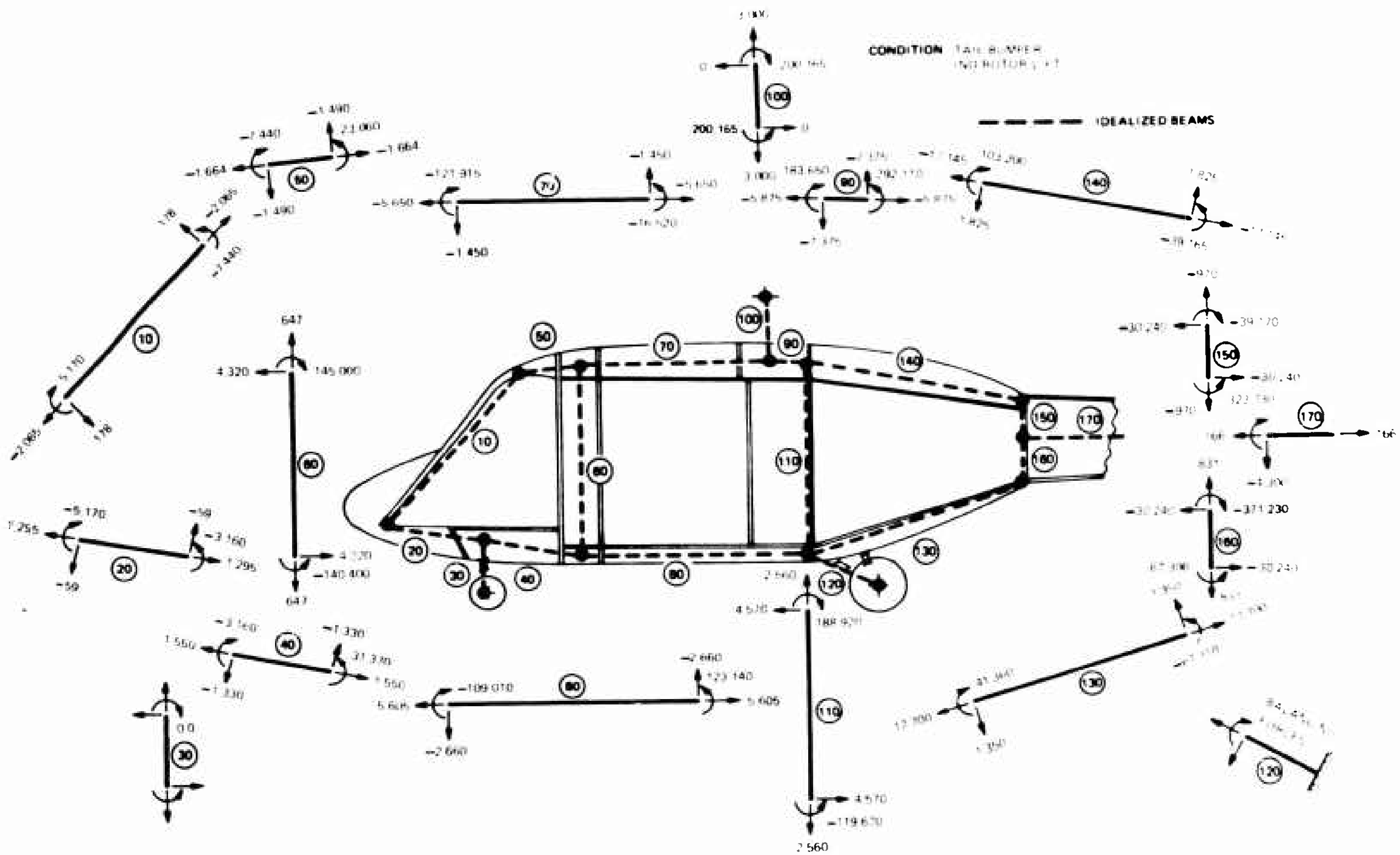


Figure 90. Internal Loads for Tailbumper Impact Condition.

values of 5 to 8, but certainly not to the extent that aluminum alloy sheet is capable of (where σ/σ_{cr} values of 15 to 20 may be achieved for maximum efficiency). The primary concerns in bonded laminated composites are the unknowns associated with repeated static-load-induced elastic buckling and the superimposition of fatigue loads on the laminate itself, and the resulting peeling-type loads on the bond joint which resists the sheet buckling at frames and stringers. Without an extensive test program, the limits to which buckling may be permitted in sheet-stringer construction may only be estimated. For the trade study performed for the tailboom, skin buckling was limited to τ/τ_{cr} = 2.5 at design limit load, a level at which the above mentioned concerns are felt to be of minor risk.

Detailed analyses were performed on selected structural items so that the impact of the application of advanced composite materials could be assessed. Included in Table 17 is a summary of the major airframe structural elements and the loading conditions which are critical or affect the design. The items which were analyzed are identified as well as those whose design is based on concepts selected and proven for the YUH-61A.

4.4.3.2 Structural Analysis Considerations in Materials Selection

The advantages of composite materials when compared with the conventional aircraft structural materials have been stated in many ways. The most apparent comparisons can be made for the mechanical properties which are summarized in Table 18. Environmental resistance, cost, and design and manufacturing flexibility are some of the other properties which may be advantageously exploited. The glass fiber composites have excellent strength and impact properties, are low in cost, but are very low in modulus. Composites of high modulus fibers have excellent static and fatigue strengths and stiffness properties, but some (boron and high-modulus graphite) remain quite expensive and most are susceptible to low-energy impact damage. Composites using the Kevlar 49 fiber possess a medium level of stiffness and relatively good impact resistance, but suffer from a low compression strength. No one fibrous composite either woven or nonwoven stands out as the panacea, but each has its place in aerospace structures. Each individual structural element or component with its strength, stiffness, and exposure requirements must be examined so that a single material or hybrid combination of materials may be selected as the optimum or near optimum for that specific application.

The structure for a MUT type helicopter is, in general, loaded to low intensities, subject to a vibratory fatigue environment, and is exposed to personnel and equipment knocks and bumps during service and maintenance. The structural concept chosen for most of the airframe is honeycomb sandwich. Nomex is the

TABLE 17. SUMMARY OF CRITICAL CONDITIONS FOR AIRFRAME STRUCTURAL ELEMENTS

Airframe	Critical Loading Condition						
	Crash			Flight	Ground Loads		Miscellaneous
	Crash-worthiness; Personnel Protection	Mass Reten-tion	Fuel Contain-ment	Air Load and Inertia	Gear	Tail Bumper	Cargo Tiedown; Seats; Airloads; Personnel, etc.
• Cockpit (c)	✓(a)	-	-	-	-	-	✓
• Main Fwd Frame & Posts, sta 80-91	✓(b)	-	-	-	✓	-	-
• Cabin Floor Panels & Substructure (c)	✓	-	-	-	✓	-	✓
• Crown Beams							
sta 91-163	-	✓(b)	-	-	-	-	-
sta 163-239	-	✓	✓	-	-	-	-
• Top Deck (c)							
sta 91-163	-	✓	-	-	-	-	-
sta 163-239	-	-	✓	-	-	-	-
• Bulkhead, sta 163	✓	✓	✓	✓	✓	✓	✓
• Fuel Tank	-	-	✓(b)	-	-	✓	-
• Tailboom	-	-	-	✓(b)	-	✓(b)	-
• Vertical Tail (c)	-	-	-	✓	-	-	-
• Horizontal Tail	-	-	-	✓(b)	-	-	-
• Secondary Structure; Doors, Fairings, Etc. (c)	-	-	-	-	-	-	✓

NOTES: (a) ✓ critical condition
(b) detail analysis for MUT
(c) Based on similarity with YUH-61A

TABLE 18. MATERIAL SYSTEMS PROPERTIES AND COSTS

$/LB				[0] Laminate										[±45] Laminate	
1974	1978	Material System	ρ (LBS/IN^3)	F_{t_u} (KSI)	F_{t_u}/ρ (10^{3}IN.)	E_t (MSI)	E_t/ρ (10^{6}IN.)	F_c (MSI)	E_c/ρ (10^6 IN.)	F_{cu} (KSI)	F_{cu}/ρ (10^{3}IN.)	G (MSI)	G/ρ (10^{6}IN.)
1.20	1.61	2024 T3 Aluminum	.100	66	660	10.5	105	10.7	107	40	400	4.0	40
.20	.27	4130 Steel	.283	150	530	29.0	102	29.0	102	145	513	11.0	39
16.00	21.40	6AL 4V Titanium	.160	130	813	16.0	100	16.4	102	126	788	6.2	39
.74	.99	18-8 Stainless Steel	.286	185	647	26.0	91	26.0	91	98	342	11.0	38
17.00	21.50	181S Fabric/Epoxy	.065	76	1170	4.0	62	4.0	62	65	1000	1.4	22
2.35	3.44	1002E Glass	.065	160	2460	5.7	88	4.6	71	86	1320	1.7*	26
17.60	20.60	1002S Glass (901)	.066	219	3320	6.3	95	6.0*	91	110	1670	1.7	26
8.50	7.50	SP250-SF-1(1014)	.068	180	2650	6.3	93	6.0*	88	110*	1620	1.8	27
6.20	7.00	1002S Glass (S2)	.066	180*	2730	6.3	95	6.0*	91	110*	1670	1.7	26
50.00	20.00	Graphite Epoxy (AS)	.055	186	3380	18.5	335	16.0	291	165	3000	4.5	82
75.00	45.00	Graphite Epoxy (HMS)	.056	102	1820	26.0	465	24.0	429	117	2090	6.5	116
65.00	35.00	Graphite Epoxy (HTS)	.056	200	3580	20.0	357	20.0	357	165	2950	5.5	98
20.00	10.00	Kevlar 49 Epoxy	.050	200	4000	11.0	220	11.0	220	40	800	3.0	61
100.00	85.00	Boron Epoxy (5.6 MILS)	.075	209	2790	29.0	390	29.0*	390	460*	6140	7.9	105
70.00	25.00	Polysulfone (AS Graphite)	.069*	155	2250	13.7	198	14.0	203	150	2180	3.6*	52
21.50	16.60	181 Kevlar 49 Fabric/ Epoxy	.050	75	1500	4.5	90	4.5	90	25	500	3.0	60

Notes:

1. Composite Strengths are Mean of Data.
2. Metals Strengths are B Basis.
3. Metals Compression Strengths are Yield Strengths.
4. Composite Costs are Prepreg Costs.
5. Cost Values are for Large Quantities (approximately 10,000 pounds).
6. All 1978 Costs include a 6% Yearly Inflation.
7. Asterisk denotes estimated value.

primary selection for the core for its corrosion and damage resistance and its relatively good tolerance of manufacturing dimensional variations. Aluminum core has been selected in a few highly loaded locations where core stiffness requirements make a Nomex core weight inefficient.

The low intensity loads result in thin face laminates to carry the loads. A minimum gauge of 3 plies (or 3 times 0.00525 = 0.0157 inch) has been selected in a (0/±45) lamination. The special properties of boron/epoxy (high modulus and high compression strength) and high modulus graphite are not necessary for the structures considered, and their costs are relatively high. The use of fiberglass epoxy is attractive from the cost, damage resistance, and damage tolerance viewpoints, but the stiffness and weight efficiency does not match the low modulus graphite (G-AS/E; T300/E, etc.*) or Kevlar 49 (K49/E) composites. The K49/E is desirable since its density is the lowest of all candidate fibers used in aircraft structure; its cost is moderate, and the impact resistance is better than the G-AS/E composite. The drawback for K49/E is its low compression strength.

The low-modulus graphite composites are in most cases the most structurally efficient, but are more costly and more damage susceptible than K49/E. A hybrid composite using both G-AS/E and K49/E to best advantage has been chosen for the majority of the airframe. The G-AS/E layers are used where stiffness, bearing, and compression strengths are necessary, mostly in the zero degree orientation. The K49/E layers are used in the ±45 degree orientations to carry the shear loads and to fill out the laminate to the 3-ply minimum previously established. The use of the K49/E keeps the material costs down and improves the impact resistance over an all G-AS/E laminate.

Another possible method of utilizing the positive aspects of a graphite/Kevlar hybrid is to mix the two fibers in the same layer--an intimate blend. It then would be possible to maximize the use of the lower cost material. This approach was not studied during this program. It is obvious, though, that the optimum amount of each fiber for weight and/or cost efficiency would differ for each specific application. Unless one mixture ratio would satisfy the needs for a majority of the structure, the cost of a number of prepreg runs, storage, coding, etc., would negate the possible advantages.

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*Note: The low-modulus, or what has been called intermediate strength graphite fiber, will be called G-AS/E, which does not mean to imply a specific fiber, but a generic one.

The properties of hybrids are such that when two different modulus fibers are mixed in a laminate, both in the same (0^o) direction, the resulting modulus is a linearly varying property with the fiber mixture ratio. The strength is usually compromised since both types of fiber cannot be loaded to their full strength at the same time. These property variations are shown for an HTS/K49 hybrid in Figure 91. When hybridizing with 0^o and 45^o directions, the unidirectional properties are dominated by the 0^o fibers and the shear properties by the angle ply fibers, with little or no compromising of properties. Charpy impact test results (see Figure 92) show the improvement in unidirectional laminates with increasing amounts of K49 fiber. Similar results could be expected with a multidirectional hybrid laminate. Ball drop tests on hybrid sandwich panels have also been performed and have shown improved damage resistance of graphite/epoxy laminates with the introduction of lower modulus materials such as K49 and fiberglass.

For the specific applications studied for the MUT airframe, it became obvious that a laminate using a hybrid of G-AS_0/K49$\pm$45 would be most weight and cost efficient. Point trade studies, discussed subsequently for the tailboom and upper deck beams, confirm the deductive conclusions reached from examining the materials properties alone.

4.4.3.3 <u>Tail Boom</u> (See Figure 74)

The tail boom has a modified circular cross-section tapering in diameter from the attachment to the mid-fuselage at sta 239 to the empennage. It is designed to support loads from the tail surfaces, tail rotor forces, and from the tail bumper, the latter being critical for the most part. Many studies have been performed on this type of low load intensity structure by Vertol and other airframe manufacturers for composite material applications. The number of candidate concepts have included:

- Thick honeycomb sandwich monocoque
- Medium honeycomb sandwich and longeron/frame semimonocoque
- Thin honeycomb sandwich and stringer/frame semimonocoque
- Skin/stringer/frame semimonocoque
- Open truss work

The composite materials considered have included fiberglass, graphite (of all moduli), boron, and Kevlar fibers in epoxy matrixes, and hybrids thereof. The net results have been relatively small differences in absolute weights, but significant

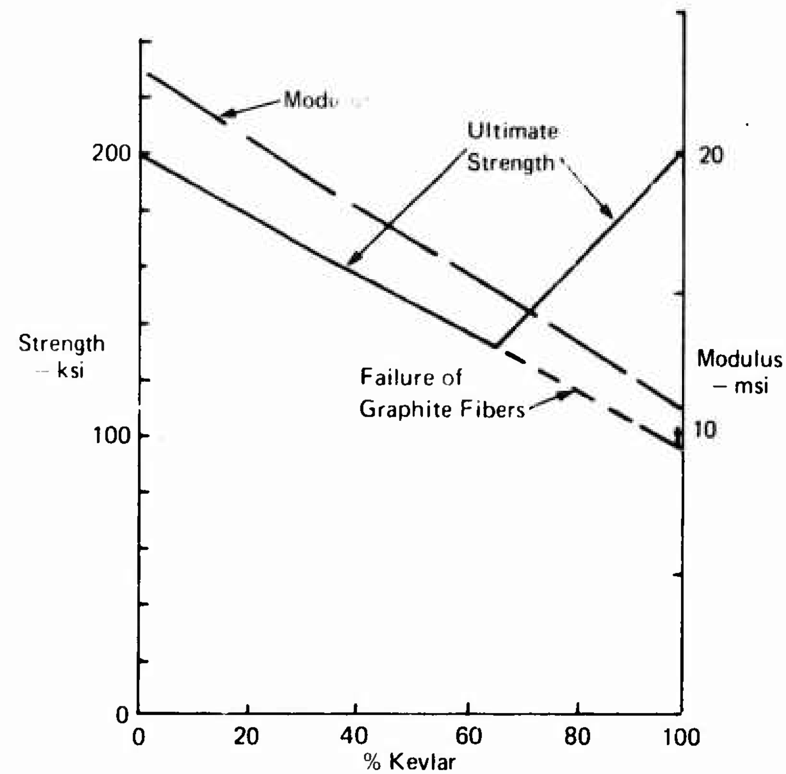


Figure 91. Strength and Modulus for Unidirectional HTS/Kevlar Hybrid Laminates.

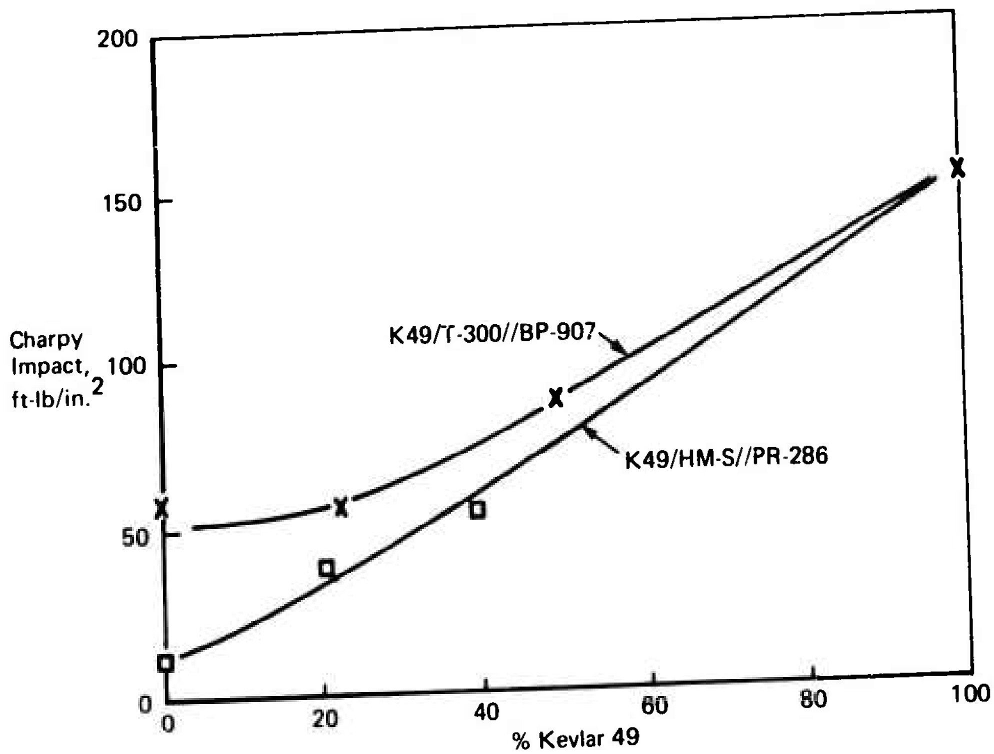


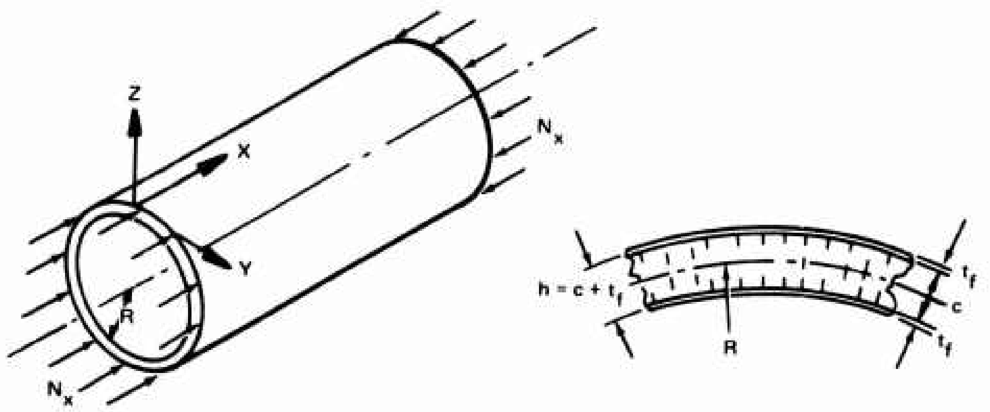
Figure 92. Charpy Impact Strength of Unidirectional Kevlar 49 Hybrid Composites.

when calculated as a percent reduction compared to metals--between 10 and 20 percent. Since weight is not the only parameter of interest, an analysis was performed to size and define the structure so that these other parameters (damage resistance and tolerance, cost, fail safety, etc.) could be assessed. From the results of previous work[3], it was determined that the most weight efficient structure should be a sandwich monocoque shell; therefore, this concept was analyzed. For comparison purposes a skin/stringer/frame concept was also sized.

For the sandwich monocoque it was determined that a minimum gage for the face sheets should be 3 plies thick (or 3 x 0.00525 = 0.0157 inch), with at least one unidirectional ply and one pair of ±45° angle plies. Nomex honeycomb is used for the core. The method of analysis used is that from Reference 4 for compression stability of a cylindrical honeycomb sandwich shell. Figure 93 shows the actual cross-section, and that used for the stability analysis. It was assumed that the critical compression loading $N_{x_{cr}}$ (in pounds/inch) would be the same for bending of the cross-section where $N_x = M/\pi R^2$. The critical condition is the tail bumper loading, where the resultant moment $\sqrt{M_z^2 + M_y^2}$ causes a fairly constant N_x due to the tapering shell, as seen in Figure 94.

The following is a simplified equation to predict the buckling stress of the sandwich cylinders.

$$(\sigma_x)_{cr} = \phi\,(\sigma_{cr})_{rc}\,\{1 - \tfrac{1}{2}\,[\phi\,(\sigma_{cr})_{rc} t_f c/G'_{cx}\, h^2]\}$$



SANDWICH CYLINDER LOADED IN UNIAXIAL COMPRESSION

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3 Study of Advanced Structural Concept for Fuselage, USAAMRDL TR73-69, Eustis Directorate, US Army Air Mobility Research and Development Laboratory, Fort Eustis, Va., Oct. 1973.

4 Advanced Composites Design Guide, Third Edition, Jan. 1973.

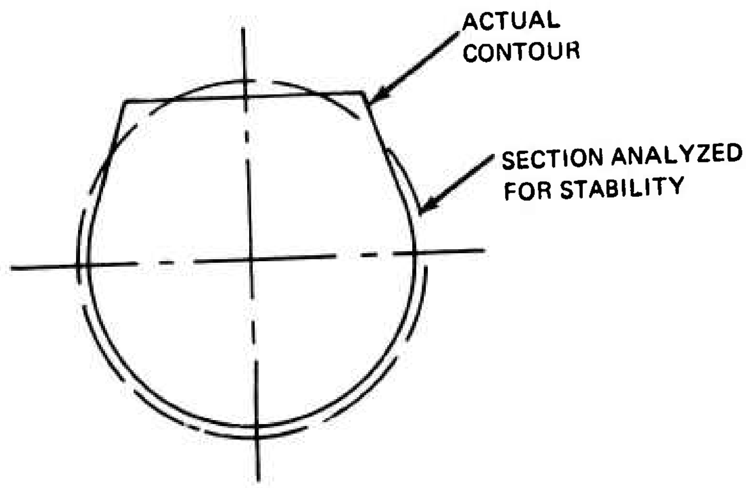
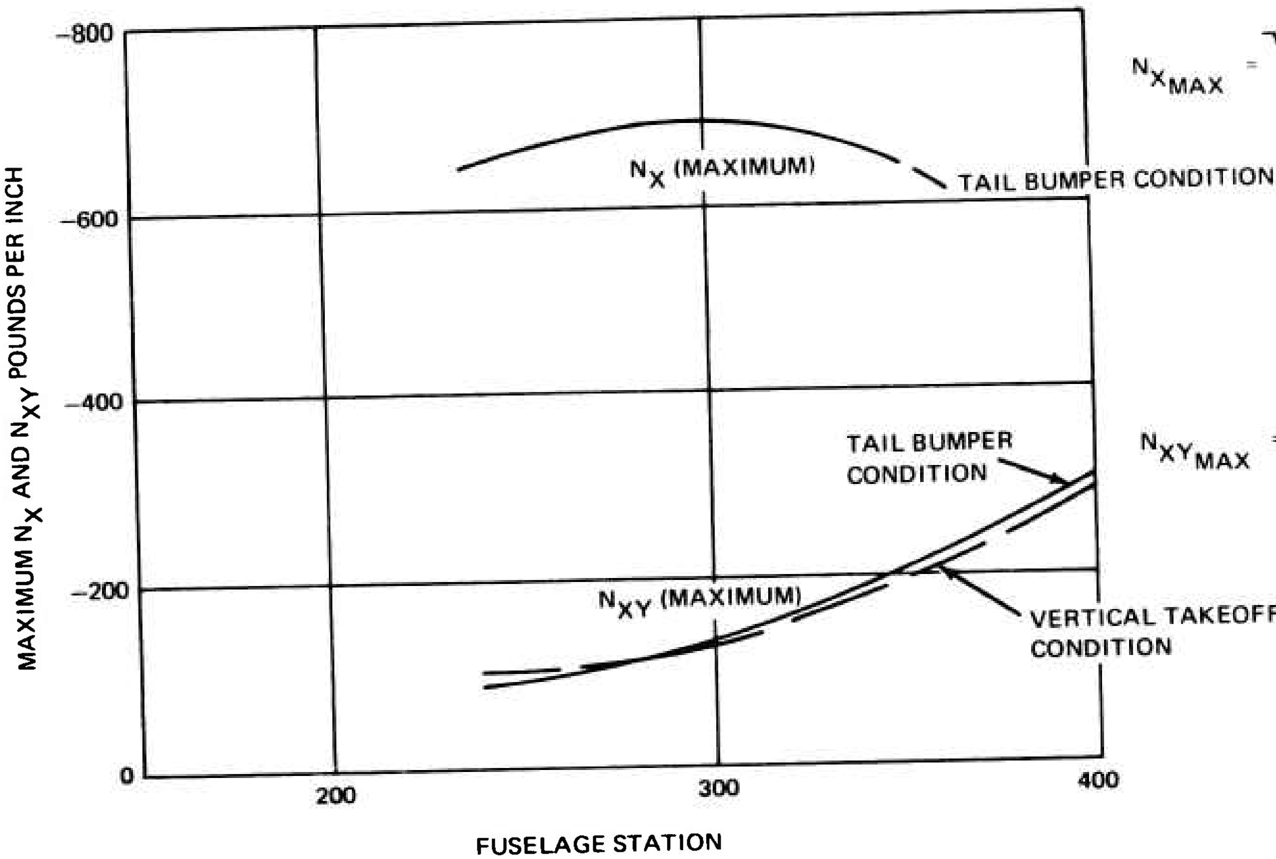


Figure 93. Tail Boom Section at Fuselage Sta. 239.



$$N_{X_{MAX}} = \frac{\sqrt{M_Z^2 + M_Y^2}}{\pi R^2}$$

$$N_{XY_{MAX}} = \frac{\sqrt{V_Z^2 + V_Y^2}}{\pi R} + \frac{M_X}{2 \pi R^2}$$

Figure 94. Maximum Compression and Shear Loads on Tail Boom Shell.

where ϕ equals the smaller of:

$\phi = 1$ or

$$\phi = \left[\frac{2G_{xy} \left(1 + \sqrt{\nu_{xy} \nu_{yx}}\right)}{\sqrt{E_x E_y}} \right]^{1/2}$$

where G'_{cx} denotes the shear modulus of the core in the xz plane; E_x, E_y, ν_{yx}, ν_{xy}, and G_{xy} are the extensional constants of the laminated face sheets; and

$$(\sigma_{cr})_{rc} = (h/R) \left[E_x E_y / (1 - \nu_{xy} \nu_{yx}) \right]^{1/2}$$

A practical minimum for the core thickness was established at 0.375 inch. Stability analyses showed that the theoretical thickness could go as low as 0.15 inch when considering the shell for overall bending. This is too thin since the shell must also have sufficient ring stiffness to support local loading from personnel and driveshaft supports. The analysis also assumes a somewhat perfect cylinder, which may not be practically achieved. The weight penalty for the additional core amounts to about 2 pounds for the tailboom.

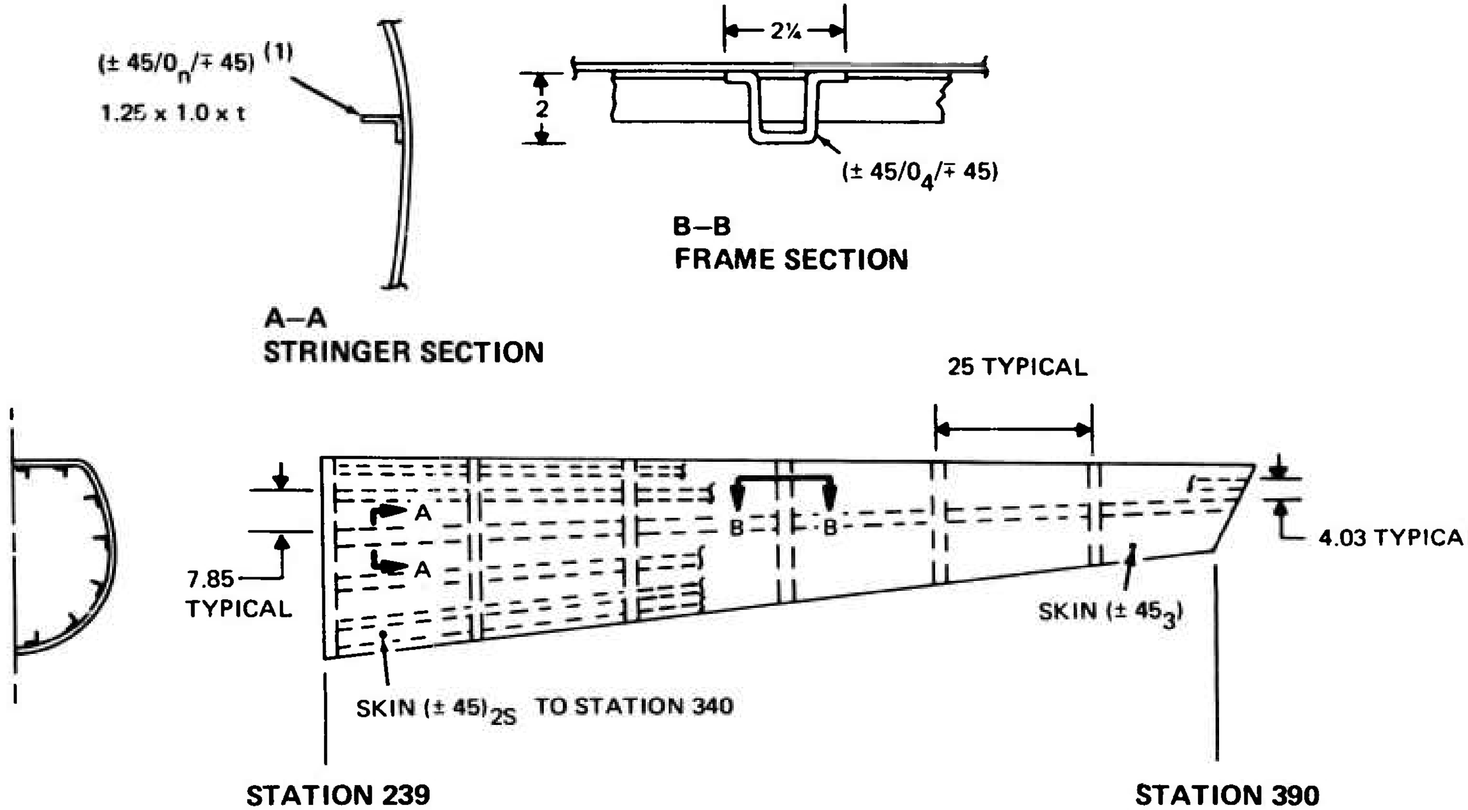
The results of the analysis are shown in Table 19. It is seen that the lowest weight shell is the hybrid using 0° graphite/epoxy and ±45° Kevlar/epoxy as a face laminate. The all Kevlar laminate requires two unidirectional plies to provide a positive margin of safety for compression. Since this peak compression stress occurs only on the upper half of the cross section for the tail bumper load condition, it is recommended that the hybrid laminate (D) be used for the upper half of the shell and that the Kevlar laminate (B) be used for the lower half, for a total shell weight of 47.1 pounds.

A sheet stringer frame concept in graphite/epoxy was analyzed to determine detail sizes and tailboom weight. The skin thickness and stringer spacing were strongly influenced by the buckling criterion imposed, that is, a maximum allowable shear of 2.5 times the critical shear at design limit load, or a $\tau/\tau_{cr} \leq 2.5$. This concept uses 14 equally spaced angle stringers to carry overall bending loads, hat section frames at 25-inch spacing to support the stringers and maintain contour, and a ±45 laminate for the skin to carry the shear loads resulting from overall shear and torsion on the tailboom shell. The final sizes are shown in a schematic in Figure 95 and weigh 55.5 pounds (including adhesive); this compares with the selected sandwich shell weight of 47.1 pounds. Almost 60 percent of the 55.5 pounds is in the skin. If the skin was allowed to go as thin as 3 pairs of ±45° plies (0.0315 inch), the total

TABLE 19. SUMMARY OF TAILBOOM SHELL ANALYSIS

Laminate	(A) G-AS/E (0/±45)	(B) K/E (0/±45)	(C) K/E (0_2/±45)	(D) Hybrid (0/±45) 0° G-AS/E, ±45 K/E
Properties				
F_{cu_x} psi	68,000	15,000	20,150	54,000
F_{xy} psi	37,000	22,000	19,250	22,000
E_x msi	7.2	4.3	6.0	6.32
E_y msi	3.4	1.27	1.22	1.68
G_{xy} msi	3.04	2.14	1.69	2.14
μ_{xy}	0.69	0.69	0.63	0.69
μ_{yx}	0.12	0.12	0.20	0.12
t_f inches	0.0157	0.0157	0.021	0.0157
Laminate Strength				
N_x (sta 300) lb/in.	-694	-694	-694	-694
f_x psi	22,100	22,100	16,520	22,100
MS (comp)	+2.08	-0.32	+0.24	+1.44
N_{xy} (sta 390) lb/in.	289	289	289	289
f_s psi	9,200	9,200	6,880	9,200
MS (shear)	+3.02	+1.39	+1.80	+1.39
Sandwich Stability (1)				
N_x (sta 239) lb/in.	-651	-651	-651	-651
$N_{x_{cr}} = \sigma_{x_{cr}}\ 2t_f$; lb/in.	-2,410	-1,402	-2,040	-1,690
MS (stability)	+2.70	+1.15	+2.13	+1.79
Weights				
Unit(3) lb/in.^{2}	0.00328	0.00312	0.00365	0.00315
Total(2) lb	49.2	46.9	54.8	47.3

(1) Core - Nomex HRH 10; 3.0 lb/cu ft; G_L = 7,000 psi; 0.375 inch thickness
(2) For total tailboom area of 15,000 sq in
(3) Includes adhesive at 0.13 lb/sq ft



(1) n VARIES FROM A MAXIMUM OF 15 PLIES IN THE UPPER HALF OF SHELL AT STATION 239 TO A MINIMUM OF 6 PLIES AT STATION 390.

Figure 95. Schematic for Skin-Stringer-Frame Semimonocoque Tail Boom Shell.

weight would be 49.1 pounds; but at design limit load, τ/τ_{cr} would equal 3.9. The skins would also be buckled at 1.0g. The performance of the buckled skin/stringer and skin/frame bonds at 1.0g under a vibratory helicopter environment is at this time an unknown and presents an unnecessary risk. Additionally, the effective shear moduli of the skins are reduced and may present a stiffness problem, since the tailboom cross section has been reduced from the baseline configuration to take advantage of the advanced materials properties and to reduce both weight and material usage.

4.4.3.4 Upper Deck Beams (See Figure 71)

The upper deck beams in the cabin section (sta 91 to sta 163) form a major part of the overall load carrying structure and support the transmission and rotor system for normal flight and emergency crash conditions. The beams are typical in that the bending loads are carried in the caps and the shear loads in the webs.

The critical condition is mass retention for a crash condition of 20g's down, 10g's forward, and 9g's lateral. A maximum bending moment of 452,000 inch-pounds occurs at sta 136, causing cap loads of ±37,700 pounds. The lower cap carries a portion of the load and acts as a shear tie to the deck, which also provides end load material.

The upper cap is a G-AS/E angle section with a thick leg buried in the sandwich shear web to carry the majority of the load and a thin leg to provide for attaching the fairing skin and center access hatch. The thick leg is primarily unidirectional plies with some angle plies to provide shear capability. The thinner leg contains sufficient ±45° angle plies to provide adequate bearing strength and carry shear loads around the holes for the mechanical fasteners.

Two possible configurations for the shear carrying web are sandwich and sheet-stiffener. The latter could be efficient if allowed to buckle elastically in a partial tension field mode, as metal structures do. The fatigue environment adjacent to the transmission and rotor make this approach unwise and risky with the current state of experience with buckled fibrous composite laminates. Sandwich shear webs were selected. A maximum shear flow of 1290 pounds per inch exists in the bay between 138 and sta 163 for the crash condition. The method of analysis used to determine the critical load in the shear panel is that from MIL-HDBK-23 where

$$F_{s_{cr}} = K \frac{\pi^2 D}{2t_f b^2}$$

Three different laminates were considered for the face sheets on a Nomex core:

Laminate	Material	t_f (in.)	F_{xy}^{su} (psi)	G_{xy} (psi)
(±45) G-AS/E	Uni prepreg	0.021	50,200	4.5×10^6
(±45) K/E	181 style prepreg	0.019	32,000	3.0×10^6
$(\pm 45_{G-AS}/45_K)$ Hybrid	G-AS uni prepreg K 181 style prepreg	0.020	40,400	3.75×10^6

For the hybrid, the 181 style Kevlar is used on the outer surface to improve damage resistance in the area where service and maintenance are performed.

The shear strength for each of the three laminates is adequate to carry the shear load. For stability, the core thickness required is 0.252, 0.387, and 0.286 inch, respectively. The hybrid sandwich is the lightest (the weight difference among the three is small) and is selected for its strength margin, damage resistance, and stiffness.

Undoubtedly these webs will be perforated with holes for the passage of lines from some of the subsystems (electrical, hydraulic, etc.). The stresses around such cutouts in the basic laminate and reinforcing layers may be analyzed using the digital computer program SY-5[5] (Boeing Vertol Program S-75).

4.4.3.5 Forward Box Frame and Cabin Posts (See Figure 63)

The forward box frame and cabin posts (at sta 81 to 91) provide the primary structural redistribution members for the forward cabin and provide for the requirements of crashworthiness.

The posts are designed for the large bending loads induced by a nose gear impact. The material selected is G-AS/E since large compression stresses exist. The posts are deep return lip C-sections and provide a cavity for the control rods to pass from the bottom to the crown section. The caps are solid $(0_2/\pm 45/0_4)_s$ laminates to carry the axial and bending loads while the web is a thin sandwich with $(0_2/\pm 45)$ faces.

The box frame is designed primarily by the crashworthiness requirements for rollover. The construction is a large hat section formed from a thin sandwich with hybrid face laminates. The hat section is closed off by a sandwich skin panel. At

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5 Eisenmann, J.R., Stress Distribution Around Cutouts, General Dynamics Report No. FZM-5555, August 1970.

the peak load area, the cross section includes $(\pm45/0_3)_s$ face laminates for the bending material, with unidirectional G-AS/E and angle ply K/E plies. The load intensity is relatively high, requiring an aluminum alloy honeycomb flex core to stabilize the panel, since the low shear modulus of Nomex would require too thick a core. The shear webs on the sides of the frame have $(\pm45/\overline{0})_s$ face sheets made of the same hybrid combination.

4.4.3.6 Fuel Cell Bay (See Figure 73)

The fuel cell structure is a curved shell with a flat top serving as the top deck, spanning from sta 163 to sta 239. The actual fuel tank terminates at a bulkhead forward of sta 239. The two basic critical loading conditions are the overall fuselage loading and a 20g crash condition for which the fuel tank walls must resist bursting pressures. The basic monocoque shell structure is a sandwich with a 0.75-inch-thick Nomex core and hybrid face sheets sized to carry the basic fuselage bending moments, shear, and torsional loads. Additionally, this shell must redistribute the loads coming in from the tailboom at sta 239 to the sta 163 bulkhead, upperdeck, and floor structures. The fuel pressure loads are resisted by the frames which are formed by the outer shell, a deep Nomex core, and an inner cap strap of laminated fiberglass. The frames are essentially circular on the bottom half of the tank, and therefore frame bending loads are not excessive. The frame inner caps are interconnected with longitudinal fiberglass straps. The interconnected framework is configured to resist extensive damage propagation from the hydraulic ram pressures created by high-energy ballistic penetration into the fuel. Since loads in the outer shell exist in the three load directions (x, y, and xy), the face sheet layers are outlined in a hybrid quasi-isotropic (0/±45/90) laminate, with 0° and 90° plies of G-AS/E and ±45° plies of K/E. Although the strengths (F_x, F_y, and F_{xy}) of this laminate are fiber dominated, the transverse strains (ε_{22}) for each layer do not exceed their allowable at limit loads for each condition.

4.4.3.7 Horizontal Tail (See Figure 77)

The horizontal tail surface is constructed of a box spar with the leading and trailing edges attached. The box spar spans from tip to tip, tapering down in cross section from the aircraft centerline to the tip. The loads are reacted at the tailboom at a pair of fixed lugs and at an actuator pivot located at the aircraft centerline. The angle of attack is varied by the extension and retraction of the actuator.

The box spar is constructed in two halves, top and bottom, each a channel-shaped section with one fitting into the other along the vertical shear webs. Ribs are located at the tip, each of the support locations, and at midspan. The top and bottom surfaces are of sandwich construction, while the shear webs are sized to be shear-resistant laminates.

The loads are typical of tail plane surfaces with the maximum loads at the root. The cross section of the spar box at the root is approximately 6.3 inches wide and 4.0 inches deep and must resist ultimate loads of

M_x = 55,000 in.-lb (beamwise)

M_z = 10,000 in.-lb (chordwise)

V_z = 2,330 lb (beamwise)

V_x = 600 lb (chordwise)

T = 1,050 in.-lb (torsion)

The method of analysis used to size both the bending and shear material is that of MIL-HDBK-23, as cited in the previous sample analyses. The upper and lower surfaces require a honeycomb thickness of 0.30 inch faced with ($\pm 45/0_2$) graphite/epoxy laminates. The face sheets are carried around the corner of the box to form a shear web laminate of $(\pm 45/0_2)_s$. At the root, the shears are sufficiently high so that the upper and lower halves must overlap completely to provide for shear resistance. Outboard, the shear loads are reduced, and the overlap required is for bondline shear transfer only.

4.4.4 STRUCTURE NOT ANALYZED IN DETAIL

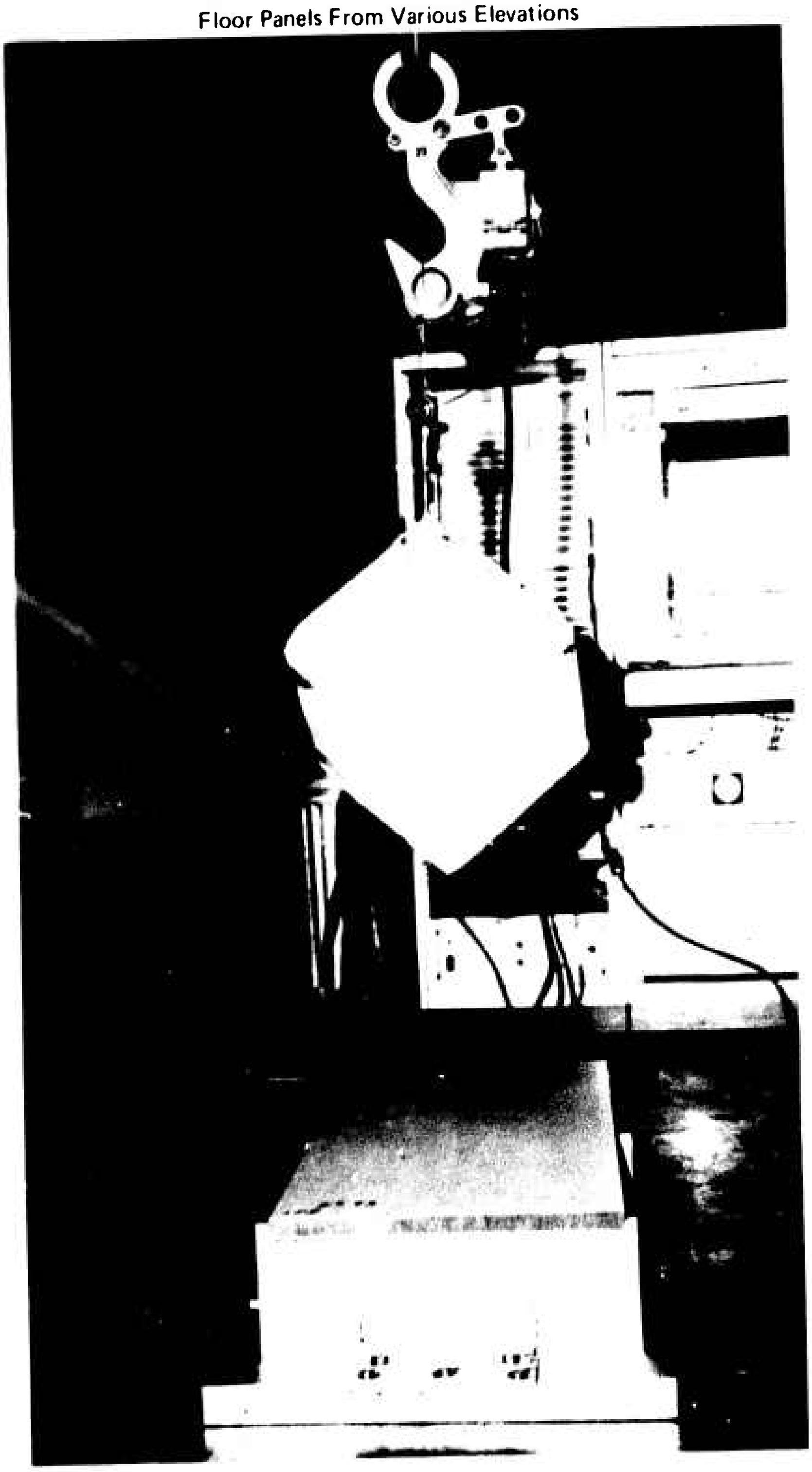
Some elements of the airframe structure have been sized based on previous work or experience which is considered applicable to this study.

The cockpit framing structure recommended for the MUT is hybrid unidirectional and woven prepreg material formed into the sections required for supporting the windshield and the door frames. The design is similar to that of the YUH-61A and to that previously demonstrated on the CH-53 cockpit canopy. Detail analyses were not needed to define the structure for preliminary design.

The cabin flooring is a sandwich panel concept design for resisting damage from personnel and cargo handling. This concept was developed through a drop test program for the YUH-61A and resulted in demonstrating the adequacy of sandwich panels with S-glass faces on a Nomex core. Figure 96 shows results of a stringent requirement for a 200-pound ammunition box drop tests on a CH-46 type floor using a rigidized aluminum alloy face sheet, a Kevlar 49 faced panel, and the S-glass faced panel. Although the S-glass panel is 4 percent lighter than the aluminum panel, it resists damage to a much greater degree.

200 Lb Box, Corner Impacted Onto Sandwich Floor Panels From Various Elevations

Upper Face, S-Glass [0/+45/90]$_S$
Lower Face, S-Glass [+45/0/−45]
4.0 Lb/Ft^3, 1.0 Inch Nomex Core
Panel Wt, 1.17 Lb/Ft^2



S-GLASS

200 LB BOX

+ 6"

+ 9"

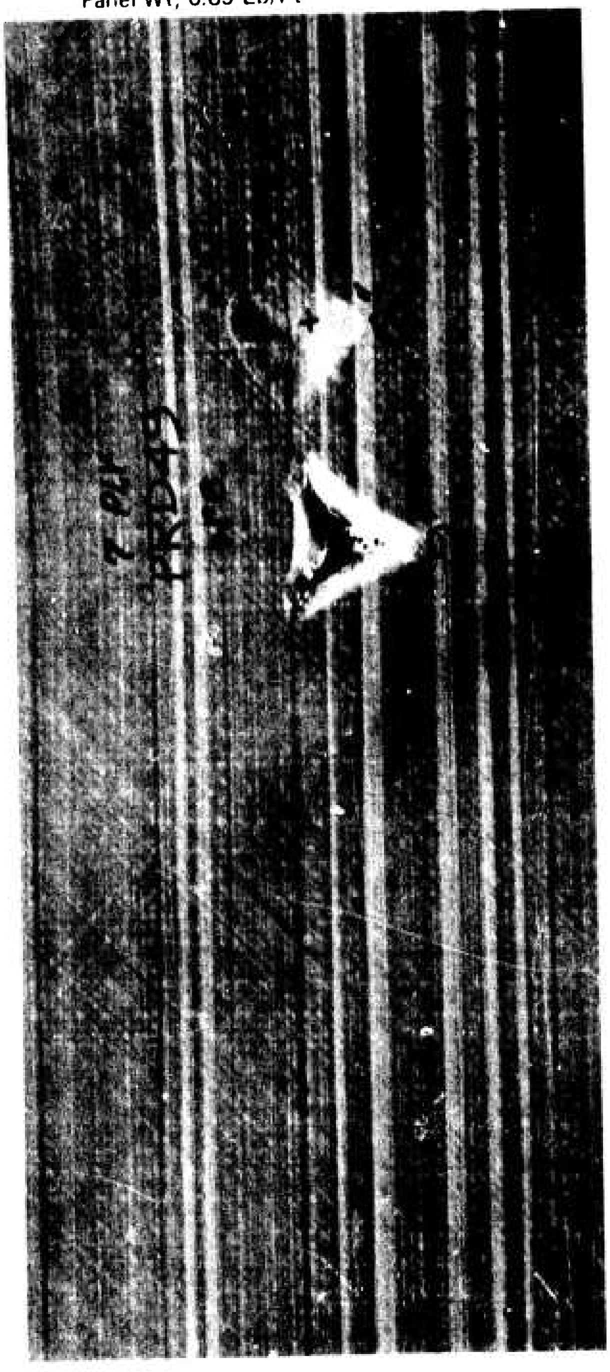
+ 12"

+ 15"

15-Inch Drop Results in 0.07 Inch Indentation

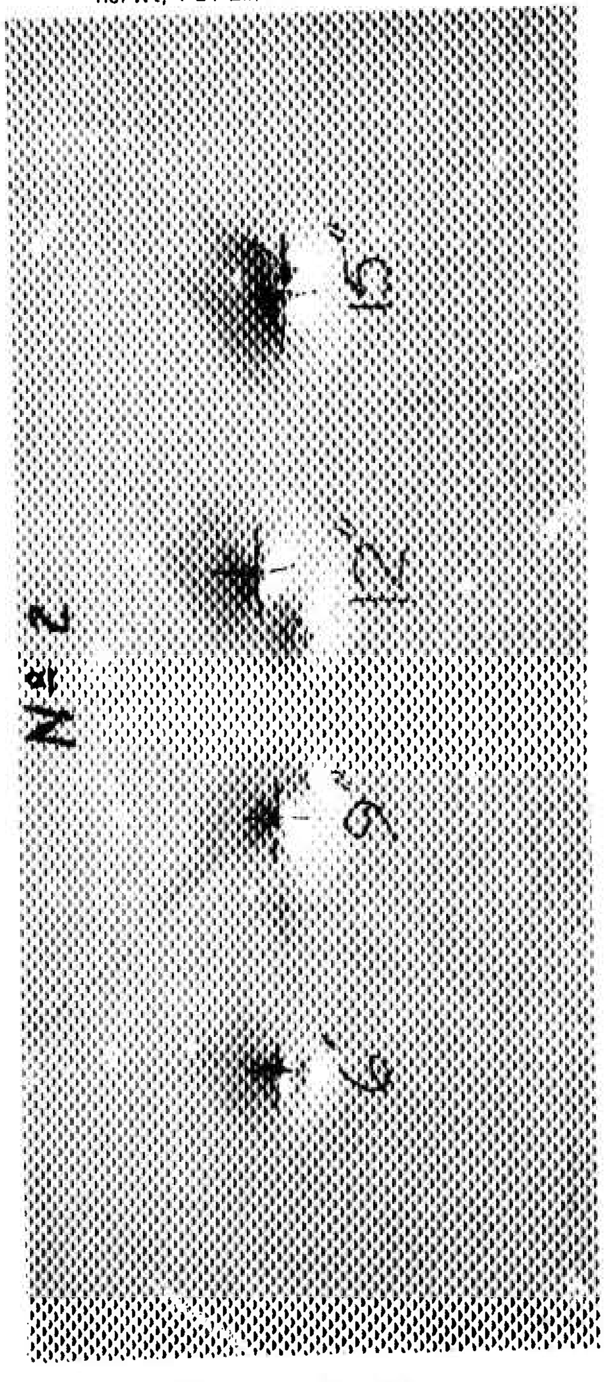
Figure 96. Drop Test Damage to Helicopter Floor Panels.

