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AN APPLICATIONS STUDY OF ADVANCED COMPOSITE MATERIALS TO THE C-130 CENTER WING BOX

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ABSTRACT

An application study of advanced filamentary composite materials to the C-130 center wing box was performed. The study included three design approaches; (1) composite reinforcement of the existing box while installed on the aircraft, (2) composite reinforcement during fabrication in accordance with a new design, and (3) all composite design. Consideration of cost, manufacturing feasibility and reliability led to the choice of approach (2). This particular approach enabled a 13 percent weight savings with boron/epoxy reinforcement of aluminum as compared to the current aluminum design. Cost of the composite material largely determined the increased cost over that of the current aluminum design. Preliminary fatigue analysis indicated the life of the composite reinforced box equal to or greater than that of the current aluminum box.

A demonstration program designed to provide flight experience for two C-130 aircraft equipped with composite reinforced center wing structure is outlined.

FOREWORD

This is the final report for NASA Contract Number NASI-9540 which was entitled "An Applications Study of Advanced Composite Materials to the C-130 Center Wing Box". The program was accomplished during a time period from November 1969 to June 1970. The program was monitored by Mr. George W. Zender, NASA Langley Research Center.

The individuals who made technical contributions to the program and their areas of activity are as follows:

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SUMMARY

Advanced filamentary composites offer great promise for major improvement in aircraft structural weight. The use of composites such as boron/epoxy and graphite/epoxy in conjunction with conventional metallic structures is an attractive means to enable early realization of much of the benefit of composites, with a minimum of both risk and development cost. It is believed that the high-cost composites can be applied to a basically complete metal structure in small amounts and in judicious locations to provide large return for low investment.

An applications study of the use of advanced composite materials as reinforcement for the center wing box of the C-130 aircraft was performed. The program established the effectiveness of advanced composite materials in providing improved structural performance over conventional aircraft structure. The C-130 center wing box provided an ideal component for the study, since a previous re-design of the box had been accomplished utilizing an aluminum build-up. This allowed demonstration of relative structural weights, manufacturing costs and producibility of the aluminum-reinforced box and the composite-reinforced boxes.

Certain material selection studies were initially accomplished to determine optimum composite and metallic material combinations for static and fatigue strength, stiffness, and thermal compatibility of the materials. Fiberglass-reinforced metallic materials were examined but proved inefficient for reinforcement because of the low modulus of elasticity of fiberglass. The boron and graphite-reinforced metallics proved to be very efficient. Graphite composites showed more efficiency when stiffness or stability considerations governed; boron composites utilized as reinforcement materials were more efficient when material strength was critical. In the C-130 wing box, fatigue strength requirements predominate in setting allowable stress levels. The graphite composites proved to be less efficient for the reinforcement of fatigue-critical structure due to the adverse effect of residual tensile thermal stresses induced in the metal from the bond operation. The boron unidirectional composite has thermal coefficients of expansion nearer those of titanium or aluminum than does graphite. Boron is therefore somewhat more

efficient for reinforcement than graphite for fatigue critical structure.

The study compared three design approaches with respect to weight savings and cost and manufacturing effects associated with using composites:

1) Reinforcement of the existing wing box while installed on the aircraft.
2) Reinforcement during fabrication in accordance with a new design.
3) All-composite design.

The all-composite approach was included as a basis for comparison.

The weight savings achieved on the reinforced designs ranged from a few percent to a maximum of 23 percent. These weight savings could have been larger if the design had been governed by either stiffness or static strength. The residual tensile stresses induced in the metallic material after bonding diminishes substantially the weight saving on fatigue-critical composite-reinforced structure. All-composite designs indicated weight savings as high as 38%.

In the selection of the preferred design approach, careful consideration of manufacturing feasibility and reliability led to the choice of the approach in which the composite reinforcement is applied during fabrication. The box would be fabricated from the wing skin and hat stiffener extrusions currently in use but would be machined to smaller cross sections prior to the addition of composite material. This particular design enabled a 13 percent weight savings for boron reinforcement and a 9 percent weight savings for graphite reinforcement. The boron-reinforced design was selected for more detailed study. A preliminary fatigue analysis substantiated the adequacy of the design and verified the weight savings.

The next logical steps in the substantiation of composite-reinforced structure is to fabricate actual component hardware and obtain static, fatigue and lone-term service evaluations. Accordingly, a program has been planned to enable the required hardware demonstration tests. The hardware program logically consists of three phases:

<u>Phase I</u> - Development of the analytical, design, and manufacturing information necessary to proceed with the detail design.

<u>Phase II</u> - Detail design.

Phase III - Fabrication and test of three center wing box structures. The first box would be subjected to static and fatigue tests. Two boxes would be installed on Air Force operational aircraft and evaluated in service for an extensive period.

The C-130 center wing box provides an excellent medium for evaluation of composite-reinforced structures for the following reasons:

1) The aluminum technology is readily at hand for comparison;
2) Appropriate analysis techniques are available and computerized;
3) Test fixtures will be available and test loads and load spectra are developed;
4) The C-130 offers economic advantage, since it is already undergoing modification;
5) The Air Force has indicated willingness to cooperate; and
6) The C-130 design lends itself to a minimum risk approach.

Completion of this proposed hardware development program will:

1) Demonstrate the effectiveness of boron composite reinforcements in enhancing the performance of conventional aircraft structures;
2) Provide the analytical, design, and manufacturing development background needed as a firm base for flight demonstration of a composite-reinforced structure;
3) Accomplish the actual design of a composite-reinforced structure;
4) Prove the feasibility of manufacture of a practical composite-reinforced structure;
5) Demonstrate the effectiveness of quality assurance procedures in manufacture and service;
6) Prove the operational serviceability of a composite-reinforced structure; and
7) Demonstrate the near-term effectiveness of composites in primary structure without the risks attending an all-composite approach.

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1.0 INTRODUCTION

The utilization of advanced composite materials as a reinforcement medium for conventional metallic structure has proved to be a useful structural concept (Refs. 1 and 2). It has been shown that significant improvements in strength and stiffness can be achieved for simple structural shapes (Refs. 2 and 3). The weight savings possible by utilizing composite reinforced metallic material on an actual structural component had not been determined until this study was accomplished.

This composite reinforcement concept builds on the existing technology for metal aircraft structures which provides better structural reliability for near term applications of composite materials. It has the highly desirable feature that integrity can still be maintained in the event of complete ineffectiveness of the reinforcing composite materials. That is, a high percentage of design limit load can be maintained with only the basic metallic structure sustaining the load. Hence, this concept provides the means for obtaining near term structural weight savings by using composites in primary structure where the risk might be too high for all composite technology. This brings the use of advanced composites for commercial and transport aircraft structure much nearer reality.

The C-130 center wing box provided an ideal component for the applications study since a redesign was previously accomplished on the box to improve the fatigue resistance. Thus, the aluminum redesign provided an excellent base line for comparison with the composite reinforced designs. Weight, cost and manufacturing comparisons could be made on a direct basis. Background data on the C-130 center wing is presented in Appendix A.

This program was accomplished in three phases: Material Selection Studies, Applications Analysis and the Demonstration Program Plan. The design data and parametric design curves generated during the Material Selection Studies are presented in Appendix B. These data are useful in assessing the payoffs involved in utilizing boron or graphite reinforced aluminum and titanium structure. The actual design studies were accomplished during the Applications Analysis Phase, and a summary of the configurations exercised is given in Appendix C. The design selected for further study is described in Section 2.0. In Section 3.0, Demonstration Program Plan, an outline and schedule is presented for a program to design, fabricate and ground and flight test three (3) reinforced C-130 center wing boxes.

2.0 APPLICATIONS ANALYSIS

During the Applications Analysis phase of the program, the weight, cost and manufacturing payoffs which could be achieved from the application of advanced composite materials to the C-130 center wing box were determined. Three different design approaches were utilized. The first approach, the Reinforcement Concept, produced a design which reinforced the existing C-130E wing box without removal of the box from the aircraft. The second approach, the Compound Composite Concept, resulted in six new wing box designs having composite material used in conjunction with, or as reinforcement for, metallic materials. The third approach, the All Composite Concept, considered two all-composite designs which were primarily utilized as a baseline for comparison of the reinforced designs. For the second and third design approaches, the old C-130E wing boxes would be discarded and new boxes fabricated.

For each concept investigated, a projected cost of $300 per pound for boron pre-impregnated material and $25 per pound for graphite was used. These material costs are Lockheed costs based on forecasted cost trends for late 1970. Generally speaking, the largest percentage of the total cost increase for the composite reinforced wing over the previously designed aluminum wing was for the material expense of the composite material. These material costs were the governing parameter in the cost studies since labor costs and other costs were very similar for all the metal/composite designs. An example of the cost data calculations is given in Appendix E.

After the initial preliminary design phases of the program were completed, a selection was made of the concept which would be most feasible to continue for a demonstration article. The predominate factor in the decision was the total cost of a hardware demonstration program rather than the cost of a production program. As assessment of the ranking of the concepts for a production environment can be obtained from Table IV. (Appendix C). In the following sections, the final configuration, Concept II-A, will be discussed in more detail and the substantiating analysis presented.

2.1 Final Wing Box Configuration

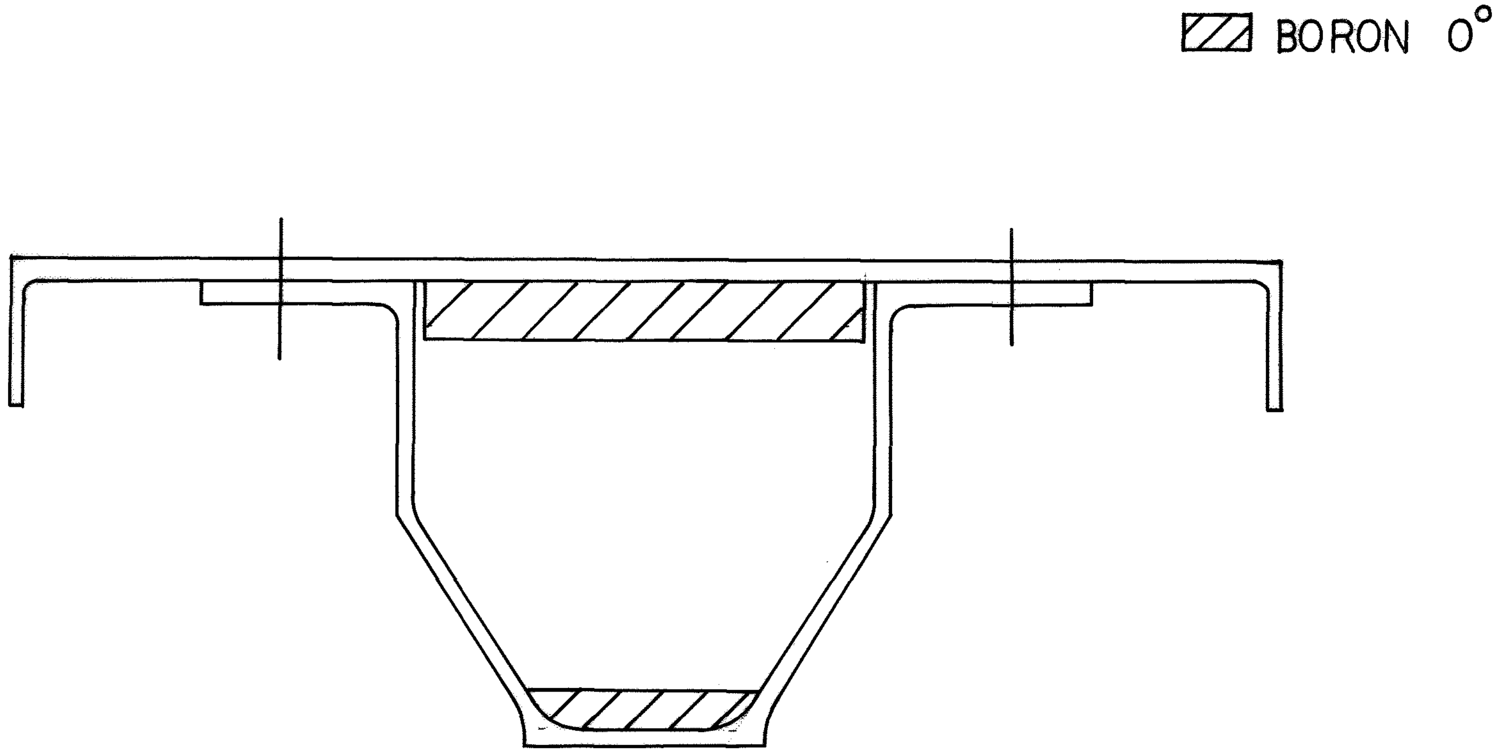
Concept II-A, with boron composite reinforcement, was selected from the various concepts considered. In order to obtain a meaningful and direct comparison between the aluminum reinforced wing box and the composite reinforced wing box, the same basic structural configurations should be used. This was a strong point in favor of configurations I-A and II-A. Concept I-A was deemed unfeasible because of excessive costs and manufacturing problems. Boron reinforcement was selected over graphite even though the costs were higher because the material properties and allowables have been substantiated to a greater extent.

Configuration II-A makes use of the C-130E aluminum skins and stringer extrusions. The stringer crowns are machined to a 0.08 inch thickness. The access cut-outs have the same shape as those on the aluminum reinforced wing box, i.e., oval instead of square. Composite material is added to the skin under the hats and to the crown of the stringer (Ref. Figures 1 and 2). The amount of composite required to reduce the stress in the aluminum to the required levels is discussed in Appendix B. The ribs and spars are the same as those of the aluminum reinforced design.

A new rainbow (end) fitting is not needed for Concept II-A (Ref. Fig. 3). The unidirectional composite in the stringer crown and the skin can be tied into the existing rainbow fitting to give a straight, continuous load path. Titanium shims would have to be bonded between composite plies in order to furnish sufficient bearing strength in the joints in the rainbow fitting area. This is illustrated in Figure 3.

Composite material was tried as reinforcement around the access door cut-outs. A combination of 0° and $\pm 45^{\circ}$ ply orientations is necessary due to high local shear flows which are produced by the redistribution of stringer loads. The aluminum skin would be a constant thickness in the cut-out region and the composite doubler would be added to the external surface. This reinforcement around the access doors only resulted in a small weight savings. For this reason, it is not recommended for use.

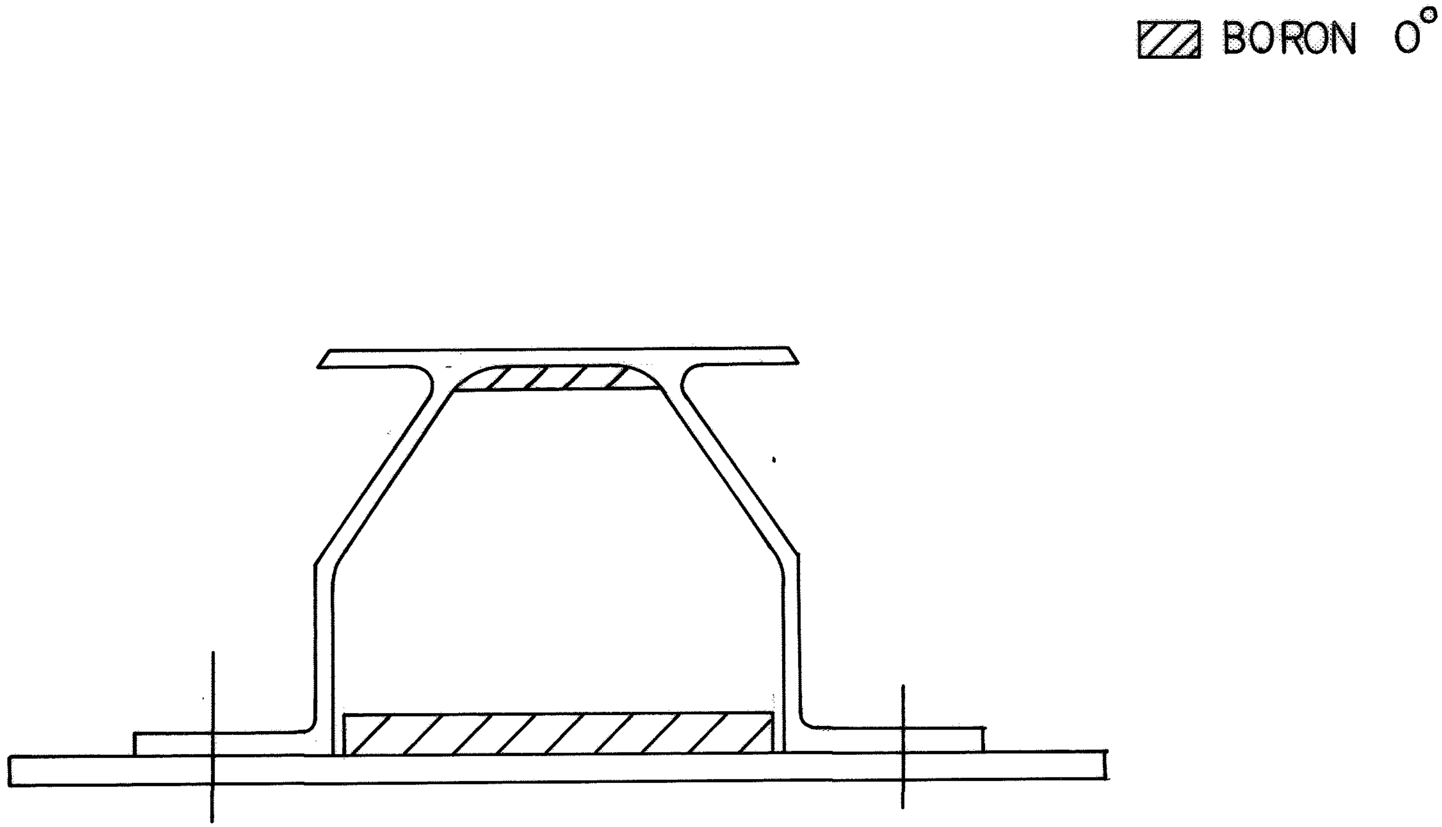
IIA



UPPER SURFACE

FIGURE 1

IIA



LOWER SURFACE

SECTION AT W STA 68

FIGURE 2

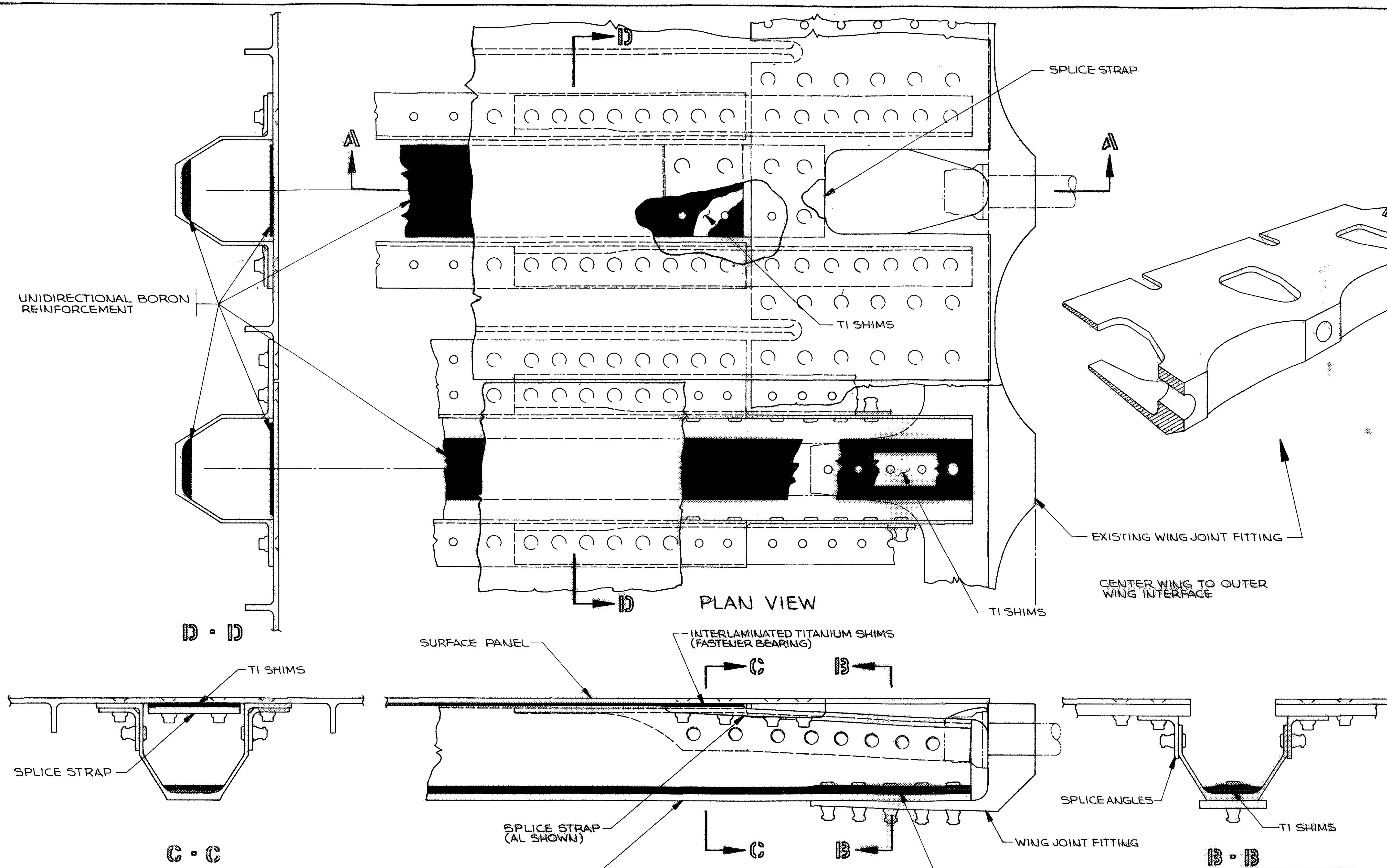


FIGURE 3

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

COMPOUND COMPOSITE 2-A
C-130 CENTER WING
INTERFACE - UPPER SURFACE
PANEL & WING JOINT FITTING

2.2 Fatigue Analysis

The fatigue analysis for the final design concept, II-A, was conducted to obtain an evaluation of the center wing fatigue life as compared with the modified C-130 center wing previously reinforced with aluminum. The primary structural parameter utilized in determining the fatigue endurance of the wing box is the moment-stress ratio. The moment-stress ratio is the ratio of the spanwise bending moment to the maximum internal stress at a particular spanwise location on the wing box. The points analyzed were the skin-stringer element with the highest stress at a particular wing station. The upper and lower skin-stringer elements are shown in Figures 1 and 2. The quality levels and wing stations chosen for the static and fatigue analysis were assumed to be the same as those used in the aluminum design. The fatigue analysis consisted of finding the moment-stress ratios for the composite reinforced wing box and comparing these values to the aluminum reinforced design. The moment-stress ratios were obtained by applying a constant spanwise bending moment to the finite element model and calculating the stresses from the internal loads at particular spanwise stations. The residual thermal stress which results from the bonding operation is added. Then the moment at each station is divided by the maximum stress which occurs at each station. A description of each step of the moment-stress ratio calculations for the fatigue analysis is given in Appendix D.

A graph of the moment-stress ratios versus wing station for the aluminum wing is shown in Figure 4. The moment-stress ratios obtained from the analysis of the composite reinforced box are plotted on the same graph. If these ratios for the composite reinforced box are the same or greater than those for the aluminum modified box it can be concluded that the compound composite box has a fatigue endurance equal to the modified C-130E wing box. This was true of all points except the point on the lower surface at W.S. 101. An additional amount of composite material would have to be added to this point to obtain a comparable result to the aluminum reinforced wing box. The closeness of the results is a good check on the finite element simulation and fatigue analysis model.

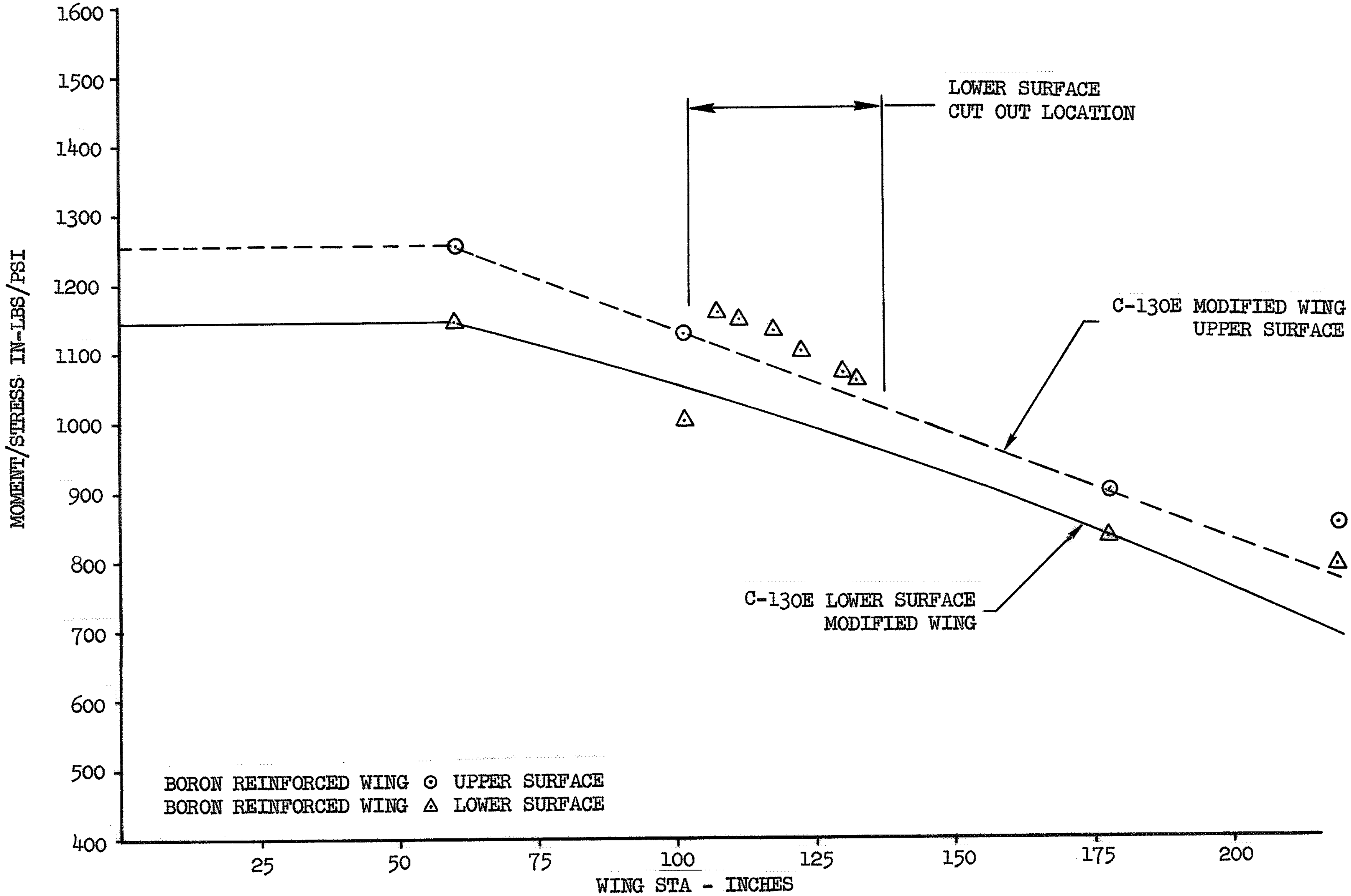


FIGURE 4 MOMENT-STRESS RATIO Vs WING STATION

3.0 DEMONSTRATION PROGRAM PLAN

The preliminary studies reported herein indicate a very attractive potential for the efficient solution to fatigue-critical structural design through use of composites as reinforcement. Thus, as required by the present contract, a Demonstration Program has been planned in detail to provide full verification of the concept. This program, described in the ensuing paragraphs, is planned to follow three phases:

- A Development Phase to provide the full engineering and manufacturing data base
- A Design Phase to develop the detailed design
- A Fabrication & Test Phase to fabricate one (1) C-130 center wing box for static and fatigue tests and two (2) boxes for flight test evaluation.

The phasing for the total program is shown in Figures 5, 6 & 7. The phasing is shown in "months from go-ahead". The program is broken down into three phases: Development, Design and Fabrication and Test. There is necessarily some overlap in the various disciplines between the phases. In the following sections, the content and scope of the required effort in each phase is outlined.

3.1 Development Phase

During this phase of the program, the majority of the development work required to provide the analytical, experimental and manufacturing background for the program will be accomplished.

3.1.1 Analytical Methods

The requirement for analytical methods of analysis to assist the detail design of the wing box should present no major difficulties. For the static analysis, strength (stress-at-a-point) and stability of the structure must be assured.

There are adequate methods available for determining the strength of metallic/composite combinations (Reference 4). These analysis methods take into account the residual thermal stresses which result from the

FIGURE 5

I. DEVELOPMENT PHASE

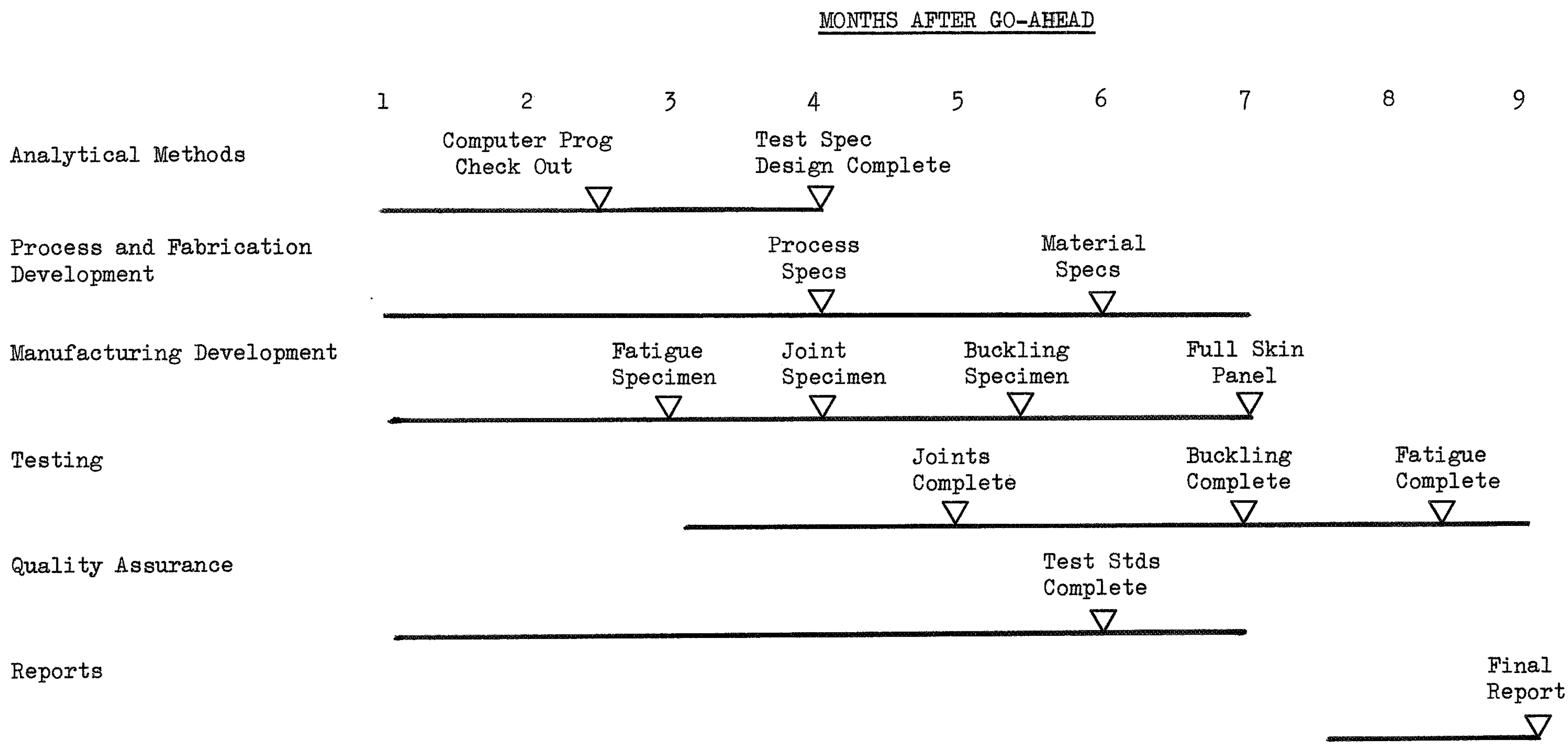


FIGURE 6

II. DESIGN PHASE

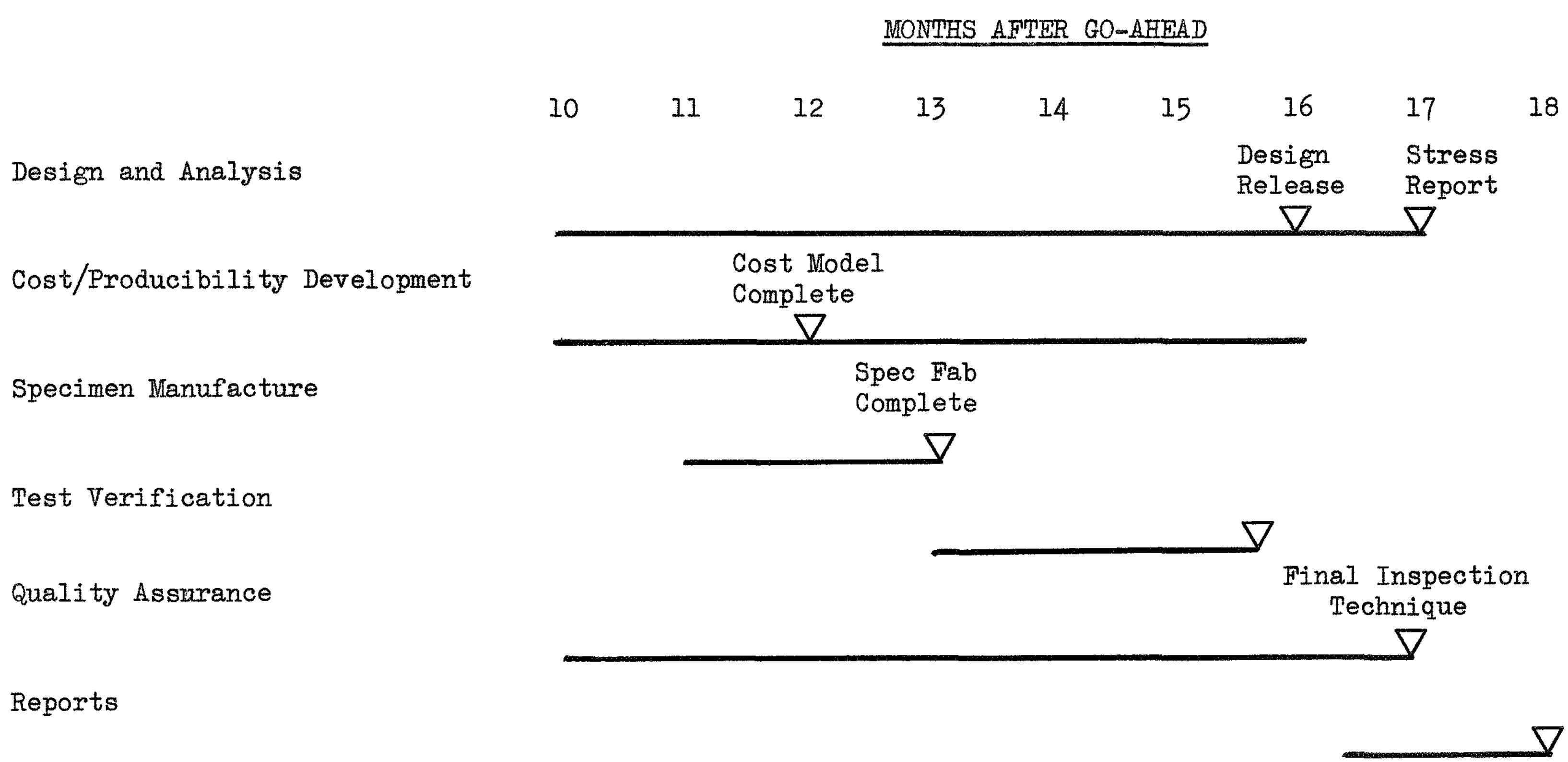
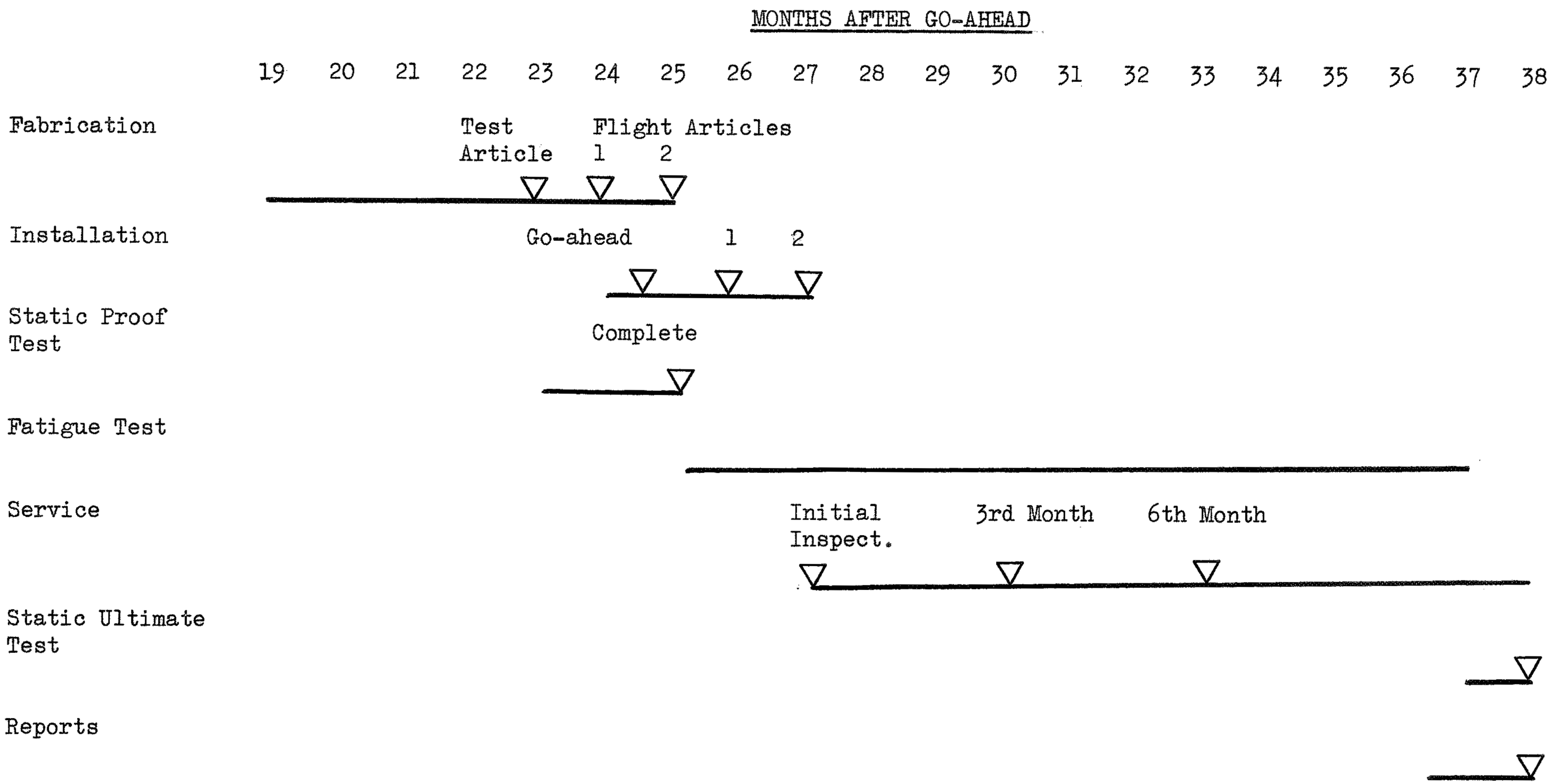


FIGURE 7

III. FABRICATION AND TEST PHASE



bonding of the metallic to the composite material. There has been experimental verification of the analysis of this program for uniaxial stress states.

An analysis method for predicting the stability of the wing structure has recently been completed on Air Force Contract (Reference 5). The program predicts the buckling loads for laminated plates which have discrete stiffeners. The energy solution includes the bending and torsional stiffness of the stiffeners. The energy from stiffener warpage is neglected; however, since the stiffeners for the final design concept are hat (closed) sections, this is a valid assumption. In order to substantiate the analytical procedures and provide empirical design data, three representative composite reinforced wing skin stiffener panels will be tested in compression. The panels will be single bay panels which require no side support.

At least three major static joint tests will be accomplished. These joints will be representative of the splice joints in the upper and lower rainbow fitting-skin area. (Figure 3).

The fatigue analysis must prove that the composite reinforced wing has as much or more fatigue life as the modified aluminum wing. It has been assumed in the fatigue analysis that all the fatigue damage in metallic/ composite combination occurs in the metallic material. This was based on the fact that S-N data on composite materials have shown much higher fatigue strength than the metallics. In some cases, run out has been achieved for loadings in excess of design limit load. Some recently acquired experimental data have verified the validity of this approach; however, a more extensive fatigue program should be accomplished. In addition, basic fatigue data should be acquired on the adhesive system.

Three wing skin-stiffener panels should be tested to the fatigue spectrum utilized on the modified aluminum box. After completing the fatigue tests, damage tolerance tests will be conducted. The three panels will include:

- Upper surface with central access door
- Upper surface rainbow joint and access door
- Lower surface rainbow joint

3.1.2 Quality Assurance

During this phase of the program, all of the inspection techniques which will be used during the program will be developed. Nondestructive test standards will be designed and fabricated. In-process inspections will be performed on all joints and sub-components which are fabricated, and receiving tests will be performed on all composite material. Fatigue specimens will be inspected during test to determine continuity of the adhesive.

Inspection of the components will be by state-of-the-art contact pulse-echo ultrasonics. Using this method, the inspection will have to be made before the hat is mechanically fastened to the skin since access to the composite surface is desirable. There are technical problems which would be encountered if the inspection were to be made through the aluminum skin. The hat may require removal if an inspection of the bond were made after the installation on the aircraft.

There is an advanced inspection method utilizing phase/amplitude sensitive ultrasonics which could be used. This method would permit inspection of the part at any time during fabrication or after fabrication.

3.1.3 Process and Fabrication Development

During this intiial phase of the program, the process and fabrication methods will be developed. The bonding and warpage problems must be assessed. The previously developed process and material specifications for the boron composite material will be utilized. However, material and process specifications must be developed for the low temperature curing adhesive. In addition to the test specimens which will be fabricated during this phase of the program, one full length composite reinforced wing skin - stiffener panel will be fabricated to determine warpage and fabrication problems. The warpage of the full skin panels as a result of bonding the composite is one of the major problem areas. The magnitude of the warpage can only be fully determined after the fabrication of a full panel. If the panels are excessively warped, a possible solution is an extended postcure of the bond.

The adhesive system which is currently planned for use in bonding the reinforcing composite to the aluminum is Shell's EPON 9614. This adhesive is a modified epoxy system impregnated on a nylon carrier. Preliminary tests have shown that the adhesive is a good candidate for a structural low temperature curing adhesive. The cure cycle presently being utilized is a two hour cure under at least 20" Hg at 190°F. This adhesive is currently in use by Lockheed-Georgia for structural repairs on the C-5A.

The material and process development program must be accomplished on the adhesive in order to qualify it for structural applications. For the process development, two cure cycles will be evaluated in order to achieve the lowest cure temperature for bonding the cured boron composite to the aluminum structure. Each cycle will be evaluated by checking lap shear (L/t of 40), creep rupture (L/t of 8) and metal-to-metal peel properties. All tests are to be conducted at 190°F and will consist of five (5) specimens for each property. The selected cure cycle will be documented in a process specification for subsequent material evaluation and component fabrication. The material development program for the adhesive will utilize the cure cycle developed during the process development. Approximately 100 lap shear and metal-to-metal peel tests will be accomplished. Environmental exposure the creep rupture tests will be utilized on some of the lap shear tests. The data generated will supplement data generated on this adhesive for previous programs. The physical and mechanical properties of the adhesive will be identified and documented in a material specification for future material procurement and control.

3.2 Design Phase

During this phase of the program, the final design will be completed and the detail drawings prepared. The drawings will be sufficient to have the final components qualified for the extended flight test program. Certain final design experimental verifications and manufacturing developments will have to be accomplished.

