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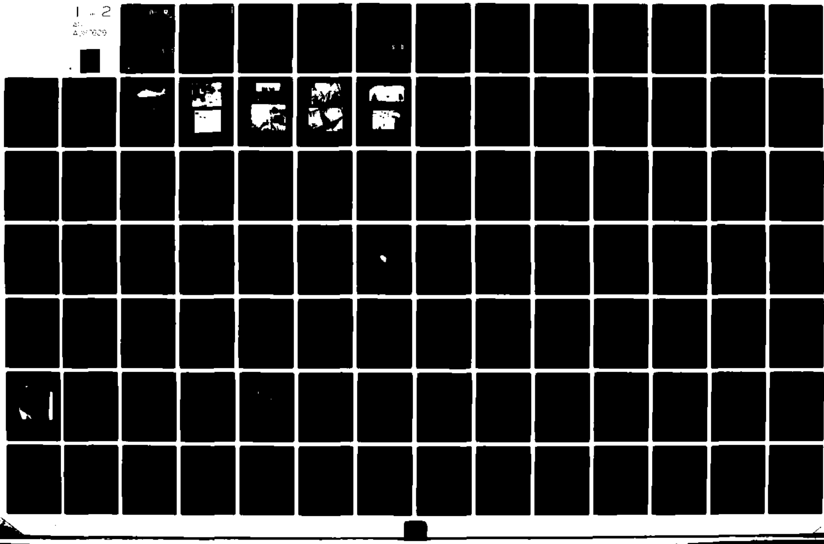

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ADVANCED STRUCTURES MAINTENANCE CONCEPTS

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June 1980

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Prepared for

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

APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

This report is the result of a contractual effort by the Applied Technology Laboratory with Sikorsky Aircraft to provide reliability and maintainability (R&M) analysis and design support during the design, development, and fabrication of a full-scale, predominantly advanced composite structure rear fuselage transition section for the UH-60A Black Hawk helicopter. The UH-60A composite rear fuselage section was being developed under a manufacturing methods and technology (MM&T) sponsored program being conducted under Applied Technology Laboratory technical direction.

The objectives of this contractual effort were to (1) perform an R&M analysis of both the metal baseline and proposed advanced composite design and identify potential R&M problem areas; (2) assess R&M design alternatives and define repair concepts; and (3) recommend a design configuration incorporating a cost-effective mix of R&M design features.

This report has been reviewed and the R&M design options and recommendations are considered to be reasonable and acceptable approaches to providing a high degree of R&M for the proposed advanced composite rear fuselage for the UH-60A Black Hawk helicopter.

The technical monitor for this contractual effort was Mr. Thomas E. Condon of the Aeronautical Systems Division, Reliability, Maintainability and Mission Technology Technical Area.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A detailed R&M analysis was conducted for an advanced composite rear fuselage for the UH-60A Black Hawk helicopter. Methods of improving the damage tolerance of the structure in areas of high impact exposure were evaluated. Cost-effective R&M options were recommended for incorporation in the design. Materials and techniques for performing rapid, low-skill-level repair of composite airframe structures in the Army field environment were investigated. Epoxide-impregnated preform patches were evaluated for over

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repair of primary structural components. Polyurethane foam kits were studied as a potential method of creating forms and mandrels for laminating repairs on the aircraft.

A modular design concept was developed for the UH-60A composite rear fuselage. Module replacement methods were defined. Studies were conducted to assess the economics of modular maintenance versus conventional types of repair. Potential improvements in combat repairability were evaluated.

PREFACE

This program was conducted by the Sikorsky Aircraft Division of United Technologies Corporation under Contract DAAK51-79-C-0027 for the Applied Technology Laboratory (ATL), U.S.Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia. The program was conducted under the technical direction of Mr. Thomas E. Condon of the Reliability, Maintainability and Mission Technology Technical Area of ATL.

The authors wish to acknowledge contributions to this program made by the following Sikorsky Aircraft personnel. Mr. Richard Corbeille of R&M Engineering assisted with the economic modeling of maintenance policies. Mr. Bruce F. Kay, Supervisor of Advanced Composite Airframe Design, provided technical guidance throughout the program.

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INTRODUCTION

Under a previous contract[1] with the Applied Technology Laboratory (ATL), Sikorsky Aircraft investigated the R&M and life-cycle cost potential of advanced composite airframe structures for Army helicopters. Critical issues affecting the R&M and life-cycle cost of advanced composite structures were established and a technique for assessing the R&M and operating cost characteristics of advanced structures concepts was developed.

It was a recommendation of that program that the Army conduct further work in the area of advanced composites R&M and that the continuation of work be associated with the development of a major composite airframe structure for helicopters. In response to that recommendation, in August 1979 Sikorsky Aircraft received a contract from ATL to develop maintenance concepts for an advanced composite rear fuselage (CRF) for the UH-60A Black Hawk helicopter. The CRF concept had been developed by Sikorsky under IR&D funding, and at the time of the contract award, was being proposed for full-scale development under MM&T (Manufacturing Methods and Technology) sponsorship and ATL technical direction. In November 1979, Sikorsky was awarded a contract* for the composite rear fuselage MM&T program.

The objectives of the Advanced Structures Maintenance Concepts program were to define and recommend cost-effective R&M design features for the CRF and to develop field repair concepts and serviceability criteria in support of the recommended design. The time-phasing of the R&M and MM&T programs (Figure 1) provided an excellent opportunity to influence the CRF design definition on a "real-time" basis. Tradeoffs involving R&M design options were being completed just as the baseline definition for the MM&T program was getting underway, insuring that R&M was considered in the design from the outset. Repair concept studies were being completed as the CRF detail design refinement was progressing, providing the opportunity to incorporate repairability design features at an early stage of design.

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1 Cook, T. N., and Kay, B. F., ADVANCED STRUCTURES CONCEPTS R&M/COST ASSESSMENTS, Sikorsky Aircraft Division, USARTL-TR-79-18, Applied Technology Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, VA, September 1979, AD 077373.

* Contract DAAK51-80-C-0001

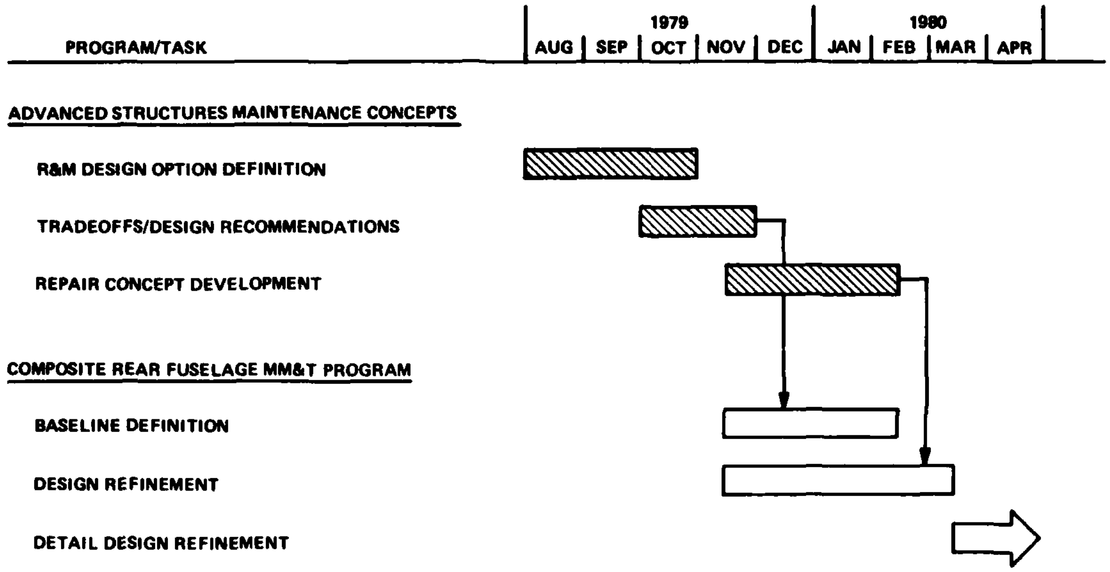

Figure 1. Program Interfaces

REAR FUSELAGE DESCRIPTIONS

CURRENT METAL DESIGN

A relatively large and complex section of the airframe, the rear fuselage of the Black Hawk helicopter is a semi-monocoque design containing a mix of heavily loaded primary structure and lightly loaded secondary structure. The principal loads are applied by fuselage shear and bending and by fuel inertia. The existing metal design contains 1,203 detail parts, 88 subassemblies and 17,000 fasteners.

A frame, skin and multi-stringer design constitutes the basic arrangement, with bulkheads, beams, and frames used to transmit the high pressure loads from the fuel cells. A honeycomb panel separates the two cells, and removable top covers provide access to the fuel cells for repair and replacement. The bulkhead in front of the fuel cells also serves as the aft bulkhead for the cabin and supports the high intensity loads from the cabin roof and floor beams. The aft frame of the rear fuselage contains the bolted manufacturing joint between the tailcone and the rear fuselage and incorporates an air retrieval/ tiedown fitting. A portion of the skin under the engine exhaust is made from titanium to provide fire protection for the structure beneath.

Compartments within the structure contain the aircraft fuel cells and fuel system plumbing and provide storage for equipment and baggage. Numerous cutouts and panels accommodate fuel filler ports, fuel drains and other external connections and provide access to interior equipment.

Interfaces with other systems include provisions for mounting the auxiliary powerplant, hydraulic accumulators, tail rotor drive shaft and other items of equipment located on the upper deck. Tail rotor controls, hydraulic lines and electrical wiring pass through and are supported within the structure. Steps and walkways are provided for maintenance personnel. Major joints and fittings form the interface between the main fuselage and tail section. From the standpoint of R&M, the rear fuselage has probably the most representative mix of design attributes of any structure in the airframe.

Figures 2 through 11 show principal details of the Black Hawk metal rear fuselage.

COMPOSITE DESIGN CONCEPT

As brought out in the Introduction, the R&M program led the CRF MM&T program by approximately three months. The R&M analysis was therefore based largely on the design concept which has been developed by Sikorsky under IR&D funding and which served as the basis for the MM&T proposal. While the concept defined the principal form and structural arrangement of the CRF, it did not define most details of the design. In order to provide a basis for evaluating R&M design options, it was necessary to fully define a baseline design. Therefore, as each area of R&M consider-

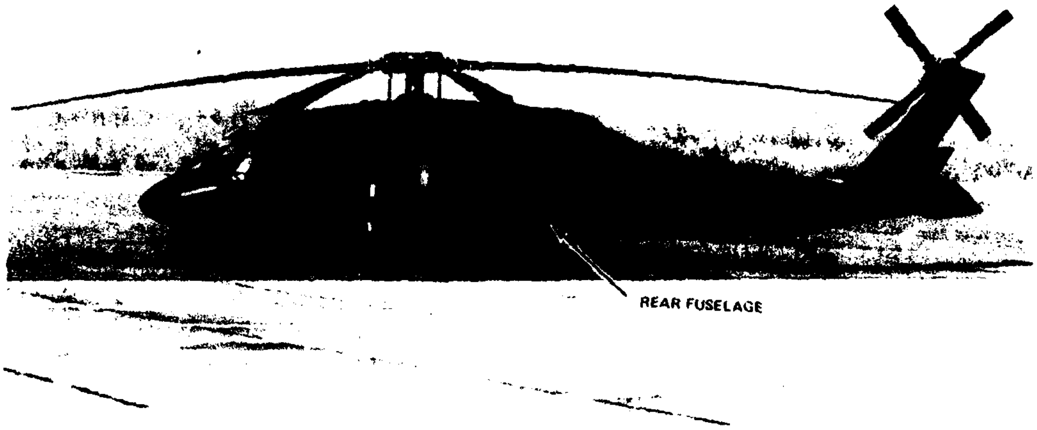

Figure 2. Black Hawk Rear Fuselage

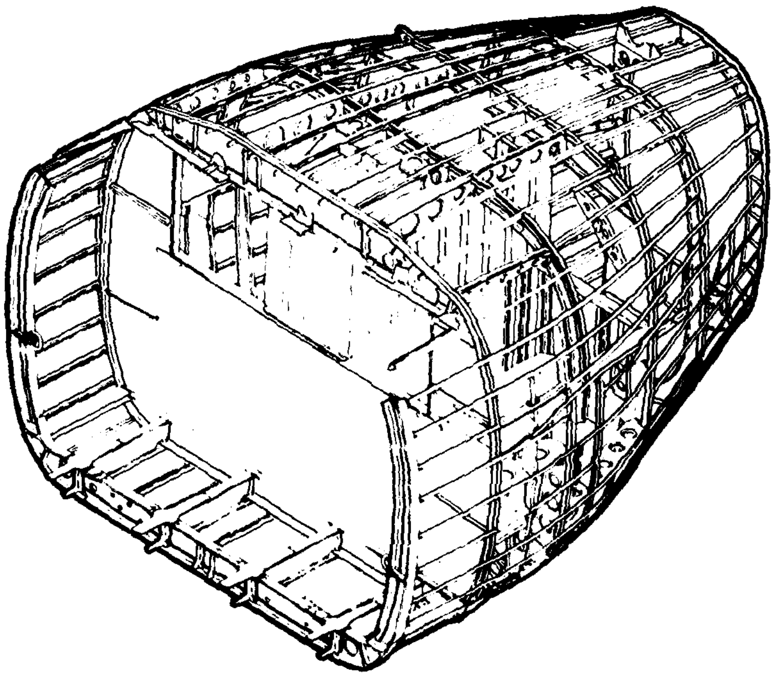

Figure 3. Metal Rear Fuselage Configuration

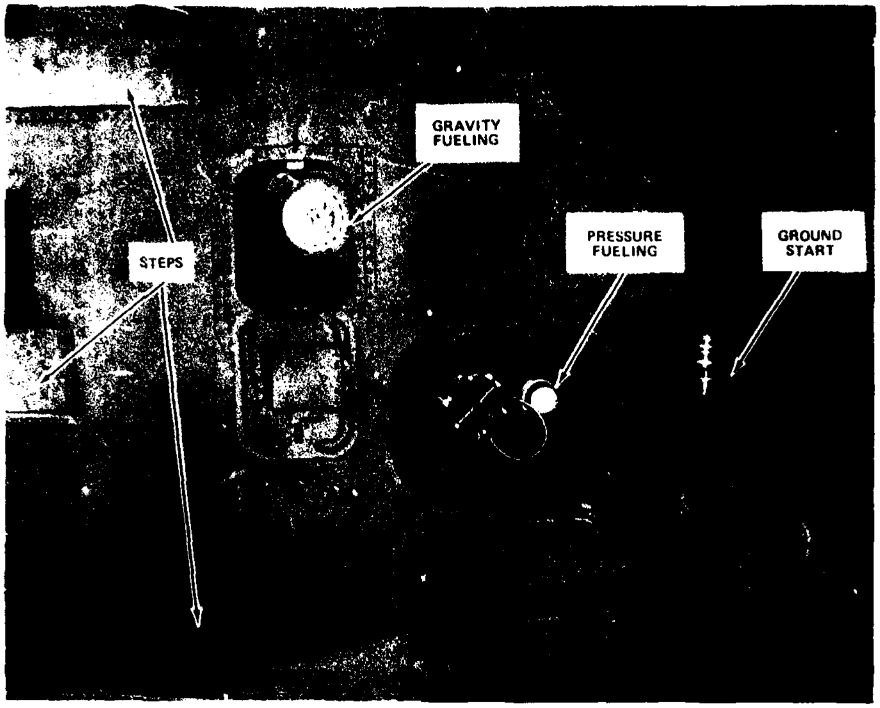

Figure 4. Access Doors and Steps, Left Side

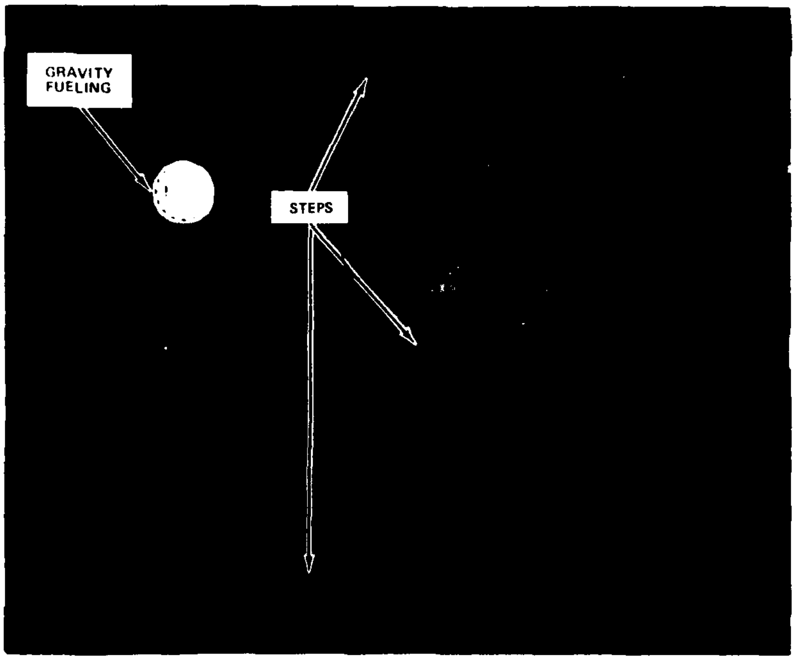

Figure 5. Access Door and Steps, Right Side

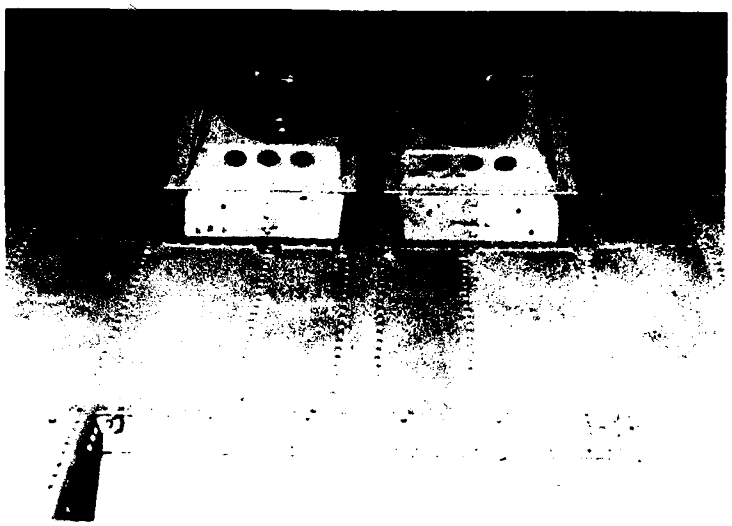

Figure 6. Fuel Sump Drain Doors, Underside

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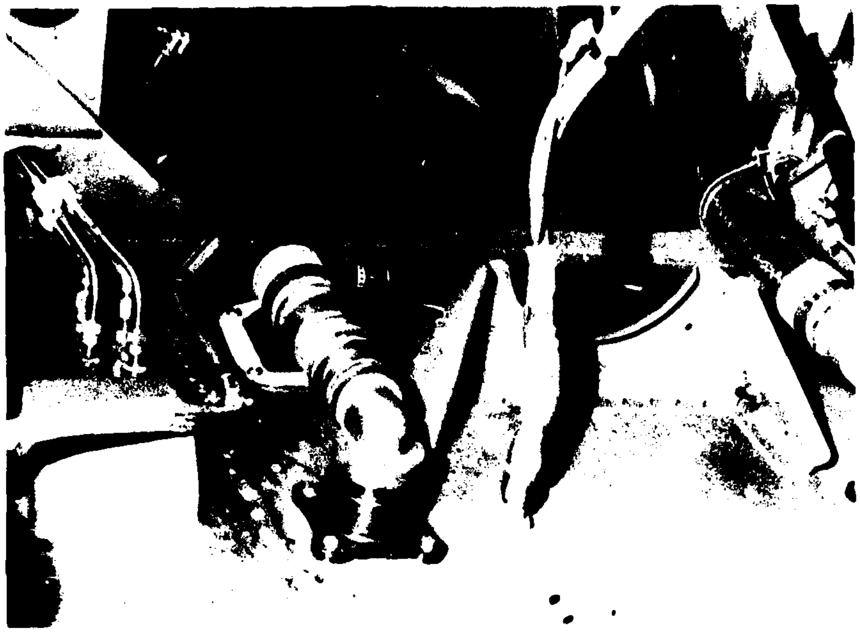


Figure 8. Upper Deck Interfaces

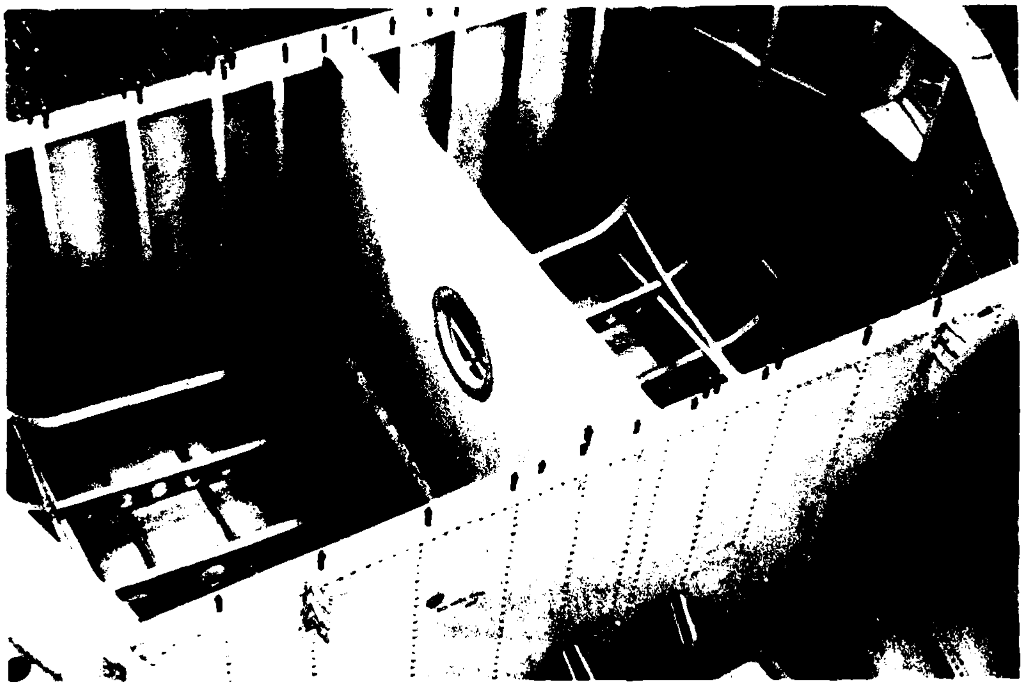


Figure 9. Fuel Cell Structure

Figure 10. Rear Fuselage Interior Structure

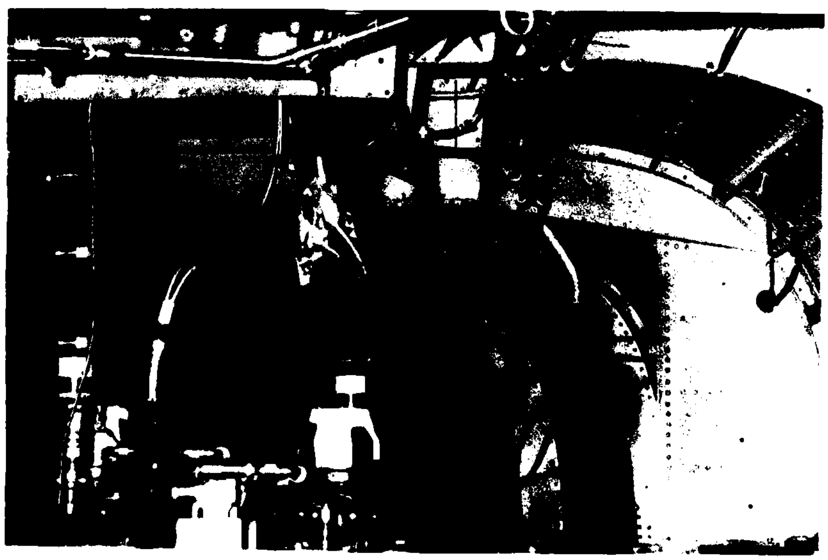


Figure 11. Interior Plumbing and Controls

ation arose, the lead designer was requested to estimate how that area of the design would be approached if the CRF was being designed for structural loading conditions exclusively, i.e., without any consideration to R&M. This avoided having him try to speculate on the R&M concerns in that area and propose a design that he thought might satisfy those concerns. It also provided a uniform basis for evaluating alternatives.

The CRF design described in this report is the baseline concept. At the time the R&M program was being completed, the MM&T program was still in the basic data phase and tradeoffs were still ongoing. However, the tradeoffs were heavily favoring a design very similar to the original concept and it was expected that the final configuration would not differ significantly from the one described in this report.

Baseline Design

The CRF (Figure 12) is predominantly a Kevlar skin-skeleton configuration comprised of an upper half and a lower half joined by a manufacturing

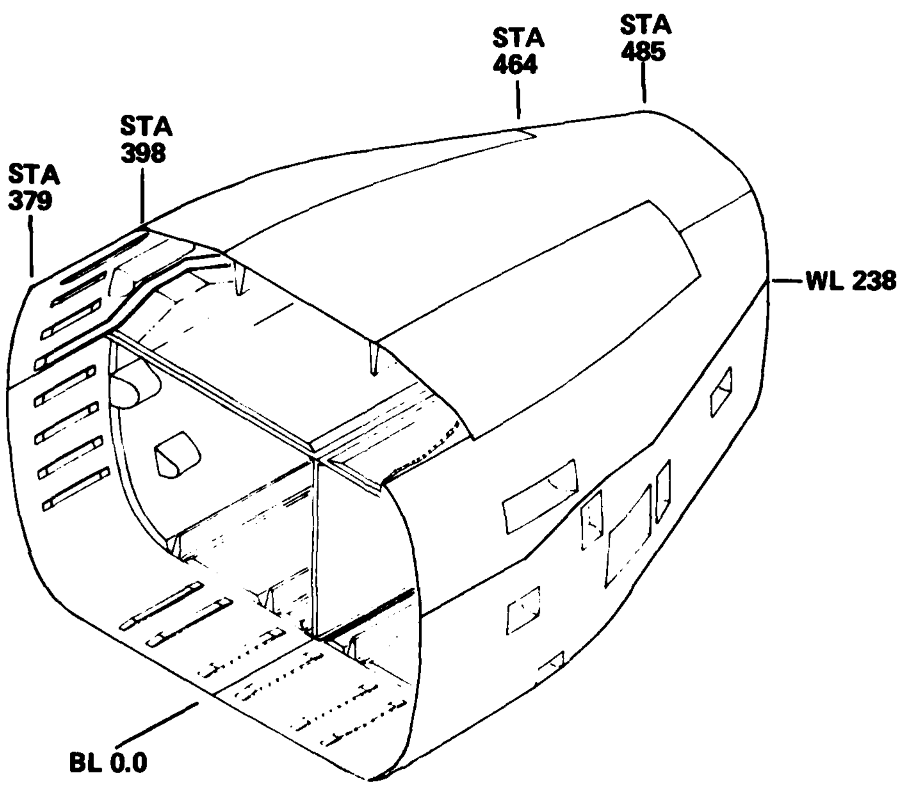


Figure 12. Composite Rear Fuselage

splice at W.L. 238. The structure consists of frames, beams, longerons, bulkheads and stiffeners that support and transmit the loads applied to the structure and provide attachments to the forward fuselage and tailcone. The arrangement and form of most major elements are controlled by interface requirements with existing structure and subsystems.

The primary aluminum structure between Stations 379 and 398 is left unchanged to meet crash load conditions. Kevlar hat sections with unidirectional graphite caps make up the primary longitudinal framing members (beams and longerons). The Kevlar outer skin is stiffened with cocured hollow hat section members. In the lower half of the CRF, the bulkhead aft of the fuel cell at Station 443.5 is a corrugated one-piece Kevlar design. Its function is to support fuel pressure loads and react the B.L. 10 and B.L. 30 beam loads. In the upper half of the CRF at this station, the frame is a one-piece precured graphite design with a constant channel section around the periphery and a "floating" upper angle attached to the skin.

In the lower half of the CRF at Station 464, the frame is designed primarily to react tie-down fitting loads and fuselage hoop stresses. It is fabricated as a press-molded, precured graphite, "C" channel half-ring. The Station 464 frame is a hat section ring stiffener design of Kevlar and graphite composition. The stiffeners stabilize the B.L. 16.50 beams and longerons and react circumferential loads. The upper and lower frame halves at Station 485 are integral with the fuselage skin and are laid up using graphite preforms to provide hard points for tailcone attachment.

The design of the upper deck is controlled by interfaces with aircraft systems and components located there and by the requirement for a fireproof skin beneath the engine exhaust and IR suppressor. These requirements dictated the retention of the titanium skins in the shoulder areas of the roof and the design of a honeycomb center panel with numerous cutouts and core densification zones. The titanium requirement creates natural manufacturing splices with the three composite subsections of the upper half of the CRF.

The CRF contains numerous cutouts, covers and doors for such items as the fuselage steps, refueling filler ports and fuel drain valves. Appendix A describes details of the CRF design in these areas.

R&M DESIGN OPTION TRADEOFF ANALYSIS

SCOPE OF THE ANALYSIS

The R&M design option analysis addressed those areas of the rear fuselage subject to design change under the MM&T contract. Areas of the rear fuselage designated to remain metal structure, such as the forward fuel cell bulkhead, were not evaluated, nor were items of the existing structure that were already of composite construction such as the fuel plumbing enclosure panels. However, estimates were made of the degree of R&M improvement that might be achieved through the introduction of improved designs for selected items of structure outside the scope of the MM&T contract.

DESIGN CONCEPT R&M ASSESSMENT

One product of the program reported in Reference 1 is a technique for assessing the R&M and life-cycle cost potential of advanced structures concepts. At the conclusion of that program, the technique was tested by applying it to a sample of representative composite structures designs. A total of 12 structural concepts were evaluated, one of which was the composite rear fuselage concept for the Black Hawk helicopter developed by Sikorsky under IR&D.

The first task of the current program entailed a reassessment of the rear fuselage design concept and the existing metal structure using the R&M assessment technique. The composite design was assessed as it was then defined, prior to any special R&M considerations, to provide a baseline against which the final configuration was compared. The results of this assessment indicated that the conclusions documented in Reference 1 relative to possible R&M concerns with a composite rear fuselage were valid.

MAINTENANCE COST ESTIMATES

One of the major conclusions of the study reported in Reference 1 is that repair of helicopter airframe structures is associated almost entirely with damage caused by external impact. Failures of an inherent nature are rare and generally minor (fatigue-generated skin cracks, popped rivets, etc.). As was brought out by that study, assessment of induced damage is a complex undertaking, since it involves a totally random process and a large number of variables.

In order to assess the value of improved damage tolerance, some measure of the potential for damage is necessary. Obviously, if the exposure to hazards is minimal and the magnitude of the expected impacts is small, improving damage tolerance has little value. Conversely, frequent exposure to severe impacts makes damage tolerance highly desirable. The ease with which repairs can be made will also affect these decisions.

At the outset of the program it was recognized that little information exists with which to estimate how often, how severely, and with what effects a structure will be subjected to external impact in service. It was concluded that such estimates must be based very substantially on engineering judgement. The procedure used to develop these estimates for the CRF is shown in Figure 13.

Estimates of Hazard Exposure and Impact Frequency

The exposure of rear fuselage structure to the various impact hazards was estimated on the basis of the frequency at which ground operations, servicing and maintenance are performed in and around that area of the aircraft. Eleven impact hazards were evaluated:

1. Stowed baggage impact on vertical surface
2. Stowed baggage impact on horizontal surface
3. Dropped tool impact
4. Dropped part impact
5. Fueling nozzle impact
6. Impact by pneumatic ground start coupling
7. Boot impact
8. Foot traffic impact
9. Edge and corner impact (dropped panel)
10. Edge and corner impact (struck panel)
11. Impact with terrain objects

Estimates were developed for the rate per 10,000 flight-hours at which specific areas of the structure are exposed to specific impact hazards. In the case of exposure to fueling nozzle impacts, for example, the fuel servicing interval for the aircraft formed the basis for the estimate. Foot traffic exposure was estimated on the basis of inspection and maintenance intervals requiring personnel to walk on specific areas of the structure. The rationale for each estimate is given later in this section of the report when the hazard exposure estimates are presented.