Upper Face Kevlar 49 [+45/0/−45/90]$_S$
Lower Face 7075-T6 0.012 Inches
4.4 Lb/Ft^3, 1.0 Inch 5052 Al-Al Core
Panel Wt, 0.89 Lb/Ft^2



9-Inch Drop Results in
Upper and Lower Face
Rupture

Upper Face 0.032 Inch Rigidized 2024-T3
Lower Face 0.016 Inch 2024-T3
4.4 Lb/Ft^3, 1.0 Inch 5052 Al-Al Core
nel Wt, 1 21 Lb/Ft^2



15-Inch Drop Results in
0.55 Inch Indentation and
4–5 Sq Inch Core Crushing

2

The upper deck spans from the cockpit to the tailboom, provides a ceiling for the cabin area, and bounds the fuel tank. Structurally it provides material for fuselage lateral shear and torsional loads and also provides supplemental vertical bending material. The overall loads are of low intensity, and therefore the face sheets are of minimum gauge. In the cabin area the lower face sheet is perforated for acoustical reasons and is not considered structural, except to provide stability through the core for the upper or load carrying face sheet. In the fuel bay, the deck has been sized to be compatible with the load carrying capability of the tailboom and fuel bay structure.

The bulkhead at fuselage sta 163 is a key structural element and provides load paths for most every critical load condition. It is the major load redistribution structure for overall vertical and torsional shear loads. It distributes the main landing gear reaction loads to the body, provides the forward face of the fuel cell, supports the main transmission and rotor loads, provides support for the cabin personnel seats, and contains crashworthiness features for roll over. Load paths are provided for all the aforementioned conditions and the sizes estimated to establish a weight for preliminary design.

4.4.5 COMPUTER AIDED DESIGN

A rigorous analysis of composite structures for the many conditions imposed on a helicopter airframe structure is a lengthy, arduous task. The design procedure to examine all the possible materials and laminations thereof is an almost impossible task if optimized weight and cost (the design goals) are to be achieved without the aid of the computer. For a preliminary design of a lightly loaded structure, as was done in this study, one may examine a few known critical conditions applied to critical locations and perform analyses sufficient to size the elements so as to give a firm basis to weight estimates and to uncover design problems.

Computer programs are available to perform many of the necessary calculations for strength, stability, cost, stiffness, etc. One such program, COOP (laminated composite analysis and optimization computer program), was used to examine a shear panel for the upper deck beam. This panel is one of the highest loaded panels in the airframe, with a maximum shear flow of 1290 pounds per inch. The results of the analyses are summarized in Figure 97 and show that minimum gauges determine the final design. The computer program was allowed to determine minimum thickness laminates, unconstrained by realistic per-ply prepreg thicknesses, resulting in face thicknesses as low as 0.0026 inch. The realistic minimum for an all ±45 degree laminate is 4 plies or 0.021 inch. The selected panel face sheet is a ($\pm 45_G/45_K$) hybrid laminate using uniprepreg graphite and 181 style woven Kevlar 49 for a total

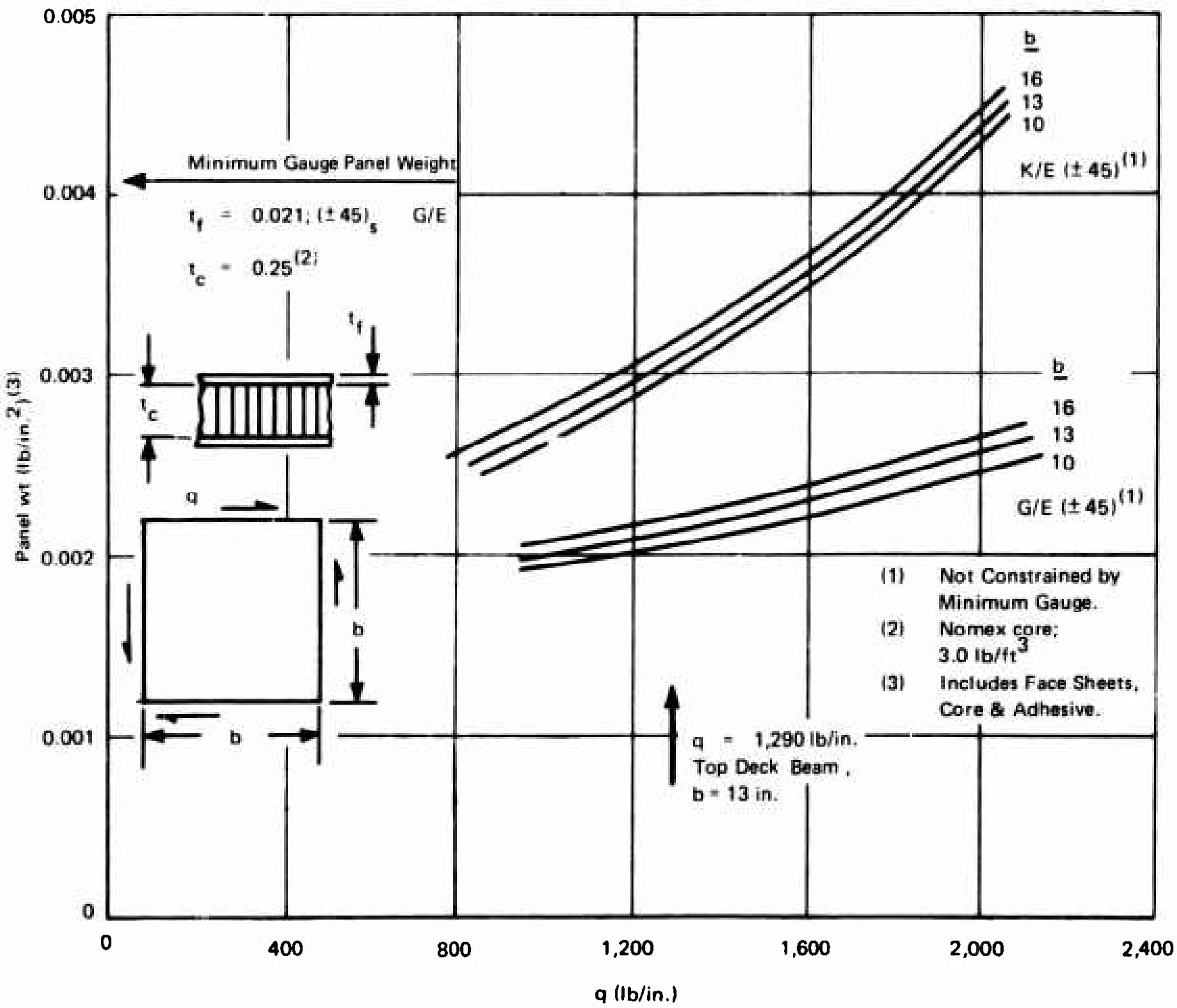


Figure 97. Sandwich Shear Panel - Optimization Study.

thickness of 0.020 inch as discussed in the section on design features.

A more general design concept synthesis computer code, SPEED[6], is also available; it is capable of analyzing and optimizing skin/stringer panels as well as sandwich panel structure. These tools are more valuable for moderate to highly loaded structures (2,000 to 10,000 pounds per inch), but they may still be used to examine the lower end of the load intensity regime (200 to 2000 pounds per inch), especially if the lower bounds on minimum gauge, number of plies, and core thickness are included in the computer code.

It should be noted that the results of the computer optimization study for the shear web do not necessarily agree with the results of the analysis whose results are reported in Section 4.4.3.4. The material properties and buckling equations are not identical. The material property data for the computer program were input to preclude laminate failure due to matrix failure in a single layer, using unrealistically high transverse tensile and shear properties for the unidirectional layer strength. Failure criteria and stability equations for the COOP program may be modified to be consistent with whatever criteria are established for the airframe or elements thereof. The inclusion of the computer results is intended to reflect current state-of-the-art analysis techniques available for advanced structural concept design with laminated composite materials.

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6 Laasko, J.H., and Zimmerman, D.K.: Synthesis of Compression Panels Having Nonuniform Stiffener Sections, AIAA/ASME/SAE 14th Structures, Structural Dynamics, and Materials Conference; Williamsburg, VA, March 20, 1973.

4.5 PARTS COUNT

1. All items which will carry a part number are counted.

2. Mechanical fasteners (rivets, bolts, anchor nuts, quick-release fasteners) are not counted.

3. Each vendor-supplied assembly, regardless of number of parts, is counted as one (e.g., actuator assembly consisting of cylinder-rod, end fittings, valves, etc.). Note: these items count as one only if they are "bought out".

4. List accounts for left-hand (LH) and right-hand (RH) items in totals (e.g., parts for both RH and LH pilots' doors are included in list totals). These are not multiplied by 2.

5. A 20-percent contingency factor is applied to the parts count of each module, thereby increasing the actual count.

Table 20 indicates the weight and parts count associated with each module, with the last column giving the module parts-per-pound ratio. To obtain a true average of parts-per-pound, the total of the module weights was divided into the total parts count to obtain the 0.84 figure shown.

This point was plotted in conjunction with the corresponding airframe weight to show an appreciable improvement over the metal state-of-the-art skin/stringer construction; it closely approximated the HLH metal honeycomb structure (see Figure 98).

TABLE 20. AIRFRAME MODULE PARTS PER LB
(COMPOSITE REFINED DESIGN AS DEFINED IN SECTION 4)

Module Number	Description	Remarks	Module Weight (lb)	Parts Count	Parts Per lb
1	Cockpit enclosure	Includes pilots doors each side	188	140	0.74
2	Underfloor structure and floor panels	Not including nose gear weight of 61 lb	149	172	1.15
3	Upper deck and fairing assembly		187	98	0.52
4	Bulkhead sta 163 and side panel assembly	Not including main gear weight of 186 lb	64	59	0.92
5	Tank support and side avionics structure	Includes ballistic protection system	132	155	1.17
6	Tailboom and vertical stabilizer box		122	40	0.31
7	Tailcone fairing		8	7	0.87
8	Tail bumper assembly		9	11	1.22
9	Horizontal stabilizer and actuator		38	29	0.76
10	Vertical stabilizer leading and trailing edge and tip fairing		16	20	1.25
11	Cabin hinged and sliding doors	Weight and parts total is for 4 doors (1 hinged and 1 sliding each side)	45	70	1.55
		TOTAL	957	801	0.84 average

NOTES: a. Includes primary and secondary structure.
b. Includes all metal or composite fittings.
c. Excludes main and nose landing gear weight and parts count.
d. Includes 20 percent contingency added on actual parts count.

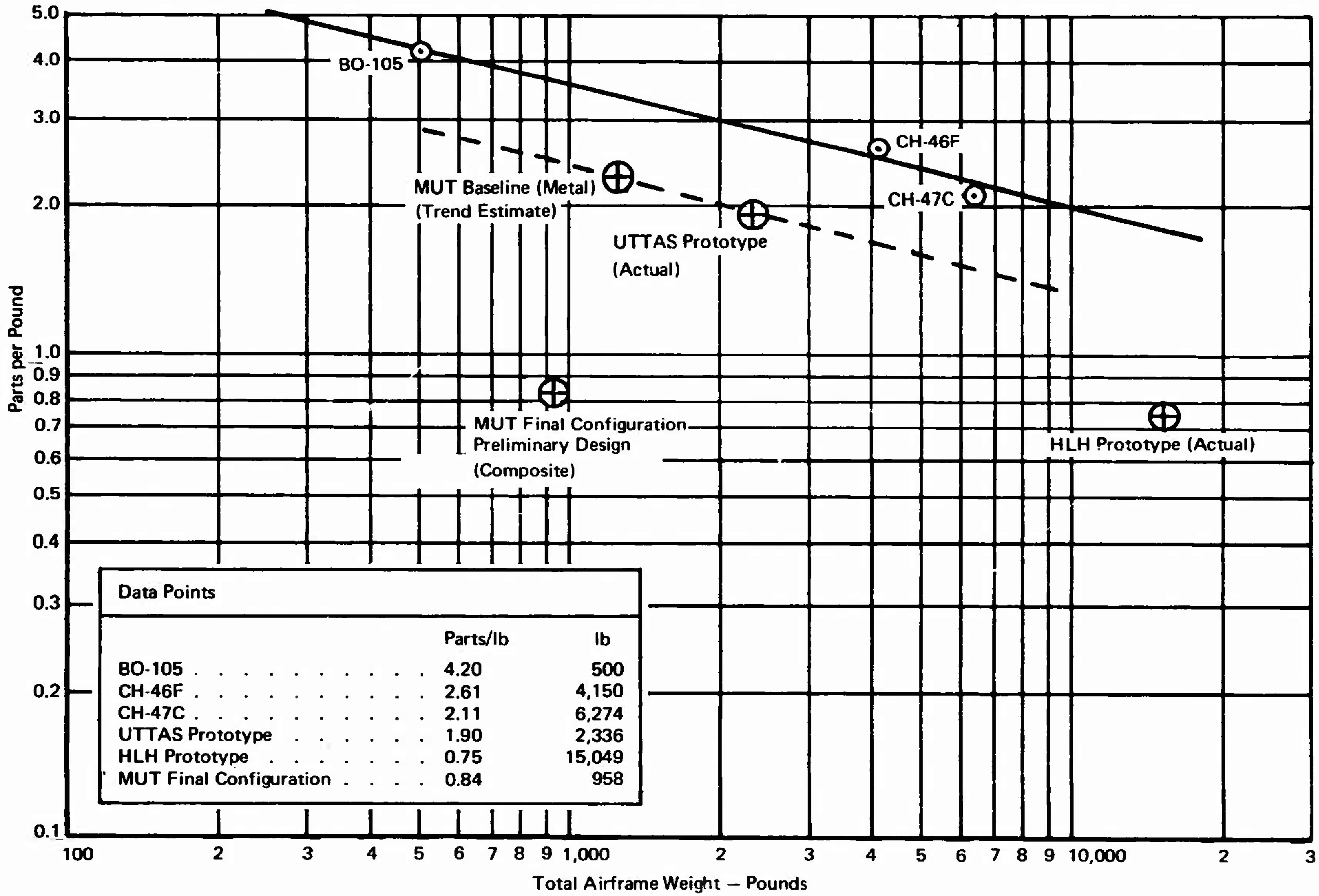

Figure 98. Airframe Parts Per Pound.

5. DETAILED SUMMARY (COMPARATIVE CHARACTERISTICS OF BASELINE DESIGN, ADVANCED STRUCTURE DESIGN, AND RESIZED ADVANCED STRUCTURE DESIGN HELICOPTER)

The baseline helicopter and the advanced structure helicopter are nearly identical in geometry and installed horsepower. They differ slightly in the tailboom and horizontal stabilizer configuration, and differ in the design of the rotor system, drive system, flight control system, landing gear, and airframe structure.

The weight estimate for the advanced helicopter was substituted for that of the baseline, and new performance was calculated using HESCOMP analyses.

Correction factors for the improvement in system weights were then introduced and HESCOMP again used to predict the size and weight of an advanced structures helicopter which would maintain the same crew and cargo compartment space provisions, utilize the advanced engine cycles of the SASS (Appendix A), and perform the SASS mission requirements satisfied by the baseline design.

A comparison chart of the three vehicles resulting from this study is presented as Table 21, which compares size, cost, weight and performance. In addition, selected performance characteristics resulting from the HESCOMP analysis are presented in Section 5.3.2.

TABLE 21. CONFIGURATION COMPARISON - GEOMETRY, PERFORMANCE, WEIGHT AND COST

Parameter	Baseline	Advanced Structure Derivative	Resized Advanced Structure
Length, Body and Tailboom (ft)	40.5	40.5	38.3
Length, Cabin (ft)	9.2	9.2	9.2
Length, Body (ft)	20.1	20.1	20.1
Length, Tailboom (ft)	20.4	20.4	18.2
Main Rotor Location (ft)	12.6	12.6	12.6
Cabin Width (ft)	8.0	8.0	8.0
Horizontal Tail			
Aspect Ratio	5.7	5.7	5.7
Area (sq ft)	21.1	21.1	17.8
Span (ft)	11.0	11.0	10.1
Mean Chord (ft)	1.9	1.9	1.8
Taper Ratio	0.566	0.566	0.566
Thickness/Chord (ft)	0.15	0.15	0.15
Vertical Tail			
Aspect Ratio	1.722	1.722	1.722
Area (sq ft)	18.3	18.2	15.4
Span (ft)	5.6	5.6	5.2
Mean Chord (ft)	3.3	3.3	3.0
Taper Ratio	0.473	0.473	0.473
Thickness Ratio	0.23	0.23	0.23
Main Rotor			
Diameter (ft)	39.0	39.0	35.8
Solidity	0.101	0.101	0.101
Disc Loading (psf)	8.0	8.0	8.0
No. of Blades	4	4	4
Blade Twist (deg)	-12.0	-12.0	-12.0
Cutout/Radius Ratio	0.23	0.23	0.23
Tip Speed (fps)	750	750	750
Tail Rotor			
Diameter (ft)	7.8	7.8	7.2
Solidity	0.227	0.227	0.226
Disc Loading (psf)	13.9	13.9	13.8
No. of Blades	4	4	4
Blade Twist (deg)	-9.0	-9.0	-9.0
Cutout/Radius Ratio	0.25	0.25	0.25
Tip Speed (fps)	700	700	700

TABLE 21. Continued

Parameters	Baseline	Advanced Structure Derivative	Resized Advanced Structure
WEIGHTS AND PERFORMANCE			
Design Gross	9544	9544	8059
Payload (lb)	960	1642	960
Fixed Useful Load (lb)	498	498	495
Fuel (lb)	1655	1655	1422
Weight Empty (lb)	6431	5749	5193
Power and Transmission Ratings			
Installed SL, STD 2 Engines (hp)	2122	2122	1797
Transmission			
Main (hp)	1836	1876	1559
Rotor (hp)	1615	1615	1369
Tail Rotor (hp)	220	220	185

SUBSYSTEM WEIGHT/COST	Wt (lb)	Cost Ratio*	Wt (lb)	Cost Ratio*	Wt (lb)	Cost Ratio*
Rotor Blades	563	1.00	434	.87	347	.82
Rotor Hub	364	1.0	316	.85	269	.85
Flight Controls	562	1.0	535	.74	455	.74
Drive System	1047	1.0	873	.80	737	.80
• Transmission	725		605		510	
• Gearboxes, Shafts, etc.	322		268		227	
Landing Gear	296	1.0	258	1.0	237	1.0
Airframe	1067	1.0	873	.73	829	.69

*NOTE:

Cost ratios refer to ratio of advanced components in system to equivalent baseline design. Where no advanced components are used, ratio is 1.0.

This study based on Boeing Vertol recurring wraparound dollar rates for labor and material in 1974. (Includes everything except profit.)

Reference Sub-System Weight/Cost Table - the last two columns of weight figures shown are participating composite structure weights only (no metal parts).

5.1 WEIGHT ESTIMATING

5.1.1 INTRODUCTION

The weights for the baseline aircraft of conventional sheet metal construction were largely derived from the YUH-61A helicopter. The weight methods in the HESCOMP program were adjusted to predict the actual weights of the YUH-61A components, thus ensuring realism in estimating weights of modern state-of-the-art helicopters. The fixed-equipment weights of the YUH-61A were adjusted to the requirements of the MUT aircraft. For comparison, the structure weights of the sheet metal baseline aircraft were separated into assembly modules which were similar to the modules of the advanced composite aircraft. Weights of the composite aircraft modules were calculated in detail, and the weight reductions resulting from the application of advanced composite material were summed to get the total reduction of structural weight.

For some components, such as the landing gear shock struts, no weight reduction was made, since design analysis indicated that the resulting dollar cost would be unacceptably high.

A direct comparison between advanced composite and conventional metal structure can be readily made for most of the structural assemblies, but it should be noted that there are areas where the structural configurations differ sufficiently to preclude a satisfactory comparison. An example of this is the fuel bay section.

Comparative module weights are given in Table 22. A conventional weight summary is shown in Table 23, which also includes the weight reduction achievable by the use of advanced composites. It will be noted that there is an apparent discrepancy between the airframe module weight total of 958 pounds (Table 20) and the body weight of 832 pounds (shown in Table 23). This is due to the conventional weight coding methods used in the latter table whereby the weights of certain elements of the structure such as brackets, support members, fairings, etc., are transferred from the actual body group and are included with other groups in the same table such as tail group, avionics group, flight controls, etc.

Table 20 includes all airframe structure, primary and secondary, in modules total weight regardless of coding method used in the other table.

Some items, mainly supports for equipment (and some other components), were not studied in sufficient depth to ascertain the desirability of using composites for their construction. Some of these items, which total about 150 pounds, will probably be made of composites in a production program, and will result in increased weight savings; however, only 15 pounds weight reduction was taken for composite equipment supports in this study.

The weights in this study are considered to be attainable without undue technical risk. A slight degree of conservatism is believed to exist in the composite aircraft weights, and since the sheet metal aircraft was based on the actual weights of the YUH-61A, the comparative weight savings shown for the composite aircraft are expected to be attainable with current state-of-the-art materials and known manufacturing processes.

TABLE 22. MODULE WEIGHTS COMPARISON

Module	Weight (lb) Sheet Metal	Weight (lb) Composite (with metal landing gear and some metal fittings)
Cockpit Enclosure	219.5	188.2
Floor Structure	258.3#	212.2*
Deck Structure	296.7	186.8
Bulkhead Sta 163 Assembly	289.6##	263.5**
Tank Support and Side Avionics Structure	159.9	131.7
Tailcone Fairing	22.5	8.8
Tail Bumper and Absorber	10.8	9.4
Tailboom	159.9	121.8
Horizontal Stabilizer	50.1	38.3
Vertical Stabilizer Fairings	24.5	16.0
Side Doors (Cabin)	54.3	45.0
TOTAL	1546.1	1221.2

\# Includes nose gear weight of 70 pounds
\## Includes main gear weight of 215 pounds
* Includes nose gear weight of 63 pounds
** Includes main gear weight of 200 pounds

5.1.2 WEIGHT SAVING OF COMPOSITE AIRFRAME INCLUDING METAL LANDING GEAR AND SPECIFIED METAL FITTINGS COMPARED TO BASELINE METAL AIRFRAME AND LANDING GEAR

Total airframe modules percent weight saving - advanced composite structural derivative over baseline metal:

$$\frac{(1546 - 1221)}{1546} \times 100 = \frac{32500}{1546} = 21 \text{ percent}$$

TABLE 23. COMPARATIVE WEIGHT SUMMARY - PRELIMINARY DESIGN

	Baseline Sheet Metal		Advanced Structural Derivative		Resized Advanced Structure Composite	
WING	--		--		--	
ROTOR	927		750		616	
TAIL	98		86		72	
SURFACES			30			
ROTOR			56			
BODY	1067		832		807	
BASIC	623		491		466	
SECONDARY	444		341		341	
ALIGHTING GEAR GROUP	296		281		237	
ENGINE SECTION	135		122		114	
PROPULSION GROUP	1891		1703		1453	
ENGINE INST'L	468		468		410	
EXHAUST & IR SYS	94		85		74	
COOLING SYS						
ENGINE CONTROLS	32		32		28	
STARTING SYS	58		58		50	
PROPELLER INST'L						
LUBRICATING (DRAINS)	2		2		2	
FUEL SYS	190		185		152	
DRIVE SYS	1047		873		737	
FLIGHT CONTROLS	562		535		455	
AUX. POWER PLANT	--		--		--	
INSTRUMENTS	133		133		133	
HYDR. & PNEUMATIC	50		50		50	
ELECTRICAL GROUP	281		279		279	
AVIONICS GROUP	400		398		398	
ARMAMENT GROUP	20		18		18	
FURN. & EQUIP. GROUP	462		455		455	
ACCOM. FOR PERSON.	309		306		306	
MISC. EQUIPMENT	61		57		57	
FURNISHINGS	41		41		41	
EMERG. EQUIPMENT	51		51		51	
AIR CONDITIONING	46		45		45	
ANTI-ICING GROUP	29		29		29	
LOAD AND HANDLING GP.	34		33		33	
WEIGHT EMPTY	6431		5749		5193	
CREW	470		470		470	
TRAPPED LIQUIDS	10		10		10	
ENGINE OIL	18		18		18	
PAYLOAD	960		1642		960	
FUEL	1655		1655		1422	
GROSS WEIGHT	9544		9544		8070	

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5.1.3 ROTOR GROUP

The rotor group consists of the rotor blades and hub. It does not include any flight-control components. The rotor weight for the baseline sheet metal aircraft is derived from trends and analyses of current rotor studies. The rotor for the two composite aircraft is based upon the bearingless main rotor (BMR) study.

The BMR study promises a weight reduction of 23 percent of blade weight and 13 percent of hub weight. The total reduction in rotor weight is 19 percent. The BMR has the blade attachment location at approximately 27.5 percent of rotor radius from the centerline of rotation, compared to a location of about 7.5 percent for conventional rotors. This reduces the actual rotor blade span and thus, rotor blade weight. Further reduction in the blade weight occurs due to the elimination of the conventional heavy metal root fitting. This fitting is replaced by a clamping device which is included with the hub. The hub design replaces the conventional pitch change mechanism with flexing straps.

Due to the difference in configuration between the rotor of the baseline metal aircraft and the two composite aircraft, the HESCOMP program should not use any reductions in the blade and hub K factors, K_{12} and K_{13}. Instead, the constants k_{PRB} and k_{PH} are changed to account for the configuration effects.

5.1.4 TAIL GROUP

The baseline horizontal tail weight is calculated on a unit weight (pounds per square foot) based on existing helicopters. The composite version is estimated from design sketches and material usage. The tail rotor weight is obtained from a trend, adjusted for YUH-61A experience. Weights of the vertical tail (tail rotor pylon) and ventral fin are included with the body group.

5.1.5 BODY GROUP

The body group weights were determined by a trend for the sheet metal basic structure. The secondary structure was determined by semianalytical methods which are based on the air load on each door, etc. The floor weight for the sheet metal aircraft was based on the weights of similar existing floors. The weights for the two composite aircraft basic structures were derived by calculating them as conventional sheet metal structures and then applying the factor derived from the composite module portion of this study. The composite secondary structure was taken directly from the composite module results and the same value was used for both composite aircraft, since only the boom changes in size between the two.

5.1.6 ALIGHTING GEAR

The alighting gear weight is determined as a proportion of gross weight. The ratio used was determined from the YUH-61A aircraft, which has a similar configuration.

For the composite aircraft, a 5-percent weight reduction was made in the alighting gear group for manufacturing the structural backup supports from composite materials. The lower fittings are metal.

5.1.7 ENGINE SECTION

The engine cowling, mounts, vibration absorbers, firewalls, and air intake comprise this group. These weights were derived as a ratio of the engine weight for a similar installation. Only the cowling was considered as a candidate for composite materials. No work platforms are built into the cowling.

5.1.8 ENGINE INSTALLATION

The engine weight was determined from curves of engine versus shaft horsepower supplied as part of the SASS of the MUT (Appendix A). The exhaust system, lubrication system, engine controls, and electrical starting system weights were developed from similar aircraft data and converted to a ratio of engine weight for HESCOMP use.

5.1.9 FLIGHT CONTROLS

The flight controls weight is derived from a combination of weights from similar aircraft and from trends. The cockpit controls and stability augmentation system weights are based on YUH-61A data. The weights of the upper (rotor head) controls and the hydraulic boost and system controls are from trends. The trends have been modified for the two composite aircraft since the interim report was submitted. Discussions with our controls and rotor groups indicated that the upper controls and hydraulic boost system for same should not vary noticeably with rotor weight. Rotor blade weight is used as a parameter in the upper controls trend, and rotor weight is used in the system controls trend, which includes the hydraulic boost system. The trend constants in the HESCOMP program were therefore adjusted to effectively increase the weights of upper controls and system controls to the same level as those for a conventional rotor.

The upper controls weight was reduced by 2 percent for the use of composite structural supports. Similarly, the system controls are reduced by 1 percent for using composites.

A fly-by-wire system is used in the two composite aircraft as a backup system for the single set of system controls. Studies indicate that approximately 20 pounds is saved by replacing the mechanical backup system by the fly-by-wire system. A slight increase in stability augmentation system (SAS) weight due to fly-by-wire integration is offset by an equal reduction by using composite structural supports for the cockpit controls. The cockpit controls sticks and pedals are of conventional construction.

5.1.10 FIXED EQUIPMENT

The fixed equipment weights are based on the YUH-61A with applicable reductions for the smaller size and specified requirements of the MUT aircraft. A 15-pound weight reduction for composite structural supports was used for the two composite aircraft.

5.2 PERFORMANCE EVALUATION

The purpose of this task in the MUT study was to define the effect of the application of specific advanced structures on (1) vehicle performance, and (2) vehicle size. An all-metal airframe vehicle was established through a trade study which examined the effect of rotor disc loading (W/A), rotor tip speed (Vtip), and design cruise speed (Vcr). This cruise speed was used to define the rotor solidity which would allow unstalled capability to perform a 1.75g maneuver (2.75g normal load factor). The mission used is described in detail in Appendix A.

Figures 99 through 104 are the results of the vehicle sizing study for the metal airframe baseline vehicle. This study was carried out using the Boeing-developed HESCOMP computer program. Figure 99 shows the variation in vehicle takeoff gross weight for design speeds of 150, 160 and 170 knots. As the cruise speed increases, the vehicle weight increases as a result of the increased solidity required at the higher speeds to maintain the 1.75g capability. The lightest weight vehicles result at the lowest design speed of 150 knots. Examining the 150-knot matrix, gross weight decreases as tip speed increases, resulting from reduced rotor solidity requirements, as shown in Figure 103, and reduced fuel requirements from the more efficient rotors. At a constant tip speed, gross weight reduces as rotor disc loading is decreased. This is caused by the improvement in hover efficiency at lower disc loading and the resulting reduction in installed power and drive system weights.

5.2.1 DESIGN POINT SELECTION

One of the design point selection guidelines stated that rotor disc loading should be within the range of 6 to 8 psf. An additional requirement for air transportability was also imposed. This second criterion requires a small, compact vehicle, therefore, a small rotor diameter or the highest allowable disc loading. The final selected vehicle (indicated on the plot by the circle) weighs 9546 pounds, has a disc loading of 8 psf, and a design cruise speed of 150 knots, with 1.75g maneuver capability. A complete geometric description is given in Table 21.

5.2.2 EFFECT OF ADVANCED STRUCTURES ON AIRCRAFT SIZE

To understand the effect of advanced structure on vehicle performance, the baseline vehicle was reweighed applying selected advanced systems to the airframe. The baseline metal configuration geometry was held constant, and weight reductions were applied to the selected airframe systems. The engines from the baseline metal vehicle were utilized in this new vehicle without modification

The detailed geometry and summary weights are shown in Table 21, column 2.

5.2.3 RESIZED ADVANCED STRUCTURE VEHICLE

To establish the effect of advanced structures on reduced vehicle size, while maintaining the same mission and performance capability, the sizing exercise used to establish the metal baseline vehicle was rerun using weight factors identified for the advanced systems. The results of this trade study are presented in Figures 105 through 110.

The trends with design cruise speed, disc loading, and rotor tip speed are the same as those for the metal aircraft. The advanced structure vehicles, however, exhibit a lower sensitivity to these parameters than do the metal vehicles, which is indicative of a lower growth factor (i.e., $\frac{\Delta GW}{\Delta WE}$).

The design point advanced vehicle selected is illustrated on Figures 105 through 110. The vehicle was selected using the same criteria as used for the metal vehicle, namely, small size and low weight. The geometric characteristics and summary weight data are given in column 3 of Table 21.

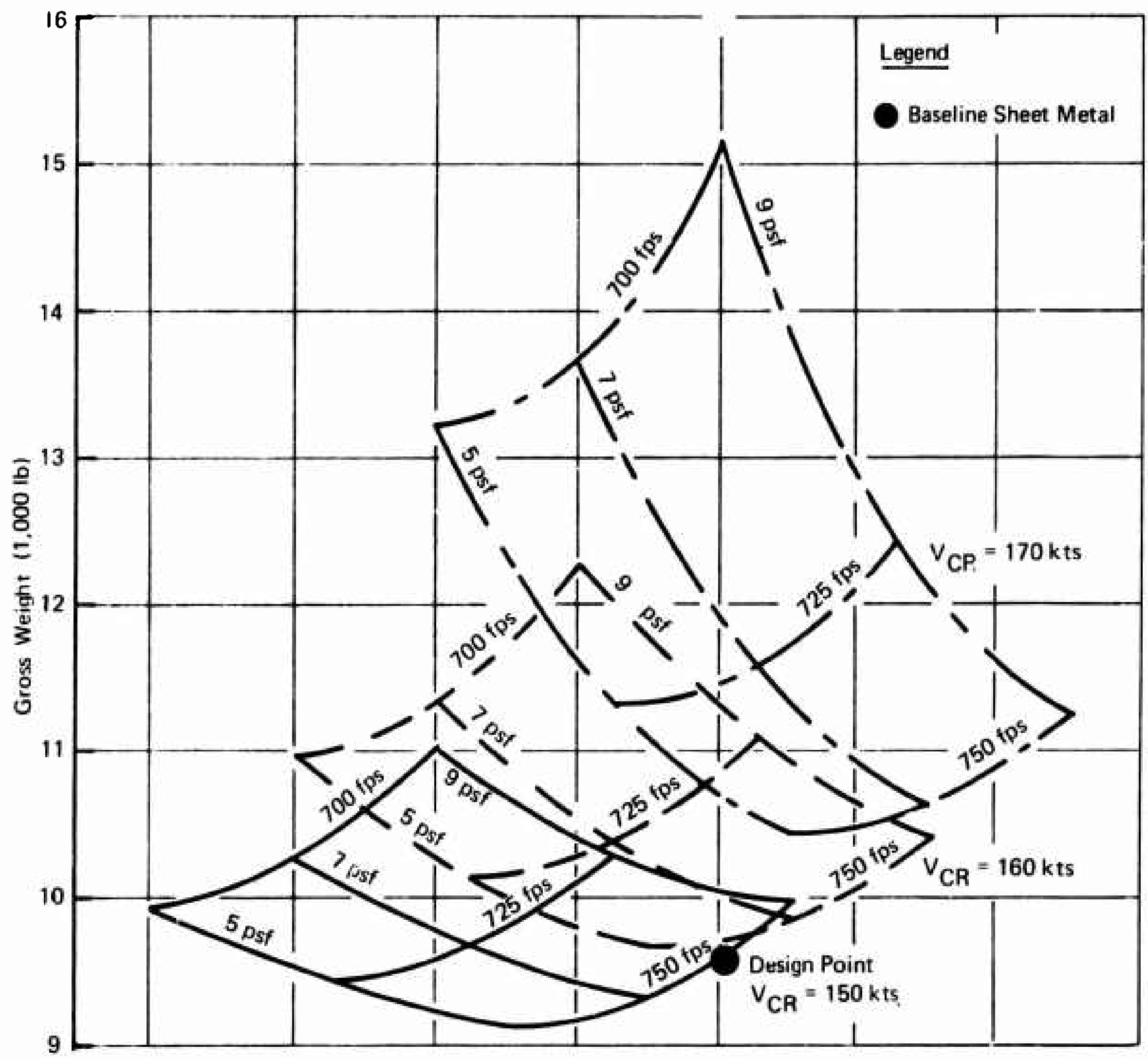


Figure 99. Effect of Tipspeed Disc Loading and Design Cruise Speed on Gross Weight.

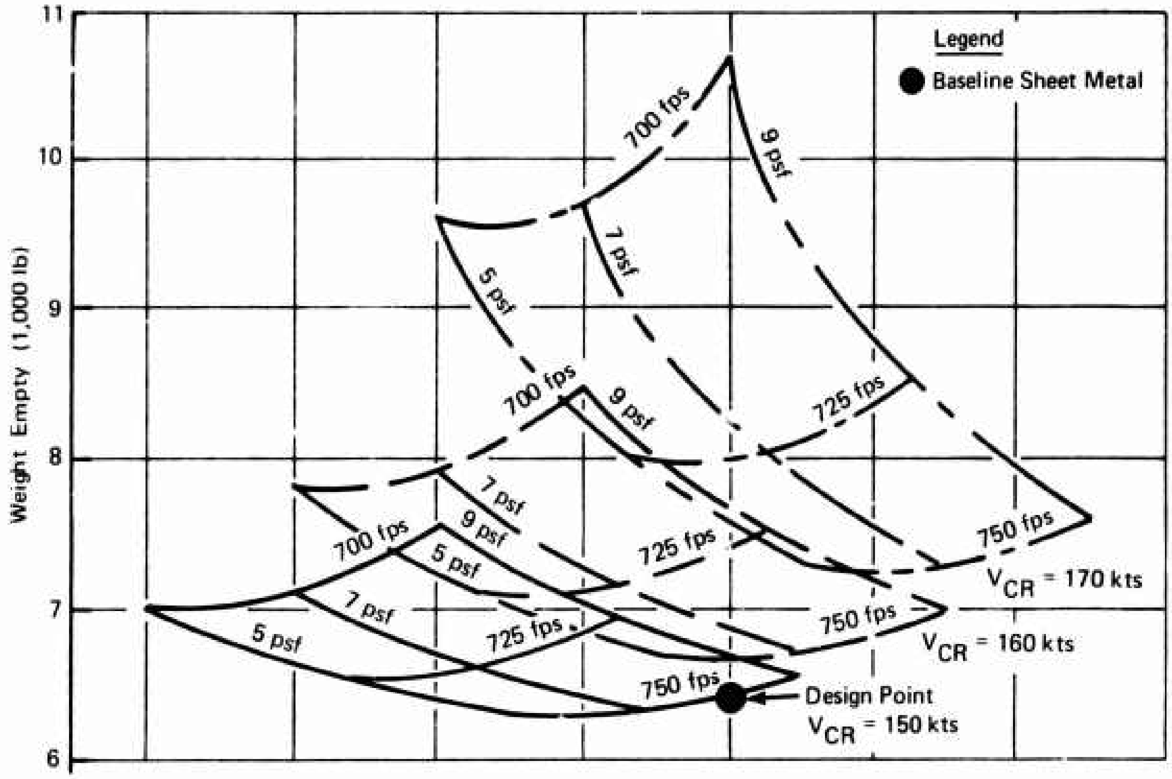


Figure 100. Effect of Tipspeed Disc Loading and Design Cruise Speed on Weight Empty.

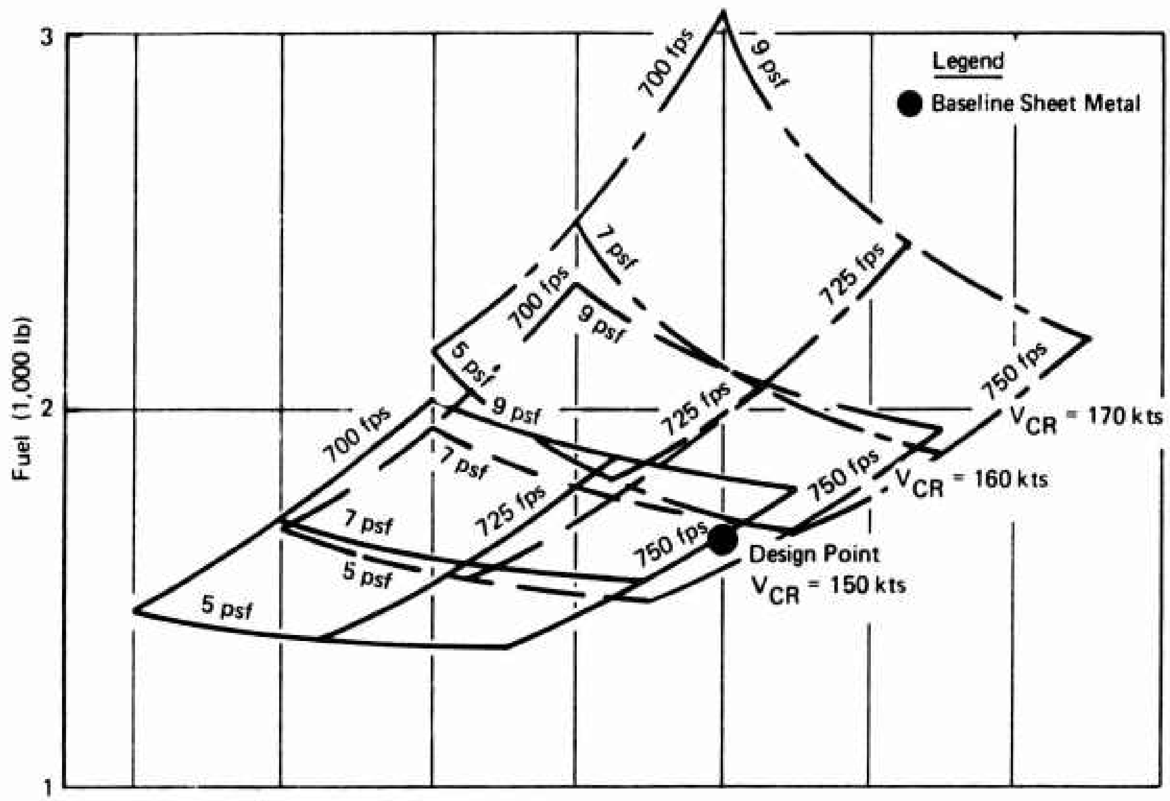


Figure 101. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Fuel.

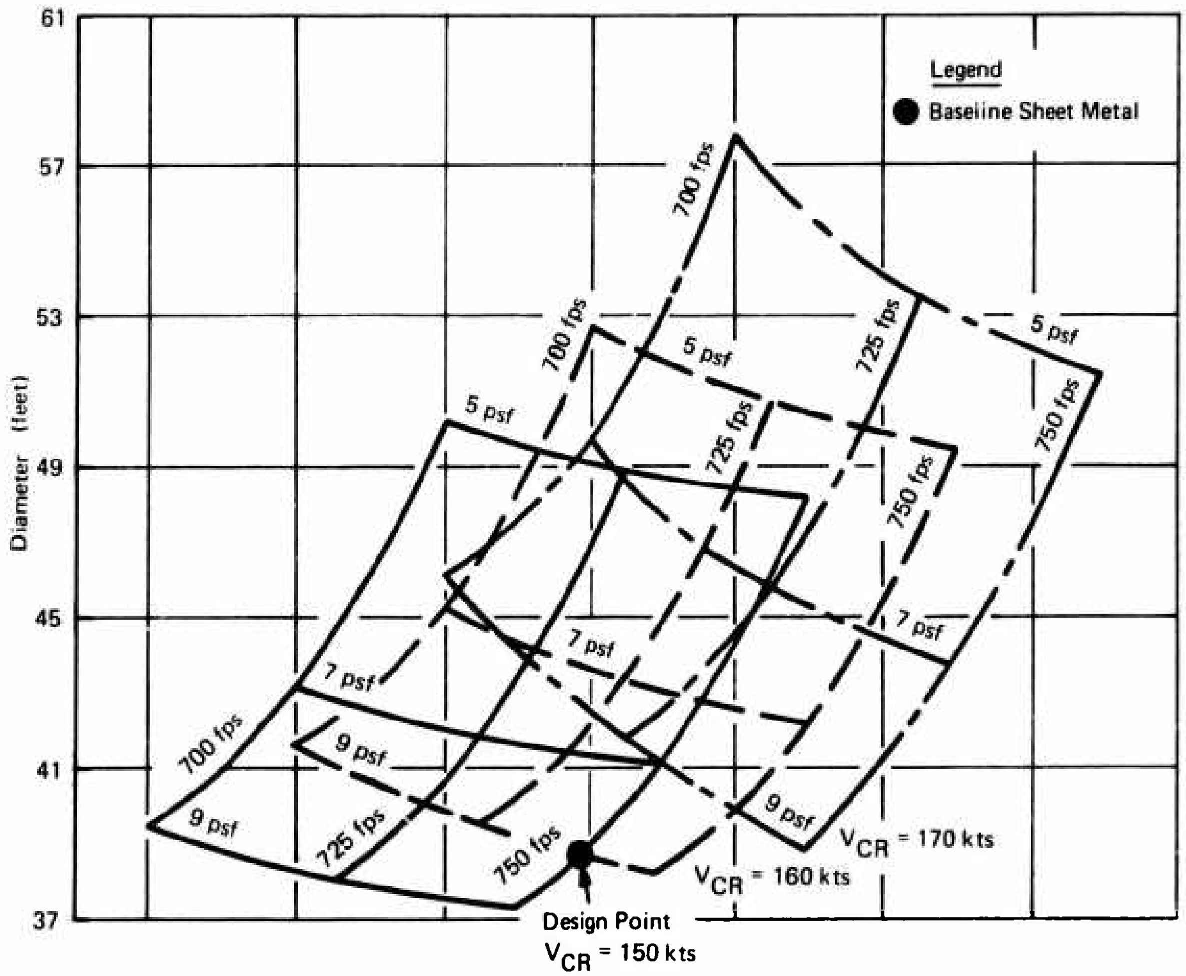


Figure 102. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Diameter.

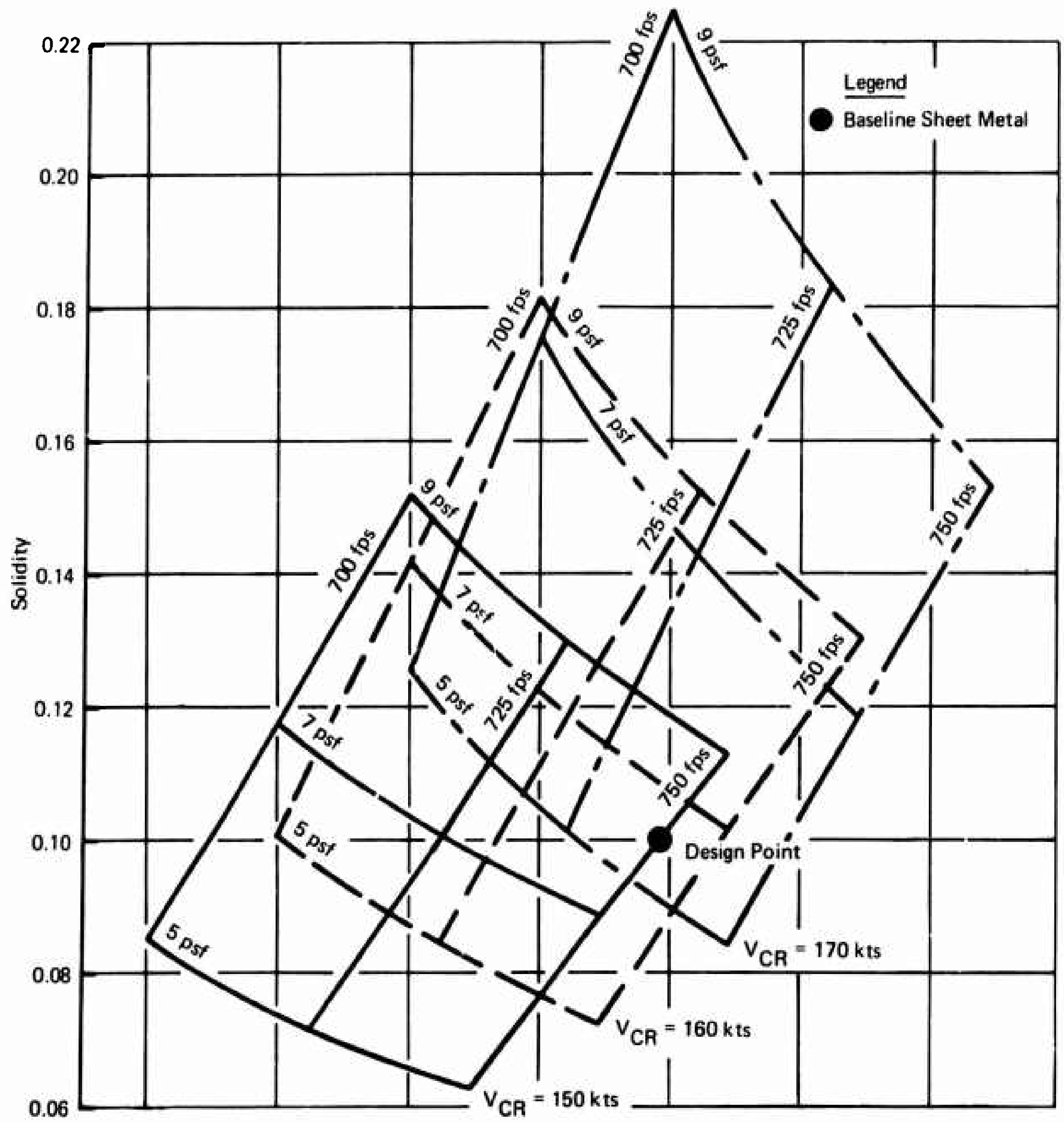


Figure 103. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Solidity.

