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SUPERCritical WING PRELIMINARY DESIGN STUDY

GENERAL DYNAMICS
FORT WORTH DIVISION

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TECHNICAL

DECEMBER 1975

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FINAL REPORT FOR PERIOD APRIL-OCTOBER 1975

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Prepared For
AIR FORCE FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio 45433

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)	This analytical effort was applied to a supercritical wing box with provisions for variable camber leading and trailing edge devices. The wing geometry and criteria are consistent with the requirements for the FB-111 aircraft or growth versions of this strategic aircraft. Cost and weight reduction of the wing box was realized thru concept innovation and application of a systematic design approach that utilizes a quantitative evaluation.				

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20. Abstract (Continued) *Pr p 1473A*

system for selecting the most promising concepts from a broad array of alternate designs.

(were considered and

Twenty-one metallic and twelve composite concepts including two baseline concepts were defined as one inch cross-section sketches. Cost and weight screening of these resulted in the selection and definition of fourteen metallic concepts, five composite concepts and a baseline on analytical assembly drawings as 48 inch long, constant section analytical models. From these, two metallic concepts and one composite concept were selected and developed into preliminary designs of complete wing boxes. Evaluation of the three designs showed weight savings of up to 27% and cost savings of up to 12.3% when compared to a baseline of the same size but configured to FB-111 construction and materials. *These*

The three advanced designs meet static strength, fatigue, fracture, thermal and flutter requirements for the FB-111 aircraft.

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F O R E W O R D

The analytical structural study reported in this document was sponsored by the Air Force Flight Dynamics Laboratory (AFFDL). This work was performed under contract F33615-75-C-3104, AFFDL Project Number 1368, Task Number 136802, Work Unit Number 13680223, "Supercritical Wing Preliminary Design Study".

Mr. Charles L. Ramsey (AFFDL/FBS) was the Air Force Project Engineer.

This study was performed at General Dynamics Fort Worth Division with D. F. Davis as Program Manager. Other principal participants in the program were: E. W. Gomez, Stress Analysis; W. C. Rister, Fatigue and Fracture; Olin E. Weiss, Structural Design (Composites); R. W. McAnally, Structural Design (Metals); C. J. Sawey, Manufacturing Engineering; R. L. Haller, Flutter Analysis; and H. E. Bratton, Information Transfer.

The study was conducted April through October of 1975.

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SECTION I

INTRODUCTION AND SUMMARY

This document reports the procedure and results of the analytical work accomplished under contract F33615-75-C-3104, "Supercritical Wing Preliminary Design Study".

This study involved the development of preliminary design definitions and evaluation of optimum metallic and composite wing box designs for a variable camber supercritical wing designated as ATW-4. The ATW-4 wing is sized and configured for the FB-111 aircraft or various growth versions of the FB-111. Results of the study analytically demonstrate the feasibility of significantly reducing the cost and weight of the wing box component while maintaining strength, fatigue and fracture characteristics consistent with requirements of the FB-111 aircraft.

The cost and weight advantages of the advanced designs developed during the study when compared to the baseline (633RW000) give the following results:

633RW001 - Laminated Skin Y-Spar Wing Box

Weight Savings 15%
Cost Savings 12.3%

633RW002 - Laminated Skin Canted Spar Wing Box

Weight Savings 12%
Cost Savings 10.7%

633RW003 - Composite Skin & Y-Spar Wing Box

Weight Savings 27%
Cost Savings -29% to -4.4% (Depending on
Assumed 1980 Gr/Ep Cost)

Figure 1 Program Flow Diagram, outlines the approach taken and shows the sequence of tasks accomplished to complete the ATW-4 wing box study. The main tasks completed during the program are summarized below:

1. Assembled baseline loads, ATW-4 geometry and criteria and reduced to a usable form.
2. Selected materials for consideration in the study and developed fatigue and fracture allowable stress curves for each material.
3. Assembled an array of 31 metallic and composite wing box concepts from other programs and through innovation of new concepts. These were defined on cross-section sketches, sized, weighed, and costed with the most promising selected for analytical assembly iteration.
4. Defined alternate concepts plus a baseline on 48 inch span analytical assembly drawings. These were analyzed, evaluated and two metallic and one composite concept were selected for input into the preliminary design iteration.
5. Preliminary design drawings were prepared defining the three selected concepts plus the baseline for the entire ATW-4 wing box from pivot to tip.
6. Static strength, fatigue, fracture and flutter analyses was conducted for each of the preliminary wing box designs.
7. Evaluative data for each design parameter in the AFFDL Merit Rating System was computed for each of the preliminary designs plus the baseline. The preliminary designs were then scored and ranked.
8. A follow-on plan that would provide the proof-of-concept was developed.

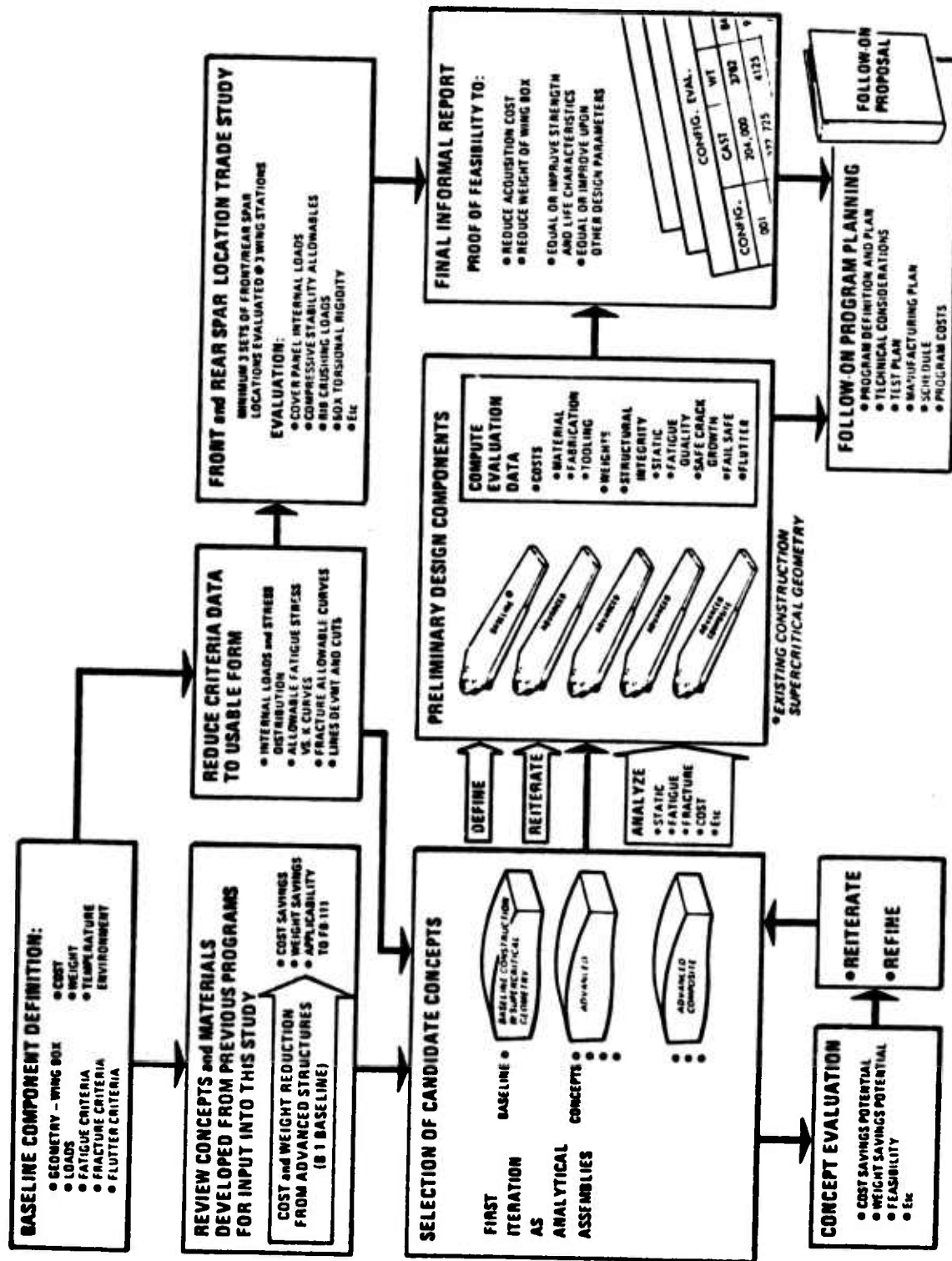


Figure 1 Program Flow Diagram for Supercritical Wing Box Study

S E C T I O N I I

B A S E L I N E D E F I N I T I O N

The baseline article is the wing box structure from pivot to tip for a supercritical wing configuration designated as ATW-4. The ATW-4 configuration incorporates variable camber leading and trailing edge devices and provides a planform area of 725.7 square feet. The planform is shown in Figure 2.

2.1 BASELINE DESCRIPTION

The wing box is defined as the primary wing structure from the wing pivot fitting outboard to the wing tip splice. It consists of seven spars, upper and lower one-piece skins, and eight primary bulkheads. The outer section is spliced to the wing pivot fitting between center spar stations 97.7 and 106.8. All wing box loads are transferred to the pivot fitting through this connection. The configuration was developed in this program by revising the F-111F wing box to the planform and airfoil for the ATW-4 configuration, deleting provisions for the external stores, and conducting trade studies to optimize the number of spars. The concept for manufacturing each element of the box assembly for the F-111F was retained in that the skins are tapered and etched and the spars and bulkheads are designed as integrally stiffened machined members. The materials used in the F-111F wing box were retained in the new baseline, such as 2024 aluminum for skins, spars, and bulkheads, and D6ac for the wing pivot fitting. The baseline structure is shown in Figures 3 and 4.

The baseline criteria includes the requirements of the FB111 aircraft, but in addition, imposes the requirements of MIL-STD-1530 and MIL-STD-83444. External loads and fatigue spectrum for the ATW-4 wing differ from those for the FB111 wing. The ATW-4 loads and fatigue spectrum are defined in FZM-12-6466 which was developed under Contract No. F33615-75-C-3018 and is included in Appendix A of this report.

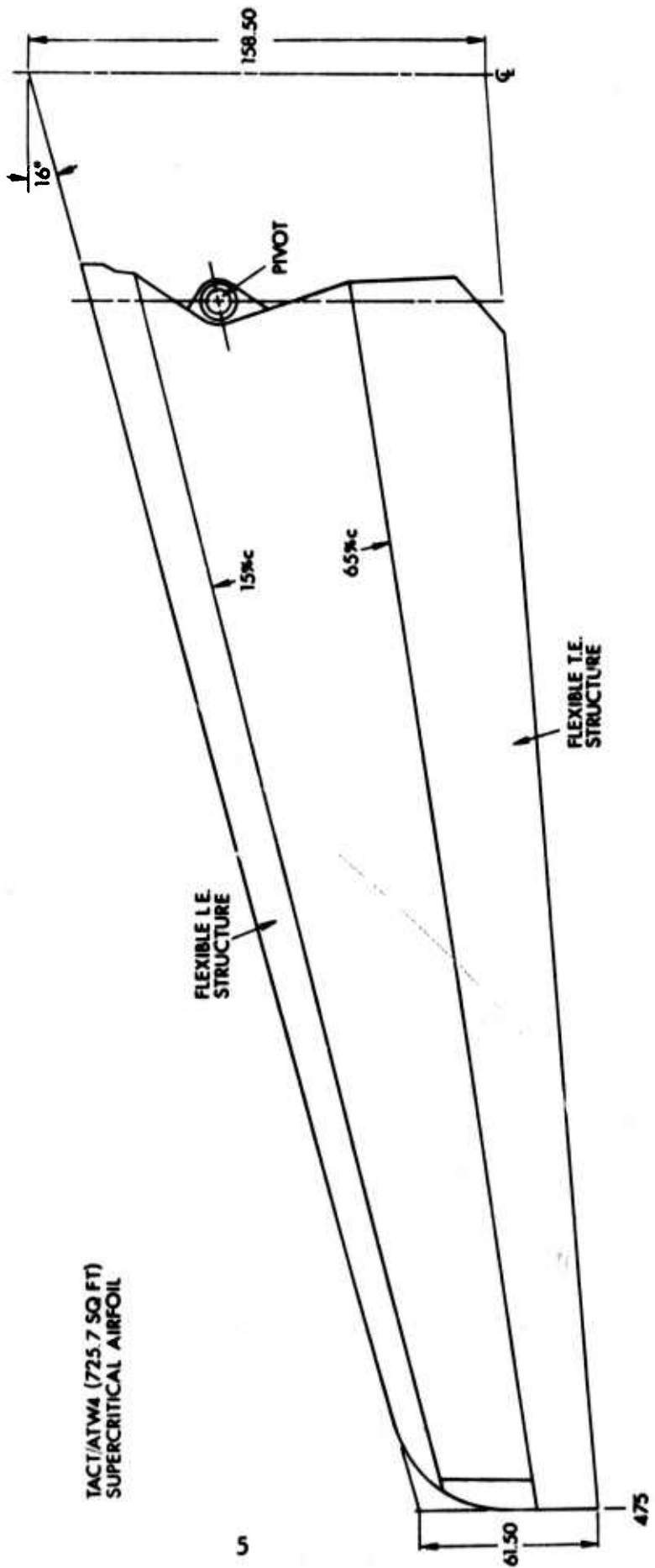
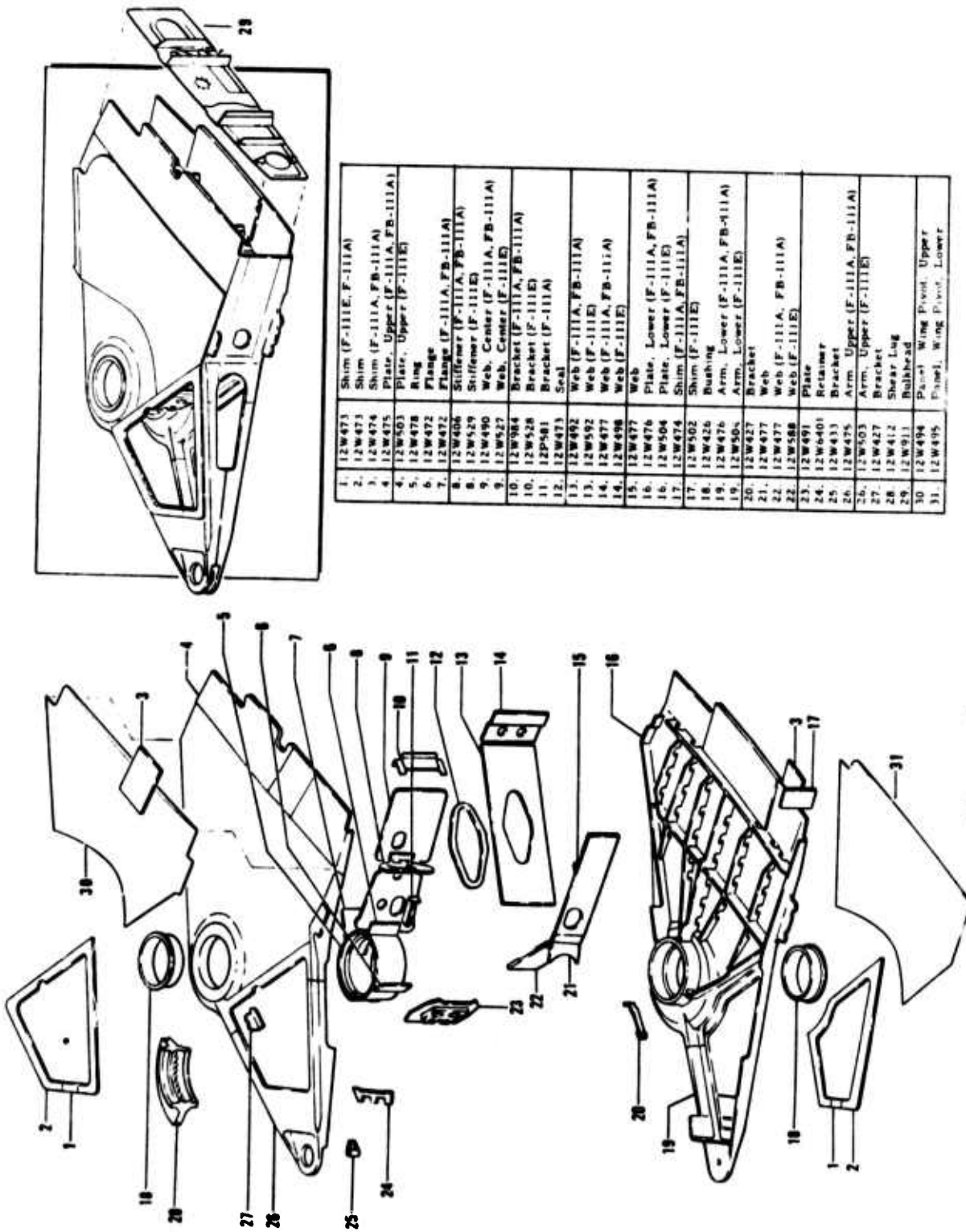


Figure 2 ATW-4 Wing Planform



1.	12W473	Shim (F-111E, F-111A)
2.	12W473	Shim (F-111A, FB-111A)
3.	12W474	Shim (F-111A, FB-111A)
4.	12W475	Plate, Upper (F-111A, FB-111A)
5.	12W503	Plate, Upper (F-111E)
6.	12W478	Ring
7.	12W472	Flange (F-111A, FB-111A)
8.	12W506	Stiffener (F-111A, FB-111A)
9.	12W529	Stiffener (F-111E)
9.	12W490	Web, Center (F-111A, FB-111A)
9.	12W527	Web, Center (F-111E)
10.	12W984	Web (F-111A, FB-111A)
10.	12W528	Bracket (F-111E)
11.	12P581	Bracket (F-111A)
12.	12W473	Seal
13.	12W492	Web (F-111A, FB-111A)
13.	12W592	Web (F-111E)
14.	12W477	Web (F-111A, FB-111A)
14.	12W498	Web (F-111E)
15.	12W477	Web
16.	12W476	Plate, Lower (F-111A, FB-111A)
16.	12W504	Plate, Lower (F-111E)
17.	12W474	Shim (F-111A, FB-111A)
17.	12W502	Shim (F-111E)
18.	12W426	Bushing
19.	12W426	Arm, Lower (F-111A, FB-111A)
19.	12W505	Arm, Lower (F-111E)
20.	12W427	Bracket
21.	12W477	Web
22.	12W477	Web (F-111A, FB-111A)
22.	12W588	Web (F-111E)
23.	12W491	Plate
24.	12W401	Retainer
25.	12W433	Bracket
26.	12W475	Arm, Upper (F-111A, FB-111A)
26.	12W503	Arm, Upper (F-111E)
27.	12W427	Bracket
28.	12W412	Shear Lug
29.	12W911	Bulkhead
30.	12W494	Panel, Wing Pivot, Upper
31.	12W495	Panel, Wing Pivot, Lower

Figure 3 Baseline Pivot Fitting

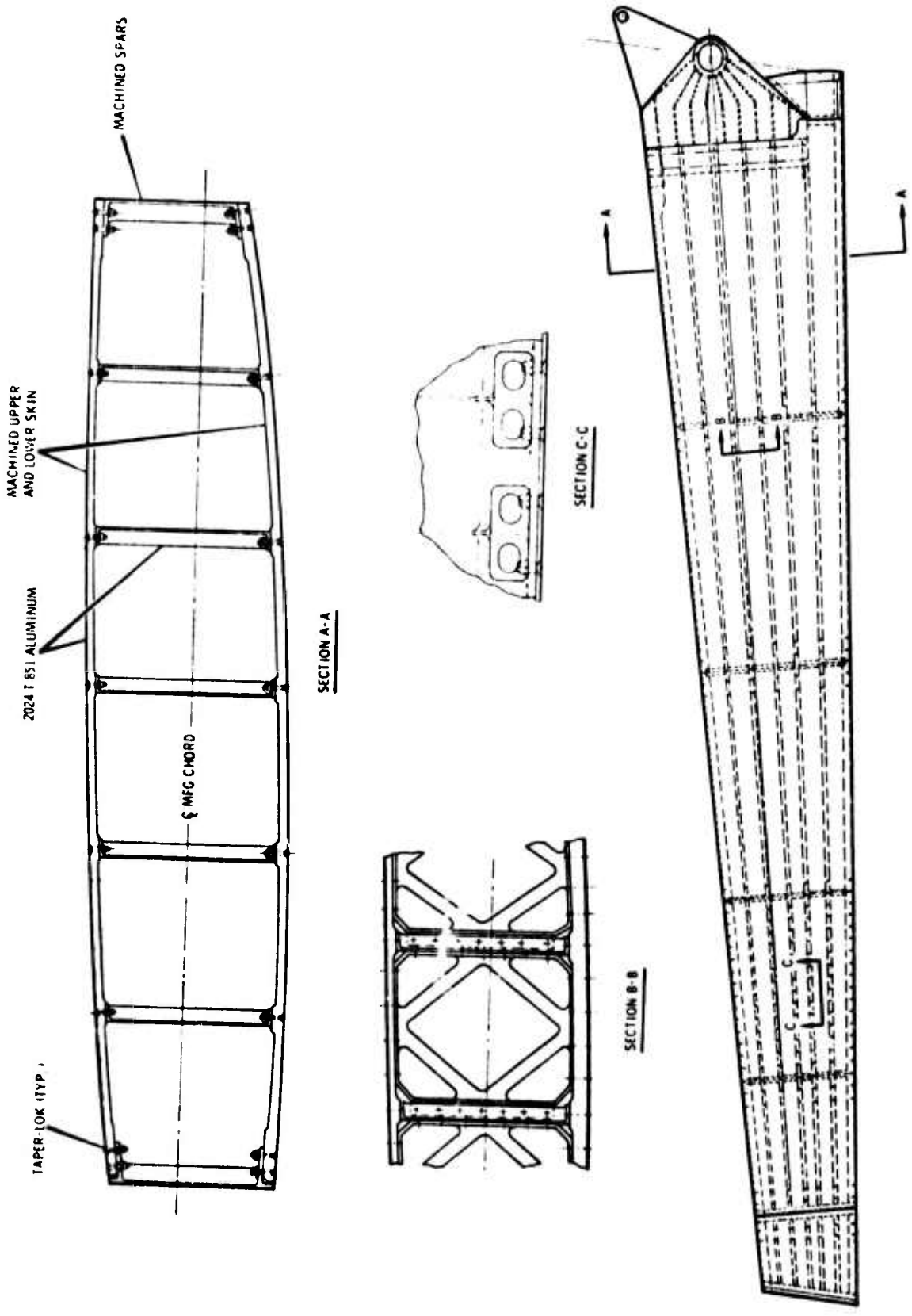


Figure 4 ATW-4 Wing Box Baseline

SECTION III

DESIGN

Design definition and analysis was conducted in sufficient detail to determine the applicability of innovative and advanced structural concepts to the main wing box of a supercritical variable camber wing configuration.

3.1 DESIGN APPROACH AND EVALUATION SYSTEM

A systematic iterative design method was applied that utilizes the AFFDL merit rating system to quantitatively evaluate each design parameter of each alternate concept after each iteration for scoring and selecting concepts for the succeeding iteration.

3.1.1 Design Approach

A methodical system developed on previous contracts (F33615-72-C-2149 and F33165-74-C-3026) was used in this program which allows a large number of ideas to be evaluated for potential application to the final design. In this process, ideas are developed into one inch span cross-section drawings; the cross-sections are analyzed and ranked; highest ranking cross-sections are further developed into forty eight inch span analytical assemblies; the analytical assemblies are analyzed and ranked; the highest ranking analytical assemblies are then drawn up as preliminary designs of full wing boxes; and finally, these preliminary designs are analyzed and ranked for final selection of the highest potential concept. Details of each step of this methodology are discussed in the following paragraphs.

3.1.1.1 Cross-Section Concept

The use of one inch span cross-sections provide a valuable iterative step in the overall design process leading to an optimum design. Advantages of the cross-section methodology are as follows: (a) the cross-section sketches provide a work sheet for integrating element concepts into workable wing designs; (b) by considering only one inch length a large variety of designs can be economically weighed, costed, and evaluated; and (c) by configuring all concepts

at a specific wing station and to the identical static load, fatigue spectrum, and fracture criteria that occurs at that station, a meaningful and equitable evaluation will result.

The initial design effort was sizing and defining applicable concepts from previous programs as one inch span cross-section analytical specimens with geometry and load taken at Load Reference Axis (LRA) Sta. 140.0. Innovative effort produced several new concepts which were also sized and defined as one inch span cross-section specimens. The one inch span cross-section concepts were weighed, costed and scored (see paragraph 3.2.1).

3.1.1.2 Analytical Assembly Design

Advantages of the analytical assembly designs as a preliminary design tool are as follows:

- o Provides on a single drawing a definition of the configuration and the critical numerical values that measure weight, cost, strength/stress levels, fatigue quality, damage tolerance and overall desirability of a design concept.
- o Serves as an instrument to coordinate the technical efforts of the various disciplines necessary to optimize and effect a complex design concept.
- o Promotes concept reiteration by showing up promising elements of one assembly that can be comined with promising elements from other assemblies to provide a new assembly with the best collection of elements.
- o The concept data block provides detailed evaluation data on each configuration of each part or element, such as a spar or skin, for better visibility as to which element is the principal driver for cost, weight, etc.
- o Provides valid data for evaluating a number of design concepts to specified design parameters on a completely uniform, equitable basis.

From the one inch cross-section concepts, fourteen metallic concepts and five composite concepts were selected for definition on analytical assembly drawings. The analytical assembly models

are 48 inch span constant section wing box assemblies designed to the geometry and loads that occur at LRA Sta. 140.0. These assemblies were sized to static strength, fatigue and fracture allowable stresses. Weights and costs were computed and other parameters evaluated. Scores and rank were computed for each concept in accordance with the AFFDL merit rating system (see paragraph 3.2.2).

3.1.1.3 Preliminary Design

After having gone through the screening of the cross-section concepts and the analytical assembly designs, the next step is to develop preliminary designs of the highest ranking analytical assemblies. These drawings are much more complex and complete than the previous drawings. The advantage of the methodology has its greatest pay-off at this point in that the concepts reviewed at this stage are all high potential ideas. The time required to develop the preliminary design for ideas that have limited potential has been avoided by the screening process. Also, the possibility that a good idea will emerge late in the program has been minimized because of the large number of concepts evaluated in the system.

Two of the most promising metallic concepts and one composite concept were selected and developed into complete wing box preliminary designs capable of meeting all specified criteria. Weights, and costs were computed and other parameters evaluated. Scores and rank were computed for each design (see paragraph 3.2.3).

3.1.2 Merit Rating System

The design parameters that were evaluated and the "weighing" value applied to each parameter in computing a weighted score was specified by AFFDL. The merit rating system is shown in Table I.

3.1.3 Evaluation and Ranking of Concepts

An important part of the design methodology used in this program was the evaluation and ranking of design concepts via a formal rating system. The objective of the rating system is to minimize personal opinion and to ensure that each area of responsibility has an opportunity to influence the design configuration chosen for the production effort.

The basic elements of the rating system are shown in Table I. The approach used to implement this system is discussed in the following paragraphs.

3.1.3.1 Structural Efficiency (40%)

Two parameters were used to evaluate the structural efficiency of a concept; cost and weight, sixty percent of the structural efficiency score is assigned to cost (24% of total ranking) and forty percent is assigned to the weight (16% of total ranking). Use of these parameters is discussed below:

Cost - Cost was computed for each concept by estimating and summing the material cost, the tooling recurring cost, and the fabrication costs. The cost score recorded in the Evaluation Summaries are:

$$\text{Cost Score} = \frac{\text{Cost of lowest cost concept}}{\text{Cost of concept being scored}} \times .24$$

Weight - Weight was computed for each concept which has been sized to the controlling criteria of static loads, fatigue, or fracture. The weight score recorded in Evaluation Summaries are:

$$\text{Weight Score} = \frac{\text{Weight of lightest concept}}{\text{Weight of concept being scored}} \times .16$$

TABLE I MERIT RATING SYSTEM

STRUCTURAL EFFICIENCY (.4)	TECHNOLOGY IMPROVEMENT (.1)	DAMAGE TOLERANCES (.2)	ABILITIES (.3)
COST (.6)	CONCEPTS (.5)	SAFE CRACK (.4)	INSPECTABILITY (.2)
WEIGHT (.4)	MATERIALS (.2)	FAIL SAFE (.6)	MANUFACTURABILITY (.2)
	MFG. (.3)		MAINTAINABILITY (.2)
			REPAIRABILITY (.2)
			*DURABILITY (.2)

*DURABILITY IS DEFINED AS THE ABILITY TO RESIST FATIGUE, THERMAL DEGRADATION, CORROSION, STRESS CORROSION, HYDROGEN CRACKING, WEAR, AND FOREIGN OBJECT DAMAGE -- MINIMIZES REPAIR AND MAINTENANCE.

3.1.3.2 Technology Improvement (10%)

The weighted technology improvement score is made up of the sum of the weighted scores from Concepts, Manufacturing Technology and Materials Technology. The weighted scores for each of the three technology parameters is defined below:

Concept Technology - The weighted concept technology score was computed for each concept by counting the number of innovations embodied in each concept and rationing the scores such that the highest ranking score equals .05.

Manufacturing Technology - The weighted score for manufacturing technology was computed by scoring the concept from 0% to 100% on the degree to which it will advance manufacturing technology and multiplying the percent value by .03.

Materials Technology - The weighted score for materials technology was determined by identifying the number of new materials and processes used in a concept and rationing the scores such that the concept using the greatest number of new materials and processes receives, the highest score of .02.

3.1.3.3 Damage Tolerances (20%)

The parameters assessed during this portion of the rating system were safe crack growth and fail safe characteristics as discussed below:

Safe Crack - Safe crack is interpreted as referring to the maximum stress in the fatigue stress spectrum consistent with stable crack growth. Each design concept was analyzed for cracks starting at both surface flaws and at holes (unless the concept was free of holes). There are four damage tolerance categories: (1) fail-safe, hole free structure; (2) fail-safe structure with holes; (3) slow crack growth (not fail-safe) structure; and (4) slow crack growth structure with holes.

The critical crack growth stress level, F_{cr} , is controlled by the damage tolerance category and the type

of material in accordance with MIL-STD-1530. The ratio of the critical crack growth stress to the maximum static tension stress is considered a measure of excess damage tolerance capability. The ratios are then divided by the maximum such ratio and multiplied by .08 to obtain the final weighted values.

Fail Safe - A count was made of the maximum number of individual structural elements in a concept that could be failed without impairing load capability. By dividing all such counts by the number in the concept with the highest number and multiplying this number by .12 the weighted score is obtained.

3.1.3.4 Abilities (30%)

The parameters that were evaluated to arrive at the abilities weighted scores are inspectability, manufacturability, repairability, and durability. All concepts were ranked on a 0% to 100% scale for each parameter by specialists in the area defined by the parameter. The score for each concept was obtained by multiplying these % numbers by .06 for the corresponding parameter.

3.2 WING BOX DESIGN

The contents of this paragraph summarizes the design work accomplished and the evaluation summary charts for this work.

3.2.1 Cross Section Iteration

Figures 5 thru 37 show the structural concepts defined as cross-section sketches. Table II, Wing Box Metallic Cross Section Concept Scoring and Ranking Summary and Table III Wing Box Composite Cross Section Concept Scoring and Ranking Summary summarizes the data used in selecting concepts for input into the analytical assembly iteration.

3.2.1.1 Metallic Cross-Section Concepts

The nineteen metallic cross-section concepts defined and evaluated embody several promising ideas to reduce cost, reduce weight, improve fatigue life and improve damage tolerance characteristics.

Alternate aluminum lower skin concepts defined and studied include the following:

- o Monolithic skin with fastener penetrations
- o Laminated lower skins with & without fastener penetrations
- o Planked (fail safe) laminated & non-planked (not fail safe).

Alternate spar concepts included several configurations to increase the allowable compressive buckling stress of the upper skin, and at the same time achieve low weight and cost for the spars. These concepts are as follows:

- o Extruded and built-up sheet metal "Y" spars
- o Corrugated sheet metal spars with wide extruded caps
- o Conventional integrally machined spars
- o Inverted "A" sheet metal spars
- o Slanting sheet metal spar web having stabilizing inter-costals with two upper extruded caps.

The upper wing box skin configurations included:

- o Machine pocketed aluminum plate
- o Non-pocketed aluminum plate
- o Aluminum honey comb sandwich panel.

The concepts showing the best total score (scoring cost at a weighting value of .60 and weight at a weighting value of .40) were as follows:

- o Laminated aluminum sheet lower skins without fastener penetrations where these skins are planked to fall into the "fail safe category of MIL-A-83444.
- o Spar designs that provide a broad area of support for the upper skin member in a manner that increases the upper skin compressive buckling stress allowable over the base spar concept.
- o Configurations of sheet metal or extrusions in lieu of parts integrally machined from heavy plate. Reducing the ratio of starting material to finished material achieves a cost savings in material and fabrication.

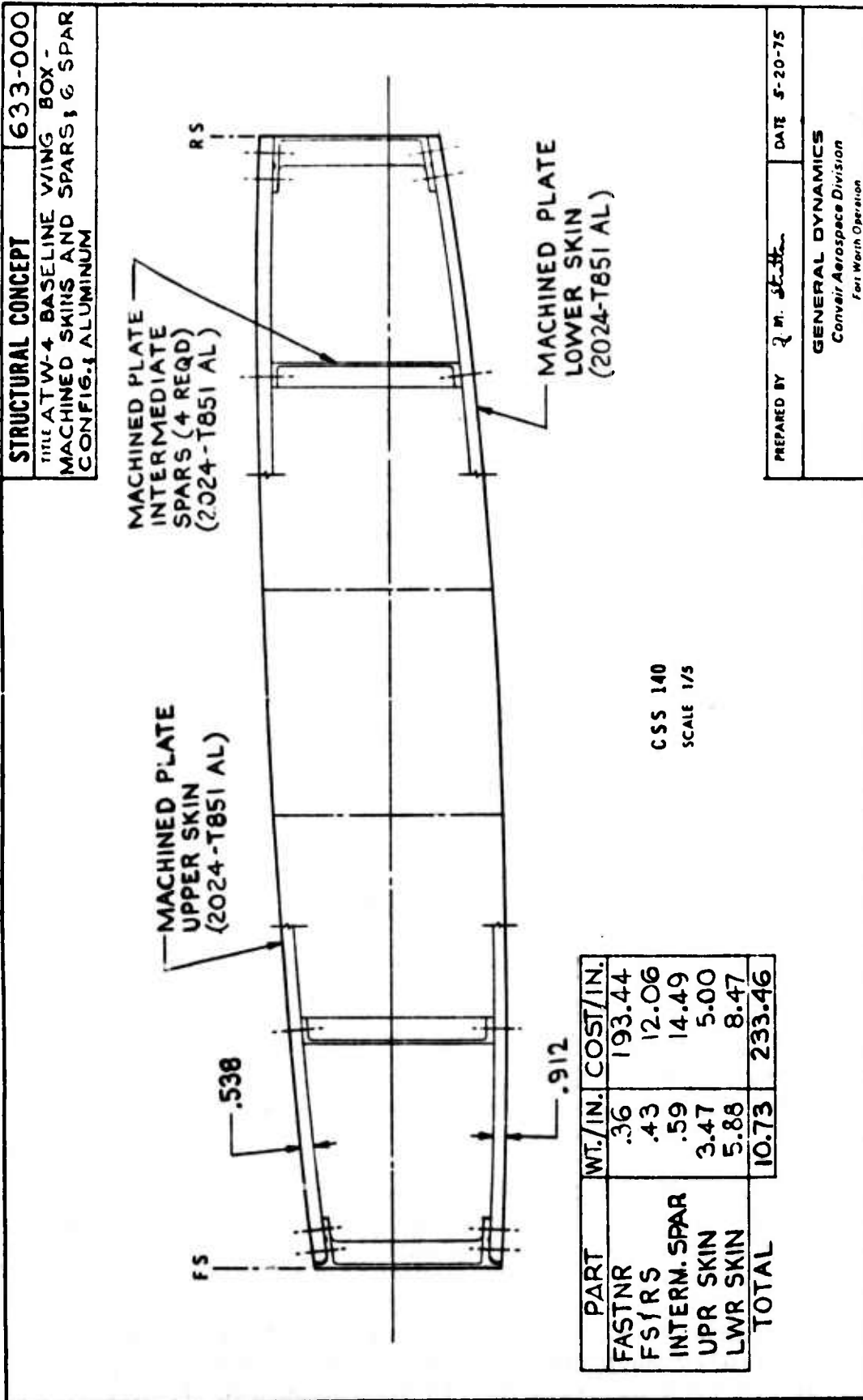


Figure 5 Six Spar Baseline

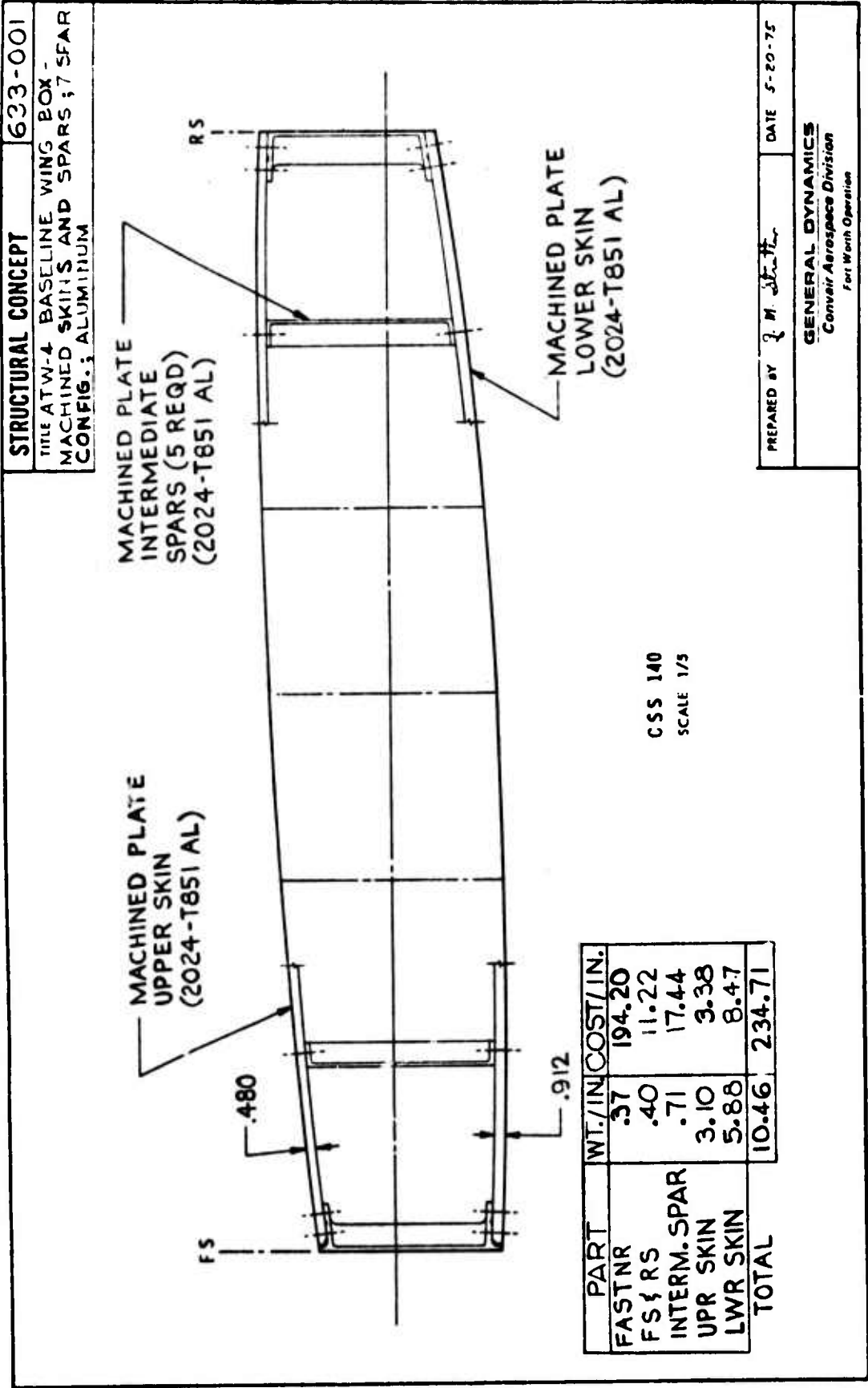


Figure 6 Seven Spar Baseline

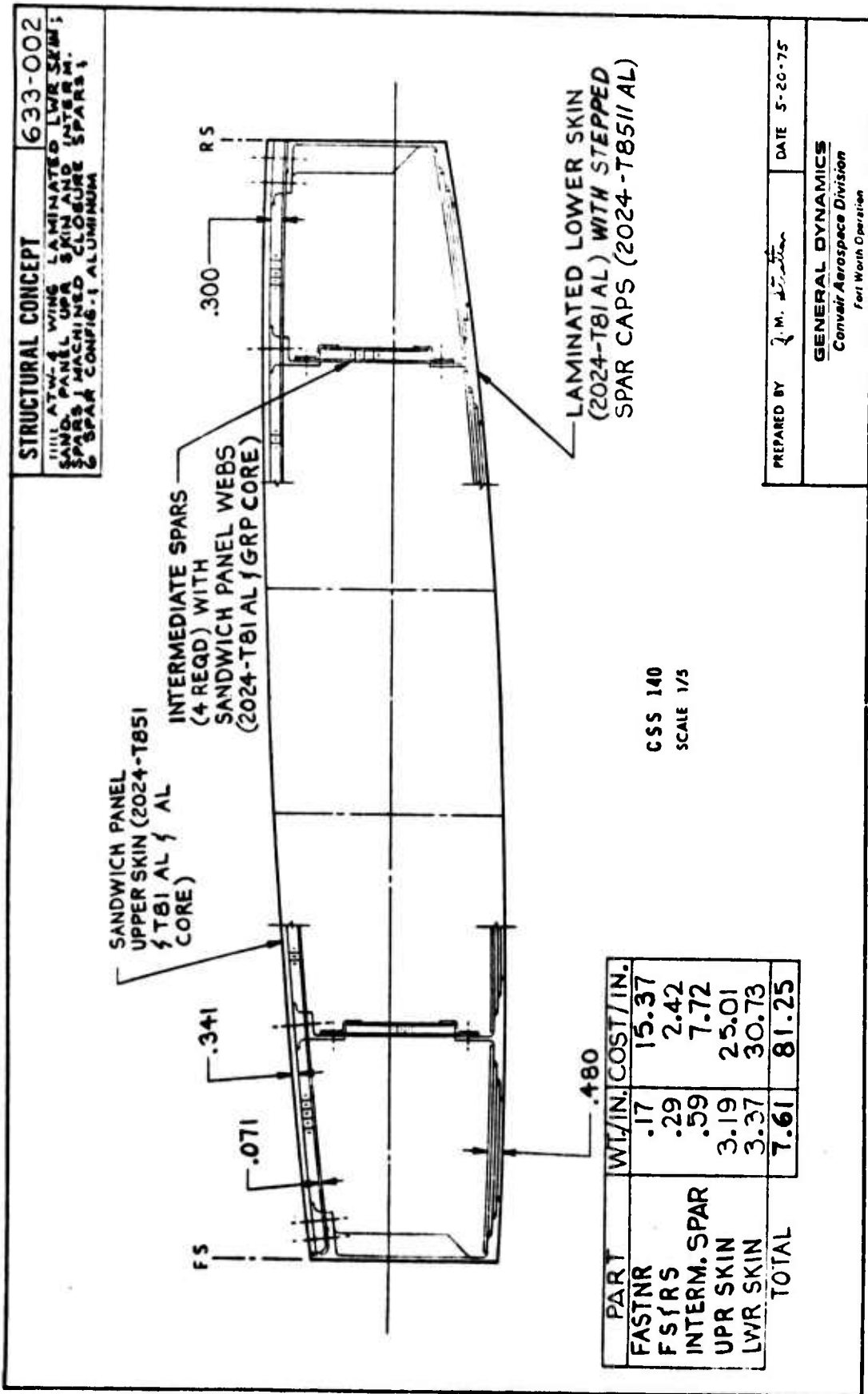
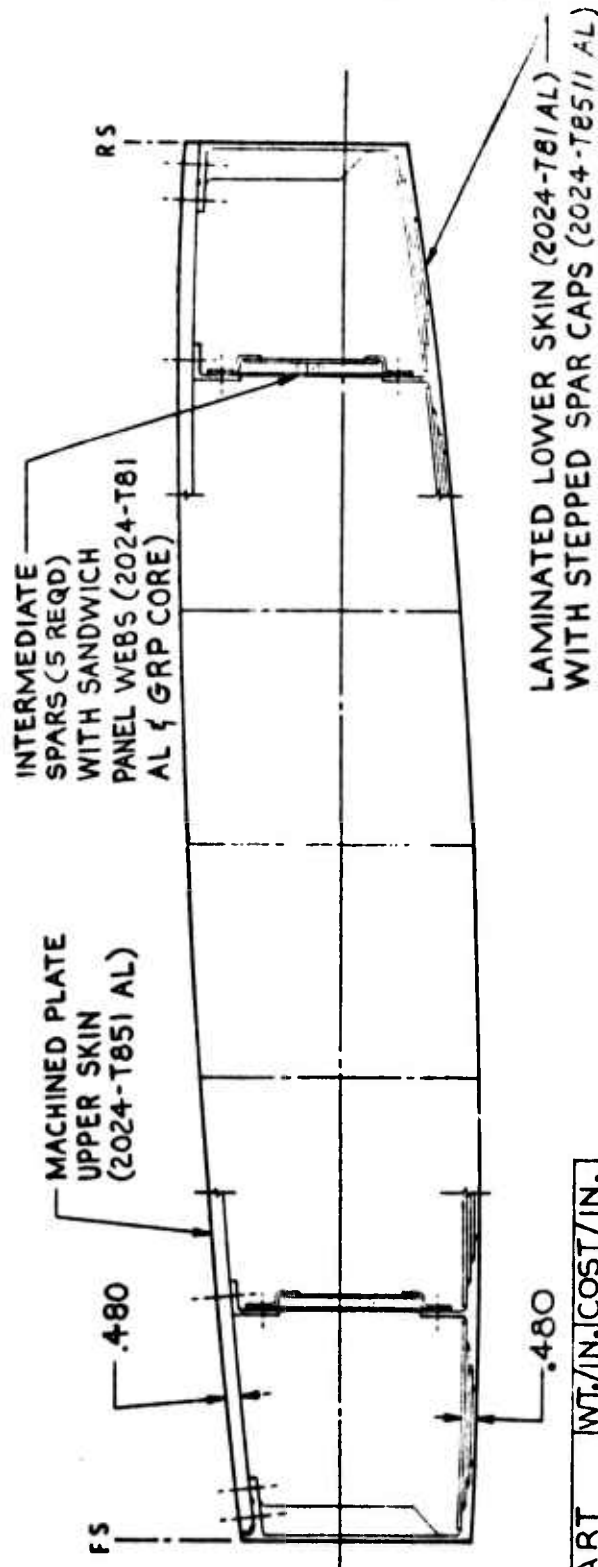


Figure 7 Honeycomb Spar and Upper Skin

STRUCTURAL CONCEPT 633-003
 THE ATW-4 WING - LAMINATED LWR SKIN;
 MACHINED UPPER SKIN & SPARS;
 SAND PANEL UPPER SPARS & 7 SPAR
 CONFIG. 3 ALUMINUM



CSS 140
 SCALE 1/5

PART	WT./IN.	COST/IN.
FASTNR	.17	16.49
FS & RS	.27	2.25
INTERM. SPAR	.73	9.56
UPR SKIN	3.10	3.38
LWR SKIN	3.36	30.64
TOTAL	7.63	62.32

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 Fort Worth, Texas

Figure 8 Honeycomb Spar Plate Upper Skin

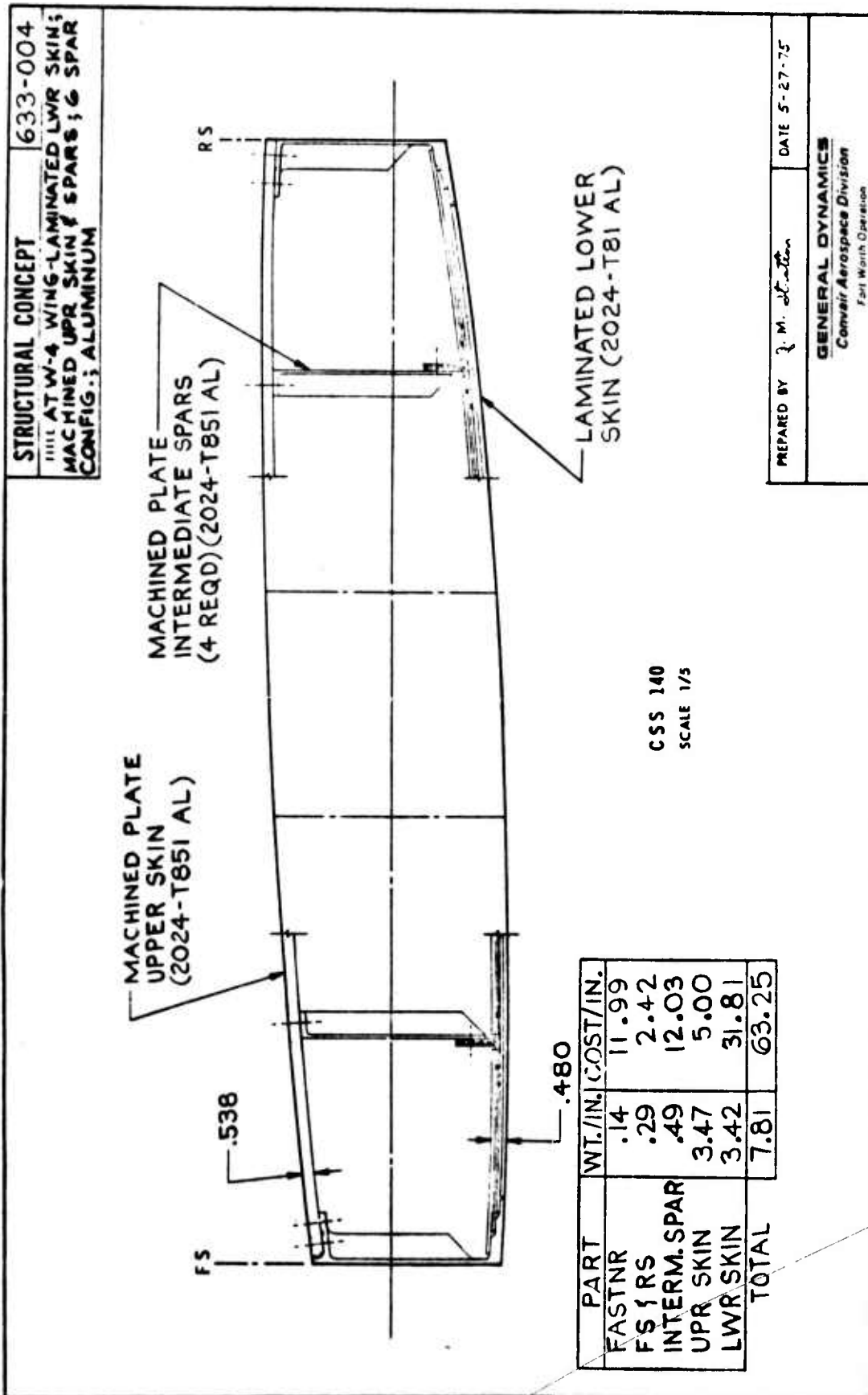


Figure 9 Machined Spars and Plate Upper Skin

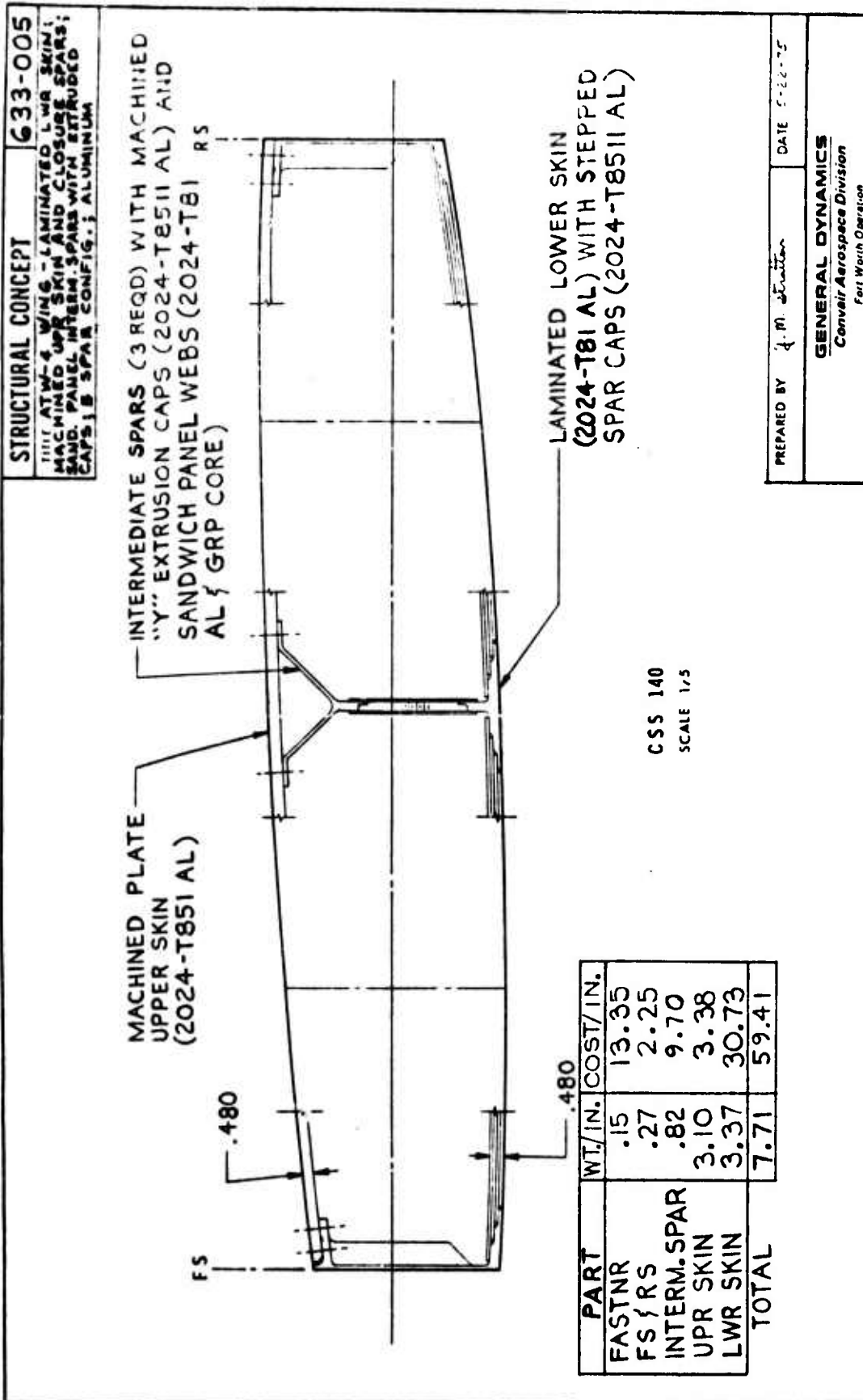


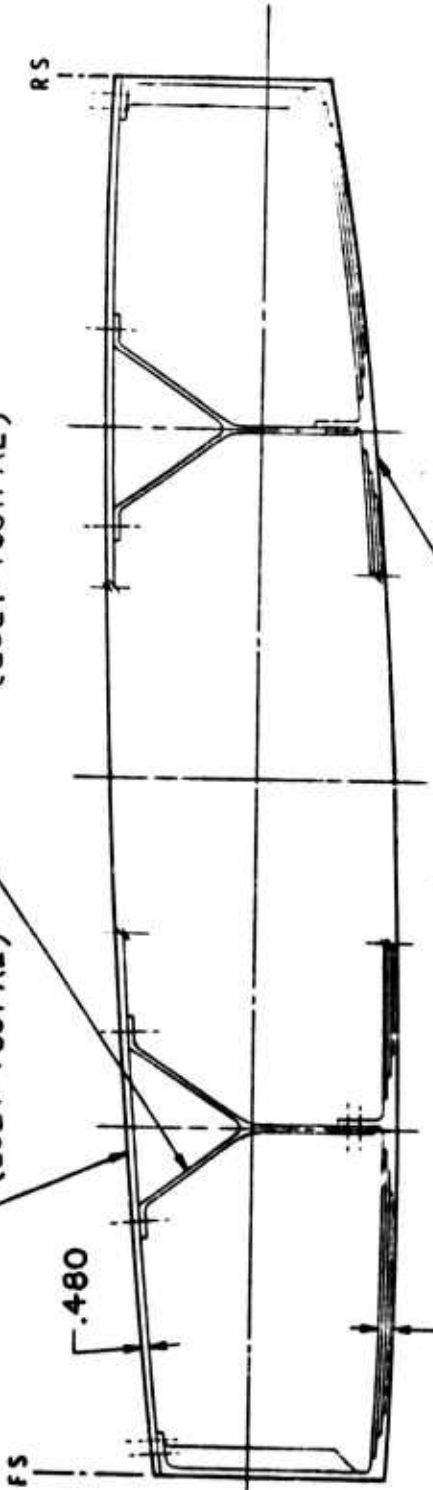
Figure 10 Honeycomb Y-Spars Plate Upper Skin

STRUCTURAL CONCEPT 633-006
 IIII ATW-4 WING- LAMINATED LWR SKIN;
 MACHINED UPR SKIN & CLOSURE SPARS;
 EXTRUDED INTERM. SPARS; S PAR;
 CONFIG. - I ALUMINUM

MACHINED "Y" EXTRUSION
 INTERMEDIATE SPARS (3 REQD)
 (2024-T8511 AL)

MACHINED PLATE
 UPPER SKIN
 (2024-T851 AL)

LAMINATED LOWER SKIN
 (2024-T81 AL) WITH STEPPED
 SPAR CAPS (2024-T8511 AL)



CSS 140
 SCALE 1/5

PART	WT./IN.	COST/IN.
FASTNR	.21	18.15
FS & RS	.27	2.25
INTERM. SPAR	1.09	2.92
UPR SKIN	3.10	3.38
LWR SKIN	3.37	30.73
TOTAL	8.04	57.43

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 Convair Aerospace Division
 Fort Worth, Oklahoma

Figure 11 Extruded Y-Spar

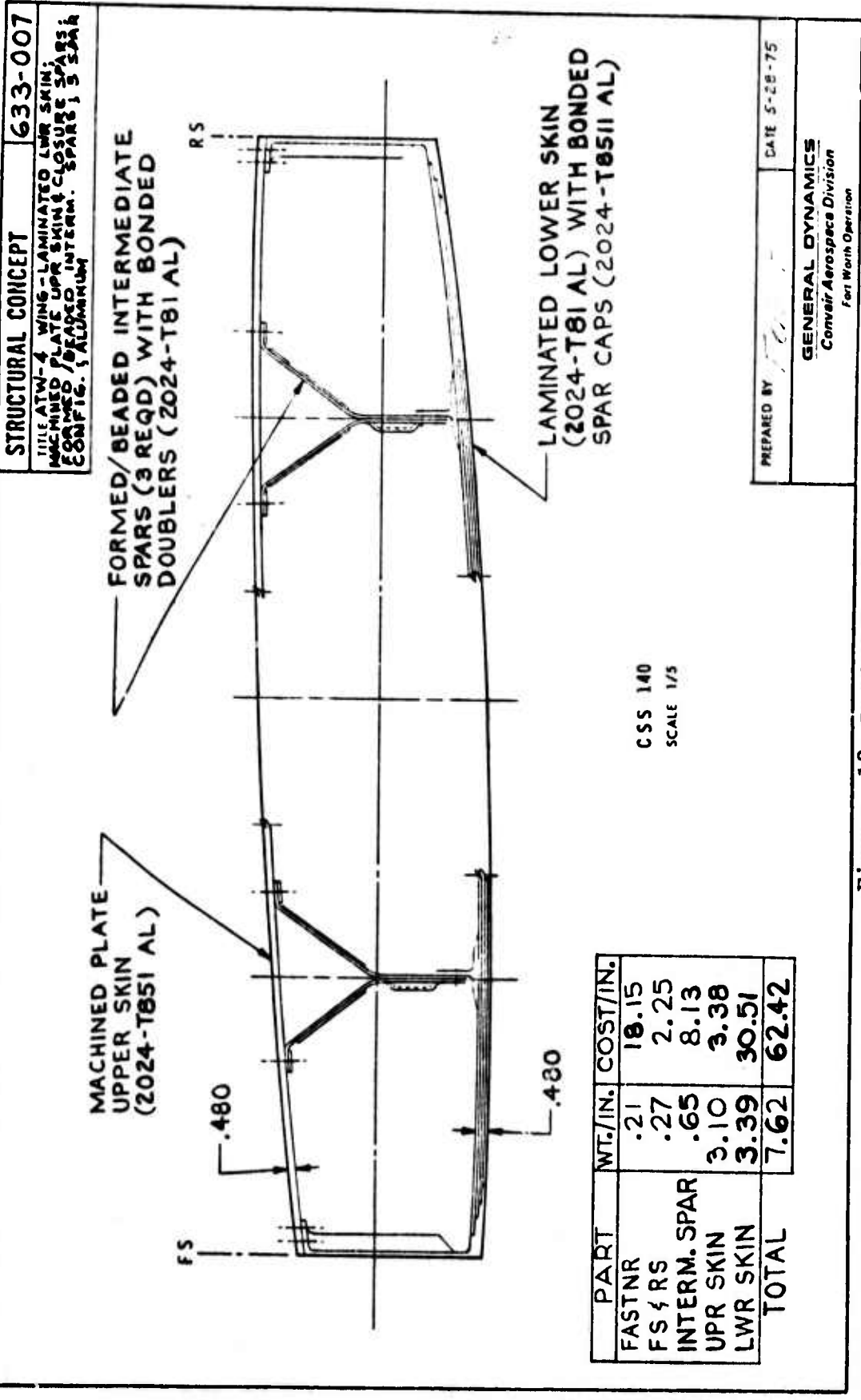


Figure 12 Beaded Y-Spar

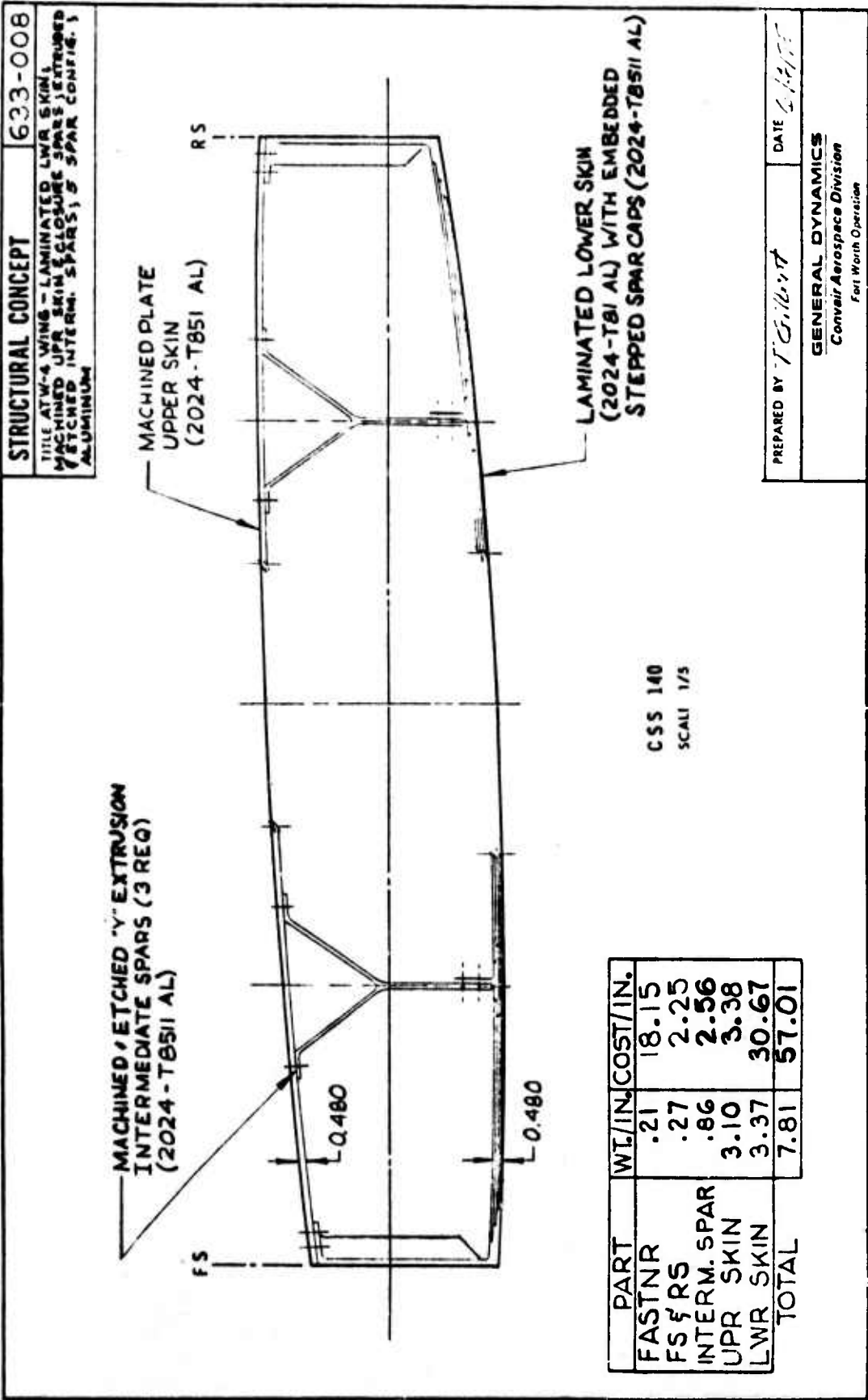
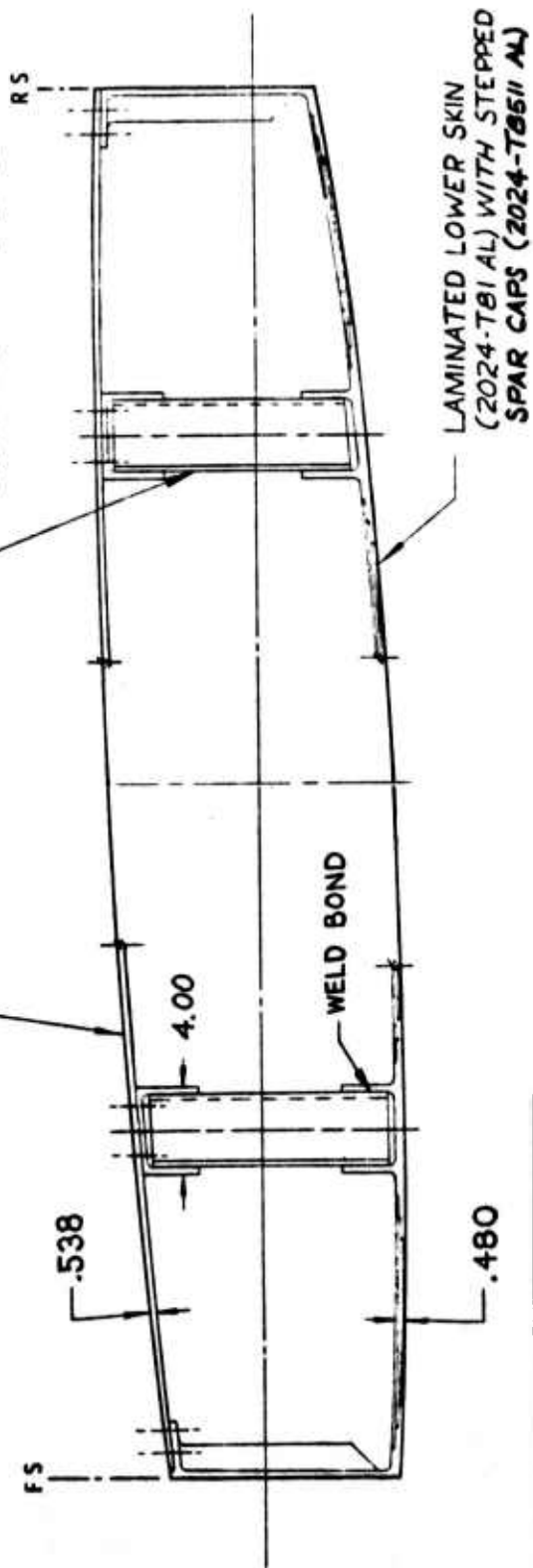


Figure 13 Extruded and Etched Y-Spar

STRUCTURAL CONCEPT 633-009

TITLE AT W-4 WING - LAMINATED LWR SKIN,
MACHINED UPR SKIN & CLOSURE SPARS,
CORRUGATED INTERM. SPARS WITH EXTRUDED
CAPS & SPAR CONFIG. 1 ALUMINUM

CORRUGATED SHEET INTERMEDIATE
SPAR WEBS (2024-T80) WITH EXTRUDED
CHANNEL CAPS (2024-T85II AL)



PART	WT./IN.	COST/IN.
FASTENER	.15	13.35
FS & RS	.27	2.25
INTERM. SPAR	.59	7.23
UPR SKIN	3.47	5.00
LWR SKIN	3.37	30.73
TOTAL	7.85	58.56

CSS 140
SCALE 1/5

PREPARED BY: [Signature]
DATE: 5-22-75
GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operations

Figure 14 Corrugated - 5 Spar

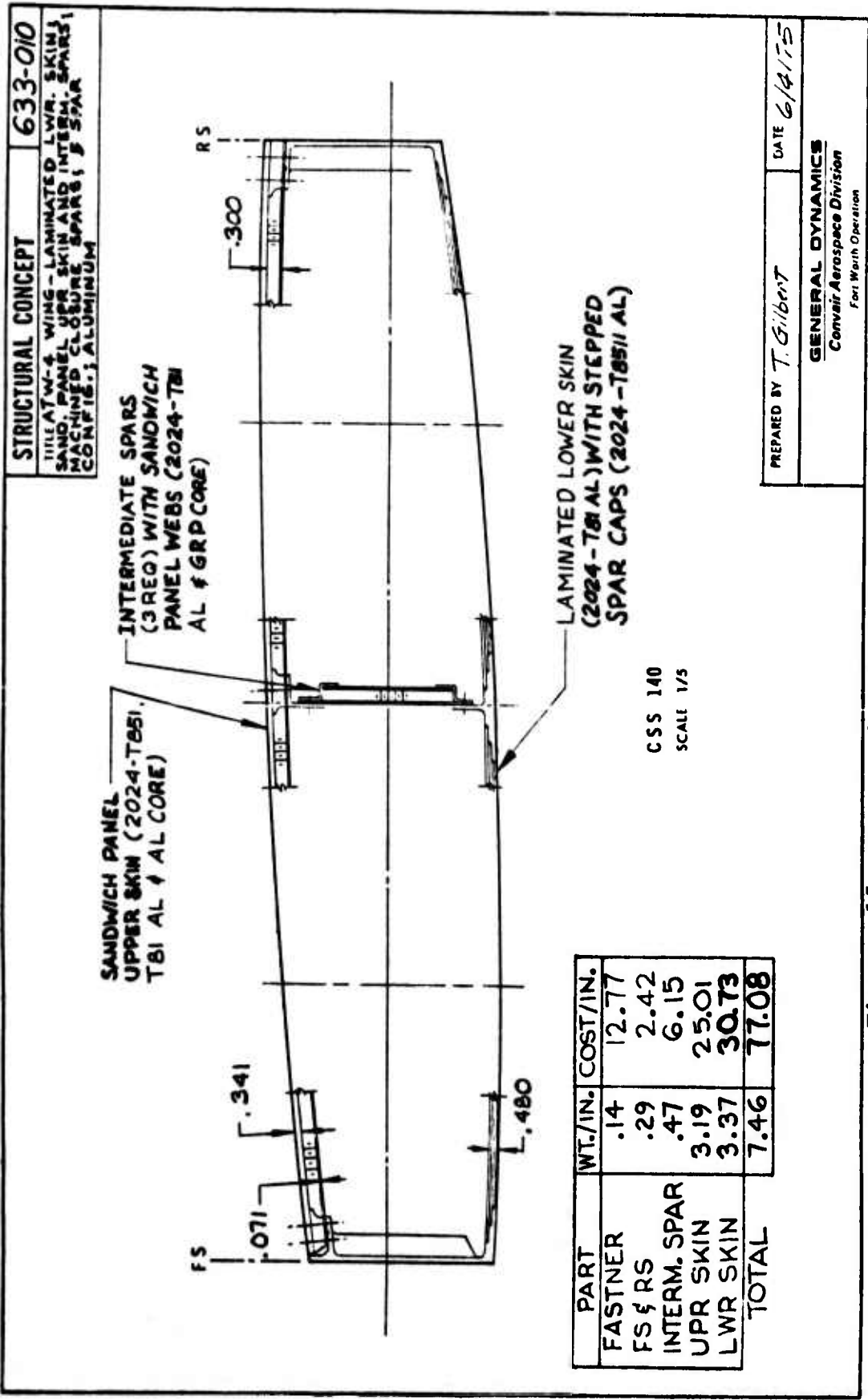
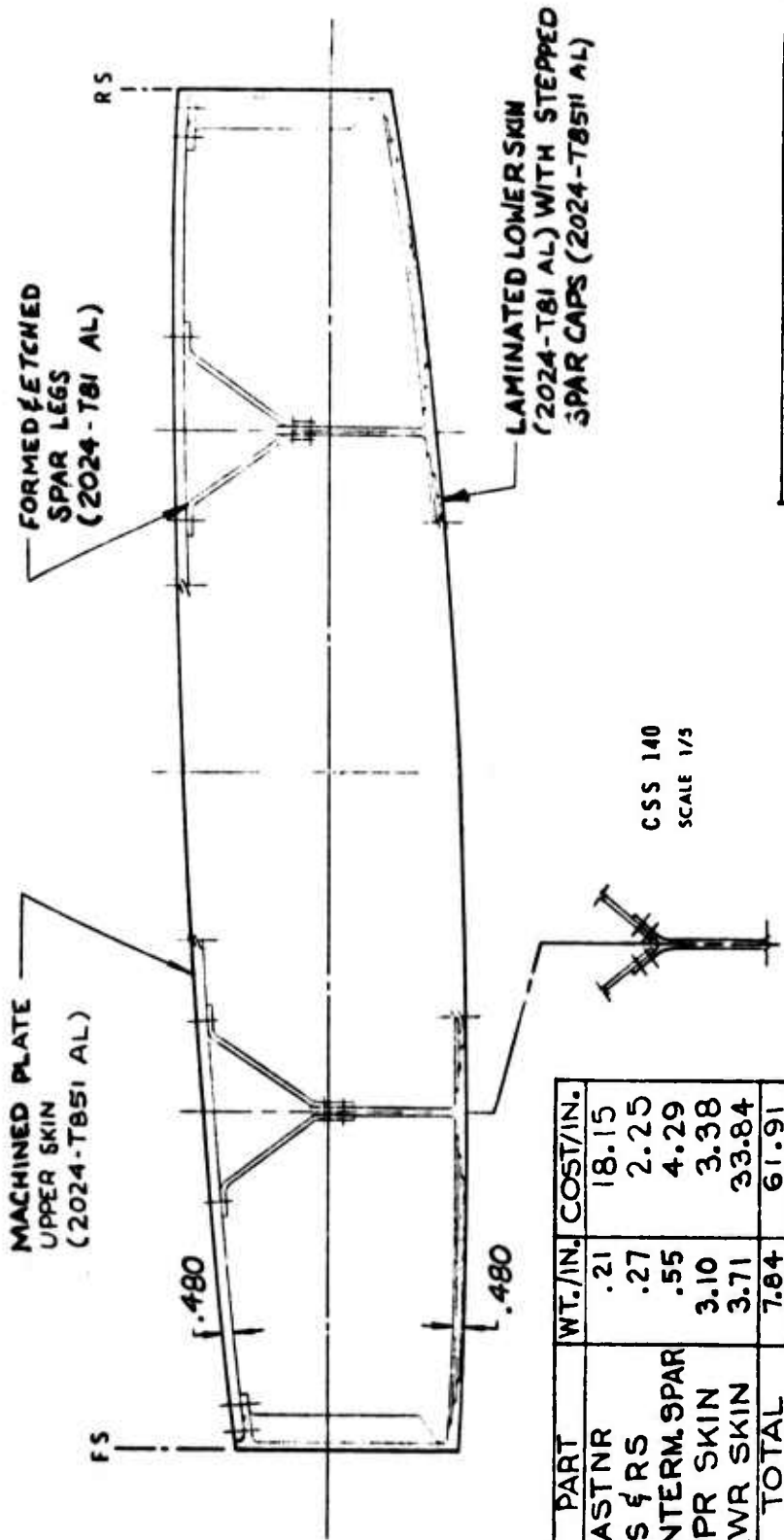


Figure 15 Honeycomb Spar & Upper Skin - 5 Spar

STRUCTURAL CONCEPT 633-011

TITLE ATW-4 WING - LAMINATED LWR SKIN,
MACHINED PLATE UPR SKIN & CLOSURE SPARS;
FORMED & ETCHED INTERM. SPARS & SPAR
CONFIG.; ALUMINIUM

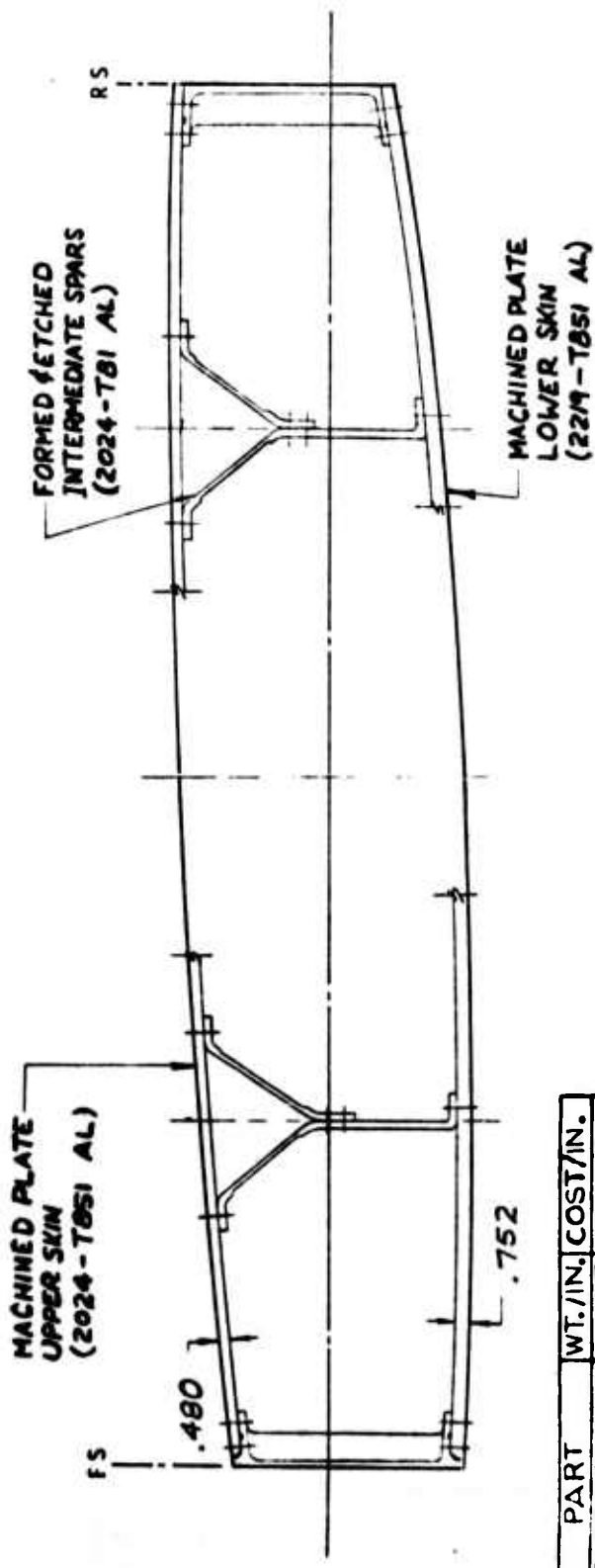


PART	WT./IN.	COST/IN.
FASTNR	.21	18.15
FS & RS	.27	2.25
INTERM. SPAR	.55	4.29
UPR SKIN	3.10	3.38
LWR SKIN	3.71	33.84
TOTAL	7.84	61.91

PREPARED BY *F. G. ...* DATE 2/14/55
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 Fort Worth Operation

Figure 16 Formed & Etched Y-Spar

STRUCTURAL CONCEPT 633-012
 TITLE ATW-4 WING - MACHINED UPR & LWR SKINS AND CLOSURE SPARS; FORMED & ETCHED INTERM. SPARS; 5 SPAR CONFIG.; ALUMINUM



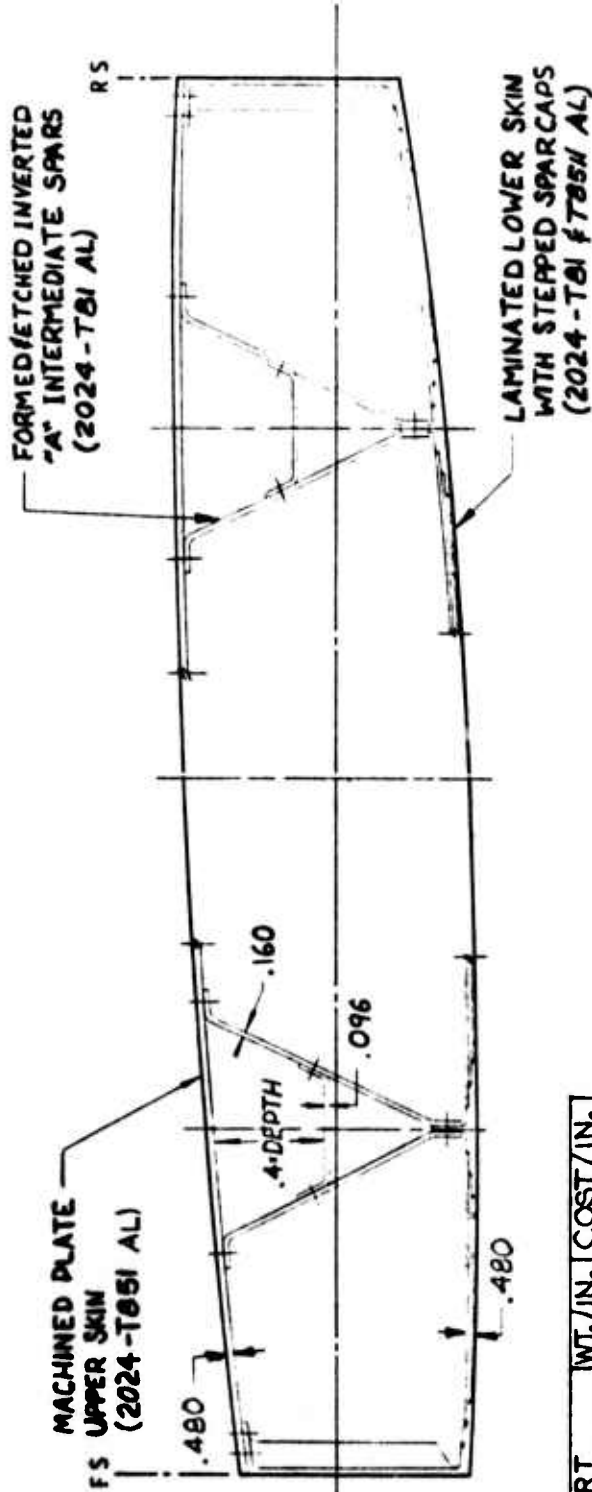
PART	WT./IN.	COST/IN.
FASTNR	.45	200.84
FS & RS	.40	11.22
INTERM. SPAR	1.03	10.30
UPR SKIN	3.10	3.38
LWR SKIN	4.85	6.98
TOTAL	9.83	232.72

CSS 140
 SCALE 1/5

PREPARED BY *T. Gilbert* DATE 6/5/75
GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 17 Formed & Etched 2 Piece Y-Spar

STRUCTURAL CONCEPT **633-013**
 FULL WING - LAMINATED LWR SKIN;
 MACHINED UPPER SKIN & CLOSURE SPARS;
 FORMED & STAGED INVERTED "A" INTERM.
 SPARS; S SPAR CONFIG.; ALUMINUM



PART	WT./IN.	COST/IN.
FASTNR	.19	18.33
FS & RS	.27	2.25
INTERM. SPAR	1.45	15.95
UPR SKIN	3.10	3.38
LWR SKIN	3.37	30.73
TOTAL	6.38	70.64

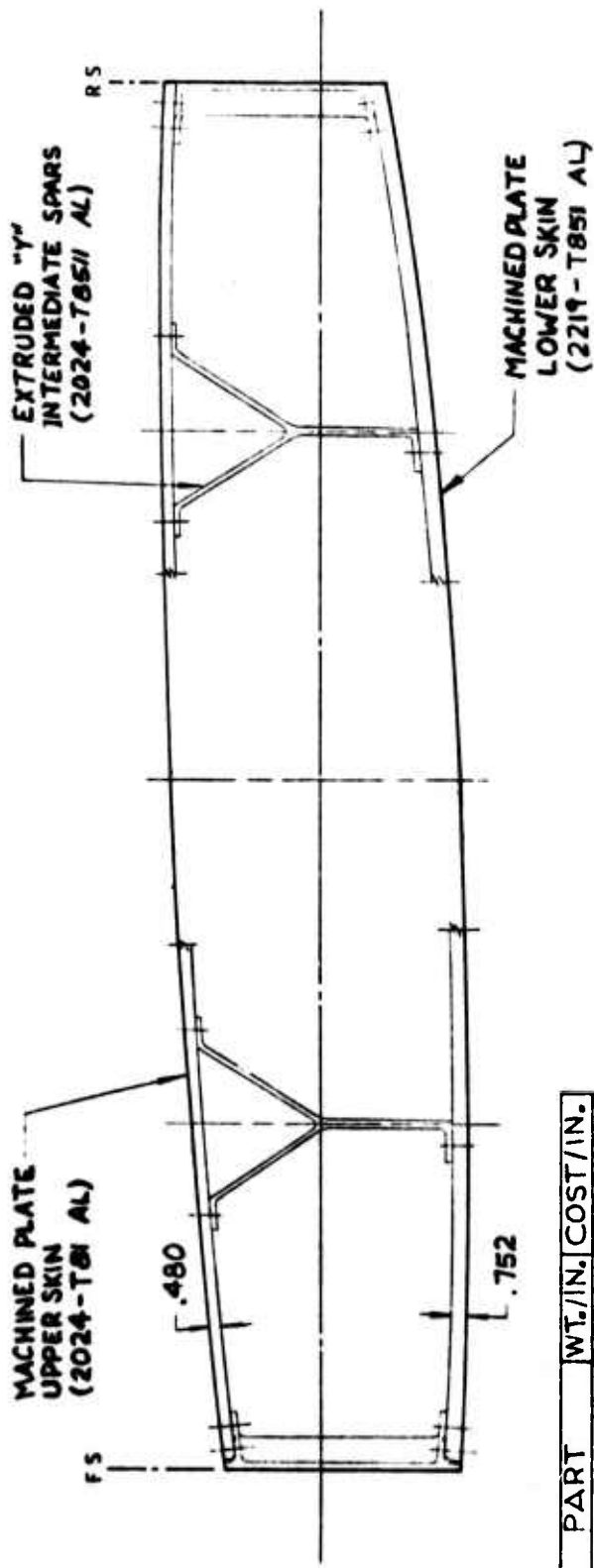
CSS 140
 SCALE 1/5

PREPARED BY / 10-6-77 DATE 6/5/77

GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 18 Inverted "A" Spar

STRUCTURAL CONCEPT **633-014**
 TITLE: ATW-4 WING - MACHINED SKINS & CLOSURE SPARS; EXTRUDED INTERM. SPARS; 5 SPAR CONFIG.; ALUMINUM



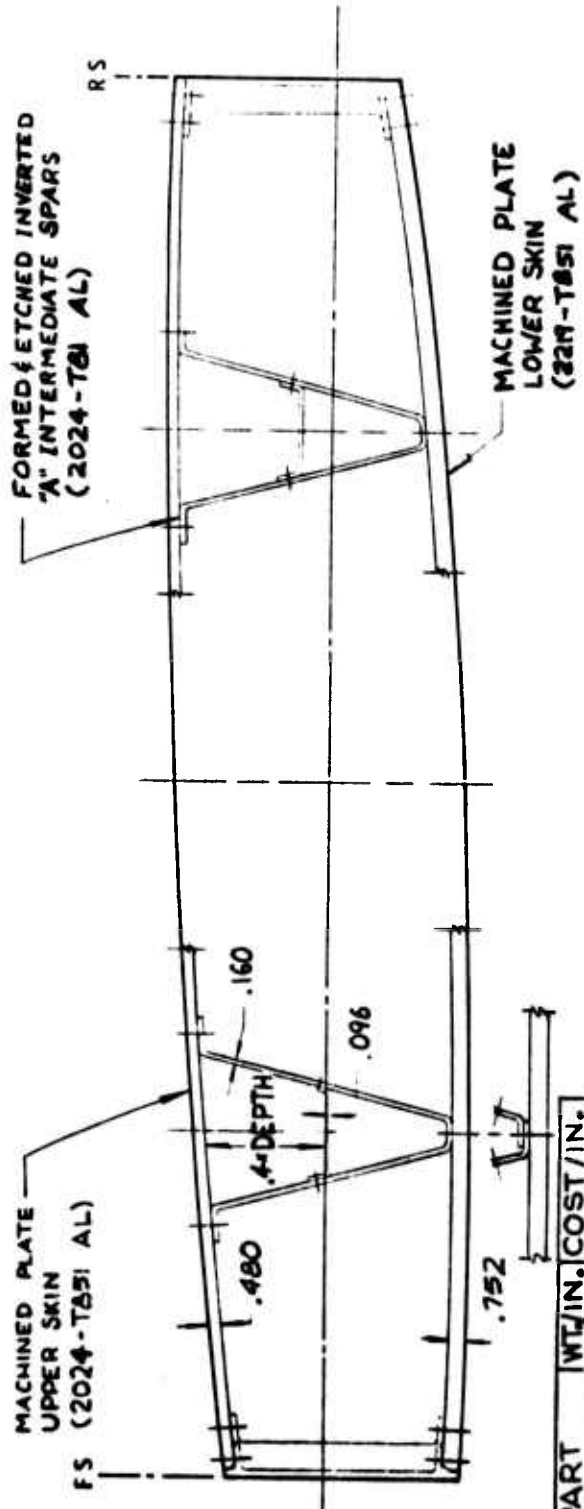
PART	WT./IN.	COST/IN.
FASTNR	.39	196.04
FS & RS	.40	11.22
INTERM. SPAR	1.12	3.00
UPR SKIN	3.10	3.38
LWR SKIN	4.85	6.98
TOTAL	9.86	220.62

CSS 140
SCALE 1/5

PREPARED BY *T. Gilbert* DATE *6/5/72*
GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 19 Extruded & Machined Y-Spar

STRUCTURAL CONCEPT 633-015
 WING A/W-4 WING MACHINED UPR & LWR
 SPARS AND CLOSURE. SPARS FORMED &
 ETCHED INVERTED "A" INTERM. SPARS, 5
 SPAR CONFIG. (ALUMINUM)



PART	WT./IN.	COST/IN.
FASTNR	.41	199.78
FS & RS	.40	11.22
INTERM.SPAR	1.45	15.23
UPR SKIN	3.10	3.38
LWR SKIN	4.85	2.98
TOTAL	10.21	236.59

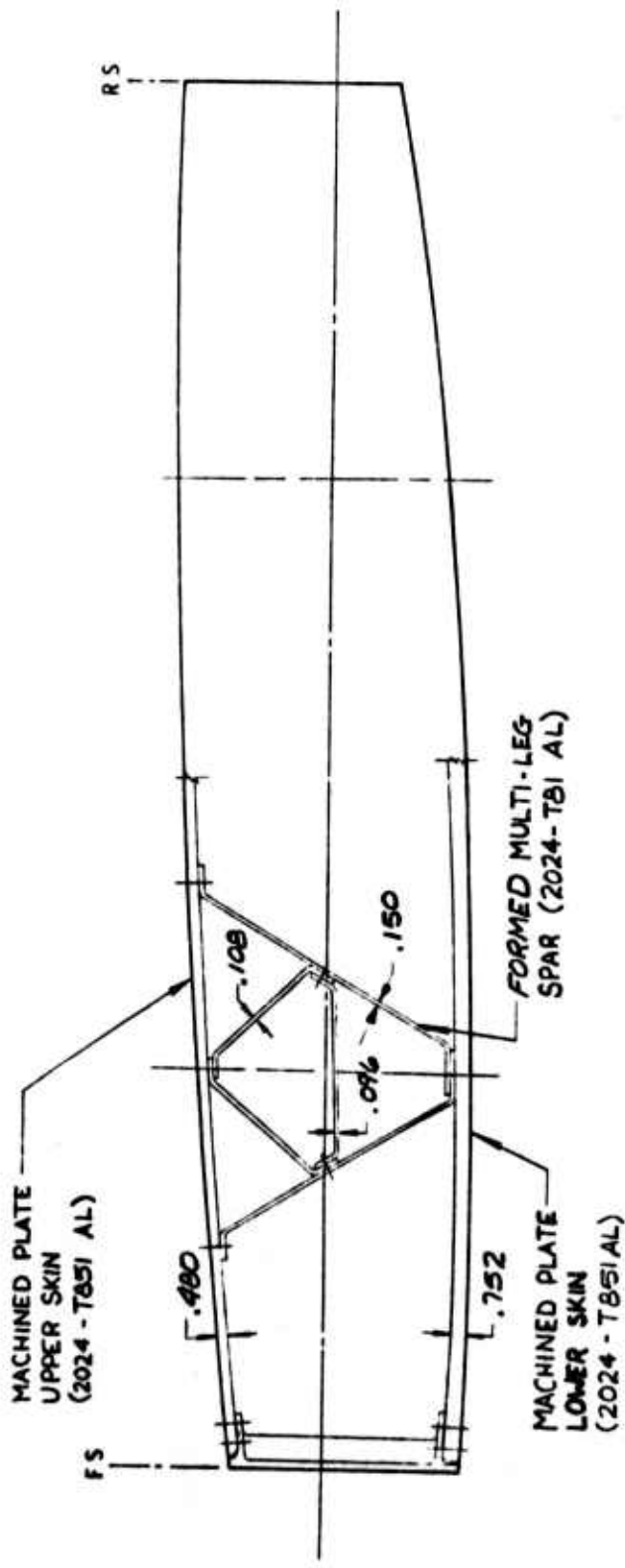
CSS 140
 SCALE 1/5

PREPARED BY T. J. ... DATE 6/5/75

GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 20 Inverted "A" Spar - Plate Skins

STRUCTURAL CONCEPT 633-016
 TITLE: ATW-4 MACHINED SKINS & CLOSURE SPARS
 1 FORMED MULTI-LEG INTERM. SPAR
 2 4 SPAR CONFIG. 1 ALUMINUM



PART	WT./IN.	COST/IN.
FASTNR	.29	116.69
FS & RS	.40	11.22
INTERM. SPAR	1.54	30.80
UPR. SKIN	3.10	3.38
LWR. SKIN	4.85	6.98
TOTAL	10.18	169.07

CSS 140
 SCALE 1/5

Figure 21 Multi-leg Spar

PREPARED BY *T. Gilbert* DATE *6/5/72*
GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

STRUCTURAL CONCEPT **633-017**

TITLE ATW-4 WING-LAMINATED LOWER SKIN;
MACHINED UPPER SKIN & CLOSURE SPARS; EXTRUDED
DIAGONAL INTERM. SPARS; SPAR CONFIG. 1;
ALUMINUM

PART	WT./IN. CO. CT/IN.
FASTNR	.27
FS & RS	.27
INTERM. SPAR	.90
UPR SKIN	3.10
LWR SKIN	3.97
TOTAL	6.51
	67.48

EXTRUDED SPAR DIAGONAL
MEMBERS (2024-T651 AL)

RS

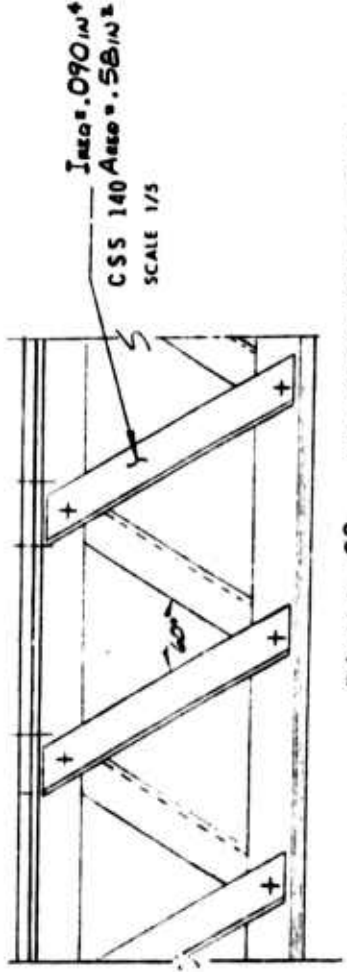
MACHINED PLATE
UPPER SKIN
(2024-T651 AL)

FS

.480

.480

LAMINATED LOWER SKIN
WITH STEPPED SPAR CAPS
(2024-T61 & T651 AL)



PREPARED BY _____ DATE 6/5/75

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth, Texas

Figure 22 Extruded Diagonal Spar

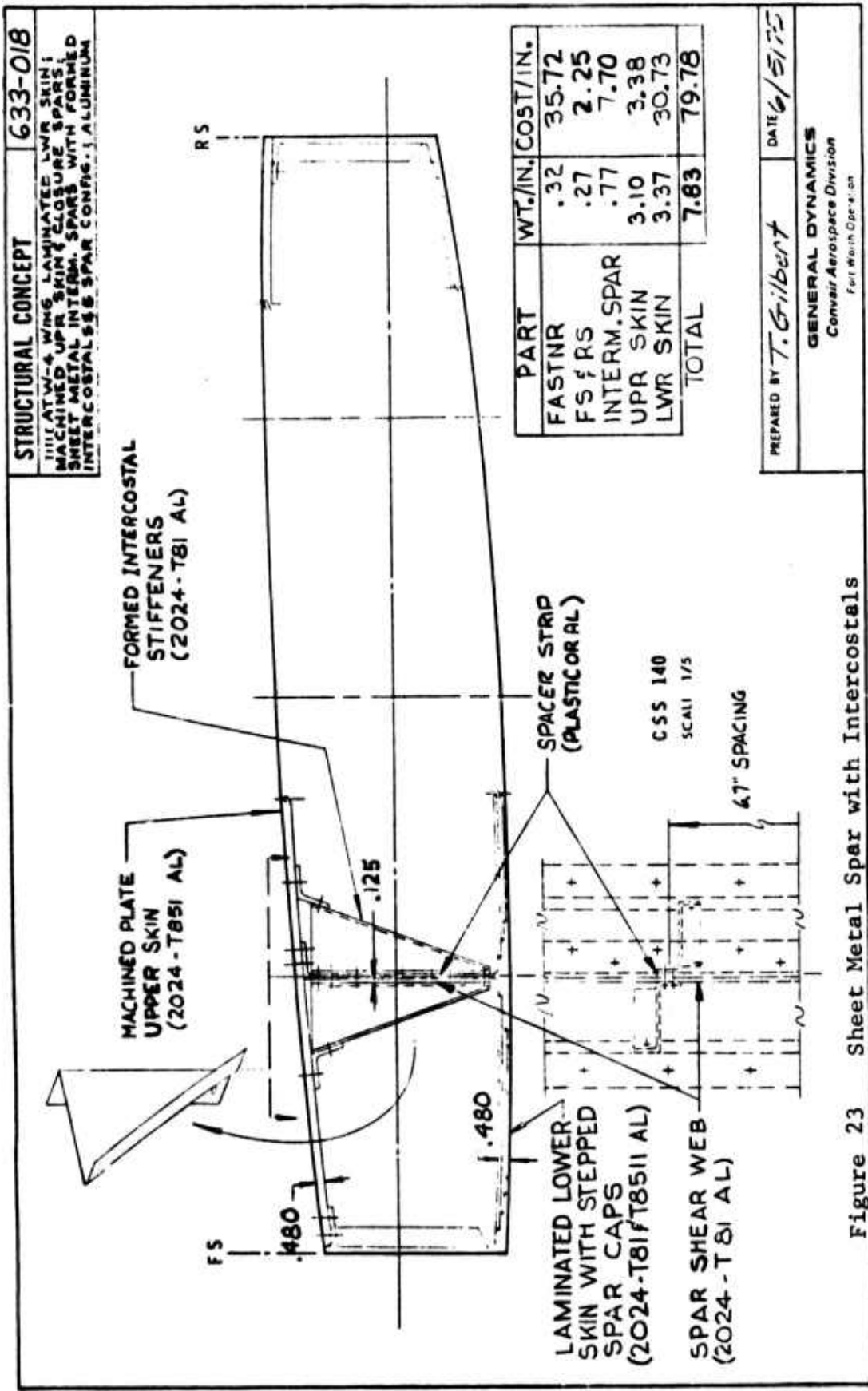


Figure 23 Sheet Metal Spar with Intercostals

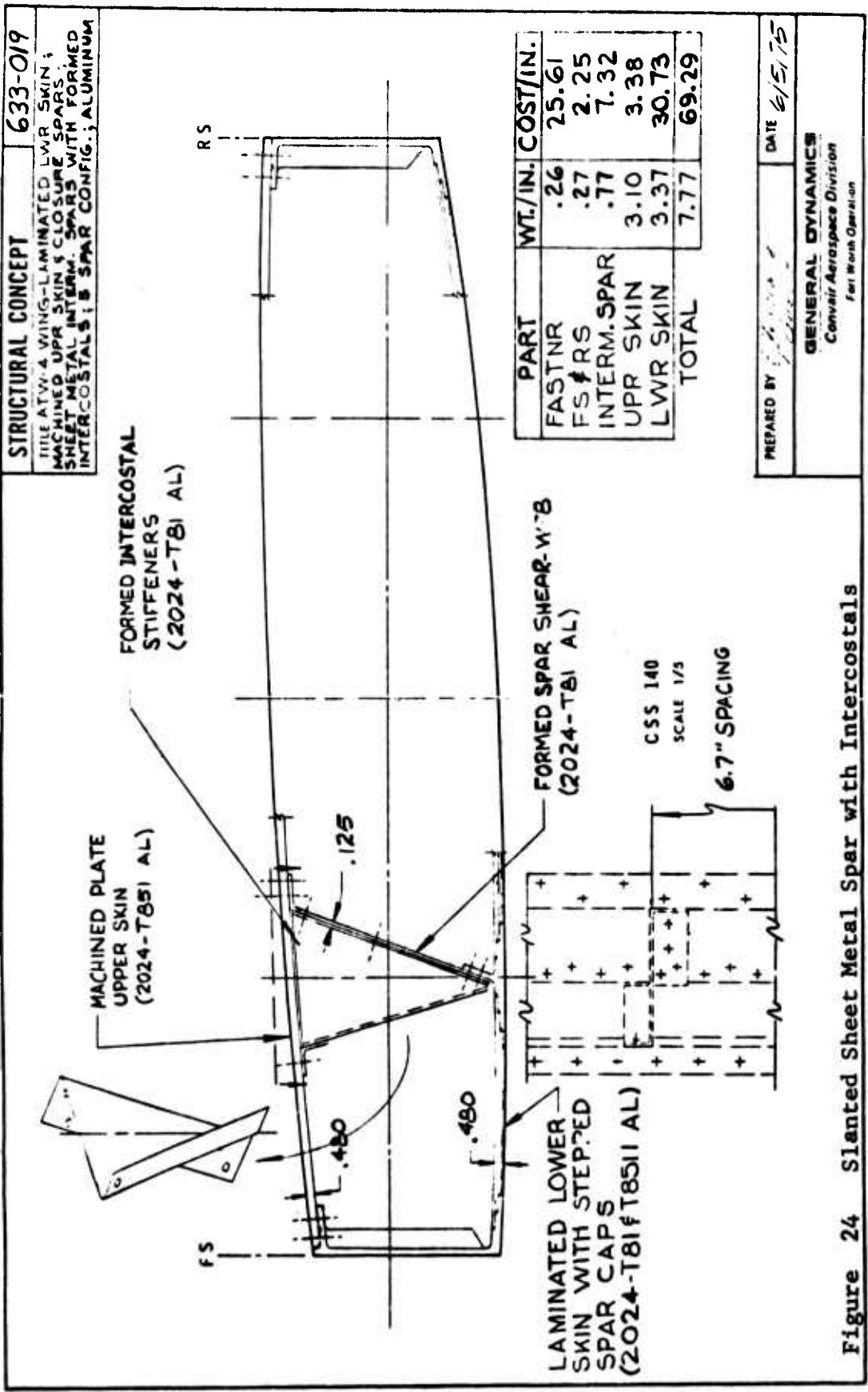
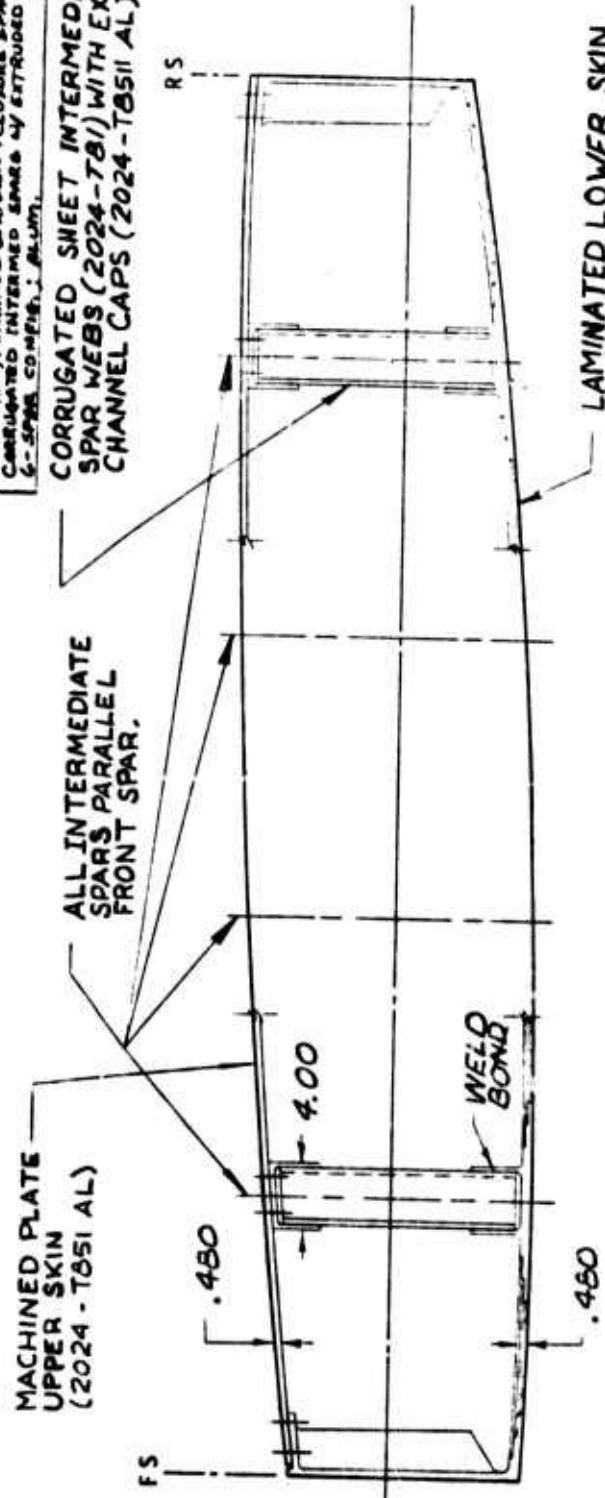


Figure 24 Slanted Sheet Metal Spar with Intercostals

STRUCTURAL CONCEPT 633-020
 TITLE ATW-4 W/F'S - PARALLEL SPAR CONCEPT. LAMINATED
 LWR SKIN; MACHINED UPR. SKIN & CLOSURE SPARS;
 CORRUGATED INTERMED. SPARS BY EXTRUDED CAPS.
 G-SPRBS. CO. #F18, J. #L-007.