Some fraction of the exposures to impact will actually result in an impact. The number usually depends on several factors and is highly variable. For example, when maintenance is being performed in inclement weather (rain, wind, cold, etc.) or in darkness, the frequency of dropped tools and parts can be expected to increase substantially over that

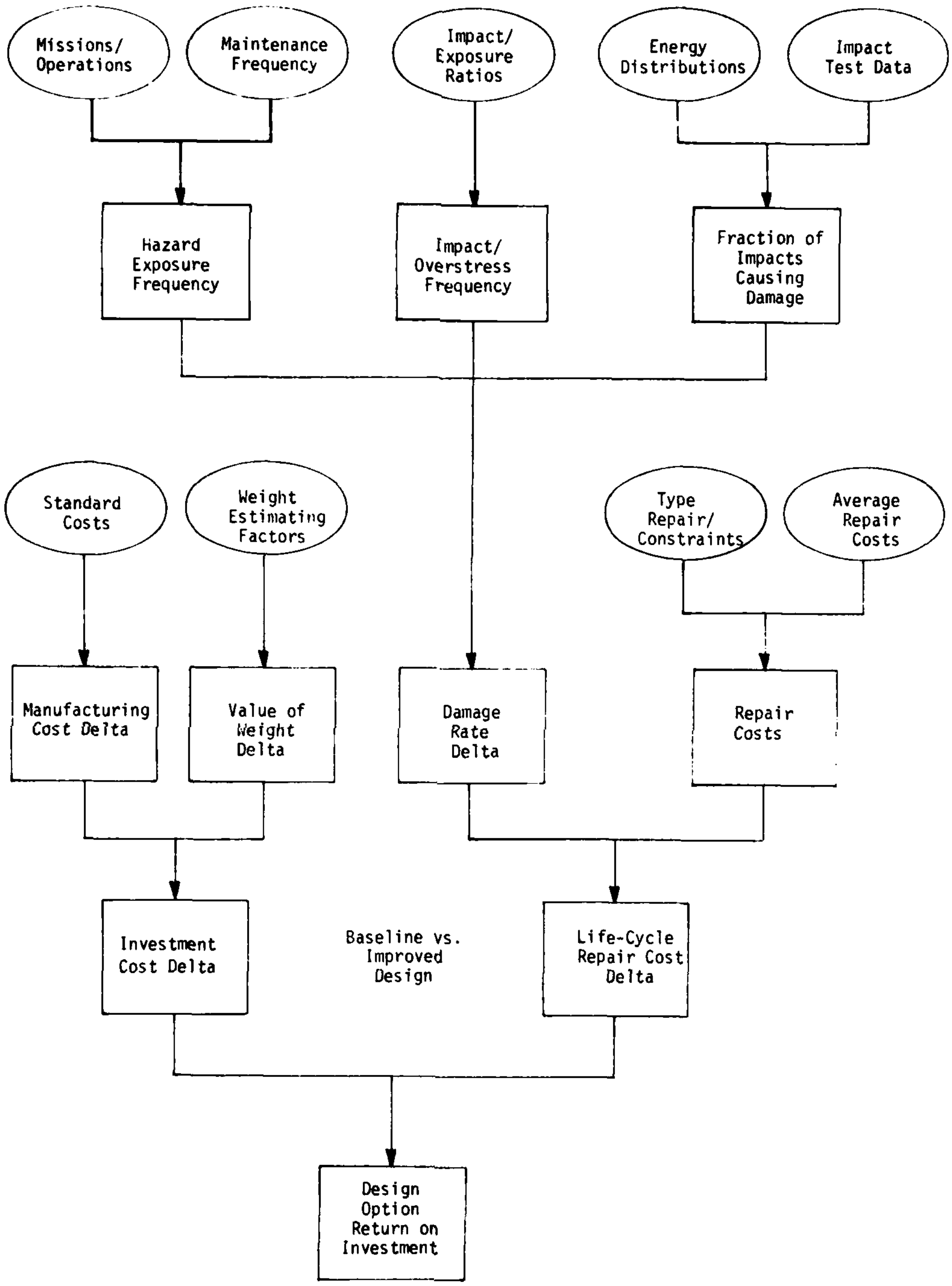


Figure 13. R&M Design Option Tradeoff Procedure

experienced under normal conditions. Predicting the frequency of such mishaps is therefore difficult. The approach taken was to assess the potential for impact on a graded scale as follows:

Damage Potential	Impact/ Exposure Ratio
Low	1/100
Moderate	1/50
Moderately High	1/25
High	1/10

Engineering judgement was used to assess the potential for impact when an area or item of rear fuselage structure is exposed to a given hazard. The corresponding impact/exposure ratio was then applied to the hazard exposure rates to obtain estimated impact rates per 10,000 flight-hours.

Estimates of Impact Intensity

Estimates of impact intensity were developed for the 11 impact hazards as follows: The type of hazard was analyzed and a condition was picked to represent the 99th percentile impact, based on such factors as the anticipated mass and velocity of the impacting objects. This is a degree of impact which engineering judgement suggests would be exceeded in 1% or fewer of the cases. With respect to impact caused by stowed baggage, for example, conditions involving dropping shoving and shifting of baggage within the baggage compartment were studied. The 99th percentile impact against a vertical surface was picked to represent a forceful shove of a heavy tool box into the compartment. It is expected that a very small percentage of the impacts will be more severe than this. In every case, the 99th percentile impact is considered to be a conservatively high estimate.

In the next step of the procedure, the distribution of impact energies associated with each hazard was approximated. It was assumed that a typical hazard (dropped tools, foot traffic, etc.) will involve many light to moderate impacts and relatively few severe impacts. In the case of dropped tools, for example, it is expected that small, frequently used hand tools such as wrenches and pliers will be dropped much more often than heavier, less frequently used tools such as drill motors and rivet guns. None of the impact energies will be less than zero of course, which yields a distribution of impact energy that is typically of the lognormal type (Figure 14). The assumption of a lognormal distribution was applied to all 11 impact hazards.

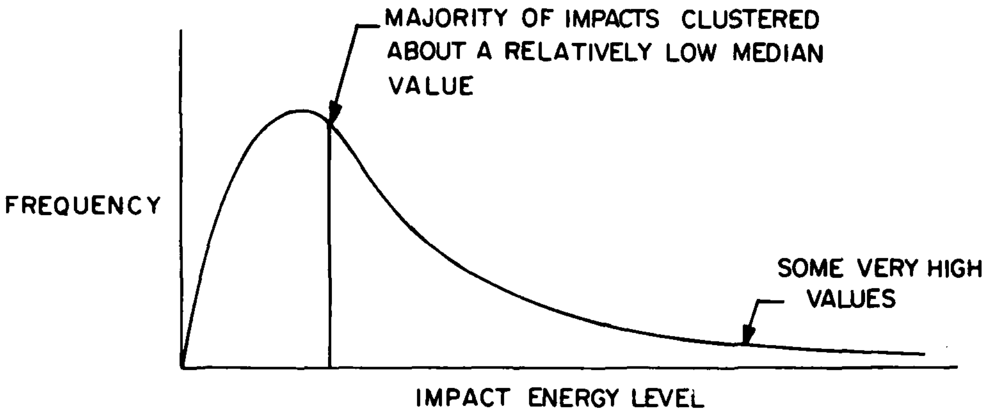


Figure 14. Assumed Shape of Impact Energy Distributions

For each hazard, the impact energy at the 1st percentile was assumed to be 10% of the impact energy at the 99th percentile. The 1st and 99th percentile impact energy points were located on lognormal probability paper and a straight line was drawn between the two points to describe the distribution of impact energies. Figure 15 shows the plotted distributions for the 11 impact hazards. The basis for estimating the 11 impact energy distributions is described in the following paragraphs.

Impact by Stowed Baggage

The baggage compartments located over the fuel cells in the interior of the structure are used to stow such items as troop seats, tie-down devices and personal gear (helmets, clothing, etc.). Among the heaviest items carried in these compartments is the crew chief's emergency tool kit, a metal box 18" x 13" x 10-1/2" in size, weighing approximately 45 pounds.

Items of structure vulnerable to damage by items thrown into or shifting about in the baggage compartments are the fuel cell covers, which form the compartment floors, the aft bulkheads, and the rear fuselage interior skin and framing. Particularly vulnerable is the enclosure for the largest of the three fuselage steps, one of which protrudes into each baggage compartment.

The 45-pound emergency tool kit is considered to have the greatest potential for causing damage to the baggage compartment. A strong mechanic could conceivably throw the tool box into the compartment, although it is more likely that the box would be dropped on the edge of the forward

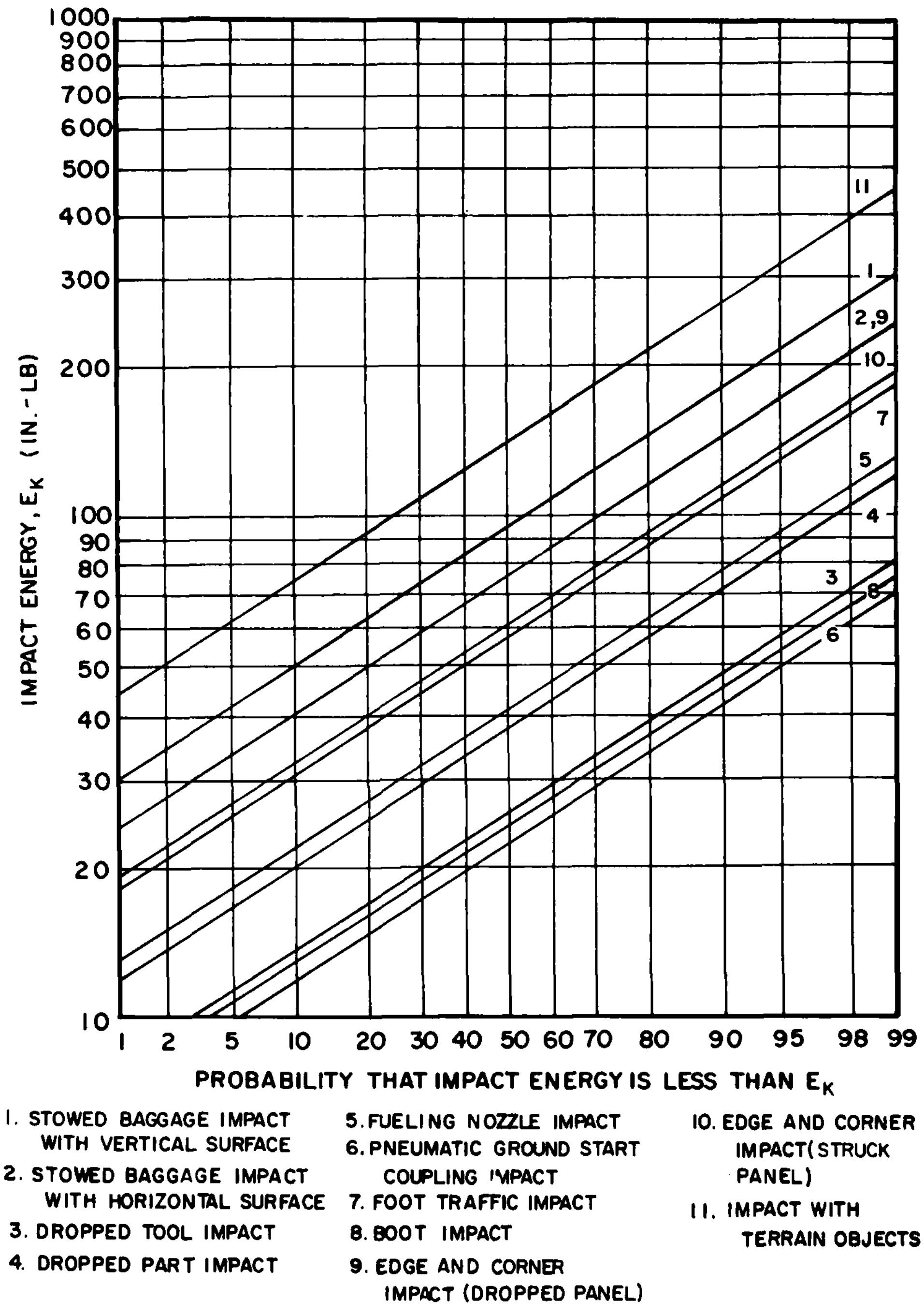


Figure 15. Estimated Impact Energy Distributions for 11 Impact Hazards

bulkhead or on the fuel cell cover and then pushed into the compartment. A forceful push could cause the box to slide across the floor of the compartment and into the rear bulkhead or the raised section of the fuel cell cover. Contact with the fuselage step enclosure and the rear fuselage interior skin and framing is also possible.

The average height at which the tool box will be dropped on the fuel cell cover is estimated at 2 to 3 inches and the maximum height at 6 inches. This would produce impact energies of from approximately 11 to 23 ft-lb. The 99th percentile impact on a horizontal surface is approximated by a drop of 5 to 6 inches (20 ft-lb). The 99th percentile maximum impact caused by the box sliding into vertical structure as a result of large aircraft rolling or pitching moments or a forceful shove of the box into the baggage compartment is estimated at 25 ft-lb (45 lb box moving at 6 ft/sec). Because of the sharp edges and corners of the box, the impact may be distributed over a relatively small surface area.

Impact by Dropped Tools

The upper deck is exposed to damage caused by dropped tools. Three items located on the upper deck will require periodic maintenance: The APU (Auxiliary Power Unit), the fire protection bottles, and the drive shaft and coupling. The fuel cell covers in the interior of the structure are also subject to impact by dropped tools, primarily as a result of work performed on the APU accumulator mounted in the roof structure over the fuel cell.

Maintenance of the APU and other components located on the upper deck is accomplished with common hand tools and equipment. A review of the commonly used tools shows that the majority weigh less than one pound. Only a few items such as rivet guns and drill motors weigh more than three pounds. The mechanic will normally be in a stooped or kneeling position when working in this area. Tools that are dropped either on the upper deck or the fuel cell covers will generally be within two feet of the surface. The 99th percentile impact is approximately by a 2 pound tool dropped from waist height (80 inch-lb). The shape of the tool will affect the amount of damage inflicted on the structure, and a heavy screw driver dropped blade first is likely to be among the most damaging impacts.

Impact by Dropped Parts

The upper deck and fuel cell covers are also exposed to impact from dropped parts and components. APU accessories will be the most frequently handled parts. A review of the APU weights breakdown shows the weight of the majority of removable parts to be under three pounds. The starter is the heaviest removable accessory, weighing approximately 10 pounds.

The mechanic will normally be in a stooped or kneeling position when working on equipment above the deck. Parts that are dropped either on the upper deck or on the fuel cell covers will generally be within two feet of the surface. The 99th percentile impact is approximated by a 3 pound part dropped from waist height (120 inch-lb). Many parts have sharp edges and corners so that the imparted energy may be distributed over a relatively small surface area.

Impact by Aircraft Refueling Nozzles

The refueling access enclosures, the surrounding structure and the access doors are vulnerable to impact by the aircraft refueling nozzles. Closed-circuit refueling is the primary servicing method used in the Army today. The closed-circuit refueling nozzle is approximately 2-1/2" in diameter and weighs approximately 5 pounds. With the gravity fueling adapter and fuel strainer attached, the nozzle weighs approximately 10 pounds. The weight of fuel in the nozzle and the section of hose in the grip of the mechanic adds another estimated 5 pounds to the weight. The aircraft has a pressure fueling port which is serviced with a larger and heavier fuel servicing nozzle, but pressure refueling is rarely used in the Army.

Routine impacts of the closed-circuit refueling nozzle are expected to be relatively light. The 99th percentile impact is estimated at 130 inch-lb (15 lb nozzle moving at 7 ft/sec). Depending on the angle at which the nozzle strikes, the impact energy could be concentrated in a small surface area, however.

Impact by Pneumatic Ground Start Coupling

The engine ground start access enclosure, the surrounding structure and the access door are exposed to impact by the pneumatic ground start coupling. The weight of the coupling and the length of hose in the grip of the mechanic is estimated at 8 pounds. Routine impacts by the ground start coupling are expected to be relatively light. The 99th percentile impact is approximated at 70 inch-lb (8 lb coupling moving at 7 ft/sec).

Foot Traffic Impact

The upper deck in the vicinity of the APU, fire protection bottles and drive shaft is vulnerable to damage by foot traffic. The routine hazard consists of normal walking, primarily small steps within a confined area. In walking, feet and leg muscles attenuate the shock so that the energy imparted to the structure is relatively light. A 99th percentile impact is approximated by a mechanic stepping quickly from the pylon cover to the deck in such a way that all of his weight is accelerated through some vertical distance before his foot contacts the deck (the equivalent of a short hop to the deck). This is approximated by a 180 lb man and a vertical drop of 4 inches. Assuming that half of the energy is absorbed by the mechanic's foot and leg, this would impart an estimated kinetic energy of 360 inch-lb. to the deck. Since the impact tends to be distributed over a relatively large surface area, the estimate for the 99th percentile impact is reduced by one-half to 180 inch-lb.

Boot Impact Against the Fuselage

The fuselage step enclosures, the surrounding structure and the step doors are vulnerable to damage by boot impact. Impacts caused by boots missing the steps will occur primarily during descent from the aircraft when it is more difficult for the mechanic to see the steps. Based on published human factors data, the 99th percentile impact caused by a boot kicking the side of the fuselage is estimated at 75 inch-lb. The energy is relatively low, but it is concentrated in the small surface area of the toe of the boot.

Edge and Corner Impact

Edge and corner impact will result either from dropping a removable structure or striking the exposed edge or corner of a structure. There are three items of removable structure that might be dropped during maintenance: the two fuel cell covers and the magnetic flux valve access panel. The fuel cell covers weigh approximately 12 pounds each. The greatest distance these covers might be dropped is from the open door of the aircraft to the ground outside (approximately 4 feet). The 99th percentile impact is estimated at 20 ft-lb. When installed, the fuel cell covers are relatively protected from edge and corner damage. The magnetic flux valve access panel is very light, weighing less than 1/2 pound. Edge and corner damage due to dropping the panel should not be significant.

Edge and corner damage may also occur as a result of striking an open hinged access door with a tool or ground equipment. There are six doors of this type in the rear fuselage as shown in Appendix A. The fuel sump drain doors located in the underside of the rear fuselage are relatively protected from impact; edge and corner damage to these doors should be negligible. Three of the four remaining doors (pressure refueling and two gravity refueling) are located on the sides of the aircraft and are hinged at the bottom. When opened they lie flat against the fuselage where they are relatively protected from impact by tools and equipment. The engine ground start access door is hinged at an angle so that when opened it protrudes away from the side of aircraft where it is vulnerable to impact.

Edges and corners of the hinged access doors are exposed to contact with servicing equipment such as the aircraft refueling nozzle and to ground equipment such as work stands. The 99th percentile impact is approximated by a 250 pound work platform contacting the edge of the door at a speed of 2 ft/sec (190 inch-lb).

Impact with Terrain Objects

The underside skin and framing is vulnerable to impact with terrain objects such as brush, stumps and rocks. The frequency of these occurrences is dependent on the terrain and the need to land the aircraft in unfamiliar locations. Combat provides the highest risk situation.

Impact with terrain objects will range from light contact with ground foliage, as will frequently happen when the aircraft settles into a field, to the severe types of impact that landing on solid objects at high aircraft sink rates would produce. The maximum touchdown sink rate for the aircraft on level ground is 10 ft/sec. A typical touchdown is estimated at 3 ft/sec.

Most of the contact with the terrain will involve foliage light enough to deflect away or break under the weight of the aircraft. The 99th percentile impact is approximated by a branch or limb that produces a force of 100 pounds against the structure before yielding to the weight of the aircraft. At a 5 ft/sec sink rate at touchdown, this represents an impact energy of approximately 450 inch-lb.

Estimates of the Fraction of Impacts Causing Damage

The impact testing reported in Reference 1 was intended to define minimum gage requirements for composites exposed to impact of light to moderate intensity. Thicknesses of from 2 to 8 plies were tested at energy levels in the range of 20 to 50 inch-pounds. At the 50 inch-pound maximum energy level, none of the 8-ply composites sustained measurable damage. Data obtained from the tests was thus confined to a relatively narrow range of values.

Reference 1 contains a set of curves developed from the impact testing which relate depth of damage to impact energy level for a population of composite materials, thicknesses and configurations. The 11 impact hazards being addressed in the design of the CRF have estimated energy distributions greatly in excess of the energy levels at which the impact testing was conducted. Attempts to locate published data covering impact testing of the same materials under similar conditions but at higher energy levels were not successful. It was necessary, therefore, to extrapolate the available data to higher energy levels and to interpolate the data for material thicknesses other than the ones actually tested. The resulting curves are shown in Figures 16 through 19.

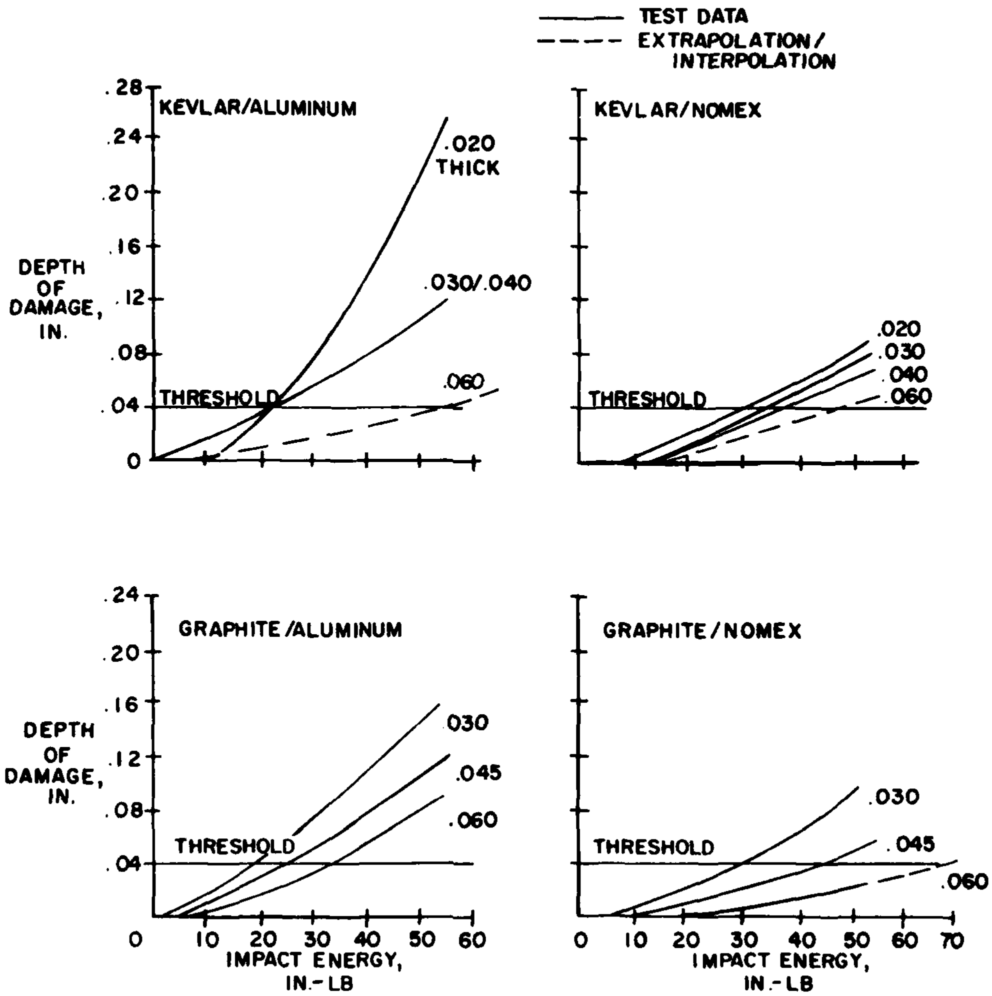


Figure 16. Extrapolated/Interpolated Impact Test Data for Kevlar and Graphite-Faced Sandwich Panels

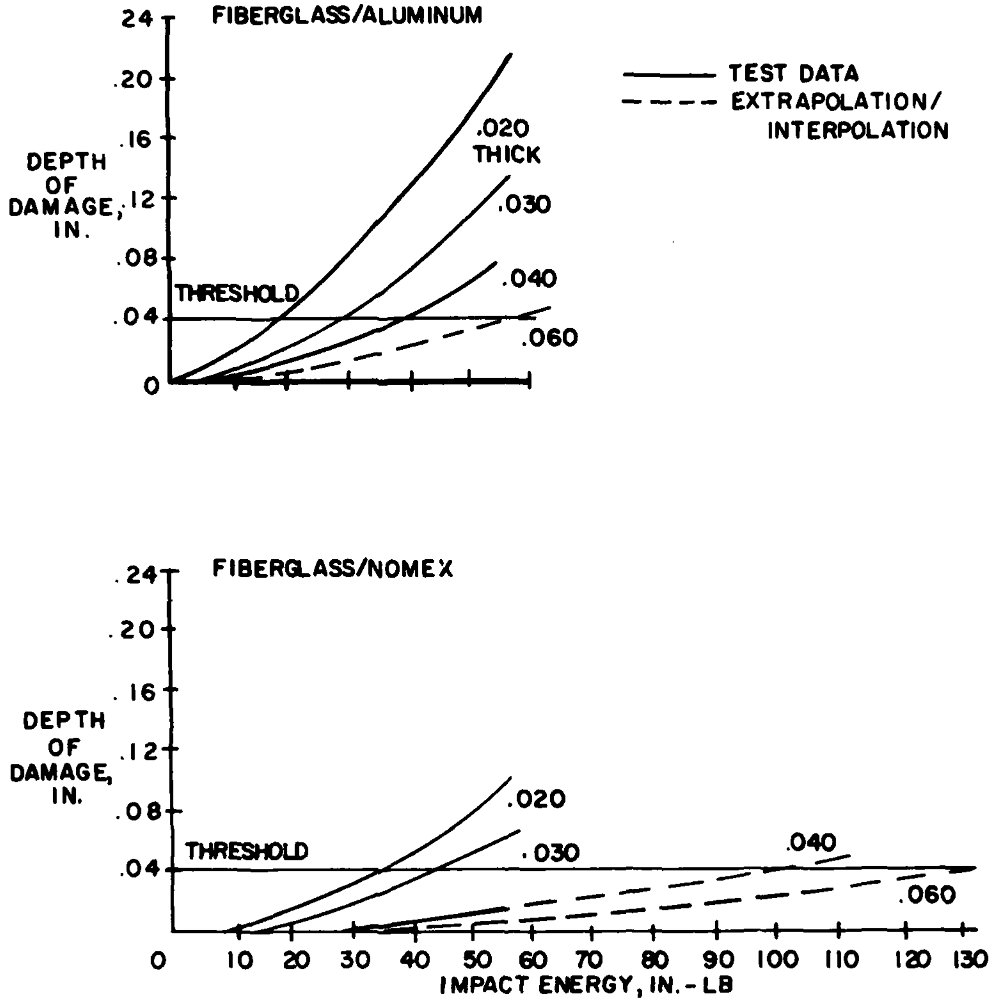


Figure 17. Extrapolated/Interpolated Impact Test Data for Fiberglass-Faced Sandwich Panels

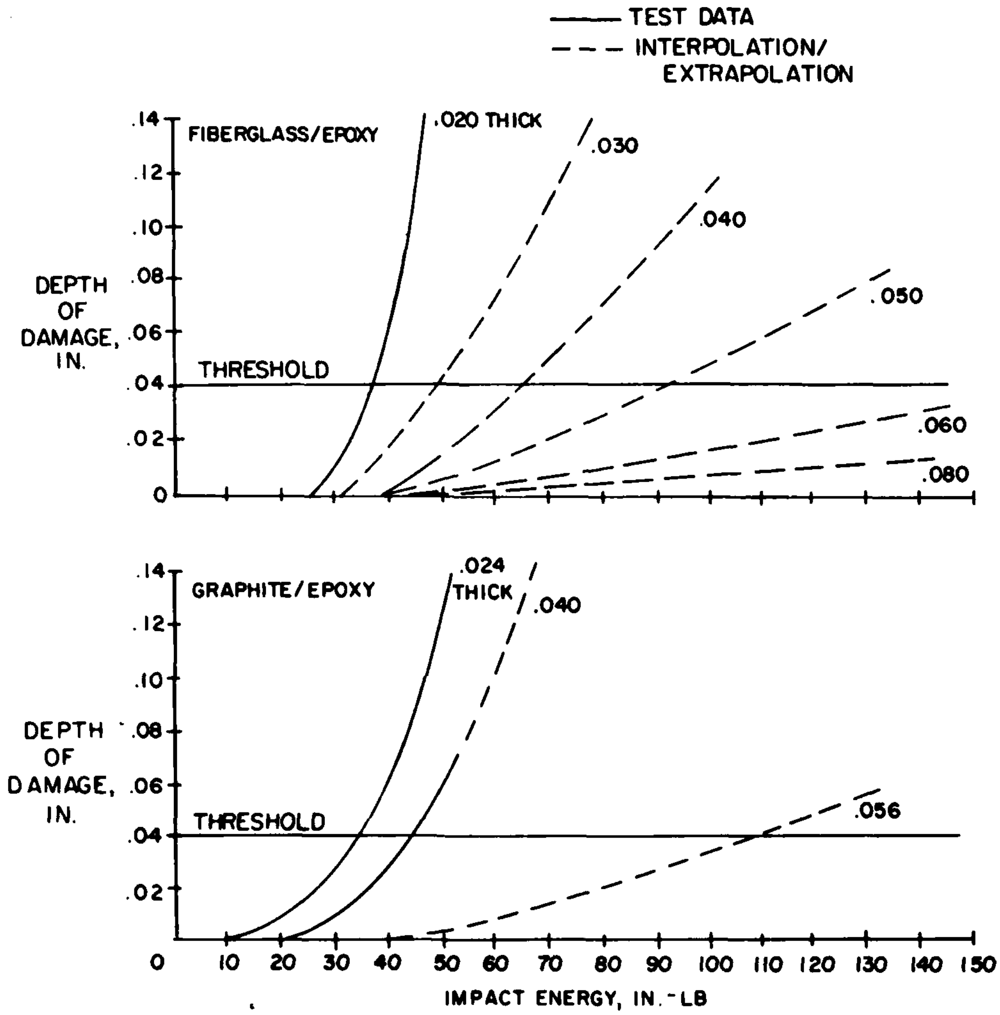


Figure 18. Extrapolated/Interpolated Impact Test Data for Fiberglass/Epoxy and Graphite/Epoxy

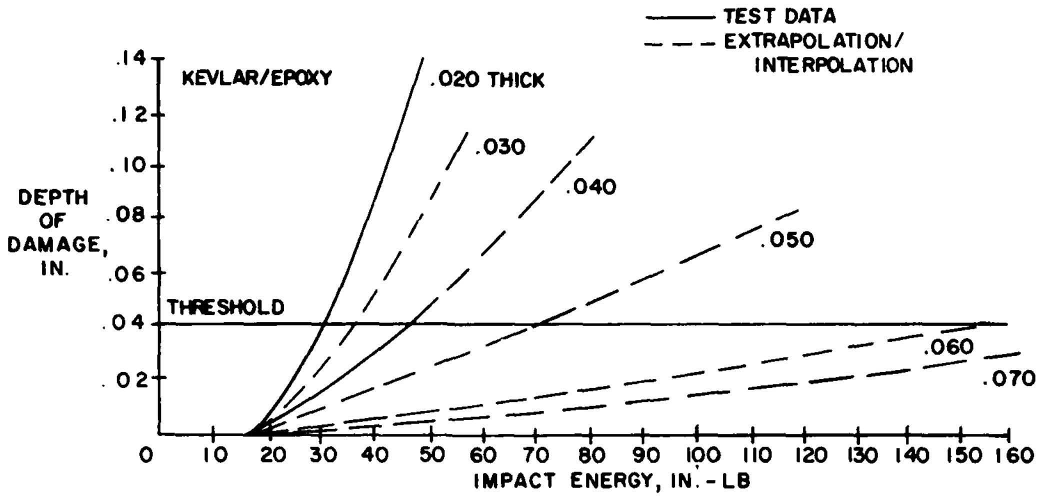


Figure 19. Extrapolated/Interpolated Impact Test Data for Kevlar/Epoxy

Figure 20 illustrates the method used to estimate the fraction of impacts causing repairable damage. For the given design (baseline or option), the impact damage curves were consulted for an identical or similar material configuration. In some cases, a comparable configuration was lacking and it was necessary to interpolate values. A depth of .020 inch was established as a threshold for damage requiring repair. Because the impact testing on which the curves are based involved untypically severe conditions (rigidly clamped specimen, precisely aligned impactor, etc.), for analysis purposes the threshold was set at .040 inch to reflect the less stringent conditions that would normally prevail in service. That damage threshold was located on the damage versus impact energy curve for the respective material and thickness, and the corresponding impact energy was found. That value was located on the impact energy distribution curve for the subject hazard and the corresponding cumulative probability value was found. The probability above that value was assumed to represent the fraction of impacts causing damage.