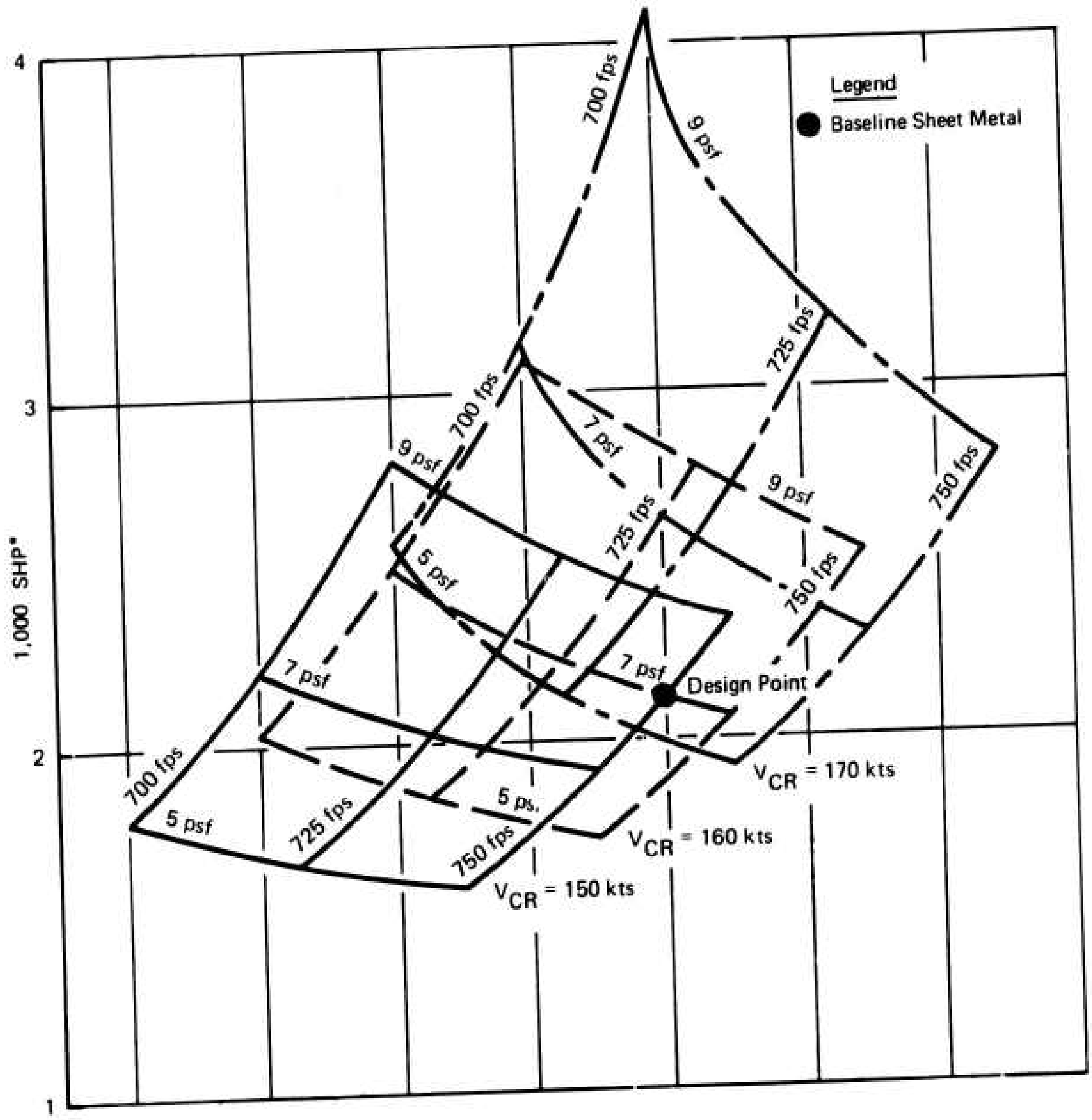


Figure 104. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Initial Power.

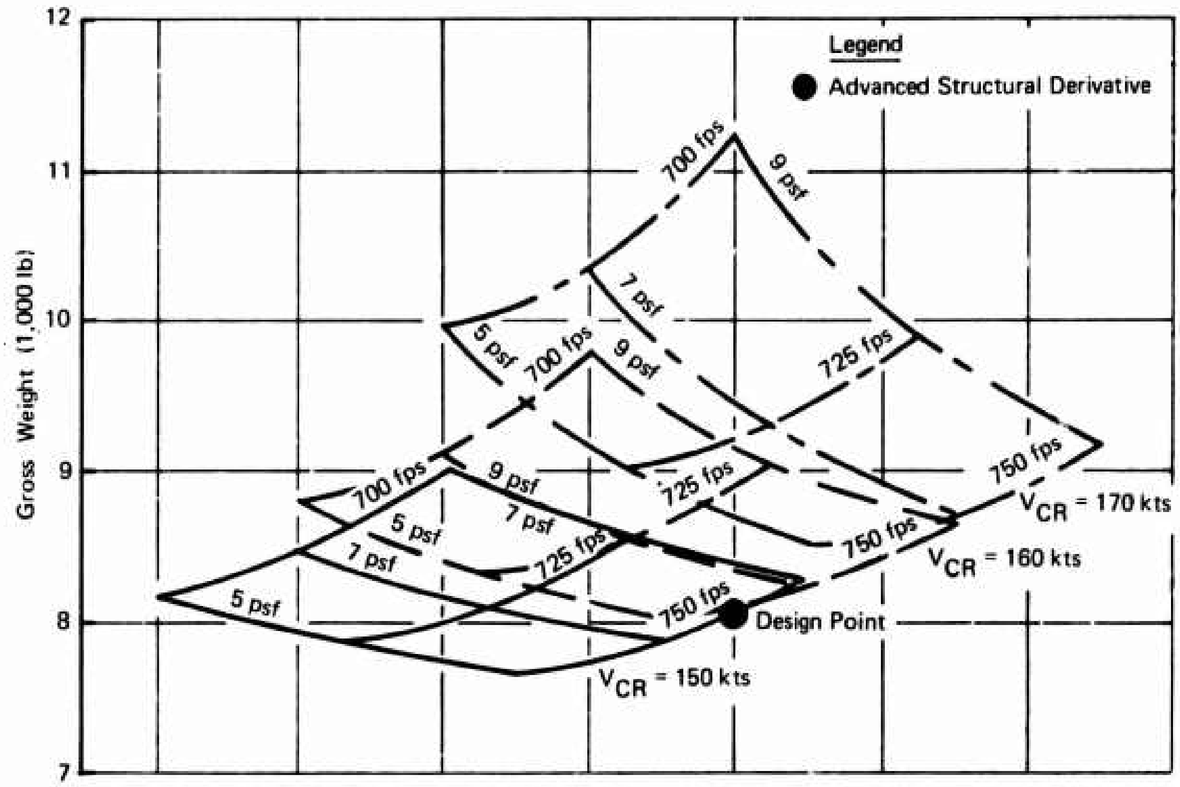


Figure 105. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Gross Weight.

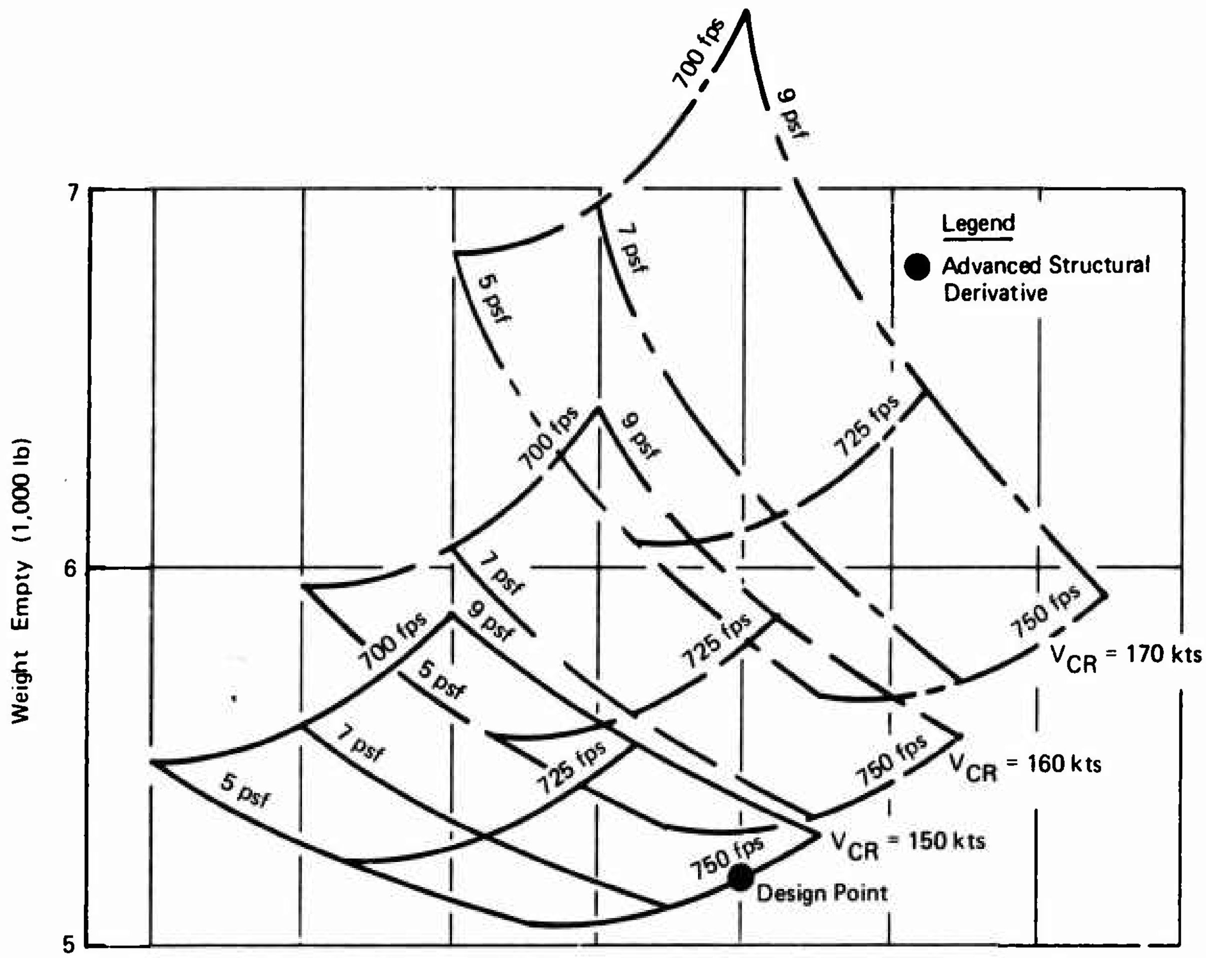


Figure 106. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Weight Empty.

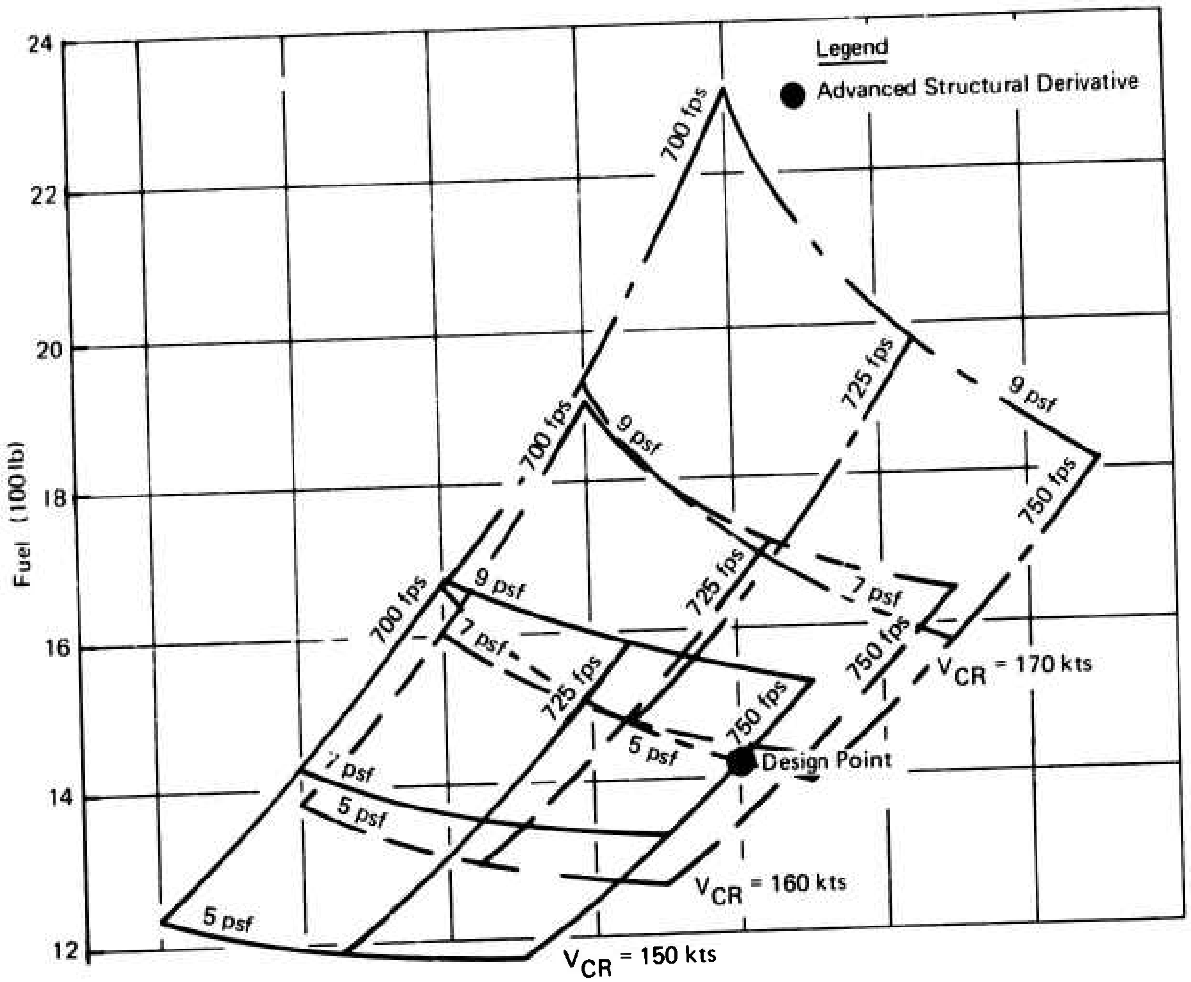


Figure 107. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Fuel.

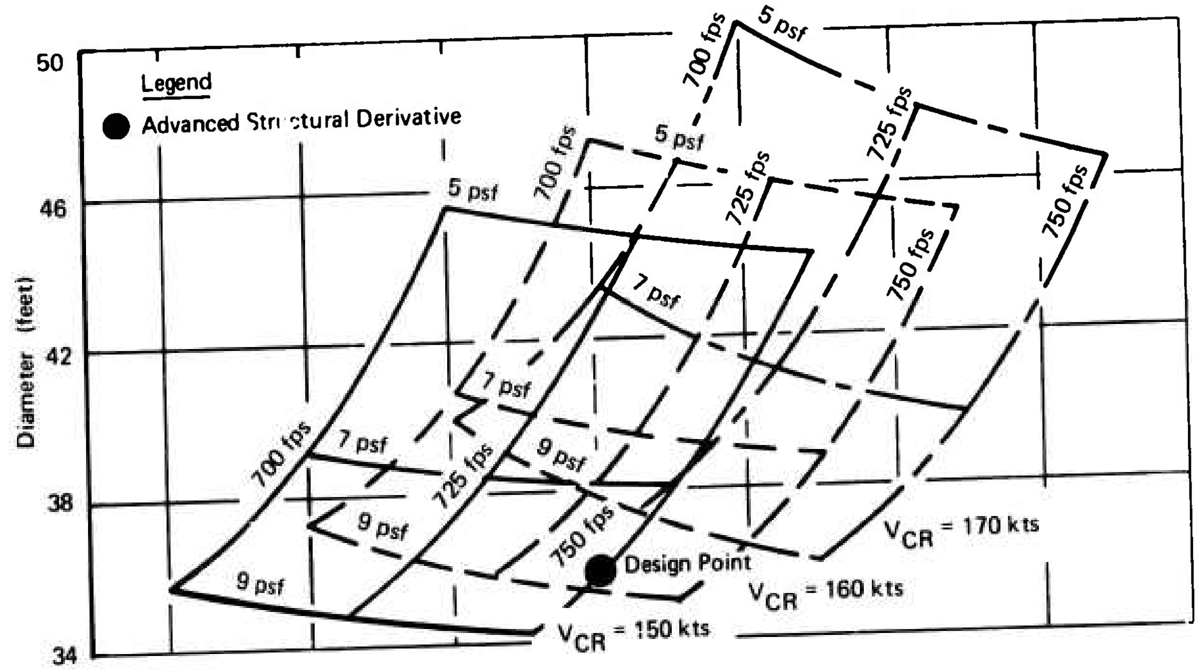


Figure 108. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Diameter.

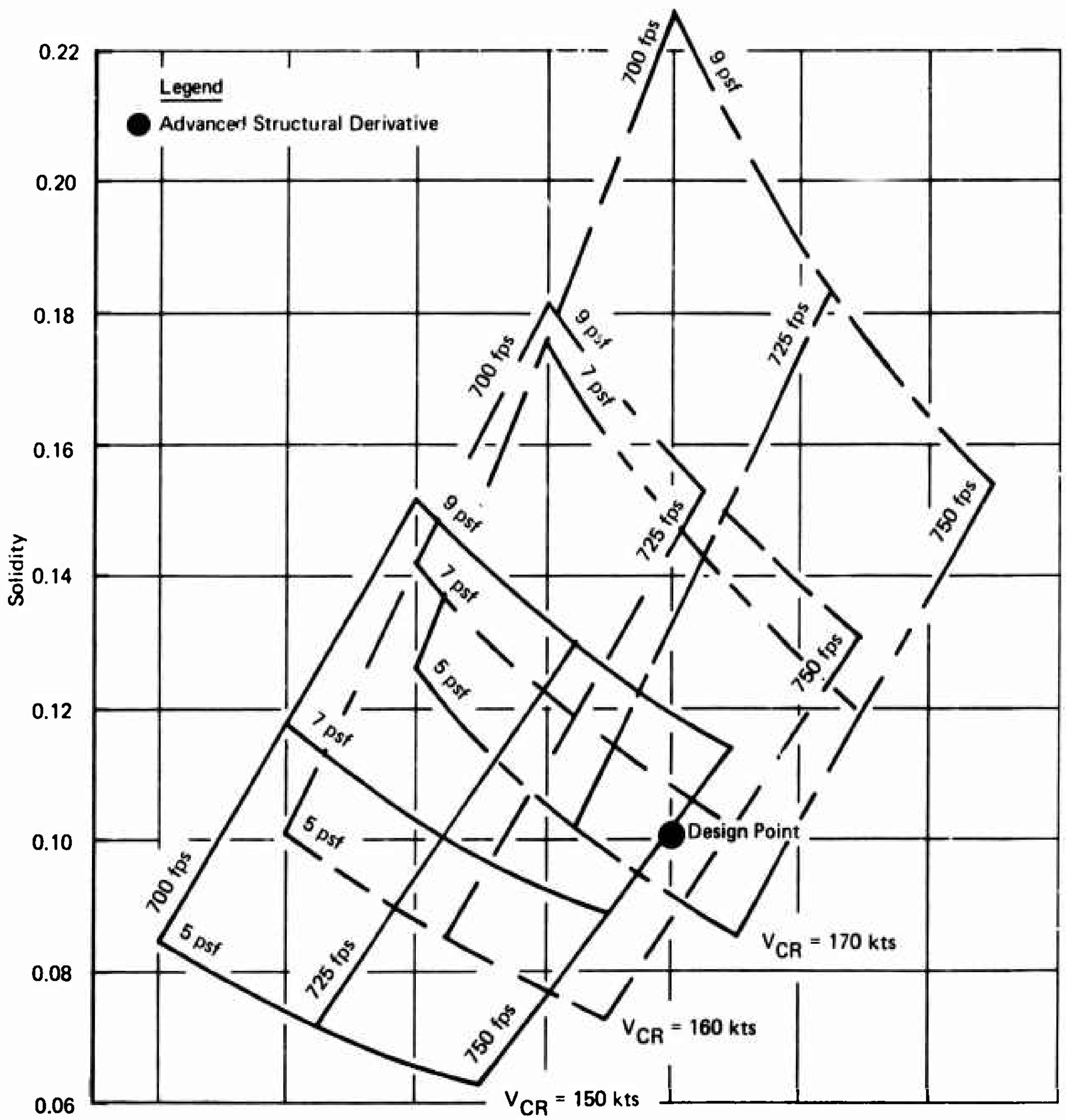


Figure 109. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Solidity.

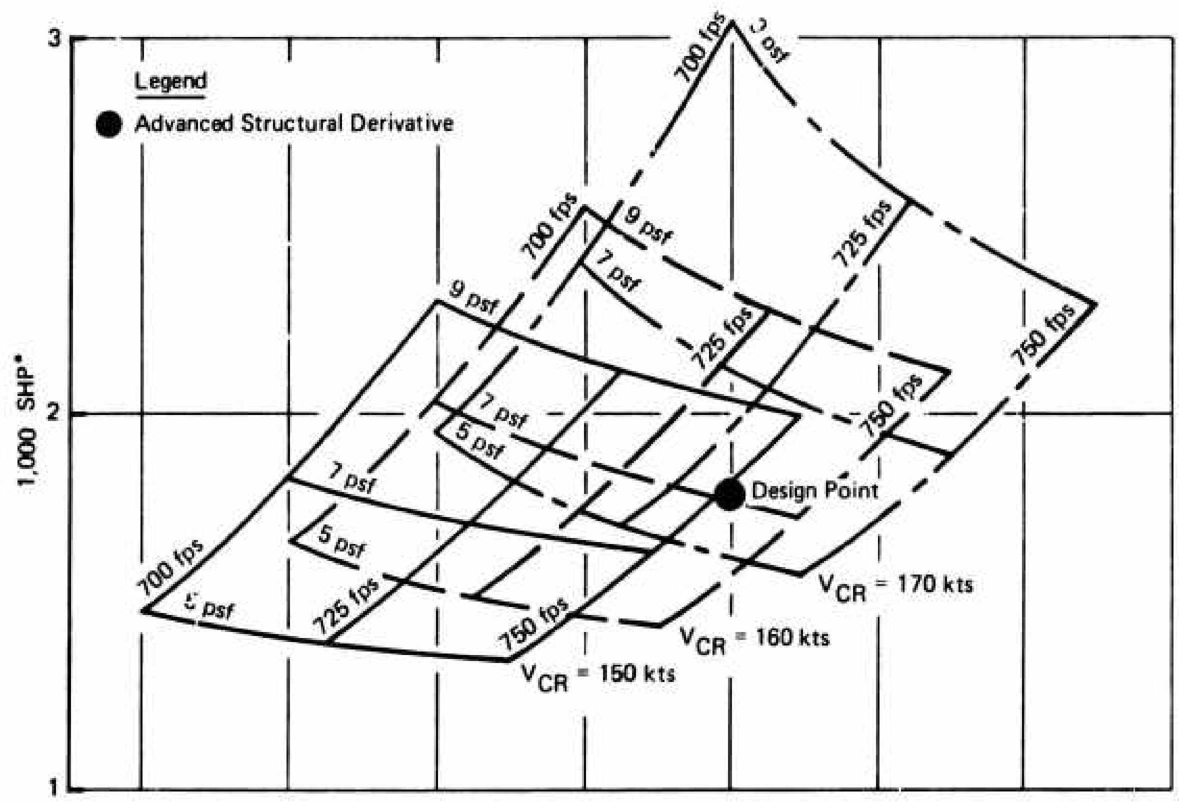


Figure 110. Effect of Tipspeed, Disc Loading and Design Cruise Speed on Initial Power.

5.3 COMPARISON OF VEHICLE SIZE AND PERFORMANCE

5.3.1 VEHICLE SIZE AND WEIGHTS

Table 21 compares the geometric data of the metal and advanced structure vehicles and the advanced structure derivative of the metal baseline helicopter. Because of common geometry, no differences appear between fuselage, tail, and rotor geometry for the baseline and baseline derivative aircraft, as indicated in columns 1 and 2. Selected advanced structure application reduced the weight empty of the baseline metal aircraft from 6431 to 5749 pounds, a reduction of 682 pounds (11 percent).

The effect on vehicle size of beginning the design approach with advanced structure can be seen by comparing columns 3 and 1 of Table 21. Maintaining the design disc loading of 8 psf to minimize rotor diameter, the overall vehicle shows a reduction in gross weight and a main rotor diameter of 35.8 feet. This, in turn, reduces the overall fuselage length to 38.3 feet from 40.5 feet. Total installed power was also reduced due to the 1474-pound saving in gross weight; the resized advanced structures vehicle requires 1797 shp compared with 2122 shp for the metal baseline. This reduced installed power, combined with reduced drag due to smaller size, improved vehicle cruise performance as evidenced by the reduction in design fuel from 1655 to 1422 pounds for the resized vehicle. Figure 111 shows the comparative size of the baseline helicopter versus a resized helicopter utilizing advanced structures.

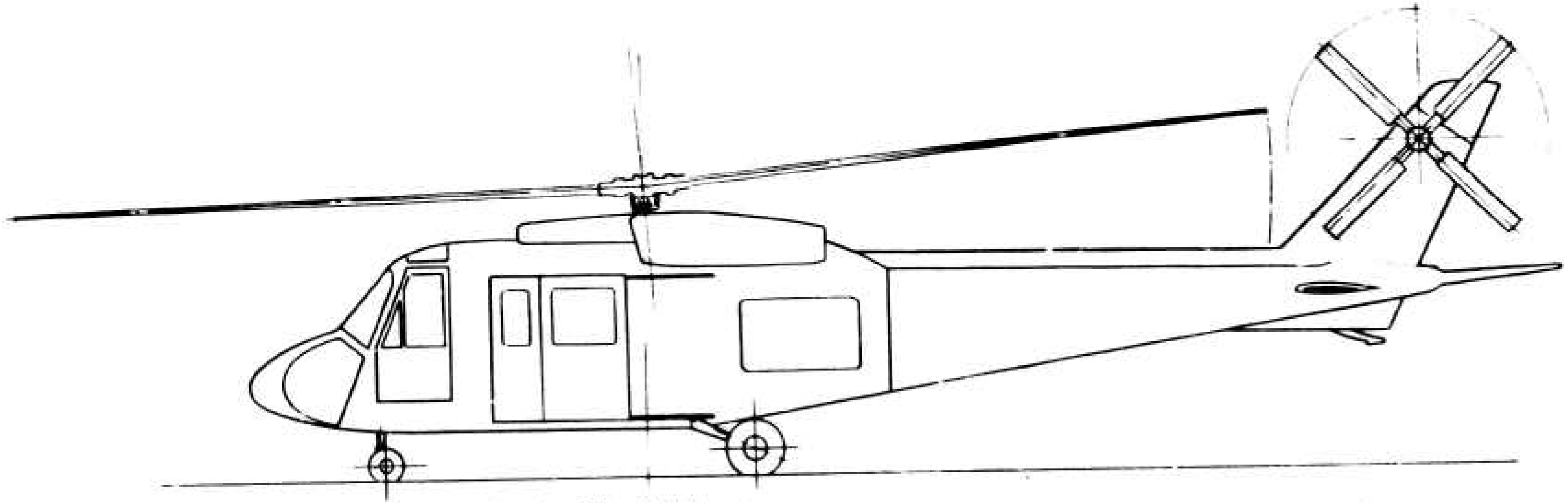
5.3.2 COMPARISON OF VEHICLE PERFORMANCE

Figures 112, 113 and 114 show the performance for the metal baseline, its advanced structure derivative, and the resized advanced structure vehicle. Each figure contains payload radius, comparisons of speed versus altitude, and hover gross weight versus altitude performance.

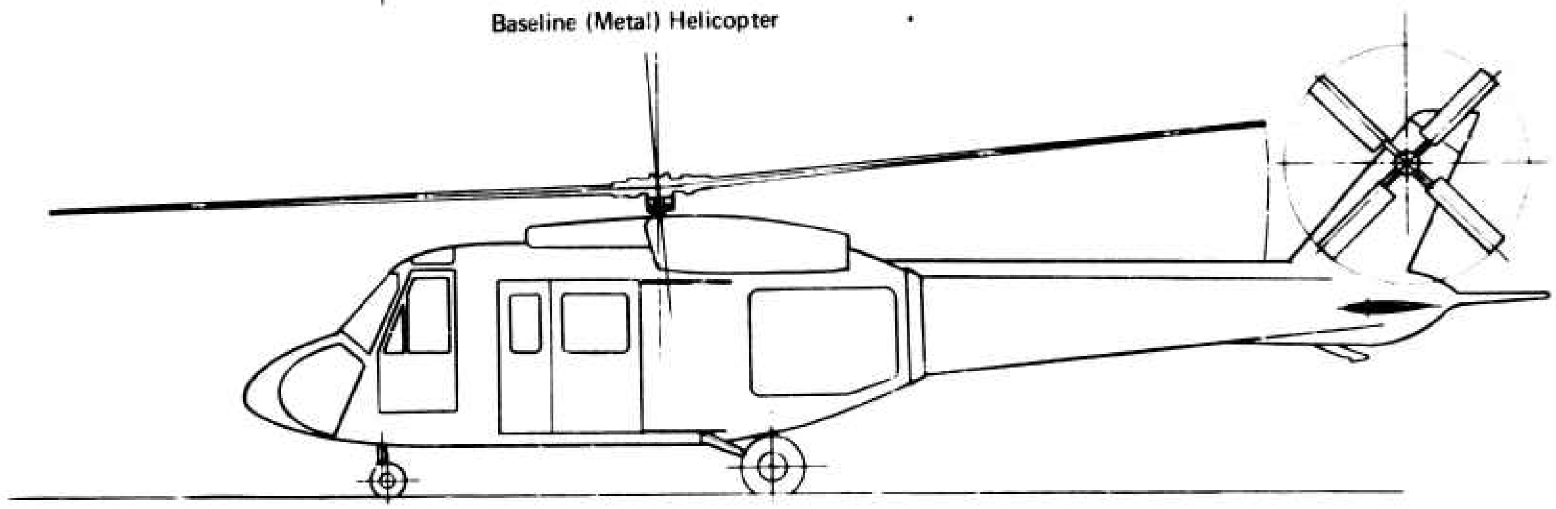
Comparison of the payload range data at design gross weight for all three vehicles indicates that the baseline metal vehicle has slightly greater payload capability at radii less than design radius. This is because of its higher fuel requirement to perform the design mission. Because it has the engines of the metal baseline, the advanced structure baseline derivative has an additional lift capability of 761 pounds at design takeoff ambient conditions (4000 ft, 95°F, 95 percent IRP), as reflected with the broken payload line at takeoff gross weight of 9546 pounds.

Speed-altitude capability is shown for the three vehicles for standard day conditions up to a 12,000-foot altitude. One g level flight maximum speeds are calculated for design gross weights at takeoff power settings of maximum continuous (NRP) and intermediate power settings (IRP). The IRP speeds are restricted below 7000 feet by the transmission torque capacity (transmission limit), which is sized to absorb 1.2 times the IRP power at design takeoff conditions. At 4000 feet the baseline vehicle has a maximum speed of 174 KTAS; its advanced structure derivative, 176 KTAS; and the resized advanced structure vehicle, 172 KTAS.

Hover capability is shown for four primary conditions, 0 and 450 fpm rate of climb on standard day conditions, an ambient temperature of 95°F and power settings at IRP and 95 percent IRP. The design condition is indicated at 4000 feet altitude, 95°F day and 95 percent IRP. The advanced structure derivative of the metal baseline indicates a 761-pound additional lift capability over that required to perform the basic mission. This is a result of the reduced disc loading and the excess installed power available with the metal baseline engines.

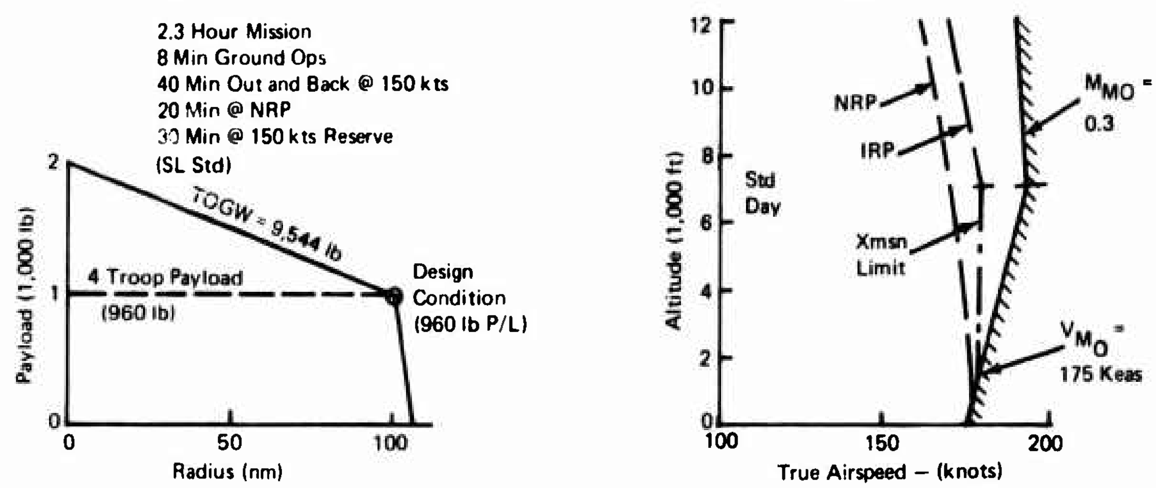


Baseline (Metal) Helicopter



Final Configuration
Composite Helicopter

Figure 111. Size Comparison of Baseline Versus Resized Advanced Structures Helicopter.



Weights

Weight Empty	6,431 lb
Fixed Useful Load	498 lb
Operating Wt Empty	6,929 lb
Payload	960 lb
Fuel	1,655 lb
Takeoff Gross Weight	9,544 lb

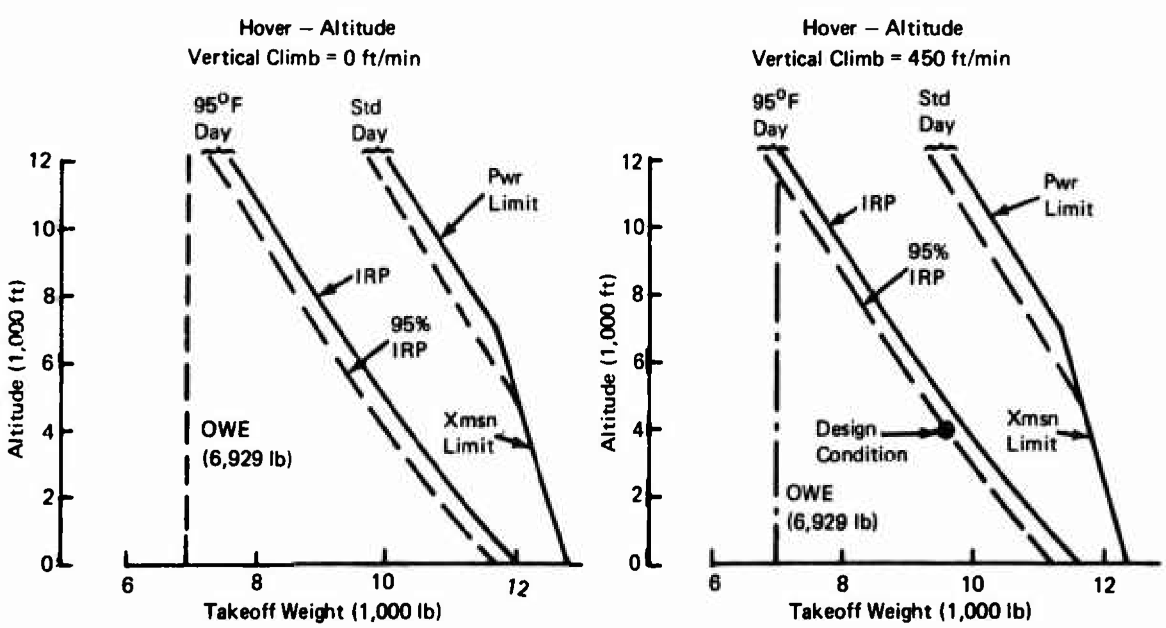
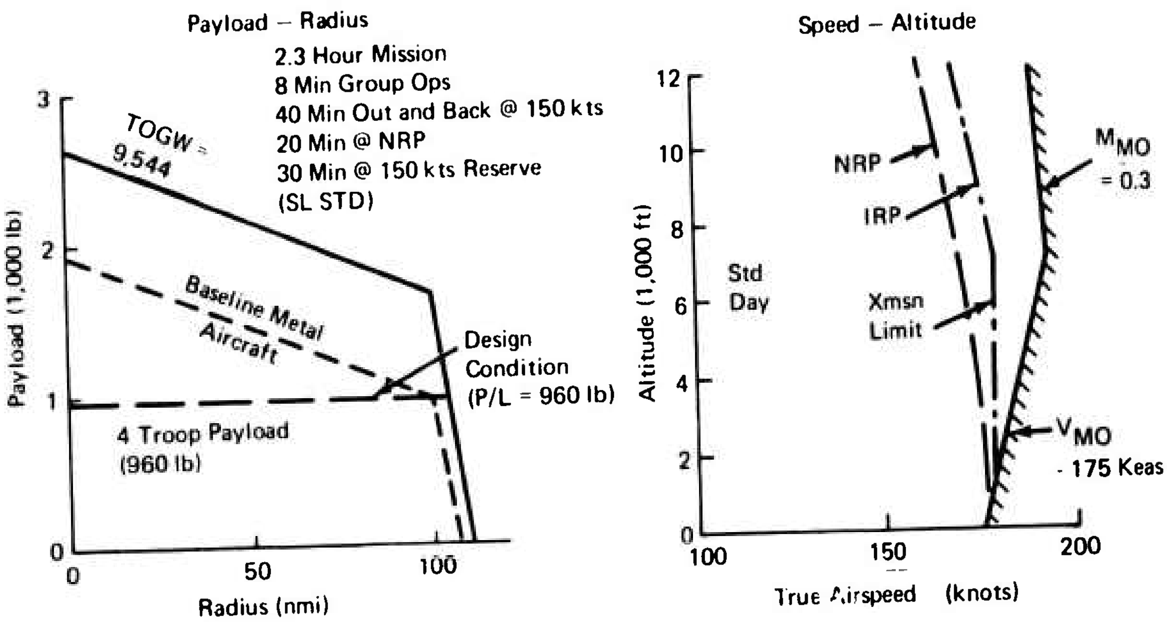


Figure 112. Metal Baseline Aircraft Performance.



Weights

Weight Empty	5,749 lb
Fixed Useful Load	498 lb
Operating Wt Empty	6,247 lb
Payload	1,642 lb
Fuel	1,655 lb
Takeoff Gross Weight	9,544 lb

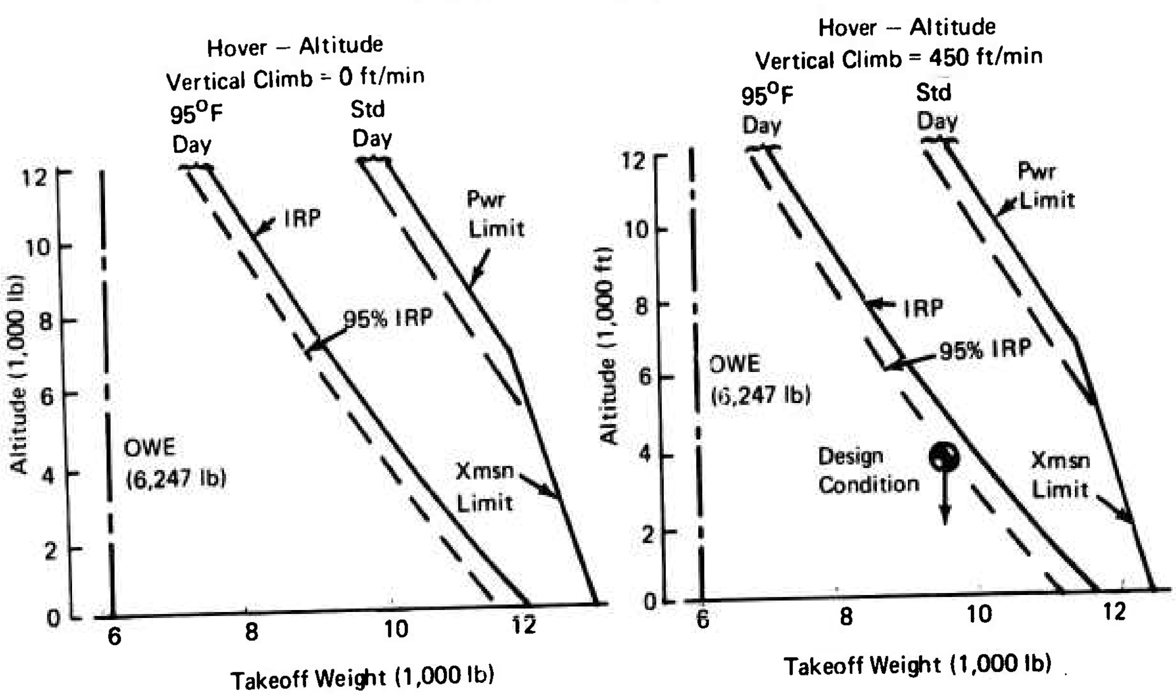
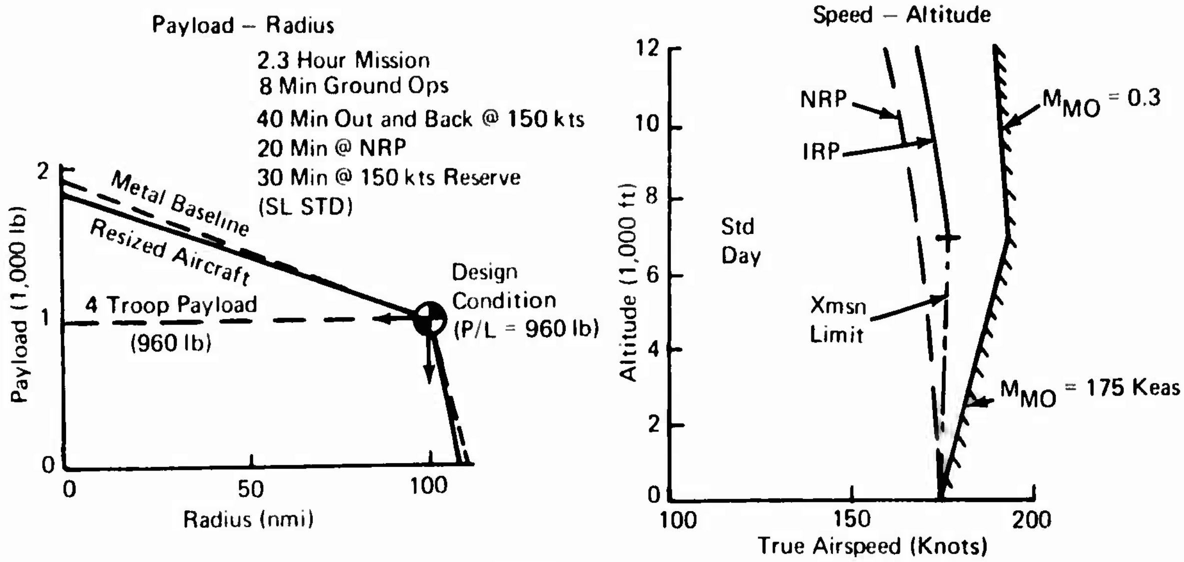


Figure 113. Advanced Structural Derivative Aircraft Performance.



Weights

Weight Empty	5,193 lb
Fixed Useful Load	495 lb
Operating Wt Empty	5,688 lb
Payload	960 lb
Fuel	1,422 lb
Takeoff Gross Weight	8,070 lb

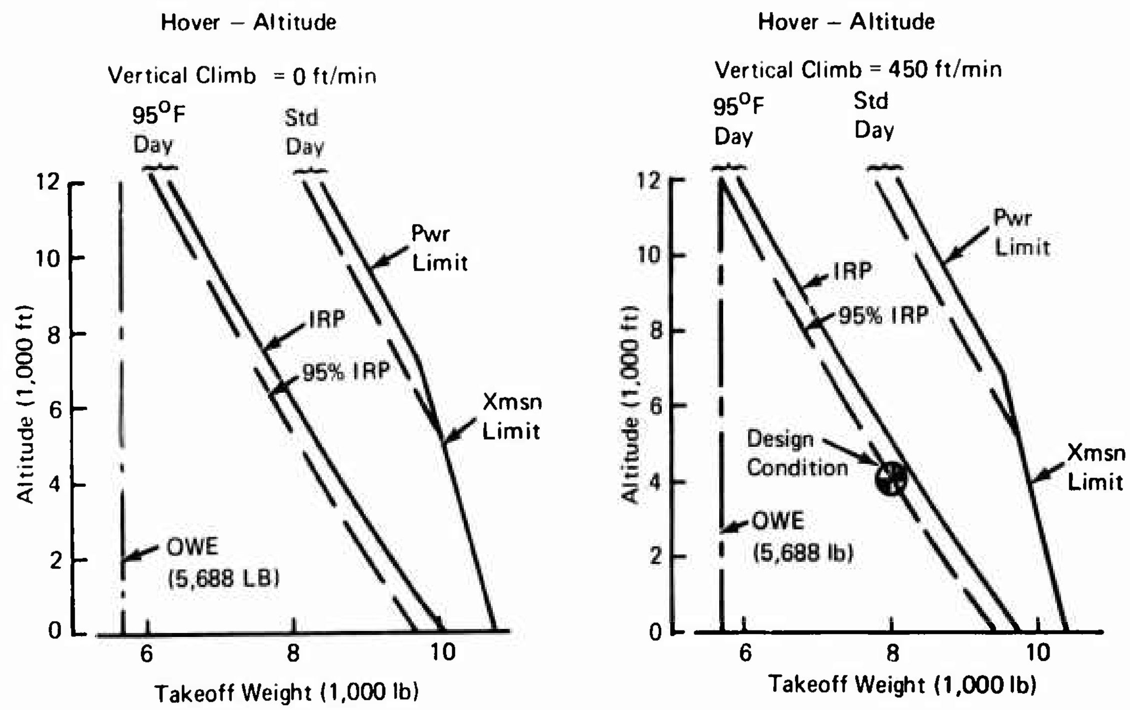


Figure 114. Resized Advanced Structure Aircraft Performance.

5.4 COST ESTIMATING

One of the primary problems in performing this contractual study was the assembling of factual data on fabrication costs of composite structure. Government agencies, led by the Air Force Materials Laboratory, have made cost finding a major R&D task in their composite structures programs.

The potential for cost-competitive composite structure in highly loaded structures in fixed-wing aircraft and missiles is enhanced by sophisticated chemical and mechanical milling and subsequent mechanical attachment of titanium and aluminum plate structures. Most helicopter structures, particularly in the MUT study size, are currently fabricated from light-gauge aluminum sheet metal. If 1974 fabrication costs of $200 per pound are considered as an average cost for highly loaded aircraft structures, an objective of $50 per pound for highly loaded composite structure is considered ambitious. However, with design-to-cost emphasis, airframe structure for U. S. Army helicopters is already in the $50-per-pound range in sheet metal.

A favorable factor is present in helicopter airframe costing, however. The low operating stress levels present in composite structure sized for minimum gauge field serviceability permits use of low-cost automated processes such as drape forming, consideration of thermoplastic matrices suitable for thermal forming, and low temperature curing adhesives (250°F or below).

Data was assembled from Boeing Vertol data (historical and current), Boeing Aerospace data, Boeing Commercial Airplane data, and government and industry data from both published and unpublished sources. A study of the MUT airframe design attempted to correlate comparable structure with in-house cost engineering studies, industrial engineering estimates, manufacturing planning, materials engineering, design engineering, weights engineering, and procurement sources.

Because of the variations in cost estimating factors and support function requirements throughout the aerospace industry, it was decided to concentrate on basic production man-hour estimating and raw material costs. From these basics, comparisons of cost of advanced structure to conventional structure can be expressed as ratios and are useful in determining cost-competitiveness.

Matched metal die production tooling was assumed, with integral heating. No credit was taken for process time reductions using methods such as dielectric curing.

Parts and assemblies were synthesized for cost estimating into three basic configurations (each ideally suited for manufacturing):

- Composite honeycomb panels (Figure 115)
- Composite skin/stiffeners (Figure 116)
- Composite beams and support structures (Figure 117)

Costs in the figures noted above are expressed in manufacturing man-hours per pound. The estimated weights, with their corresponding man-hours per pound in these graphs, yield estimated direct labor man-hours per pound per module. Composite material was estimated from basic 1974 prices predicated on design mix requirements.

The resulting total airframe cost is compared to "design-to-cost" trend curve estimates for sheet metal structure such as that shown for the baseline study aircraft, taking the baseline cost as 1.00 or unity. Comparisons are shown in Table 21.

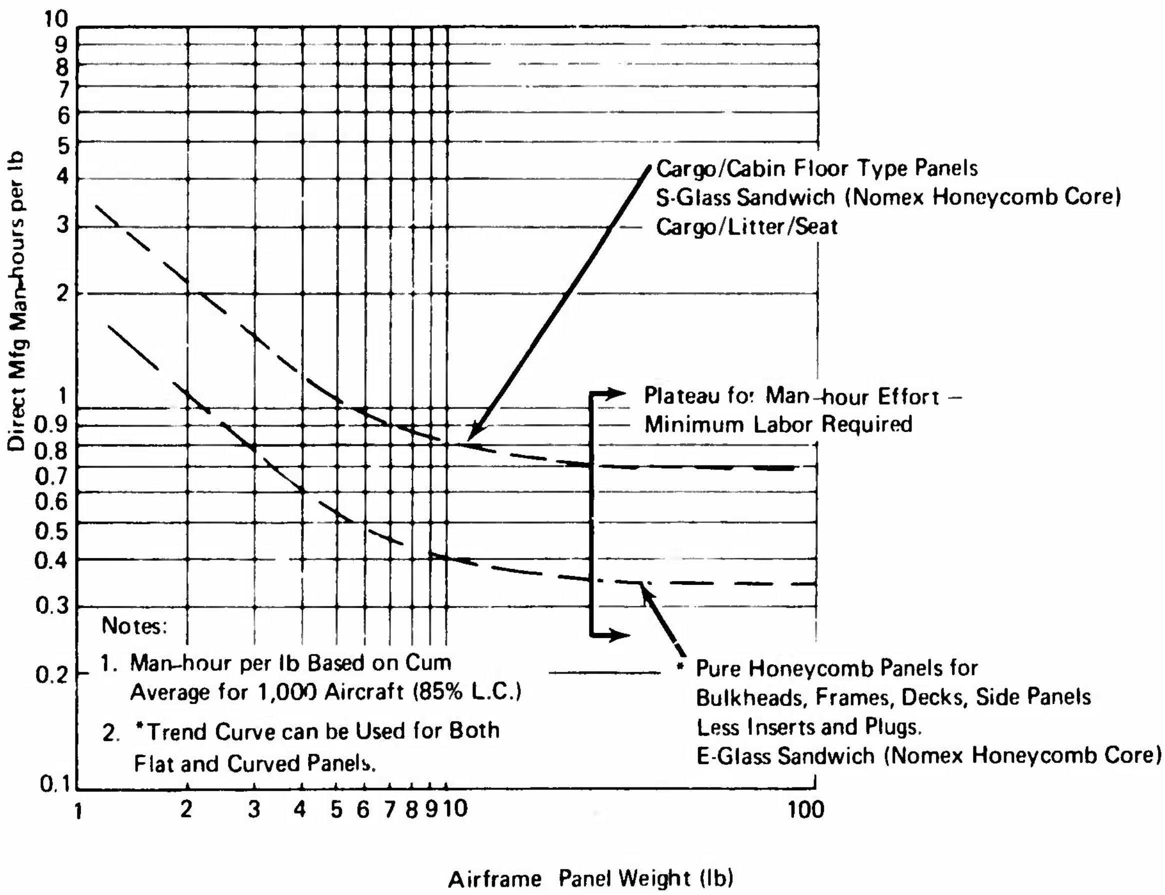


Figure 115. Manufacturing Man-Hours per Pound for Composite Honeycomb Panels of Sandwaich Construction.