CORRUGATED SHEET INTERMEDIATE SPAR WEBS (2024-T81 AL) WITH EXTRUDED CHANNEL CAPS (2024-T8511 AL).



PART	WT./IN.	COST/IN.
FASTNRS.	.18	16.02
FS & RS	.27	2.25
INTERM. SPARS	.79	9.68
UPR. SKIN	3.10	3.38
LWR. SKIN	3.37	30.73
TOTAL	7.71	62.06

CSS 140
 SCALE 1/3

PREPARED BY T. E. ... DATE 5-28-75
GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 25 Corrugated Spar

TABLE II WING BOX METALLIC CROSS SECTION CONCEPT SCORING & RANKING SUMMARY

CONCEPT	STRUCTURAL EFFICIENCY		TOTAL SCORE (1.00)	RANKING
	WEIGHT (.40)	COST (.60)		
633-000	10.73 (.278)	233.46 (.147)	.425	20
633-001	10.46 (.285)	234.71 (.146)	.431	19
633-002	7.61 (.392)	81.25 (.421)	.813	14
633-003	7.63 (.391)	62.32 (.549)	.940	5
633-004	7.81 (.382)	63.25 (.541)	.923	9
633-005	7.71 (.387)	59.41 (.576)	.963	4
633-006	8.04 (.371)	57.43 (.596)	.967	2
633-007	7.62 (.392)	62.42 (.548)	.940	5
633-008	7.81 (.382)	57.01 (.600)	.982	1
633-009	7.85 (.380)	58.56 (.584)	.964	3
633-010	7.46 (.400)	77.08 (.444)	.844	12
633-011	7.84 (.381)	61.91 (.553)	.934	8
633-012	9.83 (.304)	232.72 (.147)	.451	18
633-013	8.38 (.356)	70.64 (.484)	.840	13
633-014	9.86 (.303)	220.62 (.155)	.458	17
633-015	10.21 (.292)	236.59 (.000)	.292	21
633-016	10.18 (.293)	169.07 (.202)	.495	16
633-017	8.51 (.351)	67.48 (.507)	.858	11
633-018	7.83 (.381)	79.78 (.429)	.810	15
633-019	7.77 (.384)	69.29 (.494)	.878	10
633-020	7.71 (.387)	62.06 (.551)	.938	7

3.2.1.2 Composite Concepts

The twelve composite cross-section concepts defined and evaluated incorporate a variety of concepts that improve producibility, are economical to manufacture and that reduce weight.

Skin concepts defined and studied are as follows:

- o Planked, solid graphite epoxy layup
- o Skins with and without fastener penetrations
- o Graphite epoxy skins with and without buffer strips
- o Lower skins with integral front and rear spar
- o Intermediate lower spar caps embedded in the skin
- o Graphite epoxy sandwich skins with nomex honey comb core
- o Waffle pattern, Kevlar hat stiffened skins.

Alternate spar concepts considered were:

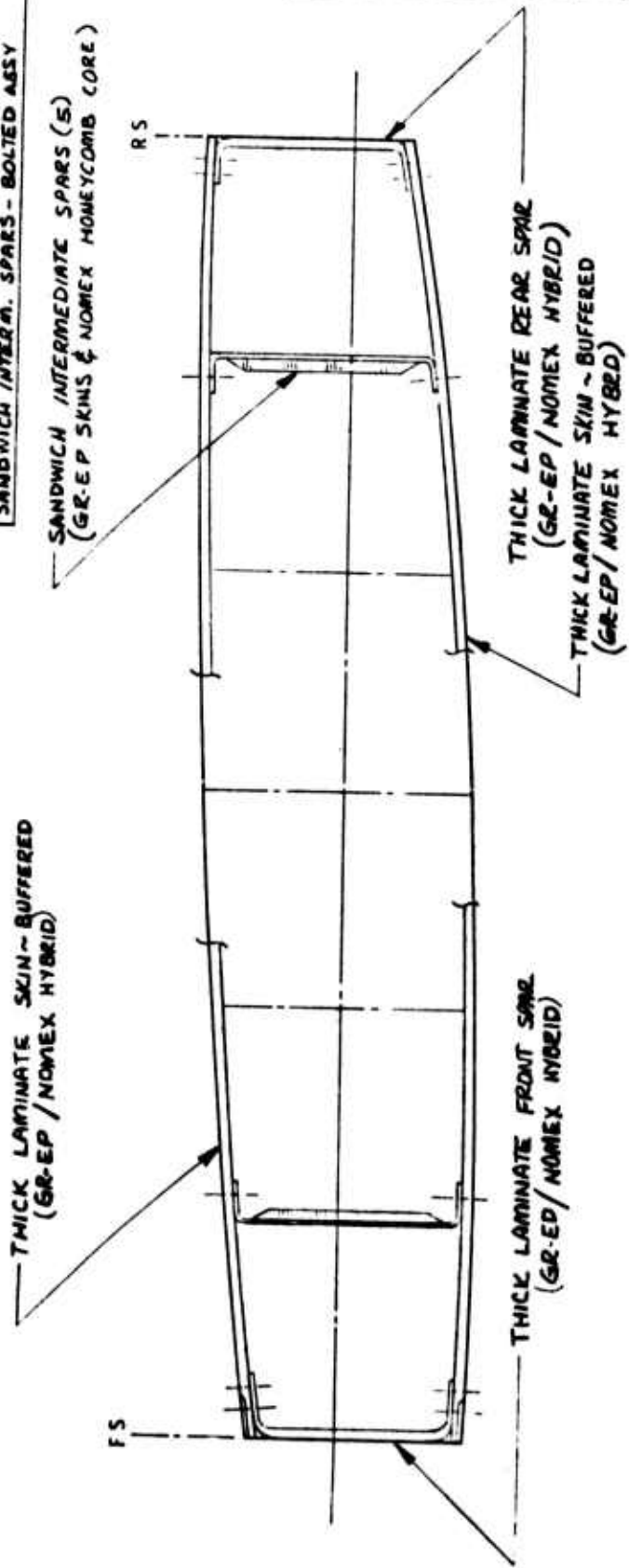
- o Sandwich intermediate spars with nomex honey comb core
- o Intermediate spars with embedded titanium lower caps
- o Sine wave spar webs
- o "Y" spars with and without solid nomex core in the webs
- o "X" spar with trussed web
- o A truss spar arrangement.

The concepts showing the most promise for low weight and cost and those selected for the composite analytical assemblies were as follows:

- o Corrugated sine wave intermediate spars with embedded lower spar cap and buffered graphite epoxy skins
- o Trussed spar arrangement with integral graphite epoxy skins

- o "Y" spars with and without embedded lower caps and solid graphite epoxy skins without buffer strips
- o Sandwich spars with lower titanium cap embedded in the lower skin. Both skins solid laminate graphite epoxy with buffer strips.

STRUCTURAL CONCEPT 633-030
 TITLE ATW-4 WING BOX-COMPOSITE
 LAMINATE SKINS & FRONT REAR SPARS
 SANDWICH INTERM. SPARS - BOLTED ASSY

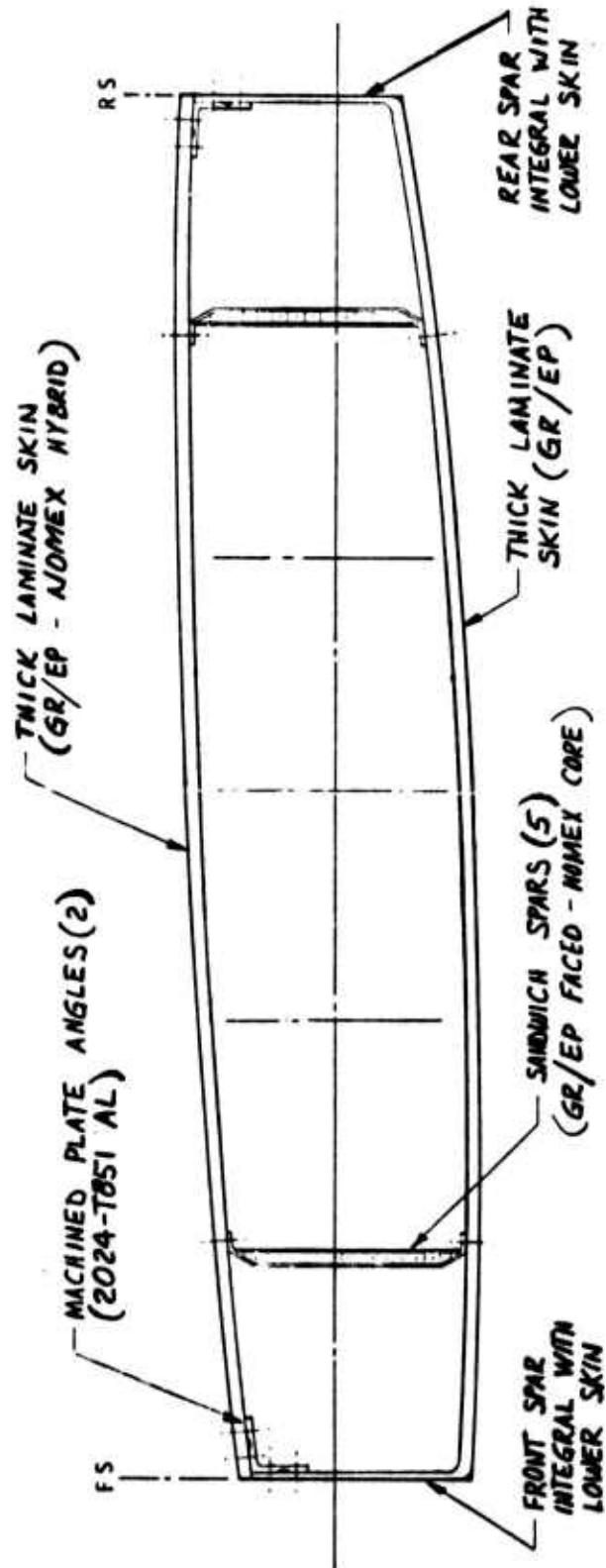


CSS 140
 SCALE 1/5

PREPARED BY RML DATE 5/25/75
 GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 26 Composite Laminated Skins & Boundary Spars

STRUCTURAL CONCEPT 633-031
 TITLE ATW-4 WING BOX - COMPOSITE
 LAMINATE SKINS & FRONT/REAR INTEGRAL SPARS;
 SANDWICH INTERM. SPARS; BOLTED ATTACHMENT



CSS 140
 SCALE 1/5

PREPARED BY J. L. H. [Signature] DATE 5-23-75

GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth, Texas

Figure 27 Composite Laminated Skins With Integral Boundary Spars

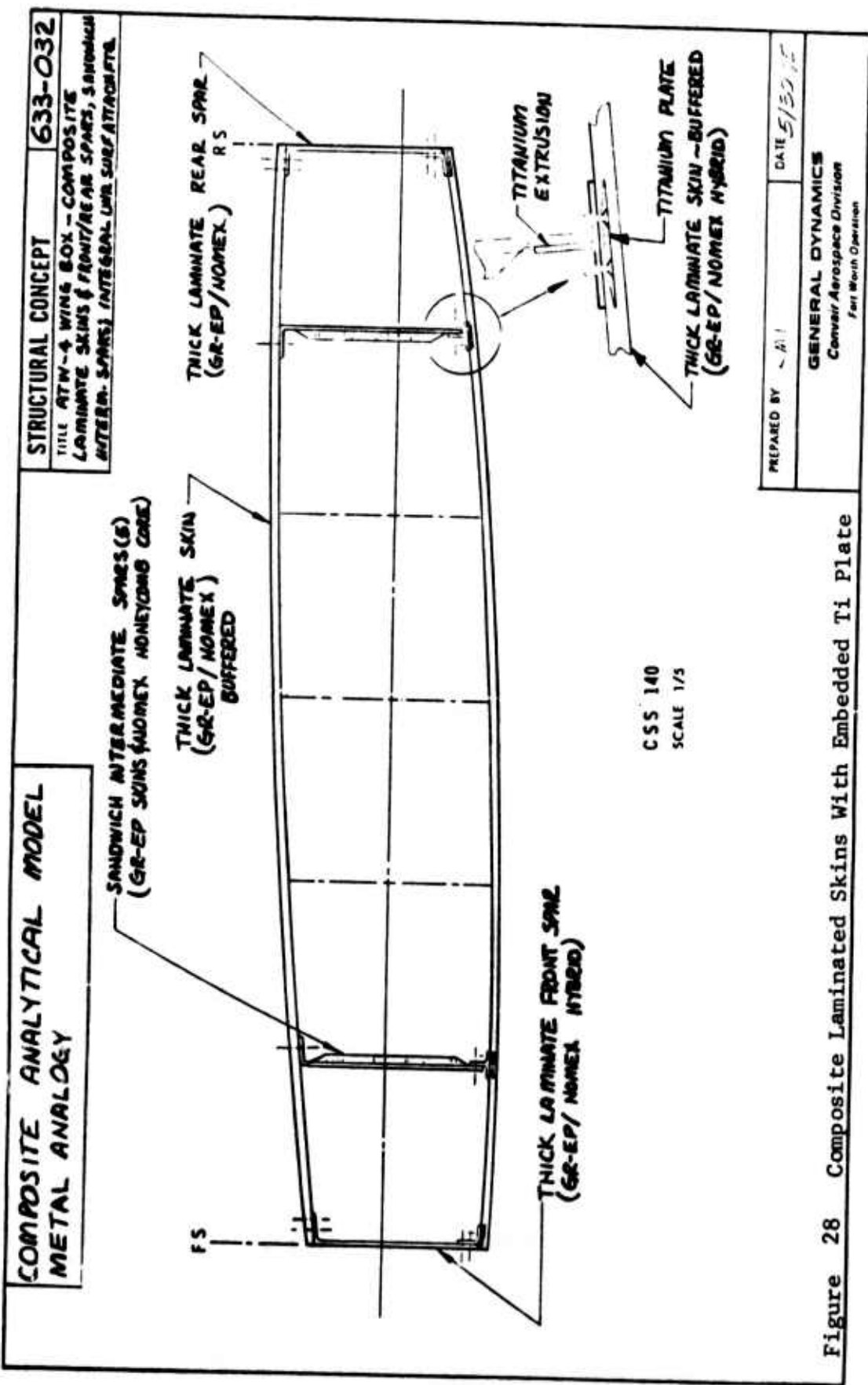


Figure 28 Composite Laminated Skins With Embedded Ti Plate

PREPARED BY: *AVI* DATE: 5/50/50

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

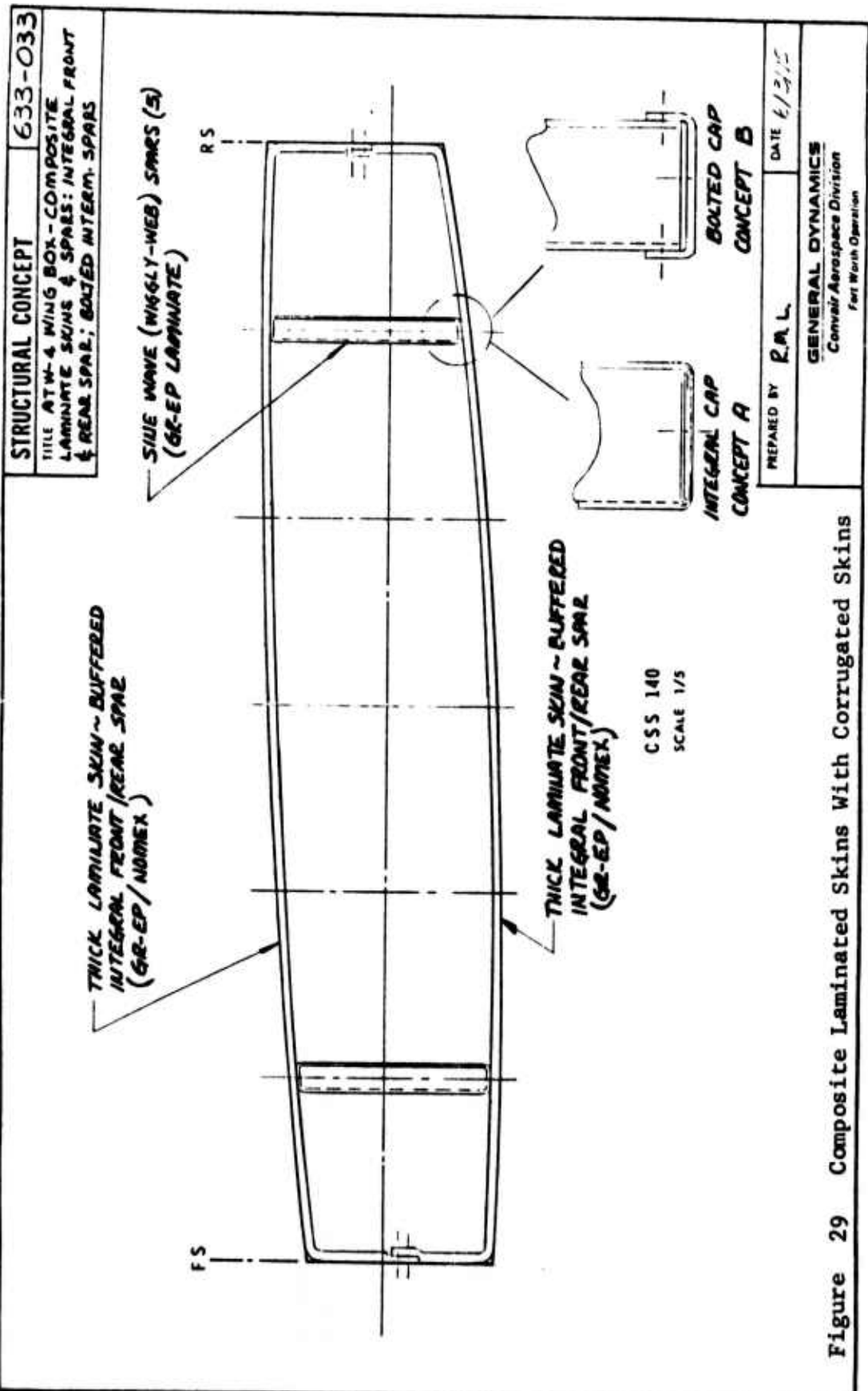


Figure 29 Composite Laminated Skins With Corrugated Skins

PREPARED BY **R.A.L.** DATE **6/2/5**

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

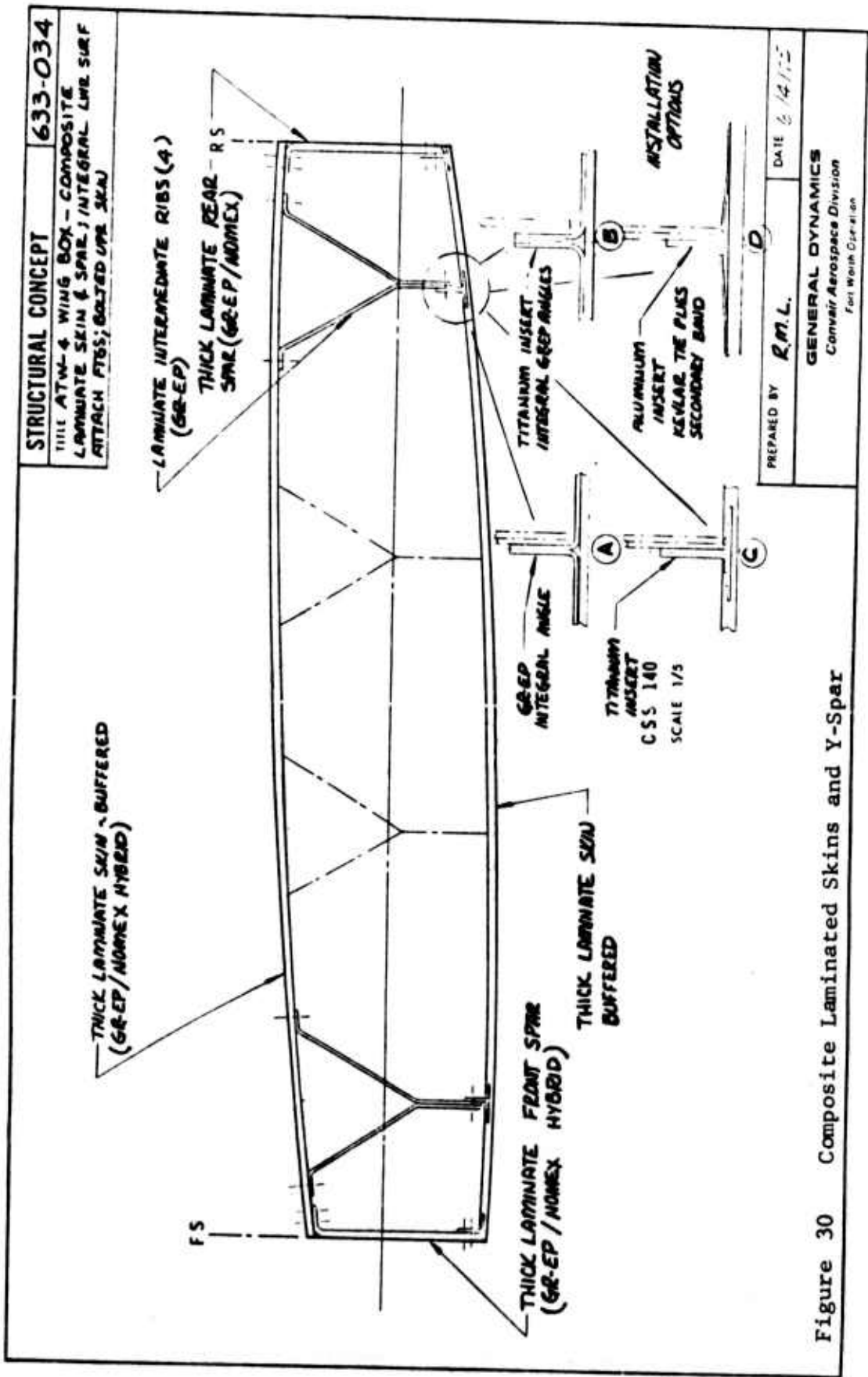
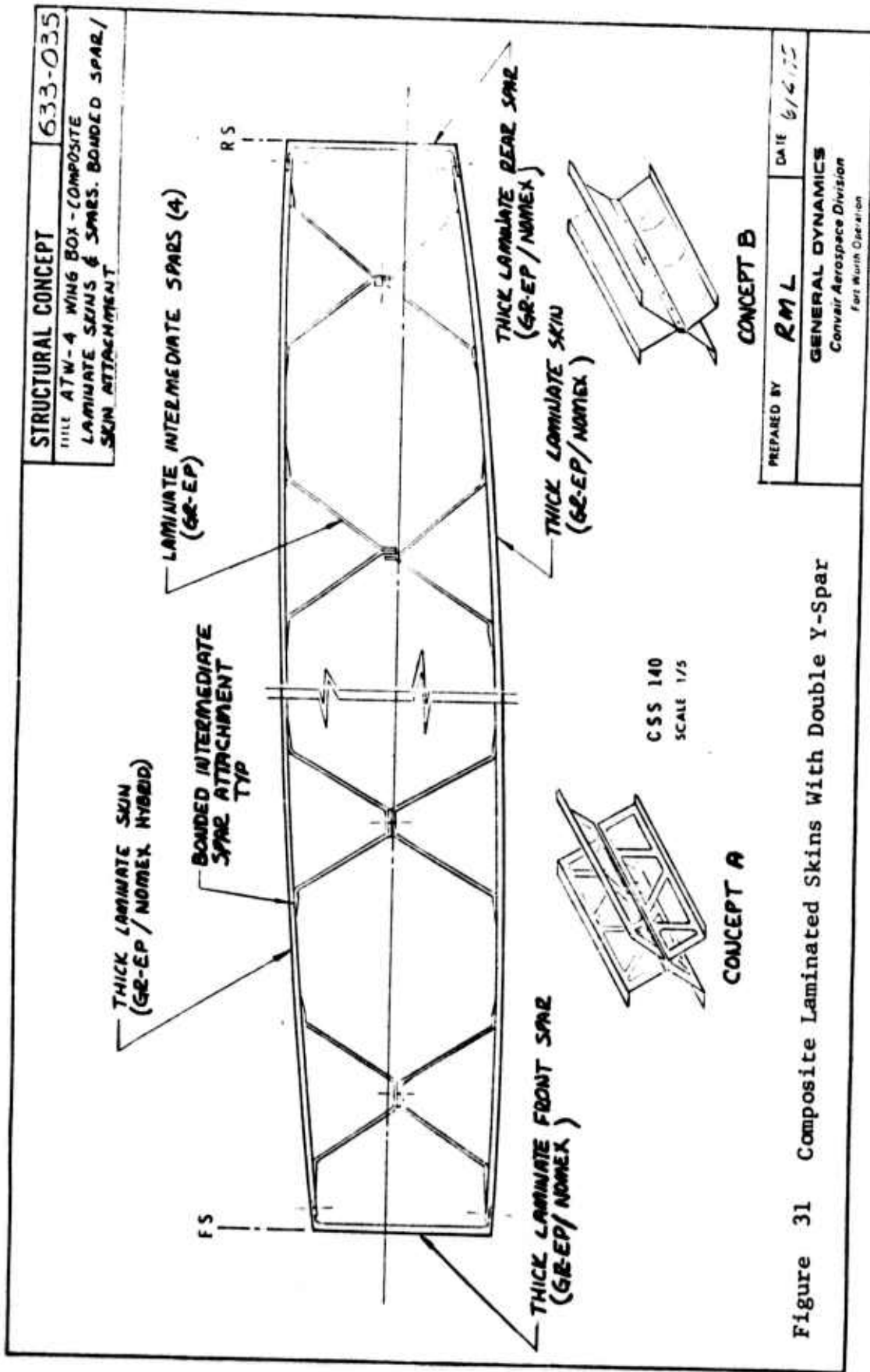


Figure 30 Composite Laminated Skins and Y-Spar

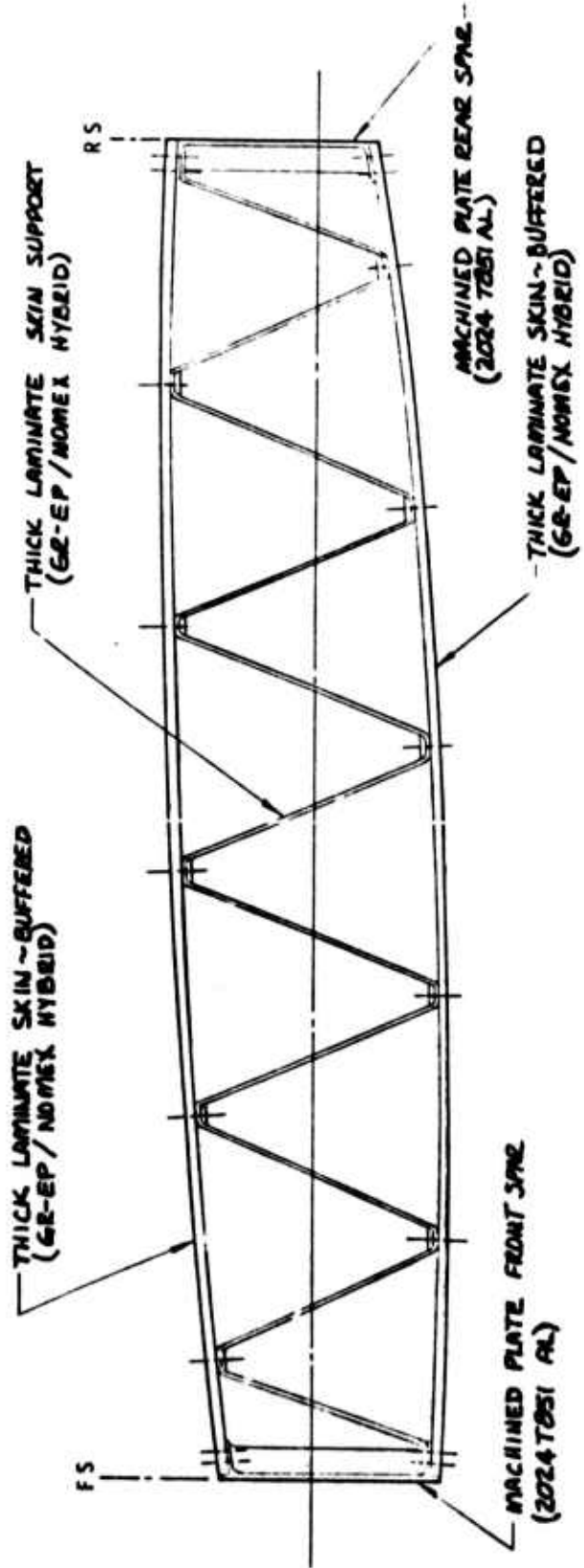


PREPARED BY **RML** DATE **6/6/75**

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

Figure 31 Composite Laminated Skins With Double Y-Spar

STRUCTURAL CONCEPT 633-036
 TILL ATW-4 WING BOX - COMPOSITE
 LAMINATE SKINS & SKIN SUPPORTS AL.
 FRONT/REAR SPAR. BOLTED ASSY



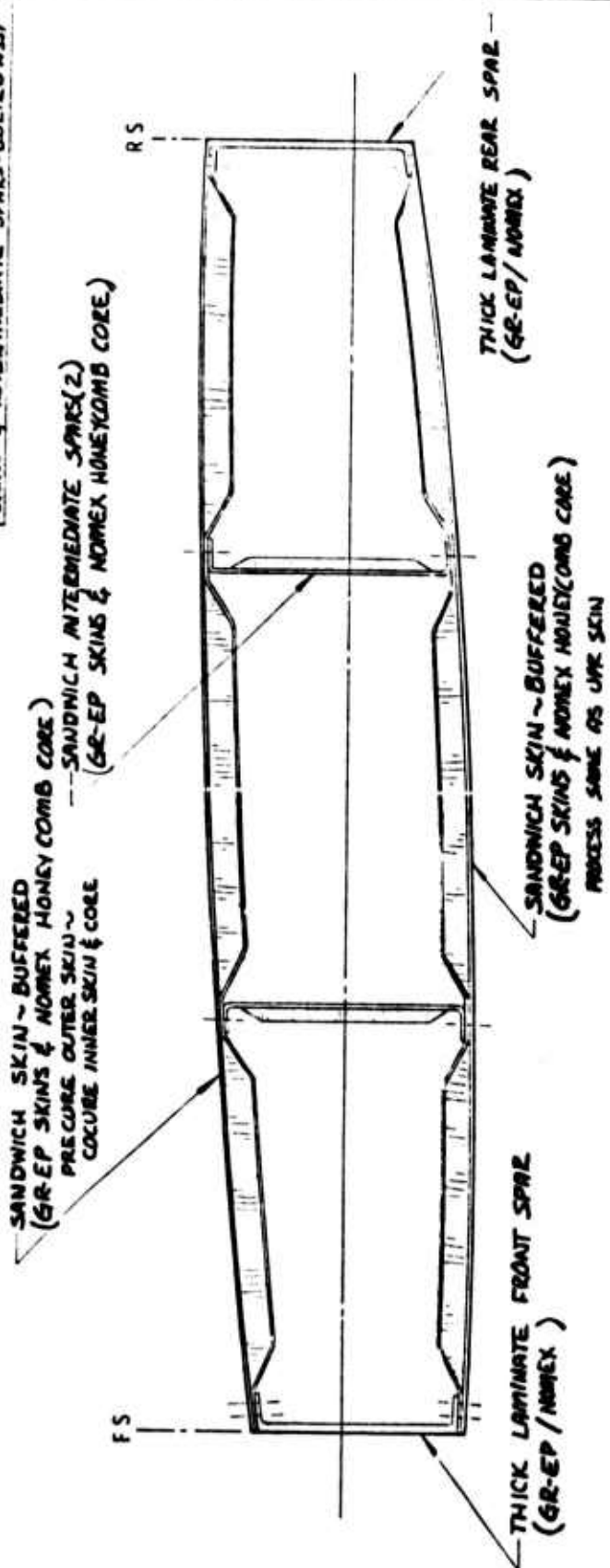
CSS 140
 SCALE 1/5

PREPARED BY RML DATE 5/29/75
 GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth, Oklahoma

Figure 32 Composite Laminated Skins & Support Structure

STRUCTURAL CONCEPT 633-037

|||| ATW-4 WING BOX - COMPOSITE
LAMINATE FRONT/REAR SPARS; SANDWICH
SKINS & INTERMEDIATE SPARS - BOLTED ASSY



SANDWICH SKIN ~ BUFFERED
(GR-EP SKINS & NOMEX HONEYCOMB CORE)
PROCESS SAME AS WPC SKIN

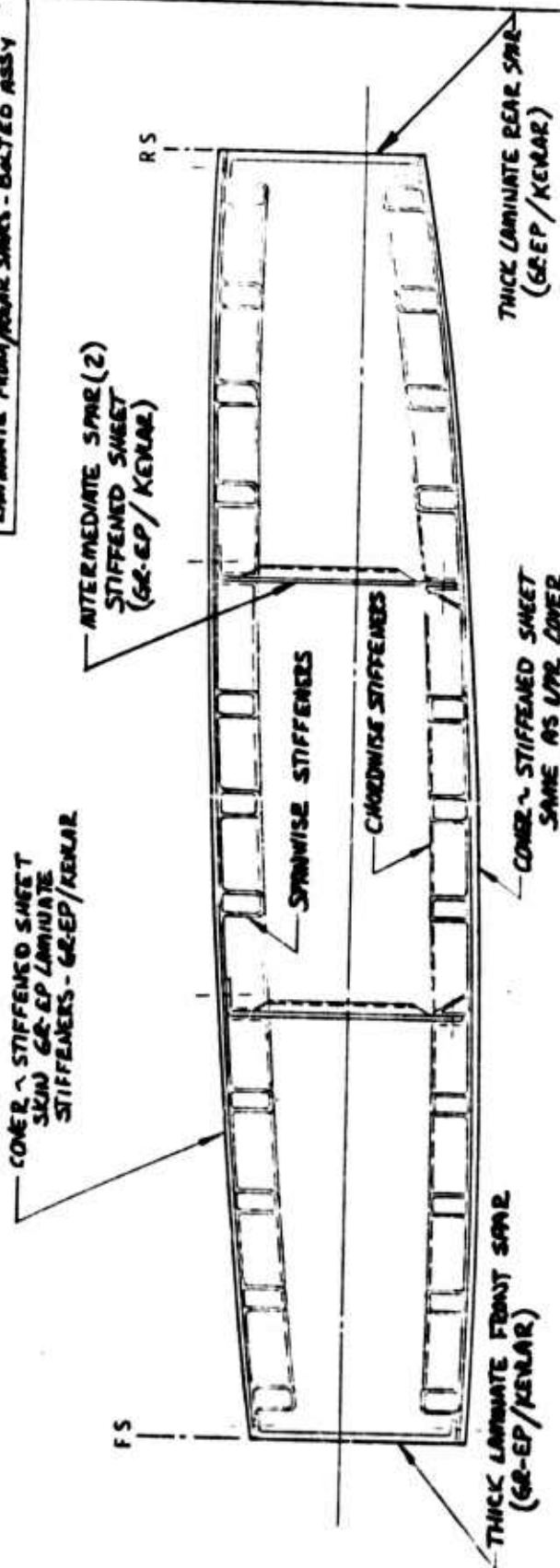
CSS 140
SCALE 1/5

PREPARED BY C M L DATE 6/4/75

GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operation

Figure 33 Composite Sandwich Skins & Spars

STRUCTURAL CONCEPT 633-038
 HILL ATW-4 WING BOX - COMPOSITE
 STIFFENED SHEET SKIN & INTERM. SPARS,
 LAMINATE FRONT/REAR SPARS - BOLTED ASSY



CSS 140
 SCALE 1/5

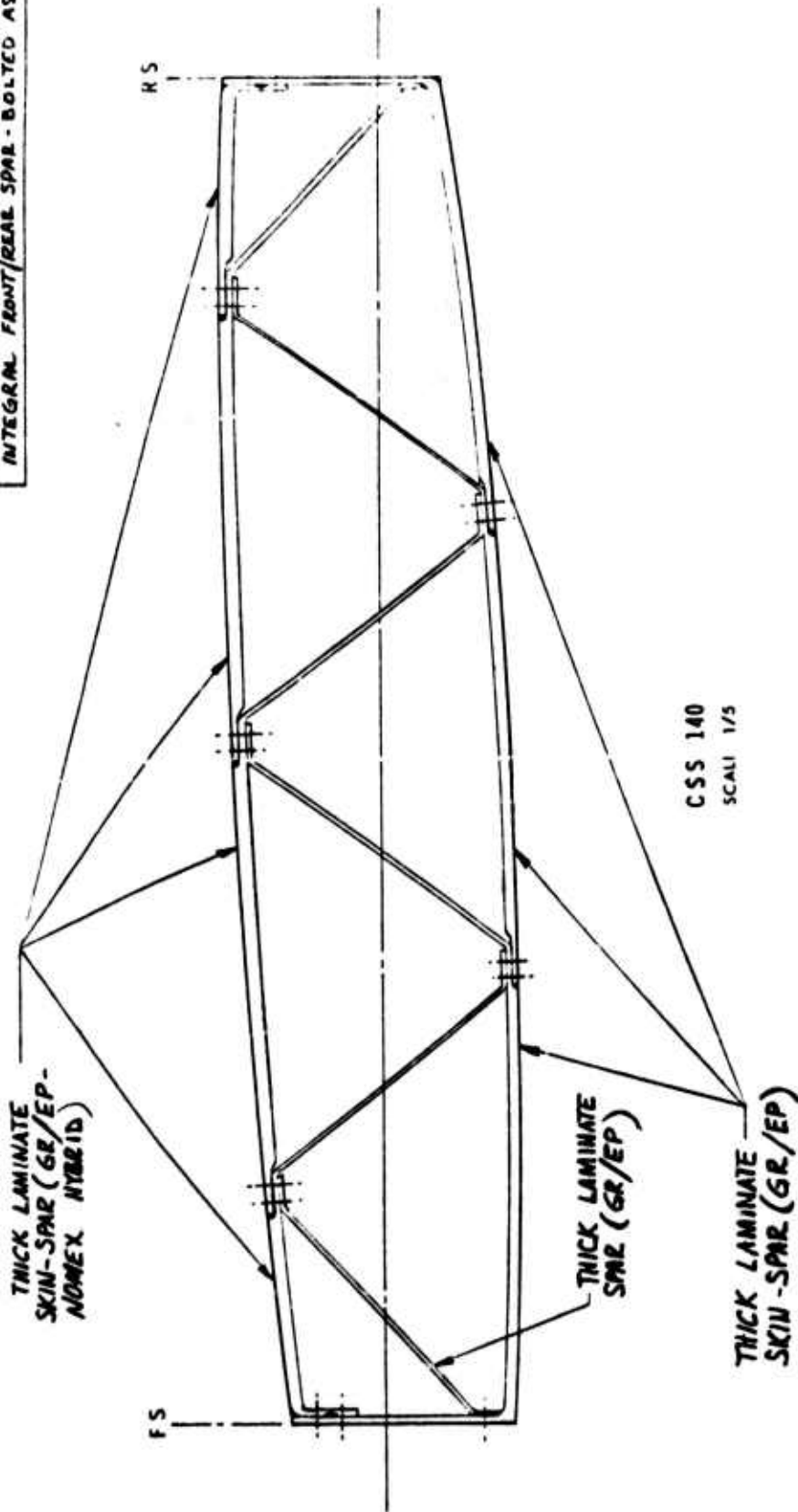
PREPARED BY R.M.L. DATE 6/5/75

GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth, Texas

Figure 34 Composite Laminated Skins & Spars With Stiffeners

STRUCTURAL CONCEPT 633-039

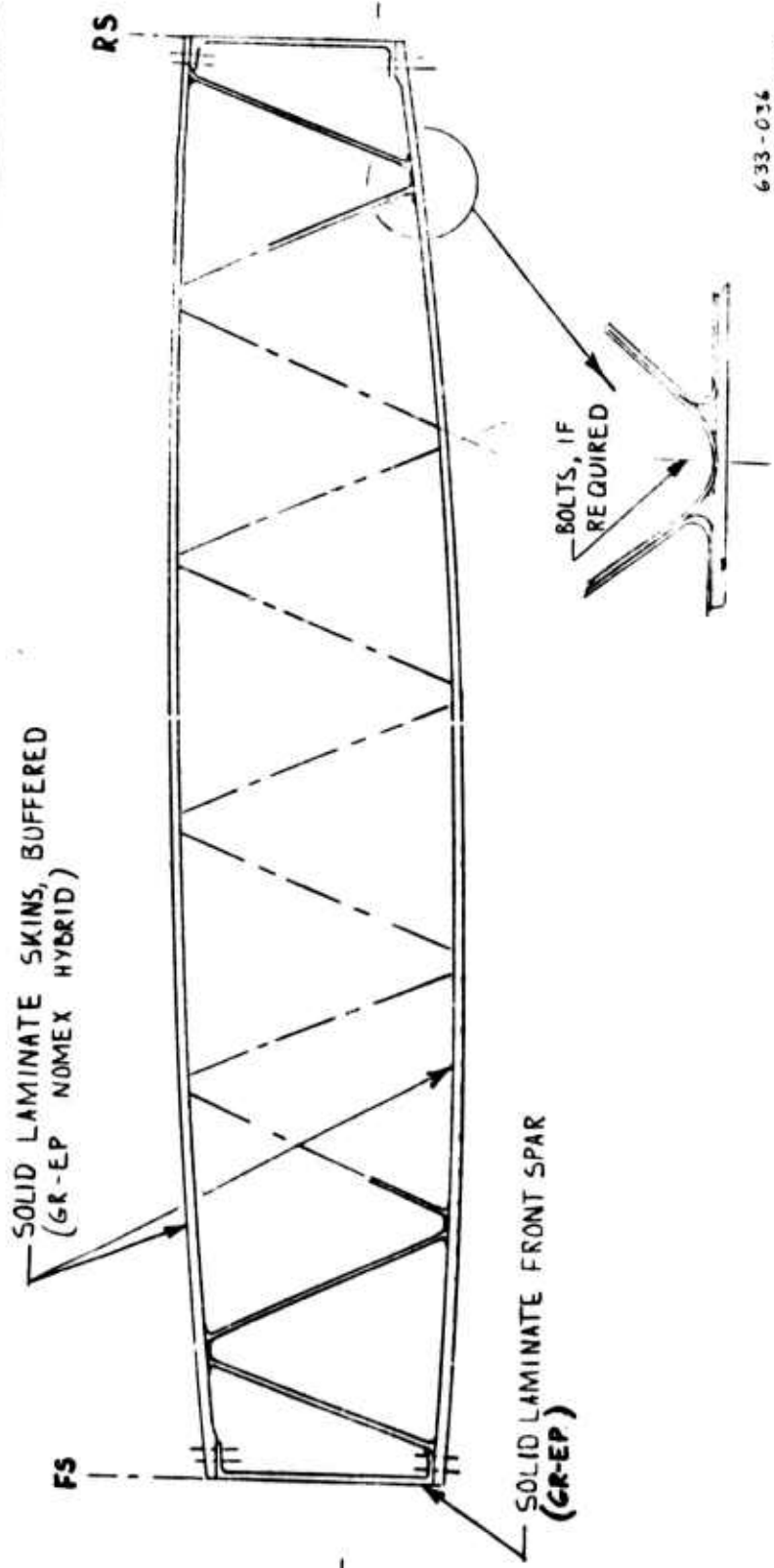
TITLE ATW-4 WING BOX - COMPOSITE
LAMINATE WING SPAR MODULAR CONSTRUCTION
INTEGRAL FRONT/REAR SPAR - BOLTED ASSY



PREPARED BY J. L. Hudson
GENERAL DYNAMICS
Convair Aerospace Division
Fort Worth Operations

Figure 35 Modular Composite Structure

STRUCTURAL CONCEPT 633-040
 TITLE ATW-4 WING BOX - COMPOSITE
 INTEGRATED SKIN WITH
 TRUSS SPAR CONCEPT



633-036
 REVISED 7/25/75

PREPARED BY RAILL. DATE 7/25/75
 GENERAL DYNAMICS
 Convair Aerospace Division
 Fort Worth Operation

Figure 36 Composite Truss Spar

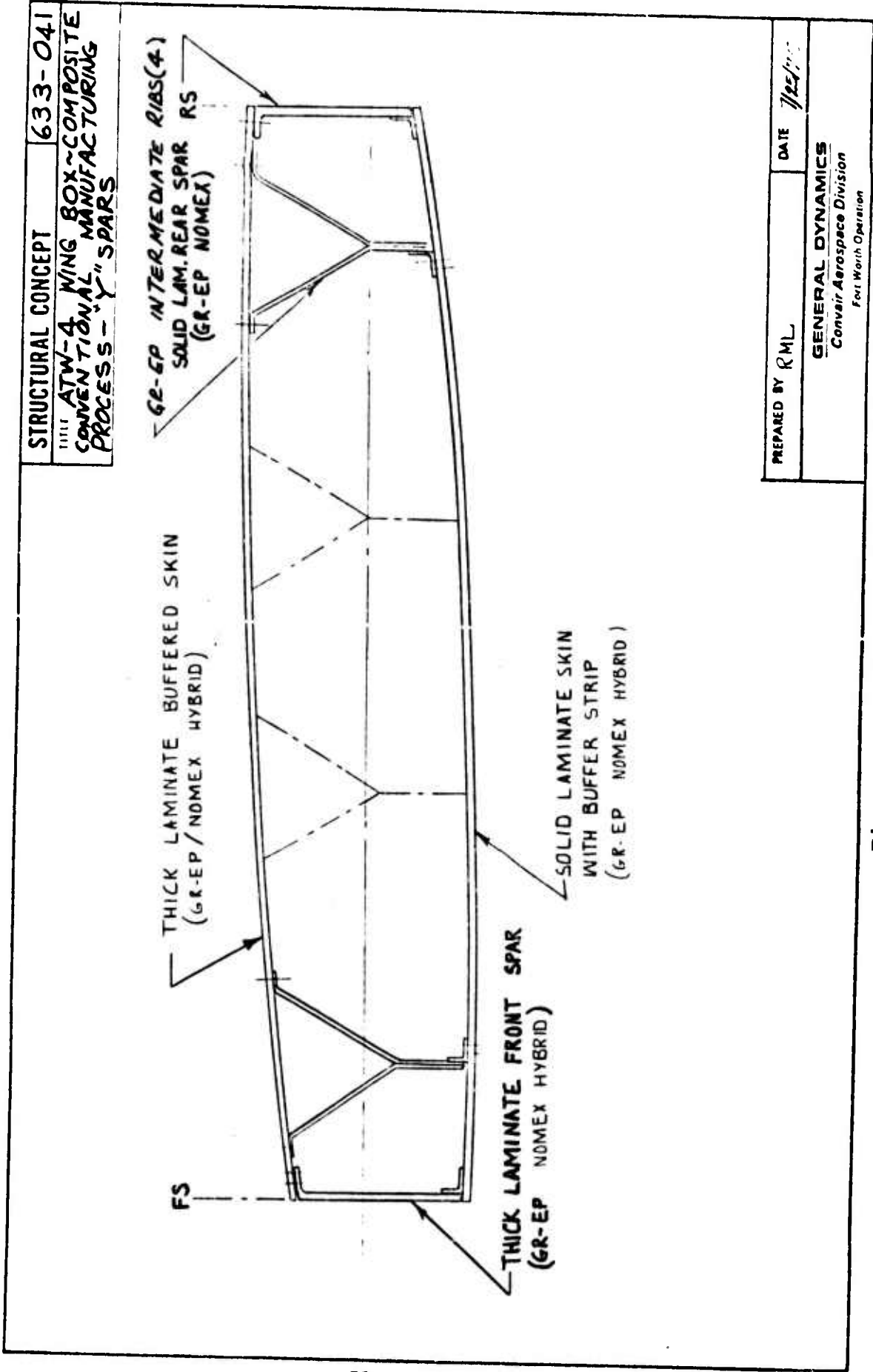


Figure 37 Composite Y-Spar

Table III WING BOX COMPOSITE CROSS SECTION CONCEPT
SCORING AND RANKING SUMMARY

CONCEPT	STRUCTURAL EFFICIENCY		TOTAL SCORE (1.00)	RANKING
	WEIGHT (.40)	COST (.60)		
633-030	.348	.334	.882	7
633-031	.348	.541	.889	6
633-032	.348	.545	.893	5 *
633-033	.400	.600	1.000	1 *
633-034	.372	.578	.950	3 *
633-035	.385	.541	.889	6
633-036	.337	.542	.879	8
633-037	.400	.578	- FUEL VOL. LOSS UNACCEPTABLE -	-
633-038	.337	.535	.872	9
633-039	.326	.533	.859	10
633-040	.337	.600	.937	4 *
633-041	.385	.578	.963	2 *

* THESE CONCEPTS INPUT INTO THE ANALYTICAL ASSEMBLY ITERATION

3.2.2 Analytical Assembly Iteration

Fourteen metallic wing box concepts, five composite wing box concepts, and a baseline wing box were defined on analytical assembly drawings. These design concepts are shown in Figure 38 thru 46 which correspond to drawing numbers 633RA000 thru 633RA008. Weight and cost for each significant detail is shown in the data block of each analytical assembly drawing.

Tables IV and V contain tabulated evaluation summaries of all design parameters specified in the AFFDL merit rating system for each analytical assembly concept studied. The costing ground rules are defined in Section VIII.

3.2.2.1 Metallic Analytical Assemblies

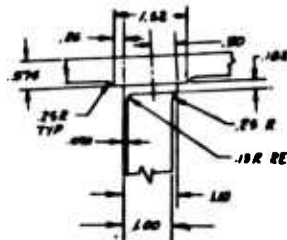
Of the fourteen promising metallic cross-section concepts that were carried into the analytical assembly iteration, the two concepts that emerged with the highest overall scores were 633RA003-801 (ranked first), 633RA001-1 (ranked second).

The 633RA003-801 analytical assembly concept is configured with laminated lower skins having no fastener penetrations and exposed, stepped spar caps; non-pocketed monolithic 2024-T851 aluminum plate upper skin; and intermediate spars having canted sheet aluminum webs stabilized by "V" shaped intercostals that support dual extruded upper spar caps. The front and rear spars are integrally machined members.

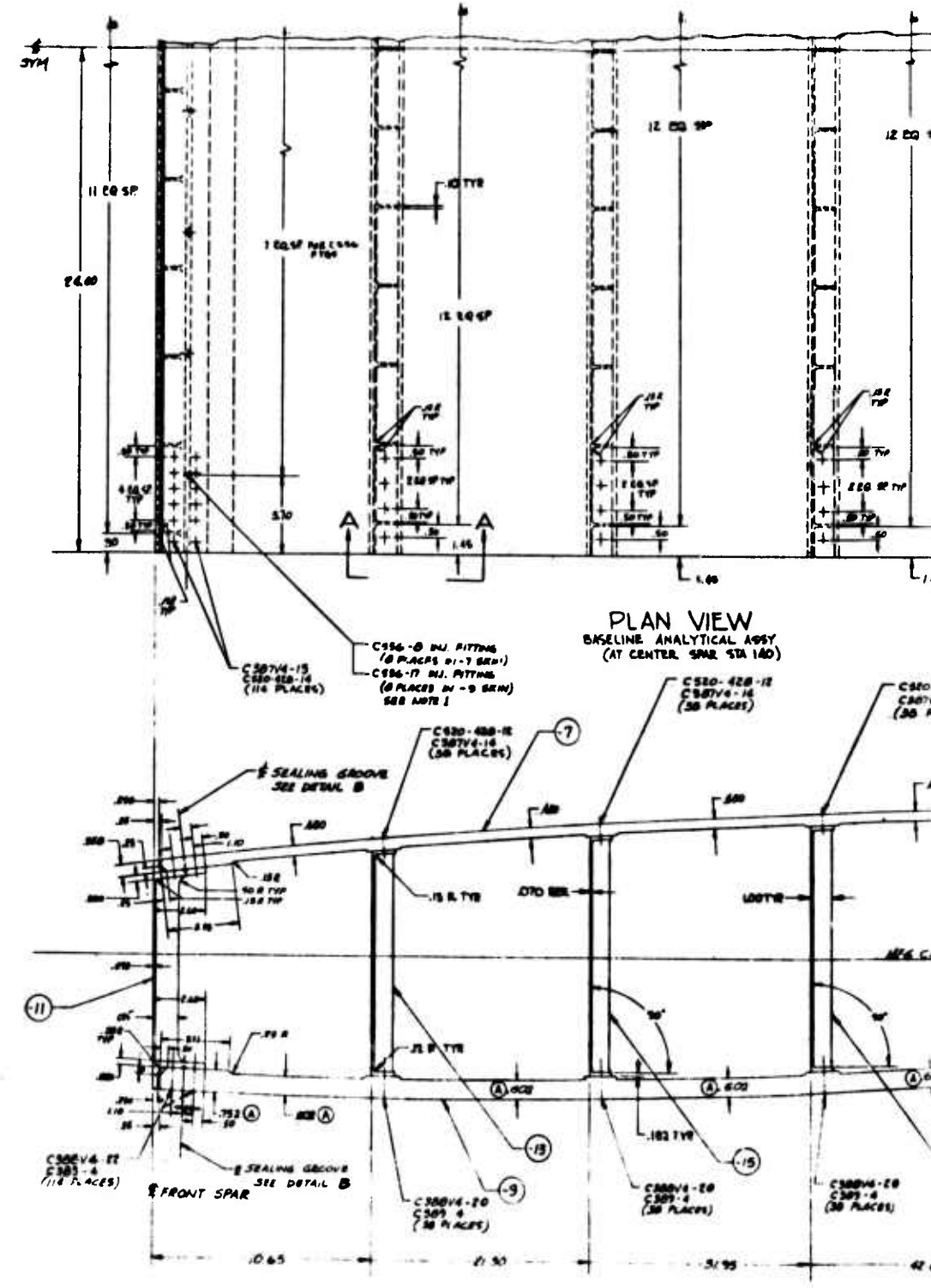
The 633RA001-1 analytical assembly concept incorporates the same upper skin, lower skin, front spar and rear spar concepts as 633RA003-801. The difference between the two designs is in the intermediate spars. The 633RA001-1 design incorporates extruded "Y" spars.

3.2.2.2 Composite Analytical Assemblies

Of the five composite analytical assembly concepts studied, the number 1 ranking design is 633RA006-3. This design incorporates solid graphite epoxy laminate upper and lower skins without buffer strips and a "Y" intermediate spar concept with an embedded lower cap and 16 layers of nomex core material in the webs. This design is more cost effective than the other composite designs because it requires less composite material than the other and permits maximum utilization of the automatic tape laying machine for fabrication.

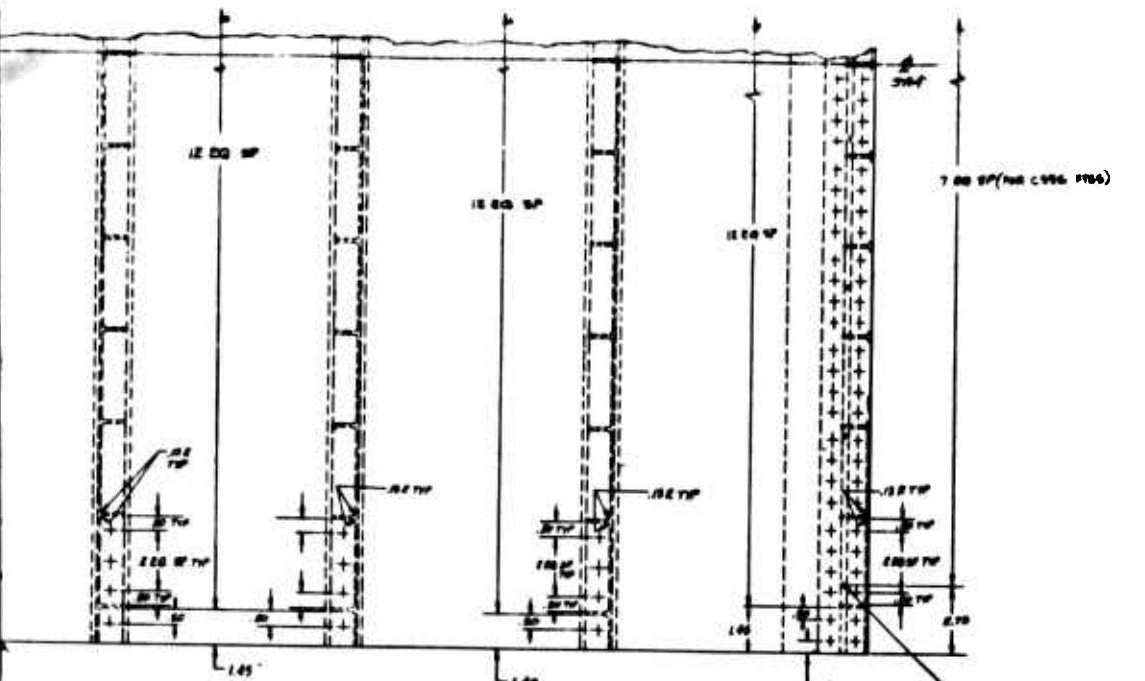


VIEW A-A
TYPICAL
(FULL SIZE)

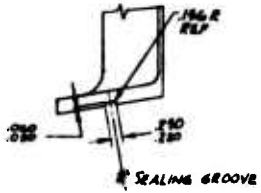
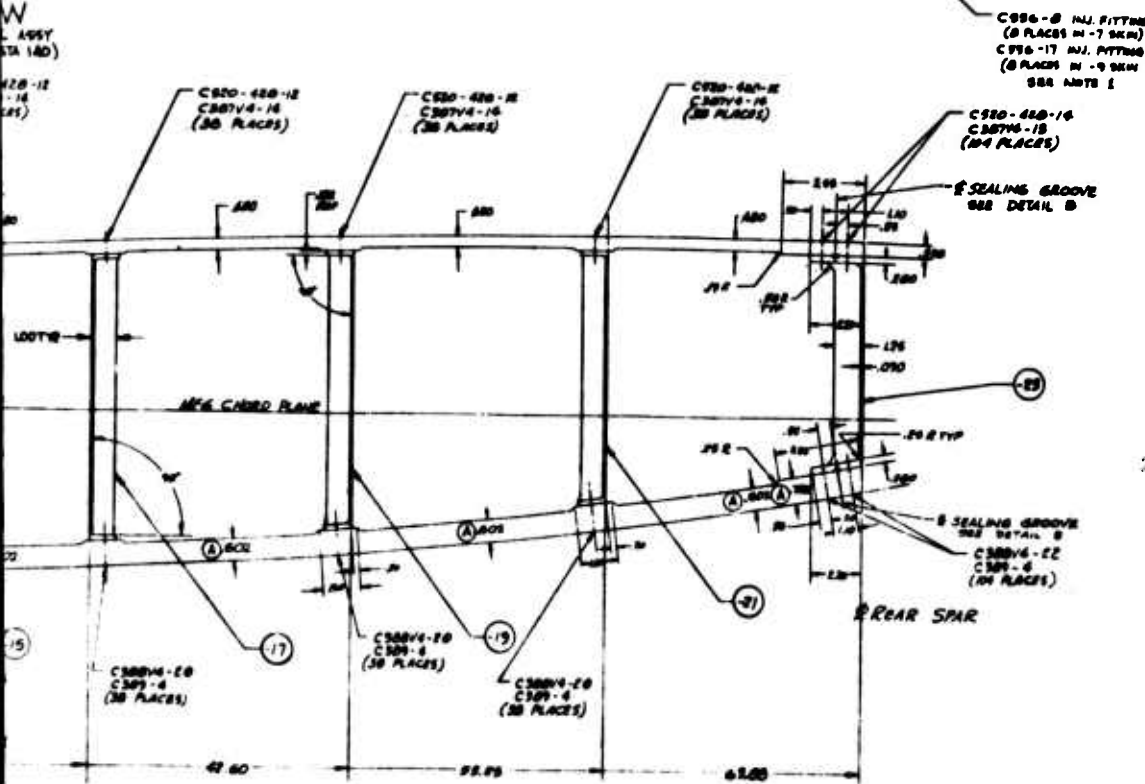


PLAN VIEW
BASELINE ANALYTICAL ASSY
(AT CENTER SPAR STA 140)

12



QTY	UNIT	DETAIL	PART NAME	NO. REQ'D	MATERIAL	QTY REQ'D
		-7	UPPER SKIN (SEE NOTE 3)	1	2024-T3B1	750X540X...
		-9	LOWER SKIN (SEE NOTE 3)	1		878X540X...
		-11	FRONT SPAR	1		1.00X11.0X...
		-15	FWD INTERM SPAR	1		1.00X11.0X...
		-15	FWD CENTER SPAR	1		1.00X11.0X...
		-17	CENTER SPAR	1		1.00X11.0X...
		-19	REAR CENTER SPAR	1		1.00X11.0X...
		-21	REAR INTERM SPAR	1		1.00X11.0X...
		-23	REAR SPAR	1	2024-T3B1	750X92.5X...
			SEALANT		FMS-1046 (SEE NOTE 8)	
			BOLT (C387V4-10)	208		
			BOLT (C387V4-14)	190		
			BOLT (C387V4-20)	208		
			BOLT (C387V4-28)	190		
			NUT (C389-4)	408		
			PRESS NUT (C320-420-12)	190		
			PRESS NUT (C320-420-16)	208		
			PL. FTS (C386-0)			
			PL. FTS (C386-17)			
MATERIAL REMOVED FOR PASTER M...						
ANALYTICAL ASSY						



DETAIL B
TYPICAL AT FRONT &
REAR SPAR, UPPER &
LOWER FLANGES
(FULL SIZE)

- NOTES:**
- INSTALL C596 COLLECTION FITTINGS PER MTG EQUALLY SPACED BETWEEN PASTERERS AS SHOWN AND CENTERED ON THE SEALING GROOVE. PLUG WITH WAS C01-06A9 SCREWS.
 - THE FAYING SURFACES AT THE UPPER SPAR AND ALL PASTERERS, SHALL BE SEALED PER PDS-1004.
 - 7 UPPER SKIN AND -9 LOWER SKIN SPAR LANDS TO BE MACHINED AND POCKETS TO BE CHAM ETCHED.

COPY PERMIT

Figure 38 Baseline Analy

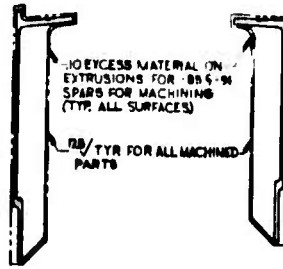
ITEM NAME	NO. REQD.	MATERIAL	GRN WGT OR PRICE	SKIN THICKNESS	PROPOSED VOLUME	EST. COST	TOTAL COST
UPPER SKIN (NOTE 3)	1	2024-T8B1	5015101490	202.24	159.27	435.76	289
UPPER SKIN (NOTE 3)	1		87819601600	328.28	198.07	726.17	743.14
FRONT SPAR	1		27711171940	166.40	11.78	272.67	638.60
INTERNAL SPAR	1		12311181940	74.99	6.80	115.77	393.88
CENTER SPAR	1		12311171940	79.76	7.18	123.14	397.29
LOWER SPAR	1		12311191940	81.15	7.28	125.24	400.79
CENTER SPAR	1		12311161940	79.00	7.18	121.95	389.19
INTERNAL SPAR	1		12311061940	72.27	6.80	111.57	376.17
LOWER SPAR	1	2024-T8B1	23019221940	129.44	10.96	180.34	536.25
PLANT		FMS-1048 (SEE NOTE 2)		1.75	1.50		
(C3874-1) 200				1.996			
(C3874-14) 190				1.919			
(C3874-17) 200				2.280			
(C3874-28) 190				1.580			
(C389-4) 400				1.805			
NUT 520-420-12 190				2.082			
1/2 NUT 420-420-16 200				2.975			
6 (C396-6) 190				2.999			
75 (C396-17) 190				1.771			
MATERIAL REMOVED FOR FASTENER HOLES					422.38		
TOTAL ASSY					416.00	224477777	212,310.00

REV	DESCRIPTION	DATE
A	CHANGED LOWER SKIN THICKNESS FROM .012 TO .002	04-29-88
B	CHANGED LOWER SKIN SPAR LAND THICKNESS FROM 1.082 TO .752	
	REVISED ALL WEIGHTS IN L/W TO REFLECT LOWER SKIN AND LOWER SPAR LAND THICKNESS CHANGES	

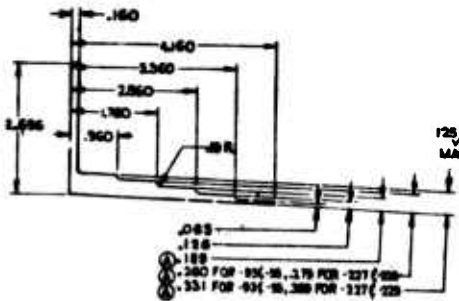
COPY AVAILABLE TO DDC DOES NOT PERMIT FULLY LEGIBLE PRODUCTION

UPPER SKIN AND LOWER SKIN SPAR BANDS TO BE MACHINED AND ROCKETTS TO BE CHEM-ETCHED.
 THE FAYING SURFACES AT THE UPPER SKIN AND ALL FASTENERS, SHALL BE SEALED PER RPS-1004
 INSTALL C996 NUTS PER FITTINGS PER FIG 2076 EQUALLY SPACED BETWEEN FASTENERS AND SHOWN AND CENTERED ON THE SEALING GROOVE. PLUG WITH HAS C91-06A9 SCREWS

PRELIMINARY DESIGN DRAWING
 AT-14 BASELINE WING BOX
 MACHINED SKINS AND SPARS;
 ALUMINUM ANALYTICAL ASSY.
 GENERAL DYNAMICS
 33-2A-000A

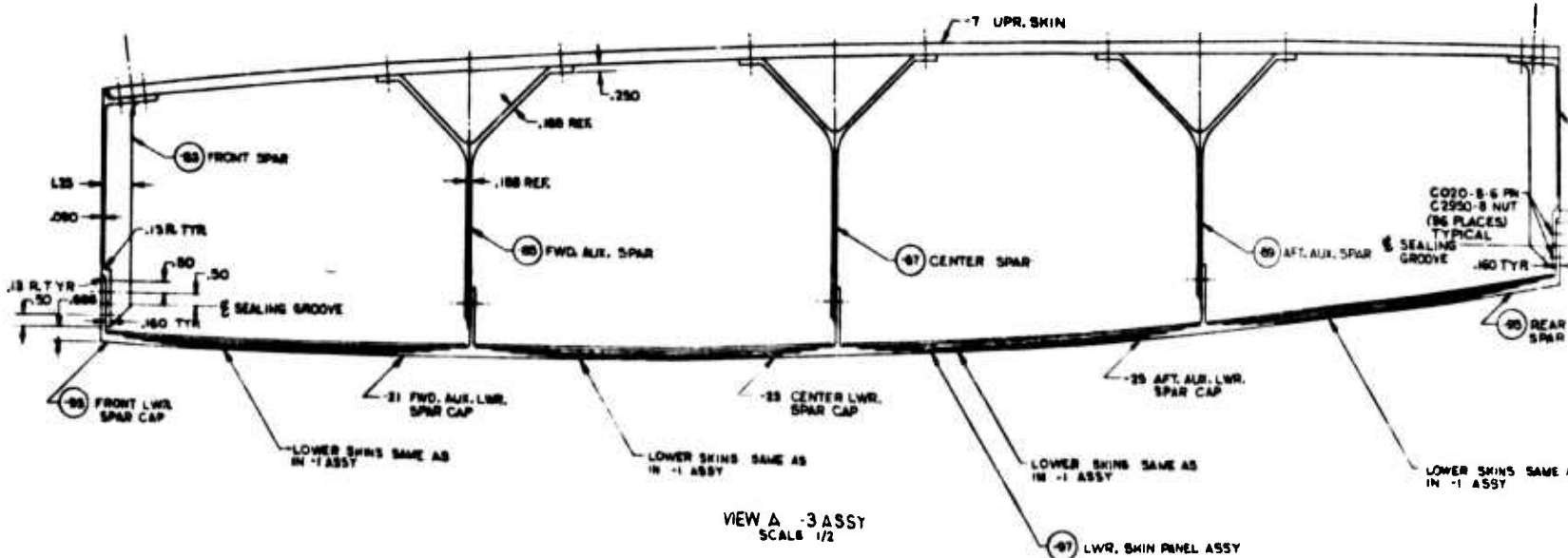


EXTRUSION DETAIL
-85 & -91
SCALE 1/2



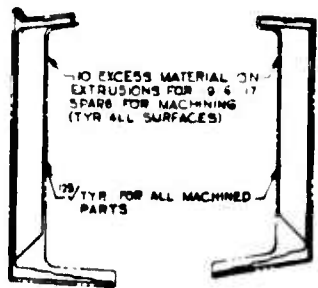
FRONT & REAR SPAR
LWR. CAP DETAIL
① -95, -95, -227, & -229

④ -99 & -231 EXTRUSION DETAIL
SCALE 1/2



VIEW A -3 ASSY
SCALE 1/2

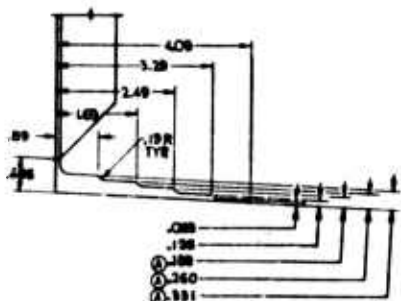
2



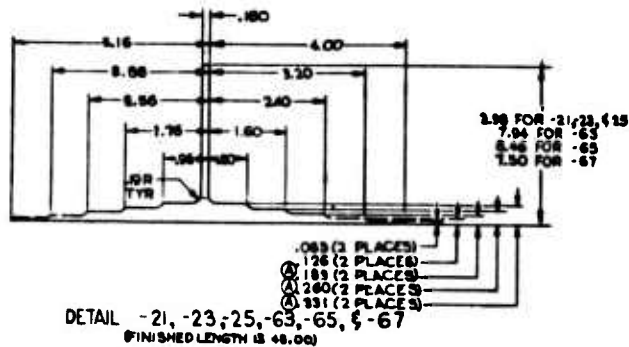
EXTRUSION DETAIL
-9 & -17
SCALE 1/2

TABLE I
DETAIL OF LOWER SURFACE WINGS SHOWN FOR -1, -9 & -17 ASSEMBLY

DASH NO.	RAW STOCK SIZE	RAW STOCK WT.	MATERIAL	FINISHED STOCK SIZE	FINISHED WT. IN LBS.
-27	.0631.31X50	2.51	2024-T81	.0631.78X48	2.37
-28	.0631.41X50	3.03		.0631.88X48	2.88
-31	.0631.11X50	3.33		.0631.06X48	3.18
-33	.0711.27X50	4.55		.0711.25X48	4.35
-38	.0711.41X50	5.13		.0711.41X48	4.87
-37	.0711.58X50	5.67	.0711.58X48	5.38	
-38	.0631.78X50	2.48	2024-T81	.0631.76X48	2.32
-41	.0631.94X50	2.99		.0631.92X48	2.81
-43	.0631.101X50	3.50		.0631.08X48	3.30
-45	.0711.126X50	4.52		.0711.124X48	4.27
-47	.0711.142X50	5.09		.0711.140X48	4.82
-49	.0711.156X50	5.59	.0711.154X48	5.31	
-59	.0631.78X50	2.48	2024-T81	.0631.76X48	2.32
-41	.0631.94X50	2.89		.0631.92X48	2.81
-43	.0631.101X50	3.50		.0631.08X48	3.30
-45	.0711.126X50	4.52		.0711.124X48	4.27
-47	.0711.142X50	5.09		.0711.140X48	4.82
-48	.0711.156X50	5.59	.0711.154X48	5.31	
-51	.0631.61X50	2.43	2024-T81	.0631.44X48	2.27
-53	.0631.01X50	2.83		.0631.01X48	2.75
-55	.0631.08X50	3.44		.0631.06X48	3.25
-57	.0711.124X50	4.45		.0711.124X48	4.21
-58	.0711.140X50	5.02		.0711.138X48	4.76
-61	.0711.154X50	5.52	.0711.152X48	5.25	



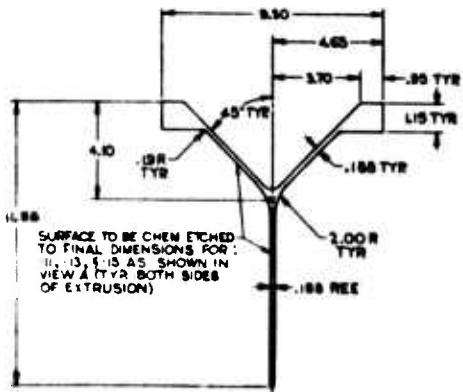
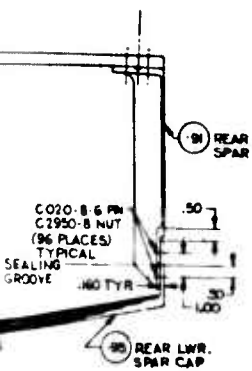
FRONT & REAR SPAR
LWR. CAP DETAIL
-9 & -17



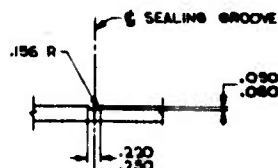
DETAIL -21, -23, -25, -63, -65, & -67
FINISHED LENGTH IS 48.00



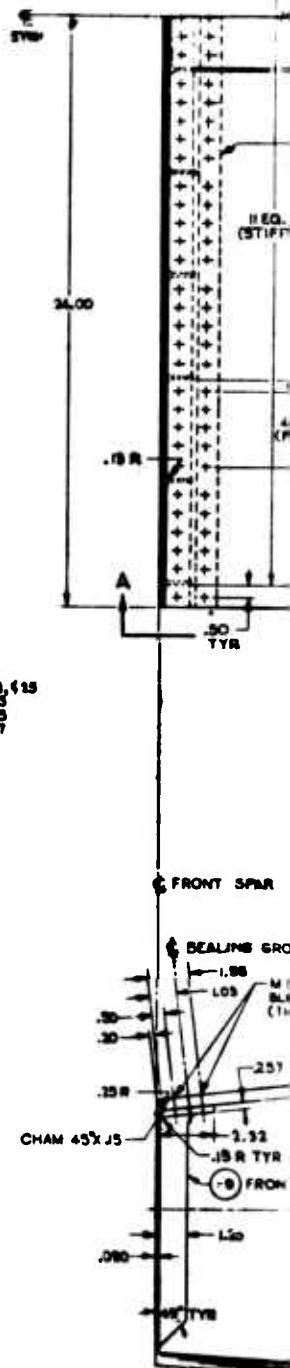
-71 & -77 EXTRUSION DETAIL
SCALE 1/2



-75 EXTRUSION DETAIL
SCALE 1/2

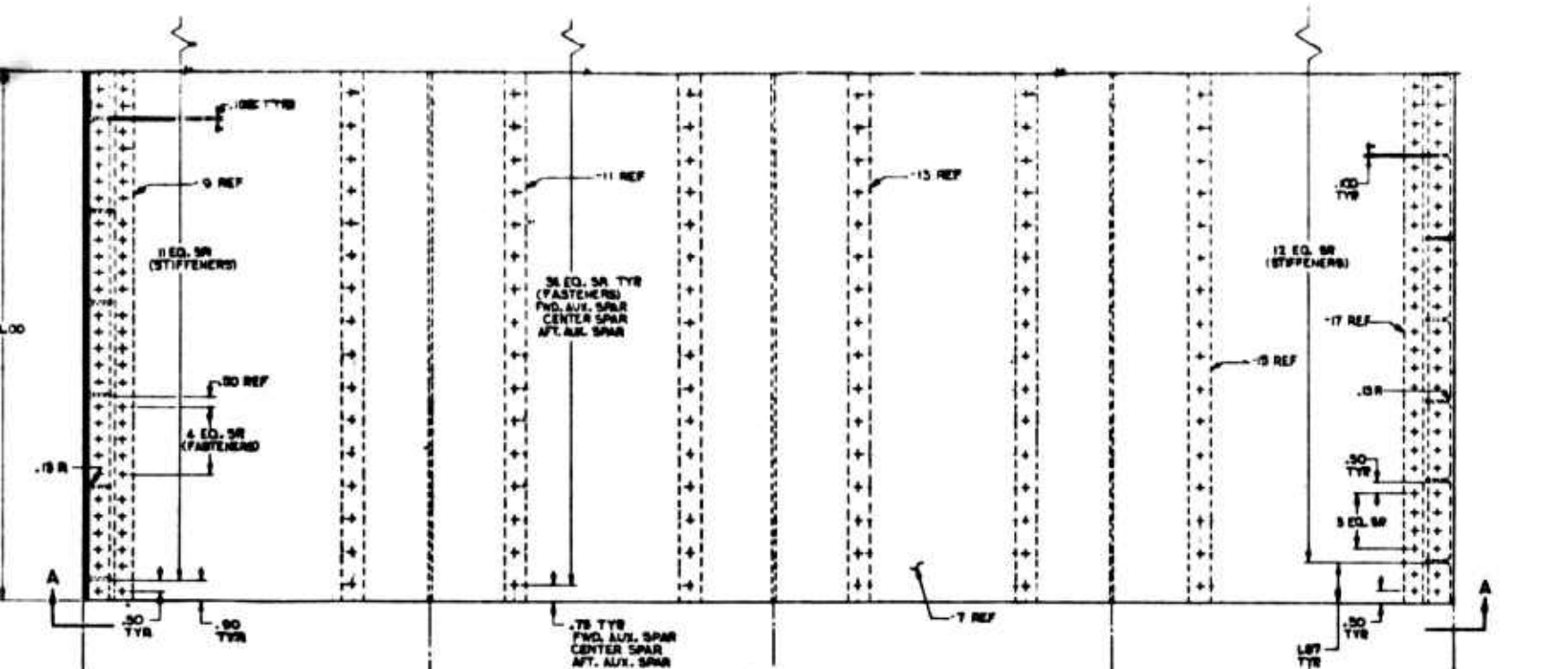


FUEL SEALING GROOVE DETAIL
(TYP FOR -9 & -17)

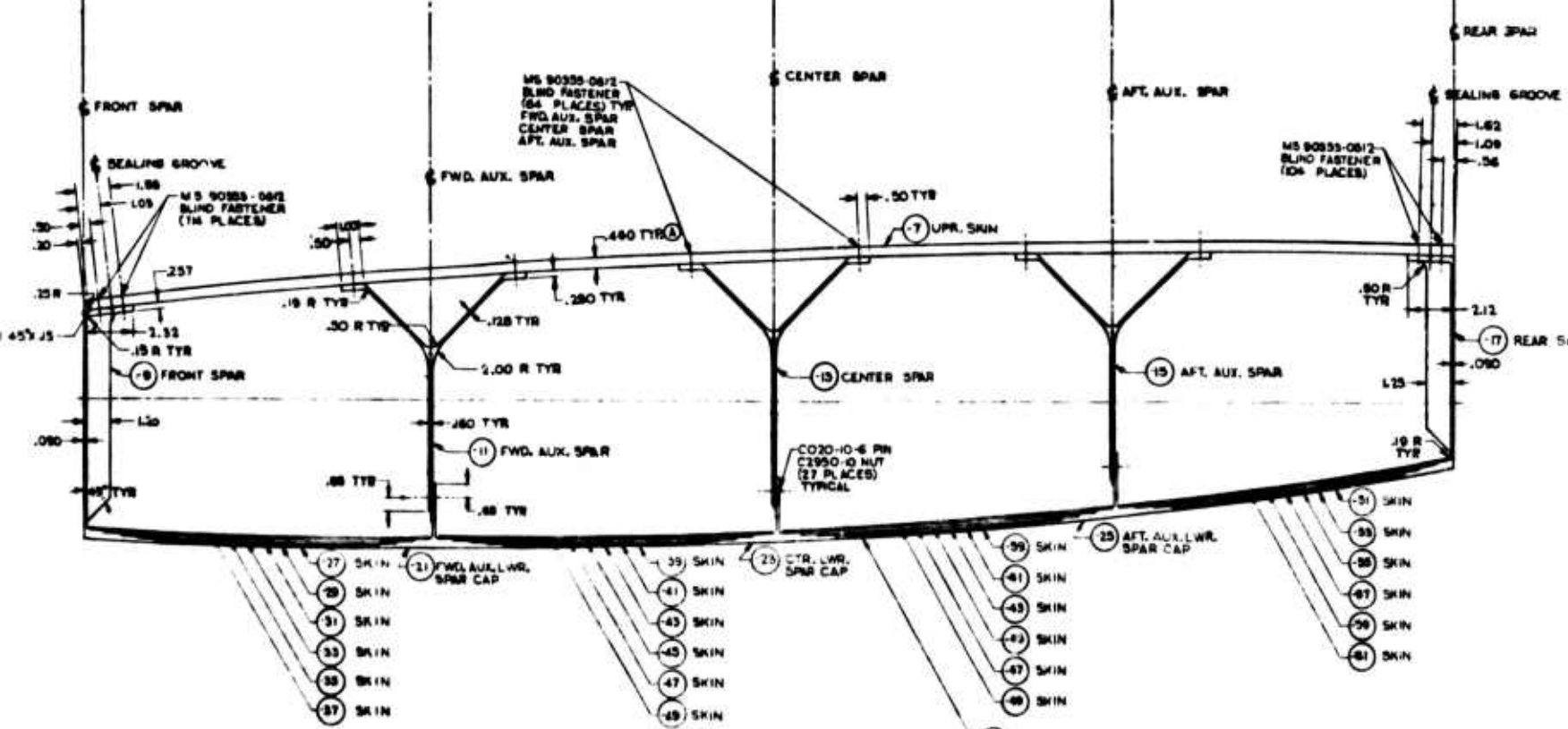


LOWER SKINS SAME AS
IN -1 ASST

3



PLAN VIEW - LASSY
(AT CENTER SPAR STA. 140)
SCALE 1/2



VIEW A - LASSY
SCALE 1/2

LIST OF MATERIALS

DASH NO.	ASBY	SUB ASSY	DETAIL	PART NAME	NO. REQD.	MATERIAL	RAW STOCK	FINISHED WT.	COST			TOTAL COST	
									DETAIL	ASBY	MATL		
				ANALYTICAL ASSY	1	SEE 1 ASSY			159.00	2799	6672	306	23772
				UPPER SKIN	1	SEE 1 ASSY			135.75	273.35	262.05		
				FRONT SPAR	1	SEE 1 ASSY			5.48	264.59	266.50		
				REAR SPAR	1	SEE 1 ASSY			8.78	239.25	255.00		
				FWD AUX SPAR ASSY	1	SEE 803 ASSY			19.67	30.22	22.1		
				CTR AUX SPAR ASSY	1	SEE 803 ASSY			15.72	31.02	22.1		
				AFT AUX SPAR ASSY	1	SEE 803 ASSY			15.32	30.25	22.1		
				LWR SKIN PANEL ASSY	1				146.88		287.07		
				FRONT LWR SPAR CAP	1	SEE 803 ASSY			6.9		181.85		
				SPAR EXTRU.	1	SEE 803 ASSY			6.09	107.78			
				FWD AUX LWR SPAR CAP	1	2024-TB51			6.09	35.83	27.45		
				CTR AUX LWR SPAR CAP	1	2024-TB51			6.09	35.83	27.45		
				AFT AUX LWR SPAR CAP	1	2024-TB51			6.09	35.83	27.45		
				SPAR EXTRU.	1	2024-TB51			6.09	35.83	27.45		
				REAR LWR SPAR CAP	1	SEE 803 ASSY			6.9		181.85		
				SPAR EXTRU.	1	SEE 803 ASSY			6.09	107.78			
				SKIN	1	SEE VIEW 2008			107.51	175.86	134.63		
				ADHESIVE	1	FMS-1018 FORM 1 (SEE NOTE 3)			8.72				
				BLIND FASTENERS MS-90358-0812	410				6.88				
				PIN (C020-B-6)	192				3.53				
				NUT (C2950-B)	81				1.50				
				PIN (C020-D-5)	81				.79				
				NUT (C2950-10)	81				.41				
				SEALANT	1	FMS-1044 (SEE NOTE 1)			1.75	1.50			
						MATL REMOVED FOR FASTNER HOLES			3.10				
													ASSY 27.50

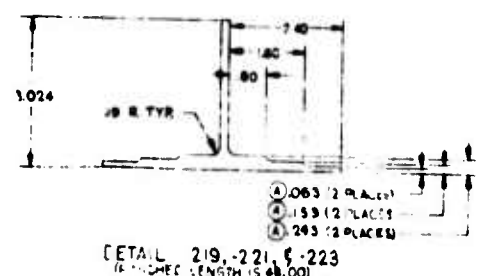
TABLE III
DETAILS OF LOWER SURFACE WING SPARS

DASH NO.	RAW STOCK SIZE	RAW STOCK WT.	MATERIAL	FINISHED STOCK SIZE	FINISHED WT. IN LBS.	NO. REQ'D
-223	0902-2-7150	5.77	2024-TB	0902-2-56748	5.48	1
-223	0907-4-5150	8.50		0907-4-38728	8.30	1
-227	0901-2-6150	4.18		0901-2-51728	4.08	1
-231	0901-4-3750	8.13		0901-4-21728	7.93	1
-238	0901-5-8150	7.03		0901-5-61728	6.83	1
-239	0901-1-4150	1.54		0901-1-31728	1.54	1
-247	0901-4-0150	6.36		0901-4-01728	6.36	1
-249	0901-5-4150	7.00	2024-TB	0901-5-26748	6.85	1

ACCESS MATL ON EXTRUSIONS FOR -214, -221, & -223 PARCAPS FOR MACHINING FOR MACHINING (TYR ALL SURFACES)

TYR FOR ALL MACHINING SURFACES

-225 EXTRUSION DETAIL

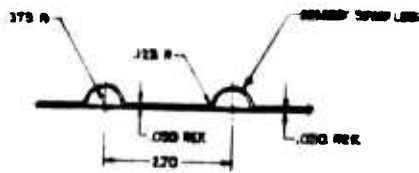


DETAIL 219, -221, & -223
(6 INCHES LENGTH IS 48.00)

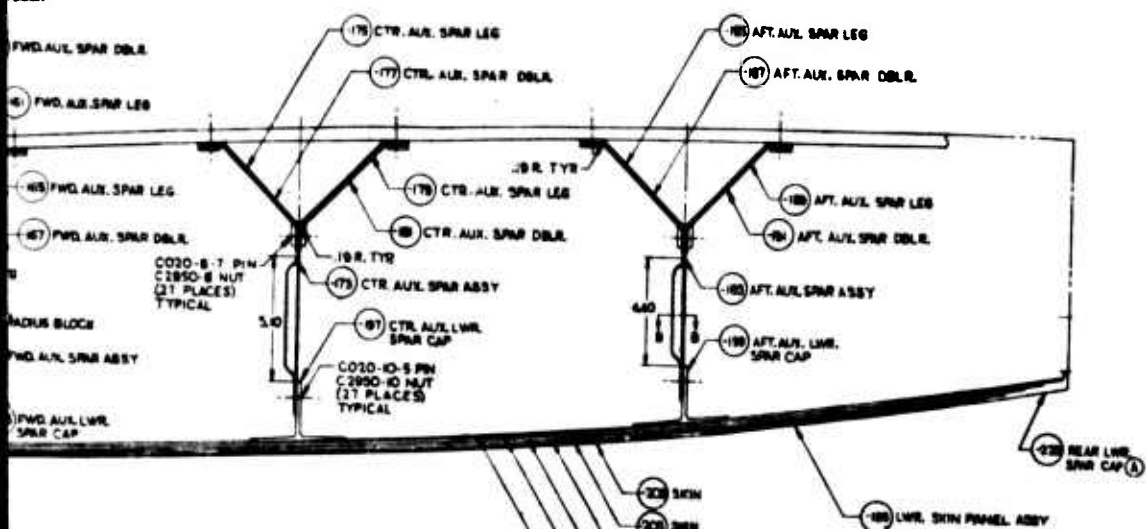
LIST OF MATERIALS

DASH NO.	ASBY	SUB ASSY	DETAIL	PART NAME	NO. REQD.	MATERIAL	RAW STOCK	FINISHED WT.	COST	TOTAL COST
				ANALYTICAL ASSY	1	SEE 1 ASSY				
				UPPER SKIN	1	SEE 1 ASSY				
				FRONT SPAR	1	SEE 1 ASSY				
				REAR SPAR	1	SEE 1 ASSY				
				FWD AUX SPAR ASSY	1	SEE 803 ASSY				
				FWD AUX SPAR LEG	1					
				FWD AUX SPAR LEG	1					
				FWD AUX SPAR LEG	1					
				FWD AUX SPAR DBLR	1					
				DOUBLER	2					
				RADIUS BLOCK	2					
				ADHESIVE	2	FMS-1018				
				PIN (C020-B-7)	27					
				NUT (C2950-B)	27					
				CTR AUX SPAR ASSY	1					
				CTR AUX SPAR LEG	1					
				CTR AUX SPAR DBLR	1					
				CTR AUX SPAR LEG	1					
				CTR AUX SPAR DBLR	1					
				DOUBLER	2					
				RADIUS BLOCK	2					
				ADHESIVE	2	FMS-1018				
				PIN (C020-B-7)	27					
				NUT (C2950-B)	27					
				AFT AUX SPAR ASSY	1					
				AFT AUX SPAR LEG	1					
				AFT AUX SPAR DBLR	1					
				AFT AUX SPAR LEG	1					
				AFT AUX SPAR DBLR	1					
				DOUBLER	2					
				RADIUS BLOCK	2					
				ADHESIVE	2	FMS-1018				
				PIN (C020-B-7)	27					
				NUT (C2950-B)	27					
				LWR SKIN PANEL ASSY	1					
				FRONT LWR SPAR CAP	1	MAKE FR				
				SPAR EXTRU.	1	2024				
				FWD AUX LWR SPAR CAP	1	2024				
				SPAR EXTRU.	1					
				CTR AUX LWR SPAR CAP	1					
				SPAR EXTRU.	1					
				AFT AUX LWR SPAR CAP	1					
				SPAR EXTRU.	1	2024				
				REAR LWR SPAR CAP	1	MAKE FR				
				SPAR EXTRU.	1	2024				
				SKIN	1	SEE TAB				
				SKIN	1					
				ADHESIVE	1	FMS-1018				
				BLIND FASTENERS MS-90358-0812	410					
				PIN (C020-B-6)	192					
				NUT (C2950-B)	81					
				PIN (C020-D-5)	81					
				NUT (C2950-10)	81					
				SEALANT	1	FMS-1044				

- 55 SKIN
- 645 SPAN (A)
- 800 SPAN (A)
- 800 SPAN (B)

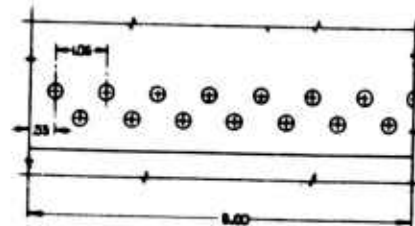


SECTION B-B
TYR OF -89, -197, & 195 SPAN ASSY

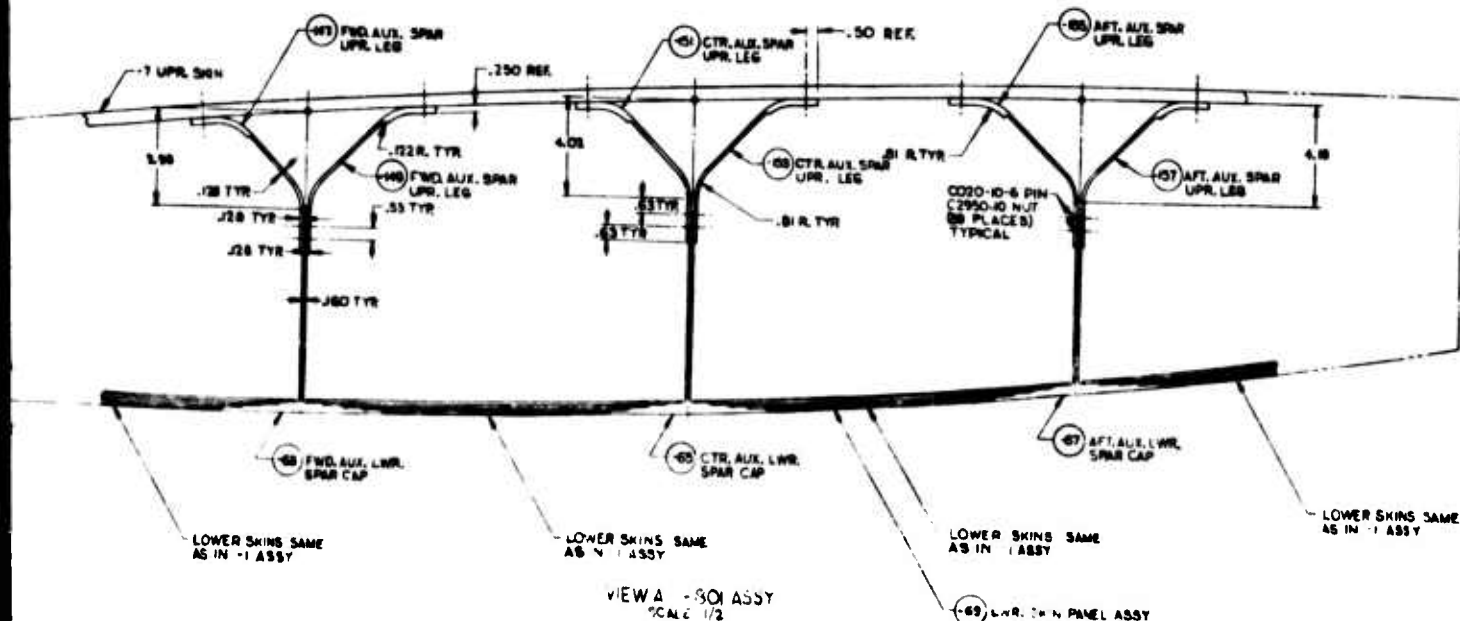


VIEW A - 803 ASSY
SCALE: 1/2

- 193 SKIN
- 194 SKIN
- 195 SKIN
- 196 SKIN
- 197 SKIN
- 198 SKIN
- 199 SKIN



INTERMEDIATE SPAN ATTACHMENTS FOR
801 ASSY



VIEW A - 801 ASSY
SCALE: 1/2

4

TABLE E
DETAILS OF LOWER SURFACE WING SKIN FOR 803 ASSY

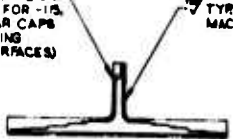
DASH NO.	QTY.	RAW STOCK SIZE	RAW STOCK WT.	MATERIAL	FINISHED STOCK SIZE	FINISHED WT.
① -103	2	.071113-50	5.58	2024 T81	.071113-48	5.28
② -105	2	.071113-50	4.98		.071113-48	4.71
③ -107	2	.071113-50	4.61		.071113-48	4.38
④ -109	2	.063157-50	5.40		.063157-48	5.20
⑤ -111	2	.063157-50	5.30		.063157-48	5.15
⑥ -113	2	.063157-50	7.38		.063157-48	7.13
⑦ -121	2	.063157-50	7.35		.063157-48	7.13
⑧ -123	2	.063157-50	5.26		.063157-48	5.11
⑨ -125	2	.063157-50	5.87		.063157-48	5.70
⑩ -127	2	.071113-50	4.87		.071113-48	4.68
⑪ -129	2	.071113-50	4.88	.071113-48	4.71	
⑫ -131	2	.071113-50	5.35	.071113-48	5.26	

DETAILS OF LOWER SURFACE WING SKIN FOR 803 ASSY

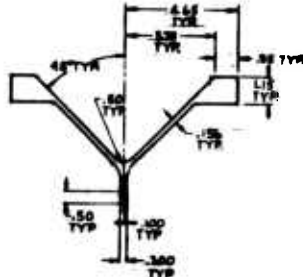
-203	1	.063155.90X50	17.78	2024 T81	.063155.70X48	17.01
-205	1	.063157.50X50	18.29		.063157.30X48	17.50
-207	1	.063159.10X50	18.80		.063159.00X48	17.99
-209	1	.090160.70X50	27.59		.090160.50X48	26.40
-211	1	.090162.30X50	28.32		.090162.10X48	27.10
-213	1	.090163.50X50	28.84		.090163.32X48	27.68

.10 EXCESS MAT ON EXTRUSIONS FOR -115, -117, & -119 SPAR CAPS FOR MACHINING (TYR ALL SURFACES)

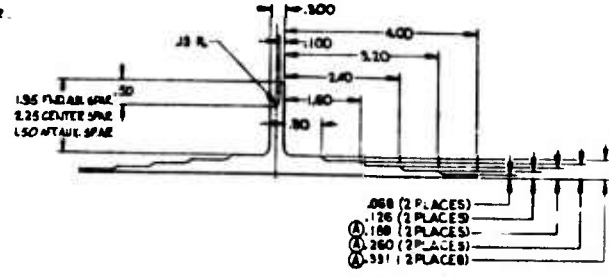
.07 TYR FOR ALL MACHINED SURFACES



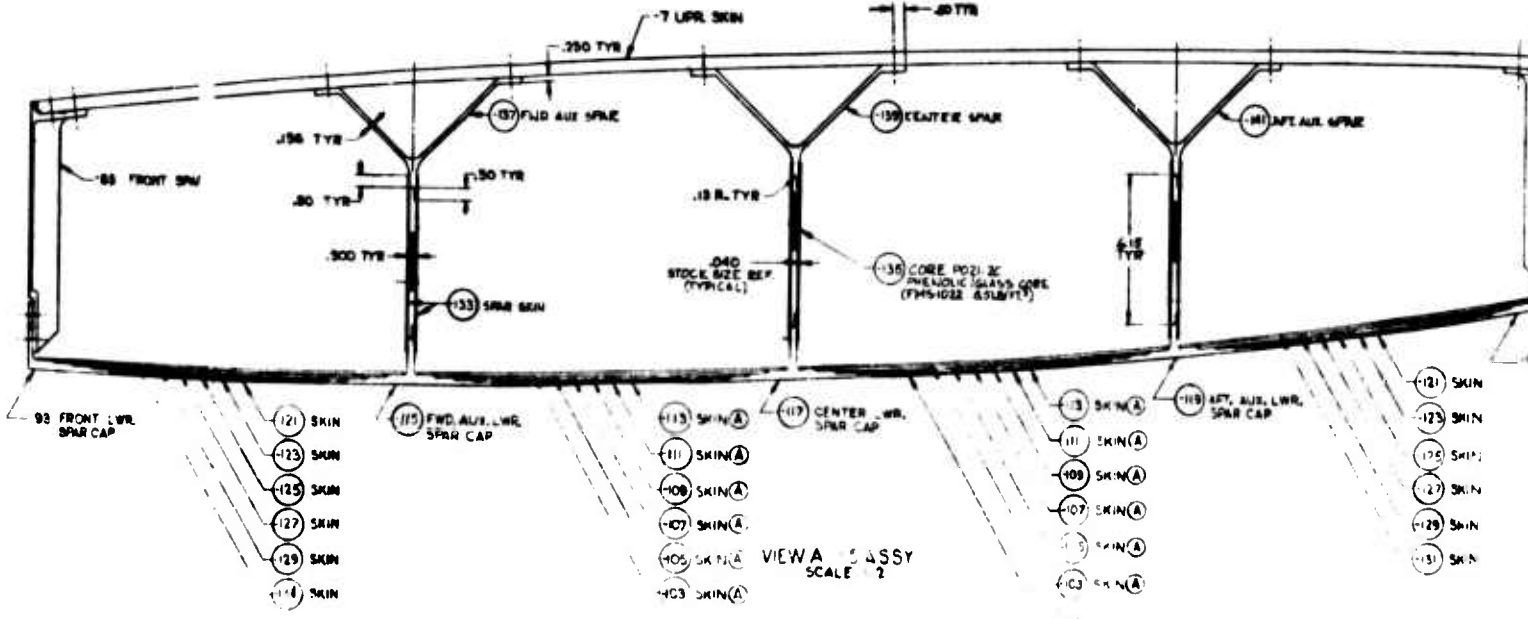
-145 EXTRUSION DETAIL
TYPICAL -115, 117, 119