3.2.1 Design and Analysis

The design and analysis of the final component will be performed to comply with existing C-130 production design requirements. Production type drawings will be released through the standard check system. A formal stress analysis report will be prepared to substantiate the ultimate strength of the center wing structure. A detailed fatigue analysis of the center wing will be prepared. The analysis will include a spanwise evaluation of the fatigue endurance and will be based on the operational usage utilized for substantiation of the aluminum modified box. At critical sections, such as cutouts and joints, fatigue damage will be presented in terms of individual missions and sources. The necessary technical manual changes will be issued, and the weight and balance corrections made.

3.2.2 Cost/Producibility Development

Design and development of the composite reinforced center wing will be monitored to determine:

- material costs
- fabrication costs
- assembly and installation costs
- tooling costs
- optimum trade-offs between cost/producibility effectiveness, weight, and structural requirements.

Excessive cost areas will be identified and potential remedial action identified.

The necessity for, and advantages of, in-process controls, inspections, and non-destructive testing for the maintenance of quality for material and fabrication will be investigated. Methods, techniques, and equipment requirements will be assessed on a preliminary basis during the design phase. Investigative efforts will be concerned with the ways and means of incorporating in-process controls, inspections and non-destructive testing and the affect on the overall operations from a cost/producibility view.

Cost/producibility effectiveness will be investigated and evaluated in the areas connected with assembly and installation as follows:

- determination of tooling required
- possible use of existing tooling, where available - evaluate advantages and problem areas associated with the use of existing tooling
- potential impact on current production operations due to introduction of new center wing boxes - schedules and cost differential variations.
- possible salvaging of plumbing, systems, etc. - affect of overall program due to special cleaning, checkout, etc., if reused.

3.2.3 Manufacturing and Experimental Test Support

In order to assure adequacy of the final design concept, certain experimental and manufacturing verifications will be made during this phase of the contract. It is anticipated that the joint areas will be most critical from a manufacturing and design standpoint. Therefore, additional joint specimens will be fabricated and tested.

3.2.4 Quality Assurance

Receiving tests and in-process inspections will be performed on all specimens and subcomponents which are fabricated during this phase. The inspection techniques for the final component will be finalized at the completion of this phase.

3.3 Fabrication and Test Phase

During this phase of the program, the actual ground test and flight components will be fabricated, tested and evaluated.

3.3.1 Fabrication and Installation

Three center wing boxes will be fabricated. The test article will not require engine mounts, trailing edges and related structure. The two flight components will be installed on two C-130E (assumed) aircraft. It is to be assumed that these boxes will be installed on two aircraft scheduled for a retrofit in the existing line. In addition, two (2) all aluminum retrofit boxes are to be fabricated and placed in storage in case the composite reinforced boxes need to be removed from the aircraft at a future date. The general fabrication methods for the full scale component are described below.

Process and Fabrication Development:

For this case, the unidirectional stiffeners of boron-epoxy material for the crown of the hat section will be laid up in a tool which conforms to the hat section. The tape will be slit to width and stacked ply-by-ply to build up to the required thickness. At the ends, where the rainbow fitting attached, titanium shims will be interleaved between plies. When the lay-up has been completed, the resin bleeder system will be installed and the section bagged for autoclave cure. Standard in-process inspection will be required for this fabrication.

The autoclave cure for these laminates is 85 psig $\pm$ 5 psig and $350^{\circ}F \pm 10^{\circ}F$ for one hour. The parts will be debagged and inspected. The process for bonding the boron laminates to the hat section requires chemical processing of the aluminum for bonding. The chromic acid anodizing process will serve as the cleaning process. The parts will be immediately primed with a noncuring epoxy primer on the underside of the hat where the bonding will be done. The boron laminates will be prepared for bonding by lightly sanding the bonding surface. Adhesive will be applied to the prepared surface of the boron laminates, and the laminates will be assembled in the hat section. The assembly will be bagged and autoclaved at 45 psig $\pm$ 5 psig and $190^{\circ}F \pm 10^{\circ}F$ for four (4) hours. The assembly will be debagged and inspected ultrasonically to check the bond integrity.

The above procedure must be repeated to effect the reinforcement of the skins. A similar lay-up procedure to the one used for the crown stiffeners will be used for the skin reinforcement. The cured boron stiffeners will be prepared for bonding by sanding the bonding surface. The aluminum skin will be chromic acid anodized as required and will be primed with a non-curing primer in the areas to be bonded. The adhesive will be applied to the boron stiffener, the stiffeners will be positioned and secured, and the assembly bagged and autoclaved. All operations will be subjected to in-process inspection with the final bond ultrasonically inspected for voids.

Tooling and Assembly Development:

The general assembly sequence and tooling requirements for the composite reinforced wing box are described in this section. Since the final design selected utilizes the existing skin and hat stiffener extrusions, the assembly sequence will be very similar to the current production sequence.

Fabrication of Skin Panels -

The skin panels are similar to the current production aluminum configuration which includes the external doubler areas. The existing aluminum extrusions will be used.

The extrusion is clamped to the bed of the spar mill by means of a vacuum chuck, and the inside surface risers and pads are milled. This surface will be the same as the production panels and production cams and cutters will be used. The panel will be turned over and clamped to the mill bed using a vacuum chuck made to nest the inner surface risers. The outer surface is milled to contour. The outer surface cut will reduce the panel to the desired thickness. Detail and profile cuts will be made to complete machining using production tools. The panels will then be shot-peened, anodized, and painted. The internal composite strip doublers will be subsequently bonded to the internal skin. Finishing sequence for the skin may vary according to requirements for bonding of composites.

Fabrication of Hat Stringers -

Stringers are designed to use the flat shape extrusion formerly used prior to the aluminum redesign. This will provide for use of an existing extrusion requiring minimum machining. Flanges will be profiled using existing production tools. Stringers will be shot-peened, anodized, and painted. The composite doublers will be subsequently bonded to the internal crowns of the hats. The finishing sequence may vary according to requirements for bonding composites.

Assembly of Wing Box -

The upper and lower surface skin panels will be assembled using existing production assembly jigs. With the composite reinforcements bonded inside the hat stringers and on the skins under the stringers, the jig locators for panels and stringers will not be affected. The external contour bars will be affected by the removal of the external doubler areas from the panels. These contour bars are installed in the first stage assembly jig and remain on the assembly, serving as transportation pick-up bars. They are used to position the assembly while performing subsequent assembly operations until the panel assembly is installed in the box beam jig. In order to keep tooling changes at a minimum, Manufacturing will assume that not more than one center wing assembly will be in work at any time and only one set of contour bars will be modified to accommodate this configuration.

The first state assembly results in the mating of panels, stringers, wing joint fittings, and installation of sufficient fasteners through panels and stringers to allow the assembly to be moved from the assembly jig to the drivematic machine. The drivematic operation mechanically installs taper-loks and rivets to fasten stringers to panels. The location and installation of the fasteners is controlled by a punched tape. If fastener size or location changes, this operation will be performed by hand in the first stage jig.

In the second stage assembly jib, the rib caps, doublers, miscellaneous clips and fasteners which are inaccessible to the drivematic are assembled. This assembly sequence is similar for both upper and lower surface assemblies.

In the box beam assembly jib, the major components of the center wing are mated (front beam, upper surface, lower surface and trailing edge assemblies). The bulkheads and other internal structural members are installed. Fuselage attach angles and engine trusses are also installed.

In the box beam pick-up position, operations are completed which are inaccessible in the jig position. External fairings and leading edge ribs are installed. Miscellaneous hardware, plumbing, bracketry, etc., is also installed. Required fuel sealing in the fuel cell cavities is performed in the center wing seal position. The final operation will be the touch-up of primer and the completion of paint system in trailing edge and dry bay areas.

3.3.2 Test Phase

A static, fatigue, and service test is to be conducted on the three boxes. The static and fatigue test is to be accomplished on one box. The service tests will be accomplished on the remaining two center wing boxes.

Static and Fatigue Tests

A static test to limit design loads will be initially accomplished and documented to permit flight of the two flight vehicles. Subsequently, a fatigue test to four (4) lifetimes will be conducted to the same spectrum previously utilized to qualify the aluminum modified boxes. Finally, a residual strength test will be conducted on the component. It is to be assumed that the fixtures being utilized to test the C-130 center wing boxes for residual strength will be adequate for this test program. This fixture will allow loads at the WS 220 joint. No nacelle or flap loads will be applied during the test. It is anticipated that fatigue damage to the specimen will occur throughout the test program and minor repairs will be required. Major failures are not considered within the scope of this program.

Service Evaluation:

The primary objective of the flight and service test program will be to provide extended service experience on boron composite reinforced primary structure.

On the two service components, a service evaluation will be conducted. The ground test evaluation will be minimal. Only minimum ground vibration tests will be conducted to qualify the components for flight. The primary emphasis will be placed on the service evaluation. The evaluation will include the effects of the following aircraft environmental conditions:

- Steady state and fatigue air loads
- Erosion, moisture, heat and ultraviolet and related conditions
- Field maintenance and routine field service

These effects and the degree of possible degradation will be determined by periodic visual and ultrasonic inspections. On damaged or degraded components, the degree of damage and/or repairability will be investigated. The two initial service inspections will be conducted after the 3rd and 6th months. These will include visual as well as ultrasonic evaluation with documentation. The subsequent two inspections will be accomplished in six month intervals.

It is anticipated that the aircraft in which the two components will be installed will be assigned to the Aeronautical System Division. This would make the serviceability monitoring program more flexible.

3.3.3 Quality Assurance

Receiving tests and in-process inspection will be performed on the three center wing boxes. In addition, the four service inspections will be conducted and documented.

3.3.4 Cost/Producibility Development

Flight test and structural test operations will be supported and monitored to:

- Correlate tests results with design requirements for a continuing appraisal of cost/producibility effectiveness
- Maintain surveillance of results of special material and process areas, such as sealant, protective finish, and attachments to ascertain conformance to requirements.
- Provide assistance and general support is special problem areas.

APPENDIX A

BACKGROUND ON C-130 CENTER WING BOX

A wing redesign was accomplished on the C-130B and C-130E wings to provide adequate structural integrity for the increased gross weight of these aircraft over that of the C-130A Model. Certain other structural changes were made in the center wing box.

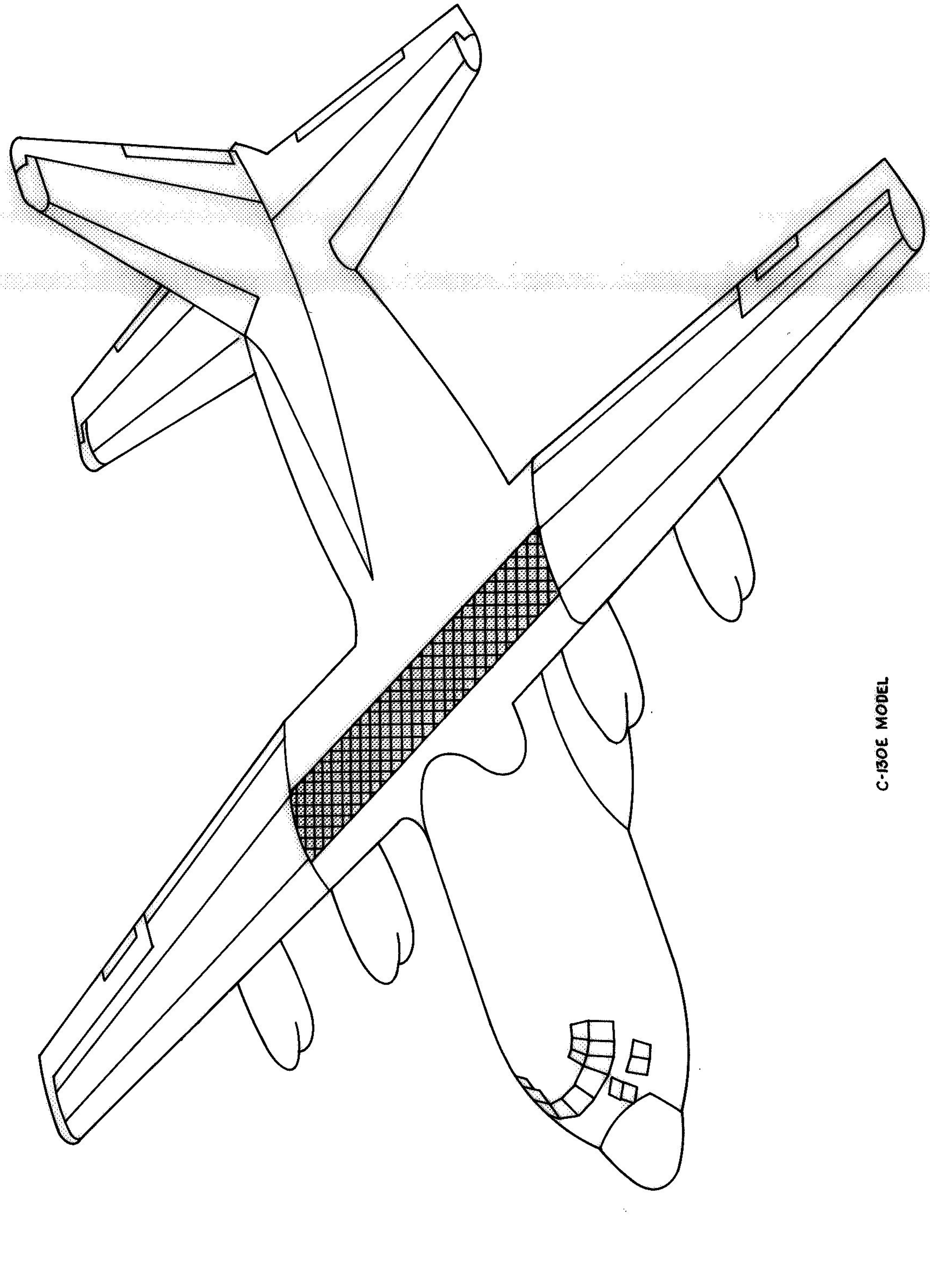
The C-130A center wing box contained no fuel cells. The C-130B and C-130E center wings were redesigned to accomodate bladder fuel cells. The location and basic structural details of the C-130E center wing box are given in Figures 8, 9, 10 and 11.

The basic C-130 center wing is a conventional single cell box beam with constant cross-section, zero dihedral and zero sweep. The wing box provides support for the outer wings, the leading edge structure, trailing edge structure, and the two inboard pylons. The structural arrangement of the box is symmetrical about wing station 0 (W.S. 0).

Main frames provide partial attachment of the wing to the fuselage at W.S. 61. The upper chord of the rib at W.S. 20 provides continuity for the fuselage upper longeron. The lower chord of the rib at W.S. 61 attaches directly to the fuselage lower longeron. Tension joints on the surface and shear attachments on the webs at W.S. 220 join the outer wing to the center wing.

The upper surface is made from extruded integrally stiffened aluminum skin and machine tapered hats. There are three access doors on the upper surface. One is at W.S. "0" providing access to the region between W.S. 61L and W.S. 61R. The other two are at W.S. 200L and W.S. 200R and provide access to the nacelle dry bay region.

The lower surface is made from machine tapered bare aluminum plate and extruded aluminum hats. There are two access doors on the lower surface. They are located at W.S. 120L and W.S. 120R and provide access to the center wing bladder fuel tank region.



C-130E MODEL

FIGURE 8

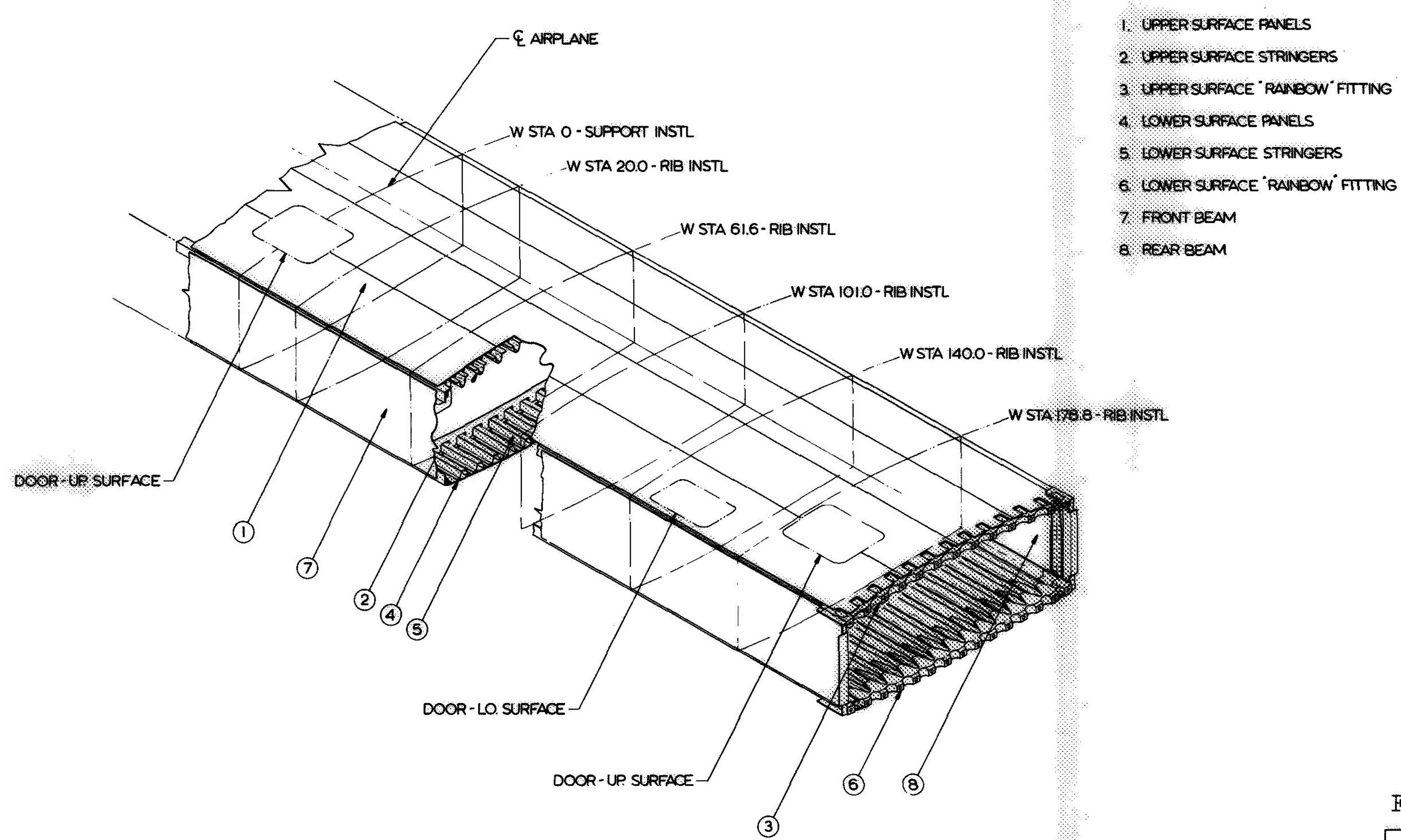


FIGURE 9

[redacted]

ISOMETRIC - C-130 CENTER WING

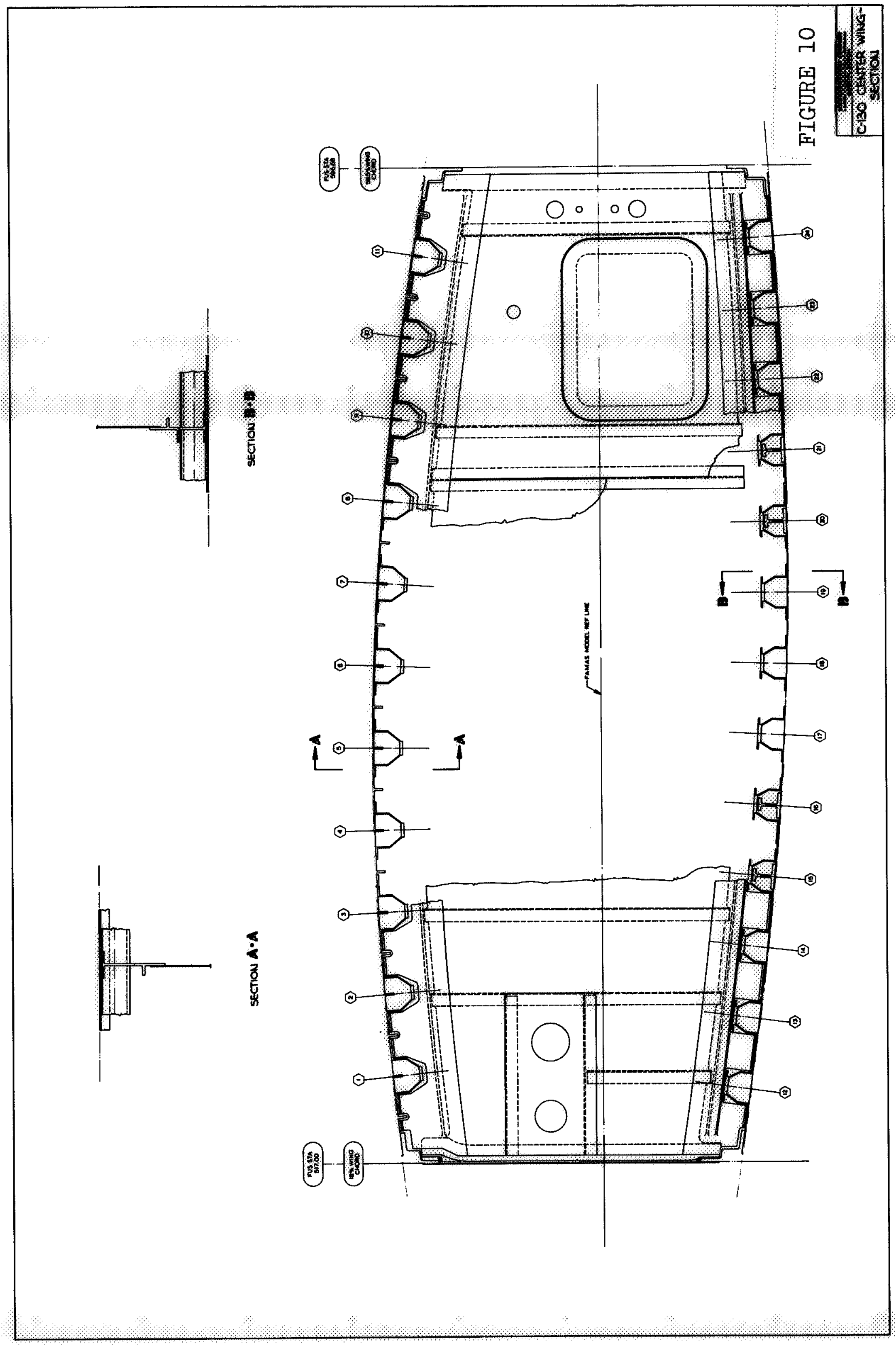


FIGURE 10

[redacted]
C-130 CENTER WING-SECTION

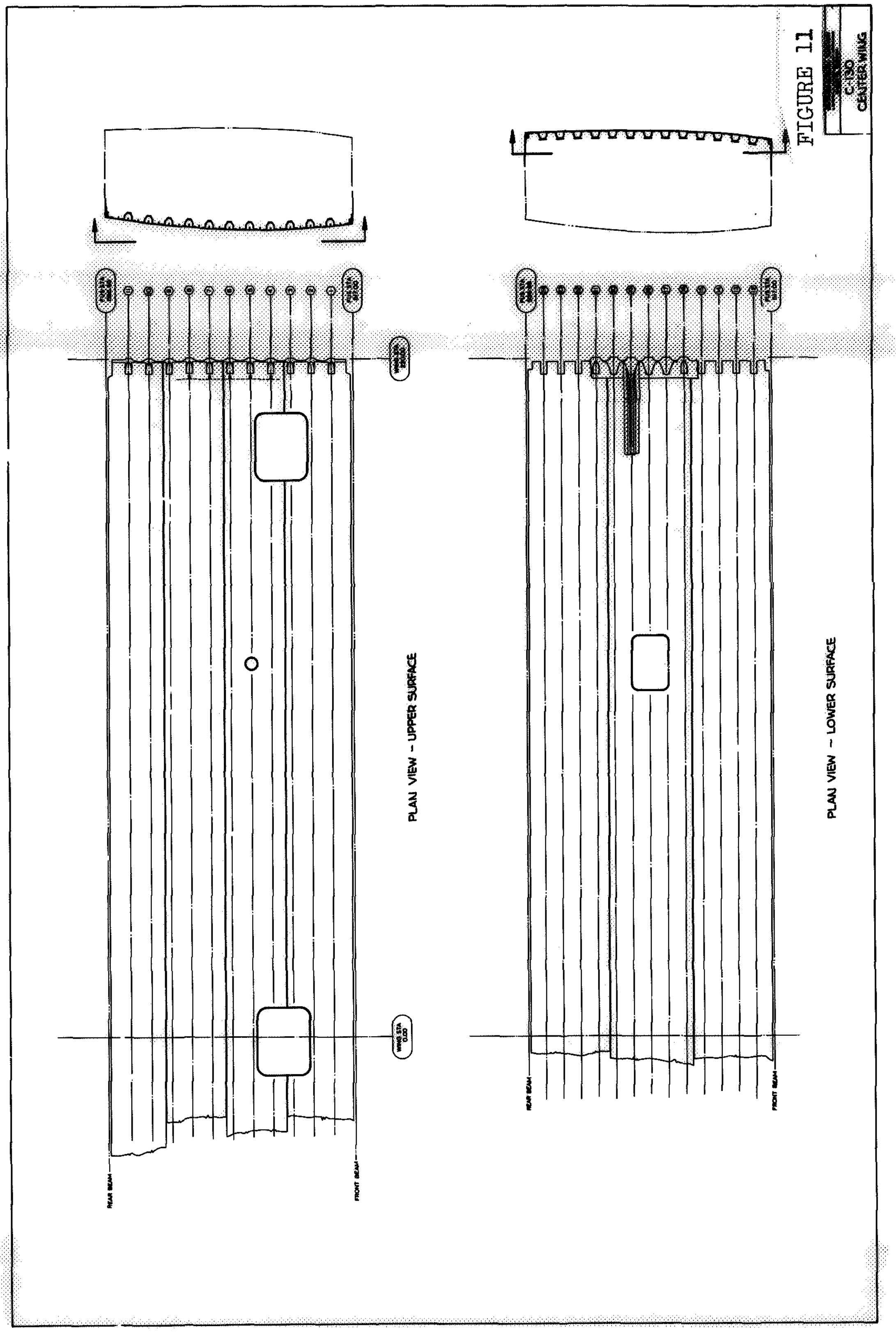


FIGURE 11
C-130
CENTER WING
PLAN VIEW - UPPER SURFACE
PLAN VIEW - LOWER SURFACE
REAR BEAM
FRONT BEAM

There are ten ribs in the center wing. The tank bulkhead at W.S. 178 is an extruded integrally stiffened aluminum panel. The intermediate ribs at W.S. 140 and W.S. 101 and the tank bulkhead at W.S. 61 are of conventional stiffened aluminum web design. The rib at W.S. 20 is a truss rib.

The C-130A aircraft was initially designed to requirements established by the R-1803 series of design specifications. These specifications did not define design standards relative to service life expectancy, fatigue endurance, fatigue loads, fatigue testing or damage-tolerant/fail safe design. After the C-130 design evolved, fatigue requirements were incorporated in the aircraft design specifications. However, fatigue tests and fatigue analyses of the various models of the C-130 have been conducted after-the-fact and under separate contracts.

The first fatigue cracks began appearing in the center wing box of C-130 fleet at approximately 6600 flight hours. Flight utilization and mission profiles indicated that the Southeast Asia shuttle missions were providing a fatigue spectrum more damaging than the spectrum assumed. The spectrum was much more severe than any previously considered for sustained operations. Generally speaking, the fatigue damage occurred in the wing skin covers around access doors, doubler termination points and fuel filler openings. The major problem areas are illustrated in Figure 12.

The previous modifications to the C-130B and C-130E center wing box to improve the fatigue life generally consisted of increasing skin and hat stiffener areas to reduce overall stress levels. The wing structural quality levels were reduced to $K_t = 4.5$ by a redesign of the geometry of the access doors, utilization of tapered interference fasteners and redesign of shear redistribution doublers around the access doors. The increase in the wing skin thickness and hat stiffener thicknesses is illustrated in Figures 13, 14, & 15. The modifications made to the wing box were to provide a structural airframe endurance of 40,000 flight hours (10,000 hours with a scatter factor of 4.0) with a quality level of $K_t = 4.5$. The detailed fatigue analysis on the redesigned aluminum wing box revealed that all but one area had a life of 40,000 hours. This area was in the lower surface at wing station 181.6.

The modifications that were made by reinforcing the C-130 center wing with aluminum material are as follows:

- o Replacement of existing 7178-T6 skin panels by thickened 7075-T73 panels, incorporating integrally machined doubler regions at cut-outs and under stringers. Fatigue critical regions were shot peened. (Ref. Figures 13-15).
- o Replacement of existing 7178-T6 hat sections stringers by reinforced 7075-T65 stringers of larger cross-sectional area. The stringers were not joggled since the skin cut-out doubler material was incorporated in the integrally machined external pads. Stringers were shot peened at outer end areas.
- o Elimination of the upper surface fuel filler cut-outs at W.S. 120 R&L.
- o Tapered interference fasteners were used in spanwise skin panel splices, in attachment of the skin panels to the spar cap flanges and in other areas considered fatigue sensitive.
- o The access cut-outs were changed to elliptical "clamp in" type doors instead of the square "bolt in" type.
- o The "rainbow" wing joint fitting at W.S. 220 was shot peened and tongs reprofiled
- o Elimination of the localized "I" section reinforcing members in the area of the W.S. 120 access cut-out on the lower surface. The required local reinforcement was replaced by the integrally machined skin doublers in combination with the larger cross section stringers.
- o Provisions were made for under-wing fuel quantity measurement necessitated by the upper surface fuel filler deletion.
- o The upper and lower caps of the front and rear beams were changed to 7075-T73 extruded aluminum.
- o The front beam lower cap was reprofiled from W.S. 179 to W.S. 220.
- o The rear beam lower cap was reprofiled.
- o Existing aluminum lockbolts attaching beam caps to webs for the front and rear beam were changed to steel lockbolts and respaced in fatigue critical areas.
- o Front beam web from W.S. 173 to W.S. 220 was changed from 17-7PH steel to 301H steel.

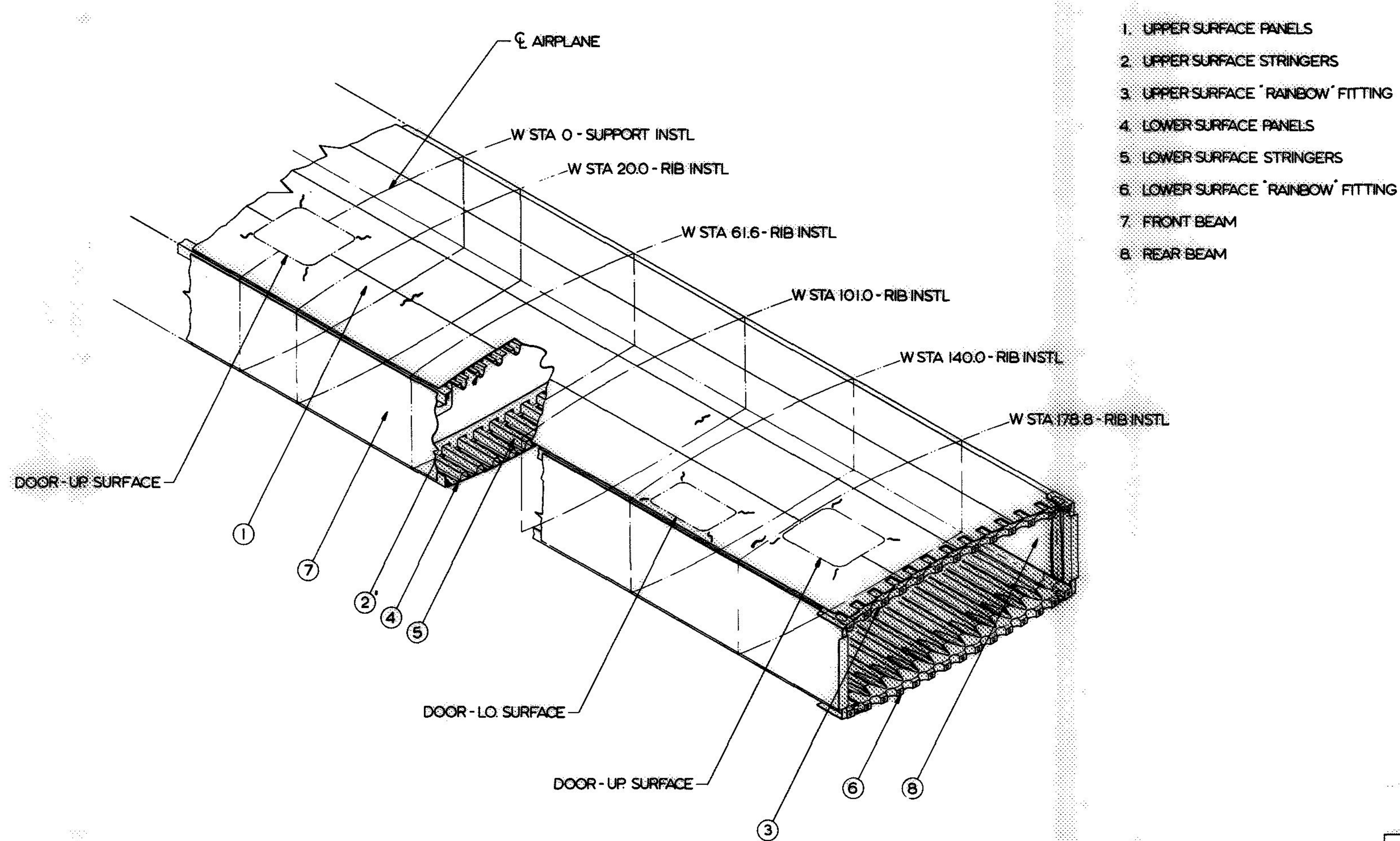


FIGURE 12

ISOMETRIC - C-130 CENTER WING

The C-130 airplane is in an extended state of production at the present time, and consequently the fabrication is performed at the efficient end of the learning curve. Consequently, the relative cost/producibility effectiveness factor of this airplane can be considered to be at an optimum. Thus, on a comparative basis, any design changes of a significant nature, such as the incorporation of advanced composite materials, will result in a considerable reduction in the cost/producibility effectiveness. It will be necessary that a high degree of efficiency of application be established for the composite material usage in an effort to counteract this loss.

The design of structures utilizing advanced composite materials as reinforcement is influenced by the type and degree of reinforcement and means of attachment of the composite material. Linked with these factors are the tradeoff factors of cost and producibility. Basically, all of the associated factors can be expressed in terms of cost required to gain a unit of weight savings. This is a measure of the cost/producibility effectiveness of a design application using composite materials.

In arriving at the tradeoff determinations, it is assumed that the design for all concepts will provide the same basic level of reliability. This requirement is mandatory in order to establish a common denominator on which to compare cost/producibility effectiveness.

For all the concepts investigated in this application study, a projected estimated cost of $300 per pound for Boron pre-impregnated material and $25 per pound for graphite was used. These material costs are Lockheed estimates based on forecasted cost trends for late 1970. Some of the concepts provided for more efficient use of the composite material, and the corresponding cost differentials for these concepts are less than for other concepts. Generally, the measure of cost/producibility effectiveness can be expressed in terms of cost increase per pound of weight saved, which in turn is related to the amount of composite material required to save a pound of weight.

The net cost of the various structural concepts is influenced to some extent by the amount of conventional material used in the remainder of the center wing structure. Aluminum integrally stiffened panels, built-up panels and titanium panels were evaluated. These factors affected the net cost on a direct basis to some extent; however, the major influence on the cost is caused by the corresponding amount of composite material that is required for each concept.

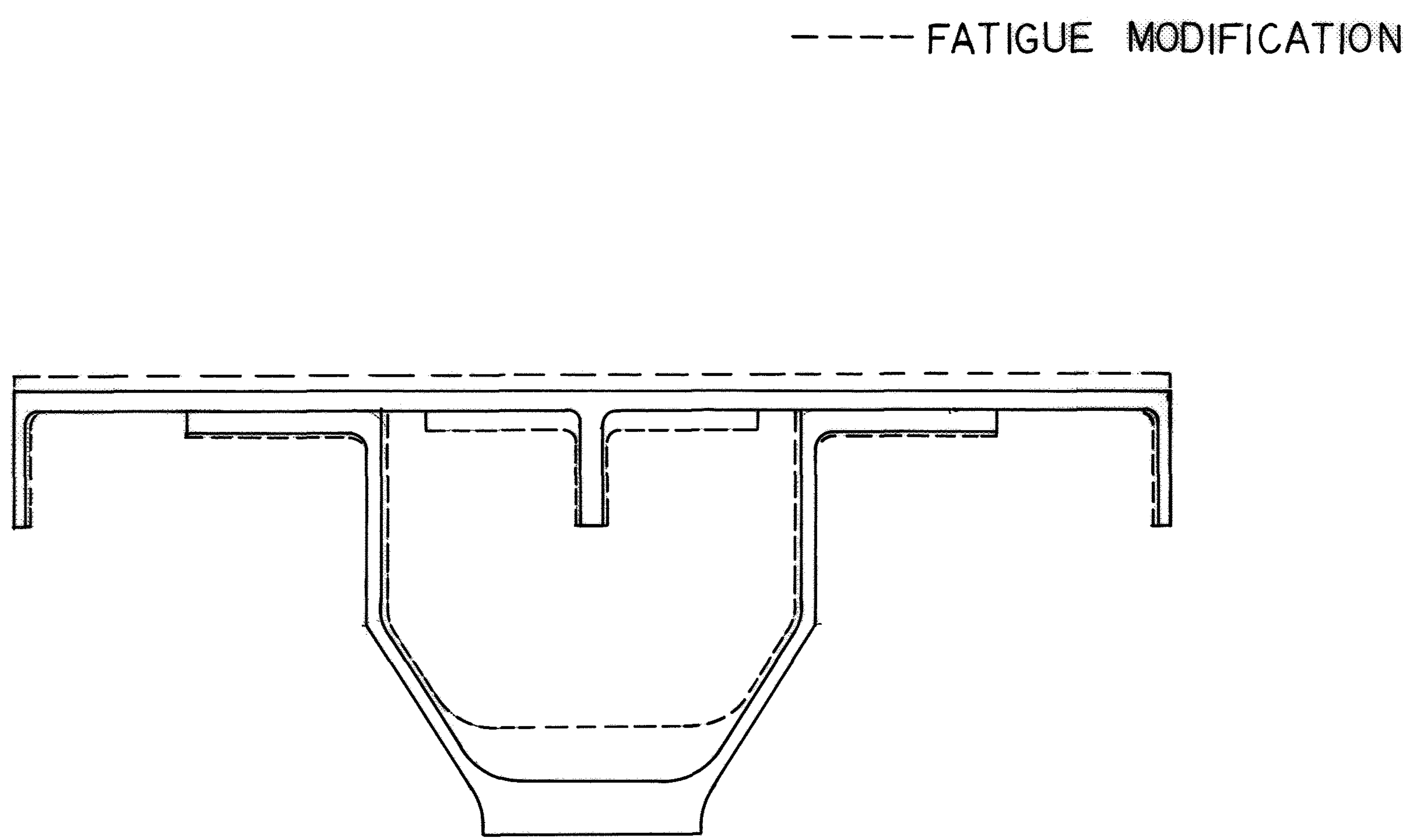


FIGURE 13A

SECTION AT W.STA 68

UPPER SURFACE

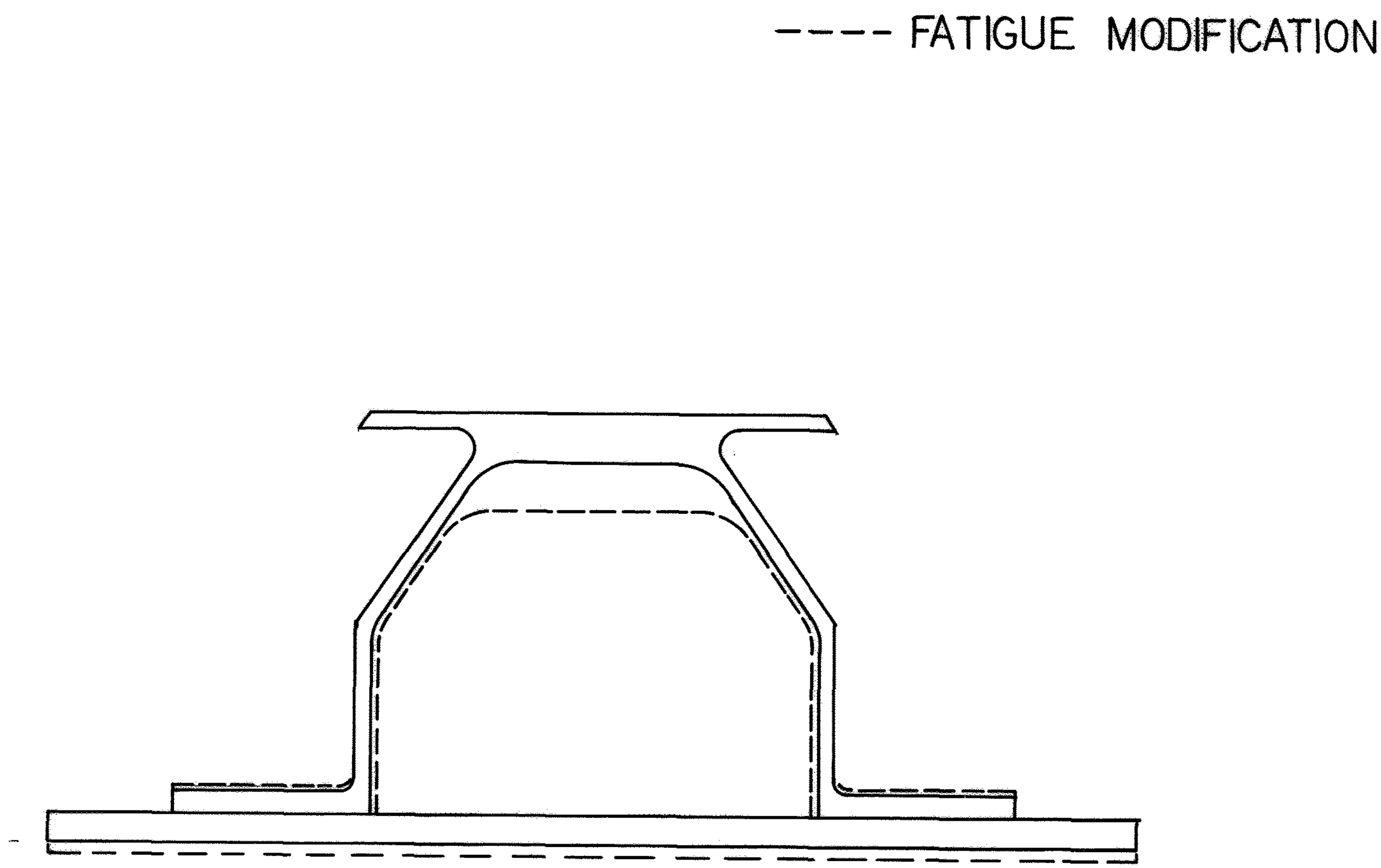


FIGURE 13B

SECTION AT W.STA 68
LOWER SURFACE

----FATIGUE MODIFICATION

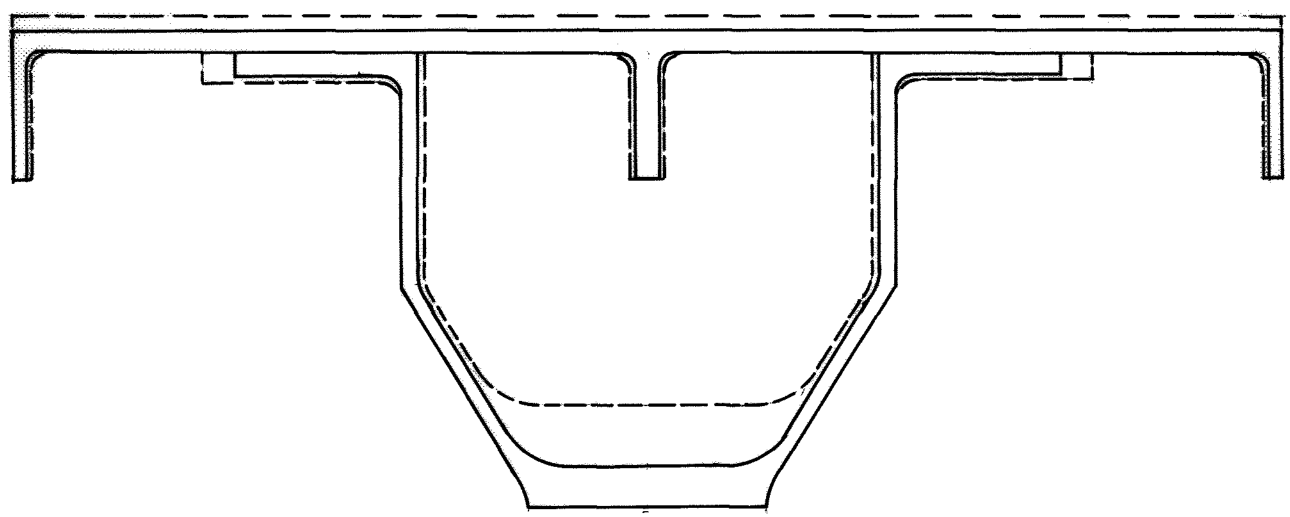


FIGURE 14A

SECTION AT W.STA 120
UPPER SURFACE

---- FATIGUE MODIFICATION

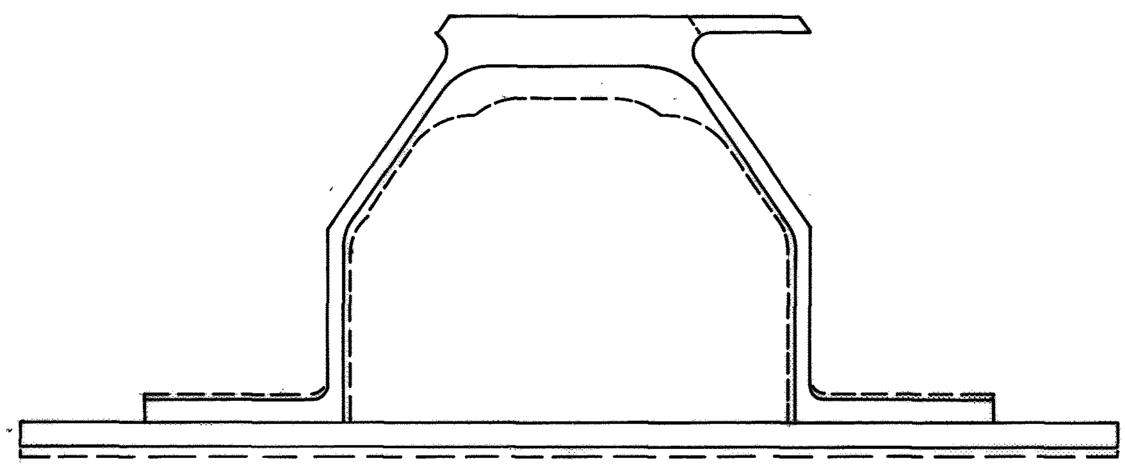
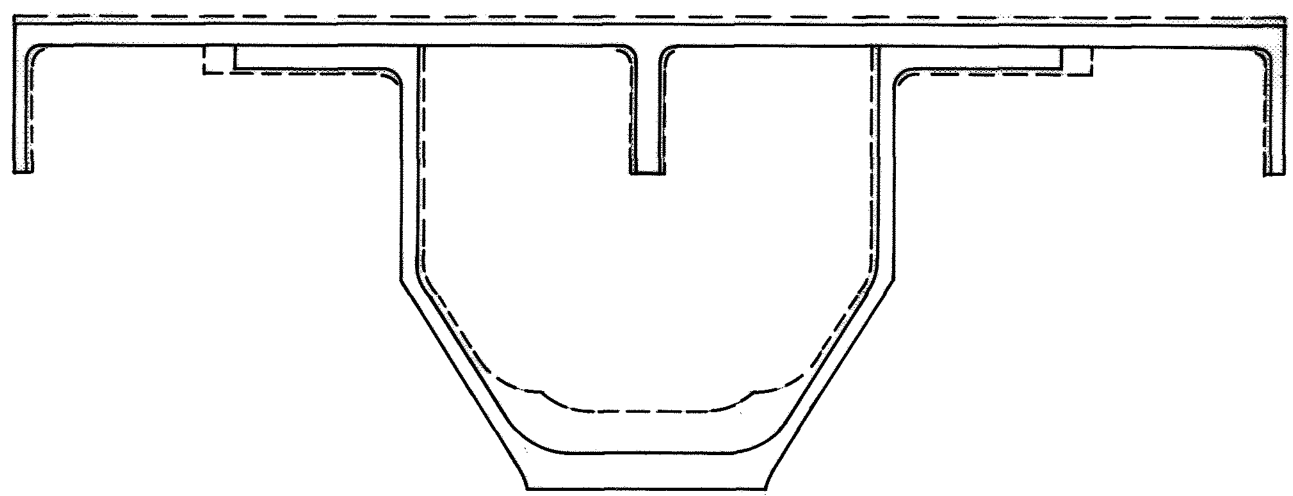