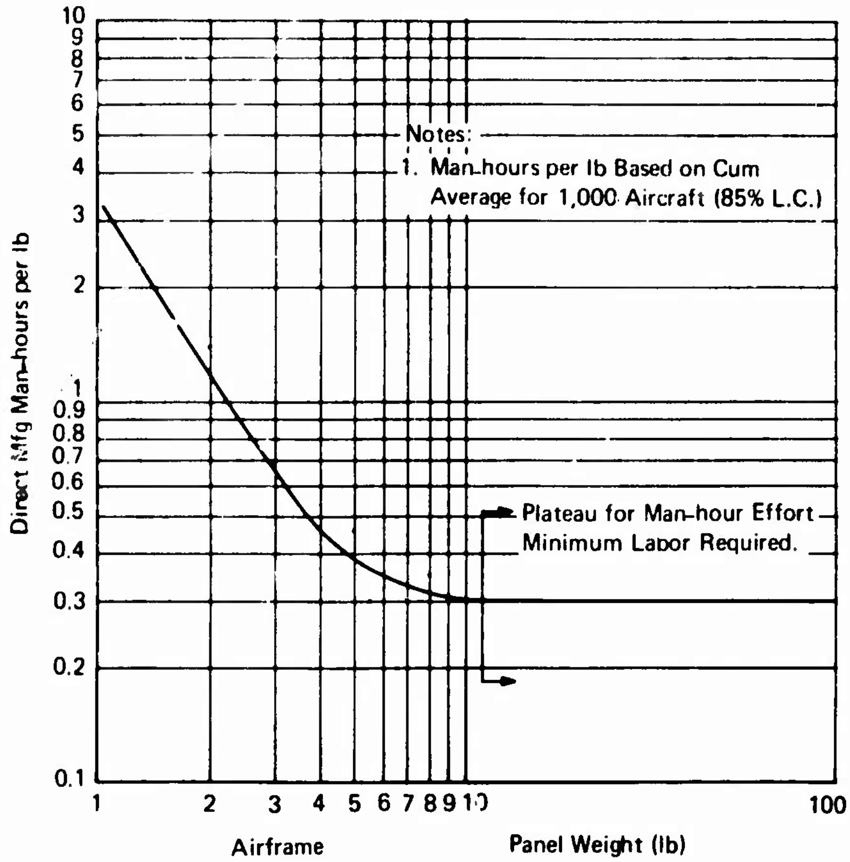


Figure 116. Manufacturing Man-Hours per Pound for Composite Skin/Stiffener Type Parts of E-Glass (Woven, Style 181).

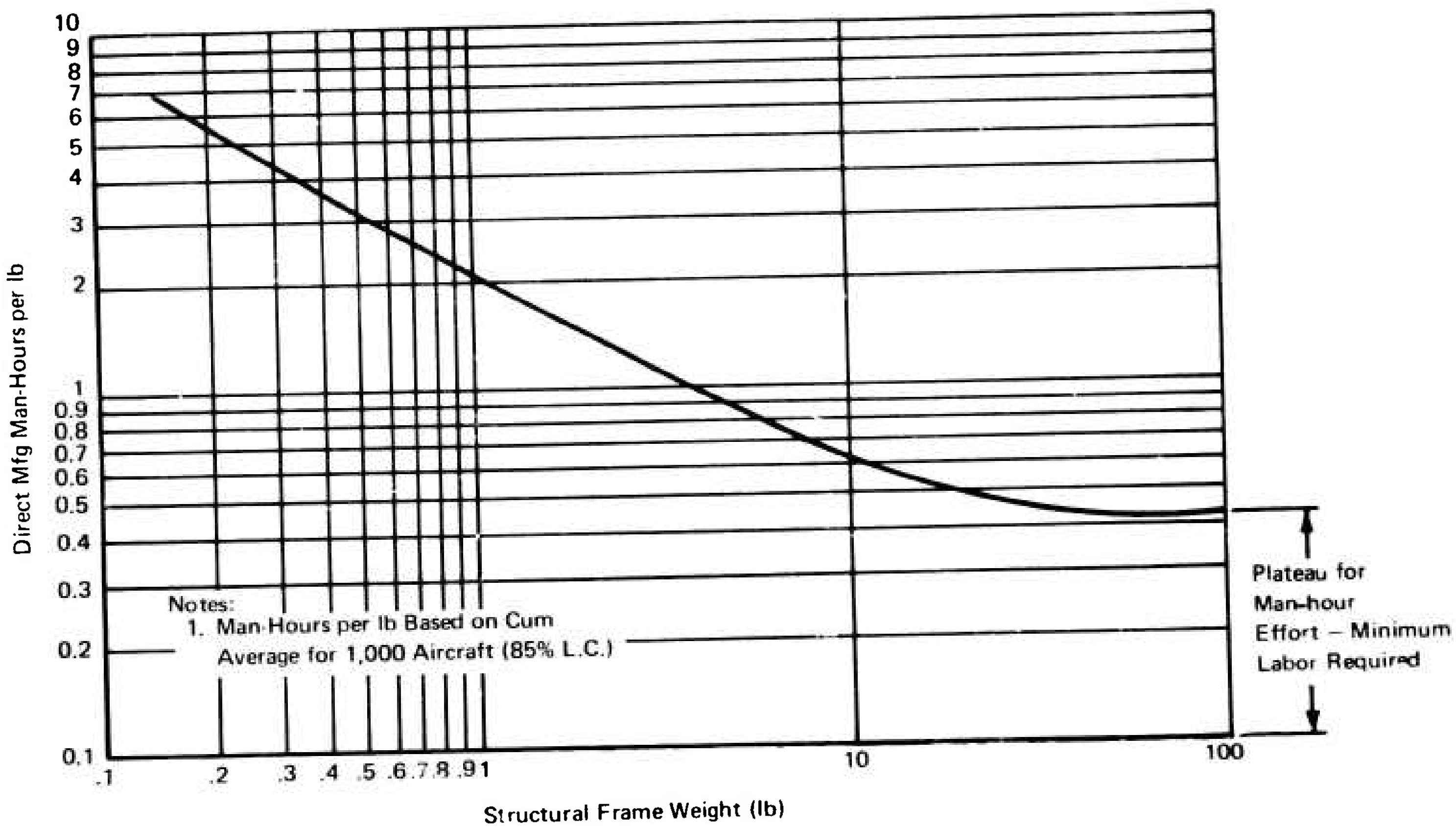


Figure 117. Manufacturing Man-Hours per Pound for Composite Box Frame Structure of Graphite/Kevlar/Epoxy.

5.5 COST ENGINEERING DATA

5.5.1 AIRFRAME

The cost estimate in Table 24 was composed from various historical Boeing and industry sources. A MUT cost methodology has been composed to substantiate the data presented. Basic considerations are as follows:

1. Labor man-hours are direct only (time to make the part) to an 85 percent learning curve experience, developed over a 1000-aircraft production.

2. Material is that type of stock that may be used as purchased to produce a desired part, such as prepreg fabric sheets. The 1974 Boeing Vertol material prices used herein are as follows:

Graphite T-300	$50.00/lb	An avg mix = $30.00/lb
Kevlar 49	$20.00/lb	
Epoxy Resin	$ 1.50/lb	
Nomex Honeycomb	$15.00/lb avg	
Aluminum Reinforced Honeycomb	$ 1.75/lb avg	
Aluminum Sheet Metal	$ 6.00/lb avg	

3. The MUT composite airframe structure final configuration is defined in Figure B-20 (in Appendix B) and in the figures that appear with the airframe module description in Section 4.3.6.

4. Cost is predicated on composite weights of parts and parts count only.

5.5.2 MAIN ROTOR BLADE

The baseline blade is a composite design with a swan-neck spar shape at the root end and a single pin attachment with a machined titanium split root end fitting. Cost estimates are based on the YUH-61 blade, a similar configuration (see Figure 43).

The proposed blade is a composite design, which deletes the swan-neck spar shape, attaches near 20 percent blade radius through two pins, and has no metal fitting (see Figure 46). The design is adaptable to high production automation. Costs were estimated based on studies in progress on the CH-46F rotor blade.

TABLE 24. COST ENGINEERING ESTIMATES FOR MUT AIRFRAME MAJOR ASSEMBLIES (PRIMARY AND SECONDARY STRUCTURE)

NOTES: 1. DIRECT LABOR HOURS AND MATERIAL READY TO FABRICATE.
2. WEIGHTS AND PARTS COUNT SHOWN ARE FOR PARTICIPATING COMPOSITE ITEMS ONLY.

Ref	Aircraft Section	Wt (lb)	Parts Count	Parts /lb	Direct Labor Man-hours per lb	Total Man-hours per Acft	Material Cost ($)	Remarks
1	Cockpit							
	1.1 Fwd box frame	43.0			0.41	17.63	45.00	3 lb honeycomb @ $15.00/lb
							1200.00	40 lb composite @ $30.00/lb
	1.2 Cockpit encl	48.2			3.16	152.06	13.75	2.75 lb honeycomb @ $5.00/lb
							109.00	45.45 lb composite @ $20.00/lb
	1.3 Nose elec door	9.3			0.55	5.12	186.00	9.3 lb composite @ $20.00/lb
	1.4 Pilot doors	33.7			0.21	7.07	85.50	5.7 lb honeycomb @ $15.00/lb
							560.00	28 lb composite @ $20.00/lb
	Unit assy (1.1, 1.2, 1.3 & 1.4)	-			-	35.10		
	Totals	134.2	60	0.5	1.6	216.98	2999.25	
2	Floor Structure							
	2.1 Floor panels	80.3			0.33	26.66	7.00	4 lb Al honeycomb @ $1.75/lb
							150.00	10 lb Nomex HC @ $15.00/lb
							1326.00	66.3 lb composite @ $20.00/lb
	2.2 Floor structure incl fittings	68.2			1.35	92.23	564.00	28.2 lb composite @ $20.00/lb
							800.00	40.0 lb composite @ $20.00/lb
	Unit assy (2.1 & 2.2)					41.10	-	
	Totals	148.5	115	0.8	1.1	159.99	2847.00	
3	Upper Deck & Buttline Beam Assy and Fairings							
	3.1 Deck and beams	132.1			1.35	178.34	396.30	26.42 lb honeycomb @ $15.00/lb
							2113.60	105.68 lb Kev/epy @ $20.00/lb
	3.2 Engine-xmsn fairing	54.7			0.33	18.05	110.00	11 lb honeycomb @ $15.00/lb
							1311.00	43.7 lb gr/Kev/epy @ $30.00/lb
	Unit assy (3.1 & 3.2)					24.60		
	Totals	186.8	70	0.4	1.18	220.99	3985.90	

TABLE 24. Continued

Ref	Aircraft Section	Wt (lb)	Parts Count	Parts /lb	Direct Labor Man-hours per lb	Total Man-hours per Acft	Material Cost ($)	Remarks
4	Bulkhead Sta 163 and Side Panel Assy (incl landing gear attachments)							
	4.1 Bulkhead	33.7			1.35	45.50	120.00	8 lb honeycomb @ $5.00/lb
							514.00	25.7 lb comp Kev/epy @ $20.00/lb
	4.2 Side panels and frames	10.5			0.41	4.31	30.00	2 lb honeycomb @ $5.00/lb
							170.00	8.5 lb comp Kev/epy @ $20.00/lb
	4.3 Ldg gear beams	19.3			1.35	26.05	579.00	19.3 lb comp Gr/Kev/epy @ $30.00/lb
	Unit assy (4.1, 4.2 & 4.3)					14.10		
	Totals	63.5	43	0.7	1.4	89.96	1413.00	
5	Tank Supports and Side Avionics Structure							
	5.1 Tank shell (int)	44.1			0.46	20.29	18.00	3 lb foam @ $6.00/lb
							60.00	4 lb honeycomb @ $15.00/lb
							744.20	37.1 lb composite Kev @ $20.00/lb
	5.2 Avionic structure	30.1			1.35	40.64	30.00	2 lb honeycomb @ $15.00/lb
							564.00	28.2 lb composite @ $20.00/lb
	5.3 Fuel bay frames	12.4			1.35	16.74	372.00	12.4 lb composite @ $30.00/lb
	5.4 Side avionics shells	23.7			0.46	10.61	57.00	3.8 lb honeycomb @ $15.00/lb
							398.00	19.9 lb composite @ $20.00/lb
	5.5 Side avionics doors	21.4			0.46	9.84	30.00	5.0 lb foam @ $6.00/lb
							328.00	16.4 lb composite @ $20.00/lb
	Unit assy (5.1, 5.2, 5.3, 5.4 & 5.5)	-			-	24.50	-	
	Totals	131.7	109	0.8	0.93	122.62	2601.20	

TABLE 24. Continued

Ref	Aircraft Section	Wt (lb)	Parts Count	Parts /lb	Direct Labor Man-hours per lb	Total Man-hours per Acft	Material Cost ($)	Remarks
6	<u>Tailboom and Vertical Stabilizer-Torque Box</u>							
	6.1 Tailboom	89.6			1.38	123.65	309.00	20.6 lb honeycomb @ $15.00/lb
							2070.00	69 lb composite @ $30.00/lb
	6.2 Vert stabilizer	17.7			1.35	23.90	60.00	4 lb honeycomb @ $15.00/lb
							411.00	13.7 lb composite @ $30.00/lb
	6.3 Tail shaft cover	12.5			1.35	16.88	52.50	3.5 lb honeycomb @ $15.00/lb
							180.00	9.0 lb composite @ $20.00/lb
	6.4 Access panels	2.0			1.2	2.4	40.00	2.0 lb composite @ $20.00/lb
	Unit assy (6.1, 6.2, 6.3 & 6.4)					45.00		
	Totals	121.8	32	0.3	1.74	211.83	3122.50	
7	<u>Tailcone Fairing</u>							
	7.1 Tailcone fairing	5.3			0.62	3.29	33.00	2.2 lb honeycomb @ $15.00/lb
							62.00	3.1 lb composite @ $20.00/lb
	7.2 Tailcone structure	3.0			1.35	4.05	90.00	3.0 lb composite @ $30.00/lb
	Unit assy (fairing and structure)	-			-	1.80	-	
	Totals	8.3	6	0.7	1.1	9.14	185.00	
8	<u>Tailbumper and Absorber Assy</u>							
	8.1 Tail skid (arm only)	5.5			1.35	7.43	165.00	5.5 lb composite @ $30.00/lb
	Unit assy (8.1)	-			-	1.00	-	
	Totals	5.5	3	0.5	1.5	8.43	165.00	
9	<u>Horizontal Stabilizer and Actuator Unit Assy</u>							
	9.1	30.2			1.35	40.77	56.25	3.75 lb honeycomb @ $15.00/lb
							793.50	26.45 lb composite @ $30.00/lb
	Unit assy	-			-	6.90	-	
	Totals	30.2	20	0.7	1.6	47.67	849.75	

TABLE 24. Continued

Ref	Aircraft Section	Wt (lb)	Parts Count	Parts /lb	Direct Labor Man-hours per lb	Man-hours per Acft	Material Cost ($)	Remarks
10	<u>Vertical Stabilizer Tip Leading-Edge Trailing-Edge Fairings</u>							
	10.1 Tip fairing	2.8			0.77	2.15	56.00	2.8 lb composite Kev @ $20.00/lb
	10.2 LE fairing	3.0			0.76	2.28	12.00 44.00	0.8 lb honeycomb @ $15.00/lb 2.2 lb composite Kev @ $20.00/lb
	10.3 TE fairing	3.6			0.52	1.87	72.00	3.6 lb composite Kev @ $20.00/lb
	10.4 IGB fairing	4.0			0.28	1.12	80.00	4.0 lb composite Kev @ $20.00/lb
	Unit assy (10.1, 10.2, 10.3 & 10.4)	-			-	5.10		
	Totals	13.4	14	1.0	0.93	12.52	264.00	
11	<u>Cabin Side and Hinged Doors</u>							
	11.1 Cabin side sliding doors	16.2			0.37	6.00	67.50 234.00	4.5 lb honeycomb @ $15.00/lb 11.7 lb composite @ $20.00/lb
	11.2 Cabin side hinged doors	12.5			0.45	5.63	48.00 186.00	3.2 lb honeycomb @ $15.00/lb 9.3 lb composite @ $20.00/lb
	Unit assy (11.1 & 11.2)	-			-	27.10	-	
	Totals	28.7	26	0.9	1.35	38.73	535.50	
	GRAND TOTAL	872.6	498	1.75	14.43	1138.86	18968.10	

5.5.3 MAIN ROTOR HUB

The baseline rotor hub is similar to the YUH-61 four-bladed hingeless (soft-in-plane) configuration and cost estimates are computed proportionally.

The advanced structure hub costs were estimated based on substitution of swan-neck fabrication costs on the YUH-61A blade into hub, and elimination of expensive metal machining, pitch bearings, and tension-torsion straps present on the baseline.

5.5.4 MECHANICAL FLIGHT CONTROLS

The existing bellcranks/quadrants are metal parts machined from bar stock and/or forgings. They may vary from simple to complex configurations.

The proposed bellcranks/quadrants made from composites in match metal mold tooling will decrease machining requirements considerably. A reduction in production labor man-hours of 6 to 25 percent is anticipated. The process considered for manufacturing would be molded composite chopped glass fibers and epoxy, reinforced with uni or fabric elements, as required.

Previous development work on survivable bellcranks/quadrants has shown cost competitiveness with machined castings/forgings, but composite push-pull tubes are not cost-competitive with metal counterparts. The number of parts in the single mechanical system is reduced over the baseline dual system, but the cost of fly-by-wire backup is estimated to offset this cost reduction.

5.5.5 MAIN TRANSMISSION

The cost of the advanced transmission concept has been estimated in relation to the baseline YUH-61A type. Estimating data are provided under separate cover (since they are proprietary at this time).

5.5.6 DRIVE SHAFTING

The baseline interconnect shafting is aluminum alloy tubing, assembled with aluminum and/or steel end adapter and Thomas coupling fittings.

The advanced structure interconnect shafting is composite tubing using an automated fabrication process with integral end adapters. Thomas couplings will join the shafts, and splined steel adapters are still required at box connections. Assume 8 tubes per shipset.

Cost estimation data for 1000 aircraft indicates a potential 20 percent cost reduction per aircraft for the advanced interconnect shafting system.

5.5.7 LANDING GEAR

A cost estimate was attempted based on existing information on metal landing gear costs and assumed first price composite costs from development program information.

Based on the limited data available, and using $50/lb for graphite epoxy and $120/lb for boron aluminum, it does not appear that labor plus composite material costs are competitive with production die-forged metal gear components. Cost competitiveness does appear to exist for small aircraft quantities because of higher machining costs with pancake billet or blocker die raw aluminum stock.

5.5.8 COST SUMMARY

The airframe (fuselage) has been identified as the system where most significant weight and cost savings may be realized (see Figure 13) and consequently the main thrust of cost estimates on MUT is directed toward airframe structure.

The majority of the airframe design features as outlined in Section 4.2 contribute directly to cost reduction; of these fourteen items, the following four appear paramount in driving down overall fabrication costs: modular assemblies; minimum parts count; reduction of mechanical fasteners; and automated processes. In this last item is included match metal die application for cost-effective manufacture of many of the structural components.

Regression analysis for airframe structure has proven that a better correlation exists for parts count than with weight when plotting the relationship to fabrication and assembly hours. Figure 118 compares the two methods.

By taking advantage of all the above cost-saving innovations, an overall reduction in cost of 27 percent over the baseline sheetmetal design is predicted for the airframe subsystem only. Obviously, less efficient cost savings for the other MUT subsystems will reduce the total helicopter composite percentage saving accordingly. The bar graph, Figure 119, compares the airframe labor and material costs for baseline and composite designs. It will be noted that although the actual composite material cost is twice the sheetmetal material cost, this is more than offset by the considerably reduced composite labor cost which is 57 percent of the sheetmetal cost. thus showing an overall airframe cost saving of 27 percent as stated above.

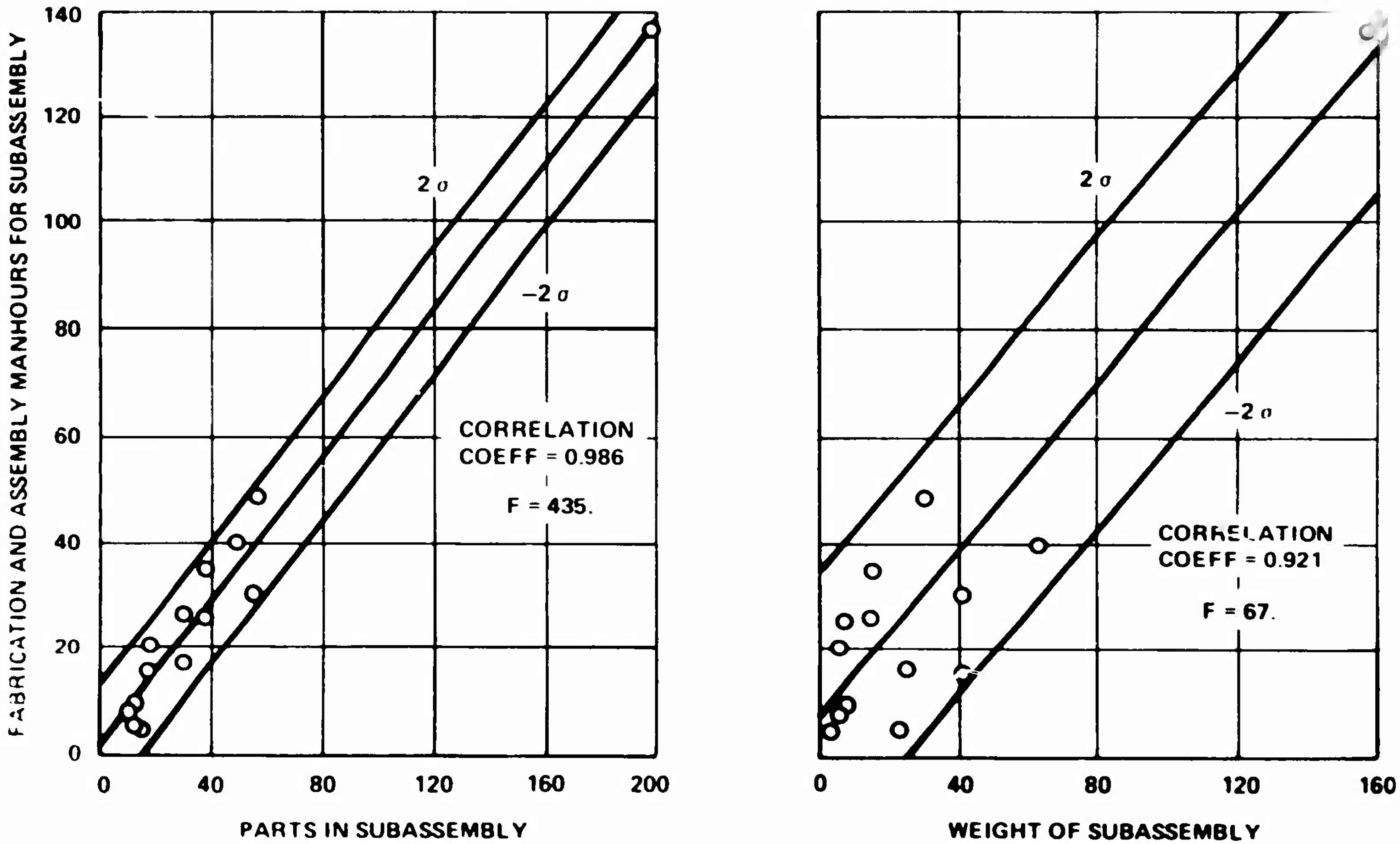


Figure 118. Scatter (Standard Deviation) of Estimate by Parts Count is Only 42 Percent of Scatter of Estimate by Weight.

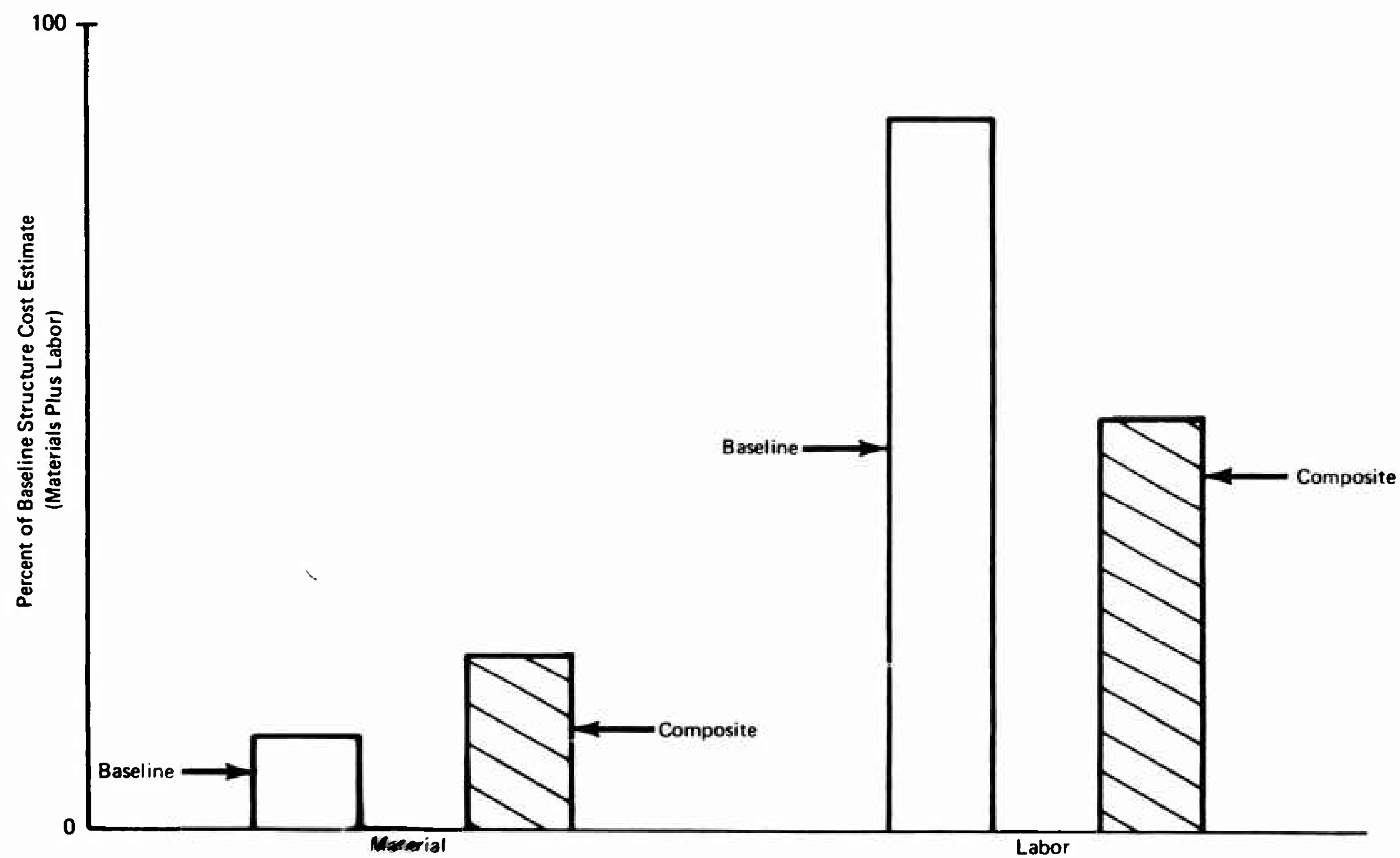


Figure 119. MUT Airframe Cost Element Comparison, Metal Versus Composite.

Composite material prices are projected to lower levels through the next decade and it is anticipated that, with the emergence of developing economical fabrication techniques, a steady downward trend of composite costs will ensue.

6. RISK/FEASIBILITY ASSESSMENT FOR ADVANCED STRUCTURES

A review of the advanced system features selected for the MUT reveals that the principal risk to be relieved in achieving the promised improvements lies in the successful development and demonstration of an all-composite hub, and in load transfer between hub and transmission. A system-by-system discussion follows (and is summarized in Table 25).

6.1 AIRFRAME STRUCTURES

Boeing experience, general industry experience, and government research in honeycomb sandwich airframe structure (both in fiberglass and advanced composites) which guided this study led the investigators to conclude that the anticipated benefits can be achieved with little risk during the 1975-80 time frame. Some of the techniques, such as use of thermoplastic resin systems, are not fully developed for production application, but sufficient R&D and IR&D work has been performed by aerospace industries to assure cost savings with limited risk in the time frame mentioned.

The principal reason that the airframe weight reductions of 30 to 40 percent identified in USAF structural development are not available to the helicopter designer is that the structures are lightly loaded, and airstream structure (skin) is designed to the minimum gauges demonstrated capable of withstanding abuse during U.S. Army operations in unimproved terrain. The fact that the composite structure is thus greatly overdesigned reduces developmental risk and makes low-cost structure fabrication methods feasible.

6.2 STRUCTURAL FITTINGS

Risk factors associated with fabricating attachment fittings for point-load applications are considered moderate, in that the anisotropic properties of the laminated fittings limit shear transfer capability and,hence, render pin-loaded hole concepts vulnerable to increased failure probability when bolt-to-hole fit-up is loose due to manufacturing tolerances, wear, etc. Metal bushings or liners can improve this situation, however, and this approach is currently in use. Another approach is to mold fittings with randomly oriented chopped-fiber reinforcement, thus producing a more isotropic structure. The drawback or penalty here is reduction in tensile and

TABLE 25. RISK/FEASIBILITY ASSESSMENT SUMMARY

MUT System (and Description)	Adequately Demonstrated	Under Development	Risk Factors		
			Low	Moderate	High
Airframe Structure					
Bonded Graphite/Kevlar/Nomex Honeycomb-Modularized Construction	Yes		X		
Structural Fittings					
Reinforced Epoxy Laminates	No	Yes		X	
Chopped Fiber Reinforced Epoxy Moldings	No	Yes		X	
All-Composite Rotor Hub					
Bearingless, Hingeless	No	Yes		X——	——X
Composite Landing Gear					
Wheels, Drag Struts, Braces	Yes		X		
Shock Strut Assemblies	No	Yes		X	
Composite Flight Controls	Yes		X		
Composite Rotor Blades					
Tape Layup	Yes	Yes	X		
Filament Wound	No	Yes			X
Advanced Drive System					
MUT Transmission					
Hub Load Transfer	No	No		X	
Other Advanced Features	Yes	Yes	X		

compressive strength despite improved interlaminar shear strength. Advanced composite technology demonstrations to date have used metal fittings at high load transfer intensity junctures, with a few exceptions. Until more development data is available, use of composite fittings must be assessed as feasible but of moderate risk.

6.3 LANDING GEAR COMPONENTS

A growing number of composite landing gear components have been designed and fabricated in fiberglass/epoxy, graphite/epoxy, boron/epoxy, boron/aluminum, and graphite/polyimide. These component developments have been quite successful, producing work on wheels, drag struts, and torsion members, and some work in cylinder fabrication. Further development in shock strut members would lower the assessed risk; but, in general, composite structure landing gear components are considered low-to-moderate risk elements at this time.

6.4 ADVANCED ROTOR HUBS

Industry- and government-sponsored development of advanced rotor hub concepts is currently in progress both in the U.S. and abroad. Concepts for bearingless, hingeless rotor hubs have been identified, and current applications to tail rotors exist. However, extensive testing of the critical main rotor hub has not yet been performed; consequently, the all-composite rotor hub must be viewed as a moderate-to-high risk area at this time, in relation to near-term application on production helicopters.

6.5 ADVANCED ROTOR BLADES

Glass composite rotor blades using ribbon or tape layup techniques in the critical spar structure have been developed to the point where they are no longer considered a significant risk element by Boeing Vertol or its military customers. Foreign helicopter manufacturers such as Messerschmitt-Boelkow-Blohm, Aerospatiale, and Westland also use fiberglass reinforced epoxy as well as glass epoxy in some structural areas. Risk is assessed as low for this construction.

Filament-wound, filament-reinforced blade structures have been fabricated under U.S. Army contract, and have also been evaluated abroad. These structures have yet to be flight demonstrated, and versions developed to date are judged to have inadequate leading-edge erosion capability to meet the procurement requirements of modern production helicopters. Until the erosion problem is solved and sufficient testing has been conducted to assure weight/balance control and dynamic matching, this promising fabrication method must be considered a moderate risk.

6.6 COMPOSITE FLIGHT CONTROLS

To produce improved reliability, composite control system components have been under development by the U.S. Army for a number of years. The results to date have indicated good strength capability and the capacity to continue operating after significant ballistic damage. Achievement of the survivability objective while maintaining cost equivalence and weight savings over metal components has been demonstrated on bellcranks and quadrants, but no means has been developed to date to fabricate composite push-pull tube assemblies as cheaply as conventional metal. The technical risk in adapting composite control system components is considered low, however. The use of redundant irreversible hydraulic boost actuators which react rotor loads at their source has virtually eliminated high-cycle fatigue as a flight control system problem, and the low operating loads reduce risk in this area even further.

6.7 DRIVE SYSTEM

6.7.1 MAIN TRANSMISSION

Work is proceeding under U.S. Army auspices to conceive, evaluate and develop advanced transmission concepts in gear train, tooth forms, gear steels, bearings, lubrication, failure detection, cooling, housing design and materials, and survivability. The advanced concept transmission selected by Boeing Vertol for application on the MUT uses technology advances previously developed or under development, and combines them in a manner which is clearly advantageous to the MUT preliminary design exercise. The one area which has not been clearly developed and demonstrated is load transfer between the hub and transmission on the compact rotor system selected for the MUT. This is a common risk element between the free planet system and the Boeing Vertol system.

The developmental risk is considered low in all main transmission areas except hub load transfer where the risk is considered moderate, pending demonstration.

6.7.2 SHAFTING

The technical risk in applying composite tail rotor drive shaft segments to the MUT in place of conventional metal is considered low as a result of U.S. Army sponsored development in this area.

The most structurally efficient system, which uses both integral flanges to replace metal adapters and integral flexures to eliminate flexible mechanical couplings, has not yet been developed and demonstrated. The risk factor in specifying such a system is classified as high at this time.

7. CONCLUSIONS AND RECOMMENDATIONS

7.1 CONCLUSIONS

A preliminary design exercise has been completed on a mid-range utility transport helicopter which has resulted in:

1. Definition of a modern state-of-the-art metal baseline design meeting current U.S. Army procurement specifications.

2. Identification of advanced structures in the airframe, drive system, flight control system, and rotor system which are lighter than and cost-competitive with state-of-the-art structure.

3. Evaluation of the performance of a metal baseline aircraft versus the same size aircraft with reduced weight advanced systems.

4. Definition of a helicopter designed from inception with advanced structures, capable of performing the identical mission of the baseline metal configuration.

From the results obtained, it has been concluded that advanced structure - using composite structure modularized airframe construction, composite main and tail rotor blades with bearingless hingeless composite rotor hubs, improved transmission concepts, composite drive shafting and composite control system components - can reduce both the size and the weight of a mid-range utility transport helicopter and be manufactured in production quantities at less cost than a typical metal helicopter designed for the same mission.

The risk in attempting to design and build a helicopter in the advanced structural configuration defined in this study is not excessive - most technology required is at hand, or is under study in ongoing or planned development programs.

More weight savings could be realized in areas such as structural fittings and landing gear, as well as structures in the hydraulic system and equipment, and electrical system, but the savings were not claimed because of lack of sufficient evidence of cost competitiveness in production, or, in the case of structural fittings, lack of development data.

7.2 RECOMMENDATIONS

Further efforts are recommended in the following areas:

1. Further efforts should be made to provide a data base for accurate production cost estimating in

   a. Filament winding costs versus tape layup costs on shafts and tailbooms

   b. Use of advanced structures such as tetracore and geodesic trusswork

2. Further data should be developed on joints and fittings.

3. Production fabrication technology methods should be further developed for use of fiber-reinforced thermoplastics.

4. Further data should be developed in identifying minimum gauge construction of composites in configurations most resistant to simulated field service damage: low velocity impact, abrasion, and hail and gravel impact.

5. Development should be accelerated in the areas of lighter weight, simpler transmission concepts, and hingeless bearingless rotor hubs.

6. The flight control system should be studied in more detail to evaluate the necessity of a fly-by-wire backup system for the single mechanical system with composite components.

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APPENDIX A

SPECIFICATION FOR AN ADVANCED STRUCTURES STUDY FOR MEDIUM-RANGE UTILITY TRANSPORT HELICOPTER (MUT)

1. Requirements.

1.1 System Definition.

1.1.1 General Description. The MUT s[illegible]l be a twin-engine rotary-wing aircraft designed to carry four combat-equipped troops and a crew of two.

1.1.2 Missions. The MUT will perform primary and secondary missions by transporting internal loads under visual and instrument conditions, day and night. External loads will be transported under visual flight conditions.

1.1.2.1 Type MUT missions. The MUT will provide the capability to perform the following type missions:

1.1.2.1.1 Primary Missions. The MUT will be used to transport special teams and/or equipment or supplies and aeromedical evacuation.

1.1.2.1.2 Secondary Missions. The MUT will be used for aviator and troop training, mobilization, development, and new and improved air mobile concepts and support of disaster relief and civic action.

1.2 Characteristics.

1.2.1 Performance Characteristics. The aircraft shall be capable of performing the missions specified herein.

1.2.1.1 Primary Mission. The following performance capabilities shall be met with the aircraft operating at sea level, standard day conditions, unless otherwise specified.

a. A minimum vertical rate of climb from an out-of-ground effect hover at 4000 feet pressure altitude, 95°F temperature conditions, utilizing not more than 95% of intermediate power, of 450 feet per minute.

b. A cruise speed, with not more than maximum continuous power, o[illegible] not less than 150 knots.

c. An endurance with mission fuel plus reserve fuel of not less than 2.3 hours based on the following:

(1) 8 minutes ground operation at idle power.

(2) 20 minutes operation at maximum continuous power.

(3) 80 minutes at cruise speed.

(4) Reserve fuel for 30 minutes at cruise speed.

1.2.1.2 One Engine Inoperative Performance. The following performance characteristics shall be met at primary mission gross weight, using not more than intermediate power on the remaining engine.

a. A level flight speed at 4000 feet, 95°F of not less than 100 KTAS.

b. Shall be capable of making a safe landing at 4000 feet, 95°F.

1.2.1.3 Maneuverability. The aircraft shall be capable of nap-of-earth operation at all air speeds up to 125 knots.

1.2.1.4 Stability and Control. The aircraft shall be designed for optimum stability and control during hovering and airspeeds throughout the flight envelope up to Vmax.

1.2.1.5 Aerodynamics. The aerodynamics of the aircraft shall not restrict pilot-desired attainment of any flight condition within the aerodynamic design envelope shown in Figure A-1. The aircraft shall exhibit safe and comfortable flight characteristics; freedom from vibration and flutter; responsive, effective, and harmonious flight control characteristics; and acceptable flying quality characteristics. At design gross weight, the rotor disc loading shall not exceed six to eight psf. The aircraft should be as externally smooth, aerodynamically faired and contoured as possible to reduce the overall drag.

1.2.2 Physical Characteristics.

1.2.2.1 Weights.

1.2.2.1.1 Weight and Balance Classification. This aircraft shall be classified 1A as defined in MIL-W-25140.

1.2.2.1.3 Operating Weight Empty. Operating weight empty shall be weight empty plus unusable fuel and oil, engine oil, miscellaneous mission equipment, and crew of 2. Operating weight empty shall be presented for both the special teams and aeromedical evacuation missions.

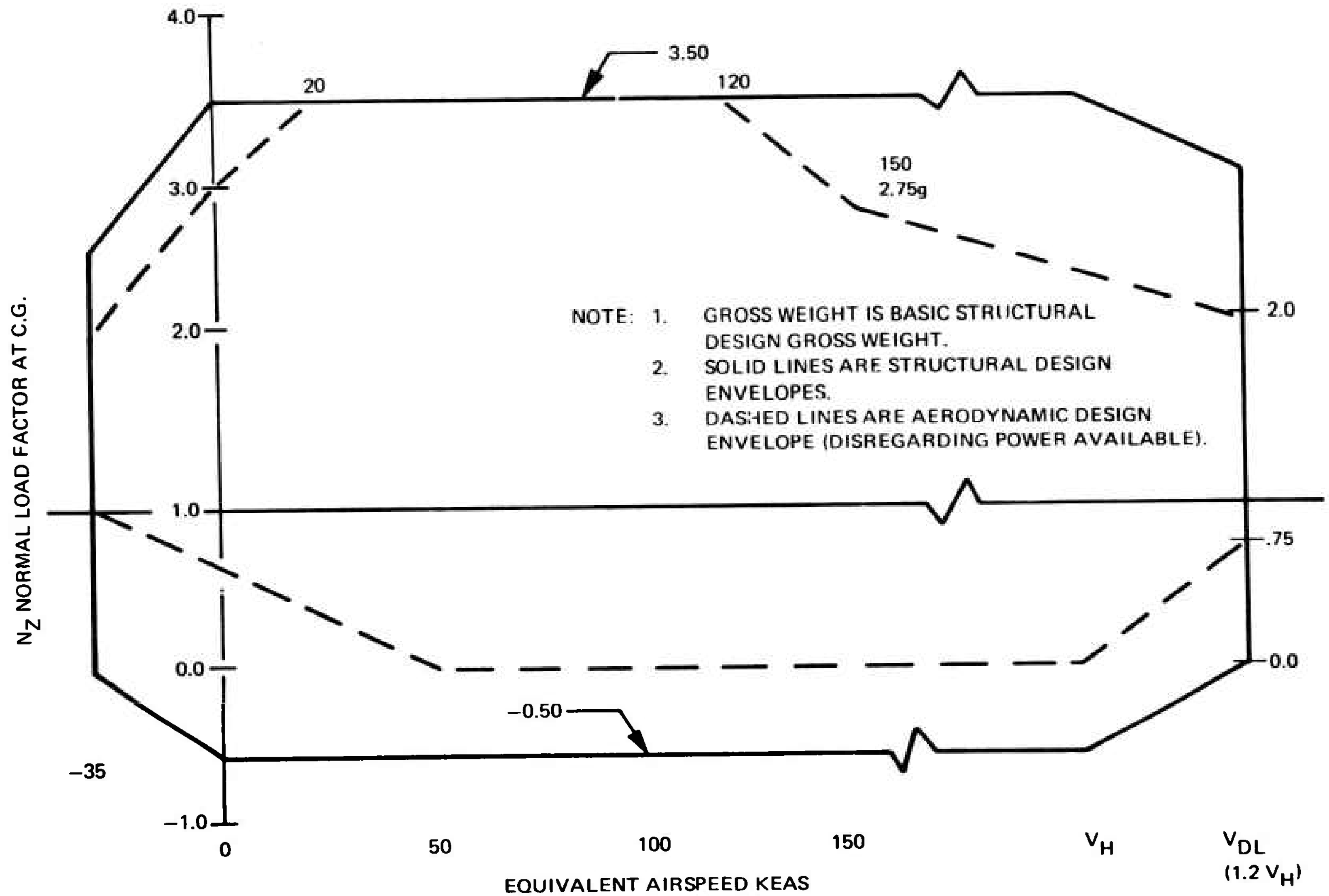


Figure A-1. MUT Design Envelopes.

1.2.2.1.4 Useful Load. Useful load shall consist of specified combinations of passengers/cargo and usable fuel.

1.2.2.1.4.1 Crew. The crew shall consist of a pilot and copilot.

1.2.2.1.5 Design Gross Weight. The design gross weight is the sum of the payload (4 combat equipped troops) and primary mission fuel load of 1.2.1.1 and the operating weight empty of 1.2.2.1.3.

1.2.2.1.6 Unit Weights. The following unit weights shall be used.

Crew weight (each) (including normal flight clothing, gloves, helmet, first aid packet, exposure suit, pistol, holster, ammunition, knife, and armor vests) -	235 lb
Combat equipped troop -	240 lb
Litter patient (includes 25 lb for litter, splints, and blankets) -	265 lb
Passenger -	200 lb
Medical attendant -	200 lb
Medical equipment -	100 lb
Fuel weights (per gallon) (grade JP-4) -	6.5 lb
Oil weight (per gallon) -	7.5 lb

1.2.2.1.7. Transportability. Physical dimensions which allow 2 MUT to be loaded aboard the C-141 aircraft and 1 MUT aboard the C-130 aircraft are required. Preparation time for loading aboard C-141 and C-130 aircraft shall not exceed 5 man-hours within a 1.5-hour period. Provisions which allow for reassembly at destination not to exceed 5 man-hours within a 2-hour period are required. Physical dimensions which allow loading into the C-5 aircraft with only rotor blades folding or removal are required. Rotor blade reinstallation shall not exceed 1 hour upon arrival at destinations.

1.2.2.2 Structural Design Criteria.

1.2.2.2.1 Strength Requirements. Unless otherwise specified, strength and rigidity shall be provided in accordance with MIL-S-8698 for a Class I aircraft. The primary structure including dynamic components shall be damage tolerant to the extent specified in 1.2.2.2.3. Other specific strength requirements are as follows:

1.2.2.2.1.1 Basic Structural Design Weight. The basic structural design gross weight for structural design purposes shall be the design gross weight of 1.2.2.1.5.

1.2.2.2.1.2 Floor Loading. Design limit floor pressures shall be 75Nz psf for crew floors and 300 Nz psf for cargo areas. The load factors Nz shall be 3.50 plus increments due to angular accelerations from maneuvers, or the load factor resulting from hard landings up to 10 feet per second, whichever is greater. In addition, the cargo floor shall have a local loading capability for a 50 psi limit load applied to a single 0.5-square-foot area (8 x 9 inches to 3 x 24 inches) within any 6-square-foot area.

1.2.2.2.1.3 Landing and Ground Loads. At the basic structural design gross weight, the design limit sinking speed shall be 10 feet per second at level ground contact and 6 feet per second for contact on any 15-degree slope.

1.2.2.2.1.4 Yield Factor of Safety. The yield factor of safety shall be 1.0.

1.2.2.2.1.5 Repairability. The structural design shall be such that repair of structural damage from .30 caliber APM2 projectiles (tumbled) with a muzzle velocity of 2750 fps and target distance of 100 yds. can be performed without requiring master jigging or special tools not normally found at direct support maintenance.

1.2.2.2.2 Fatigue. Safe-life design shall be employed as the primary means of satisfying the useful life requirements of 1.2.8. In addition, damage tolerance concepts shall be applied as a design requirement as specified below for primary structure vital to the integrity of the vehicle or the safety of personnel. A fatigue failure shall be defined as any crack caused by repeated loads which is detectable by state-of-the-art nondestructive inspection techniques.

1.2.2.2.3 Damage Tolerance. The primary structure (as defined in 3.12 of MIL-A-008860), shall incorporate materials, stress levels, and structural configurations that will minimize the probability of loss of the aircraft due to damage of a single structural element (including control subsystem and dynamic components) or due to propagation of undetected flaws, cracks, or other damage. Slow crack growth, crack arrestment, alternate loadpaths and systems, and other available principles shall be used to achieve this capability. The specific requirement for damage to flight essential structural components is that they shall preclude or accept damage from a .30 caliber APM2 projectile (tumbled) with a muzzle velocity of 2750 fps and target distance of 100 yds. and still be capable of supporting design limit loads without failure (yielding is allowed for this condition). The aircraft shall also be capable of full continuous operation until safe completion of the mission.

1.2.2.2.4 Crashworthiness. Crashworthiness shall comply with MIL-STD-1290(AV).

1.2.3 Reliability.

1.2.3.1 Aircraft Mean-Time-Between-Failure (MTBF). The aircraft shall be designed to have a MTBF of not less than 39 aircraft flight hours between mission aborting failures.

1.2.3.2 Mean-Time-Between-Removal (MTBR). Aircraft dynamic components shall have a MTBR, both for scheduled and unscheduled removals for overhaul, repair or inspection of 1500 aircraft flight hours.

1.2.4 Maintainability. Preventive and corrective maintenance tasks shall be assumed to be conducted by Army personnel with a skill level equivalent to that of an Army aircraft maintenance school graduate with 6 months on the job experience. Repair tasks or downtimes attributable to enemy action or operation of equipment outside of prescribed limits are excluded from stated maintainability requirements. However, to enhance repairability and reduce aircraft downtime due to minor crash or battle damage that is repairable at the direct or general support maintenance levels, consideration throughout engineering development should be given to providing maximum accessibility for repair of airframe structure without removal or requiring minimum removal of installed equipment. Use AMCP 706-134 as a design guide.

1.2.4.1 Mean-Time-Between-Maintenance.* The mean-time-between-maintenance of the aircraft for preventive and corrective maintenance shall not be less than 3.5 flight hours. Preventive maintenance includes daily, periodic, and special inspections of aircraft components plus any scheduled lubrication oil changes, but excludes preoperative and post operative inspections and servicing for mission turn-around. Daily inspection procedures shall be designed to allow the aircraft to be operational until periodic inspection is performed. Intermediate inspections shall not be required. The scheduled time between periodic inspection shall be at least 300 flight hours. Daily inspections can be assumed to occur every three flight hours, with the exception that no daily inspection is required when a periodic inspection is performed. Flight hours are to be measured from time of aircraft lift-off until touch down.

1.2.4.2 Replacement. Replacement of each major component shall require not more than 3.0 hours.

*Requires clarification of MTBM if daily inspection is included.