-143 EXTRUSION DETAIL
TYPICAL -115, 117, 119



DETAIL -115, -117, & -119
(FINISHED LENGTH IS 48.00)



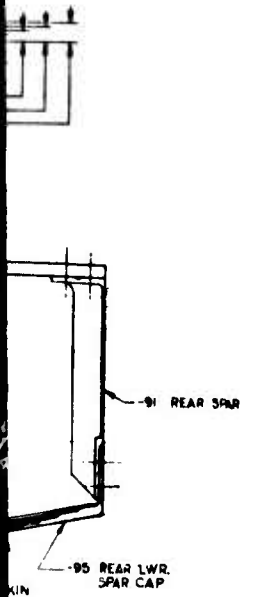
VIEW A ASSY
SCALE 2

LIST OF MATERIALS

ASSY	PART NO.	DETAIL	PART NAME	NO. REQD.	MATERIAL	PAN. THICK.	THICK. DIM.	THICK. UNIT	UNIT WT.	COST	TOTAL COST
1			ANALYTICAL ASSY	1					347.58	15789	5336
			UPPER SKIN	1	SEE - ASSY				18.78	178.40	337.04
			FRONT SPAR	1	SEE - ASSY				2.84	240.99	243.83
			REAR SPAR	1	SEE - ASSY				2.78	238.25	241.03
			LWR SKIN PANEL ASSY	1						188.4	153.8
			FRONT LWR SPAR CAP	1	SEE - ASSY				3.13		184.88
			SPAR EXTRU.	1							107.78
			FRONT LWR SPAR CAP	1	2024-T8H1				9.84		271.48
			SPAR EXTRU.	1		EXTRU 5490 16-40	31.52				138.71
			CENTER SPAR CAP	1					10.10		271.49
			SPAR EXTRU.	1		EXTRU 5490 16-40	31.52				138.71
			AFT. AUX. SPAR CAP	1					9.18		271.49
			SPAR EXTRU.	1		EXTRU 5490 16-40	31.52				138.71
			REAR LWR SPAR CAP	1	SEE - ASSY				3.13		184.88
			SPAR EXTRU.	1							107.78
			SKIN	1	SEE TABLE I				44.88		95.76
			SKIN	1	SEE TABLE I				44.82		84.94
			SKIN	2	SEE TABLE I				44.82		179.50
			SKIN	2	SEE TABLE I				44.82		179.50
			SPAR SKINS	6	2024-T81	240X040X170	1.40	PT	1.40		16.72
			COBE	3	SEE IN DETAIL	10161235154		PT	21.97		49.71
			FRONT AUX SPAR	1	MAKE FROM -45				10.84		216.77
			SPAR EXTRU.	1	2024-T8H1	EXTRU 5490 16-40	30.00				123.88
		CENTER SPAR	1	MAKE FROM -45				13.33		303.81	
		SPAR EXTRU.	1	2024-T8H1	EXTRU 5490 16-40	30.00				123.88	
		AFT. AUX SPAR	1	MAKE FROM -45				13.70		303.81	
		SPAR EXTRU.	1	2024-T8H1	EXTRU 5490 16-40	30.00				123.88	
		ADHESIVE	1	FMS-1013 (SEE NOTE 3)				10.91		23.56	
		BLIND FASTENER	410					6.88			
		MS 30353-0812	410					6.88			
		PNIC 020-B-6	192					1.11			
		NUTIC 2950-B	192					1.11			
		SEALANT	1	FMS-1044 (SEE NOTE 1)				1.75		1.50	
		MATERIAL REMOVED FOR FASTENER HOLES						2.71			
									9557	2346	

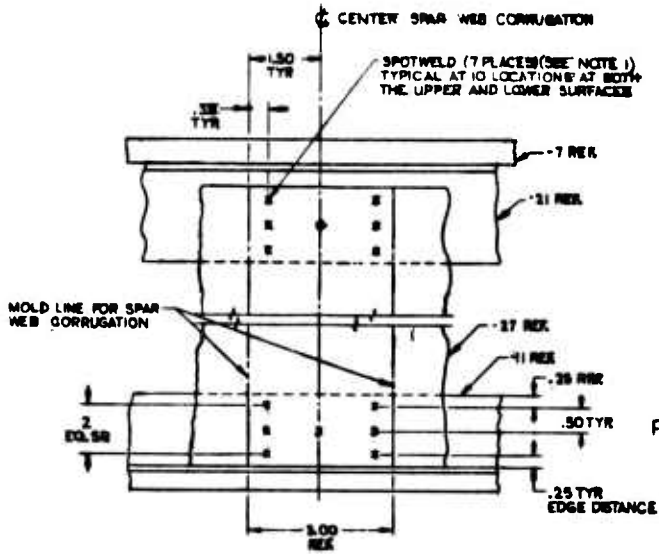
801			ANALYTICAL ASSY	1					343.05	2503	192	51066
			UPPER SKIN	1	SEE - ASSY				138.78	379.35	307.44	
			FRONT SPAR	1	SEE - ASSY				8.83	240.99	249.82	
			FRONT LWR SPAR CAP	1	2024-T81	25017.08150	8.89		4.54			
			FRONT LWR SPAR CAP	1	2024-T81	25018.02150	10.13		5.15			
			CTR. AUX. SPAR UPR LEG	1	2024-T81	25017.57150	9.30		4.75			
			CTR. AUX. SPAR UPR LEG	1	2024-T81	25017.89150	9.96		5.07			
			CTR. AUX. SPAR UPR LEG	1	2024-T81	25018.02150	10.13		5.15			
			AFT. AUX. SPAR UPR LEG	1	2024-T81	25017.94150	10.03		5.10			
			REAR SPAR	1	SEE - ASSY				8.78	239.25	248.03	
			LWR SKIN PANEL ASSY	1						180.00	152.53	
			FRONT LWR SPAR CAP	1	SEE - ASSY				3.13		184.88	
			SPAR EXTRU.	1							107.78	
			FRONT LWR SPAR CAP	1	2024-T8H1	EXTRU 5490 16-40	40.36				184.88	
			SPAR EXTRU.	1					13.77		255.15	
			CTR. AUX. SPAR CAP	1		EXTRU 5490 16-40	41.45				180.40	
			SPAR EXTRU.	1					13.02		285.10	
			AFT. AUX. SPAR CAP	1	2024-T8H1	EXTRU 5490 16-40	59.81				172.56	
			REAR LWR SPAR CAP	1	SEE - ASSY				3.13		184.88	
		SPAR EXTRU.	1							107.78		
		SKIN	1	SEE TABLE I				91.28		135.31		
		SKIN	1	SEE TABLE I				91.28		135.31		
		ADHESIVE	1	FMS-1013 (SEE NOTE 3)				6.25		15.06		
		BLIND FASTENER	410					6.88				
		MS 30353-0812	410					6.88				
		PNIC 020-D-7	167					1.00				
		NUTIC 2950-D	167					1.00				
		PNIC 020-B-6	192					3.83				
		NUTIC 2950-B	192					1.50				
		SEALANT	1	FMS-1044 (SEE NOTE 1)				1.75		1.50		
		MATERIAL REMOVED FOR FASTNER HOLES						3.56				
									9557	3238		

COPY AVAILABLE TO CDC DOES NOT PERMIT FULLY LEGIBLE PRODUCTION



PRELIMINARY DESIGN DRAWING
 MACHINED IN 316 STAINLESS STEEL
 EXTRUDED ETCHED ALUMINUM PARTS
 ALUMINUM ANALYTICAL ASSY
 GENERAL DYNAMICS CORPORATION
 2400 A
 GENERAL Dynamics Division

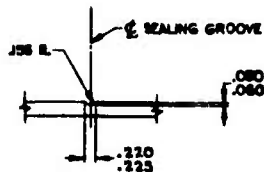
Figure 39 Extruded & Etched Y-Spar Analytical Assembly (Continued)



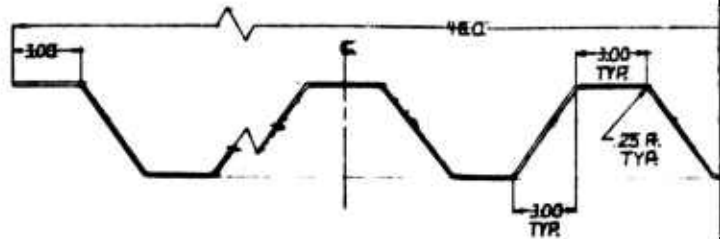
SECTION B-B

SPOTWELD PATTERN SHOWN IS TYPICAL AT THE FWD. AUX. AND AFT. AUX. SPARS

FULL SIZE



FUEL SEALING GROOVE DETAIL
TYR FOR -9, -17
FULL SIZE

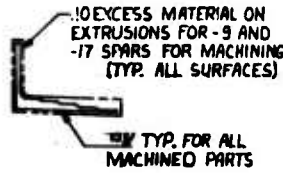


HEIGHT OF CORRUGATED WEBS*

- 25, h=11
- 27, h=11.4
- 29, h=10.6

* REFER TO END VIEW -1 ASSY
DETAIL OF -25, -27, -29 CORRUGATED WEB

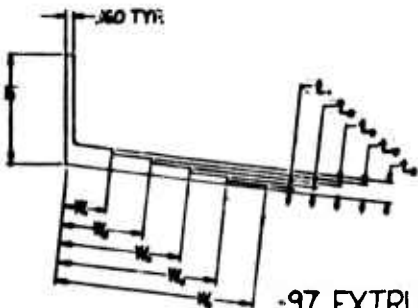
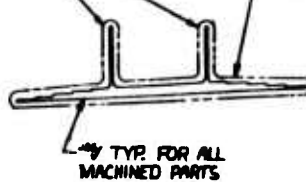
DIMENSION (IN INCHES)	FWD. AUX. SPAR (-33)	CENT. SPAR (-35)	AFT. AUX. SPAR (-37)
h	2.89	2.89	2.89
c	.063	.063	.063
e	.126	.126	.126
f	.189	.189	.189
g	.250	.250	.250
h	.331	.331	.331
i	.063	.063	.063
j	.126	.126	.126
k	.189	.189	.189
l	.250	.250	.250
m	.331	.331	.331
n	2.8	2.8	2.8
o	3.6	3.6	3.6
p	4.4	4.4	4.4
q	5.2	5.2	5.2
r	2.8	2.8	2.8
s	3.6	3.6	3.6
t	4.4	4.4	4.4
u	5.2	5.2	5.2
v (REF.)	12.0	12.0	12.0



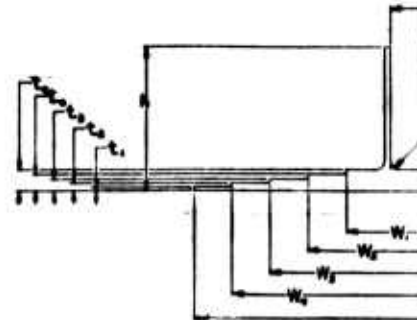
DIMENSION (IN INCHES)	FWD. SPAR (-9)	AFT. SPAR (-17)
h	2.4	2.4
c	.063	.063
e	.126	.126
f	.189	.189
g	.250	.250
h	.331	.331
i	.36	.36
j	1.76	1.76
k	2.36	2.36
l	2.36	2.36
m	4.16	4.16

.25 EXCESS MATERIAL ON EXTRUSION

.10 EXCESS MATERIAL ON EXTRUSION FOR -33, -35 AND -37 SPARS FOR MACHINING

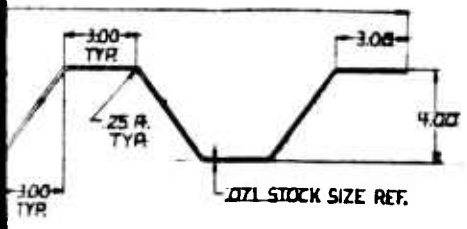


-97 EXTRU. DETAIL OF -31 AND -39 SPAR CAPS
FULL SIZE



-95 EXTRU. DETAIL

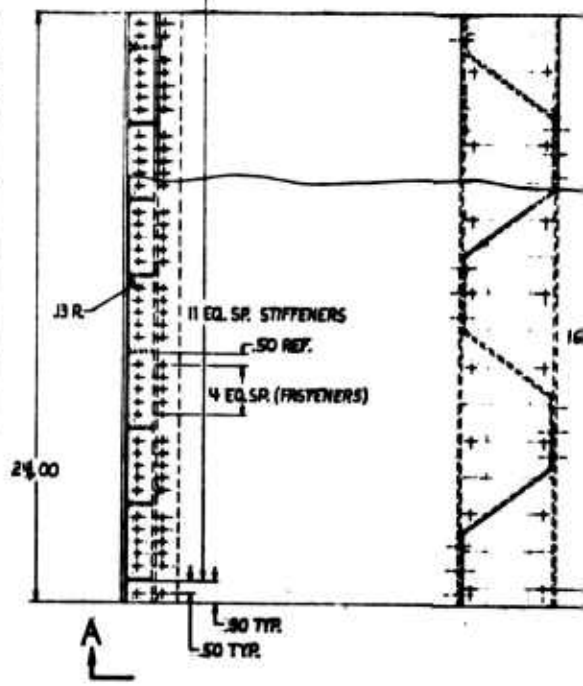
2



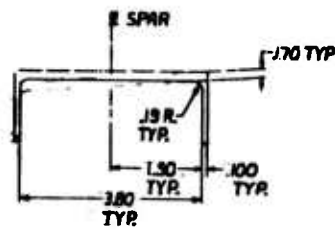
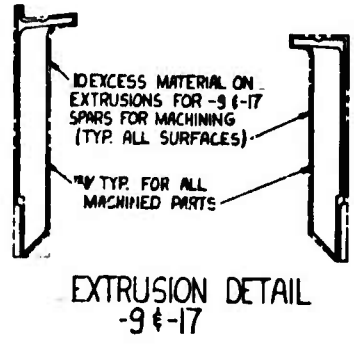
WEBS
CORRUGATED WEBS

TABLE I
DETAIL OF LOWER SURFACE WING SKIN

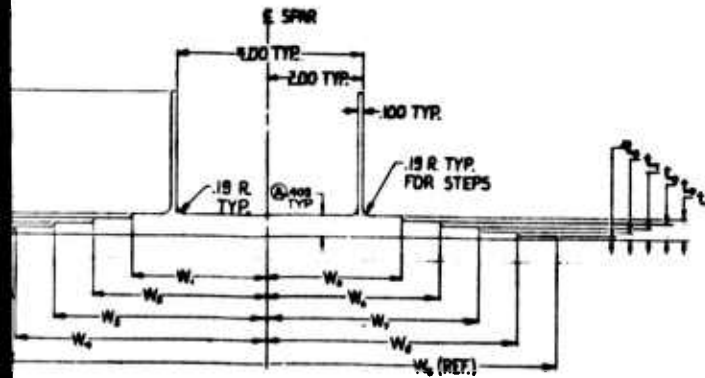
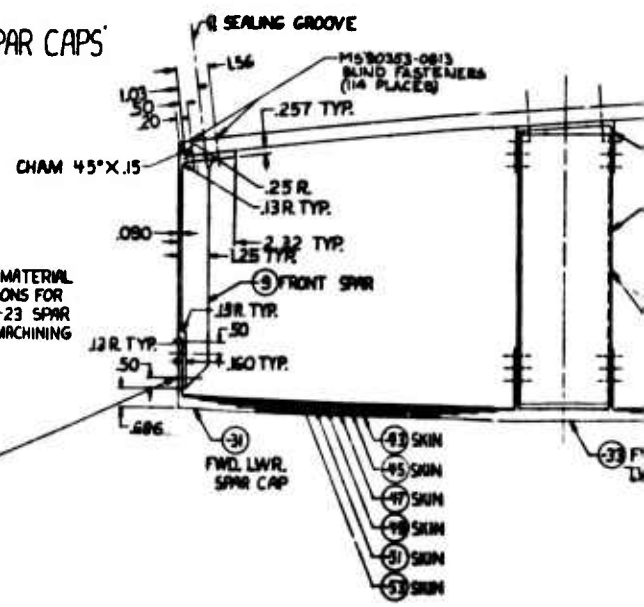
DASH NO.	RAW STOCK SIZE	RAW STOCK WT.	MATERIAL	FINISHED STOCK SIZE	FINISHED WT. IN. (LBS)
-42	.062 X 5.6 X .50	1.78	2024-T81	.062 X 5.43 X .48	1.68
-45	.063 X 7.0 X .50	2.23	2024-T81	.063 X 6.83 X .48	2.18
-46	.071 X 10.4 X .50	3.78	2024-T81	.071 X 10.23 X .48	3.68
-51	.071 X 12.6 X .50	4.30	2024-T81	.071 X 12.4 X .48	4.22
-53	.071 X 12.6 X .50	4.30	2024-T81	.071 X 12.4 X .48	4.22
-55	.063 X 3.8 X .50	1.21	2024-T81	.063 X 3.55 X .48	1.09
-57	.063 X 5.4 X .50	1.72	2024-T81	.063 X 5.13 X .48	1.58
-58	.063 X 7.0 X .50	2.23	2024-T81	.063 X 6.73 X .48	2.07
-61	.071 X 8.6 X .50	3.08	2024-T81	.071 X 8.33 X .48	2.89
-63	.071 X 10.2 X .50	3.66	2024-T81	.071 X 9.93 X .48	3.44
-65	.071 X 11.8 X .50	4.23	2024-T81	.071 X 11.53 X .48	3.99
-56	.063 X 3.8 X .50	1.21	2024-T81	.063 X 3.53 X .48	1.09
-57	.063 X 5.4 X .50	1.72	2024-T81	.063 X 5.13 X .48	1.58
-59	.063 X 7.0 X .50	2.23	2024-T81	.063 X 6.73 X .48	2.07
-61	.071 X 8.6 X .50	3.08	2024-T81	.071 X 8.33 X .48	2.89
-63	.071 X 10.2 X .50	3.66	2024-T81	.071 X 9.93 X .48	3.44
-65	.071 X 11.8 X .50	4.23	2024-T81	.071 X 11.53 X .48	3.99
-43	.063 X 6.6 X .50	1.78	2024-T81	.063 X 6.43 X .48	1.68
-47	.063 X 7.2 X .50	2.23	2024-T81	.063 X 7.03 X .48	2.18
-48	.071 X 10.4 X .50	3.73	2024-T81	.071 X 10.23 X .48	3.63
-51	.071 X 12.0 X .50	4.30	2024-T81	.071 X 11.83 X .48	4.07
-53	.071 X 12.6 X .50	4.30	2024-T81	.071 X 12.4 X .48	4.22



- AUX. SPAR
-37)
- 2.88
 - .068
 - .124
 - .189
 - .260
 - .331
 - .403
 - .474
 - .545
 - .616
 - .687
 - .758
 - .829
 - .900
 - .971
 - 1.042
 - 1.113
 - 1.184
 - 1.255
 - 1.326
 - 1.397
 - 1.468
 - 1.539
 - 1.610
 - 1.681
 - 1.752
 - 1.823
 - 1.894
 - 1.965
 - 2.036
 - 2.107
 - 2.178
 - 2.249
 - 2.320
 - 2.391
 - 2.462
 - 2.533
 - 2.604
 - 2.675
 - 2.746
 - 2.817
 - 2.888
 - 2.959
 - 3.030
 - 3.101
 - 3.172
 - 3.243
 - 3.314
 - 3.385
 - 3.456
 - 3.527
 - 3.598
 - 3.669
 - 3.740
 - 3.811
 - 3.882
 - 3.953
 - 4.024
 - 4.095
 - 4.166
 - 4.237
 - 4.308
 - 4.379
 - 4.450
 - 4.521
 - 4.592
 - 4.663
 - 4.734
 - 4.805
 - 4.876
 - 4.947
 - 5.018
 - 5.089
 - 5.160
 - 5.231
 - 5.302
 - 5.373
 - 5.444
 - 5.515
 - 5.586
 - 5.657
 - 5.728
 - 5.799
 - 5.870
 - 5.941
 - 6.012
 - 6.083
 - 6.154
 - 6.225
 - 6.296
 - 6.367
 - 6.438
 - 6.509
 - 6.580
 - 6.651
 - 6.722
 - 6.793
 - 6.864
 - 6.935
 - 7.006
 - 7.077
 - 7.148
 - 7.219
 - 7.290
 - 7.361
 - 7.432
 - 7.503
 - 7.574
 - 7.645
 - 7.716
 - 7.787
 - 7.858
 - 7.929
 - 8.000
 - 8.071
 - 8.142
 - 8.213
 - 8.284
 - 8.355
 - 8.426
 - 8.497
 - 8.568
 - 8.639
 - 8.710
 - 8.781
 - 8.852
 - 8.923
 - 8.994
 - 9.065
 - 9.136
 - 9.207
 - 9.278
 - 9.349
 - 9.420
 - 9.491
 - 9.562
 - 9.633
 - 9.704
 - 9.775
 - 9.846
 - 9.917
 - 9.988
 - 10.059
 - 10.130
 - 10.201
 - 10.272
 - 10.343
 - 10.414
 - 10.485
 - 10.556
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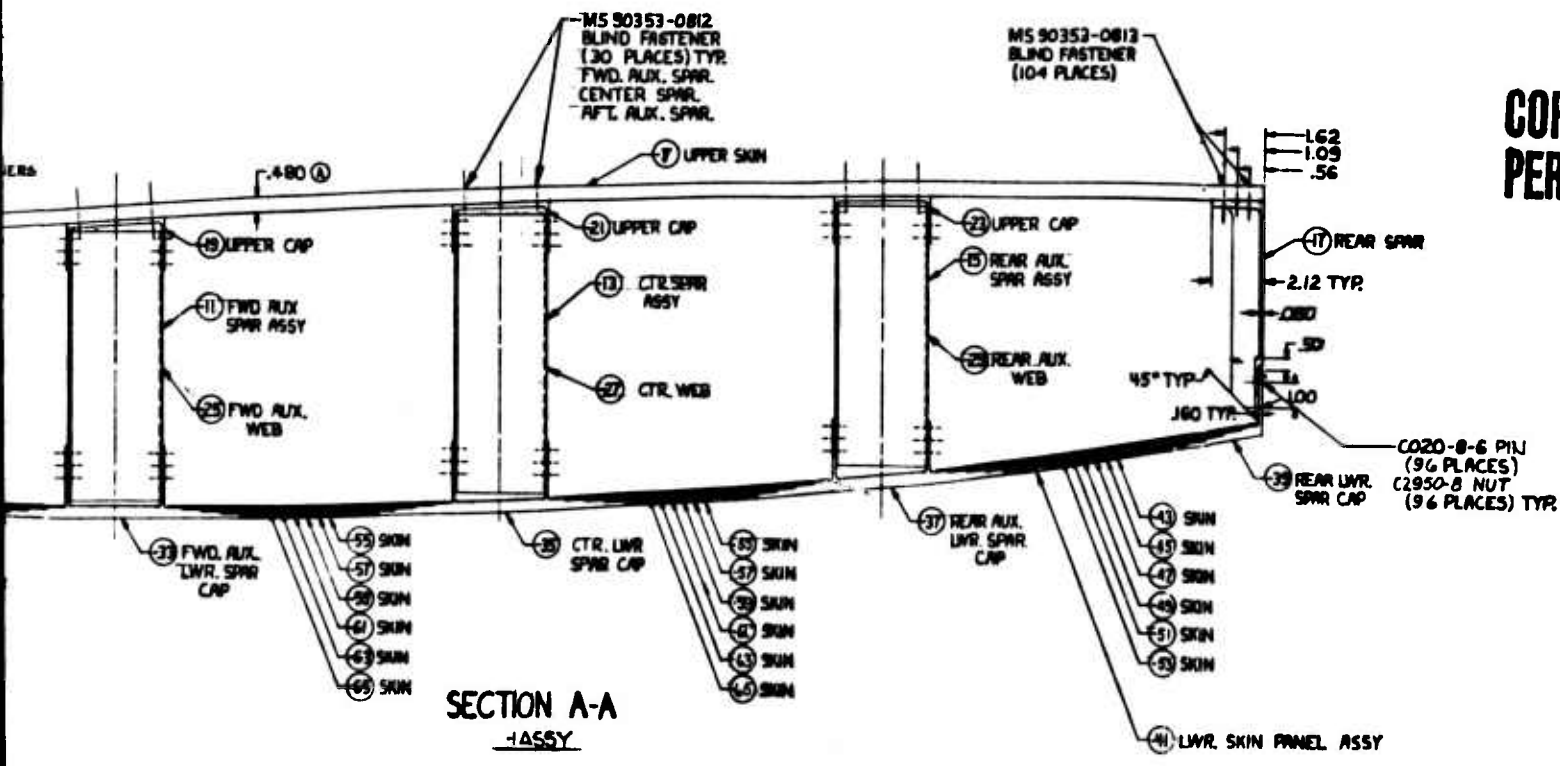
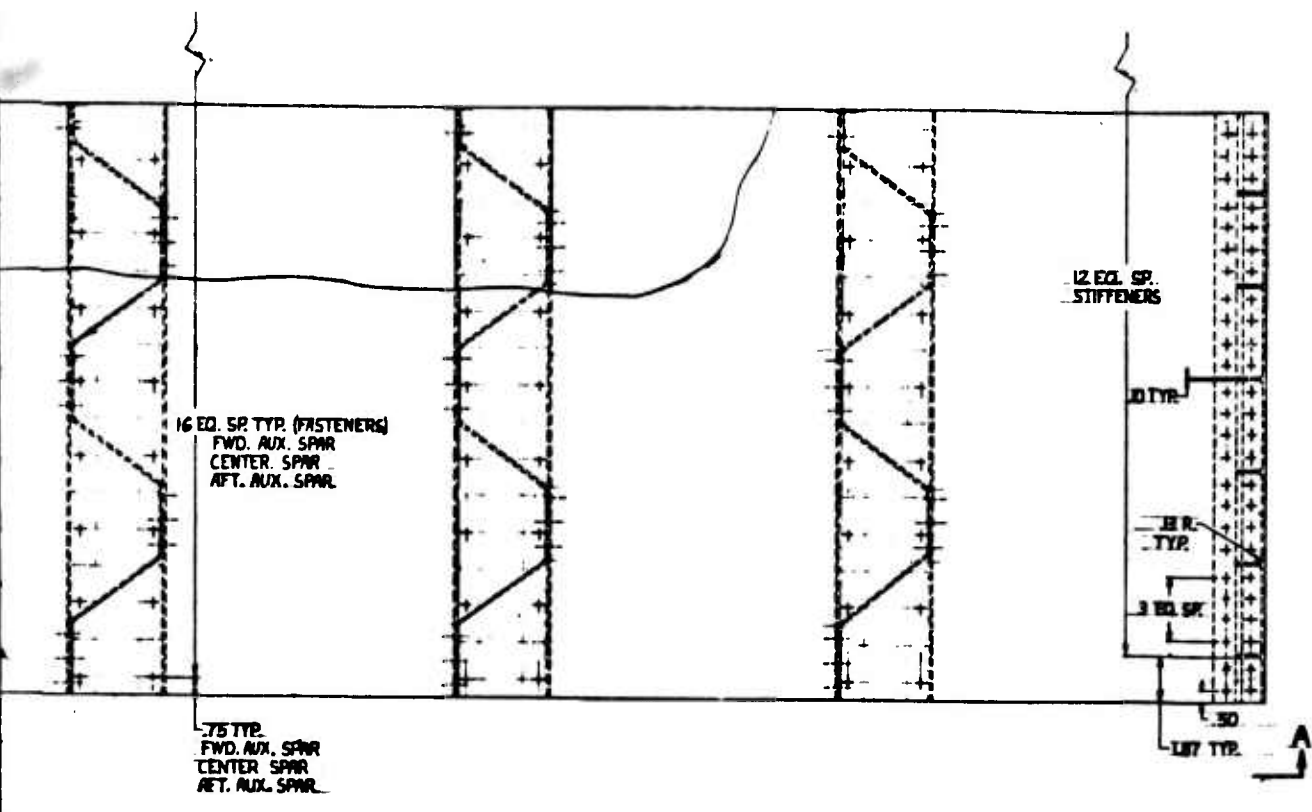
-91 EXTRU. DETAIL OF -19, -21 AND -23 SPAR CAPS
(-23 ROTATED 180°)
FULL SIZE



EXTRU. DETAIL OF -33, -35 AND -37 SPAR CAPS
FULL SIZE

3

ASBY	SUB ASBY	DETAIL	PART NAME
-1			SMALL TYPICAL ASSY
-7			UPPER CAP
-8			FRONT LWR CAP
-17			REAR SPAR
-11			FWD AUX SPAR
			ADHESIVE
-19			UPPER CAP
-21			CAP EXTRUSION
-22			FWD AUX WEB
			ADHESIVE
-18			CENTER SPAR
-21			UPPER CAP
-22			CAP EXTRUSION
-23			CENTER WEB
			ADHESIVE
-19			AFT. AUX SPAR
-23			UPPER CAP
-24			CAP EXTRUSION
-25			AFT. AUX WEB
			ADHESIVE
-81			LOWER SKIN
			PANEL ASSY
-31			FRONT LWR SPAR CAP
-37			FRONT SPAR EXTRU
-38			FWD AUX LWR SPAR CAP
-39			FWD AUX CAP EXTRU
-40			CTR LWR SPAR CAP
-41			CTR CAP EXTRU
-42			AFT. AUX LWR SPAR CAP
-43			AFT. AUX CAP EXTRU
-44			REAR LWR SPAR CAP
-45			REAR SPAR EXTRU
-46			SKIN
-47			SKIN
-48			SKIN
-49			SKIN
-50			SKIN
-51			SKIN
-52			SKIN
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-300			SKIN



**COPY AVAILAB
PERMIT FULLY**

SERIALS										
QTY	PART NAME	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED W.C.		COST			TOTAL COST \$
					NET WT	ASSY	MAINT	PRG	TOL	
1	FRONT CAP	2024-T351	4.00	148.0	35.93	1033	6407	212	5546	
1	FRONT CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	FRONT CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	UPPER CAP	2024-T351	1.00	37.50	24.00					
1	UPPER CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	UPPER CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	CENTER SPAR	2024-T351	1.00	11.40						
1	UPPER CAP	2024-T351	1.00	37.50	24.00					
1	UPPER CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	UPPER CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	LOWER SKIN	2024-T351	1.00	149.20	33.56					
1	FRONT CAP	2024-T351	1.00	37.50	24.00					
1	FRONT CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	FRONT CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	LOWER SKIN	2024-T351	1.00	149.20	33.56					
1	FRONT CAP	2024-T351	1.00	37.50	24.00					
1	FRONT CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	FRONT CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	REAR CAP	2024-T351	1.00	37.50	24.00					
1	REAR CAP EXTENSION	2024-T351	1.00	37.50	24.00					
1	REAR CAP WEB	2024-T351	1.00	28.00	24.00					
1	ADHESIVE	FMS-1013	1.00	11.34						
1	SKIN	SEE TABLE I		87.44	136.15		73.50			
1	ADHESIVE	FMS-1013, FORM I A (SEE NOTE 3)		6.81						
1	BLIND FASTENER	M8-2032-0818	90	1.31						
1	BLIND FASTENER	M8-2032-0818	218	2.86		872				
1	BLIND FASTENER	M8-2032-0818	133	1.11						
1	BLIND FASTENER	M8-2032-0818	133	1.11						
1	SEALANT	FMS-1041 (SEE NOTE 1)	1.75	1.80						
1	ADHESIVE	FMS-1041 (SEE NOTE 1)	2.19							

CHANGED 7 THICKNESS FROM .50 TO .40
 PART 440
 CHANGED LOWER SKIN THICKNESS FROM .40 TO .402
 REV. LD ALL .402-PTS IN L/W-OT REFLECT 7 UPPER SKIN THICKNESS CHANGE (LOWER SKIN THICKNESS CHANGE) - CLONES LWR-SKIN-CAP.

AVAILABLE TO DDC DOES NOT FULLY LEGIBLE PRODUCTION

- USE FMS-1013 ADHESIVE FORM I A (RELIABOND 306) IN THE FOLLOWING AREAS:
 (1) BETWEEN ALL LOWER SURFACE SKINS AND ADJACENT LWR. SPAR CAPS/ SPARS IN THE -A1 ASSY
 (2) BETWEEN ALL INTERN. SPAR WEBS (ADJACENT UPL SPAR CAPS IN THE -11, -13, & -15 AREAS)

2. 40° FINISH REQUIRED ON ALL MACHINED PARTS.

- THE FINISH SURFACES AT THE UPPER SURFACE OF THE FRONT AND REAR SPAR, THE UPPER SKIN AND ALL FASTENER SHALL BE SEALED PER FMS-1004.

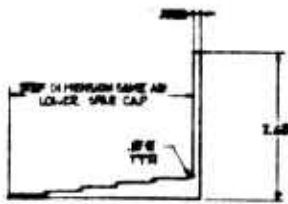
NOTES:

PRELIMINARY DESIGN DRAWING

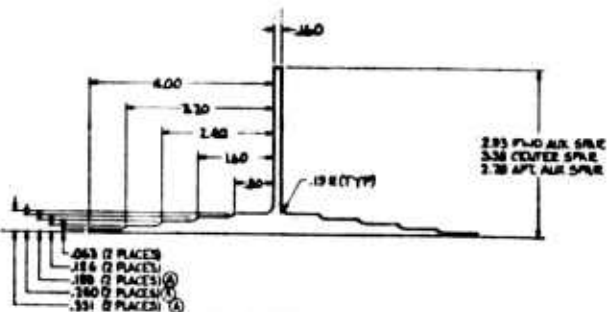
ATW - 4 WING - LAMINATED LWR. SKIN; MACHINED UPR SKIN & CLOSURE SPARS; CORRUGATED INTERN. SPARS WITH EXTRUDED CAP (ALUMINUM); ANALYTICAL ASSY.

GENERAL DYNAMICS
 Convair Aerospace Division
 633-RA002 A

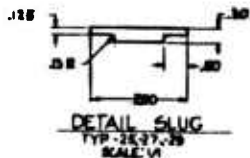
Figure 40 Corrugated Spar Analytical Assembly



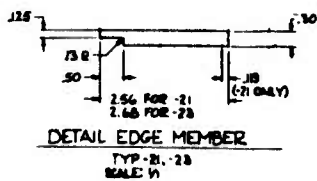
FRONT/REAR LOWER SPAR CAP
TYP - 88, -87
SCALE: 1/1



DETAIL LOWER SPAR CAP
TYP - 89, -91, -93
SCALE: 1/1



DETAIL SLUG
TYP - 25, 27, -28
SCALE: 1/1



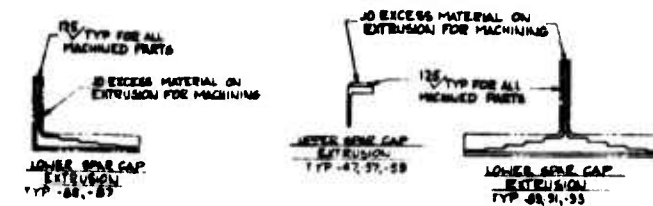
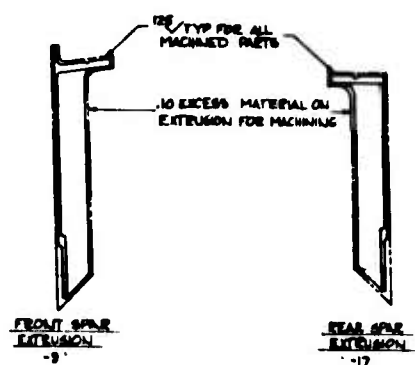
DETAIL EDGE MEMBER
TYP - 21, -23
SCALE: 1/1

TABLE I
DETAILS OF LOWER SURFACE SANDWICH CORE AND DETAILS OF INTERMEDIATE SPAR CORE

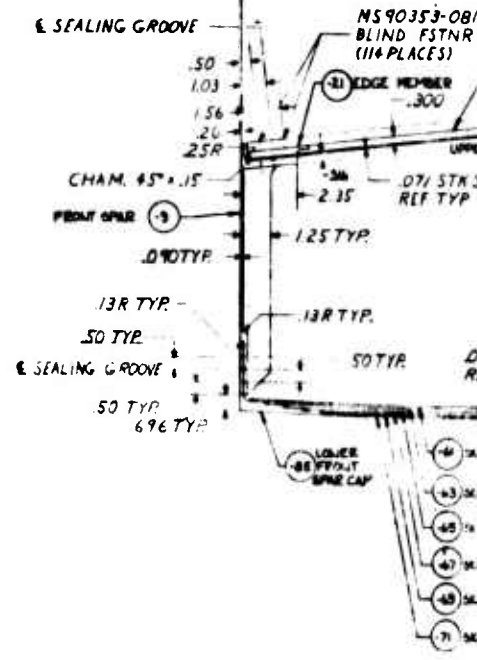
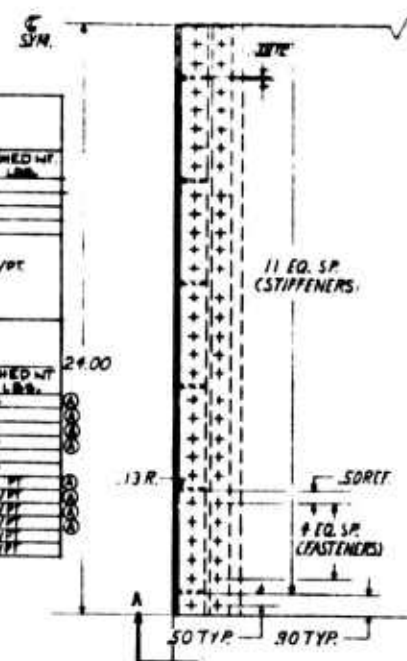
DASH NO.	REQD.	RAW STOCK SIZE	RAW STOCK WT	MATERIAL	FINISHED STOCK SIZE	FINISHED WT IN LBS.
-31		300x11x.54	1.8	PO21-10	300x11x.48	.88
-32		300x4.57x.54	1.9	AL CORE	300x4.57x.48	1.01
-33		300x4.57x.54	1.9	FMB-1070A	300x4.57x.48	1.01
-37		300x13.29x.54	1.9	B.1 LB/FT ²	300x13.29x.48	.80
-46	3	300x148x.54	.82/PT	PO21-10 PHENOLIC/BARD CORE FMB-1071 B.4 LB/FT ²	300x148x.48	.82/PT

TABLE II
DETAILS OF LOWER SURFACE HING BUSH

DASH NO.	REQD.	RAW STOCK SIZE	RAW STOCK WT	MATERIAL	FINISHED STOCK SIZE	FINISHED WT IN LBS.
-21		271x1.54x.50	5.93	3074-Y81	271x1.54x.48	5.34
-22		271x1.54x.50	5.99		271x1.54x.48	5.41
-23		271x1.54x.50	5.97		271x1.54x.48	5.39
-24		271x1.54x.50	5.97		271x1.54x.48	5.39
-25		271x1.54x.50	5.97		271x1.54x.48	5.39
-26		271x1.54x.50	5.97		271x1.54x.48	5.39
-27		271x1.54x.50	5.97		271x1.54x.48	5.39
-28		271x1.54x.50	5.97		271x1.54x.48	5.39
-29		271x1.54x.50	5.97		271x1.54x.48	5.39
-30		271x1.54x.50	5.97		271x1.54x.48	5.39
-31		271x1.54x.50	5.97		271x1.54x.48	5.39
-32		271x1.54x.50	5.97		271x1.54x.48	5.39
-33		271x1.54x.50	5.97		271x1.54x.48	5.39
-34		271x1.54x.50	5.97		271x1.54x.48	5.39
-35		271x1.54x.50	5.97		271x1.54x.48	5.39
-36		271x1.54x.50	5.97		271x1.54x.48	5.39
-37		271x1.54x.50	5.97		271x1.54x.48	5.39
-38		271x1.54x.50	5.97		271x1.54x.48	5.39
-39		271x1.54x.50	5.97		271x1.54x.48	5.39
-40		271x1.54x.50	5.97		271x1.54x.48	5.39
-41		271x1.54x.50	5.97		271x1.54x.48	5.39
-42		271x1.54x.50	5.97		271x1.54x.48	5.39
-43		271x1.54x.50	5.97		271x1.54x.48	5.39
-44		271x1.54x.50	5.97		271x1.54x.48	5.39
-45		271x1.54x.50	5.97		271x1.54x.48	5.39
-46		271x1.54x.50	5.97		271x1.54x.48	5.39
-47		271x1.54x.50	5.97		271x1.54x.48	5.39
-48		271x1.54x.50	5.97		271x1.54x.48	5.39
-49		271x1.54x.50	5.97		271x1.54x.48	5.39
-50		271x1.54x.50	5.97		271x1.54x.48	5.39
-51		271x1.54x.50	5.97		271x1.54x.48	5.39
-52		271x1.54x.50	5.97		271x1.54x.48	5.39
-53		271x1.54x.50	5.97		271x1.54x.48	5.39
-54		271x1.54x.50	5.97		271x1.54x.48	5.39
-55		271x1.54x.50	5.97		271x1.54x.48	5.39
-56		271x1.54x.50	5.97		271x1.54x.48	5.39
-57		271x1.54x.50	5.97		271x1.54x.48	5.39
-58		271x1.54x.50	5.97		271x1.54x.48	5.39
-59		271x1.54x.50	5.97		271x1.54x.48	5.39
-60		271x1.54x.50	5.97		271x1.54x.48	5.39
-61		271x1.54x.50	5.97		271x1.54x.48	5.39
-62		271x1.54x.50	5.97		271x1.54x.48	5.39
-63		271x1.54x.50	5.97		271x1.54x.48	5.39
-64		271x1.54x.50	5.97		271x1.54x.48	5.39
-65		271x1.54x.50	5.97		271x1.54x.48	5.39
-66		271x1.54x.50	5.97		271x1.54x.48	5.39
-67		271x1.54x.50	5.97		271x1.54x.48	5.39
-68		271x1.54x.50	5.97		271x1.54x.48	5.39
-69		271x1.54x.50	5.97		271x1.54x.48	5.39
-70		271x1.54x.50	5.97		271x1.54x.48	5.39
-71		271x1.54x.50	5.97		271x1.54x.48	5.39
-72		271x1.54x.50	5.97		271x1.54x.48	5.39
-73		271x1.54x.50	5.97		271x1.54x.48	5.39
-74		271x1.54x.50	5.97		271x1.54x.48	5.39
-75		271x1.54x.50	5.97		271x1.54x.48	5.39
-76		271x1.54x.50	5.97		271x1.54x.48	5.39
-77		271x1.54x.50	5.97		271x1.54x.48	5.39
-78		271x1.54x.50	5.97		271x1.54x.48	5.39
-79		271x1.54x.50	5.97		271x1.54x.48	5.39
-80		271x1.54x.50	5.97		271x1.54x.48	5.39
-81		271x1.54x.50	5.97		271x1.54x.48	5.39
-82		271x1.54x.50	5.97		271x1.54x.48	5.39
-83		271x1.54x.50	5.97		271x1.54x.48	5.39
-84		271x1.54x.50	5.97		271x1.54x.48	5.39
-85		271x1.54x.50	5.97		271x1.54x.48	5.39
-86		271x1.54x.50	5.97		271x1.54x.48	5.39
-87		271x1.54x.50	5.97		271x1.54x.48	5.39
-88		271x1.54x.50	5.97		271x1.54x.48	5.39
-89		271x1.54x.50	5.97		271x1.54x.48	5.39
-90		271x1.54x.50	5.97		271x1.54x.48	5.39
-91		271x1.54x.50	5.97		271x1.54x.48	5.39
-92		271x1.54x.50	5.97		271x1.54x.48	5.39
-93		271x1.54x.50	5.97		271x1.54x.48	5.39
-94		271x1.54x.50	5.97		271x1.54x.48	5.39
-95		271x1.54x.50	5.97		271x1.54x.48	5.39
-96		271x1.54x.50	5.97		271x1.54x.48	5.39
-97		271x1.54x.50	5.97		271x1.54x.48	5.39
-98		271x1.54x.50	5.97		271x1.54x.48	5.39
-99		271x1.54x.50	5.97		271x1.54x.48	5.39
-100		271x1.54x.50	5.97		271x1.54x.48	5.39



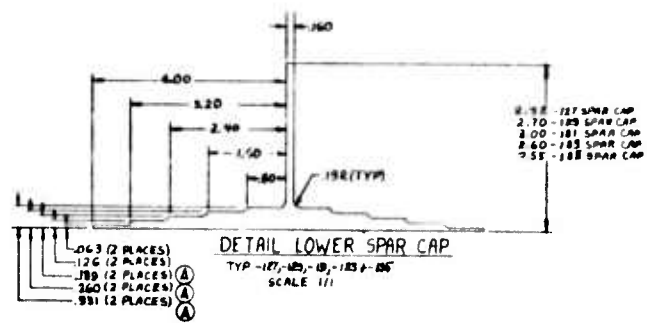
EXTRUSION DETAILS



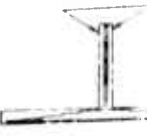
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TABLE II

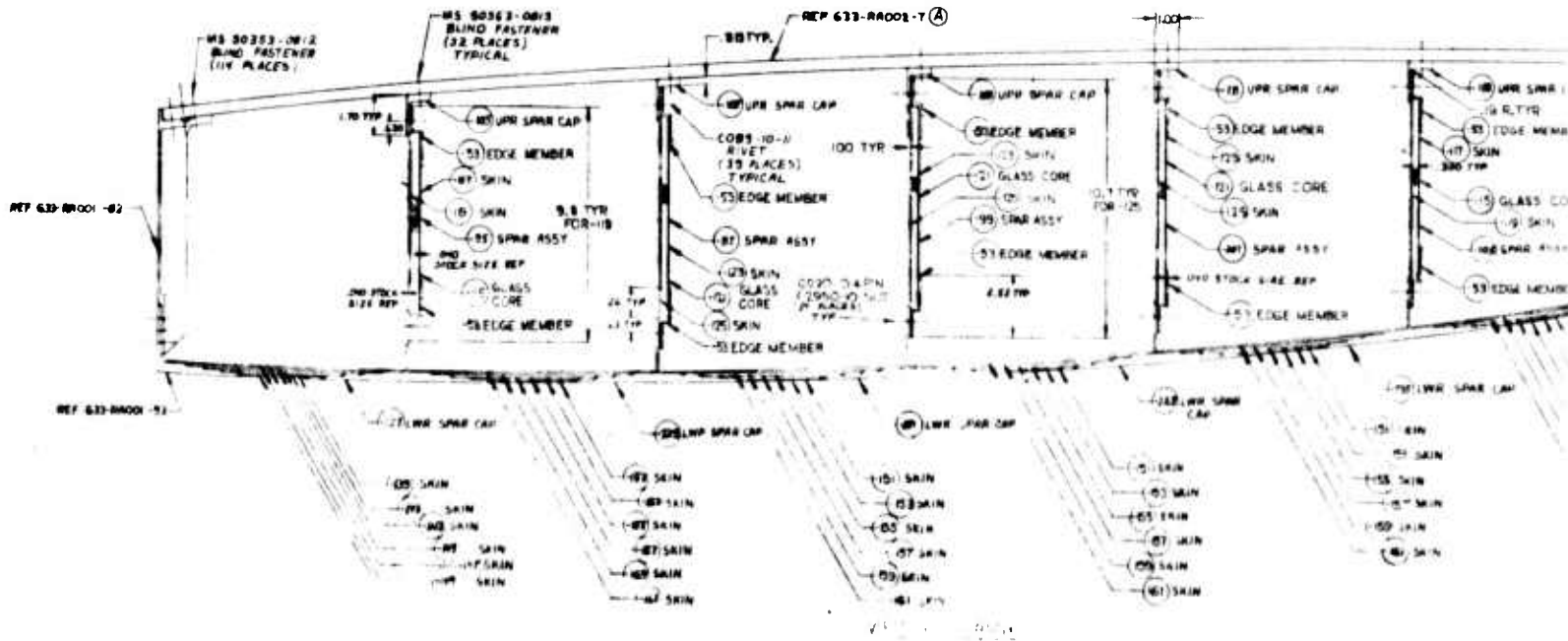
DETAIL NO.	DESCRIPTION	QUANTITY	UNIT	PRICE	TOTAL
159	0612 180	1	EA
161	0631 1850	1	EA
162	0746 2450	1	EA
163	0746 2450	1	EA
164	0631 2450	1	EA
165	0746 2450	1	EA
166	0746 2450	1	EA
167	0746 2450	1	EA
168	0746 2450	1	EA
169	0746 2450	1	EA
170	0746 2450	1	EA
171	0746 2450	1	EA
172	0746 2450	1	EA
173	0746 2450	1	EA



EXTRU DETAIL
-127, 129, 131, 133

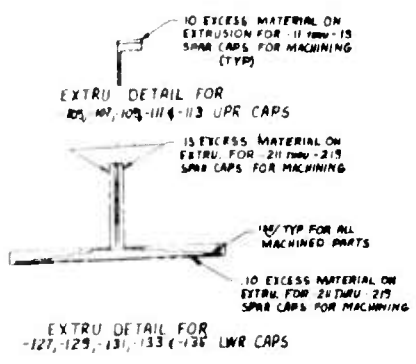


EXTRU DETAIL
-127, 129, 131, 133



Fig

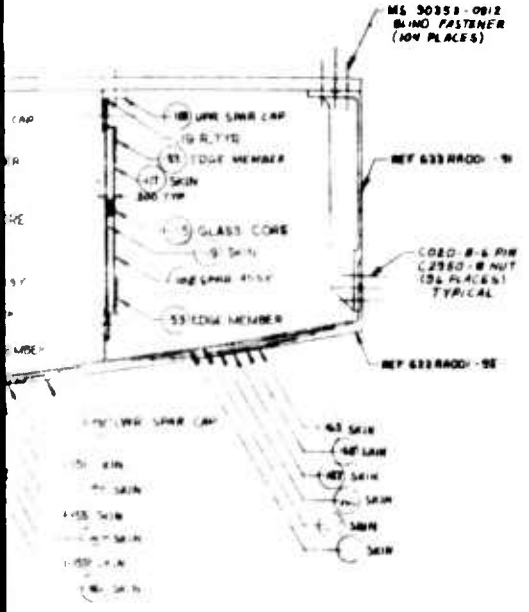
REV	DATE	BY	DESCRIPTION	QTY	WEIGHT
001			INITIAL	1	0.00
002			020771	1	0.00
003			0212 1 XVA	1.5	0.00
004			0213 1 XVA	1.62	0.00
005			0214 1 XVA	2.30	0.00
006			0215 1 XVA	2.93	0.00
007			0216 1 XVA	3.49	0.00
008			0217 1 XVA	6.9	0.00
009			0218 1 XVA	1.10	0.00
010			0219 1 XVA	1.40	0.00
011			0220 1 XVA	2.44	0.00
012			0221 1 XVA	2.99	0.00
013			0222 1 XVA	6.4	0.00
014			0223 1 XVA	1.1	0.00
015			0224 1 XVA	1.59	0.00
016			0225 1 XVA	2.34	0.00
017			0226 1 XVA	2.80	0.00
018			0227 1 XVA	3.44	0.00



NO	QTY	UNIT	DESCRIPTION	WEIGHT	VOLUME	AREA	PERCENT	REMARKS
100	1	ASSEMBLY	SPAR ASSEMBLY	100.00	100.00	100.00	100.00	
101	1	ASSEMBLY	UPPER SKIN	10.00	10.00	10.00	10.00	
102	1	ASSEMBLY	LOWER SKIN	10.00	10.00	10.00	10.00	
103	1	ASSEMBLY	GLASS CORE	10.00	10.00	10.00	10.00	
104	1	ASSEMBLY	EDGE MEMBER	10.00	10.00	10.00	10.00	
105	1	ASSEMBLY	SPAR CAP	10.00	10.00	10.00	10.00	
106	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
107	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
108	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
109	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
110	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
111	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
112	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
113	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
114	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
115	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
116	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
117	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
118	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
119	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
120	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
121	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
122	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
123	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
124	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
125	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
126	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
127	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
128	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
129	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
130	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
131	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
132	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
133	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
134	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
135	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
136	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
137	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
138	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
139	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
140	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
141	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
142	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
143	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
144	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
145	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
146	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
147	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
148	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
149	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
150	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
151	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
152	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
153	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
154	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
155	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
156	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
157	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
158	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
159	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
160	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
161	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
162	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
163	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
164	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
165	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
166	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
167	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
168	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
169	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
170	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
171	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
172	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
173	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
174	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
175	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
176	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
177	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
178	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
179	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
180	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
181	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
182	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
183	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
184	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
185	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
186	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
187	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
188	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
189	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
190	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
191	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
192	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
193	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
194	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
195	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
196	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	
197	1	ASSEMBLY	REINFORCEMENT	10.00	10.00	10.00	10.00	
198	1	ASSEMBLY	FASTENER	10.00	10.00	10.00	10.00	
199	1	ASSEMBLY	ADHESIVE	10.00	10.00	10.00	10.00	
200	1	ASSEMBLY	SEALANT	10.00	10.00	10.00	10.00	

COPY AVAILABLE TO DBC DOES NOT PERMIT FULLY LEGIBLE PRODUCTION

NOTES: FOR DETAILS OF MATERIAL SPECS OF .115 & .121 GLASS CORE SEE 'ASSEMBLY' TABLE 2, -48, SHEET 1

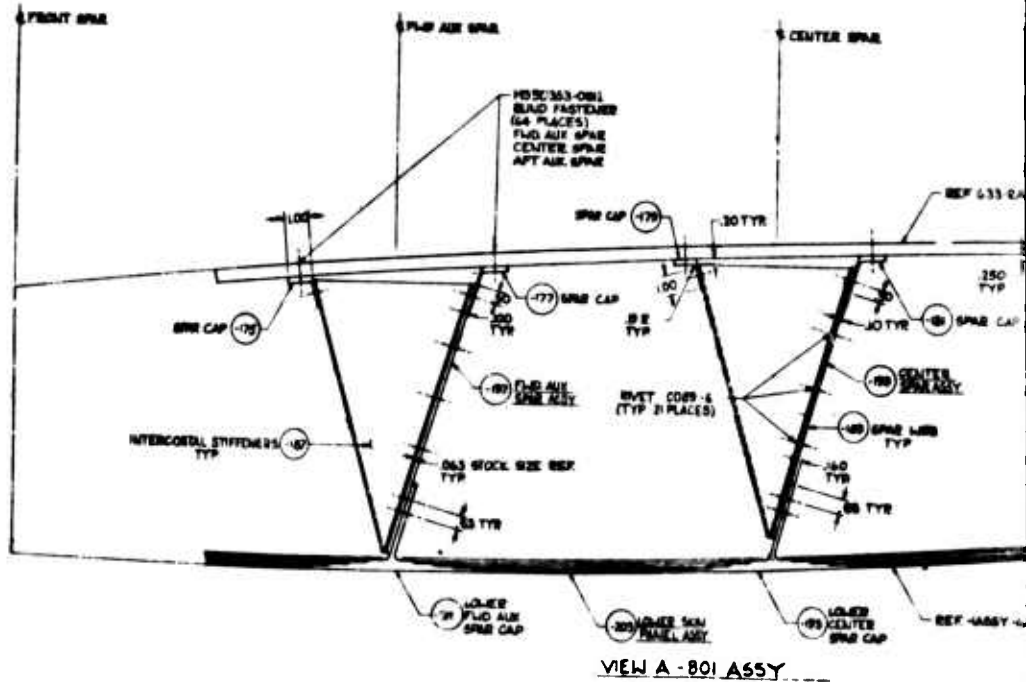
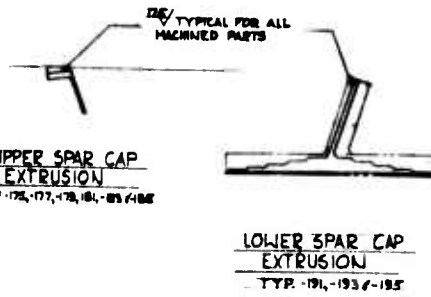


PRELIMINARY DESIGN DRAWING
 GENERAL DYNAMICS CORPORATION
 10000 BAYVIEW BLVD
 TOWSON, MD 21286
 DRAWING NO. 100-1003A
 DATE: 10/1/71

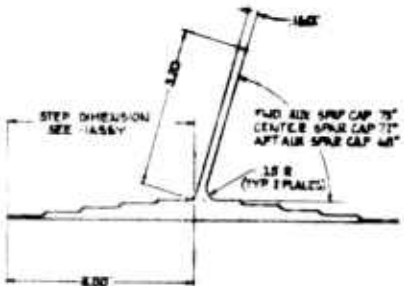
Figure 41 Sandwich Spar & Upper Skin Analytical Assembly (Continued)



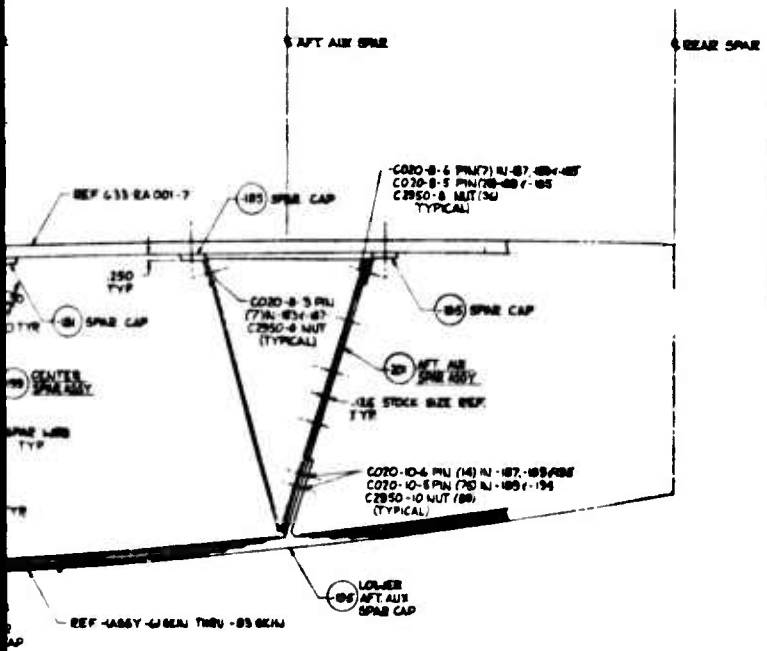
-87 INTERCOSTAL STIFFENERS DETAIL



2



DETAIL LOWER SPAR CAP
TYP - 91-03/195
SCALE 3/4



ASSY	SUB ASSY	DETAIL	PART NAME	NO REQD	MATERIAL	BAR STOCK SIZE	RAW STOCK HEIGHT	FINISHED Wt		COST			TOTAL COST	
								DETAIL	ASSY	MATL	FAB	TOOL		
-80			ALL - FINAL ASSY							320.88	192.00	23.12	0.73	573.33
			UPPER SKIN	1	1024-TB11	EXTRUDED 21 AL	2.42	30.71			17.18	21.37		
			REAR SPAR	1	REF ASS-BADD1	83-9		8.28			11.00	16.50		
-187			SPAR ASSY	1					11.5					
-179			UPPER SPAR CAP	1	1024-83T1	EXTRUDED 21 AL	2.42	1.72			17.18	21.37		
-181			INTERCOSTAL STIFFENERS	7	1024-781	0.63x0.31x1.1	1.6	33.91			3.92	16.84		
-189			SPAR WEB	2	1024-TB1	1.15x1.7x2.0	2.39	6.37			11.88	15.3		
			CO20-8-6 PIN	7										
			CO20-8-5 PIN	20										
			CO20-8-3 PIN	7										
			C2950-8 NUT	45										
			C2950-8 NUT	31										
			CO20-8 RIVET	21					11.73					
-188			UPPER SPAR CAP	1	1024-TB1	EXTRUDED 21 AL	2.42	1.72			17.18	21.37		
-181			UPPER SPAR CAP	1	1024-TB11	EXTRUDED 21 AL	2.42	1.72			17.18	21.37		
-187			INTERCOSTAL STIFFENERS	7	1024-781	0.63x0.31x1.1	1.6	33.91			3.92	16.84		
-189			SPAR WEB	2	1024-TB1	1.15x1.7x2.0	2.39	6.37			11.88	15.3		
			CO20-8-6 PIN	7										
			CO20-8-5 PIN	20										
			CO20-8-3 PIN	7										
			C2950-8 NUT	45										
			C2950-8 NUT	31										
			CO20-8 RIVET	21					11.73					
-801			UPPER SPAR ASSY	1					11.73					
-183			UPPER SPAR CAP	1	1024-TB11	EXTRUDED 21 AL	2.42	1.72			17.18	21.37		
-181			UPPER SPAR CAP	1	1024-TB11	EXTRUDED 21 AL	2.42	1.72			17.18	21.37		
-187			INTERCOSTAL STIFFENERS	7	1024-TB1	0.63x0.31x1.1	1.6	33.91			3.92	16.84		
-189			SPAR WEB	2	1024-TB1	1.15x1.7x2.0	2.39	6.37			11.88	15.3		
			CO20-8-6 PIN	7										
			CO20-8-5 PIN	20										
			CO20-8-3 PIN	7										
			C2950-8 NUT	45										
			C2950-8 NUT	31										
			CO20-8 RIVET	21					11.73					
-105			LOWER SKIN	1					136.38		108.80			
			REF ASSY - G1 SKIN THRU - 83 SKIN					80.00		151.68	109.21			
-61			SKIN	1										
-83			FRONT LONKER	1					3.28		12.24	16.29		
-87			REAR LONKER	1					3.28		12.24	16.29		
-81			LOWER	1	1024-TB11	EXTRUDED 21 AL	2.42	8.34			126.03	145.74		
-85			LOWER	1	1024-TB11	EXTRUDED 21 AL	2.42	8.34			126.03	145.74		
-196			LOWER	1	1024-TB11	EXTRUDED 21 AL	2.42	8.34			126.03	145.74		
			ADHESIVE					7.81						
			PB 90355-818	410				4.30						
			BLAD PASTER	27										
			CO20-10-6 PIN	27										
			CO20-10-5 PIN	227										
			CO20-10-3 PIN	27										
			C2950-10 NUT	87										
			C2950-10 NUT	121										
			C2950-10 NUT	121										
			SEALANT											
			MATL. REMOVED FOR PARTNERS						-13.31					

Figure 41 Sandwich Spar & Upper Skin An

THIS REPORT HAS BEEN DELIMITED
AND CLEARED FOR PUBLIC RELEASE
UNDER DOD DIRECTIVE 5200.20 AND
NO RESTRICTIONS ARE IMPOSED UPON
ITS USE AND DISCLOSURE.

DISTRIBUTION STATEMENT A

APPROVED FOR PUBLIC RELEASE;
DISTRIBUTION UNLIMITED.

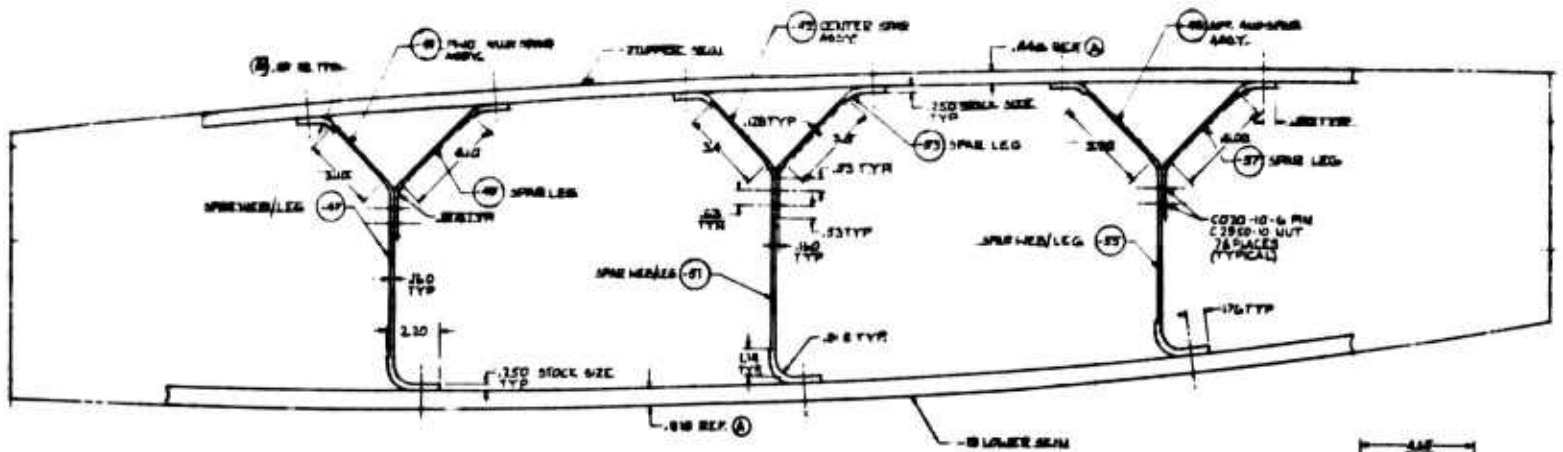
COST				TOTAL COST
MATL	FAB	TOOL	LDYRES	
1520	21.37	22.8	1.713	(A)
1751	21.37			(A)
1771	21.37			
3.92	106.84			
11.88	5.33			
1751	21.37			
1771	21.37			
3.92	106.84			
11.88	5.33			
1751	21.37			
1771	21.37			
3.92	106.84			
11.88	5.33			
	109.00			(A)
31.69	20.21			(A)
21.24	176.29			(A)
21.24	176.29			(A)
26.03	245.74			(A)
26.03	245.74			(A)
26.03	245.74			(A)
678				
assy	2432.25			

PRELIMINARY DESIGN DRAWING
 ALUMINUM LAMINATED CORE SKIN SANDWICH
 PANEL INTERMEDIATE SPARE SANDWICH PANEL
 LPR SKIN ALUMINUM ANALYTICAL ASSY

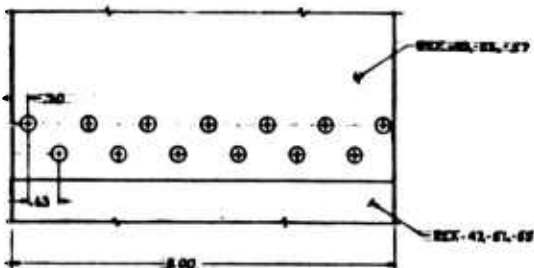
REV 12/27/64
 GENERAL DYNAMICS
 Convair Aerospace Division

REV 12/27/64
 633-RA003A
 REV 12/27/64

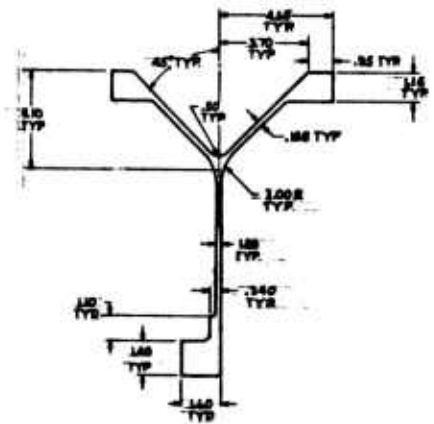
Upper Skin Analytical Assembly (Continued)



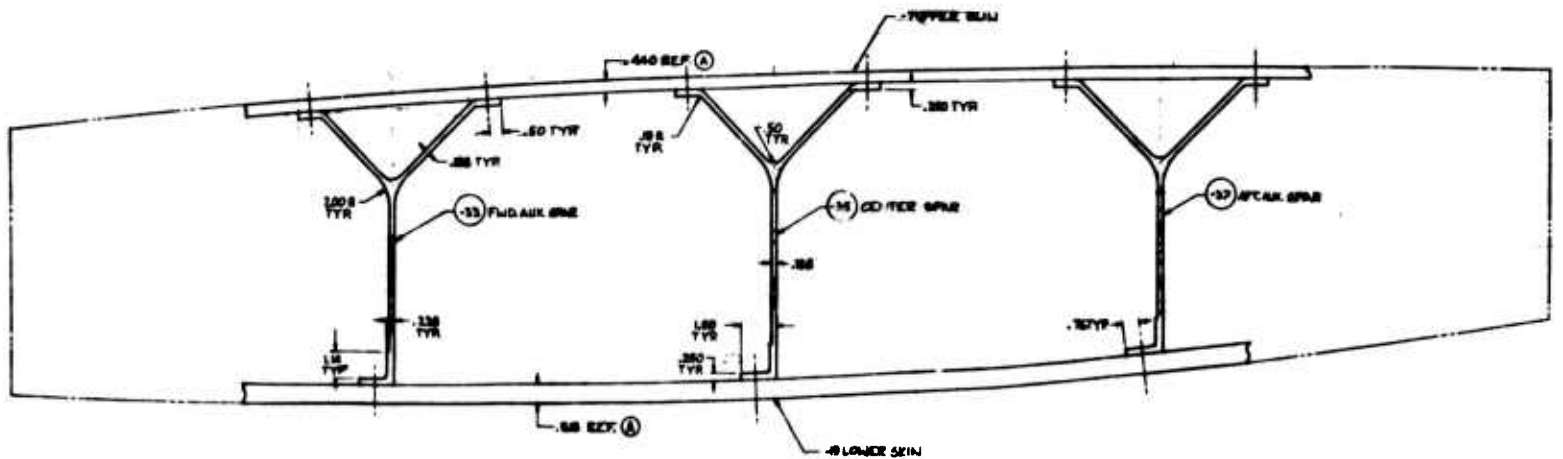
VIEW A-3ASSY



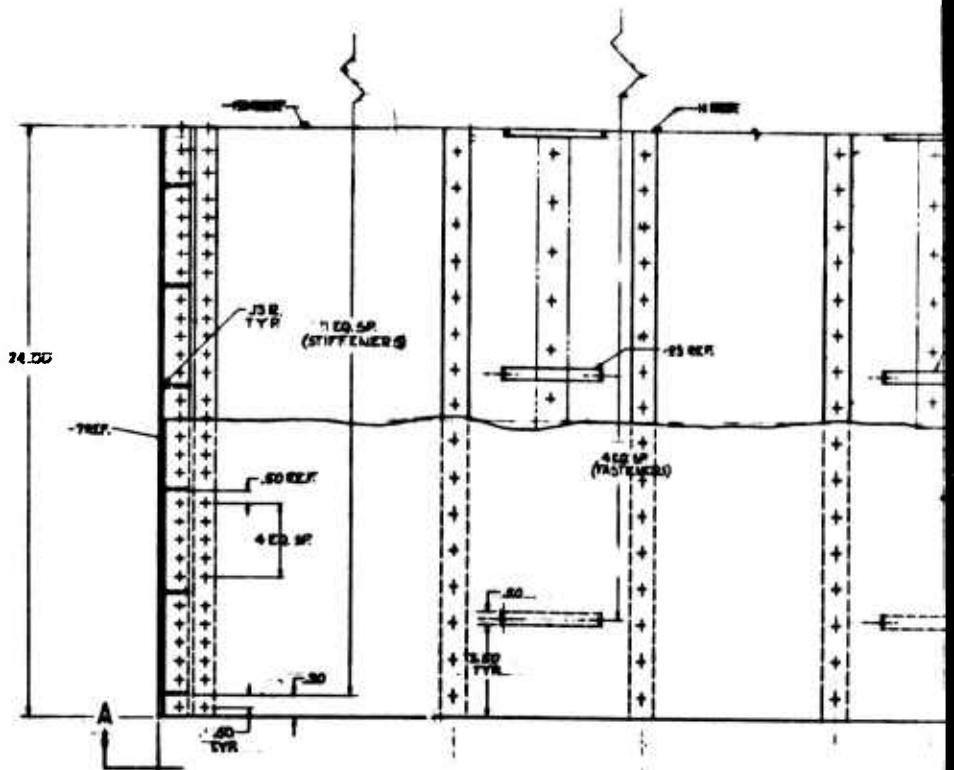
INTERMEDIATE SPAR ATTACHMENT
TYP - 81-48-46
SCALE: 1/2



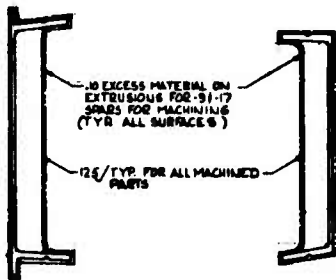
-39. EXTRUSION DETAIL
TYP-38, 36, 32



VIEW A-3ASSY

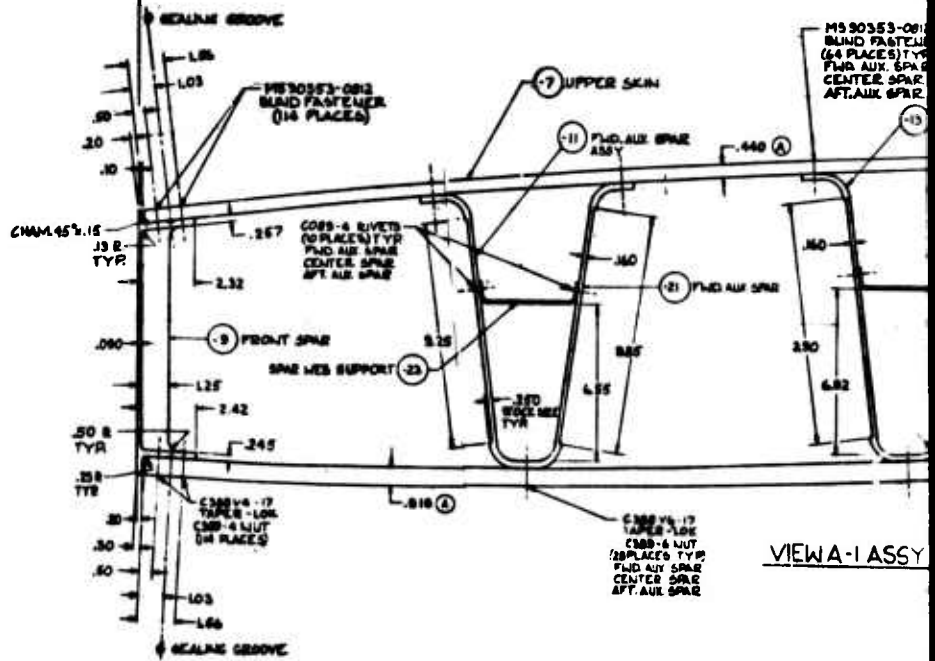


PLAN VIEW

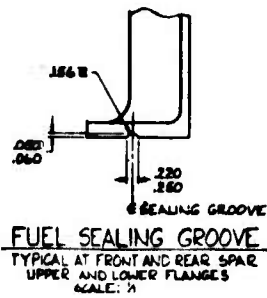


EXTRUSION DETAILS

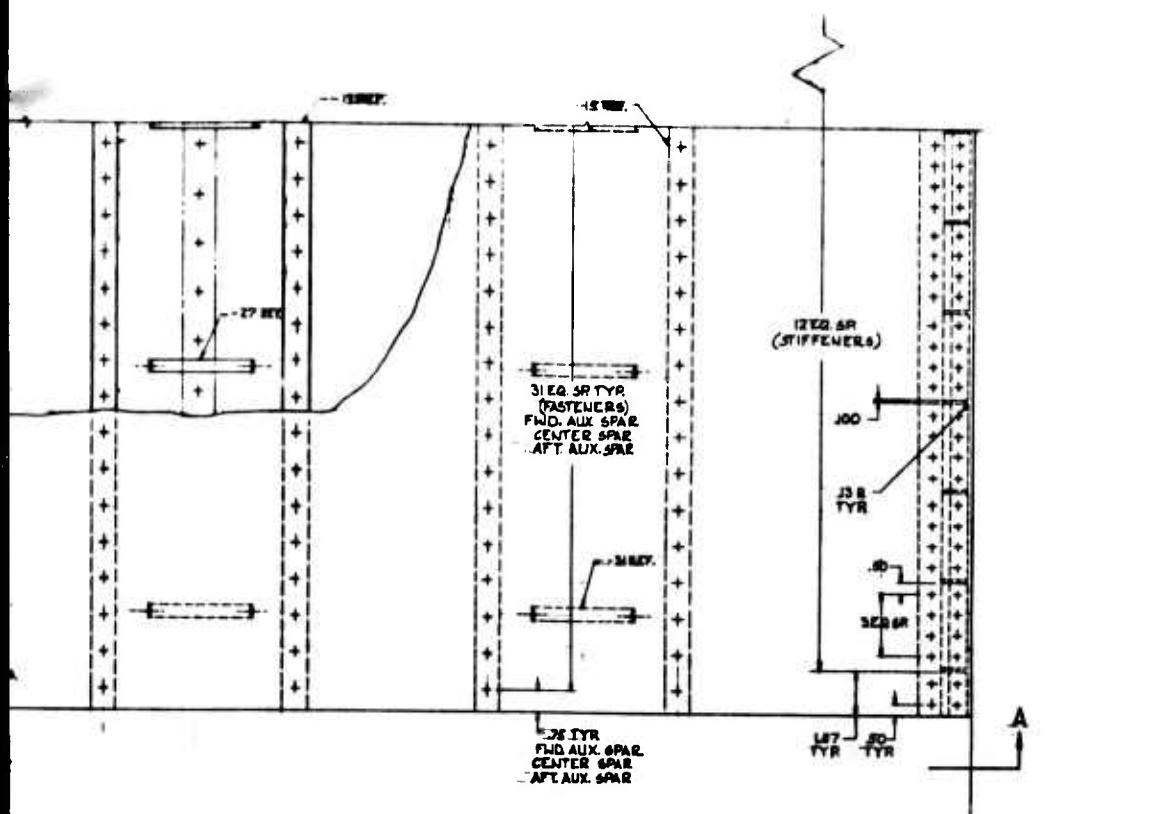
-9 1-17



VIEW A-1 ASSY



3



LIST OF MATERIALS

ASSY	SUB ASSY	QTY	PART NAME	REQD	MATERIAL	FORM SIZE	
-1		1	ANALYTICAL ASSY	1			
	-7	1	UPPER SKIN	1	2024-T851	.25x14x6	
	-9	1	FRONT SPAR	1	2024-T851	317x39x58	
-11		1	FWD AUX SPAR ASSY	1			
	-21	1	FWD AUX SPAR	1	2024-T851	250x19x170	
	-23	5	SPAR WEB SUPPORT	5	2024-T851	207x17x8	
		10	COB-4 RIVETS	10			
-13		1	CENTER SPAR ASSY	1			
	-25	1	CENTER SPAR	1	2024-T851	250x30.4x50	
	-27	5	SPAR WEB SUPPORT	5	2024-T851	207x17x8	
		10	COB-4 RIVETS	10			
-15		1	AFT AUX SPAR ASSY	1			
	-29	1	AFT AUX SPAR	1	2024-T851	350x28.7x50	
	-31	5	SPAR WEB SUPPORT	5	2024-T851	207x17x8	
		10	COB-4 RIVETS	10			
	-17	1	REAR SPAR	1	2024-T851	EXTRUSION	
	-19	1	LOWER SKIN	1	2024-T851	.750x54x68	
		810	BLIND FASTENER MS90353-0812				
		84	TAPER-LOK C389-6-17				
		218	TAPER-LOK C389-6-17				
		84	C389-6 NUT				
		218	C389-6 NUT				
			SEALANT		FMS-1044 (SEE NOTE 1)		
			MAY BE REMOVED FOR FASTENERS				
-3		1	ANALYTICAL ASSY	1			
	-7	1	UPPER SKIN	1			
	-9	1	FRONT SPAR	1			
	-33	1	FWD AUX SPAR	1	MAKE FROM -39		
	-35	1	SPAR EXTRU	1	2024-T851	EXTRUSION	
	-37	1	CENTER SPAR	1	MAKE FROM -39		
	-39	1	SPAR EXTRU	1	2024-T851	EXTRUSION	
	-37	1	AFT AUX SPAR	1	MAKE FROM -39		
	-39	1	SPAR EXTRU	1	2024-T851	EXTRUSION	
	-17	1	REAR SPAR	1			
	-19	1	LOWER SKIN	1			
			FASTENERS/MISC HARDWARE				
			MAY BE REMOVED FOR FASTENERS				
-8		1	ANALYTICAL ASSY	1			
	-7	1	UPPER SKIN	1			
	-9	1	FRONT SPAR	1			
-41		1	FWD AUX SPAR ASSY	1			
	-47	1	SPAR WEB/LEG	1	2024-T851	250x19x150	
	-49	1	SPAR LEG	1	2024-T851	250x10.20x50	
		76	CO20-10-6 PIN				
		76	C389-10 NUT				
-45		1	CENTER SPAR ASSY	1			
	-51	1	SPAR WEB/LEG	1	2024-T851	250x19x150	
	-53	1	SPAR LEG	1	2024-T851	250x9.87x50	
		76	CO20-10-6 PIN				
		76	C389-10 NUT				
-46		1	AFT AUX SPAR ASSY	1			
	-55	1	SPAR WEB/LEG	1	2024-T851	250x19.34x50	
	-57	1	SPAR LEG	1	2024-T851	250x10.02x50	
		76	CO20-10-6 PIN				
		76	C389-10 NUT				
	-17	1	REAR SPAR	1			
	-19	1	LOWER SKIN	1			
			FASTENERS/MISC HARDWARE				
			MAY BE REMOVED FOR FASTENERS				

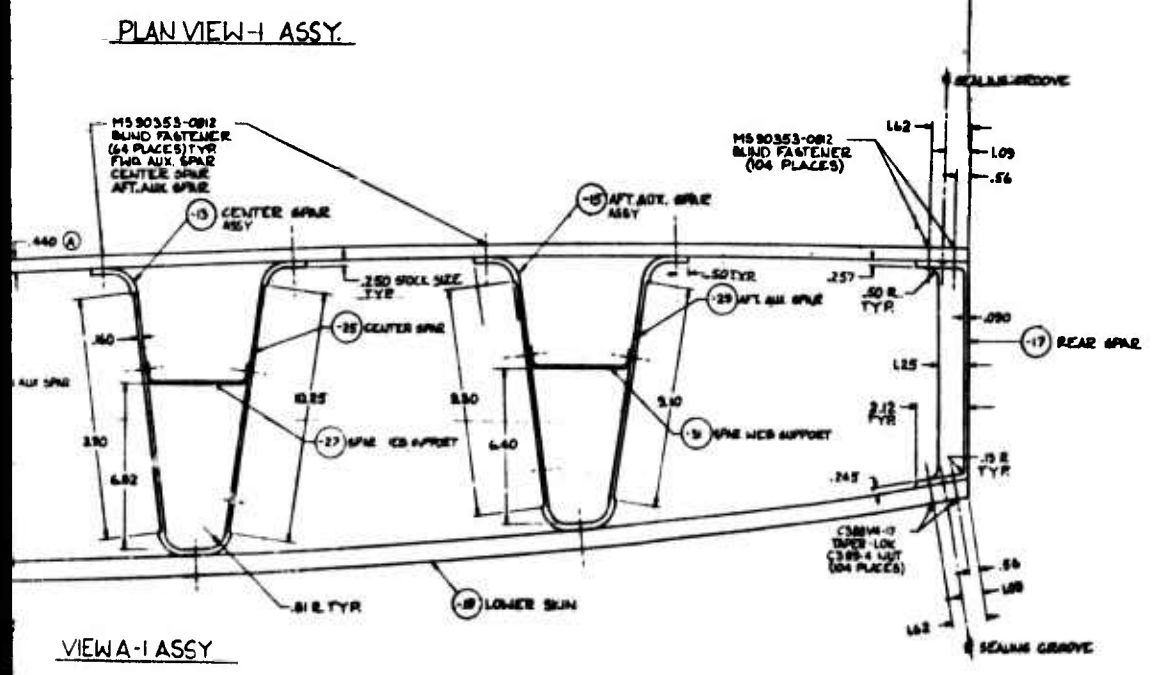


Figure 42

4

REQD	MATERIAL	RAW STOCK SIZE	WT. WEIGHT	TOTAL ASSEMBLY		COST		TOTAL COST	
				MTL	LAB	MTL	LAB		
				482.64		3770	4048	173	7991.98
1	2024-T851	220x14x.020	35.50	95.78		31.35	18.97		
1	2024-T8511	220x14x.020	35.50	10.72		490.13	192.67		
						28.94			
1	2024-T81	350x29.4x.50	37.12	25.30		34.04	52.42		
5	2024-T81	350x29.4x.50	37.12	1.04			1.87		
10									
						26.27			
1	2024-T81	250x10.2x.50	38.38	26.11		35.86	53.42		
5	2024-T81	250x10.2x.50	38.38	1.04			1.87		
10									
						24.91			
1	2024-T81	350x28.7x.50	36.76	14.75		52.80	53.42		
5	2024-T81	350x28.7x.50	36.76	1.04			1.87		
10									
						440.85	381.96		
1	2024-T8511	220x14x.020	35.50	9.03		508.44	216.30		
410				6.90					
84				2.40					
218				2.62					1787.36
84				.90					
218				.70					
	FMS-1044 (SEE NOTE 1)		176	1.50					1557.6666
	FOR FASTENERS			4.14					
1				412.11		4584	4594	150	9308
	SEE HAWBY			193.78		378.36	184.97		
1	2024-T8511	220x14x.020	35.50	10.72		490.13	192.67		
1	2024-T8511	220x14x.020	35.50	18.44		312.10	300.87		
1	2024-T8511	220x14x.020	35.50	18.76		314.10	300.87		
1	2024-T8511	220x14x.020	35.50	18.44		312.10	300.87		
1	2024-T8511	220x14x.020	35.50	9.03		440.85	381.96		
	SEE HAWBY			188.90		508.44	216.30		
				16.02					1787.36
	FOR FASTENERS			4.14					1557.6666
1				418.88		4198	4277	207	8682
	SEE HAWBY			193.78		378.36	184.97		
1	2024-T81	250x10.2x.50	38.38	12.63		30.71	42.74		
76	2024-T81	250x10.2x.50	38.38	6.37		18.71	32.08		
76				.79					
76				.38					
				21.78					
1	2024-T81	250x10.2x.50	38.38	13.76		30.06	42.74		
1	2024-T81	250x10.2x.50	38.38	6.83		16.49	32.08		
76				.79					
76				.35					
				20.78					
1	2024-T81	250x16.34x.50	31.41	13.78		31.17	42.74		
1	2024-T81	250x16.34x.50	31.41	6.31		18.48	32.08		
76				.78					
76				.33					
				20.78					
1	2024-T81	250x16.34x.50	31.41	13.78		31.17	42.74		
1	2024-T81	250x16.34x.50	31.41	6.31		18.48	32.08		
76				.78					
76				.33					
				20.78					
	SEE HAWBY			188.90		508.44	216.30		
				16.02					1787.36
	FOR FASTENERS			4.14					1557.6666

CHANGED 7 THICKNESS FROM .48 TO .50
REVISED ALL WEIGHTS IN L/W TO REFLECT .02 THICKNESS CHANGE.
CHANGED .21, .25, .28, .47, .48, .51, .53, .55, .57 RAW STOCK LENGTH IN L/W FROM 48.00 TO 50.00
CHANGED LOWER SKIN THICKNESS FROM .777 TO .615
REVISED ALL WEIGHTS IN L/W TO REFLECT LOWER SKIN THICKNESS CHANGE

3 - 21, 25, 29 OF HAWBY AND 47, 48, 51, 53, 57 AND 57 OF SASSY TO BE FORMED / CHEM-ETCHED

2 - FINISH REQD. ON ALL MACHINED SURFACES. THE FINISH SURFACES AT THE UPPER SURFACE OF THE FRONT BEARING, THE UPPER SKIN AND ALL FASTENERS SHALL BE SEALED FOR PPM-1004

NOTE:

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 Convair Aerospace Division
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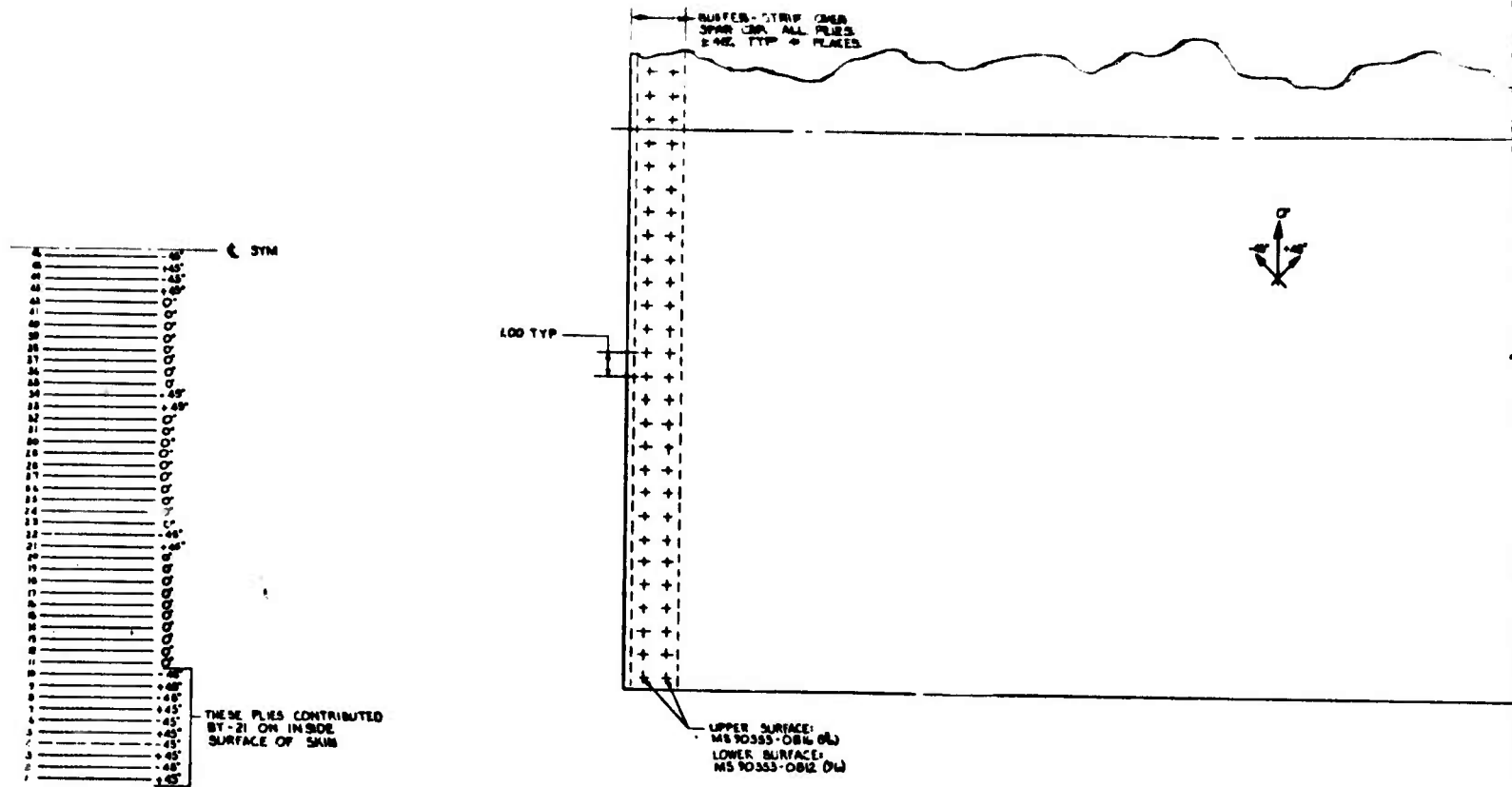
Figure 42 Inverted "A" Spar Analytical Assembly

Table IV ATW METALLIC WING BOX ANALYTICAL ASSY EVALUATION SUMMARY

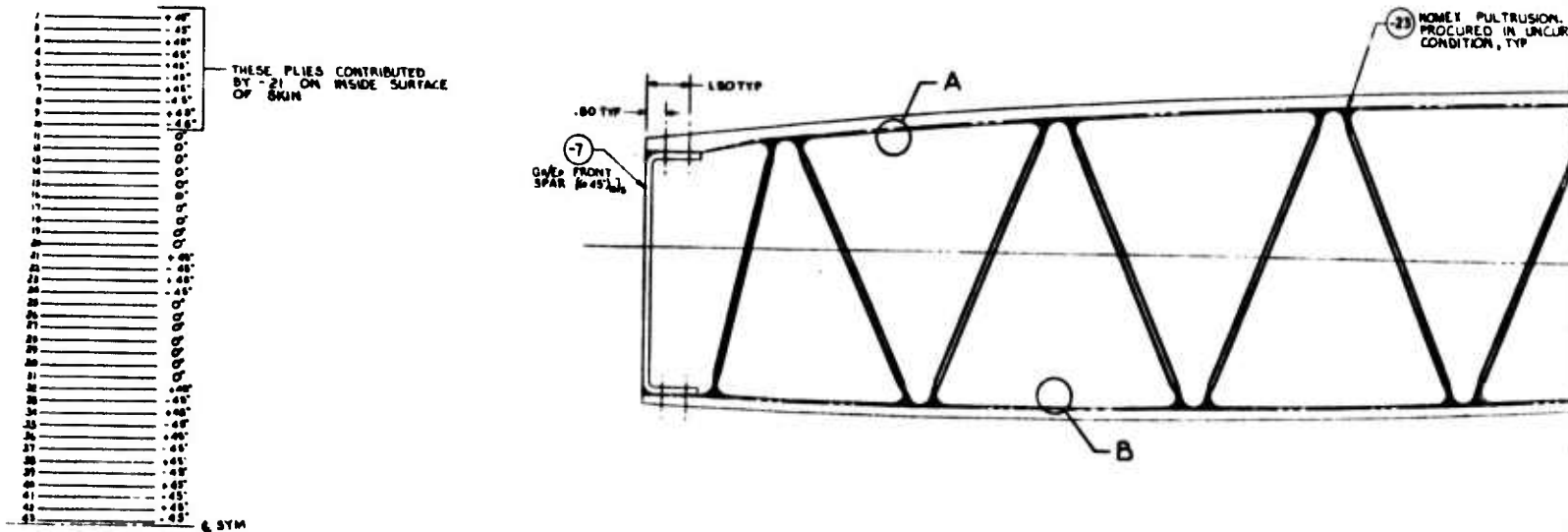
CONFIG. NO.	DESCRIPTION	STRUCT. EFFIC		TECHNOLOGY IMPROV			DAMAGE TOL.			-ABILITIES				TOTAL SCORE	RANK
		COST 1975 \$ (.24)	WEIGHT LBS. (.16)	CONCEPT (.05)	MATL (.02)	MFG (.03)	SAFE CRACK (.08)	FAIL SAFE (.12)	INSPECT- (.06)	MFG- (.06)	MAIN- TAIN- (.06)	REPAIR- (.06)	DUR- (.06)		
633-RA000-1	BASELINE WING BOX- MACHINED SKINS & SPARS; 7 SPAR CONFIG; ALUMINUM	12.310	418.09	.000	.000	.000	.075	.000	.060	.060	.060	.060	.645	9	
633-RA001-1	LAMINATED LWR SKIN; MACHINED UPR. SKIN & CLOS SPAR, EXTR & ETCHED INTERM "Y" SPARS, 5 SPAR CONFIG; ALUMINUM	10.091	340.75	.042	.000	.030	.080	.120	.045	.052	.057	.059	.839	2	
633-RA001-3	LAM. LWR SKIN; MACH UPR SKIN & CLOS SPAR; EXTR INTER "Y" SPAR; 5 SPAR CONFIG; ALUMINUM	10.774	359.38	.040	.000	.027	.080	.120	.045	.052	.057	.059	.814	3	
633-RA001-5	LAM. LWR SKIN; MACH. UPR SKIN & CLOS. SPARS; SAND WEB INTERM "Y" SPAR WITH EXTRU. CAPS; 5 SPAR CONFIG; ALUMINUM	10.992	347.88	.046	.000	.029	.080	.120	.045	.046	.057	.059	.813	4	
633-RA001-801	LAM. LWR SKIN; MACH. UPR SKIN & CLOS. SPARS; FORMED & ETCHED INTERM "Y" SPAR, 5 SPAR CONFIG; ALUMINUM	10.660	345.05	.044	.000	.027	.080	.120	.045	.044	.057	.059	.814	3	
633-RA001-803	LAM. LWR SKIN; MACH. UPR SKIN & CLOS. SPARS; FORMED/BEADED INTERM "Y" SPARS; 5 SPARS; ALUMINUM	9.741	380.13	.046	.000	.029	.078	.000	.030	.040	.057	.059	.676	12	
633-RA001-805	LAM LWR SKIN/EMBEDDED SPAR CAPS; MACH UPR SKIN AND CLOS SPARS; FORMED/BEADED INTERM "Y" SPARS, AL	9.777	359.09	.046	.000	.029	.078	.000	.030	.040	.057	.059	.683	8	
633-RA002-1	LAM LWR SKIN; MACH. UPR SKIN & CLOS SPARS; CORR. INTERM. SPARS WITH EXTR CAPS; 5 SPAR CONFIG. AL	9.546	351.83	.034	.000	.027	.080	.120	.045	.052	.057	.059	.834	3	

Table IV ATM METALLIC WING BOX ANALYTICAL ASSY EVALUATION SUMMARY (CONTINUED)

CONFIG. NO.	DESCRIPTION	STRUCT. EFFIC.		TECHNOLOGY IMPROV.			DAMAGE TOL.			-ABILITIES				TOTAL SCORE	RANK
		COST 1975 \$ (.24)	WEIGHT LBS. (.16)	CONCEPT (.05)	MATL (.02)	MFG. (.03)	SAFE CRACK (.08)	FAIL SAFE (.12)	INSPECT- (.06)	MFG. (.06)	MAIN-TAIN- (.06)	REPAIR- (.06)	DUR- (.06)		
633-RA003-1	LAM LWR SKIN; SAND PNL. UPR SKIN & INTERM SPARS; MACH CLOS SPARS; 5 SPAR CONFIG; AL	12,277	297.58	.028	.000	.016	.080	.120	.043	.044	.057	.059	.025	.788	5
633-RA003-3	LAM LWR SKIN; MACH UPR SKIN & CLOS SPARS; SAND PANEL INTERM. SPARS; 7 SPAR CONFIG; ALUMINUM	12,793	331.13	.028	.000	.014	.080	.120	.045	.050	.057	.059	.029	.776	6
633-RA003-5	LAM LWR SKIN; MACH UPR SKIN & SPARS; 6 SPARS, LAM, LWR SPAR CAPS	10,918	369.91	.024	.000	.014	.078	.000	.030	.050	.057	.059	.024	.641	10
633-RA003-801	LAM, LWR, SKIN; MACH SKIN & CLOS SPARS; SHEET METAL INTERM CANTED SPARS W/FORMED INTERCOSTALS 5 SPAR CONFIG; AL	10,733	330.04	.050	.000	.030	.080	.120	.045	.052	.057	.059	.027	.843	1
633-RA004-1	MACH UPR & LWR SKINS & CLOS. SPARS; FORMED & ETCHED INVERTED "A" INTERM SPARS, 5 SPARS AL, BOLTED LWR SKIN	7,991	432.61	.030	.000	.014	.075	.000	.060	.056	.059	.059	.029	.622	11
633-RA004-3	MACH SKINS & CLOSURE SPARS; EXTRUDED INTER "Y" SPAR; 5 SPAR CONFIG.; ALUMINUM BOLTED LWR SKIN	9,308	412.11	.026	.000	.014	.075	.000	.060	.059	.059	.059	.029	.703	7
633-RA004-5	MACH UPR & LWR SKINS & CLOSURE SPARS; FORMED & ETCHED INTERM "Y" SPARS, 5 SPARS BOLTED LWR SKIN	8,682	418.66	.024	.000	.014	.075	.000	.060	.052	.059	.059	.024	.588	13



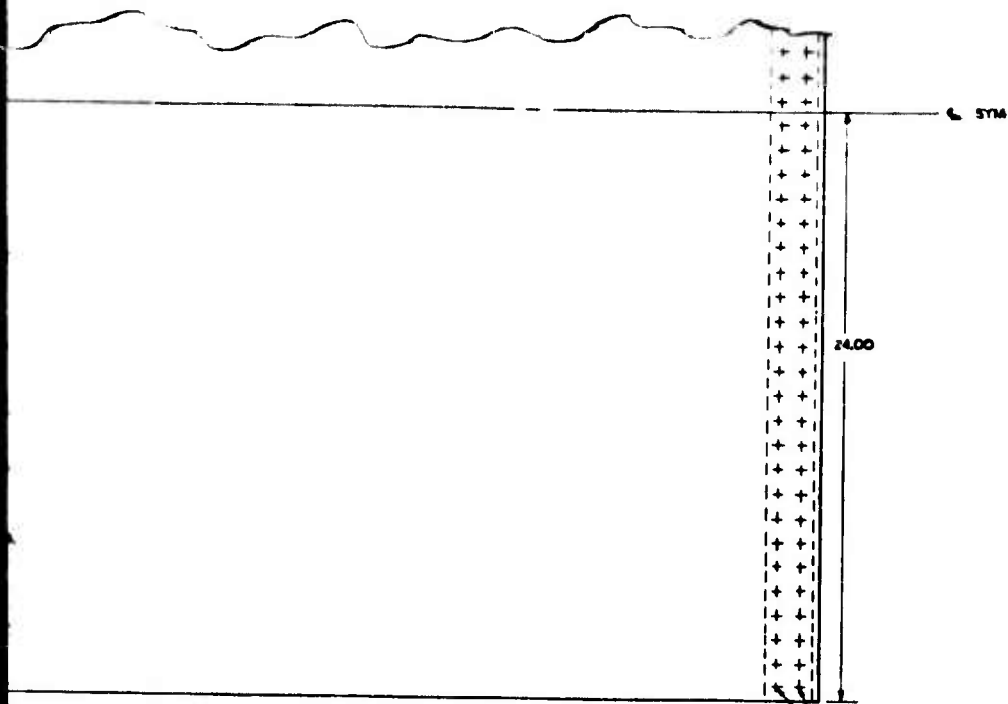
DETAIL A



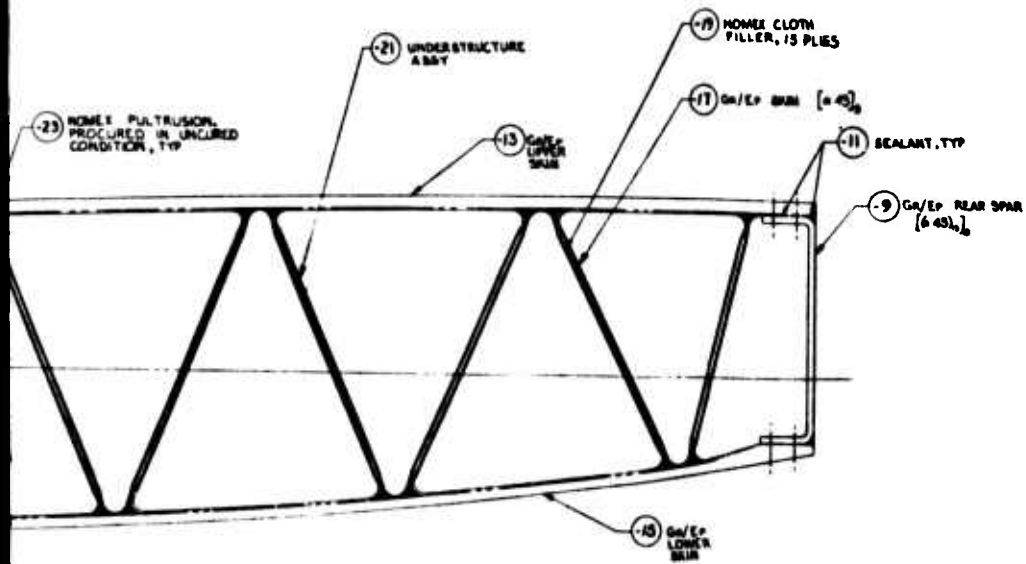
DETAIL B

-1 ASSY

2



UPPER SURFACE:
MS 90383 - OB14 (76)
LOWER SURFACE:
MS 90353 - OB14 (88)
MS 90353 - OB17 (40)