FIGURE 14B

SECTION AT W. STA 120
LOWER SURFACE



---- FATIGUE MODIFICATION

FIGURE 15A

SECTION AT W. STA 179
UPPER SURFACE

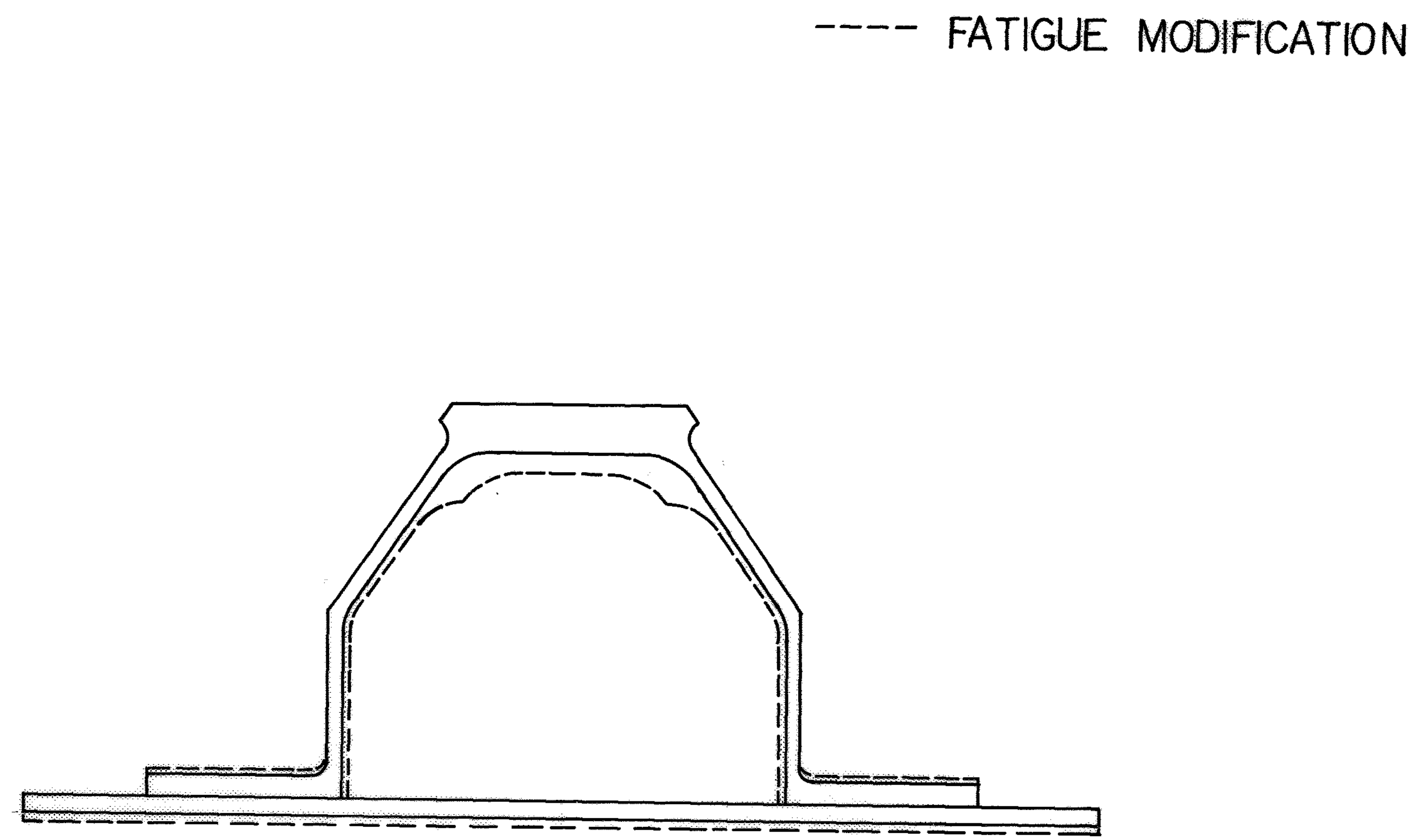


FIGURE 15B

SECTION AT W. STA 179
LOWER SURFACE

APPENDIX B

MATERIAL SELECTION STUDIES

A material selection study was accomplished in order to make preliminary assessments as to the potential structural advantages of utilizing advanced composites for reinforcement of conventional structure. These studies provided design data which assisted in the selection of optimum composite/metallic material combinations for static and fatigue strength, stiffness and thermal compatibility.

Basic Material Properties

A set of basic material properties for the composite materials was developed in order to provide a rational basis for the design studies (Table I). The material properties for the boron/epoxy, graphite/epoxy and glass/epoxy composite materials were obtained from the most current data generated by the major airframe manufacturers. The five basic lamina stress-strain curves were generated, and from these curves, the four elastic moduli and strength allowables were determined. The data is to be considered representative of "B" type allowables, since it will be used in a design environment. For this reason, some of the strength allowables may seem more conservative relative to frequently quoted average or typical values.

The properties which have been selected as representative of the boron-epoxy material are typical of the Narmco 5505 material system. The determination of the Narmco 5505 basic properties has been accomplished by various independent sources; however, those considered here are the result of the General Dynamics/Ft. Worth program of Reference 6. Large amounts of statically significant design data have been acquired on this program on the Boron/Narmco 5505 system.

The basic properties of the graphite/epoxy systems are not well defined since many of the systems are in the stage of development for fiber-matrix optimization or are not available in sufficient quantity to permit a complete characterization. For the purpose of the design study, the typical material properties are based on the Morganite II filament and were derived from two sources. The longitudinal properties, 0° tension and 0° compression, are projected from very recent test results reported by General Dynamics/Ft. Worth, Reference 5, and the remaining properties are a result of Lockheed-Georgia investigations as reported in Reference 7.

The glass/epoxy material properties included are those of the S-glass system SP 1002S (3M Company). The values provided are preliminary results derived under a Lockheed development program and have not yet been published.

TABLE I

COMPOSITE MATERIAL DESIGN PROPERTIES

		BORON/EPOXY	GRAPHITE/EPOXY	S-GLASS/EPOXY
Elastic Constants (Secant)				
Longitudinal Modulus	10^6 psi	31.5	20.0	6.5
Transverse Modulus	10^6 psi	3.1	1.55	1.6
Shear Modulus	10^6 psi	0.65	0.65	0.66
Major Poisson's Ratio		0.21	0.32	0.3
Design Strengths ("B" Allowables)				
Longitudinal Tension				
Limit	ksi	117.0	100.0	124.0
Ultimate	ksi	176.0	150.0	185.0
Longitudinal Compression				
Limit	ksi	250.0	110.0	53.0
Ultimate	ksi	375.0	170.0	80.0
Transverse Tension				
Limit	ksi	7.6	6.6	5.3
Ultimate	ksi	11.4	10.0	8.0
Transverse Compression				
Limit	ksi	22.5	18.0	17.0
Ultimate	ksi	40.0	27.0	25.5
Inplane Shear				
Limit	ksi	8.5	7.0	8.0
Ultimate	ksi	21.1	16.0	18.0
Physical Properties				
Fiber Volume Content		50%	60%	49%
Density (lb./in.^{3})		0.072	0.053	0.065
α_1 (Longitudinal Thermal Coefficient) 10^{-6} (in/in/$^\circ$F)		2.4	- 0.28	4.8
α_2 (Transverse Thermal Coefficient) 10^{-6} (in/in/$^\circ$F)		11.5	16.4	12.4

TABLE II

METALLIC MATERIAL DESIGN PROPERTIES

ELASTIC CONSTANTS

	E_{11}, psi	E_{22}, psi	G_{12}, psi	ν_{12}
Titanium	16×10^6	16×10^6	6.2×10^6	.29
Aluminum	10.3×10^6	10.3×10^6	3.9×10^6	.33

LIMIT STRAIN ALLOWABLES

	ϵ_{1t}	ϵ_{2t}	ϵ_{1c}	ϵ_{2c}	ϵ_{12}
Titanium	.0058	.0058	-.0066	-.0066	.0087
Aluminum	.0047	.0047	-.0057	-.0057	.0075

Static Strength

The static strength of materials can be compared on the basis of several strength parameters. The most pertinent parameters for aircraft design are the ultimate strength or the yield strength of the material. For metals the ultimate strength is generally quoted as a design parameter for tension whereas the yield strength is generally used for compression. For composite materials, both the yield and the ultimate strengths have been used for design throughout the industry. Actually, both the yield and ultimate strengths of materials are important since in aircraft structural design there must be no permanent structural deformation at design limit load (no yield of material) and no structural failure at design ultimate load (no material failure). Design ultimate load is 1.5 times greater than design limit load. Normally, it is not necessary to design the structure to both limit design and ultimate design loads on a strength basis. For most metals, a structure designed to ultimate loads will satisfy the criterion for no permanent deformation at limit design load automatically. The reason for this is that the metallic stress-strain curve has a yield point relatively close to the ultimate failure point ($1.5\,\sigma_y > \sigma_u$). For composite materials, the yield points are generally less than 2/3 of the ultimate failure points ($1.5\,\sigma_y < \sigma_u$); therefore, at design limit load permanent deformation could be present if the design were made to ultimate design loads with no consideration given to the yield points. However, if the structure were designed so that the composite material was at its yield point at limit design loads then at ultimate design loads the structure would have adequate strength margins of safety.

For this static strength comparison study, it was decided that the static strength of various combinations of composites and aluminum and titanium would best be assessed by examining failure envelopes of the combinations. The failure envelopes for the metallic/composite combinations were plotted utilizing an existing Lockheed computer program coupled with a plotting device (Ref. 4). A distortional energy of Hill yield criterion was utilized to predict the limit or yield failure envelopes. These envelopes are based on the composite limit allowables shown in Table I. For the metals, the ultimate tension "B" allowables divided by 1.5 were used as a limit allowable. Since an ultimate compression allowable for metals is not determinable and the use of compression yield for ultimate has shown to be very conservative for composite reinforced metals (Ref. 8), a compression limit (yield) allowable was set utilizing a value of twice the ultimate shear strength divided by the factor of 1.5. This approach is based on a Mohr's circle evaluation of the state of stress for uniaxial compression.

With the limit allowables for the metallic and composite materials shown in Tables I and II, the failure envelope computer program (Ref. 4) was utilized to compute limit strength envelopes for various percentages of composite reinforcement of titanium and aluminum. Initially, it was assumed that the bond of the composite to the metallic was accomplished at room temperature. Subsequent runs were made assuming an elevated temperature bond ($\Delta T = 200^{\circ}F$) so that the effect of residual thermal stresses in the composite and metallic material could be assessed. In both cases, it was assumed that the reinforcement was symmetrical so that no inplane (membrane) loading was coupled with out of plane (bending) deformation. To give a base of comparison, a titanium limit strength envelope is shown in Figure 16. For ultimate design loads, the envelope would be 1.5 times larger.

The trends in strengths of composite reinforced metals can be assessed by examining the envelopes for a 50% composite and 50% metallic combination.

In Figures 17 and 18, the results of reinforcement of aluminum and titanium with unidirectional boron and graphite are illustrated. It is evident that for tension-tension and tension-compression loadings the reinforcement with boron or graphite is equally efficient. For compression-compression loadings the boron reinforcement is most efficient. If these envelopes are normalized with respect to density, the results are shown in Figures 19 and 20. Here, the graphite reinforcement is superior for tension-tension loadings and boron reinforcement for compression-compression loadings. In both of these examples, a room temperature bond was assumed which left no residual thermal stresses.

FIGURE 16

TITANIUM LIMIT (YIELD) FAILURE ENVELOPE

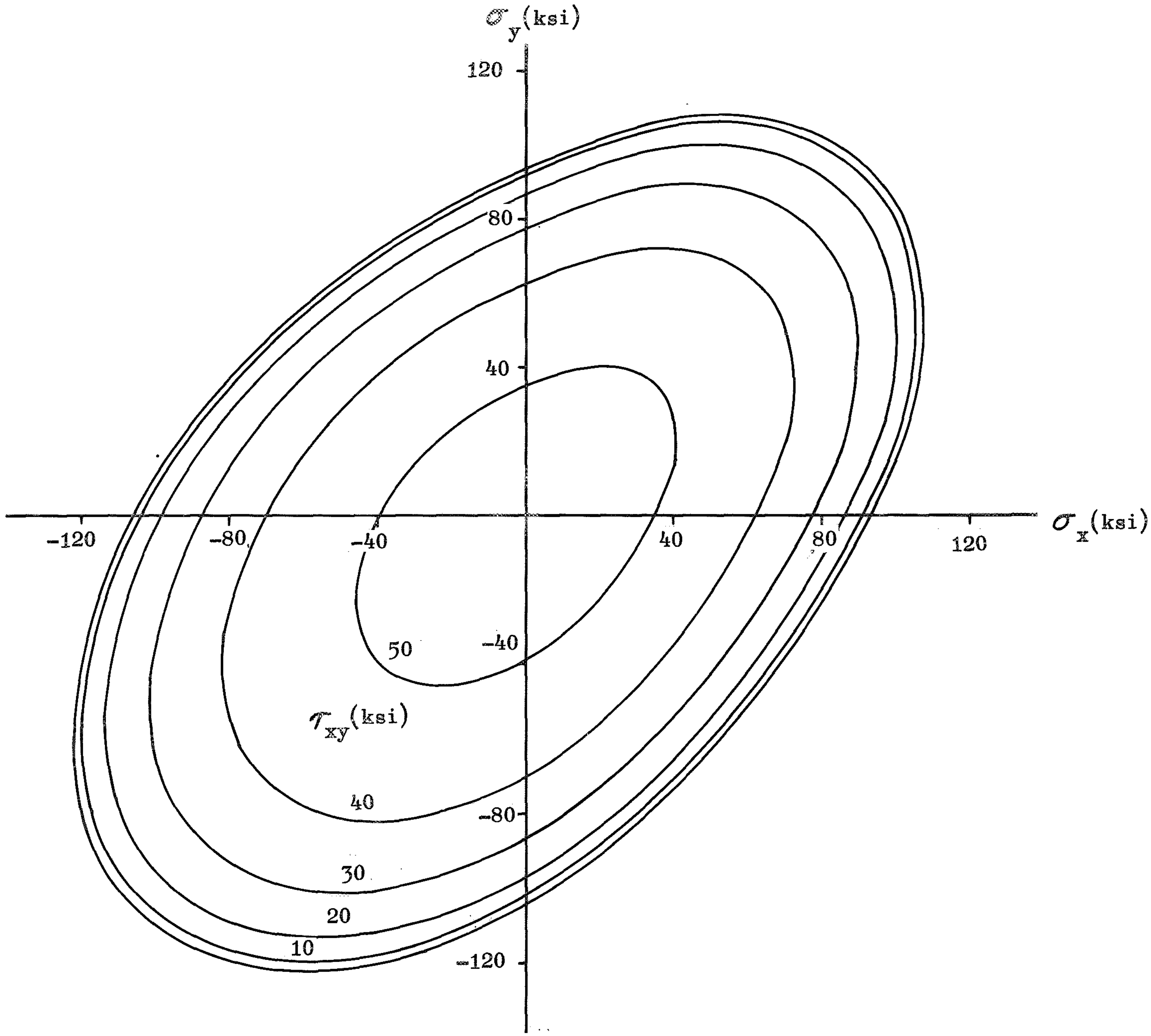


FIGURE 17

COMPOUND COMPOSITE STRENGTH ENVELOPES
(50% ALUMINUM/50% UNIDIRECTIONAL COMPOSITE)

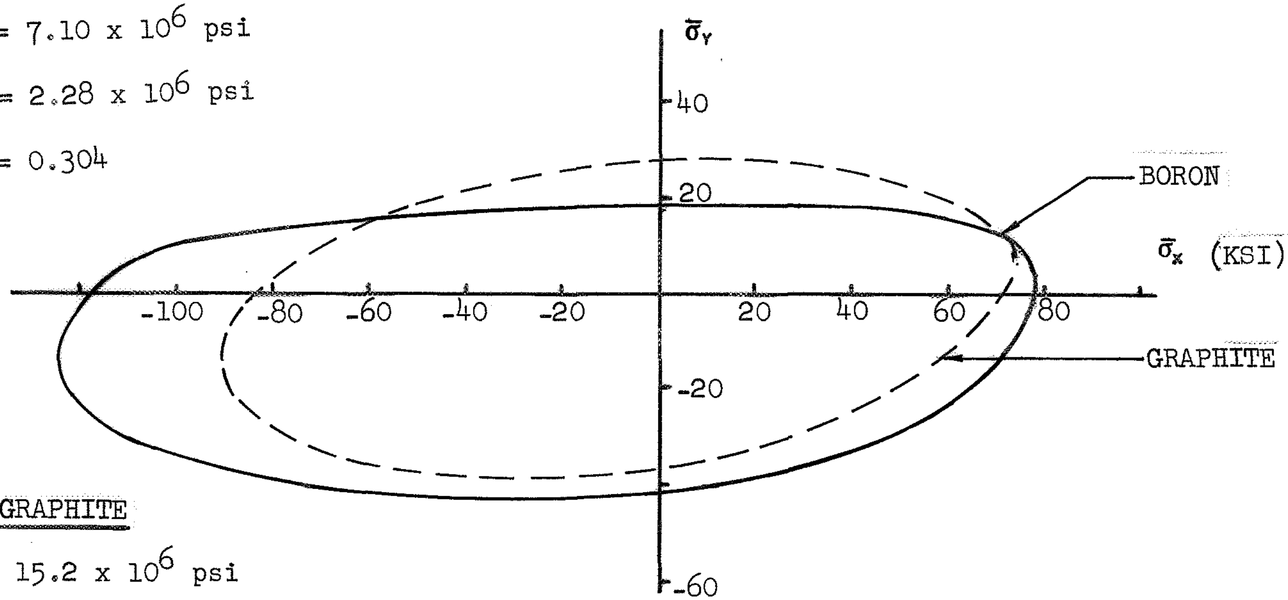
BORON

$E_{xx} = 20.9 \times 10^6$ psi

$E_{yy} = 7.10 \times 10^6$ psi

$G_{xy} = 2.28 \times 10^6$ psi

$\nu_{xy} = 0.304$



GRAPHITE

$E_{xx} = 15.2 \times 10^6$ psi

$E_{yy} = 6.27 \times 10^6$ psi

$G_{xy} = 2.28 \times 10^6$ psi

$\nu_{xy} = 0.329$

FIGURE 18

COMPOUND COMPOSITE STRENGTH ENVELOPES

50% TITANIUM/50% UNIDIRECTIONAL COMPOSITE

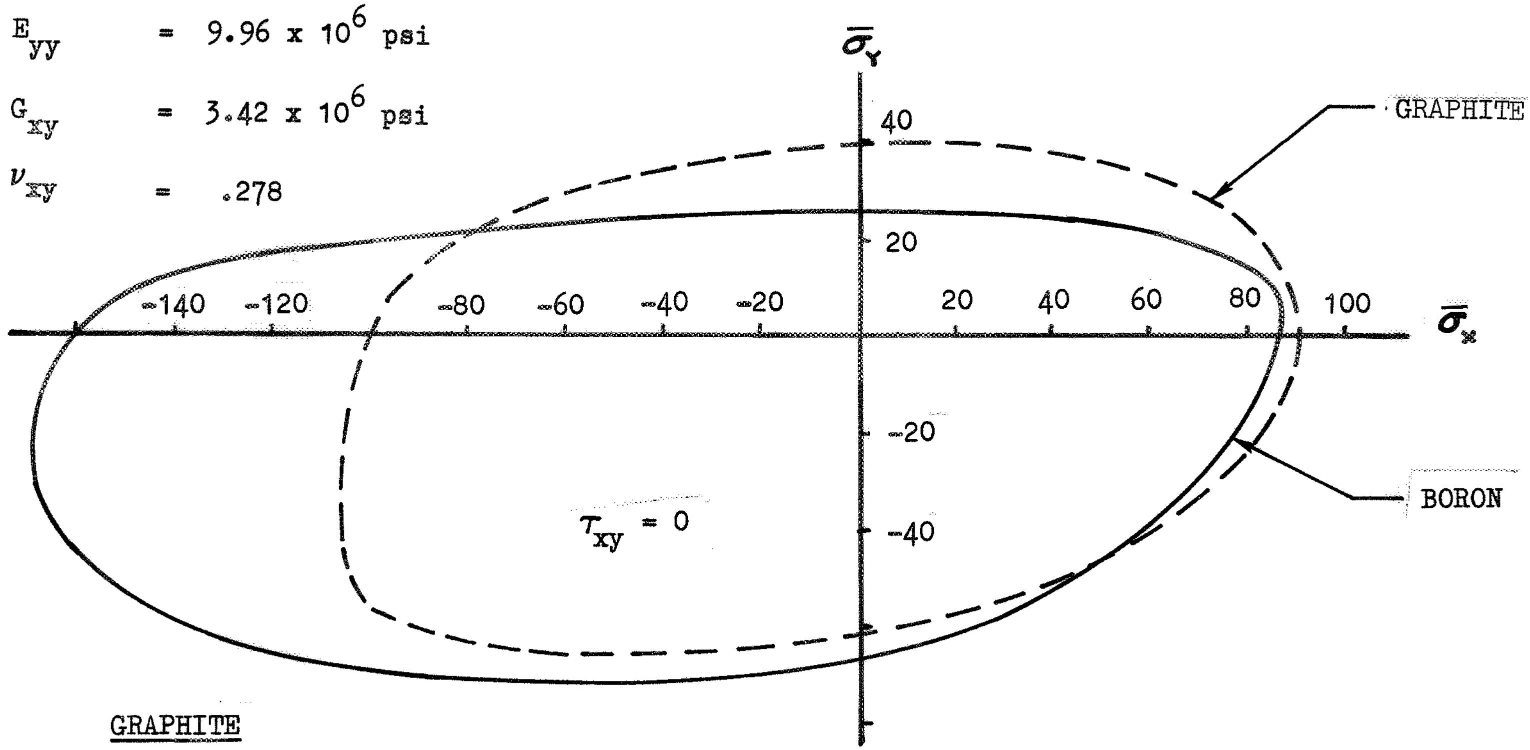
BORON

$E_{xx} = 23.8 \times 10^6$ psi

$E_{yy} = 9.96 \times 10^6$ psi

$G_{xy} = 3.42 \times 10^6$ psi

$\nu_{xy} = .278$



GRAPHITE

$E_{xx} = 18.0 \times 10^6$ psi

$E_{yy} = 9.10 \times 10^6$ psi

$G_{xy} = 3.42 \times 10^6$ psi

$\nu_{xy} = 0.292$

FIGURE 19

COMPOUND COMPOSITE SPECIFIC STRENGTH ENVELOPES

50% ALUMINUM/50% UNIDIRECTIONAL COMPOSITE

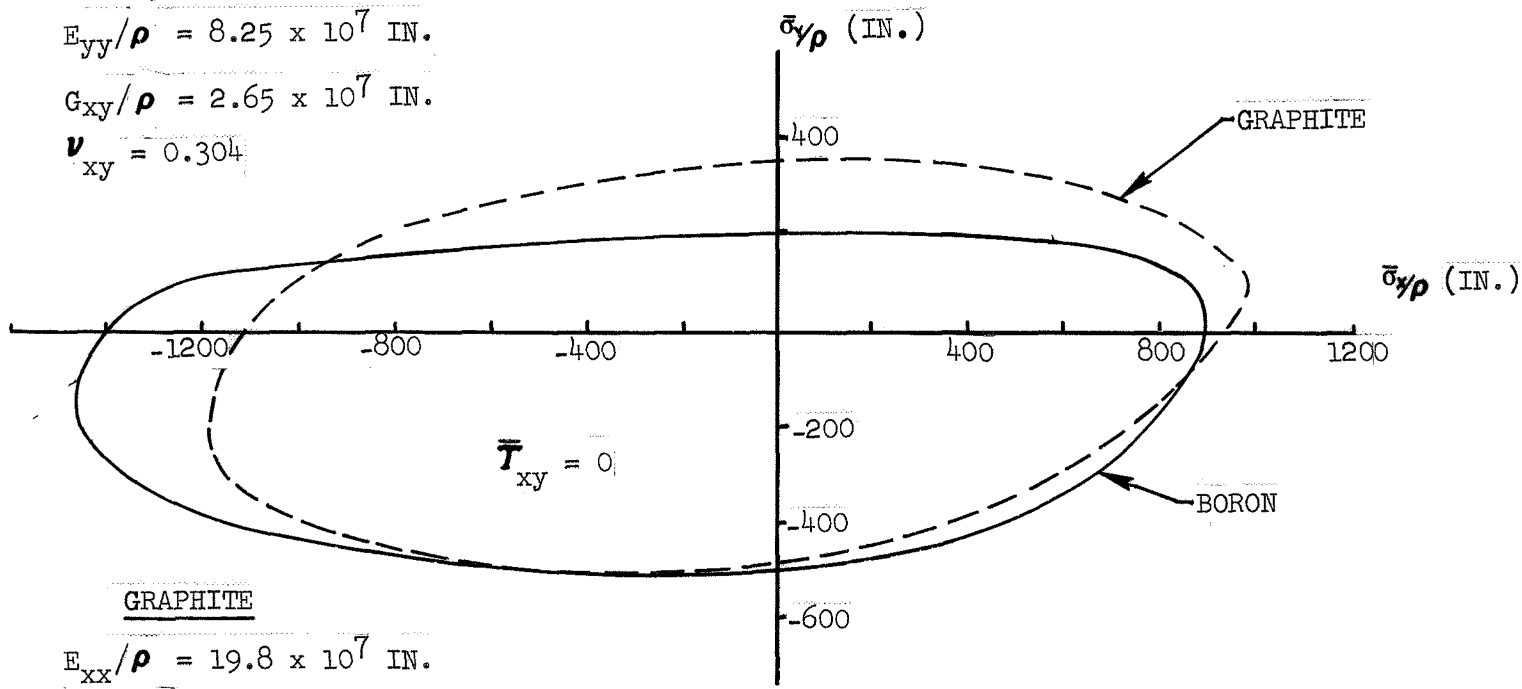
BORON

$E_{xx}/\rho = 24.3 \times 10^7$ IN.

$E_{yy}/\rho = 8.25 \times 10^7$ IN.

$G_{xy}/\rho = 2.65 \times 10^7$ IN.

$\nu_{xy} = 0.304$



GRAPHITE

$E_{xx}/\rho = 19.8 \times 10^7$ IN.

$E_{yy}/\rho = 8.15 \times 10^7$ IN.

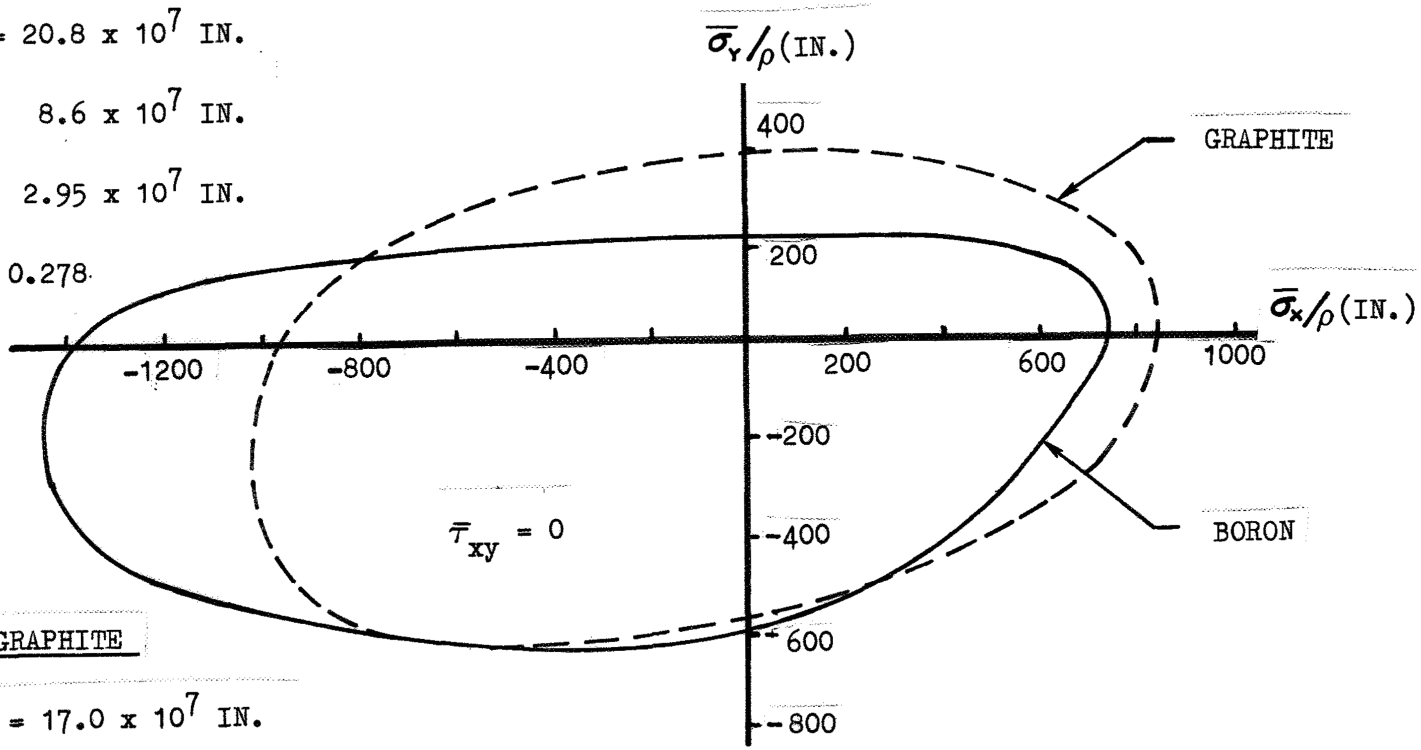
$G_{xy}/\rho = 2.96 \times 10^7$ IN.

$\nu_{xy} = 0.329$

FIGURE 20

COMPOUND COMPOSITE SPECIFIC STRENGTH ENVELOPES

50% TITANIUM/50% UNIDIRECTIONAL COMPOSITE



BORON

$E_{xx}/\rho = 20.8 \times 10^7$ IN.

$E_{yy}/\rho = 8.6 \times 10^7$ IN.

$G_{xy}/\rho = 2.95 \times 10^7$ IN.

$\nu_{xy} = 0.278$

GRAPHITE

$E_{xx}/\rho = 17.0 \times 10^7$ IN.

$E_{yy}/\rho = 8.58 \times 10^7$ IN.

$G_{xy}/\rho = 3.22 \times 10^7$ IN.

$\nu_{xy} = 0.292$

As an example of the trends in strength when the residual thermal stress is included in the analysis, the envelopes for 50% composite and 50% aluminum are shown in Figures 21 and 22. On each figure, the limit failure envelope is shown for the room temperature bond and for a bond which results in a $\Delta T = 200^{\circ}F$. It is evident that a much greater strength penalty is paid when bonding the graphite to the metallic at elevated temperatures. This difference is caused by the negative thermal coefficient of expansion, α_1, of the graphite composite which gives a greater thermal mismatch than boron composite metals.

FIGURE 21

COMPOUND COMPOSITE STRENGTH ENVELOPE

50% ALUMINUM/50% UNIDIRECTIONAL BORON

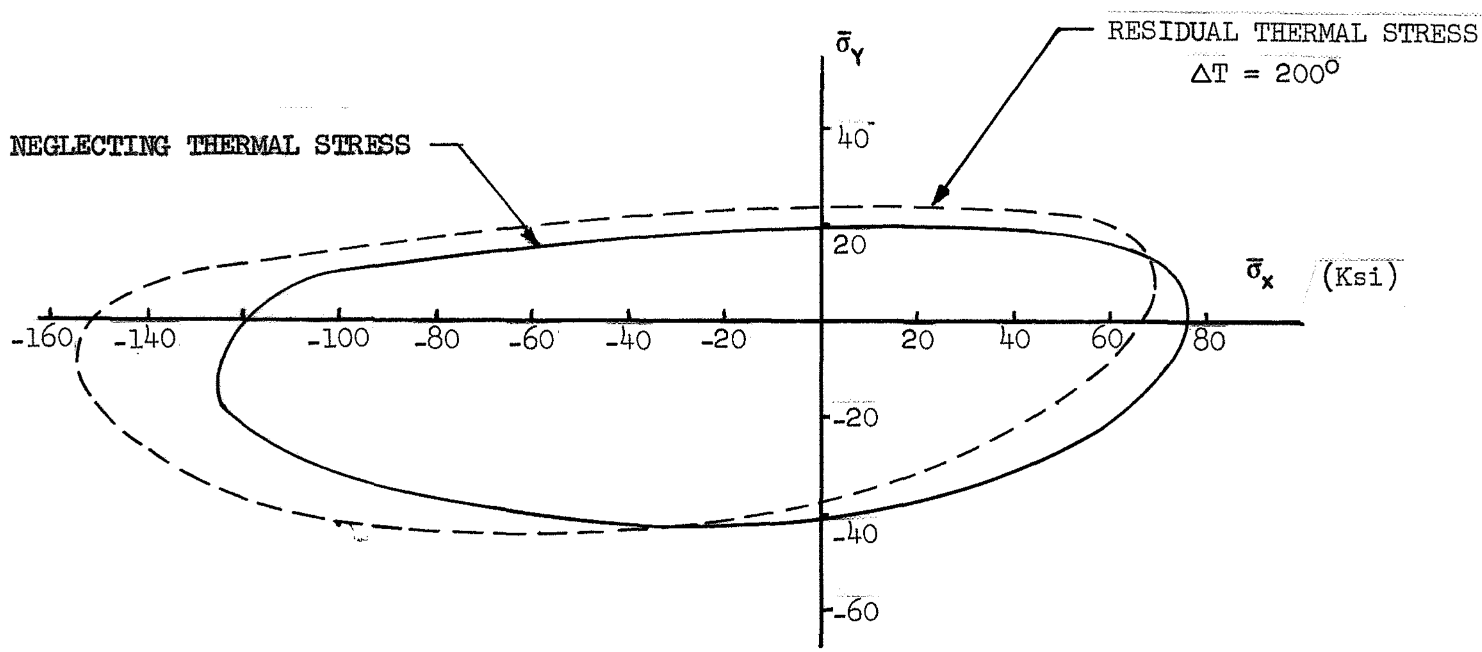
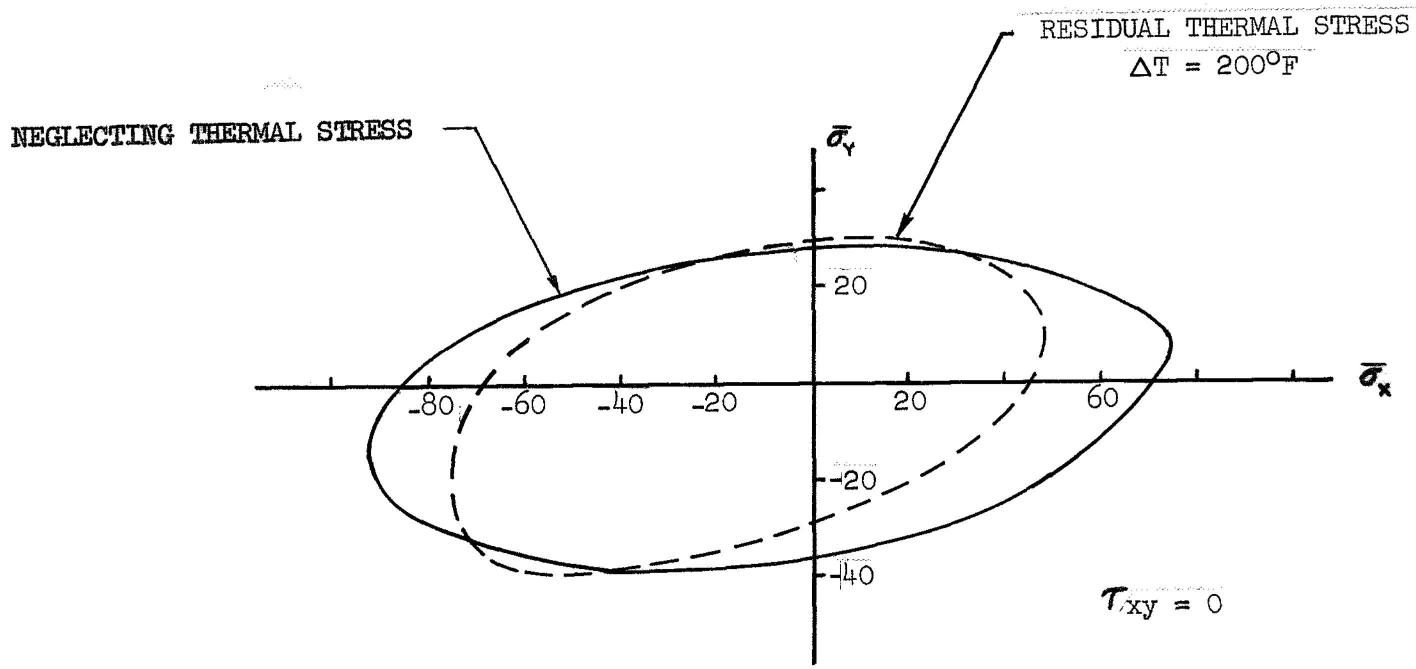


FIGURE 22

COMPOUND COMPOSITE STRENGTH ENVELOPES

50% ALUMINUM/50% UNIDIRECTIONAL GRAPHITE



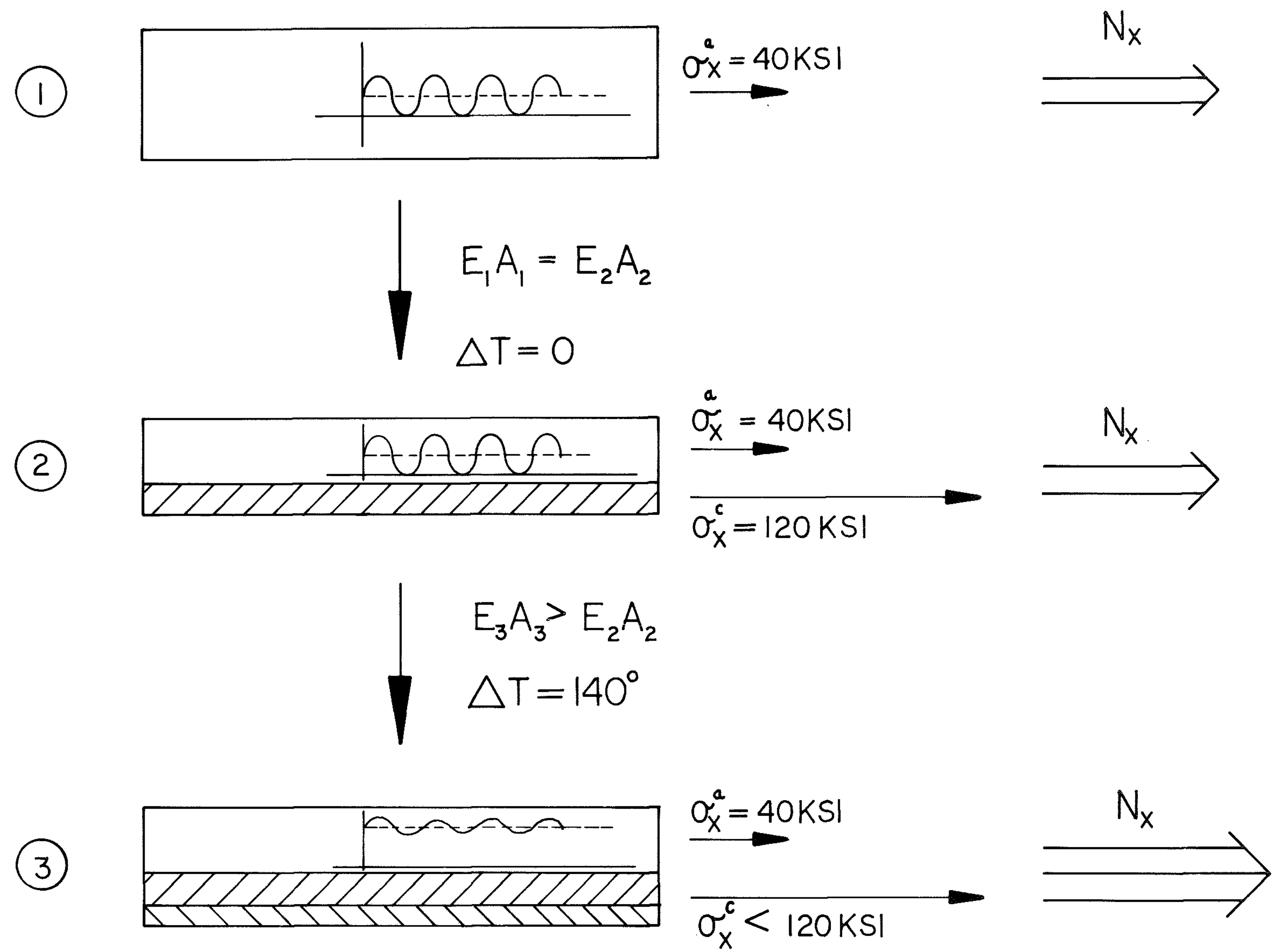
Fatigue Strength

Studies were accomplished to determine the fatigue improvements possible through composite reinforcement and the thermal compatibility problems involved. These areas were of particular interest since the C-130E center wing box is a fatigue critical design.

