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FEASIBILITY OF ARMOR MATERIAL AS BASIC AIRCRAFT STRUCTURE

By

D. G. Whinery

March 1966

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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NORTH AMERICAN AVIATION, INC.
COLUMBUS, OHIO

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U S ARMY MATERIEL COMMAND

DEPARTMENT OF THE ARMY
U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA 23604

This report has been prepared by North American Aviation, Inc. under the terms of contract DA 44-177-AMC-189(T). It consists of a study to determine the feasibility of armor material as basic aircraft structure.

The results of the study indicate that it is technically feasible to replace aircraft structure with structural armor for the protection of crewmen and aircraft components against enemy small-arms fire.

The information in this report will be utilized in the design of passive defense systems for future aircraft.

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FEASIBILITY OF ARMOR MATERIAL AS BASIC AIRCRAFT STRUCTURE

Final Report

by
D. G. Whinery

Prepared by
North American Aviation, Inc.
Columbus, Ohio

for

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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SUMMARY

The important aspects of using armor material as load-carrying structure (structural armor) are discussed. Topic coverage includes approach to design, optimization of the aircraft configuration, and detail structural design, fabrication, and repair of both opaque and transparent armor. The structural properties of the most useful armor materials are reported in the appendixes.

It is concluded that the use of armor material as basic aircraft structure is both feasible and practical. Its use will eliminate from 5 to 10 percent of the total weight of armor material in aircraft by dispensing with the parallel, non-armored structural elements.

The study indicates that for opaque armor material, ceramic-faced fiber glass is the most suitable for low and medium-hit density environments. This material also provides the best potential for effective maintenance of ballistic and structural properties over an extended service life.

For a high-hit density environment, for maximum structural load-carrying capability, and for low cost per pound of armor, dual hardness steel is the most suitable choice. Note, however, that the minimum cost of armoring an aircraft must be determined by considering all characteristics and requirements of the particular aircraft.

Use of the above materials in a crew seat, a contoured fuselage side panel, and a fuselage bulkhead illustrates the recommended application of opaque armor to aircraft structure. Use of transparent plastic armor in a face shield of three facets illustrates the recommended application of transparent armor. Other recommendations are made regarding detail design and areas of further study.

A comparison indicates that the use of structural armor saves 10 pounds compared to kit armor, in a single-place aircraft subjected to sporadic small-arms fire of either of three military projectiles.

Up to 62 pounds is saved by designing for structural armor in a single-place aircraft subjected to concentrated small-arms fire from the lower hemisphere. In addition, an aircraft design employing structural armor instead of kit armor will have fewer total parts and will have a lower cost unless an unusual pricing structure exists.

PREFACE

This study was initiated in June 1964, completed in June 1965, and performed by D. G. Whinerv of the Columbus Division of North American Aviation, Inc. The USAAVLABS technical representatives were J. P. Waller and R. Fama.

The assistance of Mr. E. N. Hegge, U. S. Army Materials Research Agency, and of suppliers of armor materials is gratefully acknowledged. Samples of armor materials were provided without cost by Aerojet-General Corporation, Azusa, California; Aeronutronics Division of Philco Corporation, a subsidiary of Ford Motor Company, Newport Beach, California; Republic Steel Corporation, Canton, Ohio; and Swedlow, Inc., Garden Grove, California.

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CONTENTS

ILLUSTRATIONS

INTRODUCTION

BACKGROUND

Designers have long recognized the load-carrying capability of armor materials. In some cases, for permanently installed armor, the armor material has been utilized to carry structural loads as well as provide ballistic protection. Sometimes, when the removal of load-carrying armor has been deemed desirable, it has been replaced with lighter weight auxiliary structure.

Developments in the field of armor materials over the past few years now make it practical to increase the armor protection on close-support aircraft.

PROBLEM

The armor material added to provide a crew member of a close-support aircraft ballistic protection from small-arms fire may weigh as much as the man himself. While the added ballistic protection is desirable, it results in a weight penalty and increased manufacturing costs. The total cost of performing the desired mission should, however, be decreased by an appropriate armor installation.

Modern ballistic armor materials typically have several times the strength properties of the structure that they could replace. Therefore, the total aircraft weight could be reduced by omitting the redundant structure which parallels the armor. Such a configuration would utilize the ballistic armor material as load-carrying structure as well as for crew protection.

The use of armor to serve also as basic structure involves two major problems: (1) a communication problem among the four technologies involved and (2) the technical problem of obtaining the most suitable configuration for accomplishing a required mission.

The four technologies involved in successfully using armor material as basic structure include:

1. Development and manufacture of armor materials.

2. Aircraft optimization, including aerodynamics, vulnerability, structure, power systems, equipment, and stores.

3. Analysis and design of aircraft structural armor.

4. Fabrication of structural armor.

The problem of optimizing the aircraft configuration to accomplish mission requirements involves decreasing the vulnerability with a minimum increase in weight and cost.

OBJECTIVE

The objectives of this study were:

1. To determine the feasibility of using armor material as basic aircraft structure.

2. To select opaque and transparent materials for primary, secondary, or transparent structure.

This study was aimed directly toward application to military close-support aircraft subjected to small-arms fire.

The report presents the results of the feasibility study and also provides an introduction to the design and manufacturing considerations involved in the use of armor materials as load-carrying structure.

APPROACH

To facilitate a methodical consideration of the problem, it was subdivided into the following categories:

1. Evaluation of state-of-the-art armor materials and materials forecast for the future.

2. Categorization of suitable armor materials as most useful for primary structure, secondary structure, or transparent structure.

3. Experimental testing of materials and structures.

4. Recommendations for applications best suited for selected armor materials.

In the section of this report titled "Theory of Application" is included a discussion of various aspects of using armor material as basic aircraft structure. The discussion includes references to design approach, optimization of aircraft configuration, detail structural design, fabrication, and repair techniques. Both opaque and transparent armor are considered.

In a following section the feasibility of using structural armor is discussed. Next, in the section "Recommended Applications", representative applications of structural armor are given. Certain recommendations are made regarding the use of armor as structural material.

All tabulated data and discussion of peripheral topics, such as structural performance and the derivation and discussion of related analyses, have been placed in the appendixes. A supplementary report contains the ballistic performance of lightweight armor materials.

CONCLUSIONS

It is concluded that the use of selected armor materials as primary, secondary, and transparent aircraft structure is both feasible and practical. The advantage is the removal of the weight of the structure parallel to or duplicating the armor material.

A comparison indicates that structural armor weighs 10 pounds less than comparable kit armor to protect the occupant of a single-occupant aircraft subjected to a minimum of small-arms fire.

Up to 62 pounds is saved by structural armor in a single-place aircraft subjected to concentrated small-arms fire from the lower hemisphere. In addition, an aircraft design employing structural armor instead of kit armor will have fewer total parts and will have a lower cost unless an unusual pricing structure exists.

Shown here are the comparative weights of the armored area in a single-occupant aircraft. The weights for four options of increasing ballistic protection are given. The options are defined on page 13.

Projectile	A		B		C	
Option	Structural Armor, Pounds	Add for Kit, Pounds	Structural Armor, Pounds	Add for Kit, Pounds	Structural Armor, Pounds	Add for Kit, Pounds
1	165	10	234	10	488	10
2	248	16	348	16	717	16
3	403	49	580	49	1232	49
4	507	62	723	62	1518	62

It is concluded that ceramic-faced, non-delaminating fiber glass armor, as represented by sample A-1,

1. Will provide minimum armor weight.

2. Will allow the most effective maintenance of ballistic and structural properties over extended service life by suitable repairs.

3. Will allow the most freedom in forming faceted, contoured armor shapes.

It is concluded that dual hardness steel armor, as represented by samples AP-1, AP-2, R-1, and R-2, defined on page 16,

1. Will provide low first cost and maximum strength.

2. Will be most resistant to concentrated small-arms fire and the resultant close impact spacing or high hit-density.

3. Will be suited to flat panels, large radius bends and panels with slight, two-dimensional contours.

RECOMMENDATIONS

The following items are grouped into recommended applications, comments on the applications, detail design recommendations, and studies recommended to improve the applications. The following four items are recommended applications:

1. Application of ceramic-faced fiber glass, as represented by sample A-1, to an aircraft crew seat structure is recommended to achieve the minimum area of armor to protect a crewman's torso as well as to utilize a ballistically efficient material. This provides suitable protection for a low hit-density threat, allows removal of armor weight by substituting a standard seat, and gives the potential of maintaining the ballistic and structural performances over an extended time by suitable repairs. A companion face shield is discussed in item 4. See Figure 1.

2. Application of ceramic-faced fiber glass, represented by sample A-1, to the structure surrounding an aircraft cockpit is recommended for increased ballistic protection in a medium hit-density threat (up to the minimum hit spacing specified for the particular armor material used). There is a potential for maintaining ballistic and structural performances by suitable repairs. The item 4 companion face shield is suitable for this application. Contoured surfaces such as a fuselage side panel can be manufactured in a configuration of a faceted contour with flat ceramic tile facets. Joining is by adhesive bonding or by self-threading inserts installed in fiber glass. See Figure 33.

3. Dual hardness steel armor, as represented by samples AP-1, AP-2, R-1, and R-2, is recommended as a COIN type fuselage bulkhead structure. This material has advantages in applications encountering high hit-density (close impact spacing) for maximum structural strength and for low cost per pound. Flat panels are desired for ease of processing. Large-radius bends and two-dimensional contour can be made at extra cost. Joining by captive fasteners (item 9) or by spot-welds (in the tough back face only) is suitable. Fusion-welding is feasible, but is accompanied by reduction of ballistic performance. Simple patch repairs can maintain ballistic performance until panel cracking occurs. Corrosion protection against the expected environment is necessary. See Figure 34.

4. Transparent armor of laminated plastic, represented by sample S-1, or of glass-faced plastic in a face shield of three facets is a recommended application. This configuration, compared to the conventional armored windshield, achieves minimum armor weight with maximum azimuth angle and elevation angle of transparent ballistic protection. The configuration

also allows a damaged and shattered shield to be pushed aside to allow unimpaired vision. The application illustrates a splined structural joint (in the plastic portion of transparent armor) applicable to conventional canopies and windshields. See Figure 35.

The following four items provide further comments on the recommended applications:

5. It is recommended that the environment be evaluated and that suitable coatings be provided for all the armor materials. In particular, corrosion protection for steel armor is necessary.

6. It is recommended that doron not be used for primary aircraft structure unless delamination is limited by stitching through the laminate or by changing the materials and manufacturing processes to minimize delamination. See Figure 23.

7. The application of armor material to an aircraft during the preliminary design phase is recommended. At this time the configuration may be adjusted to minimize the area of armor required by suitable location of items which are not ballistic resistant.

8. The use of the relationships of Appendix II in an analysis of the effect of armor on the aircraft vulnerability is recommended. This will allow the selection of the items in the aircraft (including crewmen) which should be protected and indicate where these items should be placed to meet mission objectives and to defeat the threat with minimum attrition of committed resources.

The following two recommendations relate to the detail design of structural armor.

9. It is recommended that joining of armor materials be accomplished by distributed attachment methods (such as adhesive bonding, fusion welding or multiple spot-welding) or by captive fasteners. Captive fasteners are standard or special rivets and bolts with provision for capturing the body of the fastener (following an impact on the external head of the fastener) and preventing it from becoming a secondary projectile. See Figure 5.

10. It is recommended that only shock-resistant aircraft components be mounted on armor panels.

The following five studies are recommended to improve the application of armor materials as aircraft structure:

11. Study the crewman mobility and vision requirements to determine the best transparent shield shape, to determine strength and rigidity needed by the shield to resist impact and vibration, and to determine shield retraction movement for entrance, egress, ejection, and for removal of the shield after multiple hits have obscured vision.

12. Improve the resin shear strength of non-delaminating fiber glass, as represented by sample A-1, while maintaining ballistic performance.

13. Study the improvement of manufacturing processes for ceramic tile faces. This study is expected to yield decreased cost while maintaining or improving tile performance.

14. Study the repair processes for fiber glass backed armor to find the cost of repairs and the length of time that ballistic and structural performance can be maintained. Typical projectile damage is shown in Figures 25, 26, and 27.

15. Study the problem of stopping crack propagation in dual hardness steel by drilling holes at the ends of cracks, as illustrated by the sample in Figure 27.

THEORY OF APPLICATION

GENERAL

Many considerations attend the use of armor as load-carrying structure. First, the type of close-support mission must be evaluated in terms of multiple-hit requirements versus azimuth and elevation orientations. This evaluation is necessary to determine which armor materials may be best suited to the various regions of the aircraft which require protection.

Next, an optimization procedure must be utilized to compare the penalties of added ballistic protection in terms of increased weight and cost with the advantages of the added protection in terms of increased mission effectiveness and decreased vulnerability. The comparison of penalties to advantages is made by an iterative process in which a logical series of armor applications is considered. The optimum choices for armor materials are indicated from the iterative process by determining the points at which the least weight and cost penalties are accrued in achieving the desired mission requirements.

Also involved in the optimization process are the mission characteristics of range to the target as well as takeoff and landing distance requirements. A survey of aircraft performance relationships is listed in Appendix II. Included there are expressions for performance of reciprocating-engine aircraft, but similar relationships for turboprop and turbojet aircraft may require the use of graphical methods as discussed in Reference 47. Although none of these relationships were utilized in this feasibility study, they would be required for any actual application, whether it be a preliminary design project, a production design, or a modification.

The final steps to be considered in the use of structural armor are the detail design phase, fabrication phase, and repair operation.

DESIGN APPROACH

There are several stages at which the use of armor as basic aircraft structure may be considered. These include the preliminary design stage, the production design stage, and the modification of existing aircraft. As one proceeds from the preliminary design stage toward the aircraft modification stage, it is a fact that less design freedom is available and consequently less weight and cost advantage may be gained through use of armor for structural purposes. This fact is illustrated by the following lists of limitations on the designer at each of these three stages.

The Preliminary Design Problem

The following considerations characterize the preliminary design stage.

1. The preliminary design is typically performed to develop an aircraft configuration to carry specific loads over specific mission profiles.

2. There is freedom to size the aircraft structure, the engine, and the equipment to satisfy the requirement.

3. The logical choice is the configuration which promises to deliver the required performance for the lowest cost.

The depth to which the problem could typically be analyzed at the preliminary design stage is illustrated by the following statements.

1. Considerations of reliability, fatigue life, etc., must be spelled out by specification to obtain bases for comparison for competing configurations.

2. Service-related costs may be considered if desired. If these costs are not considered, the design decisions will not reflect the costs of training, support, and transportation related to an aircraft in service.

3. The growth factor is the gross weight increment divided by the incremental weight causing the change. The "value" of a weight increment per pound may usually be expressed as the product of average aircraft cost per pound times the applicable growth factor. That is,

   value/lb. = (avg. cost in dollars/lb.) x (growth factor).

The Production Design Problem

The following considerations characterize the production design stage:

1. The payload, aircraft performance, and important components such as the engine are tied down by a procurement specification.

2. Incentive payments may be available for weight reduction below specified minimum weight. Likewise, penalties may be charged for excessive weight.

The depth to which the problem could be analyzed at the production design stage is illustrated by the following statements:

1. Service-related costs may be considered if desired.

2. At the point in production design when the engine selection is fixed, the maximum gross weight is also inherently fixed if performance is to be maintained. Because of this, the "value" of a weight increment may vary widely. If a weight underrun occurs, the "value" may be low. If a weight overrun occurs, the "value" may be high, perhaps approaching the level, where the overweight penalty is involved.

3. The effect of a weight increment on service-related costs may be considered during a production design effort provided that these costs are clearly set forth in the procurement specification or covered by incentives and penalties.

4. For a contract containing incentive payments for weight underrun, the "value" of a weight increment will usually be the sum of the incentive payment per pound plus the average cost per pound of the aircraft multiplied by the applicable growth factor. The growth factor is the gross weight change divided by the increment causing the change. That is,

   value/lb. = (payment or penalty in dollars/lb. + avg. cost in dollars/lb.)
   x (growth factor).

The Mofidication of Existing Aircraft

The following considerations characterize the aircraft modification stage:

1. The basic aircraft is available from storage for a relatively low inspection and repair cost.

2. The load-carrying capacity of the available aircraft may vary from more than adequate to marginally poor. If the capacity were more than adequate, the "value" of a weight increment would be negligible. If the capacity were marginally poor, the "value" of a weight increment would be at least equal to the aircraft cost divided by delivered payload weight.

The depth to which the problem could be analyzed at the modification stage is illustrated by the following statements:

1. Modifications are performed to suit an urgent existing need. Therefore, the effect of a weight increment on service costs may be determined in relation to the existing need. For example, the cost of fuel alone may range from 5 to 15 dollars per pound of added aircraft weight over the life of the aircraft.

2. The use of load-carrying armor may be restricted to the region of modification in order to minimize overall modification costs.

3. The "values" of a weight increment as a result of modification of an aircraft with a fixed minimum load may vary widely. The lower limiting "value" would be equal to the fuel cost over the life of the modified aircraft. The "value" might be much larger if the weight were to become critical.

4. The "value" of a weight increment when variable loads are permissible is simply the modification cost divided by the weight of delivered payload plus the related service cost increment. For this evaluation, an aircraft is assumed to have a given gross weight and mission fuel requirement. Hence, any weight increment will affect only the delivered payload. That is,

$$\frac{\text{value}}{\text{lb.}} = \frac{\text{dollar modification cost}}{\text{payload delivered}} + \text{dollar cost of service/lb.}$$

Based on the limitations described above for each of the three stages, it may be concluded that several advantages are lost if the use of armor as basic aircraft structure is not exploited during the preliminary design stage. For example, in the case of aircraft modification, certain expenses are always incurred in dismantling portions of the aircraft prior to rebuilding. In such a modification, it might well happen that simple add-on armor would be more economical than load-carrying armor if the payload were more than adequate for the intended mission.

The following discussion assumes that the utilization of load-carrying structural armor is designed into the aircraft during the preliminary design stage. The result of such an assumption is to provide an indication of the maximum benefit to be expected from the application of structural armor.

Options

The design process by which the armor configuration is determined so as to add a presumed minimum of weight and cost is iterative in nature. One proceeds by selecting several logical options and evaluating each on the basis of the important variables involved. The process of selecting the most desirable option from among all those considered is called optimization. For the purpose of discussing optimization analyses in the following section, four selected options are described here. These four options are presented in the order of increasing level of protection.

Option 1, shown in Figure 1, is an armored crew seat which provides protection of the torso only. Protection is provided on the back, bottom, and sides of the seat, as well as on the front, by a removable chest protector. The added weight and cost of providing such protection may be estimated on the basis of the 17-square-foot area of armor material required for this configuration. Either dual hardness steel or composite armor materials are suitable for this configuration. This option is particularly suited to modification of existing aircraft. It is also well suited to removable crew seats in new aircraft, since the armor weight may be easily removed if the crew seat is not needed. The protection of the torso is intended to minimize injuries which would terminate the mission. Face protection is a logical addition to the armored seat. Such face protection, made of transparent and opaque armor, would have to be developed to suit the mobility requirements of the crewmen.

Option 2 is a partially armored cockpit to provide protection for the crewmen and their equipment from attack from specific threat orientations. For certain vulnerable orientations, the armor would provide full protection for the entire body of the crewmen, including the extremities, as well as protection for vital equipment. For other less vulnerable orientations, no armor protection is provided. The weight and cost of this type of protection are dependent upon the armor area provided. The armor area might vary from 17 to 47 square feet for one crewman, including the transparent armor elements. The partially armored cockpit is especially useful when vulnerable attack orientations can be readily predicted from knowledge of specific mission techniques. For example, the mission characteristics might be such that ballistic protection is required only in the forward and aft sections of the cockpit. Again, as for option 1, either dual hardness steel or composite armor could readily be used in the option 2 configuration.