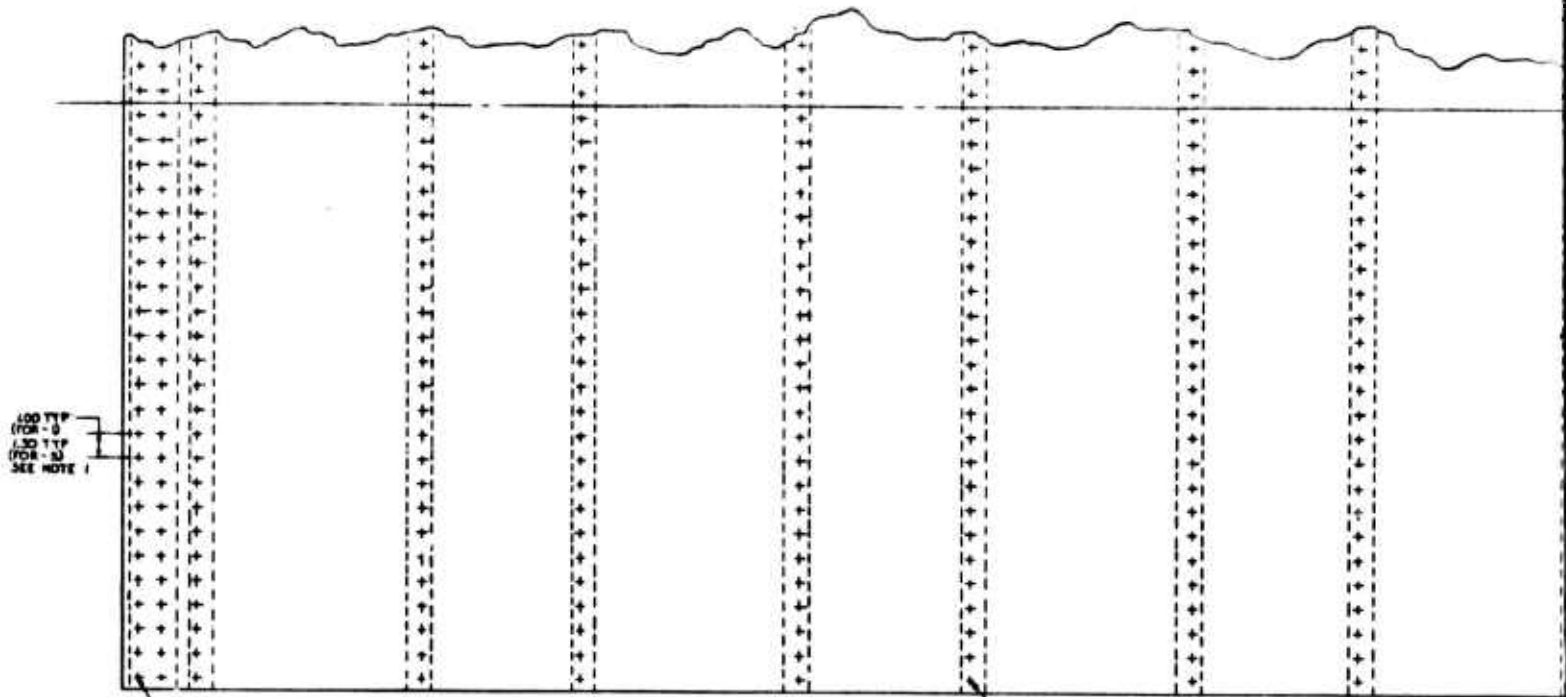


ASSY	SUB ASSY	DETAIL	PART NAME	NO. REQD	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED W/F		COST			TOTAL COST
								DETAIL	ASSY	MATL	LAB	TOL.	
		-7	FRONT SPAR	1	F-34 EP TAP	12.5" x 1.5"	1.5			1791.53	116.76	291	
		-8	REAR SPAR	1	F-34 EP TAP	12.5" x 1.5"	1.5			1791.53	116.76	291	
		-11	DELAHT	48	FMS 1048	48	5.2						
		-12	UPPER SKIN	1	F-34 EP TAP	40.0" x 1.5"	1.5			3243.14	612.48		
		-13	LOWER SKIN	1	F-34 EP TAP	40.0" x 1.5"	1.5			3243.14	612.48		
		-17	SKIN	1	F-34 EP TAP	24.5" x 1.5"	1.5			1672.28	114.74		
		-19	FILLER	1	F-34 EP TAP	4.5" x 1.5"	1.5			114.76	143.72		
		-21	UNDERSTRUCTURE	1					17.8				
		-22	FILLER	22					33				
		-23	RIVET	10					10				
		-24	RIVET	10					10				
		-25	RIVET	144					144				
		-26	RIVET	28					28				
-1									3000	3762.5227	291	43280	
										Assy 2287			

PRELIMINARY DESIGN DRAWING
 ATW-4 WING TRUSS SPAR
 GIVEP CO CURED CONSTRUCTION
 ANALYTICAL ASSY OF
 GENERAL DYNAMICS
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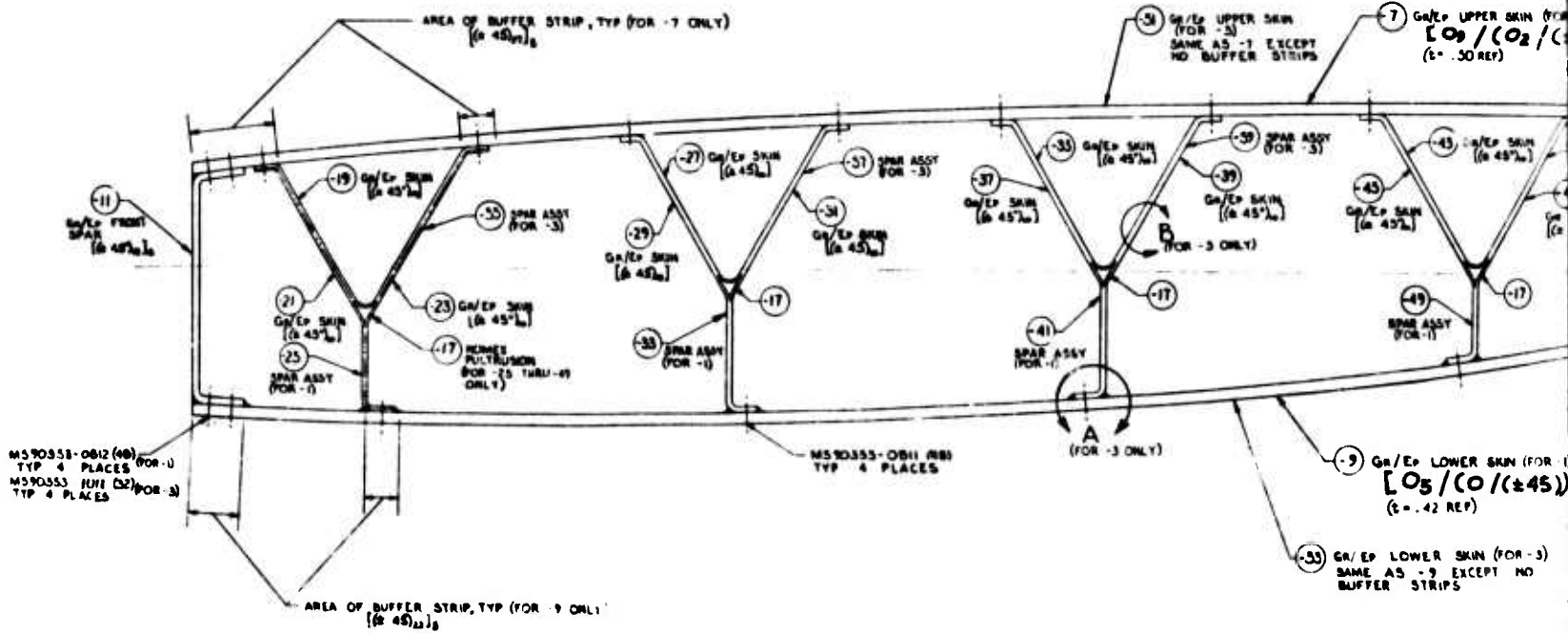
Figure 43 Composite Truss Spar Analytical Assembly

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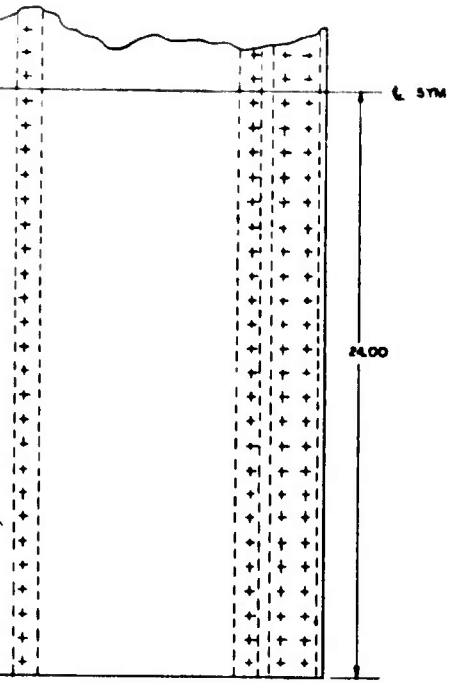
M590355-0B14 (4A) (FOR -1)
TYP 4 PLACES
M590353-1012 (32) (FOR -3)
TYP 4 PLACES
SEE NOTE 1

M590353-0615 (4B) (FOR -1)
TYP 8 PLACES
M590353-1014 (32) (FOR -3)
TYP 8 PLACES
SEE NOTE 1



M590353-0B12 (4B) (FOR -1)
TYP 4 PLACES
M590353-1011 (32) (FOR -3)
TYP 4 PLACES

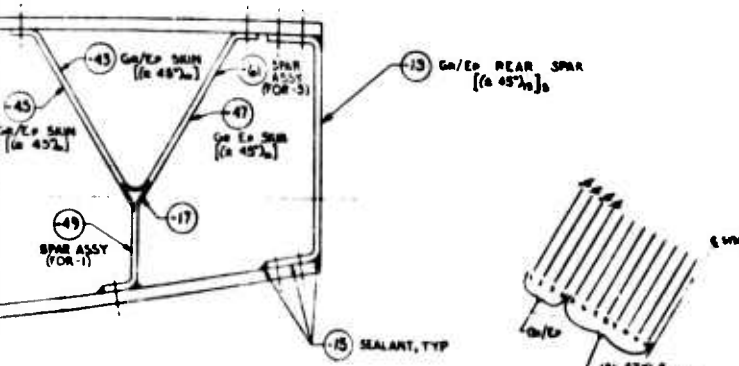
-1 ASSY
3 ASSY



ASSY	SUB ASSY	DETAIL	PART NAME	NO. REQ'D	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED DETAIL WEIGHT
		-7	UPPER SKIN	1	GR/EP TAPE	252 FT 3" TAPE	113.0	49.0
		-9	LOWER SKIN	1	GR/EP TAPE	1076 FT 3" TAPE	102.6	71.9
		-11	FRONT SPAR	1	GR/EP TAPE	1120 FT 3" TAPE	106.2	117.0
		-13	REAR SPAR	1	GR/EP TAPE	1045 FT 3" TAPE	101.1	109.9
		-15	SEALANT	AR	FM-1044	AR	1.50P	1.25P
		-17	FILLER	1	NOMEX	20 M ² 50 M	.5	.5
		-19	SKIN	1	GR/EP TAPE	320 FT 3" TAPE	7.6	5.5
		-21	SKIN	1	GR/EP TAPE	411 FT 3" TAPE	5.9	4.5
		-23	SKIN	1	GR/EP TAPE	411 FT 3" TAPE	5.9	4.5
		-25	SPAR ASSY	1				
		-17	FILLER	1	NOMEX	20 M ² 50 M	.5	.5
		-27	SKIN	1	GR/EP TAPE	498 FT 3" TAPE	7.2	5.2
		-29	SKIN	1	GR/EP TAPE	447 FT 3" TAPE	6.5	4.7
		-31	SKIN	1	GR/EP TAPE	447 FT 3" TAPE	6.5	4.7
		-33	SPAR ASSY	1				
		-17	FILLER	1	NOMEX	20 M ² 50 M	.5	.5
		-35	SKIN	1	GR/EP TAPE	498 FT 3" TAPE	7.2	5.2
		-37	SKIN	1	GR/EP TAPE	440 FT 3" TAPE	6.4	4.6
		-39	SKIN	1	GR/EP TAPE	440 FT 3" TAPE	6.4	4.6
		-41	SPAR ASSY	1				
		-17	FILLER	1	NOMEX	20 M ² 50 M	.5	.5
		-43	SKIN	1	GR/EP TAPE	515 FT 3" TAPE	7.4	5.4
		-45	SKIN	1	GR/EP TAPE	411 FT 3" TAPE	5.9	4.5
		-47	SKIN	1	GR/EP TAPE	411 FT 3" TAPE	5.9	4.5
		-49	SPAR ASSY	1				
			RIVET	192				3.1
			RIVET	384				3.3
			RIVET	192				2.8
			RIVET	192				2.1
-1			ANALYTICAL ASSY					258

		-11	FRONT SPAR	1	GR/EP TAPE	1120 FT 3" TAPE	106.2	117.0
		-13	REAR SPAR	1	GR/EP TAPE	1045 FT 3" TAPE	101.1	109.9
		-15	SEALANT	AR	FM-1044	AR	1.50P	1.25P
		-31	UPPER SKIN	1	GR/EP TAPE	6024 FT 3" TAPE	25.6	80.0
		-33	LOWER SKIN	1	GR/EP TAPE	7076 FT 3" TAPE	102.6	71.9
					GR/EP TAPE	517 FT 3" TAPE	4.6	3.3
					NOMEX CLOTH	159 FT ²	12.4	9.75
		-35	SPAR ASSY	1				
					GR/EP TAPE	316 FT 3" TAPE	4.5	3.28
					NOMEX CLOTH	151 FT ²	12.4	9.66
		-37	SPAR ASSY	1				
					GR/EP TAPE	323 FT 3" TAPE	4.6	3.36
					NOMEX CLOTH	160 FT ²	12.7	9.84
		-39	SPAR ASSY	1				
					GR/EP TAPE	511 FT 3" TAPE	4.5	3.23
					NOMEX CLOTH	156 FT ²	12.4	9.57
					RIVET	136		3.40
					RIVET	272		3.8
					RIVET	136		3.0
-3								252

7 GR/EP UPPER SKIN (FOR -1)
 $[O_9 / (O_2 / (\pm 45))_9]_5$
 (L= 50 REF)

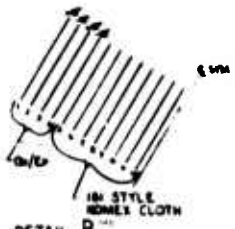


9 GR/EP LOWER SKIN (FOR -1)
 $[O_5 / (O_1 / (\pm 45))_5]_5$
 (L= 42 REF)

35, 37, 39, 41



DETAIL B FOR -3 ONLY NO SCALE



EP LOWER SKIN (FOR -3)
 SAME AS -9 EXCEPT NO UPPER STRIPS

Figure 44

W STOCK SIZE	RAW STOCK ACQHT	FINISHED WT.		COST			TOTAL COST
		DETA.	ASSY	MATL	FAB.	TOOL.	
W 3" TAC	103.0	83.0		3326	861.23		
W 3" TAP	102.0	73.0		11078	533.61		
W 3" TAP	6.0	7.0		1283	182.12		
W 3" TAP	15.1	2.7		1632	182.12		
AR	1.75	1.50					
W 3" W	.5	.5		14	1.47		
W 3" TAP	7.0	3.5		826	106.77		
W 3" TAP	5.9	4.3		643	99.30		
W 3" TAP	5.9	4.3	14.6	643	99.30		
W 3" W	.5	.5		14	1.47		
W 3" TAP	7.2	5.2		779	99.30		
W 3" TAP	6.5	4.7		690	99.30		
W 3" TAP	6.5	4.7	13.1	690	99.30		
W 3" W	.5	.5		14	1.47		
W 3" TAP	7.2	5.2		779	99.30		
W 3" TAP	6.4	4.6		688	99.30		
W 3" TAP	6.4	4.6	14.9	688	99.30		
W 3" W	.5	.5		14	1.47		
W 3" TAP	7.4	5.4		803	99.30		
W 3" TAP	5.9	4.3		643	99.30		
W 3" TAP	5.9	4.3	14.5	643	99.30		
				3112			
				2.8			
				2.7			
			258	39507	5652	149	459084
				ASSY	2796		

W 3" TAP	6.2	11.7		1753	182.12		
W 3" TAP	15.1	10.7		1622	182.12		
AR	1.50	1.25					
W 3" TAP	23.0	90.0		2346	462.60		
W 3" TAP	102.6	78.9		11078	533.61		
W 3" TAP	4.6	5.1					
W 3" TAP	12.0	5.9					
W 3" TAP	4.5	3.20	1400	784	235.04		
W 3" TAP	2.0	9.00					
W 3" TAP	4.5	3.20	1200	776	236.04		
W 3" TAP	2.7	9.04					
W 3" TAP	4.5	3.20	1318	791	235.04		
W 3" TAP	2.0	9.07					
			282	170	236.04		
				1763			
			252.6	3260	4973	149	570129
				ASSY	2411		

NOTE 5:
 1. FASTENER PATTERNS SHOWN
 ARE FOR -1, -2 FASTENER ROOTS.
 ARE SPECIFIED IN CALCULATIONS
 AND IN L.M.

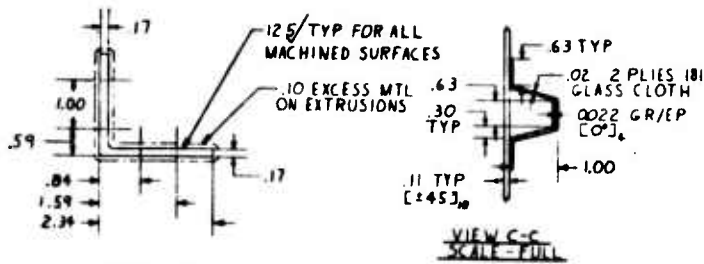
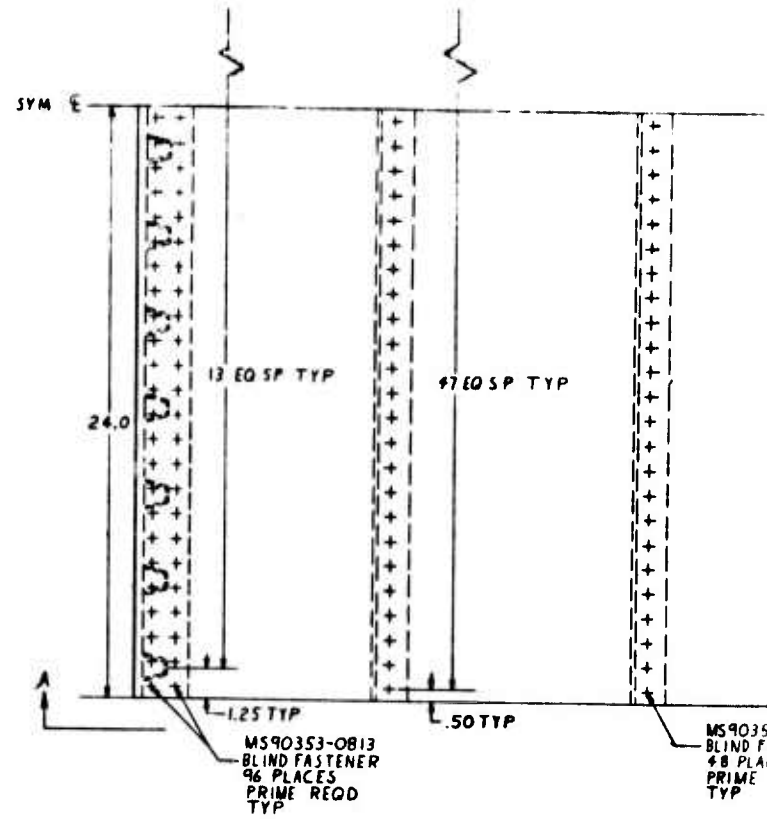
PRELIMINARY DESIGN DRAWING

ATW-4 WING-GR/EP SKINS WITH
 BUFFER STRIPS, GR/EP Y SPARS,
 ANALYTICAL ASSY OF

GENERAL DYNAMICS
 Convair Aerospace Division

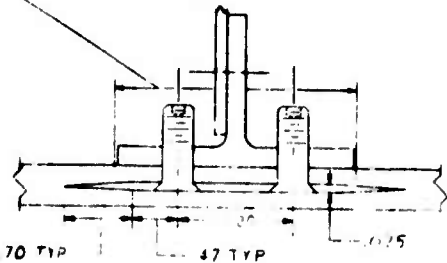
633RAC06

Figure 44 Composite Y-Spar Analytical Assembly

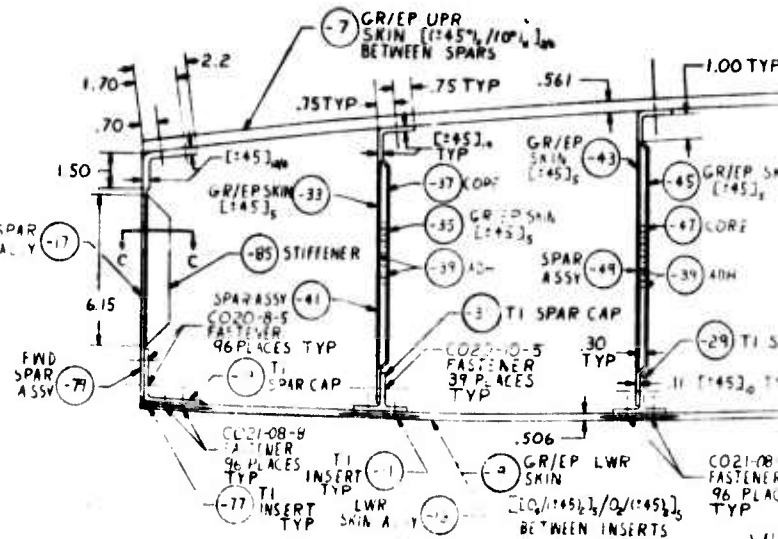
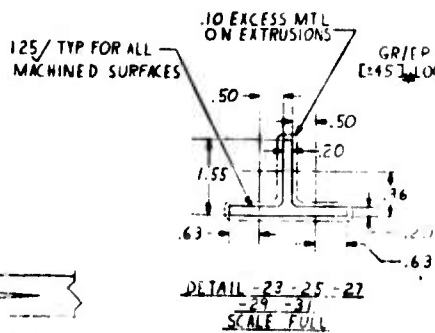


DETAIL -19
ROTATE 180°
FOR -21
SCALE FULL

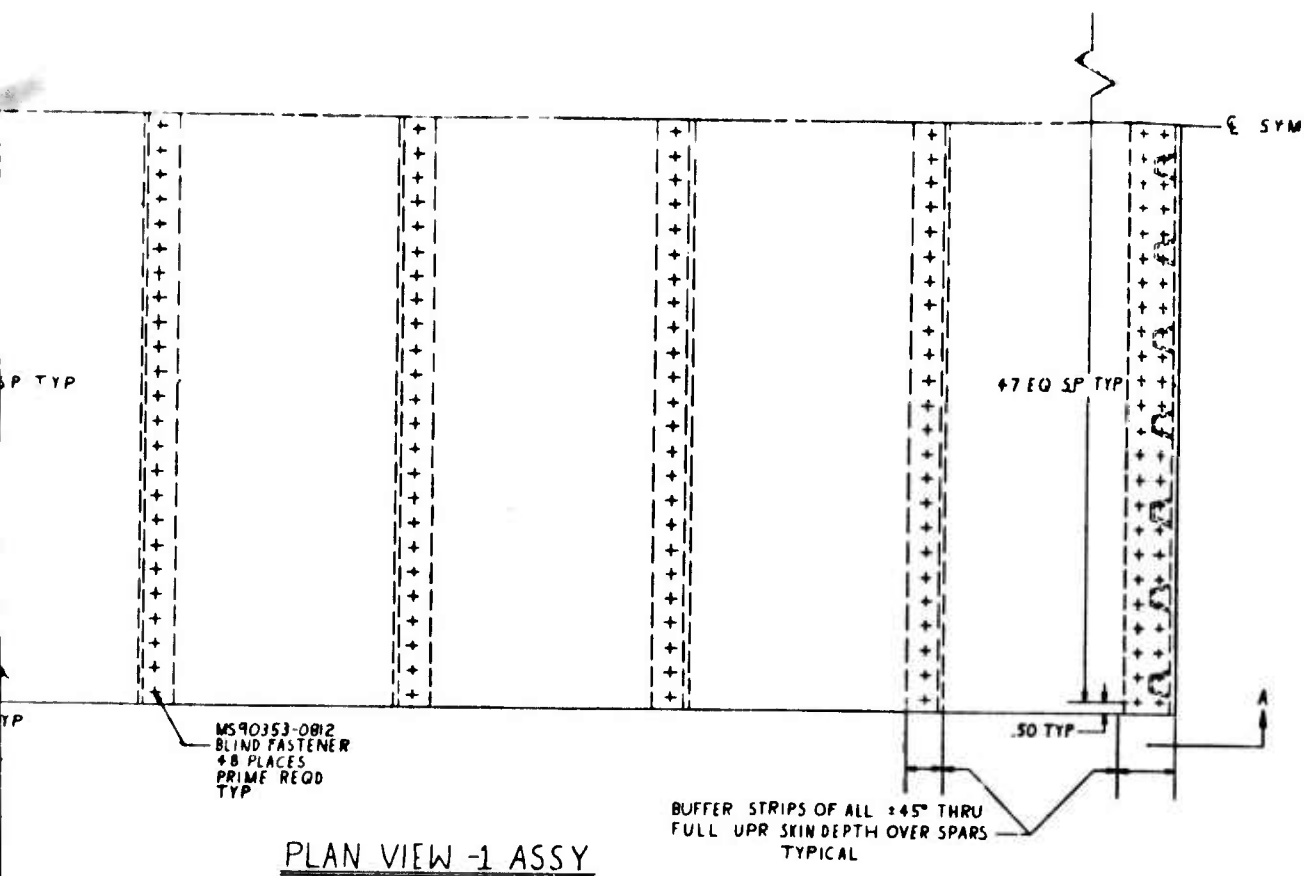
BUFFER STRIP OF ALL ±45° THRU FULL SKIN DEPTH AROUND INSERTS AS SHOWN TYPICAL



DETAIL B
SCALE 2/1



2



ASSEMBLY	DETAIL	PART NAME	NO	INCLD
-7		HING BOX OVER SKIN		
MS90353-0812		FASTENER	48	
CO21-08-B		FASTENER	142	
CR450-10		NUT	142	
CR450-10		FASTENER	142	
CR450-10		NUT	142	
-75		SEALANT	AS REQD	
-13		LWR SKIN ASSY	1	
-9		LWR SKIN	1	3"
-17		INSERT	2	6
-11		INSERT	5	6
-19		SPAR CAP	1	6
-21		SPAR CAP	1	6
-25		SPAR CAP	1	6
-27		SPAR CAP	1	6
-29		SPAR CAP	1	6
-31		SPAR CAP	1	6
CO21-08-B		FASTENER	672	
CR450-10		NUT	672	
-61		FWD AUX SPAR	1	
-33		OUTER SKIN	1	3"
-35		INNER SKIN	1	3"
-37		CORE	1	5/2"
-39		ADM	AS REQD	FM
-64		FWD AUX SPAR	1	
-63		OUTER SKIN	1	3"
-65		INNER SKIN	1	3"
-67		CORE	1	5/2"
-39		ADM	AS REQD	FM
-67		CENTER SPAR	1	
-51		OUTER SKIN	1	3"
-53		INNER SKIN	1	3"
-55		CORE	1	5/2"
-39		ADM	AS REQD	FM
-65		REAR AUX SPAR	1	
-59		OUTER SKIN	1	3"
-61		INNER SKIN	1	3"
-63		CORE	1	5/2"
-39		ADM	AS REQD	FM
-73		REAR AUX SPAR	1	
-67		OUTER SKIN	1	3"
-69		INNER SKIN	1	3"
-71		CORE	1	5/2"
-39		ADM	AS REQD	FM
-79		FWD SPAR ASSY	1	
-17		FWD SPAR	1	3"
-85		STIFFENER	14	3/8"
-83		REAR SPAR ASSY	1	
-15		REAR SPAR	1	3"
-81		STIFFENER	14	3/8"

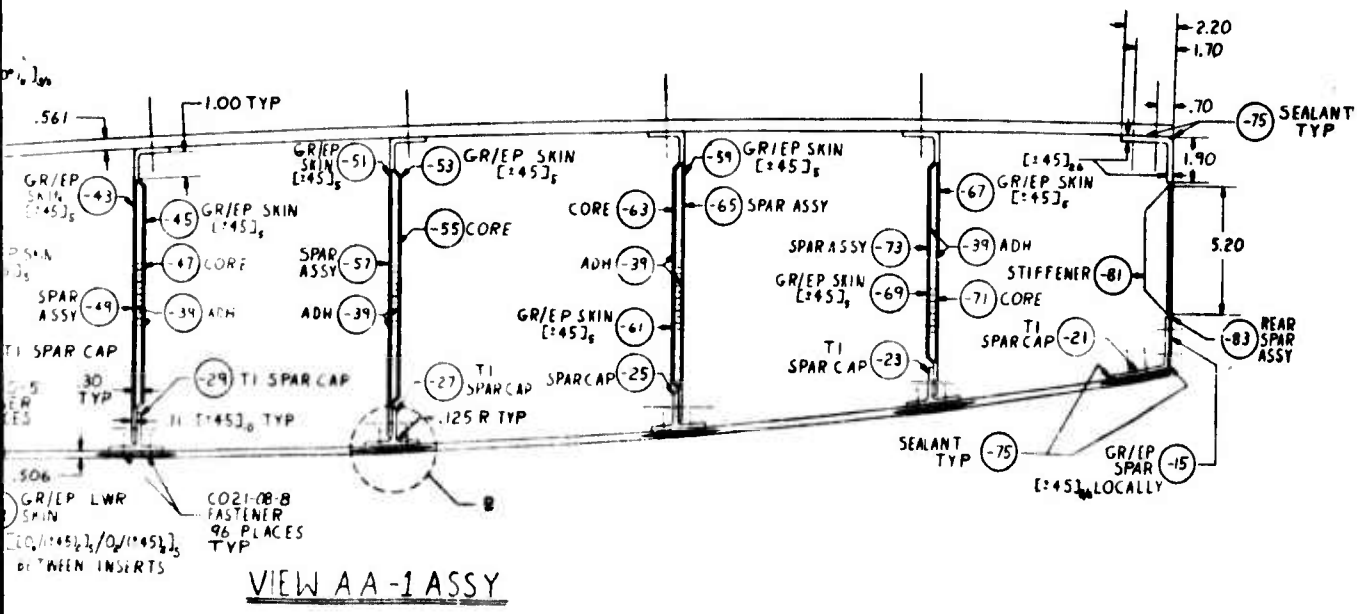
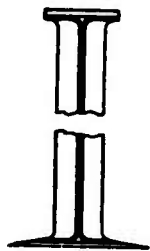
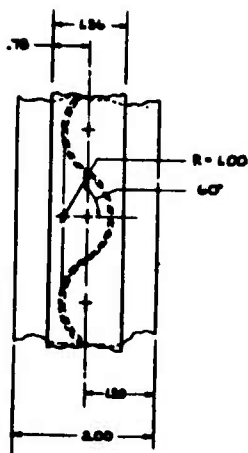
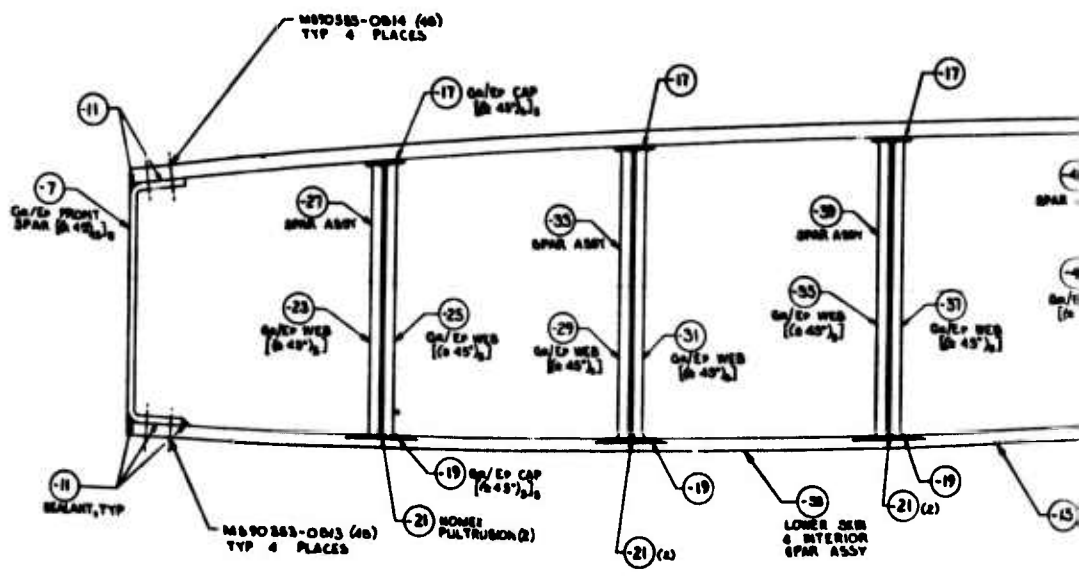
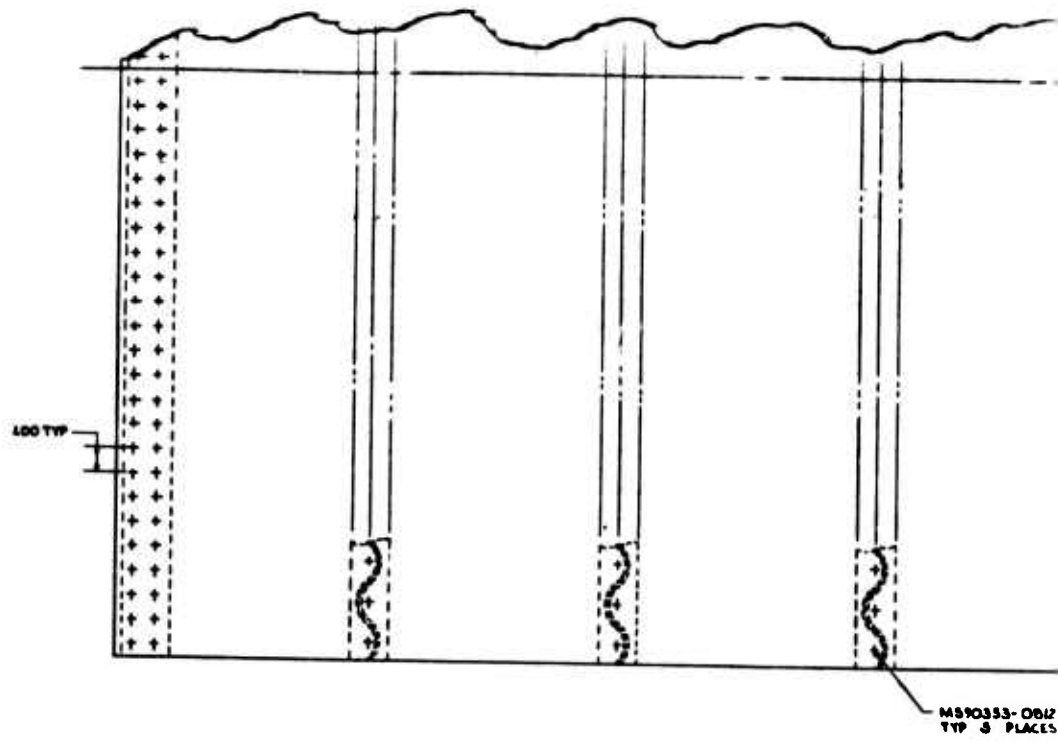


Figure 45 Composite Sandwich Spar

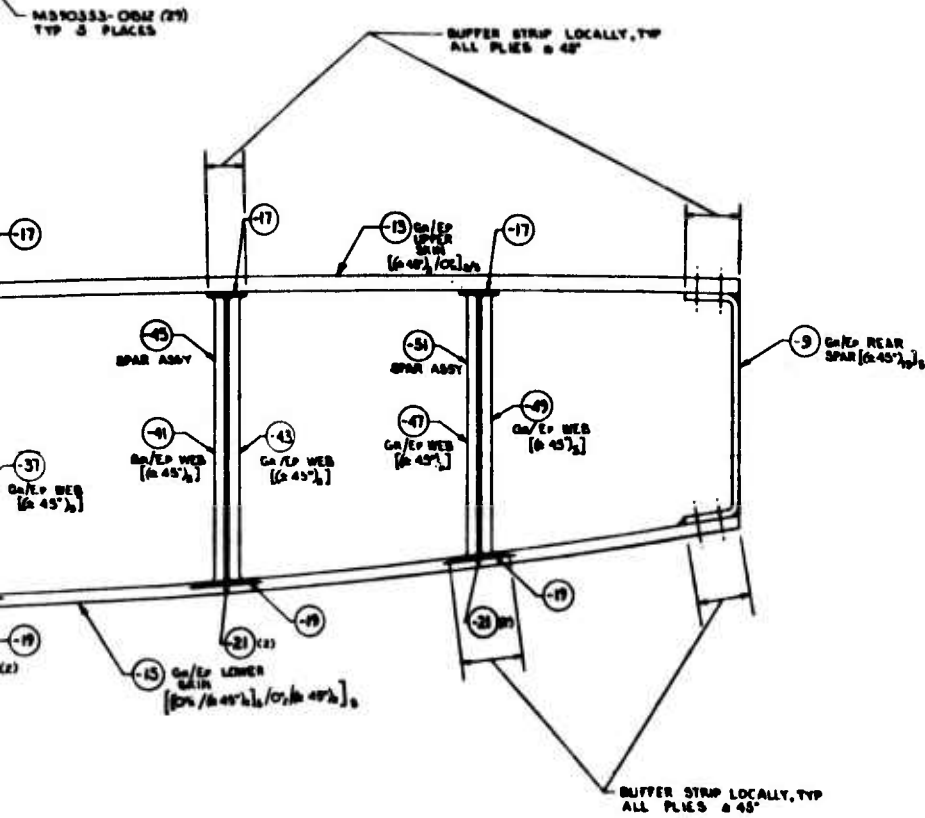
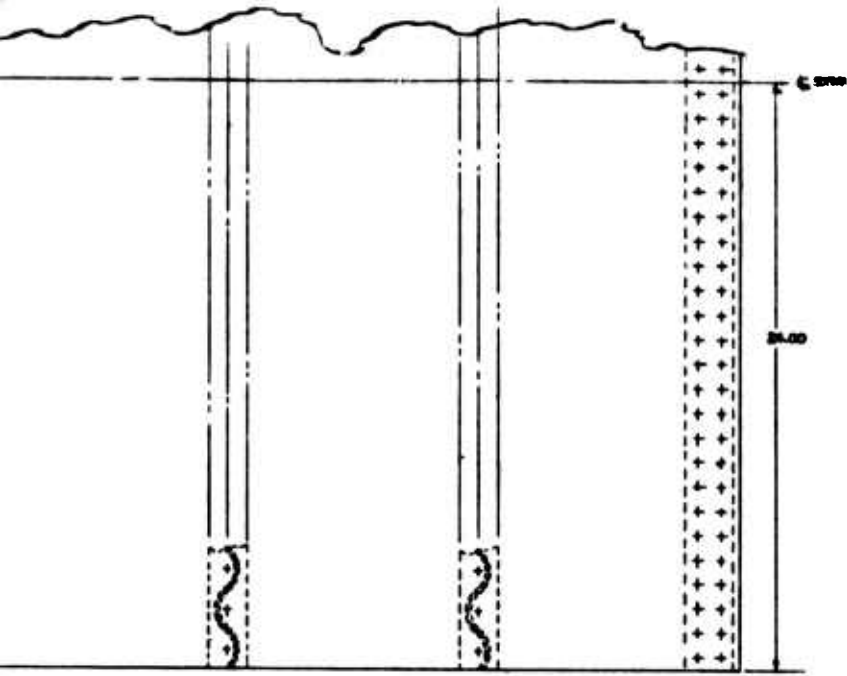
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DETAIL -27, -33, -39, -45, & -51 SPAR ASSYS SCALE - 1/1



-1 ASSY



ASSY	SUB ASSY	DETAIL	PART NAME	NO. REQ'D	MATERIAL
		-7	FRONT SPAR	1	3" GR/EP TAPE
		-9	REAR SPAR	1	3" GR/EP TAPE
		-11	SEALANT	AR	1.5" GR/EP TAPE
		-13	UPPER SKIN	1	3" GR/EP TAPE
		-15	LOWER SKIN	1	3" GR/EP TAPE
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-17	CAP	1	3" GR/EP TAPE
		-19	CAP	1	3" GR/EP TAPE
		-21	FILLER	2	3" GR/EP TAPE
		-23	WEB	1	3" GR/EP TAPE
		-25	WEB	1	3" GR/EP TAPE
		-27	SPAR ASSY	1	
		-33	THE TOP & REAR SPAR ASSY	1	
			RIVET	145	
			RIVET	192	
			RIVET	192	
			ANALYTICAL ASSEMBLY		

Figure 46 Compos

3

PART NAME	NO. REQ'D	MATERIAL	RAW STOCK SIZE	RAW STOCK WGT	FINISHED WT.		COST			TOTAL COST
					DETAIL	ASSY	MATL.	PAB.	TOOL.	
FRONT SPAR	1	3" GR/EP TAPE	1113 FT TAPE	6.1	11.6			171.55	182.77	
REAR SPAR	1	3" GR/EP TAPE	1057 FT TAPE	5.3	11.1			168.55	182.77	
SEALANT	AR	FMS 1044	AK	7.0	1.50					
UPPER SKIN	1	3" GR/EP TAPE	3560 FT TAPE	18.6	100.0			221.81	246.82	
LOWER SKIN	1	3" GR/EP TAPE	3228 FT TAPE	16.0	90.2			193.81	218.31	
CAP	1	3" GR/EP TAPE	55 FT TAPE	.80	.57			86.07	88.27	
CAP	1	3" GR/EP TAPE	53 FT TAPE	.76	.55			82.36	84.27	
FILLER	2	1/2" x 1/2" x 48" M	.03	.07				6.53	5.88	
WEB	1	3" GR/EP TAPE	220 FT TAPE	3.20	2.30			344.32	111.65	
WEB	1	3" GR/EP TAPE	220 FT TAPE	3.20	2.30			344.32	111.65	
SPAR ASSY	1					5.99			17.63	
CAP	1	3" GR/EP TAPE	55 FT TAPE	.80	.57			86.07	88.27	
CAP	1	3" GR/EP TAPE	53 FT TAPE	.76	.55			82.36	84.27	
FILLER	2	1/2" x 1/2" x 48" M	.03	.07				6.53	5.88	
WEB	1	3" GR/EP TAPE	232 FT TAPE	3.40	2.40			36.31	11.65	
WEB	1	3" GR/EP TAPE	232 FT TAPE	3.40	2.40			36.31	11.65	
SPAR ASSY	1					5.99			17.63	
CAP	1	3" GR/EP TAPE	55 FT TAPE	.80	.57			86.07	88.27	
CAP	1	3" GR/EP TAPE	53 FT TAPE	.76	.55			82.36	84.27	
FILLER	2	1/2" x 1/2" x 48" M	.03	.07				6.53	5.88	
WEB	1	3" GR/EP TAPE	232 FT TAPE	3.40	2.40			36.31	11.65	
WEB	1	3" GR/EP TAPE	232 FT TAPE	3.40	2.40			36.31	11.65	
SPAR ASSY	1					5.99			17.63	
CAP	1	3" GR/EP TAPE	55 FT TAPE	.80	.57			86.07	88.27	
CAP	1	3" GR/EP TAPE	53 FT TAPE	.76	.55			82.36	84.27	
FILLER	2	1/2" x 1/2" x 48" M	.03	.07				6.53	5.88	
WEB	1	3" GR/EP TAPE	231 FT TAPE	3.40	2.40			36.19	11.65	
WEB	1	3" GR/EP TAPE	231 FT TAPE	3.40	2.40			36.19	11.65	
SPAR ASSY	1					5.99			17.63	
CAP	1	3" GR/EP TAPE	55 FT TAPE	.80	.57			86.07	88.27	
CAP	1	3" GR/EP TAPE	53 FT TAPE	.76	.55			82.36	84.27	
FILLER	2	1/2" x 1/2" x 48" M	.03	.07				6.53	5.88	
WEB	1	3" GR/EP TAPE	213 FT TAPE	3.10	2.30			333.57	111.65	
WEB	1	3" GR/EP TAPE	213 FT TAPE	3.10	2.30			333.57	111.65	
SPAR ASSY	1					5.99			17.63	
RIVET	145				1.0					
RIVET	192				2.3					
RIVET	192				4.1					
ANALYTICAL ASSEMBLY					252			38011	18708	437218

COPY AVAILABLE TO DDC DOES NOT PERMIT FULLY LEGIBLE PRODUCTION

PRELIMINARY DESIGN DRAWING
 ATW-4 WING - CORRUGATED
 INTERIOR SPARS INTEGRAL WITH
 LOWER SKIN, ANALYTICAL ASSY OF
 GENERAL DYNAMICS
 Convair Aerospace Division
 633RA008

Figure 46 Composite Corrugated Spar Analytical Assembly

Table V ATW COMPOSITE WING BOX ANALYTICAL ASSY EVALUATION SUMMARY

CONFIG. NO.	DESCRIPTION	STRUCT EFFIC.		TECHNOLOGY IMPROV.			DAMAGE TOL.			-ABILITIES				TOTAL SCORE	RANK
		COST* 1975 \$ (.24)	WEIGHT LBS (.16)	CONCEPT (.05)	MATL (.02)	MFG (.03)	SAFE CRACK (.08)	FAIL SAFE (.12)	INSPECT- (.06)	MFG (.06)	MAIN-TAIN (.06)	REPAIR (.06)	DUR- (.06)		
633-RA005-1	TRUSS SPAR, GRAPHITE/EPOXY CO-CURED CONSTRUCTION	43,280 .210	300.70 .134	.025	.02	.03	.080	.000	.030	.045	.050	.040	.060	.724	5
633-RA006-1	GRAPHITE/EPOXY SKINS WITH BUFFER STRIPS; GRAPHITE/EPOXY "Y" SPARS	45,308 .200	258.0 .156	.025	.02	.025	.080	.000	.060	.055	.060	.06	.045	.786	3
633-RA006-3	GRAPHITE/EPOXY SKINS NO BUFFER STRIPS; "Y" SPAR WITH "NOMEX" CONE EMBEDDED LOWER CAPS	37,812 .240	252.60 .160	.050	.02	.03	.080	.000	.060	.060	.060	.06	.060	.880	1
633-RA007-1	BURIED T1 INSERT IN GR/EP IWR SKIN; SANDWICH SPARS	48,093 .189	294.22 .137	.025	.02	.025	.080	.000	.060	.040	.060	.06	.050	.746	4
633-RA003-1	GR/EP - CORRUGATED INTERIOR SPARS INTEGRAL WITH LOWER SKIN	42,721 .208	252.00 .160	.050	.02	.03	.080	.000	.060	.055	.060	.06	.060	.843	2

* ALL COSTS REVISED TO LATEST LEARNING CURVE AND RANKED WITH RESPECT TO COMPOSITE CONCEPTS ONLY.

3.2.3 Preliminary Design Concepts

The baseline concept, two advanced metallic concepts, and one advanced composite design was defined as complete wing designs from pivot to tip on preliminary design drawings. The main features of these 4 designs are described briefly below.

3.2.3.1 Baseline Configuration

The baseline concept, 633RW000 illustrated by Figure 47 embodies the same fabrication, material and joining concepts as the FB-111 wing box. However, the supercritical airfoil geometry and is larger, 725.7 sq. ft. versus 550.0 sq. ft. for the FB-111 wing. The closure and intermediate spars are machined from 2024-T851 Al plate. The upper and lower skins are machine pocketed, then chemically milled from 2024-T851 plate. The pivot fitting is a weldment of D6ac steel machined forgings and machined plate. The lower skin is joined to the spars and pivot fitting with Taper-Loc fasteners. The upper plate is joined to the spars and bulkheads by means straight shank, close tolerance fasteners.

3.2.3.2 Laminated Lower Skin "Y"-Spar Configuration

The 633RW001-1 wing box concept is illustrated by Figure 48. This wing box concept incorporate an adhesive bonded laminated 2024-T81 lower skin without fastener penetrations, extruded aluminum "Y" intermediate spars, exposed lower spar caps, and a constant tapered non-pocketed not-etched upper skin. The upper skin is attached with blind rivets in lieu of close tolerance bolts and Davis Pressnuts. The front and rear spars are machined from 2024-T851 plate. The pivot for the -1 assembly is a closed torque box to the pivot pin spool that encloses the pivot pin. The lower skin is continued inboard of the pivot to form one of three load pathes of a fail-safe lower pivot lug.

3.2.3.3 Laminated Skin Slanted Spar Configuration

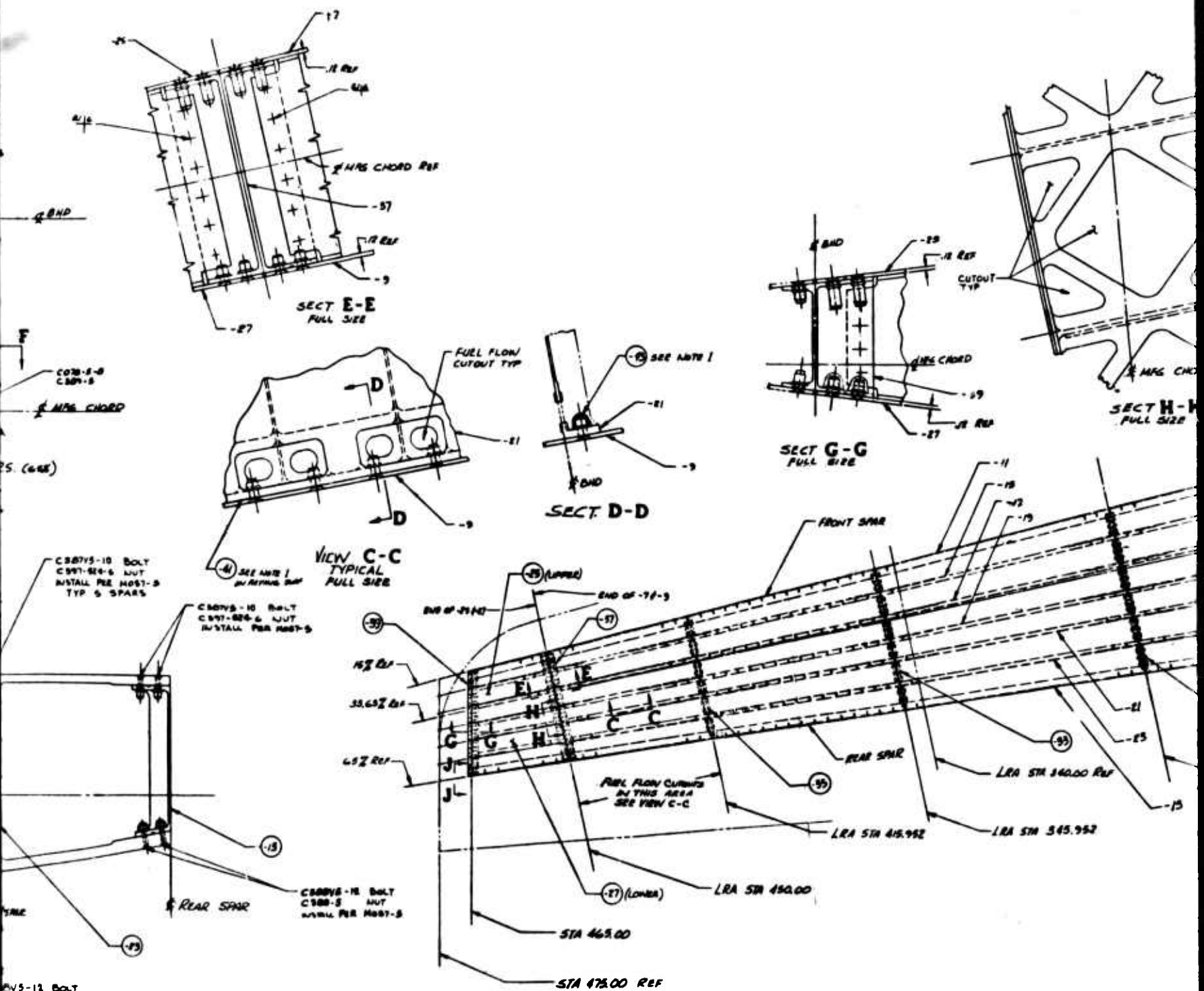
The 633RW002-1 and -3 concepts are identical except in the area of the pivot. The 633RW002-1 pivot concept is the same as the 633RW001-1 pivot concept. The 633RW002 wing box outboard of the pivot is illustrated by Figure 49. This design has a laminated adhesive bonded 2024-T81 aluminum lower skin with embedded

lower spar caps. The intermediate spars are configured of a slanted sheet aluminum web stabilized by "V" shaped sheet aluminum intercostals. Each spar has two extruded aluminum upper caps. The front and rear spars are machined of 2024-T851 aluminum plate. The upper skin is a non-pocketed non-etched constant tapered 2024-T851 aluminum plate.

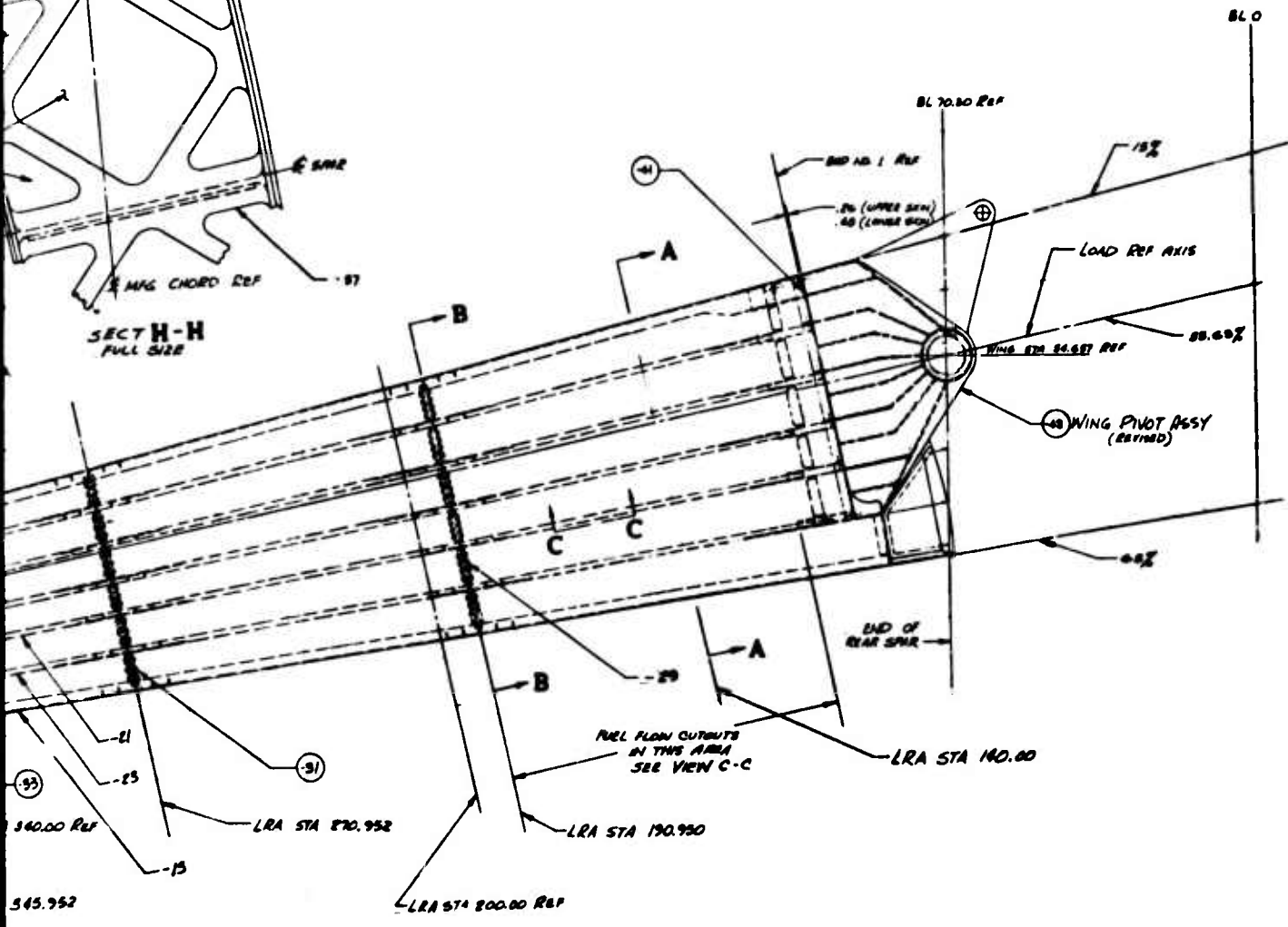
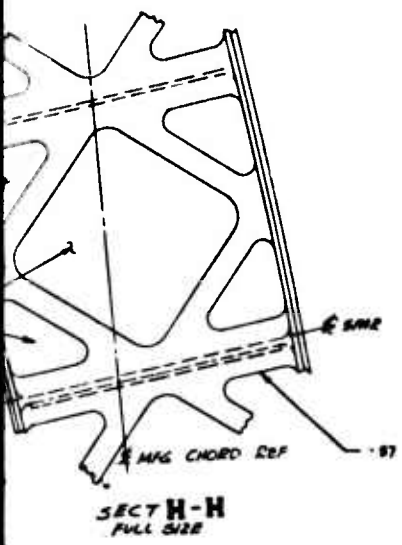
The 633RW002-3 pivot concept is similar to the -1 pivot concept except -1 is open in the area of the pivot lugs and transfers wing shear loads to the pivot spool by means of two vertical shear lugs. The lower pivot lug incorporates the same fail safe feature as the 633RW001-1 lower lug design.

3.2.3.4 Composite Wing Configuration

The 633RW003-1 and -3 designs are graphite epoxy wing box concepts. The -1 and -3 designs differ only in the pivot area. The lower skin is a solid layup design without buffer strips. The intermediate spars are of a "Y" shaped configuration with lower cap embedded in the lower skin. The lower cap design is a unique configuration that incorporates a prefabricated, embedded longitudinal member that provides transverse or chordwise bending continuity, longitudinal axial load continuity plus a shear load path in the lower skin at its intersection with the vertical web of the intermediate spars. The intermediate spars have a core of nomex laminate with graphite epoxy outer plies. The front and rear spars and the upper skin are solid graphite epoxy laminate with no buffer strips. The upper skin is attached with blind rivets. The -1 pivot concept is open in the pivot area with double shear fittings between the inboard pivot bulkhead and the spool that encapsulates the pivot pin. The -3 pivot concept is a closed torque box design. Both -1 and -3 designs are configured with the upper and lower skins forming a major load carrying layer of the pivot lugs. The metal in the pivot lug members is titanium. The 633RW003-1 and -3 designs are shown in Figure 50.



3'



DASH	SUB	ASSY
-29		
-51		
-53		
-55		
-48		
-47		
-49		
-1		

* 3, -29 TH

2 -45 WING

1. ALL SEALS SHALL BE

NOTES:

4

DASH NO.	ASSY	SUB ASSY	DETAIL	PART NAME	NO. REQD	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED WT.		COST			TOOL COST
									DETAIL	ASSY	MATL	FAB.	TOOL.	
			-7	UPPER SKIN	1	2024-T851	1.501771368	4,282.8	616.92					
			-9	LOWER SKIN	1	2024-T851	1.751771368	5,008.4	798.00					
			-11	FRONT SPAR	1									
			-13	REAR SPAR	1	2024-T851		1,558.5	511.99					
			-15	SPAR	1									
			-17	SPAR	1									
			-19	SPAR	1									
			-21	SPAR	1									
			-23	SPAR	1									
			-25	SKIN (OUTER LOWER BND)	1	2024-T851		67.17	15.66					
			-27	SKIN (INNER LOWER BND)	1									
			-29	BULKHEAD	1									
			-31	BULKHEAD	2	2024-T851		1,468.00	114.80					
			-33	BULKHEAD	2									
			-35	BULKHEAD	2									
			-37	BULKHEAD	1									
			-39	BULKHEAD	1									
			-41	BULKHEAD	1									
			-43	WING PIVOT ASSY	1	PRECIPITATION RESISTANT 5052 ALUMINUM		1,254.00	781.70					
			-45	WING PIVOT FITTING PARTS	1	2024-T851		148.00	20.90					
			-47	GENERAL SEALANT	---	PMS-1064		6.84	3.80					
			-49	FILET SEALANT	---									
				STRAIGHT SHANK BOLT (C38713-10)	1,530				214.2					
				WRENCH FASTENER (C078-5-8)	80				1.80					
				WRENCH FASTENER (C080-8-8)	242				1.60					
				TAPER LOCK WRENCH FASTENER (C38816-12)	1,530				99.00					
				WRENCH FASTENER (C38816-4)	1,530				14.80					
				NUT (C022-8)	242				.30					
				NUT (C388-5)	1,530				10.80					
				TOTAL FASTENER WT.					384.40					
			-1	WING BOX ASSY					27570					

*3.-29 THRU-35 EACH MADE UP OF 6 PIECES.

2.-43 WING PIVOT FITTING SAME CONCEPT AS FIG 2

1. ALL SEALING MATERIALS & METHODS OF APPLICATION SHALL BE AS SPECIFIED IN PMS-1064.

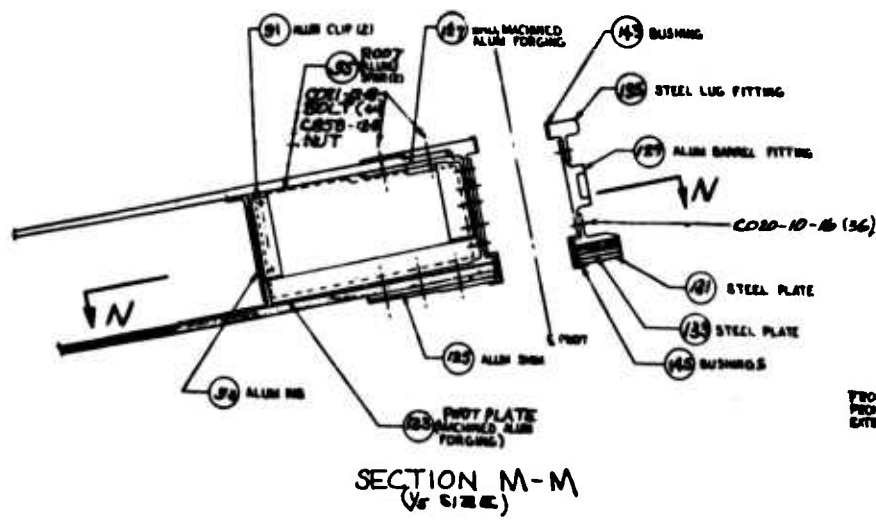
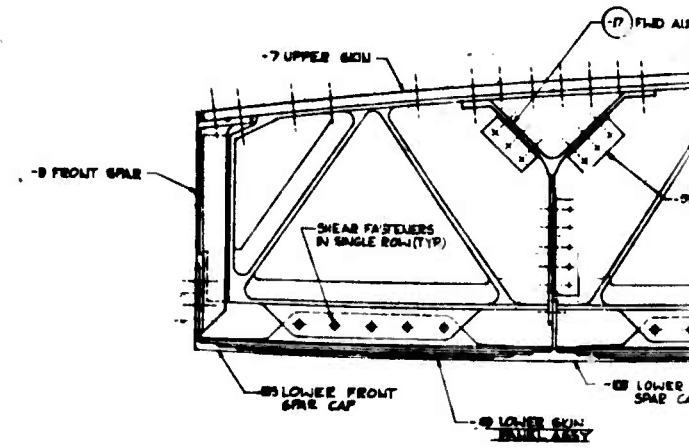
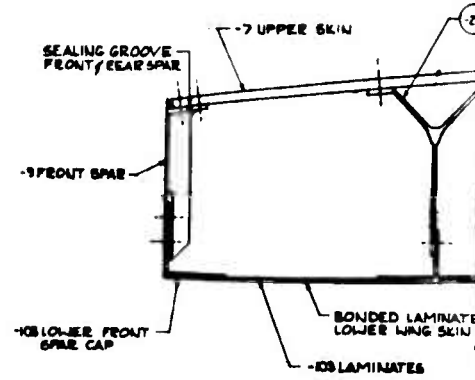
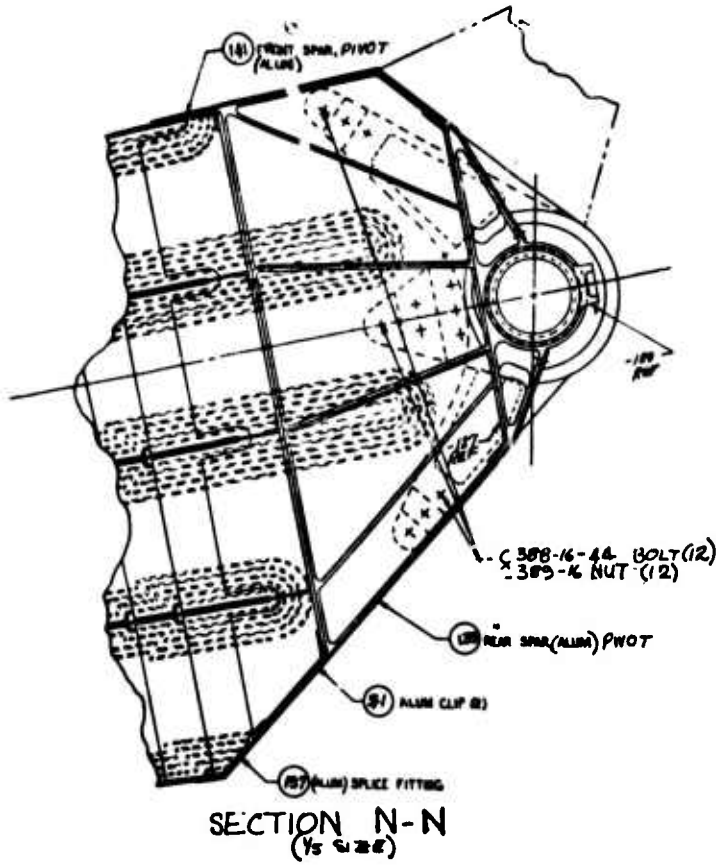
NOTES:

PRELIMINARY DESIGN DRAWING

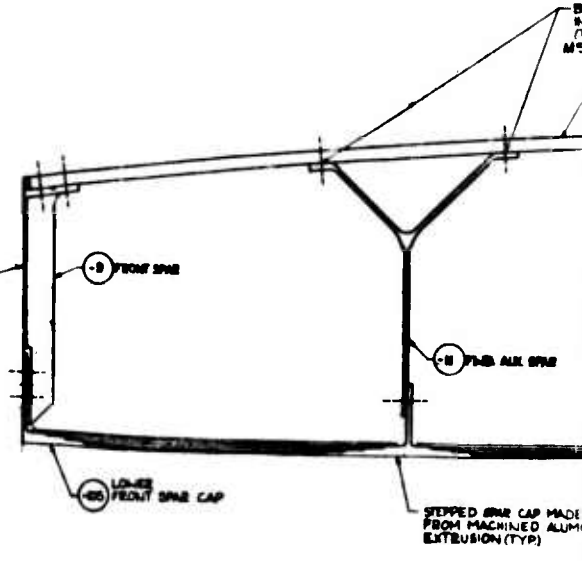
WING BOX-ATW-4
BASELINE

GENERAL DYNAMICS Convair Aerospace Division	633 R1000
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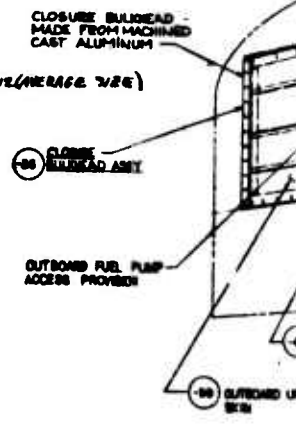
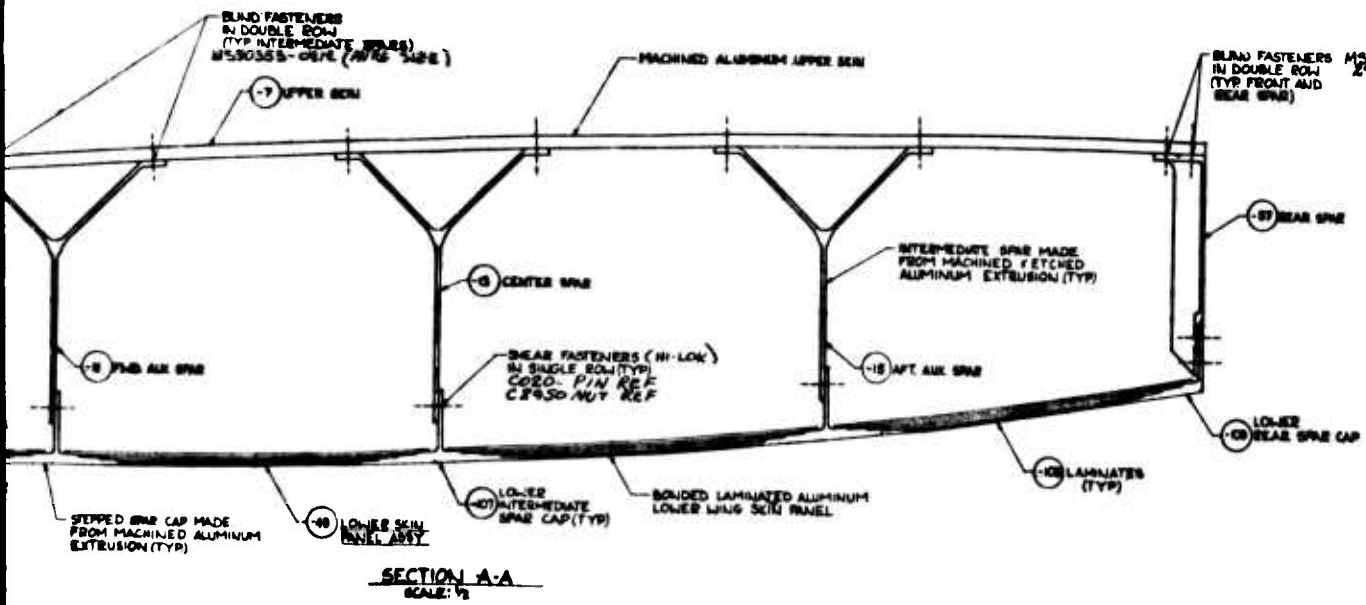
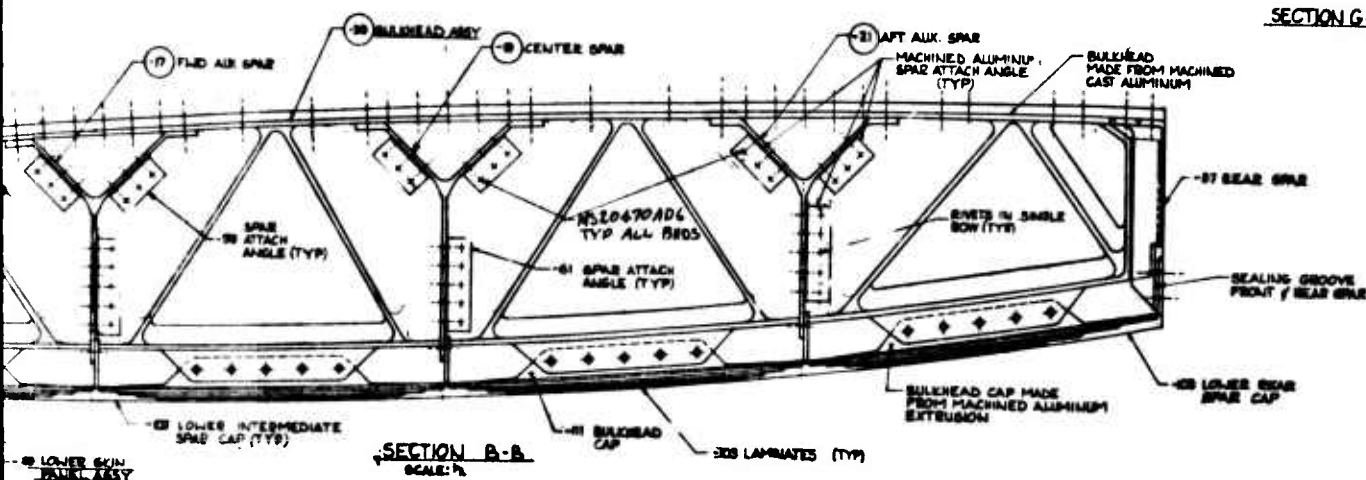
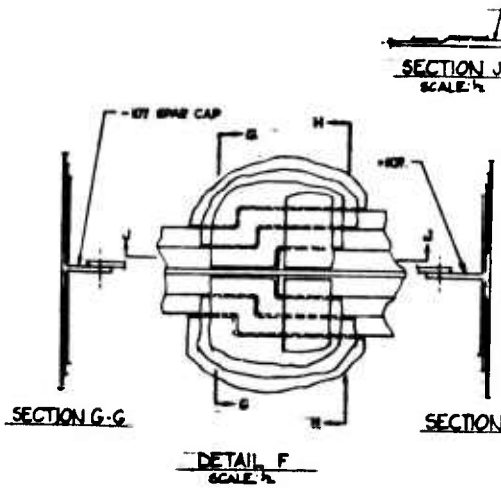
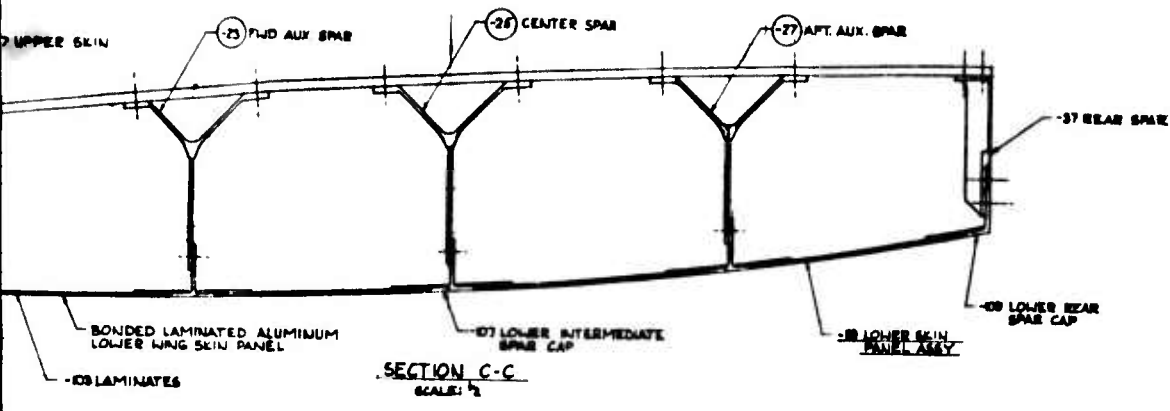
Figure 47 Baseline Preliminary Design



FRONT / REAR SPAR MADE FROM MACHINED ALUMINUM EXTRUSION (TYP)

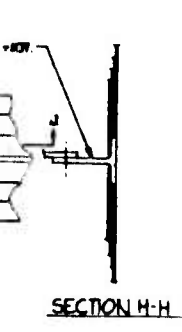


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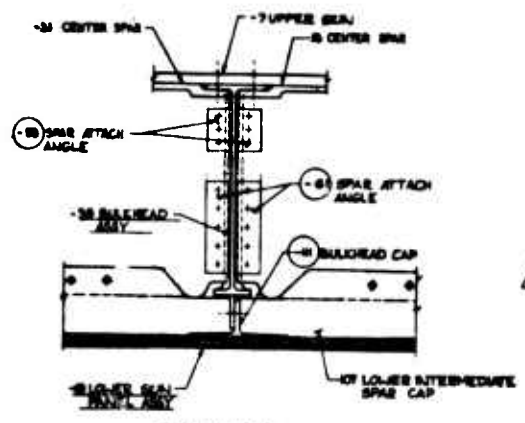


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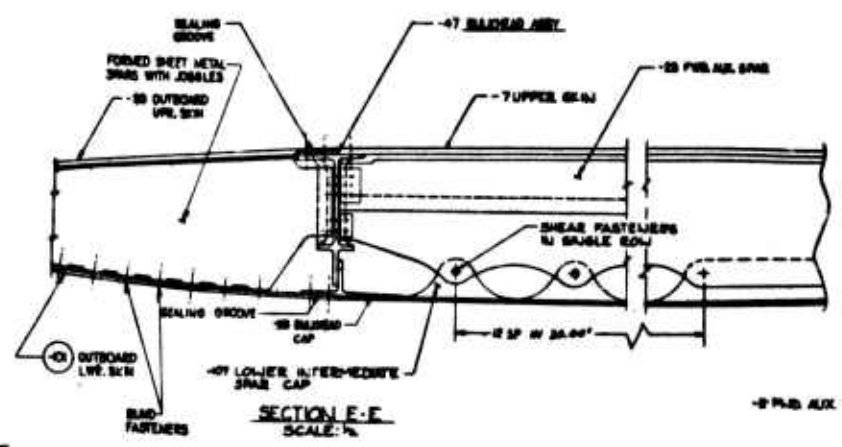
SECTION J-J
SCALE: 1/2"



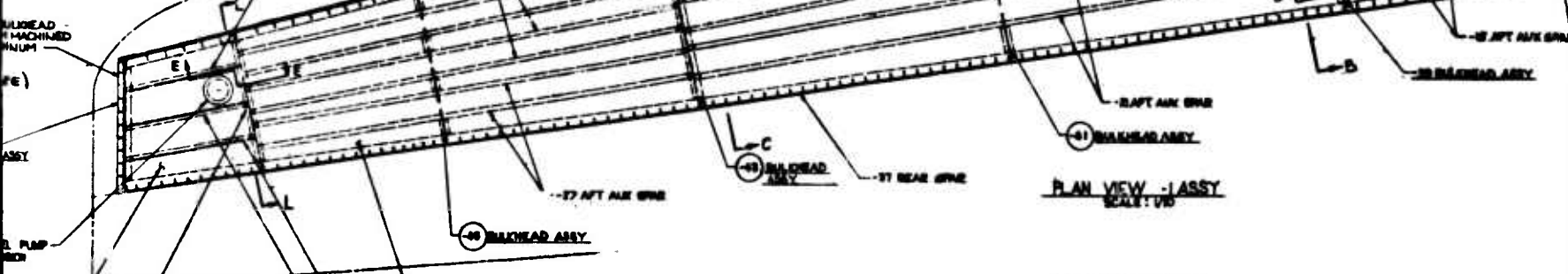
SECTION H-H



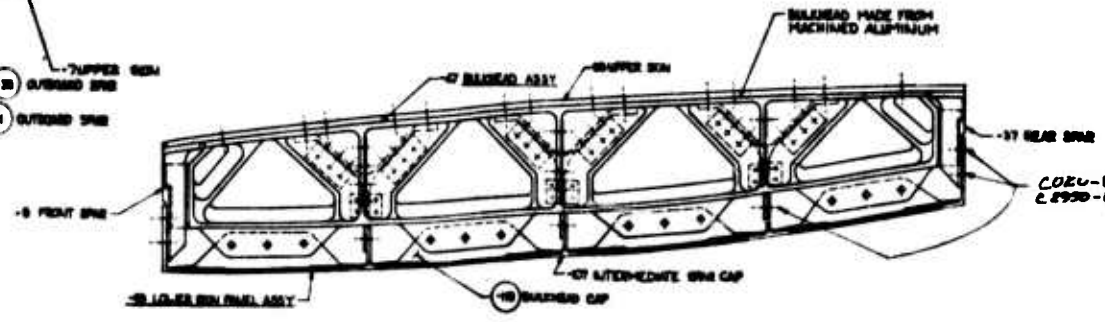
SECTION D-D
SCALE: 1/2"



SECTION E-E
SCALE: 1/2"



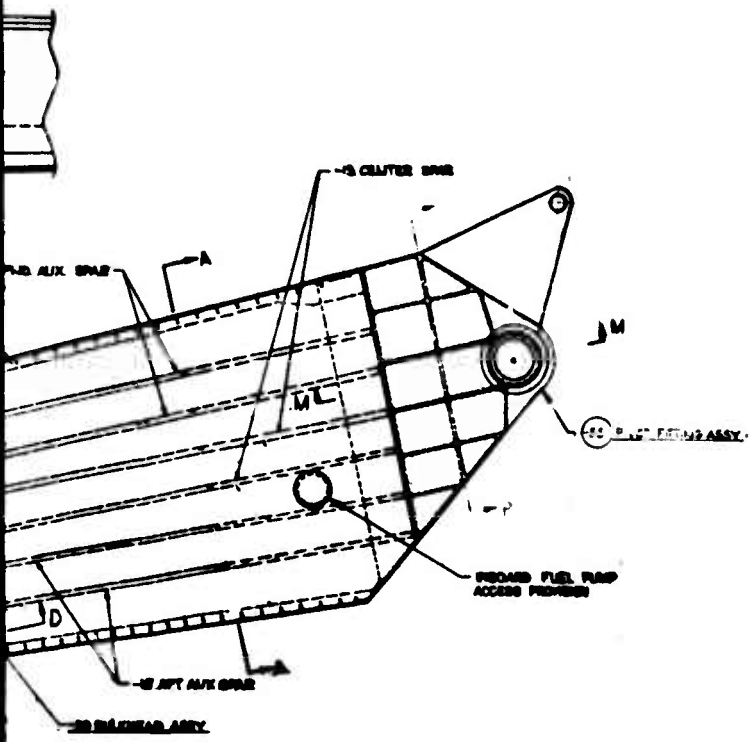
PLAN VIEW - ASSY
SCALE: 1/8"



SECTION I-I
SCALE: 1/2"

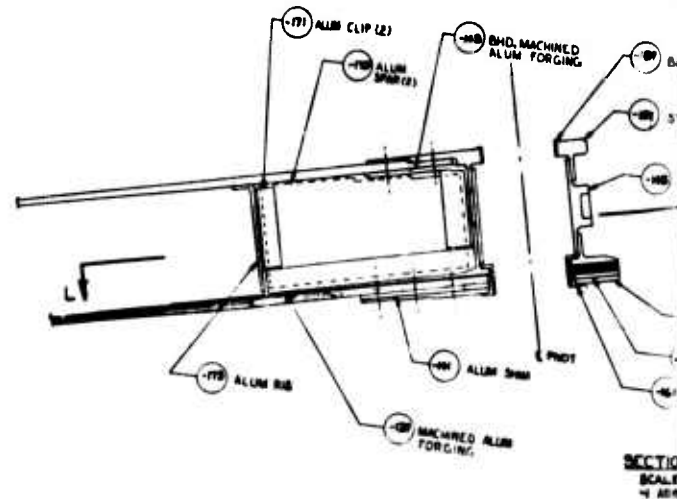
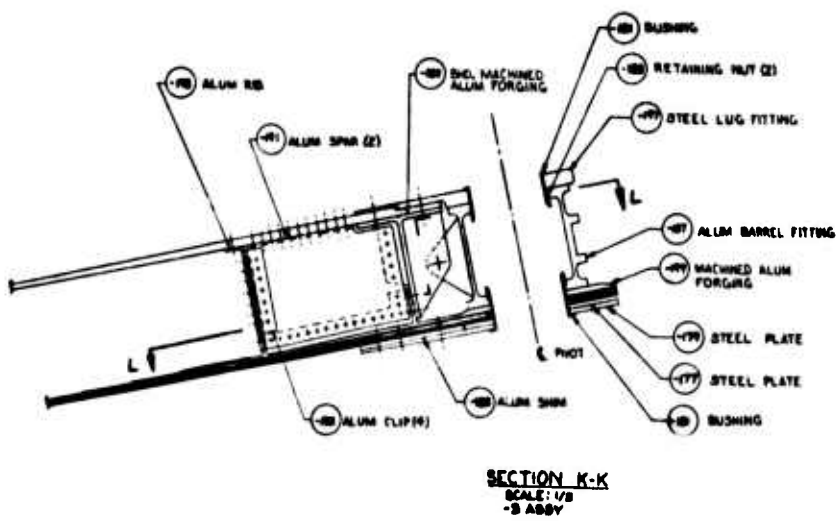
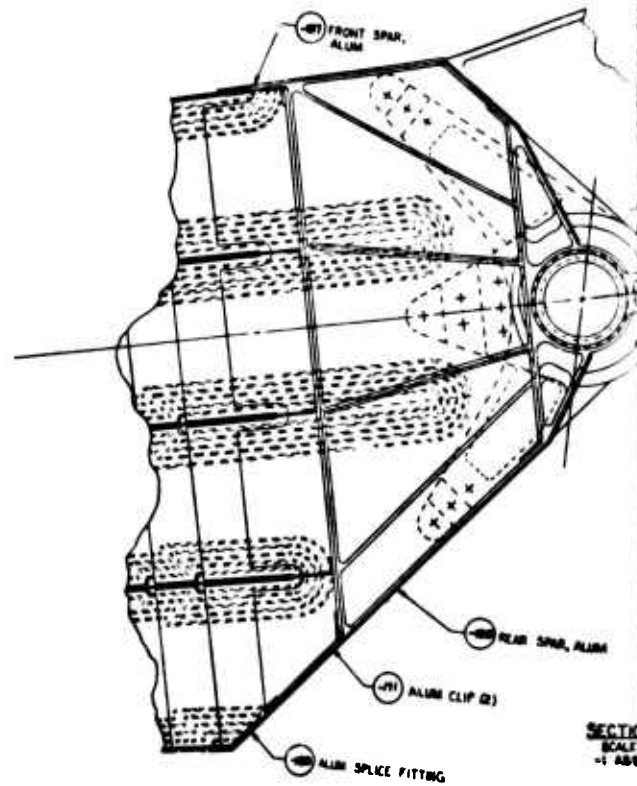
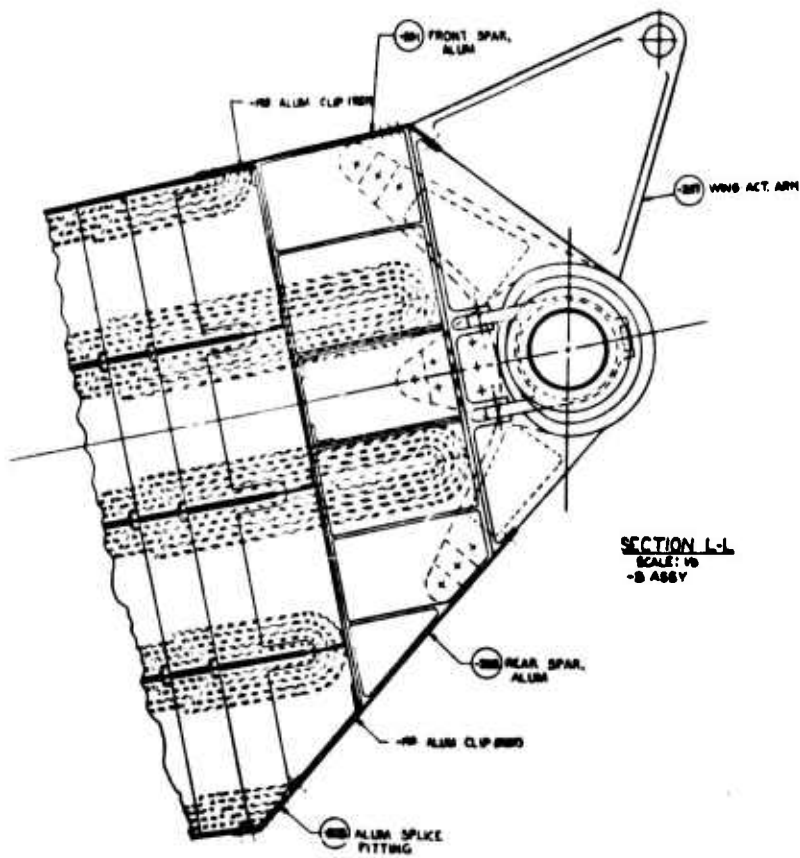
COZU-B-6 PW } AVERAGE GAGE
C-8930-B NUT } 740 pics

4

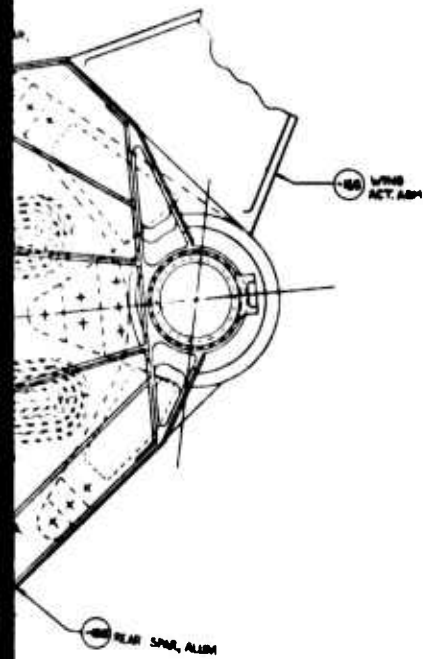


AVG } AVERAGE SIZE
MUT } 740 pics

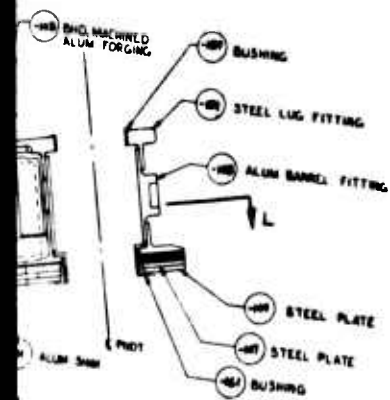
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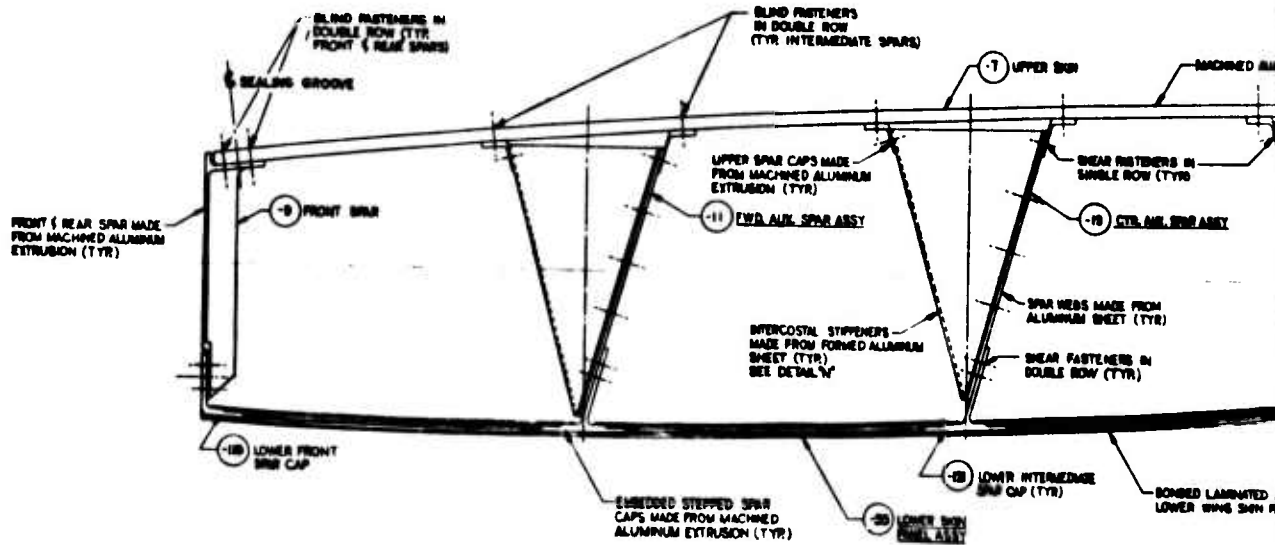
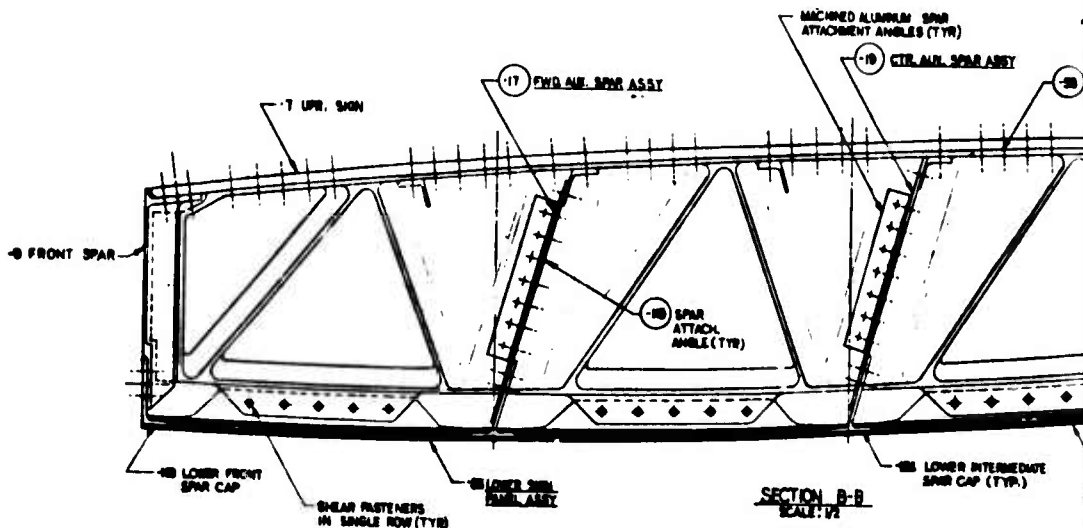
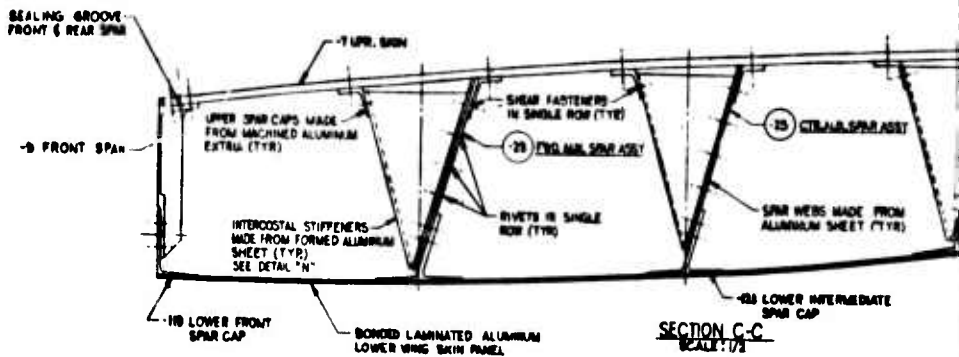
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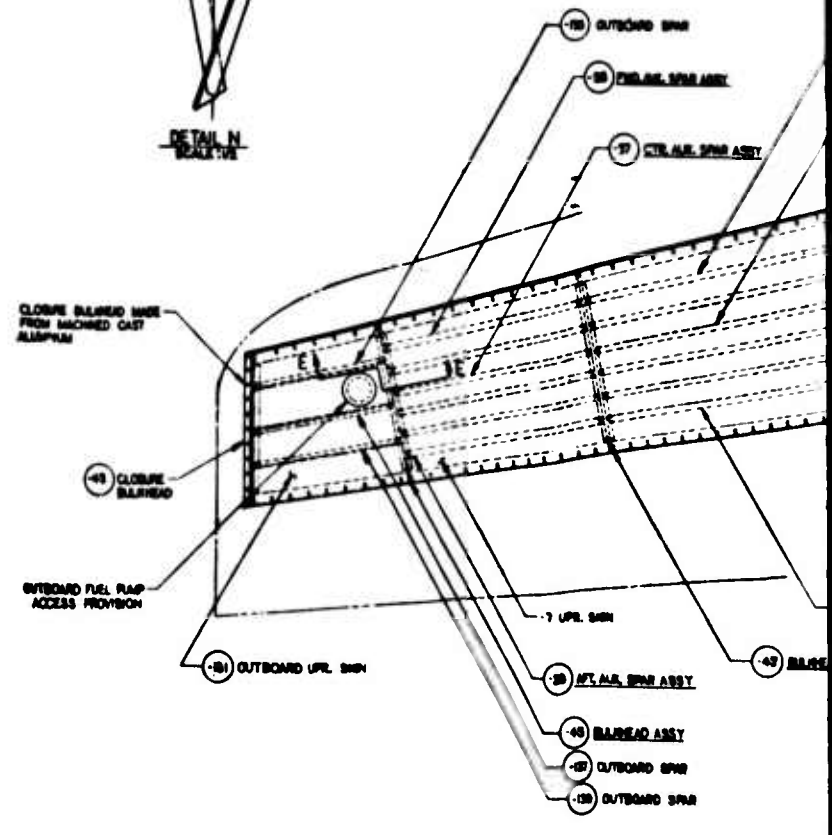
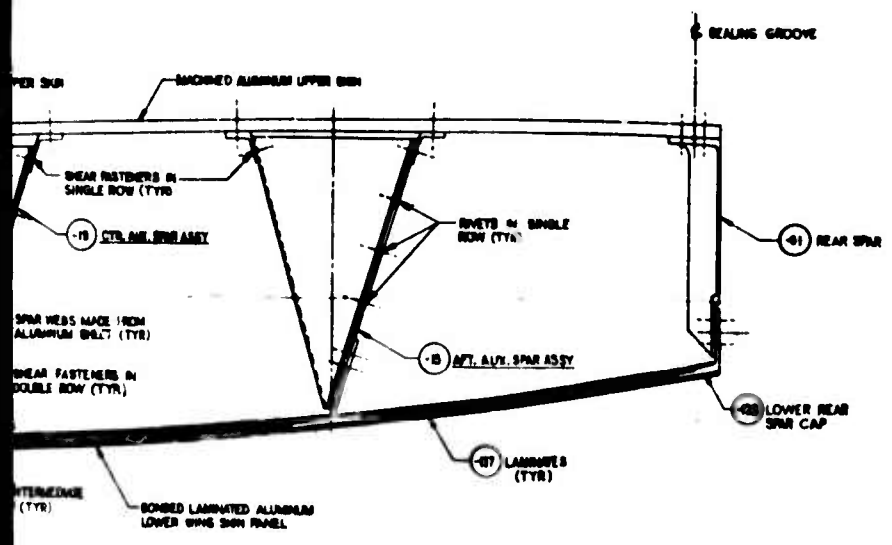
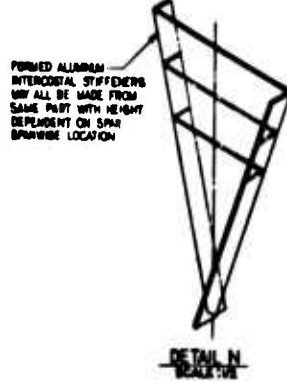
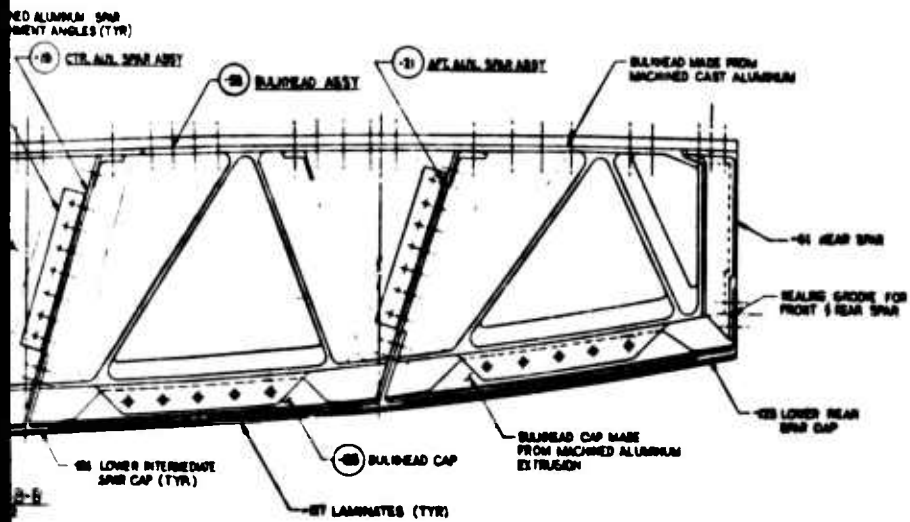
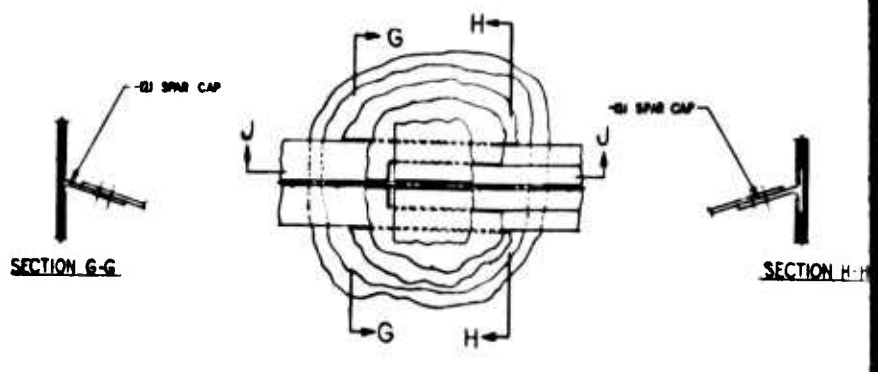
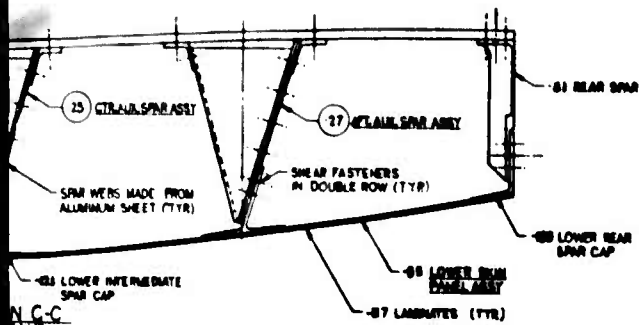
SECTION L-L
SCALE: 1/2
-1 ABBY