The residual thermal stresses induced in the metallic materials after an elevated temperature bond to unidirectional composites are tensile stresses since the thermal coefficients for composites are less than those for metallics. For a given structural member, such as a wing skin, which has certain mean and alternating loads, the end result of removing metallic material and bonding composite material is that the mean tensile stress in the metallic material is increased and the alternating stress is decreased. The actual assessment of the change in the fatigue life of the total component will depend on the magnitude of loadings, ratios of mean to alternating loads, percentage of composite reinforcement, flight spectra, and other factors.

For the initial design phases of the program, a detailed fatigue analysis could not be accomplished on all of the design concepts exercised. Therefore, the stress levels in the metallic material, including the residual thermal stresses, were kept at or below the stress levels which existed in the modified all aluminum wing boxes. This assured that each design would have at least the fatigue life of the modified aluminum box. This approach proved valid since the fatigue analysis of the final design concept showed the composite reinforced box, concept II A, had an equivalent fatigue life.

Based on this approach, a model was generated to analyze the skin stringer combinations. The model can be explained with reference to Figure 23. In Section 1 of the figure, the aluminum skin is subjected to a set of mean and alternating stresses. The stress in the skin at ultimate static design load is 40,000 psi. Upon removal of 50% of the aluminum and the addition of composite material bonded at room temperature ($\Delta T = 0$), with an equal EA for both sections, the alternating and mean stresses in the aluminum remain the same (Section 2). In actuality, the bond will be accomplished at an



REINFORCEMENT OF FATIGUE CRITICAL STRUCTURE

FIGURE 23

elevated temperature, and thus the actual stress level in the aluminum will be higher due to the residual thermal stresses. In order to lower the average stress level in the aluminum back to the level prior to the bonding of the composite, it is necessary to add an additional increment of composite. This will result in the condition shown in Section 3. The mean stress in the aluminum is higher but the alternating stresses are lower. This combination has higher static strengths or margins of safety since the EA has increased; however, any removal of material will require the stress level in the aluminum to increase. In the following paragraphs, the impact of the residual thermal stresses on the fatigue life of a metal/composite combination will be explained. Also, the impact of structural weight savings possible with a metal/composite combination design on a fatigue critical component is assessed.

As has been reported for the static stress levels in the compound composite elements, the thermal stresses induced offset much of the effects which could improve the fatigue life of the metallic element. Examples of this are given in Figures 24 and 25 where the predicted fatigue life of aluminum-boron/epxoy compound elements is given for the conditions shown. The predictions are made on the basis that all significant fatigue damage occurs in the aluminum element and that the boron/epoxy member, which has a far superior fatigue life to the aluminum, sustains relative little damage. The ΔT's indicated with the various curves represent the temperature differentials between the bonding temperature and the temperature at which the cyclic loading was applied. The trends indicated in these figures for zero mean load were also valid for the full range of positive mean loads in varying degrees.

By assuming that the major damage is imposed on the aluminum element, the fatigue life of the compound material can only be improved over the bare metal if the composite elements reduce those effects which are detrimental to the fatigue life of the aluminum. The fatigue life of the aluminum can be increased by reducing either the applied stress amplitude or the mean stress level. Since the unidirectional composite has a higher extensional modulus than the aluminum and, therefore, assumes load at a faster rate, both stress reductions can be accomplished by the addition of the composite.

FIGURE 24

EFFECT OF THERMAL STRESSES ON THE FATIGUE LIFE OF ALUMINUM & BORON ELEMENTS

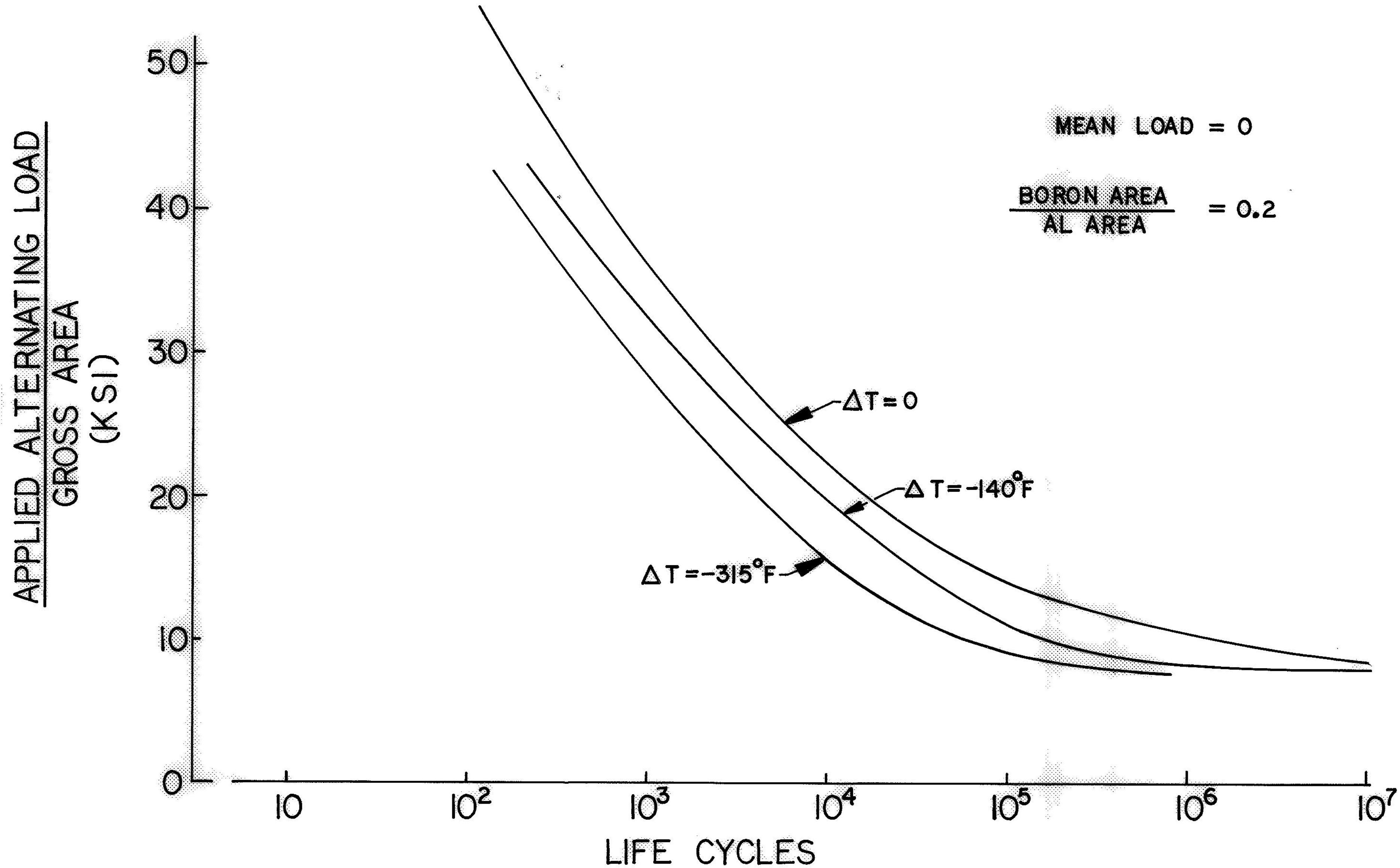


FIGURE 25

EFFECT OF THERMAL STRESS ON THE FATIGUE LIFE OF ALUMINUM & BORON ELEMENTS

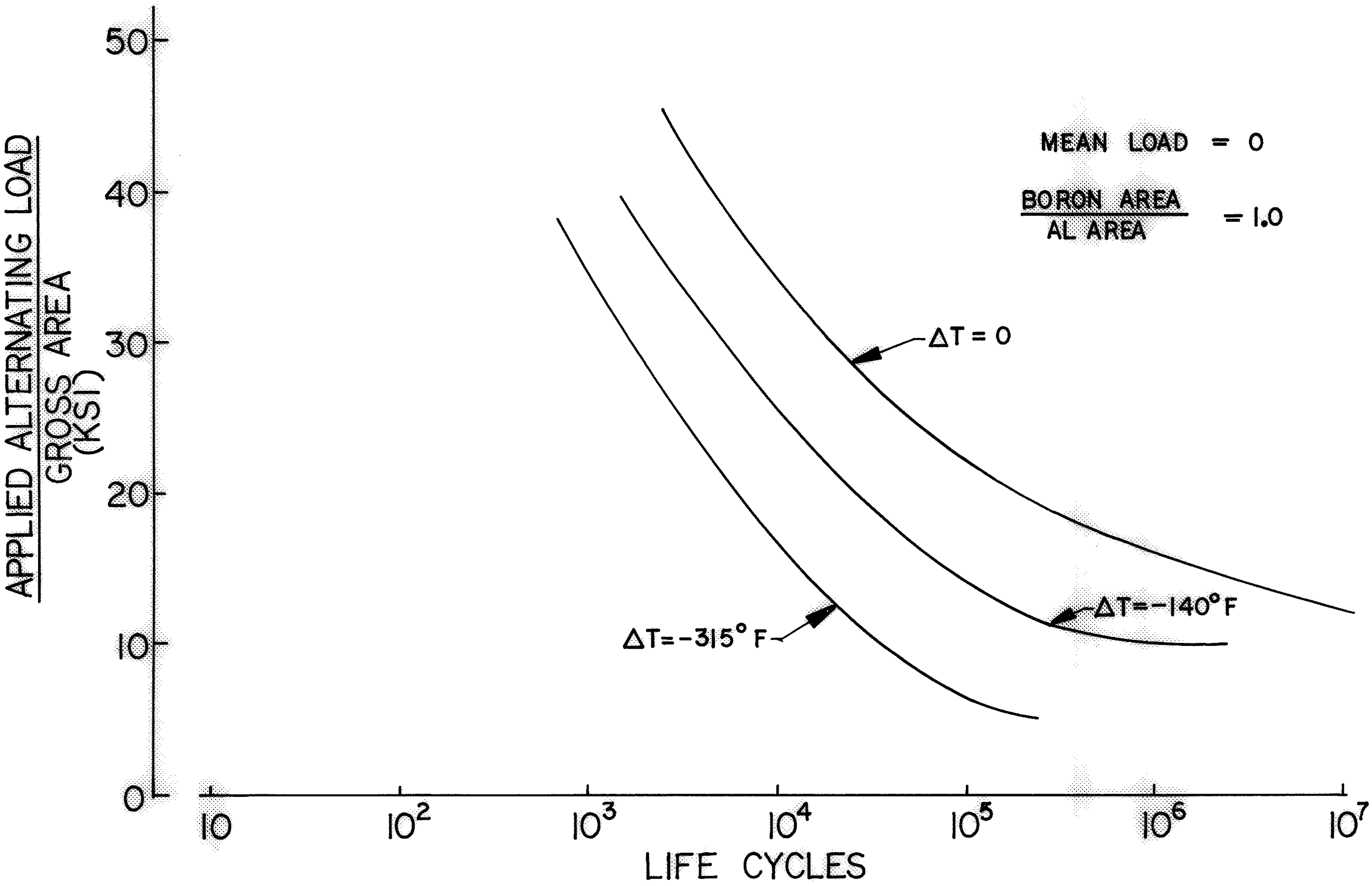


Figure 26 illustrates the means by which these changes are produced. Curve "A" represents the relation between applied load and actual aluminum stress for a compound aluminum-boron/epoxy member and curve "B" represents the relation when the boron/epoxy element for curve "A" is replaced with an aluminum element of equal weight. The offset at zero applied load in curve "A" is a result of thermal stress induced in the aluminum; the slope of the curve is determined by the relative percents of the two materials. An interesting observation is that for applied loads above the point of intersection, curve "A" generates reductions in both the mean stress and the stress amplitude more favorable than the equal weight condition; curve "B" provides a greater decrease in mean stress, but the change in stress amplitude is less effective due to the increased slope of the curve. The conditions which exist near this point are complicated by several variables which are associated with the compound element curves (as curve "A") such as slope changes. These changes are a result of different ratios of the two materials, the variation in the thermal stresses caused by changes in the material ratios and changes in the temperature differential.

Figure 27 gives an indication of the effectiveness of the addition of a unidirectional composite material of a given type and the load conditions for which each material is most desirable. Each of the enclosed areas on the diagram is marked as to the lowest weight configuration which will have a fatigue endurance of at least 10^3 cycles (the upper boundry of each area). As the stress condition moves below and to the left of the upper boundary, the life cycles in that area increases. The lower boundary of each area might exceed the defined life (10^3 cycles for this figure) but weight is greater than that of the adjacent area combination which will perform as required.

By utilizing the fatigue data and model (Figure 23) described above, certain weight parameters can be evaluated which give estimates of the structural weight savings possible with the reinforcement of fatigue critical structure. The equations on which the weight data are based are described below.

FIGURE 26

ACTUAL ALUMINUM STRESS VERSES APPLIED LOAD

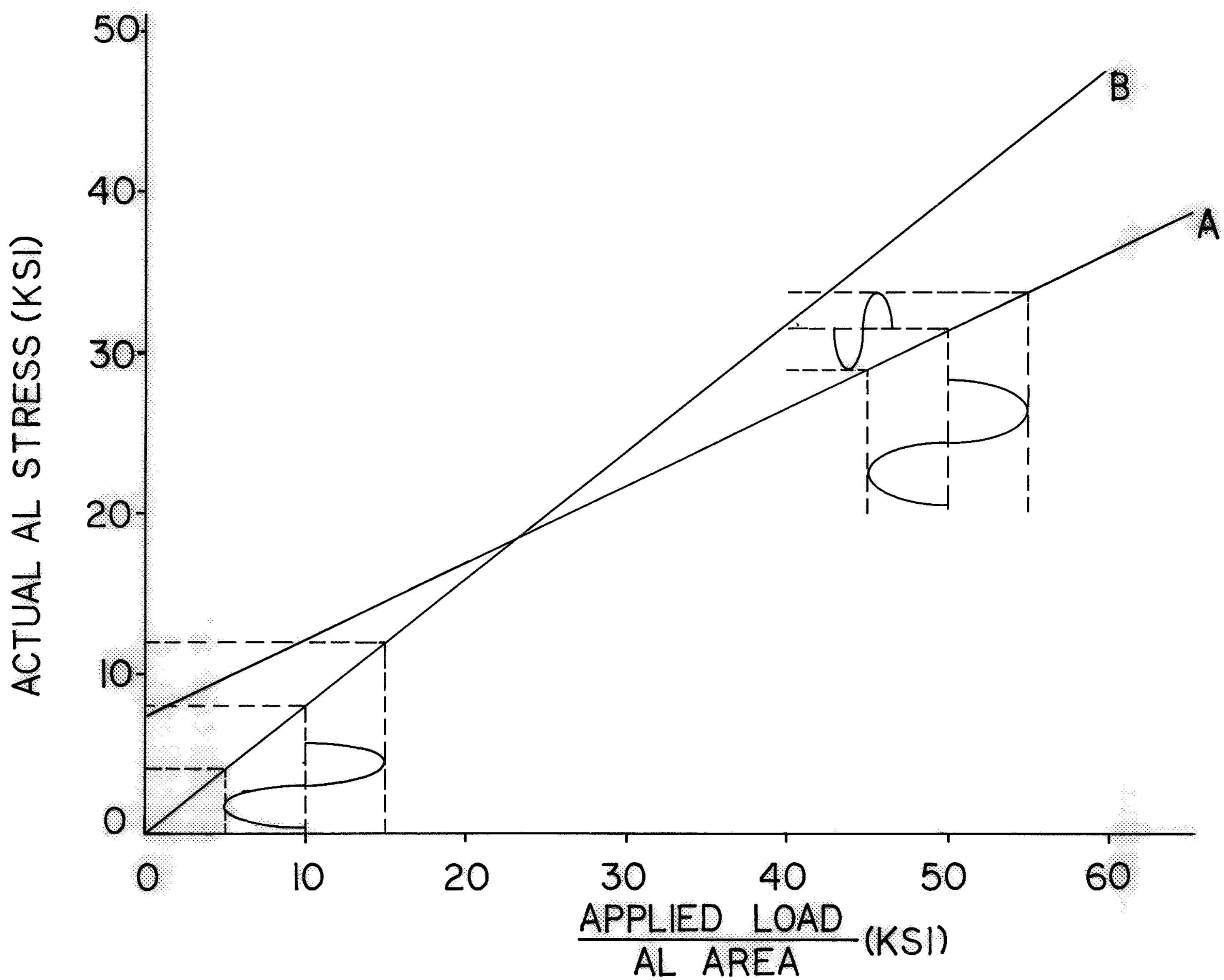
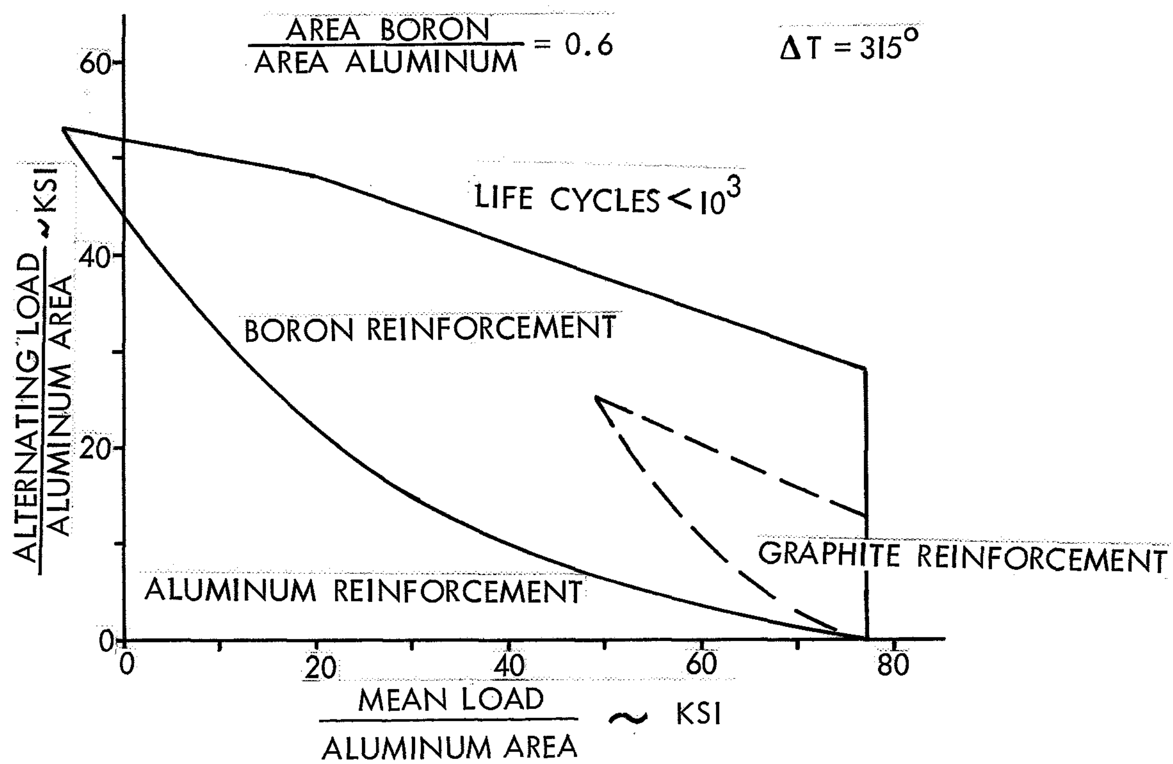


FIGURE 27



AREA BORON / AREA ALUMINUM = 0.6
ΔT = 315°
LIFE CYCLES < 10^3
BORON REINFORCEMENT
GRAPHITE REINFORCEMENT
ALUMINUM REINFORCEMENT
ALTERNATING LOAD / ALUMINUM AREA ~ KSI
MEAN LOAD / ALUMINUM AREA ~ KSI
0
20
40
60
80

The analysis begins with the assumption that the modulus-area product (EA) of the removed metallic material is equal to the EA of the composite material initially added. (Ref. Figure 23). That is, $E_1A_1 = E_mA_m$ and $A_mE_m = A_cE_c$.

For a room temperature bond ($\Delta T = 0$), the stresses in the metallic portion of the metallic/composite combination are:

$$\sigma_f = \sigma_1 = \frac{FE_m}{E_mA_m + E_cA_c} \tag{1}$$

This relationship neglects any eccentricities caused by the lack of symmetry of the lamination. For a bond at elevated temperature, the residual thermal stresses in the metallic material are:

$$\sigma_t = \frac{\Delta\alpha\Delta TE_mE_cA_c}{E_mA_m + E_cA_c} \tag{2}$$

The total stress in the metallic material as a result of combined mechanical loading and the residual thermal stress is given by

$$\sigma = \sigma_1 + \sigma_t$$

Since this stress must be kept below σ_f for the fatigue life of the combination to be at least equivalent to the unreinforced metallic, then

$$\sigma \leq \sigma_f$$

In order to return the stress level in the metal to σ_f after an elevated temperature bond, an additional amount of composite must be added (ΔA_c). This will result in an increase in EA of the total section; however, this is necessary since a reduction of EA will raise the stress level in the metallic above σ_f (Ref. Section 3, Figure 23). Thus,

$$\sigma_f = \sigma_1' + \sigma_t' \tag{3}$$

where

$$\sigma_t' = \frac{\Delta\alpha\Delta TE_mE_c\ (A_c + \Delta A_c)}{E_mA_m + E_c(A_c + \Delta A_c)} \tag{4}$$

$$\sigma_1' = \sigma_1\left(\frac{E_mA_m + E_cA_c}{E_mA_m + E_c\ (A_c + \Delta A_c)}\right) \tag{5}$$

Therefore

$$\sigma_t = \sigma_1\left(\frac{E_mA_m + E_cA_c}{E_mA_m + E_c(A_c + \Delta A_c)}\right) + \frac{\Delta\alpha\,\Delta TE_mE_c(A_c + \Delta A_c)}{E_mA_m + E_c(A_c + \Delta A_c)} \tag{6}$$

Solving equation (6) for $\Delta A_c/A_c$ gives:

$$\frac{\Delta A_c}{A_c} = \frac{\Delta\alpha\,\Delta TE_mE_c - (\sigma_f - \sigma_1)(E_m\frac{A_m}{A_c} + E_c)}{\sigma_fE_c - \Delta\alpha\,\Delta TE_m\,E_c} \tag{7}$$

Since the allowable stress level in the metallic, σ_f, is equal to σ_1 (Reference Equation 1), then:

$$\frac{\Delta A_c}{A_c} = \frac{\Delta\alpha\;\Delta TE_m}{\sigma_f - \Delta\alpha\,\Delta TE_m} \tag{8}$$

Now since

$$\Delta A_mE_m = A_cE_c$$

or

$$A_c = \Delta A_m\frac{E_m}{E_c}$$

Then Equation (8) can be solved for

$$\frac{\text{Total Composite Area}}{\text{Change in Metallic Area}}$$

This gives

$$\frac{\Delta A_c + A_c}{A_m} = \left(\frac{\Delta\alpha\,\Delta TE_m}{\sigma_f - \Delta\alpha\Delta TE_m} + 1\right)\frac{E_m}{E_c} = \left(\frac{\sigma_f}{\sigma_f - \Delta\alpha\,\Delta E_m}\right)\frac{E_m}{E_c}$$

For a unit length:

Weight = density x area

and for metal removed

$$\Delta W_m = \rho_m\;\Delta A_m$$

and neglecting adhesive weight increases for joints, the metal could be replaced by a weight of composite as follows:

$$W_c = \rho_c\;A_c$$

$$\Delta W_c = \rho_c\;\Delta A_c$$

To get a more reasonable representation of the relative weights, however, a non-optimum factor, N, is used. It is felt that a reasonable non-optimum factor is 1.15, which is a 15% increase over the theoretical oomposite weight.

Therefore,

$$W_c = N \rho_c A_c \qquad \text{and} \qquad \Delta W_c = N \rho_c \Delta A_c$$

Dividing ΔW_c by W_c and substituting the value of $\frac{\Delta A_c}{A_c}$ from equation (8) gives

$$\frac{\Delta W_c}{W_c} = \frac{\Delta \alpha \Delta T E_m}{\sigma_f - \Delta \alpha \Delta T E_m} \tag{9}$$

Adding W_c and ΔW_c, and dividing by ΔW_m gives

$$\frac{W_c + \Delta W_c}{\Delta W_m} = \frac{N \rho_c}{\rho_m} \left(\frac{A_c + \Delta A_c}{\Delta A_m} \right)$$

Substituting equation (9) for the bracket quantity, the following equation is obtained

$$\frac{W_c + \Delta W_c}{\Delta W_m} = \frac{\sigma_f}{\sigma_f - \Delta \alpha \Delta T E_m} \left(\frac{E_m}{E_c} \frac{\rho_c N}{\rho_m} \right)$$

Substracting the quantity (1) from each side of the above equation yields:

$$\frac{\Delta W_m - (W_c + \Delta W_c)}{\Delta W_m} = 1 - \left(\frac{\sigma_f}{\sigma_f - \Delta \alpha \Delta T E_m} \right) \left(\frac{E_m}{E_c} \frac{\rho_c N}{\rho_m} \right) \tag{10}$$

Multiplying both sides of equation (10) by $\left(\frac{\Delta W_m}{W_m + \Delta W_m} \right)$ yields:

$$\frac{\Delta W_m - (W_c + \Delta W_c)}{W_m + \Delta W_m} = \frac{\Delta W_m}{W_m + \Delta W_m} \left\{ 1 - \left(\frac{\sigma_f}{\sigma_f - \Delta \alpha \Delta T E_m} \right) \left(\frac{E_m}{E_c} \frac{\rho_c N}{\rho_m} \right) \right\} \tag{11}$$

Equations (10) and (11) represent the vertical and upper horizontal scales in Figures 29 - 32.

Figure (28) is a plot of weight ratio $\left(\frac{\Delta W_b}{W_b}\right)$ or $\left(\frac{\Delta W_g}{W_g}\right)$ versus design temperature differential (ΔT) for a maximum allowable stress in aluminum of 40,000 psi and a maximum allowable stress in titanium of 64,000 psi. Equation

(9) is used for this plot. For weight ratios of greater than 1.0, a weight penalty is incurred and for those less than 1.0 a weight savings is realized.

In Figures 29, 30, 31 and 32 the stress level at ultimate wing box load is allowed to vary. By assuming values for this stress level and the design temperature differential, a weight ratio can be calculated. One (1) minus the weight ratio is then calculated and plotted on the vertical scales. A fractional equivalent of the amount of metal that is to be replaced by composite is chosen. This fractional equivalent multiplied by (1-weight ratio) gives the weight savings. This weight savings is shown on the upper horizontal scale. Equations (10) and (11) are the basic equations used to complete these figures.

An example of the use of the weight charts for assessing potential weight savings of composite reinforced fatigue critical structure follows. If the stress level is 40,000 psi, the design temperature differential is 150°F and the amount of metal replaced by composite is 30%, initially start at 40,000 psi on lower horizontal scale, proceed vertically to 150°F on temperature scale, then proceed horizontally to .3 on amount of metal replaced by composite lines, and finally proceed vertically to the Weight Savings on the upper horizontal scale. The Weight Savings for this example on Figure 29 is 15%.

FIGURE 28

WEIGHT RATIO Vs TEMPERATURE DIFFERENTIAL COMPOSITE-METAL BONDED STRUCTURE

(GROUND RULES OF C-130 COMPOSITE CENTER WING BOX)

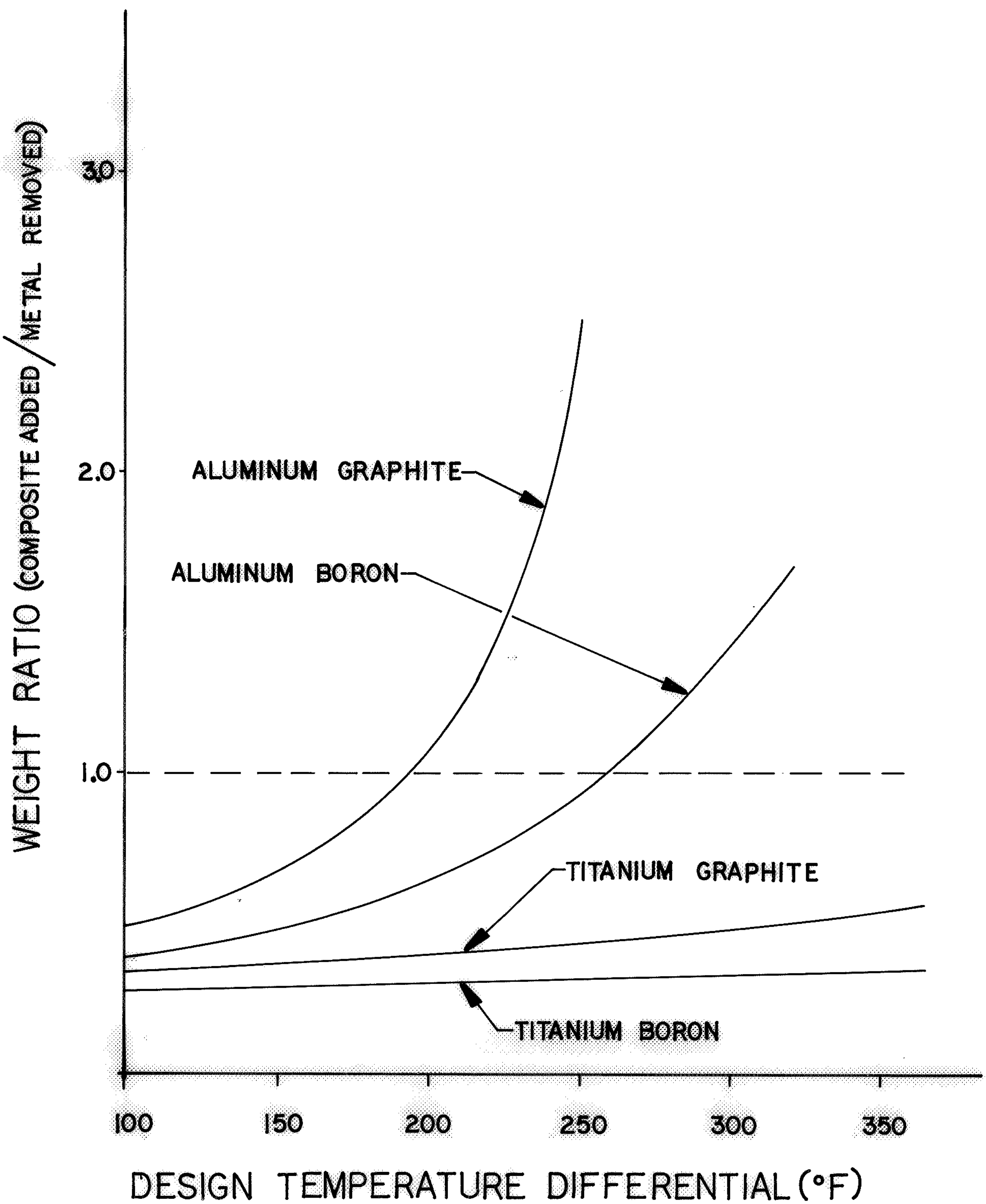


FIGURE 29

WEIGHT SAVINGS

ALUMINUM-UNIDIRECTIONAL BORON BONDED STRUCTURE Vs ALL ALUMINUM STRUCTURE

(ALUMINUM STRESS HELD CONSTANT)

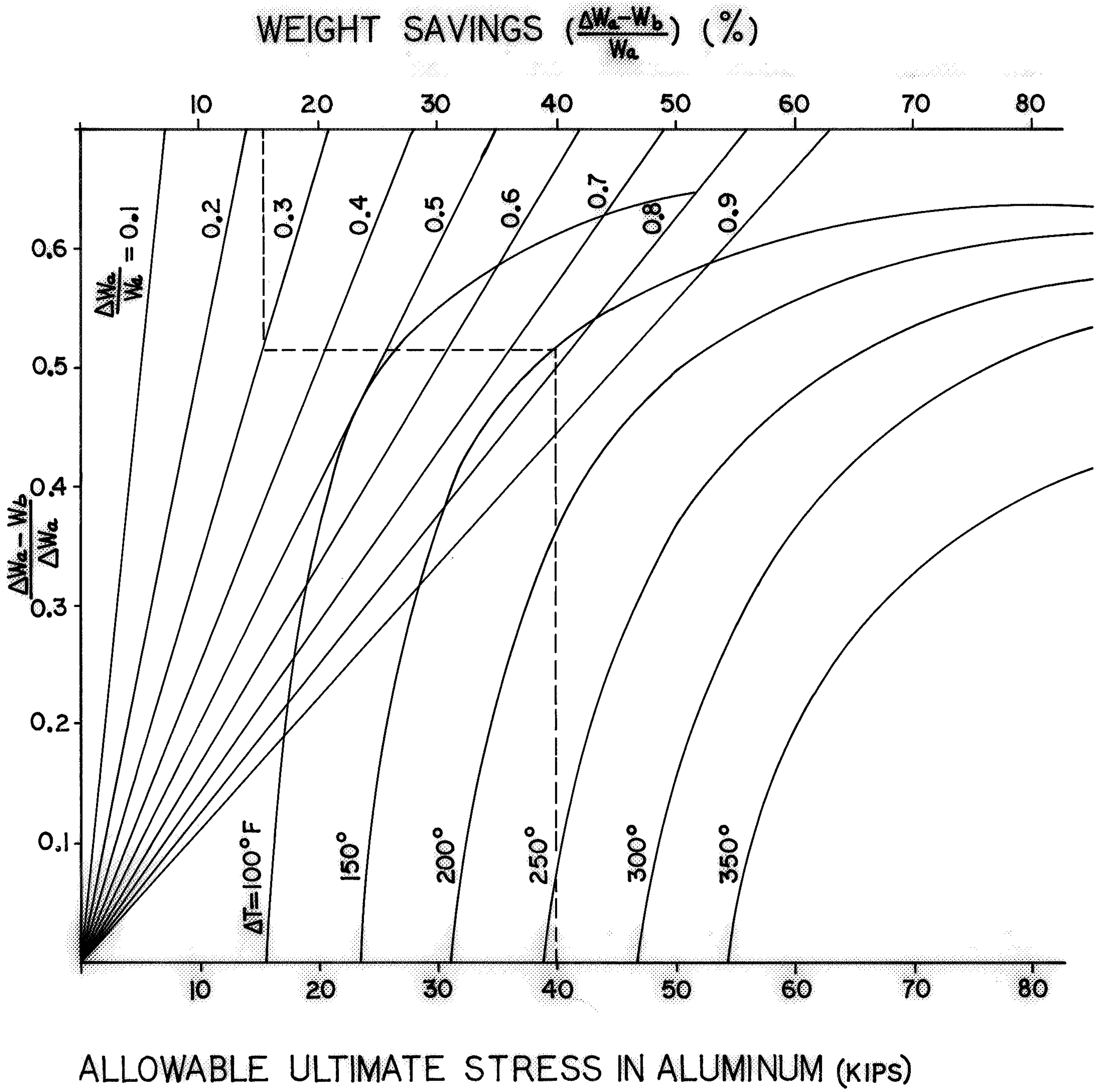


FIGURE 30

WEIGHT SAVINGS

ALUMINUM-UNIDIRECTIONAL GRAPHITE BONDED STRUCTURE Vs ALL ALUMINUM STRUCTURE

(ALUMINUM STRESS HELD CONSTANT)

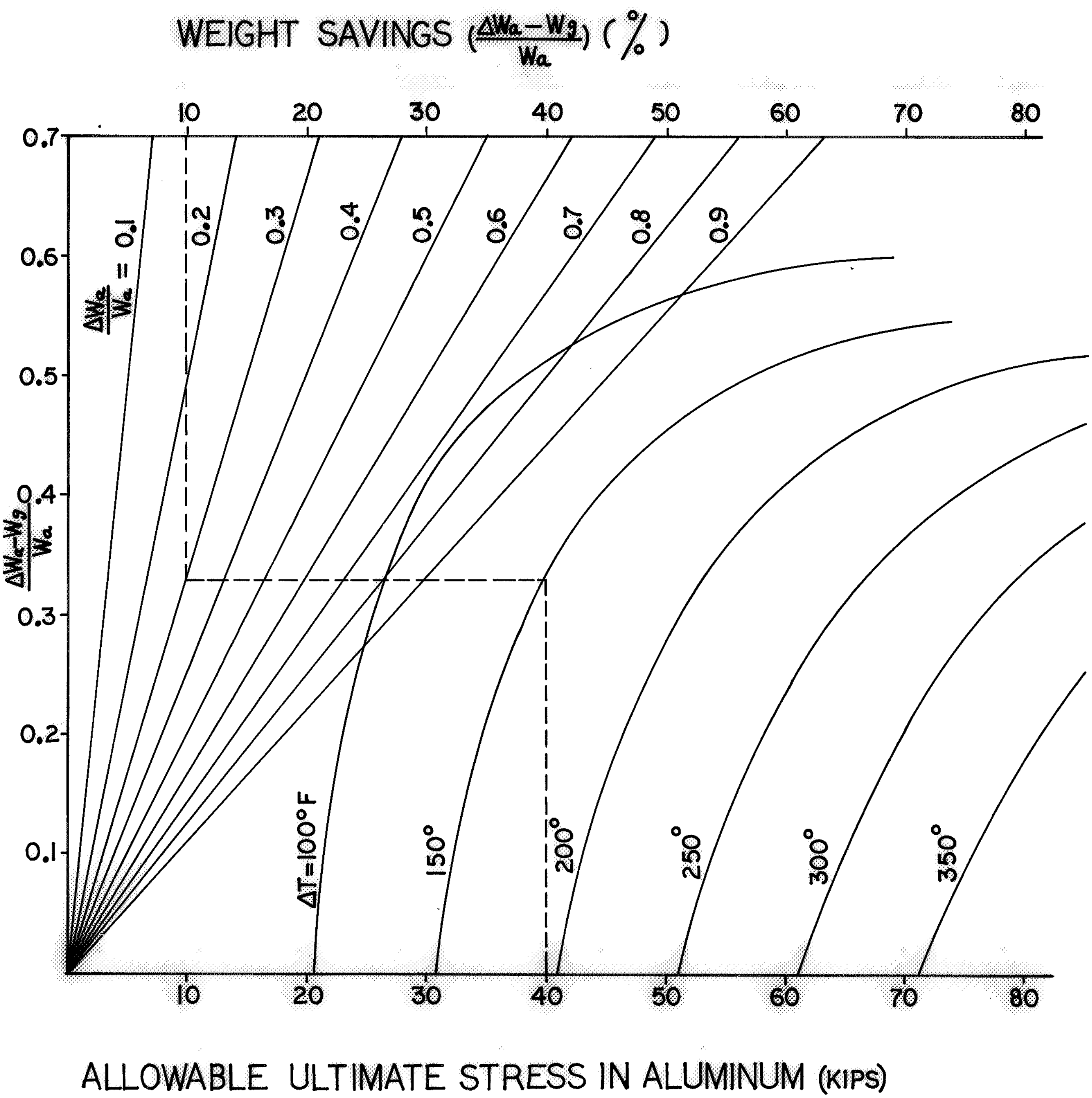
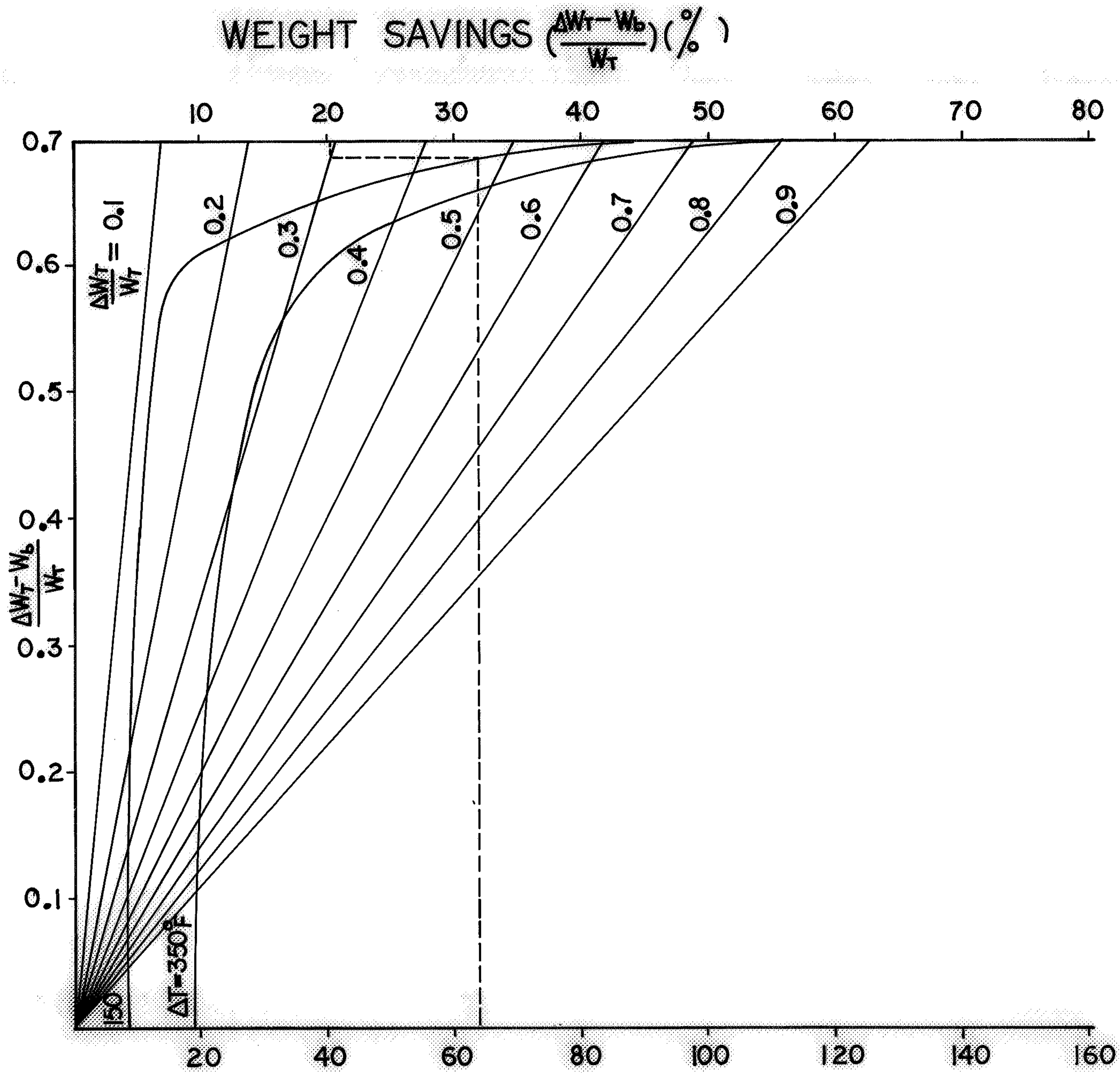


FIGURE 31

WEIGHT SAVINGS
TITANIUM-UNIDIRECTIONAL BORON BONDED STRUCTURE Vs ALL TITANIUM STRUCTURE
(TITANIUM STRESS HELD CONSTANT)



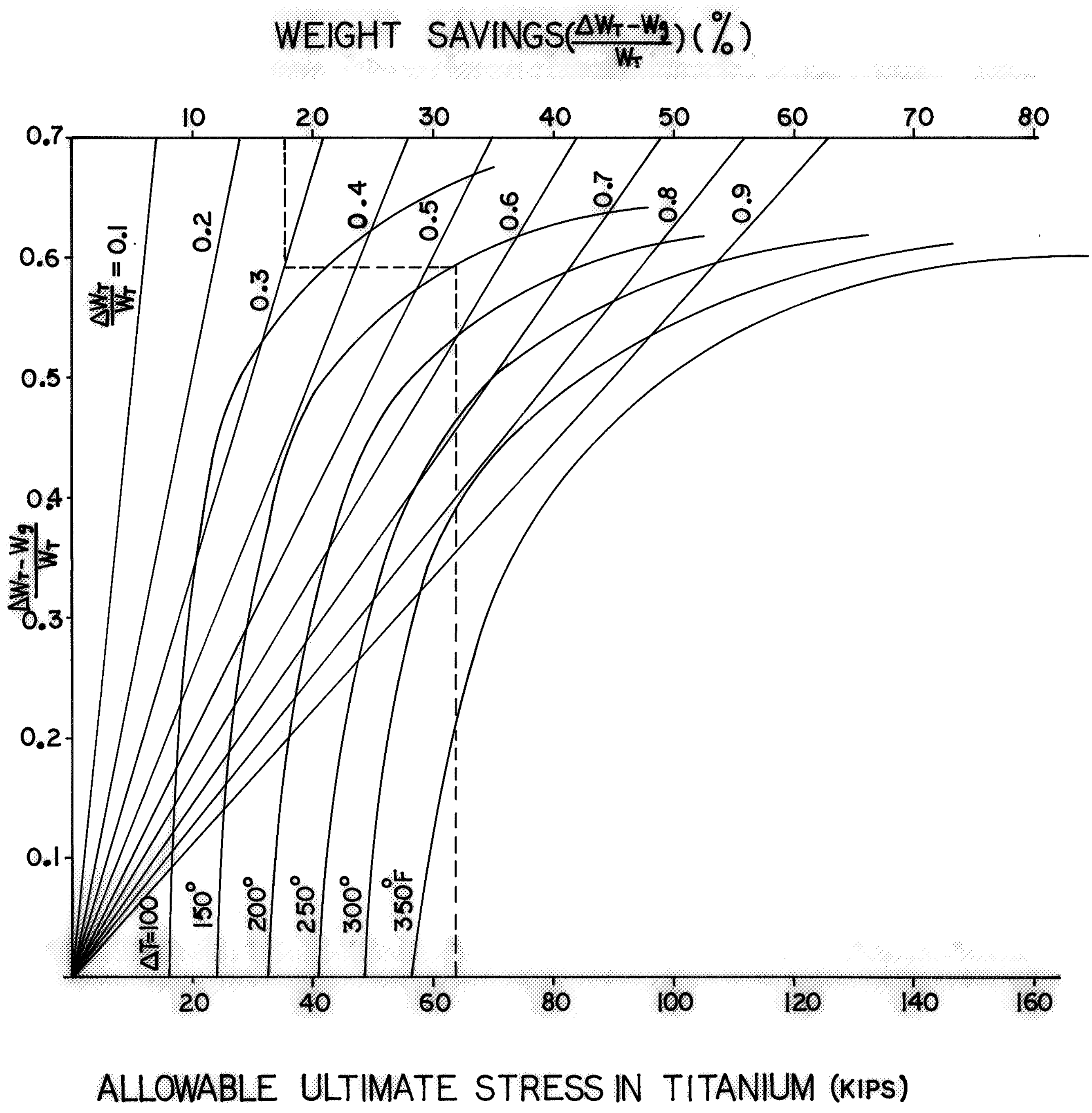
ALLOWABLE ULTIMATE STRESS IN TITANIUM (KIPS)

FIGURE 32

WEIGHT SAVINGS

TITANIUM-UNIDIRECTIONAL GRAPHITE BONDED STRUCTURE Vs ALL TITANIUM STRUCTURE

(TITANIUM STRESS HELD CONSTANT)



Stiffness

Parametric studies were initiated to determine the optimum metallic and composite material combinations for improvement of stiffness critical structure. Initial studies examined the efficiency of the reinforcement of flat panels with composite materials to improve the buckling characteristics.