A completely armored cockpit characterizes option 3. In this configuration, full armor protection is provided for the crew and contents of the cockpit, often only in the lower hemisphere of the cockpit, as shown in Figure 2. The area of armor required for one crewman is approximately 47 square feet for this type of protection, including approximately 9 square feet of transparent armor. The area of transparent armor should be held to the minimum that is consistent with visual scanning requirements. This is because transparent armor is significantly heavier and more costly than the equivalent area of opaque armor would be to provide the same protection.

Finally, option 4 is a configuration which provides not only full cockpit protection, but also gives armor protection to other vulnerable components that cannot be moved into a protected space. For example, armor protection might be afforded to a fuel system component as shown in Figure 3 or to the critical portion of an engine as illustrated in Figure 4. The armor area required for this option is clearly more than the 47 square feet which is the typical requirement for option 3.

OPTIMIZATION

Optimization may be defined as that process by which the most favorable set of conditions is selected from among the array of all possible conditions. Needless to say, a true optimum is rarely achieved. The optimization of the use of armor to provide ballistic protection is usually of the iterative or cut-and-try variety. To illustrate this, consider the following discussion.

The dependent variable or parameter of interest is the "probable mission cost". The probable mission cost may be defined as a mission total cost divided by the probability of mission success. This parameter will be a function of the particular close-support mission to be executed. For a given mission, the probable mission cost may be evaluated in terms of added weight, added cost, and perhaps other independent variables.

The iterative procedure involves calculation of the probable mission cost for several selected configurations. This is done in terms of the values of the important independent variables; in this case, the added weight and the added cost of providing armor.

The optimum configuration is then determined simply by comparing the probable mission costs for all of the selected configurations and selecting the most favorable one. A graphical display of the results of the iteration provides a clear picture of how the optimum is selected. Such a display is shown in Figure 6, where probable mission cost is plotted versus the options considered. Each option represents a selected combination of independent variables. For illustration, the four options described in the preceding section are plotted in Figure 6 with a hypothetical value of the probable mission cost. The optimum or minimum value is readily selected from the plot.

If the selected options represent a continuous progression, as do options 2 through 4, the plotted points for these cases may be connected by a smooth curve to aid in the evaluation of an optimum configuration. The dashed curve in Figure 1 illustrates this technique. Relatively independent cases, such as option 1, will, of course, not fall on the same curve.

The relationship of a particular armor configuration to the probability of mission success may be determined by vulnerability analysis. The most elementary vulnerability relationship asserts that the percentage increase in probability of mission success is directly proportional to the percentage decrease in vulnerable area. For this assertion to be valid, it must be assumed that penetration of the vulnerable area is certain to terminate the attempted mission.

If aircraft performance is to remain unchanged by the addition of armor, a gross weight increment greater than the weight of the armor alone will normally be experienced. The execution of a meaningful optimization analysis, therefore, requires knowledge of the relationship between gross aircraft weight increase and the weight of the armor alone.

Also needed to perform a meaningful optimization analysis is a means of comparing an armor material (with one ballistic efficiency and cost) to other armor materials, each with its own different ballistic efficeincy and cost. These relationships are discussed in Appendix II.

DESIGN

Design problems and techniques associated with the application of opaque armor are discussed in the following paragraphs, followed by a similar discussion for transparent armor. For each of the more useful armor materials, a description of its significant physical properties is given. Justification is also given for the experimental strength tests performed on certain materials. Some of the detail design problems, including methods of joining, are also considered.

Table 1 lists the variety of lightweight armor samples that were obtained. The specimen identification code consists of a letter indicating the manufacturer followed by a sample number.

TABLE 1

ARMOR MATERIALS TESTED

SAMPLE IDENTIFICATION	DESCRIPTION	SOURCE
A - 1	6-x-6-inch Plate of Alumina Faced Fiber Glass	Aerojet-General Corporation, Azusa, California
A - 2	6-x-6-inch Plate of Alumina Faced Aluminum Alloy	
AP - 1	6-x-6-inch Plate of Ausformed Dual Hardness Steel	Aeronutronics Division, Newport Beach, California
AP - 2	6-x-6-inch Plate of Ausformed Dual Hardness Steel	
R - 1	6-x-6-inch Plate of Heat Treatable Dual Hardness Steel	Republic Steel Corp, Canton, Ohio
R - 2	6-x-6-inch Plates of Heat Treatable Dual Hardness Steel	
S - 1	6-x-6-inch Plate of Laminated Acrylic	Swedlow, Inc., Garden Grove, California
S - 2	6-x-6-inch Plate of X1G-112 Doron	
S - 3	12-x-12-inch Plate of X1G-112 Doron	

Dual Hardness Steel

Dual hardness steel armor is the most economical of the lightweight armor materials and also the most suitable of the metal armor materials. The ultimate strength of the tough steel back is approximately 273,000 psi, and the ultimate strength of the hard front face is in the region of 400,000 psi. Significant ductility is required for good ballistic performance. It is reported that the hardened armor will withstand bending on a radius five times that of the armor thickness. No tests of material properties were considered necessary for this material because of current experience with heat-treated H-11 tool steels and published information on 9 nickel 4 cobalt steel (9-4 steel). The formability of steel during the manufacturing process prior to heat treatment allows the manufacture of curved shapes. Where space is at a premium, the minimal space required by steel armor is a desirable characteristic.

References 22 and 44 present characteristics of dual hardness steel armor. The ductile to brittle transition temperature for dual hardness steel is well below environmental temperatures expected in service. The Charpy impact test results on 9-4 steel are 47 foot-pounds and 30 foot-pounds at room temperature and -320° F, respectively, for the ductile back face material. Values for the hard front face are 10 foot-pounds and 6 foot-pounds at room temperature and -100° F, respectively.

Joining of dual hardness steel may be accomplished either by using fasteners such as bolts and rivets or by welding. Attachments which carry only moderate loads may be joined by adhesive bonding. Fasteners provide the special advantage of permitting convenient replacement of seriously damaged panels during aircraft overhaul.

If fasteners are used, they must be of the captive type to prevent their becoming secondary missiles. This could happen if a non-captive fastener were struck on the exterior end by a projectile. A retaining strap such as the one shown in Figure 5 may be used to provide the captive feature. Such retaining devices should be installed with a minimum of two fasteners in each assembly.

All welding processes are applicable to steel armor. However, spot-welding is preferred because of the localized and minimal degradation of the ballistic properties. To achieve maximum ease of replacement, it is considered best to use spot-welding to joint attachment strips to the basic armor. The spot-welding process is discussed further in the section titled "Fabrication".

The anticipated maximum stress in armor used as basic aircraft structure is less than one-tenth of the allowable stress for undamaged dual hardness steel armor. A critical problem is the determination of limits on projectile damage that can be permitted before requiring replacement of a panel of structural armor.

Because of the large strength margins, the pit marks caused by projectile impacts will generally not cause short-time fatigue failures. Typical pit marks are shown in Figure 8. Short-time fatigue failure is not a problem because the cracks which form in the hard front face due to the stress concentrating pit marks do not become significantly serious until they also progress through the tough backing material. This usually does not occur until the crack has propagated for several inches along the hard front face. Crack propagation is significantly retarded at the dual hardness interface by the excellent notch toughness of the backing material.

In spite of the above discussion, it is considered to be desirable to grind out pit marks to minimize crack-initiating stress raisers. It is also desirable to stopdrill the crack ends to slow or stop the growth of cracks in the front face due to subsequent projectile impacts. Carbide-tipped drills and rigid tools should be developed for this purpose during field service of the aircraft.

Cracks sometimes are initiated at the rough edges of the armor panel as shown in Figure 8. The crack shown did not, however, propagate through the tough back face of the panel. The back face is shown in Figure 9. It is considered to be necessary to provide smooth edges on the as-manufactured panels to minimize the development of edge-initiated cracks.

Painting of the steel armor and of damaged areas is recommended. Such a treatment helps to retard atmospheric or chemical corrosion, stress corrosion, and other corrosion-related failure phenomena.

Ceramic-Faced Aluminum

Ceramic-faced aluminum composite armor has the lowest cost and is potentially the thinnest of the composite armor materials. Fabrication of the 2024 aluminum backing panel may be completed prior to the application of the ceramic face tiles. The 2024 aluminum alloy is widely used in current aircraft construction.

Bolting, riveting, and spot-welding are all considered to be excellent joining methods. As in the case of the dual hardness steel described earlier, if fasteners are used they must be captured to prevent their becoming secondary missiles upon a direct external impact. Fusion-welding is not a permissible joining technique for this alloy.

For this type of composite armor, it was considered to be necessary to experimentally check the strains induced in the ceramic face by loads applied to the aluminum backing plate. The objective of this experiment was to insure that flight loads carried by the aluminum backing panel would not induce strains which would fracture the ceramic tiles.

The experiment was performed on a 12-inch-square ceramic-faced aluminum sample. This sample, designated A-3, was supplied by Aerojet-General Corporation. The panel was loaded edgewise in a Baldwin universal testing machine. The panel edges were carefully milled square prior to the test. The applied compressive force was then slowly increased until a peak force was noted, at which time slight panel bowing was observed. This peak force correspondended to a stress of 5400 psi in the aluminum.

During the loading process, the strains in the ceramic tiles were monitored. When the compressive stress in the aluminum reached 2700 psi, which occurred during the edge-seating period, the strain at the center of one of the ceramic tiles was read as 40 microinches per inch. This strain is equivalent to 2040 psi in dense alumina. As the aluminum stress was increased to 5400 psi peak stress, no significant change of strain in the ceramic tile was observed. During this loading interval, a strain gage on the aluminum directly in back of the ceramic tile strain gage was indicating a linear stress-strain relationship with a slope, or elastic modulus, of 10.3-million psi for the aluminum.

It may be concluded that shear in the relatively thick adhesive layer between the ceramic tiles and the aluminum backing effectively minimizes strains in the ceramic due to in-plane edge loading of the aluminum panel. However, the ceramic tiles are not isolated from bending and buckling deflections of the aluminum backing. For this reason, buckling resistance must be built into the structure when ceramic-faced aluminum is used as load-carrying armor. Analysis of this armor also indicates that neither the stiffness nor the buckling strength of the aluminum panel is significantly affected by application of the ceramic tiles.

Ceramic-Faced Doron

Ceramic-faced doron composite armor is formulated to delaminate under projectile impact. When used as a structural armor, this delamination characteristic results in a greatly decreased reliability as well as an increased cost of repair. If the formulation were changed to limit delamination, an excellent structural armor would result.

The compressive strength and stiffness were measured on sample S-2, designated by the manufacturer as Swedlow X1G-112. This sample was made of a preferred orientation glass fabric laid up with alternate plies oriented at 90° to each other to form a 1/2-inch-thick panel. A compressive force was applied to the panel edgewise in a Baldwin testing machine. An ultimate strength in compression of 9670 psi and a Young's modulus of 3.4 million psi were measured. Failure occurred by interlaminar shear accompanied by local buckling of glass fibers. The failure zone was wedge-shaped as shown in Figure 11, with delamination and ply separation occurring around and ahead of the apex of the wedge.

The damaged sample was next rotated 90° and a compressive load was again applied to the other two edges of the panel. This test yielded a value of 7830 psi ultimate strength and a modulus of elasticity of 3.5 million psi. Failure again took the form of a shear fault on the surface,as can be seen in Figure 10.

Baldwin-Lima-Hamilton foil strain gages were mounted back-to-back in the center of the panel to obtain strain readings. In the first test, one side indicated no yield by the time of failure. The other side indicated a change in modulus of elasticity at approximately 5700 psi with a subsequent linear stress-strain relationship until failure occurred. Strain readings from the second test were similar. Fiber glass does not exhibit the yield phenomenon.

<u>Ceramic-Faced Non-Delaminating Fiber Glass</u>

Ceramic-faced non-delaminating fiber glass composite armor is potentially the most easily maintainable structural armor material under multiple impact service conditions of long duration.

Tests were performed to determine both the compressive strength properties and the load-carrying ability of self-threading inserts installed in the panel. Sample A-1 was first tested under a loading intended to simulate the use of the material for basic aircraft structure. Under the first edge loading test, an ultimate compressive strength of 3580 psi was observed. Under a second edge loading of the damaged panel, at 90° to the first loading direction, a compressive strength of 3900 psi was noted. As illustrated in Figures 12 and 13, failure occurred by interlaminar shear accompanied by filament buckling.

The ceramic face partially restrained one side of the panel, causing the fault line to manifest itself as a single diagonal line rather than as a wedge-shaped cone, as was observed in the failure of doron.

This fiber glass sample is laminated with a relatively flexible resin to minimize ply separation due to ballistic impact. As a result, both the interlaminar shear and the filament buckling strength for this composition are much lower than they would be for a structural grade fiber glass laminate. In using fiber glass as structural armor, it is desirable to decrease the laminating resin flexibility to obtain higher strength, but not so much as to significantly affect the ballistic resistance.

Strain readings were taken on sample A-1 by installing Baldwin foil strain gages on both sides near the center of the 6-inch-square fiber glass sample. The stress-strain readings indicated a compressive modulus of elasticity in edge loading of 4.4 million psi. Since the layer of adhesive between the ceramic face and the fiber glass back was relatively thick, a negligible induced strain was measured in the ceramic face.

When materials such as fiber glass are tested, they often show a linear stress-strain curve up to some critical loading point where the slope of the stress-strain curve changes but remains linear up to the failure load. The slope of the stress-strain curve below this critical load is called the primary modulus of elasticity, and the slope above the critical load is referred to as the secondary modulus of elasticity. Tension tests on a coupon of sample type A-1 yielded a value of 1.0 million psi for the primary modulus of elasticity and a value of 0.8 million psi for the secondary modulus. Testing of each fiber glass material proposed for a design is necessary because modification of the fiber glass in any way has a direct effect on the physical properties.

An analysis of ceramic-faced fiber glass under compressive load was made to determine the limitations, if any, on the load-carrying ability of fiber glass resulting from the presence of the ceramic face. Ceramics have characteristically low tensile strengths. An iterative strain energy analysis was used in which actual measured stress-strain curves of the FM-1000 adhesive (Reference 33) were represented by appropriate Ramberg-Osgood equations. A linear stress-strain curve was assumed for both the ceramic and the fiber glass. The configuration used in the analysis was a 4-x-4-x-5/16-inch aluminum oxide face with 41.7-million-psi modulus of elasticity and 25,000-psi ultimate tensile strength bonded to a 5/16-inch-thick fiber glass back with a 2.9-million-psi modulus of elasticity. The 0.010-inch-thick adhesive bond layer was relatively thin compared to that of sample A-1. The results of the analysis indicated that adhesive failure between the ceramic tile and the fiber glass would limit the overall strength of the panel. The stress induced in the ceramic was larger than the stress applied to the fiber glass. The distributions of adhesive stresses and strains in the ceramic and in the fiber glass at the time of adhesive yielding are presented in Figure 15. The locations of the peak stresses existing in ceramic-faced fiber glass under structural loading are evident. Adhesive stresses are maximum at the tile edge. Induced stresses in the ceramic tile are maximum at the center of the tile. It may be concluded that a relatively thick adhesive layer is required to prevent induced shear failure in the adhesive.

Three self-threading inserts for 1/4-inch bolts were installed in the fiber glass backing panel to explore this method of joining. As with other fasteners, a capturing flange or strap is necessary to prevent these inserts from becoming secondary missiles. One of the inserts was a brass Groov-Pin Corporation Tap-Lok CZ5020-30 and the other two were H-25028-50 steel inserts manufactured to Military Standard MS35914-110. The three inserts were screwed into drilled holes with diameter tolerance of 0.300 to 0.336 inch. This installation is shown in Figure 14 with the brass insert in the middle.

Using an Instron testing machine to apply single shear loading to just one of the H-25028-50 fasteners, the recorded force-deflection curve indicated a proportional limit at a load of 550 pounds. Initial failure due to compressive bearing stresses occurred in the fiber glass surrounding the insert at a load of 1340 pounds. The fastener continued to carry load, but with evidence of further deflection. The shear loading test was discontinued to prevent destruction of the sample (see Figure 14).

Ultimate pullout load for the C25020-30 brass insert was 800 pounds, and for the H25028-50 steel fastener the ultimate pullout load was 660 pounds. After reaching the ultimate load, the fasteners quietly pulled out of the fiber glass about 1/16 inch. However, they continued to hold over half of the ultimate load. It should be noted that the external threads of the C25020-30 brass insert, which supported the largest pullout load, were coarse 3/8 - 16 threads, as compared with the finer pitch 3/8 - 20 fiber glass engaging threads of the steel inserts.

The overall conclusion is that the class of self-threading inserts represented by the ones tested should provide a reliable and an economical attachment for supporting moderate loads.

In addition to joining or attaching fiber glass armor with self-threading fasteners and inserts, adhesive bonding is also considered to be suitable for this purpose. Adhesive strengths in the range of from 500 to 1000 psi are anticipated for use with one particular fiber glass armor material. The problem of stress concentration in adhesive joints is discussed in References 33 and 42.

The replacement of an adhesively bonded armor panel may be accomplished by judicious cutting and stripping operations. For example, if two butted plates were joined with a crack-covering capstrip on one side only, the technique would be to cut along the butt joint through the capstrip with an abrasive cutoff wheel in a hand-held power saw. The remaining pieces of capstrip would then be stripped off and the surface prepared for rebonding by a filing or grinding operation. In more general terms, replacement of an armor panel involves cutting or stripping off the attaching flanges to leave a flat surface, replacing the damaged panel with a new panel, and reapplying the flange and adhesive. The adhesive may be cured by using pressure either at ambient temperature or at elevated temperature or by using heat lamps, depending on existing conditions.

<u>Transparent Armor</u>

Transparent armor design is a specialized field. However, the use of transparent armor for glazing aircraft transparencies is nearly identical to current aircraft practice in the use of glass and plastics for this same purpose. Figure 16 shows four test coupons of adhesively bonded glazing joints currently used in aircraft practice. The two coupons to the left in Figure 16 show titanium attachments bonded to glass with a ceramic adhesive. The other coupons employ an organic adhesive to bond attachments to glass and acrylic plastic. Properties of the more useful transparent materials are tabulated in Appendix III. Recent developments in organic and ceramic adhesives are reported in References 36 and 37.

In the design of armored aircraft transparencies,the large thickness of the transparent armor must be taken into account. A typical sample of transparent armor is shown in Figure 17. Transparent armor may be attached either by a hinged joint as illustrated by the left side samples of Figure 16 or by rigid attachment as shown in the upper sample of the same figure. The hinged attachment is used to prevent serious bending stresses from occurring in the transparent armor, while the rigid attachment is used to improve the rigidity if it is necessary to stabilize a canopy assembly.

FABRICATION

Lightweight armor materials are tailored to provide good ballistic resistance. The resulting armor material typically exhibits a combination of a hard front face together with a tough resilient back. Fabrication processes have been developed which are compatible with this combination of characteristics.

Dual Hardness Steel

Cutting and trimming dual hardness steel armor may be accomplished by flame or plasma cutting or by friction sawing. Standard flame and plasma cutting techniques must be modified to minimize the size of the heat-affected zone.

Limited forming operations are possible for dual hardness steel if manufacturers' recommendations are followed. A 90° bend is reported to be possible on a radius of four times the thickness for material not heat-treated and on a radius of five times the thickness for heat-treated material.

Drilling the material before hardening presents no problem. Drilling hardened material can be accomplished with carbide drills if rigid equipment to prevent drill vibration is used. The drilled holes at the ends of cracks shown in Figure 18 were made with a #30 carbide drill bit at 1500 RPM in a rigid drill press. Using a similar drill bit at 2100 RPM in an electric drill caused drill breakage and excessive drill wear.