1.2.5 Availability. Based on a monthly flight program of 120 hours, an operational availability rate of 80 percent is required.

1.2.6 Environmental Conditions. The aircraft will be subjected to worldwide extremes of climate and weather. Specific values for worldwide climatic extremes of temperature, humidity, rain, snow, sand, dust and other environmental factors shall be in accordance with MIL-STD-210, AR 70-38 and the operating environments shall be as specified below for minimum operating extremes of aircraft subsystems.

1.2.6.1 Storage. The aircraft and its equipment shall not be permanently impaired by storage in temperatures ranging from $-80^{o}F$ to $160^{o}F$.

1.2.6.2 Operation. The aircraft shall be designed for operation throughout the range of $-65^{o}F$ to $125^{o}F$, but simple modification may be necessary for operations below $0^{o}F$.

1.2.7 Survivability and Vulnerability. The aircraft shall incorporate design features to:

a. Reduce signature detection (noise, radar and infrared).

b. Reduce vulnerability of crew, critical subsystems and components.

c. Increase crashworthiness of airframe and subsystems.

To enhance survivability, the MUT shall be capable of conducting nap-of-the-earth flight during daytime visual conditions using limited navigational equipment and pilotage.

The use of integral or kit armor to achieve the desired ballistic protection should be accomplished only after:

d. All other protective design techniques for reduced vulnerability have been fully exploited.

e. The design configuration of the armor has been carefully adapted to miniaturization and concentration of the item being protected.

f. A cost/weight design trade-off study has been made for selection of the basic armor material and the armor concept selection.

1.2.7.1 Ballistic Protection. The ballistic protection of the aircrew shall be V_{05} protection against caliber .30 APM2 projectiles fired from a range of 100 yards, muzzle velocity of 2750 fps, and 0^{o} obliquity. Ballistic protection for the lower 180° hemisphere of critical components and subsystems shall be for the same threat as for the aircrew except that V_{50} protection shall be provided.

1.2.7.2 Infrared (IR) Suppression. The total infrared radiation signature of the aircraft shall be reduced to a level achievable with current state-of-the-art technology. The weight of the entire IR suppressor, including blowers, shields, etc., will be chargeable to empty weight.

1.2.7.2.1 Performance Loss. The ground rules for calculating aircraft performance with IR suppressor installed are as follows:

a. Loss of power due to turbine back pressure will not exceed 1 percent of power required to HOGE 4000 ft. 95°F at design gross weight.

b. Allowance for the loss of power for complete installation of IR suppression shall not exceed 3 percent of the power required to HOGE 4000 ft. 95°F.

c. Allowance shall be made for a 3-percent increase in engine specific fuel consumption (SFC) due to IR suppression. This allowable loss includes losses attributed to tailpipe losses (i.e., turbine back pressure), cooling air source and blower requirements, and installation weight.

d. Any device(s) provided to suppress the IR signature shall not prevent aircraft performance requirements from being obtained.

1.2.7.2.2 Engine Compartment Cooling. The installation of the IR Suppressor shall not result in engine compartment temperatures exceeding those inherent without the IR Suppressor kit installed.

1.2.7.3 Reduction of external noise. State-of-the-art design techniques shall be utilized to minimize the noise radiated from the aircraft. Particular attention shall be given to noise from blade slap, rotor rotation vortices, and gearboxes.

1.2.7.4 Radar Cross Section. Maximum cost-effective application of design techniques of the time frame shall be utilized to minimize the helicopter radar cross section. Primary effort shall be devoted to materials and design which will provide favorable transparency, absorption, and reflectivity. While self-protection to counter all potential radar threats is impractical, the combination of minimum cross section coupled with low altitude flight tactics or the use of optimum electronic countermeasures (ECM) techniques will minimize detection.

1.2.8 Useful Lives.

1.2.8.1 Retirement Life. The minimum retirement life of the MUT airframe shall be 15,000 operating hours.

1.2.8.2 Fatigue Life. The minimum fatigue life of all life limited dynamically loaded components shall be 4500 operating hours. The complete flight profile shall include an appropriate distribution of different types of maneuvers and a rational distribution of other significant parameters which affect oscillatory loading.

1.2.8.3 Mean-Time-Between-Removal. End item aircraft dynamic components shall have a mean time between removal of 1,500 flight hours (reference 1.2.3.2).

1.2.8.4 Airframe Major Overhaul. The airframe shall be designed so as not to require major overhaul in less than 4,500 flight hours.

1.3 Design and Construction.

1.3.1 Material Properties. Selection of materials and processes shall include consideration of possible impairment of physical properties due to the processing operations as well as the range of operating and storage conditions defined in this specification. Fracture toughness of materials and the possible effects thereon of processing are major considerations in the selection of materials,particularly for structural elements. Appropriate surface protection shall be provided against humidity, corrosion, rain and sand erosion, sunlight and fungus where these environmental factors can impair the ability of the aircraft to perform its mission. Properties of material shall be obtained from MIL-HDBK-5, MIL-HDBK-17, and MIL-HDBK-23 or from other sources subject to the approval of the procuring activity. Properties other than those contained in the foregoing handbooks shall be substantiated and analyzed in accordance with procedures used for corresponding data in the appropriate handbook. Minimum properties obtained from the foregoing sources shall be used for design purposes.

1.3.2 Survivability. Protection of critical subsystems and components against ballistic impact shall be accomplished primarily by the design and material selection of the components themselves. The use of armor, plastic forms and the like for this purpose is allowable provided it is demonstrated to be more efficient than other means. Design criteria presented in USAAVLABS TR 71-22 will be used as the basic criteria for the crashworthiness design of the MUT. Aircraft structure will be designed to progressively deform under crash loads up to and including the 95th percentile survivable crash forces as defined in USAAVLABS TR 71-22. During the aforementioned crash environment, crash forces experienced by the occupants shall be minimized, and occupiable space for the crew and passengers shall be retained to provide crew and passenger restraint and rapid egress from the aircraft under any conceivable postcrash attitude.

1.3.3 Standard Parts & Materials. Preference shall be given to the selection of Government approved standard parts, materials, and stock sizes. To facilitate replacement and simplify stocking, MS & AN standard parts shall be used where they suit the intended purpose. This requirement is applicable but not limited to fasteners, tubing, tie rods, cables, fittings, and accessories.

1.4 Functional Area Characteristics.

1.4.1 Body Group.

1.4.1.1 Cockpit. The cockpit shall accommodate a pilot and copilot, to accommodate the 5th through 95th percentile Army aviator as defined in ADS-3, and wearing either tropical or arctic flight clothing, survival gear, and body armor. The crashworthiness of the seats and restraint systems shall be in accordance with MIL-S-58095. A cockpit door which is jettisonable with single action release shall be provided on each side of the cockpit.

1.4.1.2 Troop/Cargo Compartment. The troop and cargo compartment shall be capable of accommodating 7 passengers in crashworthy seats. MIL-STD-1290 should be used as criteria for design of the troop compartment.

1.4.1.3 Litter Provisions. Accommodations shall be provided for three standard folding, rigid pole litters. A minimum of 18 inches vertical separation between litters is required. Crashworthy seats shall be provided for the medical attendants. The cabin design shall permit loading and unloading of litters from either side of the aircraft.

1.4.1.4 Cargo Capacity. The cargo compartment shall provide a minimum of 140 cubic feet of internal cargo volume.

1.4.2 Landing Gear. The aircraft shall incorporate wheel type landing gear to facilitate ground handling and portability. Means shall be provided for easy, rapid movement of the helicopter on the ground for purposes of servicing, maintenance, and deployment to protected areas.

1.4.3 Propulsion Subsystem.

1.4.3.1 Main Propulsion Engines. The aircraft shall be powered by two gas turbine engines with the level of technology shown in Figures A-2 and A-3. (Weight given includes 10% increase for integral inlet separator.) Engines shall be physically separated or some other provision shall be made to prevent the possibility of a single projectile damaging vital parts on more than one engine and to prevent failure of one engine from destroying the capability of the other engine.

1.4.3.2 Engine Cooling. Engine compartment ventilation design shall consider the IR suppression requirement of 3.2.10.2 to eliminate the direct radiation from hot engine components and to reduce structural temperatures. Items warmer than 400°F shall be shielded from sources of fuel spillage.

1.4.3.3 Engine Fuel Subsystem. The fuel subsystem shall be sized to the primary mission fuel of 1.2.1.1. The fuel subsystem shall be in accordance with MIL-F-38363 except for the following additions and exceptions:

a. Crashworthy features (to include extended range kit) shall be added and/or designed for following the guidelines of MIL-STD-1290 wherever feasible. Addition of such features will require exception to MIL-F-38363 in such areas as breakaway couplings/valves, frangible attachments, and fitting designs.

b. Fuel cells shall be self-sealing against .30 caliber APM2 projectiles (tumbled) with a velocity of 2750 fps and target distance of 100 yards.

c. Fuel lines shall employ ballistic protection or other methods such that the mission shall be able to be completed after a hit from a .30 caliber APM2 projectile (tumbled) with a velocity of 2750 fps and target distance of 100 yards. Suction fuel transfer from tanks to engines is required.

d. A minimum of 3 inches of void filler (polyurethane foam per MIL-P-46111 or similar material) shall be applied to all voids of 3 inches or greater, around tanks containing flammable fluids, and to all main supply lines from those tanks to components. Equal or more effective techniques may be used if desired.

e. Fuel cells shall be located to provide ease of installation, removal, and maintenance.

1.4.3.4 Drive Subsystem. The continuous rating of the main transmission shall be 120 percent of the power (torque) available from the engines operating at intermediate power at 4000

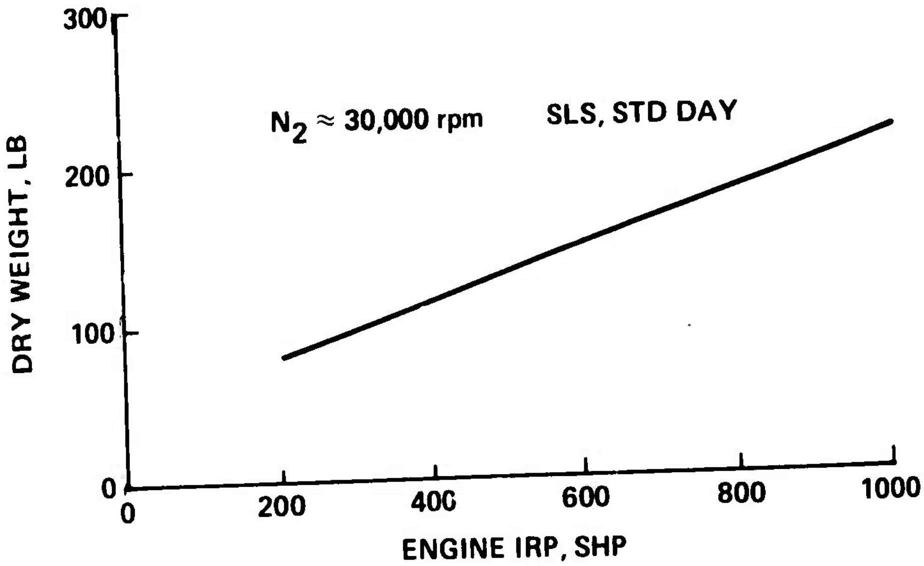


Figure A-2. Advanced Technology Engine Weight.

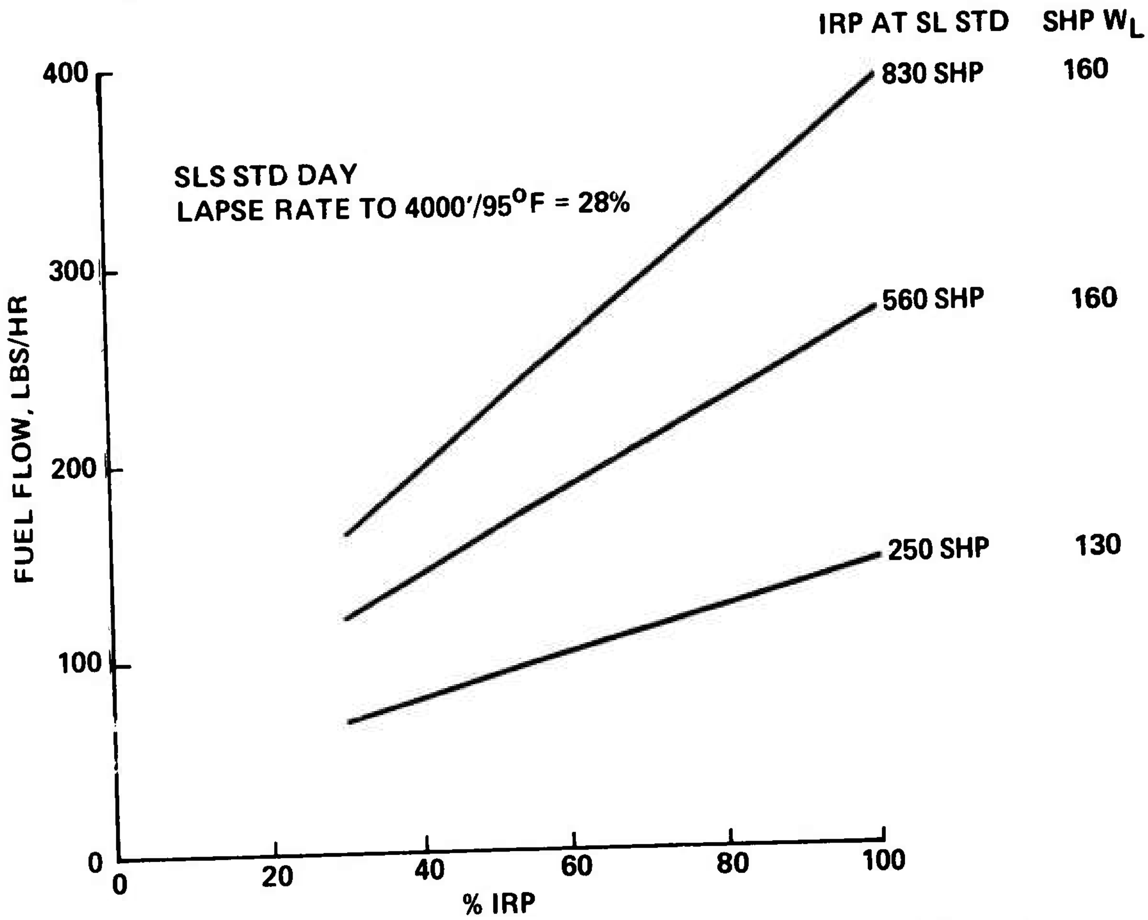


Figure A-3. Typical Performance of Three Advanced Technology Engines.

feet pressure altitude with a temperature of 95°F. The drive subsystem shall be designed for the lowest practical vulnerability to enemy weapons.

1.4.4 Instruments and Navigational Equipment. The MUT shall have installed the instruments required for instrument flights.

1.4.5 Avionics Equipment. The MUT will incorporate the UTTAS avionics package. The weight of uninstalled avionics equipment used will be 350 pounds and will include:

QTY. PER AIRCRAFT	DESCRIPTION	IDENTIFICATION	UNIT WT. (LB)
	COMMUNICATIONS		
2	VHF-FM Radio Set	AN/ARC-114	7.0
1	VHF-AM Radio Set	AN/ARC-115	7.2
1	UHF-AM Radio Set	AN/ARC-116	7.5
3	Interphone Control	C-6533/ARC	1.8
	AUTOMATIC DIRECTION FINDER, AN/ARN-89		
1	Receiver, Radio	R-1496( )/ARN-89	6.8
1	Control, Radio Set	C-7392( )/ARN-89	3.1
1	Amplifier, Impedance Matching	AM-4959( )/ARN-89	.2
1	Antenna, Loop	AS-2108( )/ARN-89	2.1
1	ADF Compensation Network		.2
	GYRO MAGNETIC COMPASS SET, AN/ASN-43		
1	Gyro, Directional	CN-998( )/ASN-43	5.5
1	Transmitter, Induction Compass	T-611( )/ASN	1.2
1	Compensator, Magnetic Flux	CN-405( )/ASN	.2
	TRANSPONDER SET, AN/APX-72		
1	Receiver-Transmitter	RT-859/APX-72	15.3
1	Control	C6280A(P)/APX	3.0
1	Mounting	MT3809/APX-72	1.7
	COMMUNICATION SECURITY SET, TSEC/KY-29		
3	Communication Security Set	TSEC/KY-28	
3	Control Indicator Assembly	C-8157/ARC	
3	Mounting	MT-3802/ARC	

QTY. PER AIRCRAFT	DESCRIPTION	IDENTIFICATION	UNIT WT. (LB)
	AUXILIARY EQUIPMENT		
1	Transponder Test Set	TS-1843/APX	2.8
1	Mounting	MT-3513/APX	.5
1	Computer, Mark XII	KIT-1A/TSEC	14.5
1	Mounting (Vibration Isolated)	MT3949A/U	1.5
	VOR RADIO SET, AN/ARN-82		
1	Receiver, Radio	R-1388/ARN-82	10.3
1	Control	C-6873/ARN-82	1.2
1	Mount	MT-3600/ARN-82	.5
1	Tactical Landing System	AN/ARN( )	32.0
1	LORAN C/D Airborne Navigation System	AN/ARN( )	30.0
	GLIDE SLOPE MARKER BEACON, AN/ARN-58		
1	Receiver, Radio	R-844/ARN-58	9.0

1.4.6 Aircraft Handling. Towing provisions shall be in accordance with MIL-STD-805 and shall permit towing under field conditions of CBR 2.5.

1.4.7 Cargo Handling. A cargo hook and controls shall be provided to facilitate acquisition, transport and release of external loads up to 2000 pounds.

APPENDIX B

MUT ENGINEERING DRAWINGS

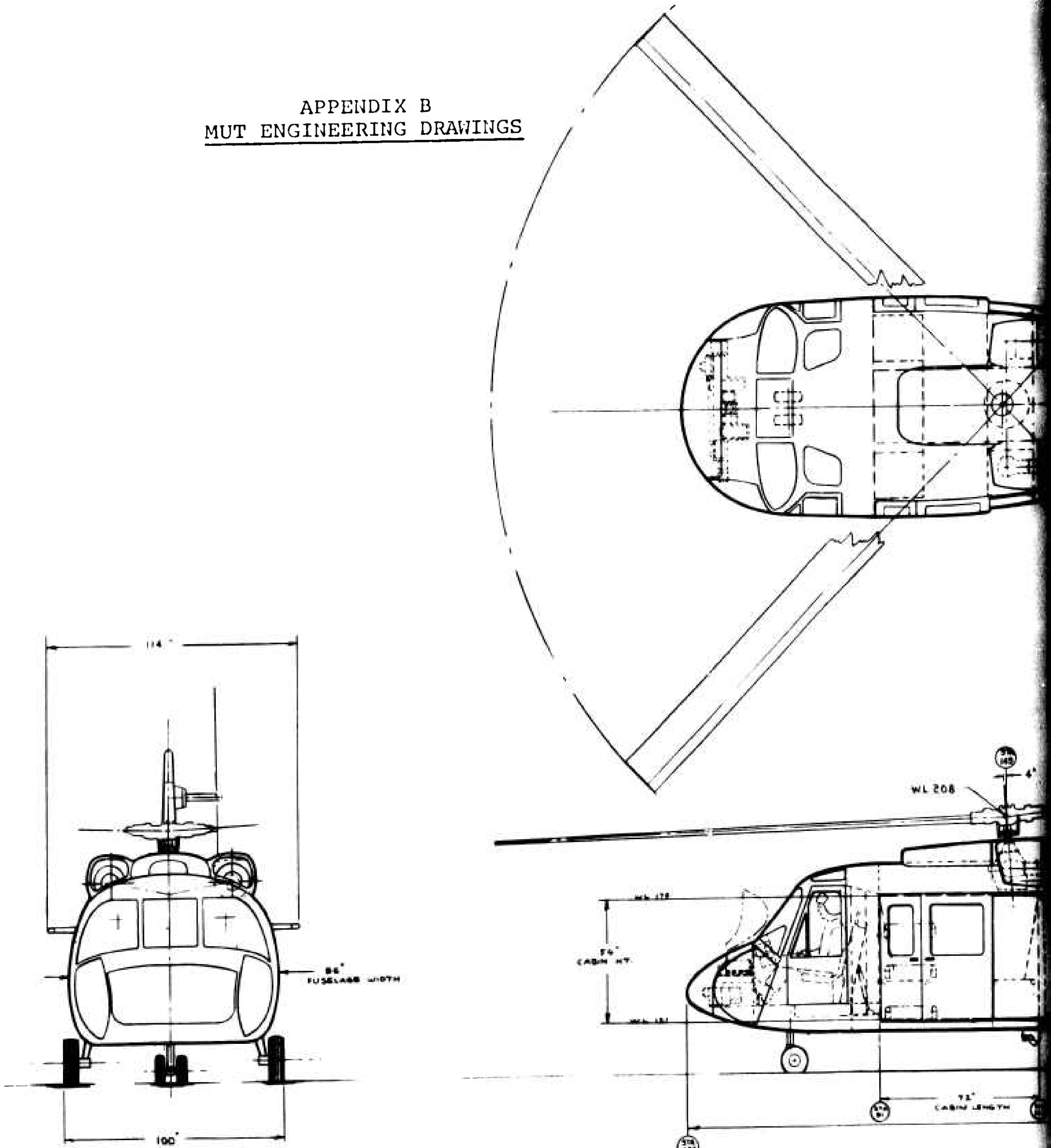
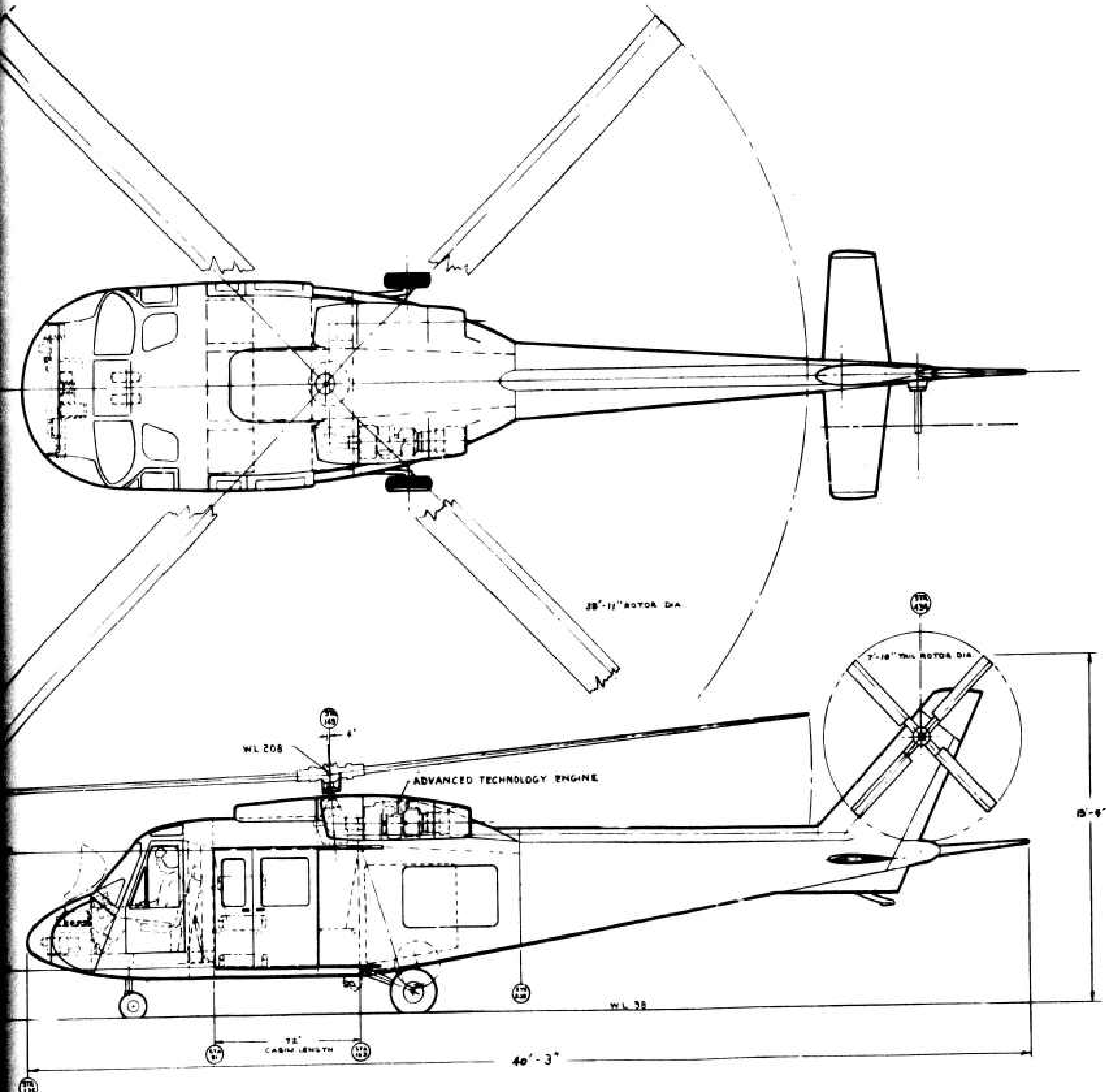


Figure B-1. Baseline MUT Three-View Drawing.



38'-11" ROTOR DIA
WL 208
4"
ADVANCED TECHNOLOGY ENGINE
7'-10" TAIL ROTOR DIA
15'-6"
WL 38
72"
CABIN LENGTH
40'-3"

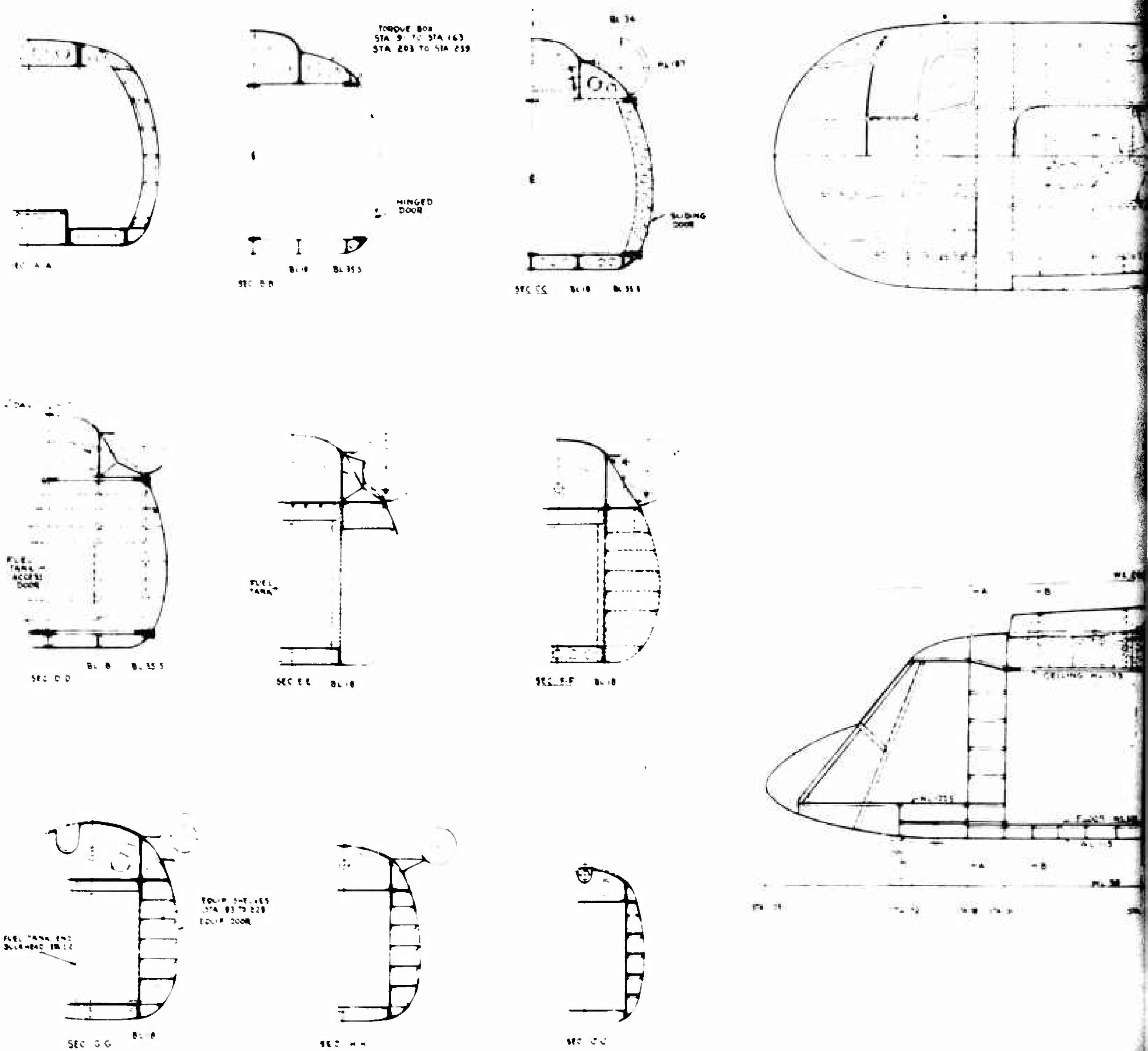
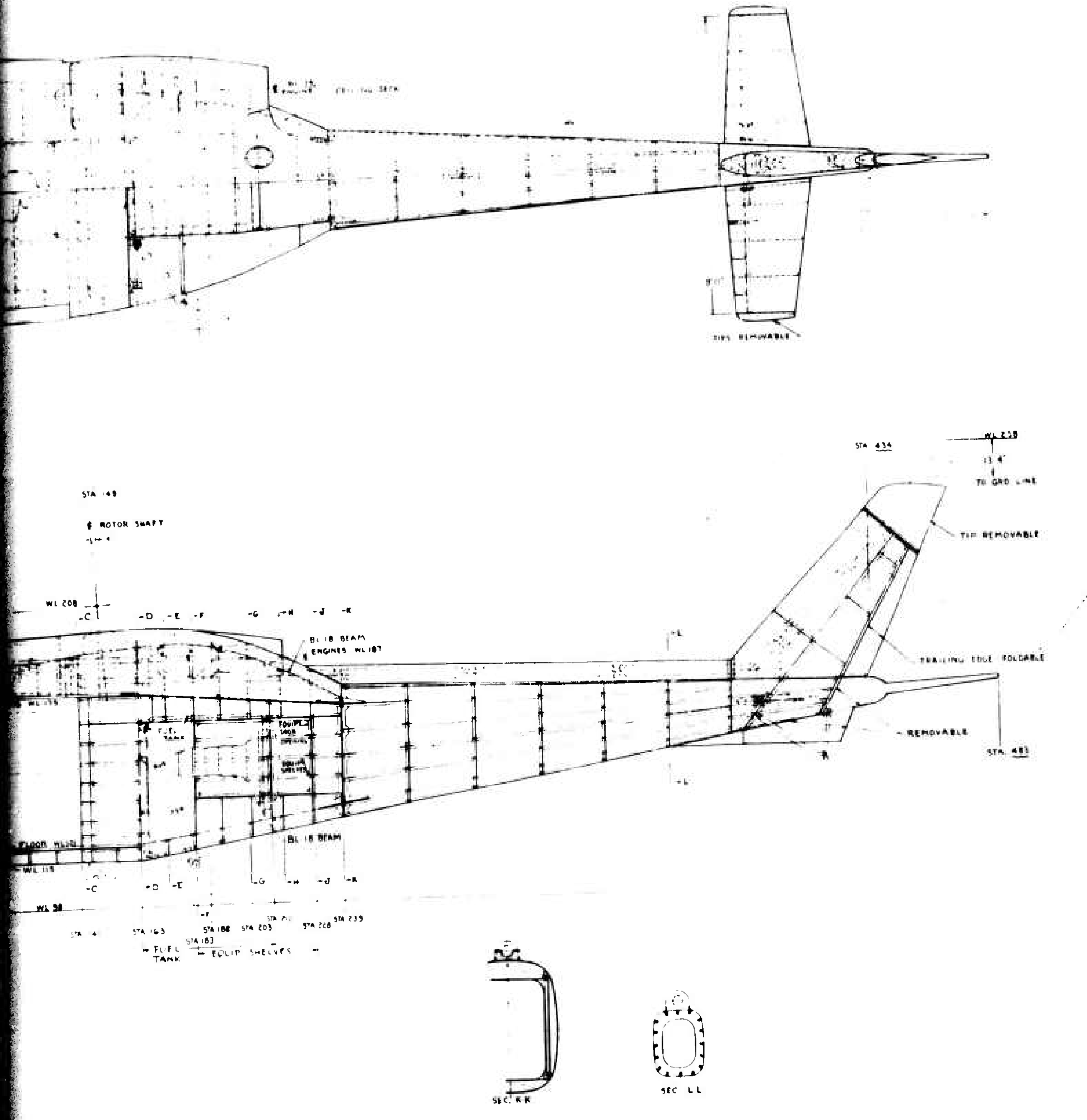


Figure B-2. MUT Baseline Airframe Structure.


TIPS REMOVABLE
STA 434
WL 258
13' 4"
TO GRD LINE
TIP REMOVABLE
STA 149
℄ ROTOR SHAFT
WL 208
C
D
E
F
G
H
J
K
L
BL 18 BEAM
ENGINES WL 187
TRAILING EDGE FOLDABLE
WL 175
FUEL TANK
EQUIP SHELVES
REMOVABLE
STA 483
R
FLOOR WL 121
WL 115
BL 18 BEAM
WL 58
STA 163
STA 188
STA 203
STA 228
STA 239
STA 183
FUEL TANK
EQUIP SHELVES
SEC. R R
SEC L L

2

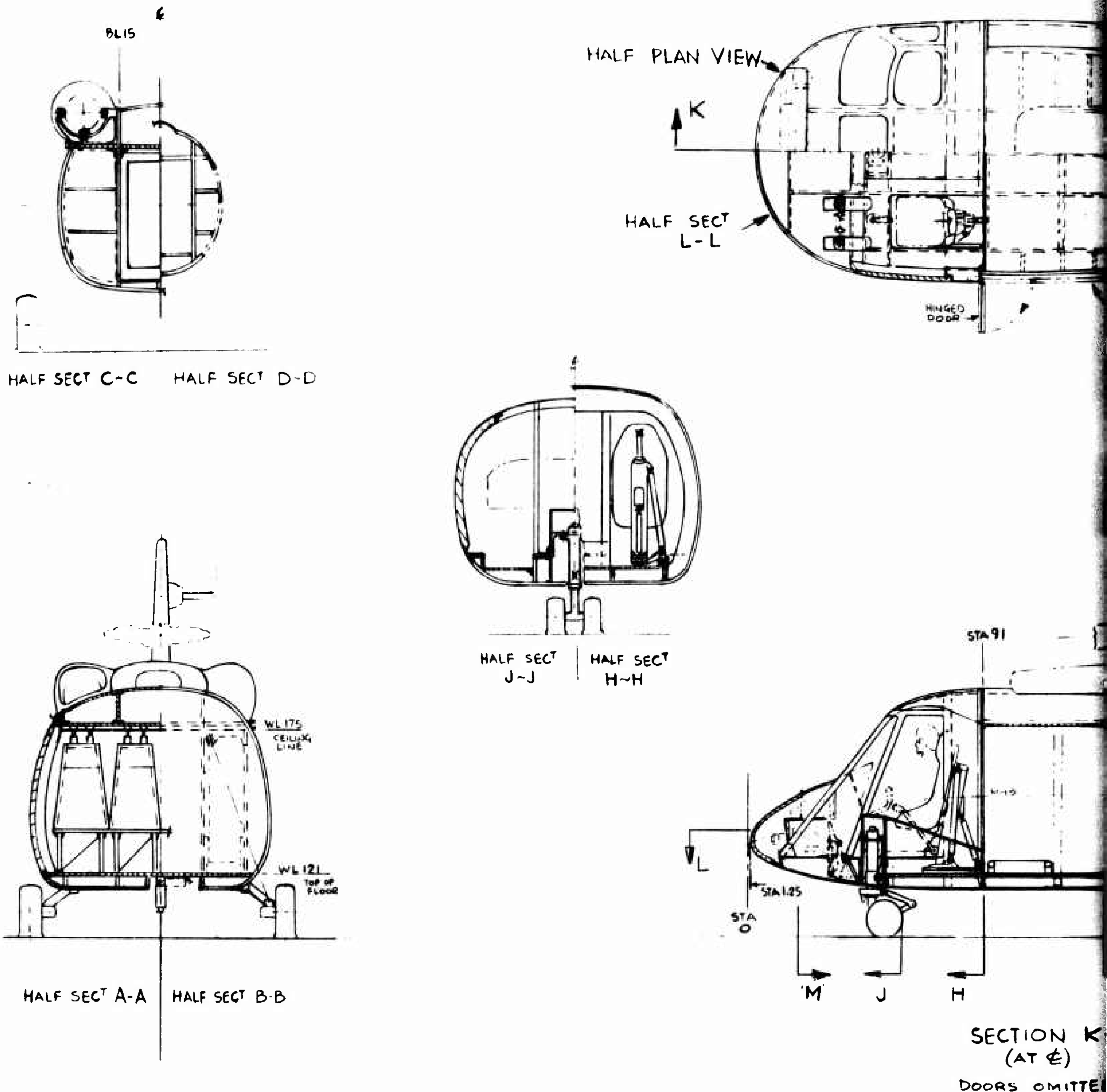
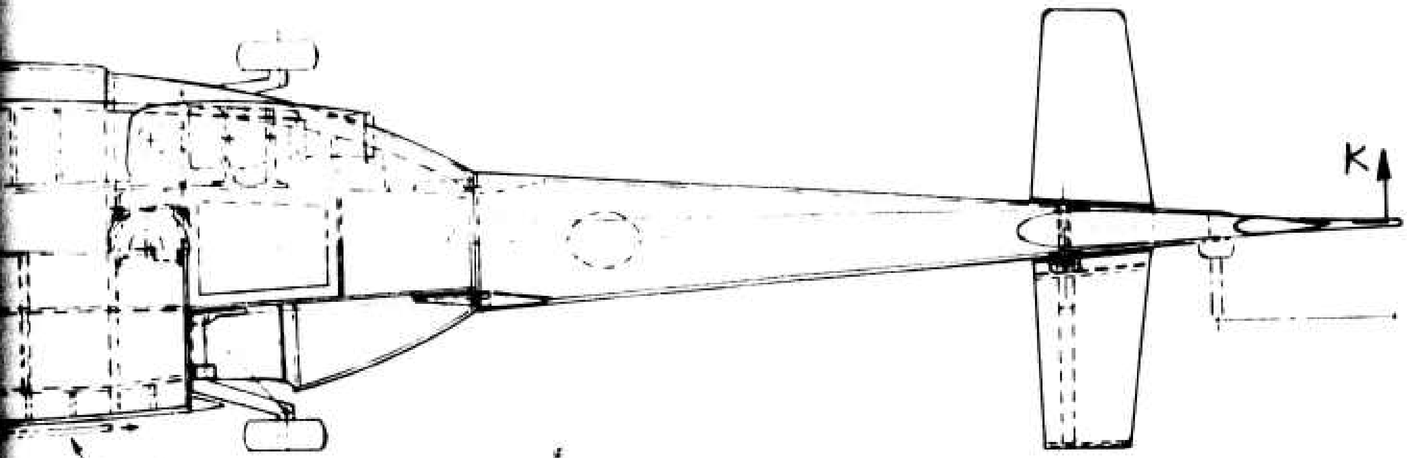
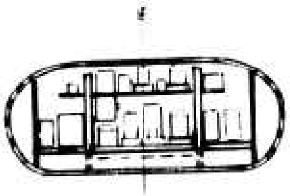
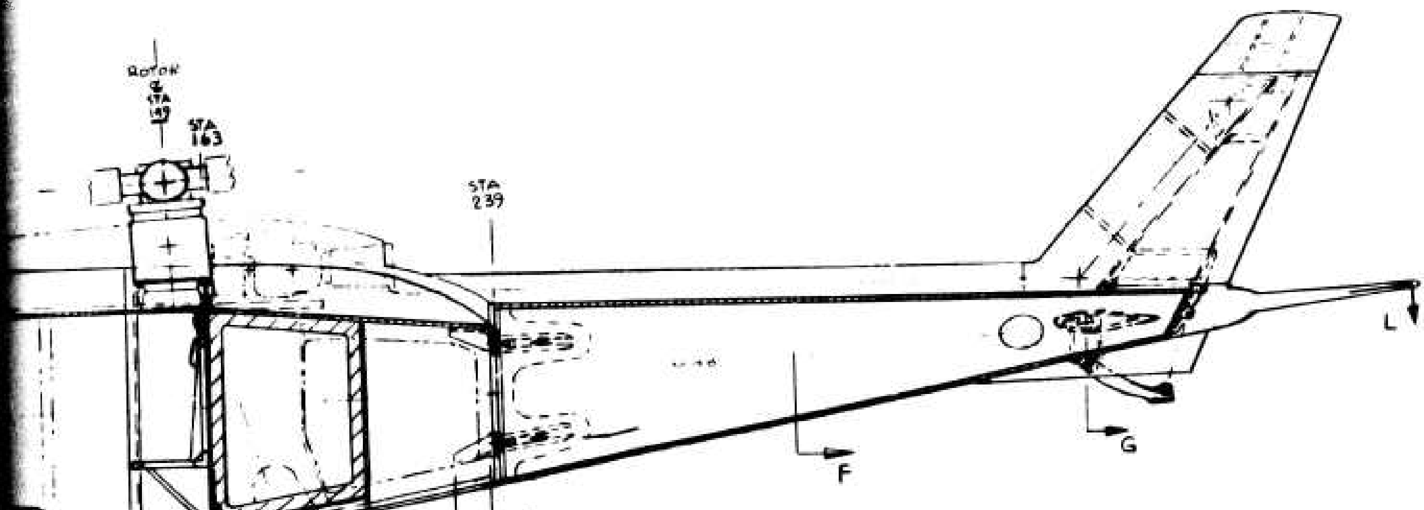


Figure B-3. MUT Advanced Composite Concept 1.

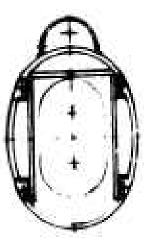




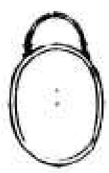
SECT M-M



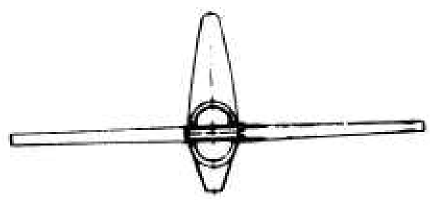
ON K-K
T ₵)
OMITTED FOR CLARITY



SECT E-E



SECT F-F



SECT G-G

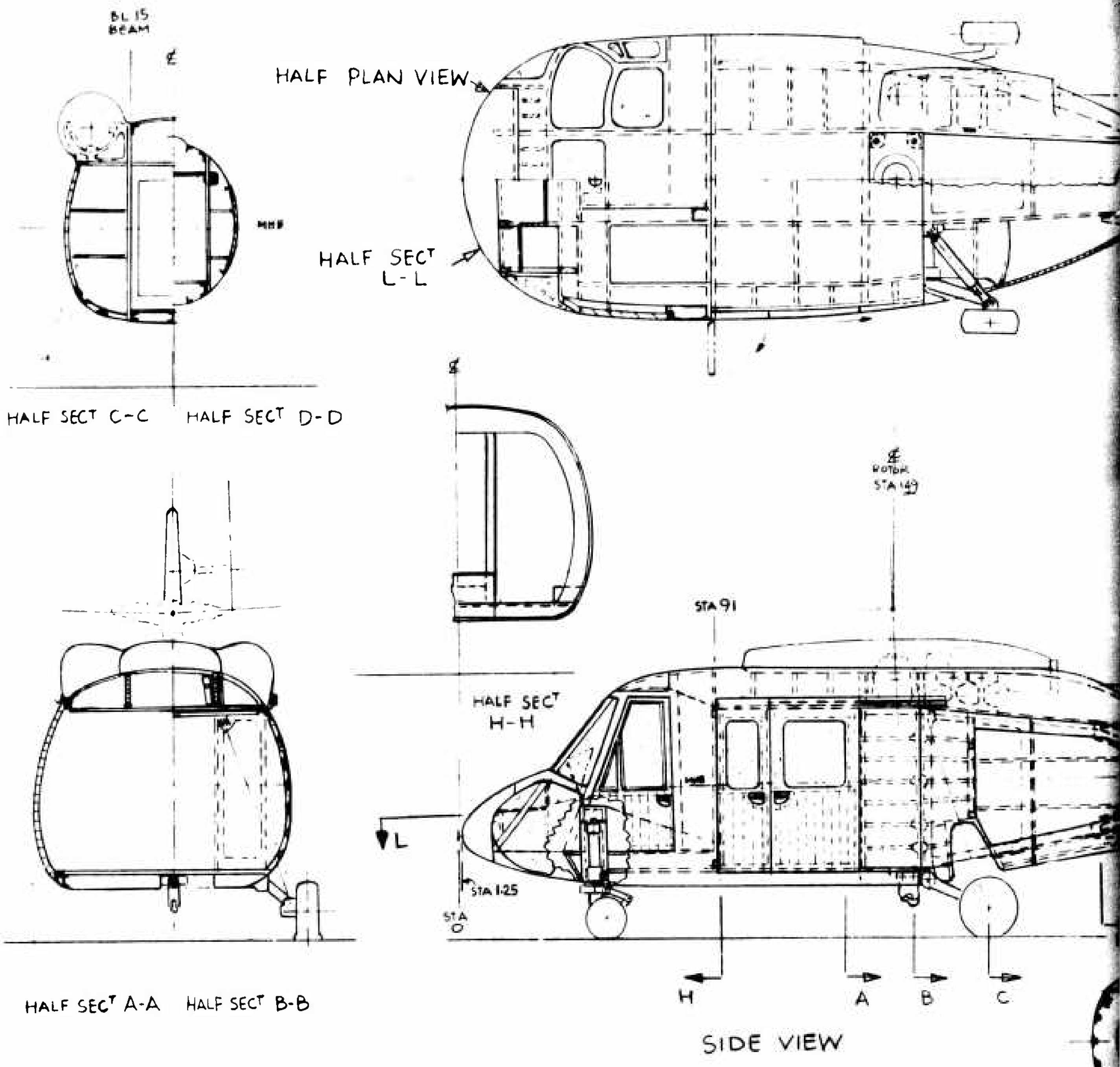
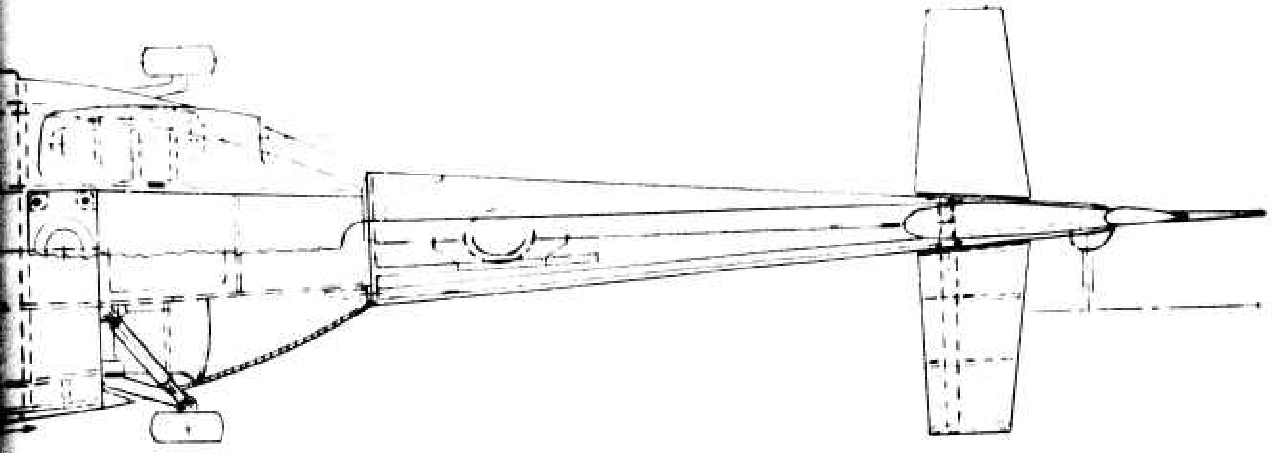
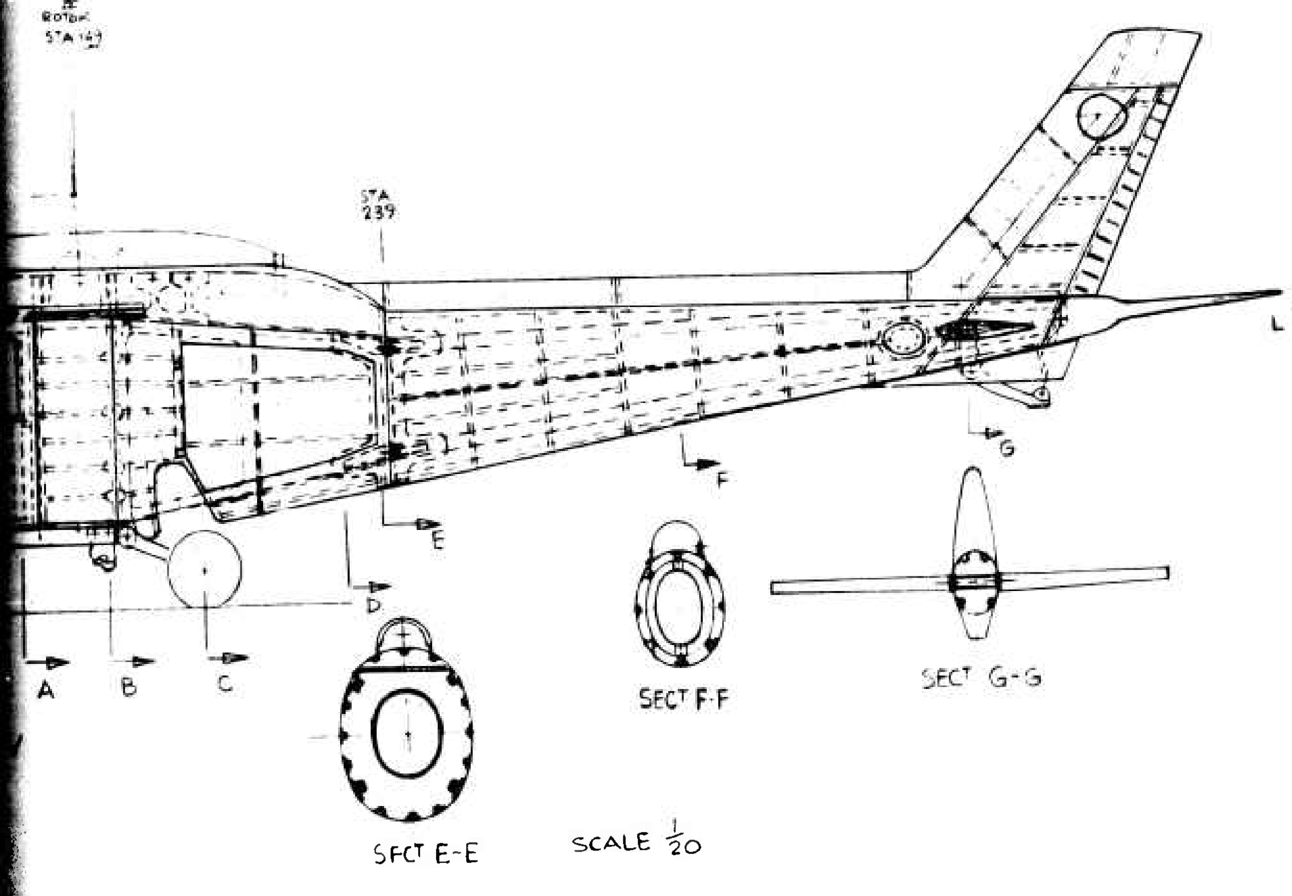


Figure B-4. MUT Advanced Composite Concept 2.