Significant structural improvements were found to be obtainable from the reinforcement of typical flat aluminum and titanium panels with unidirectional composite material. Initial buckling loads for symmetrically and unsymmetrically reinforced flat panels were determined utilizing existing computer programs which calculate the buckling loads of anisotropic plates. (Reference 9 and 10). Generally speaking, the reinforcement with the graphite/epoxy material was more efficient. It was found that for the panel dimensions assumed, the unsymmetrical reinforcement was most efficient. This was a surprising result since it has been generally shown that unsymmetrically laminated all-composite plates have lower buckling loads than the equivalent symmetrically laminated plate (Reference 11). For this study, it was assumed that the composite was bonded to the metallic plate at room temperature so no initial curvatures were induced.

The initial metallic panel dimensions assumed were a thickness of 0.12 inches and lengths $a = 20.0$ inches and $b = 10.0$ inches. Uniaxial compression on the short side was assumed. Increments of ten percent of the panel thickness were removed and an equivalent weight of unidirectional composite was added to the surface either symmetrically or unsymmetrically. The initial buckling load for the panel was calculated utilizing the computer programs referenced above. The buckling loads for the unsymmetrically reinforced panels were obtained by utilizing the "reduced stiffness matrix", $\bar{D} = D - BA^{-1}B$, for the plate (Reference 9 and 12). That is, the bending stiffness matrix, D, for the plate, which is referenced to the plate mid-plane, is replaced by the $\bar{D}$ matrix, which causes a partial uncoupling of the stress resultants and plate curvatures. The use of the "reduced stiffness matrix" $\bar{D}$ has been shown to give satisfactory agreement with buckling loads determined by more exact theory and with experimental data (Reference 11 and 12).

The results of the buckling study are presented in Figures 33-40. In Figures 33 and 34, the initial compressive buckling loads for symmetrically and unsymmetrically reinforced aluminum plates with clamped boundary conditions are presented. In Figures 35 and 36, the same data is presented for simply supported boundary conditions. The compressive buckling loads for composite reinforced titanium plates with clamped boundary conditions are shown in Figures 37 and 38. The simply supported results are given in Figures 39 and 40.

The results of these buckling studies indicate that graphite reinforcement is the most efficient from a stability standpoint. This is primarily due to the low density which gives an increased reinforcement thickness that raises the flexural rigidity since it is proporational to t^3. In other words, the effects of the lower unidirectional modulus of the graphite are more than offset by the increase in thickness of the graphite composite.

FIGURE 33

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED ALUMINUM PLATES
(SYMMETRICAL REINFORCEMENT)

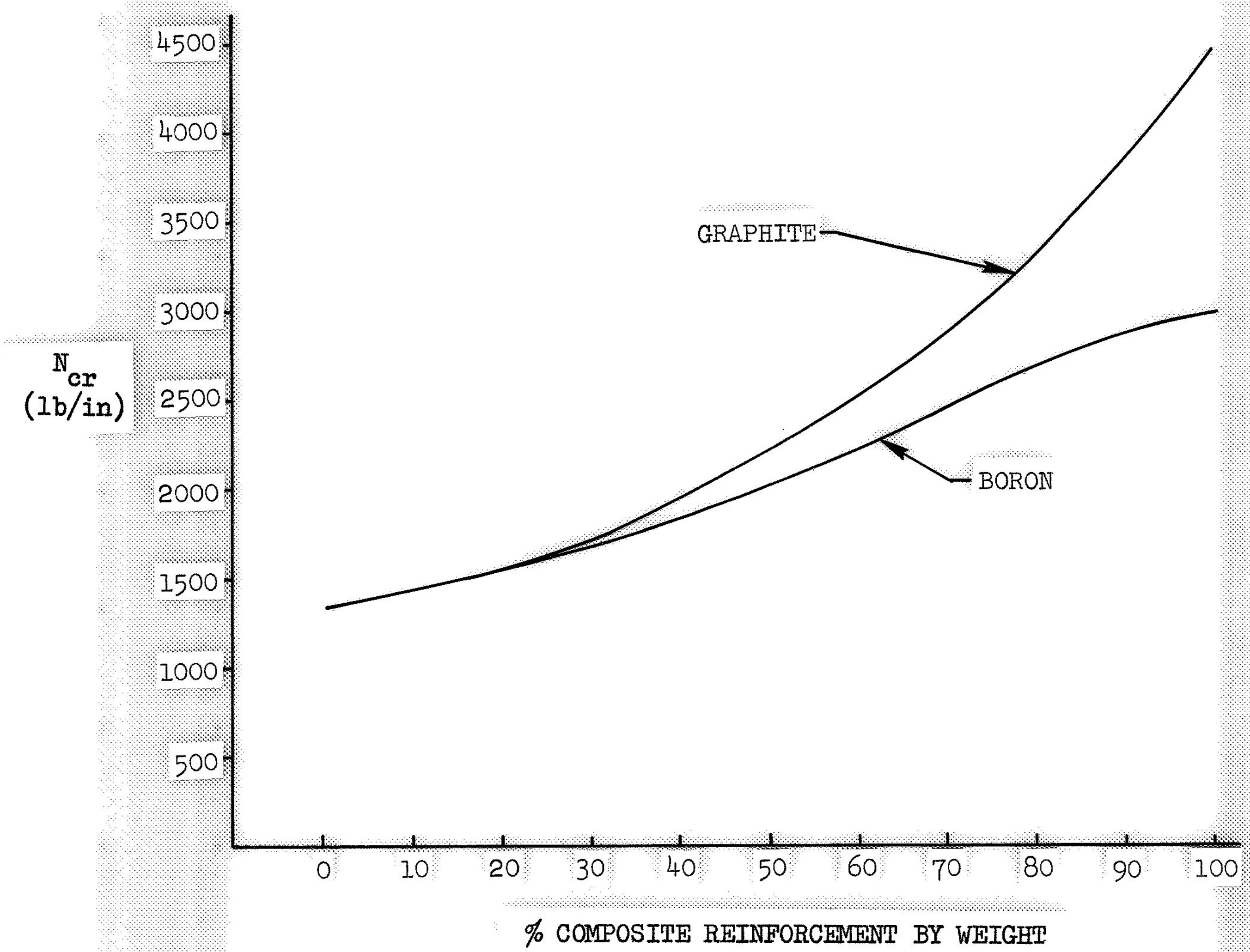


FIGURE 34

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED ALUMINUM PLATES
(UNSYMMETRICAL REINFORCEMENT)

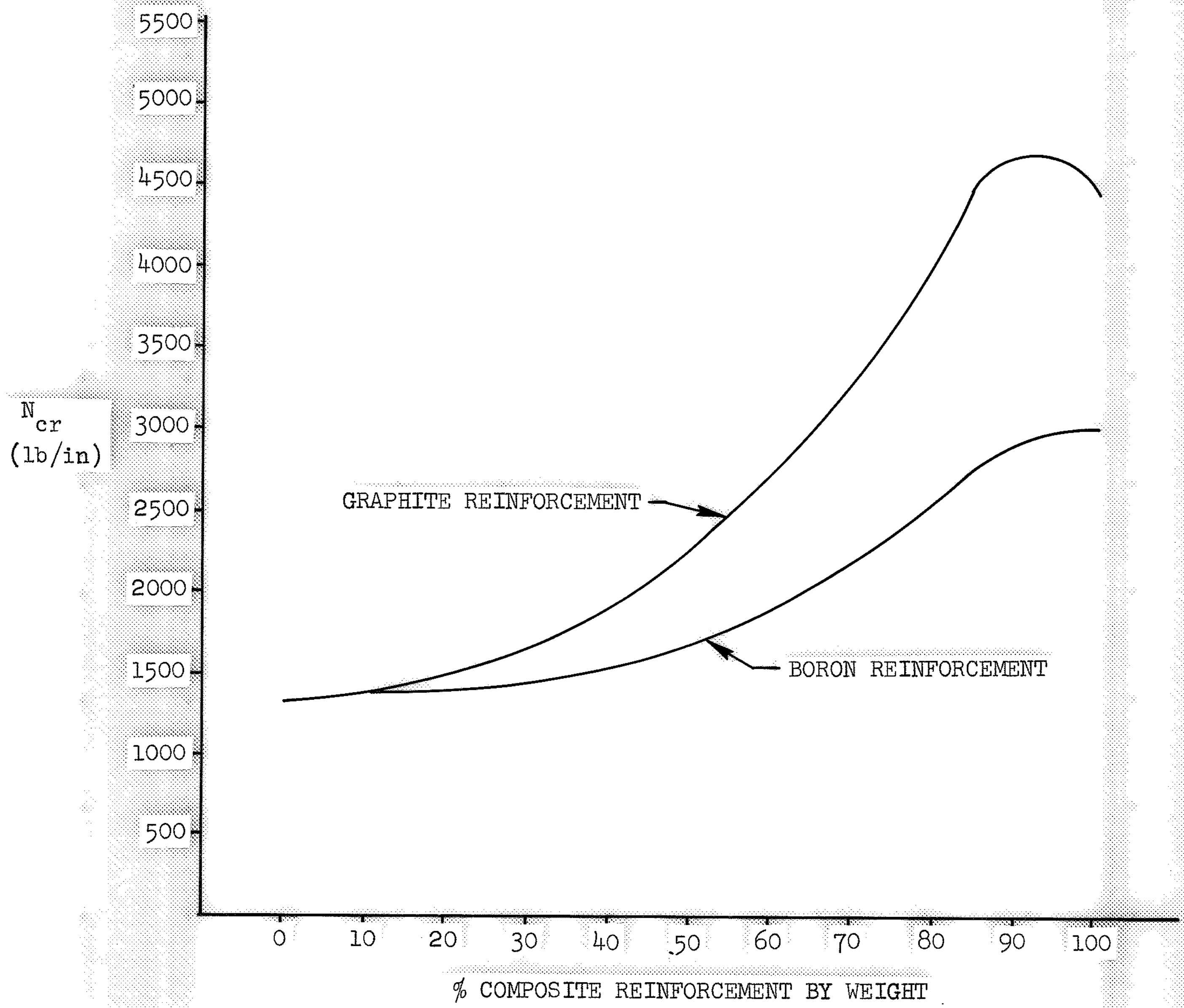


FIGURE 35

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED ALUMINUM PLATES
(SYMMETRICAL REINFORCEMENT)

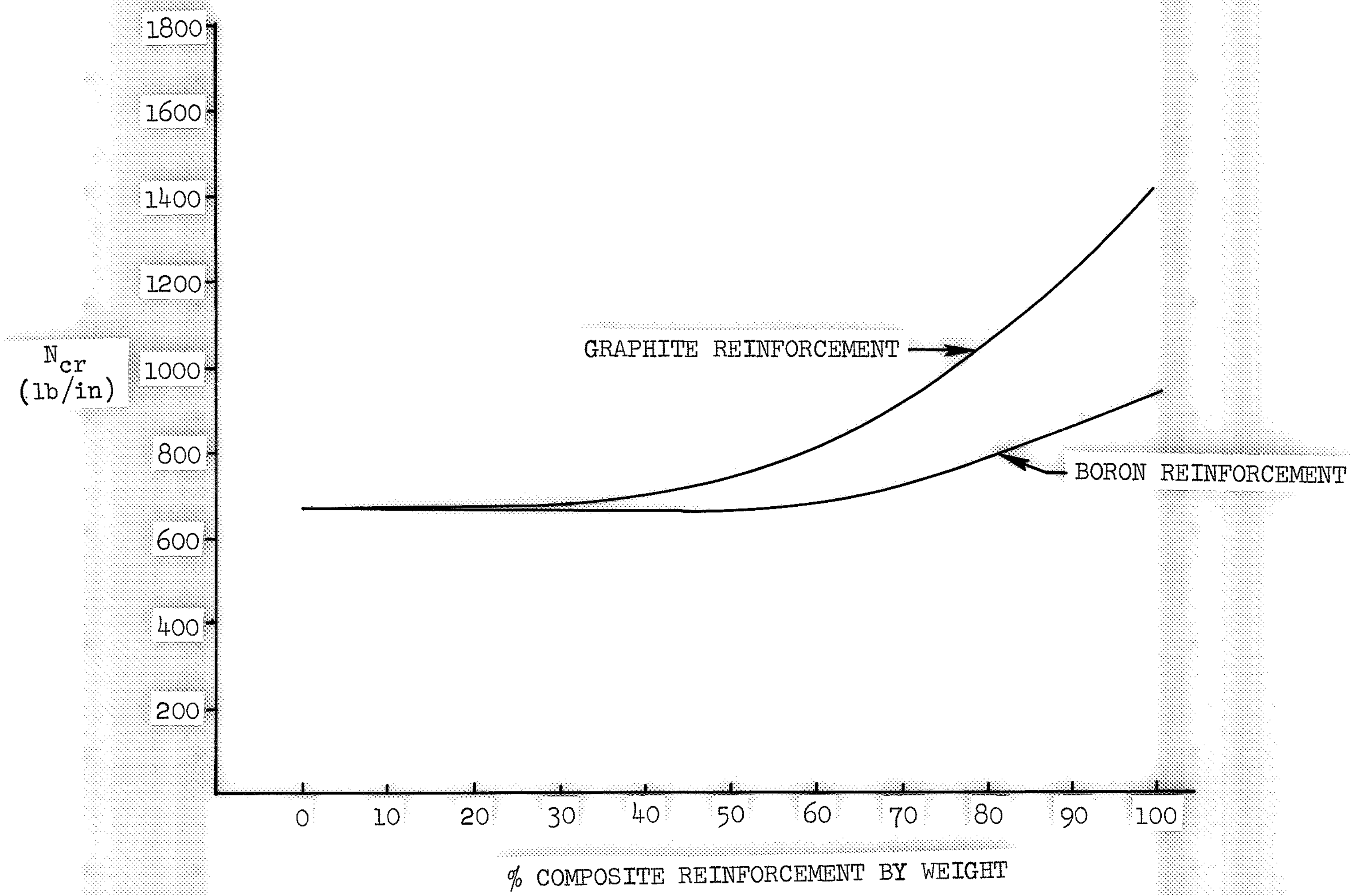


FIGURE 36

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED ALUMINUM PLATES
(UNSYMMETRICAL REINFORCEMENT)

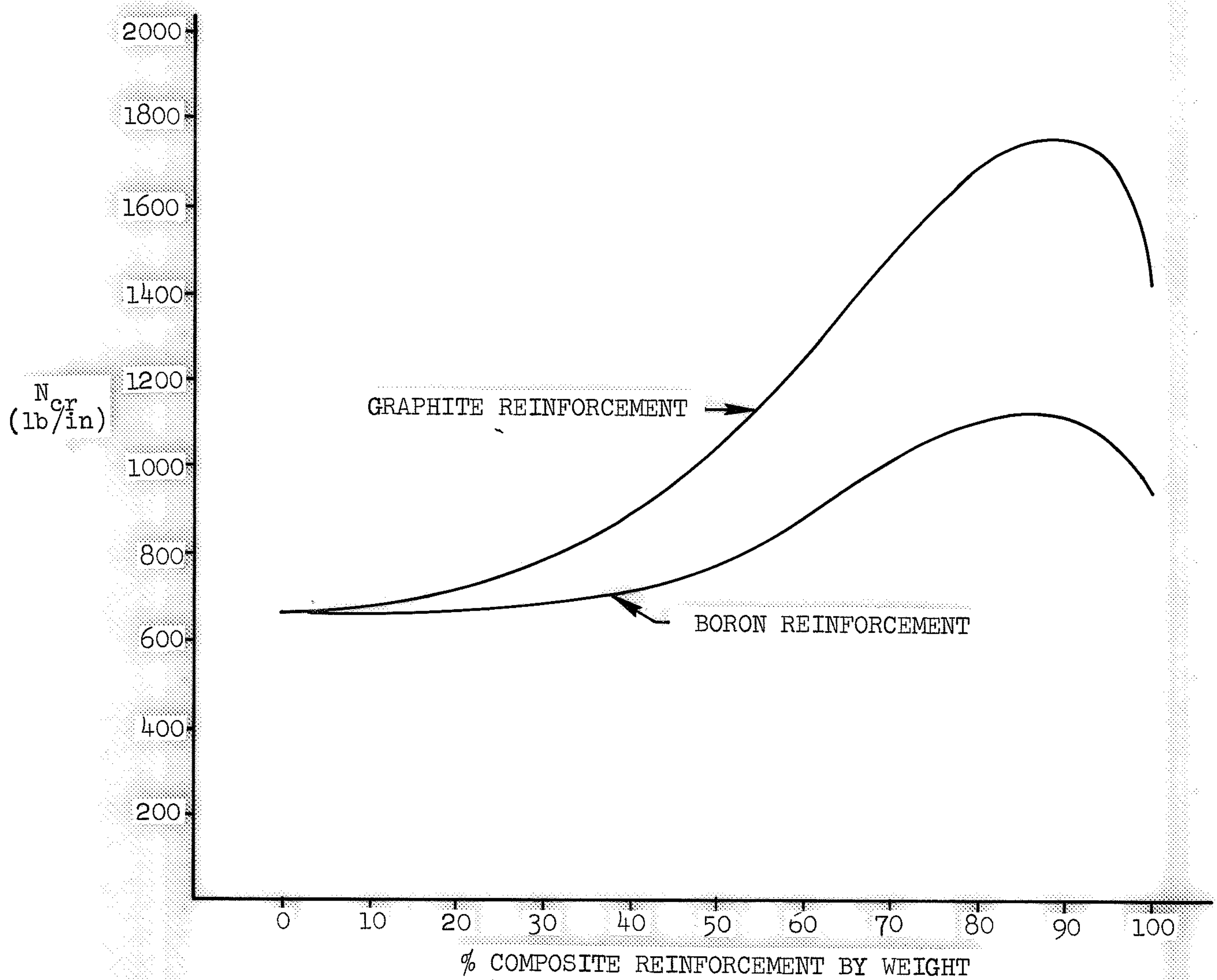


FIGURE 37

INITIAL BUCKLING LOAD FOR EQUIVALENT WEIGHT REINFORCED TITANIUM PLATES
(SYMMETRICAL REINFORCEMENT)

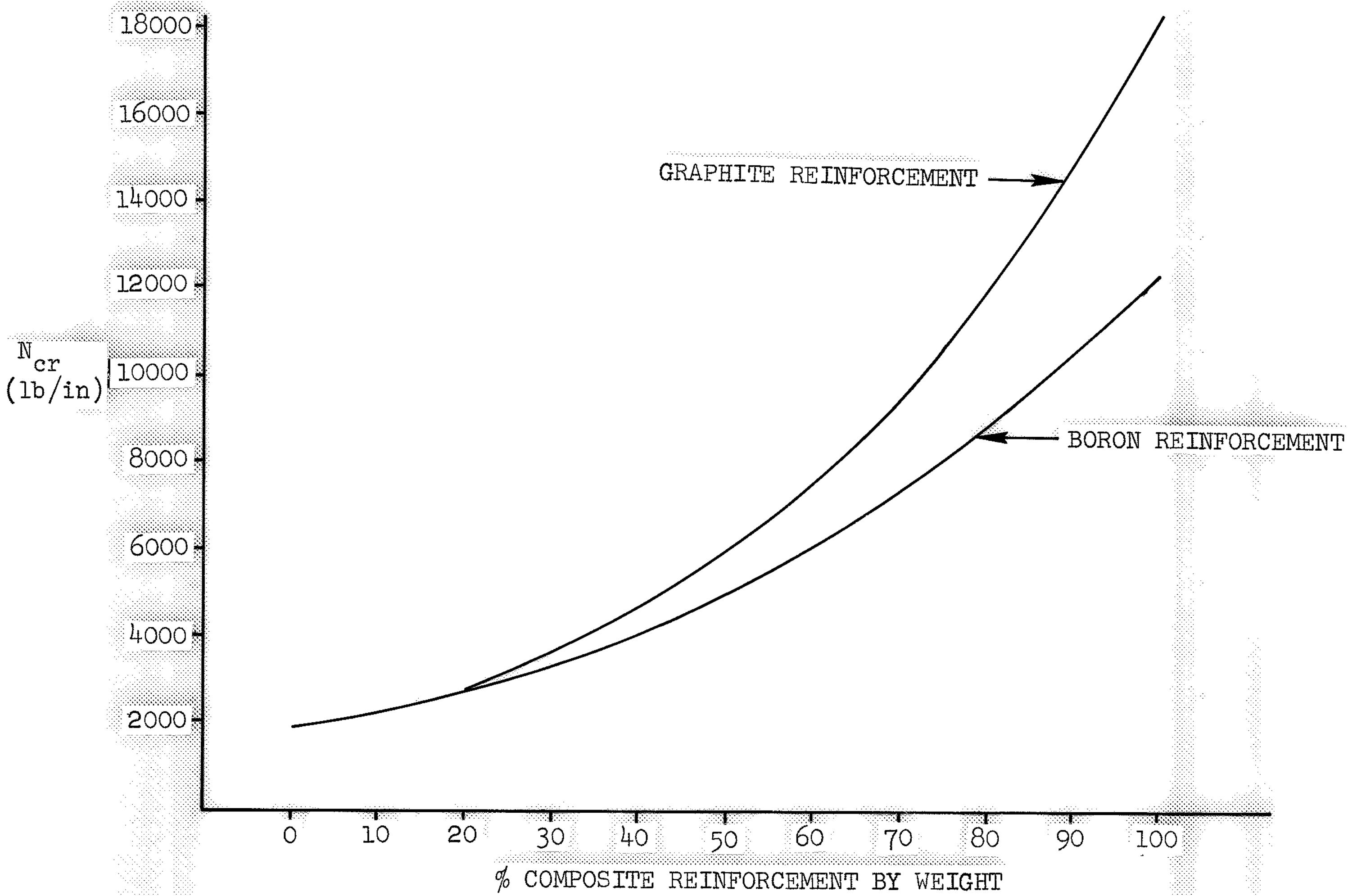


FIGURE 38

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED TITANIUM PLATES
(UNSYMMETRICAL REINFORCEMENT)

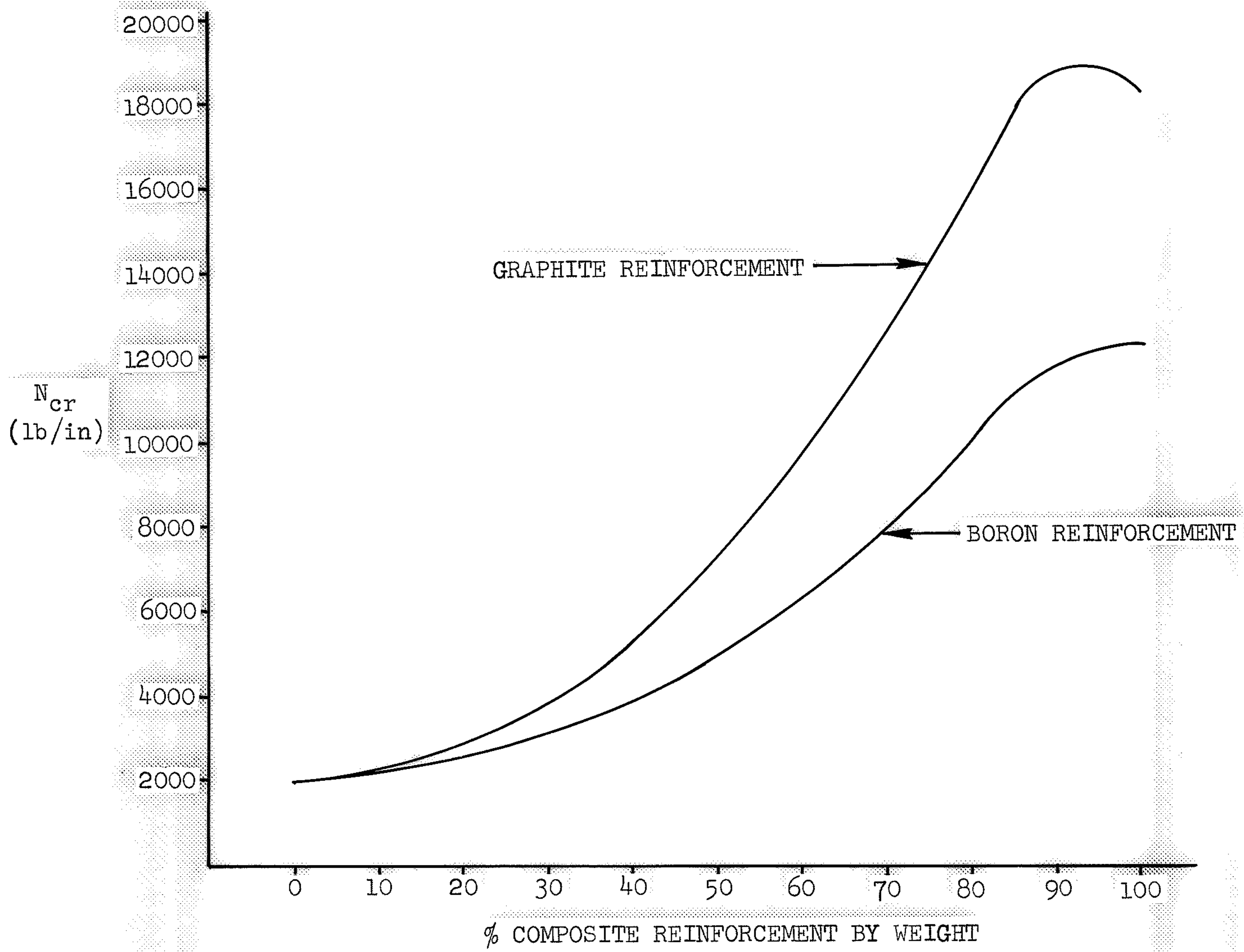


FIGURE 39

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED TITANIUM PLATES

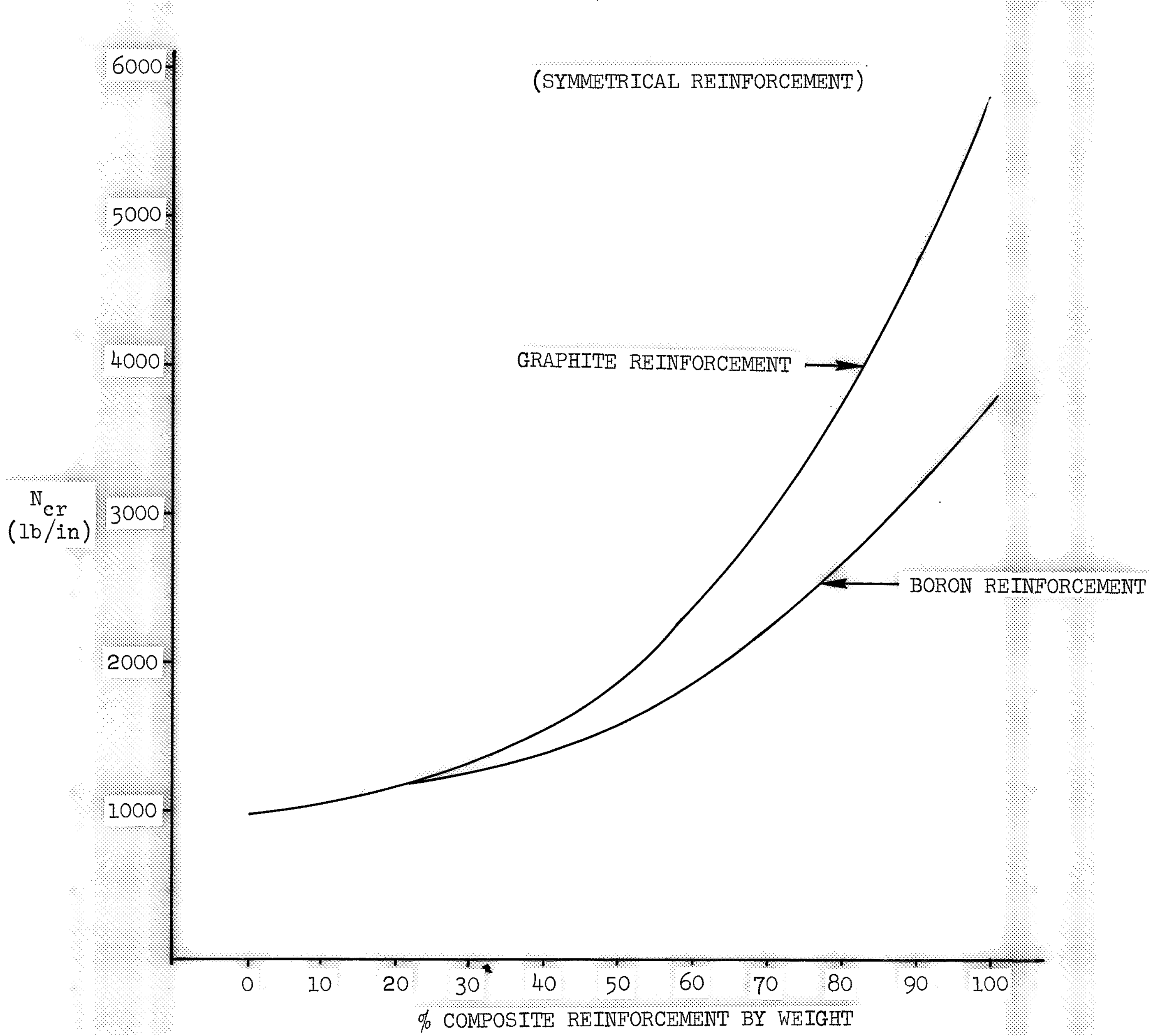
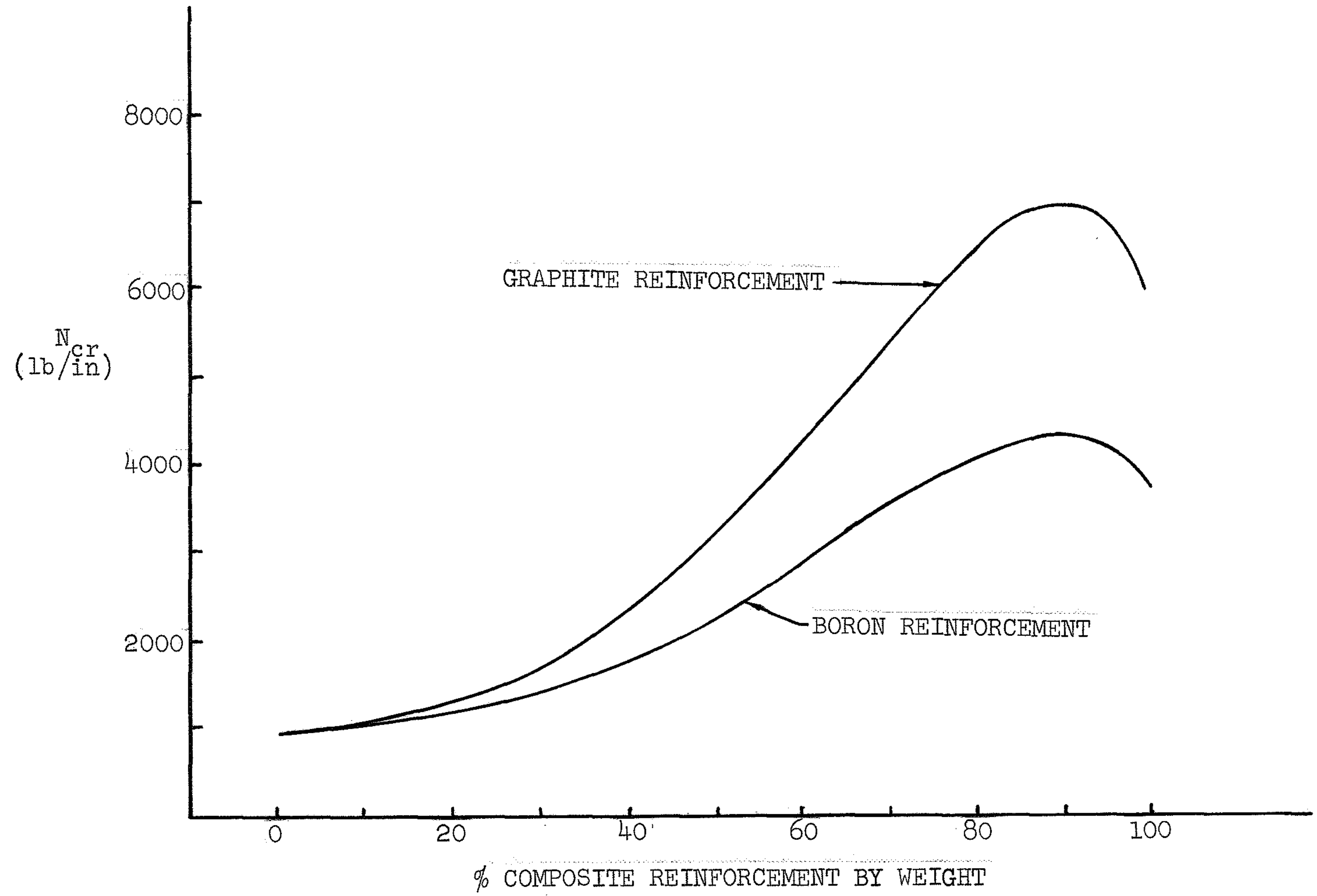


FIGURE 40

INITIAL BUCKLING LOADS FOR EQUIVALENT WEIGHT REINFORCED TITANIUM PLATES
(UNSYMMETRICAL REINFORCEMENT)



SYMBOLS FOR APPENDIX B

σ_f = Maximum stress allowed in metal for fatigue considerations (Design Ultimate Stress)

σ_1 = Stress in metal due to applied load (Initial)

σ_1' = Stress in metal due to applied load (Final)

σ_t = Thermal induced stress in metal (Initial) = 0

σ_t' = Thermal induced stress in metal (Final)

E_m = Modulus of metal

E_c = Unidirectional Modulus of composite

A_m = Area of metal

ΔA_m = Area of metal removed

A_c = Area of composite neglecting ΔT

ΔA_c = Area of composite to account for ΔT

ΔT = Temperature differential

$\Delta \alpha$ = Difference between the metal and composite coefficients of expansion

F = Applied force

W_m = Weight of unit length of metal

ΔW_m = Weight of unit length of metal removed

W_c = Weight of unit length of composite neglecting ΔT

ΔW_c = Weight of unit length of composite account for ΔT

N = Non-optimum factor = 1.15

$E_1 A_1 = E_m A_m$

$A_m E_m = A_c E_c$

APPENDIX C

DESIGN CONCEPTS

For the Reinforcement Concept, IA, a considerable amount of work is associated with wing box disassembly and in the application of surface protection and sealing of aluminum structure. A portion of the cost required for this concept is not directly related to the fabrication of a composite material structure.

The Compound-Composite Concepts offer the greatest overall cost/producibility effectiveness and generally offer the better program development plan, considering tooling requirements, development problems, and time span requirements. However, the potential level of weight savings is considerably less than that for the All-Composite group.

The All-Composite Concepts provide a considerably higher level of potential weight savings, but the cost/producibility effectiveness is considerably lower.

A more detailed summary of the information given in Figure 41 is presented in Table III. This table gives the ratio of composite material used to the total weight saved, the percentage of composite utilized, the dollars per pound of weight saved, the increase in cost of the composite designs over the aluminum reinforced box, and the percent weight saved for each concept. In Table IV, the cost/producibility effectiveness of each design is ranked.

In all of the design concepts, except IIIA and IIIB, the bending stiffness of the wing box, EI, is greater than the aluminum reinforced design. This is caused by the additional composite that was added to reduce the residual thermal stress (See Figure 23). The EI for the all-composite concept is equivalent to the bending stiffness of the C-130 wing box before modification. In all the metallic/composite concepts, except IIB, the aluminum or titanium material provides all the torsional stiffness in the box because the 0° composite orientation has negligible shear stiffness. In concepts III, torsional stiffness is obtained by using $\pm 45^{\circ}$ plies along with the 0° plies.

The weight for each concept except IIIA is based on the weight of the skin-stringer elements. The skins and stringers comprise 80% of the total weight of the wing box. In concept IIIA there was a non-optimum factor included in the weight analysis. The non-optimum factor compensates for weight increases at rib to surface panel connections and bonding of face-sheets to the core.