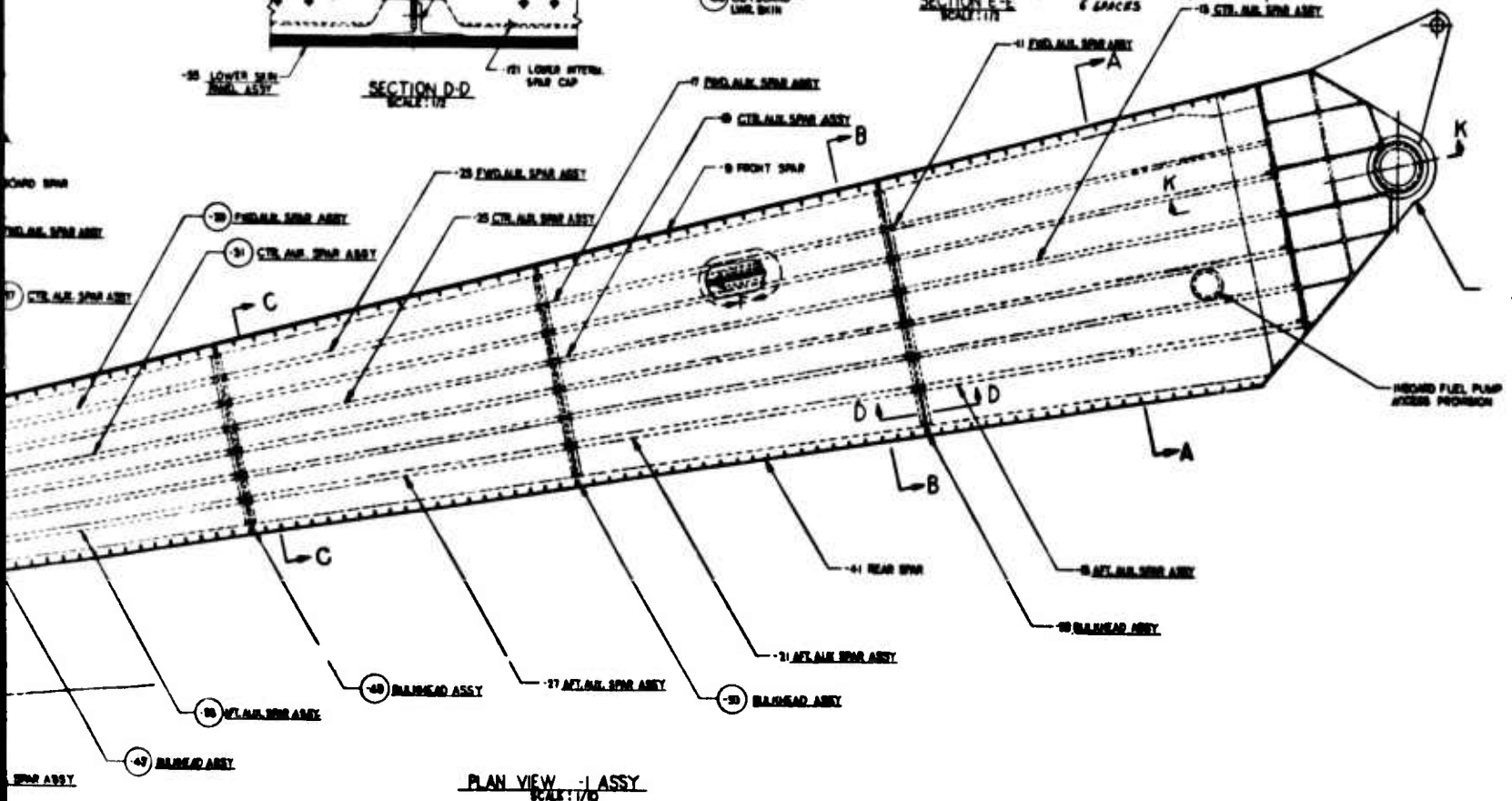
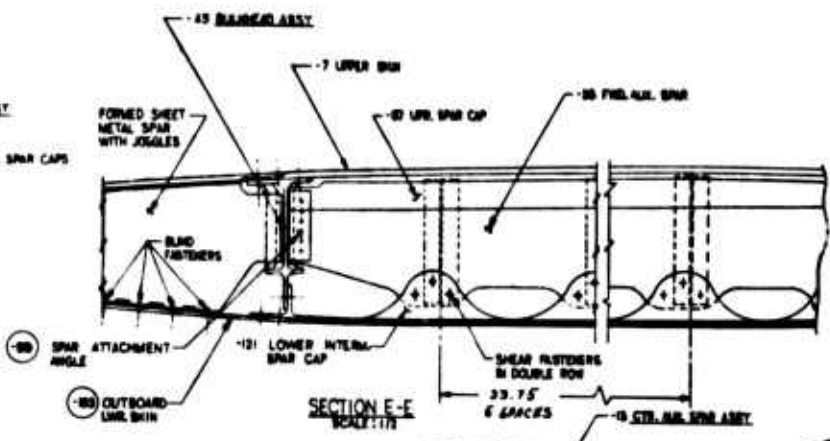
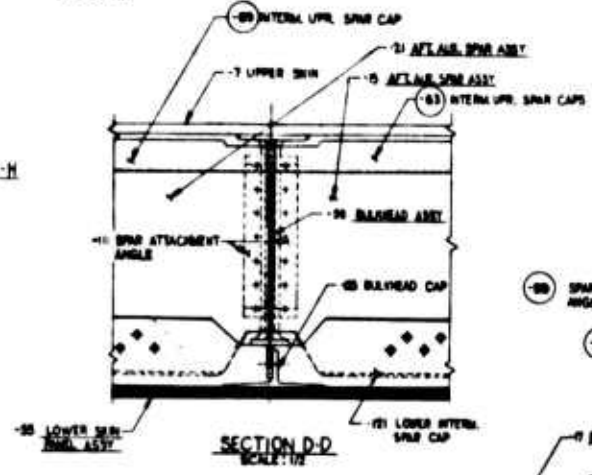
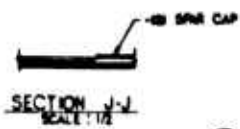
SECTION K-K
SCALE: 1/2
-4 ABBY



3



4



SPAR ASSY
HEAD ASSY
BOARD SPAR
BOARD SPAR

PROT. FITTING

HEARD FUEL PUMP ACCESS POSITION

40 SPAR CAP

SECTION J-J
SCALE: 1/2

SECTION H-H

10 INTERNAL UPFR. SPAR CAP
21 ATTACH. SPAR ASSY
7 UPPER SKIN
15 ATTACH. SPAR ASSY
10 INTERNAL UPFR. SPAR CAPS
36 BALHEAD ASSY
11 SPAR ATTACHMENT ANGLE
48 BALHEAD CAP
12 LOWER SKIN
121 LOWER INTERNAL SPAR CAP

43 BALHEAD ASSY
7 UPPER SKIN
47 UPFR. SPAR CAP
36 FIBERGLASS SPAR
FORMED SHEET METAL SPAR WITH JOGGLES
BUNG FASTENERS
48 SPAR ATTACHMENT ANGLE
121 LOWER INTERNAL SPAR CAP
SHEAR FASTENERS IN DOUBLE ROW
23.75" 6 SPACES
12 CTR. LINE SPAR ASSY

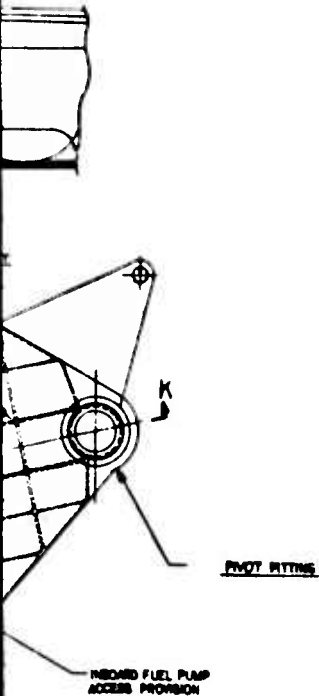
7 FIBERGLASS SPAR ASSY
10 CTR. LINE SPAR ASSY
8 FRONT SPAR
25 FIBERGLASS SPAR ASSY
25 CTR. LINE SPAR ASSY
10 FIBERGLASS SPAR ASSY
10 CTR. LINE SPAR ASSY
10 CTR. LINE SPAR ASSY
41 FIBERGLASS SPAR ASSY
A
K
K
L
HEARD FUEL PUMP ACCESS POSITION
41 REAR SPAR
8 ATTACH. SPAR ASSY
48 BALHEAD ASSY
21 ATTACH. SPAR ASSY
21 ATTACH. SPAR ASSY
30 BALHEAD ASSY
37 ATTACH. SPAR ASSY
30 BALHEAD ASSY
49 BALHEAD ASSY
C
D
D
B
A
B
C

SPAR ASSY
HEAD ASSY
BOARD SPAR
BOARD SPAR

PROT. FITTING

HEARD FUEL PUMP ACCESS POSITION

5



LIST OF MATERIALS										
ASST	SUB ASST	DETAIL	PART NAME	NO REQ'D	DESCRIPTION	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED WT. IN LBS	COST
-1	-11-R-0		INTERM. SPAR ASSY'S	1 EACH					71.76	
		-63	INTERM. UPR. SPAR CAPS	2 EACH	AL EXTR	2024-T85H	0.92x1.40x.000	17.28	17.28	
		-65	INTERCOSTAL STIFFENERS	12 EA	FORMED AL	2024-T81	0.83x1.5x0.080	33.18	15.41	
		-67	SPAR WEB	1 EACH	AL SHEET	2024-T81	12.0x90.0x.125	40.80	39.91	
			RIVET	390	COB9-S-6				0.26	
	-11-R-1		INTERM SPAR ASSY'S	1 EACH					43.61	
		-69	INTERM. UPR. SPAR CAPS	2 EACH	AL EXTR	2024-T85H	0.92x1.40x.000	17.28	17.28	
		-71	INTERCOSTAL STIFFENERS	11 EA	FORMED AL	2024-T81	0.83x1.5x0.080	33.91	10.78	
		-73	SPAR WEB	1 EACH	AL SHEET	2024-T81	10.8x80.0x0.100	23.20	23.20	
			RIVET	320	COB9-S-6				0.24	
	-11-R-2		INTERM SPAR ASSY'S	1 EACH					37.96	
		-75	INTERM UPR SPAR CAPS	2 EACH	AL EXTR	2024-T85H	0.92x1.40x.000	17.28	17.28	
		-77	INTERCOSTAL STIFFENERS	10 EA	FORMED AL	2024-T81	0.83x1.5x0.080	23.48	8.16	
		-79	SPAR WEB	1 EACH	AL SHEET	2024-T81	10.8x75.0x0.080	16.20	14.58	
			RIVET	300	COB9-S-6				0.22	
	-11-R-3		INTERM SPAR ASSY'S	1 EACH					24.23	
		-81	INTERM UPR SPAR CAPS	2 EACH	AL EXTR	2024-T85H	0.92x1.40x.000	17.28	17.28	
		-83	INTERCOSTAL STIFFENERS	3 EA	FORMED AL	2024-T81	0.83x1.5x0.080	16.77	4.38	
		-85	SPAR WEB	1 EACH	AL SHEET	2024-T81	5.0x60.0x0.080	9.07	7.14	
			RIVET	280	COB9-S-6				0.19	
	-11-R-4		INTERM SPAR ASSY'S	1 EACH					16.80	
		-87	INTERM UPR SPAR CAPS	2 EACH	AL EXTR	2024-T85H	0.92x1.40x.000	17.28	17.28	
		-89	INTERCOSTAL STIFFENERS	6 EA	FORMED AL	2024-T81	0.83x1.5x0.080	14.08	2.70	
		-91	SPAR WEB	1 EACH	AL SHEET	2024-T81	6.9x45.0x0.080	8.89	6.10	
			RIVET	280	COB9-S-6				0.16	
	-43		CLOSURE BULKHEAD	1	ALUM. CASTING	A386-T6	43.5x1.0	433	3.87	
			BULKHEAD ASSY	1					3.87	
		-97	BULKHEAD	1	MACH ALUM.	2024-T81	6.5x38.0x2.75	62.58	1.46	
		-99	SPAR ATTACH. ANGLE	3	FORMED AL ANGLES	2024-T81	2.0x0.63x3.0	0.11	0.11	
			BULKHEAD ASSY	1					4.80	
		-101	BULKHEAD	1	MACH ALUM.	2024-T81	7.8x40.0x2.75	82.50	4.49	
		-103	SPAR ATTACH. ANGLE	6	FORMED AL. ANGLES	2024-T81	2.0x0.63x4.1	0.31	0.31	
			BULKHEAD ASSY	1					6.80	
		-105	BULKHEAD	1	MACH ALUM.	2024-T81	9.0x48.0x2.75	111.38	6.14	
		-107	SPAR ATTACH. ANGLE	6	FORMED AL ANGLES	2024-T81	2.0x0.63x4.8	0.36	0.36	
			BULKHEAD ASSY	1					8.80	
		-109	BULKHEAD	1	MACH ALUM.	2024-T81	10.5x52.0x2.75	150.18	8.43	
		-111	SPAR ATTACH. ANGLE	6	FORMED AL ANGLES	2024-T81	3.0x0.89x5.7	0.43	0.43	
			BULKHEAD ASSY	1					11.76	
		-113	BULKHEAD	1	MACH ALUM.	2024-T81	12.0x60.0x2.75	186.00	11.29	
		-115	SPAR ATTACH. ANGLE	9	FORMED AL ANGLES	2024-T81	2.0x0.63x6.9	0.31	0.31	
	-55		LOWER SKIN PANEL ASSY	1					884.77	
		-117	LAMINATES	AS REQ'D	PROFILE, CHEM. ETCH A CONSTANT TAPER IN 20% OF THE TOTAL AREA OF ALL LAMINATES, & ROLL FORM CONTOUR PRIOR TO BONDING	2024-T81	0.63x70x1.49 0.63x70x1.49 0.63x70x1.50 0.71x70x1.50 0.71x70x1.55 0.71x70x1.55	674.48	886.77	
		-119	LWR FRONT SPAR CAP	1	AL EXTR.	2024-T85H	1.0x1.4x.3500	35.0	23.48	
		-121	LWR INTERM SPAR CAP	9	AL EXTR.	2024-T85H	2.0x1.4x.3500	210.0	117.52	
		-123	LWR REAR SPAR CAP	1	AL EXTR.	2024-T85H	1.0x1.4x.3500	35.0	23.48	
		-125	BULKHEAD CAP	5	ALUM. EXTR.	2024-T85H	0.97x1.40x0.080	20.0	1.43	
		-127	ADHESIVE			FM931Q3	52A.8x1.0	42.0	36.0	
		-129	PIVOT PLATE	1	MACH FORGING	2024-T85H	48x58x4	158.0	68.7	
		-141	BUSH (PIVOT FTG)	1	MACH PLATE	2024-T81	2x14x0.60	12.2	0.1	
		-143	BULKHEAD	1	MACH FORGING	2024-T81	10x10x48	40.0	36.0	
		-145	BARREL FTG	1	MACH FORGING	2024-T81	75Dx11.3	141.0	13.8	
		-147	PLATE (ALMOND)	1	MACH PLATE	10N STEEL	30x20x0.75	80.0	36.0	
		-149	PLATE (BOOMERANG)	1	MACH PLATE	10N STEEL	85x75x0.75	212.0	88.0	
		-151	PIVOT LUG, UPR	1	MACH FORGING	10N STEEL	22x49x18	448.0	166.0	
		-153	SPRICE FTG	1	MACH EXTR.	2024-T85H	0.5x1.8	0.6	0.3	
		-155	REAR SPAR PIVOT	1	MACH EXTR.	2024-T85H	2.0x1.4x44	15.0	11.3	
		-157	FRONT SPAR PIVOT	1	MACH EXTR.	2024-T85H	2.0x1.4x27	10.0	8.1	
		-159	BUSHING	1	MACH FORGING	10N STEEL	100Dx1.2	16.8	3.3	
		-161	BUSHING	2	MACH FORGING	10N STEEL	90Dx1.2	26.4	16.4	
		-163	WING ACT ARM	1	MACH FORGING	2024-T81	26x28	76.0	24.0	
		-165	MISC FITTINGS						38.1	
		-167	TANK COATING			POLYURETHANE			7.0	
		-169	SEALANT # FRON						0.1	
			BOLT	12	C388-13-04				3.04	
			NUT	12	C389-13				0.48	
			RIVET, BLIND	2320	MS 40323-082				42.3	
			PH	740	CCTC-6				4.2	
			NUT	740	C295C				1.92	

STACK NO.	RAW STOCK WEIGHT	FINISHED WT. IN LBS.	COST \$
		71.76	
	17.26	17.26	
	88.18	15.41	
	40.80	36.81	
		0.26	
		42.81	
	19.38	19.38	
	23.91	10.73	
	26.20	23.38	
		0.24	
		27.96	
	14.40	14.40	
	28.48	8.16	
	16.20	14.58	
		0.22	
		24.23	
	11.82	11.82	
	8.77	4.58	
	9.07	7.94	
		0.19	
		16.80	
	8.84	8.84	
	14.08	2.70	
	8.88	8.10	
		0.16	
		4.23	
		3.87	
		3.87	
		2.46	
		0.11	
		9.80	
		4.44	
		6.31	
		6.80	
		6.16	
		6.56	
		8.83	
		8.42	
		0.48	
		11.76	
		11.23	
		0.81	
		884.77	
		884.77	
		36.0	
		23.48	
		210.0	
		117.82	
		36.0	
		23.48	
		30.0	
		142	
		42.0	
		36.0	
		168.0	
		68.7	
		10.2	
		8.1	
		90.0	
		36.0	
		141.0	
		15.8	
		80.0	
		36.0	
		212.0	
		80.0	
		498.0	
		166.0	
		8	
		0.6	
		0.3	
		18.0	
		11.2	
		10.0	
		9.1	
		10.8	
		3.6	
		85.8	
		16.4	
		34.0	
		36.1	
		77.0	
		8.8	
		30.4	
		6.6	
		42.9	
		4.2	
		1.92	

LIST OF MATERIALS

ASST	SUB ASST	DETAIL	PART NAME	NO. REQ'D	DESCRIPTION	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FINISHED WT IN LBS.	COST \$
-1	CONT.		RIVET	300	M520426 ADG				0.2	
		-7	UPPER SKIN	1	MACHINE FROM PLATE	2024-T8B1	87" x 2 1/2" x 1/8"	198.0	638.0	
		-9	FRONT SPAR	1	MACHINE FROM EXTRU.	2024-T8B1	8 1/2" x 1"	8 1/2	67.90	
		-41	REAR SPAR	1	MACHINE FROM EXTRU.	2024-T8B1	8 1/2" x 1"	8 1/2	64.20	
		-131	OUTBD. UPR. SKIN	1	STRETCH FORM & FETCH	2024-T81	0.190 x 36" x 40"	22.0	12.88	
		-133	OUTBD. LWR. SKIN	1	STRETCH FORM & FETCH	2024-T81	0.180 x 36" x 40"	22.0	12.88	
		-135	OUTBD. SPAR	1	FORMED AL.	2024-T81	0.80" x 7" x 6"	1.80	1.40	
		-137	OUTBD. SPAR	1	FORMED AL.	2024-T81	0.80" x 7" x 6"	1.80	1.40	
		-139	OUTBD. SPAR	1	FORMED AL.	2024-T81	0.80" x 7" x 6"	1.80	1.40	
		-171	CLIP	4	MACH. EXTR.	2024-T8B1	9.0" x 3.0" x 0.18"	1.6	1.4	
		-173	RIB	1	MACH. EXTR.	2024-T8B1	1.8" x 1.8"	12.0	9.8	
		-175	BOOT SPARS	2	MACH. EXTR.	2024-T8B1	1.8" x 1.8"	9.8	7.8	
			BOLT	40	COSB-12-18				13.76	
			NUT, BLIND	40	COSB-12-18-B				7.42	
			WING BOX ASSY	1					240.0	

NOTE: -3 ASSY SAME AS -1 EXCEPT FOR SUBSTITUTIONS & ADDITIONS AS SHOWN IN -3 ASSY L.M. BELOW.

ASST	SUB ASST	DETAIL	PART NAME	NO. REQ'D	DESCRIPTION	MATERIAL	RAW STOCK SIZE	RAW STOCK WEIGHT	FIN. WEIGHT	COST \$
-3		-177	PLATE (ALMOND)	1	MACH. PLATE	10# STEEL	20" x 20" x 0.75"	177.4	8.8	
		-179	PLATE (BOOMERANG)	1	MACH. PLATE	10# STEEL	28" x 28" x 1.0"	272.4	86.9	
		-181	BUSHING	2	MACH. FORGING	10# STEEL	9.80" x 9.0"	30.0	7.0	
		-183	RETAINING NUT	2	MACH. FORGING	10# STEEL	9.80" x 9.0"	12.8	4.3	
		-185	SHIM	1	MACH. PLATE	2024-T8B1	12" x 16" x 1.0"	18.1	8.1	
		-187	BARREL FITTING	1	MACH. FORGING	2024-T8B1	18.0" x 1.8"	81.0	68.6	
		-189	BLEEKHEAD	1	MACH. FORGING	2024-T8B1	10" x 10" x 4"	68.0	48.7	
		-191	INTERIOR SPAR	2	MACH. EXTR.	2024-T8B1	18" x 1.8"	8.8	8.6	
		-193	CLIP	4	FORMED AL.	2024-T81	6.0" x 3.0" x 0.18"	1.6	1.8	
		-195	RIB	1	MACH. EXTR.	2024-T8B1	1.8" x 1.8"	12.0	9.8	
		-197	LUG FITTING	1	MACH. FORGING	10# STEEL	2.8" x 4.8" x 1.5"	200.0	96.6	
		-199	PIVOT PLATE	1	MACH. FORGING	2024-T8B1	4.8" x 8.8" x 4"	200.0	94.4	
		-201	FRONT SPAR	1	MACH. EXTR.	2024-T8B1	20" x 1.8"	8.4	8.2	
		-203	REAR SPAR	1	MACH. EXTR.	2024-T8B1	20" x 1.8"	8.4	8.4	
		-205	BRACE P.T.E.	1	MACH. EXTR.	2024-T8B1	2.0" x 1.8"	0.7	0.7	
		-207	WING ACT ARM	1	MACH. FORGING	10# STEEL	26" x 28"	96.0	24.0	
			BOLT	2	COSB-10-36				0.67	
			NUT	2	COSB-10-36				0.28	
			ALT WING BOX ASSY	1					240.0	

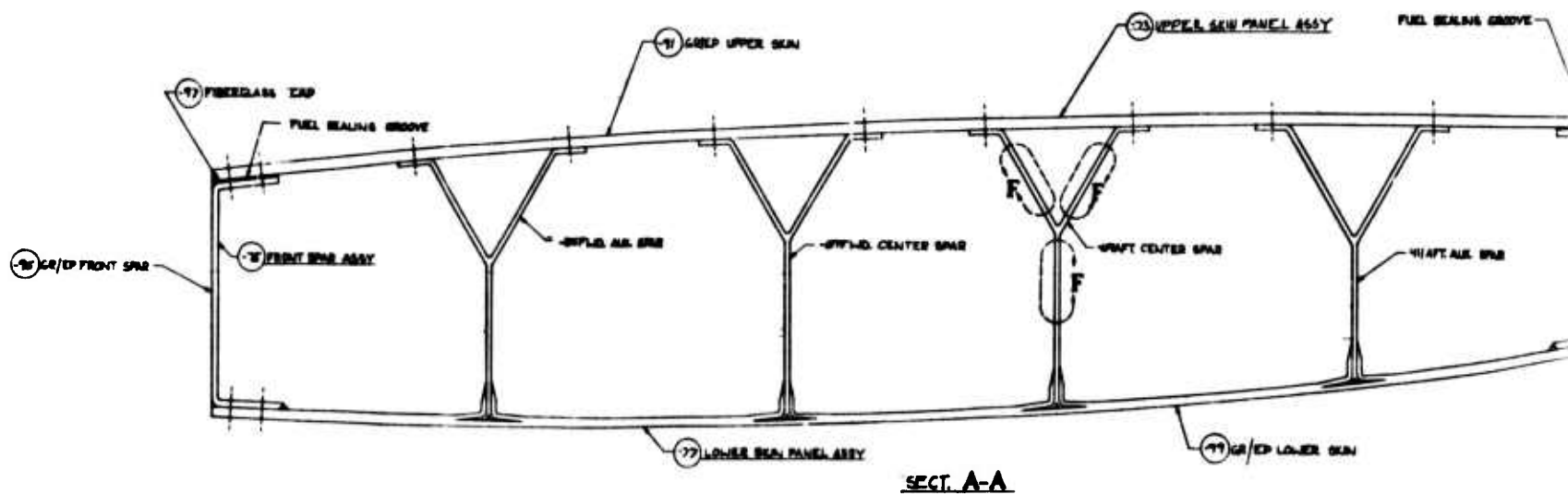
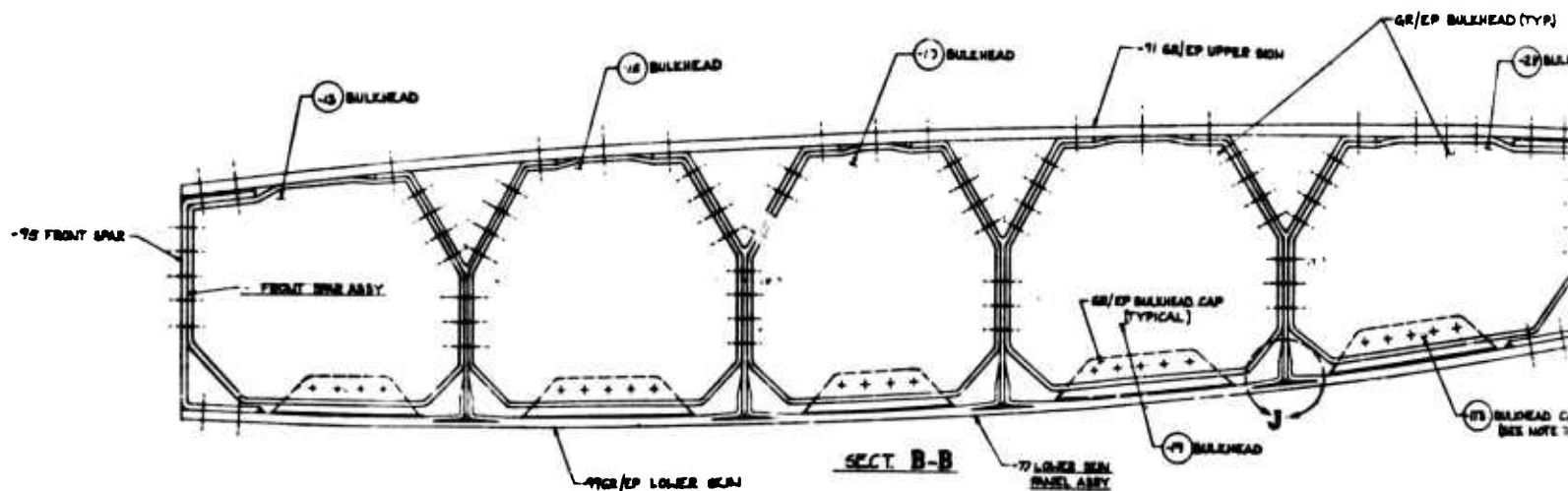
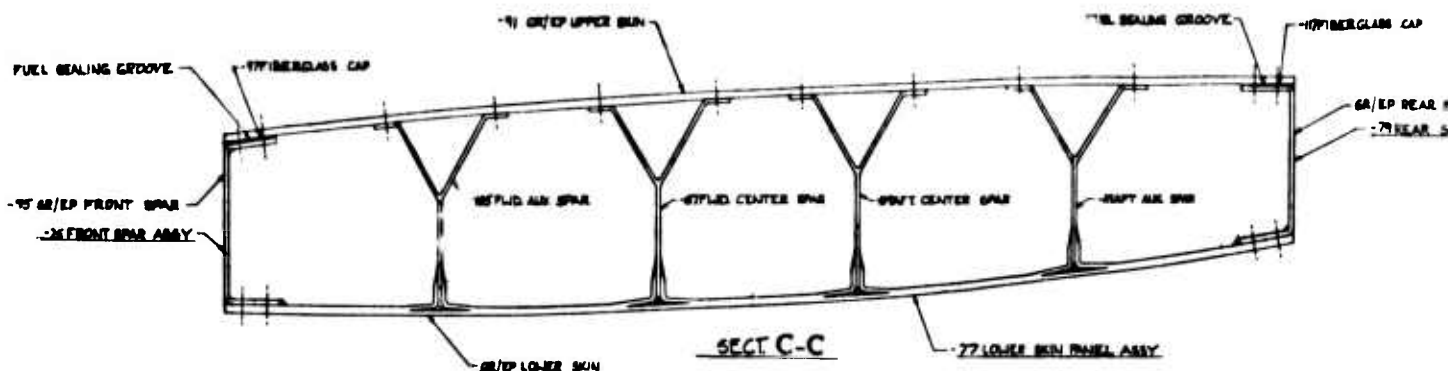
PRELIMINARY DESIGN DRAWING

ATW-4 WING BOX - LAMINATED ALUMINUM LOWER SKIN, CANTED SPARS

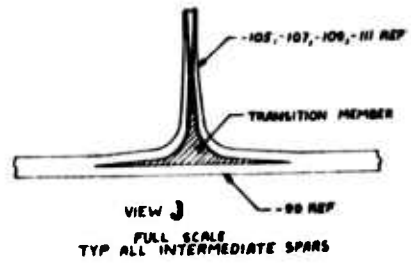
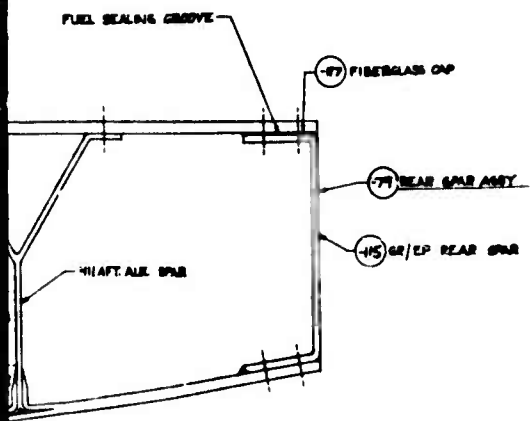
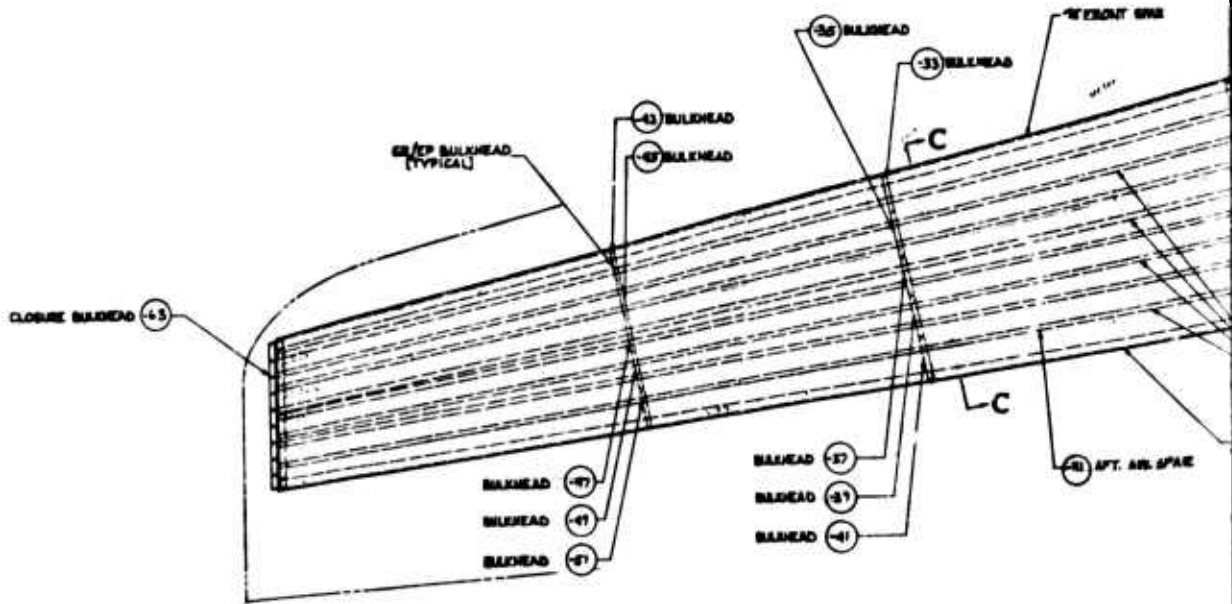
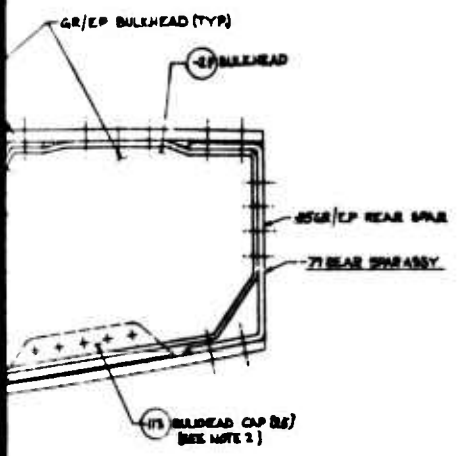
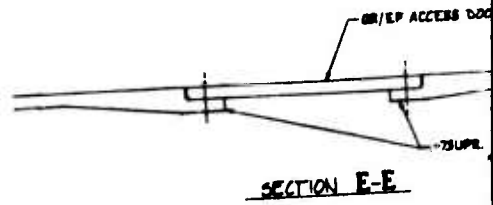
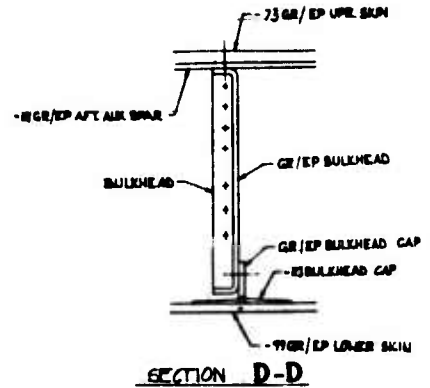
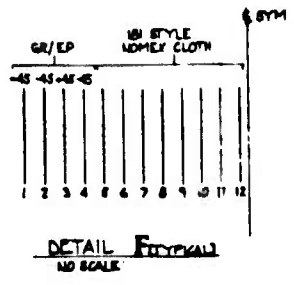
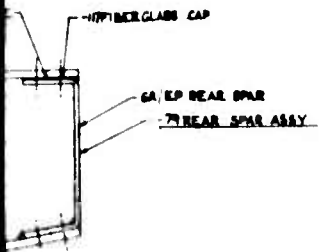
GENERAL DYNAMICS
General Aerospace Division

633-RW002

Figure 49 Slanted Spar Preliminary Design



2



ASSY	COMP NO SUB ASSY	DETAIL	PART NAME	NO REQ'D	DESCRIPTION	MATERIAL	RAW STR. SIZE	RAW STR. WT	FINISHED WT IN LBS	COST \$
-1			GR/EP WING BOX						2247.15	
		-7	FRONT SPAR (P/W)	1	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.04 IN ³	144.77	14.76	
		-9	REAR SPAR (P/W)	1	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.43 IN ³	228.00	28.40	
		-11	ELONG BND. (P/W)	1	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.67 IN ³	267.96	26.18	
		-13	BULKHEAD	25	GR/EP BND SEG	T300-S208 GR/EP S'PRE PREG TAPE	27788 FT ² OF 5" TAPE	89.39	28.40	
		-63	BULKHEAD CLOSURE BND	1	AL CASTING	ASS-76	43.3 IN ³	4.33	3.63	
		-64	S-E/E2 FASTENING	2	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.20 IN DIA BND	408.84	24.45	
		-65	DRIFT FASTENING	2	MACHINED FORG	17.4 PH	1.74 IN DIA BND	8.00	3.30	
		-69	RETAINING NUT	2	MACHINED FORG	17.4 PH	25.74 IN DIA	32.70	14.40	
		-71	UPPER SKIN PANEL ASSY	1	GR/EP WING SKIN	T300-S208 GR/EP S'PRE PREG TAPE	46.070 FT ² OF TAPE	64.68	481.80	
		-93	TI DOUBLER	2	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.			257.08	
		-95	FRONT SPAR ASSY		GR/EP FRONT SPAR	T300-S208 GR/EP S'PRE PREG TAPE	1000 FT ² OF TAPE	83.04	29.86	
		-97	CAP	1	FIBERGLASS CAP	FHS-923 CLASS 1			4.18	
		-99	LOWER SKIN PANEL ASSY	1	GR/EP WING SKIN	T300-S208 GR/EP S'PRE PREG TAPE	41.931 FT ² OF TAPE	62.1	448.60	
		-101	TI DOUBLER	1	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.00 IN DIA BND	384	24.41	
		-103	TI DOUBLER	1	MACHINED TI	TI 6AL-4V BETA ANNEALED COND.	1.00 IN DIA BND	378	24.22	
		-105	FRONT ALY SPAR	1	GR/EP INTMED	T300-S208 GR/EP S'PRE PREG TAPE	6004 FT ² OF 5" TAPE	87	28.80	
		-107	FRONT CENTER SPAR	1	WOMEX CLOTH	2128277		188	193.80	
		-109	AFT CENTER SPAR	1	WOMEX CLOTH	2128277		188	193.80	
		-111	AFT ALY SPAR	1	GR/EP INTMED	T300-S208 GR/EP S'PRE PREG TAPE	6004 FT ² OF 5" TAPE	87	28.80	
		-113	BULKHEAD CAP	25	GR/EP BND CAP	T300-S208 GR/EP S'PRE PREG TAPE	1961 FT ² OF TAPE	1.78	1.55	
		-115	REAR SPAR ASSY		GR/EP REAR SPAR	T300-S208 GR/EP S'PRE PREG TAPE	5000 FT ² OF TAPE	72.70	26.42	
		-117	CAP	1	FIBERGLASS CAP	FHS-923 CLASS 1			4.80	
		-119	ADHESIVE	1					5.30	
		-121	DRIFT FASTENING	2					4.40	
		-123	DRIFT FASTENING	2					4.40	
		-125	BLIND FASTENERS	4200					58.80	
		-127	AVG PASTERNER	182					1.82	
		-129	AVG PASTERNER	152					1.44	
		-131	AVG PASTERNER	1					4.7	
		-133	AVG PASTERNER	1					4.7	
		-135	AVG PASTERNER	1					4.7	
		-137	AVG PASTERNER	1					4.7	
		-139	AVG PASTERNER	1					4.7	
		-141	AVG PASTERNER	1					4.7	
		-143	AVG PASTERNER	1					4.7	
		-145	AVG PASTERNER	1					4.7	
		-147	AVG PASTERNER	1					4.7	
		-149	AVG PASTERNER	1					4.7	
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		-155	AVG PASTERNER	1					4.7	
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		-163	AVG PASTERNER	1					4.7	
		-165	AVG PASTERNER	1					4.7	
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		-357	AVG PASTERNER	1					4.7	
		-359	AVG PASTERNER	1					4.7	
		-361	AVG PASTERNER	1					4.7	
		-								

TABLE VI ATW WING BOX PRELIMINARY DESIGN EVALUATION SUMMARY
(1975 DOLLARS)

DRAWING NUMBER	DESCRIPTION	STRUCT. EFFIC.		TECHNOLOGY IMPROVE.		DAMAGE TOL.		ABILITIES				TOTAL SCORE (1.0)	RANK
		COST* 1975 \$ (.24)	WEIGHT* LBS. (.16)	CONCEPT (.05)	MATL. (.02)	SAFE CRACK (.08)	FAIL SAFE (.12)	INSPECT- (.06)	MFG- (.06)	MAINT- (.06)	REPAIR- (.06)		
633R4000	BASELINE - ATW-4 MACHINED AL., BOLTED STRUCT.	200K .209	2759.0 .116	.000	.000	.078	.000	.060	.05	.058	.055	.681	4
633R4001	AL. LAM. LWR. SKIN "Y" SPARS, EXPOSED LWR CAPS - CLSD PIVOT LUG AREA	174K .240	2341.0 .137	.042	.024	.080	.120	.055	.052	.057	.055	.927	1
633R4002 -1	AL. LAM. LWR. SKIN, SLANT SPARS, EMBEDDED LWR CAPS - CLSD PIVOT LUG AREA	178K .235	2421.0 .133	.042	.024	.080	.000	.045	.052	.058	.055	.784	2
633R4002 -3	AL. LAM. LWR. SKIN, SLANT SPARS, EMBEDDED LWR CAPS, OPEN PIVOT LUG AREA	185K .227	2608.0 .123	.042	.028	.080	.000	.050	.055	.058	.059	.782	3
633R4003 -1	GRAPHITE/EPOXY - "Y" SPAR, OPEN PIVOT LUG AREA	325K .000	2247.8 .143	.050	.03	.080	.000	.06	.06	.06	.06	.623	5
633R4003 -3	GRAPHITE/EPOXY - "Y" SPARS - CLOSED PIVOT LUG AREA	315K .000	2006.0 .160	.050	.028	.080	.000	.05	.052	.058	.055	.613	6

* COSTS AND WEIGHTS SHOWN ARE FOR ONE SIDE AND INCLUDE THE WING BOX STRUCTURE FROM THE TIP BHD TO AND INCLUDING THE PIVOT FITTING

TABLE VII ATW WING BOX PRELIMINARY DESIGN EVALUATION SUMMARY
(1980 DOLLARS)

DRAWING NUMBER	DESCRIPTION	STRUCT. EFFIC.		TECHNOLOGY IMPROVE.		DAMAGE TOL.		ABILITIES			TOTAL SCORE (1.0)	RANK	
		COST* 1980 \$ (.24)	WEIGHT* LBS. (.16)	CONCEPT (.05)	MATL. (.02)	MFG. (.03)	SAFE CRACK (.08)	FAIL SAFE (.12)	INSPECT- (.06)	MFG- (.06)			MAINT- (.06)
633RW000	BASELINE - ATW-4 MACHINED AL., BOLTED STRUCT.	252K	2759.0										
		.211	.116	.000	.000	.078	.000	.060	.05	.058	.055	.683	4
633RW001	AL. LAM. LWR. SKIN "Y" SPARS, EXPOSED LWR CAPS - CLSD PIVOT LUG AREA	221K	2341.0										
		.240	.137	.042	.01	.080	.120	.055	.052	.057	.055	.927	1
633RW002	AL. LAM. LWR. SKIN, SLANT SPARS, EMBEDDED LWR CAPS - CLSD PIVOT LUG AREA	225K	2421.0										
-1		.235	.133	.042	.01	.080	.000	.045	.052	.058	.055	.784	2
633RW002	AL. LAM. LWR. SKIN, SLANT SPARS, EMBEDDED LWR CAPS, OPEN PIVOT LUG AREA	234K	2608.0										
-3		.227	.123	.042	.01	.080	.000	.050	.055	.058	.059	.782	3
633RW003	GRAPHITE/EPOXY - "Y" SPAR, OPEN PIVOT LUG AREA	341K	2247.8										
-1		.000	.143	.050	.02	.080	.000	.06	.06	.06	.06	.623	5
633RW003	GRAPHITE/EPOXY - "Y" SPARS - CLOSED PIVOT LUG AREA	325K	2006.0										
-3		.000	.160	.050	.02	.080	.000	.05	.052	.058	.055	.613	6

* COSTS AND WEIGHTS SHOWN ARE FOR ONE SIDE AND INCLUDE THE WING BOX STRUCTURE FROM THE TIP BHD TO AND INCLUDING THE PIVOT FITTING

SECTION IV

STRESS ANALYSIS

A large number of structural concepts was considered for application to the ATW-4 wing box. The majority of these had been considered in depth during Contracts F33615-72-C-2149 and F33615-74-C-3026. During those contracts the concepts were evaluated using a merit rating system which was acceptable to the AFFDL. It enabled refinement of very large numbers of ideas into a manageable few which are outstanding for minimizing weight and cost while meeting the strength and durability requirements of MIL-A-83444, MIL-A-8866, and MIL-STD-1530.

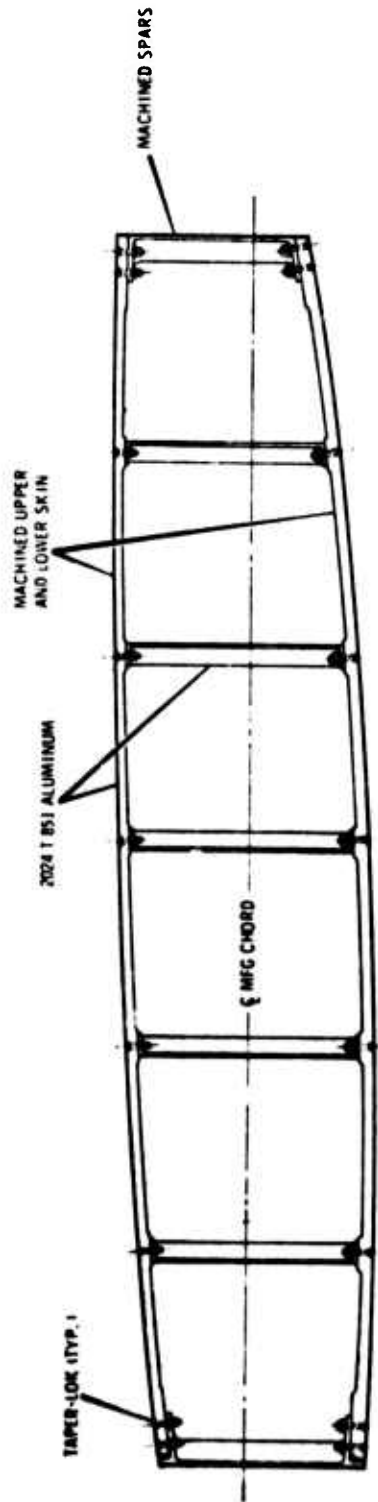
4.1 BASELINE STRENGTH CONSIDERATIONS

The current, ATW-4, baseline is similar in aspect ratio and root attachment to the baselines for the two previous contracts. For this reason, it has been possible to select directly the more promising arrangements and evaluate them for compliance with the current merit rating system.

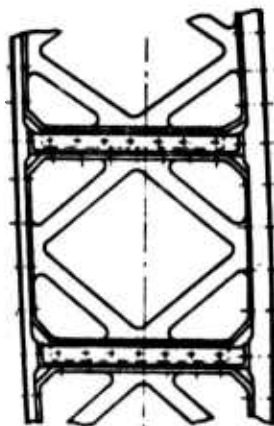
The baseline for this study is a wing with supercritical airfoil. Its aspect ratio is similar to the F-111 wing but has a greater span. The baseline then is a wing that resembled an existing wing but one which had to be defined for this program (see Figure 51). Construction of the baseline was chosen to be the same as the F-111 except that two additional spars were required to accommodate the increased chord while maintaining skin stress levels similar to those on the F-111. The baseline meets the requirements of MIL-A-83444, MIL-A-8866, and MIL-STD-1530.

4.2 DESIGN LOADS

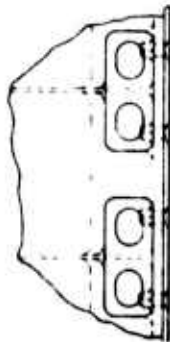
Preliminary external loadings were used to make an initial sizing of the baseline wing box for purposes of obtaining inertia data, calculating preliminary stiffness data and constructing a finite element model. These data were utilized for the preparation of the "Design Loads Data for the Variable Camber Supercritical Wing Program," FZM-12-6466 (see Appendix A).



SECTION A-A



SECTION B-B



SECTION C-C

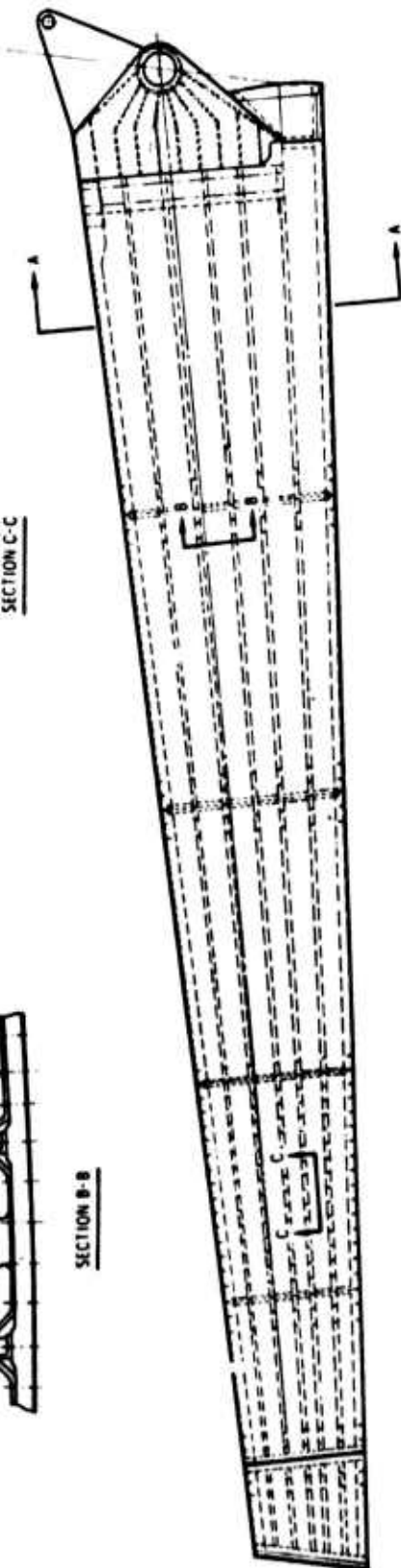


Figure 51 ATW-4 Wing Box Baseline

External loads from FZM-12-6466 were then applied to the finite element model to obtain internal loads for sizing Analytical Assemblies and Preliminary Designs of full span wing boxes. Samples of these internal loads are illustrated in Figure 52 thru Figure 54.

Fuel pressure loadings were computed using the roll rates of the F-111 with the load factors of this study and 6.0 psi constant system pressure. The resulting pressures are shown in Figure 55.

4.3 STRESS ANALYSIS AT THE ANALYTICAL ASSEMBLY LEVEL

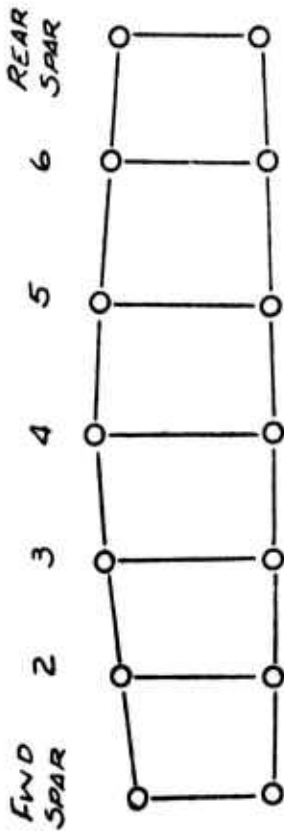
The first evaluations for this program were made at the level of "Analytical Assemblies", as first used in Contract F33615-72-C-2149: These being 48.0 inch span length boxes of full wing-box chord and of constant cross section, sized in complete detail.

All sizing was made using internal loads data obtained from the baseline finite element model and fatigue and damage tolerance designs allowables shown in Figures 56 thru 64. These design allowables were converted to equivalent static ultimate design stresses by multiplying them by the ratio of the maximum static ultimate root bending moment to the maximum root moment in the cyclic loading spectrum, but limited by the static tension strength of the material. These data are summarized in Table VIII for the Analytical Assemblies at span station 140.

Analytical Assemblies at span station 140 were given "Fail Safe" and "Safe Crack" ratings which are illustrated in Tables IX and X. The rating procedures are the same as those used during Contracts F33615-72-C-2149 and F33615-74-C-3026: Fail safe scores were given to concepts which are capable of resisting at least 115% of limit load with one major load path, such as a spar cap, failed; and the concepts with more elements at a spar cap were given higher scores. Safe crack scores were awarded proportionally to the safety factor between fracture design allowable stress and maximum static design tension stress.

The graphite epoxy Analytical Assemblies were rated with the maximum score of eight percent for Safe Crack.

SPAR CAP LOADS-FWD	CONDITION NO.			
	1	2	3	4
UPPER (KIPS)				
2	-13	-11	-12	-12
3	-9	-8	-9	-9
4	-9	-8	-9	-9
5	-9	-8	-9	-9
6	-8	-7	-8	-8
REAR	-7	-6	-7	-8
LOWER (KIPS)				
FWD	-9	-8	-9	-10
2	6	5	6	6
3	5	4	5	5
4	5	4	5	5
5	5	4	5	5
6	4	4	4	4
REAR	4	3	4	4
SPAR WEB (KIPS/IN)				
-FWD	1.7	2.0	0.6	0
2	1.3	1.3	0.8	0.6
3	1.3	1.1	1.0	0.9
4	1.1	0.8	1.0	1.0
5	1.0	0.5	1.0	1.1
6	0.8	0.2	1.0	1.3
REAR	1.0	0	1.6	2.2
UPPER SKIN (KIPS/IN)				
1-2	-21	-19	-20	-21
2-3	-22	-20	-21	-22
3-4	-22	-19	-21	-22
4-5	-21	-18	-21	-21
5-6	-19	-17	-19	-19
6-7	-17	-15	-17	-17
LOWER SKIN (KIPS/IN)				
1-2	20	18	20	20
2-3	22	20	22	22
3-4	23	21	23	23
4-5	22	20	22	23
5-6	20	18	20	21
6-7	16	14	16	16

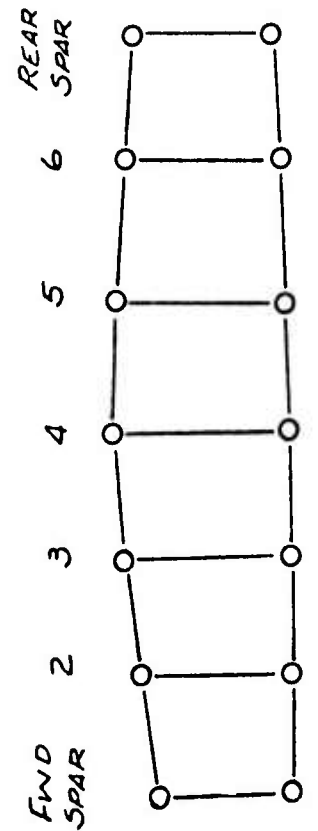


LOADS FROM MATH MODEL:

206323-H 18 July 75

Figure 52 Math Model - Internal Loads
@ S. S. 140

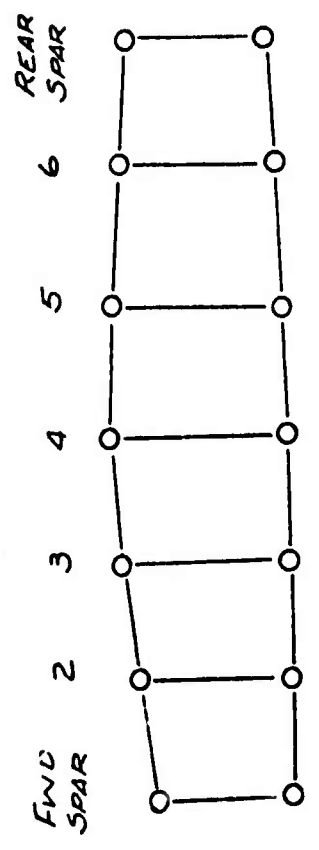
SPAR CAP LDS (KIPS)	UPPER	LOWER	SPAR WEB (KIPS/IN)	UPPER SKIN (KIPS/IN)	LOWER SKIN (KIPS/IN)	CONDITION	1	2	3	4	5	6
- FWD	2	2	- FWD	1-2	1-2	2	-	-	-	-	-	-
3	3	3	2	2-3	2-3	3	-	-	-	-	-	-
4	4	4	3	3-4	3-4	4	-	-	-	-	-	-
5	5	5	4	4-5	4-5	5	-	-	-	-	-	-
6	6	6	5	5-6	5-6	6	-	-	-	-	-	-
REAR	REAR	REAR	6	6-7	6-7	REAR	-	-	-	-	-	-
FWD	FWD	FWD	REAR	1-2	1-2	2	2	2	2	2	2	2
2	2	2	2	2-3	2-3	3	2	2	2	2	2	2
3	3	3	3	3-4	3-4	4	2	2	2	2	2	2
4	4	4	4	4-5	4-5	5	2	2	2	2	2	2
5	5	5	5	5-6	5-6	6	2	2	2	2	2	2
6	6	6	6	6-7	6-7	REAR	2	2	2	2	2	2
REAR	REAR	REAR	REAR	1-2	1-2	2	3	3	3	3	3	3
FWD	FWD	FWD	- FWD	2-3	2-3	3	0.3	0.9	0.1	0	0	0
2	2	2	2	3-4	3-4	4	0.3	0.5	0.2	0.3	0.3	0.3
3	3	3	3	4-5	4-5	5	0.4	0.5	0.3	0.4	0.4	0.4
4	4	4	4	5-6	5-6	6	0.4	0.4	0.4	0.4	0.4	0.5
5	5	5	5	6-7	6-7	REAR	0.5	0.4	0.4	0.5	0.7	0.7
6	6	6	6	1-2	1-2	2	0.5	0.3	0.6	0.9	0.9	0.9
REAR	REAR	REAR	REAR	2-3	2-3	3	0.8	0.3	1.0	1.7	1.7	1.7
1-2	1-2	1-2	1-2	3-4	3-4	4	-	-	-	-	-	-
2-3	2-3	2-3	2-3	4-5	4-5	5	-	-	-	-	-	-
3-4	3-4	3-4	3-4	5-6	5-6	6	-	-	-	-	-	-
4-5	4-5	4-5	4-5	6-7	6-7	REAR	-	-	-	-	-	-
5-6	5-6	5-6	5-6	1-2	1-2	2	4	5	5	5	5	5
6-7	6-7	6-7	6-7	2-3	2-3	3	4	5	5	5	5	5
1-2	1-2	1-2	1-2	3-4	3-4	4	5	6	6	6	6	6
2-3	2-3	2-3	2-3	4-5	4-5	5	5	6	6	6	6	6
3-4	3-4	3-4	3-4	5-6	5-6	6	5	6	6	6	6	6
4-5	4-5	4-5	4-5	6-7	6-7	REAR	5	6	6	6	6	6
5-6	5-6	5-6	5-6	1-2	1-2	2	5	6	6	6	6	6
6-7	6-7	6-7	6-7	2-3	2-3	3	4	5	5	5	5	5



LOADS FROM MATH MODEL:
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Figure 53 Math Model - Internal Loads
@ S.S. 350

SPAR CAP LDS (KIPS)	CONDITION NO.			
	1	2	3	4
UPPER				
FWD	- 2	- 2	- 2	- 2
2	- 1	- 1	- 1	- 1
3	- 1	- 1	- 1	- 1
4	- 1	- 1	- 1	- 1
5	- 1	- 1	- 1	- 1
6	0	0	0	0
REAR	0	0	0	0
LOWER				
FWD	0	0	0	0
2	0	0	0	0
3	0	0	0	0
4	1	1	1	1
5	1	1	1	1
6	1	1	1	1
REAR	1	1	1	1
SPAR WEB (KIPS/IN)				
FWD	0	0.3	0	0.1
2	0.1	0.2	0	0
3	0.1	0.2	0.1	0.1
4	0.2	0.1	0.1	0.1
5	0.2	0.1	0.1	0.2
6	0.2	0.1	0.2	0.3
REAR	0.3	0	0.3	0.4
UPPER SKIN (KIPS/IN)				
1-2	- 1.0	- 1.0	- 1.1	- 1.1
2-3	- 0.9	- 0.8	- 1.0	- 0.9
3-4	- 0.8	- 0.8	- 0.8	- 0.8
4-5	- 0.6	- 0.6	- 0.7	- 0.6
5-6	- 0.5	- 0.5	- 0.5	- 0.4
6-7	- 0.3	- 0.3	- 0.4	- 0.2
LOWER SKIN (KIPS/IN)				
1-2	.3	.2	.3	.3
2-3	.5	.5	.6	.5
3-4	.8	.7	.8	.7
4-5	.9	.9	.9	.8
5-6	1.0	1.0	1.0	.8
6-7	1.0	1.0	1.0	.9



LOADS FROM MATH MODEL:

206323-H 18 July 75

Figure 54 Math Model - Internal Loads
@ S.S. 450

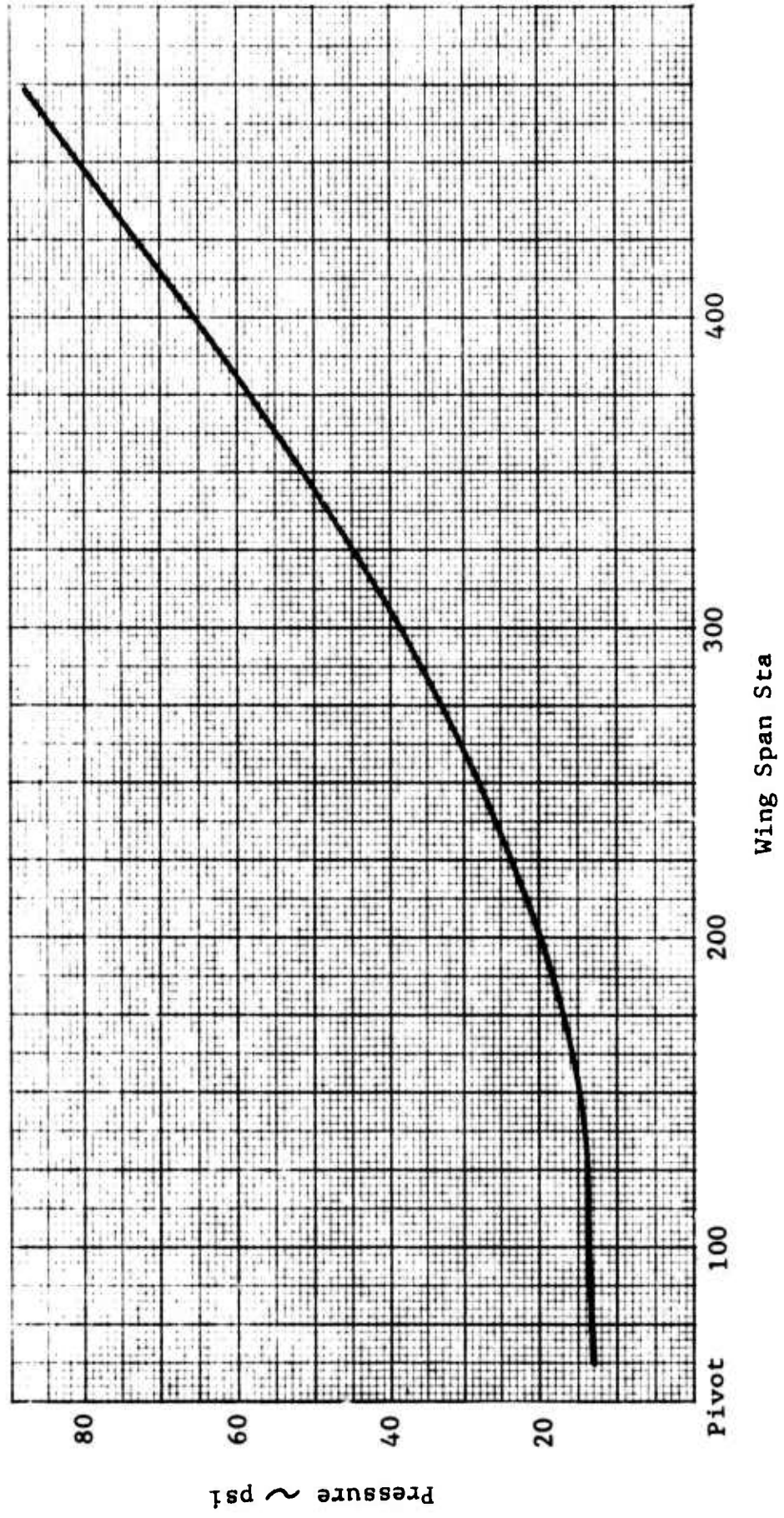


Figure 55 Ultimate Wing Fuel Pressures Due to 150°/Sec Roll Rate Plus 5.5 psi System Pressure (Wing Forward)

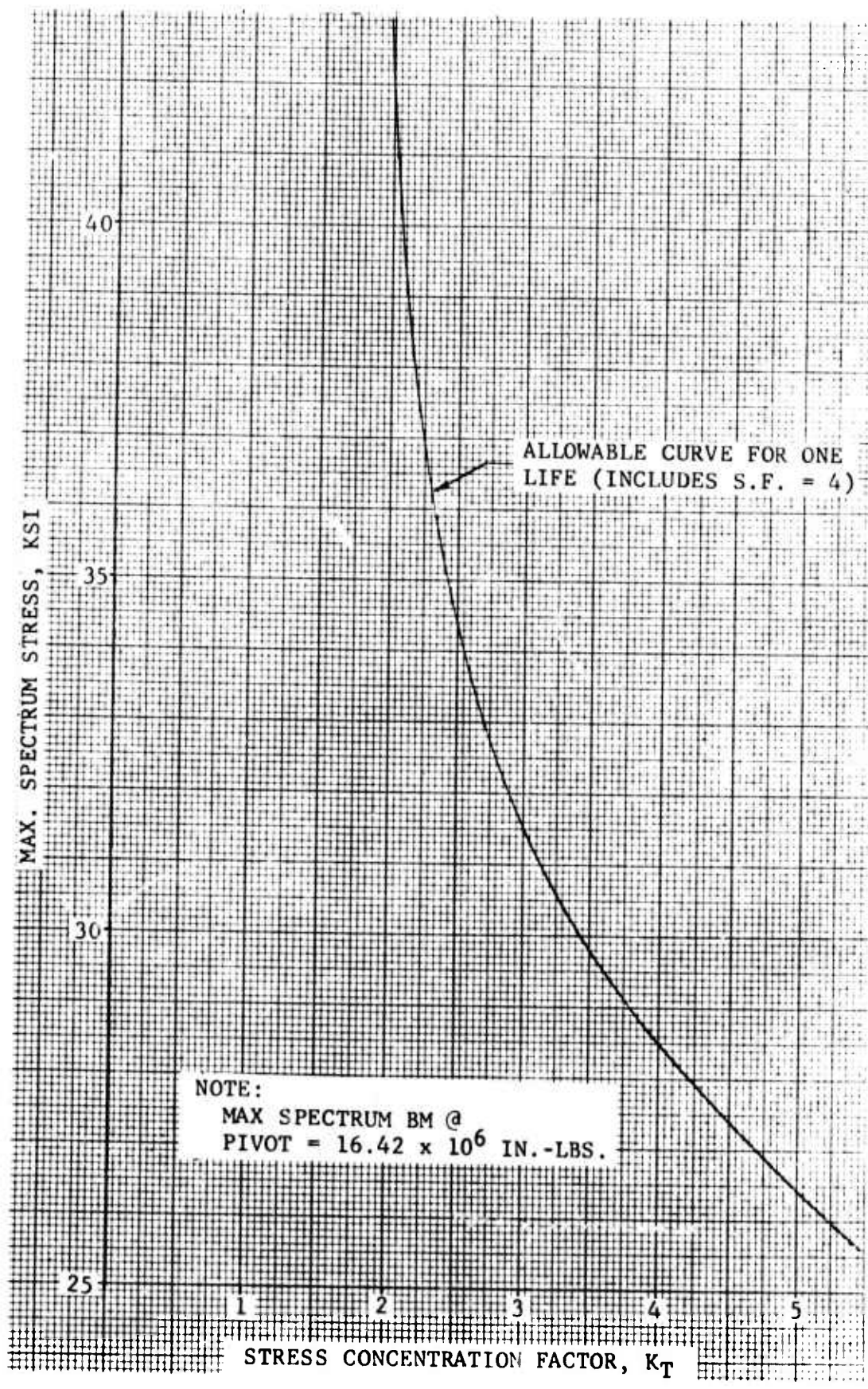
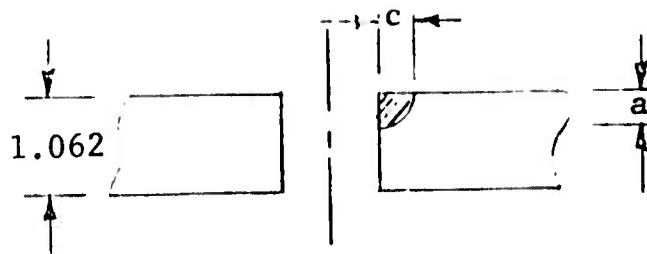


FIGURE 56 Fatigue Design Allowable Curve - Wing Box Lower Surface (2024-T851 Aluminum)



$$a/2c = 0.5$$

NOTE: MAX. SPECTRUM BM @ PIVOT = 16.42×10^6 IN.-LBS.

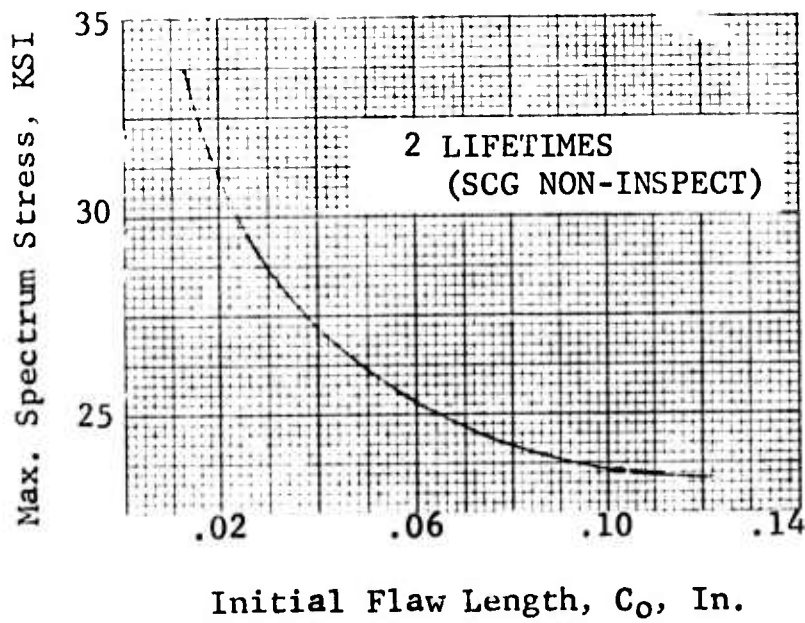


FIGURE 57 Fracture Design Allowable Curve - Wing Box Lower Surface (2024-T851 Aluminum Corner Flaw)

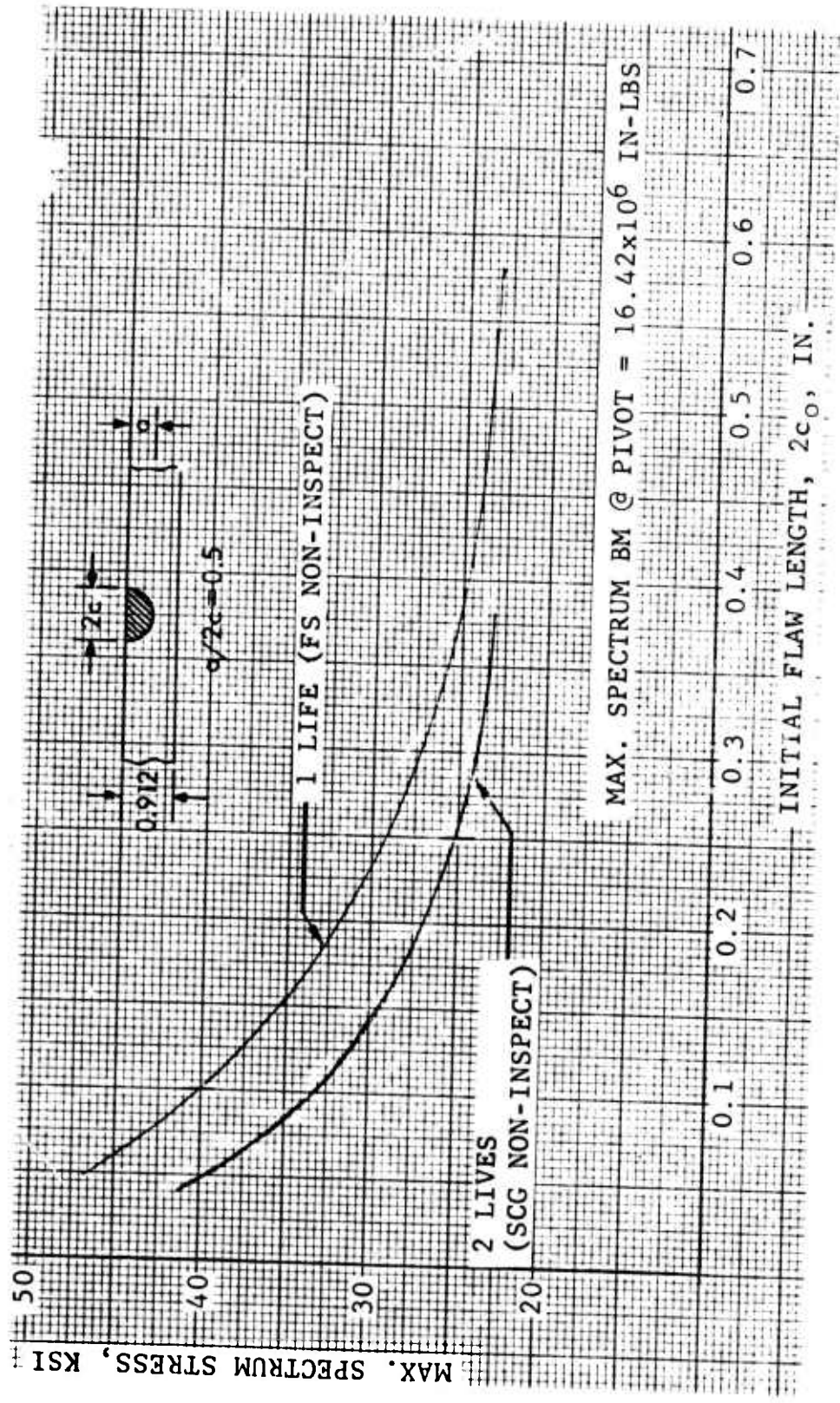


FIGURE 58 Fracture Design Allowable Curve - Wing Box Lwr Surface (2024-T851 Aluminum Surface Flaw)

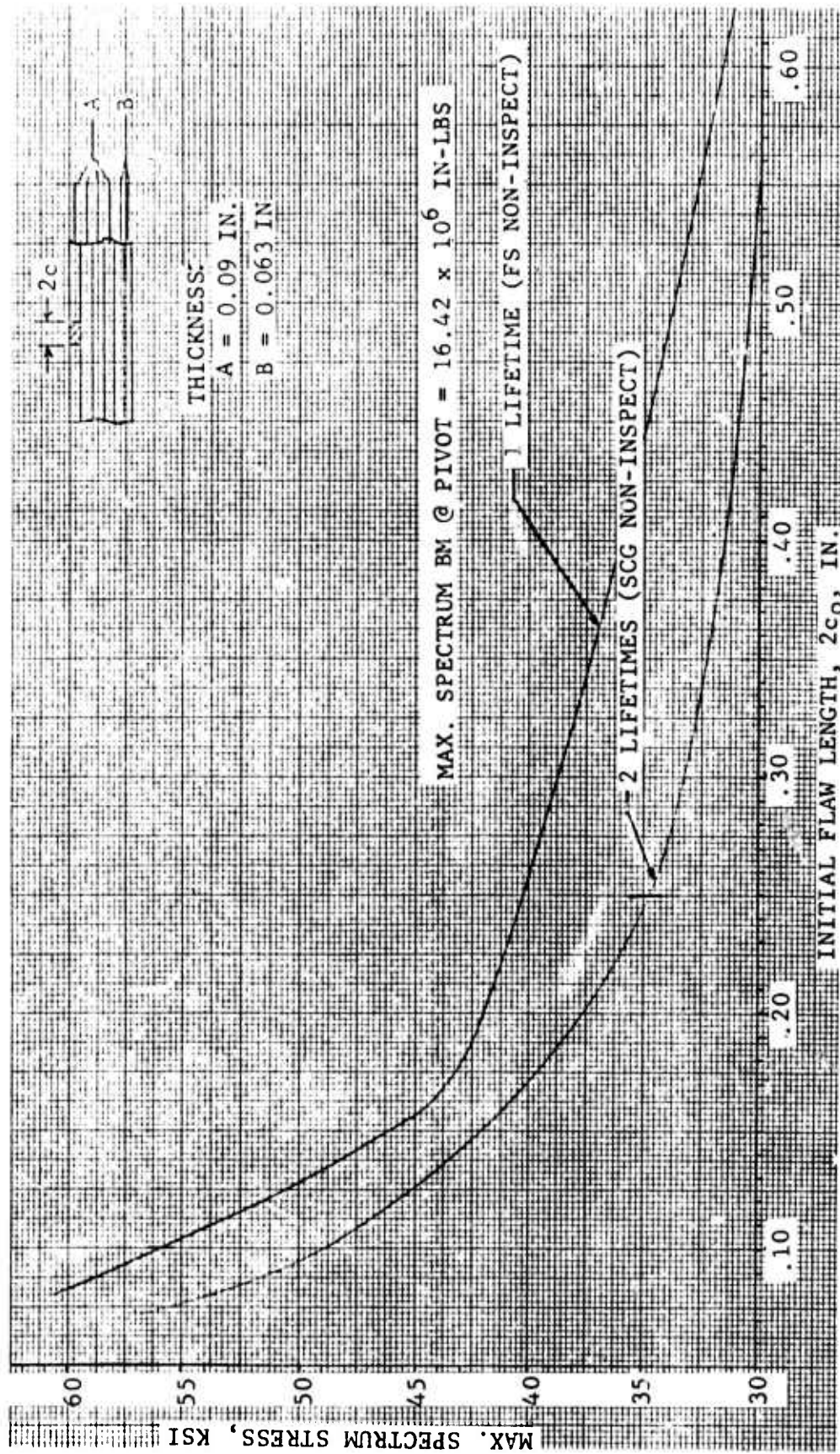


FIGURE 59 Fracture Design Allowable Curve - Wing Box Lwr Surface (Laminated 2024-T81 Aluminum Surface Flow)

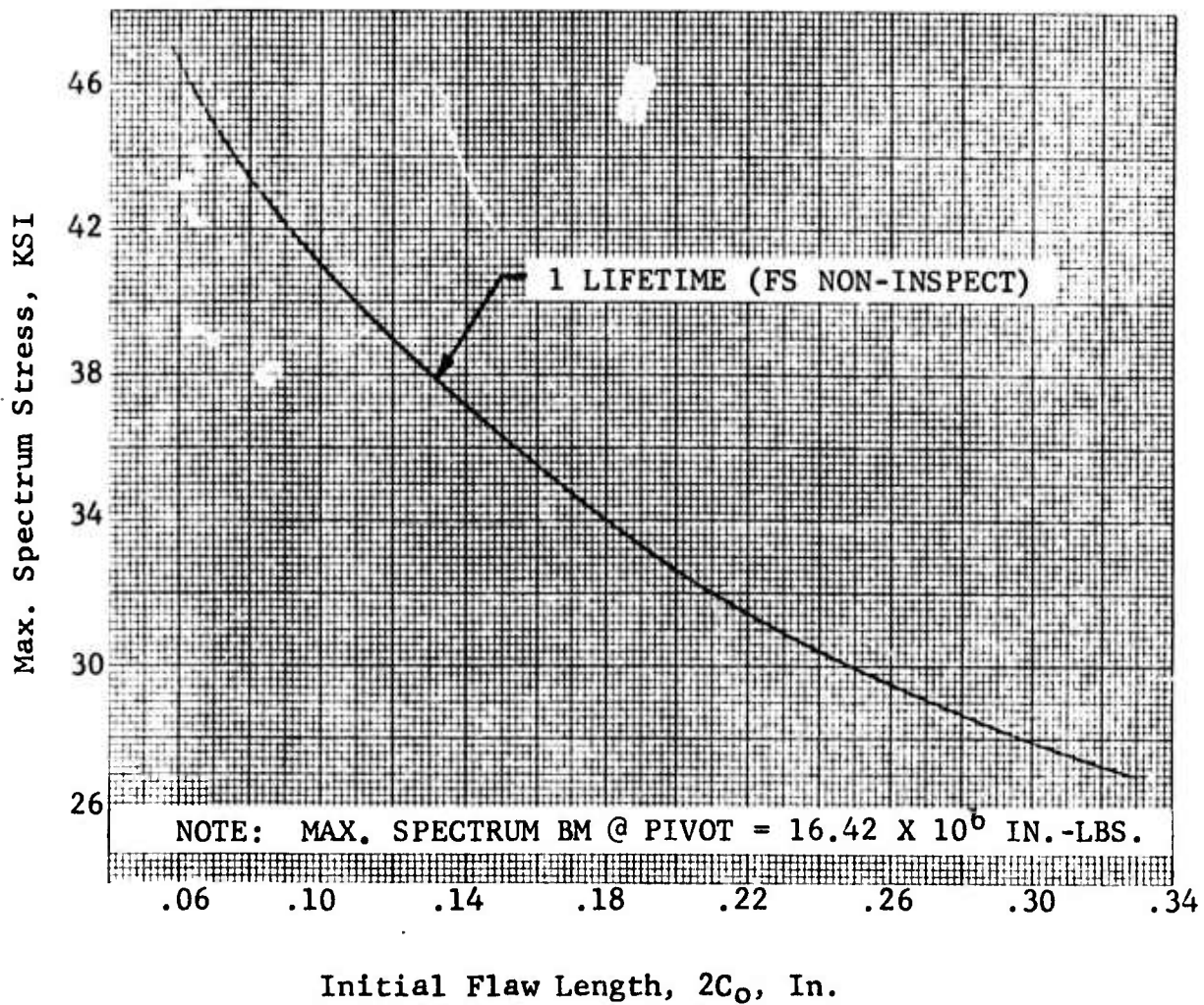
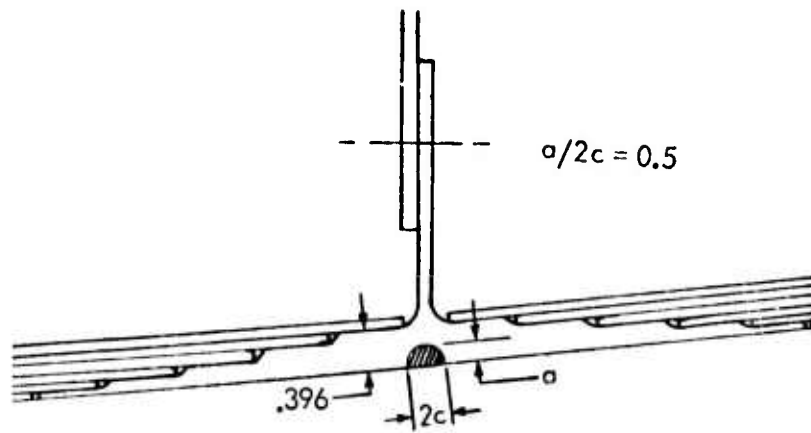
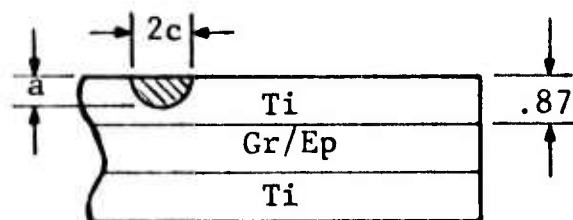


FIGURE 60 Fracture Design Allowable Curve - Wing Box
Lwr Spar Cap (2024-T8511 Aluminum Surface Flaw)

$$a/2c = 0.5$$



NOTE: Max Spectrum BM @ Pivot = 16.42×10^6 In-Lbs

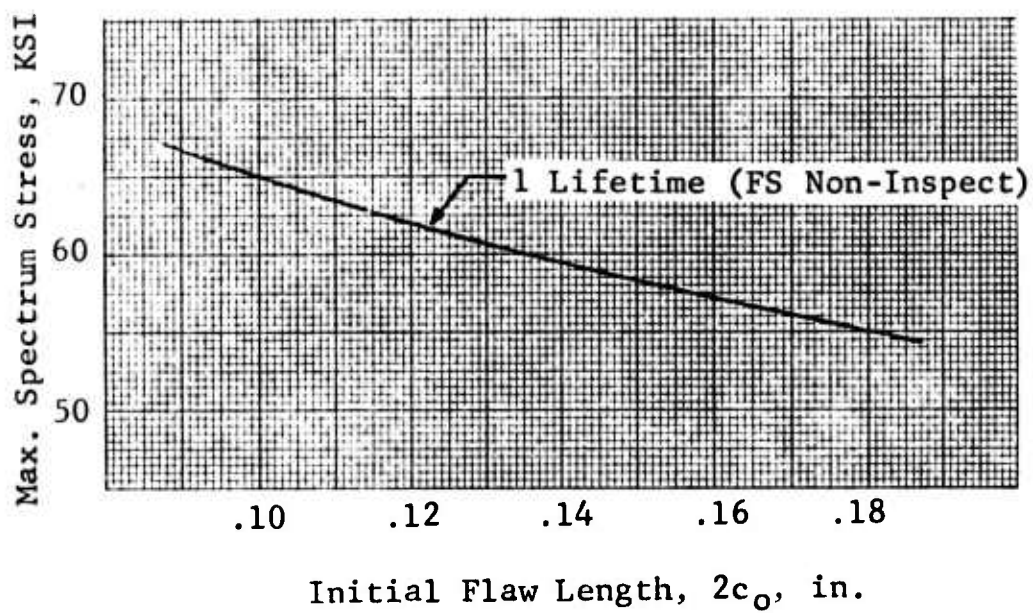
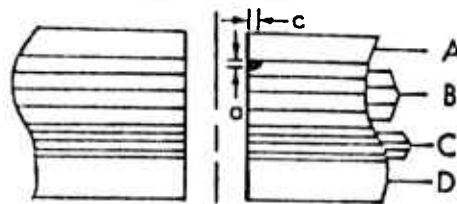


FIGURE 61 Fracture Design Allowable Curve - Wing Pivot Fitting (Ti-6AL-4V Surface Flaw)

TAPER-LOK BOLT HOLE
 $a/2c = 0.5$



THICKNESS	→ 1.0 ←	MATERIAL
A=0.2 IN		2024-T8511 AL
B=0.09 IN		2024-T81 AL
C=0.063 IN		2024-T81 AL
D=0.31 IN		10 NICKEL STEEL

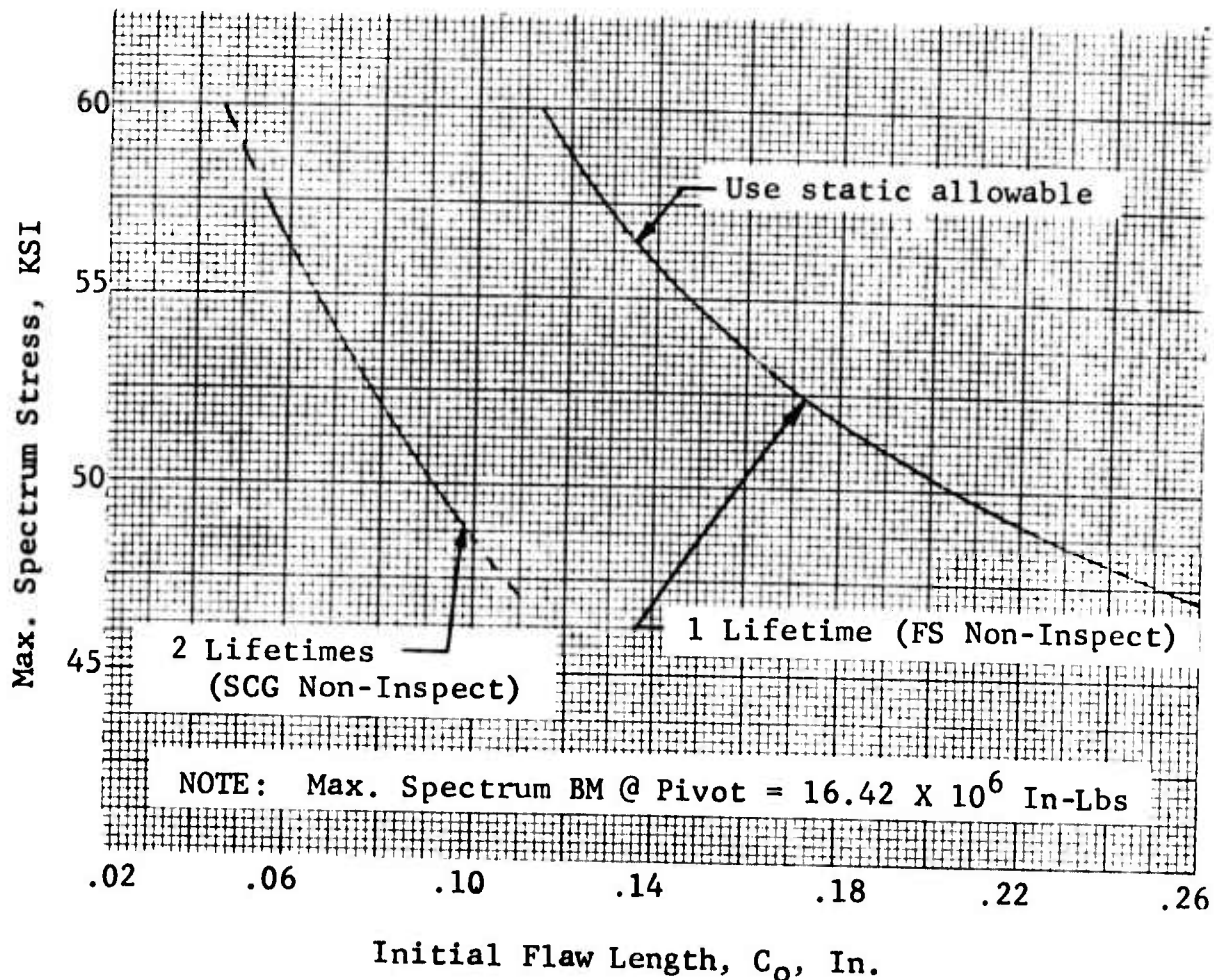
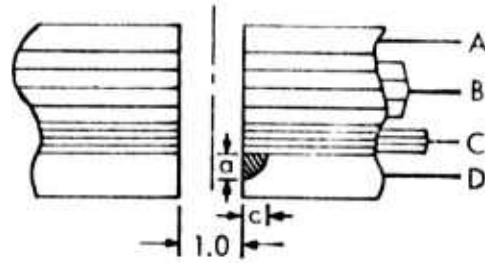


FIGURE 62 Fracture Design Allowable Curve - Wing Pivot Fitting (Laminated 2024-T81 Aluminum Corner Flaw)

$$a/2c = 0.5$$



THICKNESS	MATERIAL
A=0.2 IN	2024-T8511 AL
B=0.09 IN	2024-T81 AL
C=0.063 IN	2024-T81 AL
D=0.31 IN	10 NICKEL STEEL

TAPER-LOK BOLT HOLE

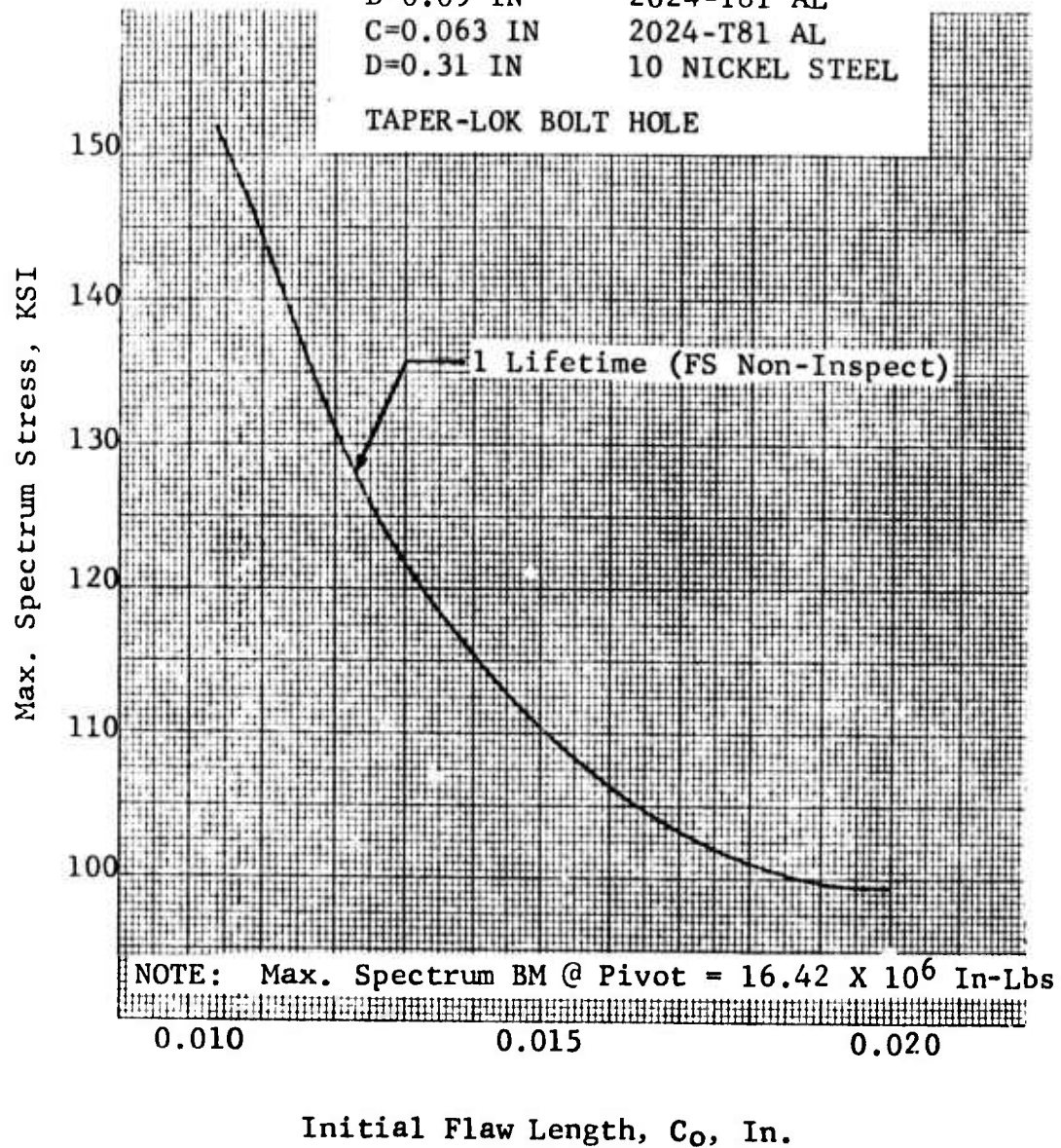


Figure 63 Fracture Design Allowable Curve - Wing Pivot Fitting (10 Nickel Steel Corner Flaw)

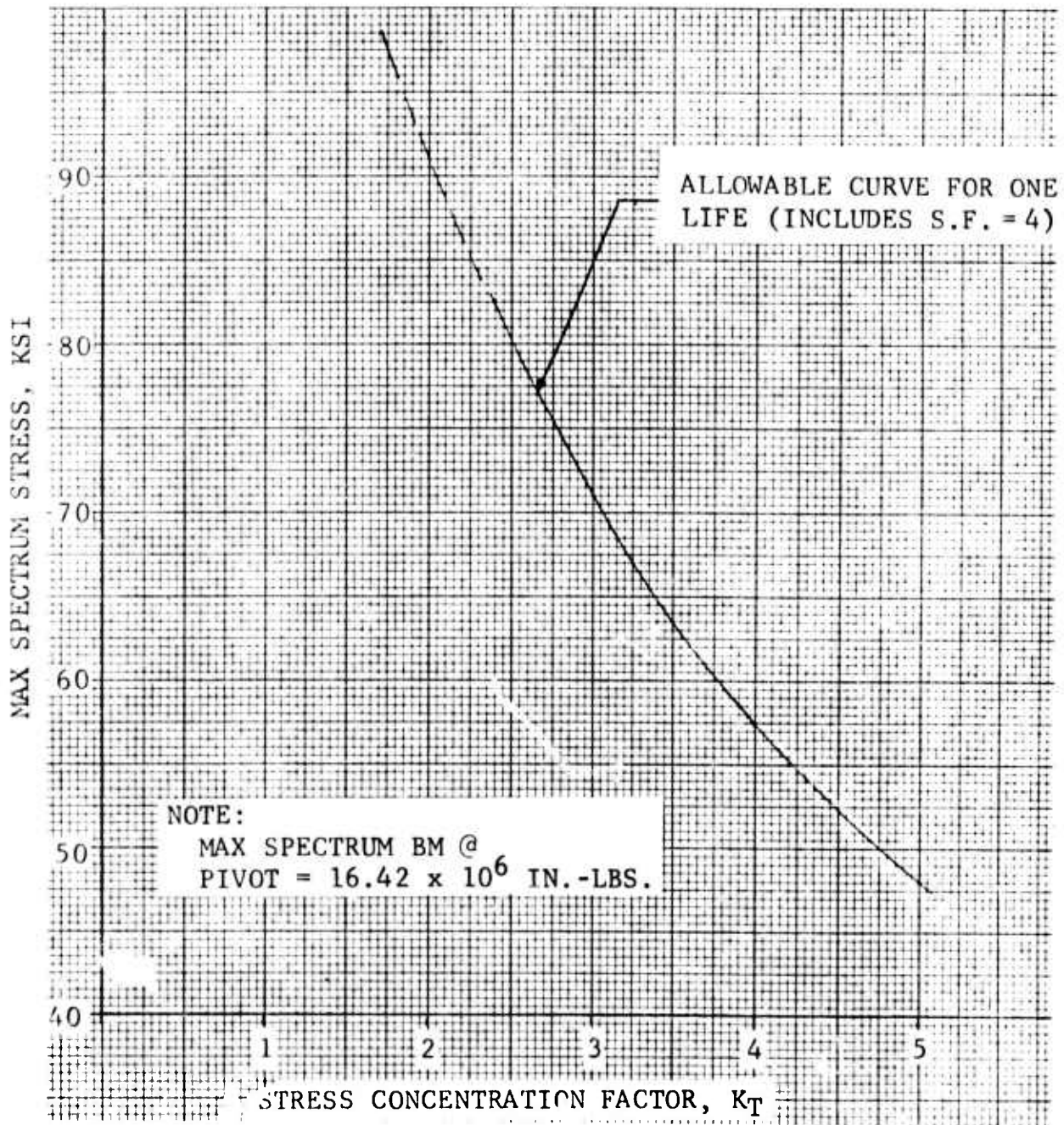


FIGURE 64 Fatigue Design Allowable Curve - Wing Box Lower Surface (6AL-4V Beta Annealed Titanium)

TABLE VIII
"ANALYTICAL ASSEMBLY" DESIGN STRESS DATA

	Lower Skin			Upper Skin			Comments		
	Material	Class	Max. Kt	S Fatigue*	S Dmg Tol.*	Material		Type	S Crit.
633RA-000-1	2024-T851	Safe Life	3.4	48600 psi	41500 psi	2024-T851	Plate	53500 psi	Baseline
633RA-001-1	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	59500	Planked & Laminated Lwr Skin/ Exposed Caps/Extruded "y" Intr Spars (3)
633RA-001-3,-5	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	59500	Planked & Laminated Lwr Skin/ Exposed Caps/Sandwich "y" Intr Spars (3)
633RA-001-801	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	59500	Planked & Laminated Lwr Skin/ Exposed Caps/Sheet "y" Intr Spars (3)
633RA-001-803	2024-T81	Safe Life	2.0	68360	57840	2024-T851	Plate	59500	Laminated Lwr Skin/Covered Caps/ Beaded "y" Intr Spars (3)
633RA-001-805	2024-T81	Safe Life	2.0	68360	57840	2024-T851	Plate	59500	Laminated Lwr Skin/Covered Caps/ Extruded "y" Intr Spars (3)
633RA-002-1	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	53700	Planked & Laminated Lwr Skin/ Exposed Caps/Corrugated Intr Spars (3)
633RA-003-1	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Sandwich	62000	Planked & Laminated Lwr Skin/ Exposed Caps/5 Intr Sandwich Spars
633RA-003-3	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	53500	Planked & Laminated Lwr Skin/ Exposed Caps/7 Intr Sandwich Spars
633RA-003-5	2024-T81	Safe Life	2.0	68360	57840	2024-T851	Plate	48400	Laminated Lwr Skin/Covered Caps/ 6 Intr Laminated Spars
633RA-003-801	2024-T81	Fail Safe	2.0	68360	66900	2024-T851	Plate	59500	Planked & Laminated Lwr Skin/ Exposed Caps/Canted Intr Spars (3)
633RA-003-801A	2024-T81	Safe Life	2.0	68360	57840	2024-T851	Plate	59500	Laminated Lwr Skin/Covered Caps/ Canted Intr Spars (3)
633RA-004-1	2024-T851	Safe Life	4.5**	45040	41500	2024-T851	Plate	59500	Plate Skins/Inverted "A" Spars
633RA-004-3	2024-T851	Safe Life	4.5**	45040	41500	2024-T851	Plate	59500	Plate Skins/Extruded "y" Spars
633RA-004-5	2024-T851	Safe Life	4.5**	45040	41500	2024-T851	Plate	59500	Plate Skins/Sheet "y" Spars

** With straight shank fasteners

* Equivalent static ultimate stress = $\frac{18.3(10^6)}{16.4(10^6)}$ (1.5) (Max allowable cyclic stress)

TABLE IX ANALYTICAL ASSEMBLY "SAFE CRACK" RATING

	$S_{cg} \frac{18.3(10^6)}{16.4(10^6)} (1.5)^*$	fx	fs	S_{max}	$\frac{S_{Safe Crack}}{S_{max}} = R$	Score: $\frac{.08R}{R_{max}}$
633RA000-1	41500	37459	5863	38355	1.082	.075
633RA001-1	66900	56930	8911	58292	1.148	.080
633RA001-3	66900	56930	8911	58292	1.148	.080
633RA001-5	66900	56930	8911	58292	1.148	.080
633RA001-801	66900	56930	8911	58292	1.148	.080
633RA001-803	57840	50218	7860	51419	1.125	.078
633RA001-805	57840	50218	7860	51419	1.125	.078
633RA002-1	66900	56930	8911	58292	1.148	.080
633RA003-1	66900	56930	8911	58292	1.148	.080
633RA003-3	66900	56930	8911	58292	1.148	.080
633RA003-5	51840	50218	7860	51419	1.125	.078
633RA003-801	66900	56930	8911	58292	1.148	.080
633RA004-1	41500	37459	5863	38355	1.082	.075
633RA004-3	41500	37459	5863	38355	1.082	.075
633RA004-5	41500	37459	5863	38355	1.082	.075
633RA005-1				38355	1.082	.075
633RA006-1						.080
633RA007-1						.080
633RA008-1						.080

= Safe Crack, Ult

* Maximum allowable spectrum stress for stable crack growth, times ratio of limit root bending to max. spectrum bending; all times 1.5 for equivalent ultimate

TABLE X
ANALYTICAL ASSEMBLY "FAIL SAFE" RATING

	<u>Minimum Number of Elements in Fail Safe Path</u>	<u>Score</u>
633RA000-1	0	0
633RA001-1	7	.12
633RA001-3	7	.12
633RA001-5	7	.12
633RA001-801	7	.12
633RA001-803	0	0
633RA001-805	0	0
633RA002-1	7	.12
633RA003-1	7	.12
633RA003-3	7	.12
633RA003-5	0	0
633RA003-801	7	.12
633RA004-1	0	0
633RA004-3	0	0
633RA004-5	0	0
633RA005-1	0	0
633RA006-1	0	0
633RA007-1	0	0
633RA008-1	0	0

4.4 STRESS ANALYSIS OF FULL SPAN WING BOXES

The full span wing box preliminary designs (see Figures 65 thru 67) have been analyzed in two portions each with an arbitrary line of demarcation at span station 100, for the metal designs; and at span station 107 for the composite design. The demarcation line permits examination of features associated with splicing, or attaching the wing to the fuselage. It also permits evaluation of the various wing box concepts without the accommodations required for attachment to the fuselage.

Sizing of the structural members was done using internal loads from the finite element model of the baseline, fatigue and damage tolerance design allowable stresses and fuel pressures as described for the Analytical Assembly phase.

Finite element models of the Preliminary Design wings were made to demonstrate the member stress levels, to examine the deformations of the boxes and to obtain the modes of vibration for the various designs, including the baseline, updated for fatigue and damage tolerance allowables. Figures 68 and 69 illustrate the stress distributions.

4.5 PIVOT FITTING STRESS ANALYSIS

The desirability of a fail-safe multi-loadpath wing pivot fitting, for possible weight and cost reduction, prompted evaluation of a three path fitting: One path in the continuous laminated aluminum skin and two additional paths through steel plates. This concept was modeled mathematically and is illustrated in Figures 70 thru 72. Figures 73 and 74 show the pivot fitting arrangements for preliminary design drawings 633RW001 and 633RW002.

A three path fitting was also used with the graphite-epoxy wing, as shown in Preliminary Design drawing 633RW003, and illustrated by Figure 75. In this case two Titanium tapered plates supplement the continuous composite skin.