In addition to the total impact energy, the area over which the energy is distributed will affect the potential for damage. (A low energy impact concentrated in a small surface area may have greater damage potential than a high energy impact distributed over a large surface area.) The amount of kinetic energy absorbed by the part is affected by the size, shape and aspect angle of the impacting object and by the thickness and elasticity of the panel. Modeling a single impact can involve a complex analysis. To evaluate the wide spectrum of cases required for design of the CRF, simplifying assumptions were necessary. Where judgement indicated that the "energy pressure" could significantly affect the damage

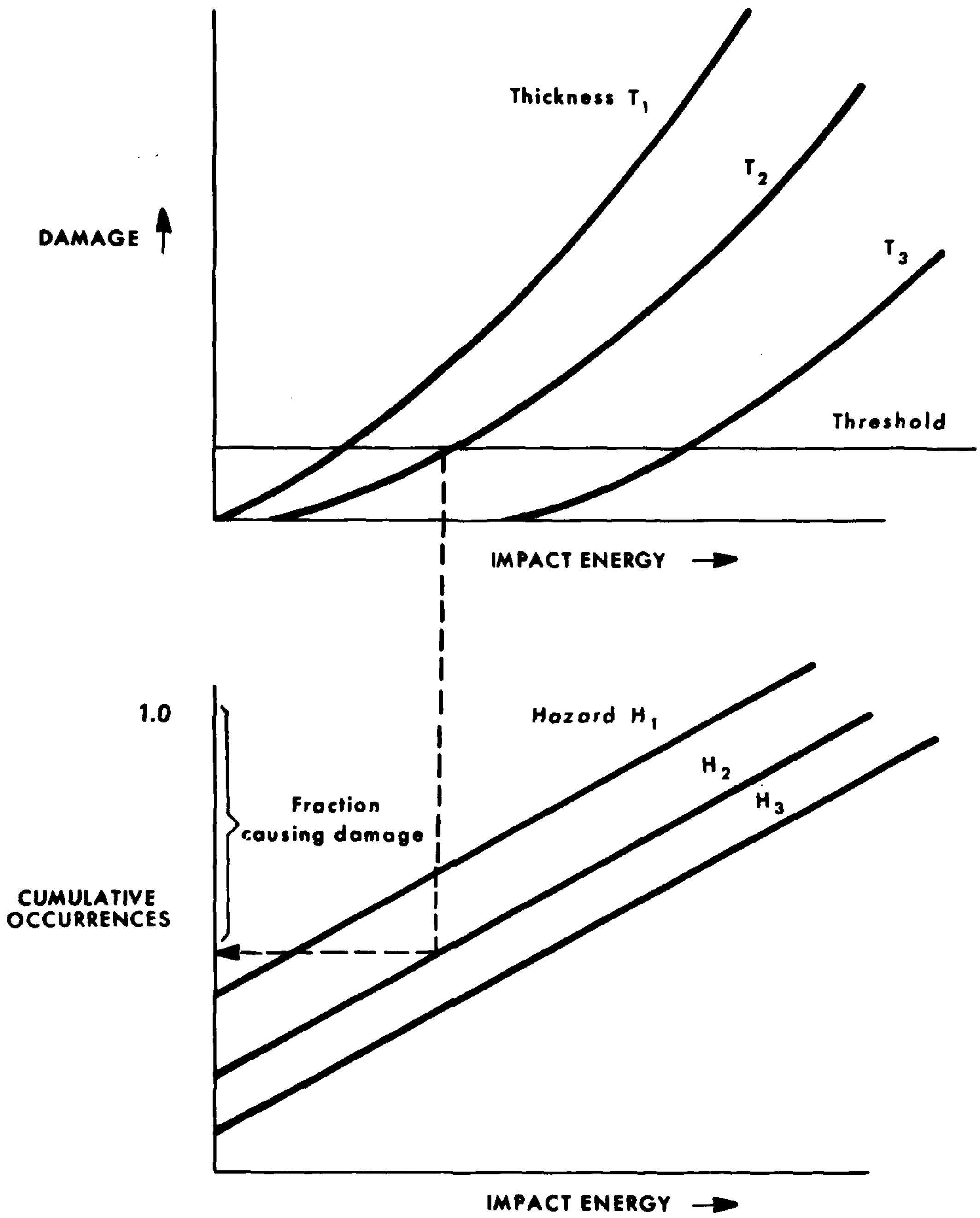


Figure 20. Method of Estimating the Fraction of Impacts Causing Damage

sustained by an impact, a factor was applied to reflect this. Thus, in the case of foot traffic where the impact is frequently distributed over a large surface area, the fraction of impacts causing damage was decreased below that which would be estimated on the basis of total impact energy alone.

Estimates of Damage Rates

The estimated fraction of impacts causing damage was applied to the estimated frequency of impact to obtain an estimated rate of damage requiring repair. Appendix A contains these estimates for the baseline CRF design and the candidate R&M design options. A page from the appendix is shown in Figure 21. The baseline is a CRF designed primarily for structural loads, i.e., without particular attention to R&M. The R&M design options are specific approaches to improving the tolerance of affected areas of the structure to the various impact hazards. The difference between the estimated damage rates for the baseline design and a given R&M design option reflects the anticipated level of improvement in damage tolerance.

Estimates of Repair Costs

Reference 1 tabulates basic types of repairs used on composite structures. Estimates were made of the average cost of labor and materials for each type of repair. Labor costs were estimated for three levels of difficulty: an unconstrained repair, an average repair and a constrained repair. The unconstrained repair is one in which the damage is completely accessible without removing aircraft components or permanently fastened structure and on which the mechanic is able to work in relative comfort (exterior damage on the side of the aircraft, for example). The average repair requires the removal of some components and/or structure to gain access to the damage and may require the mechanic to work under some handicap. The constrained repair is one which requires substantial disassembly to gain access to the damage (removal of a fuel cell for example) and which may require the mechanic to work under difficult or awkward conditions. A standard material cost was estimated for each type of repair. Field maintenance labor hours were priced at $15 per man-hour. Appendix C lists the estimated repair costs.

For each area of the structure and its associated impact hazard, an estimate was made of the generic type of repair that would be used to repair damage of that type. The fraction of total repairs of each type was also estimated. Using the average repair costs from Table 6 and the estimated damage rates, a per flight-hour repair cost was calculated for the baseline design and the R&M design options. The results of these calculations are shown in the right-most columns of Figure 21.

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type**	%	Avg. Cost($)	
Upper deck (center panels). - Impact by dropped parts.	225 (Note 6)	9 (1/25)	Baseline: 3.8 lb. aluminum honeycomb; 4 plies Kevlar.	8	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.053
			Option: 4 lb. Nomex honeycomb, 4 plies E-Glass, 2 plies Kevlar.	.3	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.002
Upper deck (center panels). - Impact by foot traffic.	1,300 (Note 7)	26 (1/50)	Baseline: 3.8 lb. aluminum honeycomb; 4 plies Kevlar.	10	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.067
			Option: 4 lb. Nomex honeycomb, 4 plies E-Glass, 2 plies Kevlar.	1	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.007
Fuselage step doors. - Impact by boot.	1,000 (Note 8)	40 (1/25)	Baseline: 3 plies Kevlar.	17	3-U 4-U	85 15	45 85	.087
			Option: 5 plies Kevlar.	7	3-U 4-U	85 15	45 85	.036
Fuselage step enclosures (interior of fuselage). - Impact by stowed baggage.	2,500 (Note 9)	25 (1/100)	Baseline: 3 plies Kevlar.	24	3-U 4-U	65 35	45 85	.142
			Option: 6 plies E-Glass.	5	3-U 4-U	65 35	45 85	.030

* Per 10,000 Flight-Hours

** U = Unconstrained; A = Average; C = Constrained

Figure 21. Sample R&M Tradeoff Analysis

ACQUISITION COST ESTIMATES

Manufacturing Cost

An estimate was developed for the difference in manufacturing cost between each R&M design option and the corresponding baseline. Material costs were estimated on the basis of changes in materials and/or thicknesses and current prices for composites and honeycombs:

Material	Cost Factor ($\$/ft^2$)
Kevlar	1.175/ply
Unidirectional Fiberglass	.557/ply
Unidirectional Graphite	4.985/ply
Woven Graphite	5.840/ply
Aluminum Honeycomb	2.50/in.
Nomex Honeycomb	6.25/in.

Labor costs were estimated on the basis of differences in the number of plys between the baseline and the option. Time standards for typical operations associated with the fabrication of composite structures were analyzed and were found to average 5 minutes/ft^2/ply. An average factory labor rate of $40/hour was used to calculate the labor costs.

Weight Valuation

The majority of R&M improvement options involve the addition of some weight to the CRF. To include this factor in the tradeoff decision, it was necessary to assign a monetary value to weight.

The monetary benefit or penalty associated with a marginal change in aircraft weight empty depends on the stage in the aircraft's life cycle at which the weight change is being considered. At the preliminary design (rubber aircraft) stage, empty weight significantly affects aircraft cost because of its influence on rotor size, installed power, fuel capacity, etc. Later in the life cycle when the aircraft is in production and these variables are relatively fixed, changes in weight primarily affect fuel consumption and performance thresholds with respect to payload, endurance, hot day performance, etc. Significant weight changes may also affect operating stresses and in some cases component fatigue lives and reliability.

Fuel consumption is one of the directly measurable costs associated with changes in aircraft empty weight. For the sea level, standard day condition, the fuel consumption derivative for the Black Hawk's 2.3 hour design mission is 32 pounds of fuel per 1,000 pounds of aircraft weight. At current fuel prices and a life-cycle utilization of 6,500 flight hours, this translates into a life-cycle cost value of approximately $15 per pound per aircraft.

Aside from the costs associated with fuel (and parts consumption if such effects can be demonstrated), the economic value of weight can be expressed in terms of the fewer or greater numbers of aircraft (or aircraft flight-hours) needed to perform the mission when aircraft are being operated at the performance thresholds (design gross weight, maximum range, etc.). Scenarios can be developed in which threshold conditions are encountered with widely varying frequency; the economic value of aircraft empty weight will vary accordingly. For purposes of evaluating R&M improvement options for the CRF, the value of a pound of weight was evaluated in terms of the number of flight-hours that would be expended or saved over the life of the aircraft through increased or reduced payload capacity at the aircraft design condition. Based on the assumption that 4% of the aircraft missions will be payload saturated (operating at maximum payload for the design condition), and a projected per flight-hour operating cost of $825, the calculated value of one pound of weight is approximately $83 over the life of the aircraft.

Added to the $15 in fuel savings, one pound of empty weight is worth approximately $100. Valuing aircraft weight at $100 per pound in today's dollars is excessive, since the effects on fuel consumption and mission performance on which the valuation is based are costs to be borne in future years. It is reasonable to discount these costs to a current value based on the time value of money. To be conservative, a nominal value of $50 per pound per aircraft was used as a measure of the investment value of weight.

DESIGN OPTION SELECTION

A life-cycle cost delta was developed for each of the proposed R&M design options, based on estimated changes in weight (valued at $50 per pound), manufacturing cost and field repair costs. In each case, the R&M improved design is compared with an artificial baseline that has been designed primarily for structural loads.

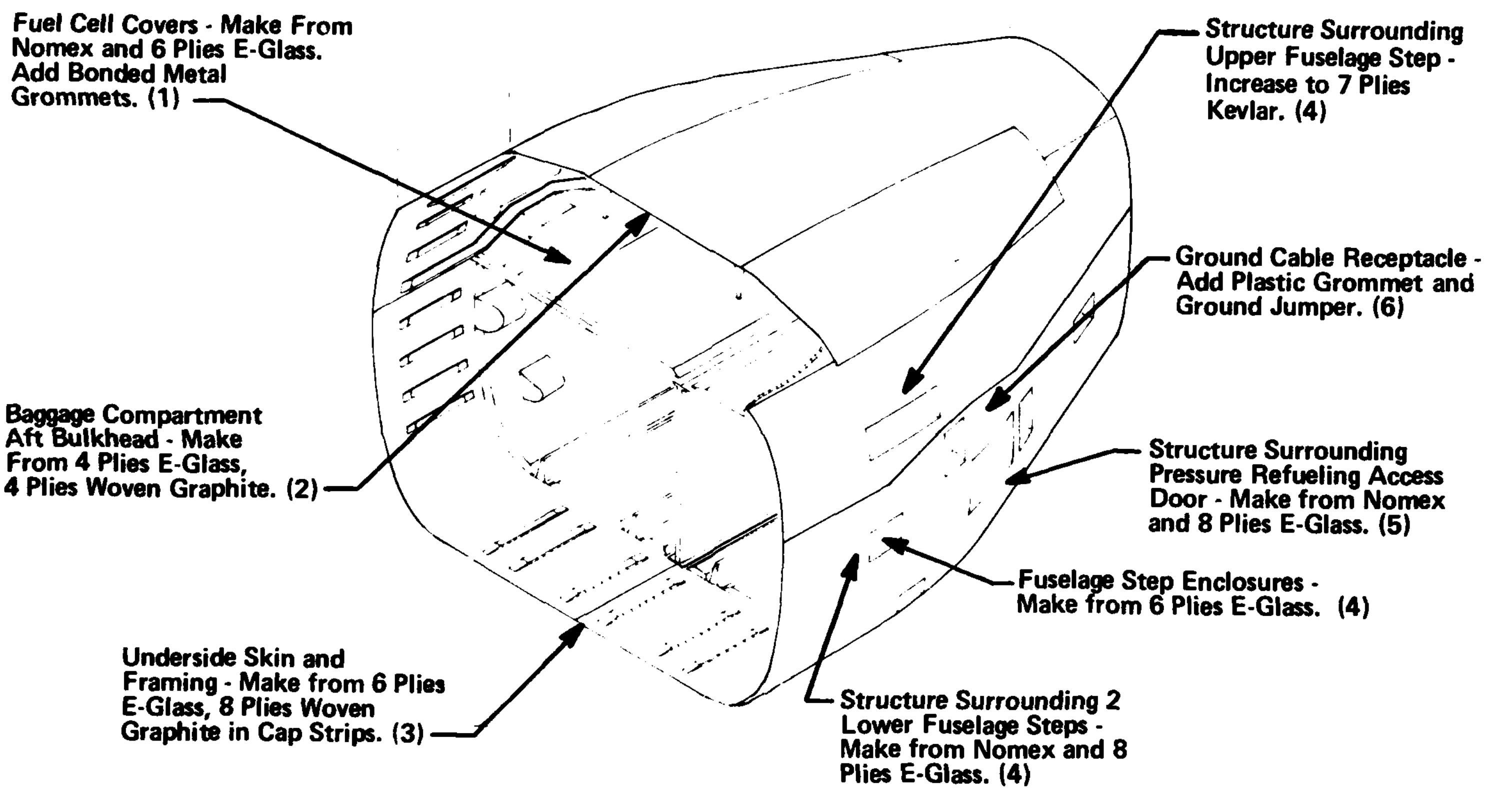
Incorporating a design attribute for the purpose of improving R&M ordinarily will involve trading a present-day investment in manufacturing cost and/or weight for a larger future savings in maintenance cost. Since the maintenance cost saving may occur many years in the future, its value must be related to the rate of return that might be realized through alternative investment of the same capital. The planned service life of the Black Hawk airframe is 20 years. Assuming that maintenance costs are distributed uniformly over this period, at an interest rate of 10%, a rate of return of approximately 2.5 to 1 would be needed to justify such an investment, i.e., a 2½-fold savings in maintenance costs over the life of the aircraft. Recognizing the uncertainty of the repair cost estimates developed under this program, a return on investment of 5 to 1 was selected as the threshold. Design options offering estimated repair cost savings of less than 5 times the estimated cost of incorporating the option were rejected as being noneconomical.

Recommended Design Options

Appendix D summarizes the R&M options considered in the design tradeoffs for the CRF and notes the set of options recommended for design incorporation, based on the rate of return guideline. With the concurrence of the CRF program manager, these were submitted to the design staff in the form of design directives.

At the time the R&M tradeoff analysis was being completed, the CRF was still in the basic data phase, and tradeoffs involving considerations such as ballistic tolerance, electrical conductivity, and manufacturing producibility were still ongoing. It was recognized that decisions resulting from these other tradeoffs could affect the relative desirability of the selected R&M design options. Consider, for example, an area for which it was recommended that E-Glass be substituted for Kevlar to improve resistance to a particular impact hazard. If, because of other requirements arising from the tradeoffs, it was found necessary to increase the thickness of the Kevlar structure substantially over that assumed as the baseline for the R&M analysis, the improvement gained by changing to E-Glass would diminish and might no longer remain a cost-effective option. The directive to the design staff was to incorporate the designated R&M options or, alternatively, to insure that the final design achieved approximately the same level of R&M. The design staff was directed to continue consultation with R&M as each decision point was reached.

Fifteen R&M design options were recommended for incorporation in the design of the CRF (Appendix D). Figure 22 illustrates some of the key recommendations. The fifteen design options (not including the modular design provisions) incur a total weight increase of approximately 22 pounds over the hypothetical baseline and a total estimated investment of $1,650, approximately 2/3 of which is represented by the $50 per pound valuation for the weight increase. The total estimated return on investment for the fifteen options exceeds $12,000; i.e., the expected maintenance cost savings over the life of the aircraft will exceed the investment cost by more than a factor of seven. Those are exaggerated estimates, since an airframe would not be designed for structural loading conditions exclusively as was assumed in the definition of the baseline. (As previously explained, this was done to provide a uniform basis for evaluating options.) The normal design process would have considered the operating environment and, as a result, many of the same design features or variations of them would have been incorporated. It is felt, however, that without the particularly thorough R&M analysis the CRF received under this program, less cost-effective decisions might have been made. Some features with a large payoff potential might have been omitted while others of questionable value were included in the design.



(1) Improved tolerance to stowed baggage, dropped tools and parts, fastener damage.

(2) Improved tolerance to stowed baggage impact.

(3) Improved tolerance to impact with terrain objects.

(4) Improved tolerance to boot impact.

(5) Improved tolerance to refueling nozzle impact.

(6) Improved tolerance to overstress due to cable snagging.

Figure 22. Key R&M Design Option Recommendations

Data developed in Reference 1 indicates that field maintenance of a UH-60A sized metal rear fuselage, including its associated secondary structure (access panels, etc.), is approximately \$1.75 per flight-hour (1980 dollars). If repairs associated with the impact damage analyzed in this program are assumed to represent 90% of the field maintenance that the CRF will require in service, and it is assumed that the recommended list of design options or their equivalent are incorporated in the design, the cost of field maintenance for the CRF will be an estimated \$1.05 per flight-hour. Figure 23 compares the estimated R&M values for the metal versus the composite rear fuselage. As shown, the frequency of maintenance for the composite structure is expected to be much lower than that of the metal structure, due mainly to the greater damage tolerance of the composites and the elimination of most of the nuisance-type repairs associated with metal airframes (loose or missing fasteners, corrosion, skin cracks, etc.). The average cost of a composite structures repair is shown to be greater than the average cost of a metal repair. Again, this is due mainly to elimination of many low-cost, nuisance-type repairs, resulting in a higher proportion of the more significant repairs and a higher average cost. If the nuisance-type repairs were eliminated from the population of metal structure repairs, the average cost of the remaining repairs would probably be greater than the average cost of the composite structures repairs.

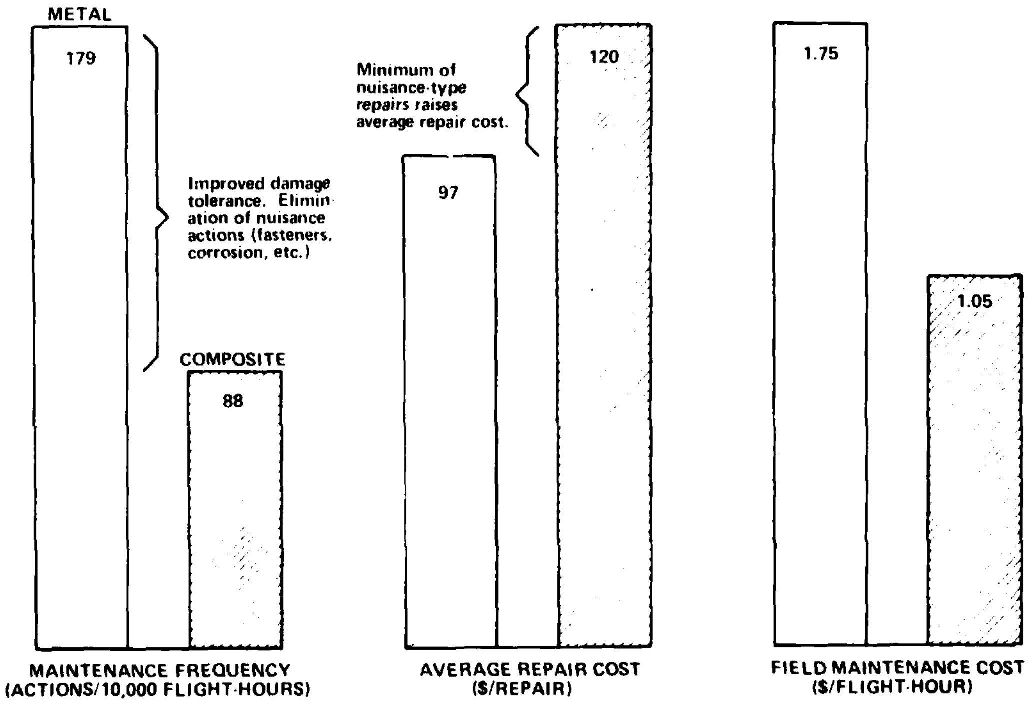


Figure 23. Estimated R&M Values for Composite Versus Metal Rear Fuselage

Overall, the composite rear fuselage will be an estimated 40% less expensive to maintain in the field. The estimates are based very substantially on engineering judgement, but do suggest that the composite structure at a mature stage of development will be significantly less expensive to maintain than the metal structure. The composite structure also has the potential for significantly reducing depot maintenance costs via the modular design approach described later in the report.

MATERIAL SURVEYS

One of the conclusions of the study reported in Reference 1 is that many of the repair techniques being developed for advanced composite structures, primarily those evolving within the fixed-wing aircraft community, will not be suited to the Army field environment. The methods under development frequently require high skills, controlled environments and rigorous quality control, none of which can be assured in the field.

In the field, aircraft repair is often accomplished in the open or under temporary shelters. Controlling the environment from the standpoint of temperature, humidity and cleanliness is seldom possible. Storage of repair materials under controlled conditions is similarly handicapped. In the field, skills and repair facilities are limited; requirements for sophisticated bonding apparatus and nondestructive inspection equipment cannot realistically be imposed.

Surveys were conducted to identify materials suited to repair of advanced composites in the Army field environment. Five categories of materials were investigated:

1. Laminating Resins
2. Reinforcing Fibers
3. Foams
4. Adhesives
5. Potting Compounds

LAMINATING RESINS

Fabricated repairs using prepregs or wet layup will require a laminating resin system. The applications will involve fiberglass, graphite and Kevlar material in both unidirectional and woven form. Ideally, for the Army field environment, the resin system should provide fast curing at low temperature and should have a long shelf life under ambient storage conditions. Workability with respect to measuring, mixing and pot life is also desired.

Resin systems of four types were investigated.

1. Conventional 2-part epoxies
2. Ultraviolet curing resins
3. Polyesters and vinyl esters
4. New epoxies (G.E. Arnox 3000 Series)

Using published data and information obtained through contacts with vendors, the four types of resin systems were rated with respect to specific requirements and objectives.

1. Mechanical Properties
   (high interlaminar shear, ultimate strain, impact resistance desired)

2. Curing Temperature
   (room temperature, moderate temperature preferred)

3. Cure Time
   (1-4 hours desired)

4. Shelf Life
   (6 months minimum)

5. Pot Life
   (½ hour minimum for wet layup)

6. Bonding Pressure
   (14 psi maximum)

7. Workability/Processability
   (minimum measuring, mixing desired)

8. Handling
   (non-toxic, non-flammable)

Properties data on resin systems is sometimes lacking, often inconsistent from one manufacturer to another, and frequently related to the properties of the matrix in a specific application (fiber type and configuration). Accurate comparisons are therefore difficult. The available information is sufficient to draw some tentative conclusions, however. Further testing will be needed to make final determinations.

Conventional 2-Part Epoxy Systems

Six 2-part epoxy resin systems (resin and catalyst) marketed by three manufacturers were examined:

Manufacturer	Resin/Agent
Shell Plastics, Houston, Texas	Epon 828/DTA Epon 828/Versamid 128 Epon 828/DMA
Emerson and Cummings, Canton, Massachusetts	Ecomold L28/Catalyst #9
Celanese Polymer Specialties, Louisville, Kentucky	Epi-Rez 50727/Epi-Cure 826 Epi-Rez 5027/Epi-Cure 826

The conventional 2-part epoxy systems meet most of the requirements for field repair, including adequate mechanical properties. They are useable only in a wet layup repair, however, and require accurate measuring and mixing at the site. Packaging methods have been devised to facilitate this task (Figure 24). Most of the 2-part epoxy systems require 24 hours to cure at room temperature and from 1-4 hours to cure at elevated temperature (typically 160°F to 240°F). Systems offering more rapid room temperature curing were found to have unacceptably short pot lives. Shelf lives for the 2-part systems range from 6-12 months, with the two Celanese products offering up to 3 years as a practical shelf life. The major disadvantages of the 2-part epoxy systems are that they are messy to work with and difficult to apply uniformly, particularly when many plies of material must be laid up. Achieving high quality structural repairs in the field using these systems may be difficult.

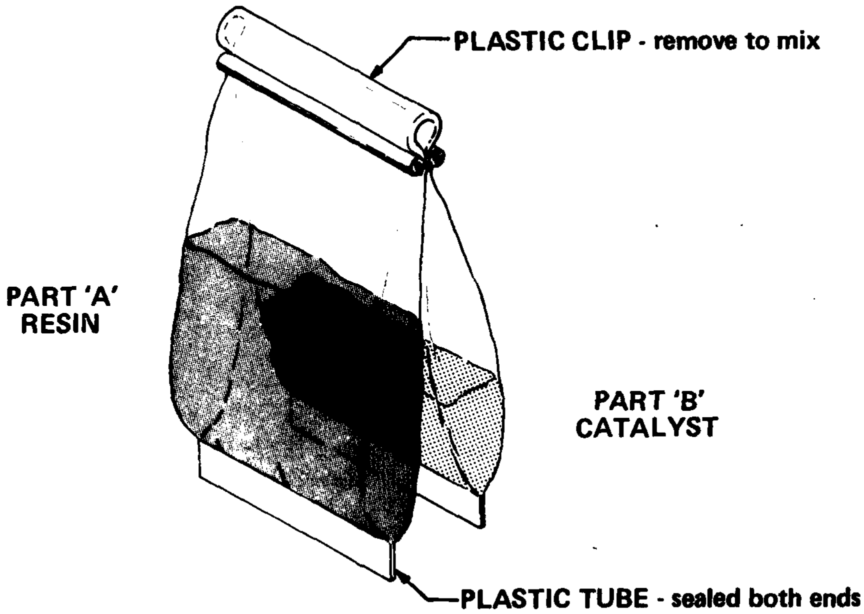


Figure 24. Packaging of 2-Part Epoxy Systems

Ultraviolet Curing Resins

A relatively new innovation in resin systems is the ultraviolet (UV) curing resin. One such product is Penwalt 4899-125 manufactured by the Penwalt Corporation, King of Prussia, Pennsylvania. The major attraction of the UV curing resin is that it cures almost instantly under ultraviolet light. Properties of the UV curing resin are not presently well characterized however, and the system is useable only with laminates that are transparent to ultraviolet light (fiberglass). Further work will be needed to assess the applicability of this system to field repair of composite structures.

Polyesters and Vinyl Esters

Polyesters are among the earliest matrix materials used in composites fabrication. Although more rapid curing than the epoxies, polyesters have poorer mechanical properties and are subject to the effects of weathering, both of which make them a poor candidate for repair of aircraft structures.

The newer vinyl esters were also investigated as a possible laminating system for field repairs. Several disadvantages with this system eliminate it as a practical candidate. Careful measuring and mixing of minute quantities of material is required, and small errors can drastically affect mechanical properties. The vinyl esters give off toxic styrene vapors and have a 90°F flashpoint, making them potentially dangerous.

New Epoxies (G.E. Arnox 3000 Series)

One new epoxide, Arnox 3000 Series, manufactured by the General Electric Company, Pittsfield, Massachusetts, appears to have great potential for use in field repair of composites. Arnox is a one-part system that offers good mechanical properties, requires no mixing, is suitable for use in a prepreg and has a relatively long shelf life under ambient storage conditions. The currently advertised shelf life is 6-9 months, although this may increase substantially with further experience, and may eventually be unlimited. Arnox requires 250°F to 300°F to initiate curing and cures in a few minutes, prospectively offering a large reduction in repair time. There is the potential for incorporating a degree of cure indicator (change of color in the matrix) to confirm proper curing of a repair.

Bleed air from the aircraft APU offers a possible source of heat for curing Arnox repairs in the field. At an ambient temperature of 75°F, the APU supplies 420°F air at a pressure ratio of 3.8:1 and a flow rate of 50 lb/min through the APU ground start port on the left side of the rear fuselage (Figure 4). It would be necessary to mix the bleed air with ambient air to cool it to the required temperature. The bleed air might also be routed through a simple venturi arrangement to create a vacuum for applying pressure to the repair. It is conceivable that the air-mixing/vacuum device might be simple enough to be fabricated in the field.

Aside from the need for relatively high temperature to initiate curing, the only known disadvantage with Arnox is a possibly low resistance to impact damage. Preliminary experiments indicate that it may be no worse than current epoxies in this respect, however. There is very little experience with Arnox to date. Its mechanical properties and its resistance to environmental effects are not well documented. The potential it offers for fast, low-skill repair of composites in the field merits further study.

REINFORCING FIBERS

The CRF will be predominantly of monolithic Kevlar construction. Graphite will be used in all of the primary structural members (beams, frames, longerons and attachment fittings). Sandwich panels made of aluminum or Nomex honeycomb with Kevlar or fiberglass facings will be used in the upper deck area and the panel on the left side of the aircraft which encloses the fuel fillers. Unidirectional or woven fiberglass will be used selectively in areas of high impact exposure.

Graphite, Kevlar and fiberglass are the three types of high performance fibers available for structural repair of composites. Graphite will be mandatory for repair of primary load paths. (The eccentricities involved in a repair joint using a lower strength material such as fiberglass would complicate the repair and cause a significant weight and stiffness penalty.) One disadvantage of graphite is that it is not amenable to repair with the new quick-curing UV resins if this system is eventually adopted for field use.