Resistance spot-welding of dual hardness armor is feasible if the spot-weld nugget is confined to the soft back. This backing material of lower carbon content has a relatively low crack sensitivity. Special welding schedules providing for preheat and postheat in the spot-welding machine must be established to prevent weld nugget or heat-affected-zone cracking. Testing of welded joints should accompany the development of the welding schedule to establish static and fatigue strengths. Resistance spot-welds should be loaded in shear only.

Fusion-welding, though it results in a localized reduction in hardness and ballistic resistance, may be successfully used as a joining method if supplier recommendations are followed. Sample R-1 was welded along a crack to determine the extent of hardness reduction in the heat-affected zone. Weld preparation consisted of routing a 60° V-groove in the back side using a carbide-tipped rotary file and a 1/32-inch load on the hard face side of the armor. The weld was made using the tungsten-inert-gas process with argon gas and 0.062-inch-diameter filler wire of the same composition as the back side base metal. Eleven passes with added weld metal were made in the groove. Then the plate was turned over and three passes without added filler metal were made to fuse the crack in the hard face. Four copper chill bars around the weld, a high manual welding speed, and cooling between passes were used to limit the size of the heat-affected zone. The finished weld is shown in Figures 19 and 20. Water cooling the copper chill bars might improve the technique. Average hardness before welding was Rockwell C58-60 on the hard face and C52-53 on the soft face. After welding, the hardness was Rockwell C40 near the weld, gradually increasing to the original hardness about 3/4 inch away from the weld. The measured hardness values are noted on Figure 21. The hardening process for ausformed steel armor does not permit re-hardening after welding. However, the development of a re-heat-treating process for 9-4 dual hardness steel armor is thought to be feasible. This process would allow regaining most of the ballistic properties of welded armor, except in the weld zone where dilution of the hard face base metal occurs during the welding process.

Other facets of steel fabrication which require special attention are dimensional change following heat treatment, crack propogation from stress raisers, and protection from corrosion and chemicals. Dimensional changes may be minimized by special quenching procedures, by permitting maximum dimensional tolerances in design, and by straightening and flattening operations. Cracking of hard steels may be controlled by elimination of stress raisers such as burrs on drilled holes, roughness, and burn spots due to overheating during machining. References 19 and 23 report two studies of this problem. Protection from corrosion and chemicals may be provided by suitable protective coatings.

Ceramic-Faced Aluminum

Fabrication of finished ceramic-faced aluminum armor is limited to cutting by abrasive cutoff wheels, adhesive bonding, and placing fasteners in blind drilled and tapped holes. If fabrication of the aluminum backing is completed prior to applying the ceramic tiles, of course, all standard production processes for trimming, forging, joining, and coating or painting may be used. One design technique for peripheral attachment of these panels is to use a lightweight bolting flange which extends outside of the armored area. With this technique the captive fasteners may be replaced by simple standard fasteners.

Spot-welding is a suitable production process for this armor material but is generally not a suitable repair process. This is because rigid process control must be maintained in spot welding this material.

Ceramic-Faced Doron

Ceramic-faced doron and fiber glass are manufactured in the same manner. Lay-up of the required number of glass fabric piles with laminating resin is followed by curing under heat and pressure. The ceramic tiles are then bonded in place on the fiber glass back using an adhesive layer as required for the desired ballistic performance. A supplementary report summarizes ballistic performance of several ceramic faces used with a doron and with a fiber glass back. The process of laying up the fiber glass back allows some design freedom in the contour or shape of the armor. The shape of the ceramic tiles is presently limited to flat, square configurations. Shaping the tiles may be accomplished by grinding or by scribe and break techniques. This limits the shape of contoured composite armor to a faceted configuration of flat tiles. The lay-up process is used for repair as well as for manufacture. Curing temperatures and pressures are dictated by the particular laminating resin. Repair with access to the face side of the armor is possible; however, access to both sides of the armor will simplify the work. Manufacturers' instructions must be followed in order to regain most of the original ballistic performance and strength.

Transparent Armor

Transparent armor fabrication is simply the assembly of completed transparent units. Use of tough transparent materials such as stretched Plexiglas or Lexan for the backing material will make the transparent unit more tolerant of stress raisers during the fabrication process. Large loads occurring in pressurized canopies require adhesively bonded edge attachments. Development of organic and ceramic adhesives for transparencies is reported in References 36 and 37. The availability of both organic and ceramic adhesive systems permits utilization of the load-bearing capability of transparent armor through a wide range of environmental conditions.

Figure 16 illustrates three types of adhesively bonded attachments in current use. The upper sample uses fiber glass bonded to plastic. The right-hand sample uses an organic adhesive together with fabric and aluminum. The two left-hand samples are two-layer and single-layer glass laminates bonded to titanium attachment strips with a ceramic adhesive.

Fabrication of glass-faced plastic requires the use of slower abrasive cutting methods than does the fabrication of laminated plastic.

REPAIR

To qualify for consideration as a structural armor, a material must be amenable to repair in the field. Maintenance of good ballistic and structural integrity is feasible when economical repair of the structural armor is possible. Repair operations include not only the renovation of extensively damaged areas, but also certain small damage regions which might give rise to propagating cracks.

Metal Armor

Projectile damage of metal armor is characterized by a small area of permanent deflection. This characteristic of metal armor makes repair of such localized damage possible by use of an economical bonded-on patch. The patch may be cut with a friction saw or a cutoff abrasive grinder. Alternatively, acetylene or plasma cutting might also be used with suitable shields and chill bars to minimize the heat-affected zone.

In bonding a patch over an armor defect, a relatively thick layer of elastomeric adhesive should be used. This is to provide enough resiliency to prevent bond separation as a result of adjacent projectile impacts. In addition, it is desirable to apply a fiber glass fabric over the patch to retain it in case of a bond failure. Local accelerations adjacent to a patch due to ballistic impacts might be high enough to rupture the patch bond, but the low-mass fiber glass fabric would probably continue to adhere to the base material and retain the patch temporarily.

The adhesives selected for this study were elastomeric sealing compounds meeting MIL-S-7502 for service to 225° and MIL-S-8802 for service to 250° F. The example repair shown in Figure 18 was made with a two-part polysulfide elastomeric sealing compound. The curing time was 24 hours at 77° F or 6 hours at 140° F. The fiber glass fabric cover selected was an open weave material to simplify the problem of getting good adhesive penetration. Before making the repair, the area was cleaned with trichloroethane rather than gasoline or kerosene, which both leave an oily residue. The patching procedure was to first apply a layer of the polysulfide adhesive, then the steel patch and, finally, to cover the whole area with fiber glass fabric impregnated with adhesive.

At the present time, the bonded-on patch appears to be the only practical field repair process. Welding and drilling processes must presently be limited to well equipped overhaul shops. An investigation of this area is reported under "Theory of Application-Fabrication".

Composite Armor

The projectile damaged area in hard faced armor varies from as small as 1 inch in diameter up to a 4-inch or larger diameter, depending on the choice of armor material. Construction characteristics and the support provided by the resilient backing material have significant effects on the size of the damaged area.

The fiber glass layers of doron are designed to delaminate to enhance its energy absorption properties. The extent of projectile damage was determined by firing a projectile at sample S-2. A ceramic face was not present on this sample. The face was simulated by a ceramic block, approximately 1 x 1 inch, centered on the target area. A 4-inch-diameter area of delamination occurred during projectile impact, as shown in Figures 22, 23, and 24.

Successful repair of the projectile damaged area must result in restoration of structural continuity as well as ballistic resistance of the composite armor. Repair of the doron may be accomplished by the conical patch method recommended for ceramic-faced fiber glass in a following paragraph. However, the large delaminated area makes repair difficult and time consuming. Assuming approximately equal ballistic characteristics, the use of a non-delaminating fiber glass back is to be preferred over a doron back. This selection should minimize cost of repairs and allow the use of threaded inserts.

The fiber glass in ceramic-faced non-delaminating fiber glass armor is designed to absorb the energy of projectile impact with a minimum of delamination. The maximum extent of projectile damage was determined by firing a projectile at sample A-2. The damage is visible in Figures 25, 26, and 27. Recent experience with non-delaminating fiber glass structures reported in Reference 58 indicates that the reduction in strength is limited to the visibly damaged area. Because of the elasticity of the laminating resin, the damaged area does not spread under moderate loads.

Repair entails the removal of the damaged fiber glass to form a tapered or stepped conical hole of 90° or larger included cone angle. Replacement fiber glass material is then laid up in the hole and cured. The small end of the conical hole may be 1-1/4 inches in diameter at the rear side and 3 to 4 inches in diameter at the front side of the fiber glass back. Following lay-up of replacement glass fabric and compatible laminating resin, the curing pressure may be applied mechanically through a thick sponge-rubber pad clamped or blocked in place. The study of repair of ceramic-faced fiber glass armor is in progress at Aerojet-General, Azusa, California. Reference 58 reports the use of a stepped patch for projectile penetrations. The resultant structural proof test to ultimate load is also reported.

Projectile damage in a ceramic-faced aluminum armor material was determined by firing a projectile at sample A-2 to obtain maximum damage. The result was cracking throughout the impacted ceramic tile and 5 to 10 scattered cracks in adjacent tile, as shown in Figure 28. The aluminum backing was torn and cracked over a 2-x-1-1/4-inch area. The damaged aluminum protrudes aft of the undamaged surface 5/8 inch at most, as shown in Figures 29, 30, and 31.

Repair of this material may be made by removing excessively damaged ceramic tile, stop drilling the damage-produced cracks in the aluminum, bonding in a new ceramic tile, and adding a local patch of armor directly over the damaged area in the aluminum. A 2-3/4-inch square of dual hardness steel armor bonded in place is considered to be the most practical patching material for field repair. Welding of the 2024 aluminum alloy is not an acceptable repair process. Captive fasteners attaching a patch to the back side are considered to be undesirable because of the large area of yielded aluminum which must be removed to obtain a flat surface on which to seat the patch.

Transparent Armor

Transparent armor is not repairable and must be replaced when it becomes excessively damaged. Transparency repairability is not as critical for flight safety as is repairability of primary load-carrying structure. The area of impact damage is limited in the more efficient glass-faced plastic armor materials because of the elastomeric nature of the plastic backing. Figure 32 (courtesy of Swedlow, Inc.) illustrates the characteristic damage that occurs in the plastic portion of transparent armor.

FEASIBILITY OF STRUCTURAL ARMOR

Use of armor materials for basic aircraft structure is both feasible and practical for most aircraft performing close-support missions. This conclusion is substantiated by the fact that most of the suitable lightweight armor materials are either similar to materials already in use or have been tested to establish structural load-carrying capability. Dual hardness steels, for example, are similar to heat-treated H-11 steel now used in aircraft. Test results for ceramic-faced aluminum, ceramic-faced doron, and ceramic-faced fiber glass are reported in the section titled "Theory of Application - Design."

The advantage to be gained by using structural armor is that the structural elements which parallel or duplicate the armor may be removed to save weight. The savings in weight of existing structure varies from 1/2 to 1-1/2 pounds per square foot for opaque armor and from 1 to 2 pounds per square foot for transparent armor. To maximize the advantage of using structural armor, it is necessary to choose a lightweight material suitable for carrying the required structural loads and also amenable to easy and economical field repair.

Optimization analyses are useful to determining the more desirable armor configurations consistent with a particular mission requirement. Development of optimization analyses is presented in Appendix II. Four armor configurations are discussed from a feasibility standpoint in the section titled "Theory of Application - Design Approach," with consideration given to weight and cost penalties.

For feasibility evaluation and further study, all armor materials to be considered for structural applications may be classified under one of three types of aircraft structure as defined below.

1. Primary structure which, if completely fractured or removed in flight, would result in loss of the aircraft.

2. Secondary structure which, if completely fractured or removed in flight, would not result in loss of the aircraft.

3. Transparent structure which provides windows with required optical qualities. Because of the brittle nature of most transparent materials, they are usually used only as secondary structure so that failure of the transparent structure would not directly cause loss of the aircraft.

The currently used armor material candidates are judged to be suitable for a particular class of structure if they pass the following test:

> An armor material is qualified for use as a particular class of structure if previous experimental or practical applications of the same or similar materials have shown adequate reliability for the required mission.

The new, untried armor material candidates are judged to be suitable for a particular class of structure if they pass the following test:

> The new armor material candidates are considered to be suitable for a particular class of structure if they are comparable to other acceptable structural materials with similar characteristics, or if suitable ballistic and/or static tests have been performed to verify their reliability.

The suitability of selected armor materials for use as primary, secondary, or transparent structure has been judged in accordance with these tests.

As an indication of its desirability, comparison of the efficiency of structural armor with kit armor for aircraft is made by standardizing the choice of materials and other design variables not related to the particular comparison. This comparison illustrates the applications as well as provides a basis for a choice between structural armor and kit armor.

For the comparison, a single-place fighter-bomber is assumed. The weight of structural armor and the additional weight for kit armor are shown for projectiles A, B, and C and for optional configurations of armor. These options are: (1) armored seat with face shield; (2) partial cockpit armor with face shield; (3) lower hemisphere cockpit armor with transparent armor below eye level; and (4) armor protection of engine components plus option 3 coverage.

Figures 1 and 2 are sketches illustrating these options. Equally efficient armor material is available for both the kit and the structure. Listed below, in the columns headed Structural Armor, are the weights of structural armor for the particular projectile and armor coverage option. Listed in the next column, Add for Kit, are the extra weights carried by the aircraft if kit armor is installed over existing aircraft structure. The added weight represents that of the aircraft structure paralleled by the kit armor.

Projectile	A		B		C	
Option	Structural Armor, Pounds	Add For Kit, Pounds	Structural Armor, Pounds	Add For Kit, Pounds	Structural Armor, Pounds	Add For Kit, Pounds
1	165	10	234	10	488	10
2	248	16	348	16	717	16
3	403	49	580	49	1232	49
4	507	62	723	62	1518	62

The table indicates that the use of armor material as basic aircraft structure saves 10 pounds in an application designed for minimum protection and a minimum threat density. Up to 62 pounds is saved by structural armor for an application providing lower hemisphere ballistic protection. In addition, the use of structural armor results in fewer total airframe parts than aircraft structure with kit armor installed. The cost of providing structural armor will be lower than the cost of equivalent kit armor unless an unusual pricing structure exists.

The estimated weight of structure parallel to opaque kit armor (and removed for structural armor) is .6 pound per square foot. This is the weight of .030-inch-thick aluminum sheet with stiffeners. This structure is considered applicable to options 1 and 2.

A heavier aircraft and aircraft structure is assumed in options 3 and 4 to deliver heavier stores in the face of a more concentrated threat of small-arms fire. For these two options the estimated structural weight is 1.0 pound per square foot. The structure is assumed to be .050-inch-thick aluminum sheet with stiffeners.

The transparent face shield assumed in options 1 and 2 replaces none of the canopy transparency. However, the transparent armor assumed for options 3 and 4 replaces 3/16-inch-thick acrylic which weighs 1.2 pounds per square foot.

RECOMMENDED APPLICATIONS

Because of the large weight penalty accompanying the addition of armor, the complete redesign of an armored area or, preferably, specification of structural armor material at the preliminary design stage of an aircraft is recommended. This practice would both minimize the weight of armor required and maximize the ballistic protection of crew and equipment.

Crew seats made of armor material are currently in use or in development. Ceramic-faced doron, ceramic-faced fiber glass, and dual hardness steel armor materials have been used for this application. Armored seats of ceramic-faced fiber glass panels joined by adhesive bonding are recommended for low hit-density missions. See the seat sketch of Figure 1. The armored crew seat concept has the operational advantage of providing adequate protection with the minimum armor area for missions with a low hit-density threat. In addition, the armor weight may be removed to perform missions in which ballistic resistance is not required. The use of armor materials in seat structure eliminates 10 to 20 pounds. It is also recommended that crewman mobility and visual requirements be studied with a view to adding lightweight transparent armor to existing armored seats. A COIN type aircraft is used here to illustrate applications.

Application of ceramic-faced fiber glass in the cockpit area to protect the crew from a medium hit-density threat is recommended. Figure 33 is a sketch showing an application of this type of armor material in a faceted fuselage side panel. This application is intended to meet the requirement of minimum added weight in an environment of medium hit-density. Panel edges are jointed by self-threading inserts. Other attachments to the panels for buckling stabilization or equipment mounting are also made by using self-threading inserts. Equipment mounted on armor panels must not be shock sensitive or equipment failure may result from projectile impact shock.

Dual hardness steel armor is recommended for areas of high hit-density and maximum structural load. A sketch showing a representative application for dual hardness steel is shown in Figure 34. The bulkhead shown is representative of those both forward and aft of the cockpit area. Edge attachment to the armor panel is made through bolting flanges spot-welded to the tough back of the armor. Flanges are brought outside the armored area when possible to avoid the need for captive fasteners. Attachments to the interior of the bulkhead are made primarily with spot-welded clips. The more critically loaded attachments are joined through use of welds or with bolts through the armor. If bolts are used, they must be backed up internally by suitable fixtures which provide the required captive fastener feature.

The armor subassembly should be protected from corrosion, chemicals and environment by a suitable protective coating. Forming the heat-treated armor to a single degree of curvature of large radius may be accomplished with relative ease. Compound curves and small radius single-degree bends may be made at increased cost. The use of armor material as basic aircraft structure should eliminate parallel basic structural weight equivalent to 5 to 10 percent of total armor weight.

Transparent armor, either glass-faced plastic or laminated plastic, is recommended for aircraft transparencies and face shields for the crewman. Armored windshields for aircraft are in widespread use and require no further discussion. A representative sketch of a face shield of transparent armor is shown in Figure 35.

It is recommended that crewman mobility and visual requirements be studied to determine shield shape and position as well as requirements for strength and rigidity under projectile impact. Also, retraction movements and spring rates for entrance, egress and ejection, and for removal of the shield after multiple hits have obstructed vision,should be studied. The use of a transparent face shield has a large weight advantage and increased visual angle of coverage. For example, a typical panel of windshield armor which would permit a visual angle of 15° in azimuth and 25° in elevation might provide protected vision over an angle of 180° in azimuth and 70° in elevation, if reconfigured as shown in Figure 35. The shield has three facets large enough to allow viewing an entire target through a single facet. The joints between the facets are made with multiple fiber glass splines to provide the strength required for resisting impact, yet limiting the visual blockage to less than a 1- inch width. Similar structural joints may be used in a faceted canopy configuration.

FIGURES

The first five figur .s and Figures 33 through 35 are examples of applications of armor as basic aircraft structure discussed in Recommended Applications and elsewhere in the report.

Figure 6 graphically illustrates a method for comparison of the relative effect of several optional ways of adding armor on average cost of performing a close-support mission. The discussion of this subject starts on page 14.

Figures 7 through 32 are examples of projectile impact damage inflicted on opaque and transparent armor materials. The discussion of the effect on strength and of repair of the damage begins on page 17.