SCALE 1/20

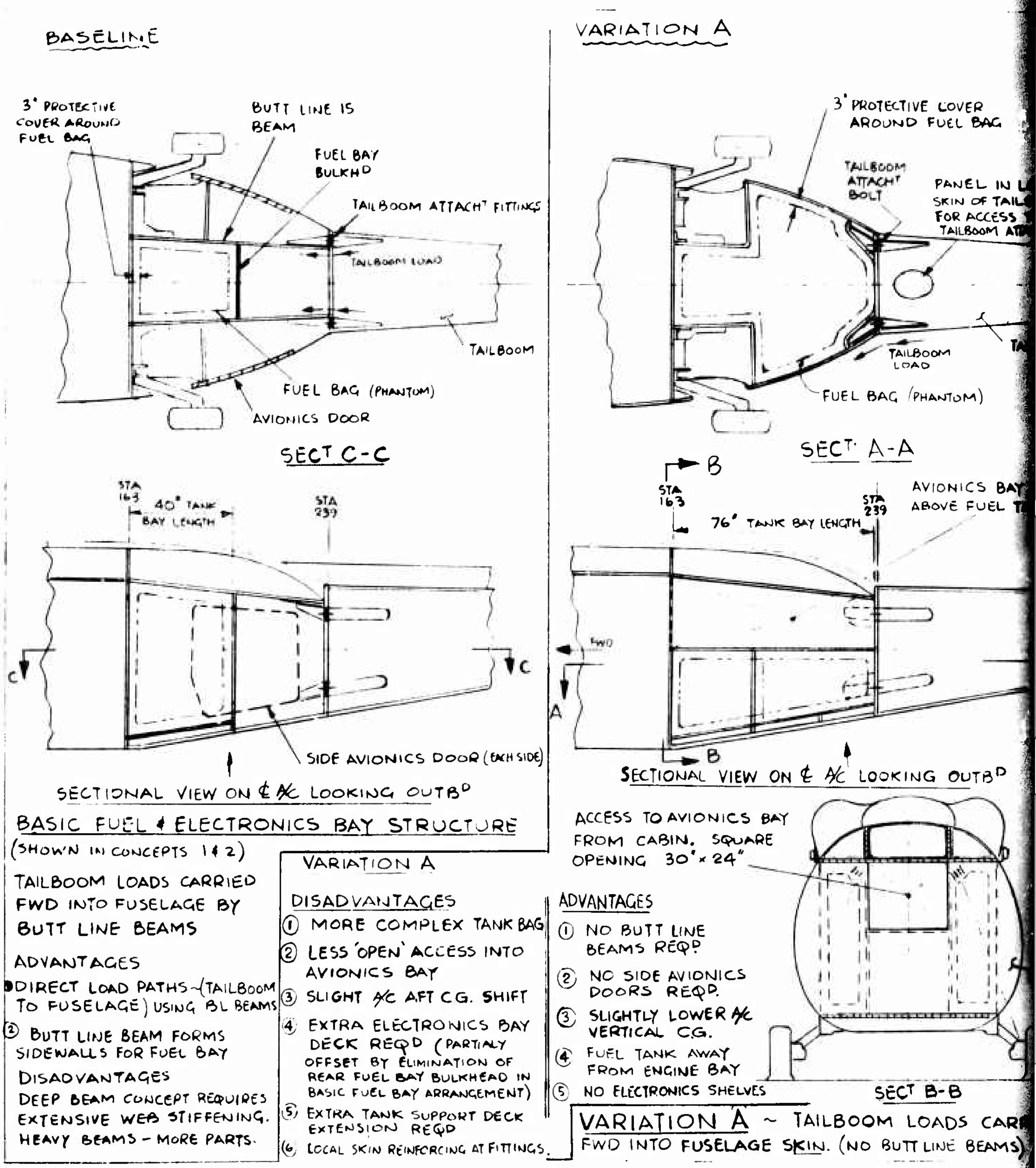
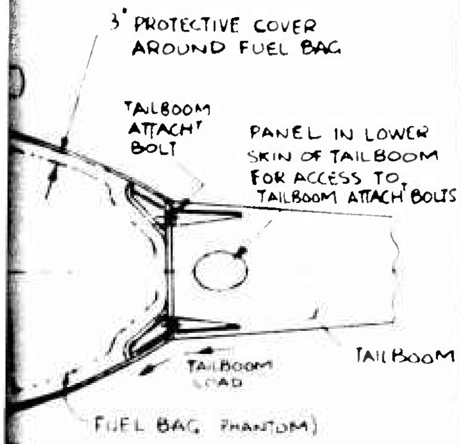
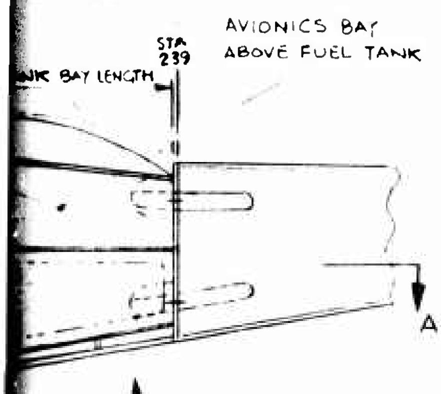


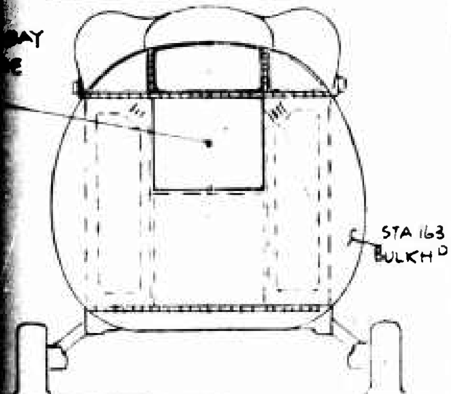
Figure B-5. MUT Concept Variations for Tank Bay and Tailboom.



SECT. A-A



ON ℄ A/C LOOKING OUTBD

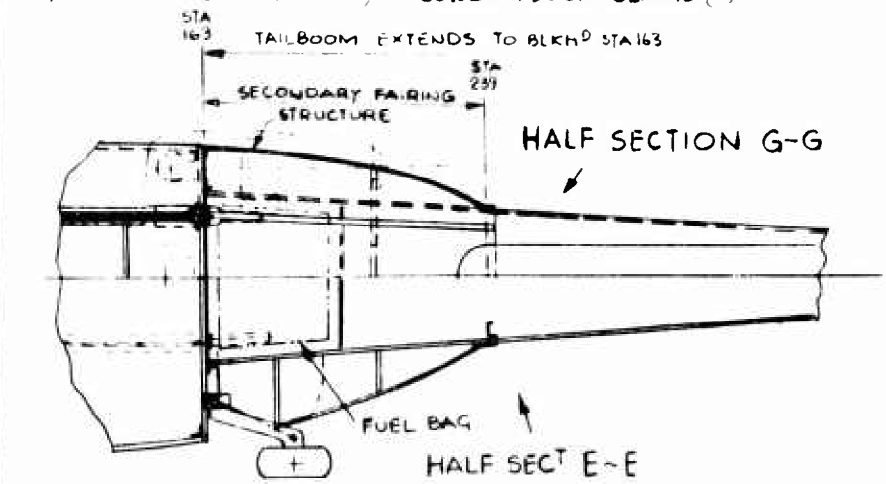


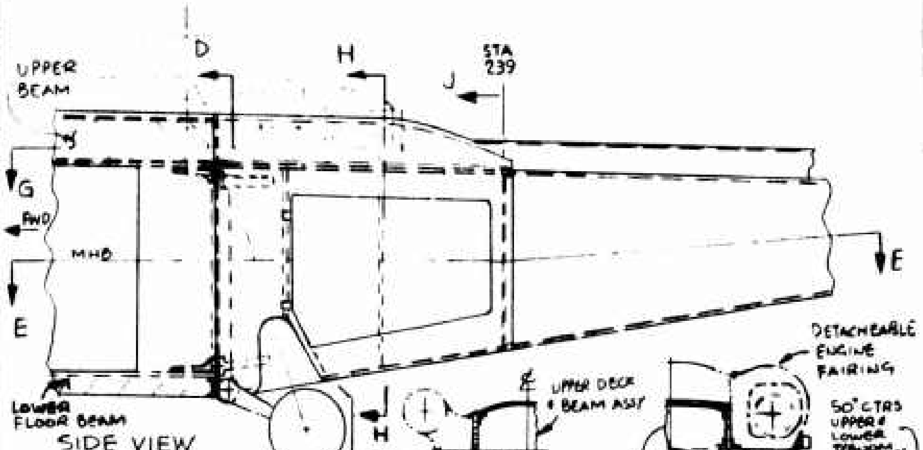
SECT B-B

~ TAILBOOM LOADS CARRIED
E SKIN. (NO BUTT LINE BEAMS)

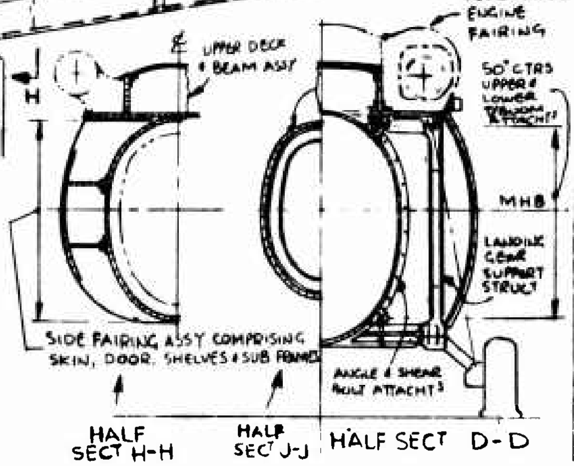
VARIATION B

TAILBOOM EXTENDED FWD TO STA 163 BENDING LOADS TAKEN BY UPPER XMSN BEAMS (2) & LOWER FLOOR BEAMS (2)





SIDE VIEW



ADVANTAGES

1. ELIMINATION OF PRIMARY STRUCTURE SEPARATE FUEL BAY SECTION (STA 163 - 239)
2. FUEL SUPPORTED BY EXISTING TAILBOOM SKIN STRUCTURE
3. NO FULL DEPTH BUTT LINE BEAMS REQD. TAILBOOM CONNECTS TO UPPER & LOWER LONGITL BEAMS

DISADVANTAGES

1. SEPARATE SECONDARY STRUCTURE FAIRING ASSEMBLIES REQD EACH SIDE FOR AVIONICS
2. HEAVILY REINFORCED BULKHEAD REQD AT STA 163
3. HIGH SHEAR LOADS GENERATED IN SKINS JUST FWD OF STA 163

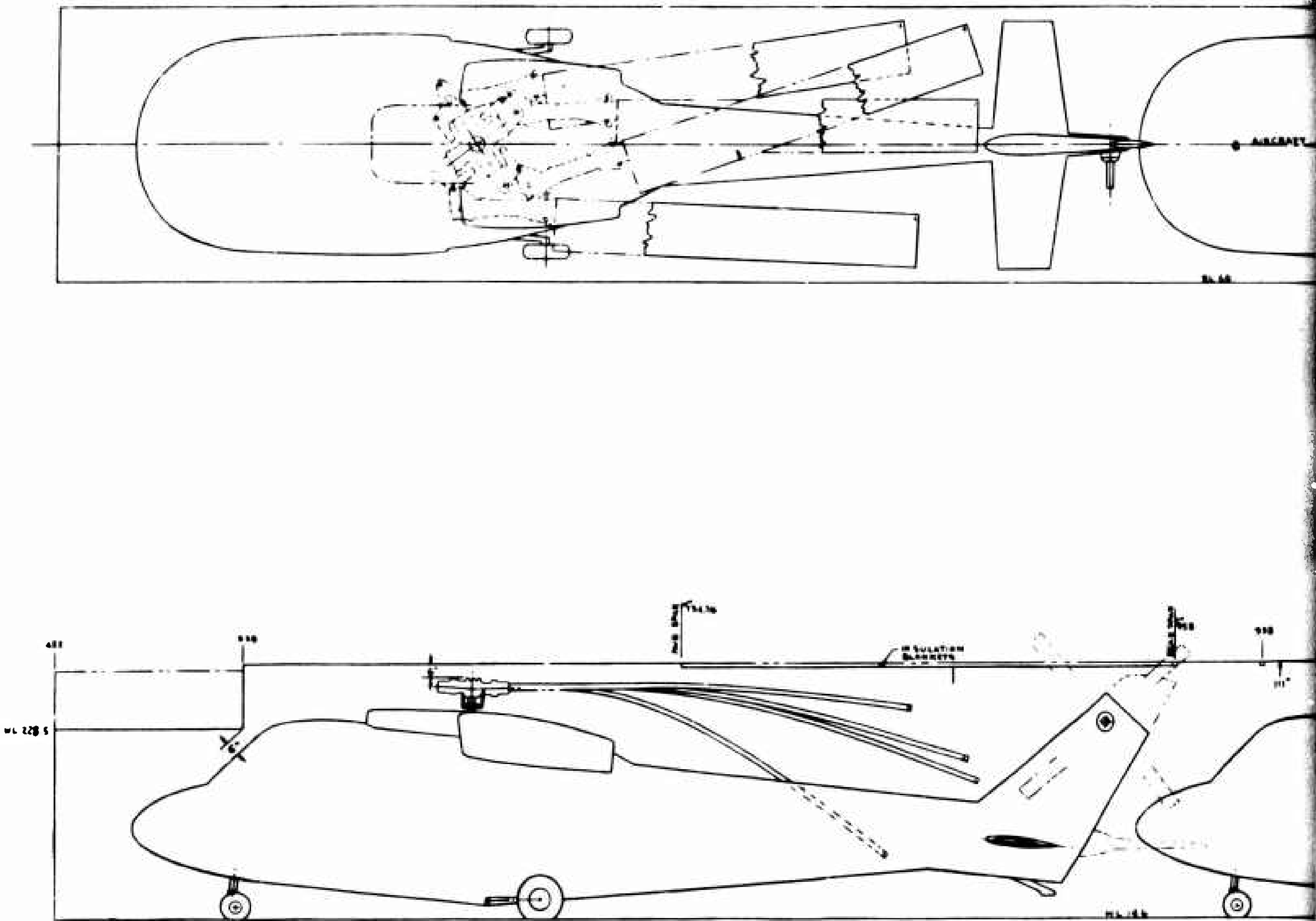
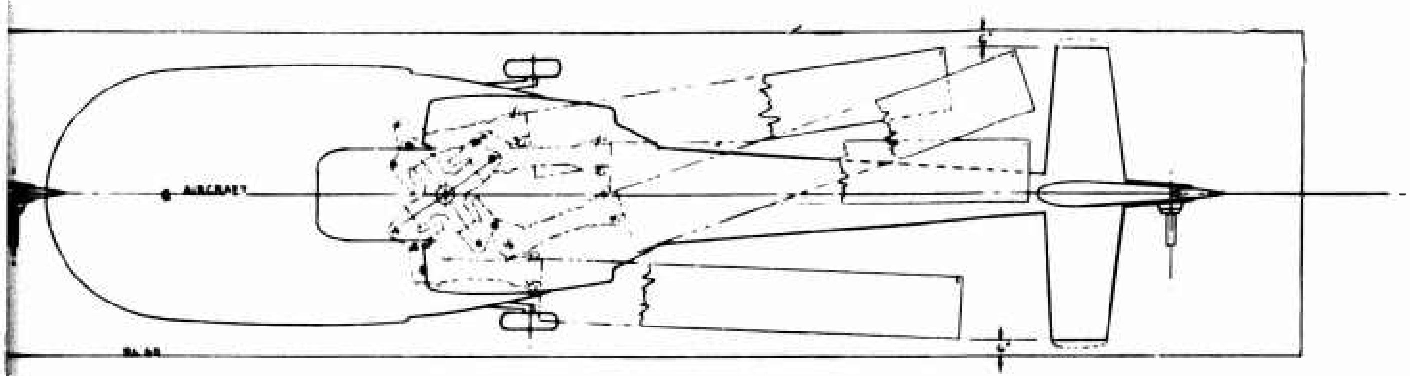
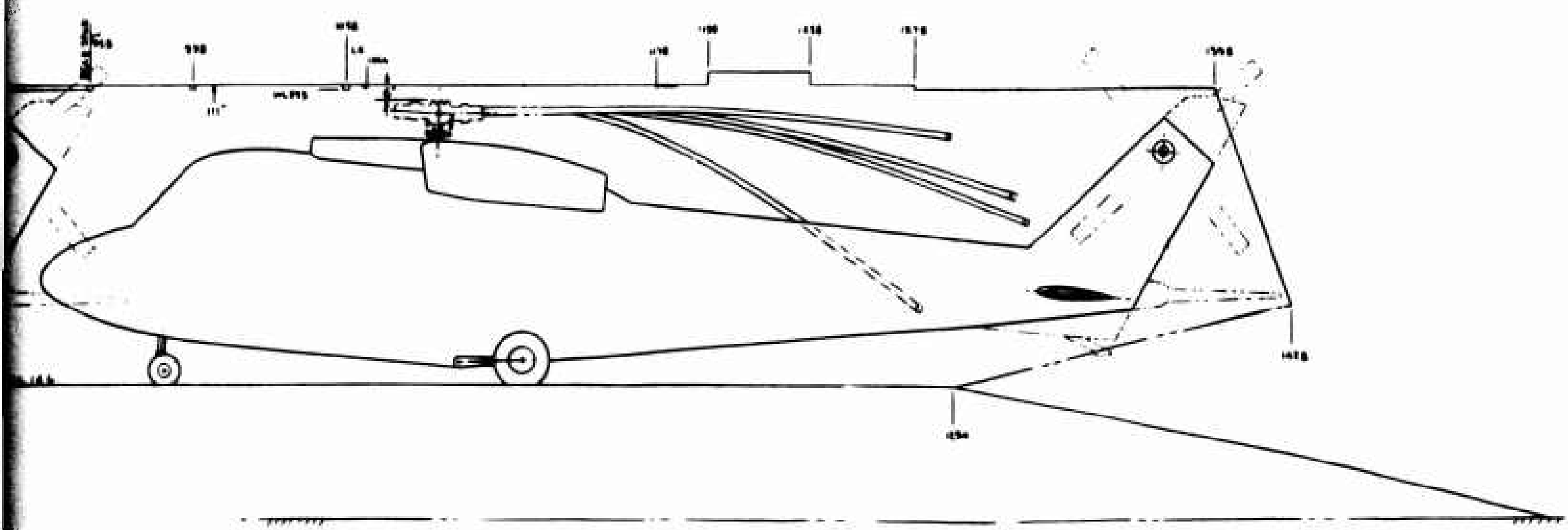


Figure B-6. Two MUT's in C-141 Aircraft.





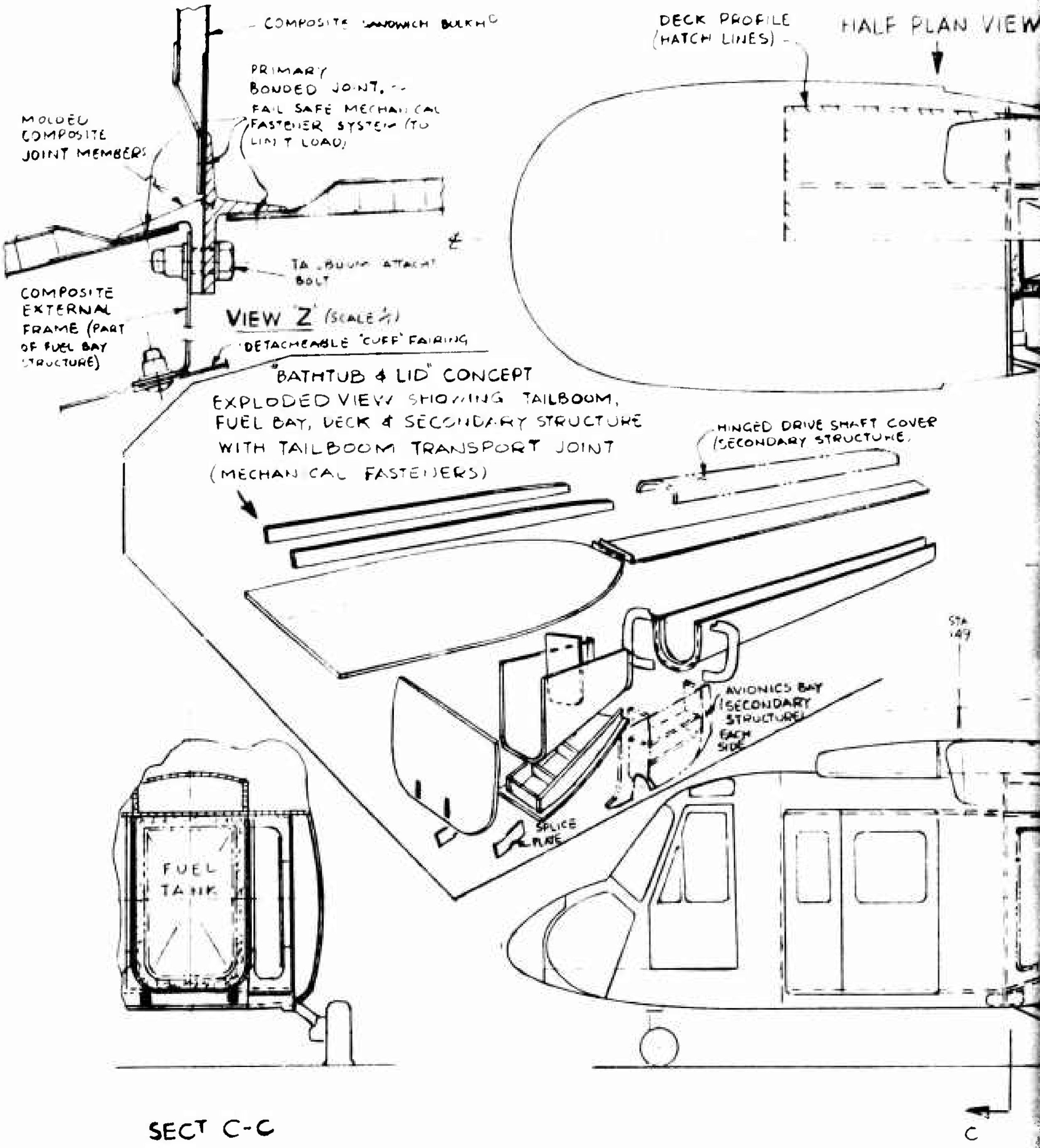
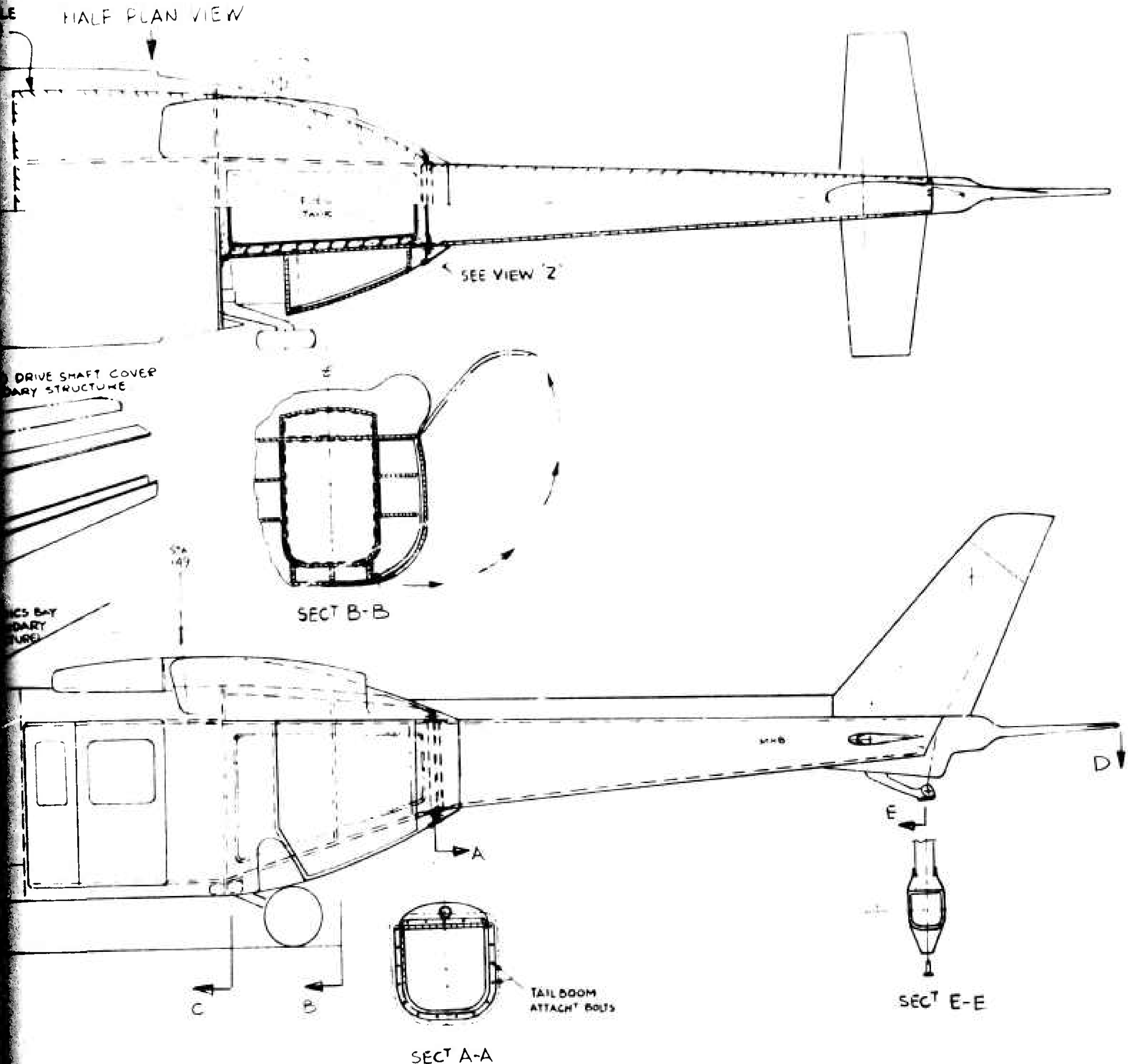


Figure B-7. MUT Integral Tailboom/Fuel Bay (Bathtub and Lid Concept)


HALF PLAN VIEW
SEE VIEW 'Z'
DRIVE SHAFT COVER
SECT B-B
STA 149
A
D
E
C
B
TAILBOOM ATTACH'T BOLTS
SECT A-A
SECT E-E

ept)

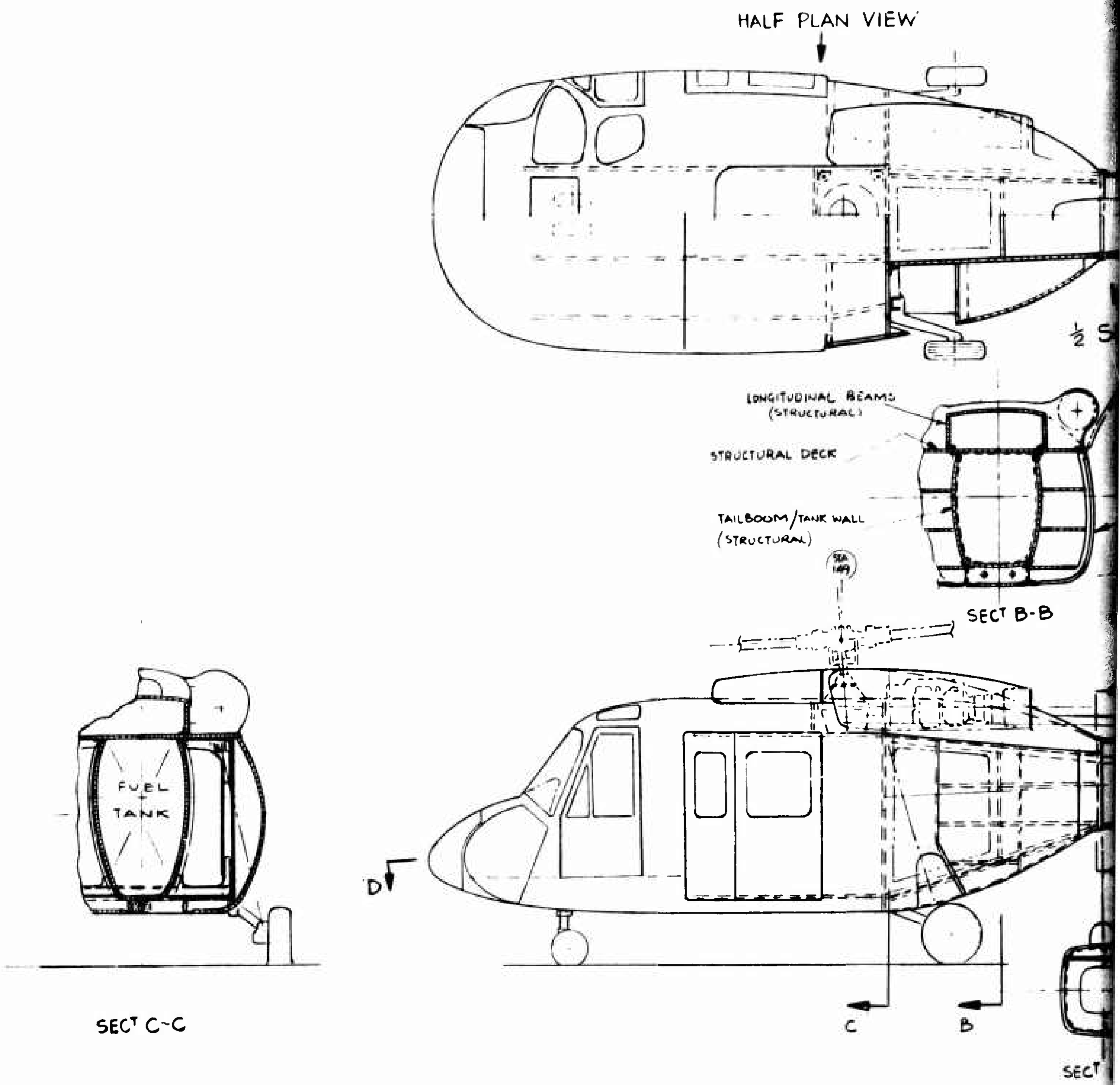
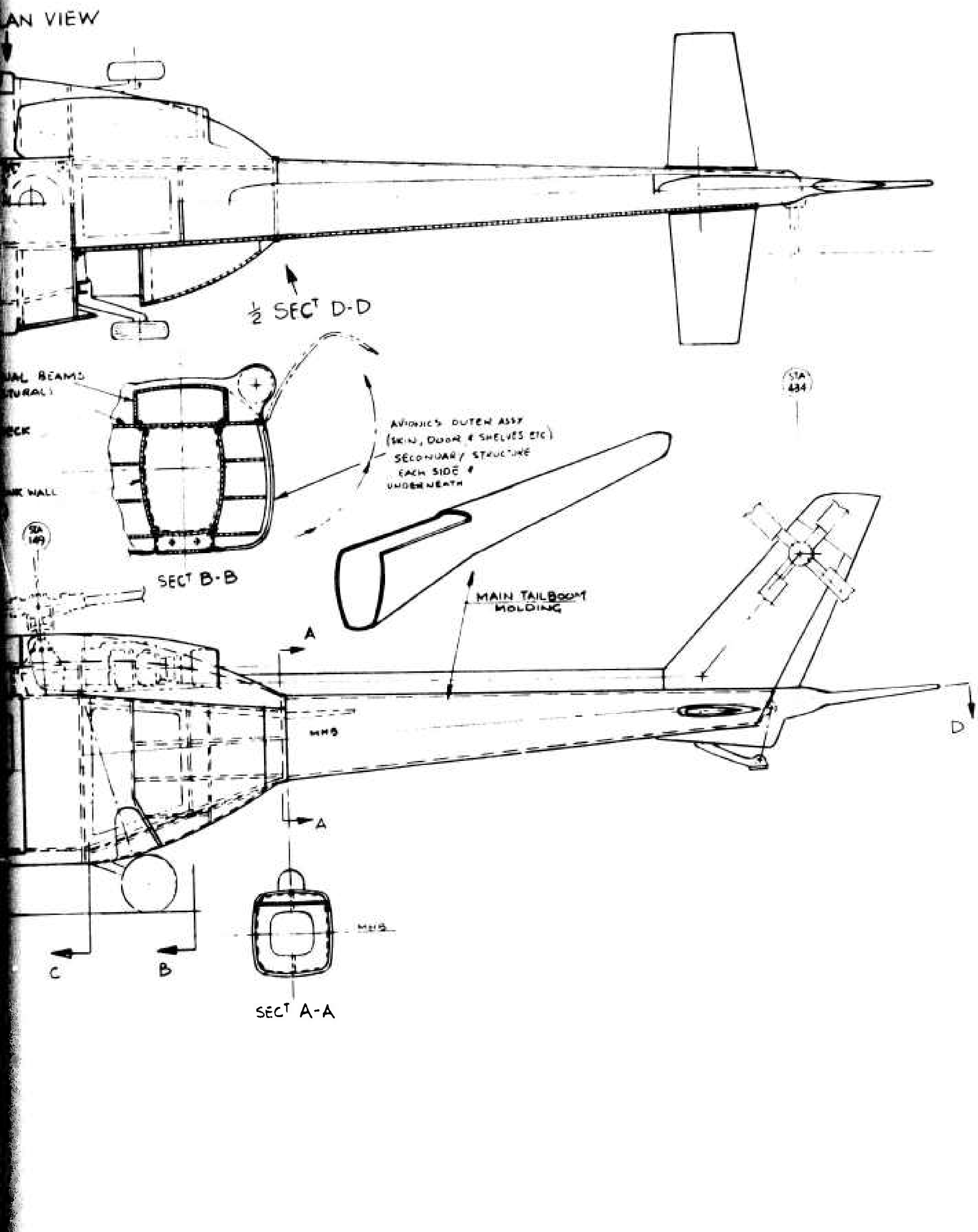
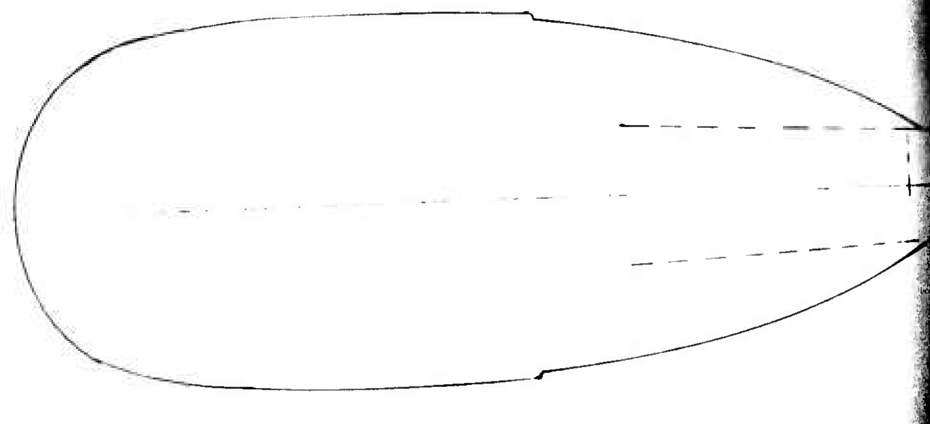
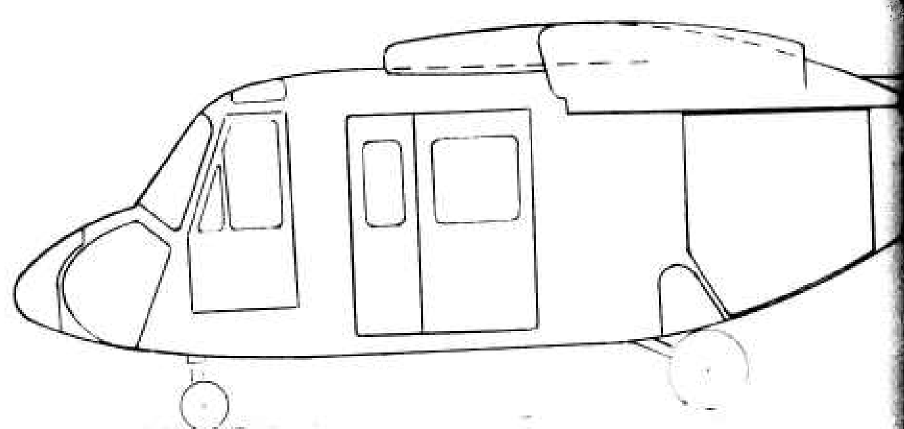


Figure B-8. MUT Integral Reduced-Section Tailboom.



AN VIEW
½ SECT D-D
AL BEAMS
TURAL)
ECK
K WALL
AVIONICS OUTER ASSY
(SKIN, DOOR & SHELVES ETC)
SECONDARY STRUCTURE
EACH SIDE &
UNDERNEATH
STA 434
STA 149
SECT B-B
MAIN TAILBOOM
MOLDING
A
A
D
C
B
SECT A-A

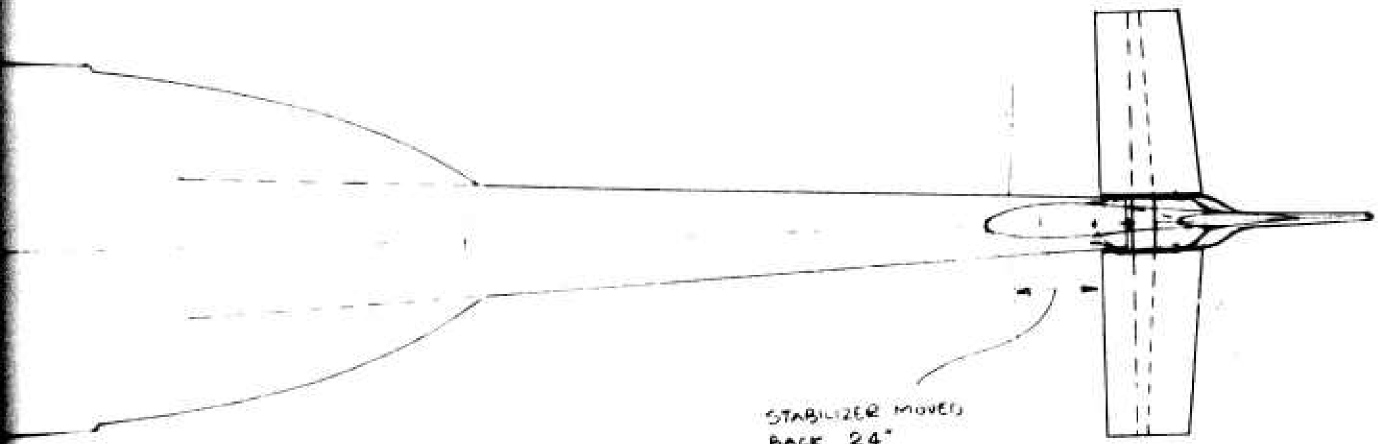


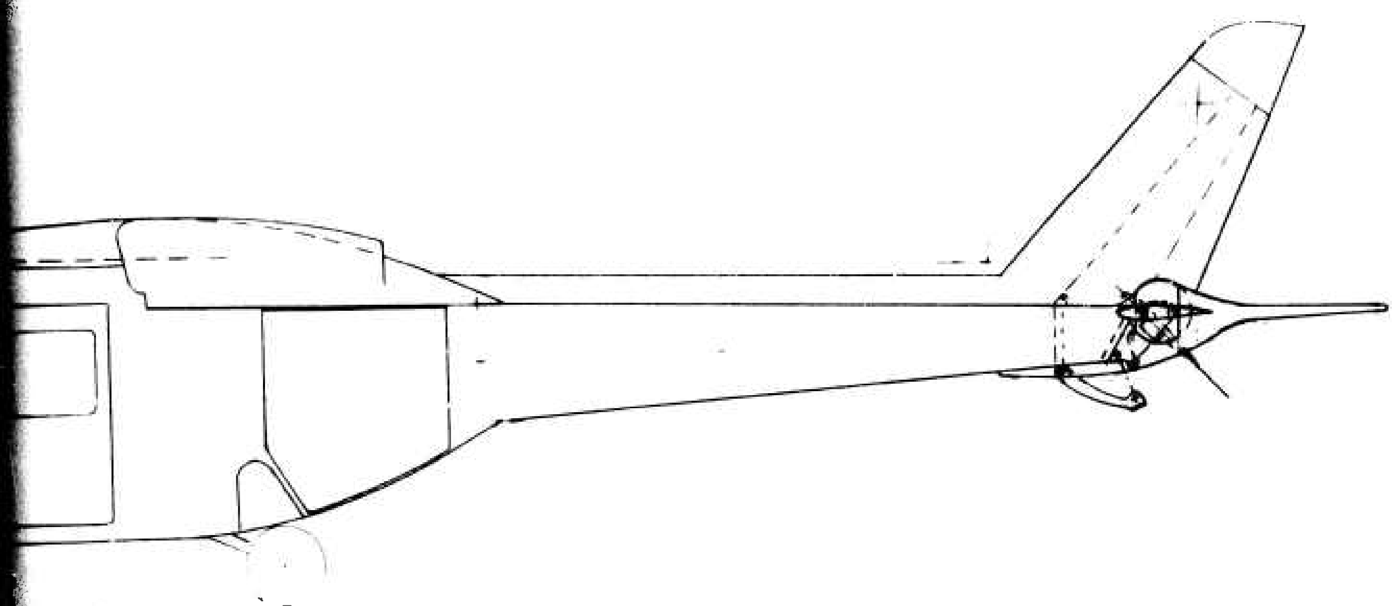


ADVANTAGES
1. SMALLER HORIZ TAIL AREA REQ'T
2. NO ACCESS HOLES & TORQUE TUBE HOLES IN TAILBOOM SHELL
3. SIMPLIFIED CONTINUOUS SPAR STABILIZER – LIGHTER & SIMPLER STAB ATTACH'T FITTINGS
4. DIRECT ACCESS TO STABILIZER ACTUATOR LOCATED OUTSIDE OF TAILBOOM
   BY REMOVAL OF TAIL STINGER FAIRING
5. TAILBOOM MOVED DOWN 3" (RELATIVE TO TAIL ROTOR) GIVING IMPROVED MAIN ROTOR TO TAILBOOM

DISADVANTAGES
LOCAL 'BUMP OUT' OF STINGER FAIRING TO FORM FLAT FOR CIRCULAR COVER PLATE ATTACHED TO STABIL

Figure B-9. MUT Horizontal Stabilizer Behind Tailboom.





SHELL

SIMPLER STAB ATTACH'T FITTINGS

IDE OF TAILBOOM

ING IMPROVED MAIN ROTOR TO TAILBOOM CLEARANCE (STILL WITHIN ACCEPTABLE FLARE ANGLE)

IRCULAR COVER PLATE ATTACHED TO STABILIZER

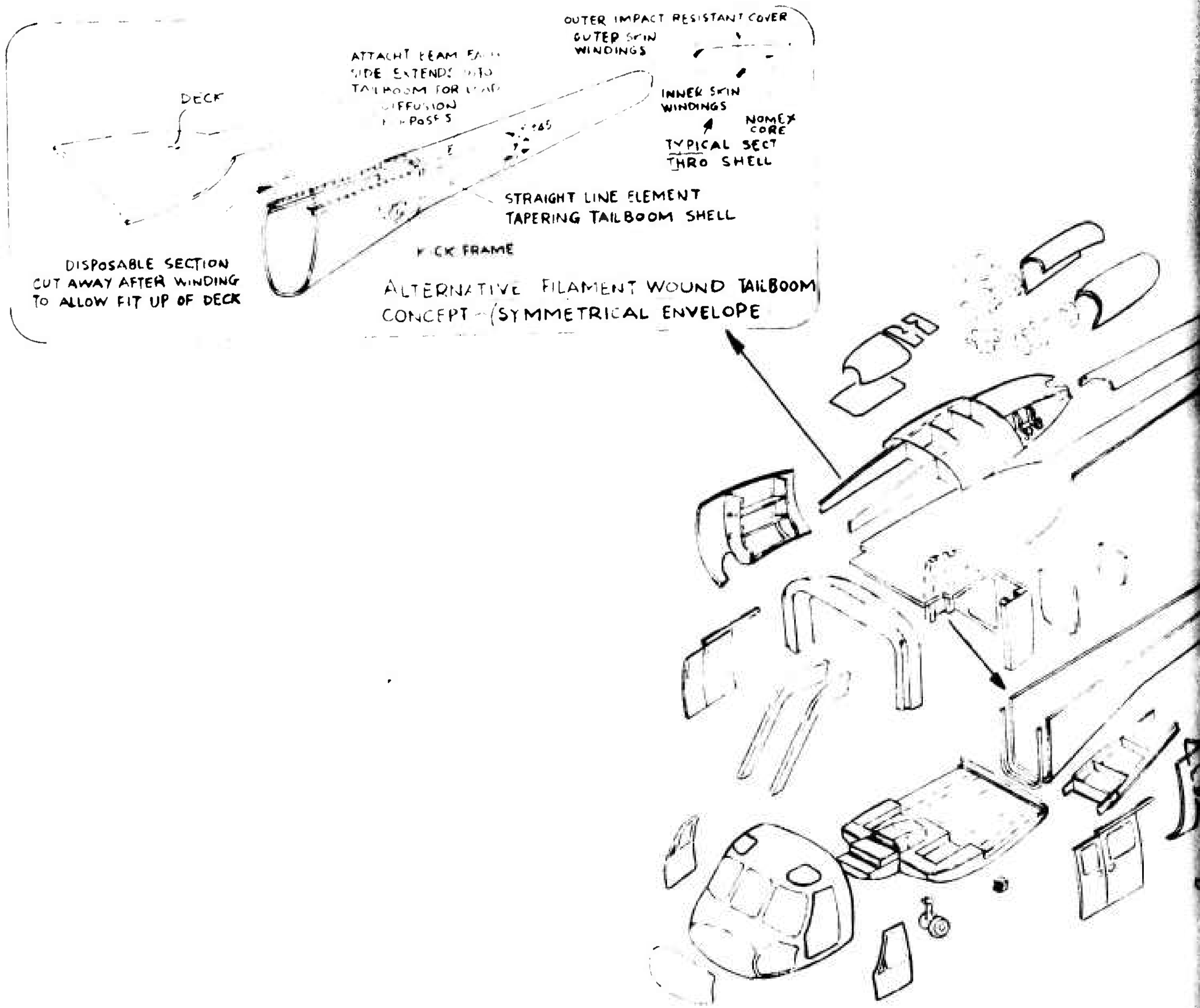
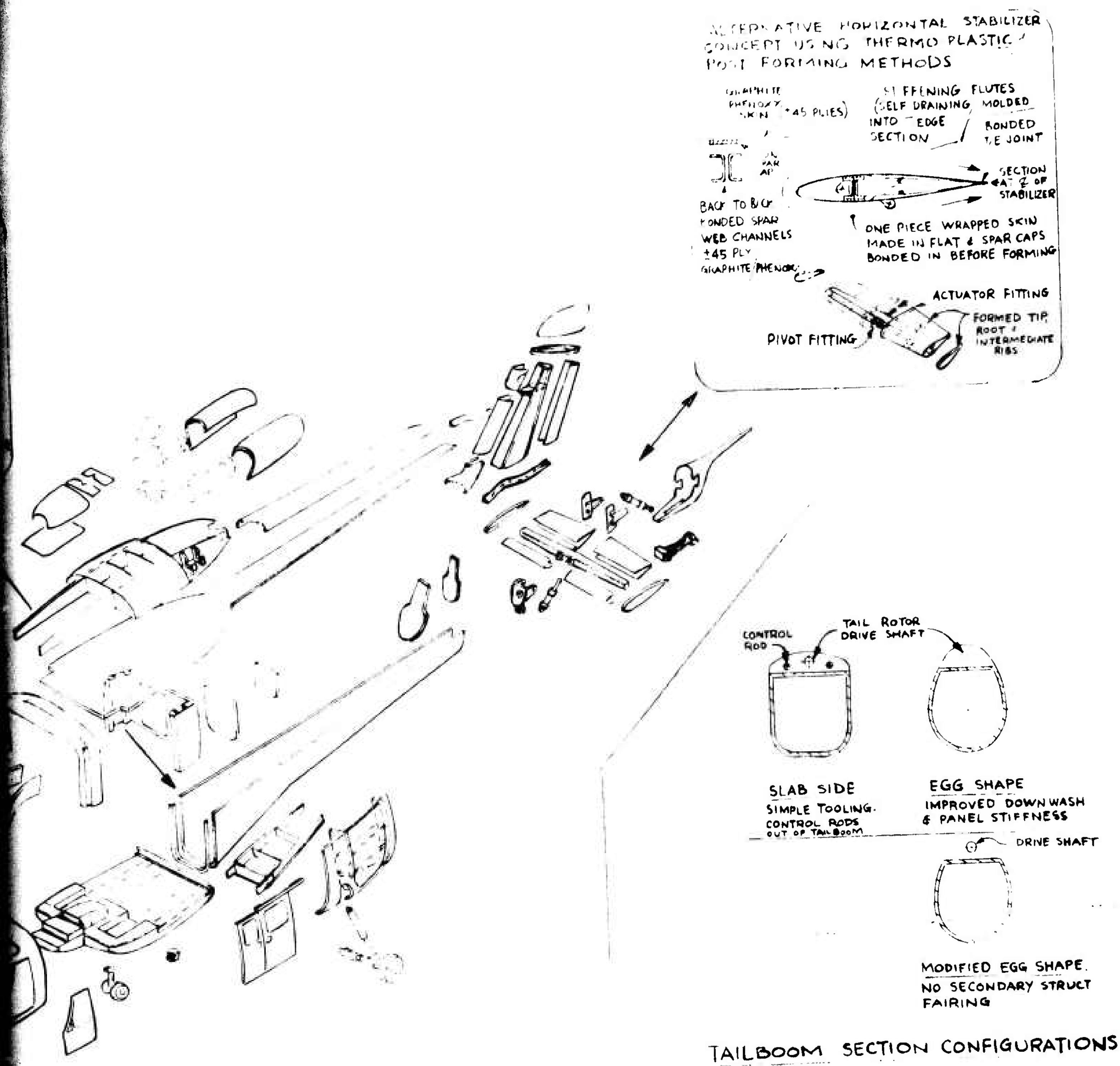


Figure B-10. MUT Exploded View of Honeycomb Sandwich Panel Concept.