The remaining structure is primarily Kevlar. Kevlar might be used to repair this structure but it exhibits rather poor wetting characteristics and poor resin-to-fiber bonding in a wet layup repair. Cutting and sanding of Kevlar also tends to be difficult. Fiberglass has a lower strength-to-weight ratio, but does not suffer these other disadvantages. Repair with fiberglass is therefore preferred.

Repair Kits

Reinforcing fibers for wet layup repair of composites in the field could be supplied in kit form. Each kit would contain an assortment of unidirectional tape and/or fabric cut to standard sizes. Several different kits containing graphite and/or fiberglass materials could be developed. Each would be designed and sized for a particular type of structure and area of damage. Alternatively, one general repair kit might be supplied from which the mechanic would select materials of various type and size according to instructions provided in the repair manual. Where fiber direction and stacking sequence are important to the repair, the tape or fiber might be stenciled, color coded or notched to aid the mechanic in obtaining the proper configuration and orientation. Two or more cross plies of material might be sewn together and supplied either dry for a wet layup or as a prepreg, depending on the resin system used.

Preform Patches

Wet layups may be difficult to accomplish when the repair is structural and involves a complex ply configuration. Achieving consistently good quality under field conditions may not be possible. A better alternative, if a suitable prepreg can be developed with Arnox or some other system, would be to supply complete preform patches. The preforms could be of hybrid configuration (mix of glass and graphite laminates) and could be specifically tailored in size, shape, taper, etc., to accommodate the specific structural elements in the CRF (Figure 25). Custom preform patches of this type, called 3-D preforms, are manufactured by Fiber Materials Corporation, Biddeford, Maine, and others. With the preform, after preparing the surface, the mechanic simply removes the single-piece patch from its sealed container, lays it in place and cures it under heat and pressure. The procedure requires minimal skill and should produce consistently high quality repairs. Fabricated in quantity, preforms should not be overly expensive and whatever added costs they do incur over a wet layup method should be greatly exceeded by the savings in labor, reduced aircraft downtime and elimination of faulty repairs. As discussed earlier, the Arnox epoxide system appears to have the potential for use in a workable prepreg for the field. The development of an Arnox impregnated preform should be further studied.

FOAMS

When damaged sections of structural members must be removed, fabricating a repair will require a form of the original part on which to laminate the repair. This might be accomplished by laminating the repair on undamaged structure in the same aircraft or another aircraft and then transferring and bonding the precured patch to the damaged part. Alternatively, the damaged section might be restored with a nonstructural material, creating a form or mandrel on which the repair is laminated and cured in place. Depending on the shape and contours of the part, forms and mandrels might be created with materials such as cardboard or plastic. When the geometry is irregular, a material such as foam which can be molded and shaped to the part would be needed.

A survey was made of available foams that could be used to make lightweight remain-in-place forms or mandrels for laminating composite repairs in the field. It was desired that the foam possess the following characteristics:

1. Mechanical Properties
   (adequate rigidity and strength; low weight)

2. Curing Temperature
   (room temperature preferred)

3. Cure Time
   (1 hour to workable state)

4. Shelf Life
   (6 months minimum)

5. Processability
   (minimum measuring, mixing)

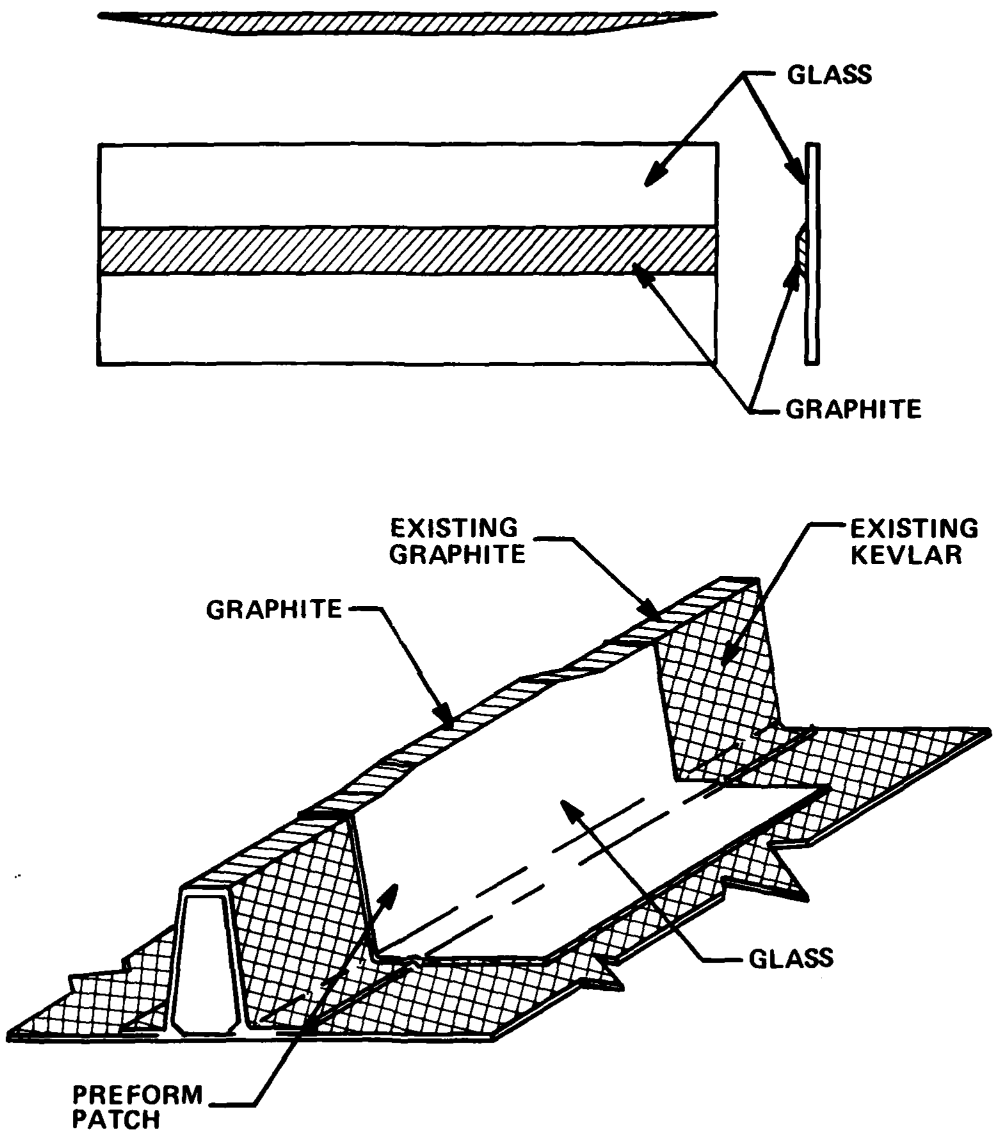


Figure 25. Preform Patches for Simplified Field Repair

6. Processing Environment
   (wide range of temperatures)

7. Storage
   (wide range of ambient conditions; transportable in unpressurized aircraft at 10,000 ft cabin altitude)

8. Handling
   (non-toxic, non-flammable)

9. Other Characteristics
   (closed cell; capable of withstanding 300°F - 350°F for 30 minutes)

Several foam vendors were contacted, all of whom recommended polyurethane foam for this purpose. These foams consist of two components: polyol and isocyanate, which when mixed create an exothermic reaction. One of the byproducts of the reaction is a gas which causes the foam to rise and expand. The density of the foam is controlled by the formulation (the temperature of the reaction and the concentration of blowing agent by-product).

A search of polyurethane foam manufacturers located a product that appears to meet all of the major requirements. Insta-Foam of Philadelphia, Pennsylvania, markets a complete urethane foaming kit, a self-contained unit consisting of two pressurized bottles containing the chemical components and the necessary tubing and mixing nozzle.

Three sizes are available, the smallest yielding about 1/2 ft^3 of foam, an amount that should be adequate for the majority of small area repairs. In an experimental application, it was found that the cured foam is very workable; cutting and sanding are accomplished easily with conventional tools. Densities range from 1.75 to 2.5 lb/ft^3. Demonstrations will be required to evaluate the processability of the foam, including the effects of ambient temperature on processing. Tests will also be needed to verify the ability of the foam to withstand the elevated temperatures at which laminates may be cured.

ADHESIVES

In some cases, it may be necessary to repair by precuring a patch and bonding it to the damaged structure. Generally, secondary bonding is less desirable than a repair that is cured in place on the damaged part. A survey was made to identify available adhesives that would be suitable for bonding precured composites in the Army field environment.

With respect to mechanical properties, high peel strength and a lap shear strength in the order of 2000 psi at room temperature are desired. (For Kevlar-to-Kevlar bonding, the fiber-to-matrix adhesion breaks down at approximately this stress level, making a higher shear capacity in the adhesive bond unnecessary.) It would be desirable to have an adhesive that cures rapidly at room temperature or moderately elevated temperature. Two commonly used adhesives were examined:

Adhesive	Manufacturer
Hysol 9309.3 NA	Hysol Division, Pittsburg, California
EC 1751	3M Company, St. Paul, Minnesota

Both adhesives possess the required mechanical properties, but neither was found to have a short cure cycle at room temperature. Handling strength is achieved in 6-12 hours. Complete curing requires several or more days at room temperature; 2 hours or less at approximately 150°F. The shelf life of both adhesives is 12 months. Pot lives are 30 to 45 minutes at room temperature.

POTTING COMPOUNDS

Repair of damaged honeycomb panels will require a material to fill voids in the honeycomb. For large area damage, the honeycomb will be replaced. For small area damage, a foam or potting compound can be used. Since the filler need not be structural, workability and fast curing are more important attributes than mechanical properties. Low density (lightweight) and moderately good compression strength are the basic properties desired. Two currently used potting compounds were found to meet these requirements:

Compound	Manufacturer
Epocast 169	Furane Division of M&T Chemical, Rahway, New Jersey
PR 1547	Products Research and Chemical Corp., Burbank, California

Both have 12-month shelf lives and pot lives of 30 to 60 minutes. Curing is achieved in approximately 12 hours at room temperature and in 1-4 hours at elevated temperatures of 110°F to 150°F.

FIELD REPAIRS

Techniques are already well-established for field repair of minor damage to secondary composite structures such as fairings, cowlings and doors. These repairs consist primarily of fiberglass patching using wet layup methods and repair of damaged honeycomb with potting compounds or foam. More advanced methods have recently been introduced for repair of composite rotor blades. These involve the replacement of large sections of skin and honeycomb using prefabricated patches and specialized bonding equipment. Composite blade repair has thus far been limited to afterbody structure; no repair of the heavily stressed spar is presently allowed.

For the Army field environment, there are presently no established techniques for repair of heavily loaded primary structure. Techniques being developed elsewhere in the military, primarily within the fixed-wing aircraft community, are generally not suited to the Army. The methods under development frequently require high skills, controlled environments and rigorous quality control, none of which can be assured in the field.

Cure-in-Place Repairs

Repair of damage of primary structural components such as frames, beams and longerons will require the development of new techniques. Components of this type typically are highly stressed. In the CRF, they are constructed predominantly of hybrids employing unidirectional graphite in beam caps and frame caps for stiffness and strength. Repair of these components, particularly tension-loaded members, will be more critical than any of the composite structures repairs now being done in the field.

The serviceability criteria mapping described later in the report identifies areas of the CRF which appear amenable to repair in the field. They include all of the composite structure with the exception of structural fittings and the intersections of framing members. It is expected that damage involving structural intersections will require a custom-engineered repair or replacement of a section of the structure. It is improbable that repair of structural fittings will be permitted at any level of maintenance. Significant damage to these fittings will probably require replacement of a portion of the CRF.

Field repairs use wet layup techniques primarily. Presently, there are no prepregs that appear suitable to the Army field environment, since they all require controlled temperature storage. However, in the section of the report covering repair materials, the new G.E. Arnox Series 3000 epoxide was discussed as a promising candidate for a workable field-level prepreg.

The method of wet layup repair of composite structure is well-documented. Repairs of this type differ mainly with respect to configuration (materials, geometry, stacking sequence, etc.). Essentially the method involves the application of successive layers of fabric or tape using a liquid resin system. The patch is allowed to cure in place, usually under pressure and elevated temperature. One of the problems of a wet layup or prepreg repair is that of reconstructing the form and contours of the damaged section preparatory to laying up the repair. Figures 26 and 27 illustrate a concept that might be used. In Figure 26, the damaged structure is accessible from both sides; in Figure 27 access to one side is blocked by a backing of rigid ballistic foam.

Step 1: The damaged section of structure is cut away and trimmed. For the blind-side repair, the loose and damaged areas of ballistic foam are carved out, leaving a cavity in the backing.

Step 2: For the two-sided repair, a mold of cardboard or plastic is fitted to the damaged hat section and taped in place. A vented container is held or fastened in place on the opposite side of the structure, and polyurethane foam is sprayed into the container and allowed to cure. The container and the mold are then removed, leaving the shape of the hat section formed by the hardened foam. For the blind-side repair, the foam is sprayed directly into the cavity in the rigid ballistic foam, unless the repair is on the side or underside of the aircraft, in which case the foam will have to be contained. The cured foam is then trimmed to the shape of the hat section, using the original structure and a hand-fashioned template as a guide.

Step 3: The surface is prepared and cleaned and the hat section is laid up and cured in place, using the materials and number of plies specified in the maintenance manual. Repairs of frames, beams and longerons include the installation of unidirectional graphite tape in the cap areas. If a suitable prepreg can be developed (see discussion of repair materials), a complete one-piece preform patch, exactly sized and configured for the component, would be installed.

Step 4: Polyurethane foam is sprayed into the cavity in the repaired hat section, allowed to cure and trimmed flush with the skin line.

Step 5: A skin patch is laid up over the closed hat section and allowed to cure, completing the repair.

Figures 28 through 30 illustrate variations of the wet layup or prepreg type of repair. In cases where the geometry is complex and/or access is particularly restricted, the concept would be to lay up and cure a patch on similar undamaged structure in the same aircraft or another aircraft, then transfer and bond the precured patch to the damaged part. This is not the preferred method, however, because secondary bonds lack the strength of co-cured repairs. In Figures 29 and 30, a concept for repairing a damaged beam/stiffener intersection is shown. As mentioned previously, this type of repair would probably have to be custom-designed.

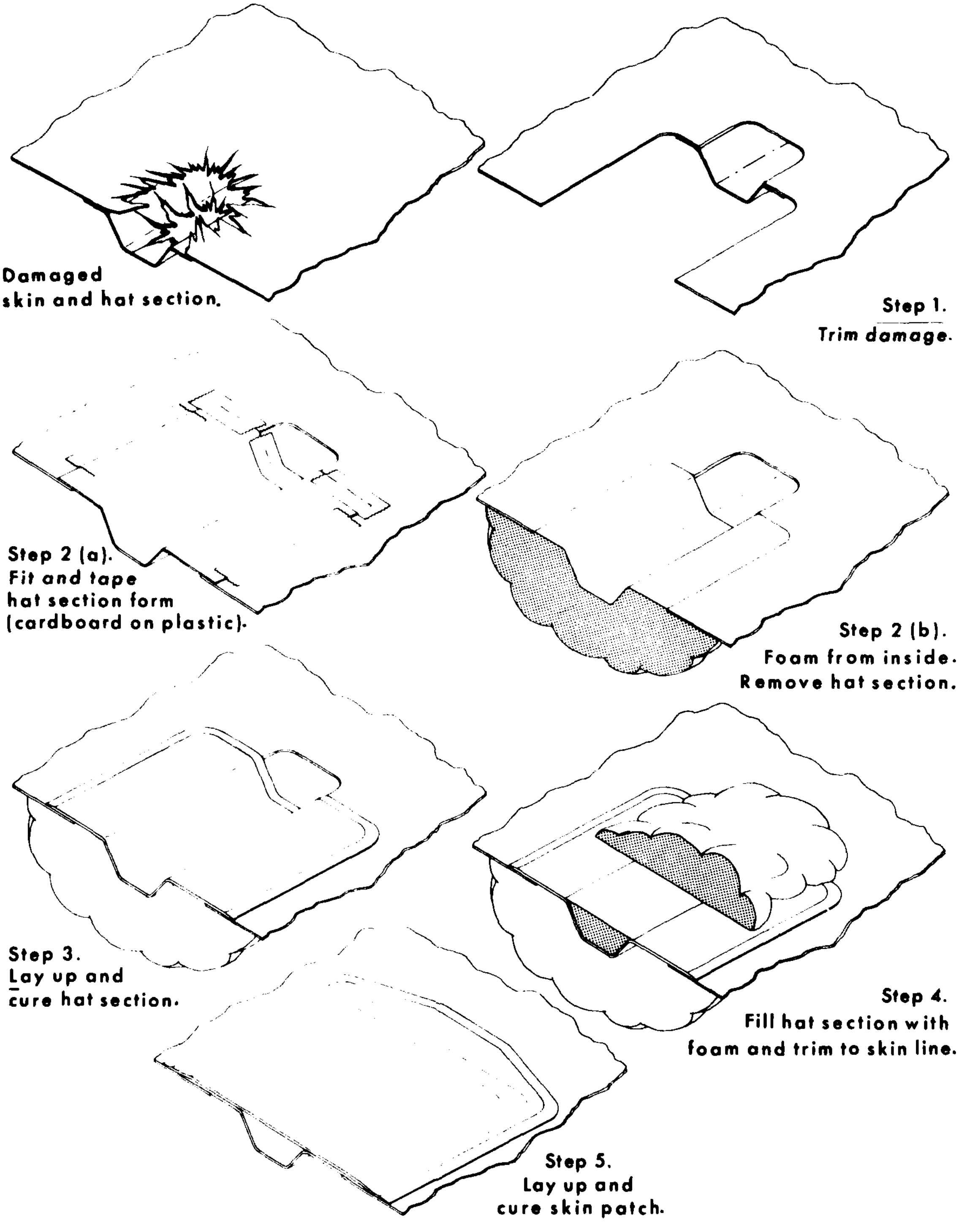


Figure 26. Two-Sided Repair of Damaged Skin and Hat Section Member Using Polyurethane Foam

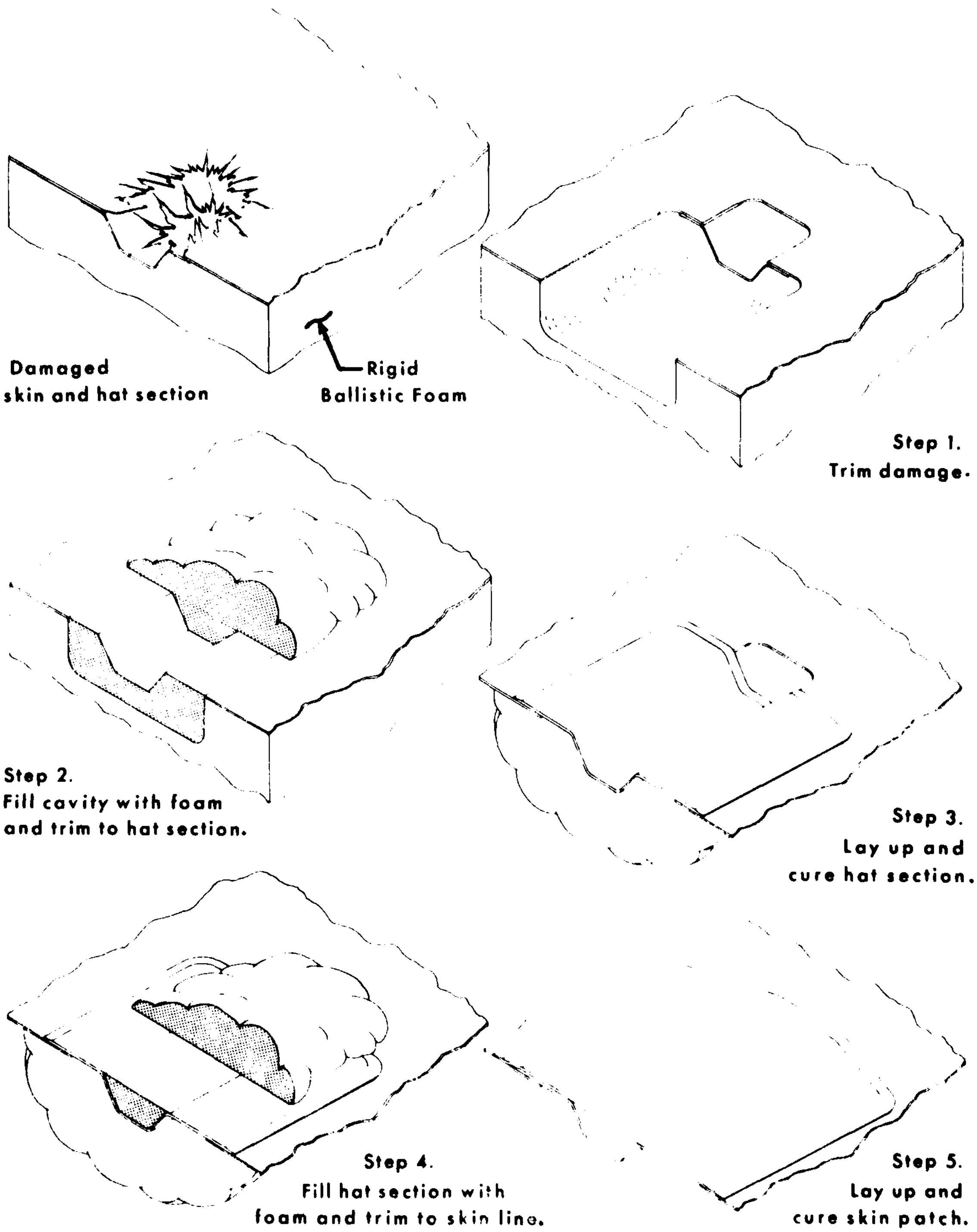


Figure 27. Blind-Side Repair of Damaged Skin and Hat Section Member Using Polyurethane Foam

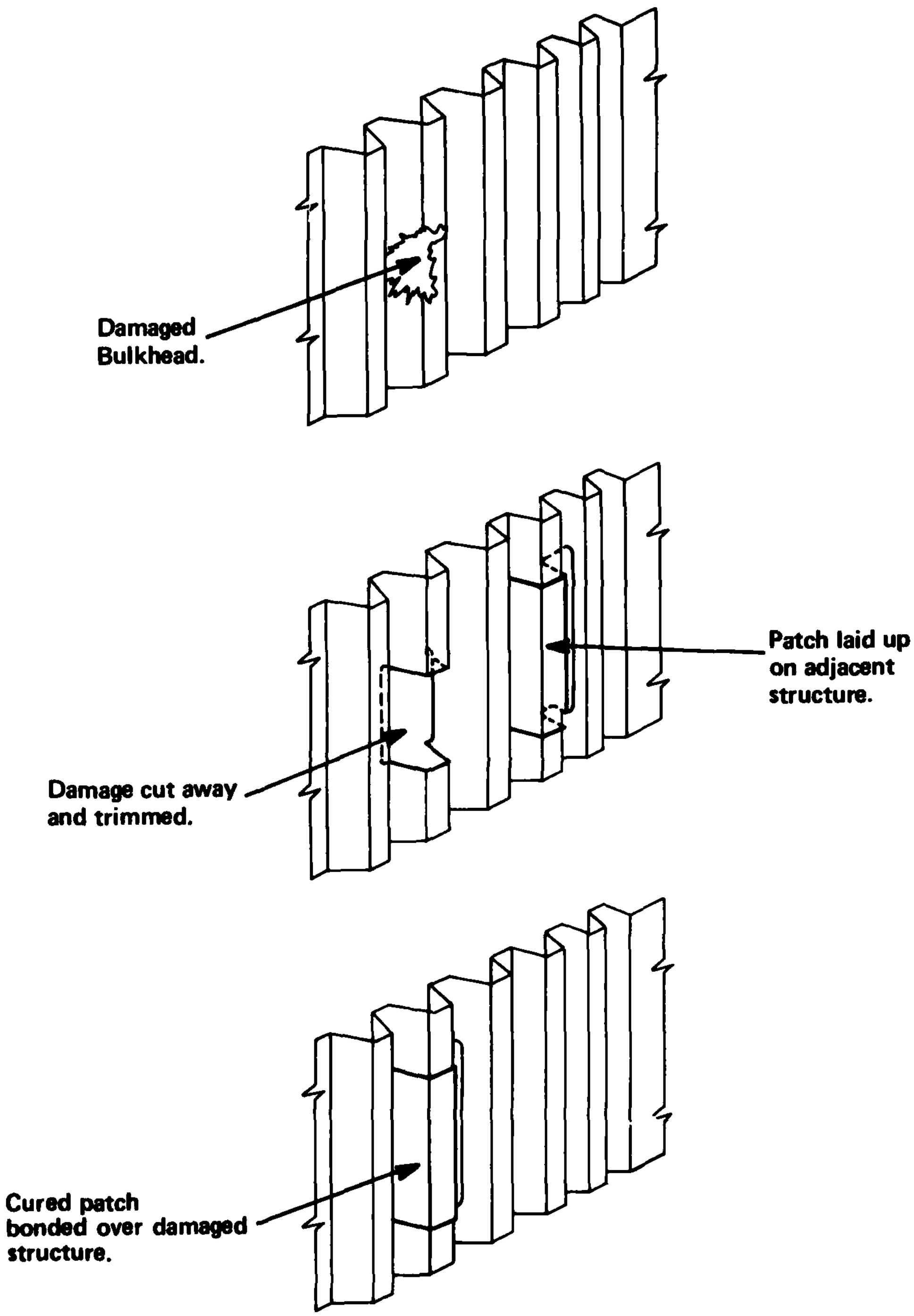
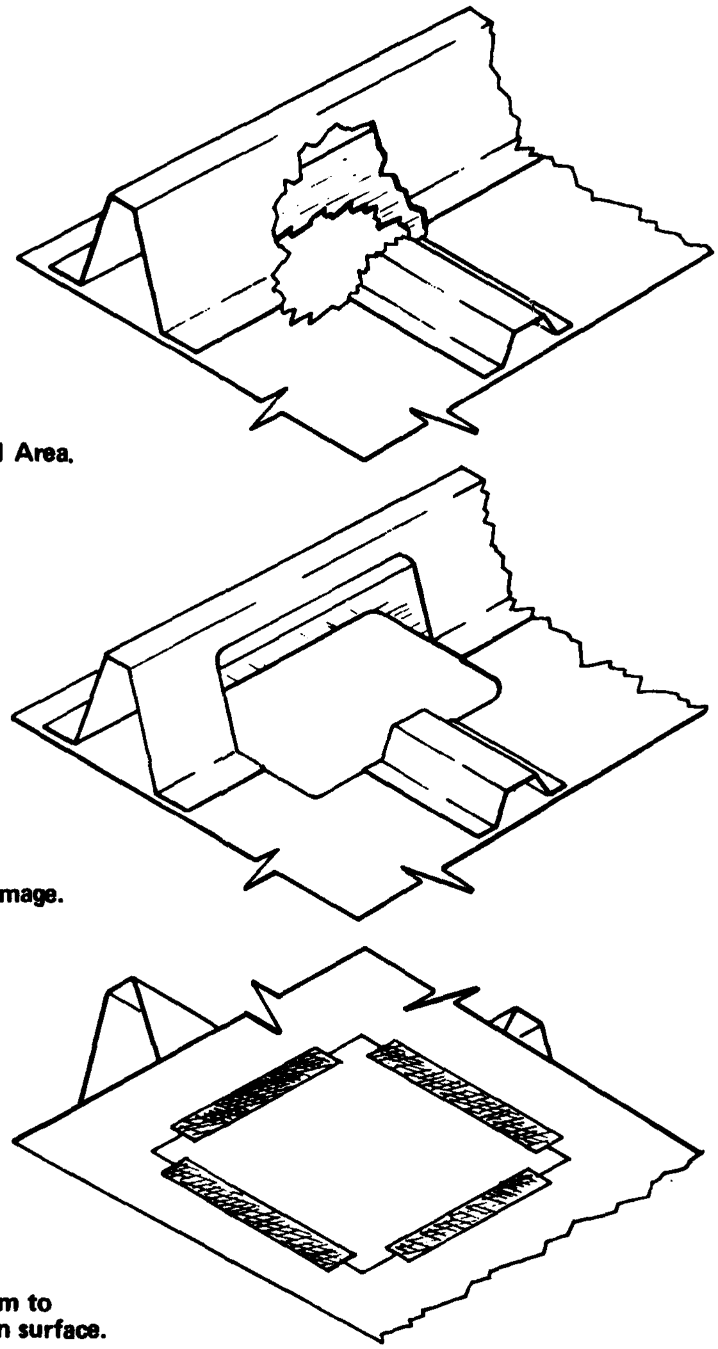


Figure 28. Repair of Corrugated Bulkhead With a Precured Patch Molded on Adjacent Structure



1. Damaged Area.

2. Trim Damage.

3. Tape form to outside skin surface.

Figure 29. Repair of Framing Member Intersection (1 of 2)

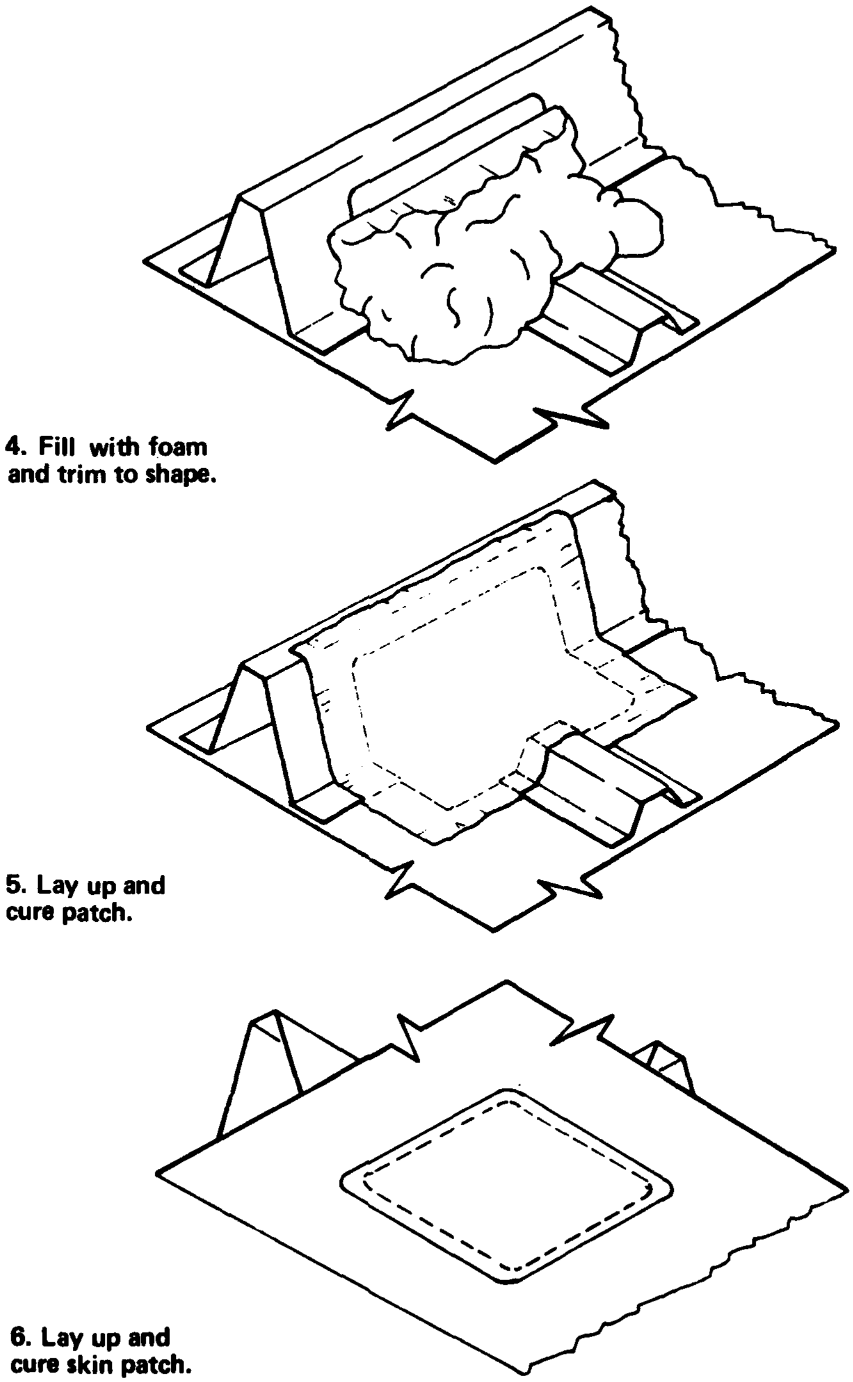


Figure 30. Repair of Framing Member Intersection (2 of 2)

MODULAR DESIGN CONCEPT

Composite airframe structures will be lighter and less costly than metal airframe structures. These improvements are made possible by the ability to fabricate major sections of the airframe as continuous monolithic structures, greatly reducing parts counts and assembly man-hours. The monolithic nature of composite structures can create problems for repair, however.