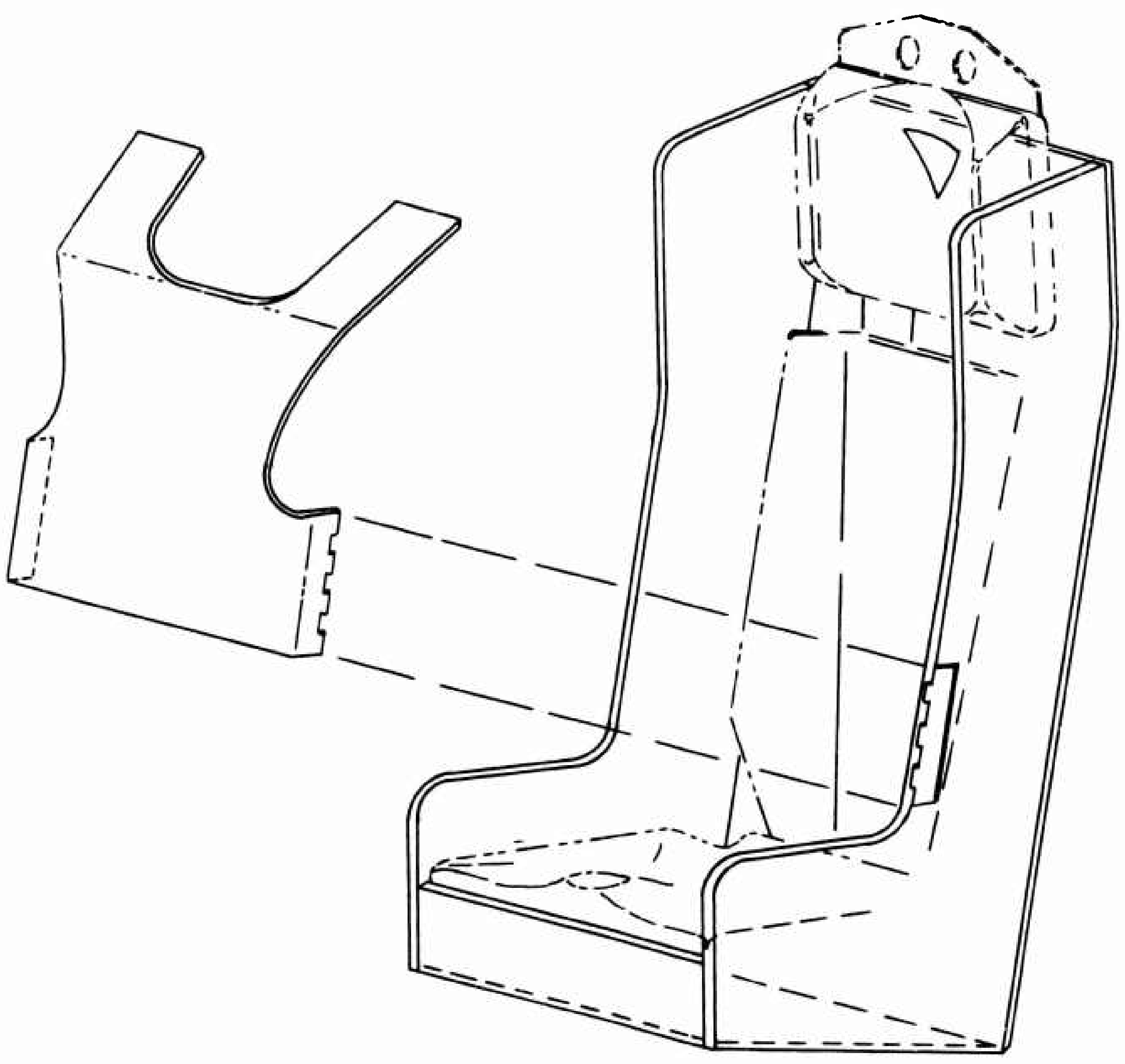


FIGURE 1 - SKETCH OF ARMORED CREW SEAT WITH CHEST PROTECTOR

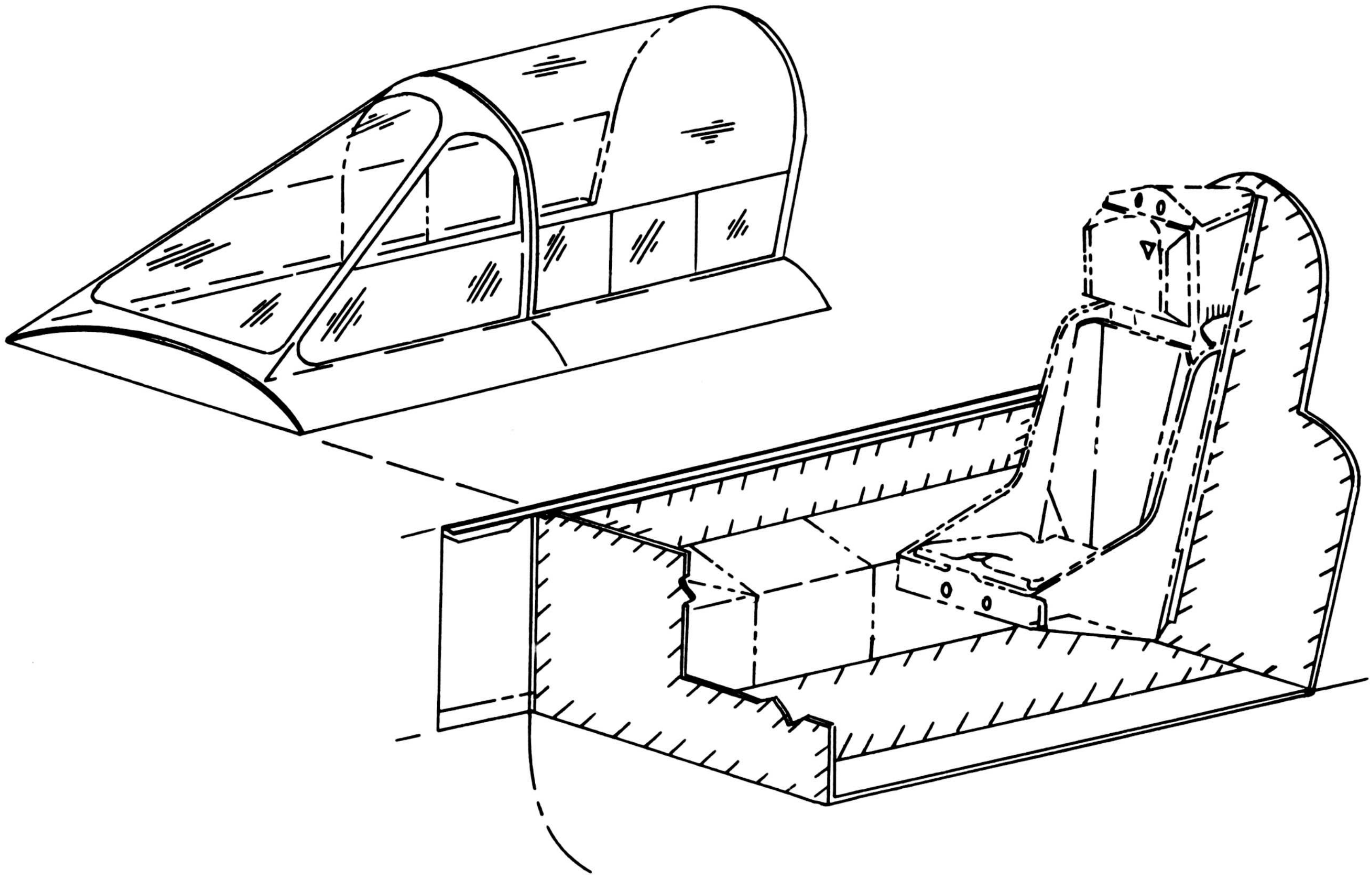


FIGURE 2 - SKETCH OF COCKPIT ARMORED IN LOWER HEMISPHERE

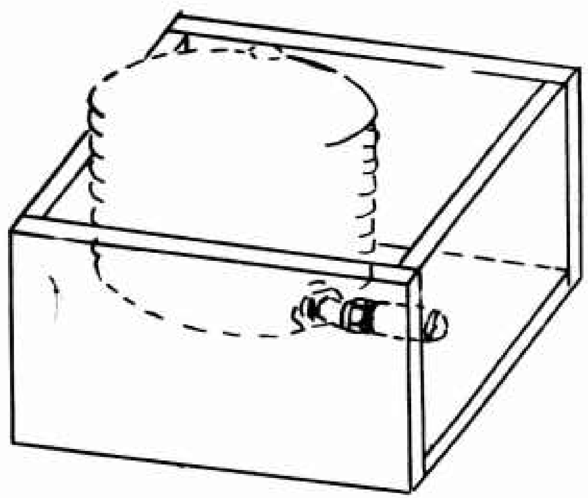


FIGURE 3 - SKETCH OF ARMORED FUEL SYSTEM COMPONENT

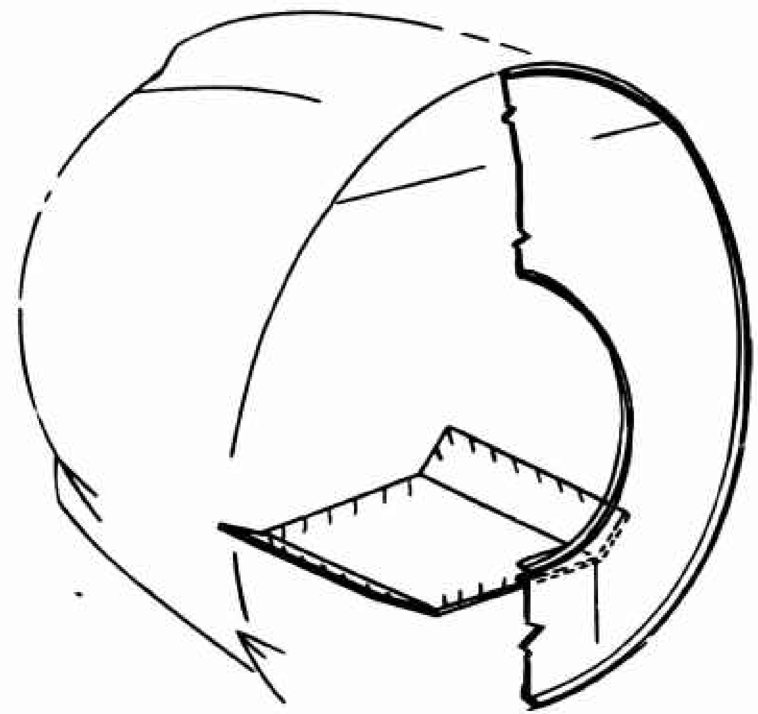


FIGURE 4 - SKETCH OF ARMOR PROTECTION FOR AN ENGINE COMPONENT

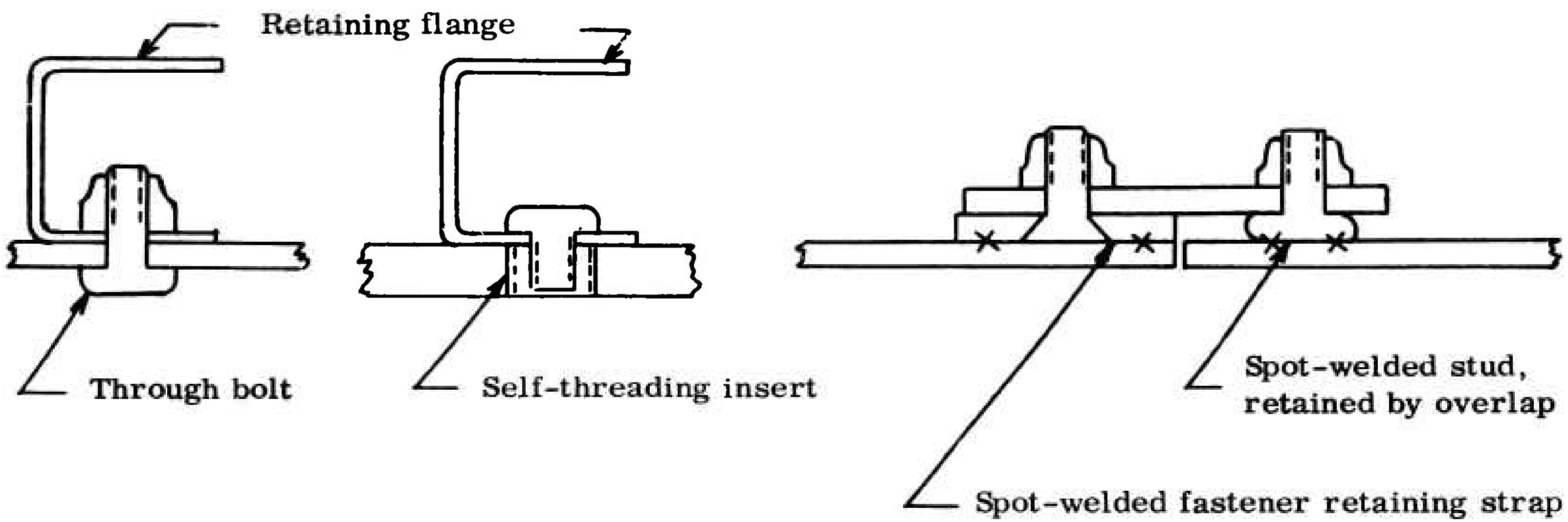


FIGURE 5 - CAPTIVE FASTENER RETAINING DEVICES

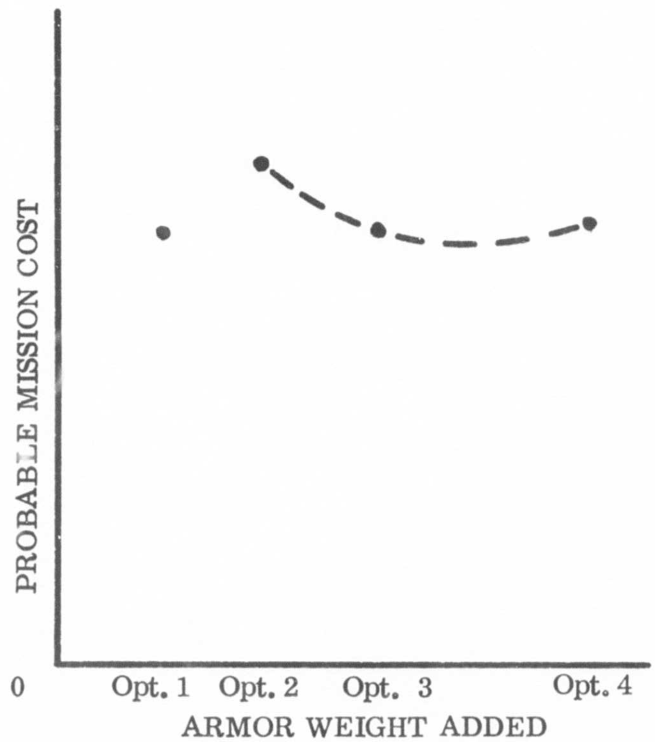


FIGURE 6 - ILLUSTRATIVE GRAPH OF PROBABLE MISSION COST vs. ARMOR WEIGHT ADDED

FIGURE 7 - IMPACT PIT MARKS IN DUAL HARDNESS STEEL, SAMPLE AP-1

FIGURE 8 - DUAL HARDNESS STEEL PERFORATION AND RESULTING CRACK, SAMPLE R-1, FRONT FACE

FIGURE 9 - DUAL HARDNESS STEEL PERFORATION, SAMPLE R-1, BACK FACE

FIGURE 10 - DORON COMPRESSION FAILURE, SAMPLE S-2

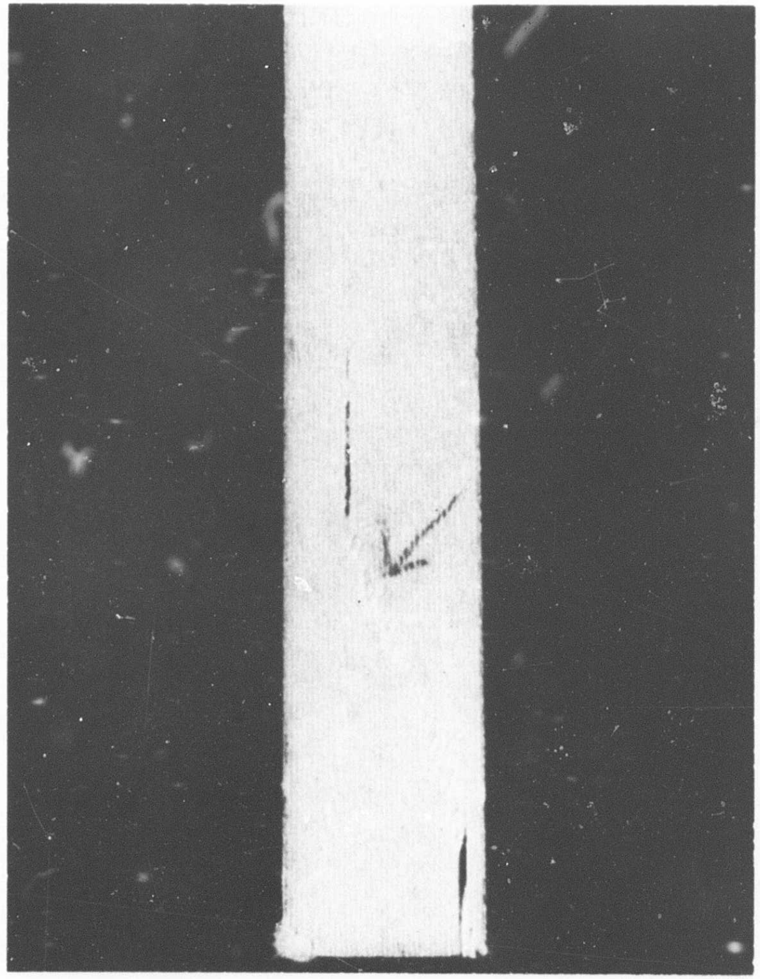


FIGURE 11 - DORON COMPRESSION FAILURE, SAMPLE S-2, EDGE

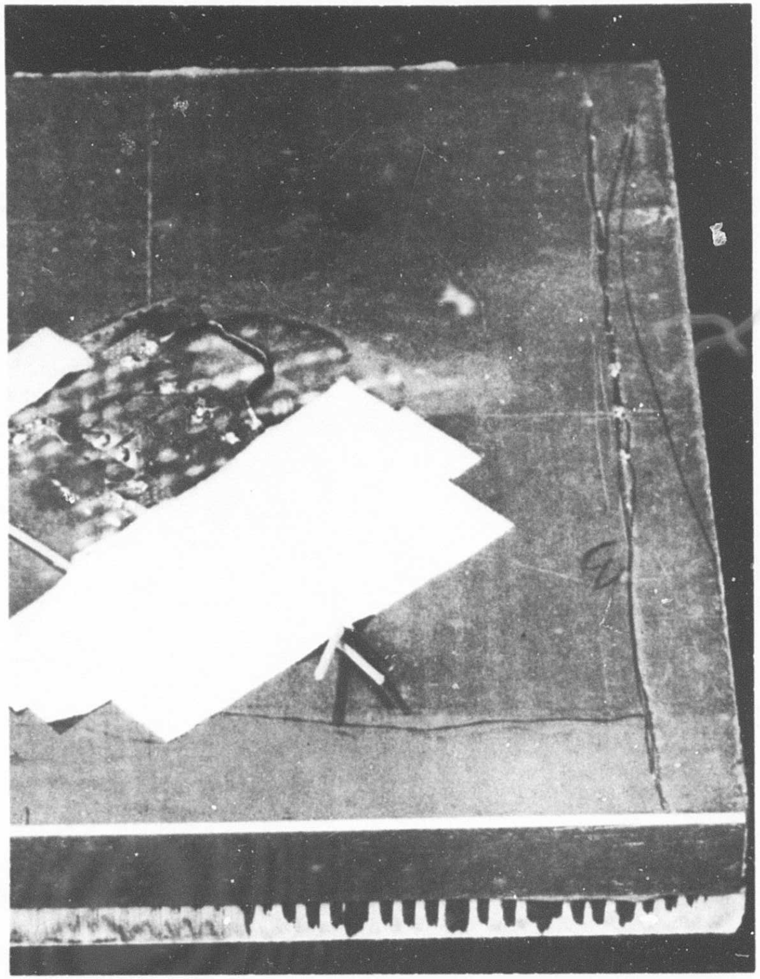