ALTERNATIVE HORIZONTAL STABILIZER
CONCEPT USING THERMO PLASTIC
POST FORMING METHODS
GRAPHITE PHENOXY SKIN (±45 PLIES)
STIFFENING FLUTES (SELF DRAINING) MOLDED INTO EDGE SECTION
BONDED T.E JOINT
SECTION AT ℄ OF STABILIZER
BACK TO BACK BONDED SPAR WEB CHANNELS ±45 PLY GRAPHITE PHENOXY
ONE PIECE WRAPPED SKIN MADE IN FLAT & SPAR CAPS BONDED IN BEFORE FORMING
ACTUATOR FITTING
FORMED TIP, ROOT & INTERMEDIATE RIBS
PIVOT FITTING
CONTROL ROD
TAIL ROTOR DRIVE SHAFT
SLAB SIDE
SIMPLE TOOLING.
CONTROL RODS OUT OF TAILBOOM
EGG SHAPE
IMPROVED DOWNWASH & PANEL STIFFNESS
DRIVE SHAFT
MODIFIED EGG SHAPE.
NO SECONDARY STRUCT FAIRING
TAILBOOM SECTION CONFIGURATIONS

Concept.

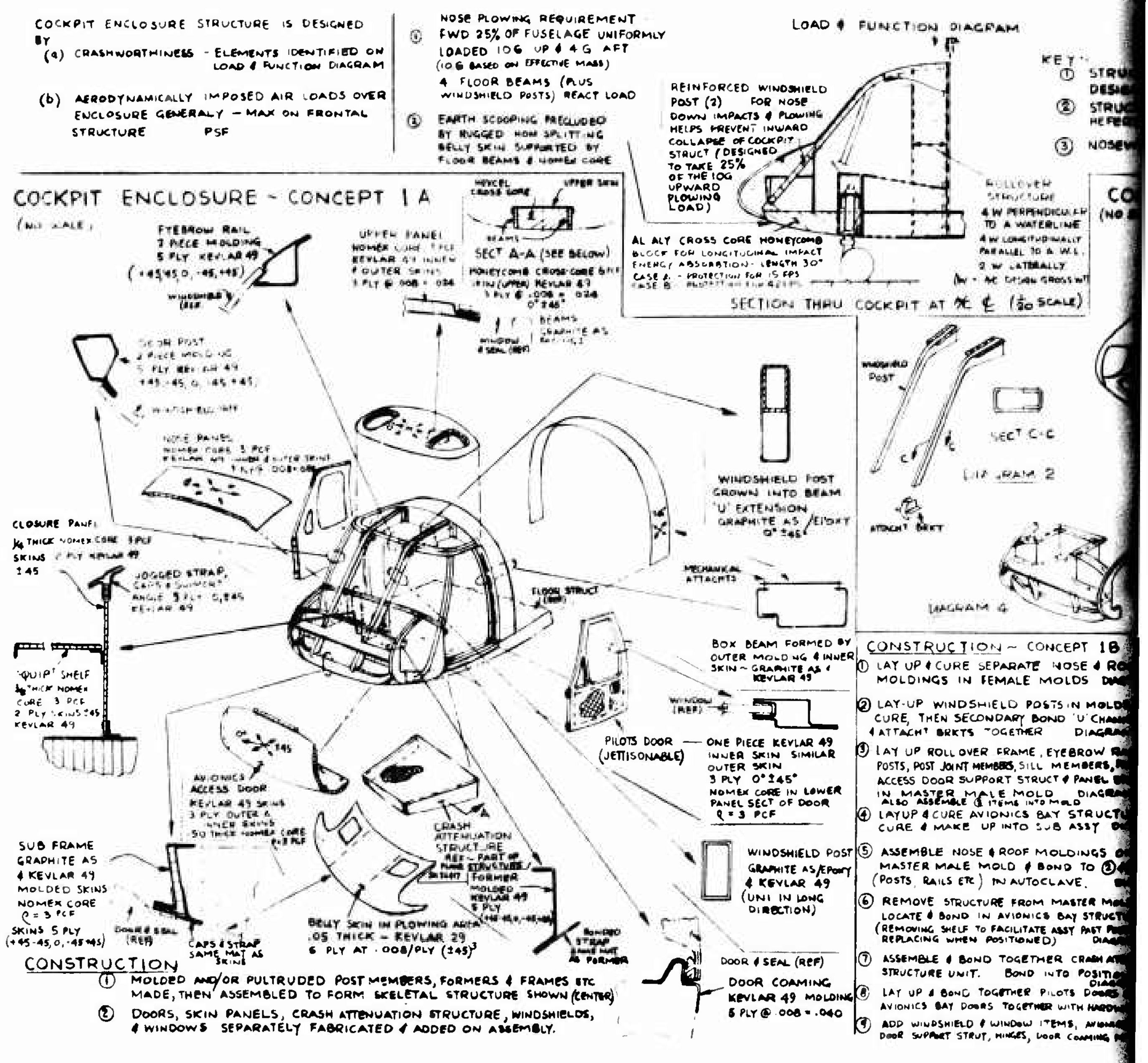
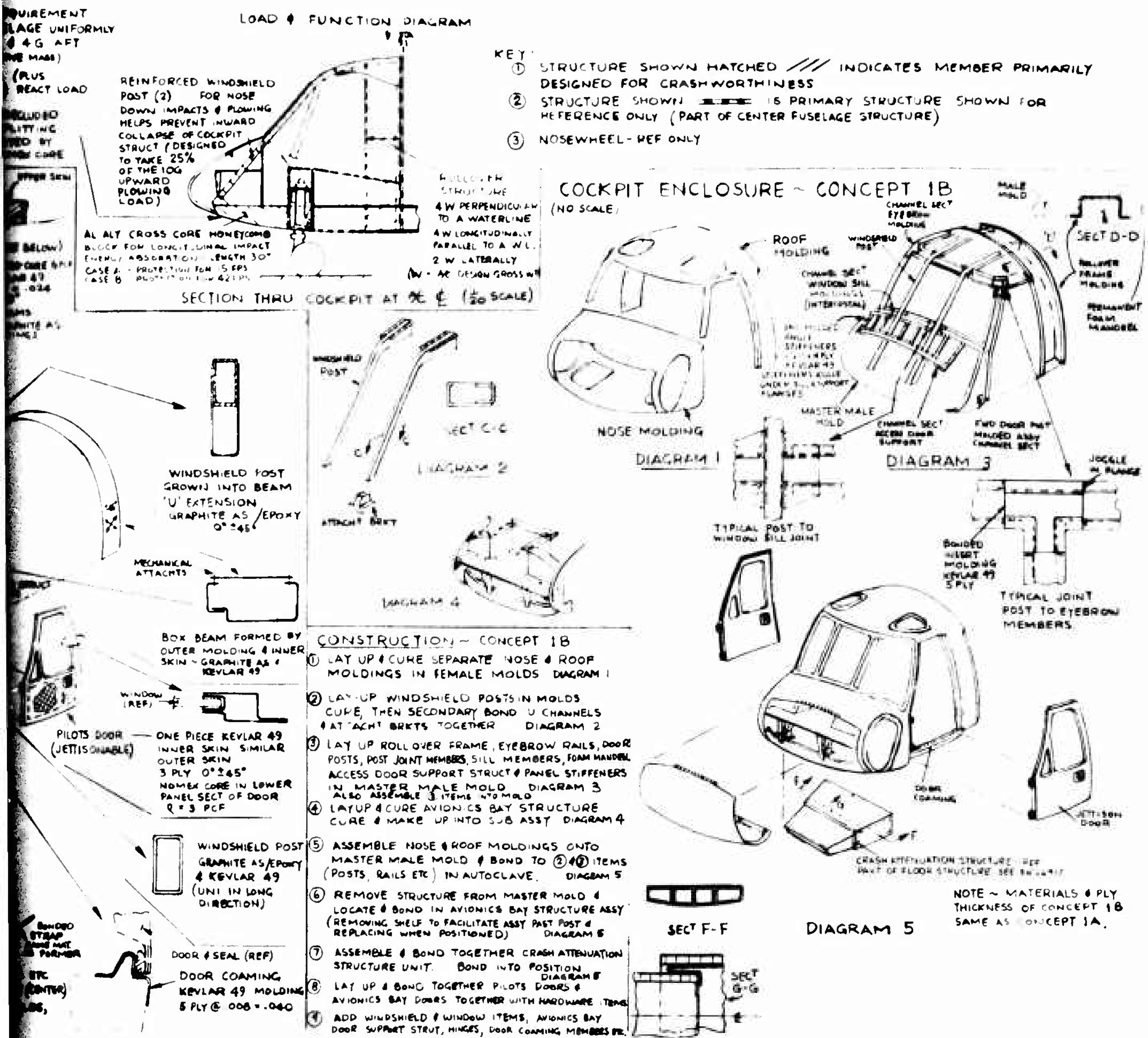
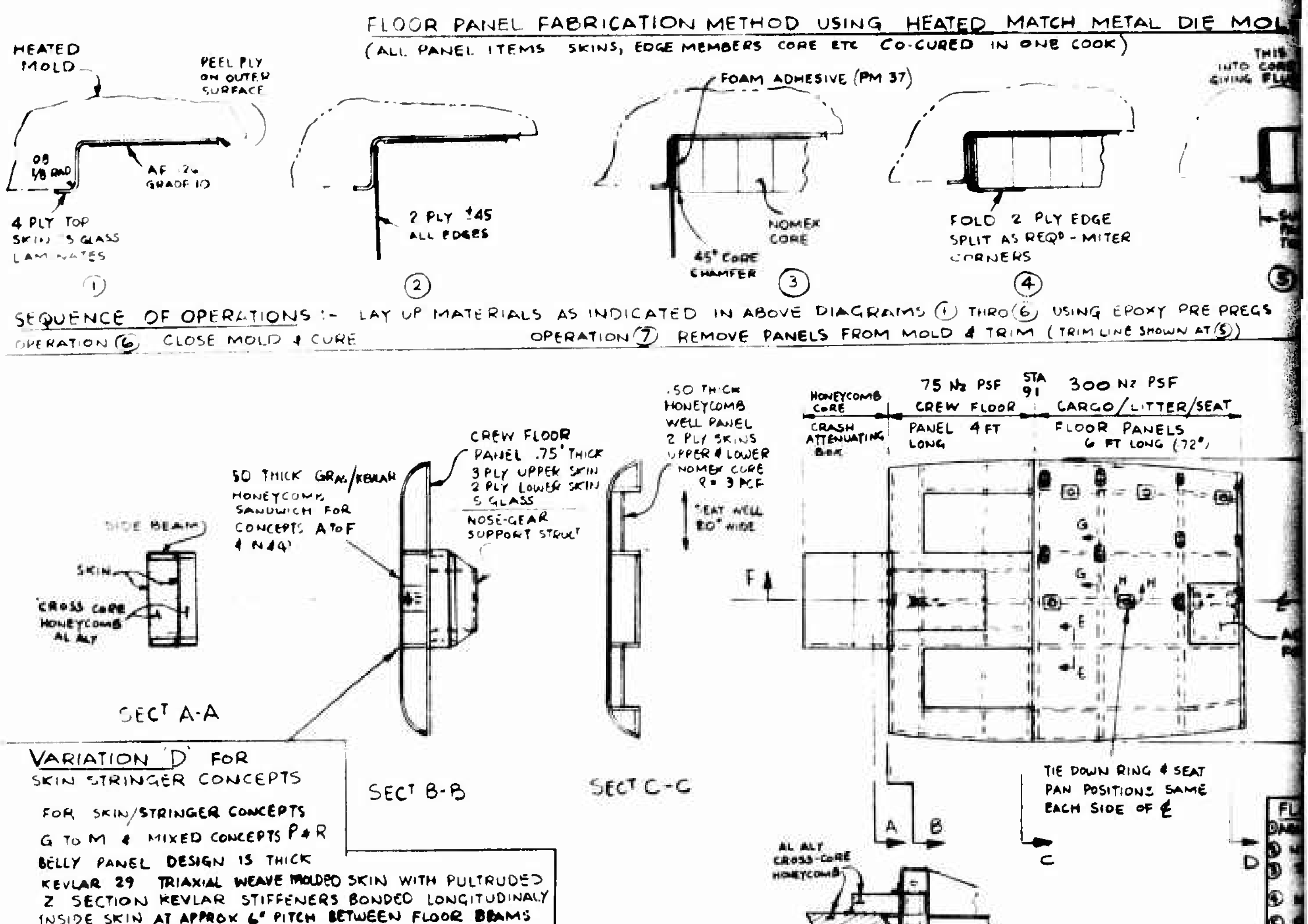


Figure B-11. MUT Cockpit Enclosure.



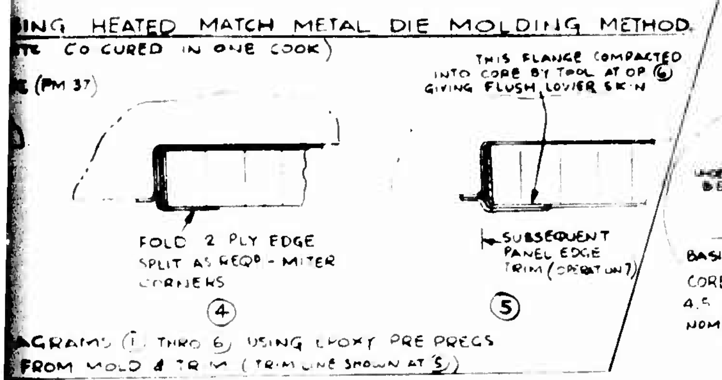
LOAD & FUNCTION DIAGRAM
KEY:
① STRUCTURE SHOWN HATCHED /// INDICATES MEMBER PRIMARILY DESIGNED FOR CRASHWORTHINESS
② STRUCTURE SHOWN ▬▬▬ IS PRIMARY STRUCTURE SHOWN FOR REFERENCE ONLY (PART OF CENTER FUSELAGE STRUCTURE)
③ NOSEWHEEL - REF ONLY
REINFORCED WINDSHIELD POST (2) FOR NOSE DOWN IMPACTS & PLOWING HELPS PREVENT INWARD COLLAPSE OF COCKPIT STRUCT (DESIGNED TO TAKE 25% OF THE 10G UPWARD PLOWING LOAD)
ROLLOVER STRUCTURE
4 W PERPENDICULAR TO A WATERLINE
4 W LONGITUDINALLY PARALLEL TO A W.L.
2 W LATERALLY
(W = AC DESIGN GROSS WT)
AL ALT CROSS CORE HONEYCOMB BLOCK FOR LONGITUDINAL IMPACT ENERGY ABSORBTION LENGTH 30"
SECTION THRU COCKPIT AT A/C ℄ (1/20 SCALE)
COCKPIT ENCLOSURE ~ CONCEPT 1B
(NO SCALE)
ROOF MOLDING
NOSE MOLDING
DIAGRAM 1
WINDSHIELD POST
SECT C-C
DIAGRAM 2
ATTACH BRKT
DIAGRAM 4
MALE MOLD
CHANNEL SECT EYEBROW MOLDING
WINDSHIELD POST
SECT D-D
ROLLOVER FRAME MOLDING
PERMANENT FOAM MANDREL
CHANNEL SECT WINDOW SILL MOLDINGS (INTERIOR)
MASTER MALE MOLD
CHANNEL SECT ACCESS DOOR SUPPORT
FWD DOOR POST MOLDED ASSY CHANNEL SECT
DIAGRAM 3
JOGGLE IN FLANGE
TYPICAL POST TO WINDOW SILL JOINT
BONDED INSERT MOLDING KEVLAR 49 5 PLY
TYPICAL JOINT POST TO EYEBROW MEMBERS
WINDSHIELD POST GROWN INTO BEAM 'U' EXTENSION GRAPHITE AS /EPOXY 0° ±45°
MECHANICAL ATTACH'TS
BOX BEAM FORMED BY OUTER MOLDING & INNER SKIN ~ GRAPHITE AS & KEVLAR 49
WINDOW (REF)
PILOTS DOOR (JETTISONABLE) — ONE PIECE KEVLAR 49 INNER SKIN SIMILAR OUTER SKIN 3 PLY 0° ±45° NOMEX CORE IN LOWER PANEL SECT OF DOOR ρ = 3 PCF
WINDSHIELD POST GRAPHITE AS/EPOXY & KEVLAR 49 (UNI IN LONG DIRECTION)
DOOR & SEAL (REF)
DOOR COAMING KEVLAR 49 MOLDING 5 PLY @ .008 = .040
CONSTRUCTION ~ CONCEPT 1B
① LAY UP & CURE SEPARATE NOSE & ROOF MOLDINGS IN FEMALE MOLDS DIAGRAM 1
② LAY-UP WINDSHIELD POSTS IN MOLDS CURE, THEN SECONDARY BOND U CHANNELS & ATTACH'T BRKTS TOGETHER DIAGRAM 2
③ LAY UP ROLL OVER FRAME, EYEBROW RAILS, DOOR POSTS, POST JOINT MEMBERS, SILL MEMBERS, FOAM MANDREL ACCESS DOOR SUPPORT STRUCT & PANEL STIFFENERS IN MASTER MALE MOLD DIAGRAM 3 ALSO ASSEMBLE ③ ITEMS INTO MOLD
④ LAY UP & CURE AVIONICS BAY STRUCTURE CURE & MAKE UP INTO SUB ASSY DIAGRAM 4
⑤ ASSEMBLE NOSE & ROOF MOLDINGS ONTO MASTER MALE MOLD & BOND TO ②&③ ITEMS (POSTS, RAILS ETC) IN AUTOCLAVE. DIAGRAM 5
⑥ REMOVE STRUCTURE FROM MASTER MOLD & LOCATE & BOND IN AVIONICS BAY STRUCTURE ASSY (REMOVING SHELF TO FACILITATE ASSY PAST POST & REPLACING WHEN POSITIONED) DIAGRAM 5
⑦ ASSEMBLE & BOND TOGETHER CRASH ATTENUATION STRUCTURE UNIT. BOND INTO POSITION DIAGRAM 5
⑧ LAY UP & BOND TOGETHER PILOTS DOORS & AVIONICS BAY DOORS TOGETHER WITH HARDWARE ITEMS
⑨ ADD WINDSHIELD & WINDOW ITEMS, AVIONICS BAY DOOR SUPPORT STRUT, HINGES, DOOR COAMING MEMBERS ETC.
DOOR COAMING
JETTISON DOOR
CRASH ATTENUATION STRUCTURE - REF PART OF FLOOR STRUCTURE
SECT F-F
DIAGRAM 5
SECT G-G
NOTE ~ MATERIALS & PLY THICKNESS OF CONCEPT 1B SAME AS CONCEPT 1A.

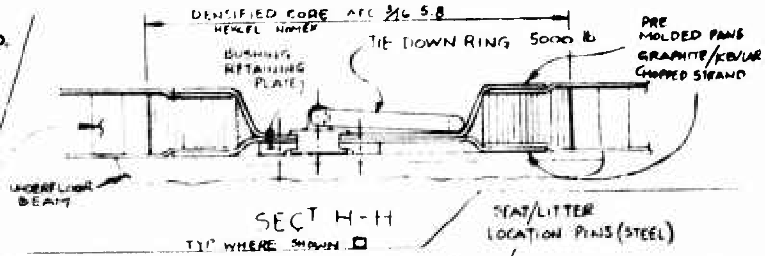


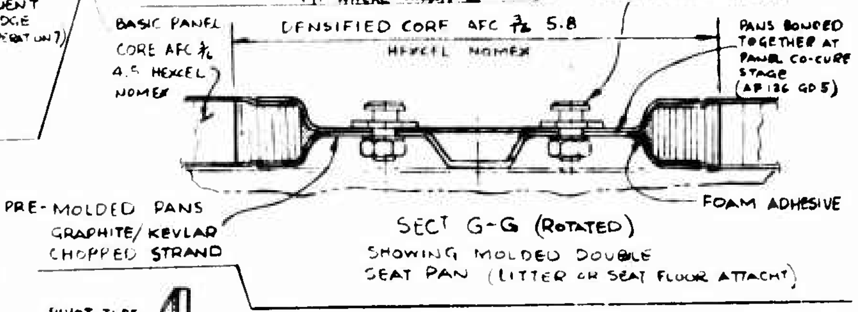
FLOOR PANEL DESIGN CRITERIA

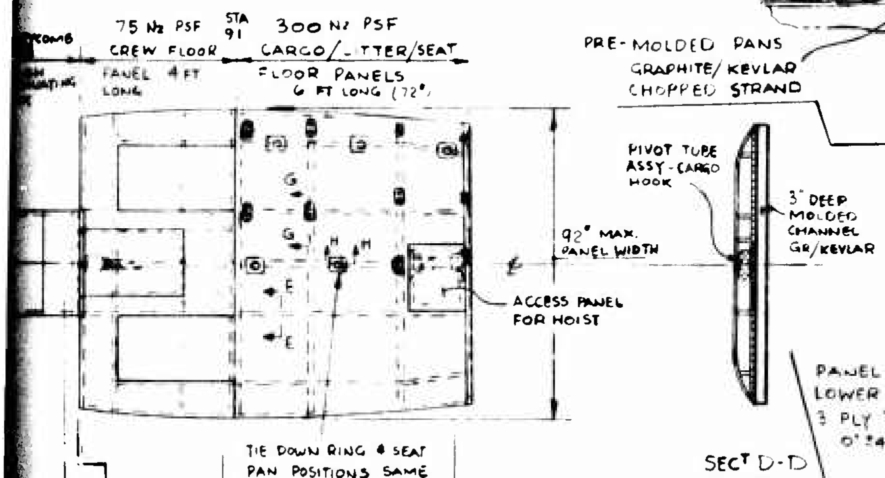
FLOOR TYPE	DESIGN LIMIT FLOOR PRESSURE	LOAD FACTOR Nz	OTHER REQUIREMENTS	LOCAL LOADING CAPACITY	PANEL IMPACT TE
CARGO FLOOR	300 Nz PSF	3.5 →	PLUS INCREMENTS DUE TO ANGULAR ACCELERATIONS FROM MANEUVERS OR THE LOAD FACTOR RESULTING FROM HARD LANDINGS UP TO 10 FPS (WHICHEVER IS GREATER)	50 PSI LIMIT LOAD APPLIED TO A SINGLE .5 SQ FT AREA (8"×9" TO 3"×24") WITHIN ANY 6 SQ FT AREA	LESS THAN .03 IN OF PANEL SURFAC CORNER DROP OF PINE BOX WITH 0.
CREW FLOOR	75 Nz PSF	3.5 →	AS ABOVE	NOT APPLICABLE	NONE

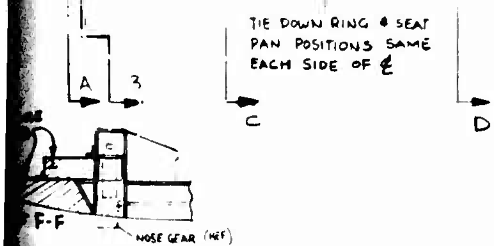
Figure B-12. MUT Underfloor Structure and Floor Panels.

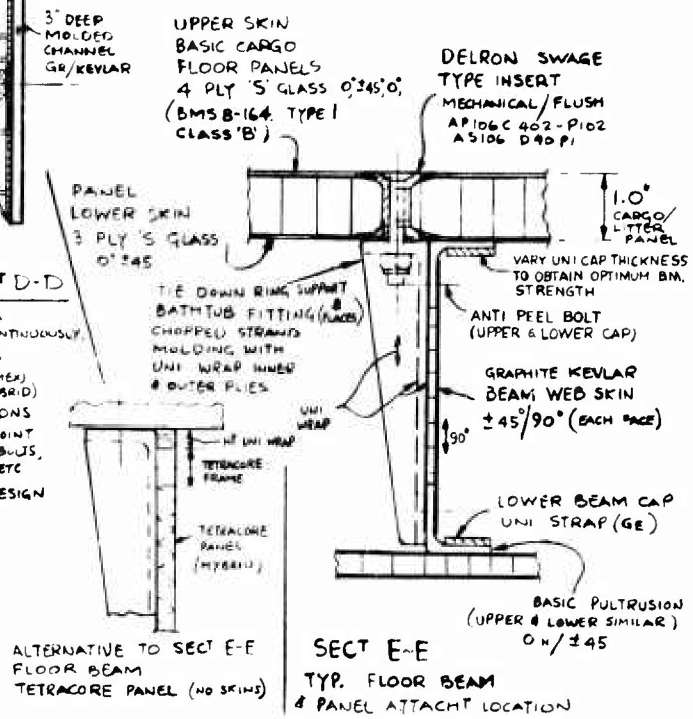












FLOOR BEAM CRITERIA

(1) ABILITY TO VARY CAP AREA CONTINUOUSLY.
(2) NO FASTENERS IN UNI CAP.
(3) THIN CORE 1/8 HONEYCOMB (NOMEX) & TETRACORE (HYBRID)
(4) MIN HAND LAYUP OPERATIONS
(5) CAPABILITY FOR REACTING POINT LOADS FROM PANEL ATTACHT BOLTS, TIE DOWN RINGS, SEAT PANS ETC
(6) FAILSAFE & CRASHWORTHY DESIGN

LOCAL LOADING CAPACITY	PANEL IMPACT TEST
50 PSI LIMIT LOAD APPLIED TO A SINGLE .5 SQ FT AREA (8"x9" TO 3"x24") WITHIN ANY 6 SQ FT AREA	LESS THAN .03 INDENTATION OF PANEL SURFACE AFTER 15" CORNER DROP OF 200 LB PINE BOX WITH 0.5" RADIUS CORNER
NOT APPLICABLE	NONE

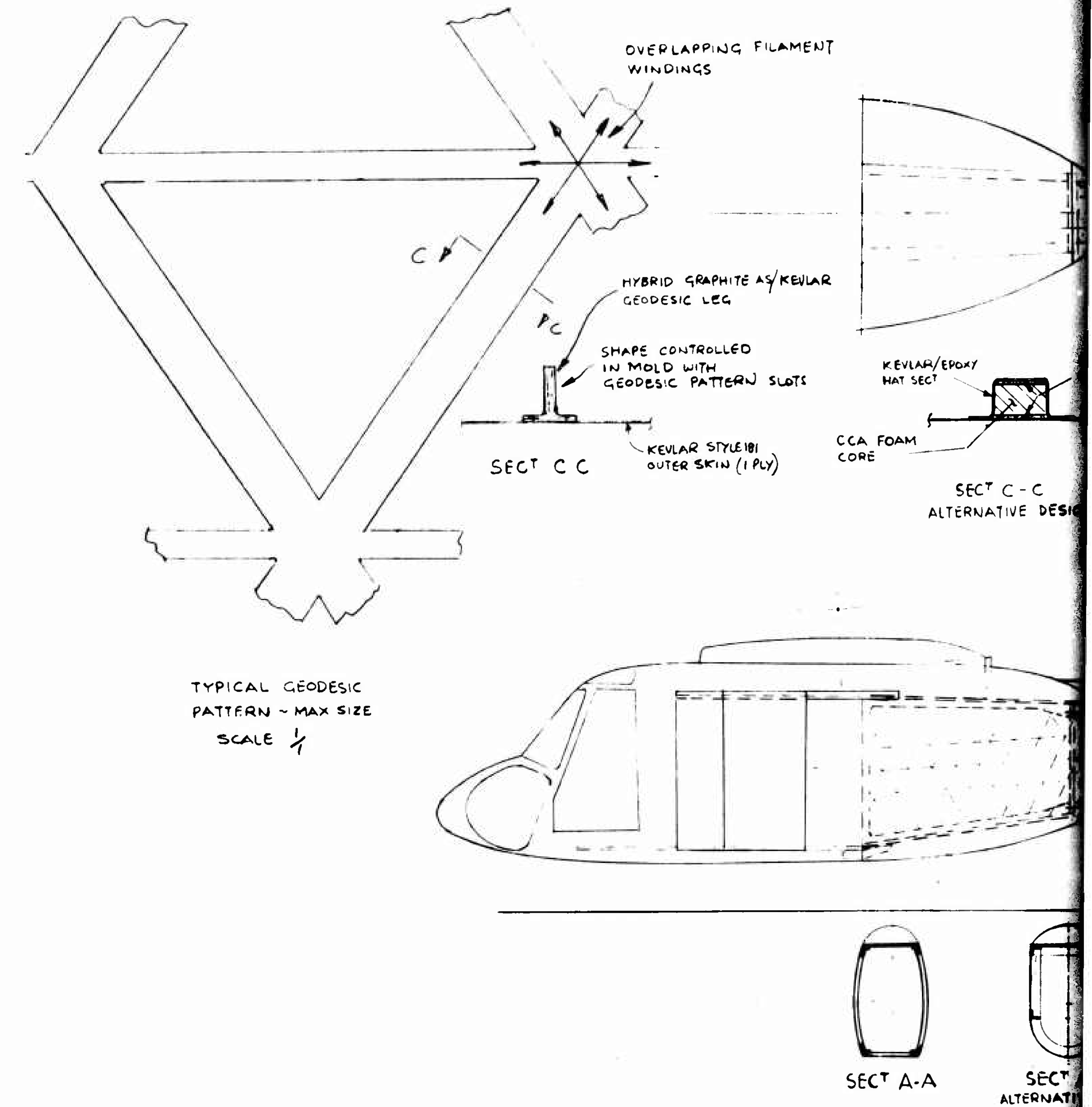
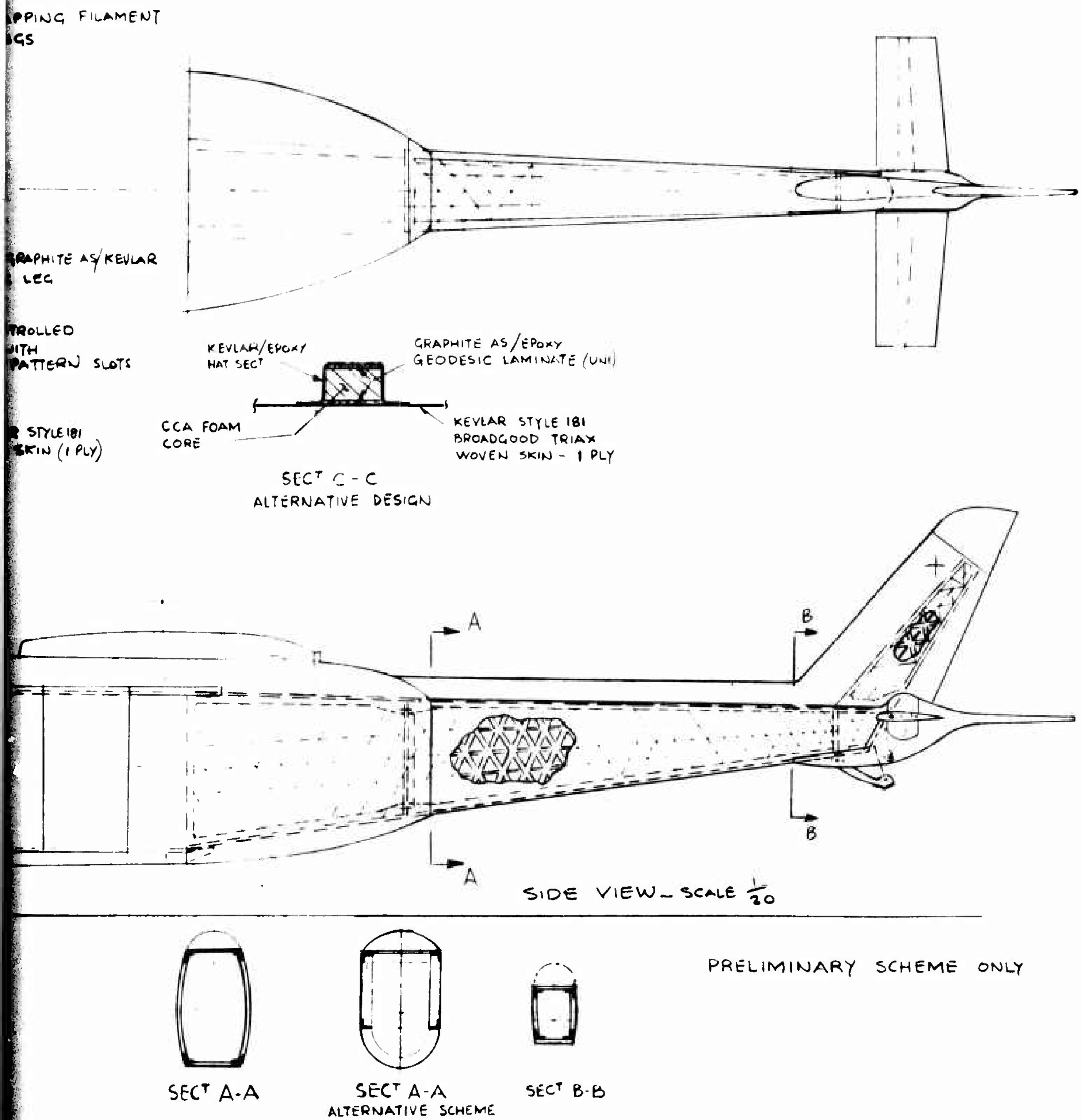


Figure B-13. MUT Mixed-Concept Geodesic Structure.



PRELIMINARY SCHEME ONLY

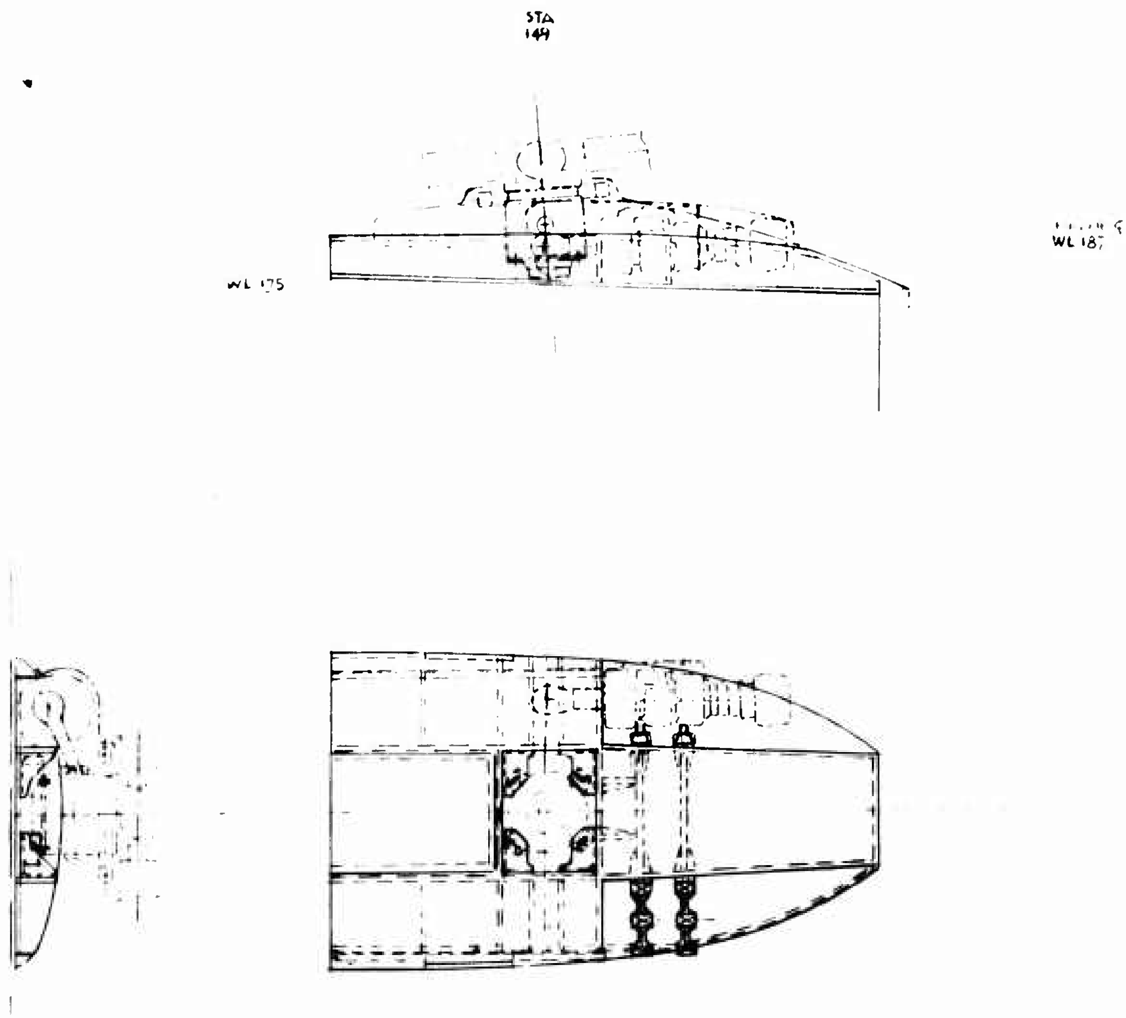


Figure B-14. MUT Airframe Upper Deck Module.

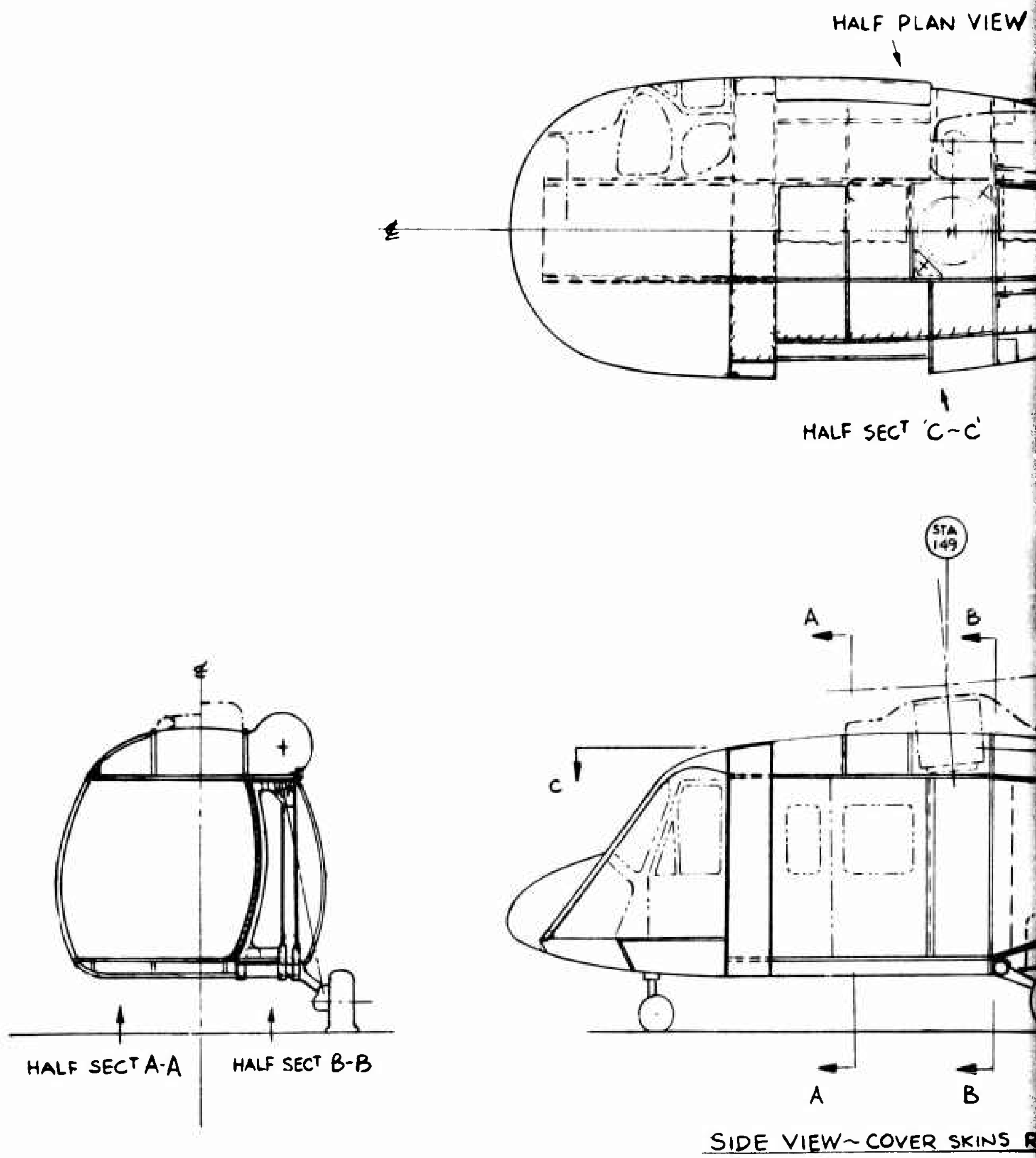
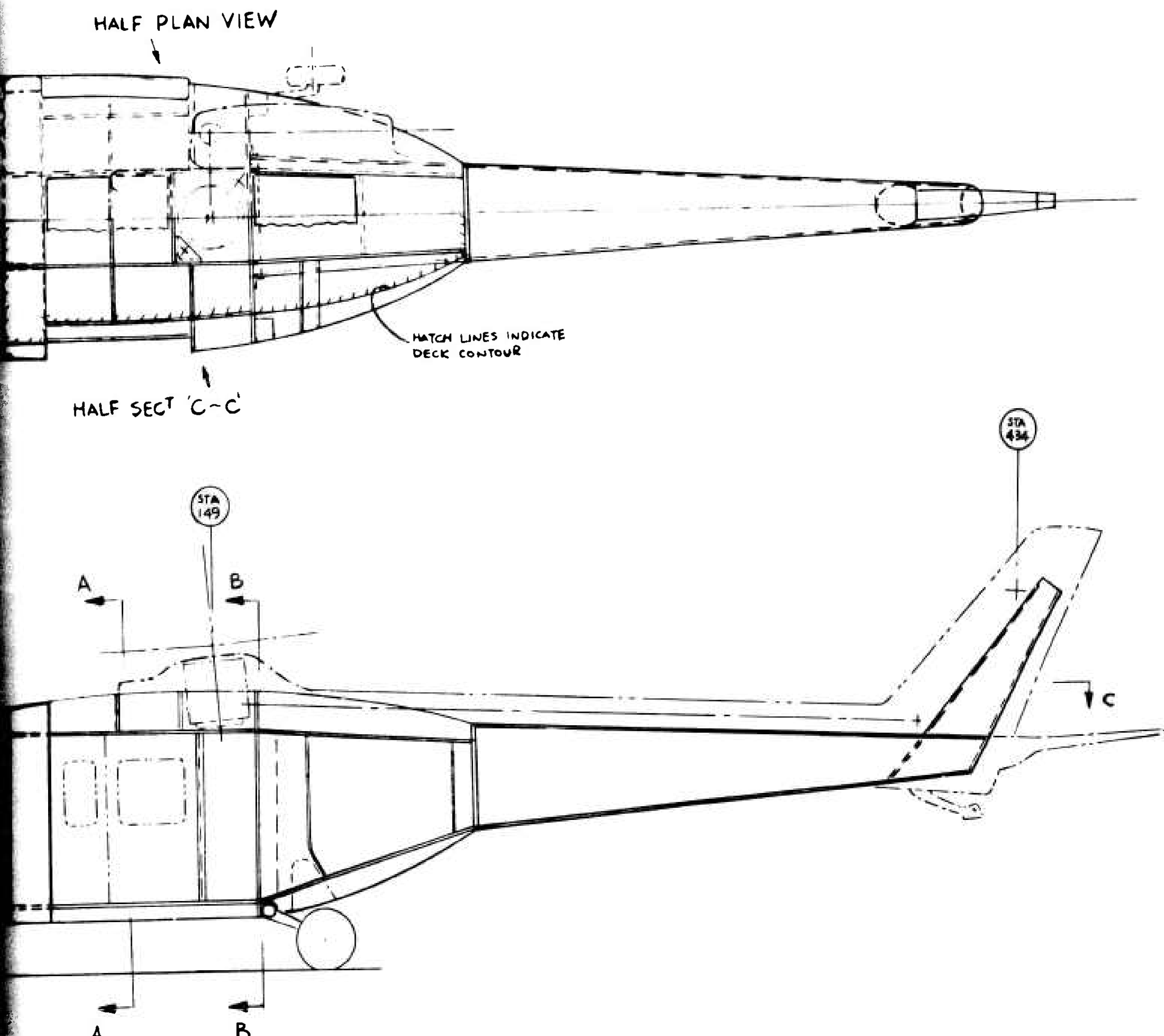


Figure B-15. MUT Composite Structure Primary Airframe Arrangement.

P.358-Blank



IDE VIEW~COVER SKINS REMOVED

gement.

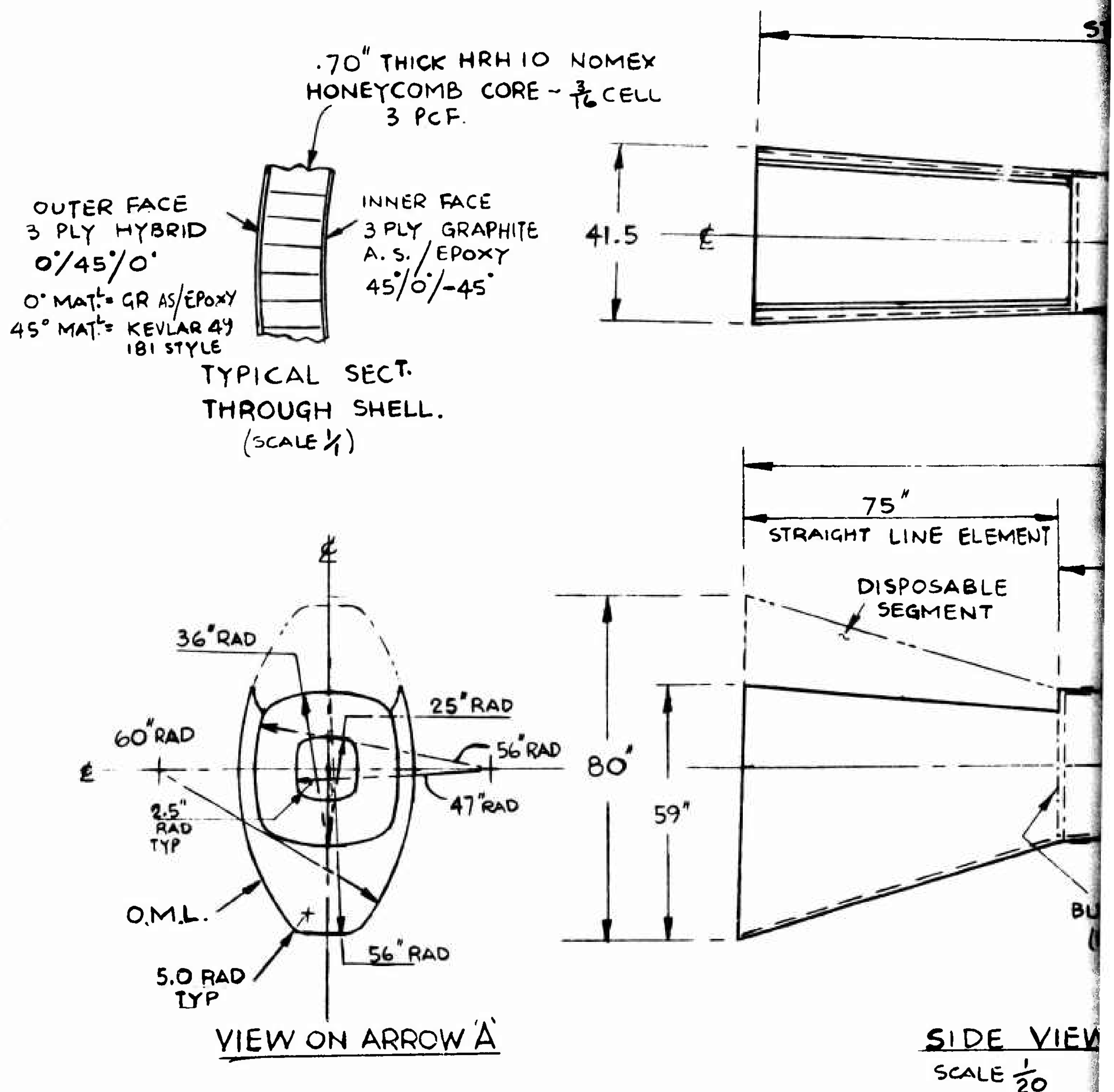
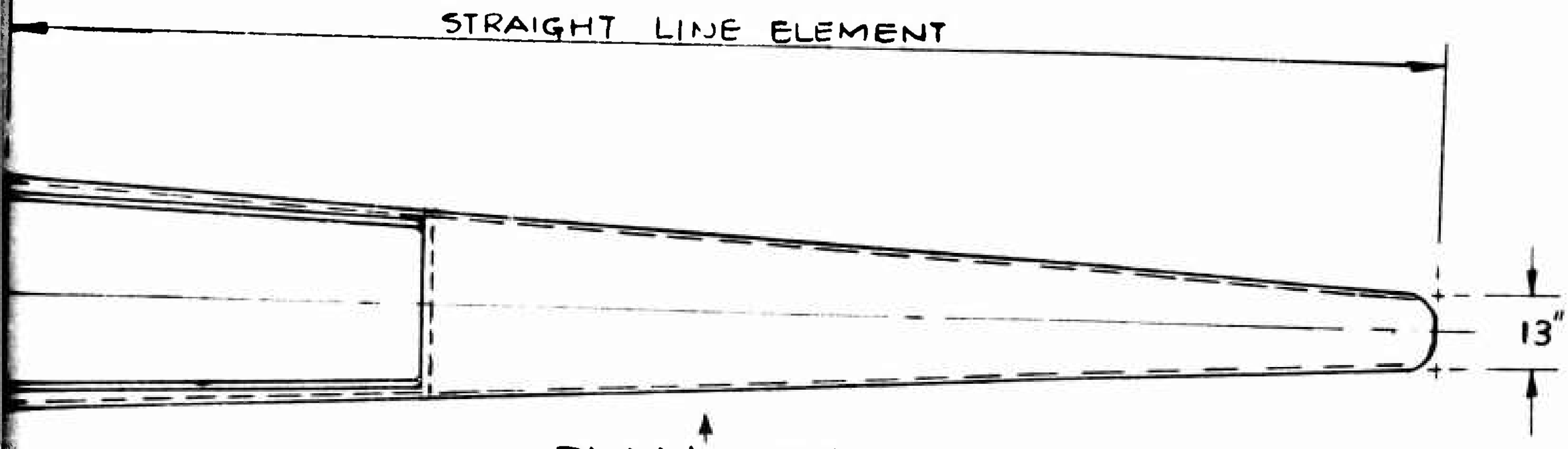
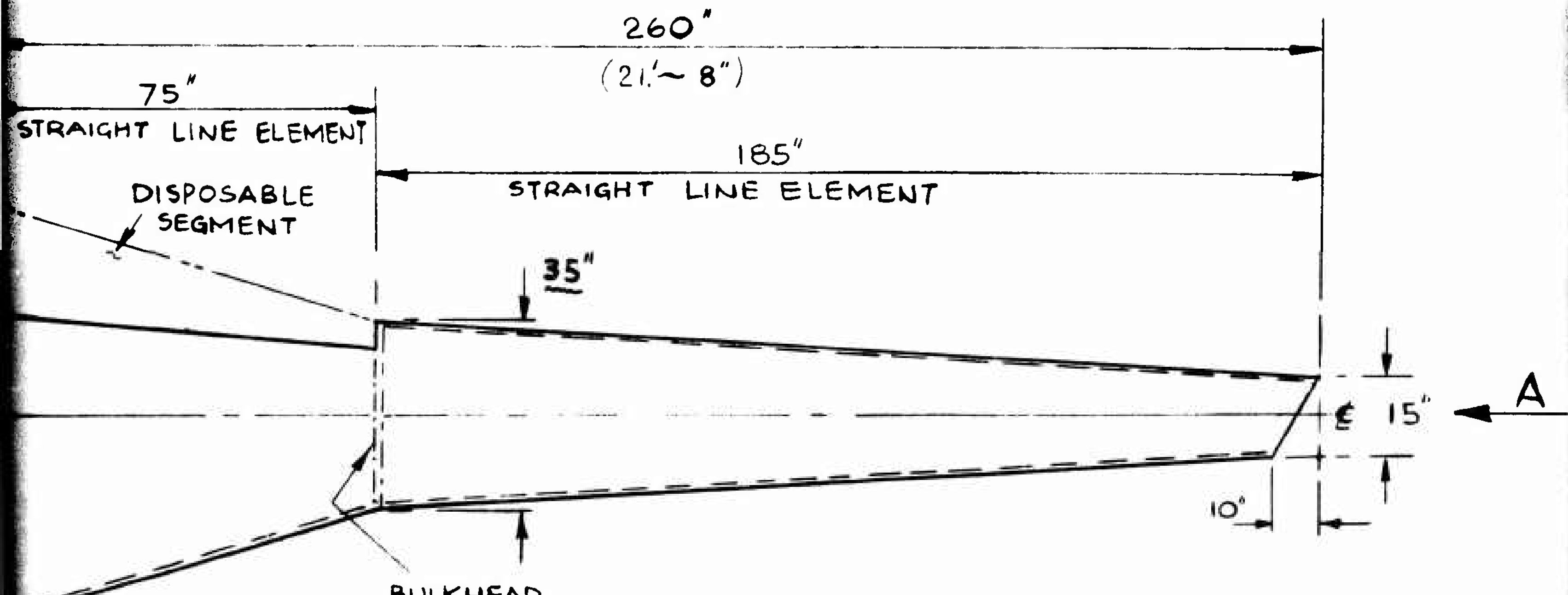


Figure B-16. MUT Tailboom Configuration for Wound Concept.