Standard field repairs of the type described in this report will be suitable for repair of most small area damage and some large area damage to the structure. When fully developed, these repairs are expected to be less complex and less costly than equivalent repairs of metal structure. Major structural repair may be significantly more difficult with composites than with metals, however. Large scale bonding operations typically require the use of an autoclave and this is impractical when the structure involved is an integral part of the airframe. Mechanically assembled repairs may also be prohibited. Unlike metal structures where convenient splice locations can almost always be found, there may be no acceptable location at which to mechanically splice a composite structure. Unless provisions are made in the basic design, holes cannot be drilled in primary composite structure without introducing unacceptable stress concentrations.

Modular design has evolved as a solution to the problem of large area structural repair. Under this concept, the CRF is designed in sections or modules of a size that can be removed and discarded in the field. When damage to the CRF extends over a large area and/or involves multiple adjacent load paths, repair is accomplished by replacing the module(s) containing the damage.

The CRF structure is continuous except at several mechanically joined manufacturing splices. The manufacturing splices are the lines of separation for the modules at these locations. Modules are separated at all other locations by cutting through integral repair strips in the structure. Replacement modules are installed with metal splice straps and mechanical fasteners.

The modular design concept provides a greater degree of structural repairability than is possible even with metals. It is possible that the entire CRF could be replaced piecemeal with modules. Replacement of modules employs techniques that are commonly used in the field, requires no special tools and can be accomplished by personnel of average skill. Other than the modules themselves, installation is accomplished with commonly available materials. Verification that a satisfactory repair has been made requires only a simple visual inspection (unlike a major field-cured structural repair whose quality may be impossible to verify in the field).

Replacement of modules may be the only practical approach to repair of large area structural damage in the field. The alternative to replacing modules would be to custom-engineer and install a repair using very strict process and quality control. Nondestructive inspection of the completed repair would probably be required. The skills and equipment required for a custom-engineered repair are beyond those normally available in the field; special repair teams would have to be dispatched from a depot or the aircraft moved to a depot for repair.

MODULE DEFINITION AND SIZING

The first step in the development of the modular design concept was to define and size the modules. Because the CRF was intended to replace an existing metal structure, certain constraints were placed on its design, particularly with regard to the structural interfaces with the aircraft cabin and tail cone. These constraints permitted less freedom of choice in the definition of the modules than would have been permitted if the CRF were designed originally as part of a total composite airframe.

Several considerations were involved with the definition and sizing of the CRF structural modules:

1. Probable locations, modes and extent of structural damage to the CRF.

2. Size and weight of individual modules from the standpoint of handling and replacement.

3. The availability of natural breaks in the structure occurring along manufacturing splices.

4. Accommodation of functional interfaces with components of other systems located on or within the CRF.

Probable Damage Considerations

Aside from infrequent incidents such as the aircraft striking a terrain object or being struck by a ground vehicle, large area structural damage to the CRF will occur primarily in combat. The major threat is the 23mm high explosive incendiary (HEI) projectile. Structural damage caused by the 23mm HEI is highly dependent on the fuse mechanism and the configuration of the structure. Two types of fuses may be encountered: superquick and time delay. With both types, the damage mechanism involves penetration by metal fragments traveling at high velocities, coupled with the effects of explosive blast and overpressure.

With the superquick fuse, detonation occurs immediately on contact with the surface, causing massive entry-side damage. For typical airframe construction, a 16-inch-diameter hole in the skin and underlying structure could be expected. Deformation of adjacent structure might also occur.

After the initial penetration, the fragments continue to travel forward and outward in a cone-shaped trajectory. Depending on the geometry of the airframe, and the components housed within it, varying amounts of interior structure will be damaged via penetrations and imbedded fragments. Some fragments may travel completely through the structure and exit through the opposite skin.

The delay fuse HEI employs essentially the same damage mechanism except that detonation occurs some distance after the initial penetration, causing less damage to the entry side (typically a clean hole) and greater damage to the interior and exit sides of the structure. The delayed fuse HEI also introduces overpressure caused by explosion within a confined volume, often resulting in buckling or rupturing of the surrounding structure.

Hostile fire should be received primarily from the lower hemisphere. The fragment pattern associated with the HEI (Figure 31) therefore tends to favor a sizing scheme incorporating small modules in the lower section and larger modules in the upper section. This was set as an objective, but other considerations tended to influence a more uniform distribution of module sizes in the final configuration.

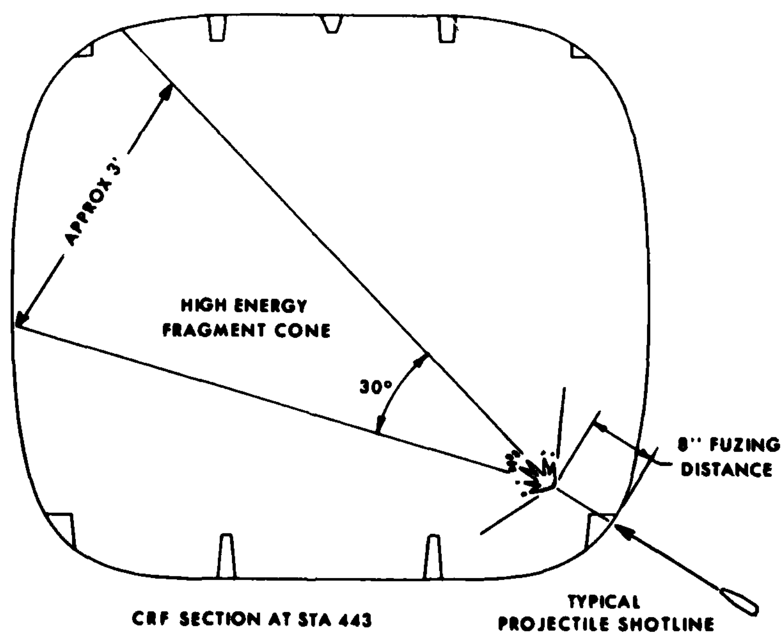


Figure 31. Typical HEI Damage Pattern

Module Size and Weight

An objective in sizing the modules was to make no one module larger in size or heavier in weight than could comfortably be handled by two men, recognizing that installation of some modules would require the mechanics to work under awkward conditions (lying under the aircraft for example).

Utilization of Natural Structural Splice Lines

A third objective in defining the module arrangement was to take advantage of existing manufacturing splices. Utilizing the mechanically joined seams in the structure to locate modules would minimize the addition of repair strips to the CRF, reduce the amount of cutting and fitting required to replace modules in the field, and eliminate the need for external splice straps in these locations.

Accommodation of Functional Interfaces

A fourth objective in defining the module arrangement was to minimize the amount of interfacing hardware that had to be removed to replace any one module. Having a module partially extend under a fuel cell was to be avoided, for example, since it would require that the fuel cell be removed even if the damage was in an area of the module outside of the fuel cell.

MODULAR DESIGN CONCEPT

The selected module configuration is shown in Figure 32. The entire CRF is divided into 18 modules, 8 in the upper half and 10 in the lower half. Mechanically joined manufacturing splices define module breaks at various locations:

1. At Stations 379 and 398 the CRF is joined to the metal cabin structure with bolts and rivets. All of the modules at the forward end of the CRF separate from the cabin at these joints.

2. At Station 485 the composite frame sections integral with the modules are bolted to the metal frame of the tailcone; all of the modules at the aft end of the CRF separate from the tail cone at these joints.

3. A manufacturing splice at waterline 238 is the line of separation for adjacent modules in the upper and lower halves of the CRF.

4. The titanium panels (Modules #3, #4) outboard of the firewalls are installed entirely with mechanical fasteners. Modules #1 and #2 are therefore fastened mechanically on both sides and at their forward and aft ends respectively.

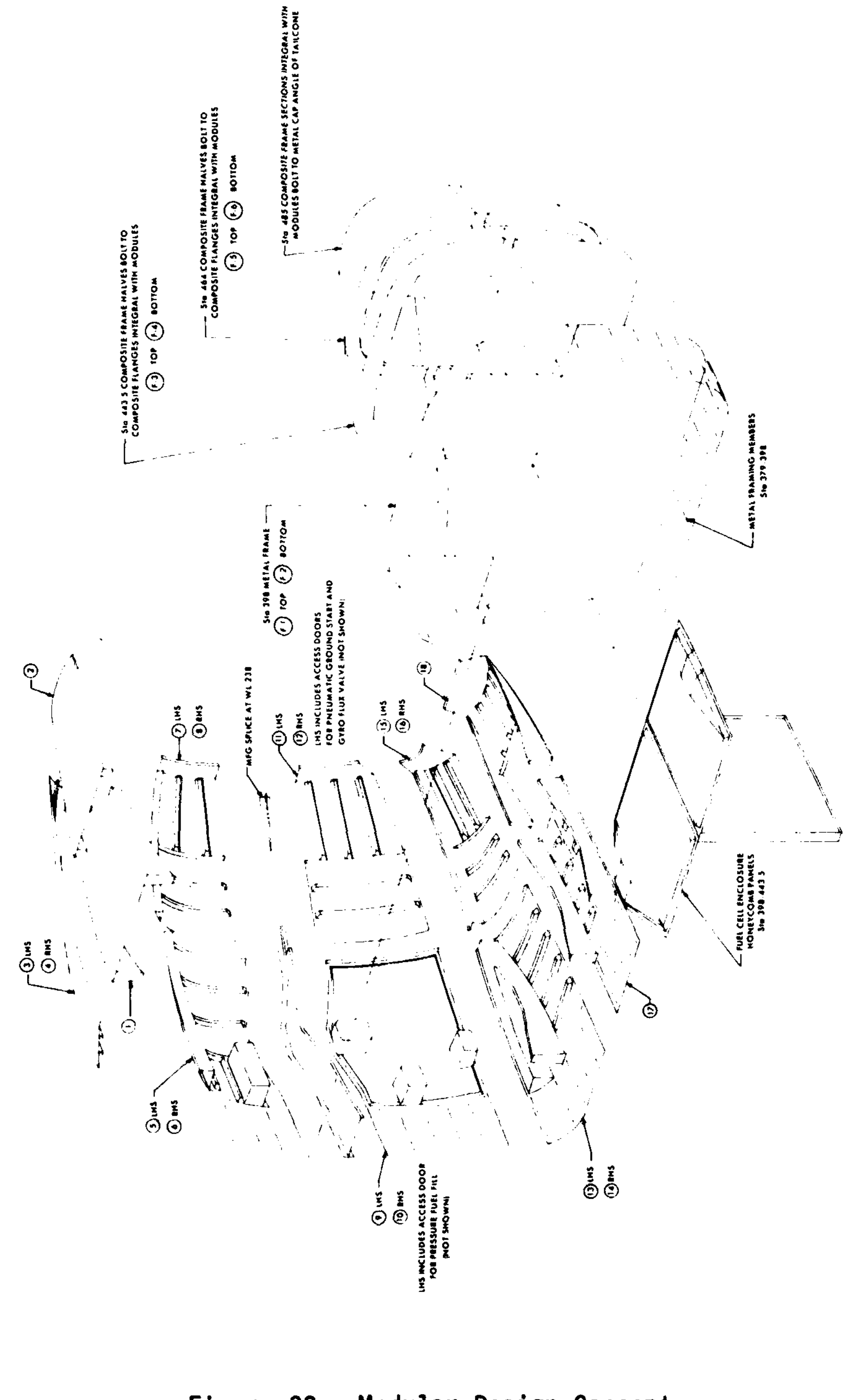


Figure 32. Modular Design Concept

The modules are separated at all other locations via integral repair strips along which cutting lines are marked. Repair strips are formed primarily by extending the flanges of structural members such as stiffeners, beams and angles (Figures 33 and 34). The typical repair strip is a 2-inch-wide band 10 plies thick (4 skin plies plus 6 flange plies).

Cutting Lines and Fastener Locations

Fastener hole locations are marked in the repair strip along both sides of the module cutting lines. These are the locations at which splice straps are riveted for installation of the replacement module. Several approaches to marking the module cutting lines and fastener locations during manufacturing were considered. (It was considered impractical to require the use of templates and/or precise measurements by the mechanic to locate them in the field.) Painting and stenciling were ruled out as being non-permanent. A punched paper or metal tape laid up in the outer ply of the structure was considered, but it was concluded that the holes in the tape would not be visible through the exterior paint. Use of paper might also adversely affect conductive coating performance, especially radar reflectivity. Two methods of providing a permanent and precise indication of the cutting lines and fastener hole locations in the manufactured part are considered feasible. Both are integral with the bonding tools and require negligible additional manufacturing labor.

With the first method, the surface of the tool is raised intermittently to form the impression of a dashed (cutting) line between two rows of dimples (fastener hole locations) in the cured part. Even if relatively shallow, the impressions should transfer through to the opposite side of the laminate, making them visible from both the outside and interior of the aircraft. If paint should obscure the impressions, locating the cutting line approximately and running sandpaper along it on the raised impression side will remove the paint from the high spots and make the lines and hole locations immediately visible.

The second method is similar except that the cutting line and hole locations are scribed into the surface of the bonding tool. When the part is laid up in the tool, surplus resin fills the impressions, transferring the dashed cutting line and fastener hole locations to the cured part in the form of a resin flashing (Figure 35). The resin flashing should be visible through paint but if obscured can be located with light sandpapering along the surmised location as described before. Scribing marks in the tool surface should be the least expensive and most durable of the two methods. The disadvantage of this method is that the cutting lines and fastener hole locations will only be visible on the tool side of the part (currently expected to be the exterior of the aircraft). However, if it were necessary to cut or drill from the opposite side of the structure, pilot holes could be drilled through the structure from the marked side to locate the points.

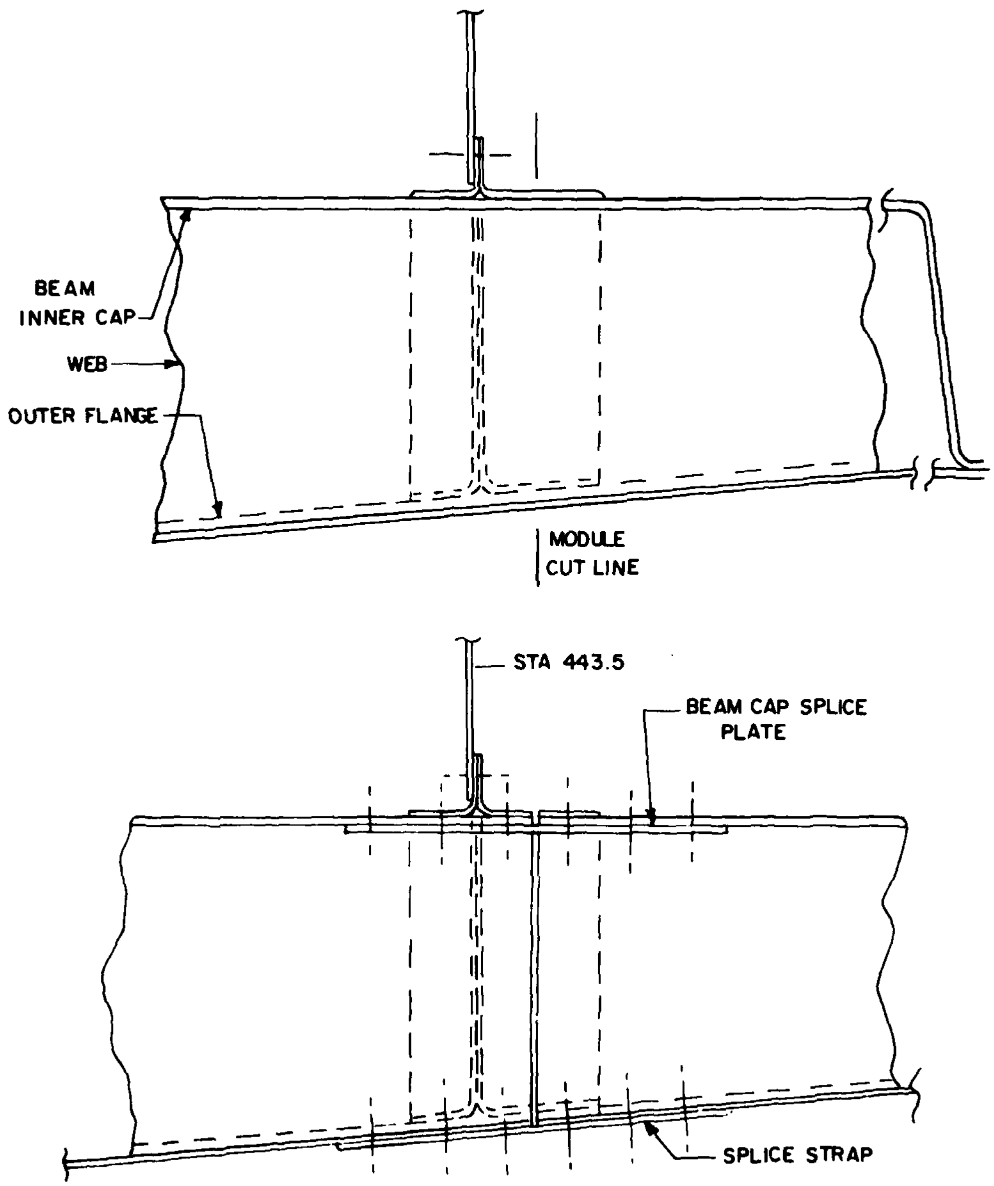


Figure 33. Beam Splicing Concept

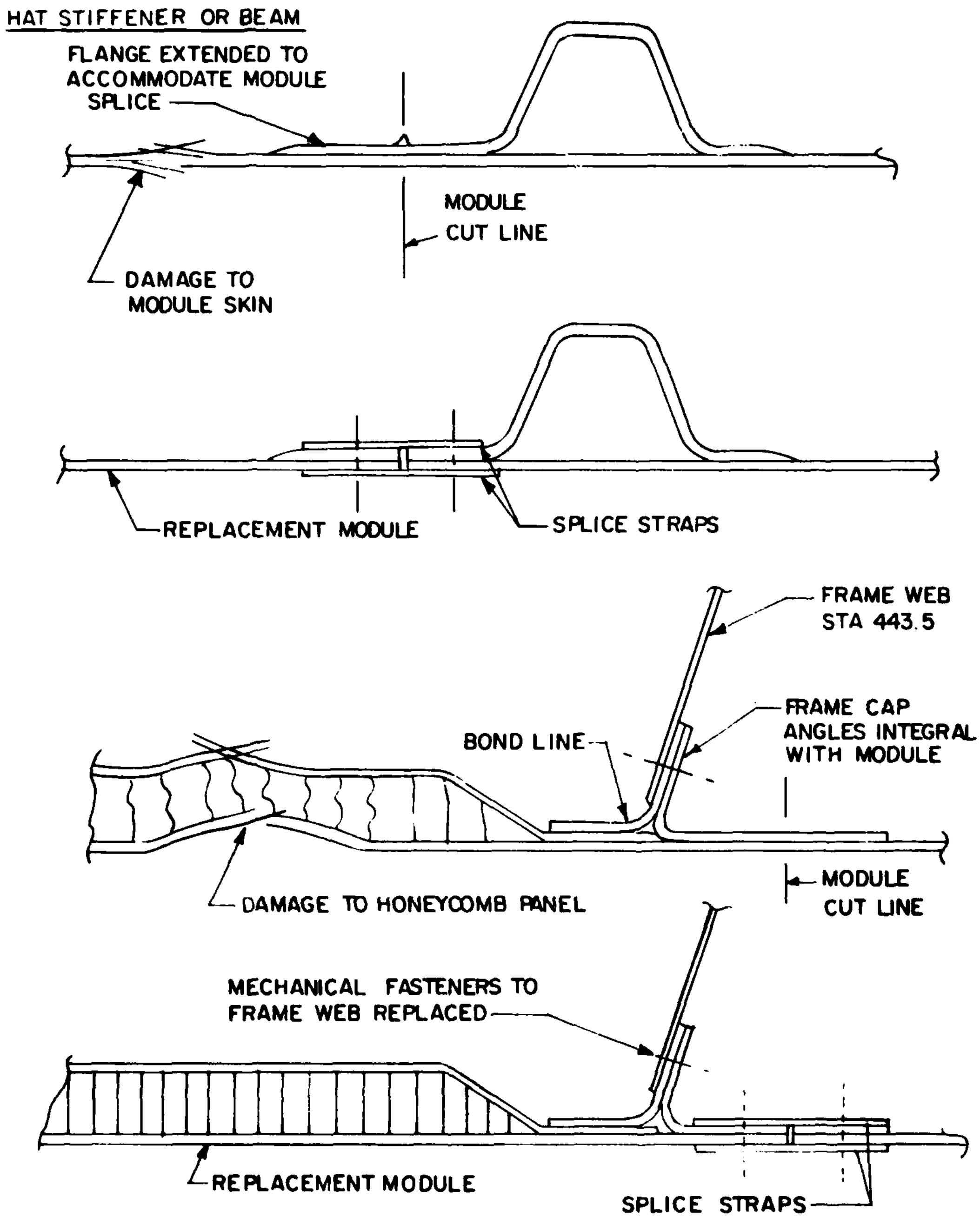


Figure 34. Module Splicing Concepts

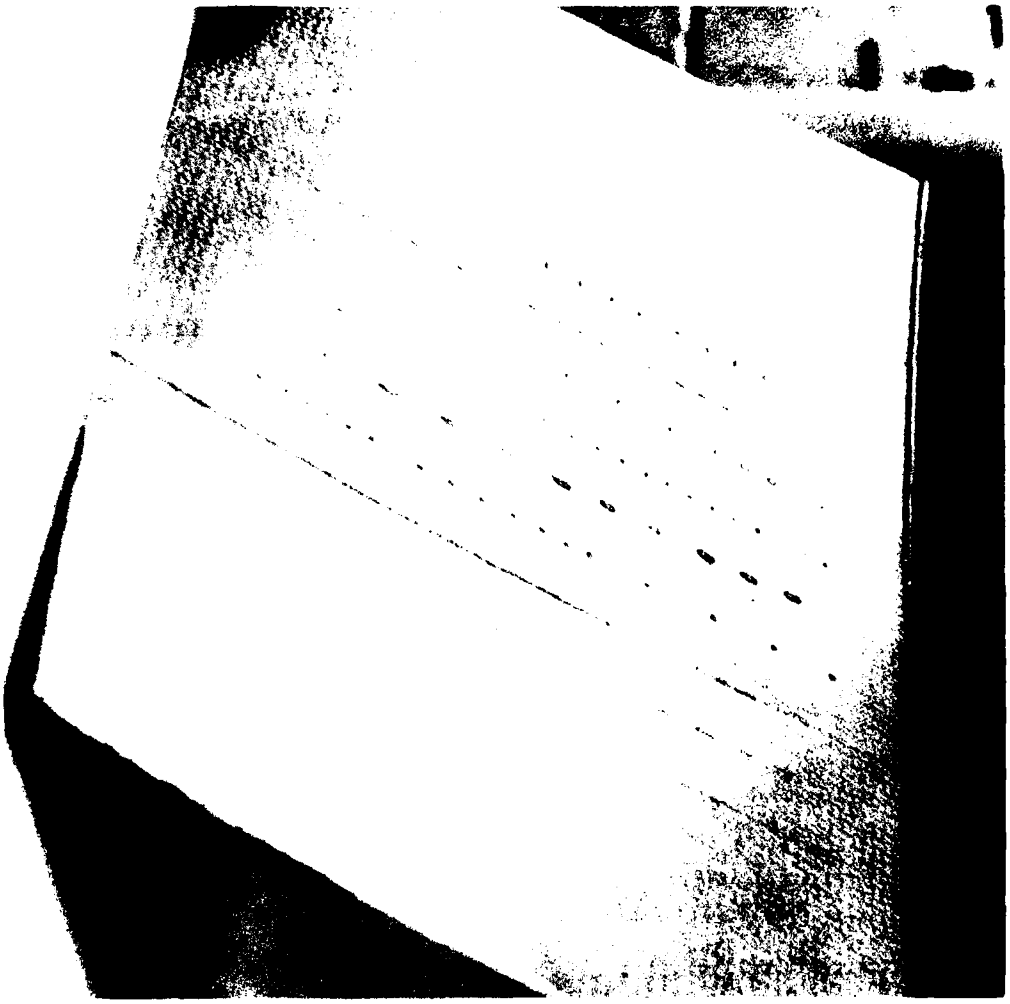


Figure 35. Cutting Lines Made by Scribed Marks in the Bonding Tool

Cost and Weight of Modular Construction

The modular design concept provides a large increase in field level repairability for a small penalty in manufacturing cost and weight. Modular construction will add an estimated 700 inches of repair strips to the CRF. Assuming that the additional material required for the repair strips will be an average of 1½ inches wide and 6 plies thick, modular construction will add an estimated 3½ pounds to the weight of the structure. Since the repair strips are created as an extension of existing material, no additional assembly steps are involved and the effect on manufacturing labor is expected to be negligible. At an average of $20 per pound, the cost of material for the repair strips is an estimated $70. The $50 per pound valuation of weight used for the R&M design option tradeoffs brings the total estimated investment in modular construction to $245.

Sold as spares, the average module might be expected to cost in the neighborhood of $1,700. There would be no cost for modules salvaged from nonrepairable aircraft, a primary source of supply during combat. The cost of other materials required to install a module (metal splice straps, fasteners, etc.) is negligible. The weight increase caused by the addition of metal splice straps and fasteners is estimated to be less than 2 pounds for the average module. A slight degradation in reliability might also be expected owing to the introduction of mechanical fasteners in place of monolithic structure.

MODULE REPLACEMENT SCHEME

Except in cases where the edge of a module is located along an existing mechanical splice in the CRF, installation is accomplished with external and internal splice straps and plates. Figures 36 through 38 illustrate a typical installation. Components mounted on a module or blocking access to either side are removed. The module splice lines are located by sight or feel and the damaged module is removed by cutting through the structure (skin and interior members) along the splice lines. (The cut is made just outside of the line so that the line is removed with the module.) A replacement module is drawn from supply or cannibalized from an unrepairable aircraft and rivet holes are drilled at indicated locations along the edges of the module and the cutout in the fuselage.

Metal tabs are temporarily attached to the module to hold it in place and shims are used to center the module in the cutout. Splice straps are cut from .050 aluminum sheet and rivet holes are drilled in the straps using the hole pattern around the module and the cutout as a guide. Straps on the edges of modules that run longitudinally with the aircraft centerline are extended beyond the corners of the module and the perpendicular station straps are butted against them. The major structural members and load paths are longitudinal and continuity in that direction is therefore more critical. The module is installed with NAS 173805 rivets using a Cherry G-55 hand riveter with a W/H640 pulling head. Washers are used where internal splice straps are absent.