CONCEPT	TYPE	CONFIGURATION	DESIGN	WEIGHT SAVED	Δ COST $/lb	Δ COST $/A/C	PRODUCIBILITY	GENERAL RECOMENDATIONS	RANK
REINFORCEMENT CONCEPT	I A		BORON/ALUMINUM	531.8	587	311,910	The upper surface is removed from ship, but remains assembled with the hat sections. The lower surface will remain assembled in the ship with ribs and other structures. Bond composite skin to aluminum face sheet and composite stiffeners to hat sections. Problem area of greatest magnitude will be cleaning the surfaces for bonding. These currently have an epoxy coating which is impervious to chemical cleaning without damage to the aluminum present, and can only be removed by abrasive techniques. Inspection for, and assurance of, a clean surface for bonding is very questionable. Another problem exist with the lower surface still intact in the ship. This will require a vacuum cure of the composite skin to the aluminum skin. Cleaning of this surface will be doubly difficult since it is on the underside of the wing outboard of the fuselage. A similar situation is present for bonding the hat reinforcements, but this is further complicated by heating the length of the wing box to 160°F on the ship. Inspection of stiffener bond to hat section essentially impossible.	It is recommended that this alternate be dropped from consideration due to inherent problems of producibility and lack of reliability in processing and lack of reliability in structure being reworked.	7
			GRAPHITE/ALUMINUM	365.1	691	252,586			
COMPOUND COMPOSITE CONCEPT	II A		BORON/ALUMINUM	642.7	376	241,907	Hat sections with reinforcing composites are all precured and mechanically assembled to aluminum skin. The procured composite skin is bonded to the skin panel. Warpage due to differences in thermal coefficients of expansion during bond cycles is not fully predictable. Methods of compensating and straightening must be evaluated.	This design concept restricts visual inspection of the composite material. However, the design is adequate for 92% of design limit load (3G) without the composite material. This design affords the most direct comparison with the previous all aluminum design since the same skin and stringer extrusions will be used.	1A
			GRAPHITE/ALUMINUM	451.4	284	128,357			1B
	II B		BORON/ALUMINUM	652.1	912	594,987	Machine from wide rib extrusion or plate. Modify rib caps as required. Precured composites are bonded to the aluminum panels. Warpage due to thermal coefficient differences must be evaluated and offset by orientation of fiber or mechanical means. Study is required in this area. Rib caps would be installed over composites requiring drilling of holes for fasteners. Countersinking would be in aluminum.	This approach is a much higher cost approach than others. The reduction of skin thickness with the use of the ±45° plies in the skin reinforcement to maintain the box GJ proved to be an inefficient design concept from a weight standpoint. Economics and weight saved eliminate this alternate in favor of others.	6
		0/±45°	GRAPHITE/ALUMINUM	199.5	1,654	329,963			
	II C		BORON/ALUMINUM	945.8	325	307,299	Aluminum - This concept is considered to be producible and represents design concepts of intergrally stiffened structure that are state of the art on current aircraft. The use of intergrally stiffened skins is advantageous from a manufacturing standpoint. Bonding precured composites to the aluminum panel is similar in approach to other alternates. The potential warpage problems must be addressed in design and fabrication. Methods of control and mechanical means of straightening must be evaluated. Titanium - This design affords all of the advantages of al. and greatly reduces the problems involving warpage due to the more compatible thermal coefficient of expansion of titanium to the composites. This further aids in the weight saving and cost areas.	Aluminum - This concept is recommended for further study. It represents the current state of the art in aluminum combined with advanced composite materials. In addition this concept represents the most advantages combination of weight, cost, design, and manufacturing development. Titanium - This approach warrants further study as an independent program.	2B
			BORON/TITANIUM	1,146.2	223	255,732			2D
			GRAPHITE/ALUMINUM	677.1	228	154,120			2A
			GRAPHITE/TITANIUM	986.6	154	151,474			2C
	II D		BORON/ALUMINUM	736.1	270	198,362	Similar to IIC.	Similar to IIC but does not afford a sufficient usage of composites to fully justify selection.	3
			GRAPHITE/ALUMINUM	549.5	203	111,664			
	II E		BORON/ALUMINUM	694.0	393	272,400	Machined plate and extrusion mechanically attached with precured composite bonded to extrusion cap. Similar to IIC in other respects.		4
			GRAPHITE/ALUMINUM	361.8	416	148,549			
	II F		BORON/ALUMINUM	814.0	293	238,365	All considerations discussed under Concept IIC apply to this concept. However this design incorporates additional complexities in the aluminum design.		5
			GRAPHITE/ALUMINUM	500.3	300	149,961			
ALL COMPOSITE CONCEPT	III A		BORON	1,906.2	487	929,047	A considerable development program is required to determine manufacturing feasibility of boron usage in the hat sections. Approximately 15 feet of boron would have to be corrugated into a final panel width of 8 feet for a length of 40 feet for a single skin concept. Using hat sections would reduce the complexity and provide a better reliability factor for positioning the 0° plies in the caps of the risers. In both methods, the stiffness of the boron would make holding the material to contour in the lay-up operation difficult. The complexity of the end transitions would require additional development. Problems would generally be greatly reduced with the use of Graphite. However major development work is still required.	This design concept is an advanced configuration and is not recommended for a flying aircraft at this stage of development.	9
			GRAPHITE	1,877.5	192	360,409			
	III B		BORON	1,528.0	590	901,466	In view of the experiences gained in the Boron Slat Program, this concept would be producible. Techniques developed for attaching the rib caps on the slat would be directly applicable to the wing box. Attachment of the rainbow fittings, as shown, would not produce any significant problems in light of previous experiences.	This design utilizing all composite face sheets and construction is an advanced concept proposal and is not recommended for primary structural applications such as the center wing at this stage of development.	8
			GRAPHITE	1,262.7	341	430,380			

FIGURE 41

TABLE III

CONCEPT COMPARISONS

Graphite	Comp. Used To Weight Saved Ratio	Composite Material Usage	$/#	Δ Cost $/AC	% Weight Saved
I A	1.7 to 1.0	14%	$ 691	$252,586	7.4
II A	1.6 to 1.0	17%	284	128,357	9.2
II B	8.8 to 1.0	37%	1,654	329,963	4.0
II C	1.4 to 1.0	23%	228	154,120	13.7
II C(Ti)	0.6 to 1.0	15%	154	151,474	20.0
II D	1.2 to 1.0	15%	203	111,664	11.2
II E	2.3 to 1.0	18%	416	148,549	7.3
II F	1.7 to 1.0	19%	300	149,961	10.2
III A	1.2 to 1.0	72%	192	360,409	38.1
III B	1.8 to 1.0	62%	341	430,380	25.6

Boron	Comp. Used To Weight Saved Ratio	Composite Material Usage	$/#	Δ Cost $/AC	% Weight Saved
I A	0.9 to 1.0	9%	$587	$311,910	10.8
II A	0.9 to 1.0	13%	376	241,907	13.0
II B	2.1 to 1.0	32%	912	594,987	13.2
II C	0.7 to 1.0	18%	325	307,299	19.2
II C(Ti)	0.4 to 1.0	12%	223	255,732	23.3
II D	0.7 to 1.0	12%	270	198,362	14.9
II E	0.9 to 1.0	15%	393	272,400	14.1
II F	0.7 to 1.0	14%	293	238,365	16.5
III A	1.1 to 1.0	72%	487	927,047	38.7
III B	1.3 to 1.0	59%	590	901,466	31.0

TABLE IV

COST/PRODUCIBILITY EFFECTIVENESS

RANKINGS

Boron

$/lb.		Δ Cost ($/AC)	
223	II C (Ti)	$198,362	II D
270	II D	238,365	II F
293	II F	241,907	II A
325	II C	255,732	II C (Ti)
376	II A	272,400	II E
393	II E	307,299	II C
487	III A	311,910	I A
587	I A	594,987	II B
590	III B	901,466	III B
912	II B	927,047	III A

Graphite

$/lb.		Δ Cost ($/AC)	
154	II C (Ti)	$111,664	II D
192	III A	128,357	II A
203	II D	148,549	II E
228	II C	149,961	II F
284	II A	151,474	II C (Ti)
300	II F	154,120	II C
341	III B	252,586	I A
416	II E	329,963	II B
691	I A	360,409	III A
1654	II B	430,380	III B

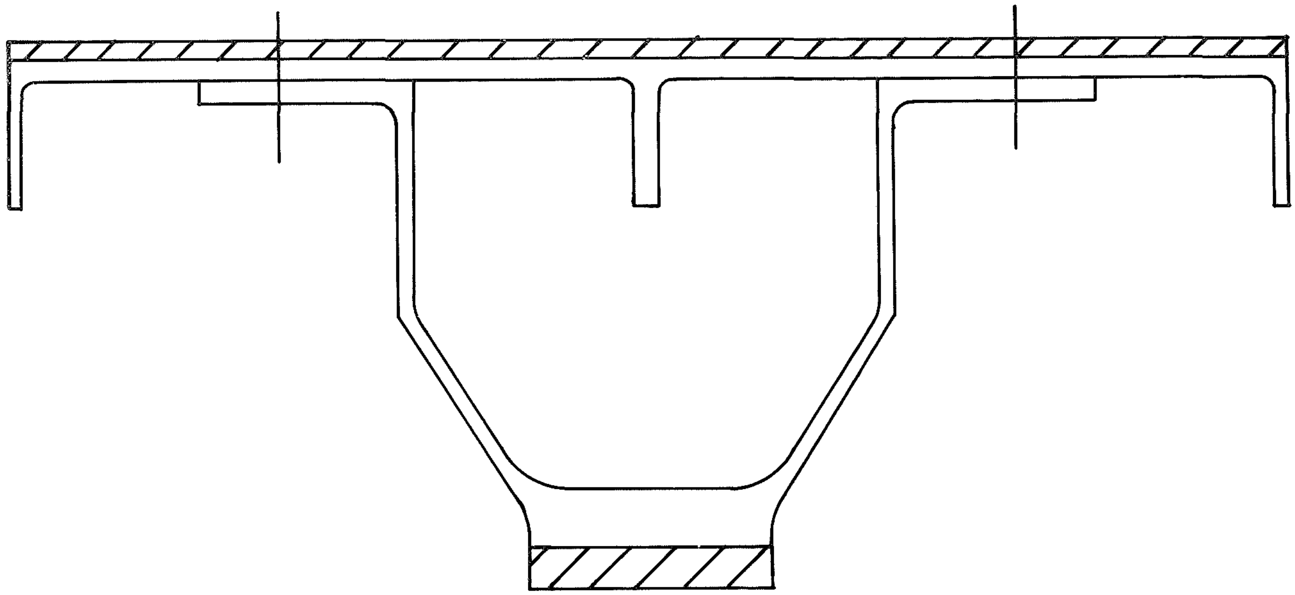
Reinforcement Concept

For this design, the feasibility was explored of reinforcing the center wing box without removal from the aircraft. This was the only possibility of completing the structural additions while remaining cost competitive with the other designs. That is, once the box was removed from the aircraft for reinforcement, it would be more advantageous to scrap the old box and build up a new center wing for replacement. The design, Concept IA, which evolved could be accomplished by removal of the upper skin and stiffeners from the box. The lower skins would remain attached to the ribs and spars in the aircraft. The composite material would be added to the skin and hat stiffeners as necessary for reinforcement.

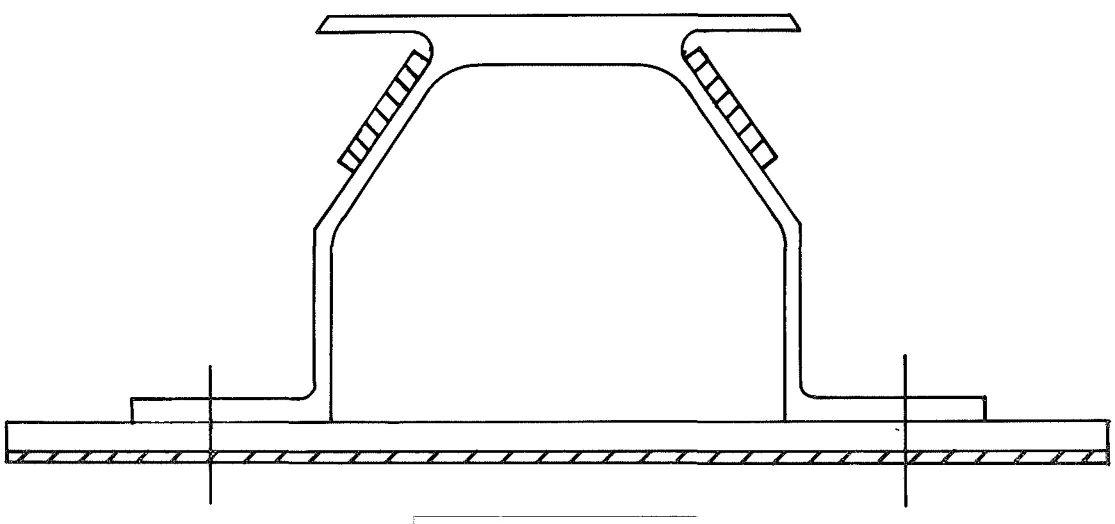
Concept I-A

This concept was exercised primarily to examine the feasibility of reinforcing a primary wing structure without removal of the total structure from the aircraft. The methods available for reinforcement of the wing box were limited since only portions of the box could be removed from the aircraft. Unidirectional composite material was added to the wing skin and hat stiffener to reduce the overall stress levels in the aluminum. The amount of composite material to be added was determined by the techniques outlined in Appendix B.

The wing skin and hat stiffener reinforcement is shown in Figure 42 for wing station 68. This is a typical section for this reinforcement concept. The upper skin panels (skins and hats) are removed from the box and reinforced as shown in Figure 42. The external reinforcement of the skins has certain disadvantages. Inspection of the aluminum for cracks would be impossible and lightening strikes might cause excessive damage to the externally applied composite material. The lower skin panels will remain on the aircraft and attached to the ribs and spars. The external reinforcement will run from wing station 68 outboard. From wing station 68 inboard, the hat reinforcement will have to be increased to compensate for the inability to add composite to the skins in this area in the fuselage.



UPPER SURFACE



LOWER SURFACE

FIGURE 42

TABLE V

CONCEPT I-A

WEIGHT SUMMARY

Boron - Aluminum

Item	Weight (Pounds)
Boron Weight	484.6
Total Weight	4396.4
Weight Saved	531.8
% Weight Saved	10.8

Graphite - Aluminum

Item	Weight (Pounds)
Graphite Weight	636.2
Total Weight	4563.1
Weight Saved	365.1
% Weight Saved	7.4

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$311,910	$587/#
Graphite	$252,586	$691/#

The weight and cost summary for this design is given in Table V. The weight savings were significant; however, the cost to weight saved ratio indicates that this concept would not be cost effective.

From a cost/producibility effectiveness view, Concept IA is basically different from the other concepts in that a greater level of effort is required to accomplish the composite material application. Among the elements that add complexity to the overall operation and thus add to the total labor are the removal and rework of wing panels, the cleaning of surfaces as a prerequisite for bonding, the composite material installation, and the subsequent inspection. For the removal of the upper wing panels, the existing fasteners must be drilled out. From this follows the possibility of damage to fastener holes which must be reworked for installation of new fasteners.

Surfaces on which composite material is to be bonded as a reinforcement must be cleaned thoroughly prior to the application. The only practical way to clean these surfaces is a vacuum blast operation which is preceded by a locally applied paint stripper and followed by a chemical etch. Stripped surfaces must be refinished and resealed where required in conjunction with the operation. The actual installation of the composite material reinforcement on the center wing structure would be difficult and time consuming due to the inaccessibility and confined nature of the work area. Extreme difficulty would be encountered in laying down the procured composite material laminates and applying the regulated heat and pressure required for bonding.

Inspection must be accomplished to determine the integrity of the material lay-up and bond line adhesion. Considering the construction of the center wing and the prospective location of the composite material reinforcement, inspection of the beef-ups would be difficult, time consuming, and of doubtful accuracy due to the inherent inaccessibility problem. Inspection would be by contact pulse-echo ultrasonics. This method will detect any disbonds between the composite and metal surfaces.

There are two primary objections to the configuration from an inspection aspect. The first is the assurance of a clean bonding surface on the hat walls and surface skins. Inspection of these surfaces would at best be questionable especially on the lower surface where cleaning would be hampered by dripping cleaner. The second objection is the inspection of the stiffener bond to the hat section wall. The inspection would be essentially impossible due to the limited access. Also, there would be no way to inspect for corrosion or cracks around the fastener holes since a composite sheet would cover the surface. The cover would prevent direct contact of the ultrasonic probe with the metal surface.

Compound Composite Concepts

The "Compound-Composite Concepts" evolved under the design assumption that significant structural changes may be made in the box but the design will result in a structure which has a composite material used in conjunction with or as reinforcement for metallic materials. This concept assumes the existing C-130E wing box is discarded after removal from the aircraft and a new reinforced box is built up.

Six designs were developed for composite reinforced aluminum wing boxes. One titanium reinforced design was exercised where a wing skin titanium extrusion could be efficiently produced. The initial concept evaluated, Concept II-A, makes use of the existing skin and stiffener extrusions which are machined to thinner cross section prior to reinforcement. This concept represents a compromise in that the weight savings and cost are not as good as concept II-C, but the design and manufacturing risks are minimized. Concept II-B proved to be an inefficient design because $\pm 45^{\circ}$ composite material was utilized in the skins. The aluminum skins were reduced in thickness to the point that the wing box torsional stiffness, GJ, was below that required, and then the $\pm 45^{\circ}$ composite orientation was added to regain the GJ. The unidirectional composite was added in the skins and stiffener. Concept II-C showed the best weight savings and one of the best cost savings, and from an overall standpoint proved to be an efficient design. Both an aluminum and a titanium reinforced box was designed for Concept II-C. Concept II-D had the least cost and least dollars per pound of weight saved for all the compound composite designs; however, the weight savings were not impressive. Concept II-E is a built-up version of Concept II-C. The weight and costs are higher, however. Concept II-F evolved because the configuration made attachment of the skins and stringers to the box end fitting (rainbow fitting) easier. In the following sections, each compound composite design concept will be discussed.

Concept II-A

This concept evolved into a design which made use of the existing wing skin and hat section aluminum extrusions. The wing skins were machined to the thicknesses of the unmodified C-130E design (Reference Figures 13-15). Thus, the wing torsional stiffness, GJ, was unchanged from the C-130E design which had adequate torsional stiffness. The crowns of the hat stiffeners were machined to the same thickness as the webs ($t = 0.08$) on the C-130E design. The small riser on the upper skin under the hats was removed, and the addition of composite to this area precludes local failure. The overall aluminum stress levels in the box were reduced by the addition of unidirectional composite material to the crown of the hat stiffeners and to the inside of the wing skins under the hats (Reference Figure 43). The reinforcing composite for the wing skins was placed internally under the hats because this would allow inspection of the external aluminum surface for damage and it would afford better protection for the composite in case of lightening strikes. The composite material cannot be inspected; however, since the structure is capable of sustaining a high percentage of limit design load without the composite material, this approach is considered to be a good compromise.

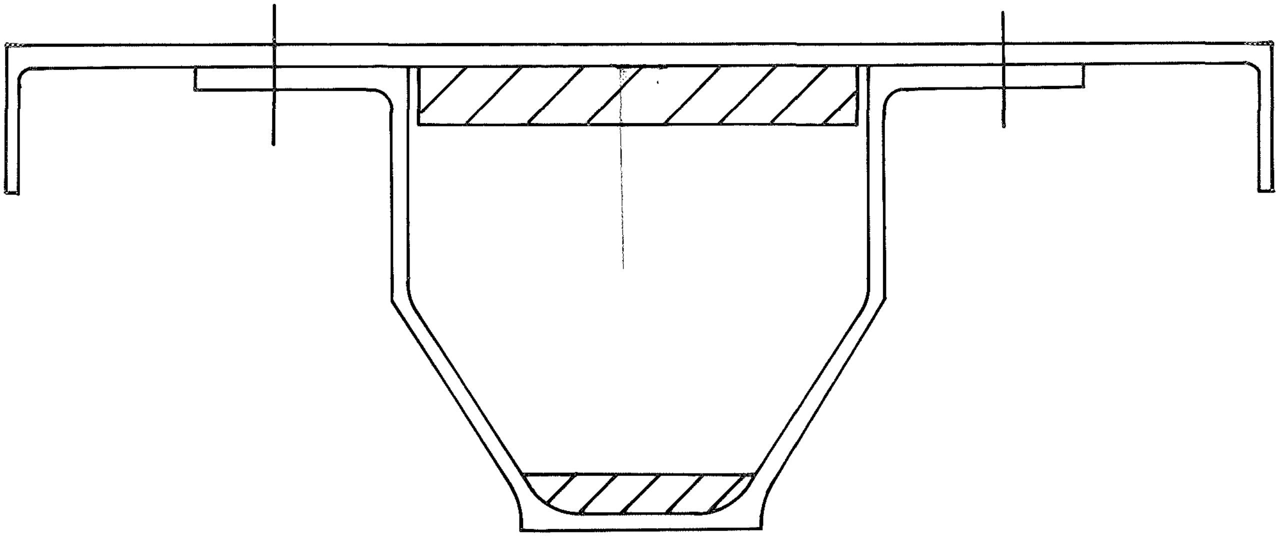
From an overall program development view, Concept II-A appears to be the most cost effective. Cost effectiveness here includes considerations for expense items of development, tooling, etc., and considers the time span factor for program conclusion. On a straight cost/producibility effectiveness basis, Concept II-A rates in the upper half of the concepts group listing for the "cost increase per pound of weight saved" and for the "total cost increase."

The hat sections are machined from basically the same extrusions as currently used in the production center wing box. The hat stiffener reinforcement strips will be cured at 85 psig and 350°F in a tool made from the hat crown section. Reinforcing composite material is subsequently bonded to the hat sections. Precured composite strips would be bonded to the skin panel under the hats at 45 psig and 190°F. The hat sections are then assembled to the aluminum skin with mechanical fasteners. Warpage due to differences in thermal expansion during the bond cycles is not fully predictable. Methods of compensating for the warpage and straightening must be evaluated.

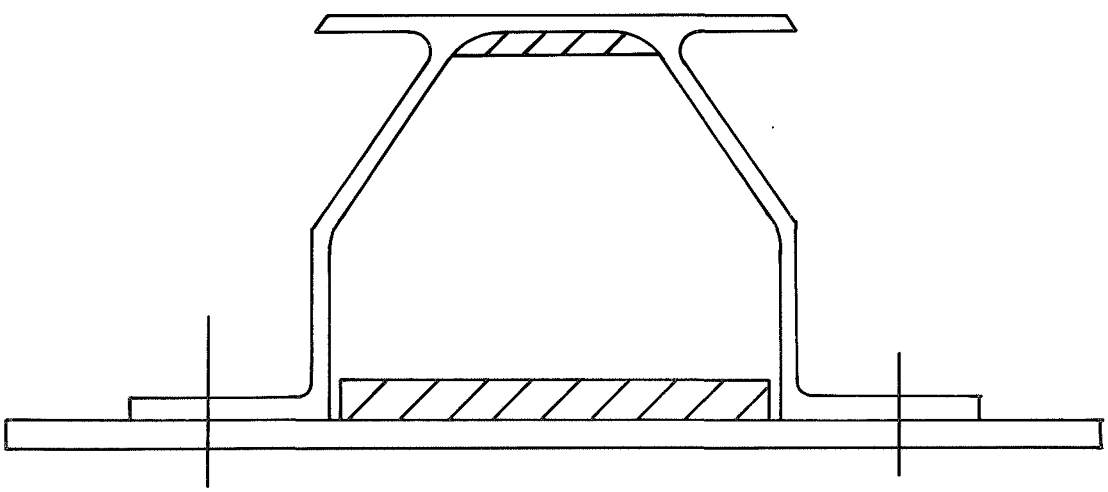
Inspection of this configuration would be by contact pulse-echo ultrasonics. Using this method, the inspection would have to be made before the hat was mechanically fastened to the skin since access to the composite surface is desirable. There are technical problems which would be encountered if the inspection were to be made through the aluminum skin. The hat may require removal if an inspection of the bond were made after the installation on the aircraft.

There is an advanced inspection method utilizing phase/amplitude sensitive ultrasonics which could be used. This method would permit inspection of the part at any time during fabrication or after fabrication.

A summary of the weight and cost data for this concept is given in Table VI. The weight savings were 9 to 13 percent, for the graphite and boron reinforced designs respectively. These weight savings are significant and would have been larger had the reinforcement been added to improve the static strength or stiffness. The cost per pound of weight saved is still larger than that required for a cost effective application.



UPPER SURFACE



LOWER SURFACE

FIGURE 43

TABLE VI

CONCEPT II-A

WEIGHT SUMMARY

Boron - Aluminum Item	Weight (Pounds)	Graphite - Aluminum Item	Weight (Pounds)
Boron Weight	577.5	Graphite Weight	752.0
Total Weight	4284.5	Total Weight	4476.4
Weight Saved	642.7	Weight Saved	451.4
% Weight Saved	13.0	% Weight Saved	9.2

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$241,907	$376/#
Graphite	$128,357	$284/#

Concept II-B

Concept II-B was the first concept exercised in a series of integrally stiffened wing panel designs (Reference Figure 44). With the thought that the most optimum structure would have a large percentage of composite material, the wing skins were sized to a thickness which would give a box GJ well below that which is required. The torsional stiffness of the box was brought back to the required level by the addition of $\pm 45^{\circ}$ plies in the skins. The $\pm 45^{\circ}$ laminate orientation has the highest shear modulus possible for any laminate orientation, and it was throught that the addition of the $\pm 45^{\circ}$ plies would prove effective. However, as shown in the weight and cost summary in Table VII, the use of the $\pm 45^{\circ}$ plies in the wing skins produced designs which were not efficient. If the design had not been fatigue critical, the results would have shown more efficient weight savings. The percentage of 0° or spanwise orientated plies was based on the requirement of keeping the aluminum stress levels below 40,000 psi in order to provide adequate fatigue life. The 0° plies were balanced between the riser and the skins in order to keep the center of gravity of the skin-riser section in the same position. This would preclude local bending taking place at the center wing-outer wing joint fitting (rainbow fitting). In Figure 45 the detail of the rainbow fitting is shown.

New extrusions would be required for this concept. They would require a considerable amount of machining especially for the rises and falls required in rainbow fitting area. The rib cap extrusions would have to be machined differently (Figure 46) to accommodate the "T" risers. This would require a new sealing technique in the wet bay areas.

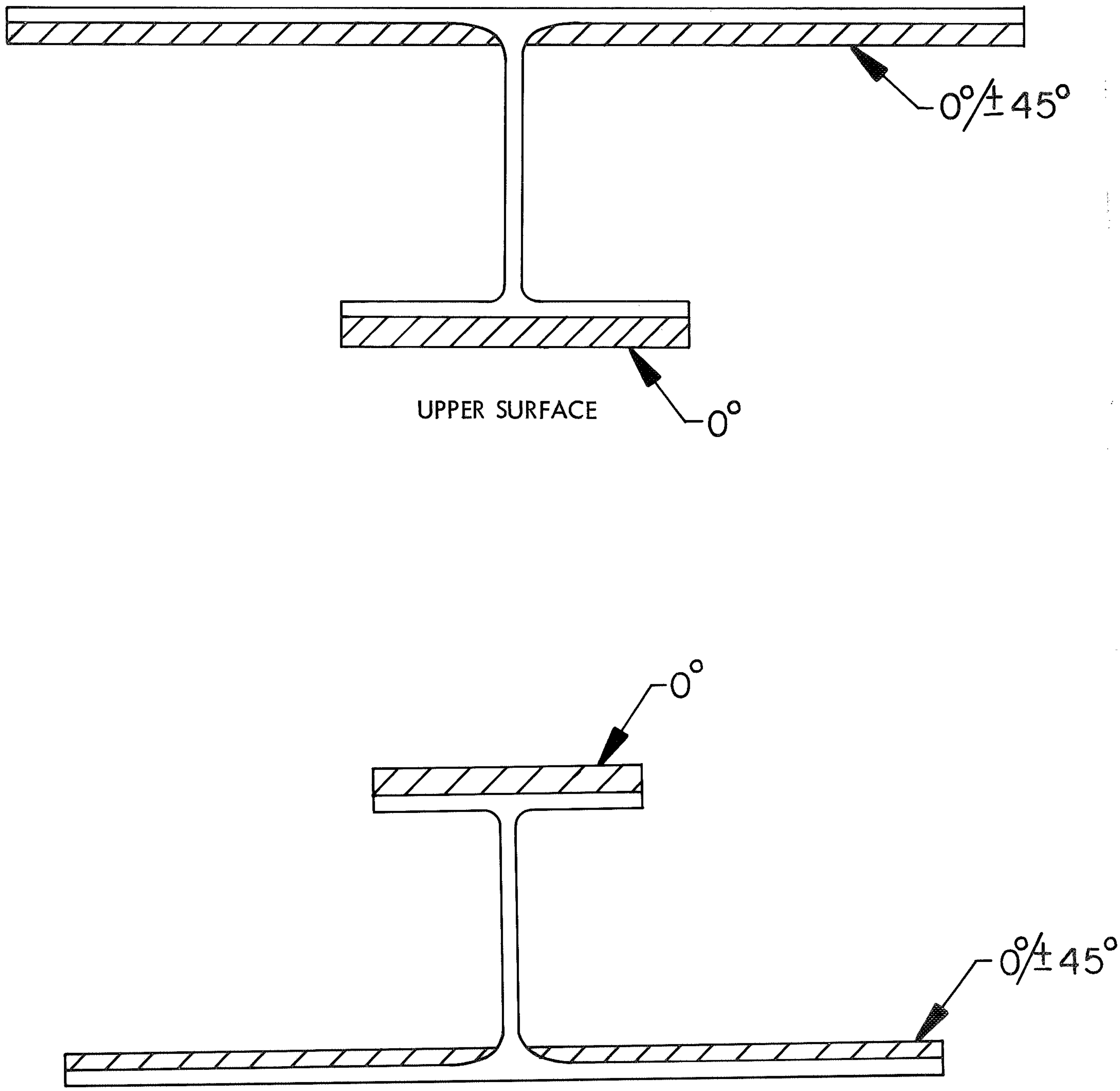


FIGURE 44

TABLE VII

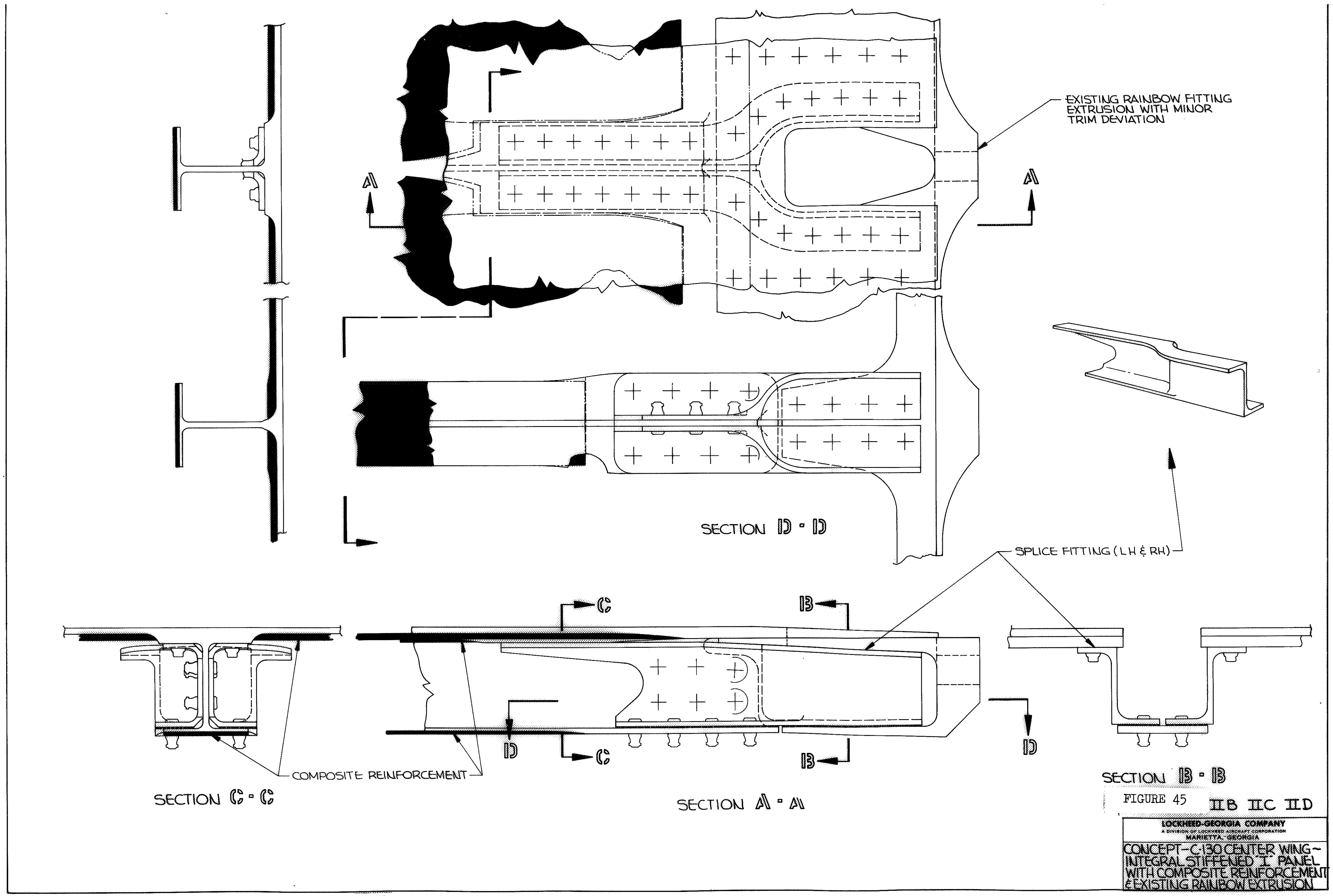
CONCEPT II-B

WEIGHT SUMMARY

Boron - Aluminum		Graphite - Aluminum	
Item	Weight (Pounds)	Item	Weight (Pounds)
Boron Weight	1351.3	Graphite Weight	1762.8
Total Weight	4276.1	Total Weight	4728.7
Weight Saved	652.1	Weight Saved	199.5
% Weight Saved	13.2	% Weight Saved	4.0

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$594,987	$ 912/#
Graphite	$329,963	$1,654/#


EXISTING RAINBOW FITTING EXTRUSION WITH MINOR TRIM DEVIATION
A
A
SECTION D - D
SPLICE FITTING (L H & RH)
C
C
B
B
D
D
COMPOSITE REINFORCEMENT
SECTION C - C
SECTION A - A
SECTION B - B
FIGURE 45
IIB IIC IID
LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA
CONCEPT-C-130 CENTER WING-INTEGRAL STIFFENED 'I' PANEL WITH COMPOSITE REINFORCEMENT & EXISTING RAINBOW EXTRUSION

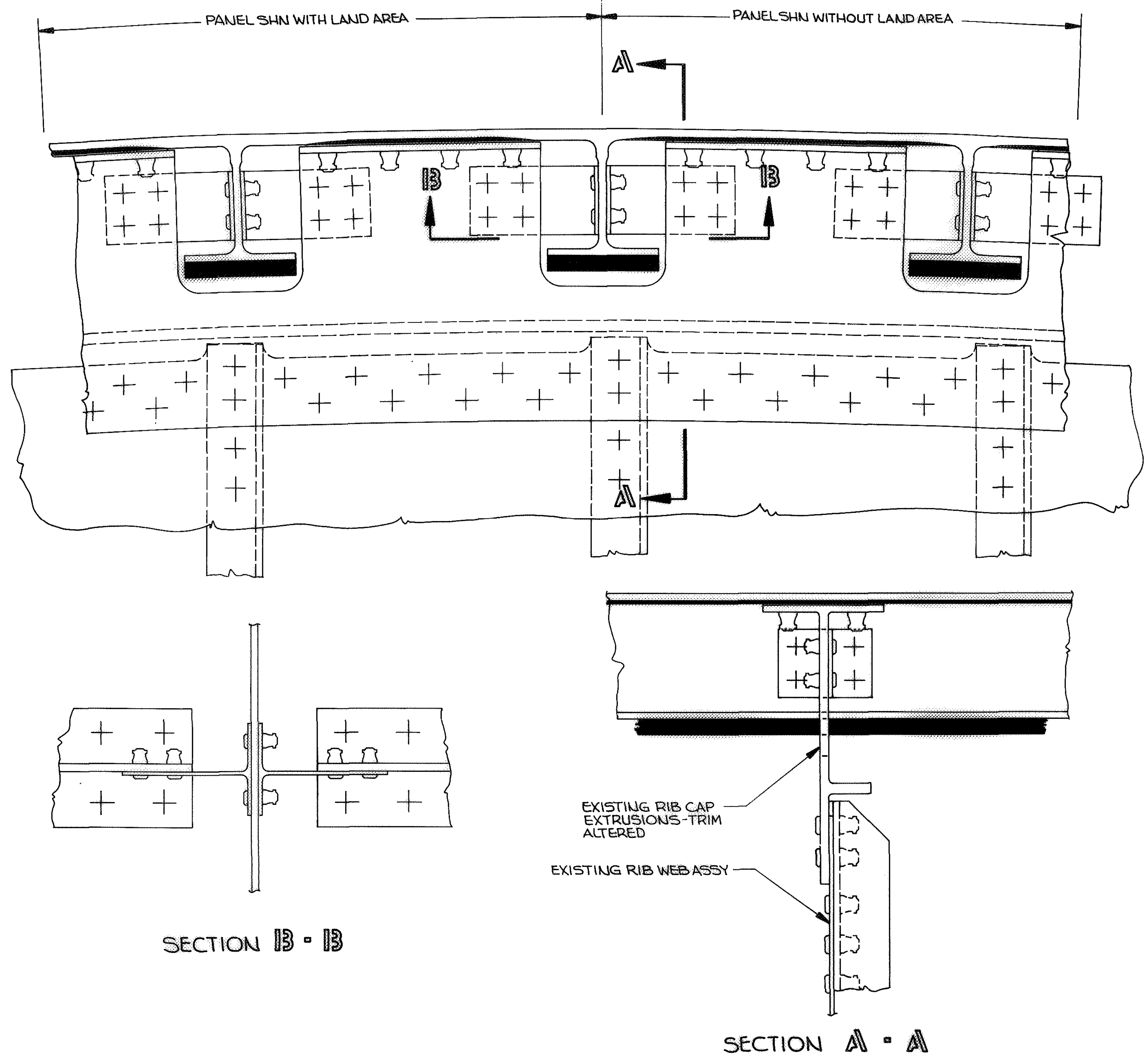


FIGURE 46

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

RIB-PANEL RISER INTERFACE/CONCEPTS II B,C,D & E

The riser reinforcement and skin panels would be cured at 85 psig and 350°F for one hour. These would be bonded to the stiffened aluminum panels at 45 psig and 190°F for four hours. The questionable area for bonding will be in the radii where the tee risers emerge from the skins. Dimensional tolerance on extrusion and expansion or contraction of parts during bond cycle could create voided areas in these radii.

The rib caps would be installed over the composites which would require drilling holes for fasteners through the aluminum and composite. Countersinking would be done in the aluminum.

The simplicity of this design would allow a straight forward utilization of contact pulse-echo ultrasonics for inspection. Any disbonds between the composite and the metal surface could be easily detected. Inspection must be performed prior to installation of the part on the aircraft.

This concept is rated in the bottom half of the group listing for the "cost increase per pound of weight saved" and for the "total cost increase". Basically, this is attributed to the fact that this concept provides a low percentage of weight saved and a relatively large amount of composite material is required to save a pound of weight.

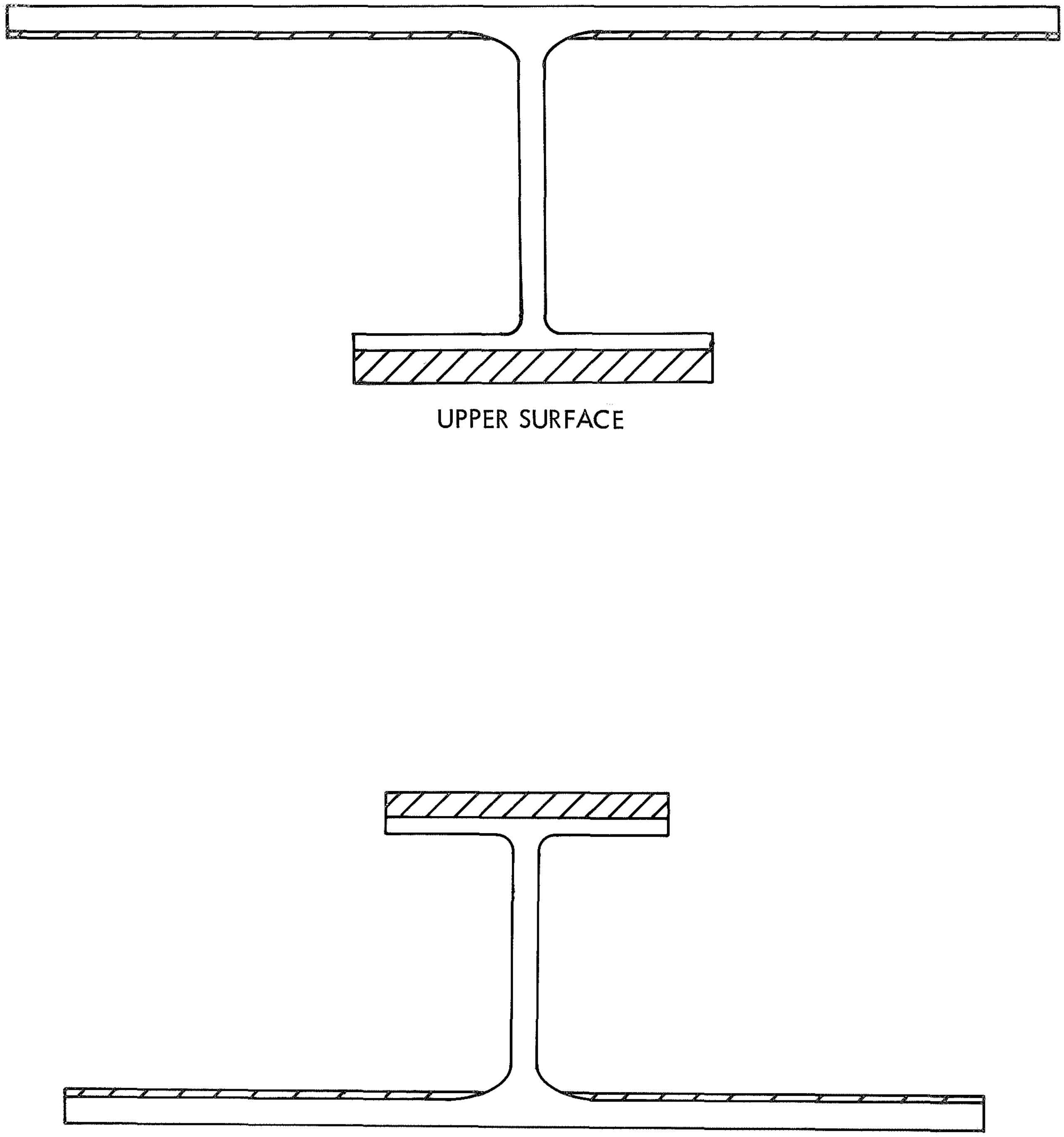
The basic structure for this concept is machined from wide rib extrusions or plate. Precured composites are bonded to the aluminum panels. Potential warpage from the bonding process and the drilling of holes through the composite and aluminum are problems that must be investigated and resolved. Cost requirements for this concept could increase depending on the complexity of these problems.

Concept II-C

Both an aluminum and titanium reinforced wing box was exercised for this concept. This concept is similar to Concept II-B except there are no $\pm 45^{\circ}$ plies in the skins (Ref. Figure 47). The aluminum skins are only reduced in thickness to equal the skin thickness on the C-130E wing box in order to maintain the box torsional stiffness, GJ. For the titanium box, the skins were reduced in thickness to the point where the box GJ was equivalent to the existing C-130E torsional stiffness. The 0° or spanwise plies are added to reduce the stress levels in the metallic material to a level which will ensure an equal fatigue life to that of the aluminum modified box. For the aluminum reinforced box the ultimate design stress levels were kept below 40,000 psi. For the titanium box, the stress levels were kept below 52,000 psi which corresponds to approximately the same percentage of the endurance limit as the 40,000 psi is for aluminum. The 0° plies were balanced between the riser and the skins in order to keep the center of gravity of the skin-riser section in the same position. This would preclude local bending taking place at the rainbow fitting. The detail for the rainbow fitting is shown in Figure 45 of the preceding section. Concept II-C utilizing the aluminum structure is rated in the middle of the group listing for the "cost increase per pound of weight saved" and for the "total cost increase". As the rating indicates, this concept provides for an efficient use of the composite material as a reinforcement (Ref. Table VIII).

Concept II-C (Ti) utilizing a basic titanium structure is rated in the upper half of the group listing for the "cost increase per pound of weight saved" and for the "total cost increase". The use of titanium in this concept provides for a greater percentage of weight savings at a correspondingly smaller amount of composite material usage. The lower percentage of composite utilization accounts for the lower costs for the titanium box. This concept affords all of the advantages of aluminum and greatly reduces the problems involving warpage due to the more compatible thermal coefficient of expansion of titanium to composites. This concept had the greatest percentage weight savings of any of the reinforcement concepts (Ref. Table IX).

The manufacturing methods and quality assurance inspection methods are identical to those described for Concept II-B.



UPPER SURFACE

LOWER SURFACE

FIGURE 47

TABLE VIII

CONCEPT II-C
(ALUMINUM)

WEIGHT SUMMARY

Boron - Aluminum		Graphite - Aluminum	
Item	Weight (Pounds)	Item	Weight (Pounds)
Boron Weight	708.0	Graphite Weight	952.4
Total Weight	3982.4	Total Weight	4151.1
Weight Saved	945.8	Weight Saved	677.1
% Weight Saved	19.2	% Weight Saved	13.7

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$307,299	$325/#
Graphite	$154,120	$228/#

TABLE IX

CONCEPT II-C
(TITANIUM)

WEIGHT SUMMARY

Boron - Titanium

Item	Weight (Pounds)
Boron Weight	468.9
Total Weight	3782.0
Weight Saved	1146.2
% Weight Saved	23.3

Graphite - Titanium

Item	Weight (Pounds)
Graphite Weight	596.0
Total Weight	3941.6
Weight Saved	986.6
% Weight Saved	20.0

COST SUMMARY

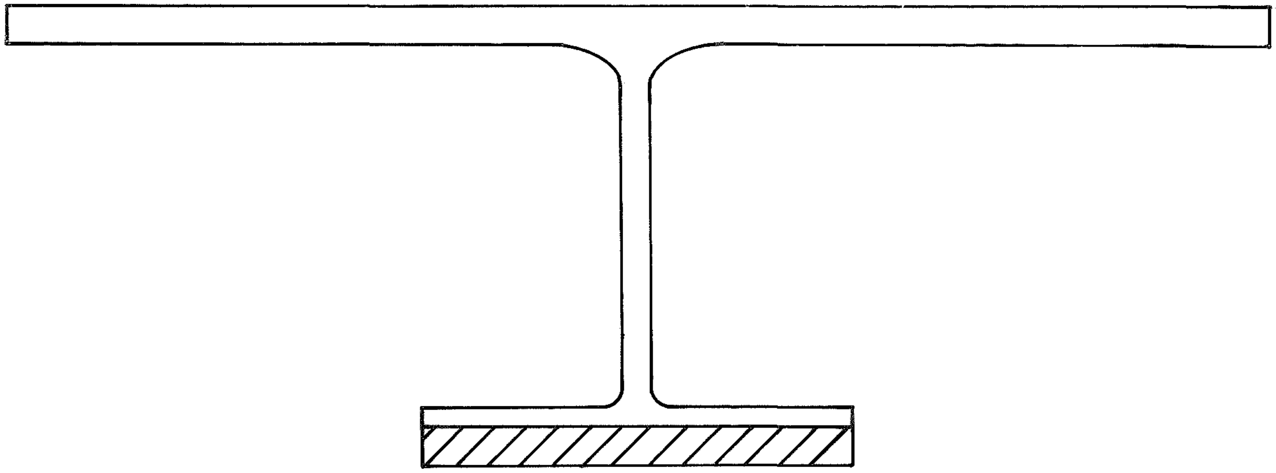
Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$225,732	$223/#
Graphite	$151,474	$154/#

Concept II-D

This concept essentially is a more conservative approach to the integrally stiffened concepts II-B and II-C. The wing box skin thicknesses were maintained to the gages used in the aluminum modified box (Ref. Figure 48). This gives a larger GJ than the unmodified C-130E wing box. The unidirectional composite is utilized in the riser only.