PLAN VIEW



SIDE VIEW

SCALE $\frac{1}{20}$

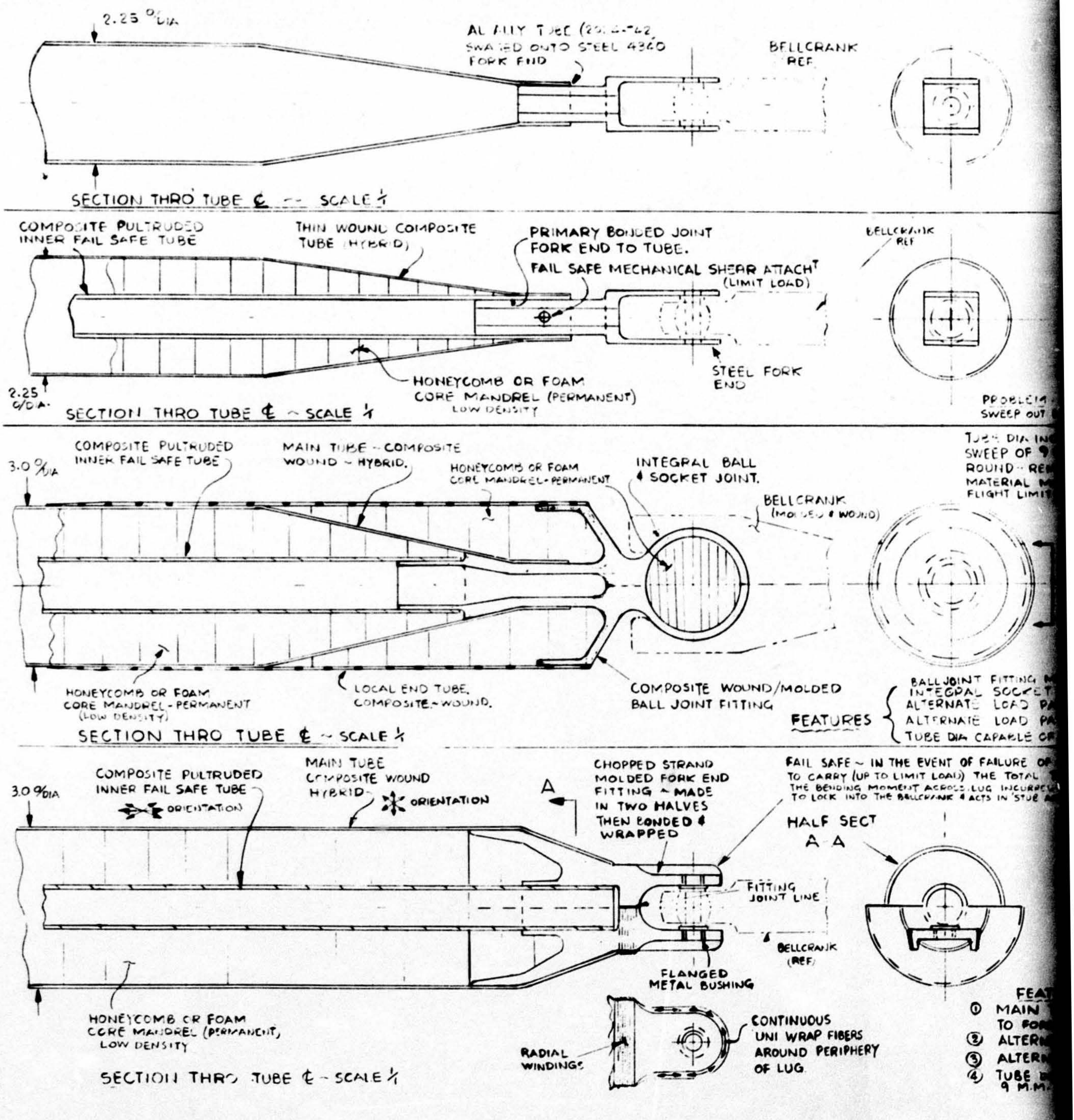
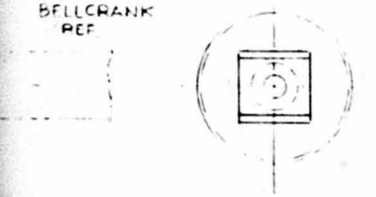


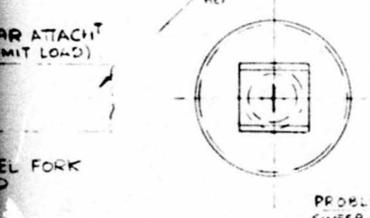
Figure B-17. MUT Composite Control Link Concepts.



REGULAR ALL METAL CONTROL TUBE DESIGN.

NOT FAIL SAFE.
NOT BALLISTICALLY TOLERANT.

NOTE.
COMPOSITE DESIGNS 1, 2, & 3 ARE CONCEPTUAL ONLY & HAVE NOT BEEN SIZED FOR SPECIFIC APPLIED LOADS



COMPOSITE CONCEPT 1. CONTROL TUBE DESIGN.

PARTLY FAIL SAFE.

PARTLY BALLISTICALLY TOLERANT

PROBLEM – TUMBLED 9 MM (.303) ROUND WILL SWEEP OUT 1.5" OF TUBE DIA – INSUFFICIENT MATERIAL LEFT TO TAKE REQD LOAD

FABRICATION SEQUENCE

a MAKE COMPOSITE PULTRUDED INNER TUBE & METAL FORK END DETAILS

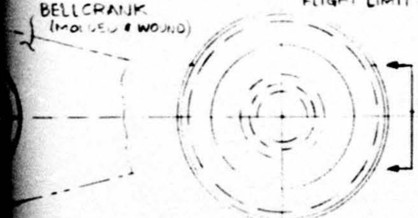
b FIT UP & BOND INNER TUBE TO FORK END USING EPOXY ADHESIVE

c MOLD ONTO TUBE & FORK ASSY THE MANDREL EITHER FOAM (POLYSTYRENE) OR NOMEX CORE WRAPPED & BUTT JOINTED WITH ADHESIVE.

d MACHINE MANDREL TO PRECISE SHAPE REQD

e WIND COMPOSITE MAIN TUBE OVER MANDREL & CURE ASSY.

TUBE DIA INCREASED IN THIS CONCEPT. SWEEP OF 9 M.M. TUMBLED ROUND – REMAINING TUBE MATERIAL MUST SUSTAIN FLIGHT LIMIT LOADS.



COMPOSITE CONCEPT 2
INTEGRAL CONTROL TUBE & BELLCRANK DESIGN.

IMPROVED. FAIL SAFETY (OVER CONCEPT 1)

BALLISTICALLY TOLERANT.

(UP TO 9MM TUMBLED ROUND)
EXCEPT FOR NECK OF BALL.

FEATURES
- BALL JOINT FITTING MECHANICALLY LOCKED TO MAIN TUBE ASSY (ALSO BONDED)
- INTEGRAL SOCKET JOINT – NO NUTS & BOLTS.
- ALTERNATE LOAD PATH SYSTEM IN TUBE.
- ALTERNATE LOAD PATH SYSTEM IN END FITTING.
- TUBE DIA CAPABLE OF ABSORBING A TUMBLED 9MM ROUND WITHOUT FAILURE.

ND/MOLDED
NG

FABRICATION SEQUENCE.

a MAKE MOLDED TUBE END BALL JOINT FITTING BY FIRST MAKING CHOP STRAND MOLDING IS TWO HALVES.

b BOND TWO MOLDED HALVES TOGETHER WITH BALL SECTION ENCLOSING A DENSE FOAM BALL.

c WIND FIBERS ARROUND COMPLETE ASSEMBLY TO FORM INTEGRAL FITTING

d CONTOUR GRIND FORMED BALL ENDS FOR ROUNDNESS THEN SPRAY WITH A TETRAFLUOROETHYLENE LUBRICANT

e THE BELLCRANK SEGMENTS COMPRISE TWO HALF MOLDINGS USING CHOPPED FIBER METHOD THESE HALVES FOR RETAINING THE BALL ARE ALSO SPRAYED WITH THE SAME LUBRICANT IN THE BALL AREA.

f THE TWO BELLCRANK MOLDINGS ARE THEN BONDED TOGETHER ENCLOSING THE BALL END OF THE BALL JOINT FITTING. FIBERS ARE THEN WRAPPED AROUND THE UNIT DIAGONALLY & RADIALLY & ASSY CURED

g. INNER COMPOSITE TUBE IS BONDED TO BALL JOINT FITTING

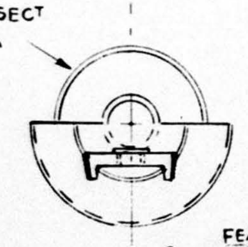
h FIT & SHAPE MANDREL AS c & d OF CONCEPT 1

j WIND COMPOSITE MAIN TUBE OVER MANDREL

k WIND LOCAL END TUBE OVER END MANDREL & CURE

FAIL SAFE ~ IN THE EVENT OF FAILURE OF ONE LUG THE REMAINING LUG IS DESIGNED TO CARRY (UP TO LIMIT LOAD) THE TOTAL TENSION, OR COMPRESSION LOAD TOGETHER WITH THE BENDING MOMENT ACROSS LUG INCURRED BY THE OFFSET LOAD. THE JOINT BOLT IS DESIGNED TO LOCK INTO THE BELLCRANK & ACTS IN 'STUB AXLE' MANNER WHEN ONE FORK END LUG FAILS.

HALF SECT
A-A



ONTINUOUS
NI WRAP FIBERS
ROUND PERIPHERY
LUG.

COMPOSITE CONCEPT 3.
CONTROL TUBE DESIGN.

IMPROVED FAIL SAFETY (OVER CONCEPT 1)

BALLISTICALLY TOLERANT

(UP TO 9MM TUMBLED ROUND)

EXCEPT FOR FITTING ENDS

FABRICATION SEQUENCE.

a MOLD FORK END FITTING IN TWO HALVES USING CHOP STRAND MOLDING METHOD WITH INTEGRAL UNI WRAP FIBERS AROUND LUG

b FIT UP & BOND INNER PULTRUDED TUBE TO ASSEMBLED FITTING HALVES WHICH ARE ALSO BONDED TOGETHER

c FIT & SHAPE MANDREL AS c & d OF CONCEPT 1

d WIND COMPOSITE MAIN TUBE OVER MANDREL

e ADD RADIAL WINDING AS SHOWN IN SKETCH & CURE

f FACE OFF LUG SURFACES & BOND IN BUSHES

FEATURES

1. MAIN TUBE SKIN MECHANICALLY LOCKED TO FORK END FITTING.
2. ALTERNATE LOAD PATH SYSTEM IN TUBE
3. ALTERNATE LOAD SYSTEM IN FITTING
4. TUBE DIA CAPABLE OF ABSORBING A TUMBLED 9 M.M. ROUND WITHOUT FAILURE OF CONTROL

2

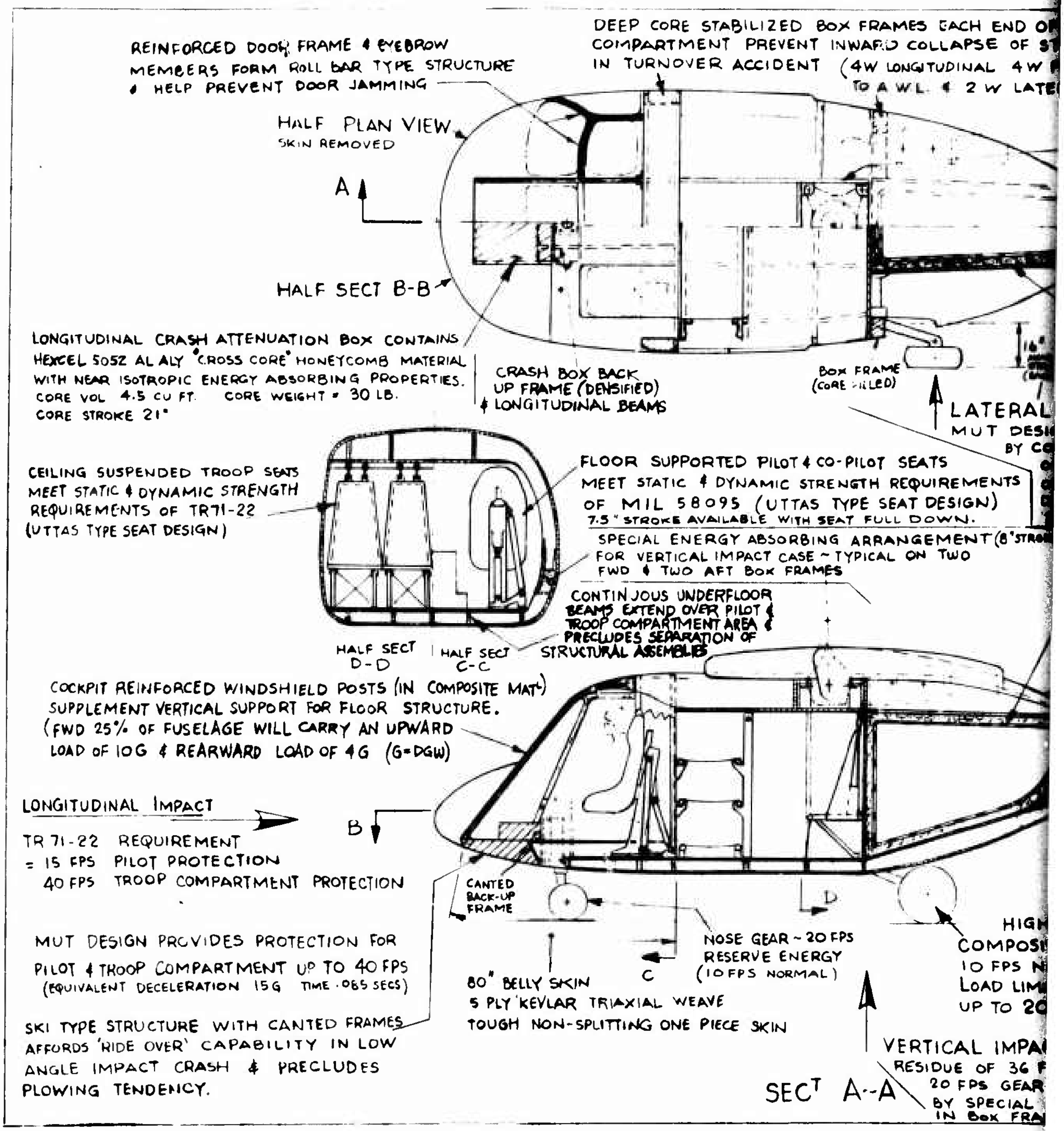
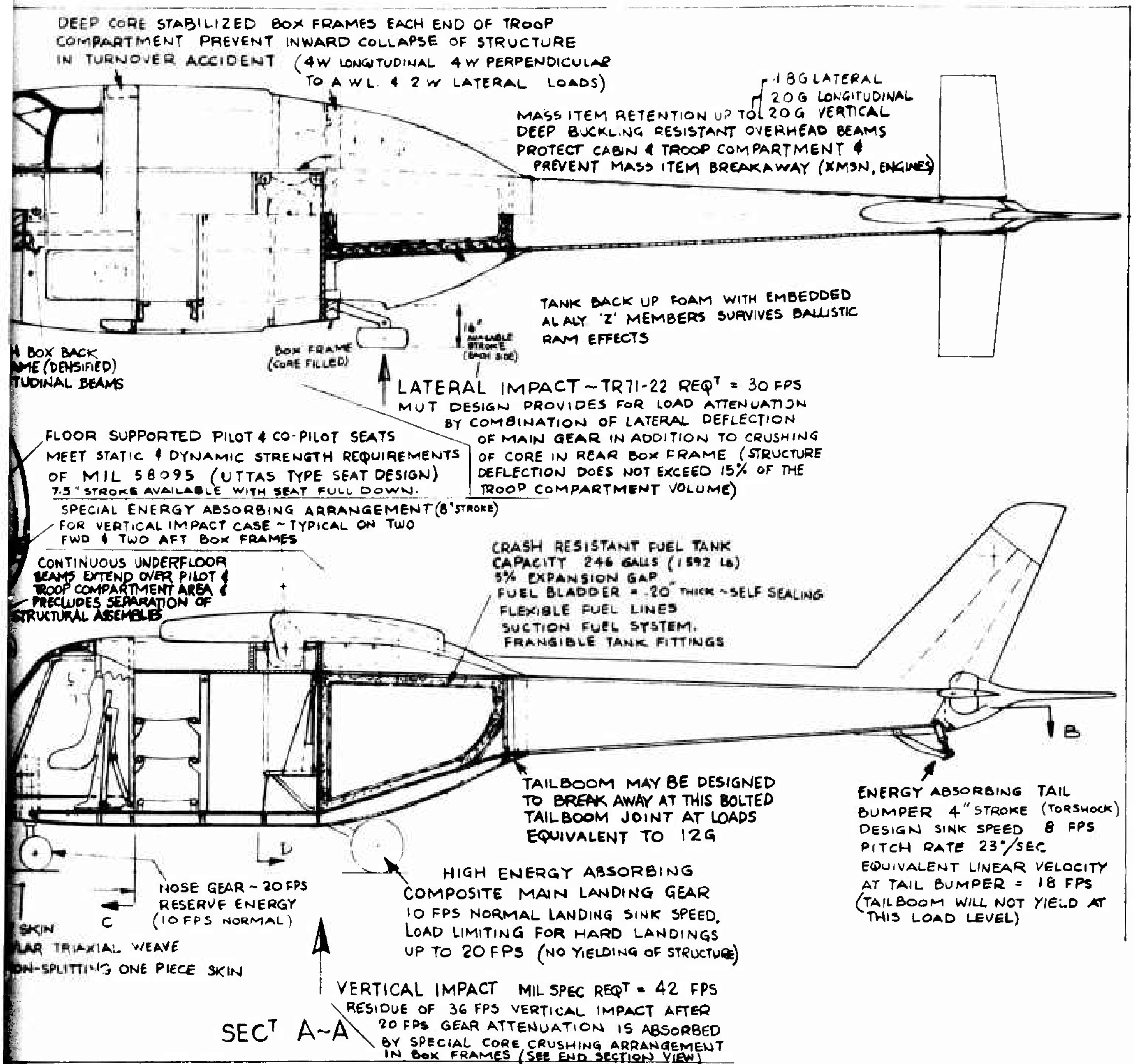


Figure B-18. MUT Airframe Crashworthiness Features.


DEEP CORE STABILIZED BOX FRAMES EACH END OF TROOP COMPARTMENT PREVENT INWARD COLLAPSE OF STRUCTURE IN TURNOVER ACCIDENT (4W LONGITUDINAL 4W PERPENDICULAR TO A W.L. & 2W LATERAL LOADS)
18G LATERAL
20G LONGITUDINAL
MASS ITEM RETENTION UP TO 20G VERTICAL
DEEP BUCKLING RESISTANT OVERHEAD BEAMS PROTECT CABIN & TROOP COMPARTMENT & PREVENT MASS ITEM BREAKAWAY (XMSN, ENGINES)
TANK BACK UP FOAM WITH EMBEDDED AL ALY 'Z' MEMBERS SURVIVES BALLISTIC RAM EFFECTS
16" AVAILABLE STROKE (EACH SIDE)
BOX FRAME (CORE FILLED)
BOX BACK
ME (DENSIFIED)
TUDINAL BEAMS
LATERAL IMPACT ~ TR71-22 REQ'T = 30 FPS
MUT DESIGN PROVIDES FOR LOAD ATTENUATION BY COMBINATION OF LATERAL DEFLECTION OF MAIN GEAR IN ADDITION TO CRUSHING OF CORE IN REAR BOX FRAME (STRUCTURE DEFLECTION DOES NOT EXCEED 15% OF THE TROOP COMPARTMENT VOLUME)
FLOOR SUPPORTED PILOT & CO-PILOT SEATS MEET STATIC & DYNAMIC STRENGTH REQUIREMENTS OF MIL 58095 (UTTAS TYPE SEAT DESIGN)
7.5" STROKE AVAILABLE WITH SEAT FULL DOWN.
SPECIAL ENERGY ABSORBING ARRANGEMENT (8" STROKE) FOR VERTICAL IMPACT CASE ~ TYPICAL ON TWO FWD & TWO AFT BOX FRAMES
CONTINUOUS UNDERFLOOR BEAMS EXTEND OVER PILOT & TROOP COMPARTMENT AREA & PRECLUDES SEPARATION OF STRUCTURAL ASSEMBLIES
CRASH RESISTANT FUEL TANK
CAPACITY 246 GALLS (1592 LB)
5% EXPANSION GAP
FUEL BLADDER = .20" THICK ~ SELF SEALING
FLEXIBLE FUEL LINES
SUCTION FUEL SYSTEM.
FRANGIBLE TANK FITTINGS
B
TAILBOOM MAY BE DESIGNED TO BREAK AWAY AT THIS BOLTED TAILBOOM JOINT AT LOADS EQUIVALENT TO 12G
ENERGY ABSORBING TAIL BUMPER 4" STROKE (TORSHOCK)
DESIGN SINK SPEED 8 FPS
PITCH RATE 23°/SEC
EQUIVALENT LINEAR VELOCITY AT TAIL BUMPER = 18 FPS
(TAILBOOM WILL NOT YIELD AT THIS LOAD LEVEL)
D
NOSE GEAR ~ 20 FPS RESERVE ENERGY (10 FPS NORMAL)
C
HIGH ENERGY ABSORBING COMPOSITE MAIN LANDING GEAR
10 FPS NORMAL LANDING SINK SPEED.
LOAD LIMITING FOR HARD LANDINGS UP TO 20 FPS (NO YIELDING OF STRUCTURE)
SKIN
AR TRIAXIAL WEAVE
N-SPLITTING ONE PIECE SKIN
VERTICAL IMPACT MIL SPEC REQ'T = 42 FPS
RESIDUE OF 36 FPS VERTICAL IMPACT AFTER 20 FPS GEAR ATTENUATION IS ABSORBED BY SPECIAL CORE CRUSHING ARRANGEMENT IN BOX FRAMES (SEE END SECTION VIEW)
SECT A~A

res.

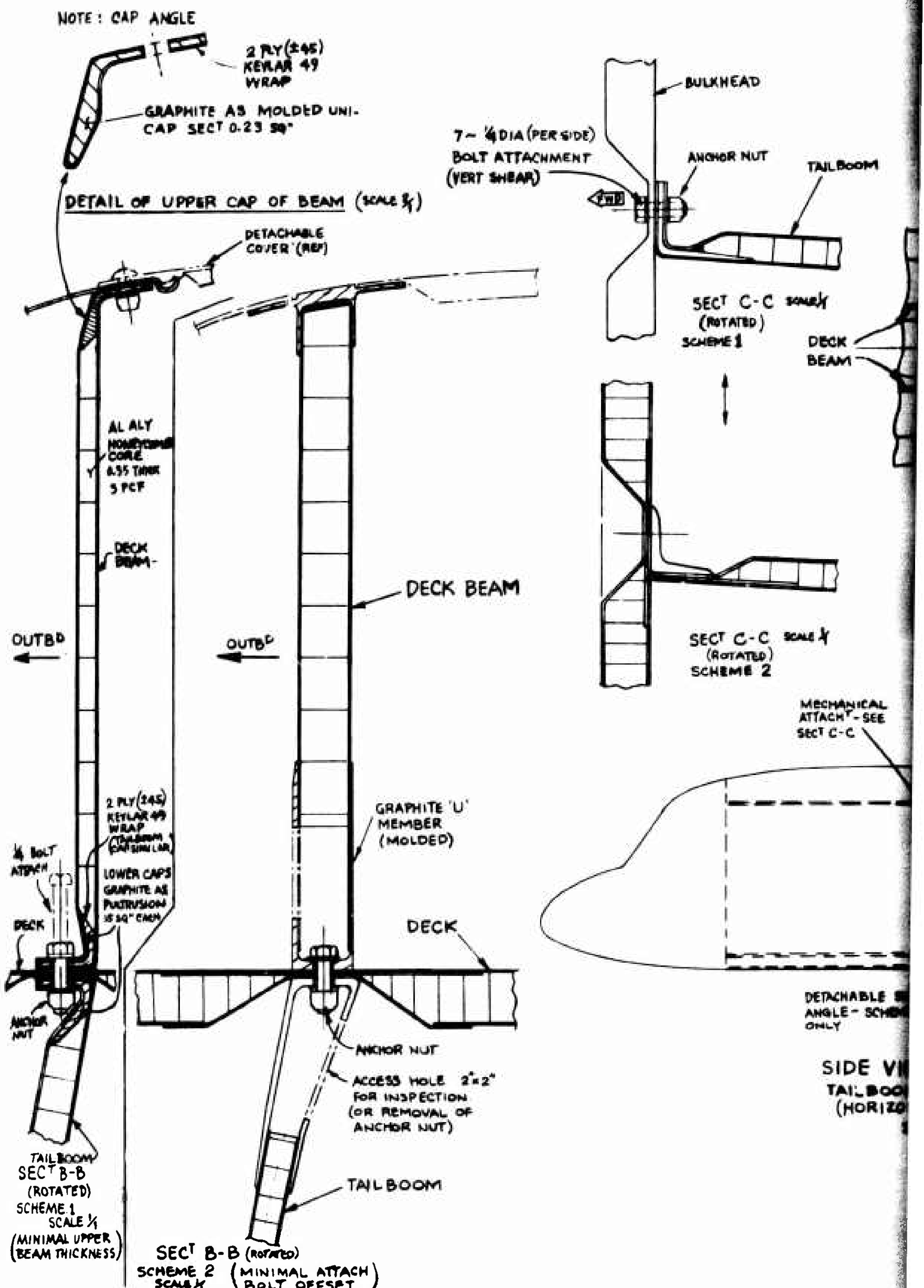
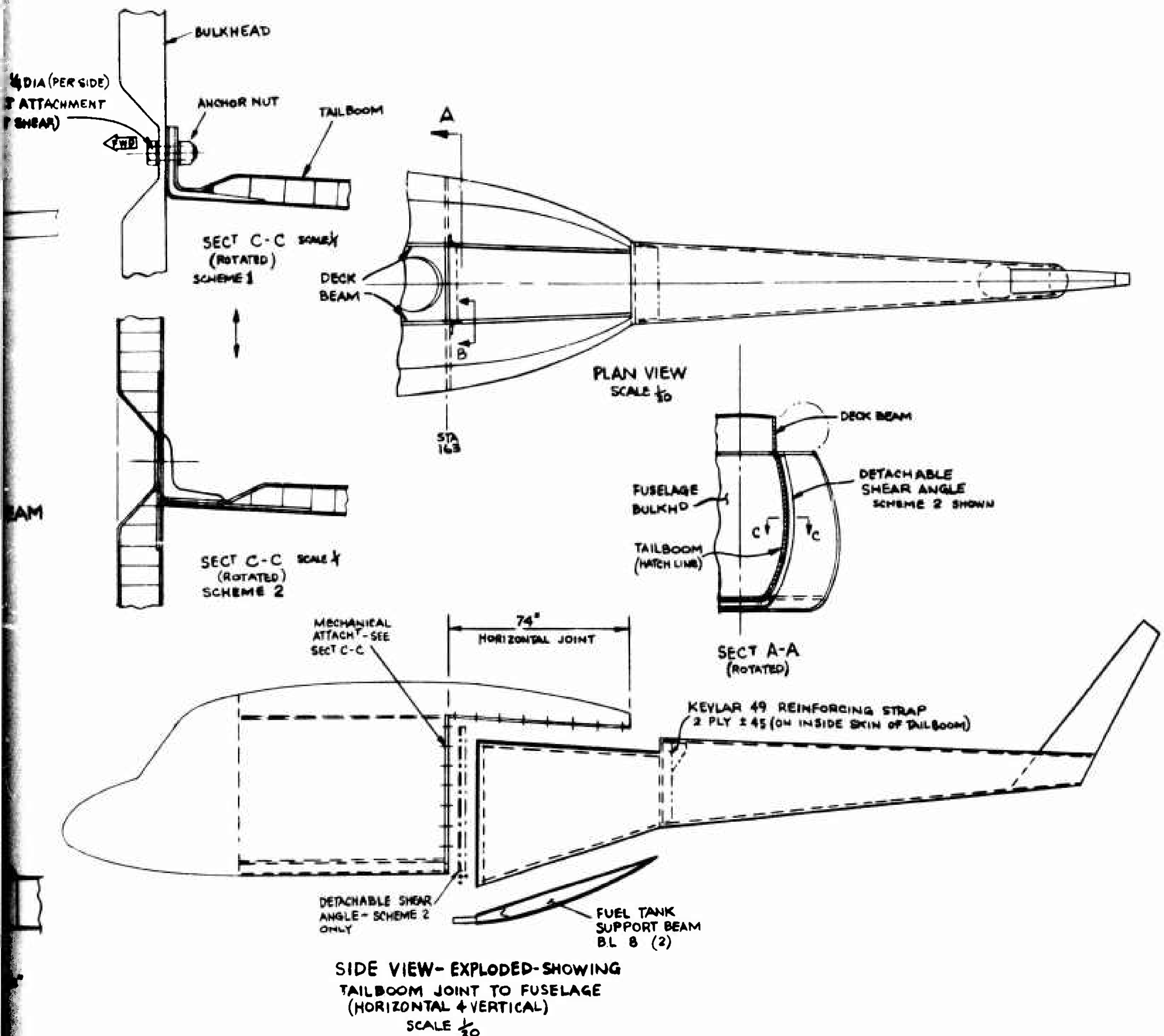


Figure B-19. MUT Integral Tailboom/Fuel Bay Joint to Fuselage Study.



o Fuselage Study.

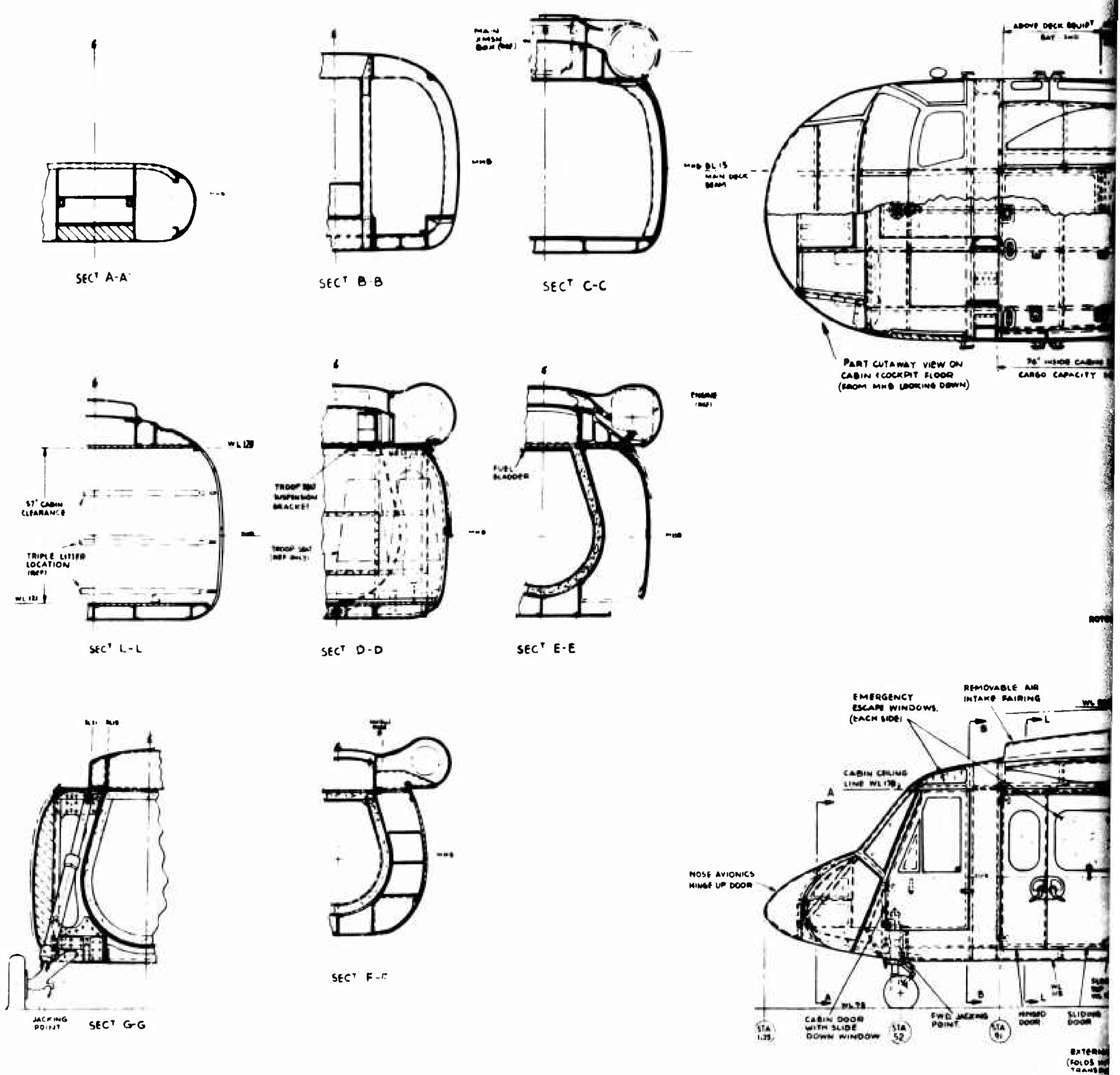
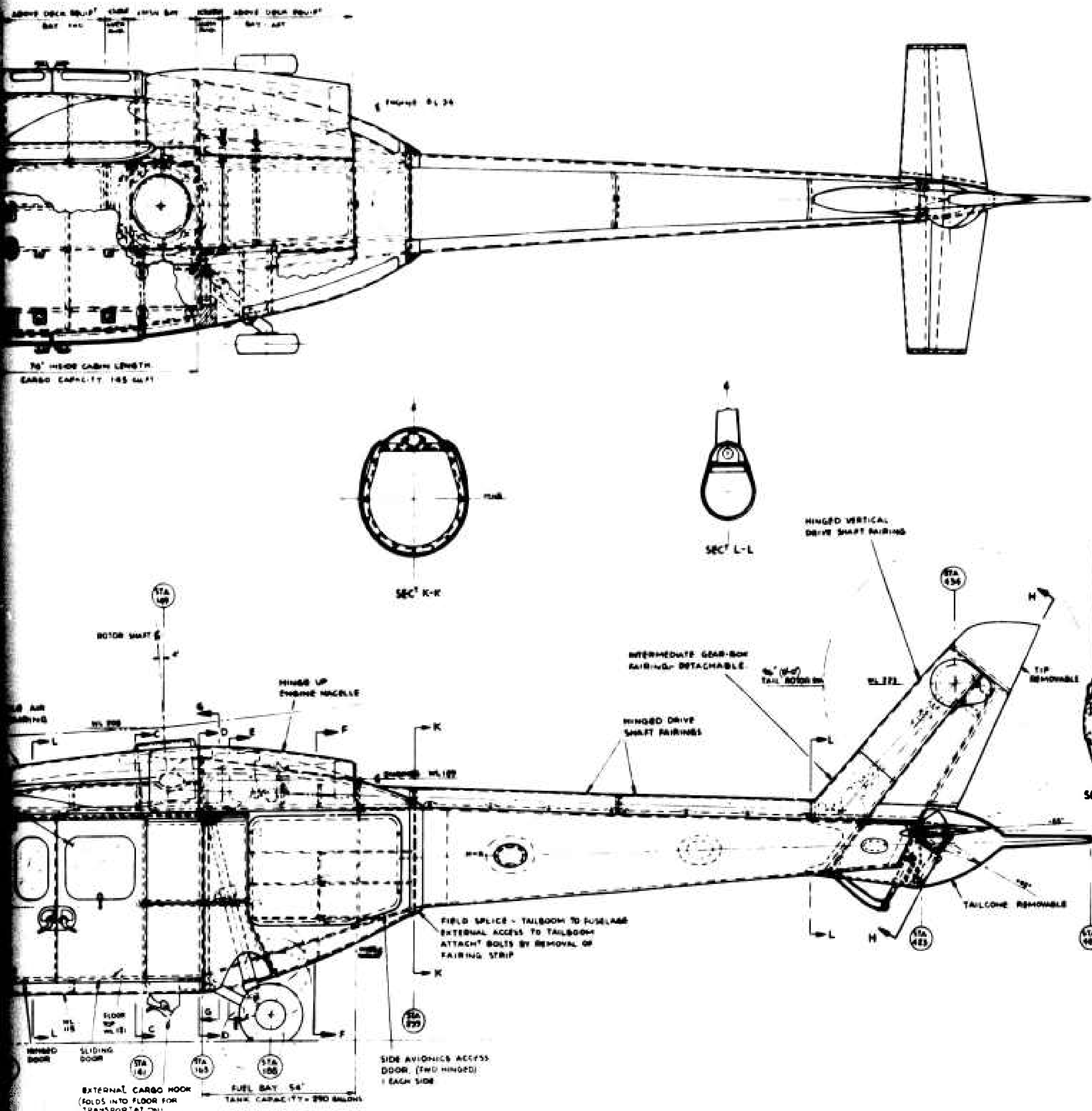


Figure B-20. MUT Composite Airframe Structure Final Configuration.



70" INSIDE CABIN LENGTH
CARGO CAPACITY 145 CU.FT
SEC^T K-K
SEC^T L-L
HINGED VERTICAL DRIVE SHAFT FAIRING
SEC^T H-H
ROTOR SHAFT
HINGE UP ENGINE NACELLE
INTERMEDIATE GEAR-BOX FAIRING- DETACHABLE
HINGED DRIVE SHAFT FAIRINGS
TIP REMOVABLE
TAILCONE REMOVABLE
FIELD SPLICE - TAILBOOM TO FUSELAGE
EXTERNAL ACCESS TO TAILBOOM
ATTACH^T BOLTS BY REMOVAL OF
FAIRING STRIP
HINGED DOOR
SLIDING DOOR
SIDE AVIONICS ACCESS
DOOR. (TWO HINGED)
1 EACH SIDE
EXTERNAL CARGO HOOK
FUEL BAY 54"

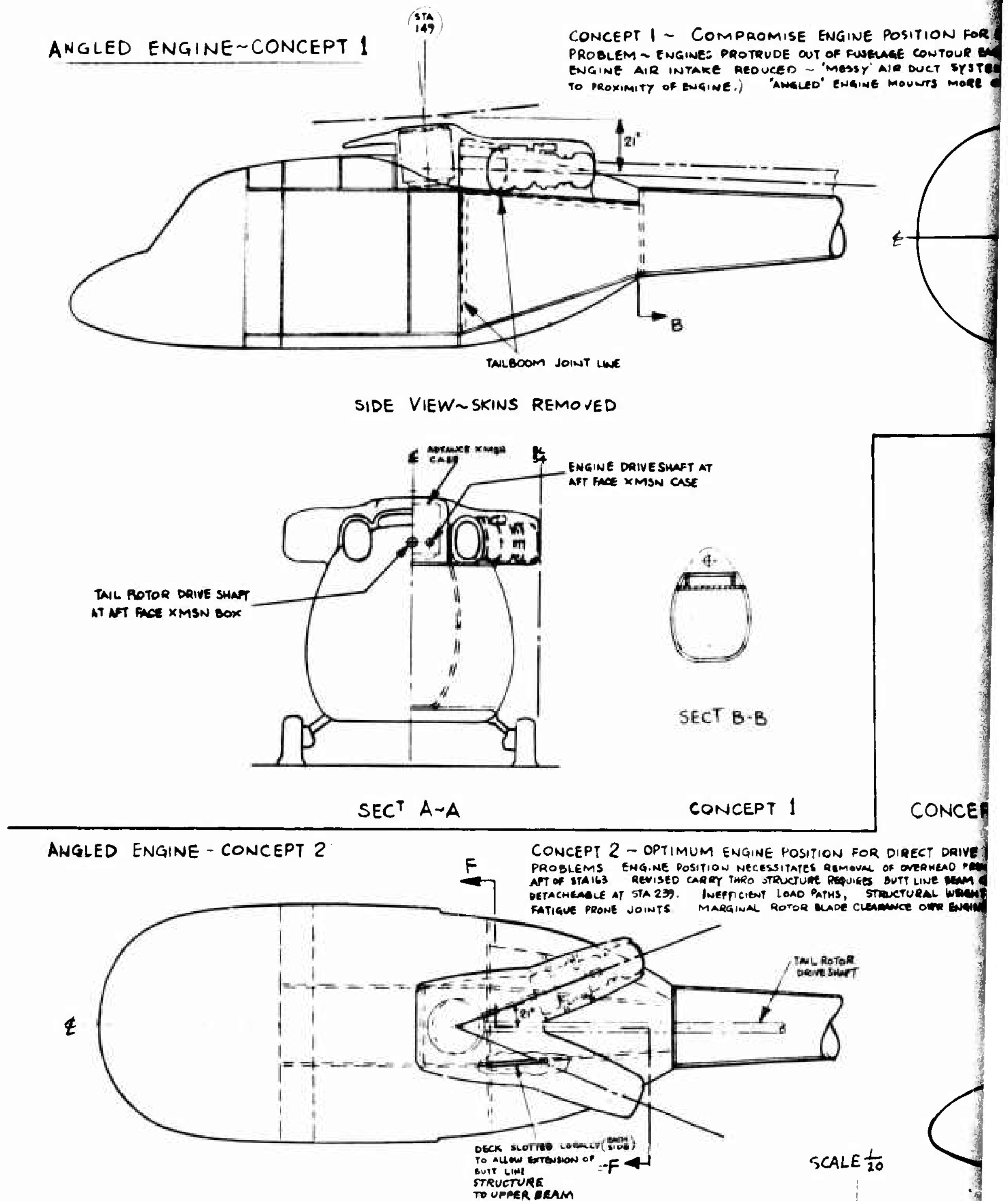
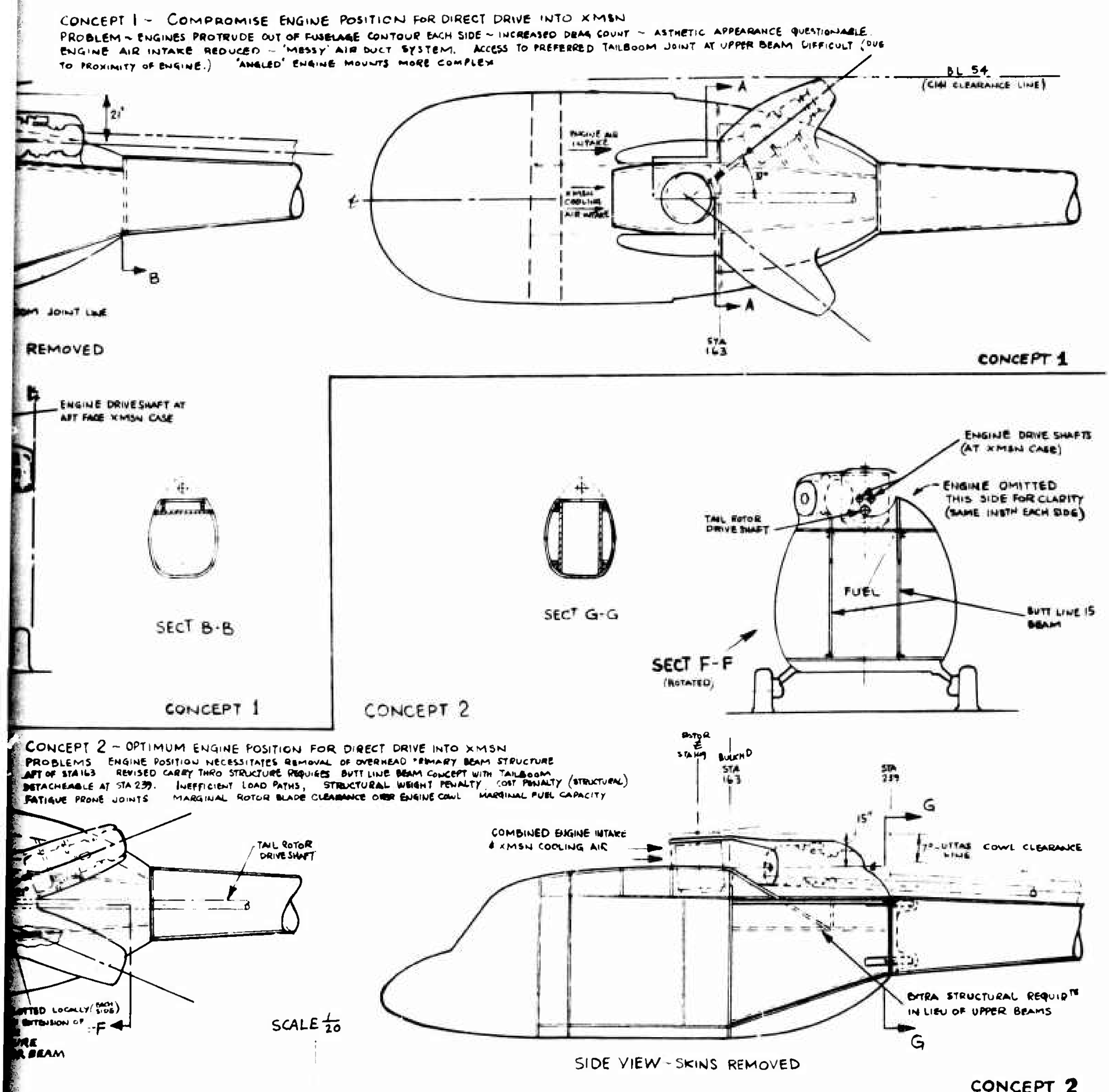


Figure B-21. MUT Direct Drive Engine Installation Concepts 1 and 2, and Required Tailboom/Fuselage Carrythrough Structure.



ncepts 1 and 2,
ough Structure.

LIST OF ABBREVIATIONS AND SYMBOLS

b	panel width, in.
CCA	cellular cellulose acetate
CR	cruise
c	core, core thickness
cr	critical
cu	compressive ultimate
D	flexural stiffness, psi
E	Young's modulus, MSI
e	fatigue endurance limit
F	allowable stress, PSI or KSI
f	face, applied stress, PSI
fps	feet per second
G	shear modulus, MSI
G-AS	graphite type AS
G-AS/E	graphite type AS in epoxy matrix
G/E	graphite epoxy
GW	gross weight, lb
HMS	high-modulus graphite
HTS	high-tensile graphite
h	height, in.
I	moment of inertia, $in.^4$
IRP	intermediate rated power

J	polar moment of inertia, in.^{4}
K	coefficient
KEAS	knots equivalent airspeed
KSI	thousands of pounds per square inch
KTAS	knots true airspeed
K-49	Kevlar 49
K-181	Kevlar 49 style 181
LC	learning curve
M	moment, in.-lb
M_{MO}	maximum operating Mach number
MS	margin of safety
MSI	millions of pounds per square inch
N	load factor, g; load intensity, lb/in.
NM	nautical miles
NRP	normal rated power
OWE	operating weight empty
P	load, lb
P/L	payload
PSF	pounds per square foot
PSI	pounds per square inch
Pwr	power
q	shear per unit length, lb/in.
R	stress ratio; mean radius of curvature, in.
rc	rigid core
S/L	sea level
Std	standard

su	shear ultimate
T	torsion, in.-lb
TOGW	takeoff gross weight, lb
t	thickness, in.
tu	tensile ultimate
V	load (shear), lb; velocity, kn
V_{MO}	maximum operating velocity
W	weight, lb
WE	weight empty, lb
WT	weight, lb
x	axis direction
y	axis direction
z	axis direction
α	angle of attack, deg
$\dot{\alpha}$	pitch rate, radians/sec
ν	Poisson's ratio
ρ	density, lb/cu in.
σ	stress, psi
τ	shear stress, psi

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