Two concepts were evaluated for reacting wing torque at the attachment to the fuselage. The first concept as illustrated by Figures 72 and 73 maintains a closed torque box up to the center of the pivot lugs. A second concept terminates the wing torque

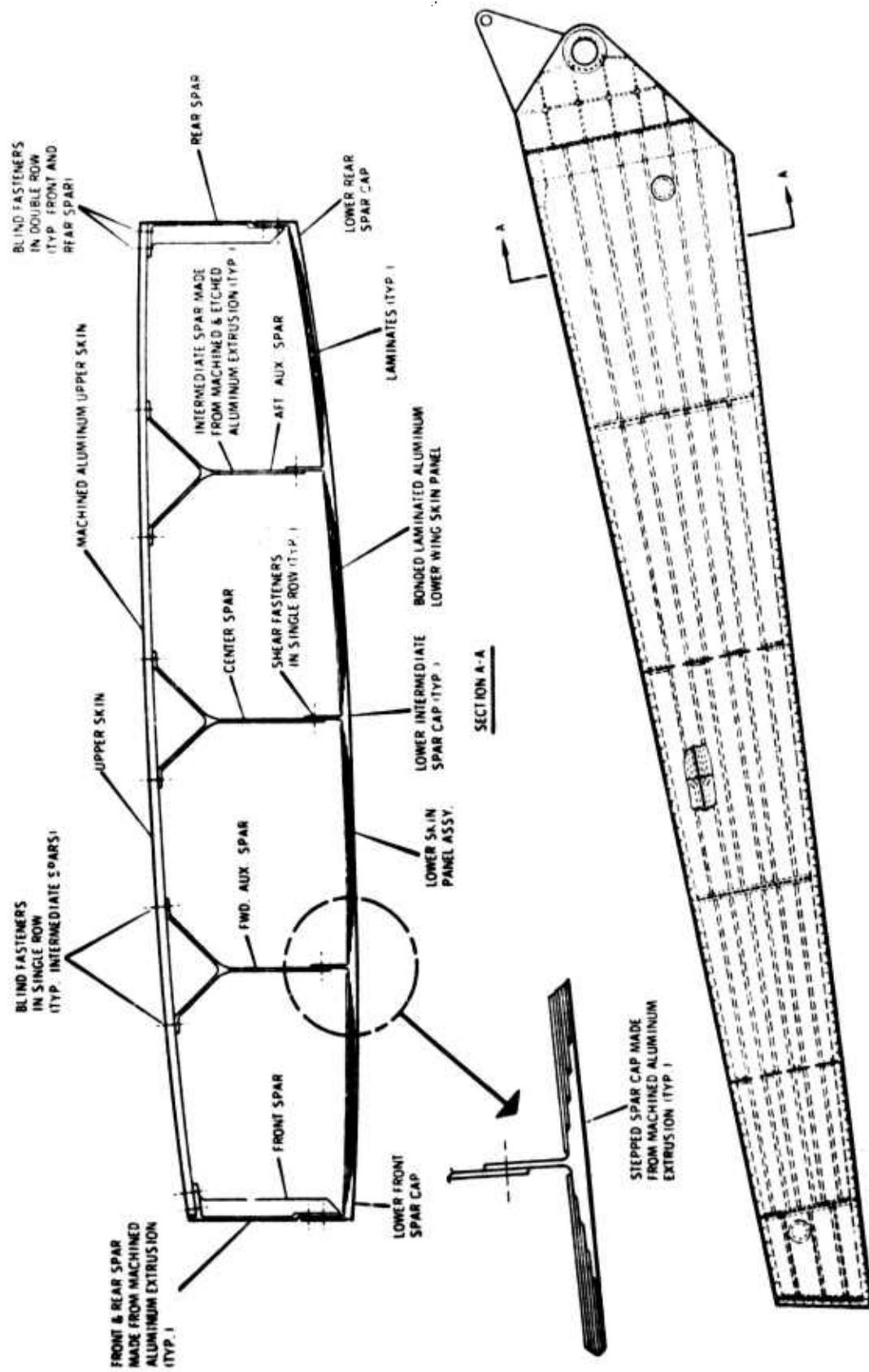


Figure 65 ATW-4 Wing Box Laminated Aluminum Lower Skin With "Y" Spars

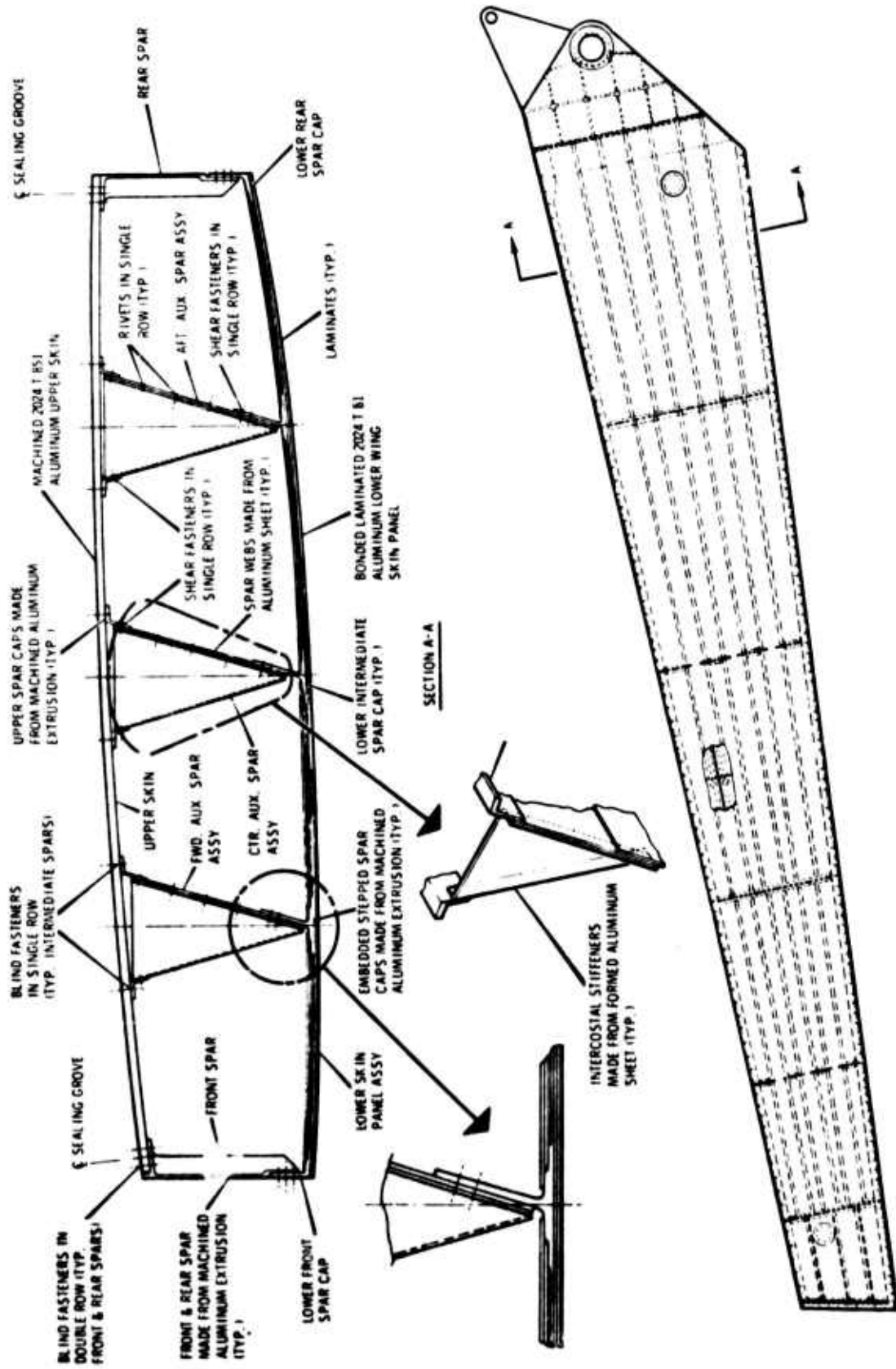


Figure 66 ATW-4 Wing Box Laminated Aluminum Lower Skin, Canted Spars

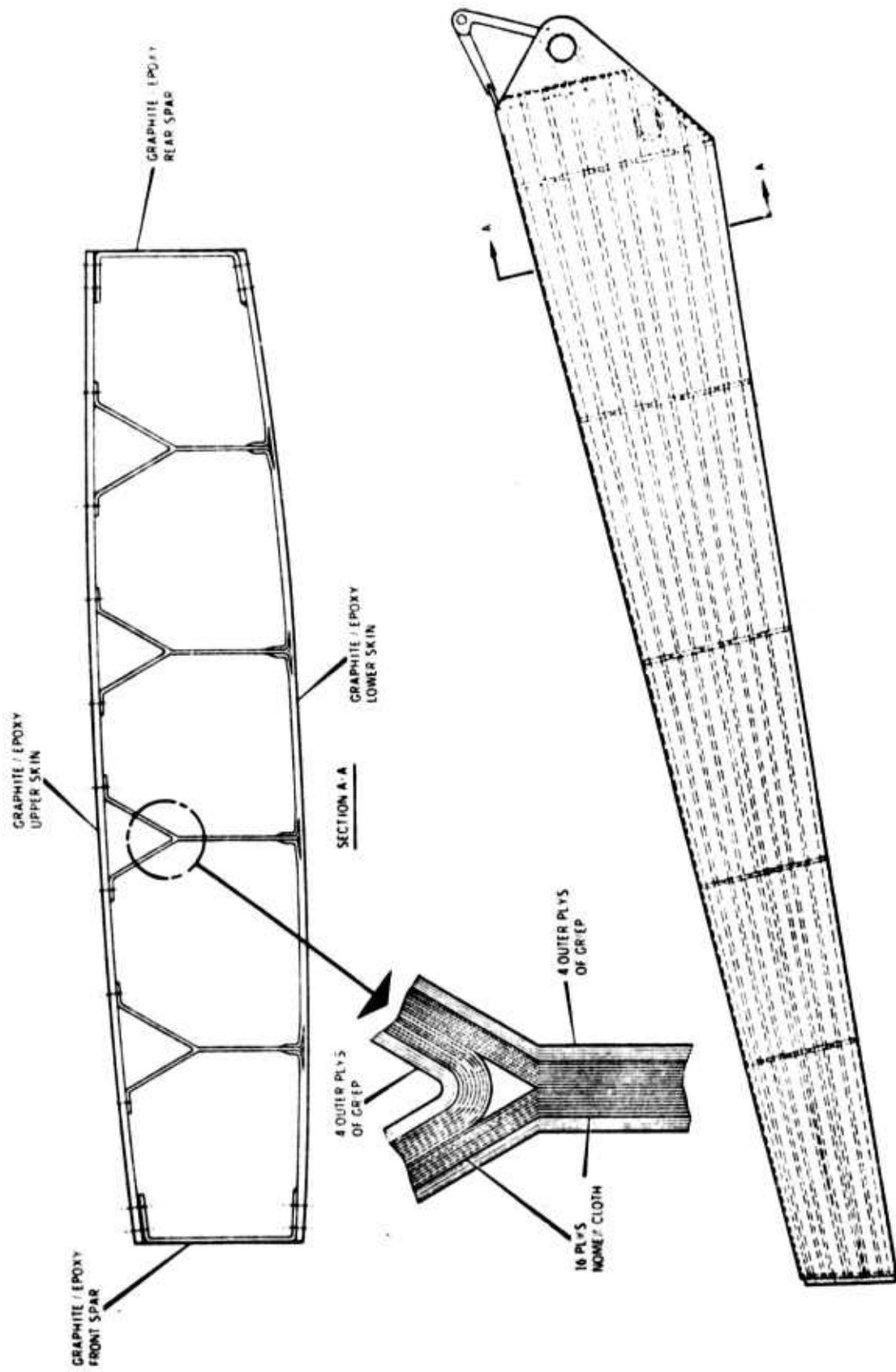
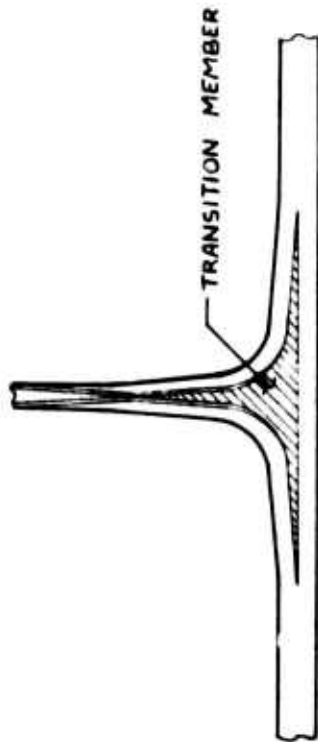


Figure 67 ATW-4 Graphite Epoxy Composite Wing Box "Y" Spars/Embedded
(Sheet 1 of 2)

633RW003 LOWER SPAR CAP DETAIL



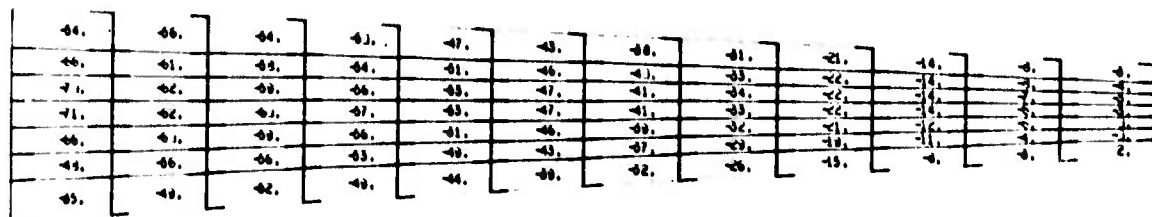
TYP ALL INTERMEDIATE SPARS

Figure 67 ATW-4 Graphite Epoxy Composite Wing Box "Y" Spars/Embedded Lwr. Caps
(Sheet 2 of 2)

63.	59.	55.	52.	49.	36.	26.	18.	12.	8.	5.	3.	2.
60.	64.	62.	59.	52.	41.	31.	21.	14.	8.	4.	3.	2.
54.	62.	62.	61.	52.	42.	31.	23.	15.	12.	5.	3.	2.
41.	51.	54.	52.	48.	37.	28.	21.	14.	12.	7.	3.	2.

STRESS 77

Figure 68 Lower Surface of Metal Y Box Finite Element Model with Condition 1 (KSI, Ultimate)



PRINCIPAL STRESS 2

Figure 69 Upper Surface of Metal Y Box Finite Element Model with Condition 4 (KSI, Ultimate)

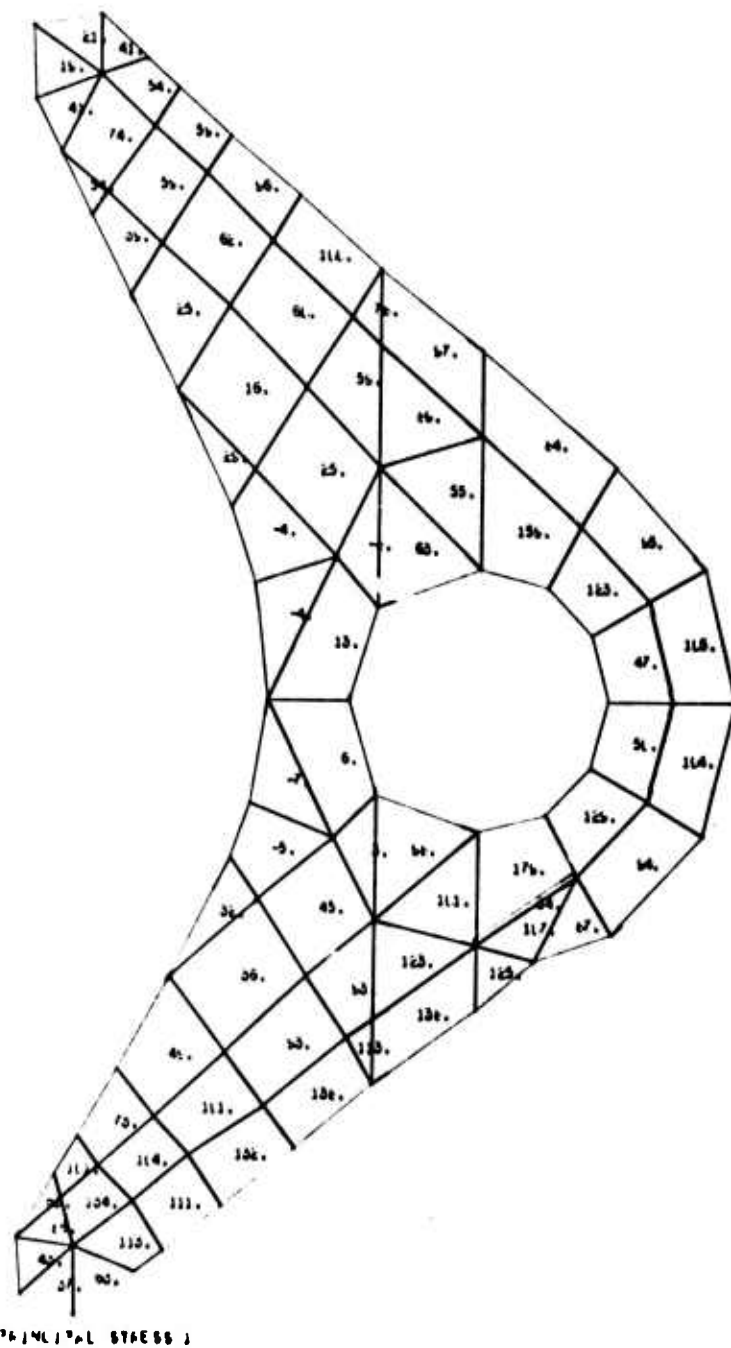
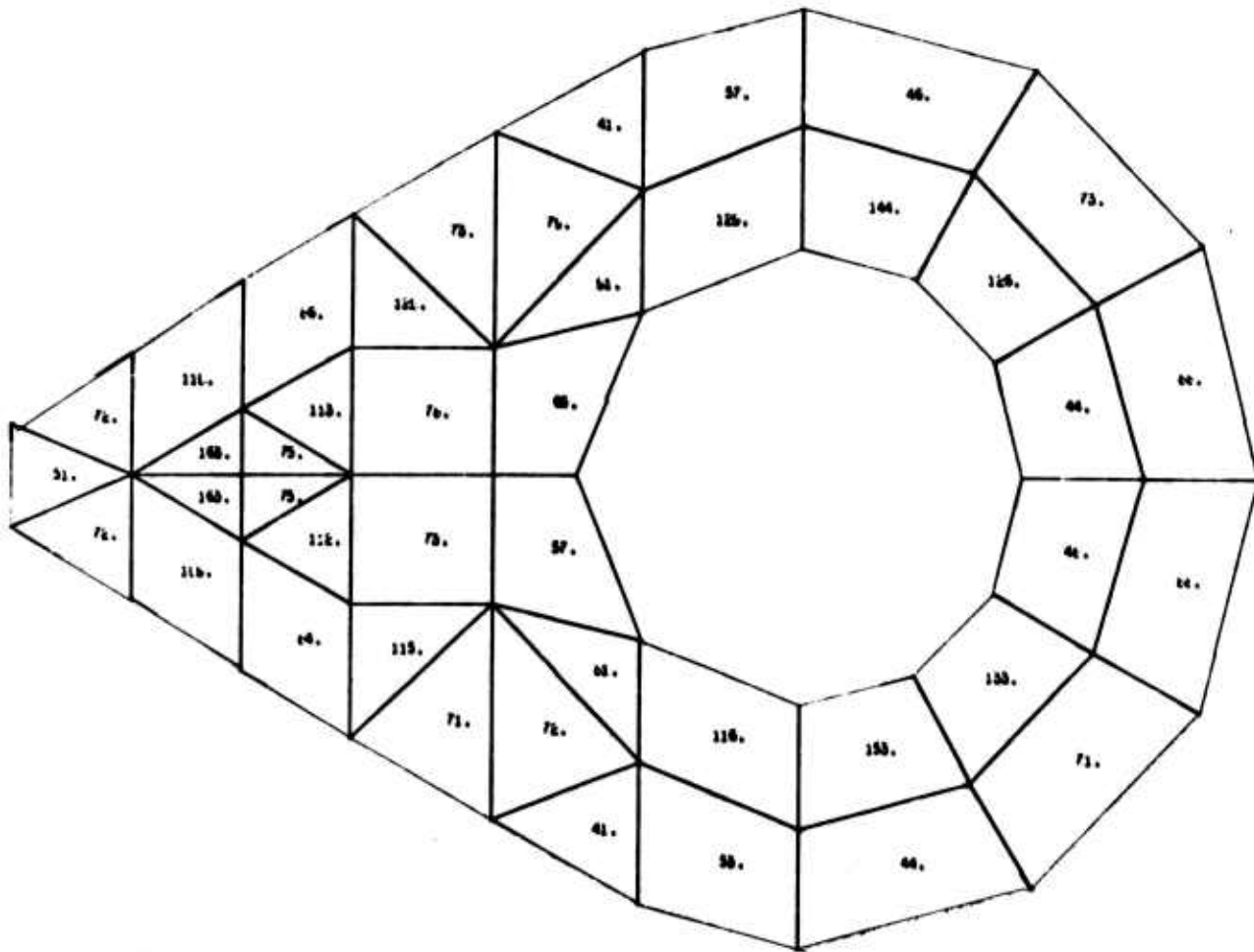
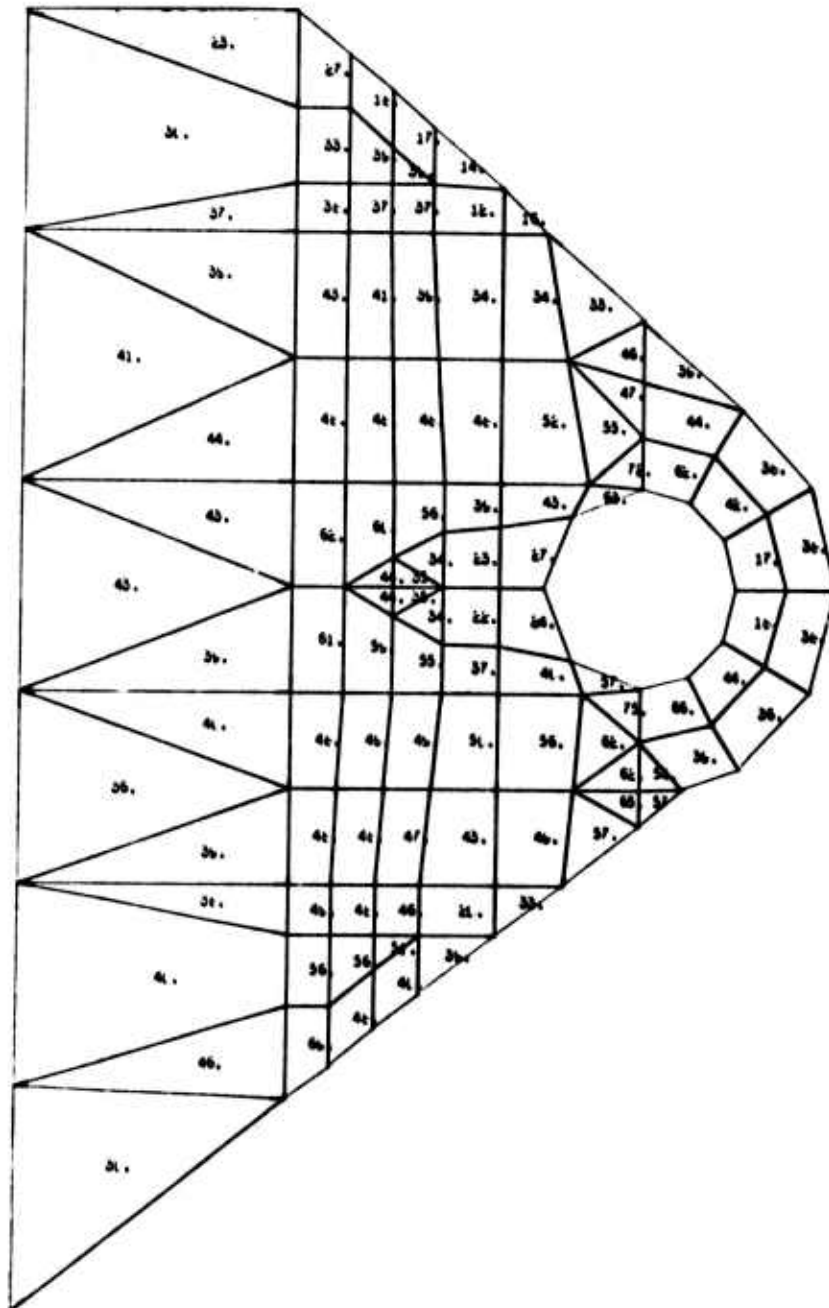


Figure 70 Failsafe Pivot Inner Steel Element



PRINCIPAL STRESS :

Figure 71 Failsafe Pivot Outer Steel Element



PRINCIPAL STRESS

Figure 72 Failsafe Pivot Aluminum Element

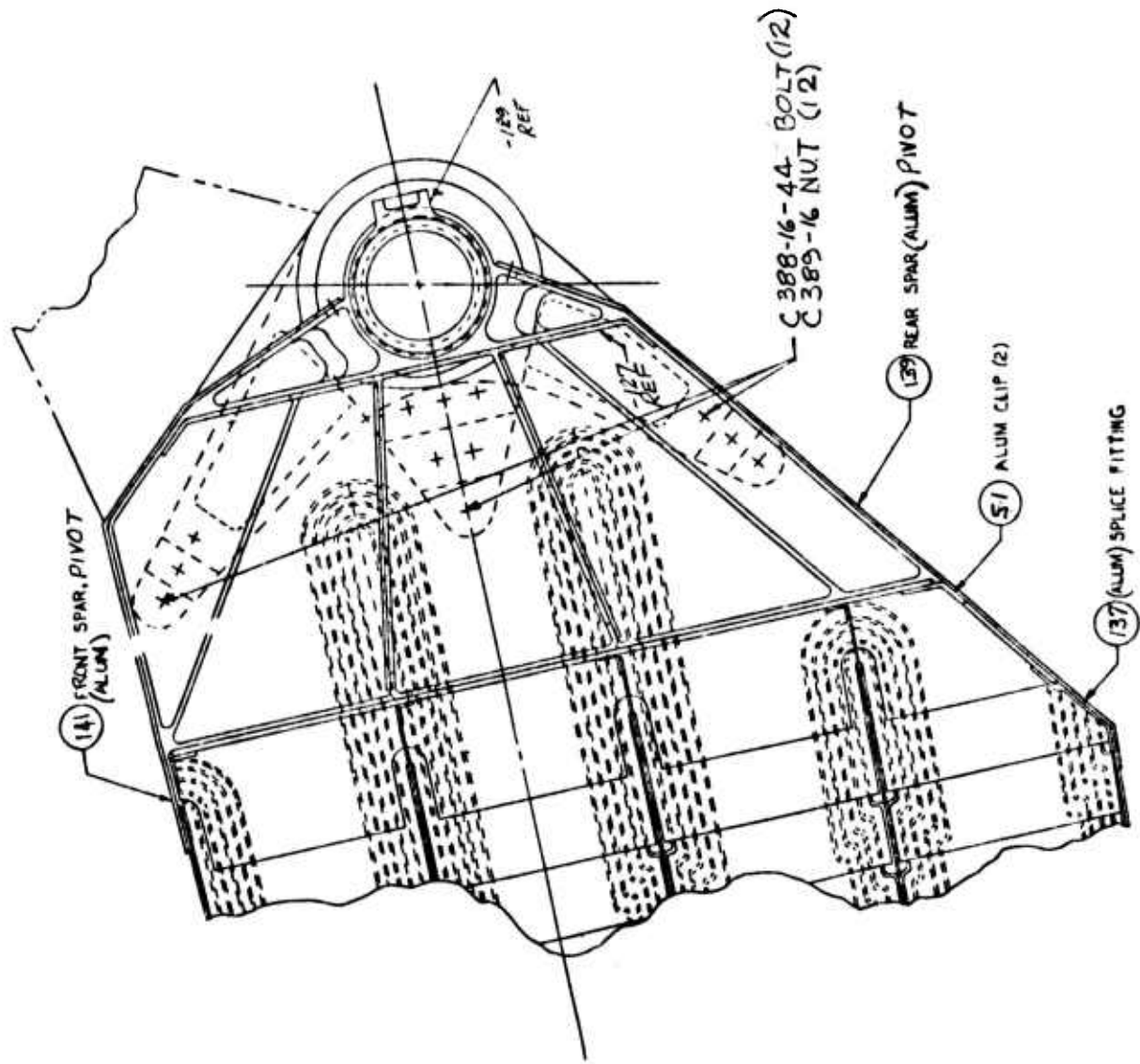


Figure 73 633RW001 Pivot Fitting (Sheet 1 of 2)

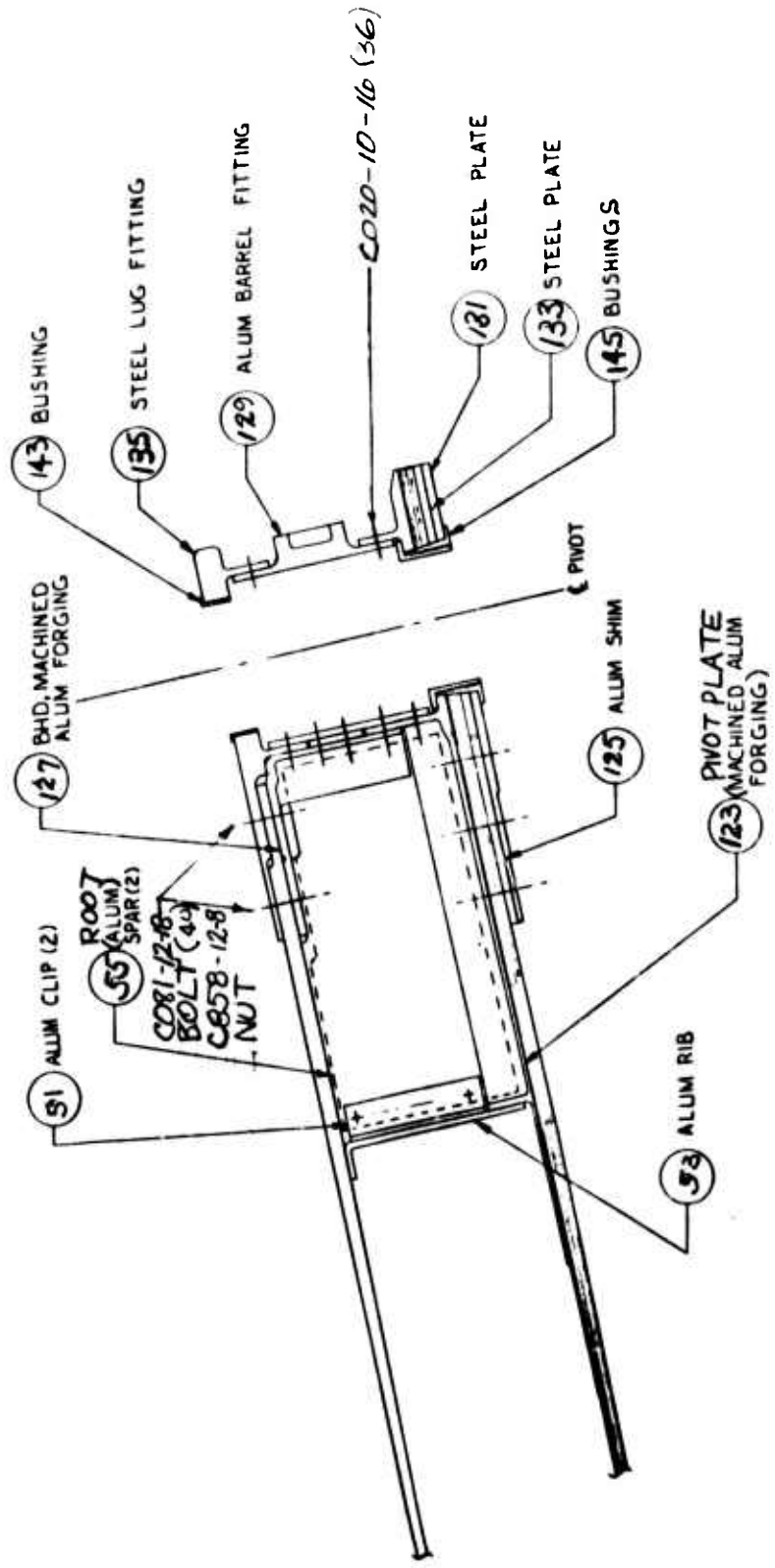


Figure 73 633RW001 Pivot Fitting (Sheet 2 of 2)

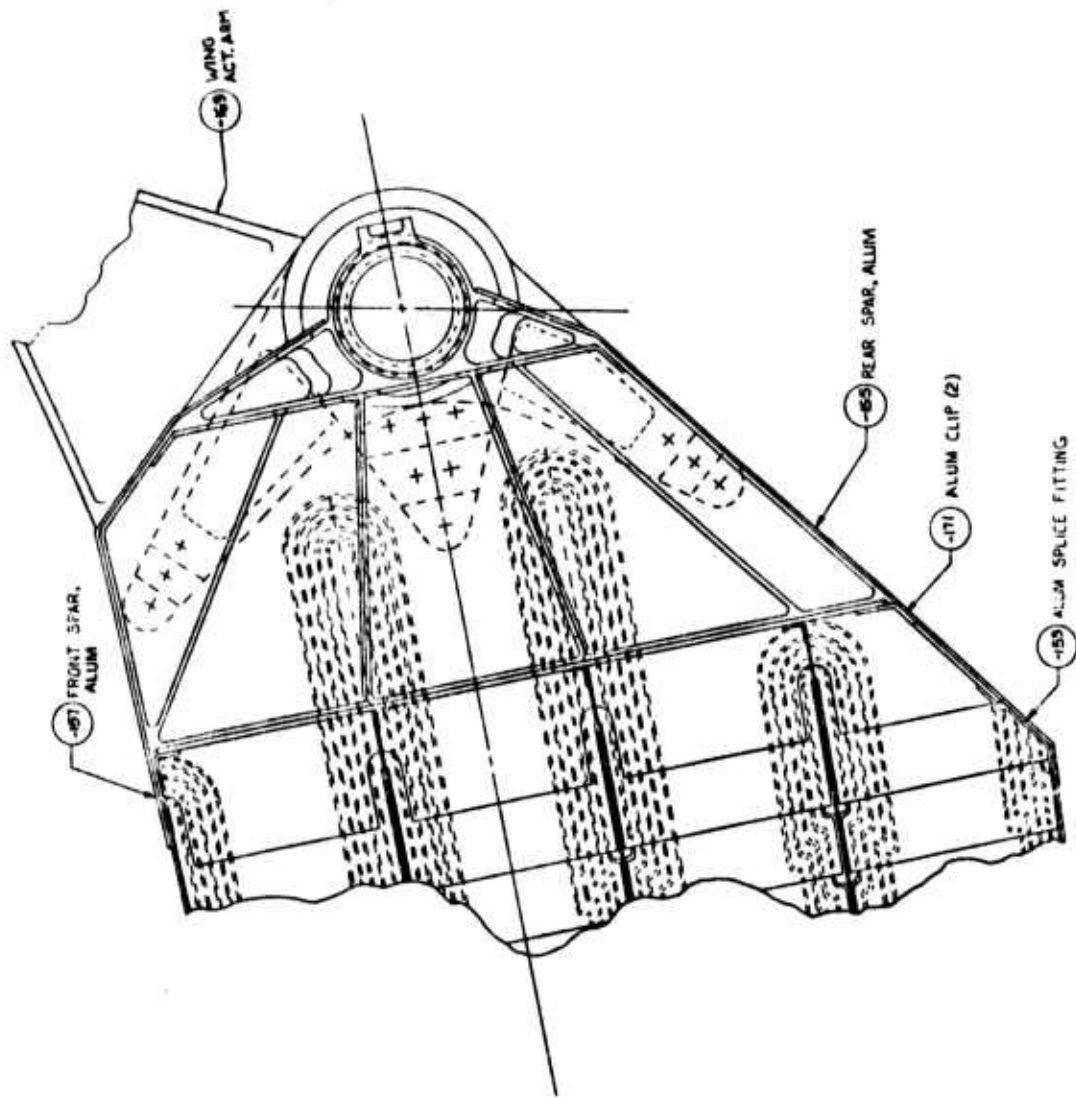


Figure 74 633RW002-1 Pivot Fitting (Sheet 1 of 2)

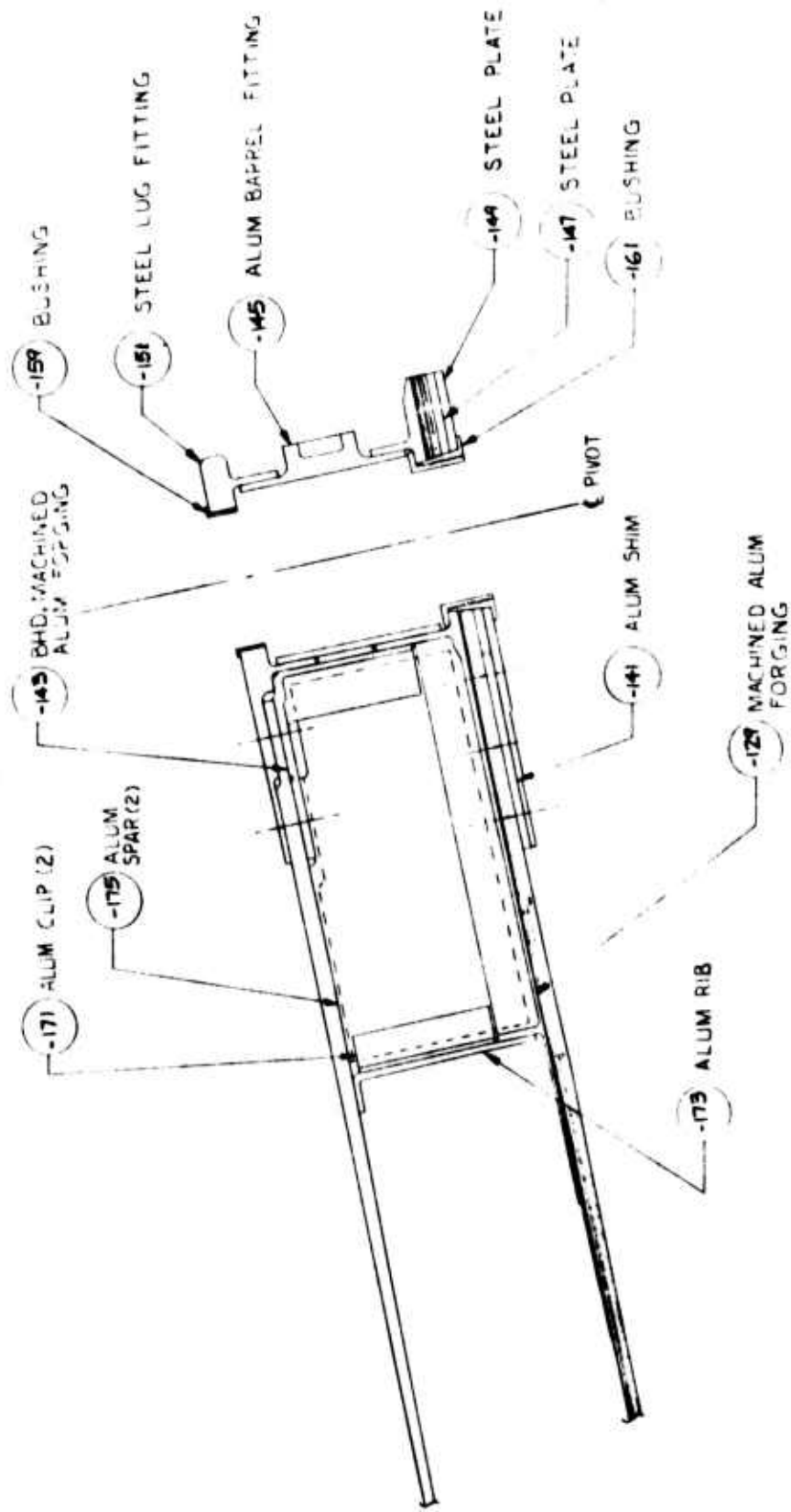


Figure 74 633RW002-1 Pivot Fitting (Sheet 2 of 2)

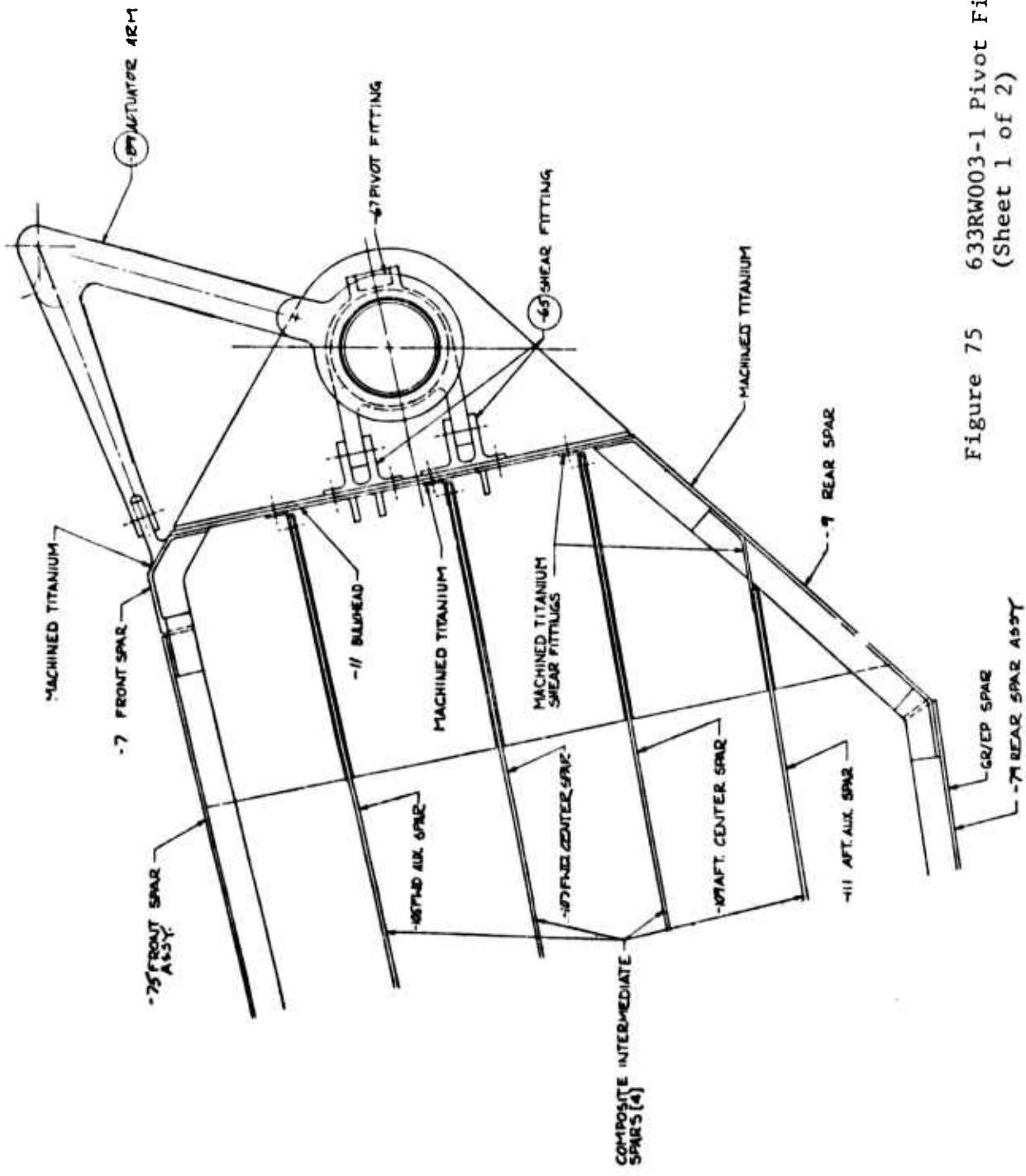


Figure 75 633RW003-1 Pivot Fitting (Sheet 1 of 2)

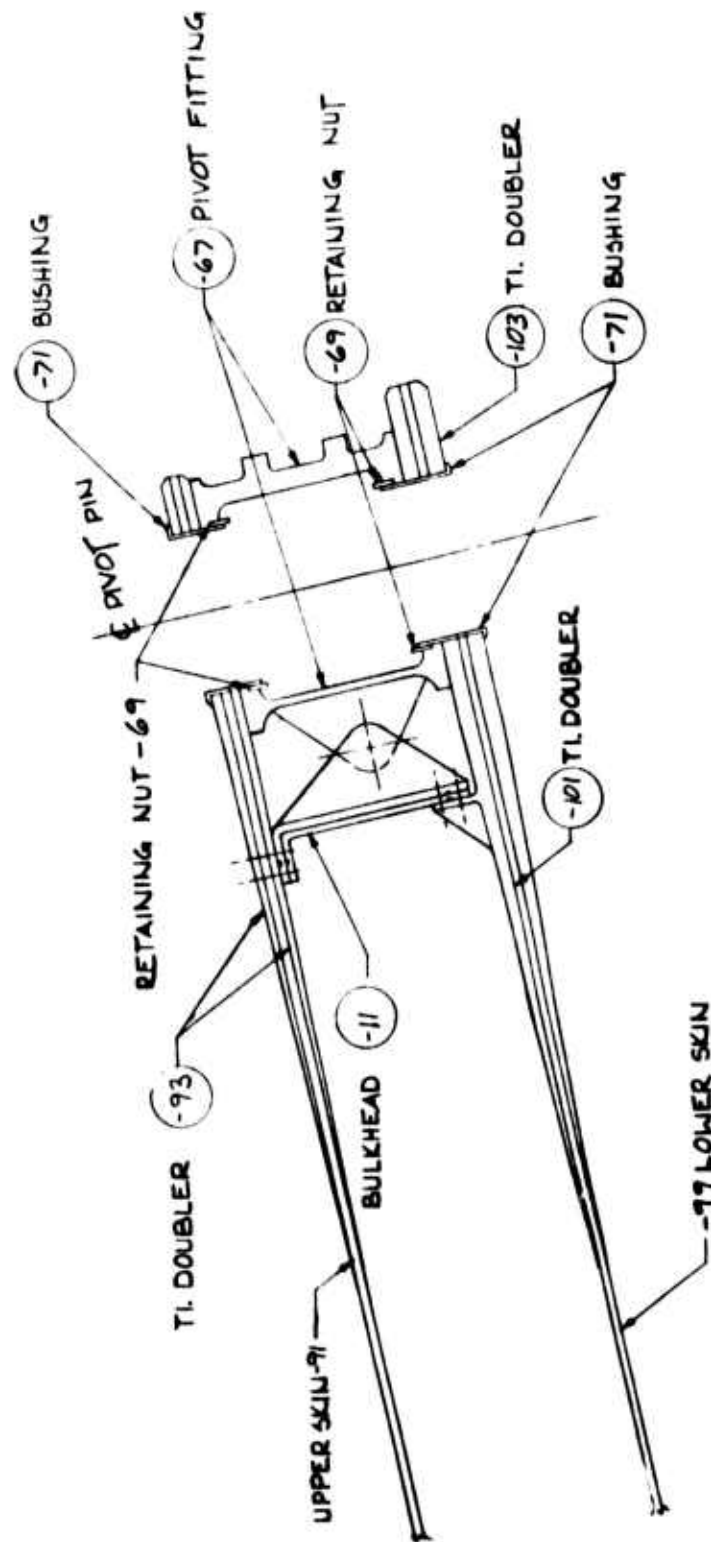


Figure 75 633RW003-1 Pivot Fitting (Sheet 2 of 2)

box outboard of the pivot as illustrated in Figure 76. This second concept permits improved accessibility to the area of the lugs, but it produces additional bending in the upper and lower skins in the vicinity of the pivot fitting.

The weights of these lugs are shown in combination with wing boxes in Table XI. One of these combinations, the composite wing box and the continuous torque box metallic pivot fittings, has not been explored fully. This combination represents a goal which should be achievable, but whose details are not essential to the study at this stage.

4.6 FLUTTER ANALYSIS

Flutter checks were made of the baseline wing box, the graphite-epoxy wing box and the Y-spar advanced metallic wing box. Results of these checks are summarized in Table XII and they indicate satisfactory flutter characteristics on the basis of having more than twenty percent speed margin above .90M at sea level with the wings swept fully forward.

The normal modes of vibration were obtained for each design by means of the program (UGO) used for mathematically modelling each design; and with the final mass distribution of the wing box and concentrated masses representing the leading and trailing edge.

Flutter speed solutions were obtained using procedure AA8 which uses the normal modes expressed as generalized mass coordinates and aerodynamic force coordinates computed using the Kernel function method.

The Kernel function relates pressure at specific points on the airfoil as functions of "downwash". The "downwash" is a function of the model deflections and is also related to the free stream flow normal to the chord plane.

Integration of these data over the airfoil forms a pressure distribution, or aerodynamic force coordinates, for each deflection mode, for use in the solution procedure, and each solution yields a value of frequency and the damping required to maintain neutral stability.

Plots of damping versus velocity provide the means to predict a critical velocity or flutter speed with respect to the minimum damping criteria. These plots are illustrated in Figures 77 through 79.

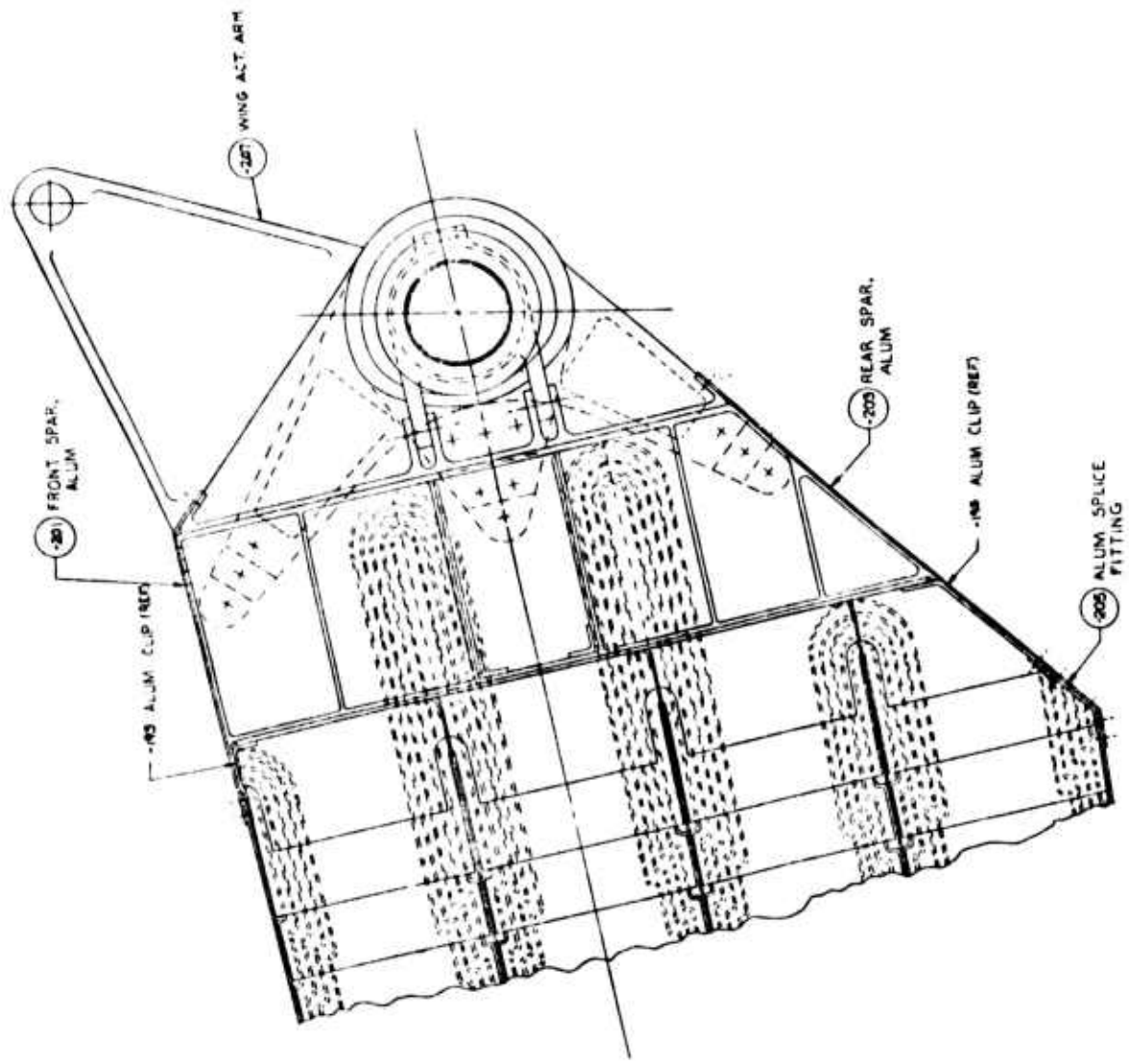


Figure 76 633RW002-3 Pivot Fitting (Sheet 1 of 2)

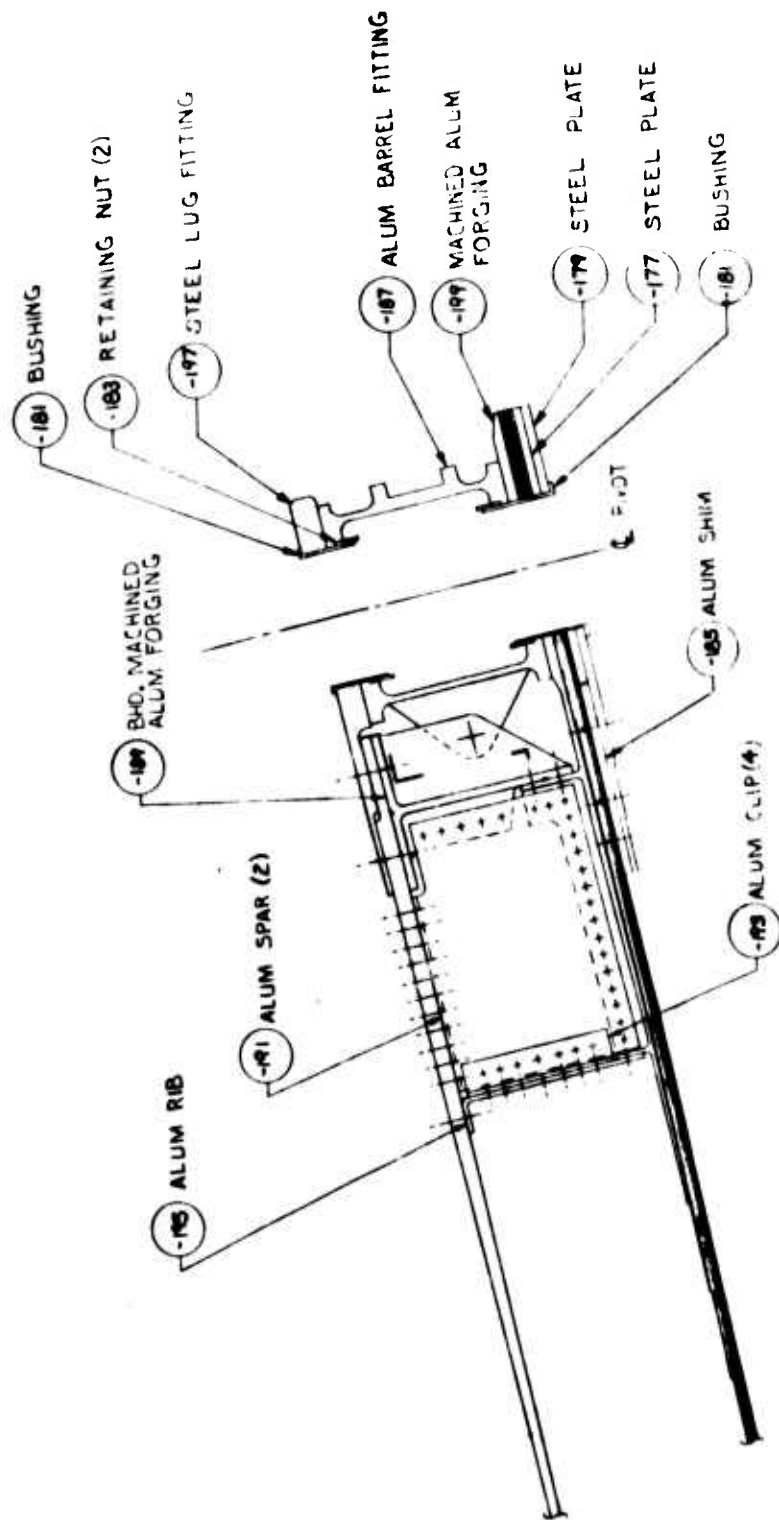


Figure 76 633RW002-3 Pivot Fitting (Sheet 2 of 2)

TABLE XI
WING WEIGHT COMPARISON (LBS. PER SIDE)

	<u>Box Outbd. of Root</u>	<u>Root</u>	<u>Recurring & Non-Opt Weight</u>	<u>Total Wing Box</u>
Baseline	1489	740	530	2759
GR/E (A)	1045	825	378*	2248
GR/E (B)	1045	583	378*	2006
3Y Mt1 (TACT Ftg.)	1228	770	530	2528
3Y Mt1 (Box Ftg.)	1228	583	530	2341
3V Mt1 (TACT Ftg.)	1298	770	530	2598
3V Mt1 (Box Ftg.)	1298	583	530	2411

* Includes 48 lbs. for fuel barrier material

TABLE XII

RESULTS OF FLUTTER CHECKS

<u>Configuration</u>	<u>Flutter Speed</u> (S.L. Density, KTS)	<u>Flutter Frequency</u> (HZ)
Baseline Wing Box	915	11.5
Graphite-Epoxy Wing Box	965	13.3
Adv. Metallic (Y-Spar) Wing Box	880	9.4

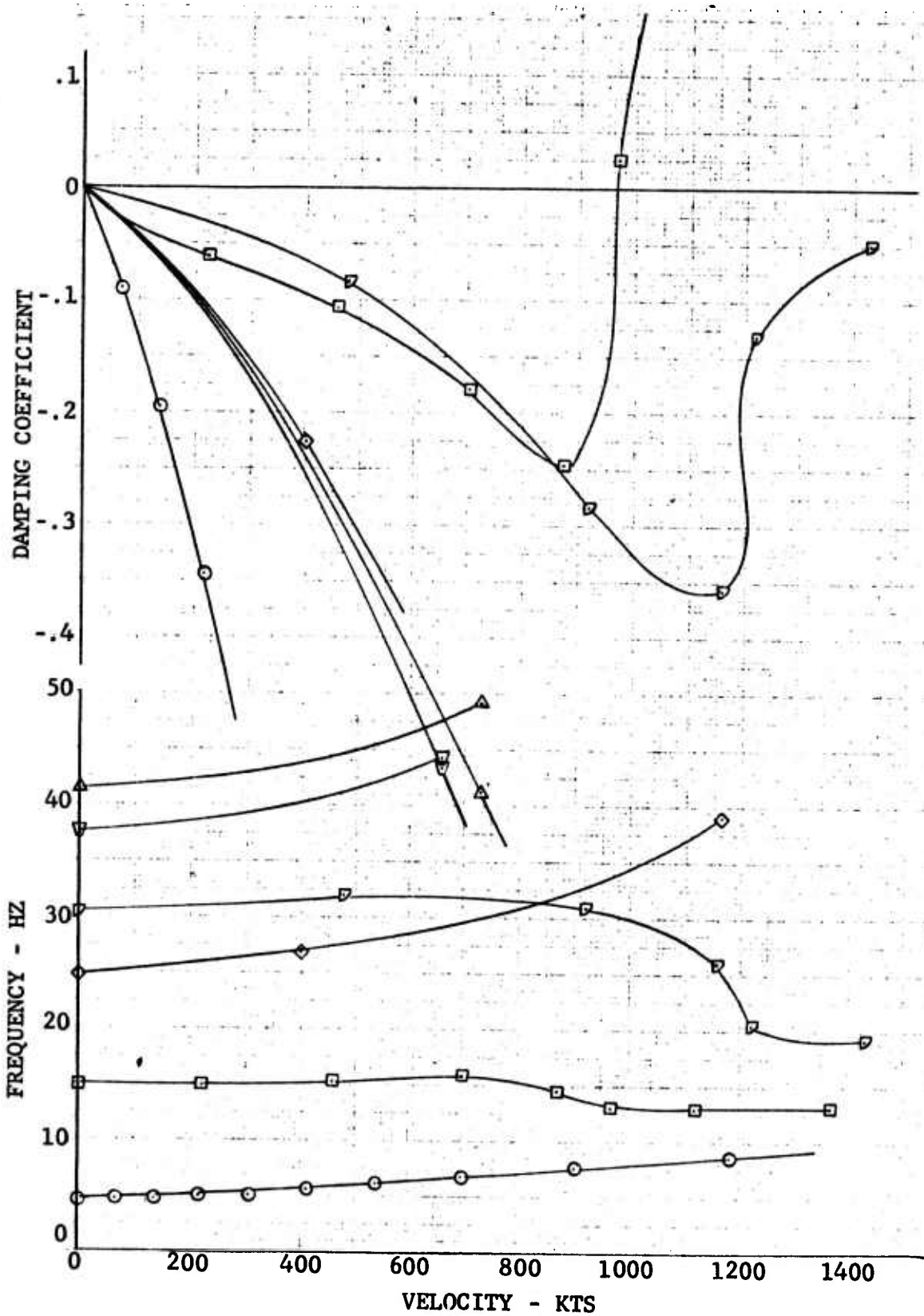


Figure 77 ATW-4 Composite Wing Mach No. 0.9 Seal Level

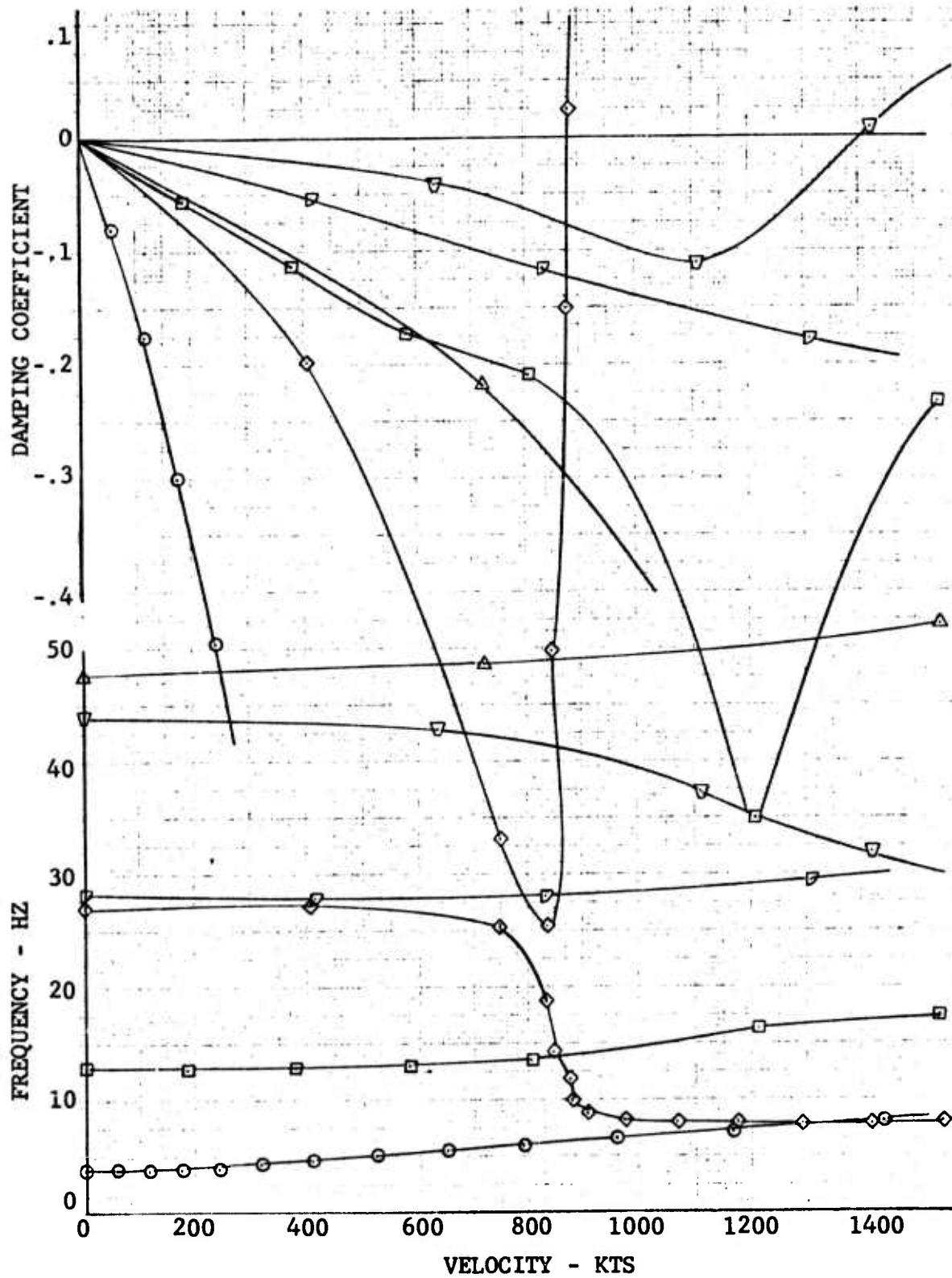


Figure 78 ATW-4 Advanced Metallic Wing Mach No. 0.9 Sea Level

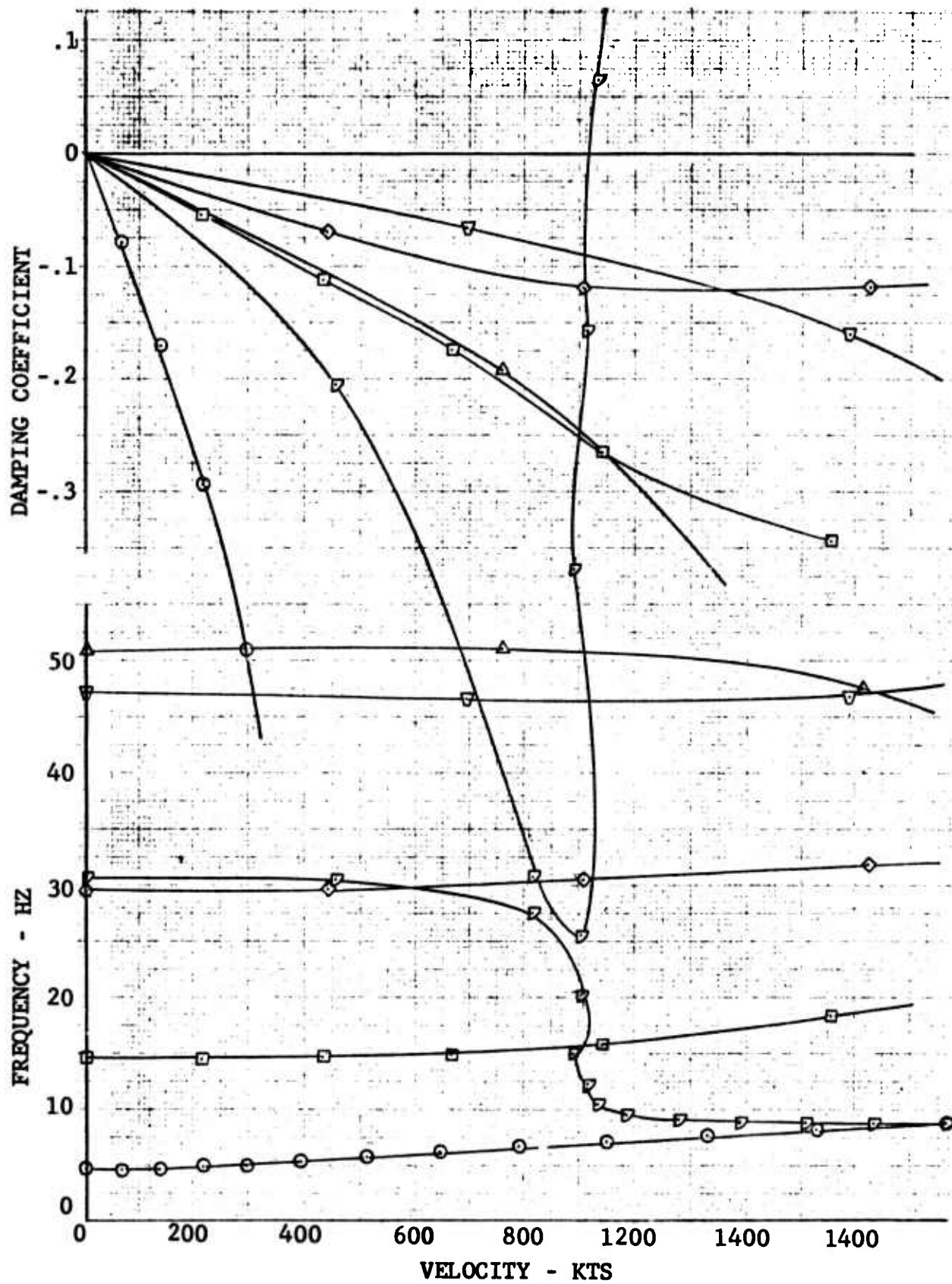


Figure 79 ATW-4 Baseline Wing Mach No. 0.9 Sea Level

4.7 "FAIL SAFE" AND "SAFE CRACK" SCORES FOR FULL SPAN WINGS

"Fail Safe" and "Safe Crack" scoring of the preliminary designs was made using the same procedure that was used for the "Analytical Assemblies" and the results are shown in Tables XIII and XIV.

TABLE XIII
 PRELIMINARY DESIGN "FAIL SAFE" RATING

	<u>MINIMUM NUMBER OF ELEMENTS IN FAILSAFE PATH</u>	<u>SCORE</u>
633RW000	0	0
633RW001	4	.120
633RW002-1	0	0
633RW002-3	0	0
633RW003-1	0	0
633RW003-3	0	0

TABLE XIV

PRELIMINARY DESIGN "SAFE CRACK" RATING

	$\frac{18.3(10^6)}{16.4(10^6)} \frac{1.5^*}{1.5^*}$	$\sigma_{\text{Safe Crack}} (Ult)$	σ_{Max}	$\frac{\sigma_{\text{Safe Crack}}}{\sigma_{\text{Max}}} = R$	Score = $(.08) \frac{R}{R_{\text{Max}}}$	Rank
633RW000	41500	41576	.998	.078	2	
633RW001	66900	65232	1.026	.080	1	
633RW002-1	57840	56398	1.026	.080	1	
633RW002-3	57840	56398	1.026	.080	1	
633RW003-1	-	75844	-	.080**	1	
633RW003-3	-	75844	-	.080**	1	

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** Based on the assumption that the advanced composite design is as good as the best.

* Maximum allowable spectrum stress for stable crack growth, times ratio of limit root bending to max spectrum bending; times 1.5 for equivalent ultimate.

S E C T I O N V

F A T I G U E A N A L Y S I S O F A D V A N C E D T R A N S O N I C W I N G (A T W - 4)

Fatigue analyses have been performed to demonstrate analytical compliance of the three preliminary wing designs with the baseline fatigue requirements. The following discussions provide a summary of fatigue analysis procedures and results.

5.1 FATIGUE CRITERIA & PROCEDURES

The fatigue life requirements used for this contract are identical to those of the baseline and are summarized below:

1. The structural service life shall be 1280 flights.
2. The fatigue design life shall be the structural service life times a scatter factor of 4.
3. The AFFDL furnished General Dynamics with an analytical fatigue loads spectrum of B-1 wing pivot bending moments. This B-1 wing pivot bending moment spectrum was adjusted to yield wing pivot bending moments compatible with the ATW-4 wing geometry and configurations as discussed in paragraph 4.0 of FZM-12-6466. The analytical fatigue loads spectrum for the ATW-4 wing box is shown in Table XV and Figure 80. The analytical design sections were selected at a wing chord plane perpendicular to the load reference axis (LRA) as shown in Figure 80. Hence, the wing pivot bending moment spectrum was adjusted by the appropriate factors for each mission segment to yield the wing bending moment spectrum 68" outboard of the wing pivot for the fatigue analyses. These factors are presented in Table XV under the column heading $BM_p/BM(68")$.

The fatigue loads spectrum in Table XV was range pair cyclic counted using computerized procedures developed for the AMAVS program (Contract No. F33615-73-C-3001). The wing pivot bending moment was used for all analyses.

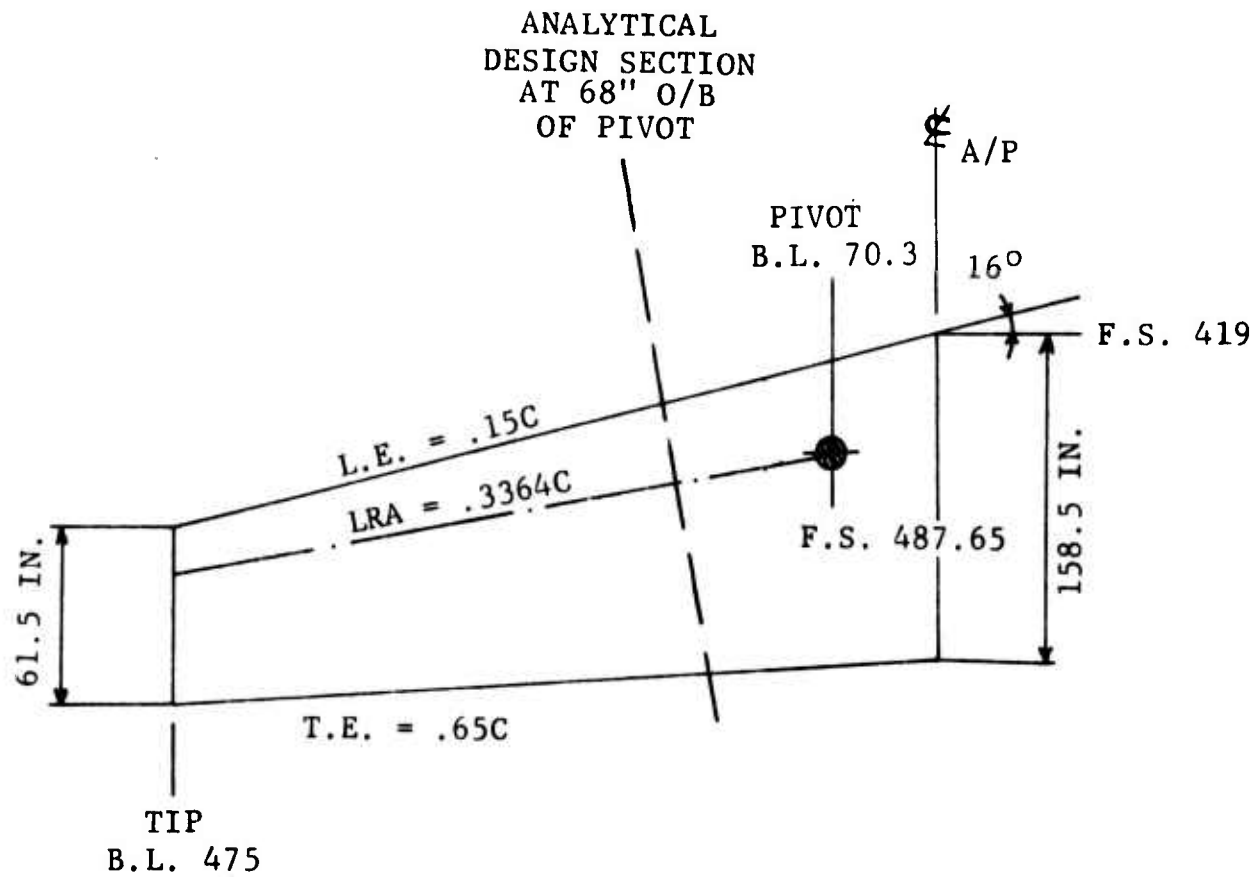
The safe-life concept was used as the primary means of satisfying the fatigue life requirements. The fatigue analyses were generally done according to the following procedure:

1. Fatigue control points were selected.
2. Control point unit stresses (stress per unit of load) were established.
3. Fatigue test (S-N) data were selected for each control point for the appropriate material and stress concentration (K_T).
4. A fatigue stress spectrum was established for each control point by combining the repeated loads and occurrences with the unit stress data.

TABLE XV ANALYTICAL WING PIVOT BENDING MOMENT SPECTRUM FLIGHT-BY-FLIGHT COMPOSITE MISSION

LOAD STEP	MISSION SEGMT.	BNP BY (68")	BNPx10 ⁶ in.-lb. Limit Ld	WING SWEEP ANGLE	TAK(O/F)	% OF CONDITION		PIVOT BmPx10 ⁶ MAX	PIVOT BmPx10 ⁶ MIN.	CYCLES/ MISSION
						MAX.	MIN.			
1	GROUND	1.5	-2.9624	16°	58.69	11.5	60.8	-1.8	-1.8	1
2						85.1	51.5	16.62	10.06	0.01
3	POST TAKE-OFF	1.5	17.93	16°	70.13	M	51.5	14.96	10.06	0.1
4						59.3	51.5	11.58	10.06	2
5						51.5	41.4	10.06	8.08	2
6						60.5	56.8	11.8	11.09	2
7						60.9	36.2	11.89	7.07	1
8						50.5	44.5	9.86	8.69	29
9	CLIMB	1.5	18.3	16°	38.97	M	37.0	10.25	6.77	1
10						64.5	56.0	11.8	10.25	22
11	CRUISE					56.0	50.5	10.25	9.24	22
12	&					69.0	46.5	12.63	8.51	1
13	REFUEL					46.5	31.0	8.51	5.57	1
14						54.0	46.5	9.88	8.51	58
15						46.5	42.5	8.51	7.78	58
16	FLY-UP	1.5	16.106	65°	126.15	M	25.5	11.03	4.11	1
17						46.7	26.5	7.52	4.27	1
18						64.7	48.2	10.42	7.76	0.1
19			16.106	65°		55.1	28.0	8.87	4.51	1
20						45.1	34.5	7.26	5.56	7
21	TERRAIN	1.5			126.15	G	8.1	7.83	1.3	1
22						48.6	8.1	7.83	1.3	1
23	FOLLOWING					32.1	18.1	5.17	2.92	132
24						28.6	-4.4	4.61	-7.1	1
25						19.6	8.4	3.16	1.35	132
26						65.4	3.7	10.53	.596	1
27						49.5	6.8	7.97	1.1	9
28						32.1	14	5.17	2.255	95
29	PRELANDING	1.5	17.828	16°	70.13	M	51.7	16.42	9.22	0.01
30						82.9	-13.8	14.78	-2.46	0.1
31	GROUND	1.6	-2.9624	16°	58.69	M	71.3	12.71	9.22	1
32	TAKE-OFF	1.5	17.93	16°	70.13		60.8	-1.8	-1.8	1
33	CLIMB	1.5	18.3	16°	38.97		49.4	13.12	8.86	1
34						80.5	56.0	14.73	10.25	1
35						51.7	34.5	9.22	6.15	1
36						59.9	51.7	10.68	9.22	19
37	PRELANDING	1.5	17.828	16°	70.13	M	51.7	9.22	8.22	19
38						76.8	47.9	13.69	8.54	1
39						65.7	56.2	11.71	10.02	4
40						67.0	33.6	11.94	5.99	1
41						60.6	39.9	10.8	7.11	9
42						57.4	43.1	10.23	7.68	48
43	GROUND	1.6	-2.9624	16°	58.69		53.0	9.45	8.22	294
44						11.5	60.8	-1.8	-1.8	8
						15.4	50.2	-4.56	-1.665	154

NOTES: (1) THIS COMPOSITE MISSION TABLE CONTAINS 1143.32 CYCLES PER MISSION & 1,463,449.6 CYCLES PER LIFE.
 (2) LEGEND: M = MANEUVER LOAD G = GUST LOAD BNP = BENDING MOMENT AT WING PIVOT
 (3) THE WING BENDING MOMENT IS IN THE LOAD REFERENCE AXIS SYSTEM. (+) BM IS WING TIP UP. SEE FIGURE 6-1.
 (4) BM (68") = WING BM 68" O/B OF WING PIVOT.



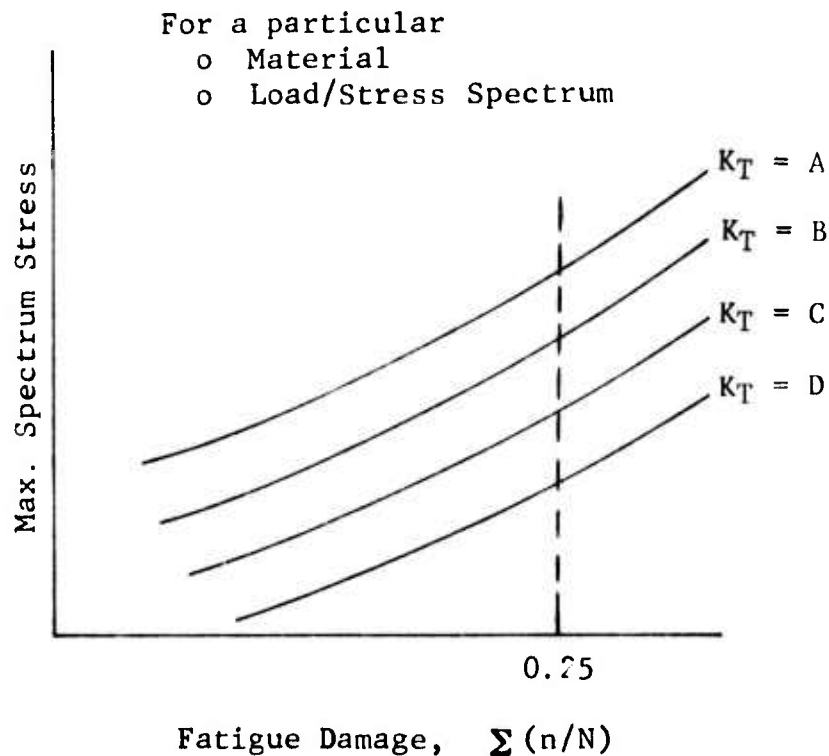
Notes:

1. LRA = Load Reference Axis
2. Wing Bending Moment is in the Load Reference Axis System.

FIGURE 80 Baseline Wing Geometry

5. The fatigue damage calculations were made for the stress spectrum of (4) using the S-N data selected in (3) and using Miner's cumulative damage rule which is $\sum(n/N)=1.0$ at failure.

Flexibility and rapid determination of fatigue damage was accomplished for this program by expanding the above procedure on a parametric basis as shown schematically in the figure below.



The approach in the above figure was used to compile a library of fatigue data which reflected the applicable material and stress level. The data in the above figure was then cross plotted at $\sum(n/N)=0.25$ to produce fatigue damage data as shown schematically in Figure 81. The fatigue damage for an intermediate K_T value can be readily determined from Figure 81.

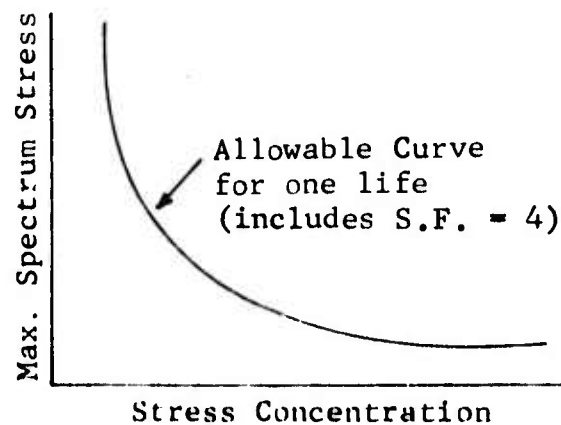


Figure 81 Fatigue Damage Data Schematic

Fatigue S-N data curves were adjusted to account for the reduction in fatigue strength due to elevated temperatures which occur in certain usage segments. The operating temperatures were not available for the various mission segments in the fatigue spectrum. Therefore, an arbitrary 5 per cent reduction in all fatigue S-N data was assumed to apply to all the control points.

The fatigue design allowable curves introduced in this section essentially indicate the interacting relationships between stress, stress concentration factor (K_T), and fatigue life for the baseline fatigue loads spectrum. These curves were developed considering each of the materials utilized in the design of the wing lower surface. The fatigue allowable curves are presented in Appendix (B).

5.2 PRELIMINARY DESIGN FATIGUE ANALYSIS

Fatigue design allowable stresses were determined for each fatigue control point in the three preliminary wing box designs and in the baseline wing box. These allowable stresses are the maximum allowable spectrum stress based on the calculated K_T of each control point and are summarized in Table XVI. As previously stated, Miner's Rule was used in the fatigue analyses for developing the fatigue design allowable curves. Each of the designs were sized to meet or exceed the fatigue requirements summarized in paragraph 5.1. For the slow crack growth structure the final allowable design stress was generally based on the damage tolerance requirements of Section VI whereas the final allowable design stress for the multiple load path-fail safe structure was generally based on the fatigue requirements.

Figures 82 through 87 show the general location of the selected control points. Table XVI indicates the K_T used for design and the net section max allowable spectrum stress at each control point. The max spectrum wing pivot bending moment is 16.42×10^6 in.-lbs. The max design limit wing pivot bending moment is 18.3×10^6 in.-lbs.

The selected control points are located in the wing lower surface of each design because the lower surface has primarily tension loads. The upper surface is primarily loaded in compression or some small tension loads; consequently, the upper surface has been statically designed primarily for compression buckling requirements.

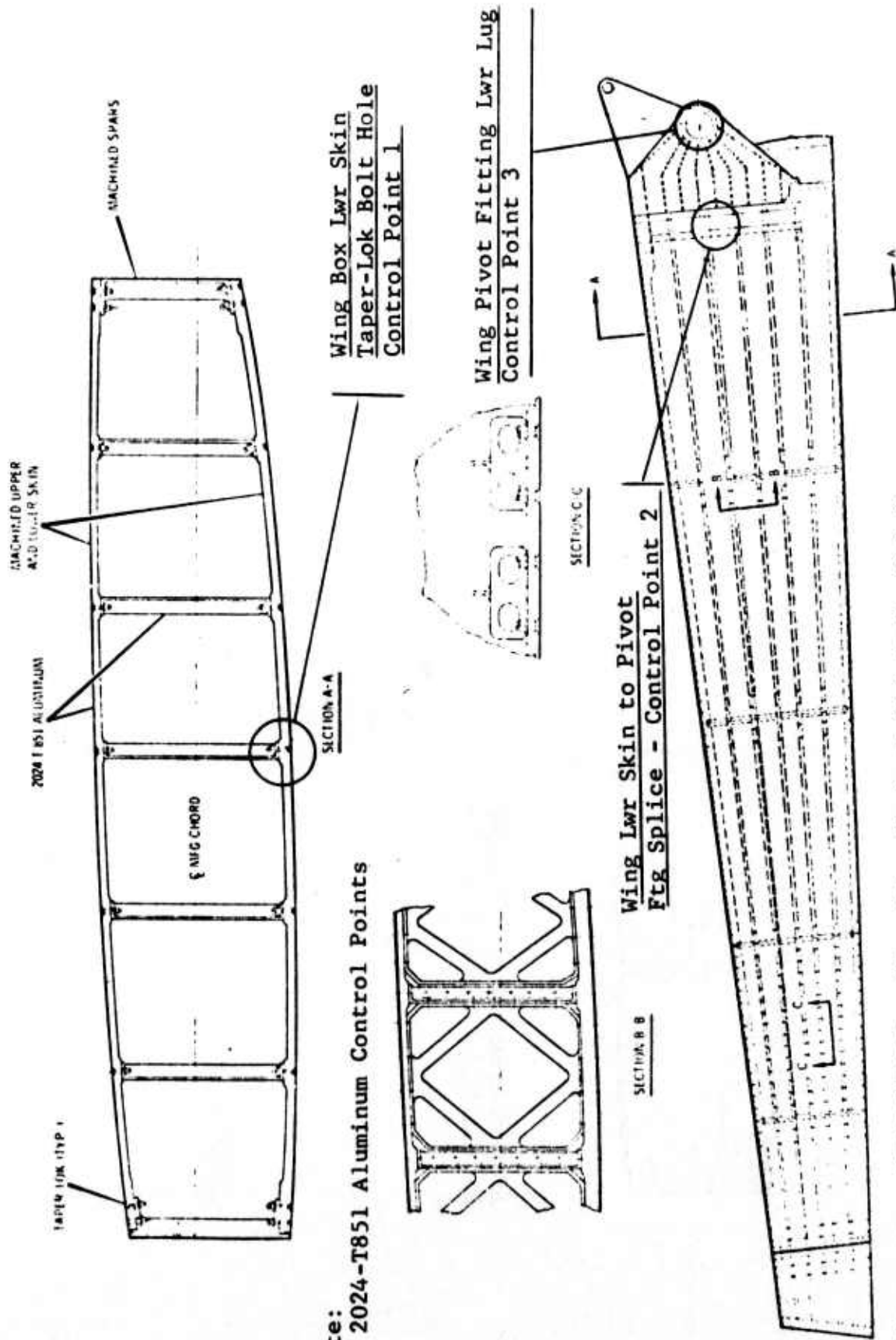
The selection of control points was based on a review of the stresses and a review of the final preliminary design drawings to locate areas of known or potential stress concentrations. Control point locations other than those documented in this section would probably require evaluation during a detail design effort.

TABLE XVI

FATIGUE ANALYSIS
PRELIMINARY WING DESIGNS

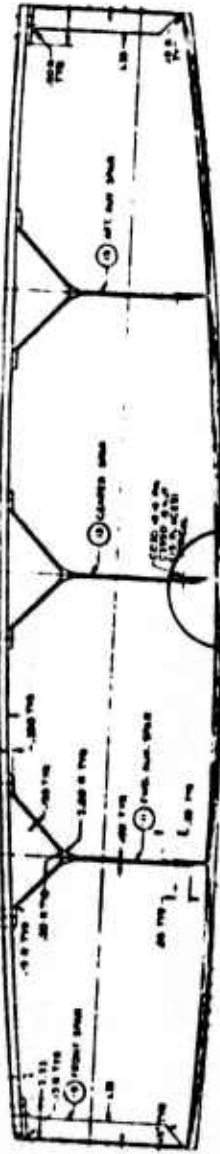
	* Wing Configuration and Fatigue Control Points	Kt	Max. Allowable Spectrum Stress, KSI
633-RW000	Baseline Wing Box, Machined Skins & Spars, Plate 1. Wing Box Lwr Skin Taper-Lok Bolt Hole - Aluminum 2. Wing Lower Skin to Pivot Fitting Splice - Aluminum 3. wing Pivot Fitting Lower Lug - D6ac	3.2 3.2 4.5	30.7 30.7 99.4
633-RW001	Laminated Lwr Skin, Extruded "Y" & Machined Spars 4. Spar Web to Cap Attach Bolt Hole - Aluminum	3.5	29.9
633-RW001 and 633-RW002	Wing Pivot Attachment 5. Laminated Wing Pivot Attach Lower Lug - 10 Ni (HY 180) 6. Laminated Wing Pivot Attach Lower Lug - Aluminum 7. Laminated Wing Pivot Taper-Lok Hole - Aluminum 8. Laminated Wing Pivot Taper-Lok Hole - 10 Ni (HY 180)	5.0 4.5 3.7 4.8	131 27.4 29.2 150
633-RW002	Laminated Lwr Skin, Canted Spars 9. Spar Web to Cap Attach Bolt Hole - Aluminum	3.5	29.9
633-RW003	Adhesive Bonded Gr/Ep and Titanium Composite Wing Pivot Attachment 10. Wing Pivot Attachment Lwr Surf Skin Bolt Hole - Titanium	2.4	82

* The fatigue damage is ≤ 1.0 for a scatter factor of 4.0.



Note:
 o 2024-T851 Aluminum Control Points

Figure 82 633-RW000 Baseline Wing Control Points 1, 2 & 3



Note:

o 2024-T8511 Aluminum
Control Point

Spar Web to Cap Attach Bolt
Hole - Control Point 4

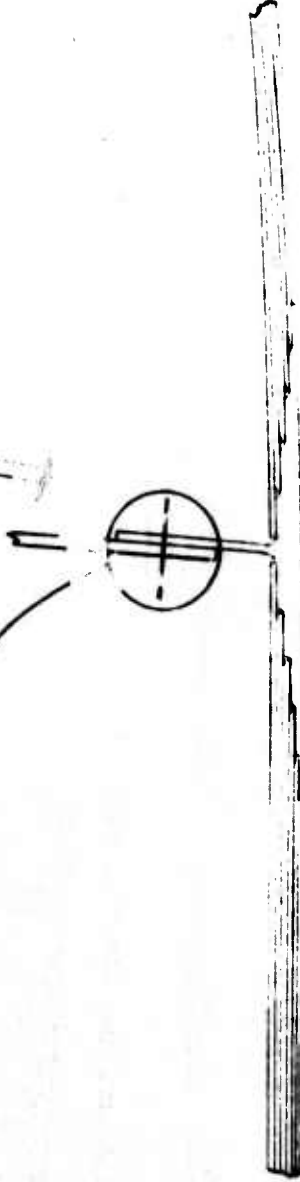
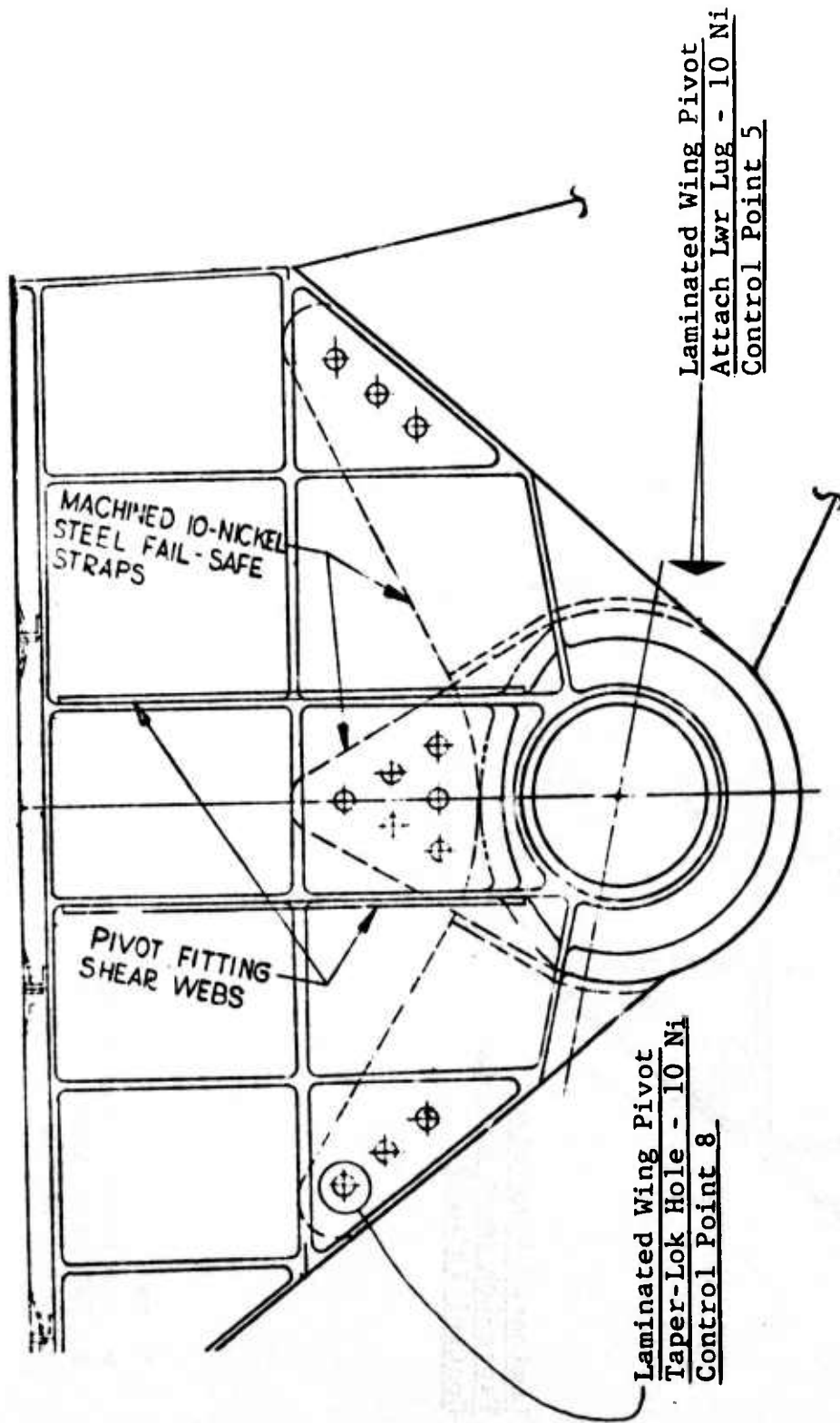


Figure 83 633RW001 Preliminary Wing Box Design -
Control Point 4



- Notes:
- o Wing pivot strap
 - o 10 Nickel Steel Control Points
 - o One inch taper-lok hole

Figure 84 633-RW001 and 633-RW002 Wing Pivot Attachment - Control Points 5 & 8

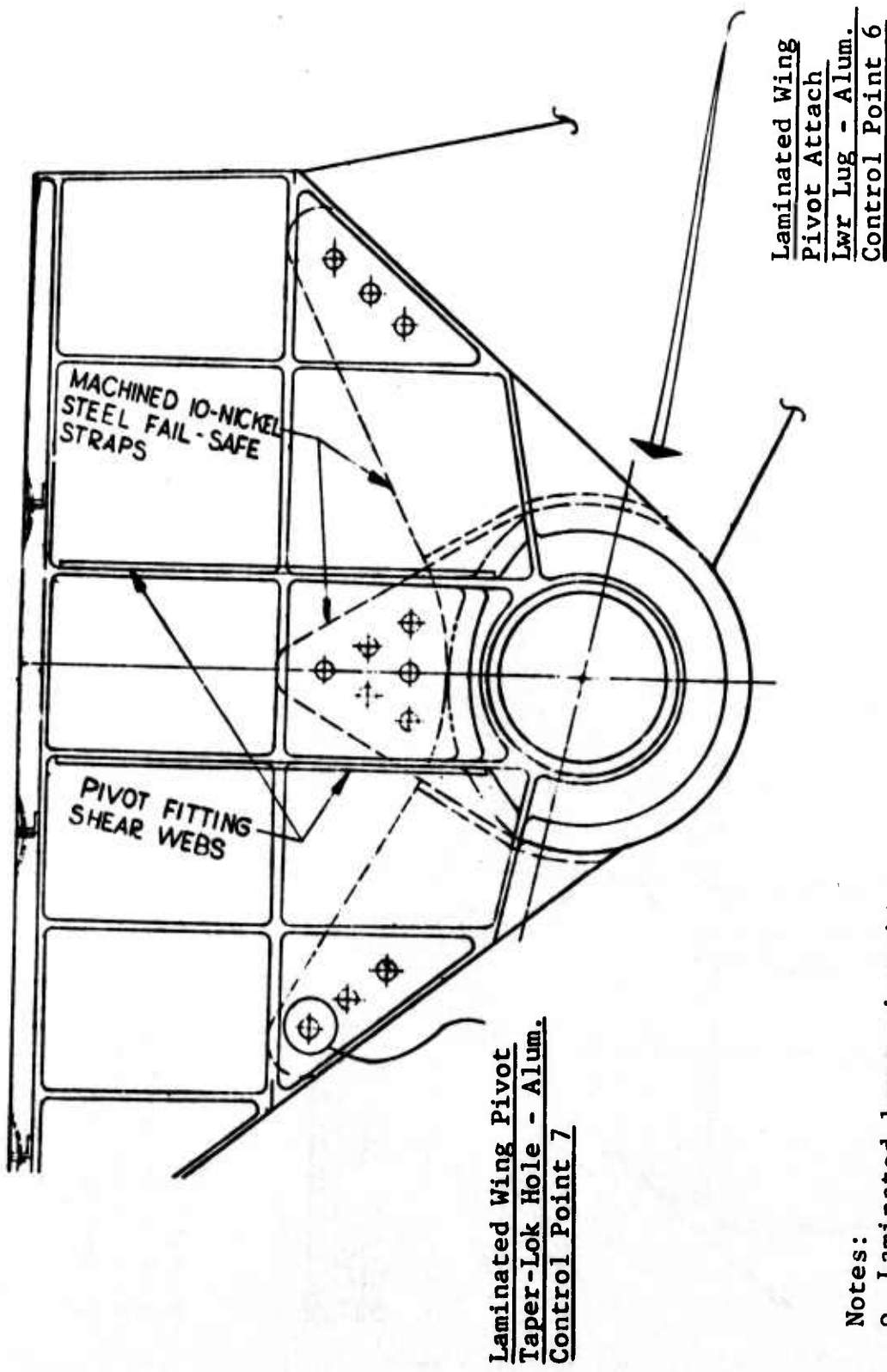
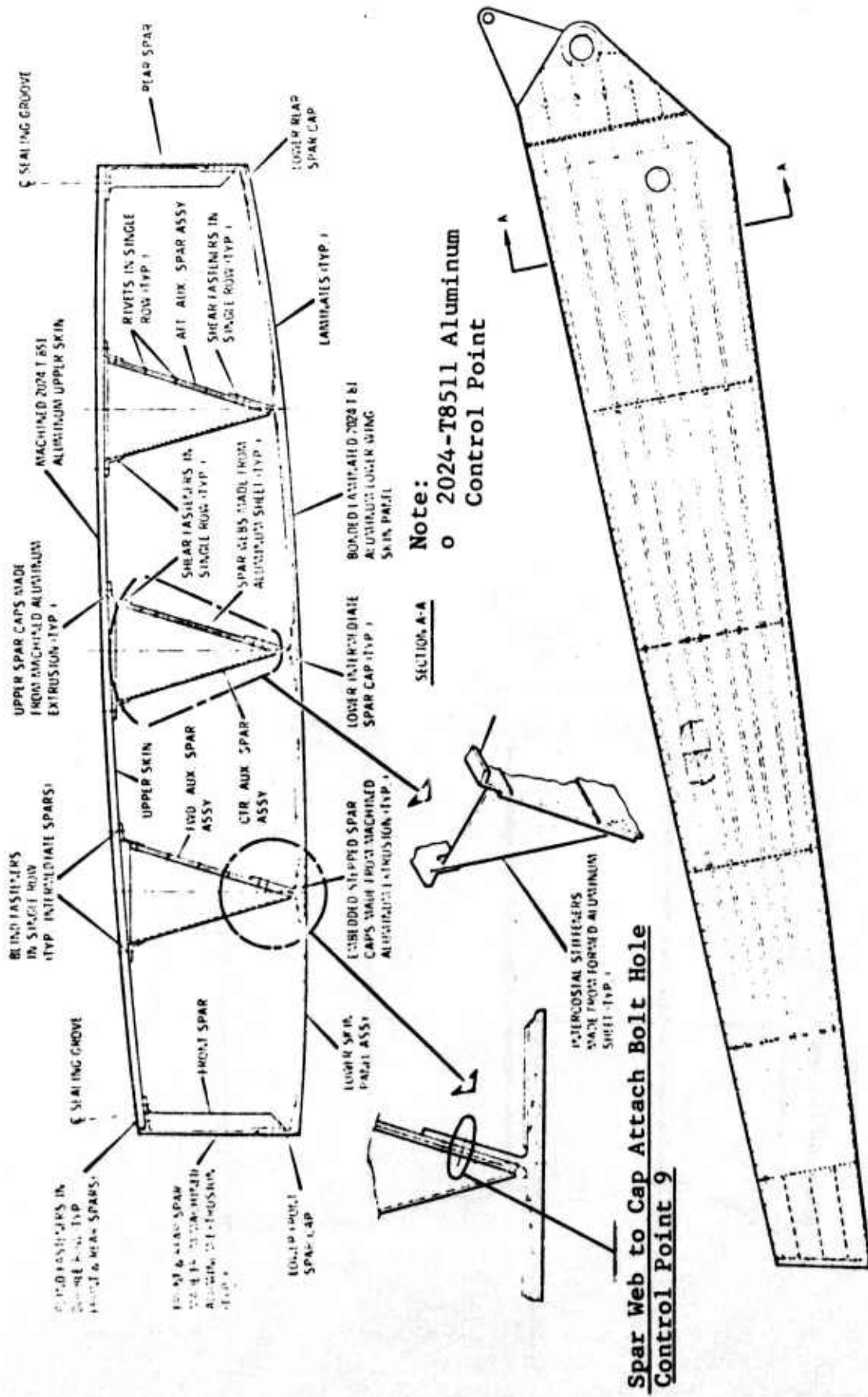


Figure 85 633-RW001 and 633-RW002 Wing Pivot Attachment - Control Points 6 & 7



Note:
 o 2024-T8511 Aluminum
 Control Point

SECTION A-A

Spar Web to Cap Attach Bolt Hole
 Control Point 9

Figure 86 633-RW002 Preliminary Wing Box Design Control Point 9

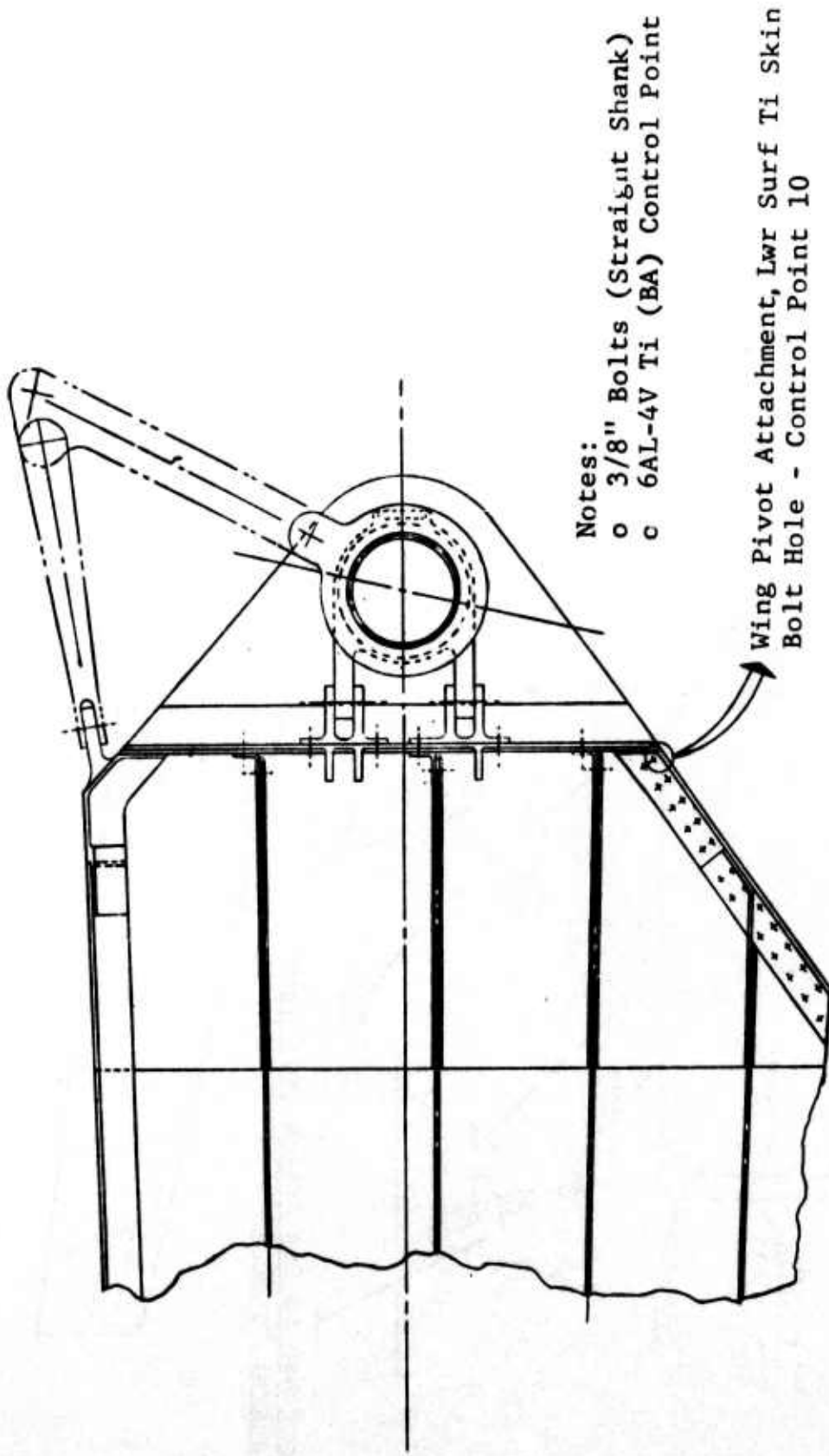


Figure 87 633RW003 Wing Pivot Attachment Control Point 10

Stress concentrations (K_T) were determined using the guidelines discussed in paragraph 5.2.2. The best way to evaluate stress concentration effects is by spectrum testing of components or full scale hardware. This type of test establishes a fatigue quality parameter, K_f , which reflects fretting, residual stress, fabrication, installation quality, etc. that may increase the theoretical stress concentration based on the structural geometry at the control point. In lieu of tests results, the K_T 's assigned for this preliminary wing design program are thought to be conservative but within the bounds of good engineering judgement.