FIGURE 12 - CERAMIC-FACED FIBER GLASS, SAMPLE A-1, FACE SHOWING COMPRESSION FAILURE FAULT LINE

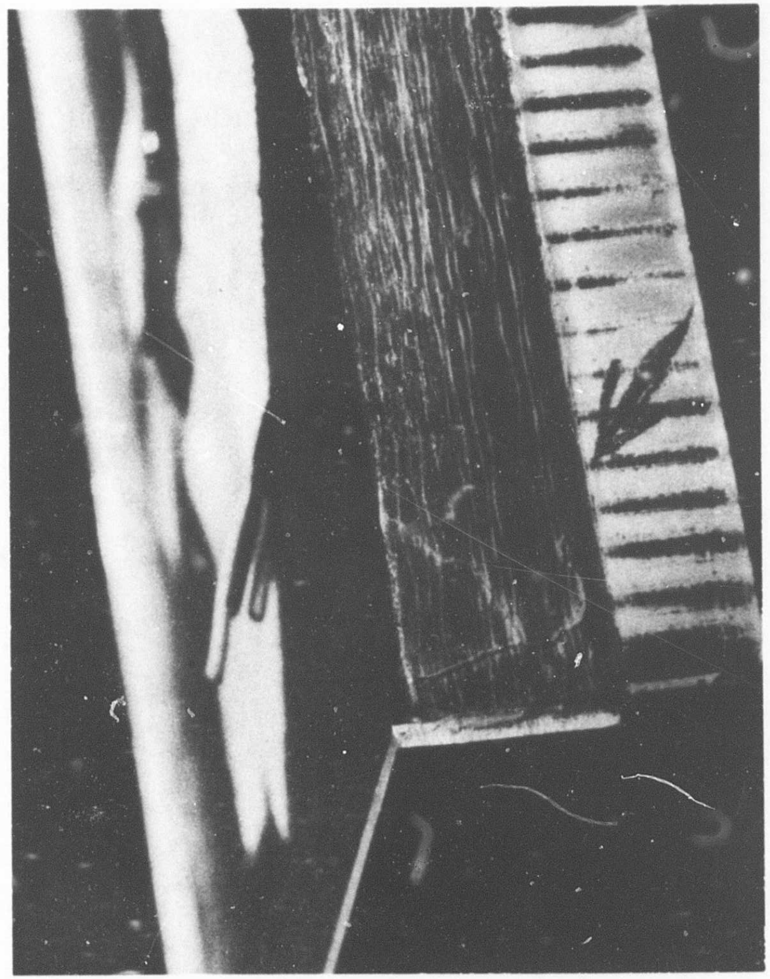


FIGURE 13 - CERAMIC-FACED FIBER GLASS, SAMPLE A-1, EDGE SHOWING COMPRESSION FAILURE FAULT LINE

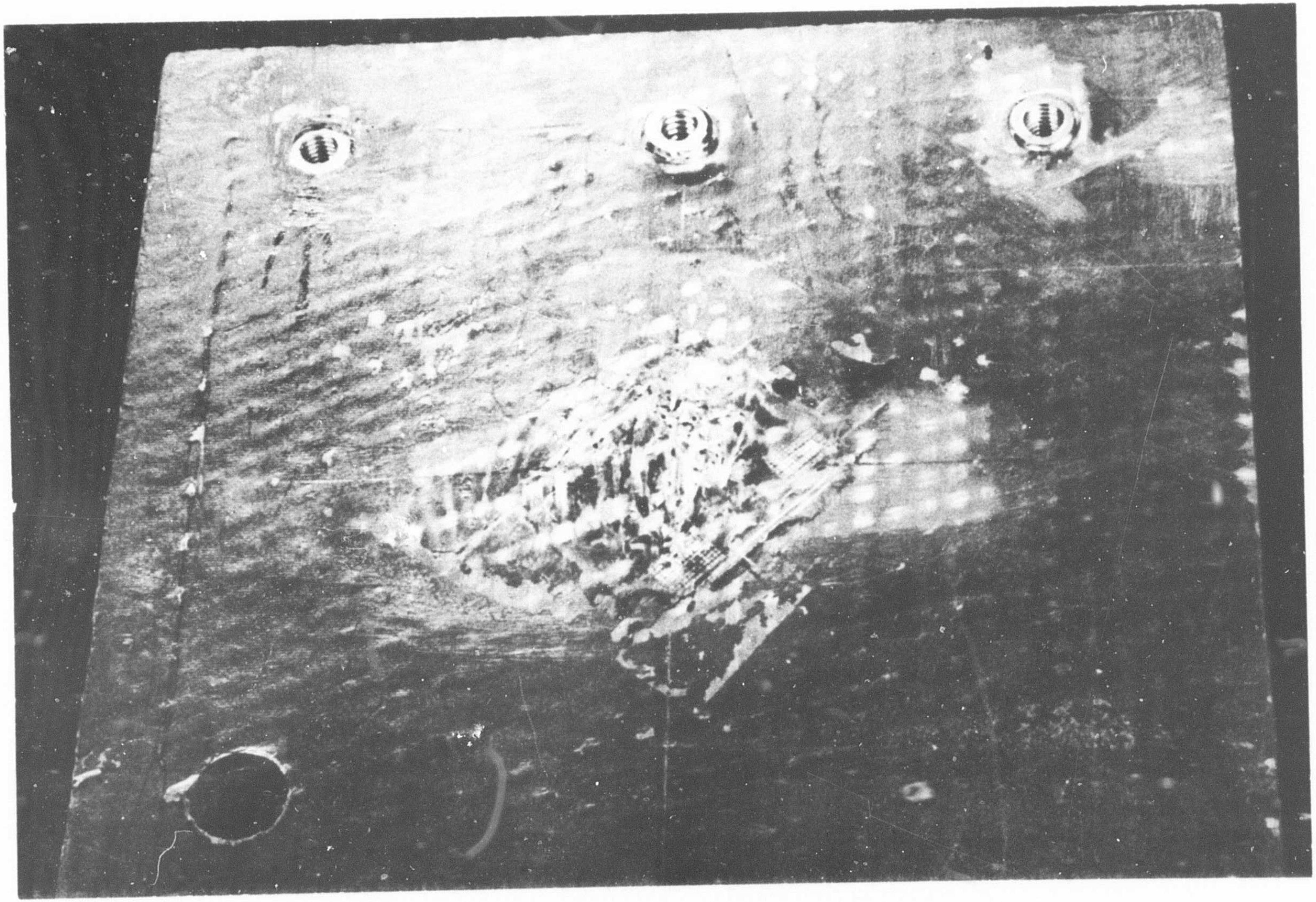


FIGURE 14 - CERAMIC-FACED FIBER GLASS, SAMPLE A-1, SELF-THREADING INSERTS AFTER ULTIMATE LOAD TEST

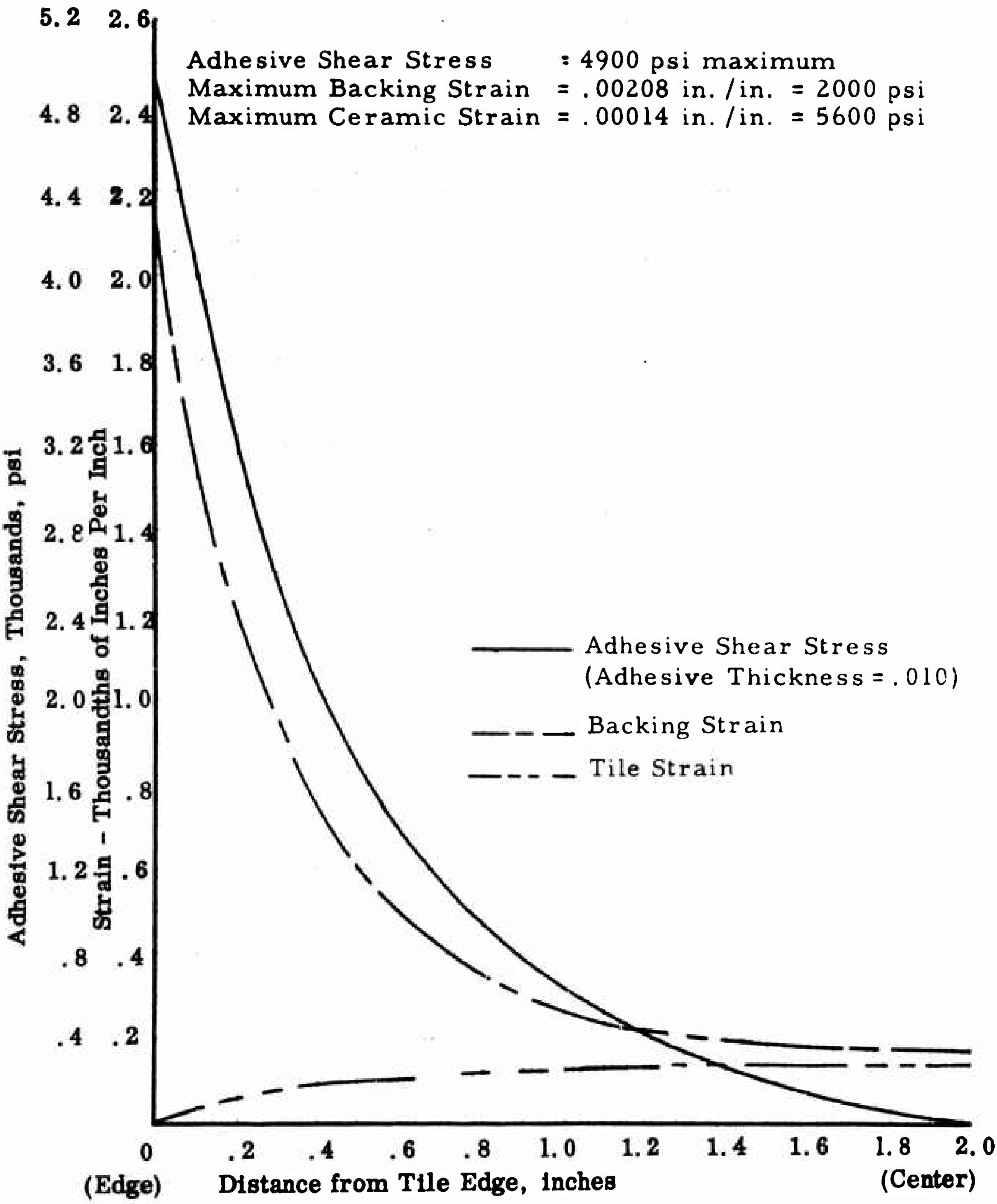


FIGURE 15 - CERAMIC-FACED FIBER GLASS STRESS AND STRAIN UNDER LOAD APPLIED TO FIBER GLASS

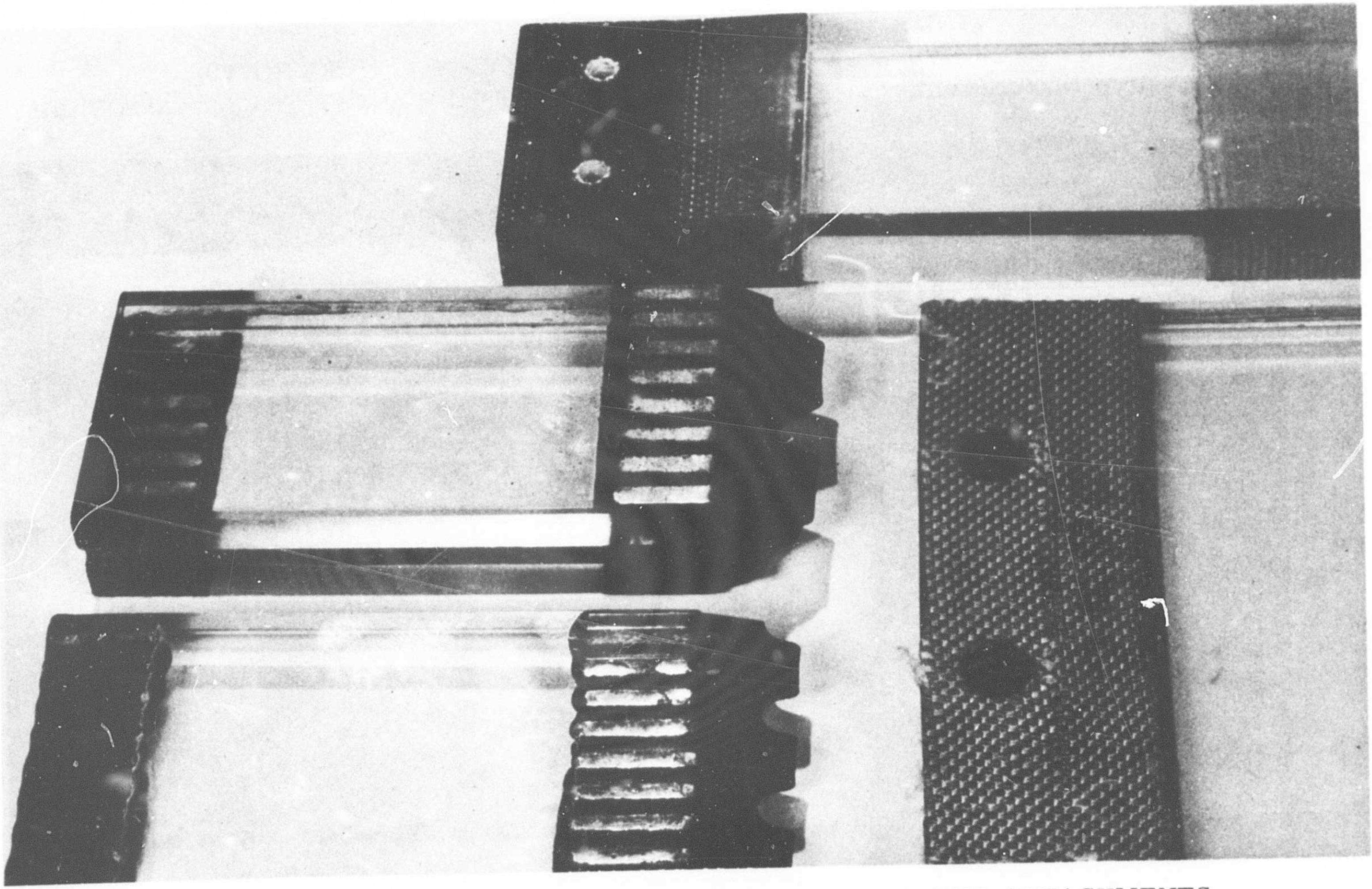


FIGURE 16 - COUPONS OF TRANSPARENT MATERIAL WITH BONDED ATTACHMENTS USING CERAMIC AND ORGANIC ADHESIVES

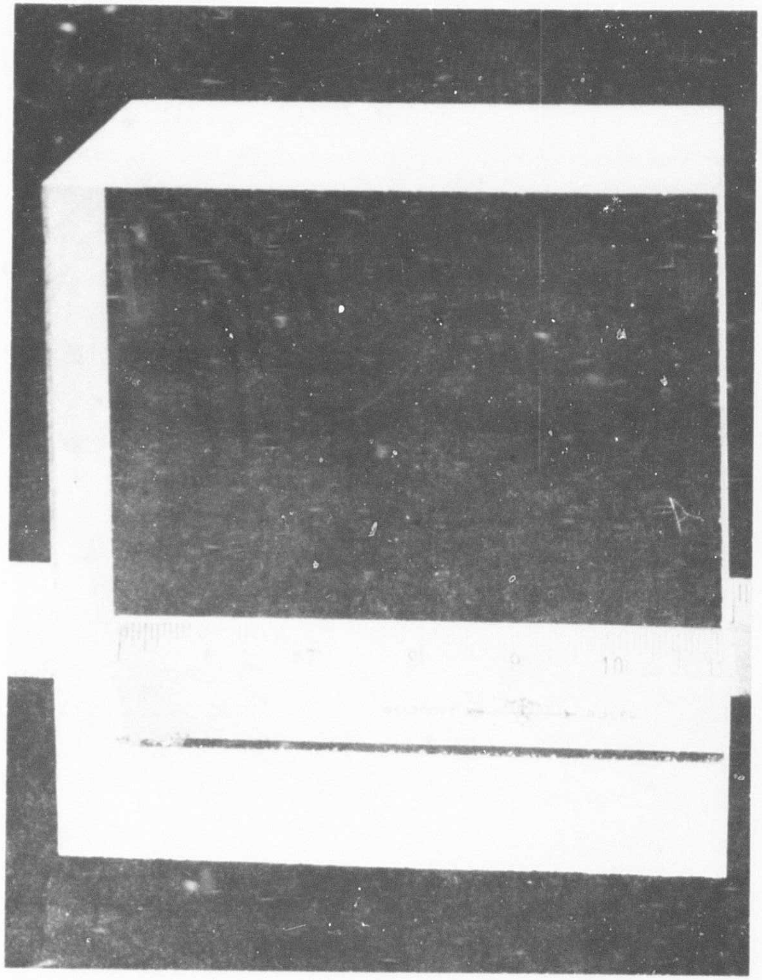


FIGURE 17 - TRANSPARENT ARMOR, SAMPLE S-1

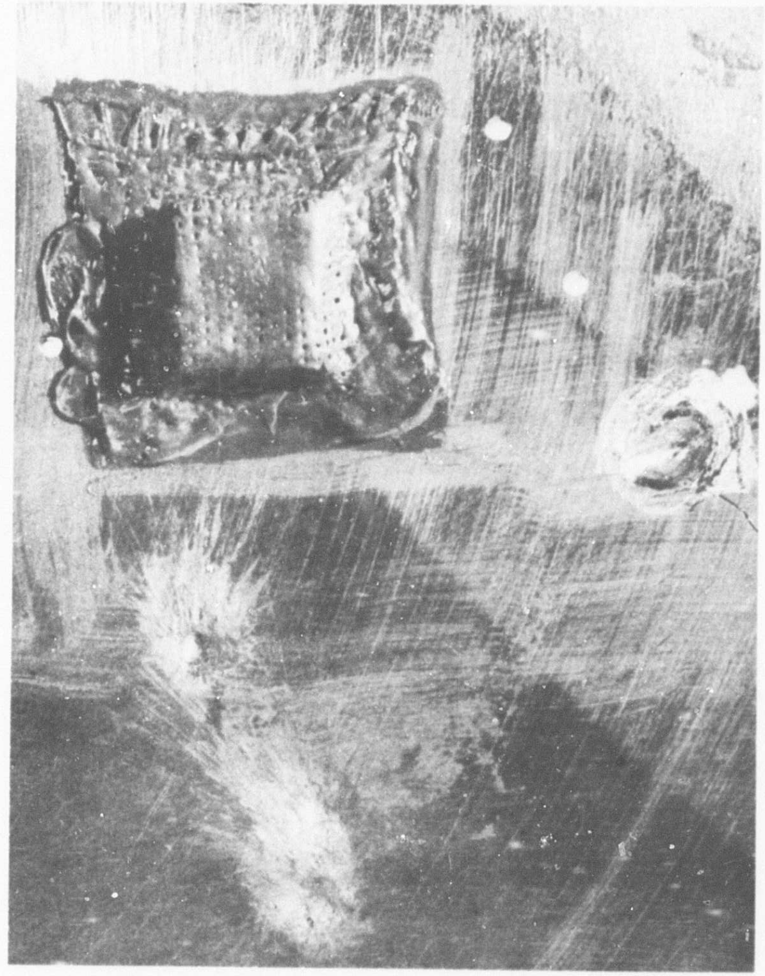


FIGURE 18 - EXAMPLE REPAIR OF DUAL HARDNESS ARMOR, SAMPLE AP-1

FIGURE 19 - DUAL HARDNESS 9 - 4 STEEL WELD, FACE SIDE

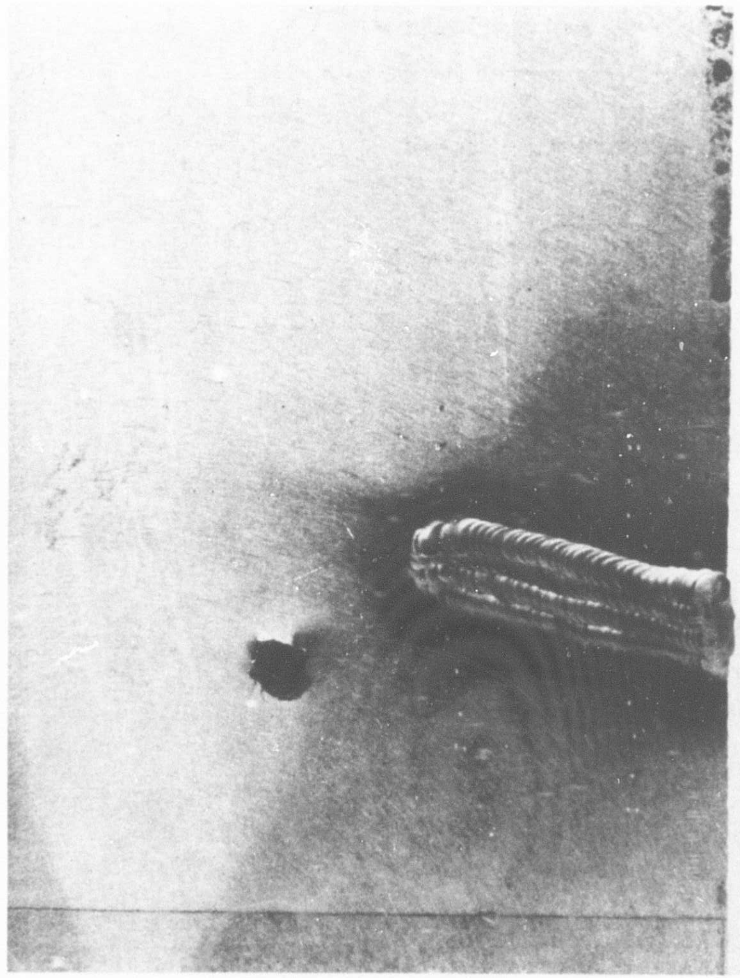
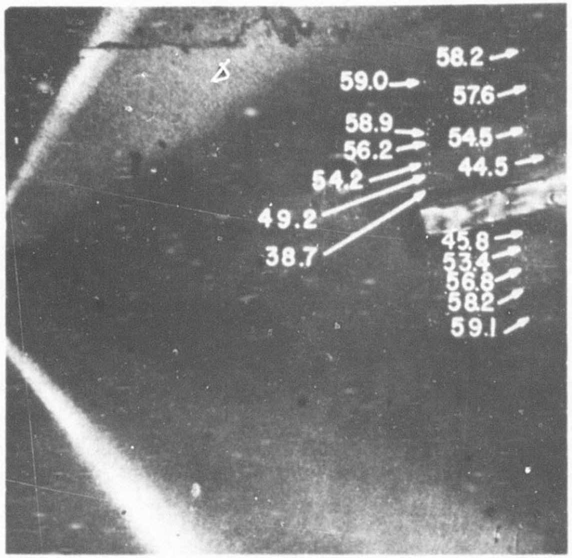
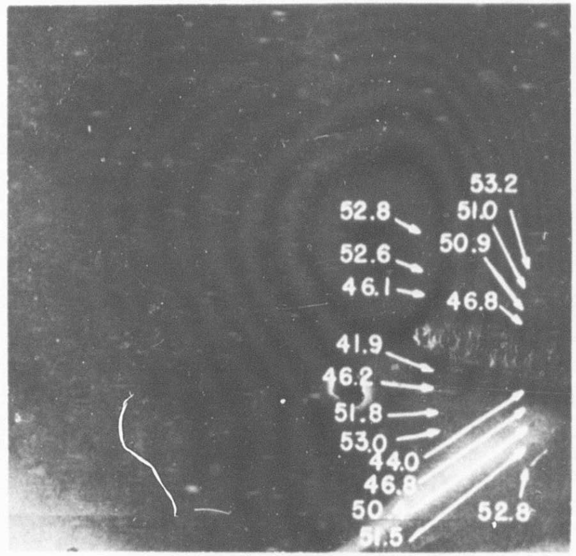