Concept II-D is rated well toward the top in group listing for the "cost increase per pound of weight saved" and for the "total cost increase". This concept can be considered to be near optimum in cost/producibility effectiveness for a composite/aluminum structure. It leads the list in ratings for total cost increase for both boron and graphite composite materials. It is near the top in ratings for cost increase per pound of weight saved ratio. The percent of weight saved is nominal, but the percent of composite material used in the new design is also nominal, resulting in a favorable cost to weight ratio. The details of the weight savings and costs are shown in Table X.

The manufacturing and quality assurance techniques and problems are essentially the same as those for concepts II-B and II-C.



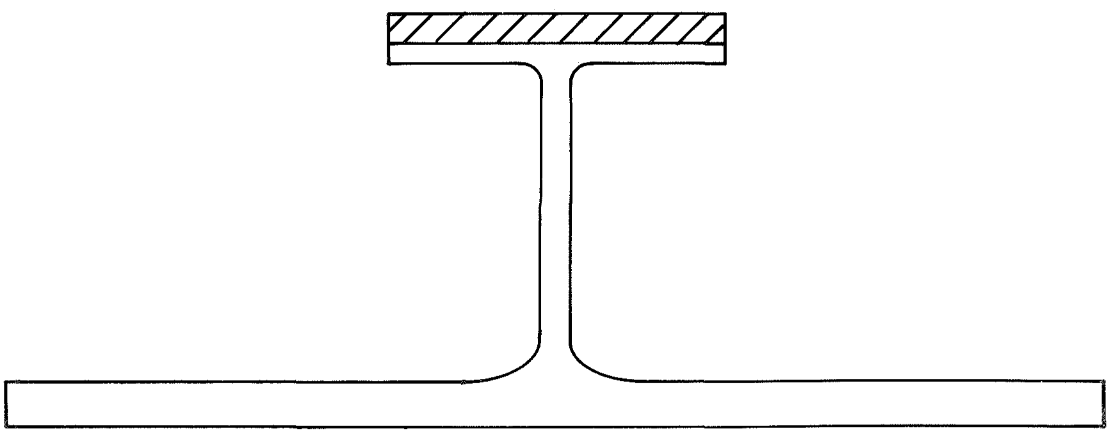


FIGURE 48

TABLE X

CONCEPT II-D

WEIGHT SUMMARY

Boron - Aluminum Item	Weight (Pounds)	Graphite - Aluminum Item	Weight (Pounds)
Boron Weight	494.5	Graphite Weight	664.2
Total Weight	4192.1	Total Weight	4378.7
Weight Saved	736.1	Weight Saved	549.5
% Weight Saved	14.9	% Weight Saved	11.2

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$198,362	$270/#
Graphite	$111,664	$203/#

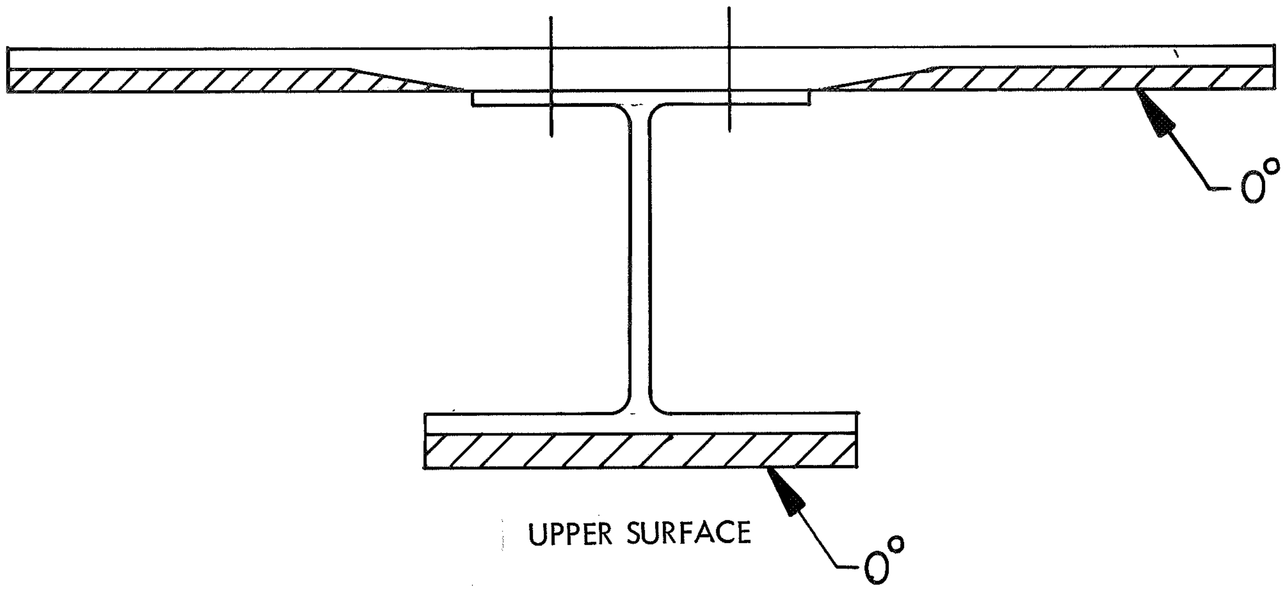
Concept II-E

This concept is essentially a "built-up" version of Concept II-C (Ref. Figure 49). The stiffeners and the skins can be reinforced individually and assembled in a conventional manner with a very small amount of fasteners through the composite reinforcement.

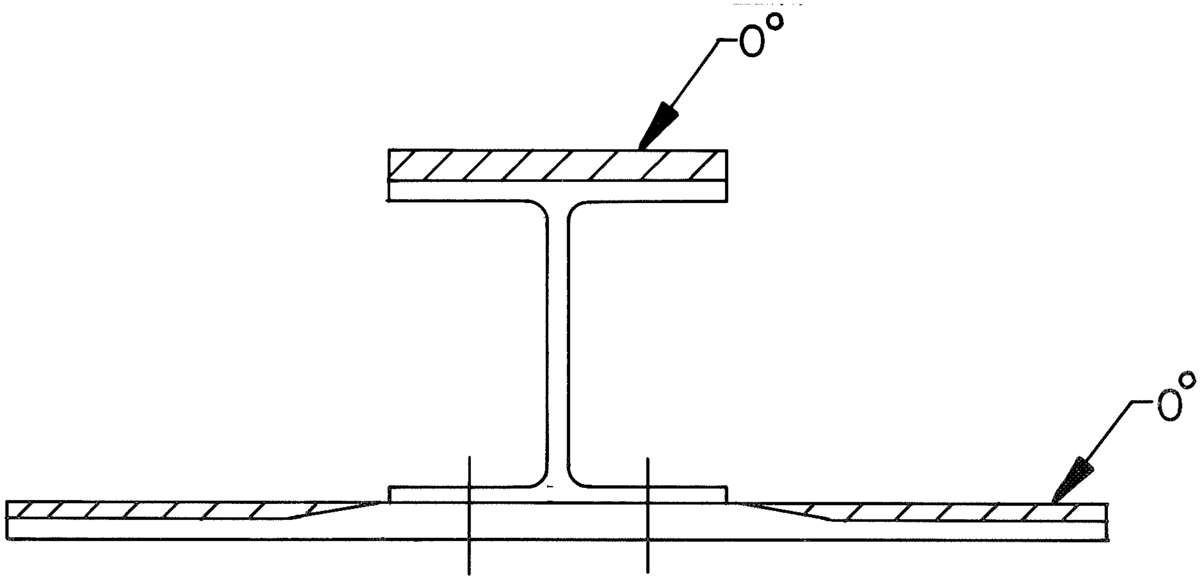
The machining of the flat plate for the skin possesses no problems. The machining tolerance required to assure continuity between composite reinforcement and the skin in the tapered areas would be difficult. The machining of the I-beam to a taper on the ends to match the rainbow fittings would require a special set-up and programming. Installation of fasteners must be by hand since the Drivmatic head will not fit in the open area between the flanges of the I-beam. The remaining manufacturing details will be similar to Concept II-B.

Concept II-E is rated in the middle of the group listing for the "cost increase per pound of weight saved" and for the "total cost increase". As the rating indicates, this concept provides only a partially efficient use of the composite material. A breakdown on the cost and weight of this concept is given in Table XI.

The simplicity of this design would allow a straight forward utilization of contact pulse-echo ultrasonics for inspection. Any disbonds between the composite and the metal surface could be easily detected. Inspection must be performed prior to installation of the part on the aircraft.



UPPER SURFACE



LOWER SURFACE

FIGURE 49

TABLE XI

CONCEPT II-E

WEIGHT SUMMARY

Boron - Aluminum		Graphite - Aluminum	
Item	Weight (Pounds)	Item	Weight (Pounds)
Boron Weight	649.1	Graphite Weight	844.4
Total Weight	4234.2	Total Weight	4566.4
Weight Saved	694.0	Weight Saved	361.8
% Weight Saved	14.1	% Weight Saved	7.3

COST SUMMARY

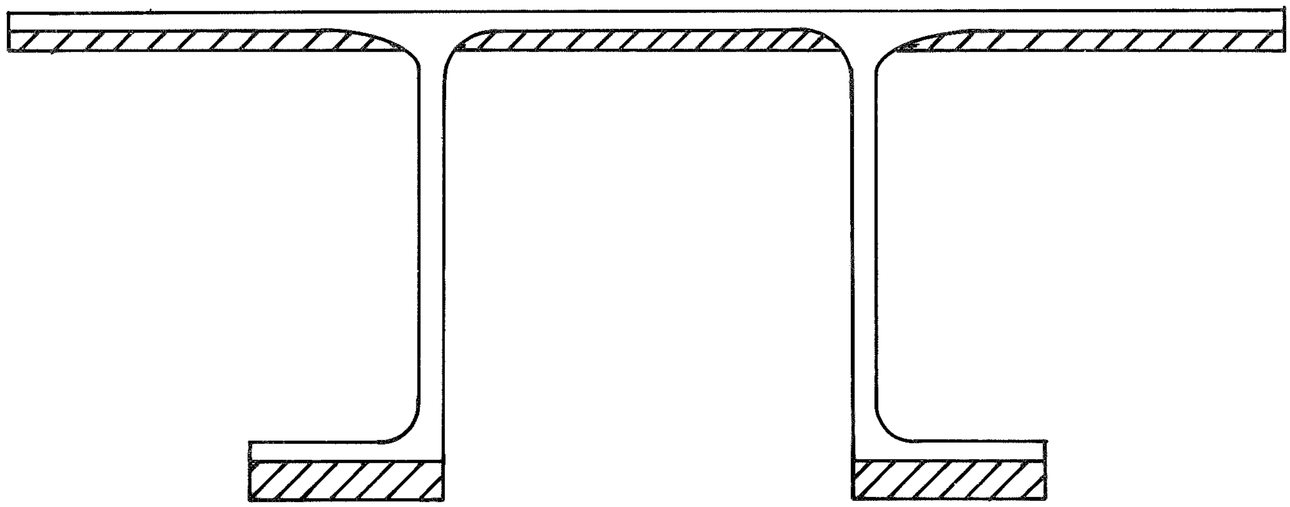
Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$272,400	$393/#
Graphite	$148,549	$416/#

Concept II-F

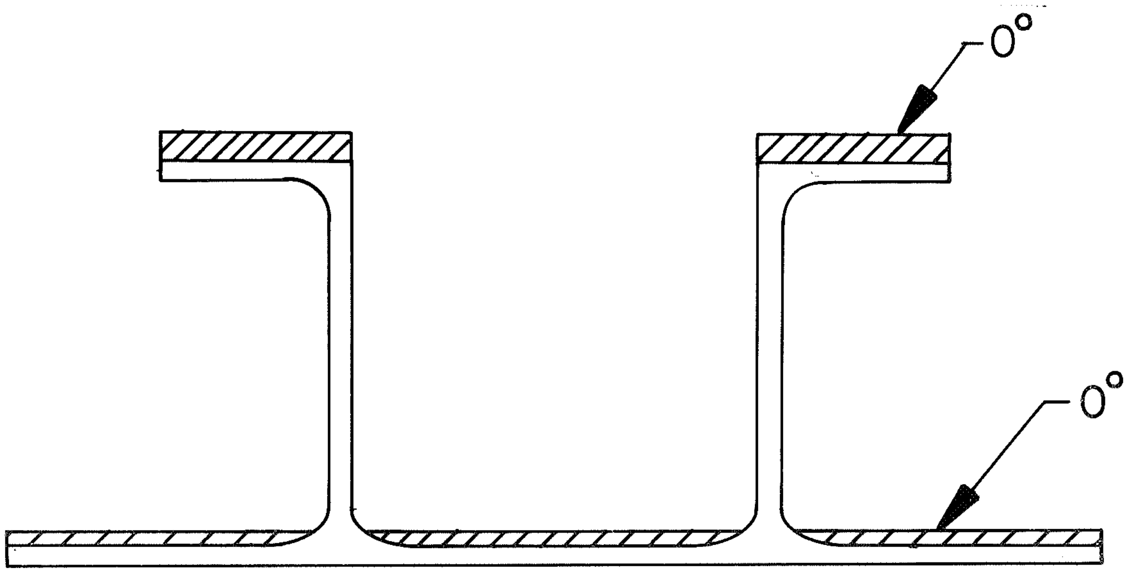
This concept evolved because it presented a skin-stringer design which could be easily attached to the rainbow fitting (Ref. Figure 50). The skin-stringer structure is made of integrally stiffened panels with a double vertical stiffener at each location where a hat section is in the current center wing box (Ref. Figures 51 and 52). The additional complexities in this aluminum structure design could present more extensive bonding and warpage problems. It appears that the possibility of incurring additional costs as a result of more complex processing operations exists for this concept to a greater extent than for other compound composite concepts.

The design concept II-F rates near the top of the group listing for boron reinforcement and in the middle for graphite reinforcement for the "cost increase per pound of weight saved" and the "total cost increase". Due to the structural characteristics of the composite materials in combination with the particular cross section of the aluminum structure, boron offers the more efficient use of composite material on a comparative group listing basis. The cost and weight summary is given in Table XII.

This concept would allow a straight forward utilization of contact pulse-echo ultrasonics just as in II-B, II-C, II-D, and II-E. The design configuration limits access to certain areas of the part so special care should be taken to remove all of the acoustic couplant used during the ultrasonic inspection to prevent contamination. In most cases the couplant would be water and it would not be desirable to have it sealed inside the finished article.



UPPER SURFACE



LOWER SURFACE

FIGURE 50

TABLE XII

CONCEPT II-F

WEIGHT SUMMARY

Boron - Aluminum Item	Weight (Pounds)	Graphite - Aluminum Item	Weight (Pounds)
Boron Weight	576.2	Graphite Weight	861.4
Total Weight	4114.2	Total Weight	4427.9
Weight Saved	814.0	Weight Saved	500.3
% Weight Saved	16.5	% Weight Saved	10.2

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$238,365	$293/#
Graphite	$149,961	$300/#

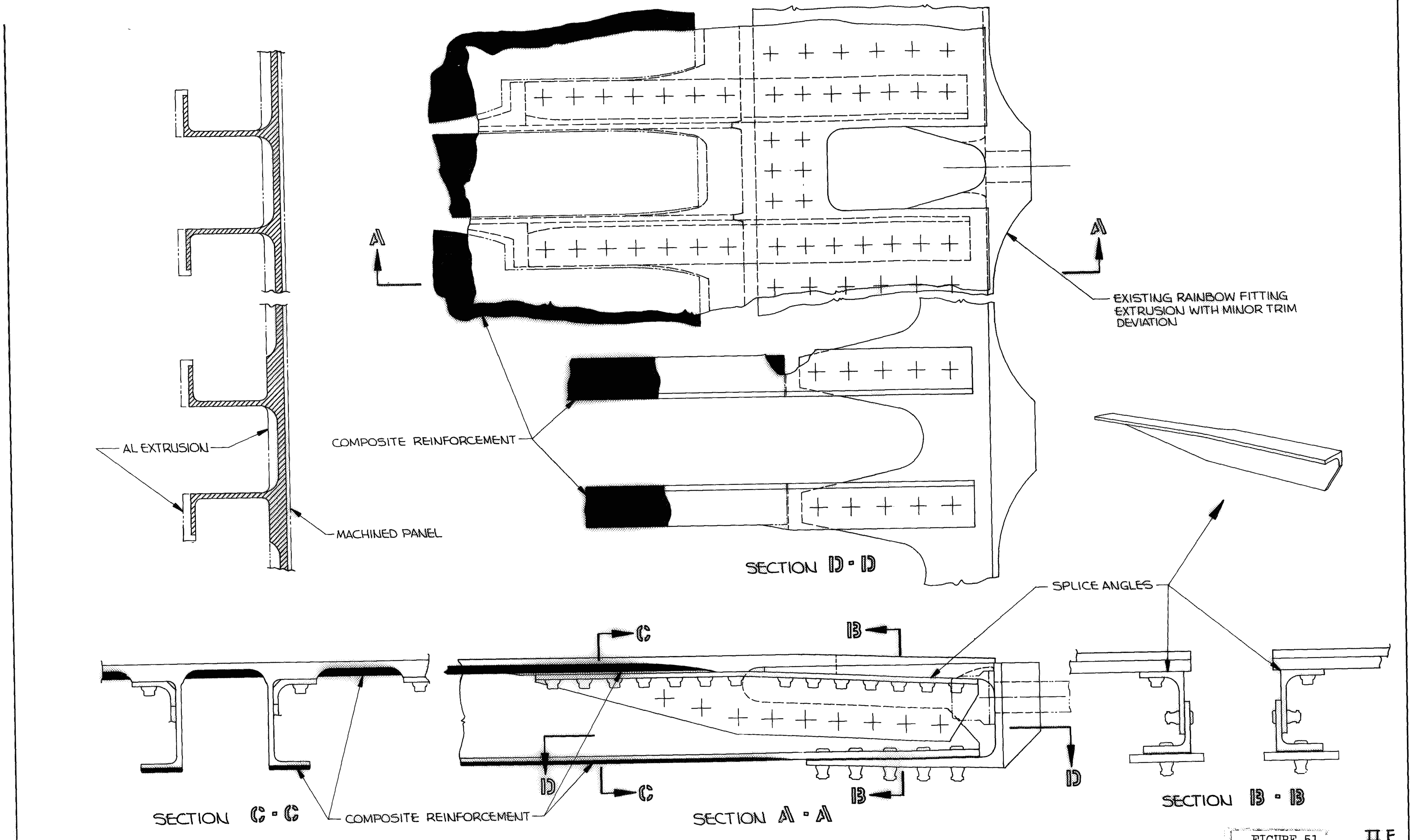


FIGURE 51 II F

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

CONCEPT-C-130 CENTER WING-INTEGRAL STIFFENED π PANEL WITH COMPOSITE REFORCEMENT & EXISTING RAINBOW EXTRUSION

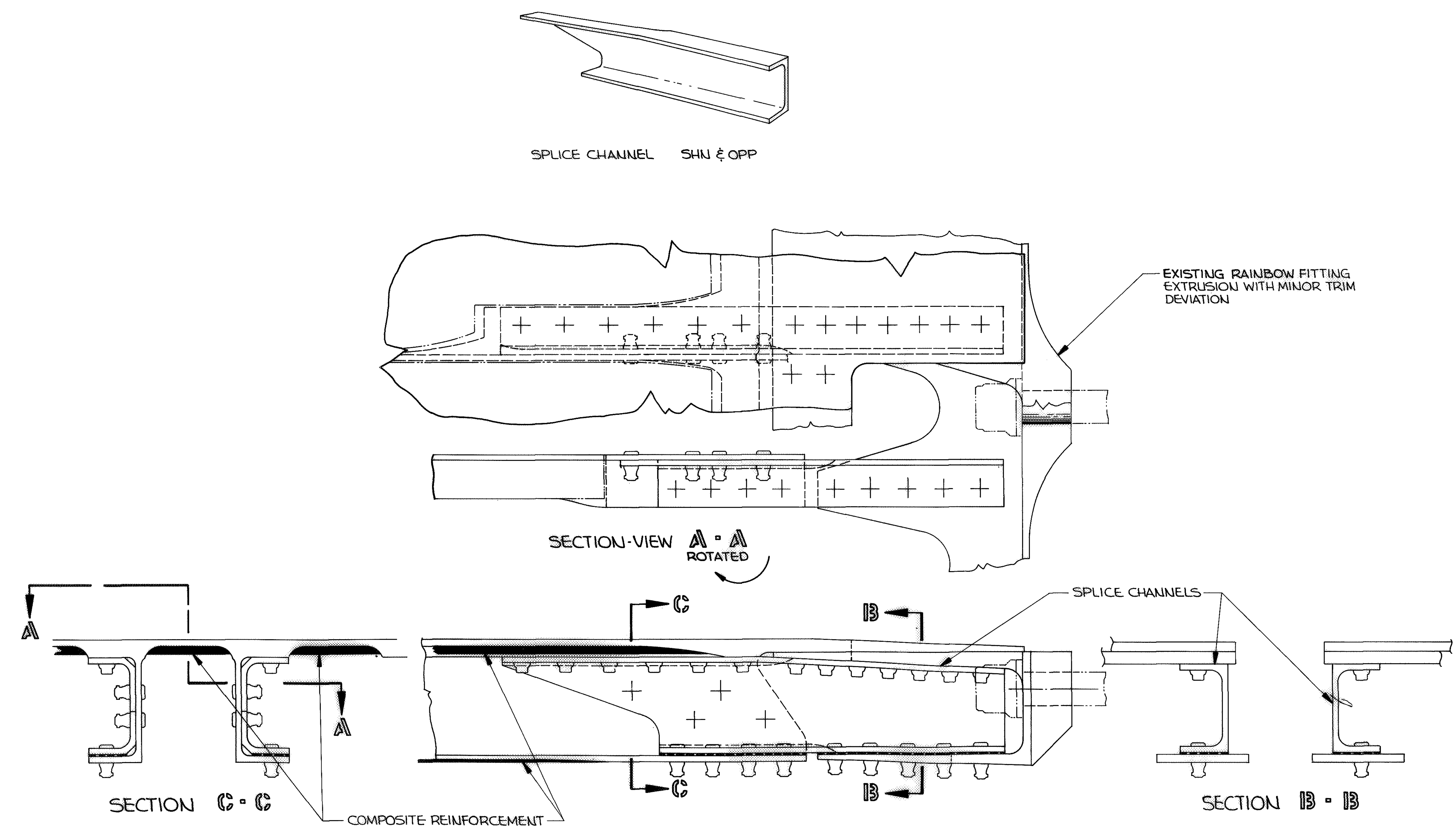


FIGURE 52 . II F ALT

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

ALTERNATE PANEL-
RAINBOW ATTACHMENT

All Composite Concepts

The all composite concepts were evaluated primarily to provide a baseline for comparison of the reinforcement concepts. These designs were exercised in sufficient detail to give a weight, cost, manufacturing and quality assurance summary. The designs were made to maintain the C-130E wing bending stiffness, EI, and torsional stiffness, GJ. Generally speaking, the overall designs were not strength critical.

Concept III-A

Concept III-A is an "all composite" design of beaded bonded structure (Figure 53, 54 and 55). This concept is distinctively different from all the preceding concepts in that the factors of weight saved, the weight of composite material used, the cost increase per pound saved, and the total cost increase are all on a higher level. However, the ratio of composite material used to weight saved is not distinctively different. Generally, the cost/producibility effectiveness of this concept is not as good as that of the other concepts. It should be noted, however, that in cases where large weight savings is mandatory, the cost/producibility effectiveness would possibly appear in a better light. The cost and weight summary is given in Table XIII.

Concept III-A offers the greatest percent weight savings of all concepts for both boron and graphite composite material usage. The ratios of composite material usage to weight saved and the percent of composite material in the new design are very close for both materials. However, since boron material cost is forecasted at twelve times the cost of graphite material, the cost effectiveness for graphite is better in this concept. The greater physical properties of boron do not offset the unit material cost.

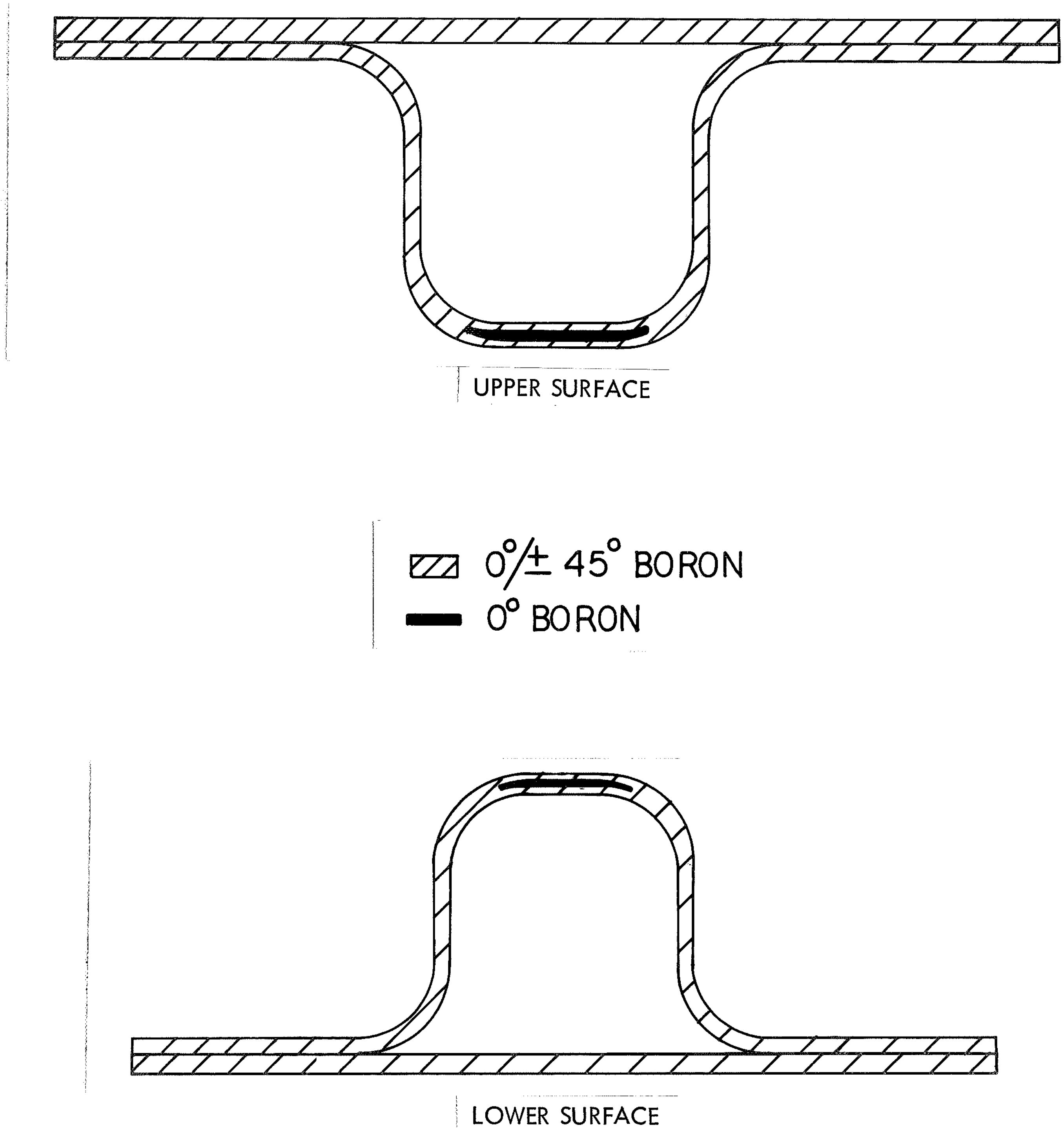


FIGURE 53

TABLE XIII

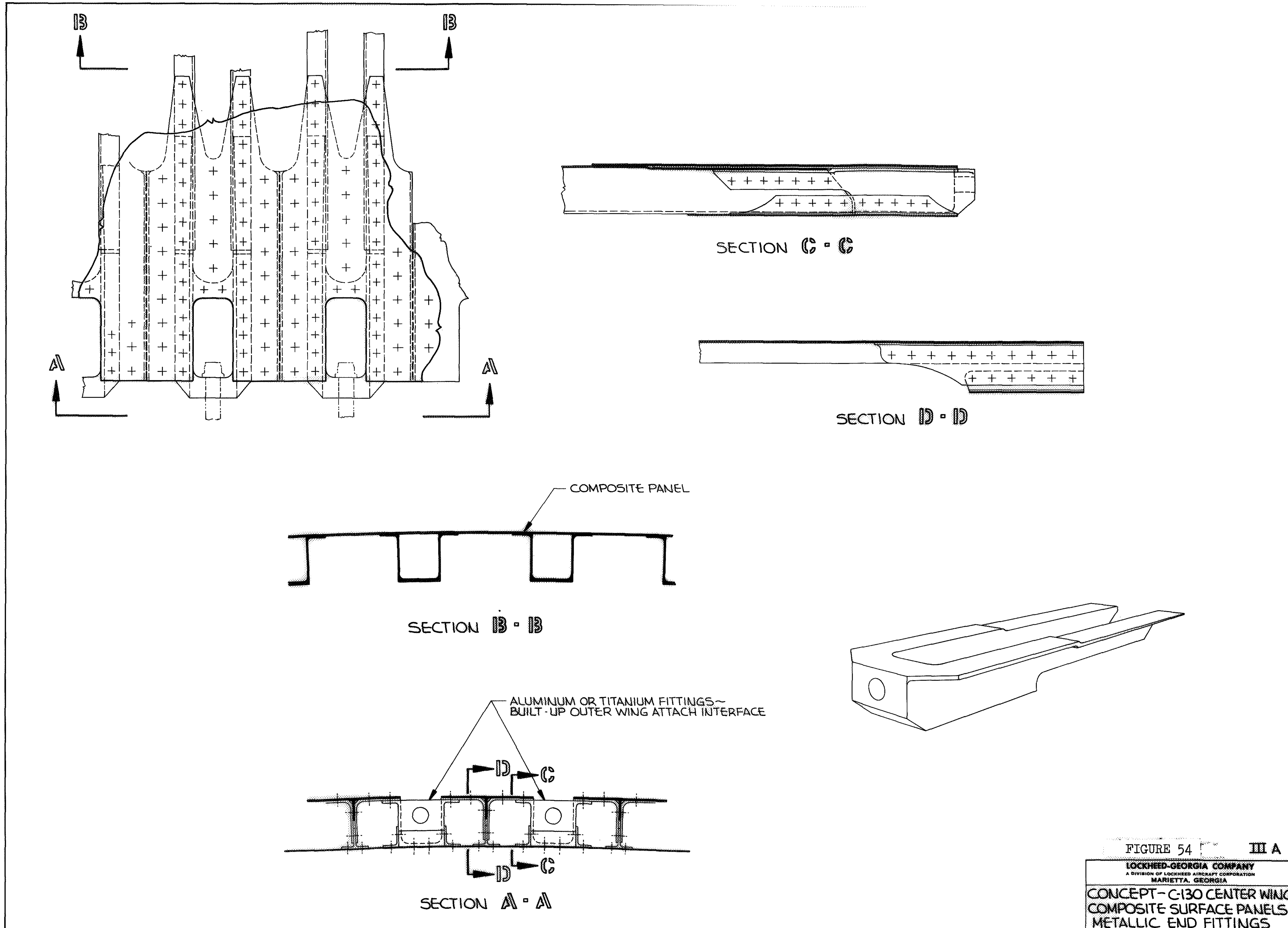
CONCEPT III-A

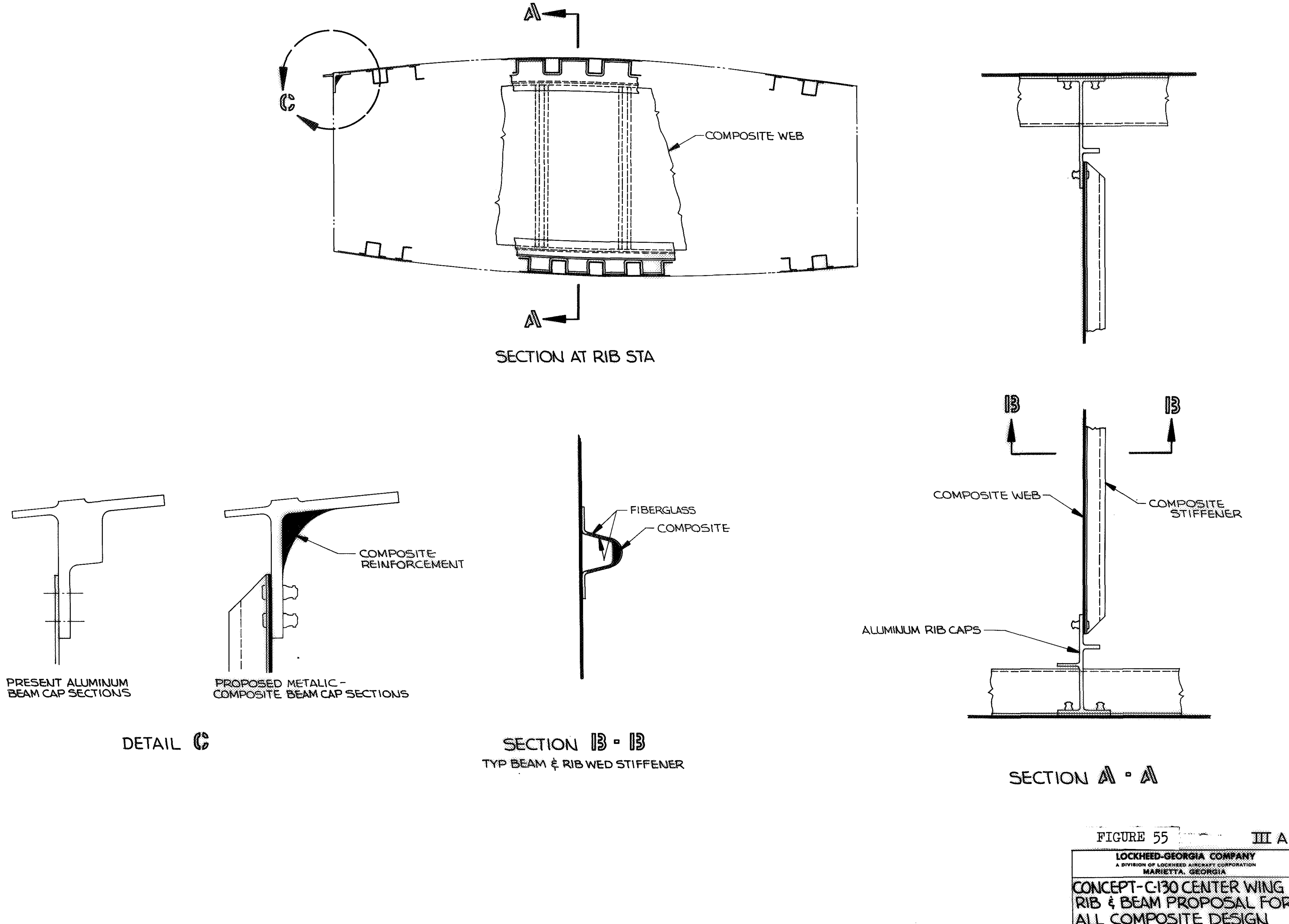
WEIGHT SUMMARY

Boron Item	Weight (Pounds)	Graphite Item	Weight (Pounds)
Boron Weight	2172.4	Graphite Weight	2199.0
Total Weight	3022.0	Total Weight	3050.7
Weight Saved	1906.2	Weight Saved	1877.5
% Weight Saved	38.7	% Weight Saved	38.1

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$929,047	$487/#
Graphite	$360,409	$192/#


B
B
A
A
SECTION C - C
SECTION D - D
COMPOSITE PANEL
SECTION B - B
ALUMINUM OR TITANIUM FITTINGS~
BUILT-UP OUTER WING ATTACH INTERFACE
D
C
D
C
SECTION A - A
FIGURE 54
III A
LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA
CONCEPT - C-130 CENTER WING-
COMPOSITE SURFACE PANELS-
METALLIC END FITTINGS



A
COMPOSITE WEB
C
A
SECTION AT RIB STA
B
B
COMPOSITE WEB
COMPOSITE STIFFENER
ALUMINUM RIB CAPS
SECTION A - A
FIBERGLASS
COMPOSITE
SECTION B - B
TYP BEAM & RIB WED STIFFENER
COMPOSITE REINFORCEMENT
PRESENT ALUMINUM BEAM CAP SECTIONS
PROPOSED METALIC-COMPOSITE BEAM CAP SECTIONS
DETAIL C
III A
LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA
CONCEPT-C-130 CENTER WING RIB & BEAM PROPOSAL FOR ALL COMPOSITE DESIGN

FIGURE 55

Major development work is required for this concept to determine feasibility of boron usage in the hat sections. The stiffness of boron would make holding the material to contour in the lay-up operation difficult. The cost/producibility effectiveness for this concept could be adversely affected if development problems encountered are extensive.

A considerable development program is required to determine manufacturing feasibility of boron usage in the hat sections. Approximately 15 feet of boron would have to be corrugated into a final panel width of 8 feet for a length of 40 feet for a single skin concept. Using hat sections would reduce the complexity and provide a better reliability factor for positioning the 0° plies in the caps of the risers. In both methods, the stiffness of the boron would make holding the material to contour in the lay-up operation difficult. The complexity of the end transitions would require additional development.

The inspection of this advanced configuration would be relatively straight forward. Contact pulse-echo ultrasonics would be used to detect any voids in the bond lines and also any delaminations in the boron sections themselves.

There is a phase/amplitude sensitive ultrasonic system which could also be used to inspect this configuration. The system has been very successful in detecting delaminations and voids in filamentary composite panels. The in-process inspection of this configuration should present no problems.

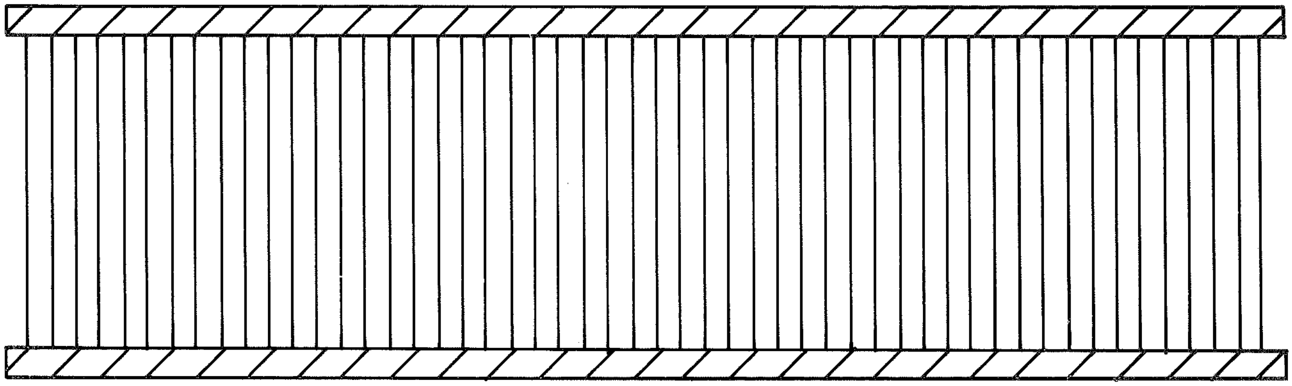
Concept III-B

This concept was selected because previous industry usage of composites had shown better payoffs with honeycomb construction. This is primarily due to the fact that the composite can be worked to such high stress levels that the only practical method of stabilization is through honeycomb construction. For this wing box, the loadings were of such a nature that the honeycomb design was actually heavier than the bonded-beaded design. This was primarily caused by the weight of the honeycomb core and the associated potting. The general structural arrangements are shown in Figures 56, 57, and 58.

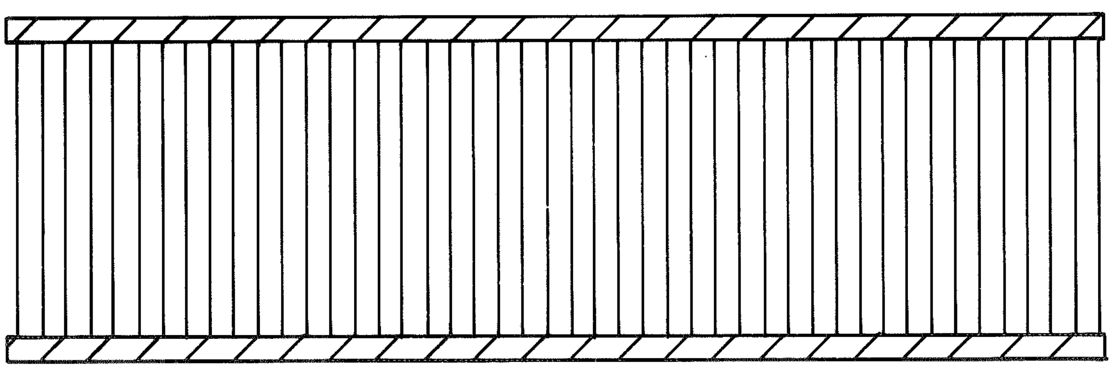
Concept III-B falls in the same category with concept III-A in that while the cost/producibility effectiveness is not as good as the other concepts, the level of potential weight savings is greater (Ref. Table XIV). In this concept, the percent of weight savings using boron material is somewhat higher than for graphite, but the cost increase per pound saved and the total cost increase factors are considerably higher. This is attributed to the higher forecasted cost of boron material.

In view of the experience gained on other composite material programs, this concept is considered to be producible, and the development program is not expected to be as extensive as that for Concept III-A.

In view of the experiences gained at Lockheed on the Boron Slat Program, this concept would be the most reproducible. Techniques developed for attaching the rib caps on the slat would be directly applicable to the wing box. Attachment of the rainbow fittings, as shown, would not produce any significant problems in light of previous experiences.