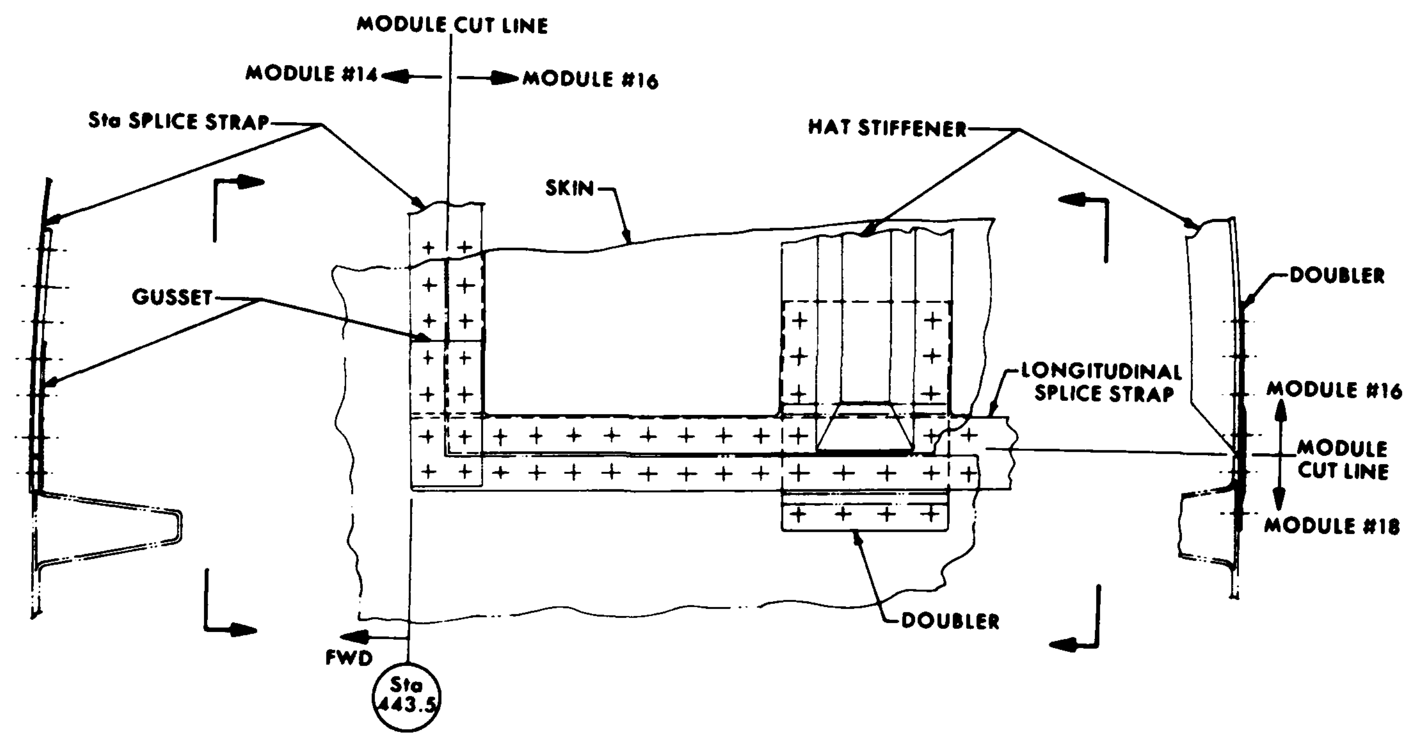


Figure 36. Splice Concept for Inboard Corner of Module 16 at Sta. 443.5 (View Looking Down)

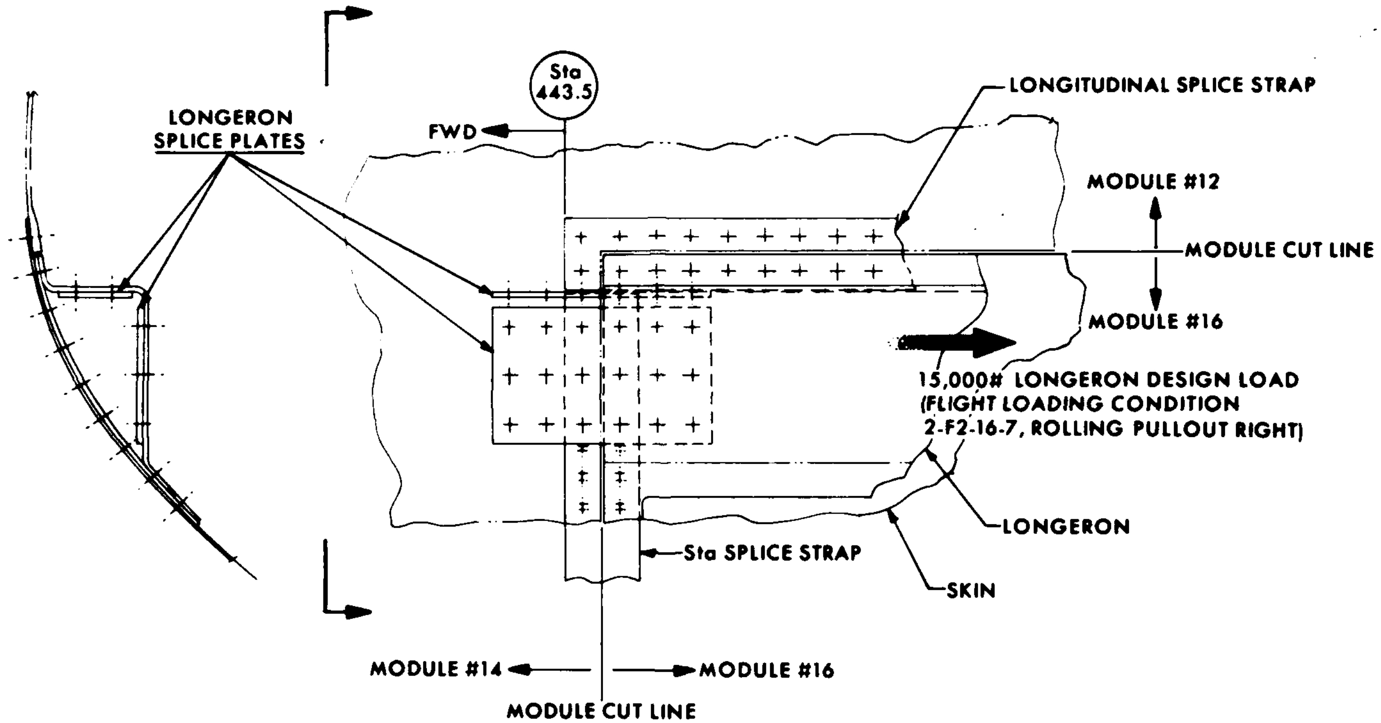


Figure 37. Splice Concept for Outboard Corner of Module 16 at Sta. 443.5 (View Looking Outboard)

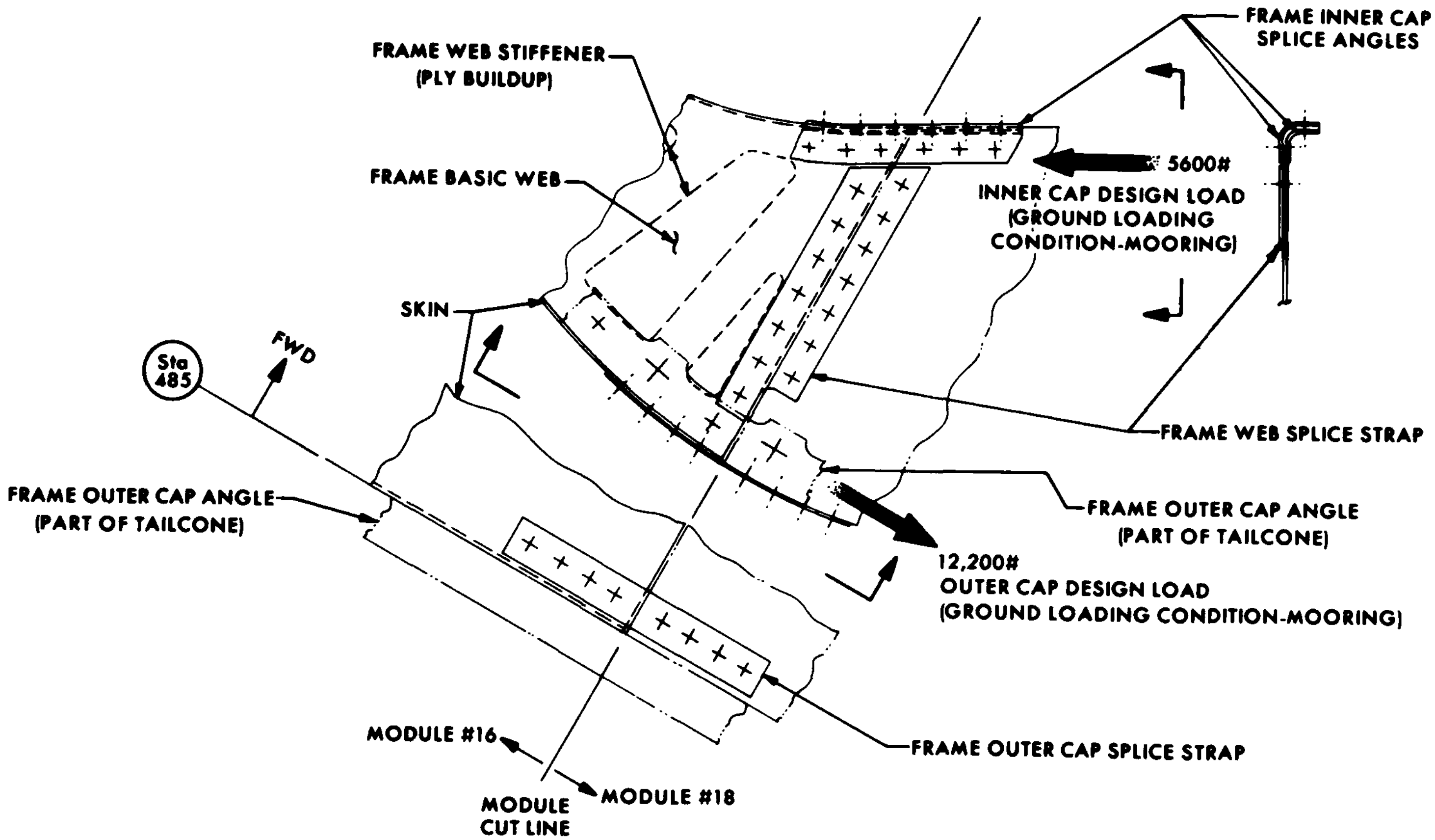


Figure 38. Splice Concept for Inboard Corner of Module 16 at Sta. 485 (View Looking Aft)

Module 16 (Figure 32) is used to illustrate the types of splices that are typically required to install a module. At the forward inboard corner (Figure 36), a simple gusset riveted through to the external straps completes the structural tie at the corner. Aft of the corner, a doubler is shown riveted over the longitudinal strap to aid in carrying the stiffener loads across the cut line. At the forward outboard corner, the longeron is shown spliced (Figure 37). The splice plates may be installed either on the interior of the longeron as shown or on the exterior, depending on access. Figure 38 shows the frame splice at the aft outboard corner of the module. Later in this section of the report, an alternate module configuration that would eliminate splices in the Station 485 frame is discussed.

Weight Penalty

The weight of the aluminum splice straps and fasteners for the average module installation is estimated at 2 pounds. At an average module weight of 10 pounds, this amounts to approximately a 20% increase. Efficiency improves when multiple adjacent modules are installed. For two adjacent modules, the average weight increase is estimated at 15%.

Frame Replacement

The composite frames at Stations 443.5 and 464 are installed with mechanical fasteners along the outer cap and at the Waterline 238 manufacturing splice. The upper and lower frame halves can be replaced independently by drilling out the fasteners. It will be necessary to properly support the aircraft so as to maintain structural alignment during the replacement operation. For localized damage, it will be possible to replace a section of the frame using a mechanical splice similar to that shown in Figure 38.

For any repair of CRF involving the removal of a major piece of structure (module, frame half, etc.), it will be necessary to remove the weight of the aircraft from the tail wheel by supporting the aircraft under the fuselage. In the field, rubber bags of the type used to store water and fuel for the aircraft might be used for this purpose. A possible approach is shown in Figure 39.

ALTERNATE MODULE CONFIGURATION

Figure 32 shows the area of the CRF aft of Station 443.5 composed of 8 modules, 3 in the upper half and 5 in the lower half. Replacement of any of these modules requires that splices be made in the composite frame sections that are integral with the modules. A typical splice is shown in Figure 38. The geometry of the structure allows splices in the frame inner cap and web to be made without difficulty. However, at the frame outer cap, the proximity of the stringers and beams doesn't provide space

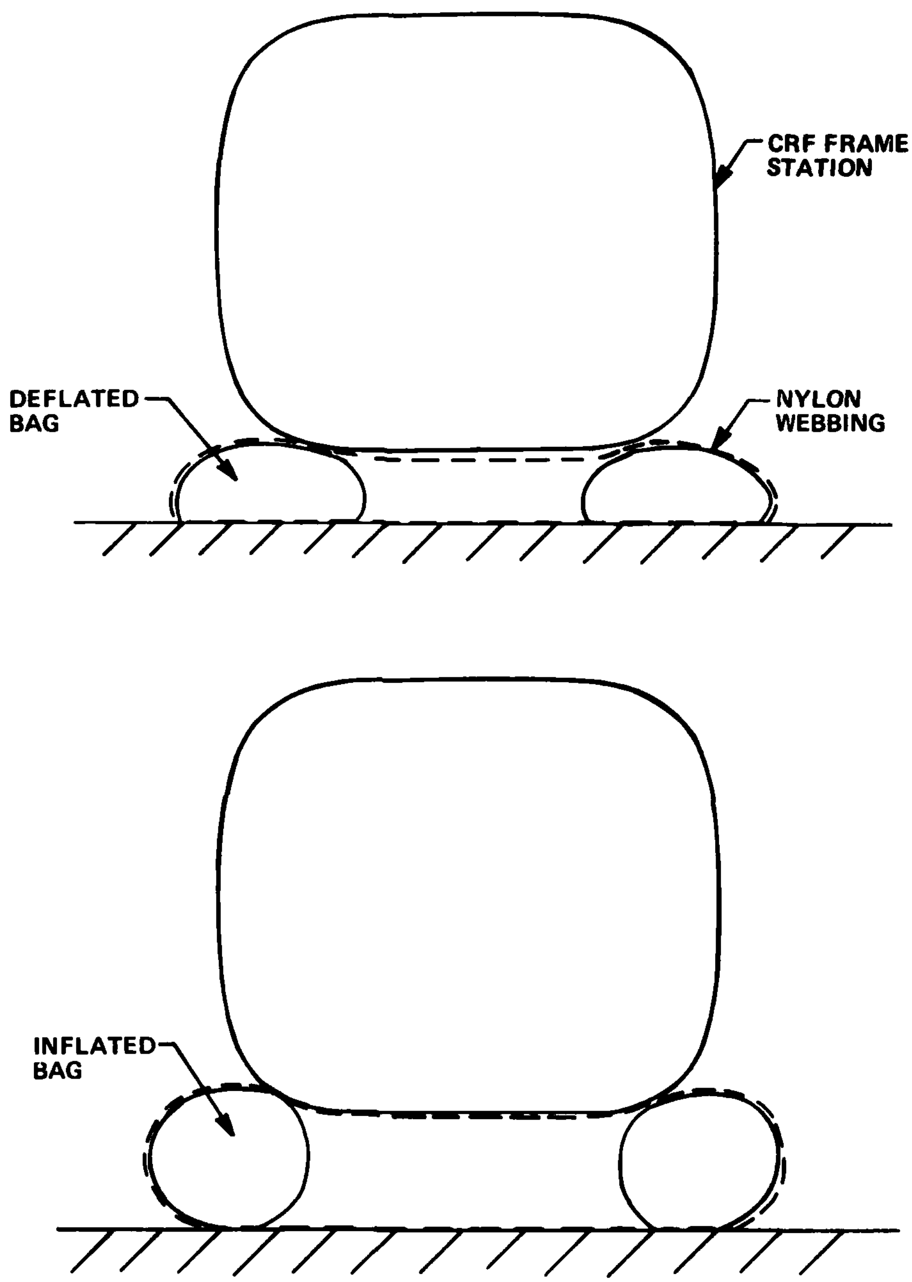


Figure 39. Concept for Supporting the Fuselage During Replacement of a Frame

for an internal splice and it is necessary to use an external strap as shown in Figure 38. The external strap may not be structurally adequate in some locations.

A possible solution to this problem would be to reconfigure the modules aft of Station 443.5. Figure 40 shows a possible arrangement. A module cut line is located at the Station 464 frame and the longitudinal cut lines aft of this frame are eliminated. This divides the entire bay aft of Station 464 into 2 modules, an upper half and a lower half, and eliminates the need for splices in the Station 485 frame. The total number of modules remain the same in the lower half. One additional module may be introduced in the upper half depending on where the station cut is located in the upper deck.

The continuous modules in the aft bay area will be largest of the set. Module 18 in Figure 40 is approximately 54" x 35" x 21" and is estimated to weigh 30.5 pounds. Two men should easily be able to handle and install a module of this size.

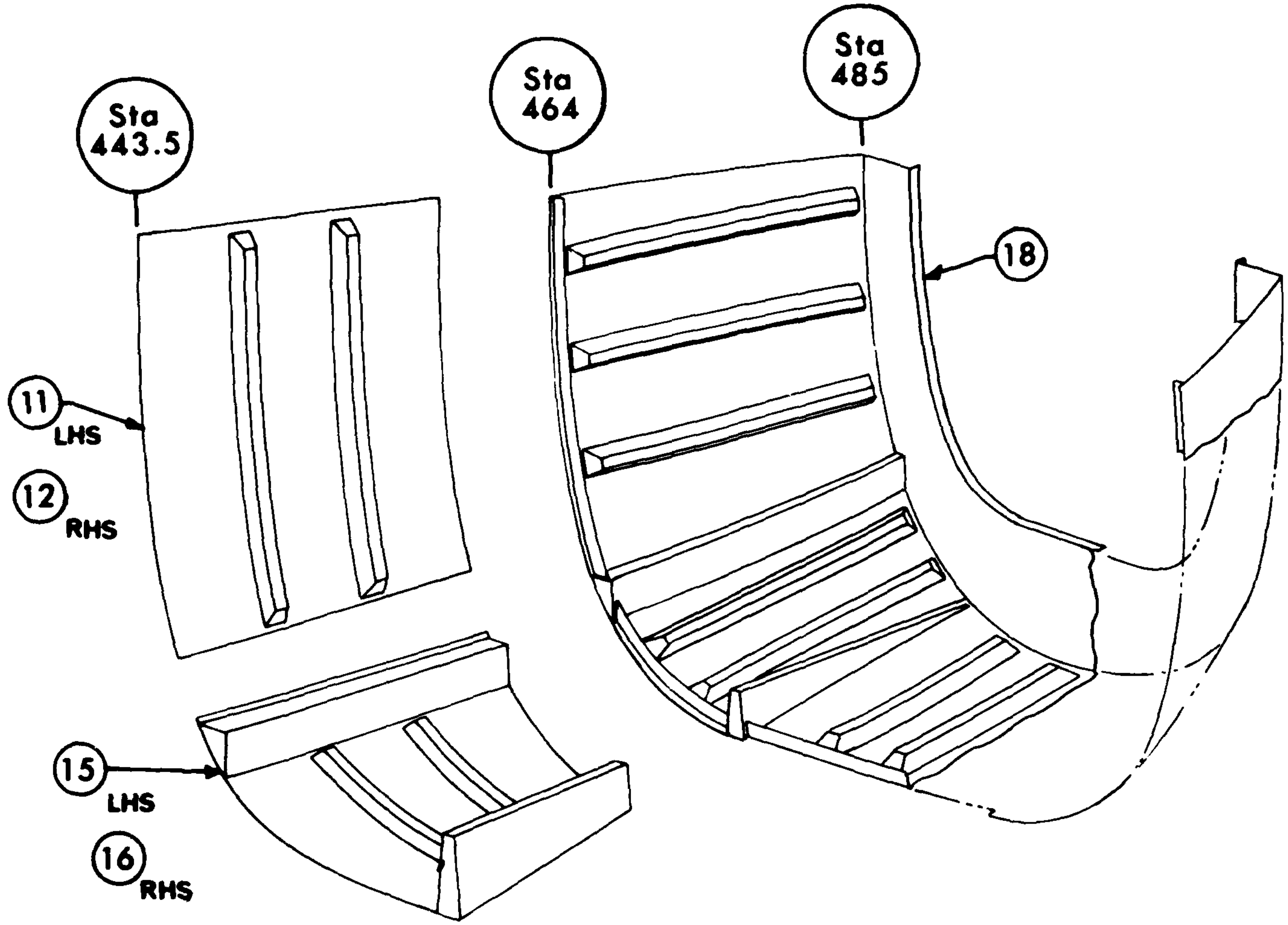


Figure 40. Alternate Module Configuration for the Aft Bay Section, Lower Half

MODULAR DESIGN CONCEPT COST ANALYSIS

ESTIMATED COSTS OF REPAIR ALTERNATIVES

To judge the relative cost-effectiveness of the modular design concept, estimates were made of the cost of alternative methods of large area repair, both for metals and composites. In both cases, the damage was represented as a 12-inch-diameter hole through an area of the structure where multiple framing members intersected. This is believed to be a relatively modest representation of large area damage; repairable damage much more extensive than this is possible. In both cases, it was assumed that cannibalization of replacement parts would be impossible and that a repair would have to be custom-designed.

Custom-Engineered Metal Repair

The assumed damage to the metal structure is shown in Figure 41. Sections of a frame and three stringers are lost. The custom-designed repair is shown in Figure 42. In the first step of the repair, a skin patch is cut, formed to the contour of the fuselage and riveted in place. Filler pieces are then installed, and nested stringer sections, formed and fitted to the original stringers, are riveted in place. Finally, an angle and channel are formed, fitted into the missing frame section, and riveted in place using metal plates as fillers. The repair requires a considerable amount of hand-forming and fitting of parts which may be particularly difficult when heavier gage sections must be nested within existing members. Tools either have to be made or considerable trial-and-error fitting of parts will be necessary.

Sikorsky's Industrial Engineering Department was requested to estimate the man-hour and material costs that would be required to engineer and install a repair of this type under factory conditions. Engineering and shop labor were estimated at 20 man-hours and 105 man-hours respectively. The shop estimate was increased by 50% to reflect the absence of factory tooling and the lower experience level that would prevail at an Army repair facility. Engineering labor was estimated at $35 per hour and mechanics' labor at $25 per hour. The total estimated cost of the repair, including a modest cost for materials, is $5,000.

Custom-Engineered Composite Repair

Figure 43 shows the assumed 12-inch-diameter damage in the identical area of the composite rear fuselage. Sections of the Station 464 frame, a stringer and a longeron are lost. The custom-designed repair is shown in Figure 43. In the first step of the repair, the damaged structure is cut away and trimmed. A backing plate is fabricated and installed and a fiberglass skin patch is prepared, laid up over the hole in the structure and cured. Next, dams are erected around the missing section of the longeron, and foam is sprayed into the section and allowed to cure. The

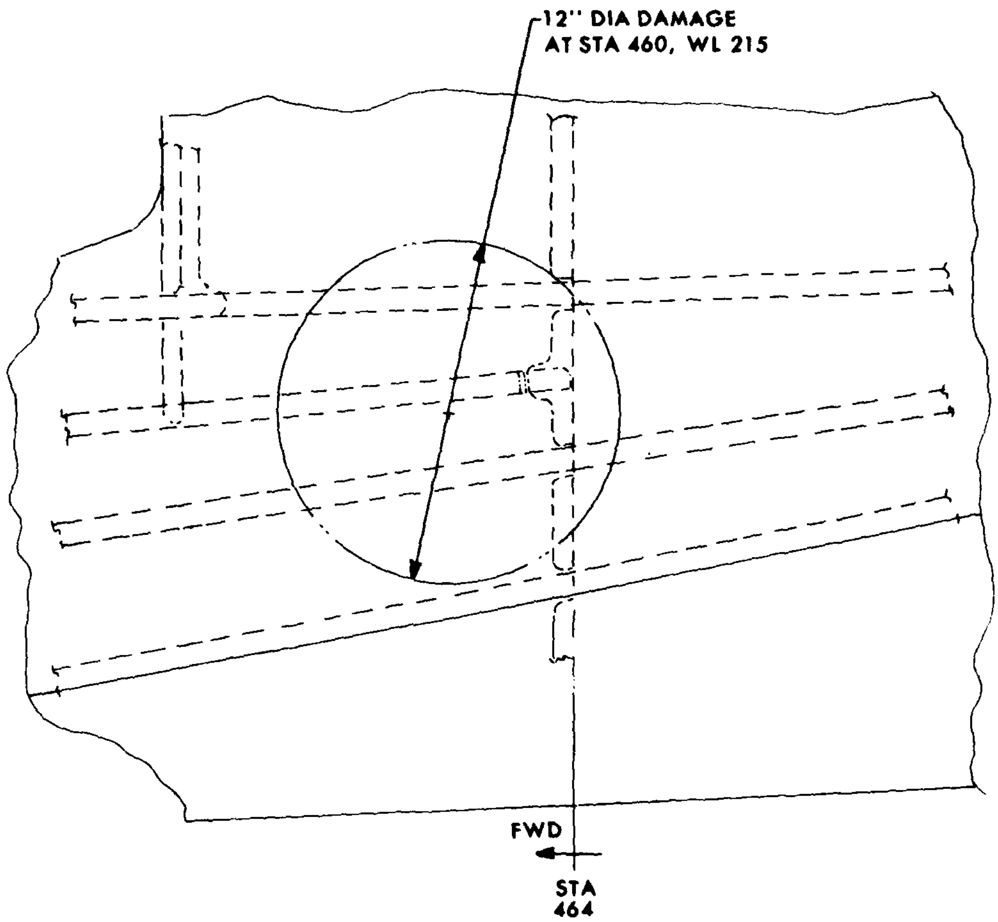


Figure 41. Assumed Large Area Damage to the Rear Fuselage at Sta. 464 (Left Side, View Looking Inboard)

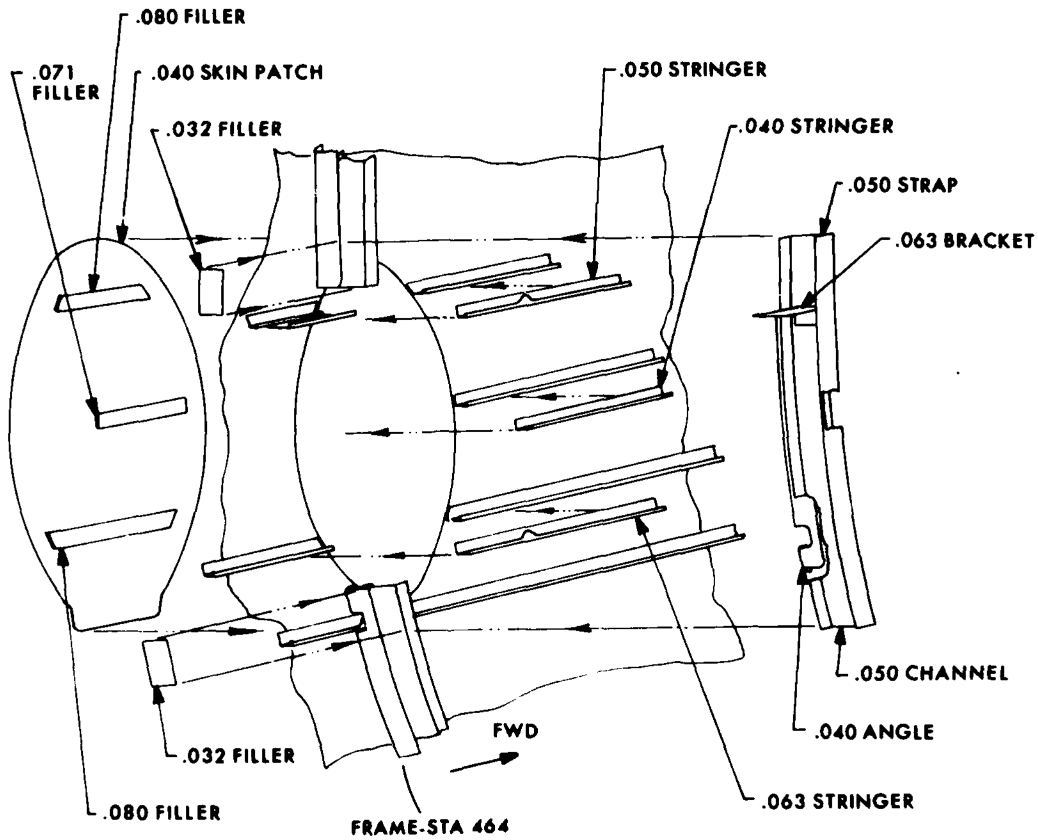
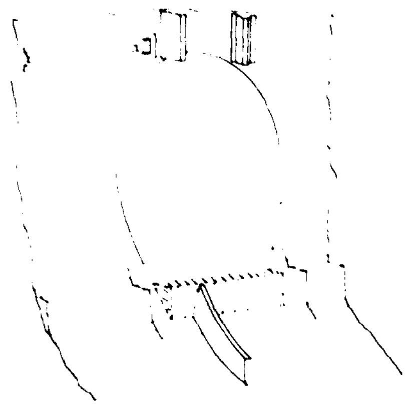
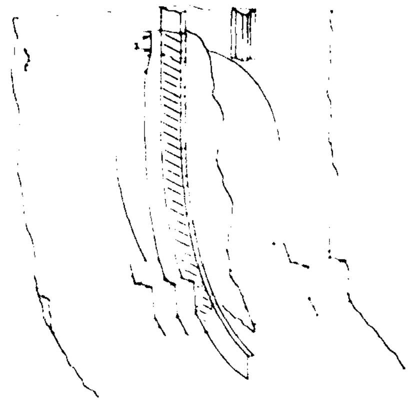


Figure 42. Custom-Engineered Repair of Large Area Damage to the Metal Rear Fuselage at Sta. 464 (Left Side, View Looking Outboard and Forward)

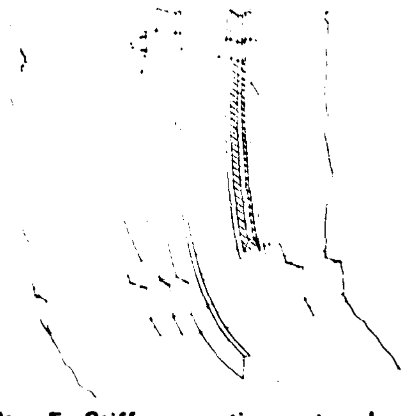
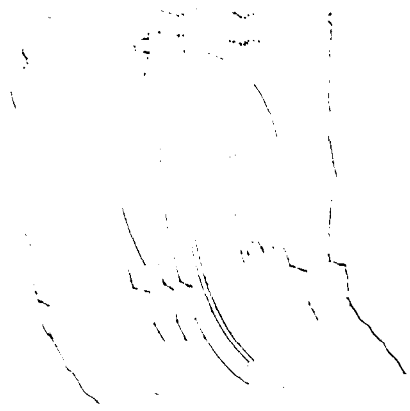
Step 1. Damage cut away and trimmed.

Step 2. Fiberglass skin patch applied (scarfed and tapered lap joints).



Step 3. Longeron section restored with foam. Longeron repaired with fiberglass and graphite tape.

Step 4. Frame section restored with foam. Frame repaired with fiberglass and graphite tape.



Step 5. Stiffener section restored with foam. Stiffener repaired with fiberglass patch.

Completed repair.

Figure 43. Custom-Engineered Repair of Large Area Damage to the Composite Rear Fuselage at Sta. 464 (Left Side, View Looking Outboard)

cured foam is then cut to the shape of the longeron and patch material is prepared, laid up over the foam mandrel and allowed to cure. The frame and stiffener are repaired next using wet patches laid up over fabricated foam mandrels in the same manner that the longeron was repaired. A thorough nondestructive type of inspection is made at each stage of the repair.

Engineering and mechanics' labor for this type of repair were estimated at 30 man-hours and 60 man-hours, respectively, including the 50% factor for factory versus Army facility. Because of the expected difficulty of achieving high quality wet layup repairs of thick primary structure, it was assumed that 25% of repair operations would be found faulty upon inspection and would have to be repeated. Using the same hourly rates as before, the cost of the custom-engineered composite repair, including an estimated cost of $155 for materials, is $3,080. It should be stressed that a composite structures repair of the type described is purely hypothetical. Whereas the mechanical properties of a metal repair can be predicted with high confidence, it is uncertain at this point if wet layup repairs of large area damage to primary composite structure can be achieved with consistently high quality, even under depot conditions. It is uncertain also whether such repairs can be made to restore sufficient strength and stiffness to the structure. Further development work will be needed to prove the feasibility of this type of repair.

Module Replacement

The cost of the composites structure portion of the CRF will be an estimated $23,000. The total cost of the CRF includes metal components, ballistic foam, fuel cell liners and the labor associated with their fabrication and installation. The CRF is divided into 18 modules. Replacement modules might be procured individually or, alternatively, complete upper and lower halves of the CRF could be procured from which modules would be sectioned at the depot and shipped to the field. The latter approach allows multiple adjacent modules to be supplied as a unit if needed, but probably invites surpluses of the less frequently used modules.

For the purpose of comparing costs with custom-engineered repairs, it was assumed that the average module will cost 1/18th of the cost of the composite structure, plus 35% for handling and packaging - a total of approximately $1,725. (There will be no cost for a module salvaged from an unrepairable aircraft in combat.) Based on the procedure previously outlined, it was estimated that the average module replacement will require 30 man-hours. No engineering is required at the time of repair. The total estimated cost of an average module replacement, including an estimated $210 for splice straps and fasteners, is $2,385. (Field labor was priced at $15 per man-hour; the higher cost of metal parts for the module replacement versus the custom-engineered metal repair is due to the requirement to use expensive blind fasteners whose material is compatible with the composites.) The module replacement is definitely within the capability of field maintenance units, so that there will be no additional costs of dispatching repair teams from the depot or moving the aircraft to a depot as would be required for a custom-engineered repair.

A summary of the estimated costs for the metal and composite custom-engineered repairs and the composite module replacement is given below.

	Labor	Materials	Total	
Custom-Engineered Metal Repair	$4,985	$ 15	$5,000	Plus the cost of dispatching a repair team from the depot or transporting the aircraft to the depot
Custom-Engineered Composite Repair	2,925	155	3,080	
Module Replacement	450	1,935	2,385	Plus the cost of stocking modules and transporting them to the field.

MODULAR DESIGN CONCEPT COST ANALYSIS

Economics of Peacetime Use of Modules

An analysis was made of economics of a modular design approach for the CRF. Four maintenance policies were evaluated (Table 1). Under Policy 1, the CRF is designed without replaceable modules. Standard repairs are accomplished on the aircraft in the field. When damage exceeds the

TABLE 1. CRF MAINTENANCE POLICIES

Type of Maintenance	Level of Maintenance			
	Policy 1	Policy 2	Policy 3	Policy 4
Standard Repairs	Field	Field	Field	Field
Module Replacement	N/A	Field	Field	Depot
Module Stocking	N/A	Field	Depot	Depot
Custom-Engineered Repairs	Depot	N/A	N/A	N/A
CRF Section Replacement	Depot	N/A	N/A	N/A

limits of a standard field repair, the aircraft is shipped to a depot where a custom-engineered repair is installed or, in the case of very extensive damage, an entire half-section of the CRF is replaced.

Under Policies 2 and 3, the CRF is designed with replaceable modules. The same types of standard repairs are accomplished in the field as would be accomplished under the first policy. When damage exceeds the limits of a standard repair, the damaged module is removed and replaced in the field.