FIGURE 20 - DUAL HARDNESS 9 - 4 STEEL WELD, BACK SIDE



HARD FACE



SOFT BACK

FIGURE 21 - DUAL HARDNESS 9-4 STEEL, REDUCTION OF HARDNESS BY WELDING

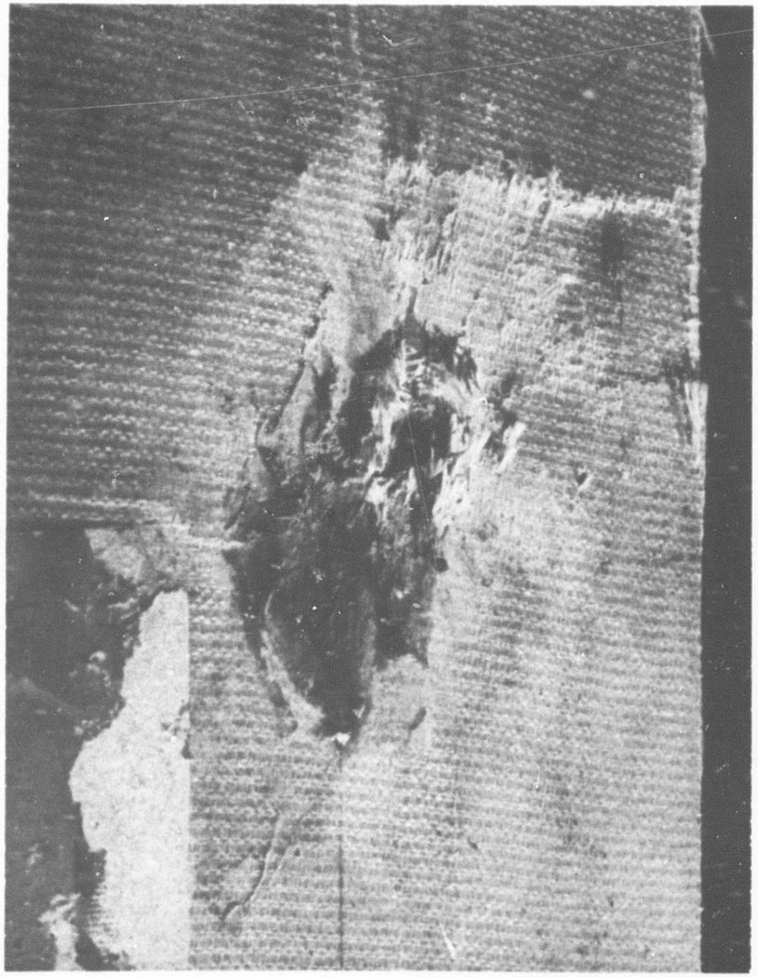


FIGURE 22 - DORON, FRONT

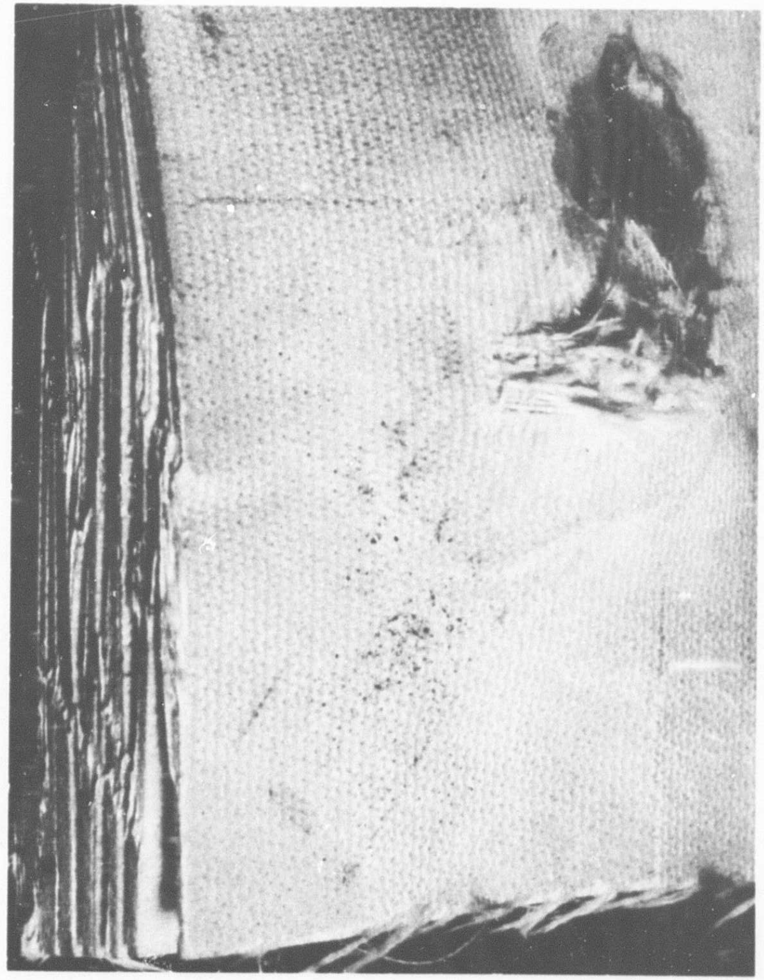


FIGURE 23 - DORON, BACK

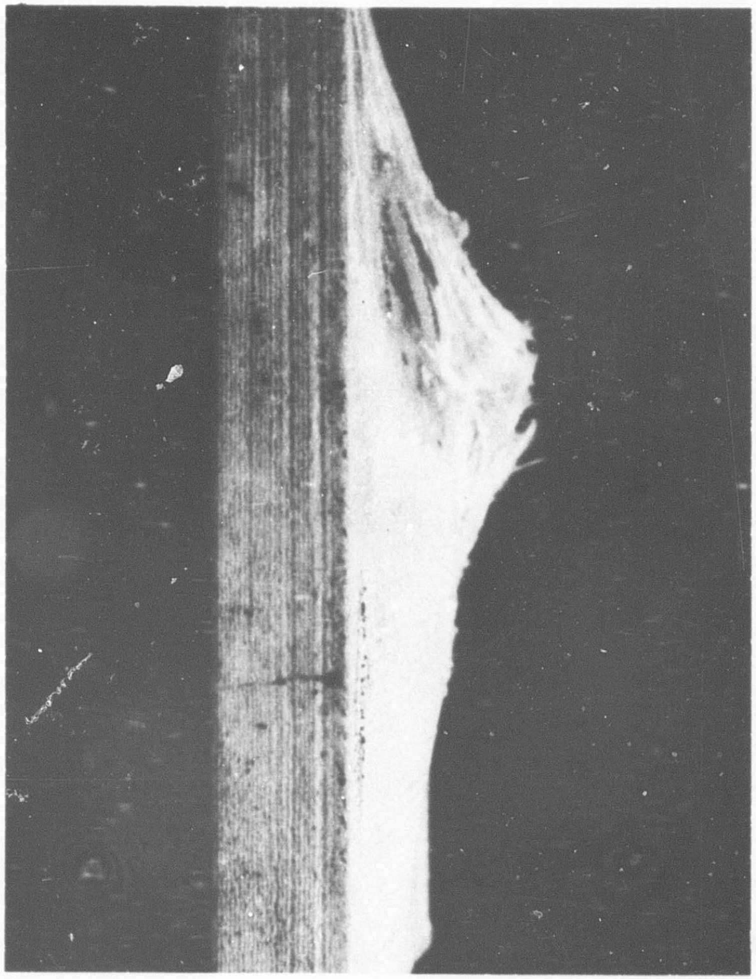


FIGURE 24 - DORON, EDGE

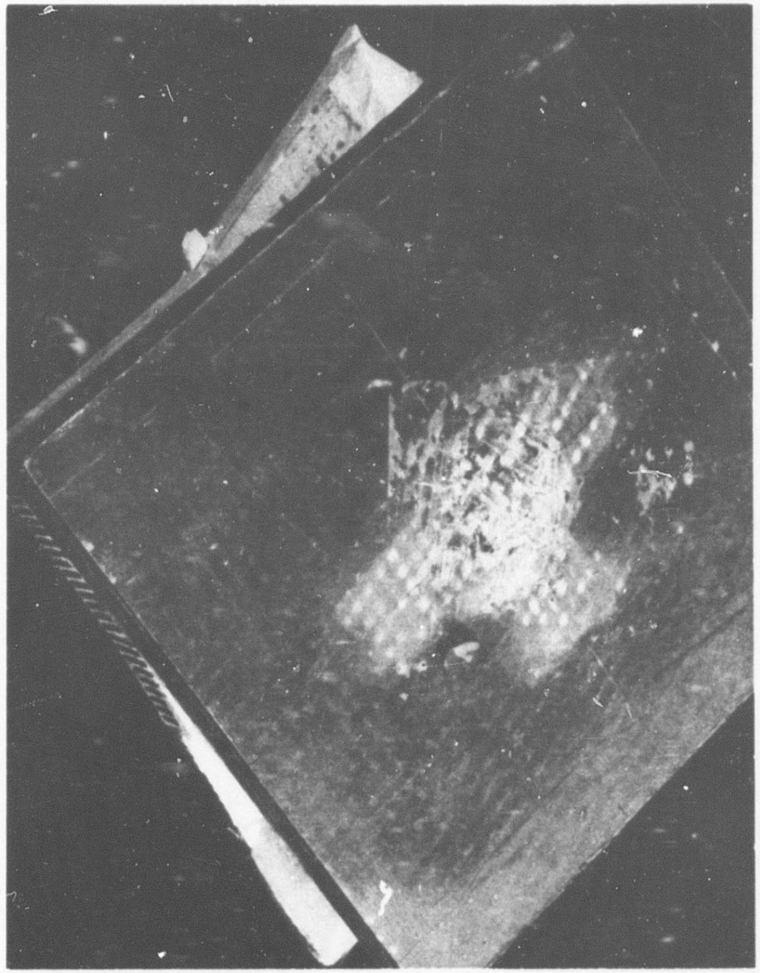


FIGURE 25 - CERAMIC-FACED FIBER GLASS, FRONT

FIGURE 26 - CERAMIC-FACED FIBER GLASS, EDGE

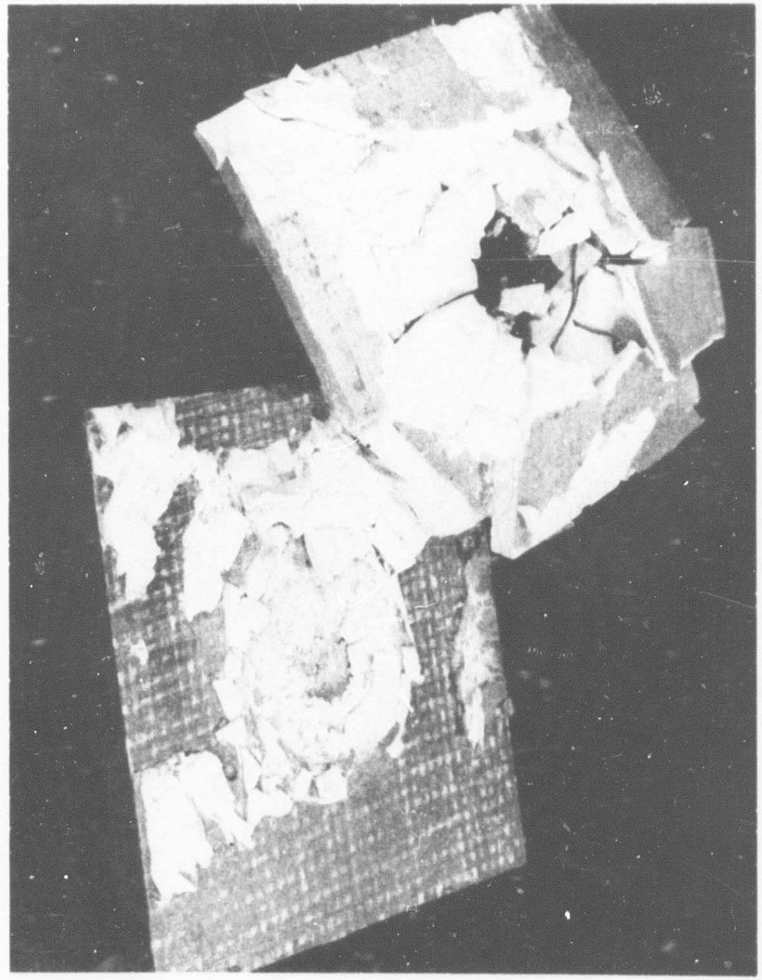


FIGURE 27 - CERAMIC-FACED FIBER GLASS, BACK

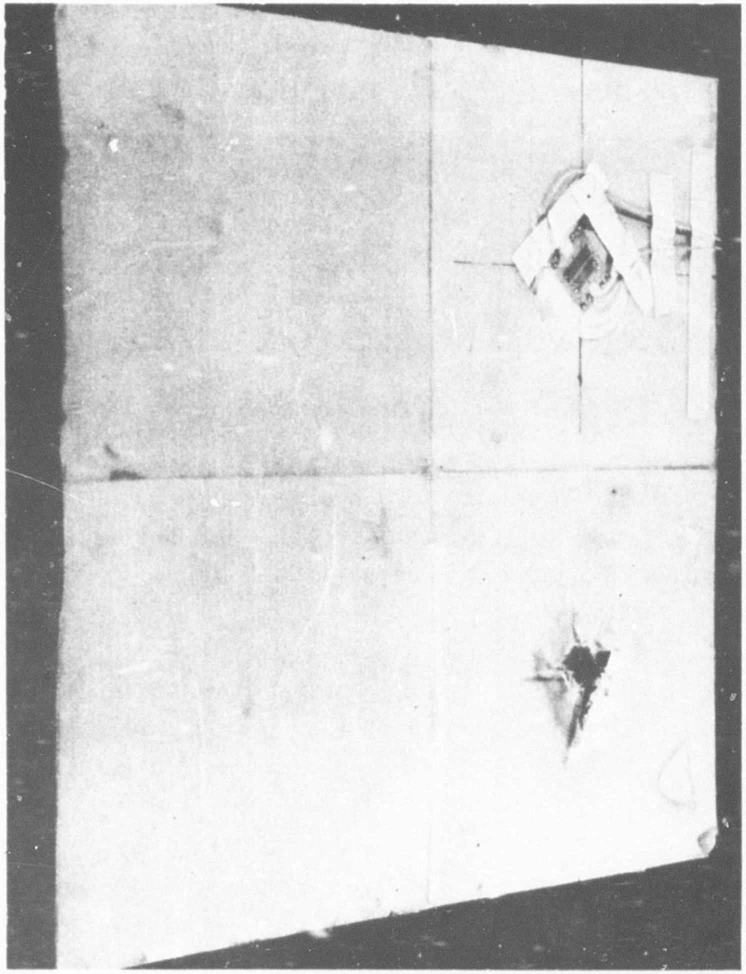


FIGURE 28 - CERAMIC-FACED ALUMINUM, FRONT

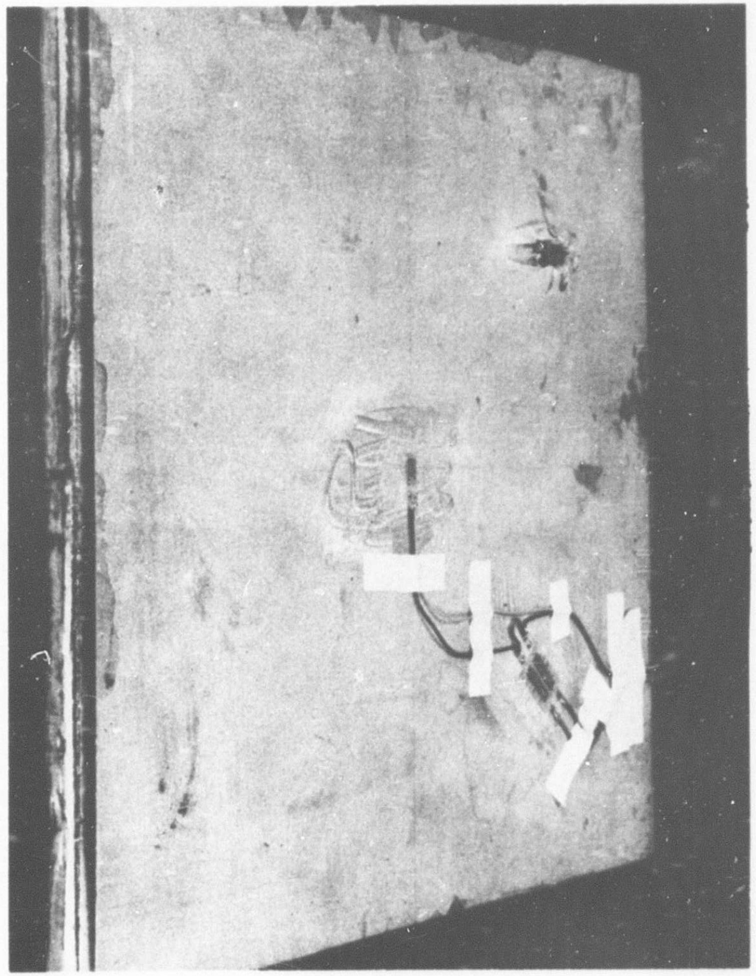


FIGURE 29 - CERAMIC-FACED ALUMINUM, BACK

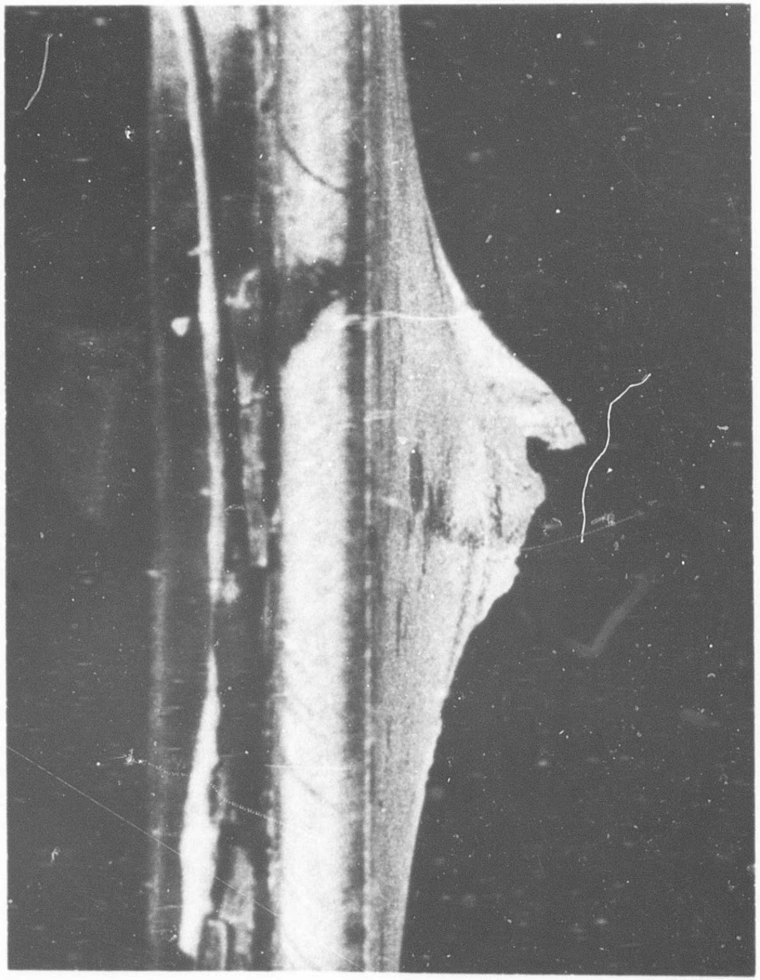


FIGURE 30 - CERAMIC-FACED ALUMINUM, EDGE

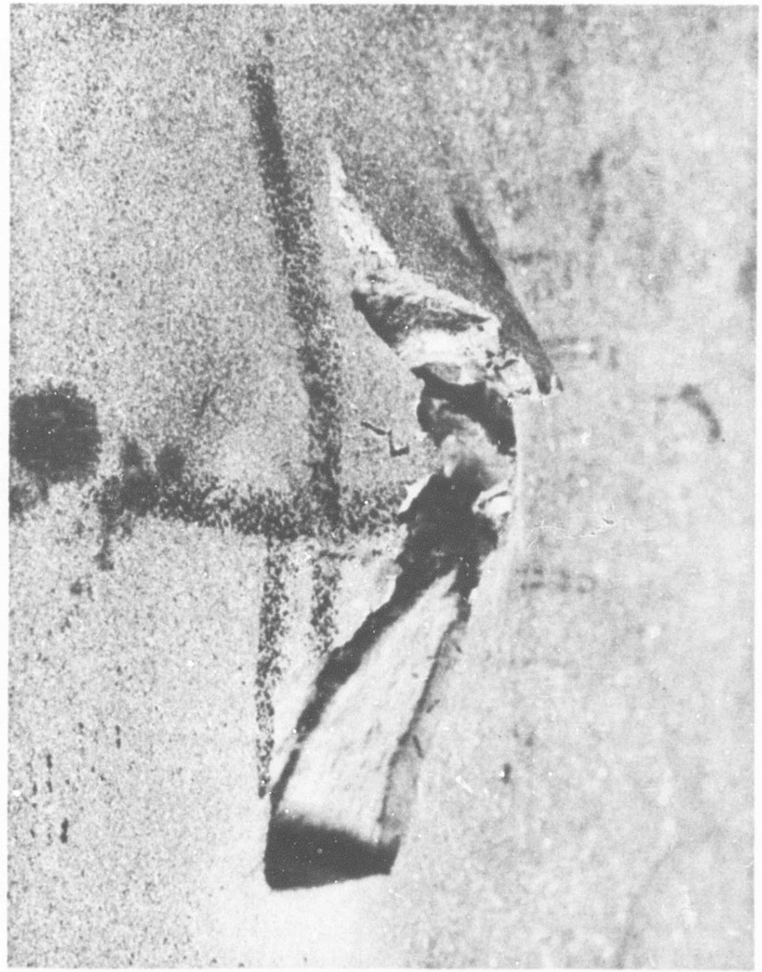


FIGURE 31 - CERAMIC-FACED ALUMINUM, BACK CLOSEUP

FIGURE 32 - TRANSPARENT ARMOR IMPACT DAMAGE

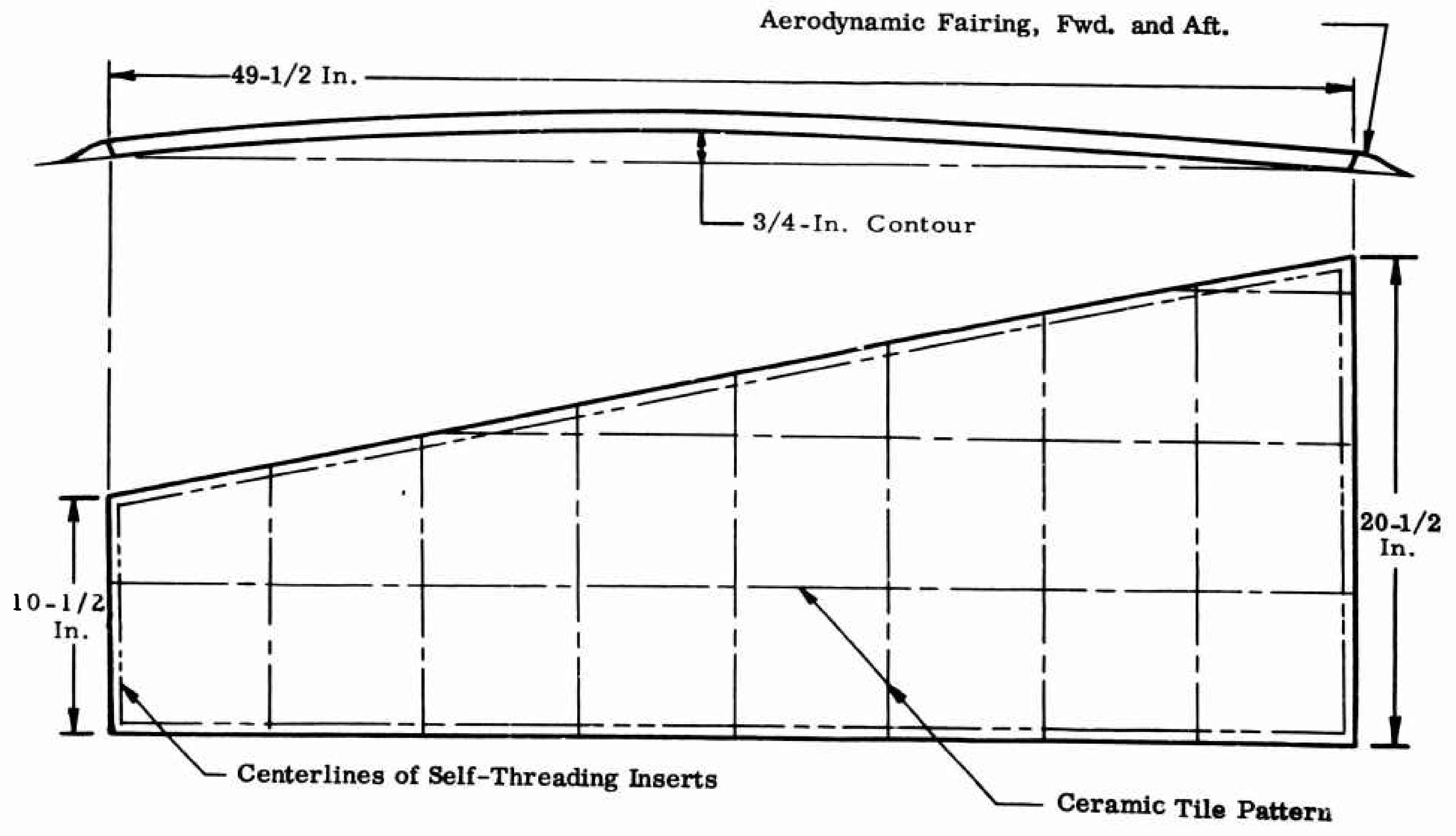


FIGURE 33 - STRUCTURAL ARMOR, FACETED FUSELAGE SIDE PANEL, CERAMIC-FACED FIBER GLASS

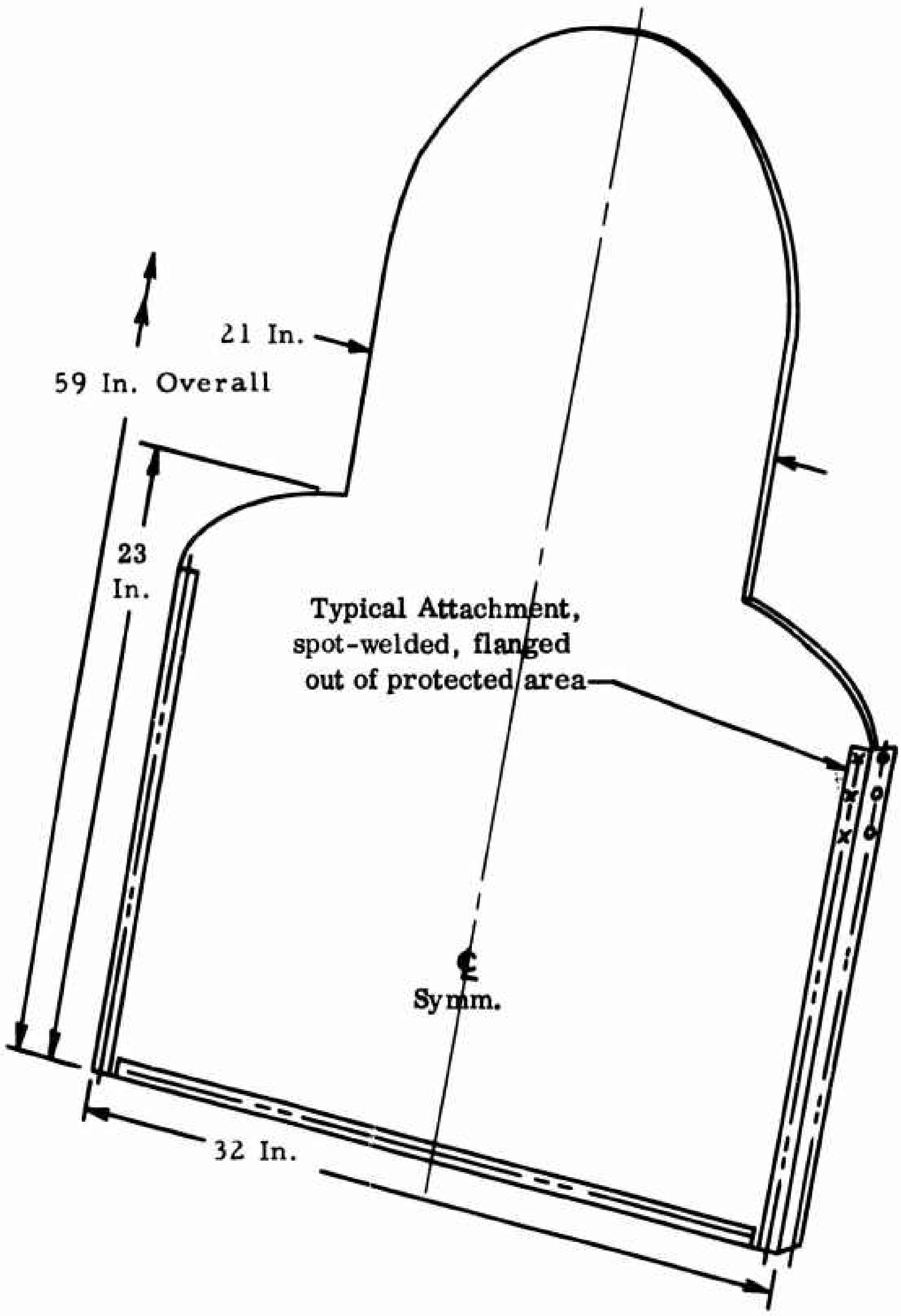


FIGURE 34 - STRUCTURAL ARMOR, FUSELAGE BULKHEAD, DUAL HARDNESS STEEL
(VIEW LOOKING AFT AT BULKHEAD AFT OF SEAT)

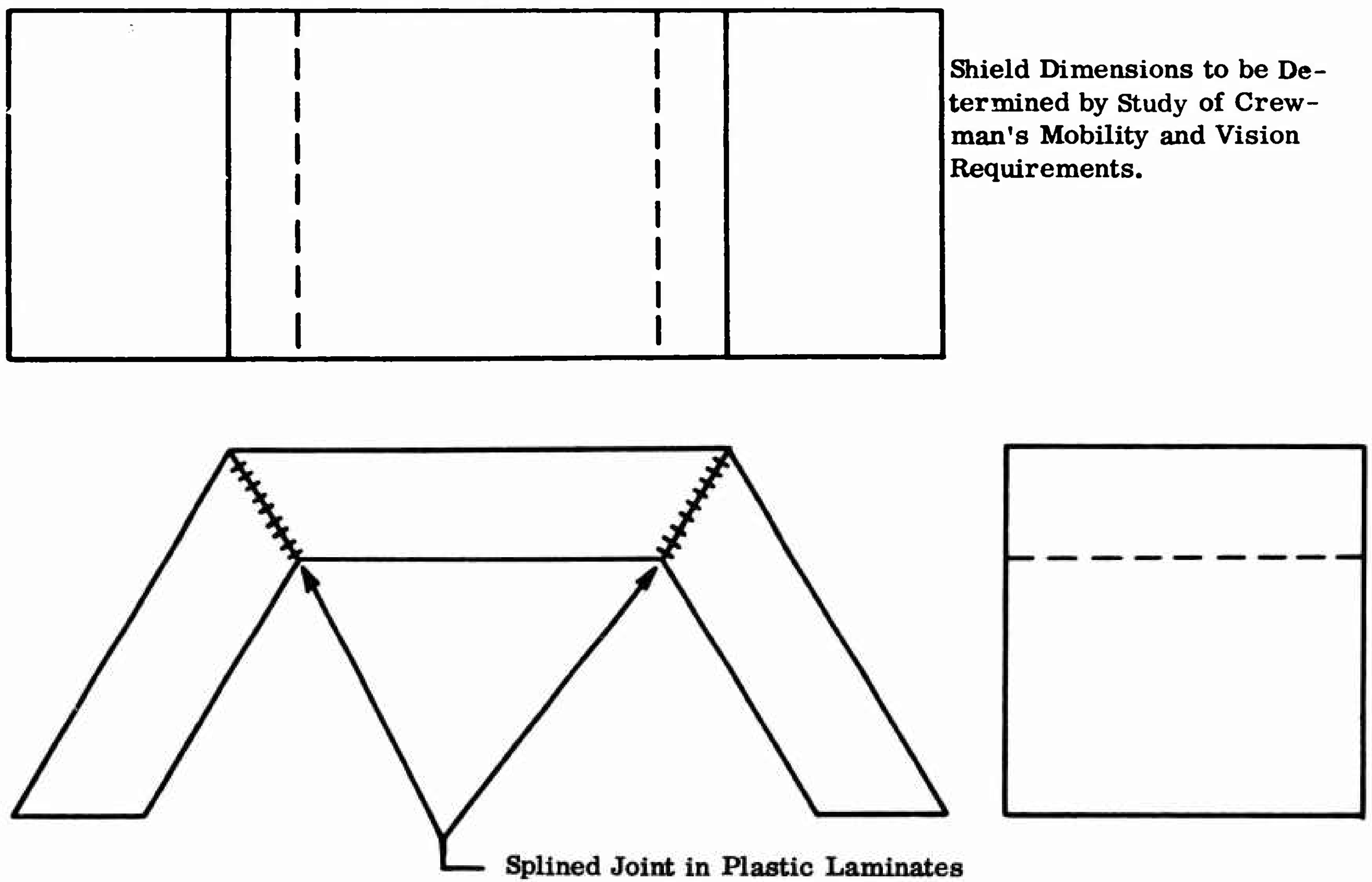


FIGURE 35 - TRANSPARENT ARMOR FACE SHIELD

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APPENDIX I

FUNDAMENTAL PROJECTILE PENETRATION THEORY

The projectile penetration theory presented here allows an understanding of the relationship between the structural properties of armor materials and their ballistic properties. The theory also provides a method of interpolating and extrapolating ballistic performance data with considerable confidence in the results.

Using the straightforward approach of Reference 43, sum the longitudinal forces on a projectile passing through a relatively thick target.

ΣF_x = 0 = inertia force + constant resisting force + resisting force proportional to velocity (strain rate force) + resisting force proportional to velocity squared (kinetic force).

Inertia force on the projectile is expressed as m v dv/dx to allow separation of variables later. The constant force is the resisting stress, σ, times the projectile velocity, v. The force is multiplied by the projectile frontal area to obtain the resisting force proportioned to velocity.

The resisting stress, $\frac{1}{2}K\rho v^2$, from the inertia of the displaced armor material is dependent on the shape of the projectile and on the density and mechanical properties of the armor material. Thus the kinetic force is $\frac{1}{2}K\rho v^2 A$.