5.2.1 Fatigue S-N Data

Notched fatigue S-N data was available for 2024-T851 and for 10 Ni Steel, but no S-N data was available for 2024-T81/-T8511. The following table shows a summary of the S-N data used in the four ATW-4 wing box designs:

TABLE XVII S-N DATA SUMMARY

Material	S-N Data Used for Analysis
2024-T851 2024-T81 2024-T8511	2024-T851 Aluminum, 2" Pl, Long. Grain, $K_T=2, 3, 4,$ and 5 available from F-111 fatigue programs.
10 Ni Steel (HY 180)	10 Nickel Steel, 0.5" Pl, Longitudinal Grain, $K_T=1.0, 2.4$ and 5 available from AMAVS program.
6AL-4V Ti (Beta Annealed)	6AL-4V Titanium (Beta Annealed), 0.59" Pl, Longitudinal Grain, $K_T=1.0, 2.4,$ and 5.0 available from AMAVS program.

5.2.2 Stress Concentration Factors

The prime objectives of this program have been cost and weight reduction. The 2024-T81 sheet material is cost effective. The laminated structure allows a considerable weight reduction. Except for the wing pivot fitting, generally the laminates were bonded thus eliminating the need for bolts thru the lower surface of the wing box. Consequently, there are no stress concentrations for bolt holes in most of the lower wing surface. However, in the baseline wing box structure of F-111 construction, bolts were used to attach the wing skins to the spars and to the wing pivot fitting. These bolts caused stress concentrations resulting in lowering the

allowable fatigue stress. Also, the wing lower surface fuel pump cutout was eliminated thus removing another area of stress concentration. Both bolted and laminated types of construction were used and evaluated during this preliminary design program.

Stress concentration factors for bolted load transfer joints were calculated using a procedure taken from Metal Fatigue by Sines and Waisman. This procedure accounts for the ΔK_T resulting from loaded fasteners as shown in Figure 88.

Using Sines and Waisman's technique, a K_T of 2.9 was calculated with the following formula for the stress concentration at a taper-lok in the baseline wing box lower surface skin.

$$K_T = K'_T + .75 \left(\frac{f_{br}}{f_{ten}} \right)$$

$$K_T = 1.9 + .75 \left(\frac{40}{30} \right)$$

$$K_T = 2.9$$

The $K'_T = 1.9$ is based on F-111 fatigue developmental tests for an unloaded taper-lok bolt. However for the baseline wing box lower surface skin, a K_T of 3.2 was used based on a F-111 fatigue test failure.

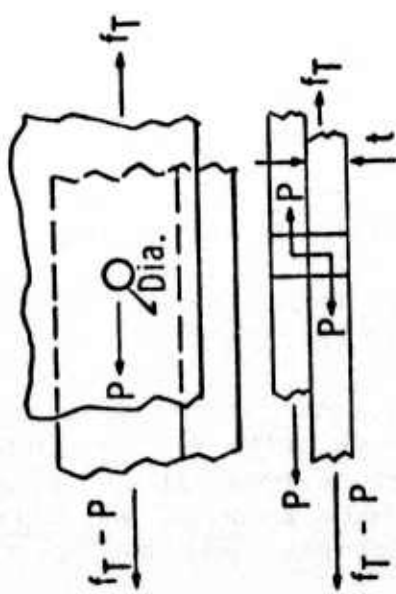
A K_T of 3.2 was also used for the stress concentration at a taper-lok hole in the splice of the wing box lower surface to the wing pivot fitting lower surface. This K_T was chosen because a K_T of 3.2 was calculated from the F-111 fatigue test failure at a taper-lok hole in a splice at this same area of the wing and with similar geometry.

A K_T of 4.5 was used for the wing pivot fitting lower lug based on a 3/8 scale spectrum loaded fatigue test specimen of similar geometry evaluated during the F-111 fatigue development test program.

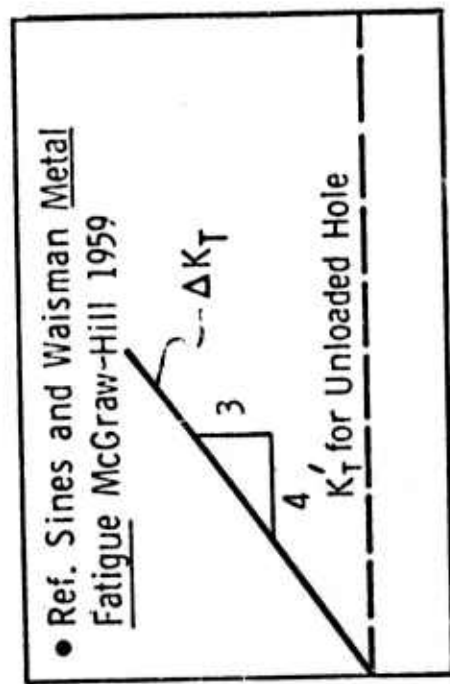
For the splicing of the lower spar cap to the center spar web of the laminated structure, a K_T of 3.5 was used. This was calculated by using the Sines and Waisman's technique and by using Stress Concentration Design Factors by R. E. Peterson.

Wing designs other than the baseline and the Graphite-Epoxy (Gr/Ep) composite designs use taper-lok bolts in the wing pivot attachment. These taper-lok bolts cause local stress concentrations at the holes. The Sines and Waisman's technical and Stress Concentration Design Factors by R. E. Peterson were used to determine

• SINGLE SHEAR BOLT LOADS



f_T = Plate Stress
 P = Bolt Load



• Ref. Sines and Waisman Metal Fatigue McGraw-Hill 1959

Total K_T

• TOTAL $K_T = K'_T$ FOR UNLOADED HOLE + ΔK_T FOR LOADED HOLE EFFECT
 WHERE $\Delta K_T = .75 \left(\frac{\text{Bearing Stress}}{\text{Net Tension Stress}} \right) = .75 \left(\frac{\text{Bolt Load} / 8 \text{ Bolt Dia.} \times \text{Plate Thickness}}{f_T, \text{ Net Section}} \right)$

- K'_T Unloaded Holes
- Interference Fit Fasteners - $K_f = 1.9$ (BASED ON F-111 FATIGUE DEVELOPMENT TESTS)
- Non-Interference Fit Fasteners - $K_T = 3.0$ (OPEN HOLE)

Figure 88 Loaded Fastener Stress Concentration Factor

the stress concentration at these bolt holes in the laminated structure. A K_T of 3.7 was calculated for a hole in the aluminum portion of the structure. A K_T of 4.8 was calculated for a hole in the steel portion of the structure.

The Gr/Ep composite wing design used two plies of 6AL-4V Ti in the wing pivot attachment. The two plies of titanium are bolted and adhesively bonded to the Gr/Ep composite. These straight shank bolts cause local stress concentrations at the bolt holes. A stress concentration factor of 2.4 in the titanium was determined by using the Sines and Waisman technique with Stress Concentration Design Factors by R. E. Peterson. The bearing stress in the titanium was determined to be relatively low resulting in a small K_T . Thus the K_T of 2.4 was determined. Consequently, taper-lok bolts were deemed not necessary to reduce the stress concentration.

Generally the Gr/Ep composite is not fatigue and fracture critical when good engineering design is practiced; therefore, no fatigue and fracture allowables were developed for this material for the preliminary wing design.

S E C T I O N V I

D A M A G E T O L E R A N C E O F A D V A N C E D T R A N S O N I C W I N G (A T W - 4)

Fracture mechanics was utilized throughout this program as the primary technology in providing damage tolerant design. Fracture analyses were performed for the purposes of (1) developing fracture design allowable data and (2) verifying that the final selected designs meet the specified damage tolerance criteria. The detail damage tolerance requirements utilized for this program are specified in MIL-A-83444.

6.1 D A M A G E T O L E R A N C E C R I T E R I A

Damage tolerance requirements are specified in MIL-A-83444 for slow crack growth structure (monolithic structure) and two types of fail safe structure. In this program the structure was classified as slow crack growth or multiple load path-fail safe structure. These requirements basically specify initial flaw sizes, residual strength loads, and flaw growth limits. A non-inspectable category was chosen as the basis for sizing the wing box structure. One design service life is 1280 flights.

6.2 D A M A G E T O L E R A N C E E V A L U A T I O N

Fracture design allowable curves were initially prepared for surface flaws and bolt hole flaws in the materials considered for design to aid in the sizing of structural parts. The emphasis was on those readily available materials which would be most cost effective and which possess relatively good fracture properties.

Preliminary analyses were prepared for the wing box lower surface parts which appear to be fracture critical. For example, fracture critical parts are those whose failure would cause loss of the aircraft or severe operating penalties. Analyses were also prepared for points in the wing pivot attachment area for both the baseline structure and a viable alternate for the baseline wing pivot attachment area. This analysis reflects the preliminary part thicknesses, fastener sizes, and stress levels. Maximum principle stresses were calculated for areas of biaxial loading.

The cyclic loads spectra used for crack propagation analyses was identical to that used in the fatigue analyses. This spectrum is shown in Table XV. Flaw growth was calculated in 10 flight block increments. Loads experienced once per 10 flights or once per 100 flights were applied in the proper sequence.

6.2.1 Fracture Design Allowables

Fracture design allowable curves are included in Appendix (C) and were developed for each material and flaw type as shown schematically in Figure 89. The following four steps are a detailed explanation of the analytical technique used to develop the fracture design allowable stress.

1. For an anticipated flaw type and thickness, crack growth analyses were performed to establish a series of crack growth curves representing a range of stress levels.
2. From step 1 the maximum initial flaw size was determined which would permit one interval of growth (one or two lifetimes) as a function of maximum stress in the spectrum. The interval of growth selected was a function of the specified period of unrepaired service usage for the applicable degree of inspectability. The specified period of unrepaired service usage for the noninspectable multiple load path-fail safe structure is one design lifetime. The specified period of unrepaired service usage for the noninspectable slow crack growth structure is two design service lives.
3. The allowable spectrum stress was plotted as a function of initial flaw size.
4. The allowable spectrum stress level was determined from step 3 in accordance with the initial flaw size and period of unrepaired service usage requirements specified in MIL-A-83444.

Procedures and assumptions used in the fracture analysis effort are discussed in paragraphs 6.2.2 through 6.2.4. The basic fracture data utilized for analysis are discussed in paragraph 6.2.6.

6.2.2 Flaw Growth Model

The basic flaw growth model used for fracture analysis is described as follows:

$$a_n = a_0 + \sum_{n=1}^N (C_p) \quad (da/dN) = f(\Delta K) \quad \text{where}$$

a_n = Crack length after "n" load applications

PRELIMINARY FRACTURE DATA DEFINED FOR EACH MATERIAL CANDIDATE

- o da/dN , K_{Ic} , K_{Isc} , K_{Isc}
- o AVAILABLE DATA FROM LITERATURE OR PRIOR TEST PROGRAMS

GENERATE CRACK GROWTH ANALYSES FOR EACH CANDIDATE MATERIAL

- o SURFACE AND BOLT HOLE FLAWS
- o WHEELER RETARDATION MODEL
- o THERMAL & CHEMICAL ENVIR. ACCT. FOR
- o RANGE OF MAX. SPECTRUM DESIGN STRESSES, f_x

CRITICAL FLAW SIZE AT RESIDUAL STRENGTH LOAD

FLAW SIZE, IN.

NUMBER OF FLTS.

DEVELOP FRACTURE ALLOWABLE CURVES FOR EACH FLAW TYPE & MAT'L CANDIDATE

ALLOW. MAX. SPECTRUM DESIGN STRESS, KSI

INITIAL FLAW SIZE, IN.

2 LIFETIMES (SCG)

1 LIFETIME (FAIL SAFE)

Figure 89 Fracture Allowable Approach

a_0 = Initial crack length

C_p = Spectrum interaction parameter that reflects retardation of flaw growth

da/dN = Rate of crack growth

ΔK - Range of stress intensity during a load cycle

The spectrum retardation parameter, C_p , is the basis for the Wheeler model developed for use during the F-111 Recovery Program. The Wheeler model is intended to account for spectrum interaction through empirical correlation to establish a retardation exponent "m". The value of "m" is varied repetitively until the analysis produces a crack growth curve which forms a lower "m" bound of spectrum environmental test results.

The value of "m" used for this program was zero because determining "m" was not within the scope of this task. Also, an "m" value of zero eliminates the C_p parameter from the calculated flaw growth which yields conservative crack growth rates (shorter time to critical crack length).

6.2.3 Stress Intensity and Flaw Growth

Stress intensity expressions used for fracture analyses are shown in Figure 6-2. These expressions include equations for stress intensity and crack lengths for various types of flaws. The secant correction was used to account for finite width when required.

The bolt hole models account for geometric stress concentration at the edge of the hole but do not account for effects of the fastener system. In the GKT model, the stress concentration (K_T) assumed for the hole was identical to that used for the fatigue analysis. However, provision was made to handle the cases where stress concentration times the maximum spectrum or operating stress (MOS) exceeds material yield stress. The maximum value of GKT (see Figure 90) was then defined as the ratio (σ_{ys}/σ_{max}). This definition is based on the reasoning that peak stresses are limited to yield strength of the material because of plastic flow.

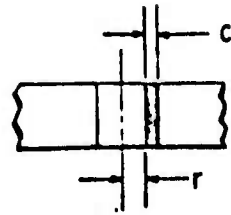
The semicircular shape was assumed for analyses involving part-through surface flaws and corner flaws. An $a/2c=0.5$ and a flaw shape parameter ($Q=2.46$) was used for the fracture analyses.

Backface correction factors (M_K) for part-through flaws were used and were based on curves developed by Boeing Aircraft Corporation.

• BOLT HOLES (Bowie Model)

$$K = \sigma \sqrt{\pi c} F(c/r)$$

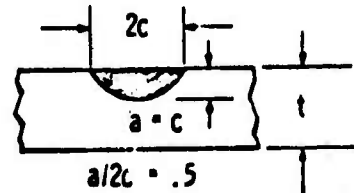
$$C = \frac{K^2}{\pi \sigma^2 [F(c/r)]^2}$$



• SURFACE FLAW (Part through)

$$K = M_K \frac{1.1 \sigma \sqrt{\pi a}}{\sqrt{\phi^2 - .212 (\sigma/\sigma_y)^2}}$$

$$a = \frac{K^2 [2.46 - .212 (\sigma/\sigma_y)^2]}{1.21 \pi \sigma^2 M_K^2}$$



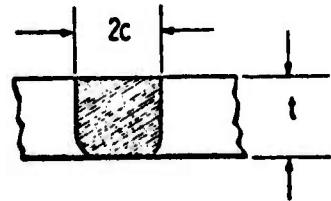
• SURFACE FLAW (through the Thickness)

$$K = \sigma \sqrt{w \tan \left(\frac{\pi a}{w} + \frac{K^2}{2w \sigma_y^2} \right)}$$

DERIVED (FOR $w \geq 6''$)

$$2c = \frac{1}{\pi} \left(2.0 - \frac{\sigma^2}{\sigma_y^2} \right) \left(\frac{K}{\sigma} \right)^2 \quad \text{Plane Stress}$$

$$2c = \frac{1}{\pi} \left(2.0 - \frac{\sigma^2}{3\sigma_y^2} \right) \left(\frac{K}{\sigma} \right)^2 \quad \text{Plane Strain}$$



• SEMICIRCULAR CORNER CRACK

$$K = \frac{1.2 \sigma \sqrt{\pi a}}{\phi} (GKT)$$

$$a = \frac{K^2 \phi^2}{1.44 \pi \sigma^2 (GKT)^2}$$

$$GKT = G_{min} + (G_{max} - G_{min}) \text{Exp} \left[\text{Ln}(.01) \frac{a}{\text{DIA}} \right]$$

$$G_{min} = 1.0, \quad G_{max} = K_T$$

For $K_T \times \sigma_{max} \geq \sigma_{ys}$, $G_{max} = \sigma_{ys} / \sigma_{max}$ By Definition

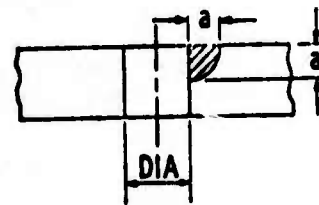
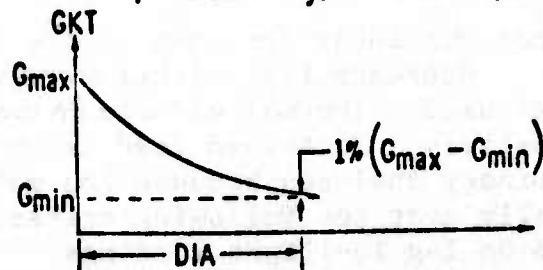


Figure 90 Stress Intensity Expressions

Flaw growth calculations were made using a General Dynamics developed IBM 370 computer program identified as TR9. This program produces results almost identical to those produced by the AFFDL-TR-70-107 CRACKS program. TR9 was used because it involves somewhat less computer run time for a typical analysis.

6.2.4 Initial Flaw Sizes

The initial flaw sizes and types are given specifically in MIL-A-83444. For this preliminary design analysis, the degree of inspectability was assumed to be In-Service Non-inspectable Structure. The following is a brief summary of the initial flaw sizes and types for noninspectable structure:

1. Slow crack growth (monolithic) structure
 - (a) The initial crack length, $2c_0$, for a surface flaw is assumed to be 0.25 inches in length.
 - (b) The initial crack length, c_0 , for a cracked hole is assumed to be 0.05 inches in length.
2. Multiple load path-fail safe structure
 - (a) The initial crack length, $2c_0$, for a surface flaw is assumed to be 0.10 inches in length.
 - (b) The initial crack length, c_0 , for a cracked hole is assumed to be 0.02 inches in length.

6.2.5 Material Fracture Data

The basic fracture data required for a comprehensive fracture analysis should include the following data for each material utilized:

1. Fracture Toughness - K_{IC} , K_C , K_{ISCC}
2. Crack Growth Data - da/dN , da/dt

Crack growth data was not available for some of the desired thicknesses and environments. Consequently, either conservative or representative da/dN data was used. Thermal effects on crack growth were neglected for the analyses. Sustained load crack growth was neglected for the preliminary analyses because the metallic materials selected will usually meet the following criteria when sustained stresses are based on 1-g in-flight loadings.

$$\frac{\text{Max. 1-g Sustained Tensile Stress}}{\text{Max. Operating Tensile Stress}} < \frac{K_{ISCC}}{K_{IC}}$$

However, additional data must be developed before ascertaining the importance of sustained load crack growth on the Graphite-Epoxy (Gr/Ep) composite materials.

The following is a summary of the fracture toughness data assumed for the preliminary design analyses:

Material	K_{IC} , ksi (in.) ^{1/2}	K_C , ksi (in.) ^{1/2}
2024-T851 Alum.	22.2	35.4
2024-T81 Alum.	60	60 for 1 ply 170 for 4 ply
2024-T8511 Alum.	22.2	35.4
10 Nickel Steel	160	160
6AL-4V Titanium (Beta Annealed)	80	193

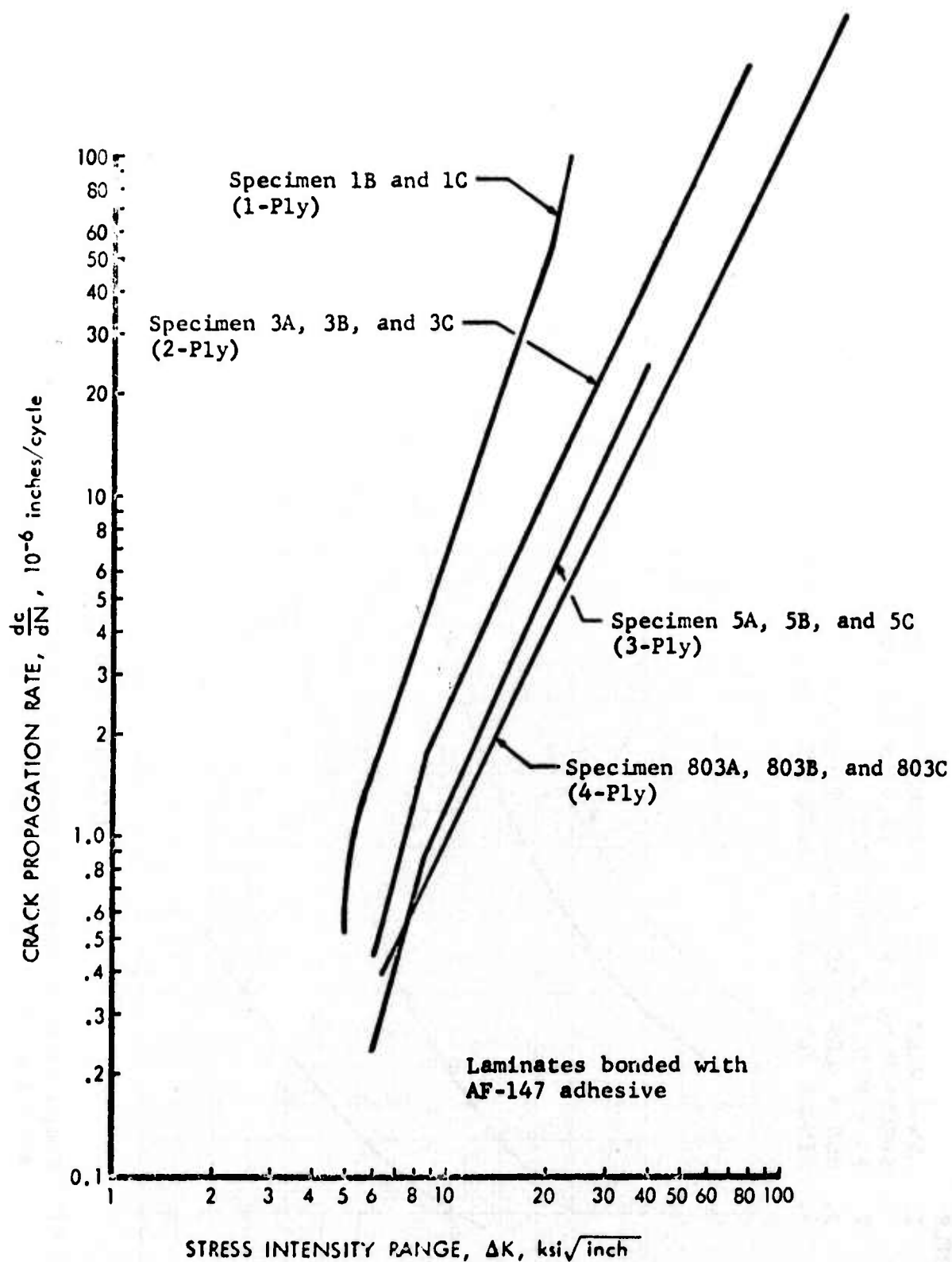
The toughness values shown in this summary represent values in the L-T direction. All values are typical of the raw stock sizes for the forms specified for the more critical parts evaluated such as the wing box lower surface.

Flaw propagation data (da/dN) used for the analyses is summarized in Table XVIII. When flaw growth data was not available for the desired material thickness, the growth data from a thicker part of the same type material was used for a conservative analysis. In Research Report ERR-FW-1584, flaw growth data is available from small test specimens of 4 ply bonded laminates made from 2024-T81 aluminum. Where applicable the decrease in flaw growth rate due to the bond lines in laminated panels was used to greatly increase the allowable fracture stress. Figure 91 shows a plot comparing flaw growth data for 1, 2, 3, and 4 ply bonded laminates of 2024-T81 aluminum. Figure 92 compares the fracture design allowables stress for sheet, plate, and 4 ply bonded laminates made from 2024 aluminum. Figure 93 compares the structural life of 2024 aluminum sheet, plate, and 4 ply bonded laminates.

TABLE XVIII
 PRELIMINARY DESIGN ANALYSIS
 FLAW GROWTH DATA SUMMARY

Material	(dc/dN) or (da/dN) Data Assumed
2024-T851 Aluminum Plate	AFML-TR-66-291 CNT Data, R = 0.33, 310 cpm, RT Air, Foreman Eq.
2024-T81 Aluminum Laminated Sheet	ERR-FW-1584 Data, Center Cracked Outer Ply, 4 Ply Laminate, Adhesive Bonded, R = 0.1, RT Air, Paris Eq.
2024-T8511 Aluminum Extrusion	AFML-TR-66-291 CNT Data, R = 0.33, 310 cpm, RT Air, Foreman Eq.
10 Nickel Steel	AFFDL-TR-74-17 CT Data, R = 0.1, 60 cpm, STW, Foreman Eq.
6AL-4V Titanium (Beta Annealed)	AFFDL Damage Tolerant Design Handbook MCIC-HB-01, CT Data, R = 0.5, 70F, STW, 6 cpm, L-T Direction, Foreman Eq.

CNT - Center-notch tensile specimen
 RT - Room temperature
 CT - Compact specimen
 STW - Sump tank water



Fatigue Crack Propagation of 1-Ply, 2-Ply, 3-Ply, and 4-Ply Laminates of Adhesive Bonded 2024-T81 Aluminum, .100 Inch Sheet, R = 0.1, Dry Air, L-T Direction

Figure 91

NOTES:

1. ATW-4 WING PIVOT BM SPECTRUM FOR WING LOWER SURFACE. MAXIMUM WING SPECTRUM BM @ PIVOT = 16.42×10^6 IN-LBS.
2. FS = FAIL SAFE STRUCTURE.
3. SCG = SLOW CRACK GROWTH STRUCTURE.
4. SINGLE PLY OR SHEET THICKNESS = 0.090; PLEATE THICKNESS = 0.912.

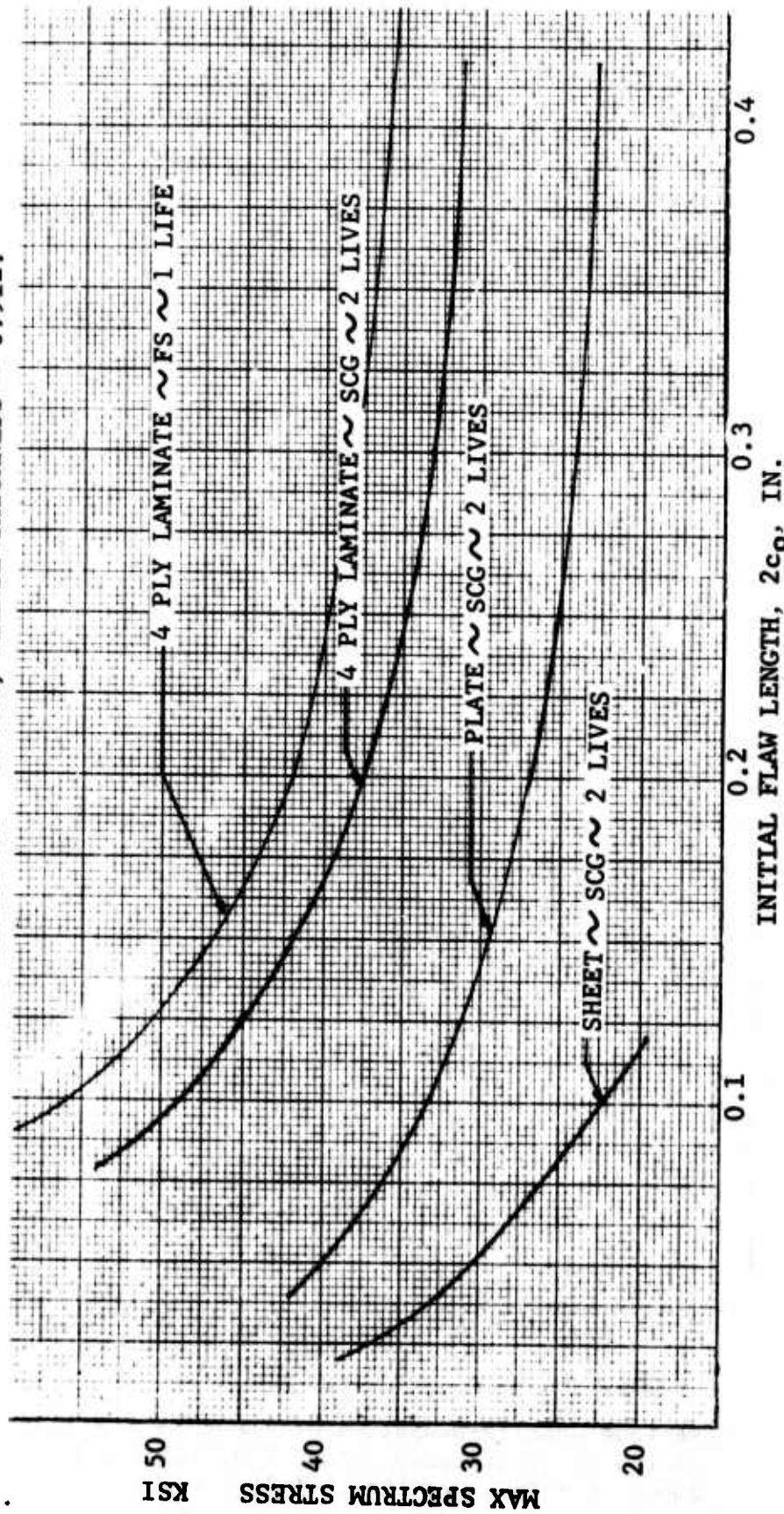
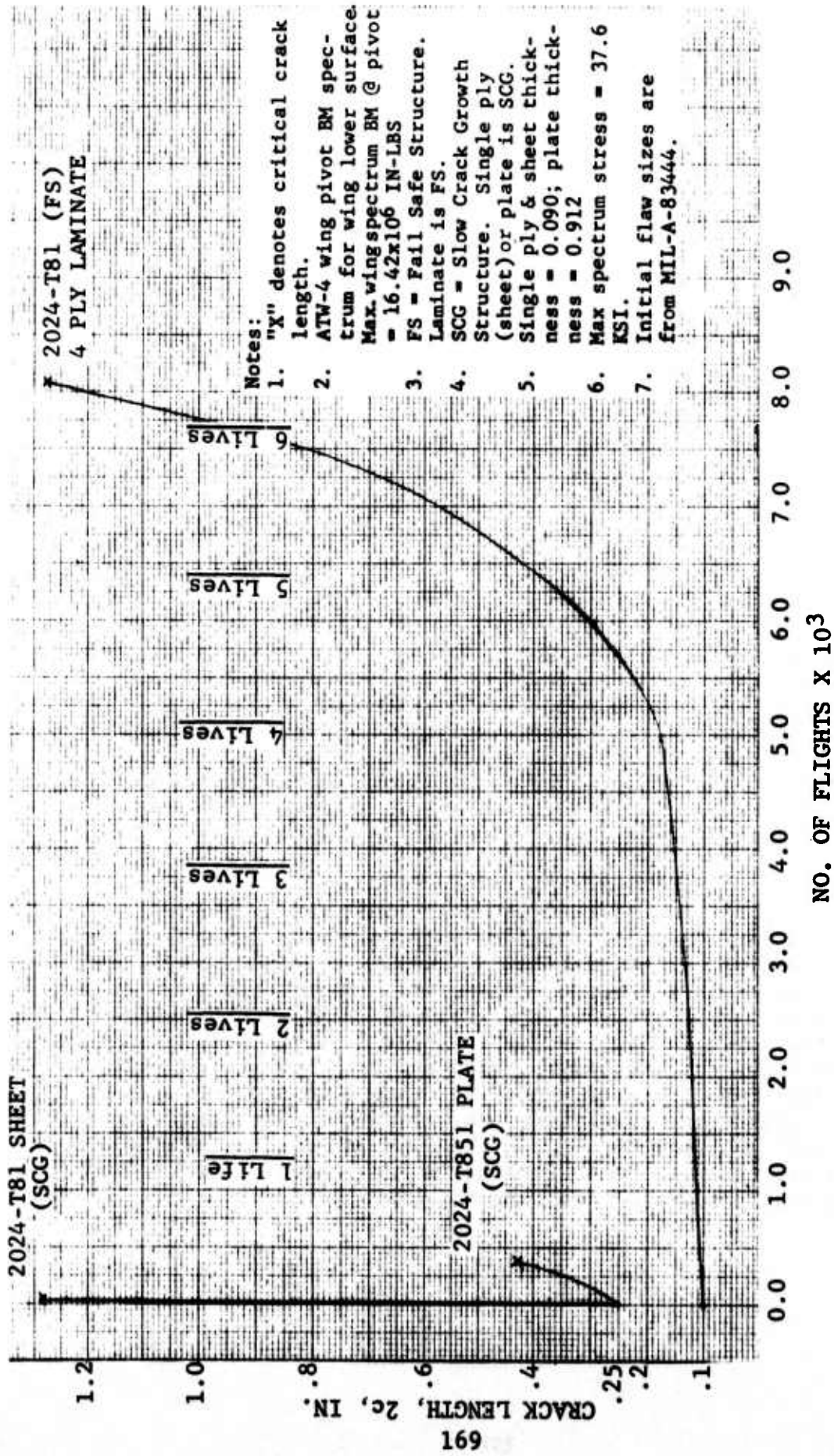


Figure 92 Comparison of Fracture Design Allowables Sheet vs. Plate vs. Laminate Noninspectable Structure 2024 Aluminum Surface Flaw



- Notes:
1. "x" denotes critical crack length.
 2. ATW-4 wing pivot BM spectrum for wing lower surface. Max. wingspectrum BM @ pivot = 16.42x10⁶ IN-LBS
 3. FS = Fail Safe Structure. Laminate is FS.
 4. SCG = Slow Crack Growth Structure. Single ply (sheet) or plate is SCG.
 5. Single ply & sheet thickness = 0.090; plate thickness = 0.912
 6. Max spectrum stress = 37.6 KSI.
 7. Initial flaw sizes are from MIL-A-83444.

Figure 93 Comparison of Structural Life Sheet vs. Plate vs. Laminate Noninspectable Structure 2024 Aluminum Surface Flaw

6.2.6 Residual Strength Load Determination

The residual strength load requirement specified in MIL-A-83444 for noninspectable structure is denoted as P_{LT} , which is the maximum one time load occurring in 20 lifetimes as determined from average load exceedance data.

Wing pivot bending moment cumulative exceedance plots were constructed from the fatigue loads spectra data shown in Table XV and documented in FZM-12-6466. The exceedance plot was increased by a factor of 20 to develop exceedances for 20 lifetimes as specified in MIL-A-83444 for non-inspectable structure. The two points on the plot for the two largest wing pivot bending moments were used to make a straight line extrapolation of the plot to the one time occurrence level. This exceedance plot is shown in Figure 6-6. Using the extrapolated data for the 20 lifetime occurrences, the residual strength load requirement, P_{LT} , is 23.5×10^6 in.-lbs. of wing pivot net bending moment. The design limit wing pivot net bending moment is 18.3×10^6 in.-lbs. MIL-A-83444 states that if P_{LT} is greater than the design limit load for one lifetime, then the residual strength load need not be greater than 1.2 times the maximum operating load for one design lifetime. Therefore, P_{LT} for these fracture analyses was determined to be $1.2 (16.4 \times 10^6) = 19.68 \times 10^6$ in.-lbs. of wing net bending moment instead of the 23.5×10^6 in.-lbs. determined from the exceedance plot.

6.2.7 Preliminary Wing Design Analysis

Preliminary fracture analyses sufficient to evaluate the baseline wing box, two metallic wing boxes, and one composite wing box are presented in this section. All four wing boxes were designed to conform to the MIL-A-83444 damage tolerance requirements for in-service non-inspectable structure. Even though some of the structure is inspectable, all the analyses were accomplished as if the structure was In-Service Non-inspectable Structure. For each of the analyses the fracture allowable stress for the maximum spectrum bending moment was developed to conform to the residual strength requirement specified in MIL-A-83444. The four wing box designs are the baseline box and the final three configurations selected according to their ranking in the preceding task. Generally the analyses are of the wing box lower surface.

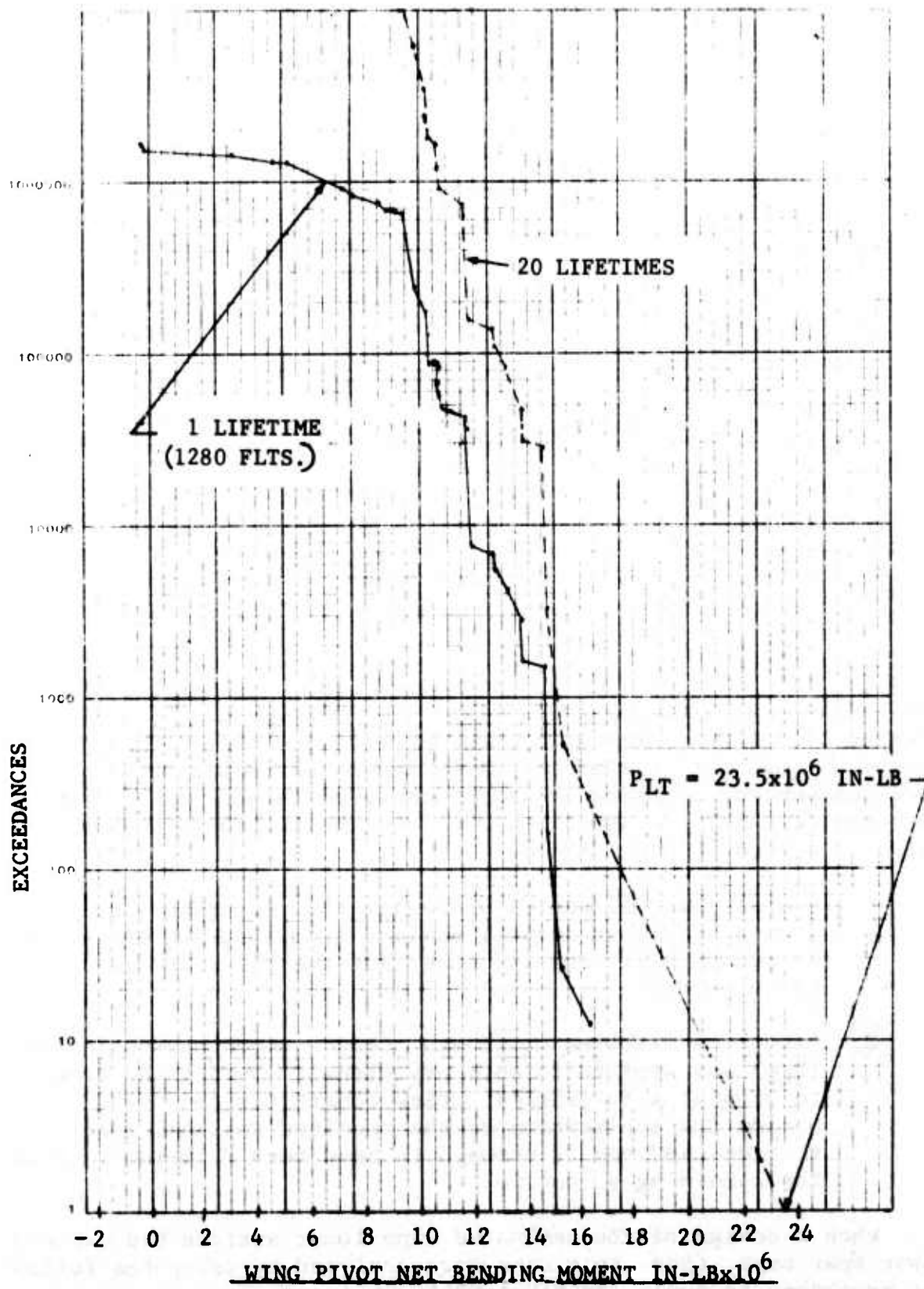


Figure 94 Cumulative Frequency Distribution of Wing Pivot Net Bending Moment

6.2.7.1 Baseline Wing Box (633-RW000)

The baseline wing box, 633-RW000, shown in Figures 95 and 96 was classified as monolithic structure because the whole lower surface skin is one piece construction in the chordwise direction. Failure of the wing lower surface skin would cause loss of the aircraft. Consequently, this lower skin was required to sustain two design service lives without the specified initial flaw sizes attaining critical crack size. Corner flaws and surface flaws were both analyzed for the lower surface of the baseline wing box.

6.2.7.2 Wing Box with Laminated Skin and Exposed Spar Cap (633-RW001)

The ATW-4 wing box design with the laminated lower surface skin and with the exposed lower spar caps as shown in Figure 97 and 98 was classified as multiple load path-fail safe structure. Crack arrest capability was provided in this design by ending each ply at the adjacent spar; hence, a chordwise crack in a ply would not propagate into or past the adjacent ply or spar during the specified inspection interval because of the inability of the crack to propagate through the bonding material. For a similar reason, a crack in the spar cap would propagate until reaching the bond line and the stop.

When a design of the laminated wing lower surface had exposed lower spar caps, the laminated lower wing surface panels were treated as multiple load path structures, i.e., each ply was considered a load path within the laminated panel, and each ply was made independent by using bonding techniques rather than fasteners. Then the critical ply was sized according to the following criteria which is stated in MIL-A-83444.

1. A crack was allowed to propagate for one design lifetime from a specified initial surface flaw. During the crack propagation for one lifetime, all the plies are assumed to be bonded.
2. After one lifetime of growth, the required residual load (PLT) was applied to this ply without causing the crack to propagate to critical crack length for this ply. During the application of the required residual load, this cracked ply is assumed to have been delaminated from the remaining structure.

When a design of the laminated wing lower surface had exposed lower spar caps, these spar caps were analyzed by using the following procedure to apply the MIL-A-83444 criteria.

Notes:

- o Lower wing skin
- o Surface flaw
- o Slow crack growth structure
- o 2024-T851 Aluminum
- o Non-inspectable structure
- o $a/2c=0.5$

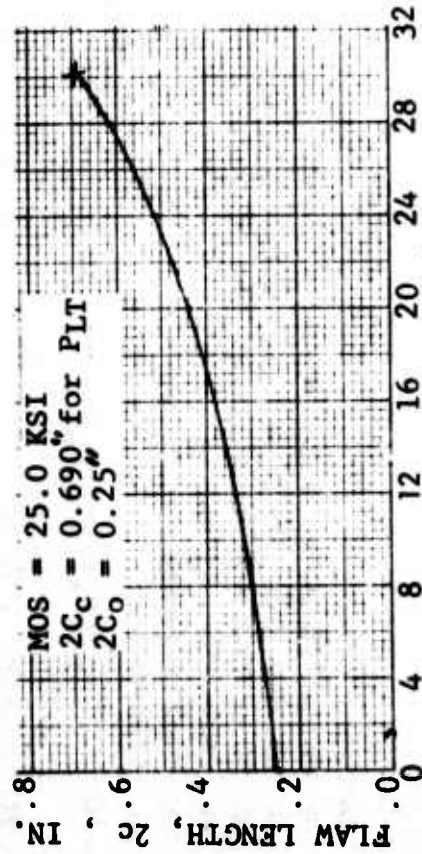
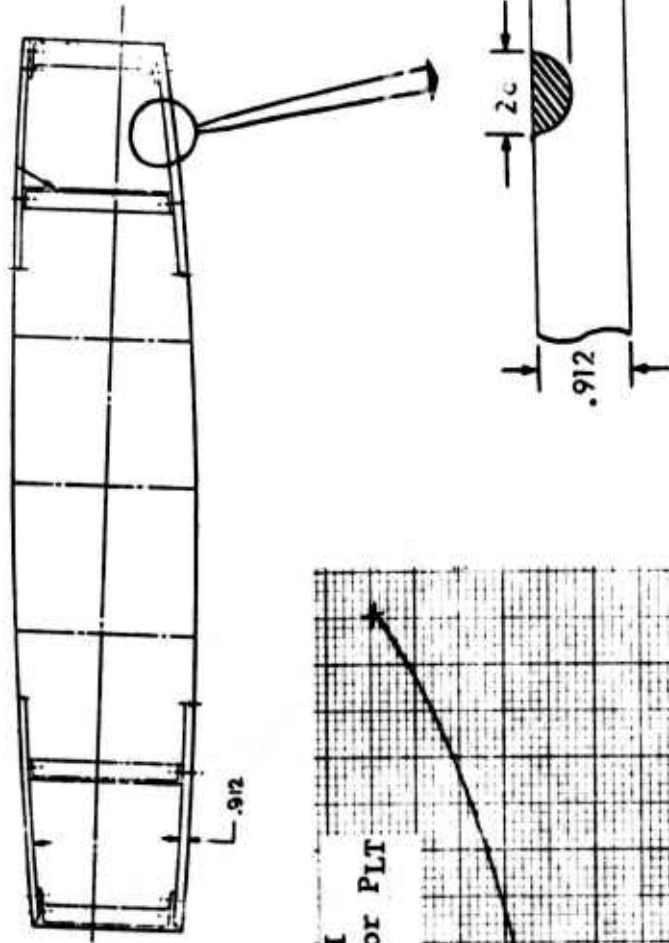


Figure 95 633-RW000 Baseline Wing Box Preliminary Fracture Analysis - Control Point A

- Notes:
- o Monolithic lwr wing skin
 - o Corner flaw in taper lok hole
 - o Slow crack growth structure
 - o 2024-T851 Aluminum
 - o Non-inspectable structure
 - o $a/2c=0.5$

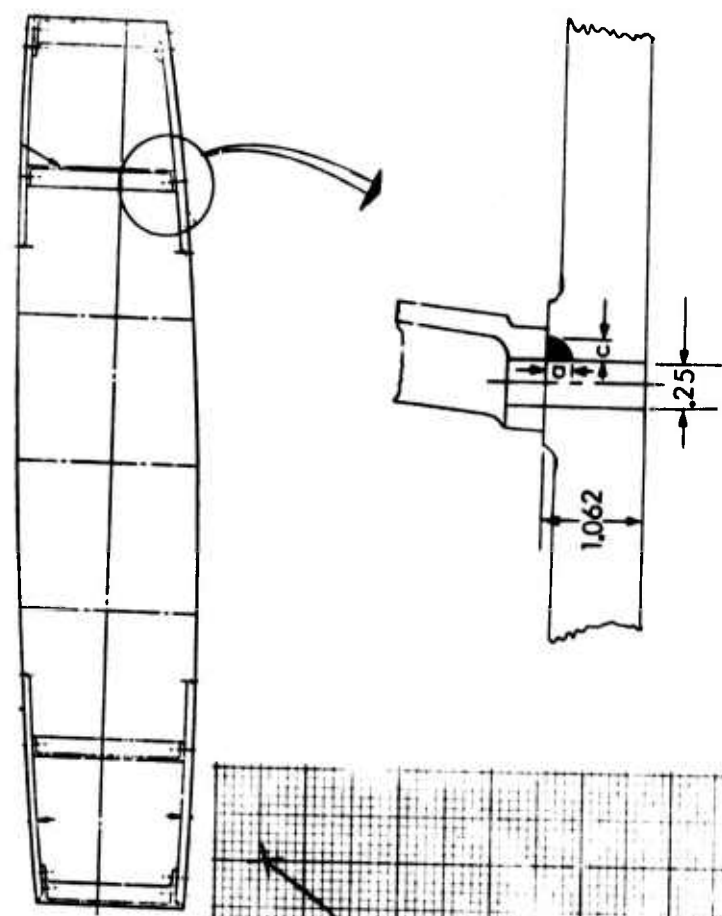
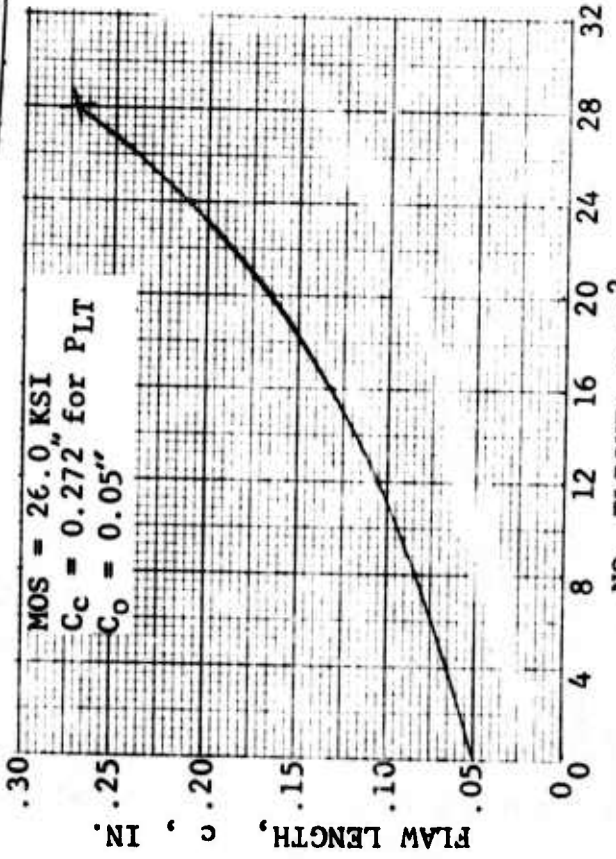
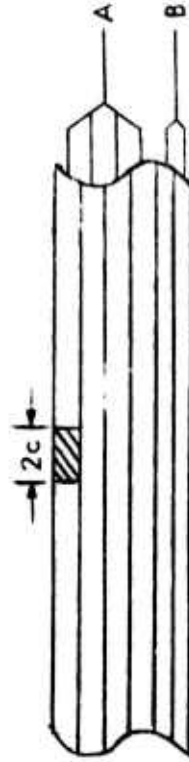
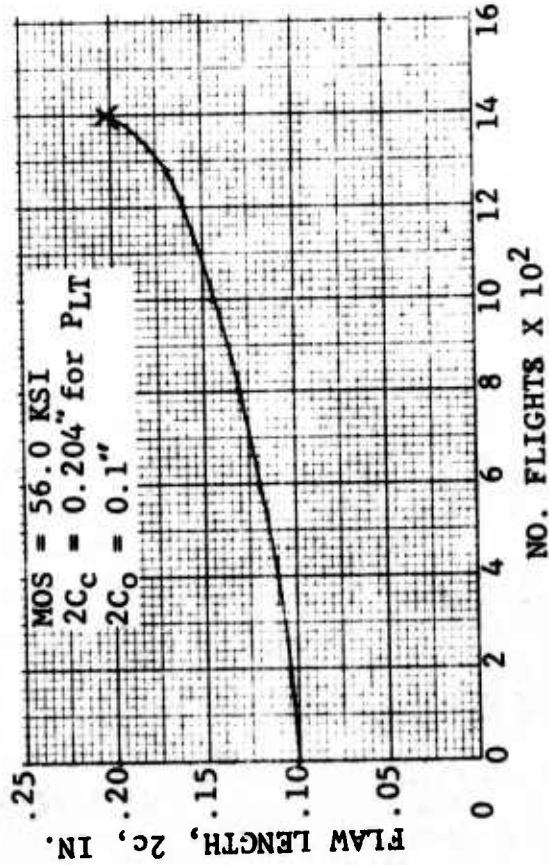
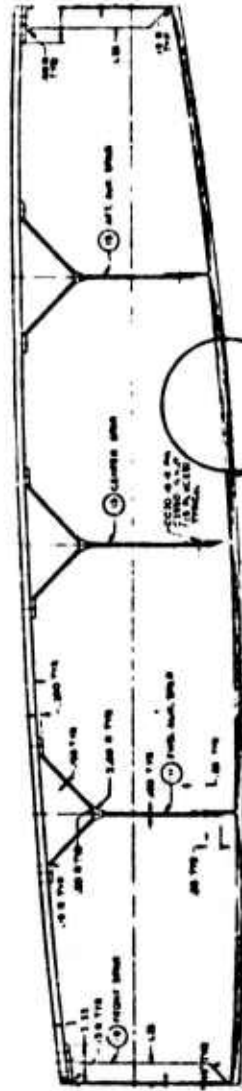


Figure 96 633-RW000 Baseline Wing Box Preliminary Fracture Analysis - Control Point B

- Notes:
- o Laminated lower wing skin
 - o Surface flaw in one ply
 - o Multiple load path-fail safe structure
 - o 2024-T81 Aluminum
 - o Non-inspectable
 - o $a/2c=0.5$



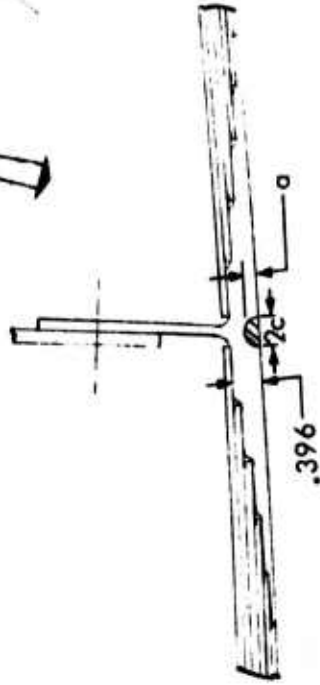
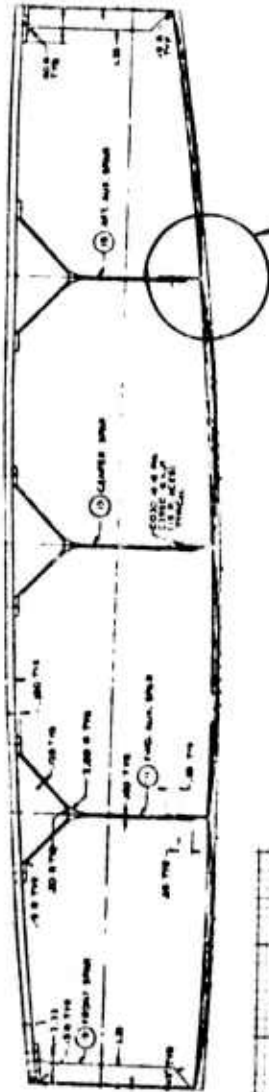
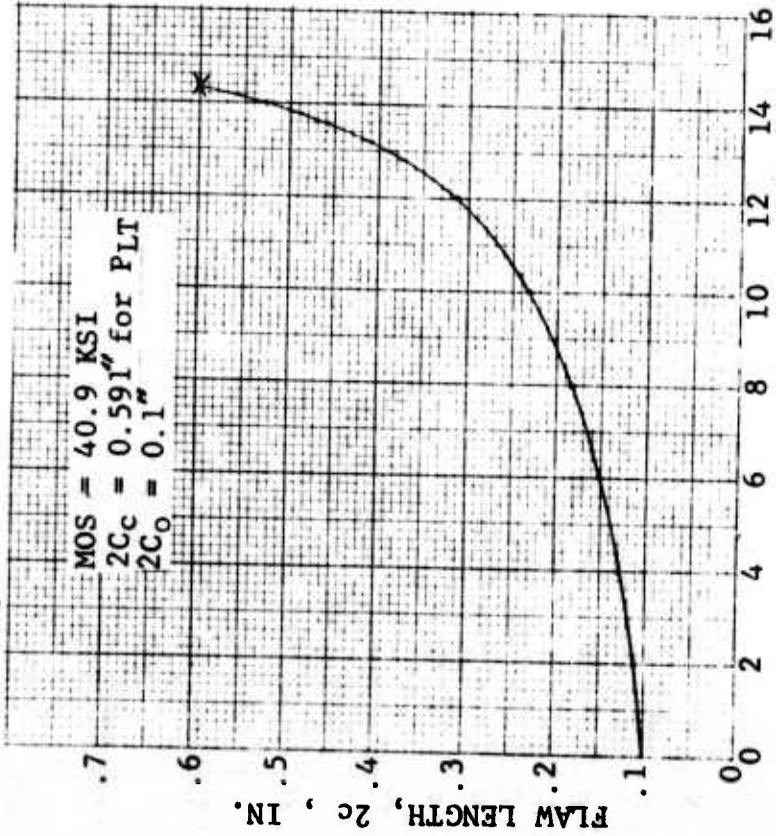
Thickness
 A=0.09 in.
 B=0.063 in.

Figure 97 633-RW001 Wing Box Preliminary Fracture Analysis
 Control Point C

Notes:

- o Lower spar cap
- o Surface flaw
- o Multiple load path - fail safe structure
- o 2024-T8511 Aluminum
- o Non-inspectable structure
- o $a/2c=0.5$

MOS = 40.9 KSI
 $2C_c = 0.591''$ for PLT
 $2C_o = 0.1''$



NO. FLIGHTS X 10²

Figure 98 633-RW001 Wing Box Preliminary Fracture Analysis
 Control Point D

1. Crack propagation data for this type of spar cap was unavailable because of the stepped forward and aft flange of the cap and because of the laminated effect at the forward and aft edges of the cap flange. As shown in Figure 91, the laminated structure (bonded plies) has a slower crack growth rate than the monolithic structure (single ply). The crack growth rate for this exposed spar cap is thought to be between the rate for the laminated plies and the rate for plate material. Therefore, crack propagation data for the plate material was used as a conservative crack propagation rate for this spar cap.
2. Because the spar cap could fail without loss of the aircraft, the spar cap was treated as multiple load path-fail safe structure.
3. Therefore, a specified initial surface flaw was allowed to propagate for one lifetime while assuming the cap was not bonded to the plies.
4. After one lifetime of growth, the required residual load (P_{LT}) was applied to the spar cap without causing the crack to propagate to critical crack length. During the application of the required residual load, this cracked cap was assumed to have been delaminated from the remaining structure.

6.2.7.3 Wing Box with Laminated Skin and Imbedded Spar Cap (633-RW002)

Another ATW-4 wing box design has a laminated lower surface skin with imbedded lower spar caps as shown in Figure 99 and 100. The exterior ply of the lower wing surface is continuous from the front spar to the rear spar. Some of the interior plies terminate at the spar caps. Because loss of the exterior ply would result in loss of the aircraft, the exterior ply of this wing box design was treated as slow crack growth structure. This exterior ply was sized according to the MIL-A-83444 criteria by using the following procedure:

1. A crack was allowed to grow for two design lifetimes from a specified initial through-the-thickness flaw while assuming this ply was bonded to the remaining structure.

Notes:

- o Laminated lower wing skin
- o Surface flaw in one ply
- o Slow crack growth structure
- o 2024-T81 Aluminum
- o Non-inspectable structure
- o $a/2c=0.5$

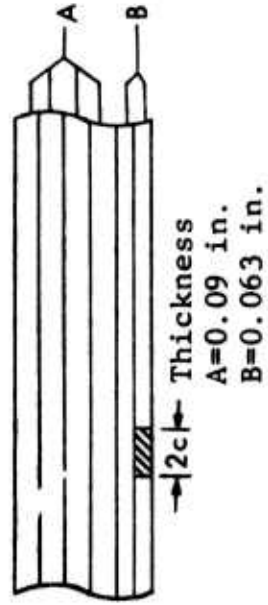
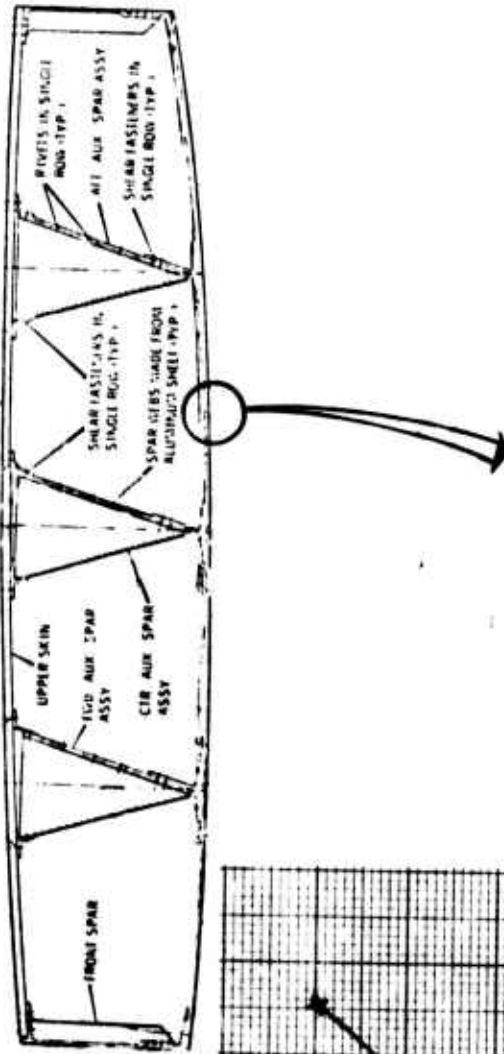
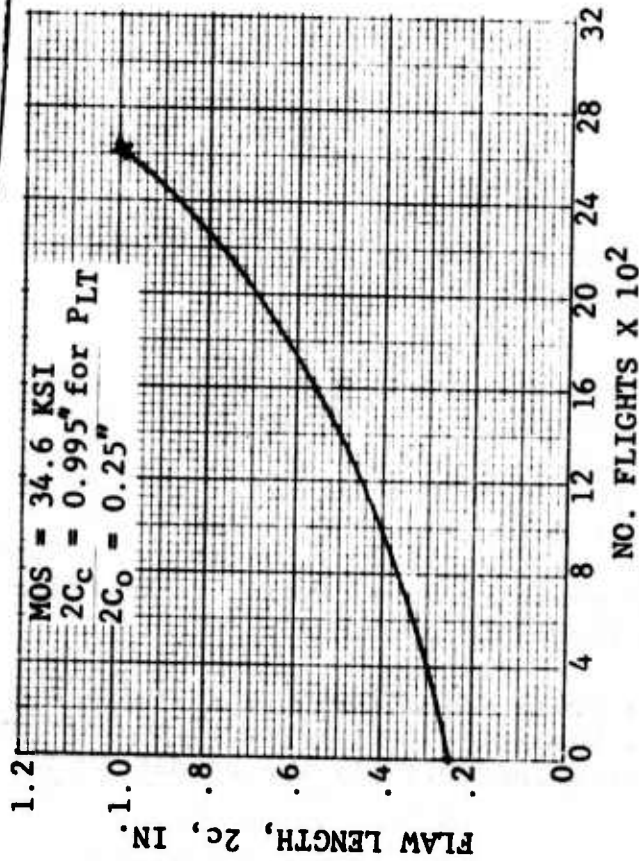
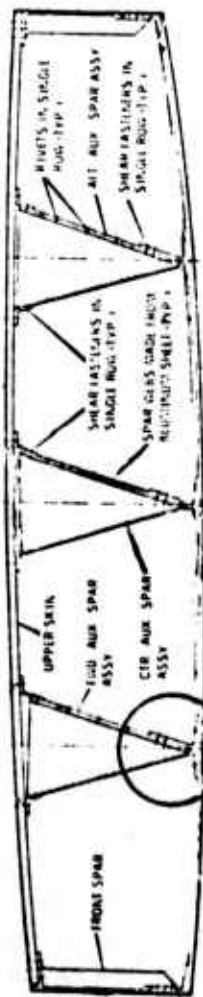


Figure 99 633-RW002 Wing Box Preliminary Fracture Analysis Control Point E



Notes:

- o Wing lower surface spar cap
- o Surface flaw
- o 2024-T8511 aluminum control point
- o Non-inspectable structure
- o Multiple load path-fail safe structure
- o No crack growth analysis was made.
- o The allowable fracture stress was assumed to be the same as the allowable stress in the sheet material bonded to the spar cap. See section 6.2.7.3 for the rationalization.
- o Refer to Figure 6-11 for assumed crack growth rate curve.
- o MOS = 34.6 ksi

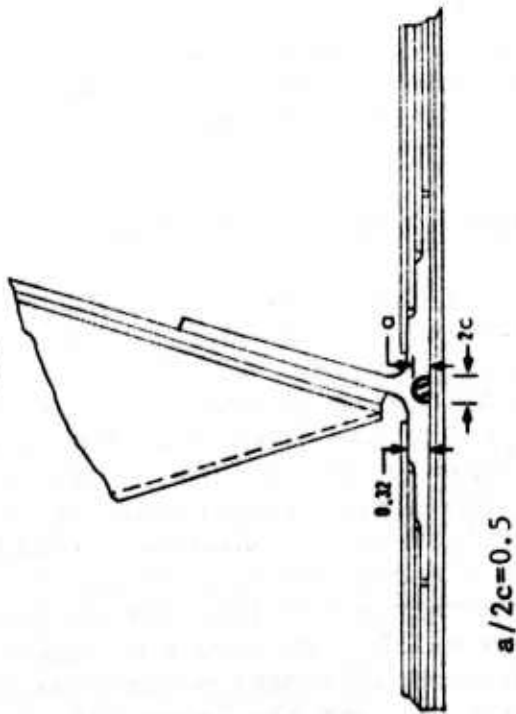


Figure 100 633-RW002 Wing Box Preliminary Fracture Analysis
Control Point F

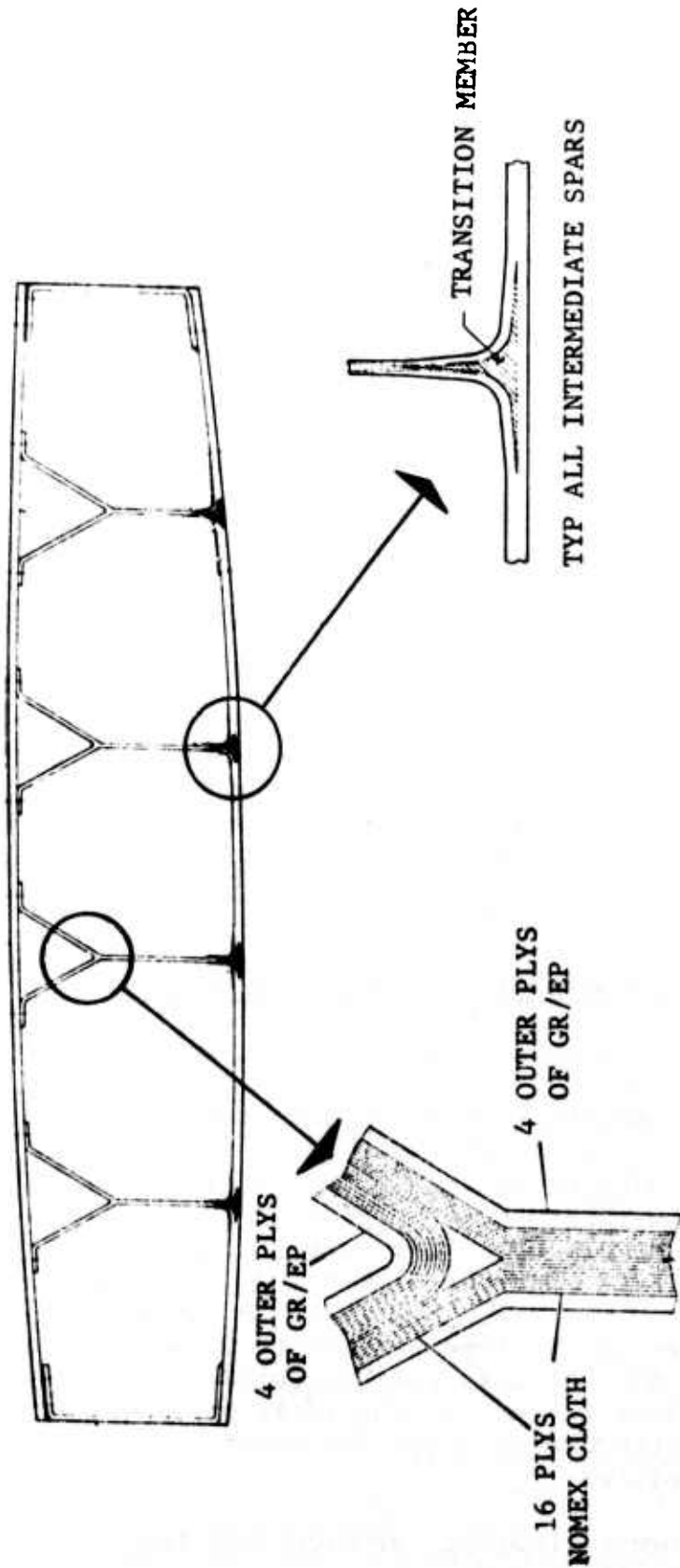
2. After two lifetimes of growth, the required residual load (PLT) was applied to the cracked ply without causing the crack to propagate to critical crack length. During the application of this residual load, this cracked ply was assumed to have been delaminated from the remaining structure.

For the wing box design having a laminated lower surface skin with imbedded lower spar caps as shown in Figures 99 and 100, no fracture analysis of the imbedded spar cap was made because no crack growth data was available. However, a fracture allowable stress was determined using the following rationalization:

1. No crack growth data was available for a stepped fore and aft flange of a spar cap.
2. No crack growth data was available for a spar cap bonded to or in a laminate.
3. However, the crack growth rate in an imbedded cap is thought to be between the crack growth rate for the exposed spar cap and the laminated skin panel.
4. The imbedded cap and adjacent plies of skin must strain at the same rate to remain bonded. Hence, the allowable operating stress is the same in the cap as in the adjacent plies of skin. Consequently the fracture allowable stress for the cap was assumed to be the same as the fracture allowable stress for the exterior ply of the skin panel.

6.2.7.4 Graphite-Epoxy (Gr/Ep) Composite Wing Box (633-RW003)

The ATW-4 Graphite-Epoxy composite wing box design, 633-RW003, as shown in Figure 101 has Gr/Ep skins and imbedded Gr/Ep "Y" spars. The upper wing surface skin is bolted to the spars. The lower wing surface skin is bonded to the precured intermediate spars and bolted to the front and rear spars. Ordinarily the bolts are in Nomex buffer strips, which have only $\pm 45^\circ$ plies, thus reducing the stress concentrations at the bolt hole and prolonging the fatigue life. However, a small amount of test data developed during a current program has indicated the Nomex buffer strips are unnecessary for this design because the bolt bearing stresses are so small. To save cost these buffer strips were omitted. This composite design was classified as slow crack growth structure because the lower skin is one piece construction having no spanwise buffer strips or splices. An assumption has been made that the graphite epoxy wing box will be



- Notes:
- o Gr/Ep Composite Spars & Skins
 - o "Y" Spars with Imbedded Lower Caps
 - o Slow Crack Growth Structure
 - o Non-inspectable Structure
 - o No crack growth analysis was made.
- The fracture allowable stress was assumed to be the same as the static allowable stress because dry Gr/Ep composite structure is generally not fracture critical when good engineering design is practiced. See Section 6.2.7.4 for a discussion.

Figure 101 633-RW003 Wing Box Design

prevented from absorbing moisture, fuel, and etc. by painting all the wing surfaces, installing a fuel bladder inside the wing box, and etc. Thus the Gr/Ep would be kept dry. Generally a Gr/Ep composite exposed to a high humidity or fuel and to an elevated temperature greater than 200°F loses much of its strength for shear and compression loads but retains most of its strength for tension loads. Usually shear loads and compression loads do not cause fatigue or fracture problems. Dry Gr/Ep composite material is very resistant to fatigue and fracture. The static design allowable stress was used for the fatigue and fracture design allowable stress for three reasons: (1) the Gr/Ep composite would be kept dry, (2) shear and compression stresses generally do not cause fatigue or fracture problems, and (3) dry Gr/Ep composite structure is generally not fatigue or fracture critical. Thus no fracture analysis has been made for the preliminary design of the Gr/Ep composite wing box. A detailed fracture analysis will be accomplished for this Gr/Ep composite wing box if this design is selected for the final ATW-4 wing.

6.2.7.5 Baseline Wing Pivot Fitting

The baseline wing design has a wing pivot fitting with similar geometry, similar construction, and the same kind of material as the F-111 design. The lower surface of the wing pivot fitting is classified as slow crack growth structure. An assumption was made to use the same fracture allowable stresses for the baseline wing pivot fitting as was used for the F-111 wing pivot fitting.

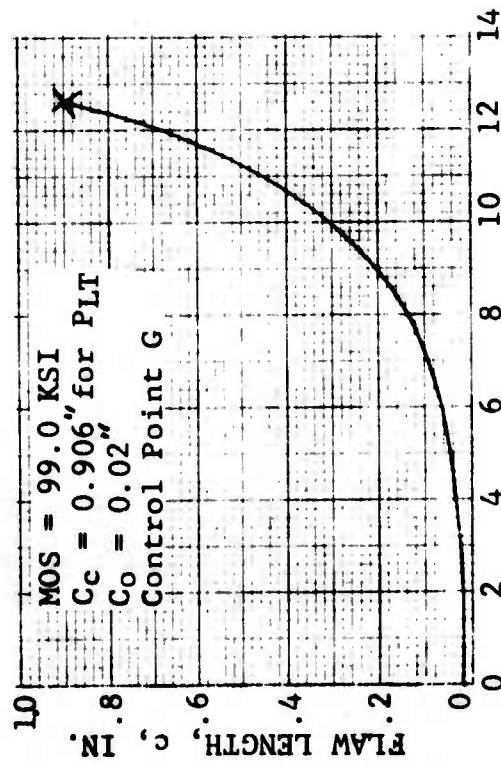
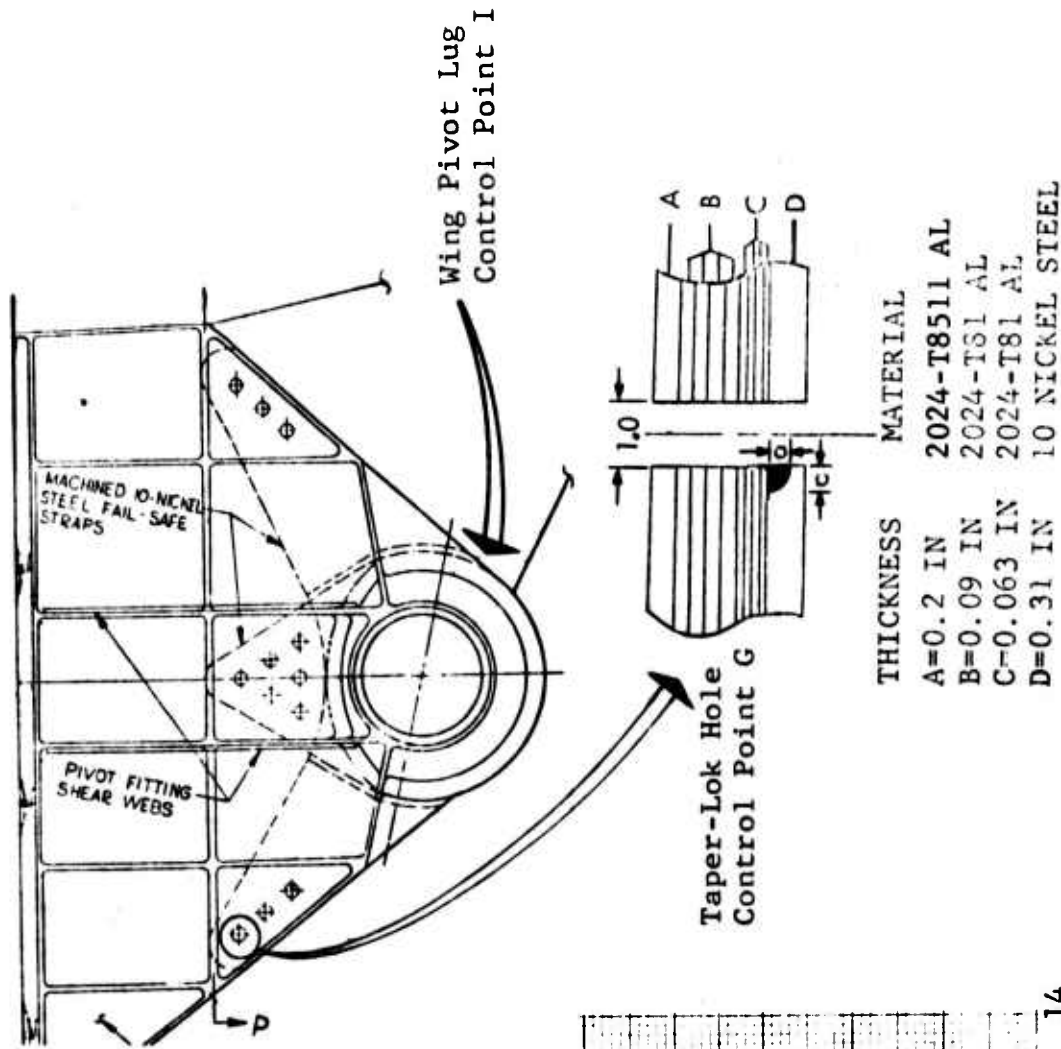
6.2.7.6 Metallic Wing Pivot Attachment (633-RW001 and 633-RW002)

The ATW-4 metallic wing designs have a wing pivot attachment classified as multiple load path-fail safe structure. The wing pivot attachment consists of laminated 2024 aluminum plies bolted to two 10 Ni steel plates by using 1" taper-lok bolts as shown in Figures 102 and 103. This wing pivot attachment is fail safe because the laminated aluminum plies or one of the steel plates can fail without loss of the aircraft. Three control points were chosen for fracture analysis. As shown in Figure 102, one control point is located at the 1" taper-lok bolt hole in the end of the 10 Nickel steel plate that has a shape similar to a boomerang. A second control point is located at this same taper-lok bolt hole but in one of the aluminum plies instead of the steel plate as shown in Figure 103. The procedure used for analyzing both of these control points is as follows:

1. Assume an initial corner flaw, c_0 , of 0.02 inch long per MIL-A-83444 criteria.

Notes:

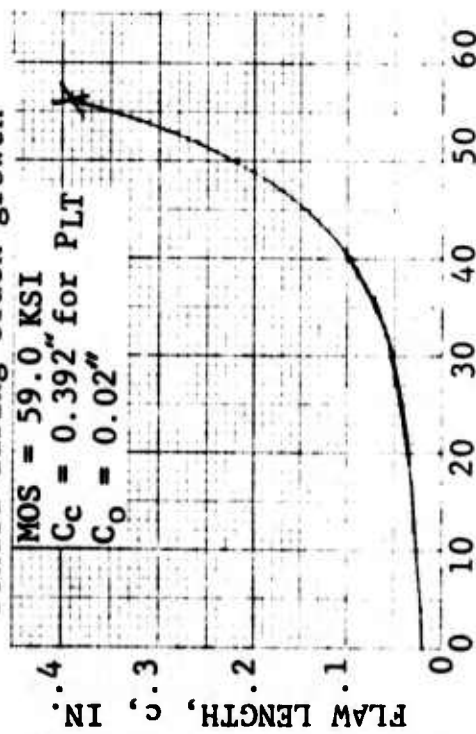
- o Wing pivot strap
- o Corner flaw in taper-lok hole
- o Corner flaw in wing pivot lug
- o Multiple load path-fail safe structure
- o 10 Nickel Steel Control Point
- o One inch taper-lok hole
- o Non-inspectable structure
- o $a/2c=0.5$



NO. FLIGHTS X 10²
 Figure 102 633-RW001 and 633-RW002 Wing Pivot Attachment - Control Points G and I

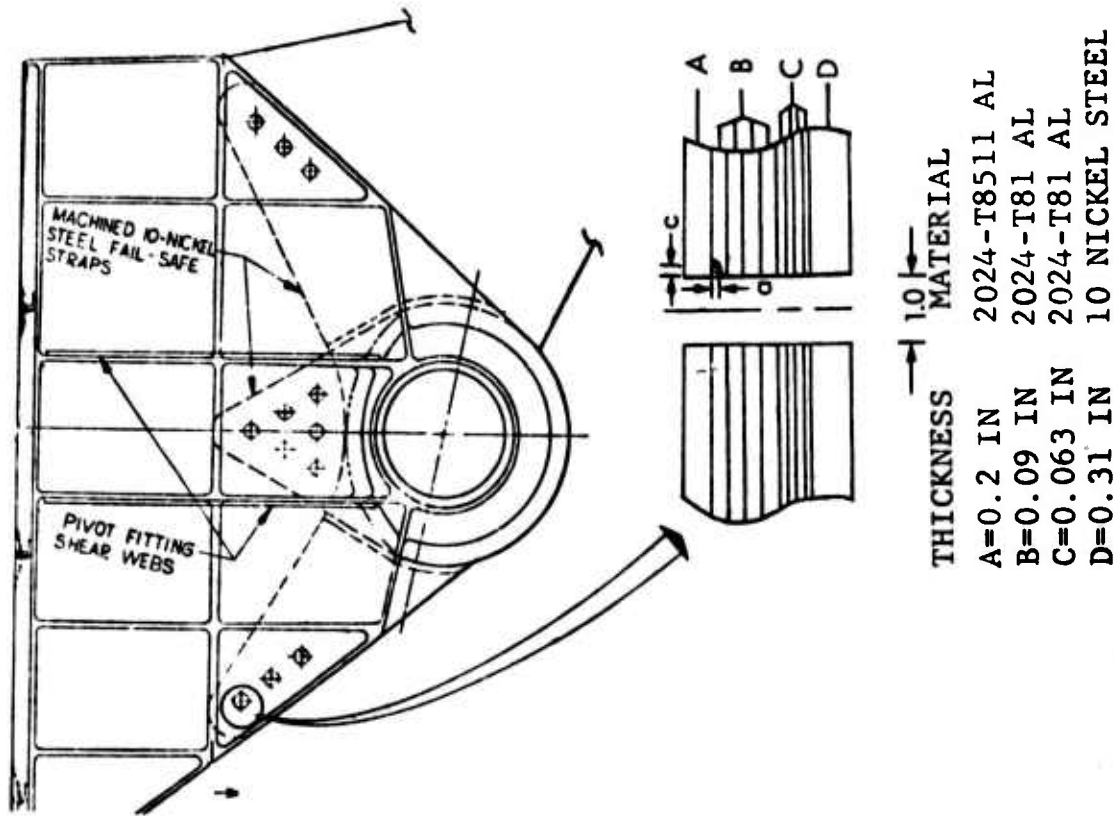
Notes:

- o Laminated lower wing skin
- o Corner flaw in taper-lok hole
- o Multiple load path-fail safe structure
- o 2024-T81 Aluminum Control Point
- o One inch taper-lok hole
- o Non-inspectable structure
- o $a/2c=0.5$
- o Assumed material "A" was not bonded during crack growth



NO. FLIGHTS X 10²

Figure 103 633-RW001 and 633-RW002 Wing Pivot Attachment - Control Point H



2. This initial flaw was allowed to propagate for one design service life.
3. The aluminum laminate was sized such that the crack in the cracked ply would not grow to critical crack length when the MIL-A-83444 required residual load was applied after one design service life. When the residual load was applied, the aluminum ply was assumed not to have been attached to the remaining structure.

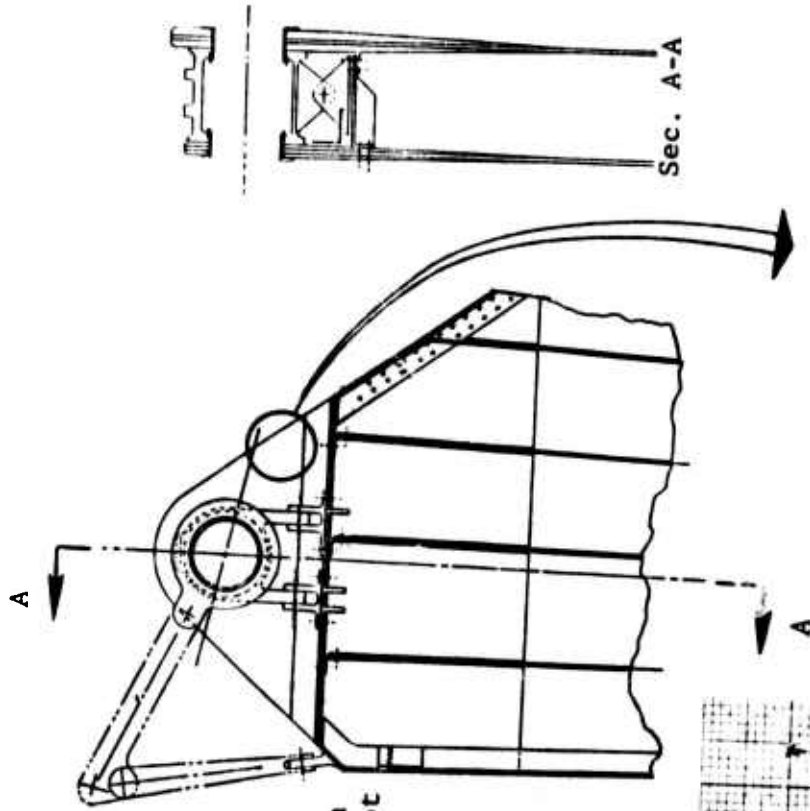
The 10 Nickel steel plate was sized such that the crack in the plate would not propagate to critical crack length when the MIL-A-83444 required residual load was applied.

The third control point is located in the 10 nickel steel portion of the ATW-4 wing pivot attachment lug as shown in Figure 102. For this preliminary fracture analysis, the allowable stress for the wing pivot lug was assumed to be the same value as was used in the 1" taper-lok bolt hole control point in the 10 Nickel steel plate as discussed in the first paragraph of this section. A detailed fracture analysis will be accomplished for this 10 Nickel steel wing pivot attachment lug if this design is selected for the final ATW-4 wing.

6.2.7.7 Graphite-Epoxy (Gr/Ep) Composite Wing Pivot Attachment (633-RW003)

The ATW-4 Gr/Ep composite wing design has a wing pivot attachment that is different from the metallic wing designs. The lower surface of the wing pivot attachment of the Gr/Ep composite wing has Gr/Ep composite sandwiched between two plies of 6AL-4V Titanium (Beta Annealed) as shown in Figure 104. The outboard portion of the titanium is both bolted and adhesively bonded whereas the area near the wing pivot pin is only adhesively bonded. This fracture analysis concerns the area of bonded titanium plate without bolts. This structure is classified as multiple load path-fail safe structure because one ply of 6AL-4V Ti or the Gr/Ep composite could fail without loss of the aircraft. A fracture analysis of the titanium plate was performed using the following procedure:

1. Assumed bonding effects were negligible.
2. Assume an initial surface flaw, $2c_0$, of 0.10 inch long per MIL-A-83444 criteria.



Notes:

- o Adhesive bonded Gr/Ep and Titanium
- o Surface flaw in Titanium Wing Pivot Attachment Lwr. Skin
- o Multiple load path-fail safe structure
- o Ti 6AL-4V (BA) Control Point
- o Non-inspectable structure
- o $a/2c = 0.5$

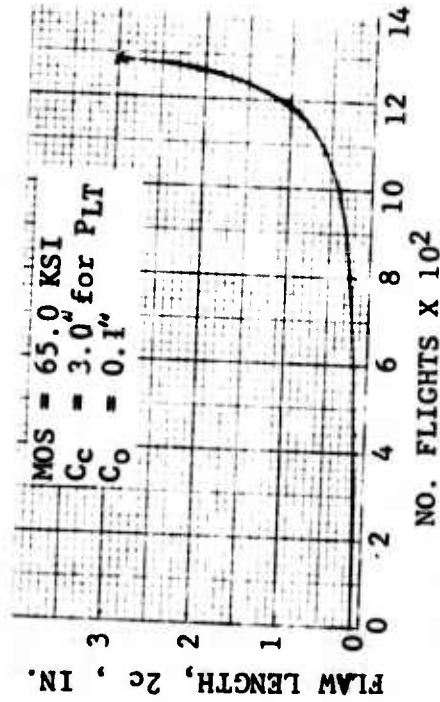


Figure 104 633-RW003 Wing Pivot Attachment Control Point J

3. This initial flaw was allowed to propagate for one design service life.
4. The cracked interior titanium ply was sized such that the crack would not grow to critical crack length when the MIL-A-83444 required residual load was applied after one design service life.

6.3 DISCUSSION

Figures 95 through 104 show the flaw growth analyses for each assumed failure mode. These failure modes are part-through, through the thickness, and bolt hole flaws as shown on the figures. Maximum operating stress (MOS) are indicated and correspond to the maximum spectrum wing pivot bending moment of 16.42×10^6 in.-lbs.

Table XIX presents a summary of the fracture design allowable stresses for the control points in the baseline wing design and the preliminary ATW-4 wing designs.

Appendix C contains the preliminary fracture design allowables data for the baseline and the other wing designs. This data is presented for surface flaws and corner flaws in each of the designs. This data consists of plots of initial flaw length versus the maximum spectrum or operating stress for one or two design lifetimes. The fracture allowable design stress is determined by entering the allowable curve at the initial flaw size specified in MIL-A-83444 and then reading the maximum spectrum stress for the appropriate number of lifetimes. This data is presented for selected control points of surface flaws and corner flaws.

6.4 CONCLUSION

The fracture allowable design stresses are much larger for the bonded laminated structure than for the monolithic structure. As was stated before, all the fracture analyses were done in accordance with the MIL-A-83444 damage tolerance criteria for In-Service Non-Inspectable Structure. The crack growth is much slower for a 4-ply lamination of 2024-T81 sheet material than for a single ply of the same material. A comparison of the flaw growth data for 1, 2, 3, and 4 ply bonded laminates of 2024-T81 aluminum is shown in Figure 91. Where applicable, the decrease in flaw growth rate due to the bond lines in laminated panels was used to greatly increase the fracture allowable stress values. A comparison of fracture design allowable stresses for 2024 aluminum sheet, plate, and bonded laminate is shown in Figure 92. For an assumed initial flaw size in Figure 91, the max. fracture

TABLE XIX

FRACTURE ANALYSIS RESULTS FOR PRELIMINARY WING DESIGNS

CONTROL POINT	WING CONFIGURATION & FLAW DESCRIPTION	TYPE STRUCT.	*MAX. OPERATING TENSILE STRESS ksi
A B	633-RW000 -- Baseline Wing Box, Machined Spars & Skins Part-through surface flaw in the wing lwr skin Corner flaw in lwr skin taper-lok hole at intermediate spar	SCG SCG	25.0 26.0
C D	633-RW001 -- Laminated Lwr Skin, Extruded "y" & Machined Spars Through the thickness surface flaw in singly ply Part-through surface flaw in lwr spar cap	FS FS	56.0 40.9
E F	633-RW002 -- Laminated Lwr Skin, Canted Spars Through the thickness surface flaw in exterior ply Part-through surface flaw in lwr spar cap	SCG FS	34.6 34.6
G H I	633-RW001 and 633-RW002 -- Wing Pivot Attachment, Laminated Lwr Skin Corner flaw in lwr skin taper-lok hole - 10 Nickel Steel Corner flaw in lwr skin taper-lok hole - 2024 Aluminum Corner flaw in wing pivot attachment lug - 10 Nickel Steel	FS FS FS	99.0 59.0 99.0
J	633-RW003 -- Adhesive Bonded Gr/Ep & Titanium Composite Wing Pivot Attachment Surface flaw in titanium wing pivot attachment lwr skin	FS	65.0

*Fatigue Allowable

allowable operating stress is much larger for the 4-ply bonded laminate than for the plate or sheet material. To further illustrate the advantages of bonded laminates for increasing fracture resistance, a comparison of structural life of 2024 aluminum sheet, plate, and 4-ply bonded laminate is shown in Figure 93. For an assumed max. operating stress, the sheet and plate material show less than $\frac{1}{2}$ design service life whereas the 4-ply bonded laminate shows greater than 6 design service lives. Consequently, the bonded laminate designs were controlled by the fatigue requirements whereas the monolithic structure design was controlled by the fracture requirements.

The Gr/Ep composite material is fatigue and fracture resistant when kept in dry air; therefore, provisions should be incorporated in design to prevent moisture absorption during the composite material service life. However, more data is needed to determine the importance of environmental effects on the Gr/Ep composite.

S E C T I O N V I I

S P A R L O C A T I O N S E N S I T I V I T Y S T U D I E S

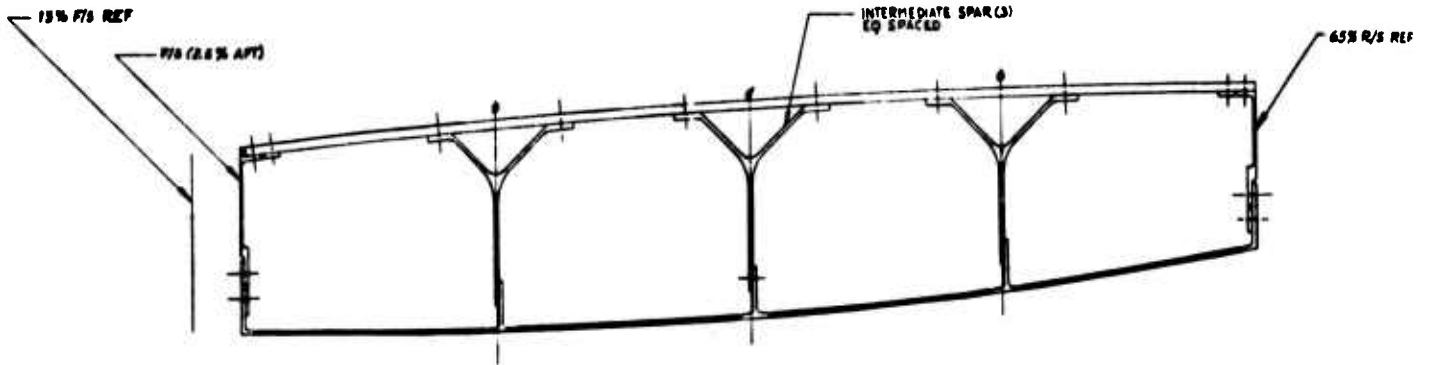
Front and rear spar location sensitivity studies were performed. The Front Spar was relocated from .150C to .175C and .200C, while holding the Rear Spar at .650C; and the Rear Spar was relocated from .650C to .700C and .750C, while holding the Front Spar at .150C. Figures 105 thru 108 show these spar location changes.

The studies were performed on the laminated lower skin box concept with three extruded "Y" internal spars, using the existing finite element model with the necessary spar relocation and the material thicknesses unchanged.

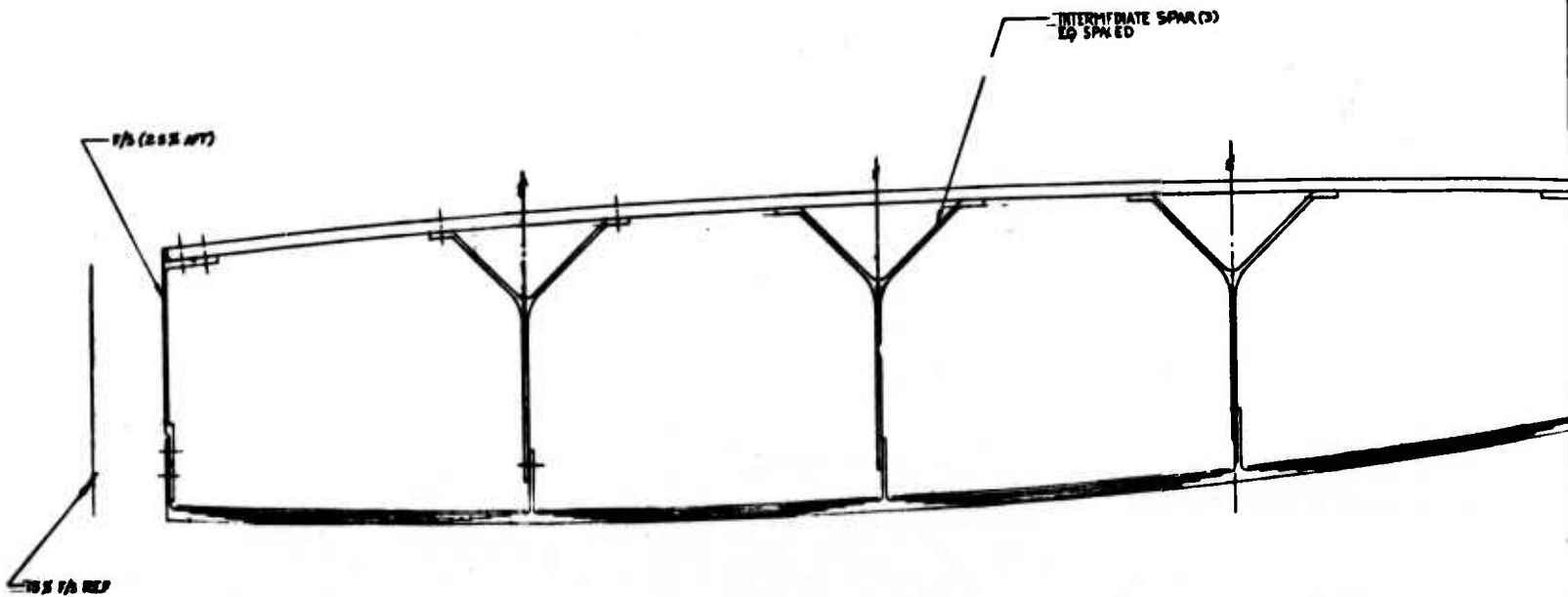
Stresses for the relocated spar case were used to adjust thicknesses to restore the stresses to their original values and the sum of the effects of thickness change is shown in Table XX for each relocation case. Table XX also shows the fuel volume changes associated with the spar location changes.

It has been determined that changing spar locations would not re-rank the three preliminary designs. The most significant effect of changing the spars is the change in fuel volume. Moving the rear spar to .75C will add 1386 pounds of fuel per aircraft.

It is interesting to note that moving the rear spar aft increases fuel volume but also increases the box weight. The weight increase is due to the fact that aft of .65C the added skin material has a reduced centroidal distance and also to the fact that the outboard portion of the wing skins is sized by fuel pressure. Extra chord width outboard, then, does not reduce skin stresses at all.