Policy 2 differs from Policy 3 with respect to the location at which replacement modules are stocked. Under Policy 2, modules are stocked locally and under Policy 3 at central warehouses. It was assumed in the case of Policy 2 that replacement modules would be located at 20 field sites, each supporting an average of 50 aircraft, and that only one each of the high-usage modules (6 of 18) would be stocked at each site. A replacement for any of the 12 low-usage modules would be obtained from the supply pipeline when needed. In the case of Policy 3, it was assumed that spare modules would be located at 3 central warehouses each supporting 1/3 of the fleet, and that each warehouse would maintain an insurance stock of 18 modules. The inventory at each warehouse was assumed to consist of 2 each of the 6 high-usage modules plus 1 each of 6 low-usage modules (an average of 1 1/2 each of the 12 low-usage modules being distributed among the 3 warehouses). As in the case of Policy 2, other demands would be satisfied from pipeline spares.

Policy 4 also adopts the modular design concept except the replacement of modules is accomplished at depot rather than in the field. When damage to the CRF exceeds the limits of a standard field repair, the entire aircraft is sent to a depot where the module containing the damage is replaced. This eliminates shipping and stocking modules for field replacement, but incurs the disadvantage of having to transport the entire aircraft rather than an individual module. As is the case with Policy 1, inability to repair in the field causes the aircraft to be removed from the operator's custody which tends to greatly increase aircraft out-of-commission time.

The economic analysis of these four concepts was accomplished with a computer model developed by Sikorsky for analysis of reliability improvement warranties (RIW). The RIW model is a simplified life-cycle cost model that projects the life-cycle cost of aircraft components based on defined maintenance policies and known or predicted values for R&M and logistics. R&M inputs to the model include the mean-time-between-failure (MTBF), mean-time-to-repair (MTTR) at each maintenance level, fraction of failures repairable at each maintenance level, and fraction of failures beyond repair (scrap). Logistics-related inputs include the number of sites to be supported, number of spares per site, spares replenishment time, and the costs of support equipment, training and labor. The projected service life and utilization of the aircraft are also stipulated. Unit acquisition and initial spares costs are included in the model, while RDT&E costs, which are considered sunk costs, are not. Table 2 lists the significant variables used in the cost analysis.

TABLE 2. KEY COST VARIABLES FOR MODULAR DESIGN CONCEPT ECONOMIC ANALYSIS

Variable	Value
Number of Aircraft	1,000
Life-Cycle (Years)	20
Flight-Hours/Aircraft/Year	325
CRF Composite Structure Cost ($)	23,000
Average Module Cost ($)	1,725
Mean-Flight-Hours Between Damage	50
Percent of Damage Beyond Field Repair Limits	1
Percent of (Non-Modular) CRF Damage Beyond Economical Repair at Depot	10
Percent of Removals Scrapped	
CRF Half-Section	100
Module	100
Average Man-Hours	
Field Repair	4
Module Replacement	30
Custom-Engineered Repair (Eng. and Shop)	105
CRF Replacement (upper or lower half)	320
Average Labor Cost ($)	
Field	15
Depot Shop	25
Depot Engineering	35
Average Material Cost ($)	
Field Repair	30
Custom-Engineered Repair	155
Average Transportation Cost (Round Trip)	
Aircraft	2,500
Module	200
Average Spares Replacement Time (Months)	6

On the basis of the factors listed in Table 2, the RIW model calculated life-cycle maintenance cost savings for the three modular maintenance policies over the non-modular maintenance policy (Policy 1).

	Fleet Life-Cycle Cost Savings
Policy 2	\$ 7.5 Million
Policy 3	\$ 7.7 Million
Policy 4	\$ 4.4 Million

The sensitivity to changes in key cost variables was tested (Figure 44). As discussed earlier in the report,in order to properly reflect the time value of money, a return on investment (ROI) of 5 was used as the basis for selecting cost-effective R&M design options for the CRF. The same rationale applies to the modular design concept, since an investment in the form of increased aircraft weight must be borne immediately to realize a savings in maintenance costs over future years.

The increased acquisition cost associated with modular design of the CRF was estimated earlier at \$245 per aircraft. At \$245 per aircraft, a rate of return of 5 to 1 would require a maintenance cost savings of \$1.2 million over the life of the fleet. Reference to Figure 44 shows that based on this preliminary analysis, modular design easily provides the required return on investment for peacetime operation of the aircraft. The economic analysis has considered only the direct life-cycle costs of maintaining a modular rear fuselage structure versus a conventional one. Not included in the comparison is the potential impact of the differing policies on aircraft availability.

In Figure 44 it is shown, for example, that over the service life of the fleet, stocking insurance modules at the local level (Policy 2) is more expensive than stocking all modules at the depot (Policy 3). This conclusion ignores the improved responsiveness and reduced downtime that a local source of supply would permit. It may be assumed, for example, that all damage to the CRF requiring replacement of a module will render the aircraft unflyable. If an average of 30 days is required to requisition and receive a module from the depot, and a local source of supply would avoid 75% of these transactions, at an MTBR of 5,000 hours, the equivalent of 80 aircraft years of downtime could be saved over the life of the fleet. This has a theoretical economic value of four complete aircraft or approximately \$6 million - a large savings over the cost of local insurance stocks of modules. Similar reasoning could be applied to show that, because of the much longer out-of-commission time associated with the movement of aircraft to depots, Policy 4 is much more expensive relative to Policies 2 and 3 than the direct costs alone would indicate.

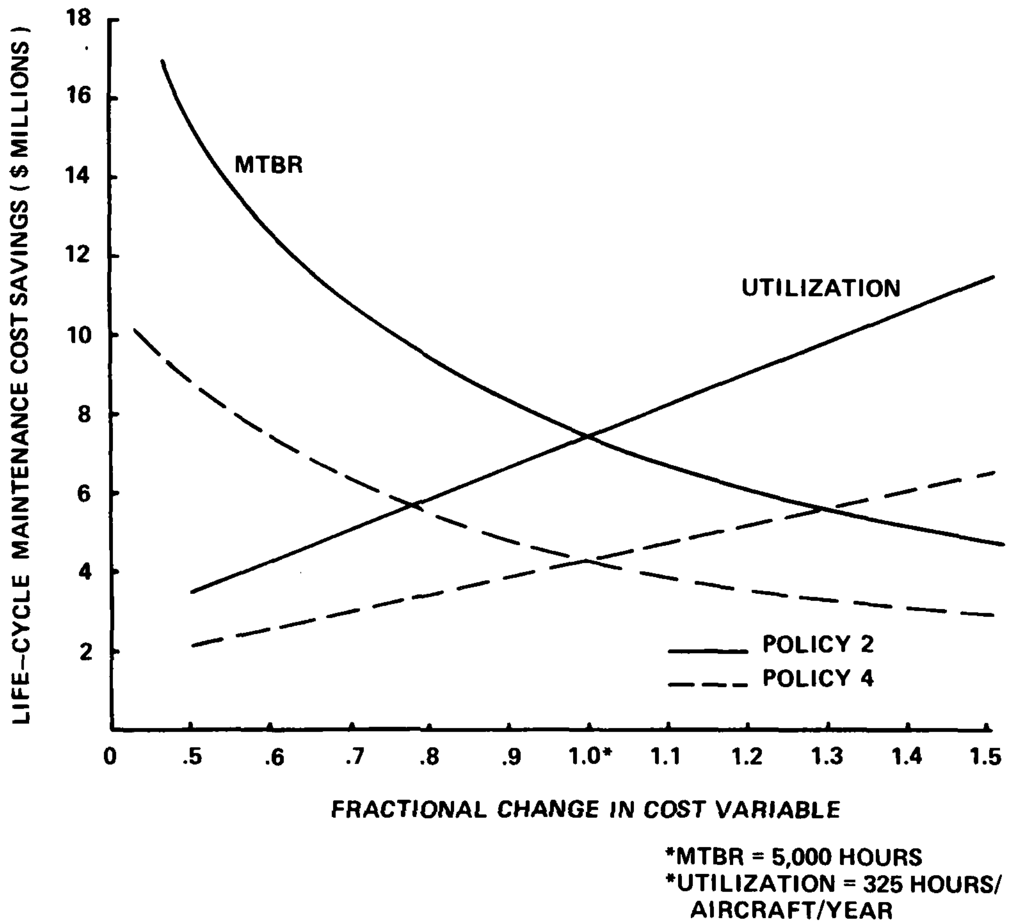


Figure 44. Sensitivity of Modular Maintenance Life-Cycle Cost Savings to Variations in Aircraft Utilization and Module Replacement Rate

The economic value of improved aircraft readiness is, however, largely theoretical, particularly in peacetime. In peacetime, utilization of aircraft is low, and average availability is already much greater than is needed to perform the mission. The contention that a fractional percentage improvement in availability will allow the Army to operate with a smaller fleet of aircraft and thus actually realize a savings would be difficult to support.

ECONOMICS OF COMBAT USE OF MODULES

Combat presents the greatest risk of sustaining major structural damage to the CRF. The explosive projectiles especially have the potential for causing structural damage too large to repair with fabricated patches or cannibalized parts. In a short, intense conflict, returning the aircraft for repair at a division or corps level maintenance base or a depot facility (if such facilities exist in the combat theater) may be tantamount to permanent loss of the aircraft. The time required to transport the aircraft to a depot, custom-engineer and install a repair, and return the aircraft to the field commander may exceed the duration of the conflict. Even in a protracted conflict where depot-level repair may be feasible, it will be necessary to replace aircraft that are removed from service for prolonged periods of time. The value of the modular design concept in combat can be measured in terms of the reduced numbers of pool aircraft that the more rapid repair turnaround would theoretically permit.

At high levels of utilization, small reductions in maintenance downtime can significantly affect the availability of aircraft for combat. Figure 45 shows the average reduction in repair downtime that will provide a 10:1 return on the investment in modular construction of the CRF, assuming that the improved availability allows a proportional reduction in the total numbers of aircraft procured. For example, at 100 sorties per month and an average of 2,000 sorties between combat damage to the CRF, the break-even point is about 24 downtime hours. That is, an average reduction of 24 hours in repair downtime makes the module replacement concept cost-effective. Module replacement should easily require this many fewer hours to perform than a major structural repair, especially if the aircraft has to be transported to a rear echelon maintenance base for the repair. At higher sortie rates and/or higher combat damage rates, a break-even point is reached with even smaller downtime savings.

Actually, although modular construction is shown to be cost-effective for any reasonable set of assumptions, justification of the concept on a cost basis may be academic. The alternative to module replacement for repair of large area combat damage to the CRF may require skills and resources that are unavailable in the combat environment. The time required to perform major structural repairs in a fast-moving conflict may also be prohibitive. If the alternative to replacing a module is to abandon or scrap the aircraft, very few replacements will justify the investment.

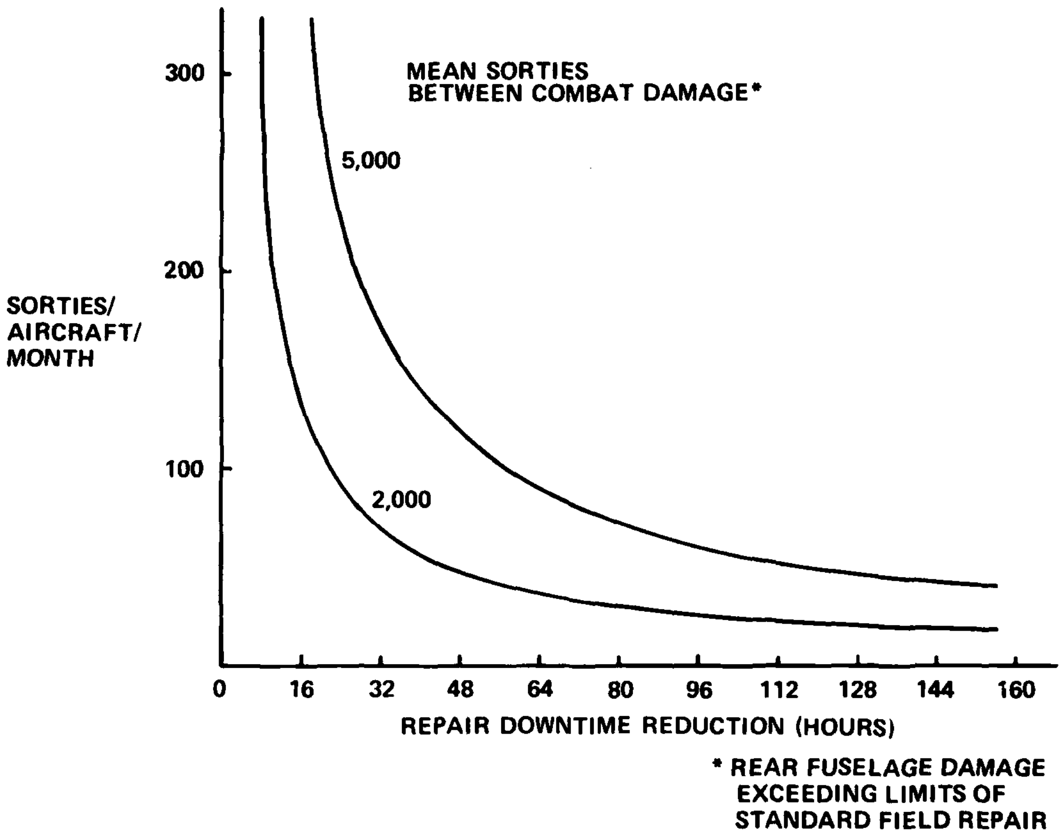


Figure 45. Break-Even Curves for Investment in Modular Design for Combat Repair

SERVICEABILITY CRITERIA: REPAIR AND INSPECTION

REPAIRABILITY ASSESSMENT

The repairability of an airframe structure can be expressed in terms of the types of repair allowed, the number of repairs allowed, and the level of maintenance at which they are accomplished. Repairability is influenced by four major design factors:

1. Type of Structure
2. Design Loading Condition
3. Margin of Safety
4. Repair Access

An analysis was conducted to assess the potential repairability of the CRF with respect to these four design factors. A mapping procedure was used to determine the variations in each attribute throughout the structure. The mapping was done entirely on the basis of relative areas measured in the four cardinal views of the structure - top, bottom and two sides. No weighting factors were applied to account for differences in probability of damage between views or areas. That level of analytical detail would be of value at the detail design stage.

Type of Structure

The CRF was defined in terms of seven generic types of structure. All structural elements of a generic type were considered to be similar in materials, form and construction for the purpose of assessing repairability. Variations in laminate layup, orientation and thicknesses were not addressed, since these affect primarily detail aspects of the repair rather than the nature of the repair. The generic types of structure were found to be apportioned by presented area as shown at the top of the following page.

	Type of Structure	Percent of Presented Area
1.	Skin	32.1
2.	Honeycomb Panels	14.8
3.	Skin and Stiffener	9.6
4.	Skin and Beam/Frame	10.2
5.	Skin and Intersection of Framing Members	6.8
6.	Structural Fittings	1.1
7.	Sheet Metal	25.4
		100.0

Design Loading Condition

Primary airframe structure is designed to one or more flight loading conditions, ground loading conditions and crash conditions. For the CRF, the following conditions size the structure:

Flight

Rolling pullout left

Rolling pullout right

Yaw recovery

8000 lb cargo hook load

Ground

Jacking

Mooring

Hard Landing

Crash - combinations of longitudinal, vertical and lateral loads

Separately applied

Simultaneously applied

Individual structural members may be designed to one critical loading condition, although more typically several conditions size the member. Maps of the CRF were drawn to portray the type of loading (tension, compression, bending, shear) on each member or section of a member at the critical design condition. Where more than one load was instrumental in sizing a member, the most critical was used in the analysis. (In general, members which experience internal tensile stresses due to the applied loads will be the most difficult to repair.) The type of loading on structural members at the critical design conditions was found to be apportioned as follows:

Type of Loading*	Percent of Presented Area
Tension	11.9
Compression	6.5
Bending	1.4
Shear	32.1
Undifferentiated	48.1
	100.0

* At the critical design condition

Structures were left undifferentiated with respect to load in areas where that determination was not considered to have an effect on the criticality of repair. All sheet metal and honeycomb damage was assumed to be repairable with currently available techniques. Damage to the intersections of framing members was assumed to require custom-engineered repair for which loading conditions would have to be determined at the time of repair. Damage to structural fittings was assumed to be nonrepairable under all loading conditions, except by replacement of a module or half section of the CRF.

Loading Condition and Safety Margin

In addition to the type of loading in a member, repairability is influenced by safety margin. (Safety margin is defined as $F/\sigma-1$, where F is the appropriate material allowable stress and, for flight-loaded structures, σ is the stress due to the application of the design ultimate load (1.5 times the largest load expected in service)). From a design standpoint, the most efficient structures are those with the lowest margins. They are also the most difficult to repair, since repair efficiency must approach 100%.

Based upon past designs, estimates were made of the safety margins to which structural members sized by critical flight loading conditions would be designed. For this preliminary assessment, margins associated with members sized by ground and crash loading conditions were ignored on the presumption that they represent less serious safety considerations. For a formal analysis made at the detailed design stage, safety margins throughout the structure would be considered.

Maps were drawn to portray the estimated safety margins that would be present in the members designed to flight loading conditions. Margins were estimated as $<25\%$, 25% - 50%, and $>50\%$. The following apportionment by presented area was obtained:

Loading Condition	Safety Margin	Percent of Presented Area
Flight	$<.25$	38.6
Flight	.25 - .50	27.0
Flight	$>.50$	2.7
Ground	-	4.9
Crash	-	26.7
		100.0

Although desirable from the standpoint of structural efficiency, for practical purposes, structural members cannot be designed to provide uniformly low margins for their entire length. To do so would require varying material thicknesses and structural sections with variations in load intensity. This would complicate manufacturing and add substantially to costs. Therefore, many structural members have varying design margins over their length, making repair less critical in some areas than in others.

Repair Access

Structures backed by ballistic foam were classified as having one-sided access for repair. All other structures were considered to have two-sided access. The measured percentages of presented area by type of access are as follows:

Type of Access	Percent of Presented Area
One-Sided	33.6
Two-Sided	66.4
	100.0

Combinations of Variables

The above four variables were incorporated into three sets of maps:

1. Type of load in member

2. Design loading condition and margin of safety

3. Type of structure and repair access

Figure 46 is a typical example. The maps were examined and all observed combinations of the four variables were listed, a typical combination being: (skin and beam) loaded in (tension) at the critical design condition with a (25%-50%) safety margin and (one-sided) repair access. Some combinations, such as skins loaded in bending at the critical design condition, obviously could not occur. Other combinations could be grouped for the purpose of repairability assessment, e.g., all combinations involving sheet metal structure. The total presented area for each observed combination of parameters was measured and recorded.

Assessment of Small Area Damage Repairability

For each observed combination of structure, loading condition, design margin and repair access, a determination was made relative to the type of repair that would be required for small area damage. Small area damage was defined as a hole, crack, delamination, etc., three inches or smaller in diameter (7 $in.^2$). Three types of repair were considered:

Standard Field Repair. A repair made by the mechanic in the field using standard tools, kits and materials in accordance with instructions provided in the maintenance manual. Standard repairs will differ for small area and large area repair. For large area repair, pressure curing will be mandatory.

Custom-Engineered Repair. A repair that requires engineering definition and authority. Custom-engineered repairs will not be performed in the field without depot or factory assistance.

Module Replacement. Replacement of a structural module in accordance with procedures described earlier in this report. Module replacement can be accomplished by field personnel using instructions provided in the maintenance manuals.

In the case of standard field repairs and custom-engineered repairs, the degree of repairability was based on a known capability or the assumption that an adequate repair capability can be developed. Further work will be needed to actually develop all of the repair capability that was assumed in this analysis. Table 3 is a sample of observed design variable combinations used in the analysis of repairability for small area damage.

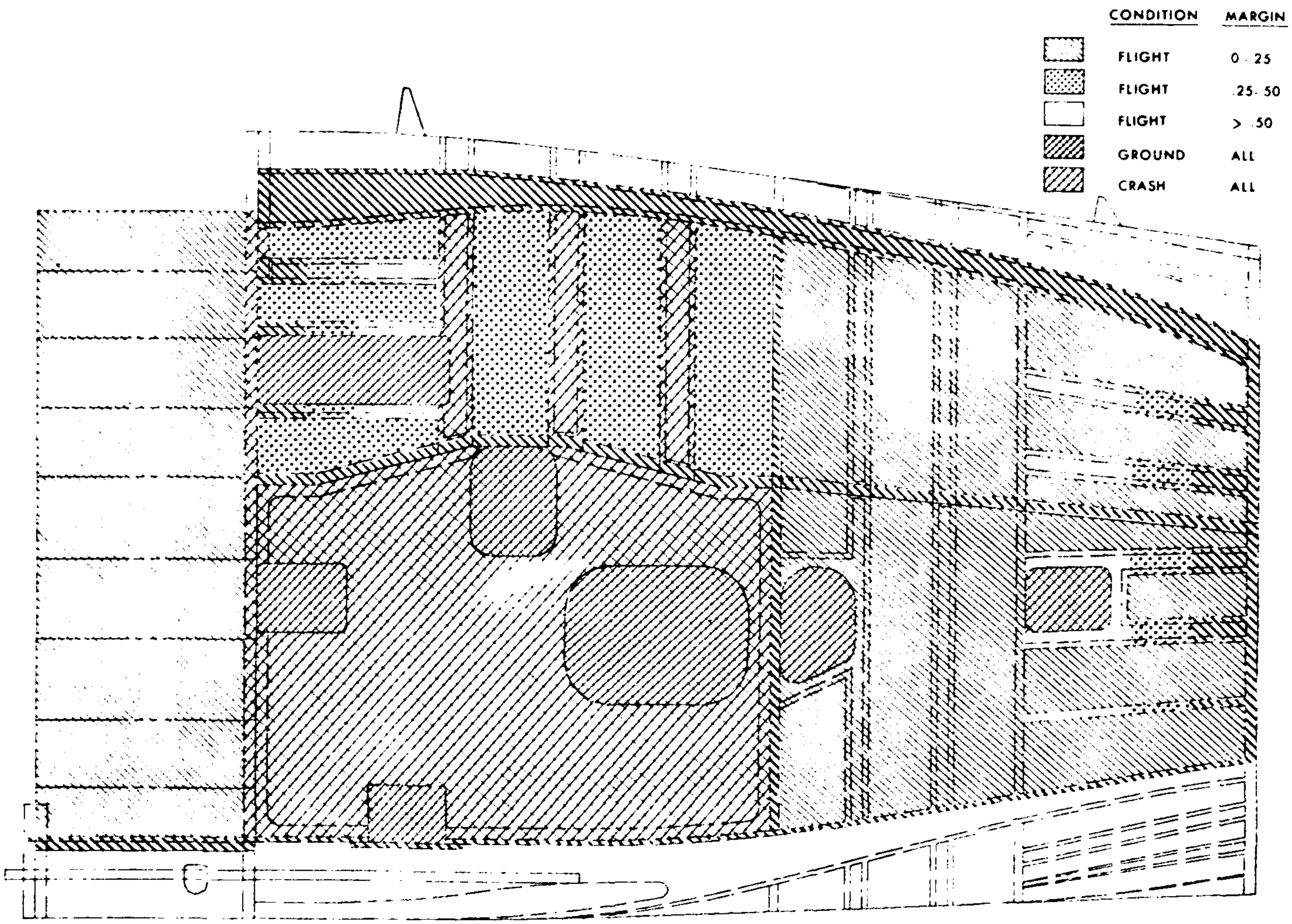


Figure 46. Typical Repairability Criteria Mapping - Flight Condition and Safety Margin, Left Side

TABLE 3. SAMPLE CRF REPAIRABILITY ANALYSIS																							
Comb. No.	Type of Structure							Design Loading Cond. & Margin					Type Load				Sides Access		Area Inch^2	Area %	Repair		
	Sheet Metal	Honey-comb	Skin	Skin/ String	Skin/ Beam	Skin/ Inters	Fit-ting	Flight <.25	Flight .25-.50	Flight >.50	Ground	Crash	Tens.	Comp.	Shear	Bend	1	2			Std.	Cust.	Mod.
1	X							(All Combinations)						(All)				X	6320	25.4	X		
2		X										X		(All)			X		3676	14.8	X		
3			X					X							X		X		290	1.2	X		
5			X						X						X		X		1600	6.4	X		
8			X								X				X			X	70	0.3	X		
12				X				X					X					X	668	2.7	X		
18				X					X					X				X	126	0.5	X		
22				X						X				X				X	195	0.8	X		
27				X								X	X				X		480	1.9	X		
31					X			X						X			X		232	0.9	X		
37					X				X				X				X		190	0.8	X		
49					X							X	X				X		60	0.2	X		
55						X		(All Combinations)						(All)			(All)		1704	6.8		X	X
56							X	X						(All)				X	272	1.1			X

It was determined that standard field repairs can be used for all small area damage except damage affecting structural fittings and the intersections of framing members. This represents an estimated 92% of the CRF by area.

Custom-engineered repair or replacement of a module (if modular design is adopted) can be used to repair small area damage affecting the intersections of framing members. This represents an additional 7% of the structure by area. For small area damage, it was determined that structural fittings are the only components that would not be repairable with standard or custom-engineered repairs, an estimated 1% of the structural area. Repair of damage of these fittings would require replacement of a module, or in the absence of a modular design, replacement of the entire upper or lower half of the CRF.

With modular design, 100% of small area damage to the CRF is repairable in the field. Without modular design, an estimated 92% of the structural area is repairable (99% if custom-engineered repairs are done in the field). It is important to emphasize that these estimates are based strictly on an assessment of the structure by presented areas. No weighting has been done to reflect the relative probability of damage to structural areas. If the required information was available and relative damage rates could be assessed, it is likely that the percentage of small area damage repairable with standard field repairs would significantly exceed 99%, since the elements of structure not amenable to standard repair tend to be relatively protected from damage.

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Analysis of Large Area Damage Repairability

An analysis was conducted to assess the repairability of large area damage to the CRF. Large area damage was defined as a hole 12 inches in diameter (although significantly larger damage could be experienced). Repairability was assessed with respect to the same four variables, i.e., structural element, loading condition, design margin and repair access. For large area damage, it was necessary to adjust the measured areas of the structure possessing each combination of the four design variables. Whereas small area damage could be classified and measured as damage at discrete points, large area damage had to be classified according to the most serious damage in a zone. For example, it was judged that 12-inch-diameter damage will always involve at least one framing member so that variable combinations involving skin only could be eliminated from consideration. Similarly, any large area damage within 8 inches of Station 485 was assumed to involve a structural fitting.

Repairability of large area structural damage was classified with respect to the same three generic types of repair: standard field repair, custom-engineered repair and module replacement, recognizing however that the nature of the repairs within the first two categories would differ substantially for large area versus small area damage.

It was estimated that for 50% of presented area of the CRF, large area damage can be repaired with standard field repairs. This area represents primarily sheet metal structure and honeycomb panels. It also includes skin/stringer and skin/beam structure which is not loaded in tension or bending at the critical design condition and/or has a moderate to large safety margin. The remainder of the structure, with the exception of damage involving structural fittings, can be repaired with custom-engineered repairs or, if the design is modular, with replacement of modules. Added to the standard field repairs, this encompasses 96% of the structural area. Damage involving structural fittings, the remaining 4% of the presented area, can only be repaired via replacement of a module or replacement of the upper or lower half of the CRF.

Overall Repairability Assessment

Figure 47 shows the assessed repairability of the CRF for both small area and large area damage. To emphasize once again, the percentages are based strictly on presented areas and do not account for the probability of damage in each area.

In Figure 48, the level of repairability is shown as a function of the percentage of large area damage to the total damage sustained. Again, the illustration is based on the relative repairability by area and does not account for the probability of damage to various areas. (This factor could be introduced in a detailed analysis of the completed design.) Figure 48 illustrates, however, that the ability to replace modules becomes extremely valuable at high rates of large area damage such as would be experienced in combat.

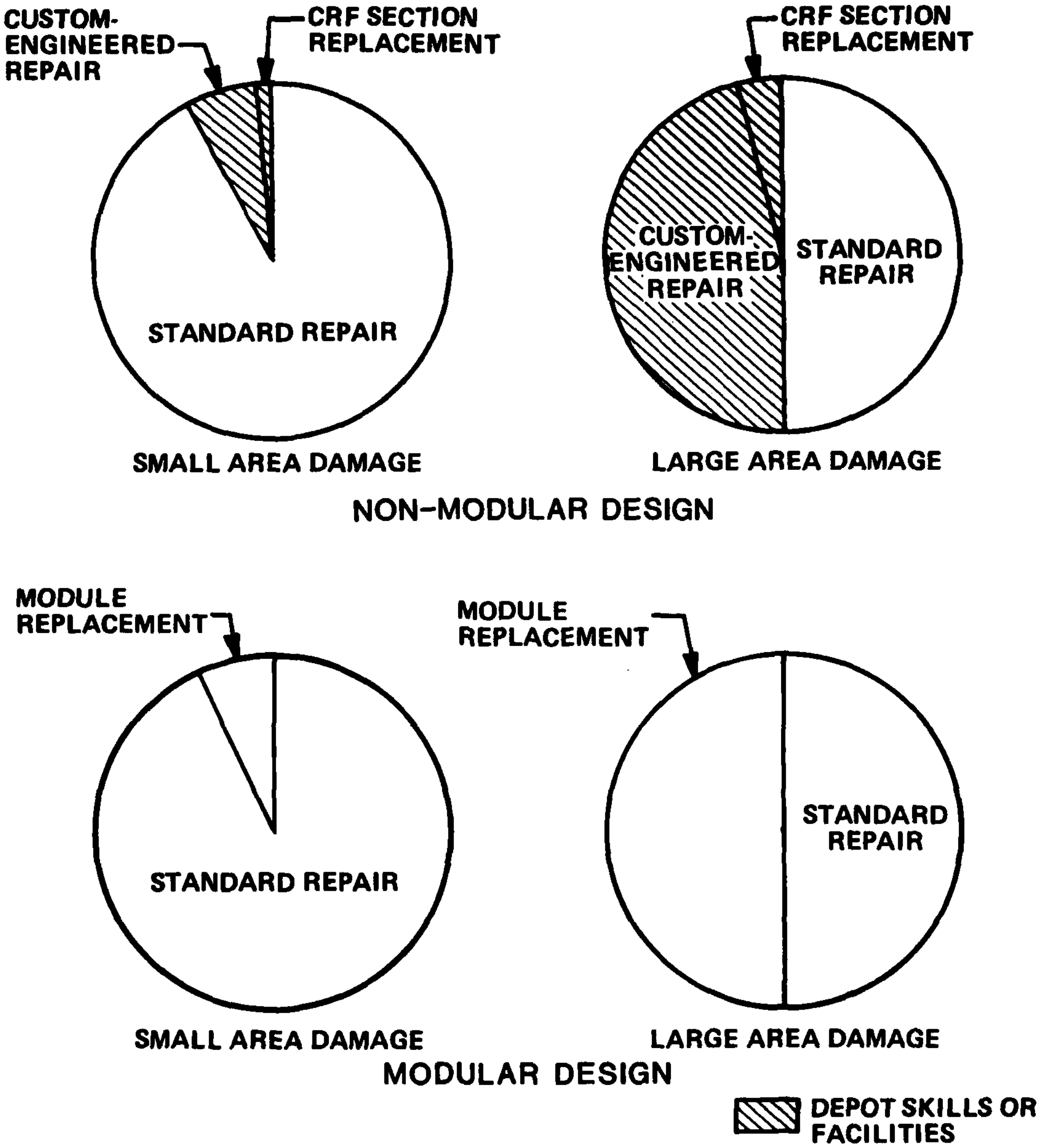

Figure 47. Estimated CRF Repairability for Presented Areas of the Structure

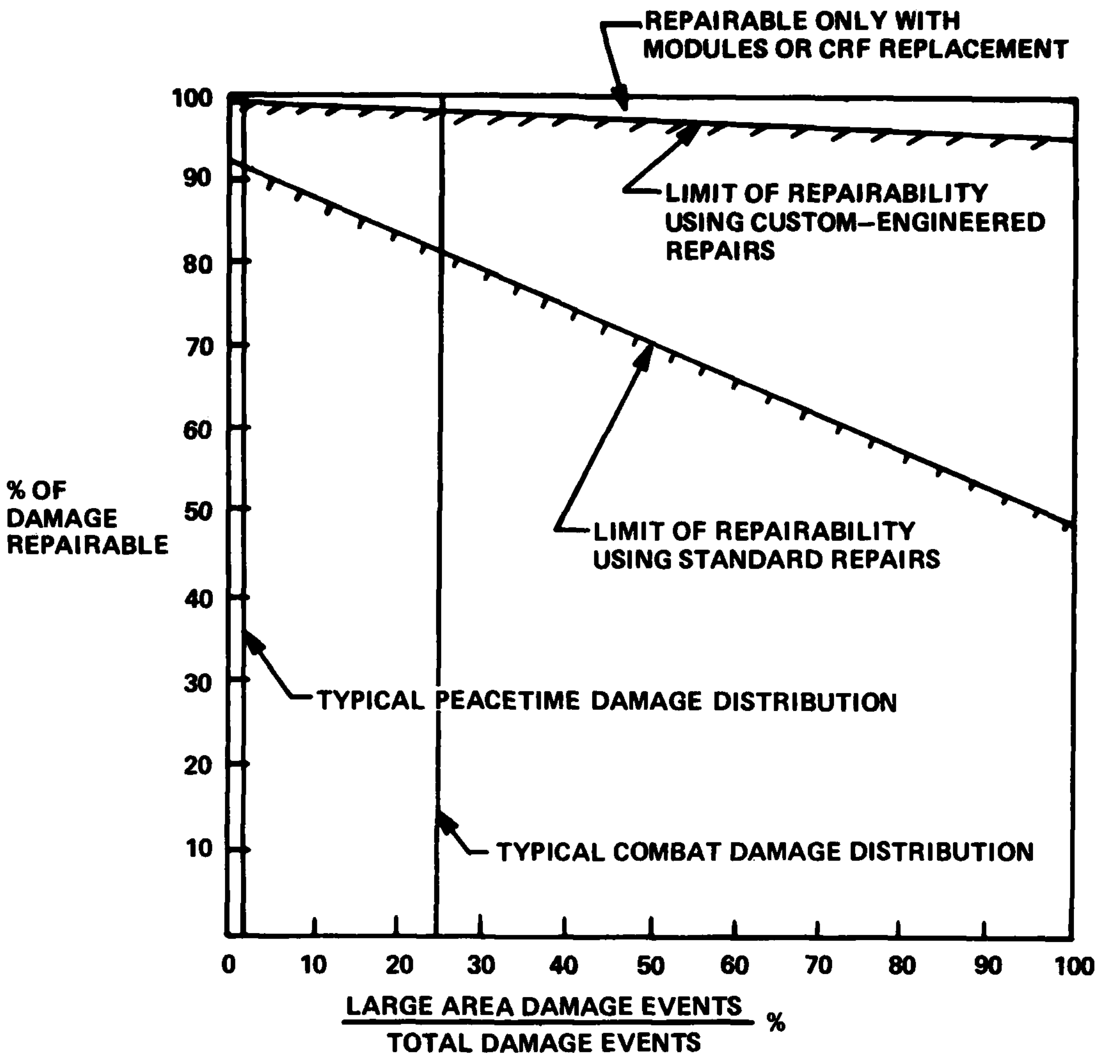

Figure 48. CRF Repairability Versus Percent Large Area Damage

INSPECTION CRITERIA

The analysis of CRF repairability using the mapping technique just described identified the areas of the structure where repair will be most difficult and/or most critical. The requirement for inspection and the criticality of inspection are related to the same set of variables. It is known that structural members loaded in tension are generally more critical from the standpoint of repair than are members loaded in compression and that members with small design margins are more critical than those with large margins. These conditions also present problems for inspection, both from the standpoint of detecting damage in service and of verifying the quality of repair.