The parameter, K, is dependent on projectile shape and on the interaction of the armor material and the projectile material. Reference 43 derives the value of 1 for a flat faced projectile, 1/2 for a sphere, and .122 for a caliber .30AP core. Experience with curve fitting lightweight armor material performance data indicates that the value of K is relatively unimportant compared to σ and C. The armor material density is ρ.

Friction on the sides of an elongated cylindrical projectile is zero until the projectile decelerates to a few hundred feet per second. The inertia forces existing in the displaced target material carry the displaced material away from the body of an undeformed armor piercing projectile. Reference 30 concludes that unconfined glass may crack and displace laterally fast enough to allow free projectile flight inside the growing crack a short time after crack initiation.

Rewriting the summation of forces, we have

$$-\sigma A - CvA - \tfrac{1}{2}K\rho v^2 A = m\ v\ dv/dx$$

By integrating this equation (by separtion of variables) over the armor thickness, t, and over the change (through that thickness) from initial projectile velocity, v_i, to final velocity, v_f, we have

$$\int_0^t -A\ dx/m = \int_{v_i}^{v_f} v\ dv/(\tfrac{1}{2}K\rho v^2 + Cv + \sigma)$$

The solution involves two cases,as noted below:

Case 1 for $C^2 > 2K\rho\sigma$

Areal density, psf $=$ $144\rho tg$, where g $=$ acceleration of gravity.

$$144\ \rho tg = \frac{144mg}{KA} \log_e \left[\left(\frac{\tfrac{1}{2}K\rho v_i^2 + Cv_i + \sigma}{\tfrac{1}{2}K\rho v_f^2 + Cv_f + \sigma} \right) \right.$$

$$\left. \left(\frac{(K\rho v_i + C + \sqrt{C^2 + 2K\rho\sigma})\ (K\rho v_f + C - \sqrt{C^2 - 2K\rho\sigma})}{(K\rho v_i + C - \sqrt{C^2 - 2K\rho\sigma})\ (K\rho v_f + C + \sqrt{C^2 + 2K\rho\sigma})} \right)^{C/\sqrt{C^2 - 2K\rho\sigma}} \right]$$

Case 2 for $C^2 < 2K\rho\sigma$

$$144\ \rho tg = \frac{144mg}{KA} \left[\log_e \left(\frac{\tfrac{1}{2}K\rho v_i^2 + Cv_i + \sigma}{\tfrac{1}{2}K\rho v_f^2 + Cv_f + \sigma} \right) \right.$$

$$\left. + \frac{2C}{\sqrt{2K\rho\sigma - C^2}} \left(\tan^{-1} \frac{K\rho v_f + C}{\sqrt{2K\rho\sigma - C^2}} - \tan^{-1} \frac{K\rho v_i + C}{\sqrt{2K\rho\sigma - C^2}} \right) \right]$$

This complete solution, involving two cases, allows the calculation of armor material density from given parameters σ , c,and K. However, with present technology we cannot yet directly determine the values of these parameters. In practice it is necessary to plot areal density versus assumed values of v_i, v_f, σ , c,and K and find which plot matches actual measured ballistic performance. The values of σ, c,and K were obtained in this manner.

By considering a simplified expression obtained by setting C = 0 and $v_f = 0$ the following expression for ballistic limit may be obtained:

$$V_{50} = v_i = \sqrt{\frac{2\sigma}{K\rho} \left(e^{K\rho At/m} - 1 \right)}$$

By substitution of the infinite series for e^x and using only the first and largest term of the series, we have the approximation

$$V_{50} = \sqrt{2\sigma At/m}$$

Dividing by areal density = 144 ρtg,we have an approximate expression for the efficiency of armor performance in relation to weight of the armor:

$$V_{50}/\text{areal density} = \sqrt{2\sigma A/(tm)}/(144\rho g)$$

APPENDIX II

WEIGHT AND COST MINIMIZATION METHODS

AIRCRAFT PERFORMANCE RELATIONSHIPS

The weight and cost minimization of an aircraft design requires the consideration of aerodynamics, structural analysis, costs of manufacturing and the particular weight distribution of the aircraft. The relations between several measures of performance (as developed in Reference 47) for aircraft powered by reciprocating engines are given below. Following that are two derivations, then a discussion of these relationships to minimize weight and cost.

The first derivation relates the weight increment (cause) added to an aircraft during preliminary design to the resulting increase in weight (effect) of those components which must increase to maintain required performance (such as fuel, power plant, and airframe). The second derivation determines the cost of saving a pound of weight by using more efficient and expensive ballistic armor. Comparison of this cost with the value of weight saved is discussed.

For reciprocating engine powered airplanes, the aircraft performance and the aircraft characteristics are related as follows:

Range	$= 375 \left(\frac{L}{D}\right)_{max} \frac{\eta}{c} \log_e \frac{W_o}{W_1}$
Endurance	$= 37.9 \frac{\eta}{c} \frac{C_L^{3/2}}{C_D} \sqrt{\frac{\sigma S}{W_o}} \left[\left(\frac{W_o}{W_1} \right)^{\frac{1}{2}} - 1 \right]$
Takeoff ground run	$= W_{V_{TO}} / (64 T_E)$
Takeoff over 50-foot obstacle	$= W_{V_{TO}} / (51 T_E)$
Landing over 50-foot obstacle	$= 118 (W/S) / (\sigma C_{L_{max}}) + 400$

where $(L/D)_{max}$ is the aircraft aerodynamic efficiency

η = propeller propulsive efficiency

c = engine fuel consumption pounds per horsepower hour

W_0 = takeoff gross weight

W_1 = final landing weight

C_L = overall lift coefficient

C_D = overall drag coefficient

σ = air density, actual density/sea level standard density

S = wing area

v_{TO} = takeoff velocity, feet per second

T_E = effective or net thrust at $.7v_{TO}$

The above equations are good estimates of performance of aircraft powered by reciprocating engines. The performance of aircraft powered by turboprop and turbojct is also discussed in Reference 47.

AIRCRAFT WEIGHT GROWTH FACTOR

The ratio of a gross weight change or increment divided by the weight increment initiating that change is defined as the growth factor. For the derivation of the weight growth factor accompanying an incremental weight change, use the approximation that certain aircraft elements vary in proportion to small changes in gross weight. These aircraft elements are fuel, power systems,and structural systems. Note that a weight change in these elements represents a change in the fuel consumption, power system efficiency,or structural system efficiency. Other aircraft elements such as cockpit furnishing and required load are independent of the efficiency of those systems previously mentioned.

Gross Weight, GW = constant weight, C + fuel, F + power, P + structure, S.

If the constant weight items are increased by an increment, ΔC, the gross weight must increase until the new constant weight, $\Delta C + C$, equals the original fraction of gross weight:

$$(\Delta C + C)/(\Delta GW + GW) = C/GW$$

Solving for the growth factor from an increment in constant weight,

$$GF_c = \Delta GW/\Delta C = GW/C$$

If a variable weight item changes by an increment, an efficiency change is evident. This change in efficiency also changes the growth factor from that associated with a change in constant weight items.

The gross weight must increase to reduce the fraction of constant weight items by the same increment that the variable weight fraction increases. Note here that it is not important which variable weight element changed in efficiency.

$$F/GW = C/GW - C/(GW + GW)$$

Solving for ΔGW and dividing by ΔF, we have the growth factor from an increment in variable weight. The subscripts indicate fuel, f, power system, p, and structure, s.

$$GF_f = \Delta GW/\Delta F = GW/(C - \Delta F)$$

$$GF_p = GW/(C - \Delta P)$$

$$GF_s = GW/(C - \Delta S)$$

Where factual data is available to indicate that other relationships are applicable, a more accurate estimate may be made by including the available data. For example, in a particular case, it may be determined that airframe strength is adequate to carry additional load without an increase in airframe weight. For this example, airframe weight would simply be included in the constant weight items.

COST OF WEIGHT SAVED

For an aircraft preliminary design, a derivation is presented of the additonal or indirect weight and cost increase encountered when an airplane configuration is resized to carry added armor weight and to maintain the specified performance.

To achieve the minimum weight of aircraft armor choose the configuration requiring the least area of armor for the desired protection;then choose the armor material having the minimum weight per square foot to provide the desired ballistic protection. Then check to see that the armor selected is adequate to carry the structural loads in the area of the armor.

The aircraft growth factor for a particular set of applicable assumptions will make the aircraft empty weight increase (resulting from the addition of armor weight) larger than the actual added armor weight. But the relative weight efficiency of the candidate armor materials will remain in the order of their weight per square foot to provide the required ballistic protection. For the case when the armor material selected is not adequate to carry existing structural load, either added structural material or a different armor material must be used.

In contrast to the above choice for least armor weight, the choice of armor material for least cost is complicated by the necessity of considering ballistic efficiency, aircraft growth factor, armor material cost and aircraft average cost. The choice of armor material for least cost involves selecting the material with the least cost per pound and calculating the cost per pound saved using the next most efficient armor material (with a higher cost per pound of armor). Then the cost per pound saved is compared with the value or worth of a pound saved. The following derivation is similar to that of Reference 32.

$$\text{Cost Increase} = W_2 \times d_2 - W_1 \times d_1$$

$$\text{Cost/lb} = \frac{\text{Cost increase}}{\text{Weight decrease}} = \frac{W_2 d_2 - W_1 d_1}{W_1 - W_2} = \frac{d_2 - d_1\left(\frac{W_1}{W_2}\right)}{\frac{W_1}{W_2} - 1}$$

where d = cost dollars/lb.

W = weight for required ballistic protection. Subscript 1 refers to the standard or reference material. Subscript 2 refers to a more expensive and more efficient material.

For an example of the use of the derived formula, consider material 1 at \$2/pound which weighs twice as much as material 2 at \$40/pound for the same ballistic protection.

$$\text{Cost/pound} = \frac{\$40 - \$2(2)}{2 - 1} = \$36\text{/pound saved.}$$

OVERALL WEIGHT AND COST MINIMIZATION

We must compare the cost of saving a pound of weight with the resulting value of saving that weight in context with an aircraft preliminary design which maintains the required aircraft performance. For example, the cost (plus transportation) of fuel saved is obviously the worth of a decrease in aircraft fuel required resulting from increased power plant efficiency or decreased range requirements.

To find an increment (caused by added armor weight) in the delivered cost of aircraft which must maintain the required performance, multiply the added weight times the applicable growth factor times the average cost per pound of the particular aircraft. The growth factor has been discussed previously. Further discussion is presented in References 18, 32, 35, and 52.

The cost of various types of aircraft are discussed in References 5, 15, 31, 32, 55, and 59.

APPENDIX III

ARMOR MATERIAL STRUCTURAL PROPERTIES

ALUMINUM PLATE

Alloy	2024-T4	7178-T6	5456-H321	5083-H113
F_{tu}, ksi	T 64	T 84	T 46	T 44
F_{ty}, ksi	T 40	T 73	T 30	T 28
F_{cy}, ksi	T 43	T 76	33	29
F_{su}, ksi	40	50	27	25
F_{bru}, ksi (e/D=2)	124	151	84	80
F_{bry}, ksi (d/D=2)	74	111	53	50
Elongation %	12	8	12	12
E_t, 10^6psi	10.5	10.3	10.2	10.2
E_c, 10^6psi	10.7	10.5	10.4	10.4
G, 10^6psi				
Density, lb/in.^{3}	.1	.102	.096	.096
Hardness				
Class of Structure	P	P	P	P

E = Young's modulus
F = allowable stress
G = shear modulus
L = longitudinal
LT = long transverse (forging)
P = primary structure
S = secondary structure
T = transparent structure, transverse

N.S. = Not specified. The values not listed here are not defined by this specification - use values on an alloy with similar F_{tu} and F_{ty}.

Subscripts

c = compression
br = bearing
t = tension
u = ultimate
y = yield

ALUMINUM PLATE

Alloy	2219-T81	M693	MIL-A-45225 (Forging)	MIL-A-46027 3/4 to 2 in. th.
F_{tu}, ksi	T 61	(Similar to 2219-T81)	L 42 LT 40	L 45
F_{ty}, ksi	T 44		L 33 LT 32	L 37
F_{cy}, ksi	T 47		N. S.	N. S.
F_{su}, ksi	35			
F_{bru}, ksi(e/D=2)	119			
F_{bry}, ksi(e/D=2)	88			
Elongation %	6		L 7 LT 4	9
E_t, 10^6psi	10.5			
E_c, 10^6psi	10.8			
G, 10^6psi				
Density, lb/in.^{3}	.102			
Hardness				
Class of Structure	P	P	P	P

B = average, not minimum values
* = not suitable for long time stress application. endurance limit at 10^8 cycles is 8000 psi

	MAGNESIUM		STEEL				
	13Li -6A	LA141 *	MIL-S-11356 Class I	MIL-S 12560	MIL-S--13823	4340	H-11
F_{tu}, ksi	31B	18	140	160	165	150	270
F_{ty}, ksi	27B	13	N. S.	N. S.	N. S.	132	220
F_{cy}, ksi	23B	18				145	250
F_{su}, ksi						95	164
F_{bru}, ksi (e/D = 2)		40				287	510
F_{bry}, ksi (e/D = 2)		25				218	364
Elongation %	18	10				9	4
E_t, 10^6 psi	6.5	6.2	28.0	28.0	28.0	29	28
E_c, 10^6 psi	6.5	6.5	28.0	28.0	28.0		
G, 10^6psi						11	
Density, lb/in^3	.049	.049	.28	.28	.28	.283	.28
Hardness		RE 65	RC 32 1/4- 1 1/4th.	RC 36.5 1/4 - 1/2th.	RC 36.5 1/4-1/2th.	RC 34	RC 54
Class of Structure	P	P	P	P	P	P	P

NOTE: H-11 steel is currently used in basic aircraft structure.

TITANIUM

	7 A1-12ZR	8A1-1MO-1V	4A1-4MN	6A1-4V	5A1-5Sn-5Zr
F_{tu}, ksi	130	130	155	160	120
F_{ty}, ksi	120	120	140	145	110
F_{cy}, ksi			140	154	
F_{su}, ksi			95	100	
F_{bru}, ksi(e/D = 2)			275	286	
F_{bry}, ksi (e/D = 2)			230	232	
Elongation %	10	10	7	10	10
E_t, 10^6psi	16.5	18.5	15.5	16.0	16.0
E_c, 10^6psi			16.0	16.4	
Density, lb/in.^{3}	.165	.158	.163	.160	.166
Hardness					
Class of Structure	P	P	P	P	P

	ACRYLIC		POLYCARBONATE
	PLEX 55 AS CAST	PLEX 55 STRETCHED	
F_{tu}, ksi	10.1	11.5	9-10.5
F_{cy}, ksi	19.5	16.9	11
F_{su}, ksi	9.5	9.5	9.2
F_{bru}, ksi			
F_{bry}, ksi			
Elongation%	3.5	2.6	60 - 100
E_t, 10^6psi			
E_c, 10^6psi	.5	.52	.24
G, 10^6psi			
Density, lb/in^3	.043	.043	.043
Hardness			
Class of Structure	T	T	T

	FIBER GLASS			DORON	FIBER GLASS NON-DELAMINATING
	181-E EPOXY	181-E POLYESTER	143-E POLYESTER		
F_{tu}, ksi	97(0)	38(0) 17.8(0)	75(0) 8.9(0)	51	17 Without Ceramic Face
F_{cy}, ksi	30(0) 31.6(90)	45(0) 19(90)		9.6	3.6 Without Ceramic Face
F_{su}, ksi					
F_{bru}, ksi					
F_{by}, ksi					
Elongation %					
E_t, 10^6psi					
E_c, 10^6psi		2.9(0) 2.8(90)	4.9(0) 1.5(90)	3.4	1.0 Pri. .88 Secd.
G, 10^6psi					
Density, lb/in^3	.065	.065	.065		
Hardness					
Class of Structure	P	P	P	S	P

(90) = Angle from longitudinal axis, degrees
143-E = 143 weave of E glass laminated with same orientation for al[illegible]es
181-S = 181 weave of S glass laminated with same orientation for all plies

	CHEMCOR ABRADED	CHEMCOR UNBRADED	SODA-LIME ANNEALED ABRADED	SODA-LIME ANNEALED UNABRADED	SODA-LIME TEMPERED ABRADED	SODA-LIME TEMPERED UNABRADED
F_{tu}, ksi *	40.4	53.3	6.9	11.3	22	29.6
F_{cy}, ksi						
F_{su}, ksi						
F_{bru}, ksi						
F_{by}, ksi						
Elongation %						
E_t, 10^6psi	12	12	10	10	10	10
E_c, 10^6psi						
G, 10^6psi	4.9	4.9				
Density, lb/in.^{3}	.088	.088	.091	.091	.091	.091
Hardness						
Class of Structure	T	T	T	T	T	T

* Modulus of Rupture

ALUMINUM OXIDE

	ZI-4	AD-85	AD-94	AD-99	AD-995	A402	GM_cB-352
F_{tu}, ksi			25-27	34-35			
F_{cy}, ksi	135.5	292.4	308	315.4	308.2		350
F_{su}, ksi							
F_{bru}, ksi							
F_{by}, ksi							
Elongation %							
E_t, 10^6psi							
E_c, 10^6psi	15.7	33.21	41.7	50.79	51.46	53	50
G, 10^6psi							
Density, lb/in^3	.096	.124	.131	.138	.138	.137	.137-.138
Hardness	55.1*	76.4*	77.8*	82.7*	83.3* 2000**		
Class of Structure	S	S	S	S	S	S	S

(*) = Rockwell 45N scale (**) = Knoop Hardness-100 scale

	BORON CARBIDE	SILICON CARBIDE
	HD	Self-Bonded
F_{tu}, ksi	50	25
	420	82
F_{cy}, ksi		
F_{su}, ksi		
F_{bru}, ksi		
F_{by}, ksi		
Elongation %		
E_t, 10^6psi		
E_c, 10^6psi	65	38 - 68
G, 10^6psi		
Density, lb/in^3.	.091	.112
Hardness	2800**	2740**
Class of Structure	S	S

(*) = Rockwell 45N scale

(**) = Knoop Hardness-100 scale

AVAILABLE ARMOR MATERIAL COST PER POUND

Aluminum oxide and Aluminum		2.15
Roll bonded ausformed steel,	D-6 to Hy Tuf	2.30
	H-11 to 5M-21	2.90
Aluminum oxide and doron		3.50
Titanium 6A1-4V Eli armor		5.55
Aluminum oxide and rigid fiber glass		7.00(Est.)
Boron carbide and aluminum		38.25
Boron carbide and doron		39.60

Note: Costs are approximate only and must be checked before use in design.

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13 ABSTRACT The use of selected armor materials as primary, secondary, and transparent aircraft structure was studied and determined to be both feasible and practical. Solutions to the several problems in design, fabrication, and repair of both opaque and transparent armor are presented. A comparison indicates that use of structural armor instead of kit armor saves 10 lbs. to 62 lbs. for a single-place aircraft subjected respectively to sporadic and to concentrated small-arms fire throughout the lower hemisphere. In addition, an aircraft design using structural armor will have fewer total parts and will have a lower cost unless an unusual pricing structure exists. Ceramic-faced, non-delaminating fiber glass is recommended for minimum weight, for the most effective maintenance of ballistic and structural properties over extended service life, and for faceted contours. Dual hardness steel armor is recommended for low first cost, for maximum strength, for high hit-density, and for flat or slightly contoured panels.

Further study on the subjects of crewman mobility and vision requirements, of increased resin shear strength in fiber glass, of manufacturing processes for ceramic tile and of stop-drilling cracks in dual hardness steel armor is recommended.

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14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Armor						
Aircraft Seats						
Airplane Panels						
Transparent Panels						
Metal Panels						
Ceramic Materials						
Glass Textiles						
Composite Materials						
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Structural Materials						

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