UPPER SURFACE



LOWER SURFACE

FIGURE 56

TABLE XIV

CONCEPT III-B

WEIGHT SUMMARY

Boron Item	Weight (Pounds)	Graphite Item	Weight (Pounds)
Boron Weight	2019.7	Graphite Weight	2265.8
Total Weight	3400.2	Total Weight	3665.5
Weight Saved	1528.0	Weight Saved	1262.7
% Weight Saved	31.0	% Weight Saved	25.6

COST SUMMARY

Material	Center Wing Cost Increase	Cost/Weight Ratio
Boron	$901,466	$590/#
Graphite	$430,380	$341/#

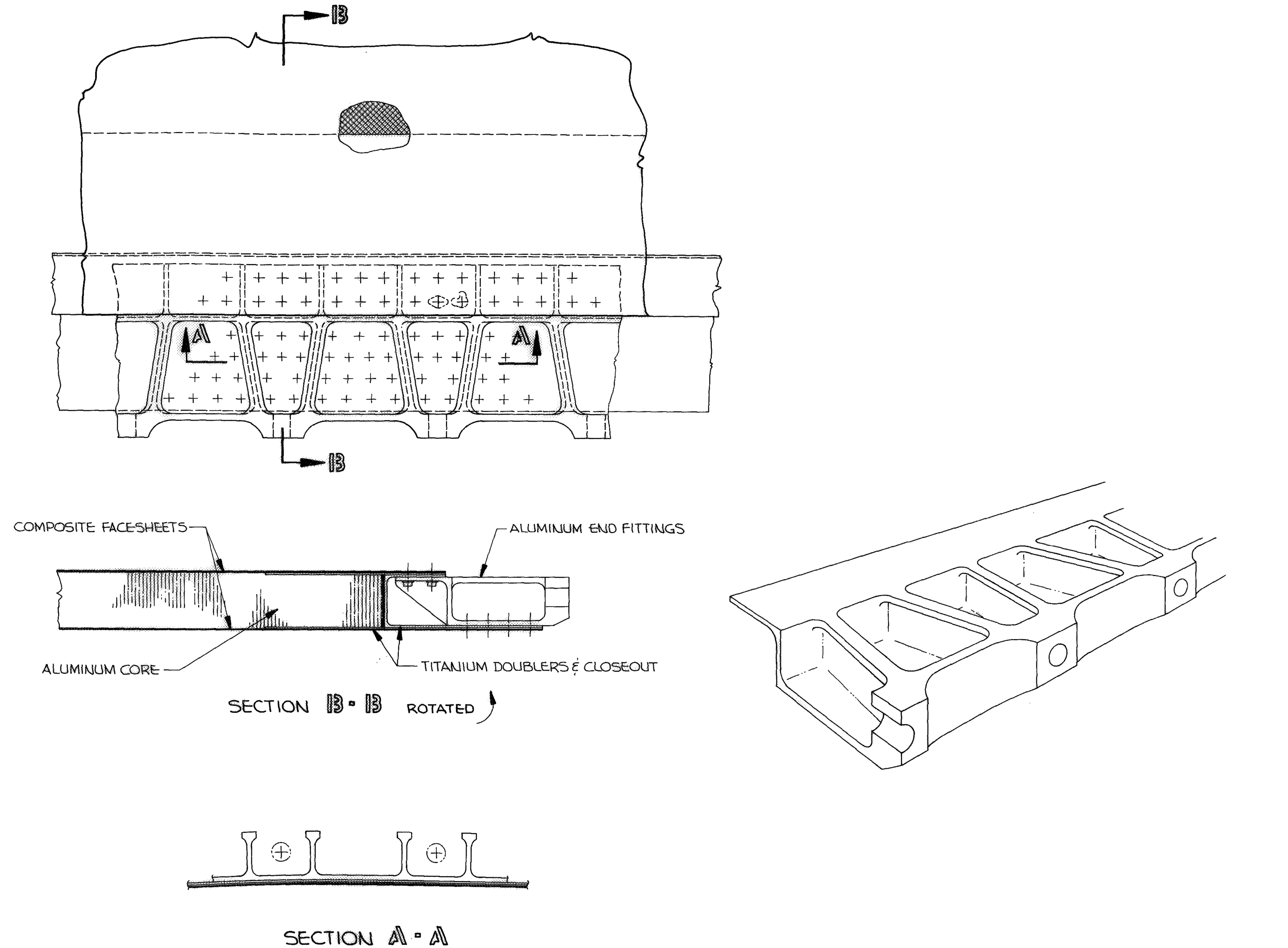


FIGURE 57 III B

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

CONCEPT-C-130 CENTER WING-
COMPOSITE FACESHETE-AL
CORE HONEYCOMB

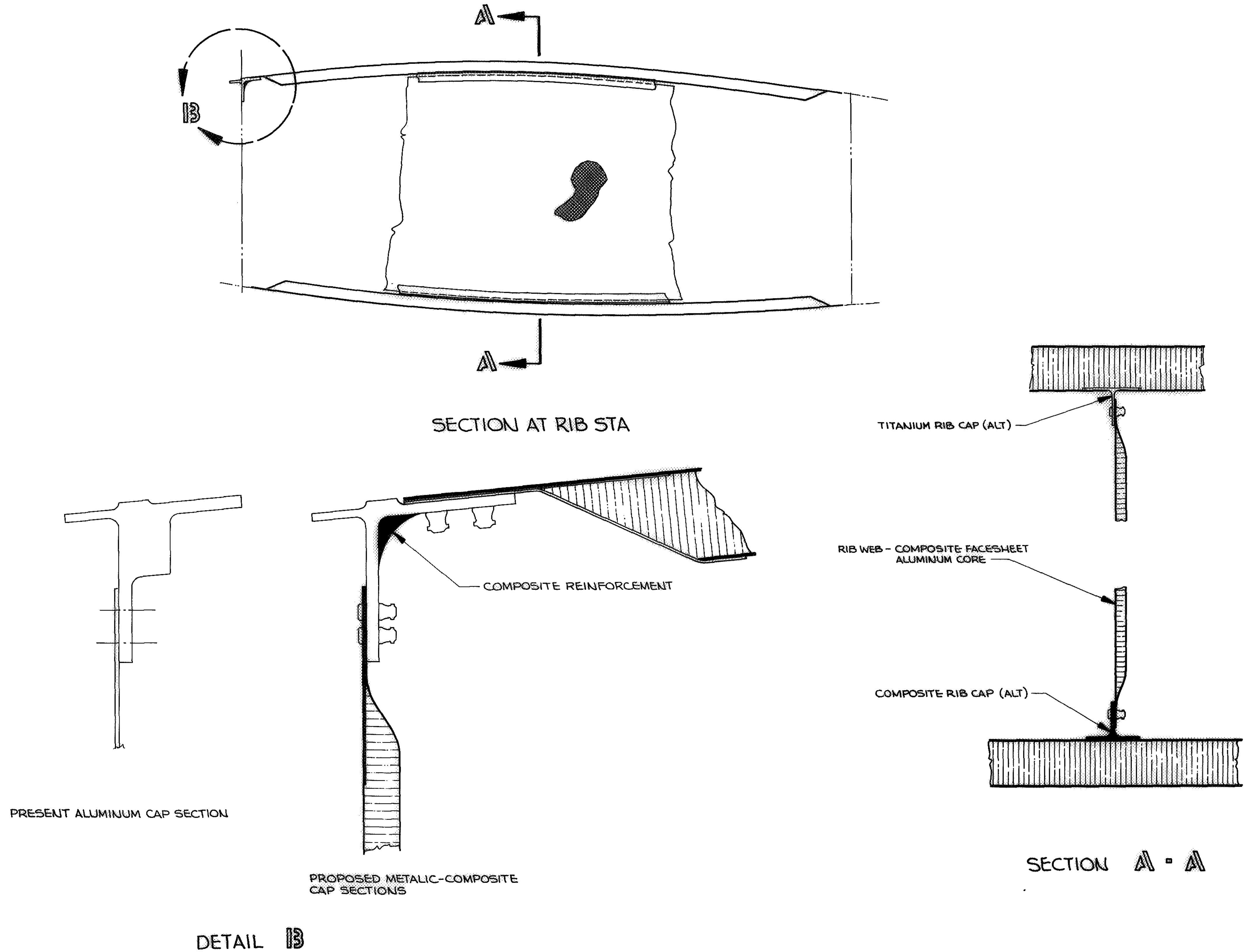


FIGURE 58 III B

LOCKHEED-GEORGIA COMPANY
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION
MARIETTA, GEORGIA

CONCEPT-C-130 CENTER WING RIB & BEAM PROPOSAL FOR COMPOSITE HONEYCOMB DESIGN

The inspection of this configuration would be straight forward. The inspection techniques used during the Boron Slat Program would apply directly to this design. Using contact pulse-echo ultrasonics, boron face sheet-to-core inspections have become routine. Care would have to be taken not to contaminate the core with the acoustic couplant used with this technique.

This configuration, just as III-A, could also be inspected with the phase/amplitude sensitive ultrasonic system using either the through-transmission or contact method. Voids and disbonds could be detected simultaneously on both sides of the structure using the through-transmission technique.

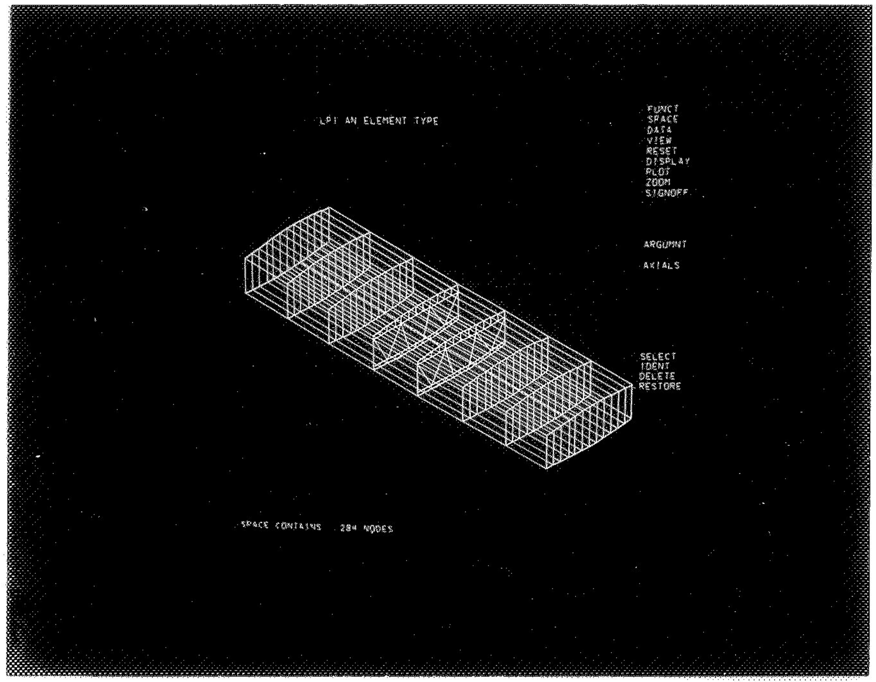
APPENDIX D

FATIGUE ANALYSIS OF FINAL DESIGN

The original stress analysis on the C-130E center wing box was performed utilizing a "unit beam" method of analysis. Basically, the technique includes the effects of spanwise bending (Mc/I) and torsion (T/2A) in the analysis. The chordwise loading is normally neglected for strength or stress-at-a-point calculations. In order to design structure with composite materials, it is necessary to have accurate structural loads in all directions, since the anisotropic nature of the materials can cause the strengths of the materials in orthogonal directions to differ by orders of magnitude. For this reason, it was decided to utilize a finite-element model of the structure to define more completely the internal loads for the structure.

The generation of the finite element model of the center wing box was easily accomplished. Use was made of an existing Lockheed-Georgia computer program which automatically generates the finite-element geometry for a wing structure by merely designating rib stations, spar stations and model coordinates at the root and tip ribs. The model geometry was checked utilizing a computer graphics display. A picture of the model as displayed on the computer graphics scope is shown in Figure 60. The spar caps, rib caps, spar web stiffeners, rib caps, rib web stiffeners, stringers and the wing box support members were simulated by axial load carrying members. The spar webs, rib webs and covers were simulated by quadrilateral shear panels. There are twelve ribs and two spars in the model. The ribs are spaced approximately forty inches apart and the spars are separated by a constant distance of approximately eighty inches. The contour of the surfaces are approximated by straight line segments from stringer to stringer. Since the C-130 center wing box has thirteen stringers on the lower surface and eleven stringers on the upper surface, a computer program was written to "lump" skin-stringer areas on the lower surface to align directly below the upper surface stringer locations. This program employed a beam distribution method to obtain "lumped" areas. A check was made on the moments of inertia about the x and z axes for stations along the wing before and after "lumping" of the areas which were used in the finite element model. The differences were less than one percent. A computer program based on the inverse transformation of the area "lumping" technique was also written to "delump" skin-stringer internal loads calculated in the model. The details of the finite element model are shown in Figure 61.

The internal loads for two C-130 external load cases were computed using the finite element computer program. These cases were compared to the internal loads computed using the same load cases for the unit beam computer program. The two load cases were a wing up-bending case due to a balanced maneuver and a wing down-bending case due to a 3G taxi condition. Besides being a check on the reasonability of the model internal loads, the comparison also served as a check on the "lumping" of skin-stringer areas and the "delumping" of stringer internal loads. The flexibilities used in the unit beam model were also used in the finite element model for the comparison. The spanwise internal loads differed by 0% to a maximum of 10% for the down-bending case and from 0% to a maximum of 5% for the up-bending case when compared with unit beam internal loads at various points. These percentages are not an indication of error but should be considered as an indication of the level of structural approximation inherent with each model. The chordwise internal loads could not be checked because the unit beam program does not compute chordwise or rib loads.


LPI AN ELEMENT TYPE
FUNCT
SPACE
DATA
VIEW
RESET
DISPLAY
PLOT
ZOOM
SIGNOFF
ARGUMNT
AXIALS
SELECT
IDENT
DELETE
RESTORE
SPACE CONTAINS 284 NODES

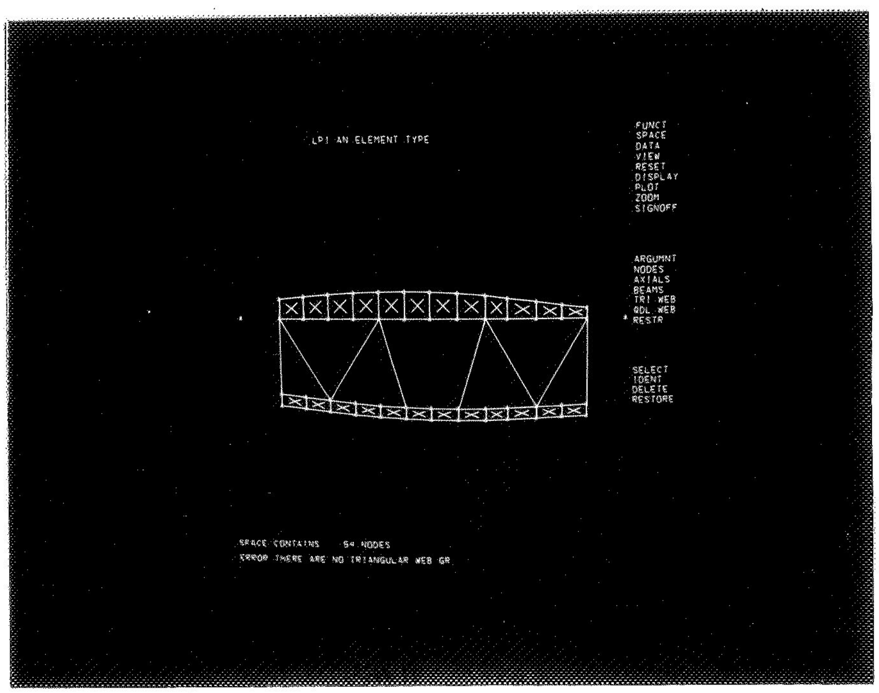
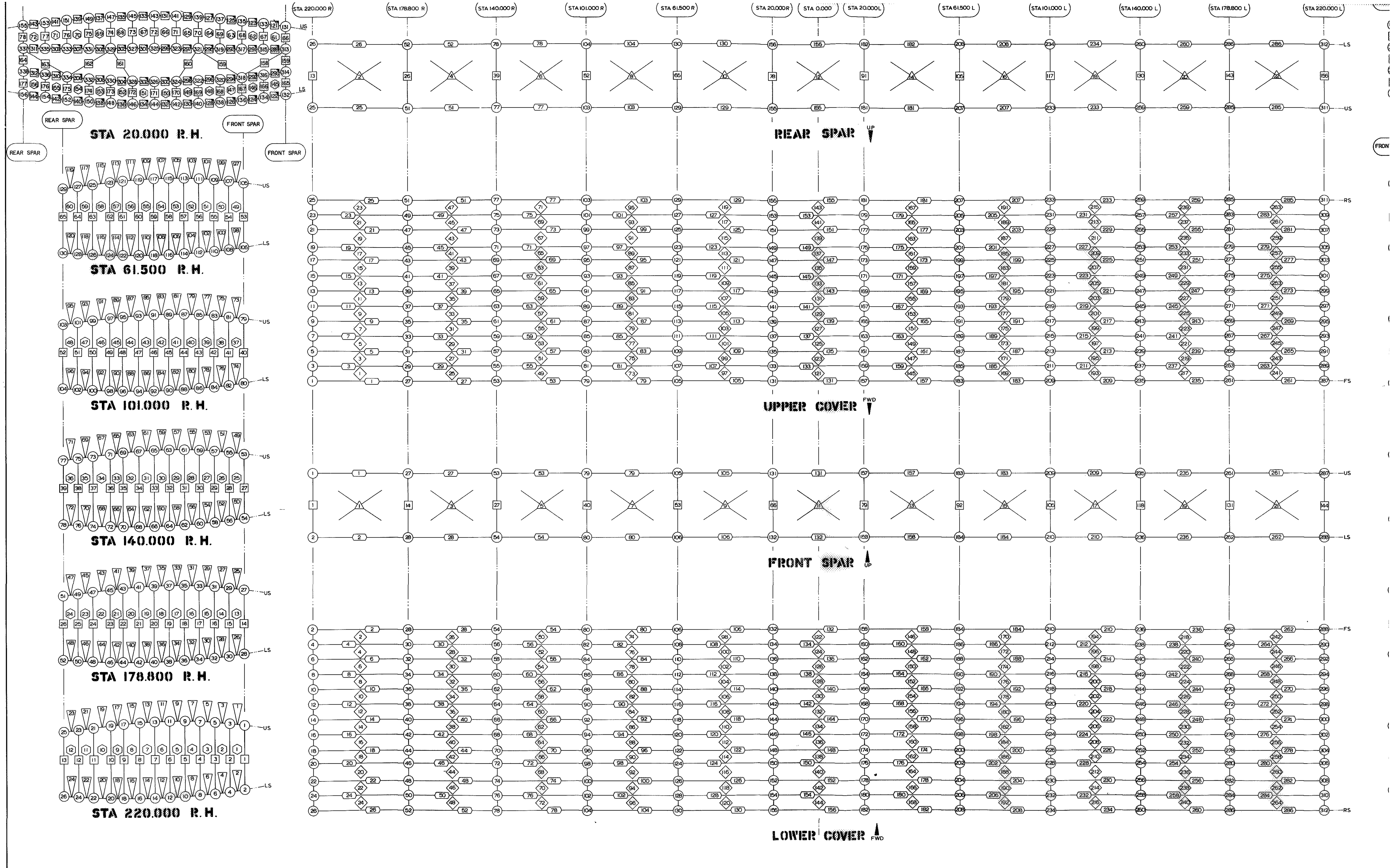

LPI AN ELEMENT TYPE
FUNCT
SPACE
DATA
VIEW
RESET
DISPLAY
PLOT
ZOOM
SIGNOFF
ARGUMNT
NODES
AXIALS
BEAMS
TRI WEB
QDL WEB
RESTR
SELECT
IDENT
DELETE
RESTORE
SPACE CONTAINS 54 NODES
ERROR THERE ARE NO TRIANGULAR WEB GR

FIGURE 60



STA 220.000 R
STA 178.800 R
STA 140.000 R
STA 101.000 R
STA 61.500 R
STA 20.000 R
STA 0.000
STA 20.000 L
STA 61.500 L
STA 101.000 L
STA 140.000 L
STA 178.800 L
STA 220.000 L
REAR SPAR
UP
UPPER COVER
FWD
FRONT SPAR
LOWER COVER
STA 20.000 R. H.
STA 61.500 R. H.
STA 101.000 R. H.
STA 140.000 R. H.
STA 178.800 R. H.
STA 220.000 R. H.

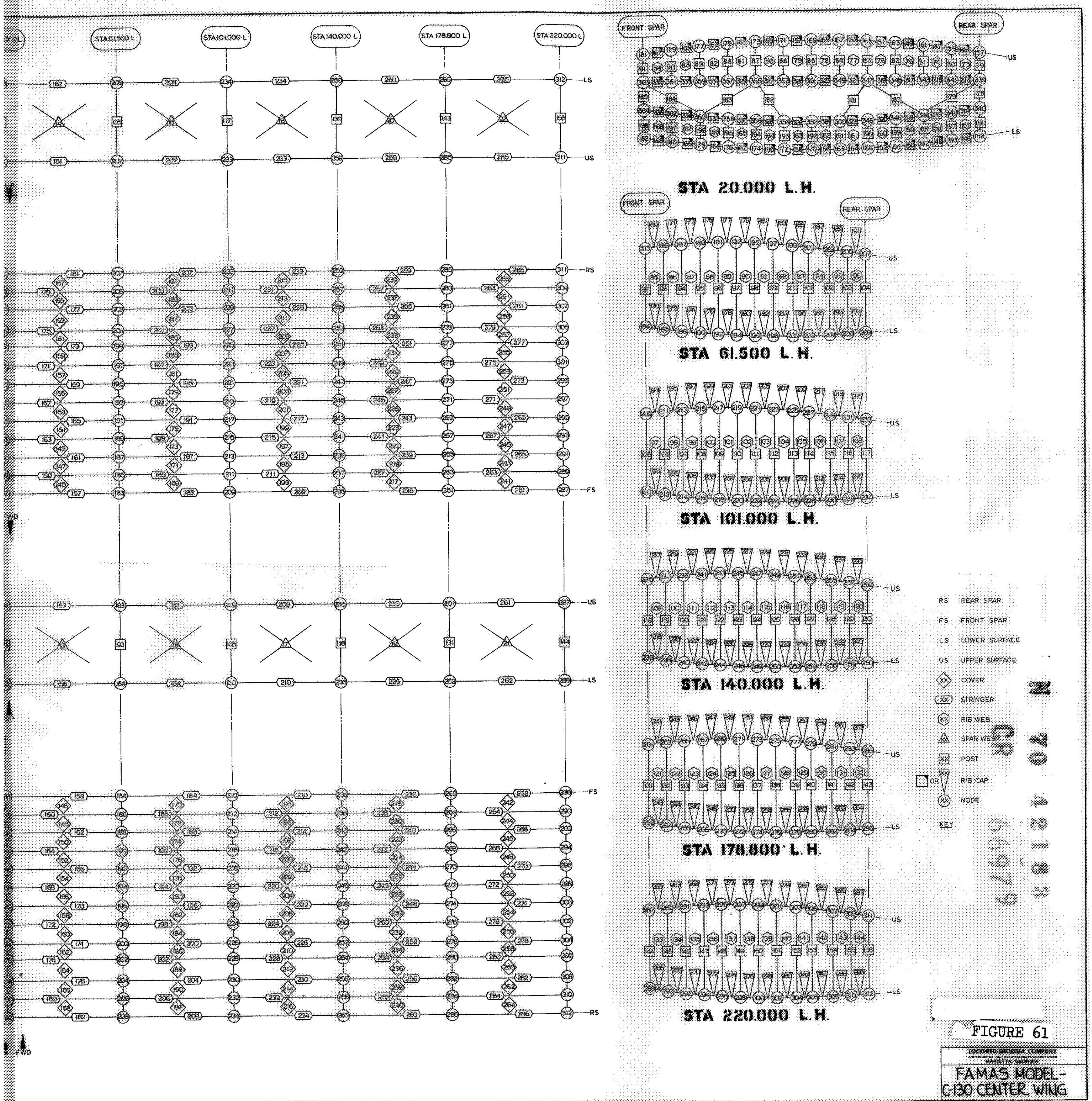


FIGURE 61

LOCKHEED-GEORGIA COMPANY
MARIETTA, GEORGIA

FAMAS MODEL-
C-130 CENTER WING

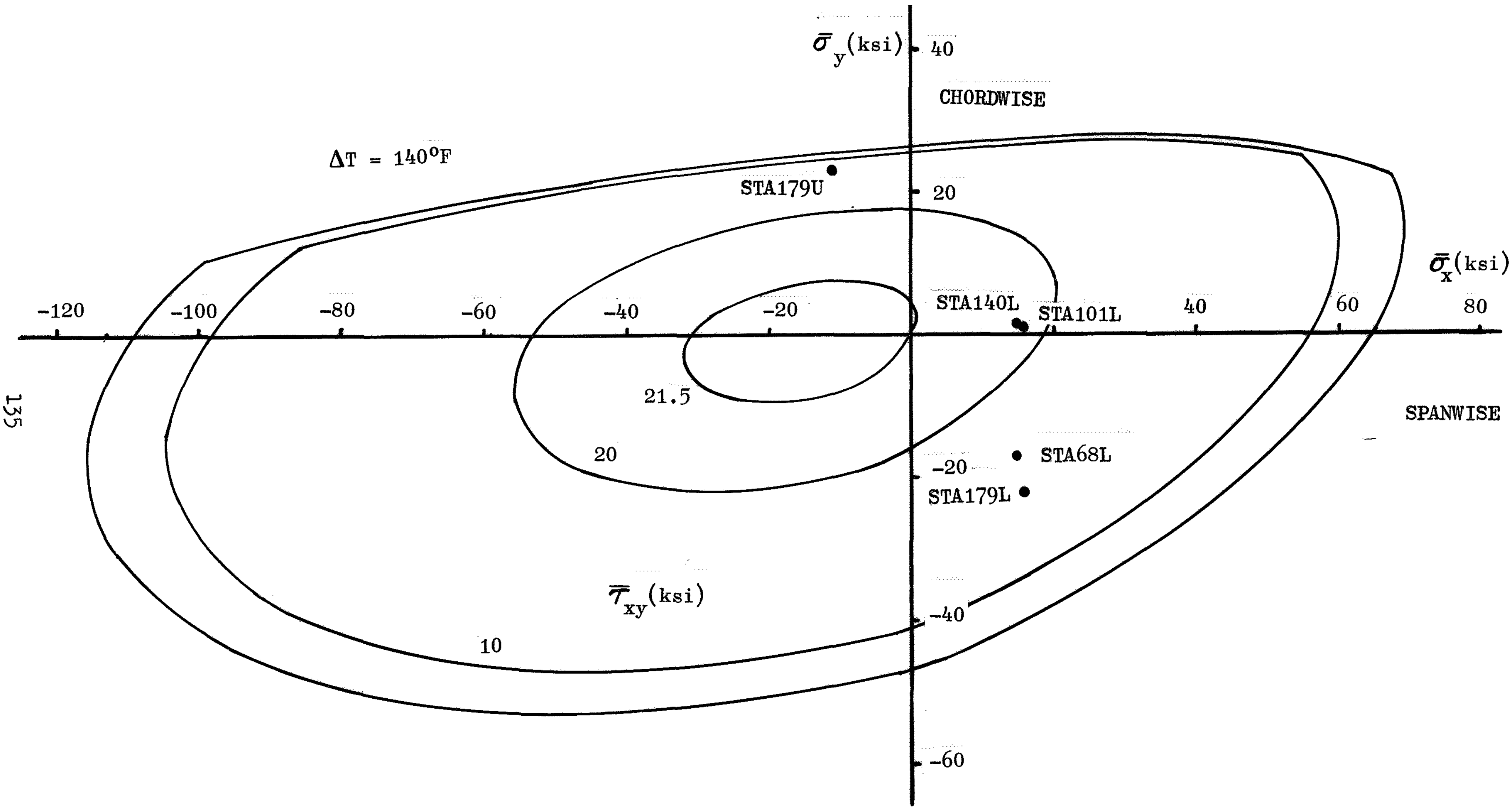
An illustration of typical stress states in the skins for a static load case is given in Figure 62. This load case is a 3G taxi maneuver. The limit load failure envelope is given for the 30 percent boron reinforcement. As can be seen in the Figure, there are substantial chordwise stresses.

The flexibilities used in the finite element model for the fatigue analysis were derived in the following way. Unidirectional boron composite material was added to the C-130E original aluminum skins and stringer crowns after the crowns were machined to a thickness of eight hundredths of an inch (See Figures 1 & 2). The composite added to the skin was concentrated under the stringer. This was done in order to keep the fasteners in the skin from being covered and to be able to inspect the aluminum skin for cracks. Also, this is a more desirable position for lightning strike protection. The amount of composite added to each skin-stringer element was calculated by the analysis presented in Appendix B. By initially assuming there would be no residual thermal stress in a bonded aluminum-composite material, the amount of composite needed would be the ratio of the modulus of elasticity of aluminum to the modulus of elasticity of composite material times the difference between the area in the aluminum reinforced element and the aluminum in the composite reinforced element. This would guarantee the stresses in the aluminum in the composite reinforced section to be equal to those in the aluminum design for the same amount of applied load. Since residual thermal stresses are present due to the elevated temperature bond, an additional amount of composite was needed to reduce the stress in the aluminum. One of the variables that determines the amount of composite required is the ultimate static stress level of each skin-stringer element in the aluminum reinforced design. Since each skin-stringer element in the aluminum reinforced design has a different ultimate static stress level, a constant average

FIGURE 62

COMPOUND COMPOSITE LIMIT FAILURE ENVELOPE

ALUMINUM (70%) / BORON (30%)



INCREASE IN COMPOSITE AREA VS. TEMPERATURE DIFFERENTIAL
(ALUMINUM-UNIDIRECTIONAL BORON COMPOSITE BONDED STRUCTURE)

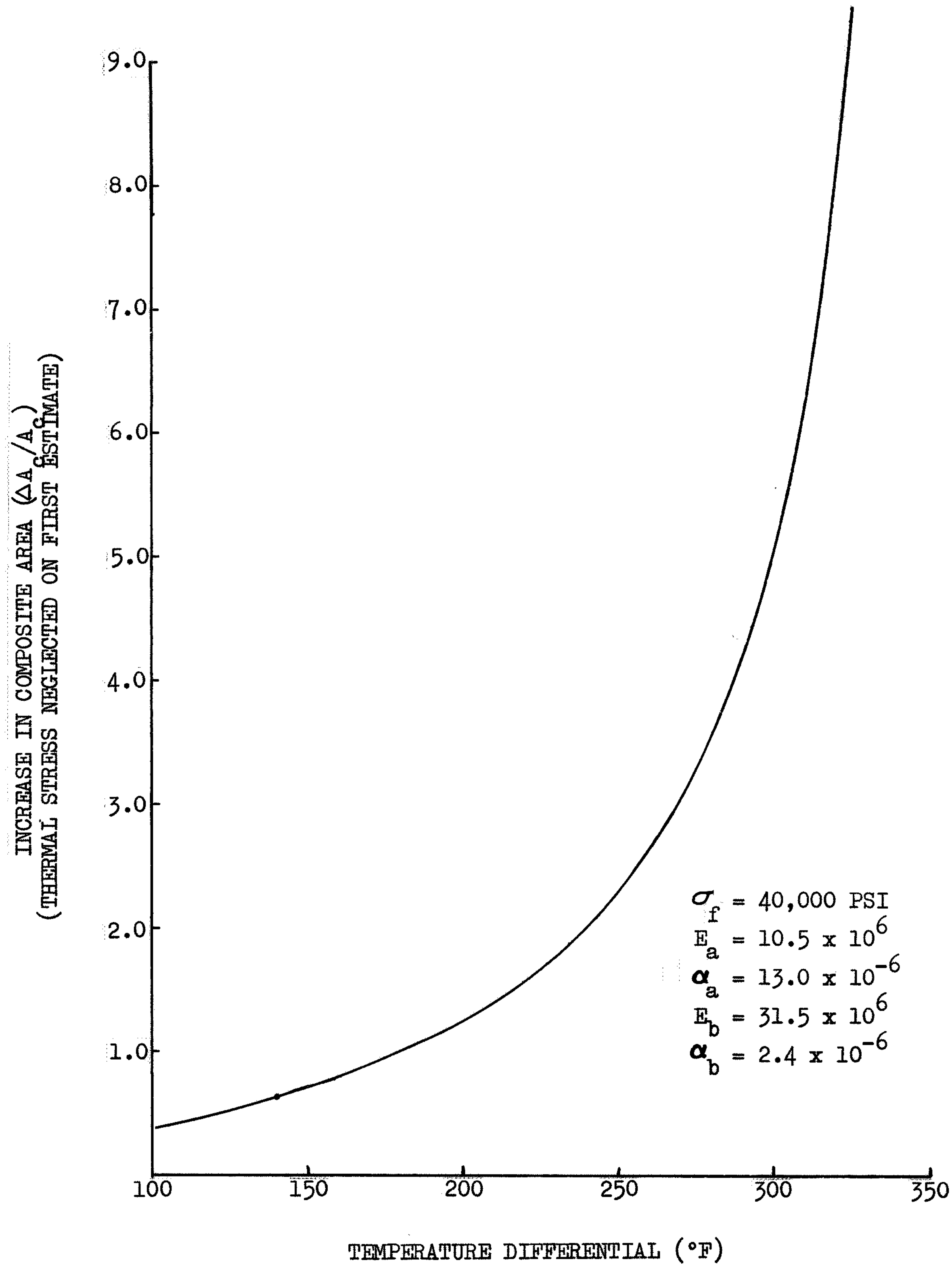


FIGURE 63

maximum value of 40,000 psi was chosen in order to reduce the tremendous amount of computation that would have been required otherwise. Having chosen a constant stress, the amount of composite needed varies only with the difference in temperature between the bonding temperature and the average operating temperature of the skin-stringer element on the aircraft and the amount of composite material added under the assumption of no thermal residual stress (See Figure 23). A value of $\Delta T = 140^{\circ}$ was assumed for the analysis. This is based on an average operating temperature of $50^{\circ}F$ ($160^{\circ}F$ to -65°) and a bond temperature of $190^{\circ}F$. The composite material needed to return the stress in the aluminum back to 40,000 psi was 1.64 times the composite material needed if no residual thermal stress existed (Ref. Figure 63).

The flexibilities of the boron composite material and the aluminum were input into the wing box finite element model. The upper surface flexibilities were input directly into the model and the lower surface flexibilities were input after "lumping". For convenience in distinguishing the proportion of the load in each material, duel model elements were created for each element in the model. The composite flexibilities were added to these elements. For the duel elements where composite material was not added (rib webs, spar webs, rib caps), a zero flexibility was input to the model.

The load case used to generate moment-stress ratios for the fatigue analysis was a ten million inch pound spanwise bending moment applied at W.S. 220 on the model. By using this load case, internal structural loads were calculated by the computer. The internal plus residual thermal loads in the aluminum portion of the skin-stringer elements were used for the moment-stress calculations. The composite material was assumed to have better fatigue endurance than the aluminum; consequently, only the aluminum structure was considered fatigue critical.

For the moment-stress calculations, the same wing stations were used in the composite reinforced wing as were used for the aluminum reinforced wing (W.S. 61, W.S. 101, W.S. 178, and W.S. 220). In the aluminum wing box fatigue analysis, a varying spanwise bending moment from a C-130 external load condition was used. The load case used to generate the internal loads

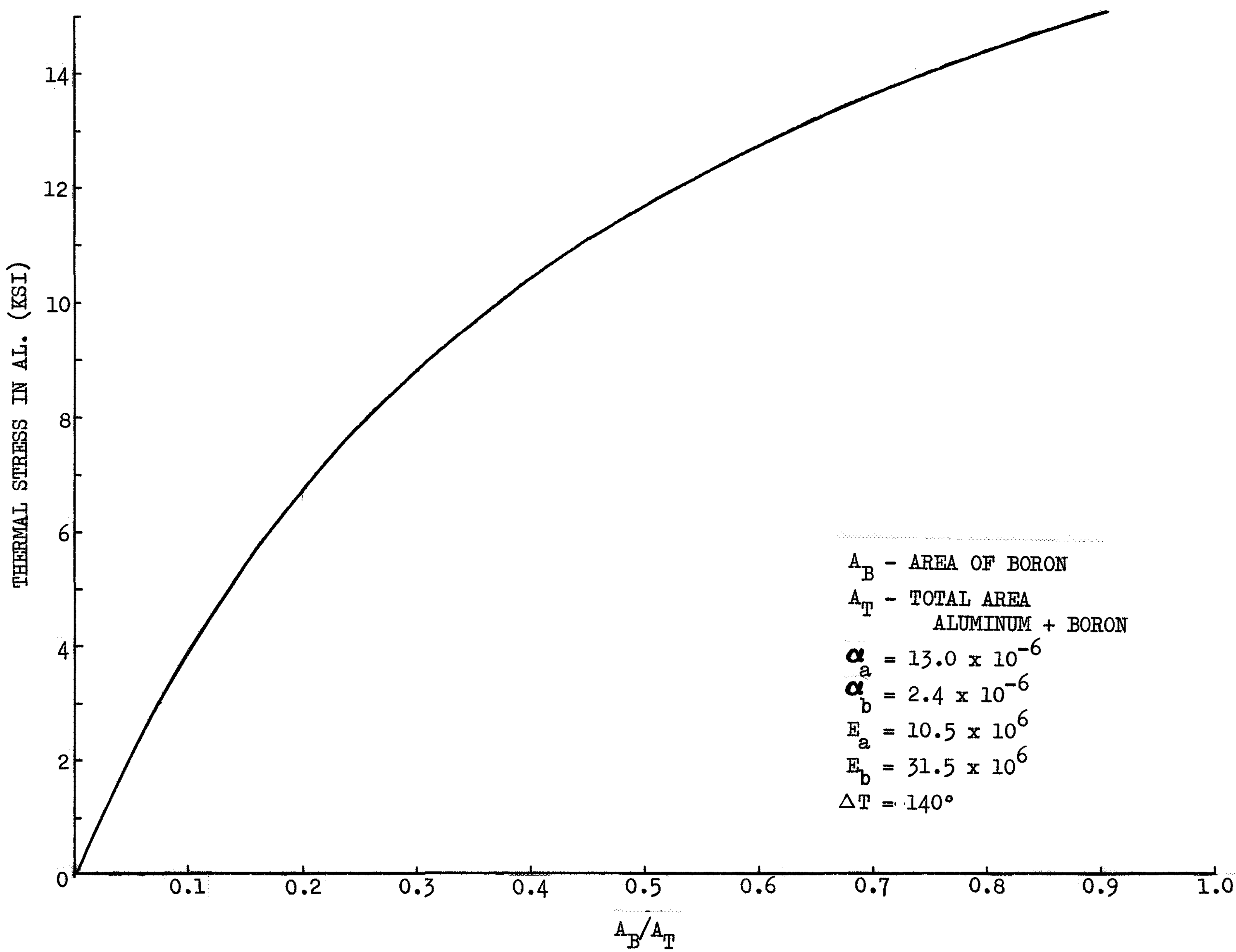

THERMAL STRESS IN AL. (KSI)
0
2
4
6
8
10
12
14
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
A_B/A_T
A_B - AREA OF BORON
A_T - TOTAL AREA
ALUMINUM + BORON
$\alpha_a = 13.0 \times 10^{-6}$
$\alpha_b = 2.4 \times 10^{-6}$
$E_a = 10.5 \times 10^{6}$
$E_b = 31.5 \times 10^{6}$
$\Delta T = 140°$

FIGURE 64

for the fatigue analysis of the composite reinforced wing was a ten million inch pound (unit) spanwise bending moment applied at W.S. 220 on the finite element model. The internal stresses calculated from the unit finite element load condition had to be modified by the ratio of the all aluminum wing box spanwise bending moment at a particular wing station to the ten million inch pounds bending moment. The residual thermal stresses were calculated for each skin-stringer element at the stations used for the analysis (Ref. Figure 64). For each station to be analyzed, the stresses due to applied load plus the thermal stresses were calculated for each element. The value of the highest total stress for each wing station was divided into the moment used in the aluminum reinforced analysis at that station. The values obtained were the moment-stress ratio for each station.

In the area of cut-outs in the wing skins, a separate detailed finite element model was used to obtain moment-stress ratios (See Figure 65). The numbers shown on the finite element model are node numbers. The internal loads at specified stations from the finite element wing model were applied as external loads on the cut-out model. The aluminum skin doublers around the cut-outs were replaced with boron composite laminates consisting of 60% 0° and 40% $\pm 45^\circ$ plies. This laminate combination has about the same shear modulus as aluminum. All unidirectional plies were found to be inadequate for reinforcement around the cut-outs because high shear flows occur due to local load redistributions. Hence, the $\pm 45^\circ$ plies were added to give the shear rigidity required.

The cut-out at W.S. 120 on the lower surface was the only cut-out analyzed in this phase of the program. The moment-stress ratios were obtained from internal loads calculated by the detailed model in a similar manner as previously described. The weight savings achieved by utilizing the boron composite doubler around the cut-out was approximately 5 percent. It is felt that this weight savings is not worth the risks involved in providing that such a reinforcement technique is feasible on actual hardware.

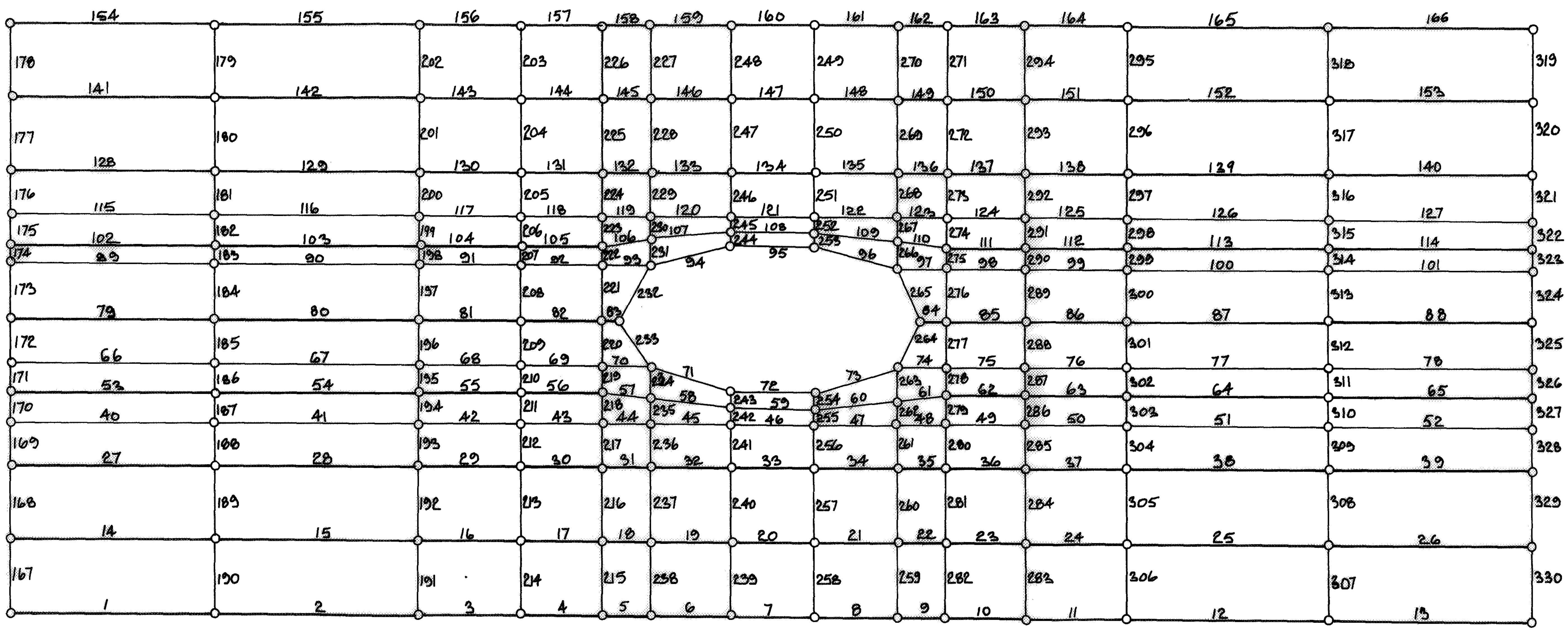


FIGURE 65

WS120.5 CUT-OUT

CENTER WING LOWER SURFACE AXIAL MEMBER MODEL

APPENDIX E

PRELIMINARY COST ESTIMATE FORMULA

$$C_B \times W_B + C_F - C_A \times W_A + C_R \times W_R = C_N$$

OR:

$$C_{BC} - C_{AC} + C_{AR} = C_N$$

WHERE:

$$C_N - C_C = C_I$$

$$\frac{C_I}{W_S} = C/P$$

NOMENCLATURE:

C_B	=	Cost of boron material per pound
W_B	=	Weight of boron material used
C_F	=	Cost of fabrication
C_A	=	Cost of aluminum structure per pound
W_A	=	Weight of aluminum structure removed
C_R	=	Cost of remaining structure per pound
W_R	=	Weight of remaining structure
C_N	=	Net Cost of center wing utilizing boron material
C_{BC}	=	Cost of boron construction
C_{AR}	=	Cost of aluminum structure remaining
C_C	=	Cost of "all-aluminum" center wing
C_I	=	Cost increase by utilizing boron material
W_S	=	Weight saved by utilizing boron material
C/P	=	Cost increase per pound of weight saved
C_{AC}	=	Cost of aluminum construction

EXAMPLE PROBLEM - CONCEPT IIA:

WEIGHT SAVED BY USE OF BORON COMPOSITE MATERIAL: 642.7#

WEIGHT OF BORON MATERIAL USED: 477.5#

COST OF BORON MATERIAL USED: $173,250
(577.5# X $300/#)

COST OF BORON STRUCTURE FABRICATION: $107,196

TOTAL COST - MATERIAL AND LABOR: $280,446
($173,250 + $107,196)

WEIGHT OF EQUIVALENT ALUMINUM STRUCTURE REMOVED: 1,200.2#
(577.5# + 642.7#)

COST OF EQUIVALENT ALUMINUM STRUCTURE REMOVED: $20,744
(1,200.2# X $17)

TOTAL COST OF RESULTING COMPOSITE STRUCTURE: $259,702
($280,446 - $20,744)

WEIGHT OF OTHER MATERIAL: 3,659.1#

COST OF OTHER MATERIAL: $62,205
(3,659.1# X $17)

NET COST - CENTER WING REINFORCED WITH BORON MATERIAL: $321,907
($259,702 + $62,205)

COST INCREASE OF CENTER WING WITH BORON MATERIAL: $241,907

COST INCREASE PER POUND SAVED: $376/POUND
($241,907 642.7#)

NOTE: Figures are preliminary estimates that represent unit cost of center wing at an assumed production volume and boron material at $300/Pound.