From the preliminary analysis conducted for this program, it has been concluded that inspection and repair will be of greatest concern in the rear bay area of CRF where concentrated loads from the tail are distributed into the forward structure and in structural members along the right side of the CRF which are loaded in tension at the critical design condition (rolling pullout right). For the current Black Hawk, periodic inspection of the interior structure of the rear fuselage is required at intervals of 500 flight-hours. It is expected that the CRF will be inspected at the same interval.

One of the concerns associated with the introduction of advanced composite airframe structures is the possibility of suffering manufacturing flaws or induced defects that become structurally unsound before they become visible to inspection. As was brought out in Reference 1, it is a characteristic of composites under some conditions to suffer internal dalamination from an impact without revealing surface damage from the impact. Under some loading conditions, it may be possible for such delaminations to progress to a serious state before becoming visible.

Ideally, the structure should be designed to retard such propagation or to insure that flaws or defects will always become visible before reaching a critical state. An investigation of this subject is covered in the Recommendations section of this report.

CONCLUSIONS

1. The potential for service-induced damage has been significantly reduced in the design of the UH-60A Composite Rear Fuselage (CRF) through incorporation of R&M design features in areas of high impact exposure. It is estimated that improved damage tolerance and the elimination of most nuisance-type repairs will make the CRF 40% less expensive to maintain in the field than the existing metal design.

2. Modular design can significantly enhance the field repairability of composite airframe structures. The concept is shown to be cost-effective for peacetime operation of helicopters. Major improvements in aircraft availability for combat are also predicted.

3. Epoxide-impregnated preform patches show great promise for rapid, low-skill-level repair of composite structures in the Army field environment. Polyurethane foam kits are a prospective approach to creating simple, lightweight forms and mandrels on which to laminate or bond composite repairs on the aircraft.

RECOMMENDATIONS

It is recommended that the Army undertake development of techniques for rapid, low-skill-level repair of composite airframe structures in the field. This should include further investigation of rapid curing epoxide for use in field-level prepregs, considering such factors as processability, mechanical properties and shelf life. It should also include demonstrations to assess the feasibility of developing complete one-piece preform patches for repair of primary structure. The use of polyurethane foam to create lightweight remain-in-place forms and mandrels for laminated repairs on the aircraft should be developed as part of this effort. The concept of using APU bleed air as a source of heat and pressure for curing composite repairs in the field should also be investigated. Low temperature curing systems for wet layup repair should be studied as an alternative to the fast-cure epoxide prepreg.

It is recommended that the Army fully develop the modular design concept for composite airframe structures. This should include developing a reliable, low-cost method of permanently inscribing the module cutting lines and fastener locations via provisions integral with the bonding tools. Methods and tools for cutting, drilling, and trimming various types of composite structures in the field should be defined. Mechanical splicing techniques for skin and interior structural members should be designed and demonstrated. Tests should be conducted to verify the structural adequacy of module splicing techniques in critical areas of the structure.

One of the concerns associated with the introduction of advanced composite airframe structures is the possibility of suffering manufacturing flaws or induced defects that may propagate to a structurally unsound condition before becoming visible to inspection. Ideally, the structure should be designed so that flaws or defects will always become visible before reaching a serious state. It is recommended that the next phase of work include an investigation of this potential problem. This should include tests wherein the possible types of defects are introduced into representative parts and the parts are tested to relate structural deterioration to visible damage. The purpose of this testing will be to verify that no serious conditions can occur prior to the onset of visual evidence or, alternatively, to identify the measures that should be taken in design to avoid these conditions.

TABLE 4. R&M DESIGN OPTION TRADEOFF ANALYSIS

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type**	%	Avg. Cost($)	
Structure surrounding pressure refueling access door. - Impact by aircraft refueling nozzle.	9,500 (Note 1)	190 (1/50)	Baseline: 3.8 lb. aluminum honeycomb panel; outer face 3 plies Kevlar, 3 plies graphite.	63	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.315
			Option: 4 lb. Nomex honeycomb panel; outer face 8 plies E-Glass.	2	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.010
Pressure refueling access enclosure. - Impact by aircraft refueling nozzle.	9,500 (Note 1)	190 (1/50)	Baseline: 6 plies Kevlar.	2	3-U 4-U	85 15	45 85	.010
			Option: 6 plies E-Glass.	2	3-U 4-U	85 15	45 85	.010
Structure surrounding gravity refueling access door. - Impact by aircraft refueling nozzle.	500 (Note 2)	10 (1/50)	Baseline: 3.8 lb. aluminum honeycomb panel; outer face 3 plies Kevlar, 3 plies graphite.	3	1-U 3-U 4-U 5-U	20 30 30 20	35 45 85 105	.015
			Option: 4 lb. Nomex honeycomb panel; outer face 8 plies E-Glass.	.1	1-U 3-U 4-U 5-U	20 30 30 20	35 45 85 105	.001

* Per 10,000 Flight-Hours
** U = Unconstrained
A = Average
C = Constrained

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Gravity refueling access enclosure. - Impact by aircraft refueling nozzle.	500 (Note 2)	10 (1/50)	Baseline: 6 plies Kevlar.	.1	3-U 4-U	85 15	45 85	.001
			Option: 6 plies E-Glass.	.1	3-U 4-U	85 15	45 85	.001
Box on interior of gravity refueling access door. - Impact by aircraft refueling nozzle on ground equipment.	500 (Note 2)	10 (1/50)	Baseline: 3 plies Kevlar.	6	3-U 4-U	85 15	45 85	.031
			Option: 4 plies E-Glass.	2	3-U 4-U	85 15	45 85	.010
Structure surrounding engine ground start access door. - Impact by pneumatic ground start coupling.	200 (Note 4)	4 (1/50)	Baseline: 4 plies Kevlar; 3 plies graphite.	.1	3-U 4-U	85 15	45 85	.001
			Option: 4 lb. Nomex honeycomb panel; 8 plies E-Glass, 2 plies Kevlar, 2 plies graphite.	.1	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.001
Engine ground start access enclosure. - Impact by pneumatic ground start coupling.	200 (Note 4)	4 (1/50)	Baseline: 5 plies Kevlar.	.1	3-U 4-U	85 15	45 85	.001
			Option: 6 plies E-Glass.	.1	3-U 4-U	85 15	45 85	.001

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Upper deck (outboard panels) - Impact by dropped tools.	290 (Note 5)	6 (1/50)	Baseline: 5 plies Kevlar.	.2	3-A 4-A	85 15	60 115	.001
			Option: 6 plies E-Glass.	.1	3-A 4-A	85 15	60 115	.001
Upper deck (outboard panels) - Impact by dropped parts.	225 (Note 6)	5 (1/50)	Baseline: 5 plies Kevlar.	.1	3-A 4-A	85 15	60 115	.001
			Option: 6 plies E-Glass.	.1	3-A 4-A	85 15	60 115	.001
Upper deck (outboard panels). - Impact by foot traffic.	1,300 (Note 7)	26 (1/50)	Baseline: 5 plies Kevlar.	10	3-A 4-A	85 15	60 115	.068
			Option: 6 plies E-Glass.	1	3-A 4-A	85 15	60 115	.007
Upper deck (center panels). - Impact by dropped tools.	290 (Note 5)	12 (1/25)	Baseline: 3.8 lb. aluminum honeycomb; 4 plies Kevlar.	7	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.047
			Option: 4 lb. Nomex honeycomb, 4 plies E-Glass, 2 plies Kevlar.	.1	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.001

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Upper deck (center panels). - Impact by dropped parts.	225 (Note 6)	9 (1/25)	Baseline: 3.8 lb. aluminum honeycomb; 4 plies Kevlar.	8	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.053
			Option: 4 lb. Nomex honeycomb, 4 plies E-Glass, 2 plies Kevlar.	.3	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.002
Upper deck (center panels). - Impact by foot traffic.	1,300 (Note 7)	26 (1/50)	Baseline: 3.8 lb. aluminum honeycomb; 4 plies Kevlar.	10	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.067
			Option: 4 lb. Nomex honeycomb, 4 plies E-Glass, 2 plies Kevlar.	1	1-A 3-A 4-A 5-A	20 65 10 5	45 60 115 145	.007
Fuselage step doors. - Impact by boot.	1,000 (Note 8)	40 (1/25)	Baseline: 3 plies Kevlar.	17	3-U 4-U	85 15	45 85	.087
			Option: 5 plies Kevlar.	7	3-U 4-U	85 15	45 85	.036
Fuselage step enclosures (interior of fuselage). - Impact by stowed baggage.	2,500 (Note 9)	25 (1/100)	Baseline: 3 plies Kevlar.	24	3-U 4-U	65 35	45 85	.142
*Per 10,000 Flight-Hours			Option: 6 plies E-Glass.	5	3-U 4-U	65 35	45 85	.030

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Fuselage step enclosures. - Impact by boot.	1,000 (Note 8)	40 (1/25)	<u>Baseline</u>: 3 plies Kevlar.	17	3-U 4-U	65 35	45 85	.100
			<u>Option</u>: 6 plies E-Glass.	.8	3-U 4-U	65 35	45 85	.005
Structure surrounding two lower fuselage steps. - Impact by boot.	1,000 (Note 8)	40 (1/25)	<u>Baseline</u>: 3.8 lb. aluminum <u>honeycomb</u>, 3 plies Kevlar, 3 plies graphite.	19	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.095
			<u>Option</u>: 4 lb. Nomex honey-<u>comb</u>; 8 plies E-Glass.	2	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.010
Structure surrounding upper fuselage step. - Impact by boot.	1,000 (Note 8)	40 (1/25)	<u>Baseline</u>: 5 plies Kevlar.	14	3-U 4-U	65 35	45 85	.083
			<u>Option</u>: 7 plies Kevlar.	.4	3-U 4-U	65 35	45 85	.002
Baggage compartment aft bulkhead. - Impact by stowed baggage.	2,500 (Note 9)	25 (1/100)	<u>Baseline</u>: 3 plies Kevlar, 3 <u>plies</u> unidirectional graphite.	19	3-U 4-U	50 50	45 85	.124
			<u>Option</u>: 4 plies E-Glass, 4 <u>plies</u> woven graphite.	5	3-U 4-U	50 50	45 85	.033

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Baggage compartment interior skin and framing. - Impact by stowed baggage.	2,500 (Note 9)	25 (1/100)	Baseline: 5 plies Kevlar/ 4 plies Kevlar, 3 plies unidirectional graphite.	18	3-U 4-U	50 50	45 85	.117
			Option: 6 plies E-Glass/ 6 plies E-Glass, 4 plies woven graphite.	5	3-U 4-U	50 50	45 85	.033
Fuel cell covers. - Impact by dropped tools.	18 (Note 10)	1 (1/25)	Baseline: 3.8 lb. aluminum honeycomb, 2-6 plies Kevlar.	.6	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.003
			Option: 4 lb. Nomex honeycomb, 2-6 plies E-Glass.	.1	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.001
Fuel cell covers. - Impact by dropped parts.	18 (Note 10)	1 (1/25)	Baseline: 3.8 lb. aluminum honeycomb; 2-6 plies Kevlar.	.9	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.005
			Option: 4 lb. Nomex honeycomb; 2 - 6 plies E-Glass.	.1	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.001

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Fuel cell covers. - Impact by stowed baggage.	2,500 (Note 9)	50 (1/50)	Baseline: 3.8 lb. aluminum honeycomb; 2-6 plies Kevlar.	49	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.245
			Option: 4 lb. Nomex honeycomb; 2-6 plies E-Glass.	16	1-U 3-U 4-U 5-U	20 65 10 5	35 45 85 105	.080
Fuel cell covers; edges and corners. - Impact due to dropping.	10 (Note 11)	1 (1/50)	Baseline: 6 plies Kevlar.	.1	3-U 4-U	50 50	45 85	.001
			Option: 6 plies E-Glass.	.1	3-U 4-U	50 50	45 85	.001
Fuel cell covers. - Fastener overtorque on pullout.	10 (Note 11)	46 (1/25) (Note 16)	Baseline: 6 plies Kevlar.	12	2-U	100	40	.048
			Option: Add bonded metal grommets.	2	2-U	100	40	.008
Pressure refueling access door; face panel. - Impact by aircraft refueling nozzle.	9,500 (Note 1)	95 (1/100)	Baseline: 3 plies Kevlar.	30	3-U 4-U	85 15	45 85	.153
			Option: 5 plies E-Glass.	5	3-U 4-U	85 15	45 85	.026

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage Rate*	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Gravity refueling access door; panel face. - Impact by aircraft refueling nozzle.	500 (Note 2)	5 (1/100)	Baseline: 3 plies Kevlar.	3	3-U 4-U	85 15	45 85	.015
			Option: 5 plies E-Glass.	.3	3-U 4-U	85 15	45 85	.002
Engine ground start access door; panel face and edges. - Impact by pneumatic ground start coupling.	200 (Note 4)	4 (1/50)	Baseline: 3 plies Kevlar.	1	3-U 4-U	85 15	45 85	.005
			Option: 5 plies E-Glass; flange edge.	.1	3-U 4-U	85 15	45 85	.001
Engine ground start access door; edges and corners. - Impact by pneumatic ground start coupling.	200 (Note 4)	4 (1/50)	Baseline: 3 plies Kevlar.	4	3-U 4-U	85 15	45 85	.021
			Option: 5 plies E-Glass; flange edge.	.1	4-U 4-U	85 15	45 85	.001
Fuel sump drain doors; edges and corners. - Impact by tools and ground equipment.	200 (Note 12)	2 (1/100)	Baseline: 3 plies Kevlar.	2	3-U 4-U	85 15	45 85	.010
			Option: Add 3 ply Kevlar edge doubler and flange edge.	.1	3-U 4-U	85 15	45 85	.001

*Per 10,000 Flight-Hours

TABLE 4 (Continued)

Structure/Hazard	Estimated Rate*		Baseline Design/ R&M Improvement Option	Est. Damage* Rate	Estimated Repair			Repair Cost ($/FH)
	Exposure	Impact			Type	%	Avg. Cost($)	
Magnetic flux valve access panel, edges and corners. - Impact due to dropping.	22 (Note 13)	1 (1/25)	Baseline: 4 plies Kevlar.	.3	3-U 4-U	85 15	45 85	.002
			Option: Add 2 ply Kevlar, edge doubler.	.1	3-U 4-U	85 15	45 85	.001
Magnetic flux valve access panel. - Fastener overtorque or pullout.	22 (Note 13)	9 (1/25) (Note 16)	Baseline: 4 plies Kevlar.	2	2-U	100	40	.008
			Option: Add bonded metal grommets.	.5	2-U	100	40	.002
Ground cable receptacle. - Overstress due to snagging cable.	10,000 (Note 14)	100 (1/100)	Baseline: Permanently fastened receptacle.	25	2-U	100	40	.100
			Option: Add plastic grommet and ground jumper.	5	2-U	100	40	.020
Underside skin and framing. - Impact with terrain objects.	5,000 (Note 15)	50 (1/100)	Baseline: Primarily 5 plies Kevlar with 4 plies unidirectional graphite in cap strips.	49	3-A 4-A 4-C 9-C 10-C	10 50 30 5 5	60 115 145 380 1100	.887
			Option: Primarily 8 plies E-Glass with 8 plies woven *graphite in cap strips.*	15	3-A 4-A 4-C 9-C 10-C	10 50 30 5 5	60 115 145 380 1100	.272

*Per 10,000 Flight-Hours

NOTES ACCOMPANYING TABLE 4

Note

1. Based on one hour average mission between refueling cycles. Estimate 95% of fueling via pressure or closed-circuit.

2. Based on one hour average mission between refueling cycles. Estimate 5% of fueling via gravity.

3. Based on daily fuel sampling via gravity port; average of 1½ 1-hour missions per flying day.

4. Estimate 1 in 50 engine starts via ground power unit (versus APU).

5. Based on predicted unscheduled maintenance frequency for the APU, fire protection bottles, drive shaft and coupling and other components and items of structure located on or in the vicinity of the upper deck.

6. Estimate that all replacement actions (55% of total) and half of repair actions (45% of total) involve handling of parts.

7. Based on frequency of 10-hour daily inspection plus frequency of unscheduled maintenance on upper deck.

8. Estimate that 50% of the inspections and maintenance actions on the engines and upper deck structure and components involve the use of the rear fuelage steps. Half of total frequency allocated to each set of steps.

9. Estimate that half of all missions involve stowing and/or carrying articles in one of the two baggage compartments.

10. Based on unscheduled maintenance frequency for the APU start accumulator.

11. Based on removal frequency for fuel cells and/or interior fuel system plumbing. Half of total predicted frequency allocated to each.

12. Estimate 1 in 50 missions it is necessary to drain fuel from sumps.

13. See Table 5.

14. Used during every aircraft refueling operation.

NOTES (Cont'd)

15. Estimate an average of 2 landings per one hour mission and that 25% of landings are made on unprepared areas.

16. Impact (overstress) frequency considers the total number of fasteners for all panels of a given type.

TABLE 5. FREQUENCY OF ACCESS PANEL AND DOOR USE

Component/Panel	Qty.	Type	Location	Reason for Removal	Est. Removal Freq.*
LF/ADF Antenna	1	Removable (8 screws)	Underside	Periodic inspection and repair of antenna.	21
UHF/AM Antenna	1	Removable (20 screws)	Underside	Periodic inspection and repair of antenna.	20
IFF Antenna	1	Removable (6 screws)	Underside	Periodic inspection and repair of antenna.	21
Aft Pylon Fairing	1	Removable (32 screws)	Upper Deck	Periodic inspection and replacement of drive shaft/coupling. Repair or replacement of fairing.	40
Magnetic Flux Valve Access Cover	1	Removable (10 screws)	Left Side	Periodic inspection and repair of valve.	22
Upper Cabin Door Track Fairing	2	Removable (7 screws)	Left and Right Sides	Repair or replacement of fairing.	2
Fuel Cell Covers	2	Removable (57 screws)	Interior	Removal of fuel cells/ interior plumbing.	10
Gravity Refueling Access Door	2	Hinged; single latch	Left and Right Sides	Daily fuel sampling. Gravity refueling.	6,700 (Note 2)
Pressure Refueling Access Door	1	Hinged; two latches	Left Side	Pressure refueling; closed circuit refueling.	9,500 (Note 1)
Pneumatic Ground Start Access Door	1	Hinged; single latch	Left Side	External air source engine starting.	200 (Note 4)
Fuel Sump Drain Doors	2	Hinged; single latch	Underside	Fuel sump draining.	200 (Note 12)

*per 10,000 flight-hours

See notes, page 113.

TABLE 6. GENERIC REPAIR METHODS AND ESTIMATED COSTS

No.	Type Repair	Basic Procedure	Applicable Damage	Comments	Est. Material Cost	Estimated Cost of Repair		
						Uncon-strained	Average	Con-strained
1.	Injection Repair	Drill hole pattern through skin over discrepant area. Heat repair area and flow liquid resin into the void. Cure under heat and pressure.	Localized edge damage and delaminations not accompanied by filament fractures.	Used only on service-incurred delaminations. (Fabrication voids usually contain a flaw that prevents rebonding).	5.00	35.00	50.00	65.00
2.	Fastener Hole Repair	Apply doubler or fill hole with machineable potting compound or bonded-in-place metal plug. Redrill fastener hole.	Oversized, elongated or mislocated fastener holes where larger fastener cannot be accommodated.	Potting compound or aluminum plug not usable for loaded holes in primary structure. Aluminum specifically not usable with graphite.	10.00	40.00	55.00	70.00
3.	Skin Patch	Remove damaged skin and clean surface area. Apply 1 or 2 plies of fiberglass using wet layup. Cure at room temperature or under heat and pressure.	Localized cuts, tears or nicks in secondary structure that do not penetrate more than two laminate plies.	Primarily a cosmetic repair. Not used where significant structural loads are present.	15.00	45.00	60.00	75.00
4.	Cure-in-Place Structural Repair (Surface or Flush)	Remove damaged structure and clean surface. Cut to size and wet lay up laminates on parent structure. Cure in place at room temperature or under heat and pressure.	Cracks, cuts, tears and punctures of skin panel or structural member where the shape and/or contour of the structure prohibits use of a pre-cured patch.	Layup comprised of composite tape or fabric or titanium foil laminates and epoxy film adhesive. Ply stacking and orientation must match parent structures or patch must be made over-strength.	25.00	85.00	115.00	145.00
5.	Potting Compound Repair (Sandwich Panels)	Removed damaged skin and carve out a cavity in the core, creating an undercut around the skin cutout. Pack potting compound into the cavity and cure under heat and pressure. Sand flush with skin and install skin patch over repair.	Small diameter tear or puncture in honeycomb or foam core sandwich panel causing core damage in excess of that allowed for simple skin patch repair.	Creating skin undercut may be troublesome. Potting compound may tend to pack into honeycomb cells. Heavier than honeycomb plug. Requires two operations: plug and skin patch. More difficult than prefabricated plug/patch.	30.00	105.00	142.50	180.00

TABLE 6 (Continued)

No.	Type Repair	Basic Procedure	Applicable Damage	Comments	Est. Material Cost	Estimated Cost of Repair		
						Uncon-strained	Average	Con-strained
6.	Core Plug Repair (Sandwich Panels)	Remove damaged skin and carve out a cylindrical cavity in the core. Cut a core plug from honeycomb material, fit to cavity, and bond in place with a wafer in the floor of the cavity using paste epoxy adhesive. Cure under heat and pressure. Install skin patch over core plug.	Large diameter tear or puncture in honeycomb sandwich panel in excess of that permitted for potting compound repair. Used where prefabricated plug/patch unavailable.	Forming plug from honeycomb stock may be difficult with field tools. Requires two operations: plug repair and skin patch. More difficult than prefabricated plug/patch.	30.00	105.00	142.50	180.00
7.	Complex Repair	Combination of two or more simple repairs, frequently involving fabrication of a mold shaped to the parent structure on which to lay up the repair.	Damage to multiple structural elements (stringers, stiffeners, etc.) and/or multiple load paths.	Magnitude and direction of the loads on the structure and the composition of each structural element (plies, orientation, stacking sequence) must be known or determined. Generally impractical for field.	80.00	230.00	305.00	380.00
8.	Cannibalization Repair (Non-Modular)	Cut and remove damaged structure from aircraft. Remove an equivalent piece of structure from a second fuselage and install with structural splices.	Extensive damage to permanent structure.	Requires an entire fuselage or fuselage sections from which replacement parts can be cut. Structure must be amenable to splicing.	500.00*	800.00	950.00	1100.00

*If obtained from stocked spare.

TABLE 7. R&M DESIGN OPTION SUMMARY

R&M Improvement Option	Estimated Weight/Cost Deltas*				Design Recommen-dation
	Weight (lb)	Mfg. Cost ($)	Invest. Cost ($)	LC Repair Cost ($)	
Structure surrounding pressure refueling access door. Use 4 lb. Nomex honeycomb with 8 ply E-Glass facing. (Improved tolerance to refueling nozzle impact.)	+ .39	- 11	+ 9	- 1,980	Yes
Pressure refueling access enclosure. Make from 6 ply E-Glass. (Improved tolerance to refueling nozzle impact).	+ .34	- 9	+ 8	Neg.	No
Structure surrounding gravity refueling access door. Use 4 lb. Nomex honeycomb panel with 8 ply E-Glass facing. (Improved tolerance to refueling nozzle impact).	+ .26	- 8	+ 5	- 90	No
Gravity refueling access enclosure. Make from 6 ply E-Glass. (Improved tolerance to refueling nozzle impact).	+ .52	- 12	+14	Neg.	No
Box in interior of gravity refueling access door. Make from 4 ply E-Glass. (Improved tolerance to impact by refueling nozzle and ground equipment).	+ .06	+ 1	+ 4	- 137	Yes
Structure surrounding engine ground start access door. Use 4 lb. Nomex honeycomb panel with facing of 4 plies E-Glass, 2 plies Kevlar, 2 plies graphite. (Improved tolerance to impact by pneumatic ground start coupling).	+ .67	+ 15	+50	Neg.	No
Engine ground start access enclosure. Make from 6 plies E-Glass. (Improved tolerance to impact by pneumatic ground start coupling).	+ .19	+ 1	+11	Neg.	No

*Versus Baseline; See Appendix A

TABLE 7 (Continued)					
R&M Improvement Option	Estimated Weight/Cost Deltas*				Design Recommendation
	Weight (lb)	Mfg. Cost ($)	Invest. Cost ($)	LC Repair Cost ($)	
Upper deck (outboard panels). Make from 6 plies E-Glass. (Improved tolerance to dropped tools, dropped parts, foot traffic).	+2.33	+10	+127	-397	No
Upper deck (center panels). Use 4 lb. Nomex honeycomb panel with outer facing of 4 plies E-Glass, 2 plies Kevlar. (Improved tolerance to dropped tools, dropped parts, foot traffic).	+3.69	+76	+261	-1,020	No
Fuselage step doors. Make from 5 plies Kevlar. (Improved tolerance to boot impact).	+ .47	+27	+ 51	-332	Yes
Fuselage step enclosures. Make from 6 plies E-Glass. (Improved tolerance to stowed baggage impact and boot impact).	+2.88	+78	+222	-1,340	Yes
Structure surrounding two lower fuselage steps. Use 4 lb. Nomex honeycomb with 8 ply E-Glass facing. (Improved tolerance to boot impact).	+ .45	-13	+ 10	-553	Yes
Structure surrounding upper fuselage step. Make from 7 plies Kevlar. (Improved tolerance to boot impact).	+.40	+23	+ 43	-527	Yes
Baggage compartment aft bulkhead. Make from 4 plies E-Glass, 4 plies woven graphite. (Improved tolerance to stowed baggage impact).	+.50	+ 6	+ 31	-592	Yes
Baggage compartment interior skin and framing. Make from 6 plies E-Glass, 4 plies woven graphite. (Improved tolerance to stowed baggage impact).	+2.28	+59	+173	-546	No

*Versus Baseline; See Appendix A

TABLE 7 (Continued)					
R&M Improvement Option	Estimated Weight/Cost Deltas*				Design Recommen-dation
	Weight (lb)	Mfg. Cost ($)	Invest. Cost ($)	LC Repair Cost ($)	
Fuel cell covers. Make from 4 lb. Nomex honeycomb, 2 - 6 plies E-Glass. (Improved tolerance to dropped tools, dropped parts, stowed baggage impact, and impact due to dropping).	+1.75	+11	+ 99	-1,112	Yes
Fuel cell covers. Add bonded metal grommets. (Improved tolerance to fastener damage).	+ .10	+50	+ 55	- 260	Yes
Pressure refueling access door. Make from 5 plies E-Glass. (Improved tolerance to refueling nozzle impact).	+ .36	+15	+ 33	- 825	Yes
Gravity refueling access door. Make from 5 plies E-Glass. (Improved tolerance to refueling nozzle impact).	+ .26	+ 9	+ 22	- 85	No
Engine ground start access door. Make from 5 plies E-Glass. (Improved tolerance to face and edge impact by pneumatic ground start coupling).	+ .24	+ 9	+ 21	- 156	Yes
Fuel sump drain doors. Add 3 ply Kevlar edge doubler and flange edges. (Improved tolerance to edge and corner impact by tools and equipment).	+ .05	+ 3	+ 6	- 59	Yes
Magnetic flux valve access panel. Add 2 ply Kevlar edge doubler. (Improved tolerance to edge and corner impact due to dropping).	+ .06	+ 5	+ 8	Neg.	No
Magnetic flux valve access cover. Add bonded metal grommets. (Improved tolerance to fastener damage).	+ .01	+ 5	+ 6	- 39	Yes
Ground cable receptacle. Add plastic grommet and ground jumper. (Improved tolerance to overstress due to snagging cable).	+ .05	+25	+ 28	-520	Yes

*Versus Baseline; See Appendix A

TABLE 7 (Continued)					
R&M Improvement Option	Estimated Weight/Cost Deltas*				Design Recommendation
	Weight (lb)	Mfg. Cost ($)	Invest. Cost ($)	LC Repair Cost ($)	
Underside skin and framing. Make from 8 plies E-Glass with 8 plies woven graphite in cap strips. (Improved tolerance to impact with terrain objects).	+14.46	+306	+1,029	- 3,998	Yes

*Versus Baseline; See Appendix A