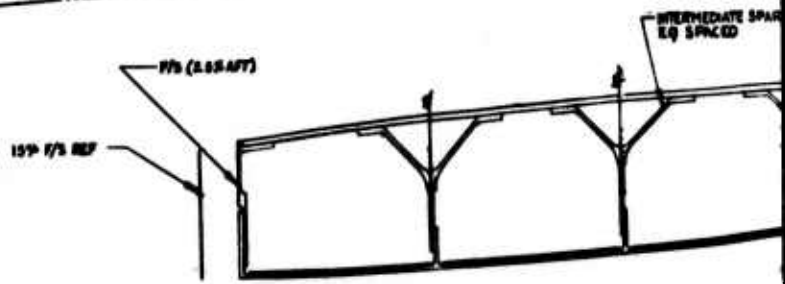
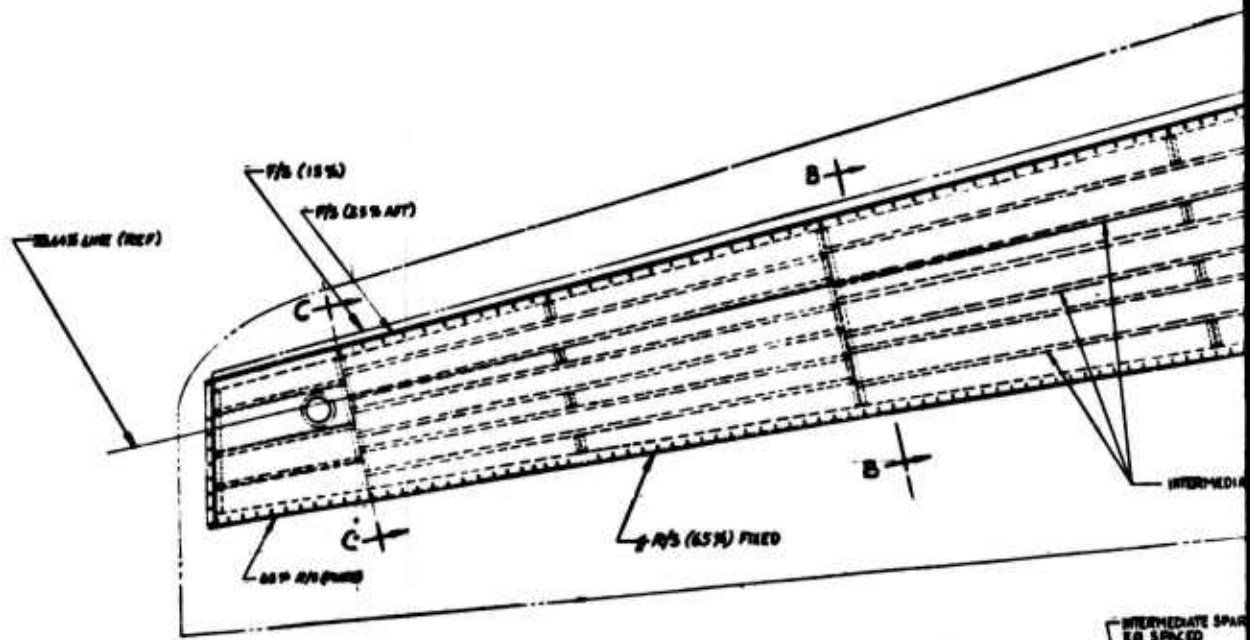
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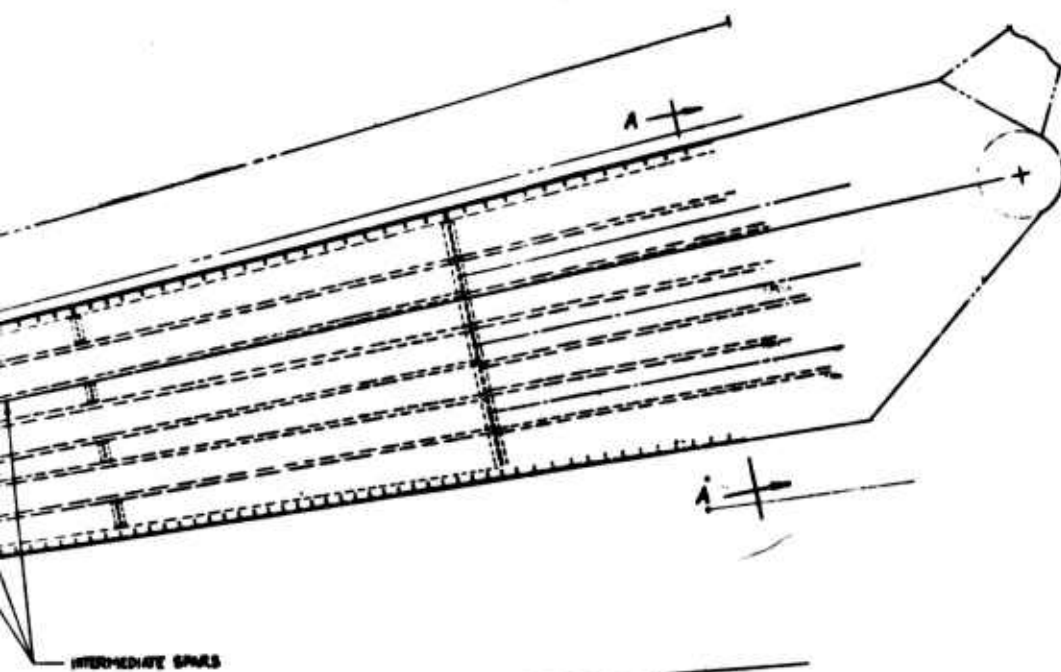
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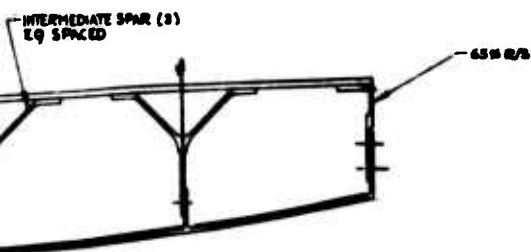
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SECTION C-C
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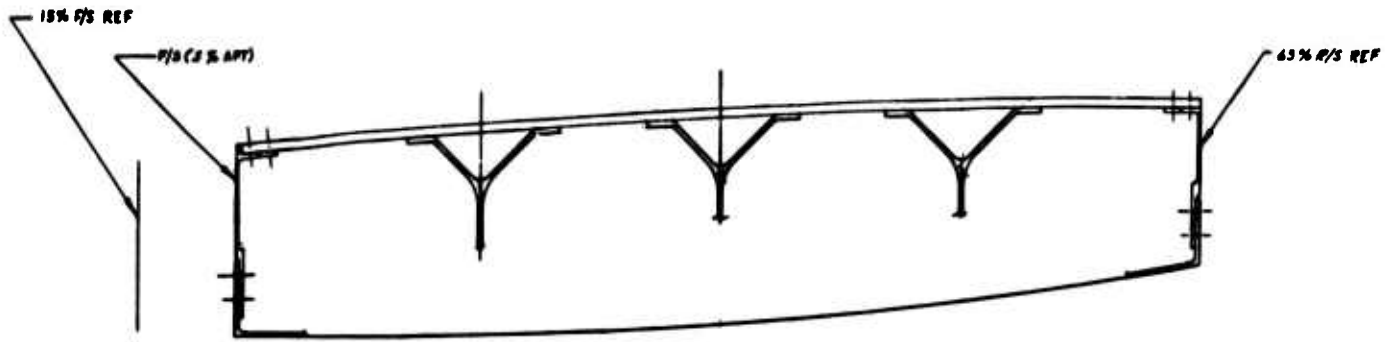
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FIXED R/S 60%, F/S MOVED AFT 25%



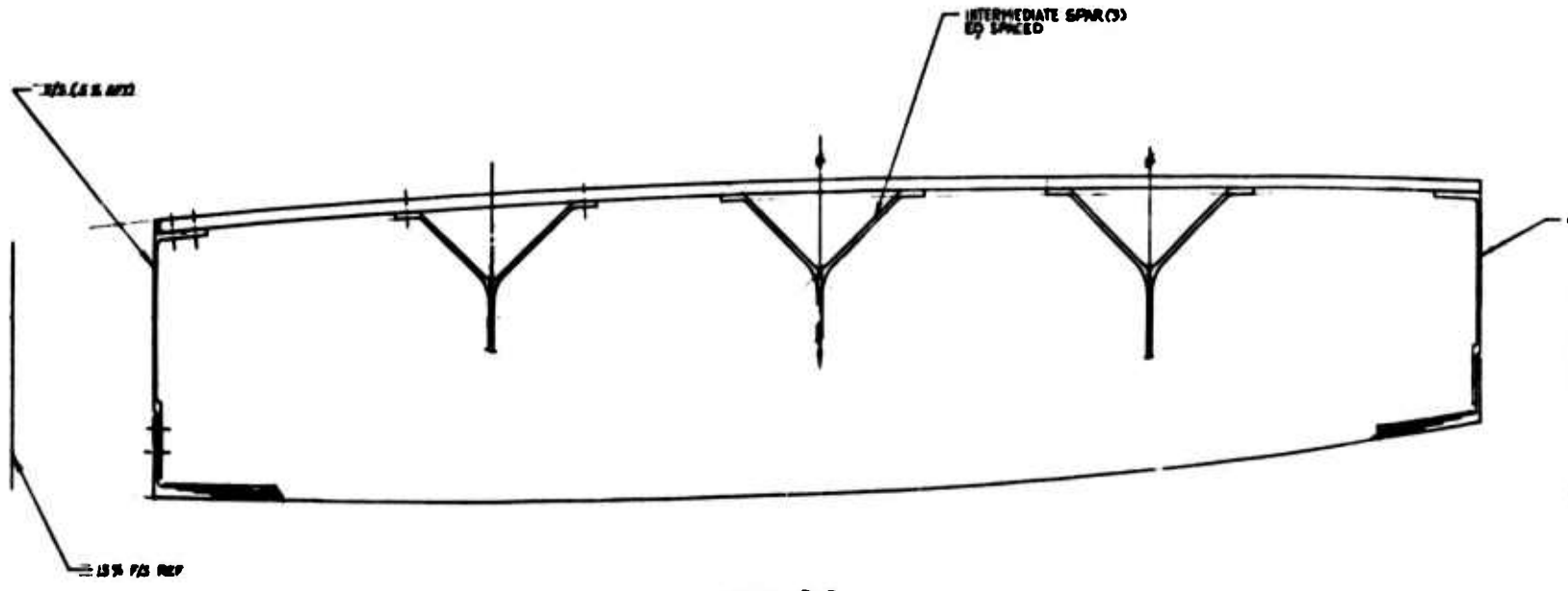
PRELIMINARY DESIGN DRAWING	
WING BOX - SPAR LOCATION SENSITIVITY LAYOUT	
GENERAL DYNAMICS Corvus Aerospace Division 101 10th Street Grand Rapids, MI 49508	633RW200

FIGURE 105 FRONT SPAR MOVED AFT. 2.5 %

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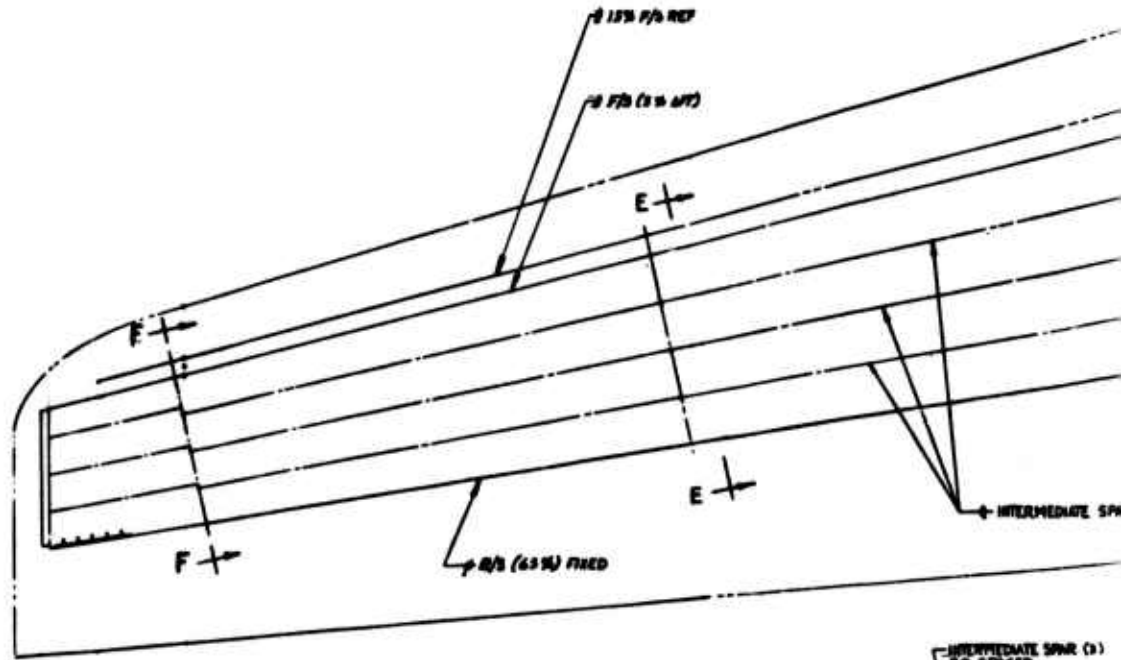
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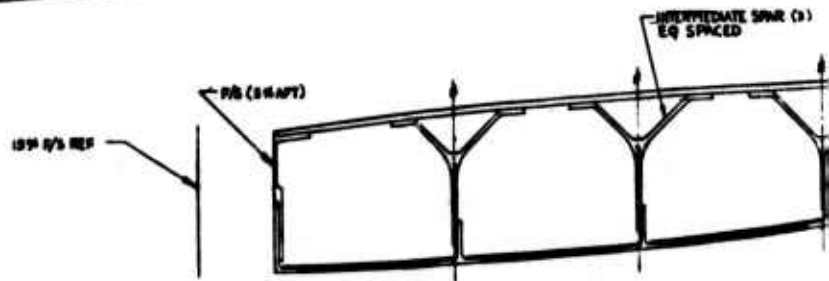
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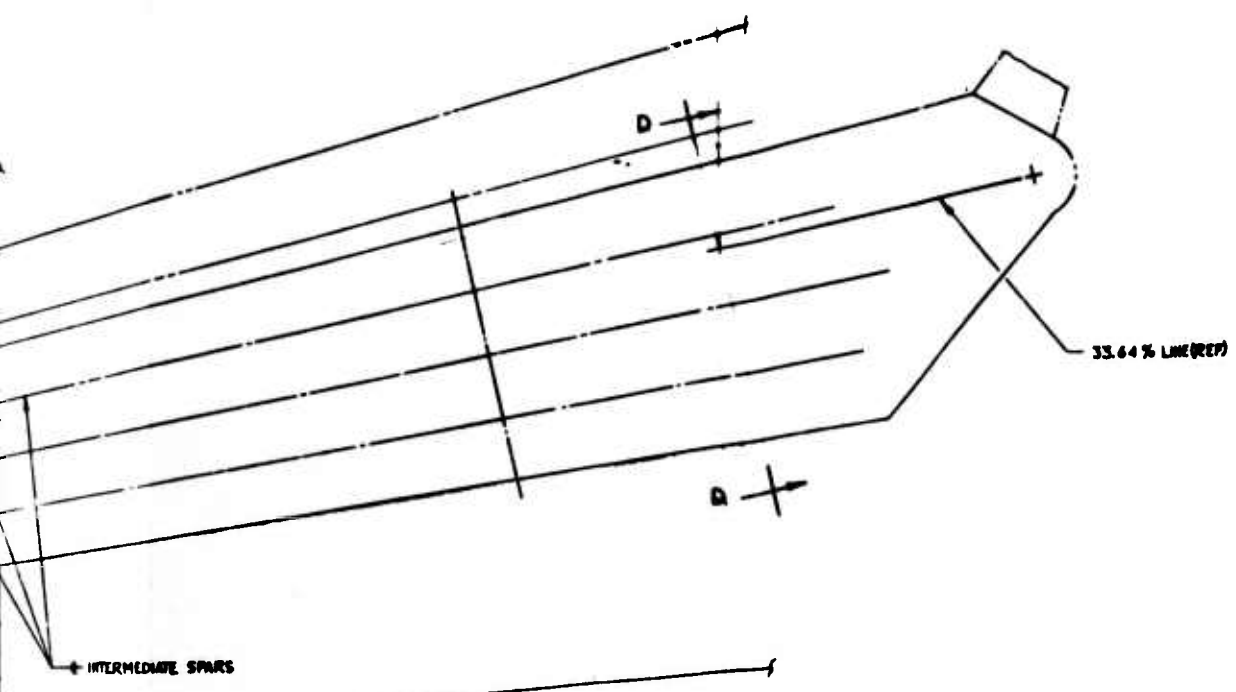
65% R/S REF



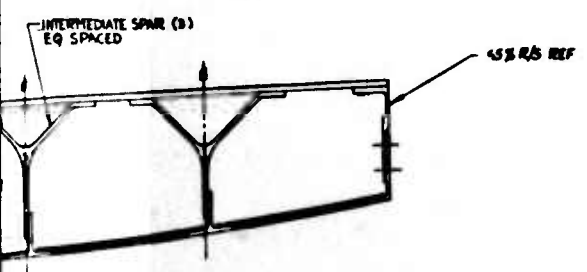
65% R/S REF



SECTION F-F
SCALE 1/8
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SAME AS C-C EXCEPT AS SHOWN



PLAN VIEW SCALE 1/80
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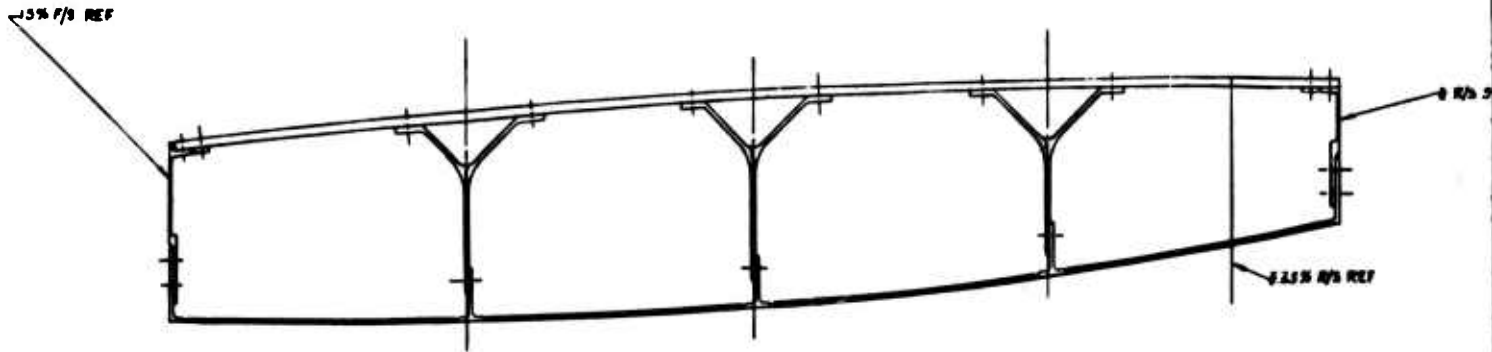


F-F
 1/2
 CONCEPT AS SHOWN

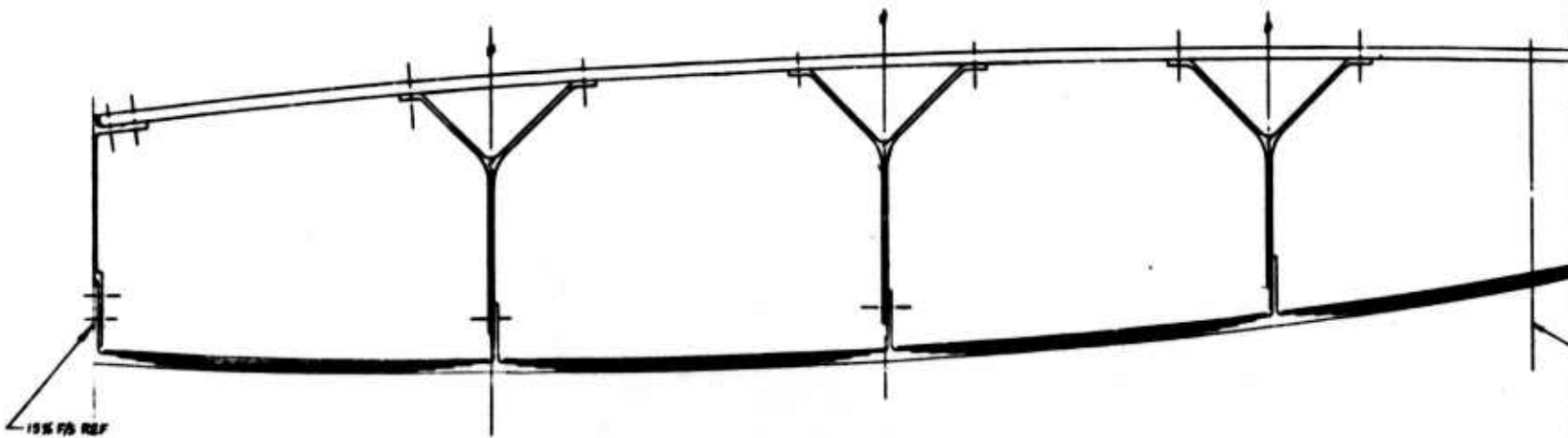
PRELIMINARY DESIGN DRAWING		
WING BOX - SPAR LOCATION SENSITIVITY LAYOUT		
GENERAL DYNAMICS	633RW200	
Convair Aerospace Division		

FIGURE 106 FRONT SPAR MOVED AFT. 5%

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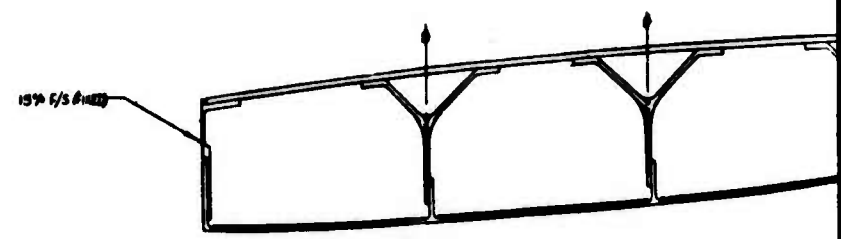
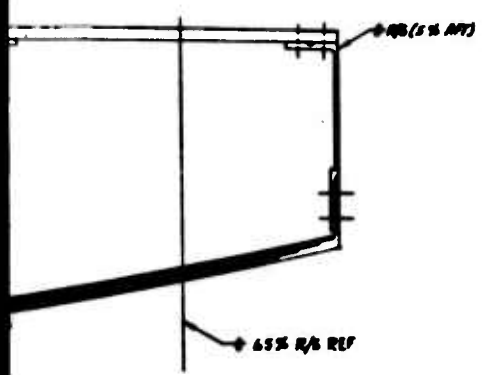
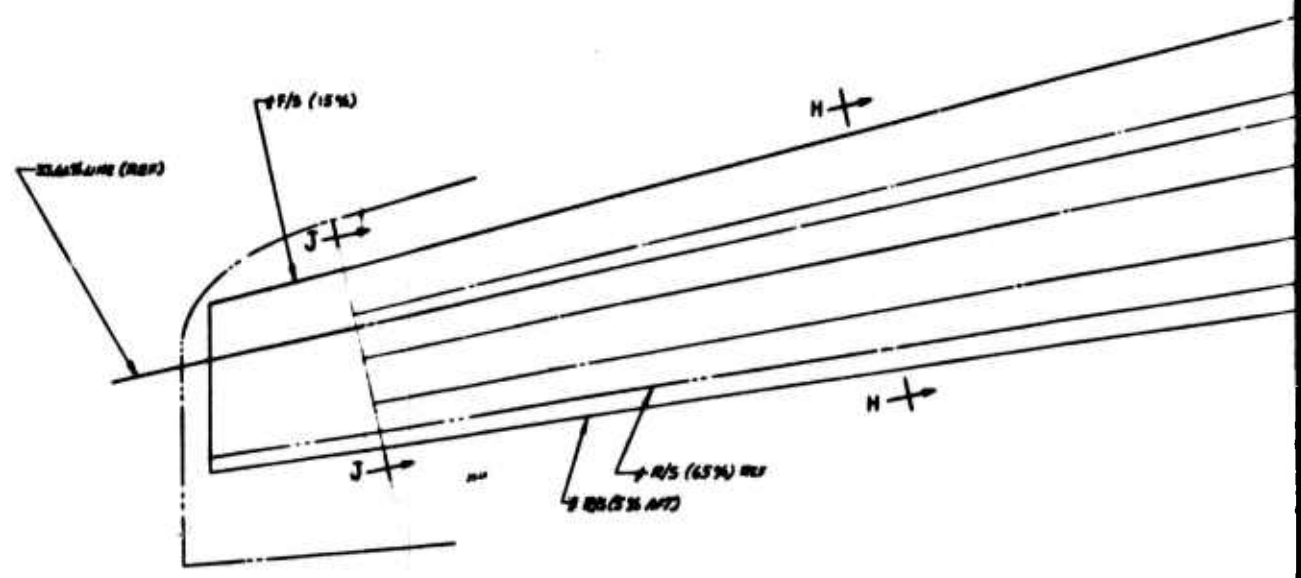
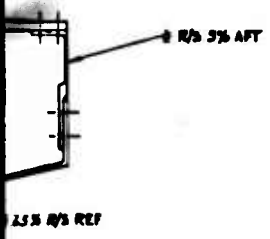


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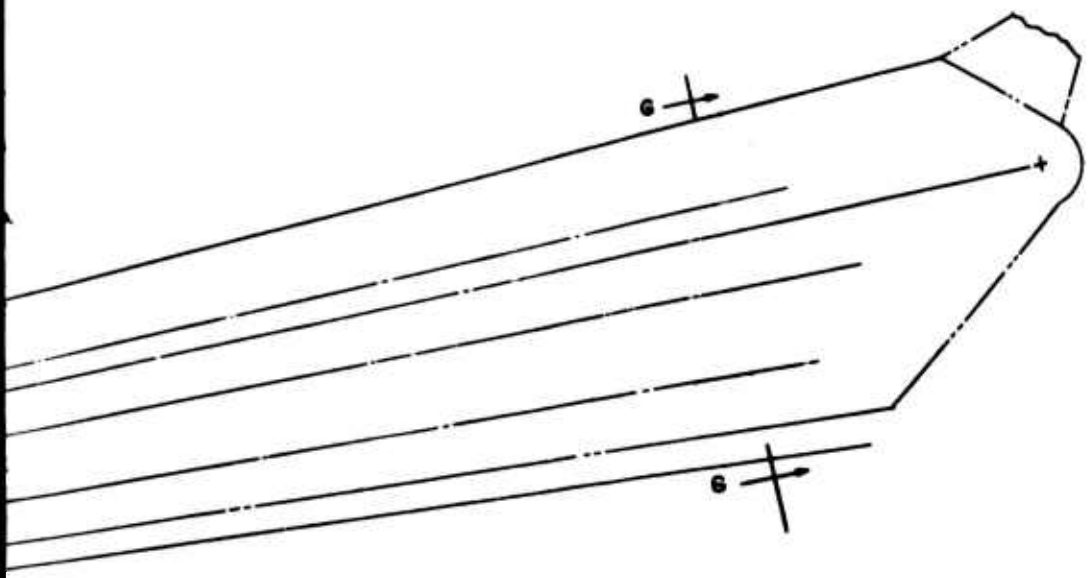


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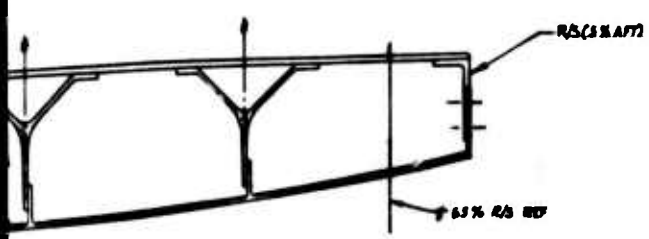
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SECTION J-J
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83 63 600



PLAN VIEW SCALE 1/10
FIXED F/S(5%), R/S MOVED AFT 5%



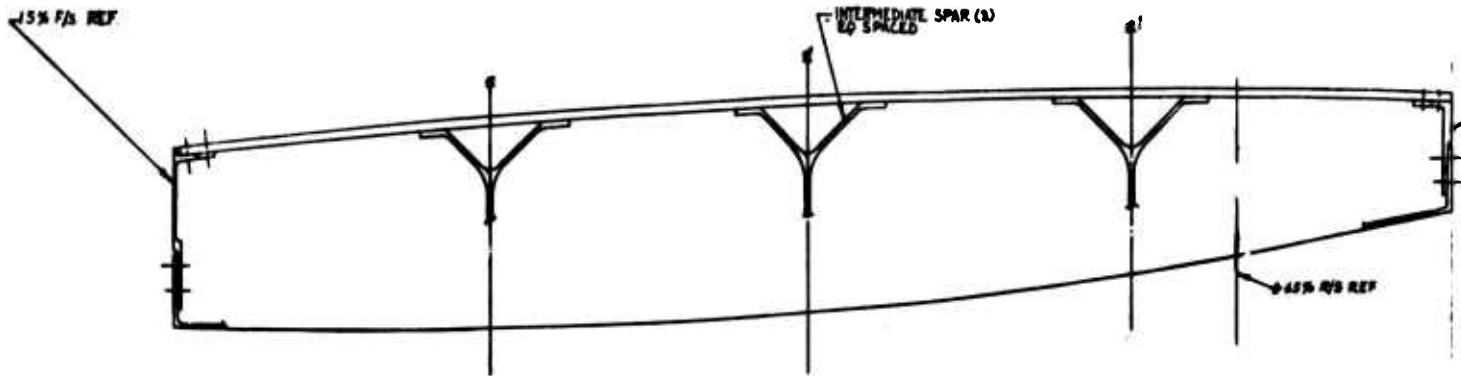
PRELIMINARY DESIGN DRAWING

WING BOX- SPAR LOCATION
SENSITIVITY LAYOUT

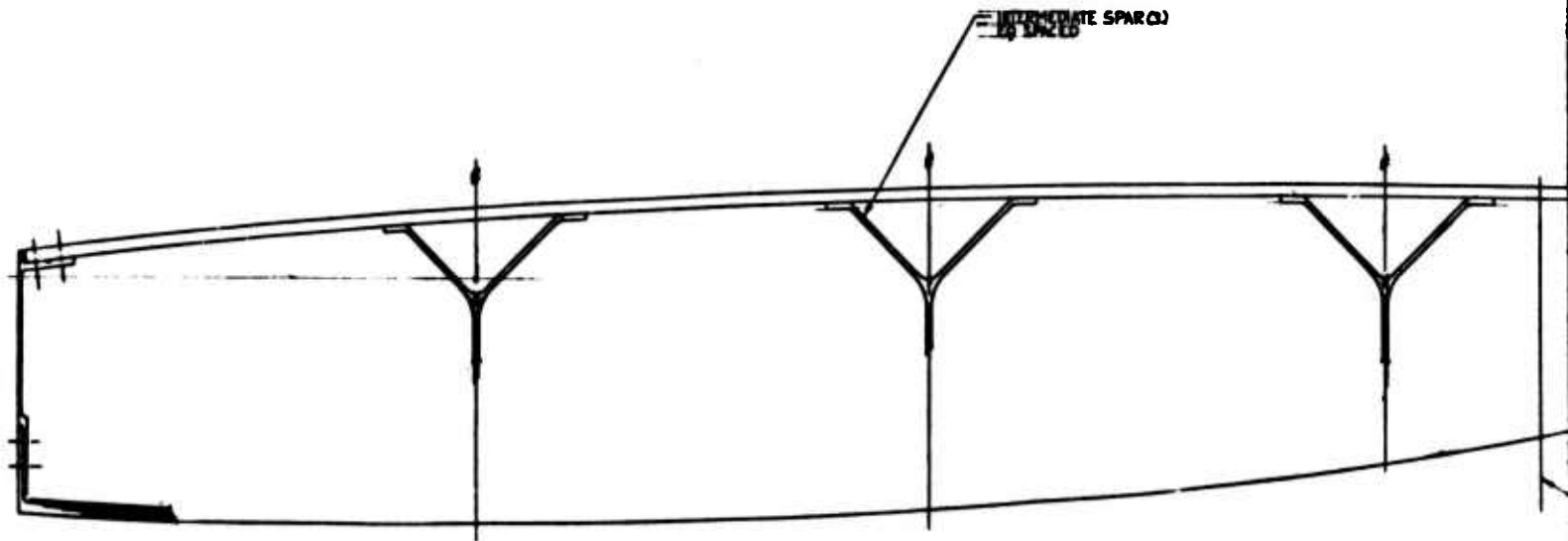
GENERAL DYNAMICS L'Ormeau Aerospace Division	633 RW 200
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FIGURE 107 REAR SPAR MOVED AFT 5%

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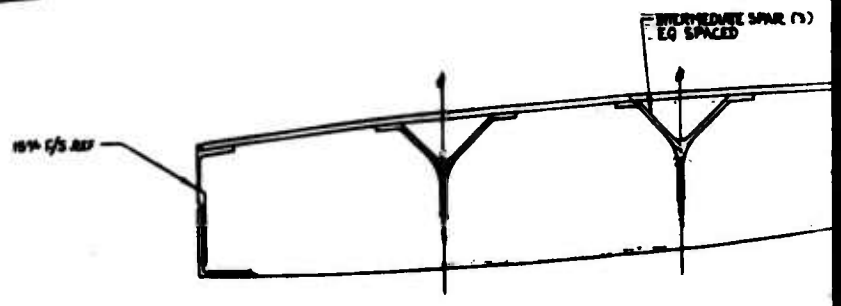
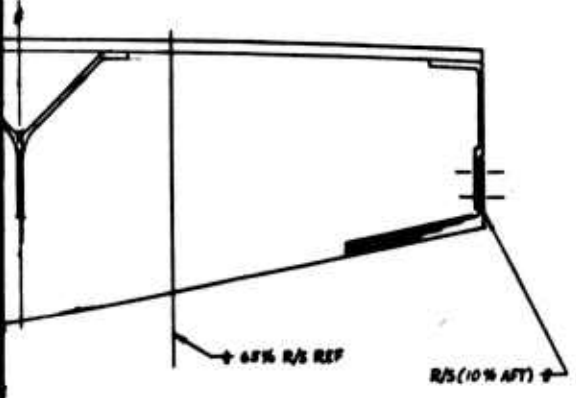
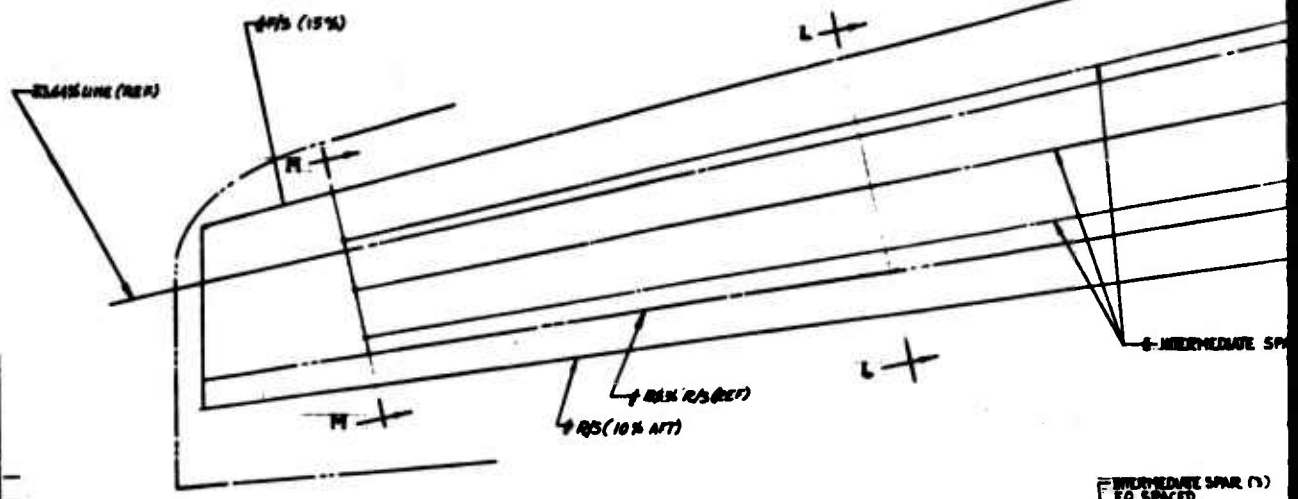
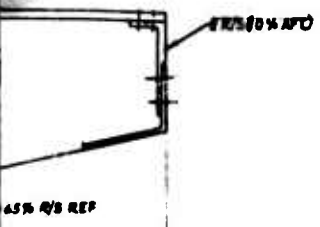


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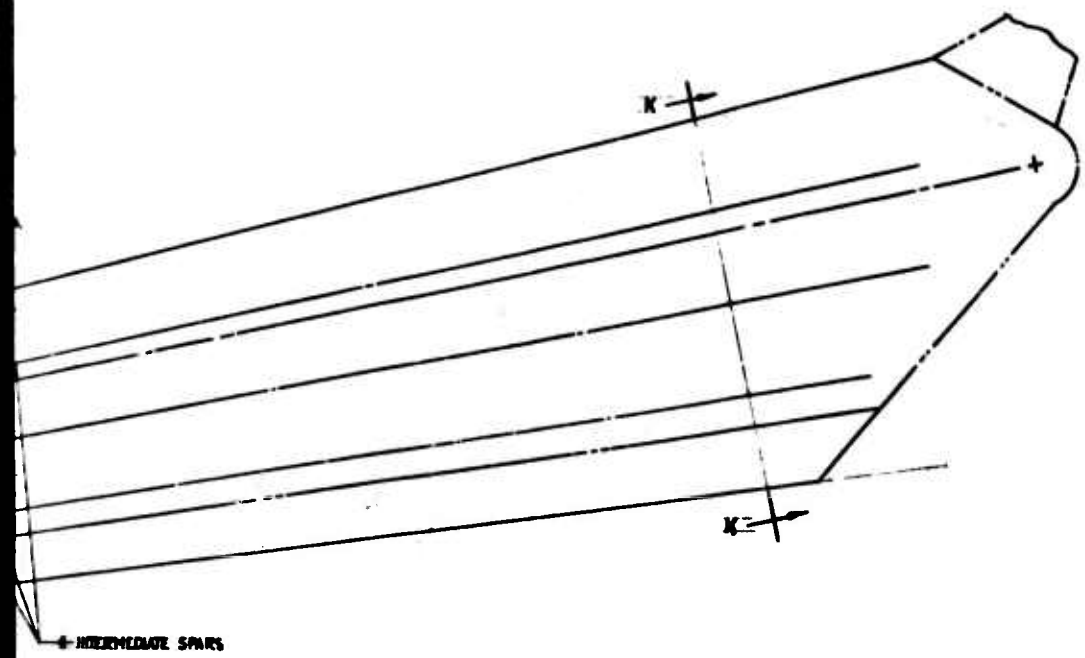


SECTION K-K
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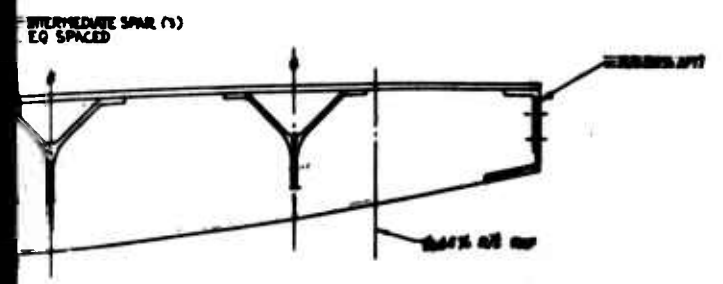
21



SECTION M-M
SCALE 1/2"
AS SHOWN
SAME AS J-J EXCEPT AS SHOWN



PLAN VIEW SCALE 1/8"
 FIXED F/S(5%), R/S MOVED AFT 10%



AS SHOWN

PRELIMINARY DESIGN DRAWING

WING BOX- SPAR LOCATION SENSITIVITY LAYOUT

GENERAL DYNAMICS Aircraft Division	633RW 200
---------------------------------------	-----------

FIGURE 108 REAR SPAR MOVED AFT 10%

TABLE XX

FRONT AND REAR SPAR RELOCATION SENSITIVITY**

	<u>WING BOX WEIGHT CHANGE</u> (Lbs/Side)	<u>TIP DEFLECTION*</u> (in.)	<u>TIP TWIST*</u> (Deg.)	<u>VOLUME CHANGE</u> (in ³ /side)
<u>F. S. To .175C</u>	-30.7	59.68	+3.17	-11022
<u>F. S. To .200C</u>	-48.7	61.13	+3.18	-18952
<u>R. S. To .700C</u>	+46.5	55.58	+2.63	+12306
<u>R. S. To .750C</u>	+116.8	54.51	+2.30	+24633
<u>F. S. At .15C,</u> <u>R. S. At .65C</u>	Reference	57.39	+2.93	Reference

** EFFECT ON WING BOX ONLY

* USING CONDITION 4 FOR REFERENCE. TIP UP AND L.E. DOWN ARE POSITIVE, BOTH ULTIMATE

S E C T I O N V I I I
V A L U E E N G I N E E R I N G

Throughout this program Value Engineering in consonance with grass roots estimating personnel have provided accurate cost data, evaluation and trade study inputs to design personnel for each concept, at each phase of the program, identifiable by element within the concept. This information was used to identify areas requiring additional design and trade study effort as shown in Figure 109.

8.1 C O S T I N G G R O U N D R U L E S

Costing groundrules were as follows:

- o Estimates to be made for recurring acquisition costs
- o Production rate of four ship sets (8 units) per month
- o Average cumulative costs for 200 ship sets in 1975 dollars for the analytical assembly concepts
- o For the preliminary design concepts compute 1975 and 1980 average cumulative costs for 1, 4, 40, 100 and 200 ship sets
- o Use General Dynamics rates and cost structure.

The costs developed included recurring costs such as:

- | | |
|-------------------------|----------------------------------|
| o Manufacturing | o Tool Engineering Maintenance |
| o Product Assurance | o Tool Manufacturing Maintenance |
| o Configuration Control | o Quality Assurance |
| o Materials | o General & Administrative |
| o Overhead | o Tool Material Maintenance |

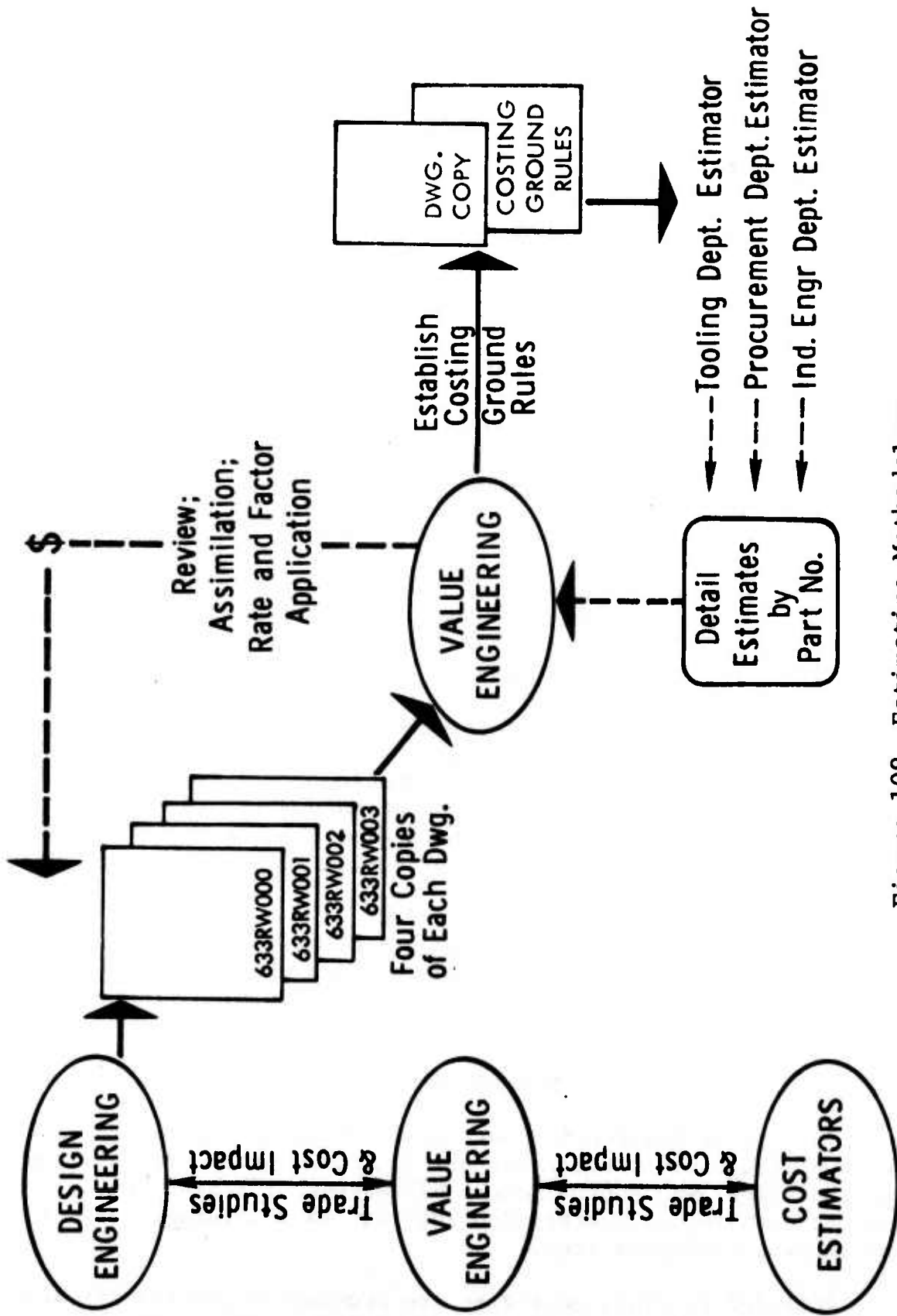


Figure 109 Estimating Methodology

The estimates did not include non-recurring costs such as:

- o Engineering Design & Development Costs
- o Basic and Augmentation Tooling
- o Production Aid Tooling
- o Flight Testing

8.2 MATERIAL COSTS

Material costs were projected to a cumulative average for 200 ship sets from a list of material (LM) on the released full size component and analytical assembly drawings. Start and finish weights and dimensions were provided for costing purposes.

Procurement department estimating personnel estimated each line item in detail using recent purchase orders, vendor published catalogues or direct vendor quotes. Material form, dimensional requirement and "extras" such as heat treat, cut to length, etc., were considered in the estimates.

Historical material factors were added to the basic estimate to reflect hidden costs such as attrition, freight in and material overhead. Costs for quantities of one through 40 were adjusted to reflect a 95% learning curve. Because materials are usually purchased in yearly lots, only one half of the value for the 95 percent cost curve was used to project average unit costs for 100 and 200 unit quantities.

The material cost data was made available to the designers through a series of master costing drawings which were left open in the design area for review, evaluation, and comparison.

8.3 MANUFACTURING COSTS

Industrial Engineering personnel, utilizing labor standards and applicable realization factors considered the material form, material type, and approximately 82% average learning curves to project cumulative average 200 ship set manufacturing costs for the various configurations.

Included in those estimates are provisions for historically hidden costs such as processing, allocations, rework and shop liaison.

An additional factor was applied to these direct labor hours to arrive at a projected cost for quality assurance.

This data was then made available to the designers through the same series of master costing drawings as was described in section 8.2 under material costs.

8.4 TOOLING COSTS

Projected recurring costs include provisions for various "maintenance type" tooling costs such as tool manufacturing, tool engineering, tool materials, and tooling QA charges. These costs were derived through use of empirical ratios applied to detail estimates for non-recurring tool manufacturing and tool augmentation costs.

These detail estimates include the basic tools which would be necessary to fabricate the full size components in addition to the augmentation tools necessary to arrive at a four ship set per month production rate.

8.5 COMPARISON OF ANALYTICAL ASSEMBLY COSTS BETWEEN BASELINE AND ADVANCED CONCEPTS

Table XXI is a tabulation of cumulative average costs for 200 units of each advanced analytical assembly concept and the baseline in 1975 dollars.

8.6 COMPARISON OF COSTS BETWEEN BASELINE AND ADVANCED CONCEPTS FOR PRELIMINARY DESIGN CONCEPTS

Table XXII is a tabulation of cumulative average costs of 1, 4, 40, 100, and 200 unit wing boxes for advanced concepts and the ATW-4 baseline shown in 1975 and 1980 dollars. Illustration of these designs are shown in Figures 110 through 113.

TABLE XXI. SUMMARY ANALYTICAL ASSY COST
WING BOX

(1975 DOLLARS)

ASSY NO	MATL \$	MFG \$	TOOLING \$	TOTAL
633RA000-1	4244	7777	289	12310
633RA001-1	3395	6442	254	10091
633RA001-3	3612	6917	245	10774
633RA001-5	3589	7067	336	10992
633RA001-801	3503	6965	192	10660
633RA001-803	3168	6291	282	9741
633RA001-805	2799	6672	306	9777
633RA002-1	3033	6301	212	9546
633RA003-1	4956	7058	263	12277
633RA003-3	3699	8835	259	12793
633RA003-5	2999	7643	276	10918
633RA003-801	3520	6985	228	10733
633RA004-1	3770	4048	173	7991
633RA004-3	4564	4594	150	9308
633RA004-5	4198	4277	207	8682
633RA005-1	37762	5227	291	43280
633RA006-1	39507	5652	149	45308
633RA006-3	32690	4973	149	37812
633RA007-1	36838	11043	212	48093
633RA008-1	38011	5508	202	43721

Average for 200 Units

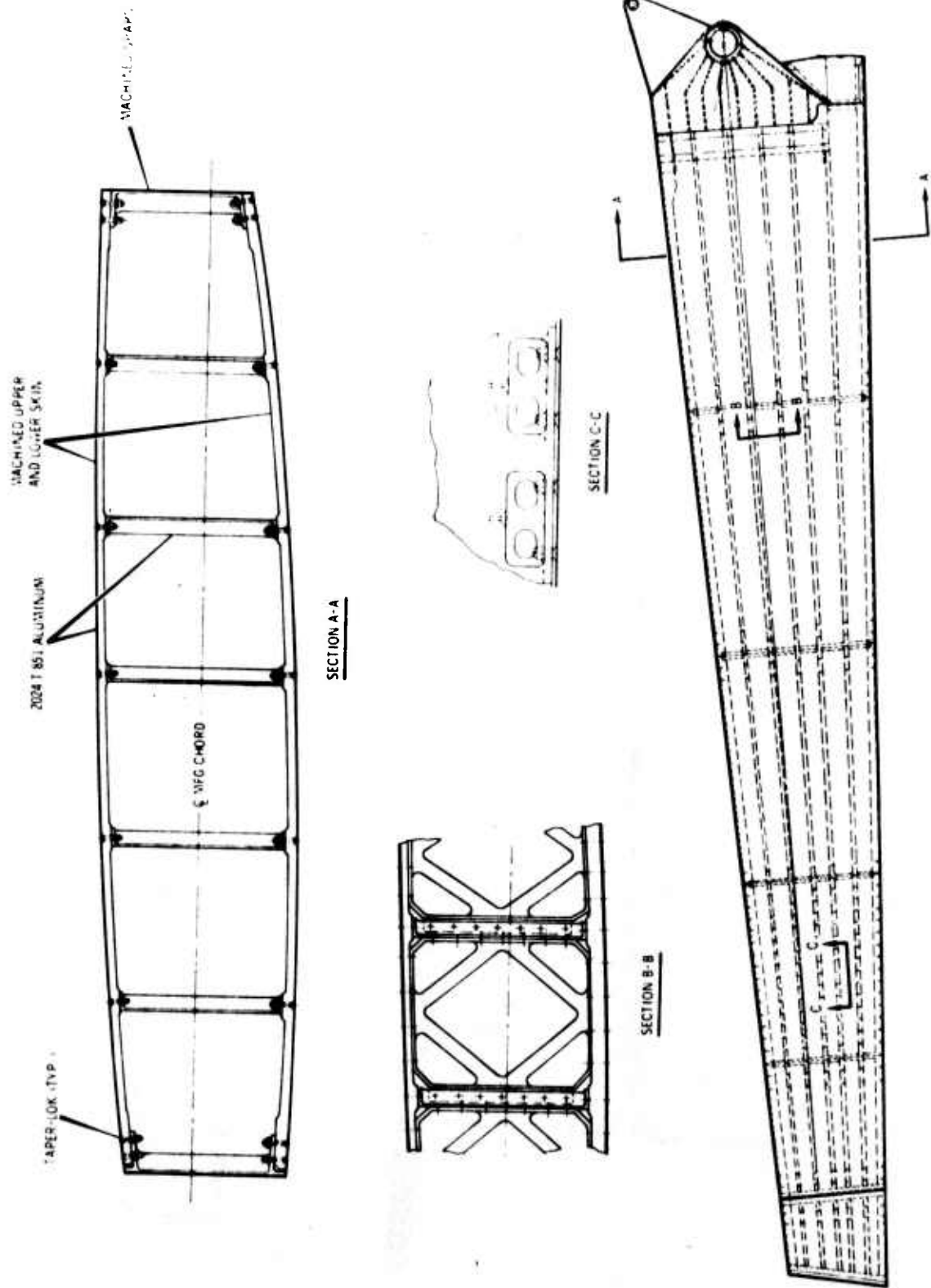


Figure 110 ATW-4 Wing Box Baseline

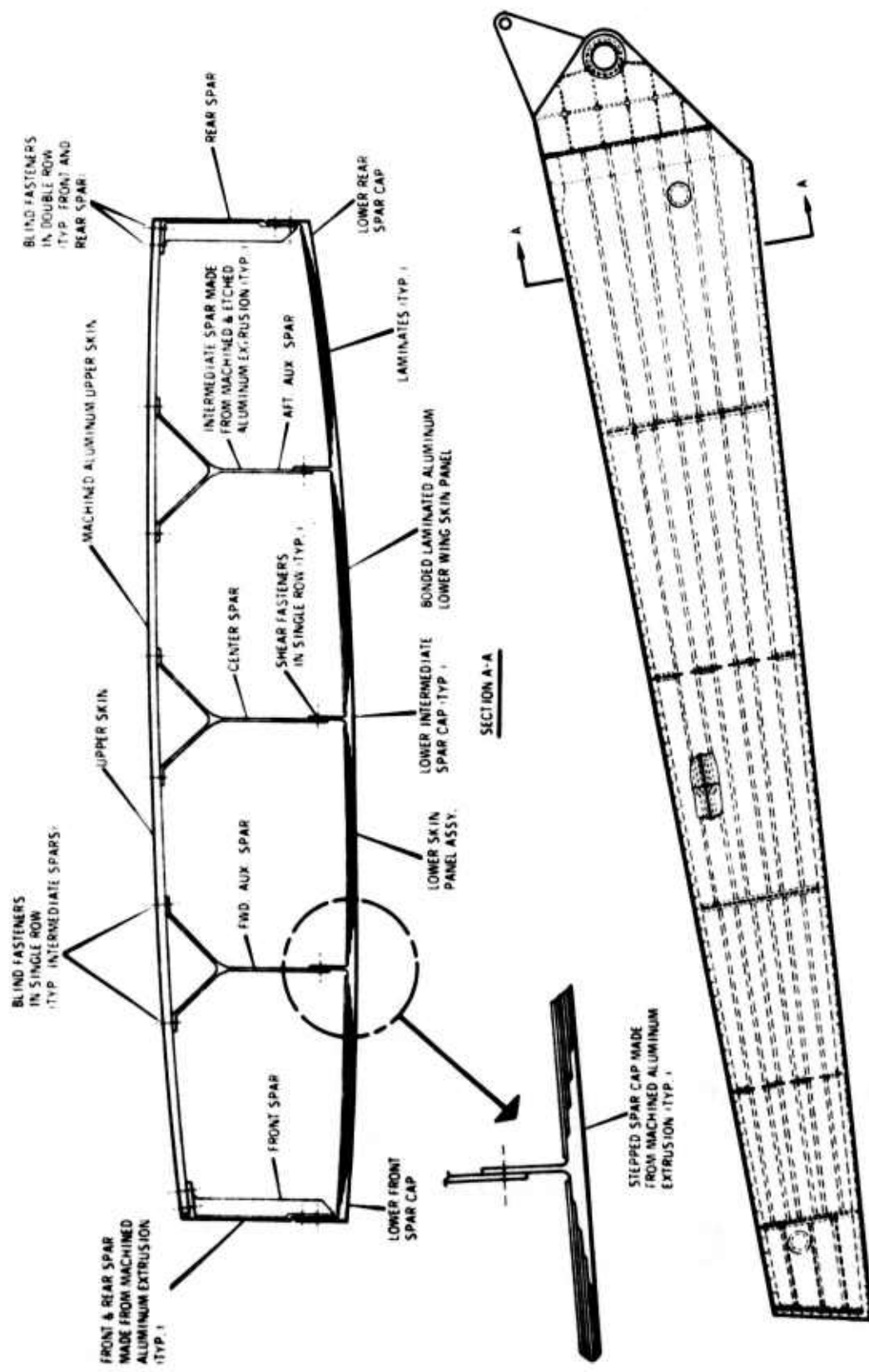


Figure 111 ATW-4 Wing Box-Laminated Aluminum Lower Skin With "y" Spars

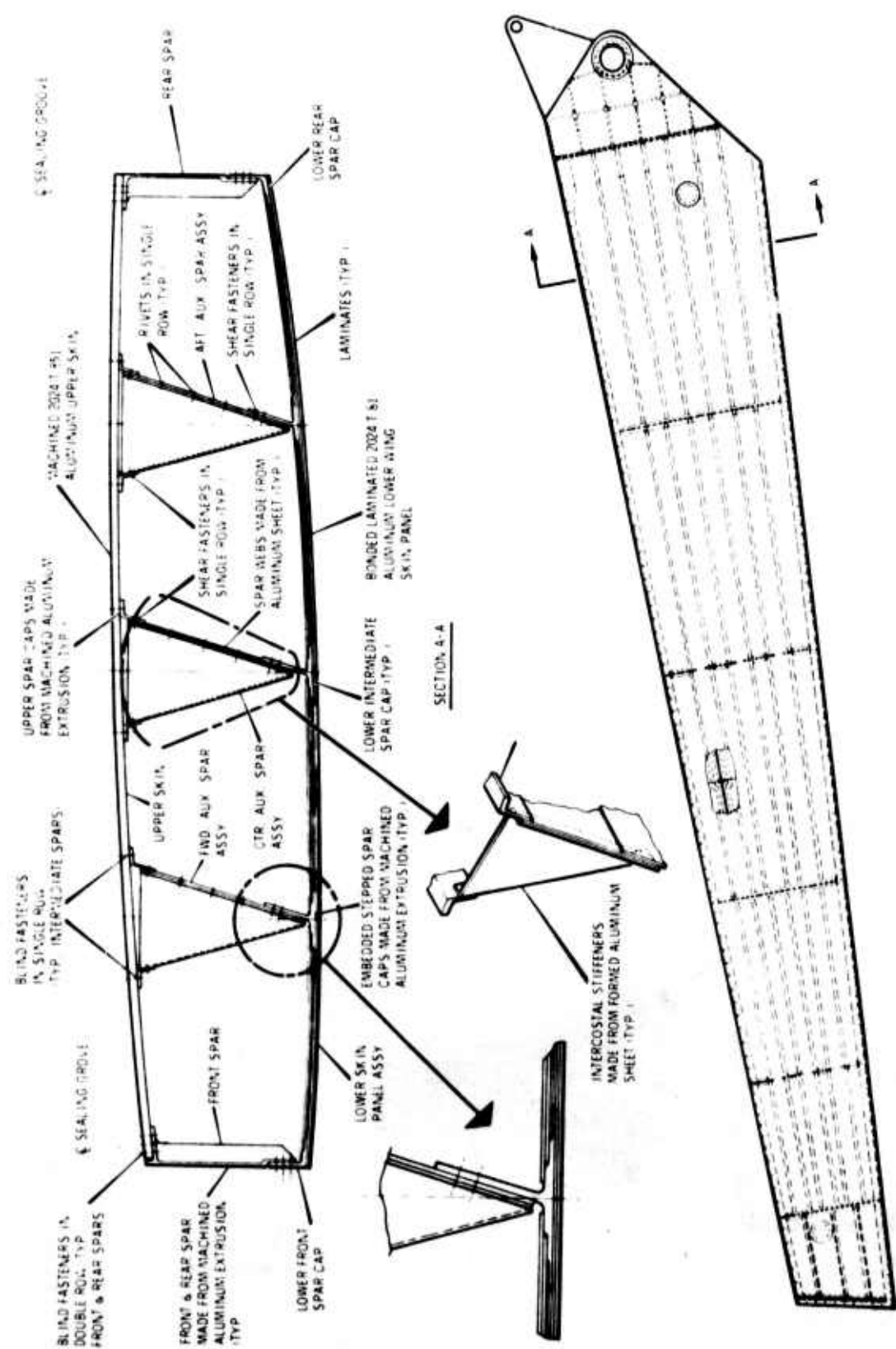


Figure 112 ATW-4 Wing Box-Laminated Aluminum Lower Skin, Canted Spars

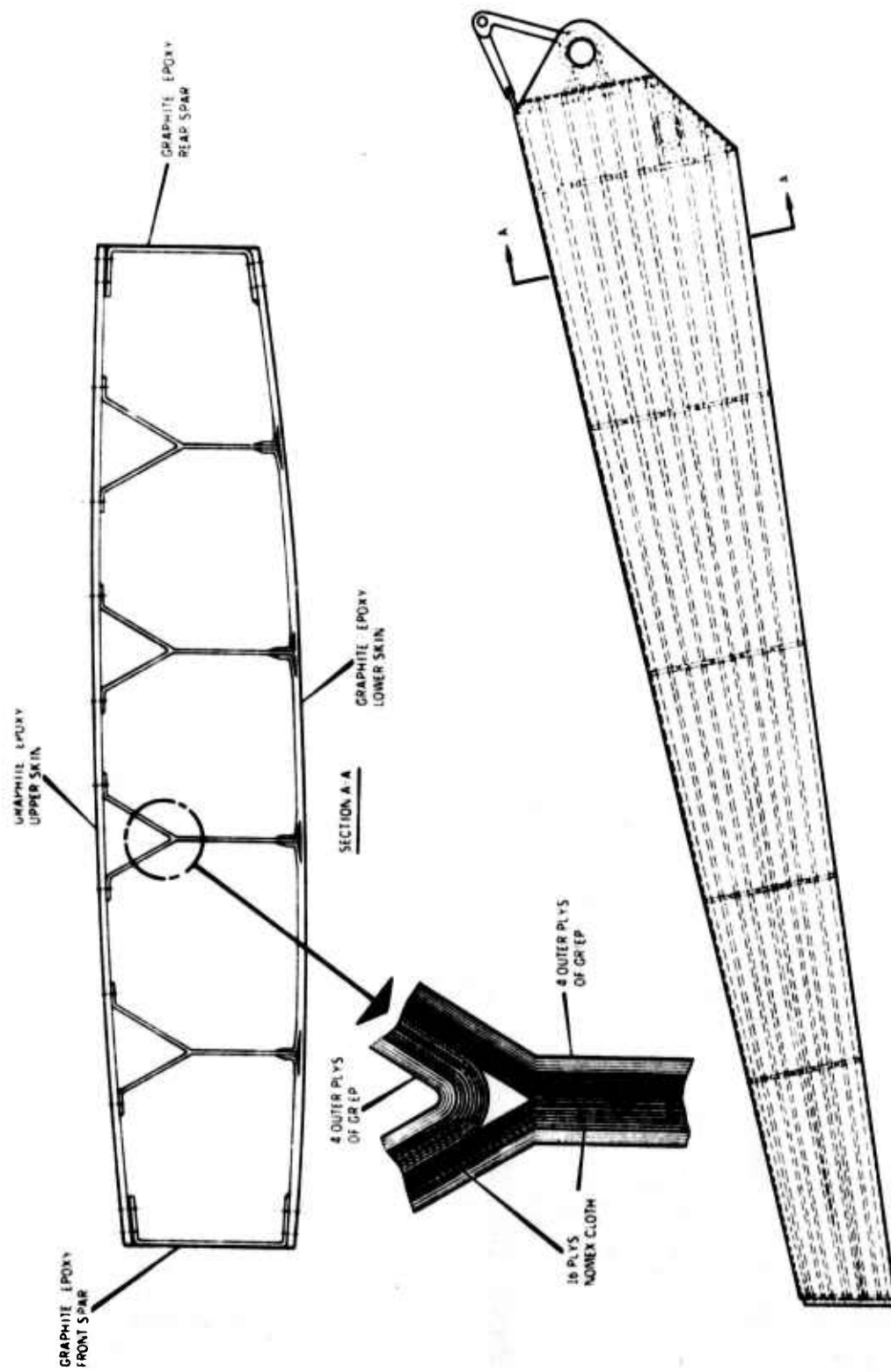


Figure 113 ATW-4 Graphite Epoxy Composite Wing Box "Y" Spars/Embedded Lwr. Caps

**Table XXII Wing Box Cost Analysis Supercritical
Wing Preliminary Design Study**

Configuration	Lot Size	Average Unit Values (000)					
		1975 Dollars			1980 Dollars		
		Cost	Earnings	Price	Cost	Earnings	Price
Baseline Design Drawing No. 633RW00-1	1	578	59	637	728	73	801
	4	465	47	512	585	59	644
	40	287	29	316	362	36	398
	100	234	23	257	294	30	324
	200	200	20	220	252	25	277
Supercritical Design Drawing No. 633RW001-1	1	496	50	546	626	63	689
	4	398	40	438	503	50	553
	40	248	25	273	314	31	345
	100	203	20	223	257	26	283
	200	174	18	192	221	22	243
Supercritical Design Drawing No. 633RW002-1	1	519	52	571	654	65	719
	4	416	41	457	524	52	576
	40	256	26	282	324	32	356
	100	208	21	229	264	26	290
	200	178	18	196	225	23	248
Supercritical Design Drawing No. 633RW002-3	1	542	54	596	682	68	750
	4	434	44	478	547	55	602
	40	267	27	294	336	34	370
	100	216	22	238	274	27	301
	200	185	18	203	234	23	257
Supercritical Design Drawing No. 633RW003-1	1	651	65	716	727	73	800
	4	554	55	609	610	61	671
	40	398	40	438	427	43	470
	100	355	35	390	375	37	412
	200	326	33	359	341	34	375
Supercritical Design Drawing No. 633RW003-3	1	621	62	683	689	69	758
	4	528	53	581	578	58	636
	40	383	38	421	406	41	447
	100	342	34	376	357	36	399
	200	315	32	347	325	33	358

SECTION IX

MANUFACTURING

The method of manufacturing of the wing boxes developed in this study was determined at both the analytical assembly level and at the preliminary design phase. This included a listing of all tools required to manufacture the analytical assemblies and the preliminary design boxes as well as an estimate of the non-recurring cost of these tools. This estimate was necessary in order to determine the recurring cost to maintain these tools in the production lots required by the contract as well as define the manufacturing plan for use by industrial engineering to develop the manufacturing costs.

9.1 ANALYTICAL ASSEMBLY STUDY

Each of the analytical assemblies was evaluated to determine the manufacturing processes that were required and the tools necessary for fabrication. Added to this was the additional tools required to meet the production rate requirements. From this information the tool manufacturing hours were estimated. Table XXIII summarizes a typical tooling summary for 633RA003-801 analytical assembly. The total non-recurring tooling mfg. hours were then determined for each analytical assembly, including tool engineering hours and tool material costs. From this information the total recurring costs were determined. A summary of these costs are shown in Table XXIV.

9.2 PRELIMINARY DESIGN STUDY

The preliminary wing box designs were analyzed to determine the manufacturing processes and tools that would be required for the fabrication of a complete wing box. In addition, the extra tools required to meet the production rate requirements were also considered. Total tooling recurring costs were determined and manufacturing plans were outlined for each preliminary wing box design in order to develop the manufacturing costs and to assist in the evaluation of each of the wing box configurations. The manufacturing plan for the baseline design, the top ranked design, and the composite wing design are being discussed in this section.

TABLE XXIII 633RA003-801 TOOL SUMMARY

TASK ITEM	TOOL CODE	UNIT HOURS	ADD FOR RATE		REMARKS	
			NO. TOOLS	TOTAL HOURS		
-7 SKIN	FLTP CKTP RTEX		1 1 1	6 12 40	SAW (EXCESS) BRAKE FORM ROUT TO SIZE	
-83 F. SPAR	[PDTA MLFX	80	2	160	MILL COMPLETE	
-91 R. SPAR		80	2	160		
-175, -177 UP.AUX.SPAP CAP -179, -181 UP.AUX.SPAP CAP -183, -185 UP.AUX.SPAP CAP	[MLFX	80	6	480	MILL COMPLETE	
-191 LWR.AUX.SPAP CAP		[PDTA MLFX	60	3		180
-193 LWR.AUX.SPAP CAP			80	3		240
-195 LWR.AUX.SPAP CAP						
-187 INTERCOSTAL	FLTP TOTP HPFM	6 6 18	1 1 2	6 6 36	SHEAR TO SIZE BRAKE ONE FLG FINISH FORM	
-189 SPAR WEB	SCS			-	SHEAR TO SIZE	
SKINS (-1 ASSY)	CKTP FLTP	8 6	6 12	48 72	SHEAR FOR 4 SKINS ROLL CONTOUR SHEAR TO SIZE	
-203 LWR SKIN ASSY	PRFX VFBX BNFM		1 1 1	120 80 240	PREFIT DETAILS BOND PROCEDURE SAME AS 633RA001	
-801 ASSY	ASFX		1	240	LOCATE DETAILS, DRILL, INSTALL FASTENERS	
				2126	TOTAL ESTIMATED TOOL MFG. HRS.	

TABLE XXIV TOOLING SUMMARY FOR ANALYTICAL ASSEMBLIES

ITEM	NON RECURRING				RECURRING					
	TOOL	MFG.	TOOL	ENG.	TOOL	MFG.	TOOL	ENG.	TOOL	MAT'L
633RA000-1		2904		1540		1567		1830		\$3526
633RA001-1		2382		1262		1286		1500		2894
633RA001-3		2286		1212		1235		1440		2778
633RA001-5		3104		1645		1676		1956		4401
633RA001-801		1768		937		955		1114		2507
633RA001-803		2622		1390		1416		1652		3186
633RA001-805		2860		1516		1545		1802		3476
633RA002-1		1976		1046		1066		1244		2400
633RA003-1		2454		1301		1325		1546		2981
633RA003-3		2410		1278		1302		1518		2930
633RA003-5		2576		1366		1392		1624		3132
633RA003-801		2126		1127		1148		1340		2583
633RA004-1		1612		854		871		1016		1960
633RA004-3		1414		750		764		871		1719
633RA004-5		1930		1023		1042		1216		2345
633RA005-1		2660		1410		1436		1676		4308
633RA006-1		1360		720		735		857		2205
633RA007-1		2058		1029		1112		1297		2502
633RA008-1		1884		1000		1018		1187		2291

9.2.1 Preliminary Manufacturing Plan for Baseline 633-RW00C

The manufacturing process for the baseline design is defined in Table XXV and schematically shown in Figure 114. The methods of fabrication are discussed below:

9.2.1.1 Upper and Lower Skin Fabrication

The upper skin and lower skin are sculptured plate made from 2024-T851 aluminum alloy.

Basic tapers will be machined on a numerical control skin mill. Contour will be formed on a 1000 ton numerical control brake. The skin will then be etched and routed to finish dimensions.

9.2.1.2 Spar Fabrication

The wing spars are all integrally machined structure made from 2024-T851 aluminum alloy extrusions.

9.2.1.3 Bulkhead and Rib Fabrication

The bulkheads and ribs are all integrally machined structure made from 2024-T851 aluminum alloy.

9.2.1.4 Pivot Fitting Fabrication

The pivot fitting is a welded assembly containing a mechanically attached shear web (Fig. 115).

The welded assembly consists of an upper plate, a lower plate, shear ring, and shear webs machined from D6Ac forgings and plate.

The mechanically attached shear web is machined from 6Al-4V titanium plate.

9.2.1.5 Final Assembly of Wing Box

A major assembly and drill fixture commonly known as a "wing buck" will be required to perform final assembly operations of the box.

TABLE XXV BASELINE 633-RW000 WING BOX MANUFACTURING ANALYSIS

DESCRIPTION	QTY	MATERIAL		FAB. METHOD
		TYPE	SIZE	
-7 & -5 UPR & LWR SKINS	2	2024-T851	1.50 & 1.75 X 77 X 368	PURCHASE ROUGH SIZE, MILL TAPER, ETCH LANDS, ROUT PERIPHERY
-11 & -13 FRONT & REAR SPARS	2	2024-T8511	2.75 X 12 X 380	EXTRU. - N/C MILL TO SIZE
-15, -17, -19, -21, -23 AUX. SPARS	5	2024-T8511	1.25 X 12 X 350	EXTRU. - N/C MILL TO SIZE
-25 & -27 UPPER & LOWER OUTBOARD SKINS	2	2024-T851	.125 X 32 X 36	SHEAR OVERSIZE, ROLL CONTOUR, ROUT TO SIZE
-29, -31, -33, -35 BULKHEADS	24	2024-T851	3.0 X 6.5 X 6 TO 3.0 X 12 X 11	PURCHASE OR SAW ROUGH SIZE, N/C MILL COMPLETE
-37 & -39 BULKHEADS	2	2024-T851	3.0 X 7 X 28	PURCHASE OR SAW ROUGH SIZE, N/C MILL COMPLETE
- OUTBOARD AUX. SPARS	5	2024-T8511	1.25 X 6 X 28	EXTRU. - N/C MILL TO SIZE
- PIVOT ASSEMBLY				SEE FIGURE 9-2
- 1 ASSEMBLY				<ul style="list-style-type: none"> o POSITION LWR.SKIN, o ATTACH PIVOT FITTING o INSTALL AUX.SPARS, o BULKHDS, FRT.& REAR SPARS o INSTALL PLUMBING & MISC. EQUIPMENT o INSTALL UPPER SKIN o INSTALL OUTBD SPARS, o MISC.HARDWARE, & SKINS

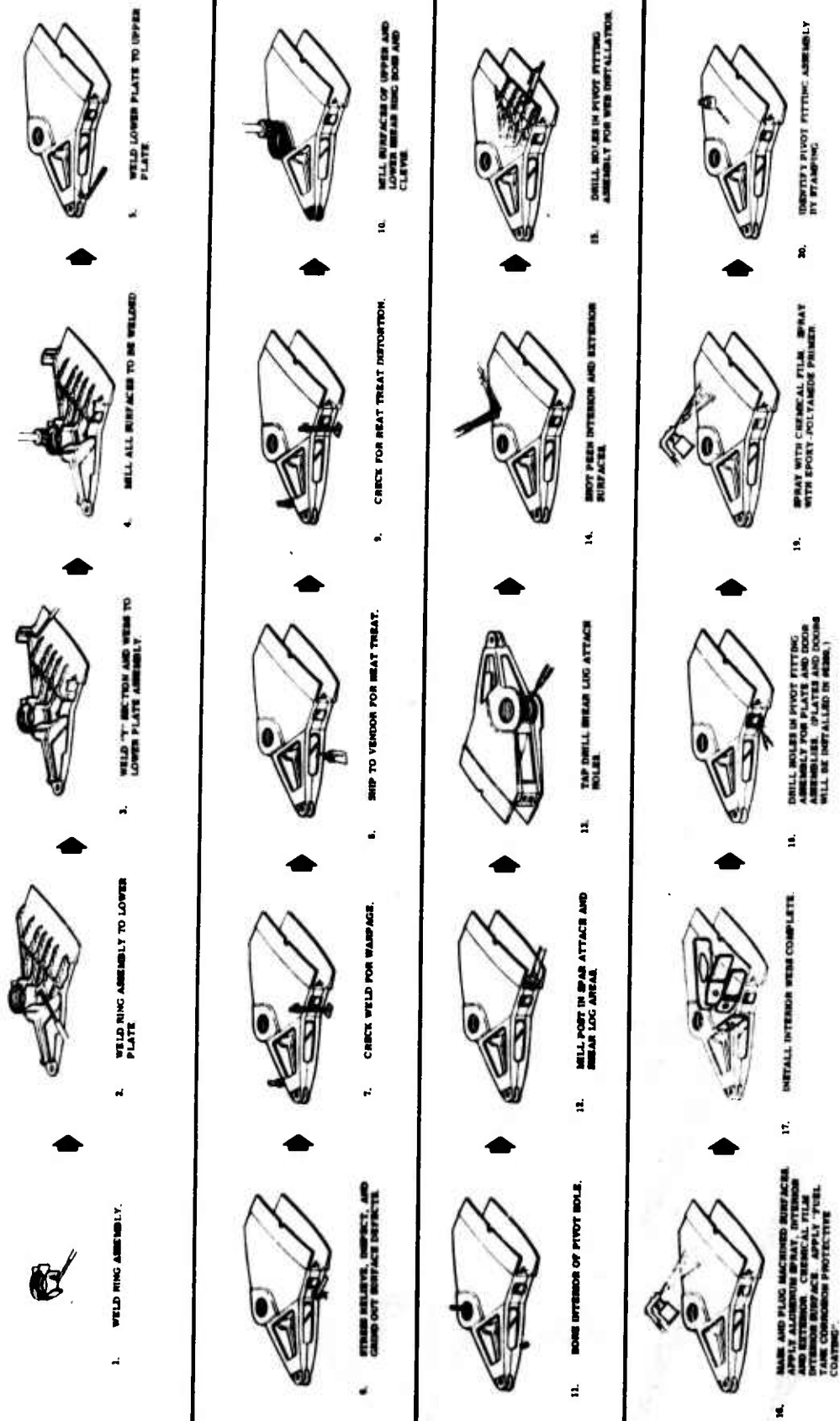


Figure 115 Baseline 633-RW000 Pivot Fitting Manufacturing Sequence

The fixture will first be used to receive and locate the pivot fitting assembly, bulkheads, and spars.

The fixture will then be used to provide a method for positioning, holding, and locating hole drilling tooling for drilling holes common to the upper skin, understructure, and lower skin.

Upon completion of the hole drilling operations on the lower skin, all faying surfaces formed by the front and rear spars will have fuel sealing applications and all understructure will be permanently attached to the lower skin with taper-lok bolts. Following the hole drilling operations on the upper skin, the skin is removed and self-locking nuts are installed in the upper attach surfaces of the understructure. All faying surfaces formed by the front and rear spars will then have fuel sealing applications and all understructure will be permanently attached to the upper skin with bolts.

9.2.2 Preliminary Manufacturing Plan for 633-RW001

The manufacturing approach for the number one ranked design is discussed below. The process is defined in Table XXVI and is also shown schematically in Figures 116 through 126.

9.2.2.1 Front and Rear Spar Fabrication

The front and rear spars are one-piece, integrally machined structure made from a 2024-T8511 aluminum alloy extrusion. Numerically controlled milling machines are used to pocket and contour the spar (Fig. 116).

9.2.2.2 Intermediate Spar Fabrication

The intermediate spars are one-piece, machined and chem-etched structure made from a 2024-T8511 aluminum alloy extrusion.

The upper caps (flanges) are contour machined to the inner surface of the upper skin and machine tapered. The spar web is routed to the proper depth profile. The spar web and diagonal legs are then chem-etched to finish dimensions.

The manufacturing steps are shown in Figure 117.

TABLE XXVI 633-RW001 WING BOX ANALYSIS
(Page 1 of 4)

DESCRIPTION	QTY	MATERIAL		FAB. METHOD
		TYPE	SIZE	
-7 UPPER SKIN	1	2024-T851	1.0 X 77 X 360	PURCHASE ROUGH SIZE, MILL TAPER, BRAKE FORM CONTOUR, ROUT PERIPHERY
-9 & -37 FRONT & REAR SPAR	2	2024-T8511	2.50 X 11 X 370	ROUGH EXTRUSION - N/C MILL COMPLETE
-11, -13, -15, -17, -19, -21, -23, -25, -27 AUX. SPARS	9	2024-T8511	Y EXTRU. LENGTH RANGE: 72 TO 128	ETCH LEGS TO THICKNESS, MILL "Y" BOSSES, ROUT LEG & FUEL CUTOUTS
-29, -31, -33 OUTBOARD SPARS	3	2024-T81	12 X 26	SHEAR TO SIZE, HYDROPRESS FORM, HEAT TREAT, STRAIGHTEN
-35 CLOSURE BULKHEAD <u>-39 ASSY</u>	1		CASTING	N/C MILL UPPER & LOWER SURFACES
-57 BULKHEAD	1	2024-T851	3.0 X 12 X 60	SAW ROUGH SIZE, N/C MILL COMPLETE
-59 & -61 ATTACH ANGLE <u>-41 ASSY</u>	18	2024-T8511	2 & 4 IN. LENGTH	EXTRU. - SAW TO LENGTH
-63, -65, -67, -69 BULKHEADS	4	2024-T851	3.0 X 11 X 15	SAW ROUGH SIZE, MILL COMPLETE
-71 & -73 ATTACH ANGLE	18	2024-T8511	2 & 4 IN. LENGTH	EXTRU. - SAW TO LENGTH

TABLE XXVI 633-RW001 WING BOX ANALYSIS
(Page 2 of 4)

DESCRIPTION	QTY	MATERIAL		FAB. METHOD
		TYPE	SIZE	
<u>-43 ASSY</u>				
-75 BULKHEAD	1	2024-T851	3.0 X 10 X 46	SAW ROUGH SIZE, N/C MILL COMPLETE
-77 & -79 ATTACH ANGLE	18	2024-T8511	2 - 4 IN. LENGTH	EXTRU. - SAW TO LENGTH
<u>-45 ASSY</u>				
-81, -83, -85, -87 BULKHEAD	4	2024-T851	3.0 X 10 X 14	SAW ROUGH SIZE, MILL COMPLETE
-89 & -91 ATTACH ANGLES	18	2024-T8511	1.5 - 3	SAW TO LENGTH
<u>-47 ASSY</u>				
-93 BULKHEAD	1	2024-T851	3.0 X 7 X 35	SAW ROUGH SIZE, N/C MILL COMPLETE
-95 & -97 ATTACH ANGLE	18	2024-T8511	1.5 - 3	SAW TO LENGTH
-99 & -101 OUTBOARD UPPER & LOWER SKINS	2	2024-T851	32 X 36	SHEAR OVERSIZE, ROLL CONTOUR, ROUT TO SIZE
<u>-49 ASSY</u>				
-103 LAMINATES	32	2024-T851	APPROX. LENGTH RANGE: 40 - 380	SHEAR TO LENGTH WITH WIDTH FOR SEVERAL PARTS. ROLL FORM, TAPER ETCH, ROUGH SAW TO SEPARATE PARTS, ROUT TO SIZE

TABLE XXVI 633-RW001 WING BOX ANALYSIS
(Page 3 of 4)

DESCRIPTION	QTY	MATERIAL		FAB. METHOD
		TYPE	SIZE	
-105, -107, -109 LOWER SPAR CAPS	5	2024-T8511	EXTRUSIONS - 380	N/C MILL TO SIZE - ROUT FUEL CUTOUTS
-111, -113, -115, -117, -119 BULKHEAD CAPS	5	2024-T8511	EXTRUSIONS - 30 TO 60	N/C MILL TO SIZE - ROUT FUEL CUTOUTS

ASSEMBLY -49

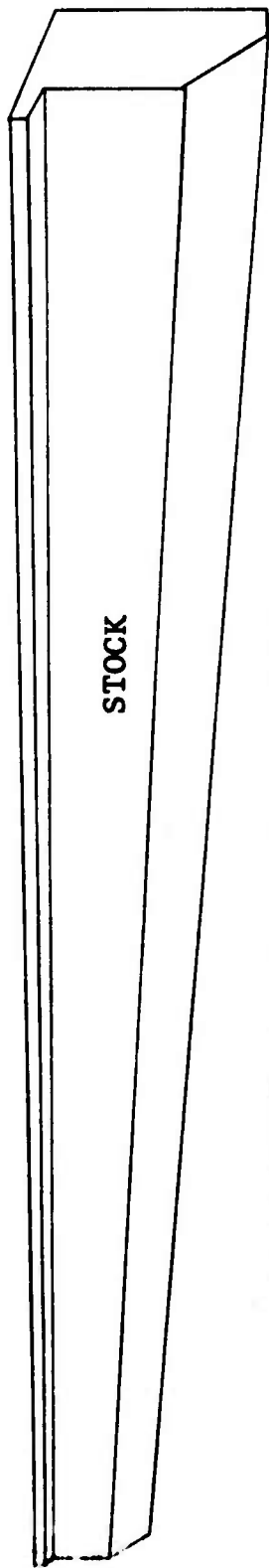
NOTE: PRIOR TO BONDING, TRAPPED AIR MUST BE REMOVED FROM LAMINATES BY APPLYING VACUUM TO ASSEMBLY AND OUTER SURFACE OF BLANKET WITH SLIGHTLY HIGHER PRESSURE ON OUTER SIDE. MODIFICATION OF AUTO-CLAVE OR A SPECIAL TOOL FOR THIS PURPOSE WILL BE REQUIRED.

-53 (?) PIVOT FITTING
ASSY. UPPER STEEL
FORGING LWR ALUM
FORGING RING
SPLICE BULKHEAD
FAIL SAFE STRAPS
SPAR ATTACH ANGLES
STEEL BUSHINGS

PREFIT SPAR & AUX. SPAR CAPS, SKINS, AND BULKHEAD CAPS. CLEAN FOR BONDING, APPLY ADHESIVE AND ASSEMBLE DETAILS, POSITION ASSY IN BONDING FIXTURE AND INSTALL VACUUM BLANKET. (SEE NOTE) BOND IN AUTOCLAVE. CLEAN TO REMOVE EXCESS ADHESIVE

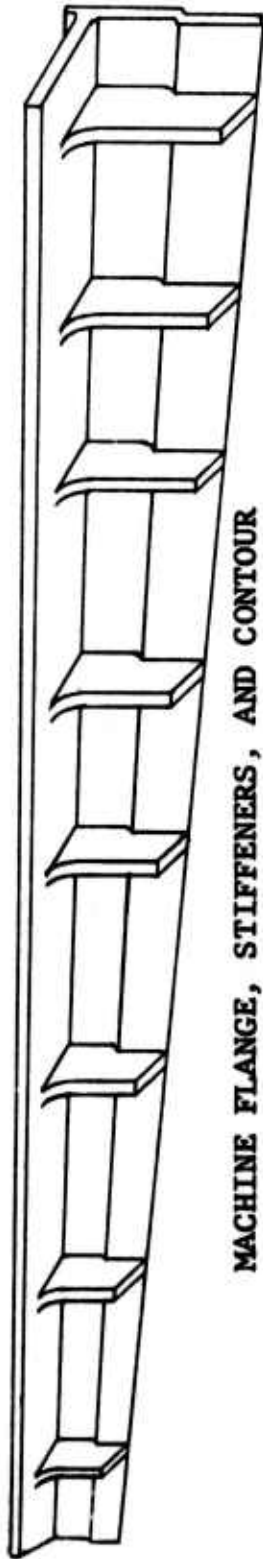
TABLE XXVI 633-RW001 WING BOX ANALYSIS
(Page 4 of 4)

DESCRIPTION	QTY	MATERIAL		FAB. METHOD
		TYPE	SIZE	
<u>-1 ASSEMBLY</u>				<ul style="list-style-type: none"> o PREFIT LWR. ALUM. FORGING TO -49 SKIN ASSY. CLEAN AND BOND TO SKIN IN AUTOCLAVE o INSTALL PIVOT DETAILS o INSTALL SPARS, BULKHEADS, PLUMBING, MISC. EQUIPMENT AND UPPER SKIN o INSTALL OUTBD. SPARS, CLOSURE BULKHEAD, MISC. HARDWARE, AND SKINS o SIZE PIVOT BUSHINGS



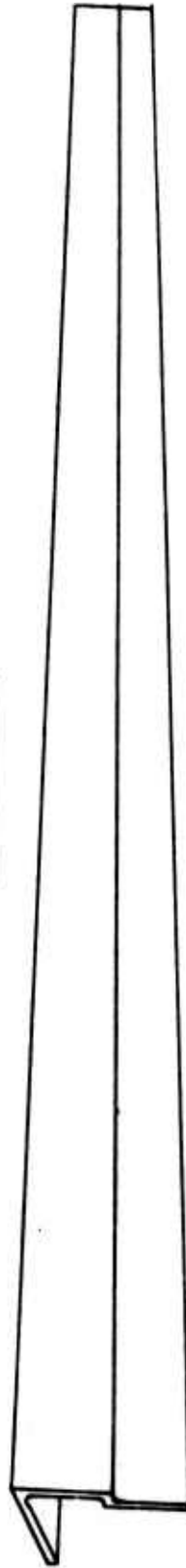
STOCK

MAKE FRONT AND REAR SPAR FROM ALUM. EXTRUSION



MACHINE FLANGE, STIFFENERS, AND CONTOUR

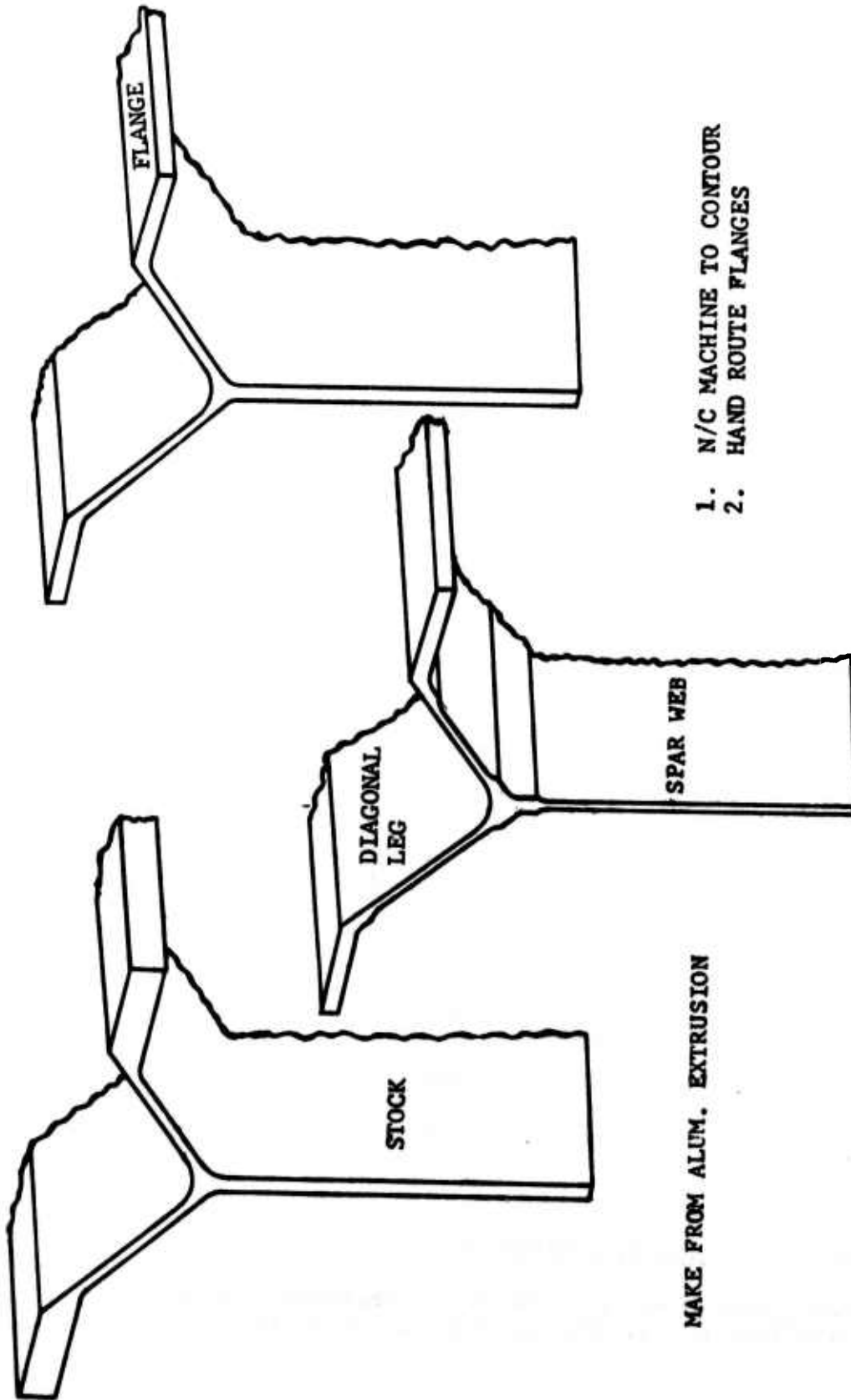
FRONT SPAR



MACHINE FLANGE, STIFFENERS, AND CONTOUR

REAR SPAR

Figure 116 Basic Manufacturing Front and Rear Spars



3. CHEM-ETCH SPAR WEB AND DIAGONAL LEGS

Figure 117 Basic Manufacturing Intermediate Spars

9.2.2.3 Lower Skin Laminates

The laminates will be formed from 2024-T81 aluminum alloy sheet stock, ranging from .063 to .071 inch thickness.

Laminates are to be sheared, rolled to contour, taper etched, routed, and deburred in preparation for bonding operation (Fig. 118).

9.2.2.4 Lower Spar Cap Fabrication

The lower spar caps are tapered and stepped "Tee" type structure machined from 2024-T8511 aluminum alloy extrusions.

The lower caps are contour machined to the outer contour of the lower wing surface. Numerical control machining techniques will be used to generate maximum effective metal removal and ultimate dimensional control.

The manufacturing steps are shown in Figure 119.

9.2.2.5 Upper Skin Fabrication

The upper skin is a one-piece tapered plate containing the entire upper surface contour and machined from 2024-T851 aluminum alloy.

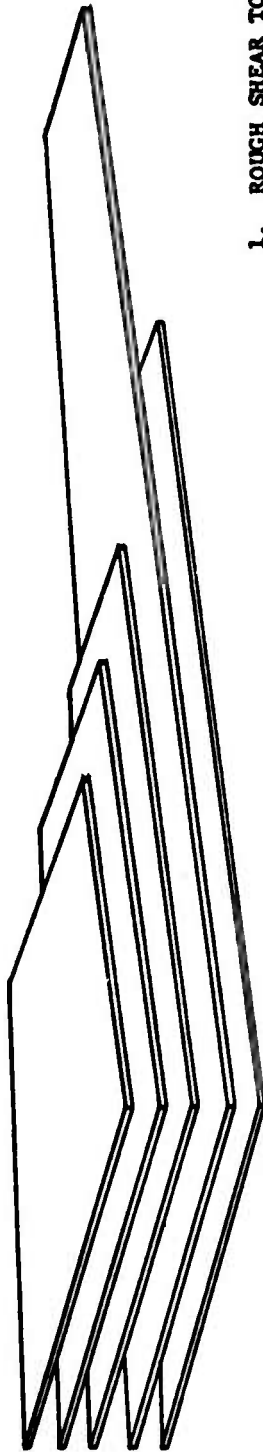
Basic tapers on the inside skin will be machined on a numerical control skin mill. Contour will be formed on 1000 ton numerical control brake. The skin will then be routed to finish dimensions (Fig. 120).

9.2.2.6 Fabrication of Bulkheads and Ribs

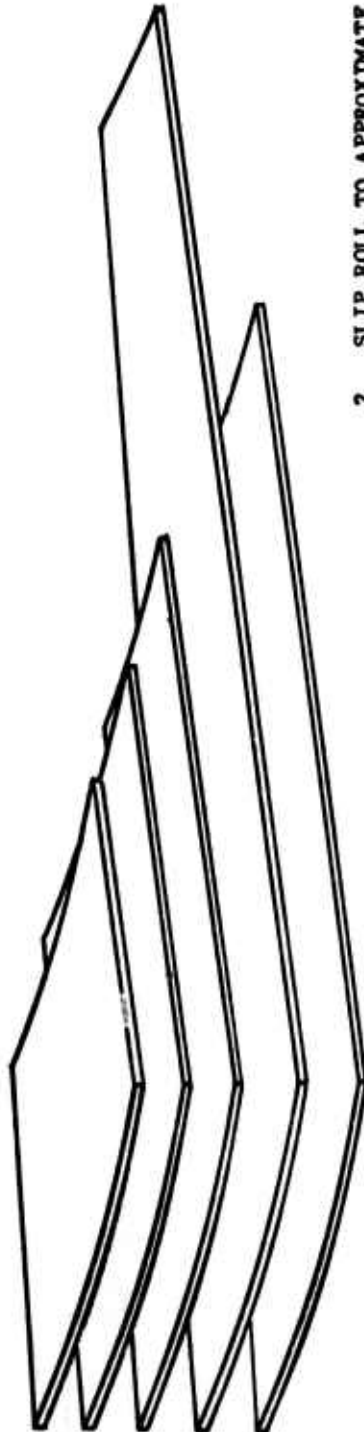
Major bulkheads and ribs may be fabricated by machining from plate or by machining from castings. Production quantity and rate will determine the most economical method. Figure 121 depicts the two methods.

9.2.2.7 Lower Pivot Plate Fabrication

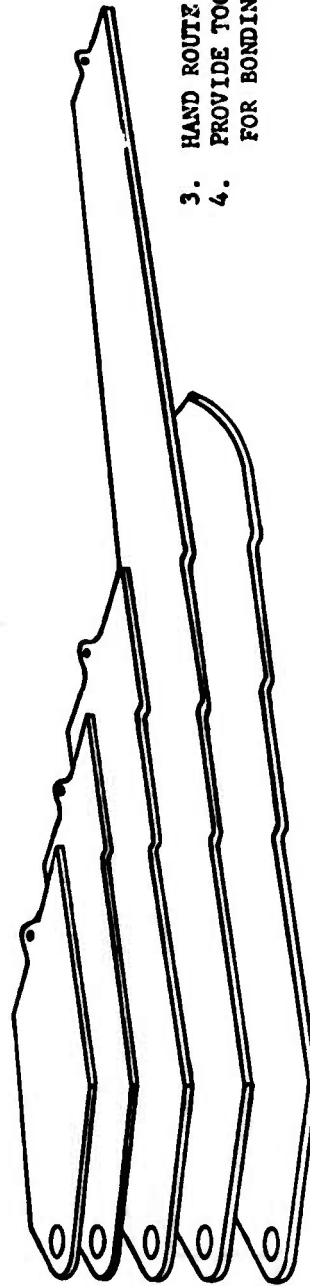
The lower pivot plate is a one-piece integrally machined structure made from a 2024-T851 aluminum alloy forging.



1. ROUGH SHEAR TO SIZE

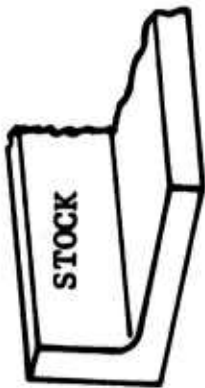


2. SLIP ROLL TO APPROXIMATE CONTOUR

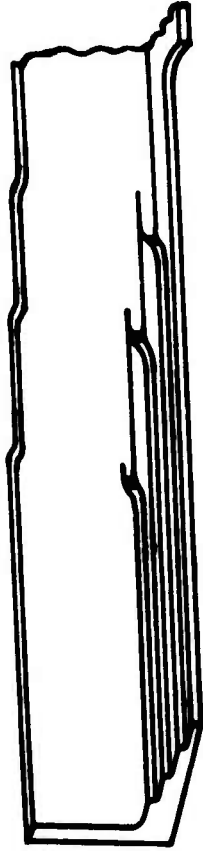


3. HAND ROUTE TO SIZE
4. PROVIDE TOOLING TABS
FOR BONDING OPERATION

Figure 118 Basic Manufacturing Lower Skin Laminates

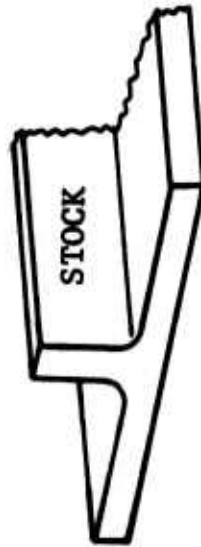


MAKE FROM ALUM. EXTRUSION

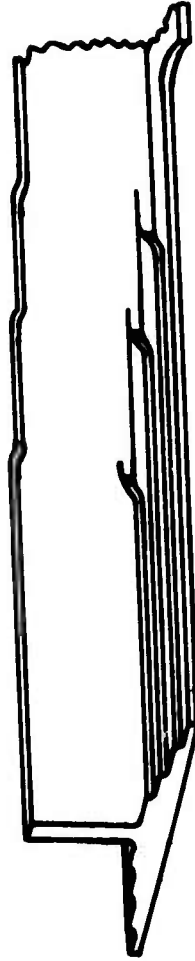


1. MACHINE STEPS & CONTOUR
2. HAND ROUTE FLANGES

FRONT & REAR
LWR SPAR CAPS



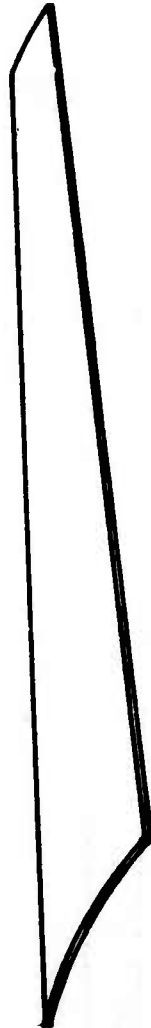
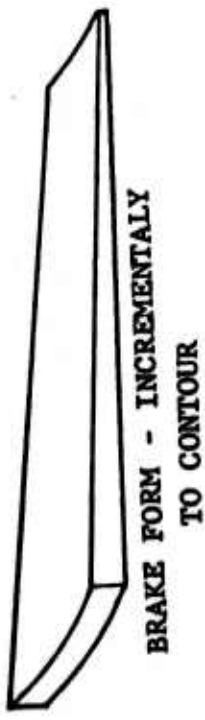
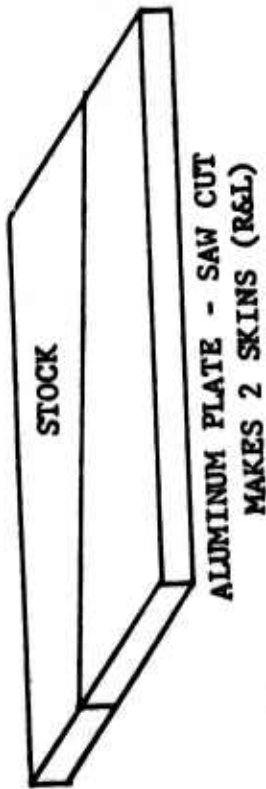
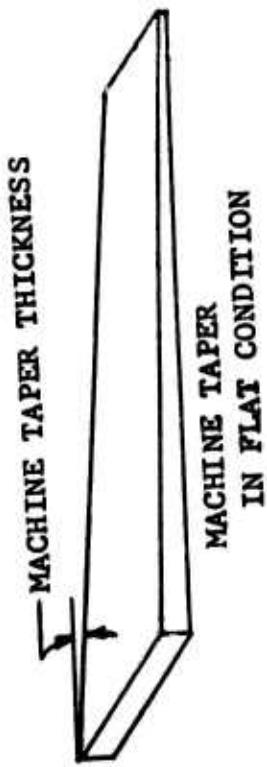
MAKE FROM ALUM. EXTRUSION



1. MACHINE STEPS & CONTOUR
2. HAND ROUTE FLANGES

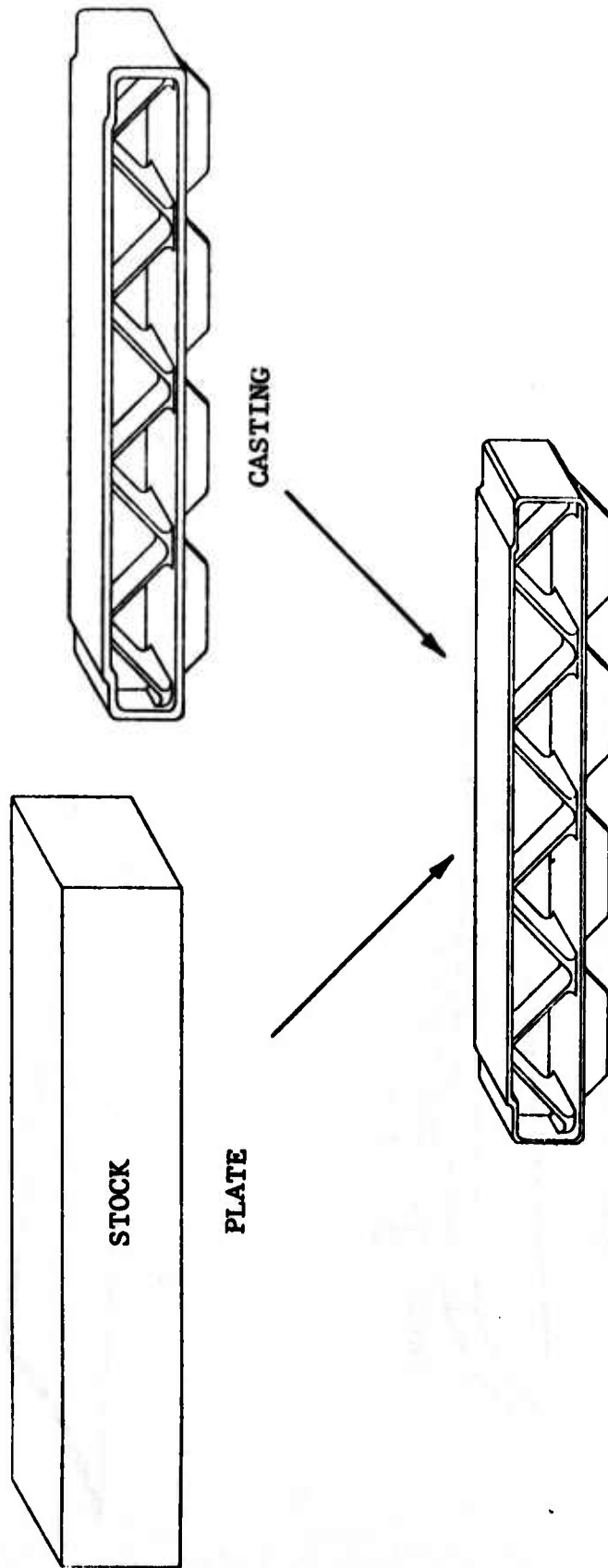
INTERMEDIATE
LWR SPAR CAPS

Figure 119 Basic Manufacturing Lower Spar Caps



- 1 ROUTE TO PERIMETER
- 2 MACHINE TAPER - INBOARD END

Figure 120 Basic Manufacturing Upper Skin



MACHINE FROM PLATE
OR
MACHINE FROM CASTING

Figure 121 Basic Manufacturing Bulkheads and Ribs

The outer surface of the pivot plate is contour machined to the inner surface of the inboard lower skin laminates. The inner surface of the pivot plate is machined to finish dimensions.

The manufacturing steps are shown in Figure 122.

9.2.2.8 Pivot Fitting Fabrication

The pivot fitting consists of an upper pivot plate, shear ring, root spars, failsafe straps, and steel bushings.

The upper pivot plate is a one-piece integrally machined structure made from a HY-180 steel forging. The outer surface of the pivot plate is contour machined to the outer contour of the upper wing surface.

The shear ring is a one-piece integrally machined structure made from a 2024-T851 aluminum alloy forging.

The root spars are integrally machined structure made from a 2024-T8511 aluminum alloy extrusion.

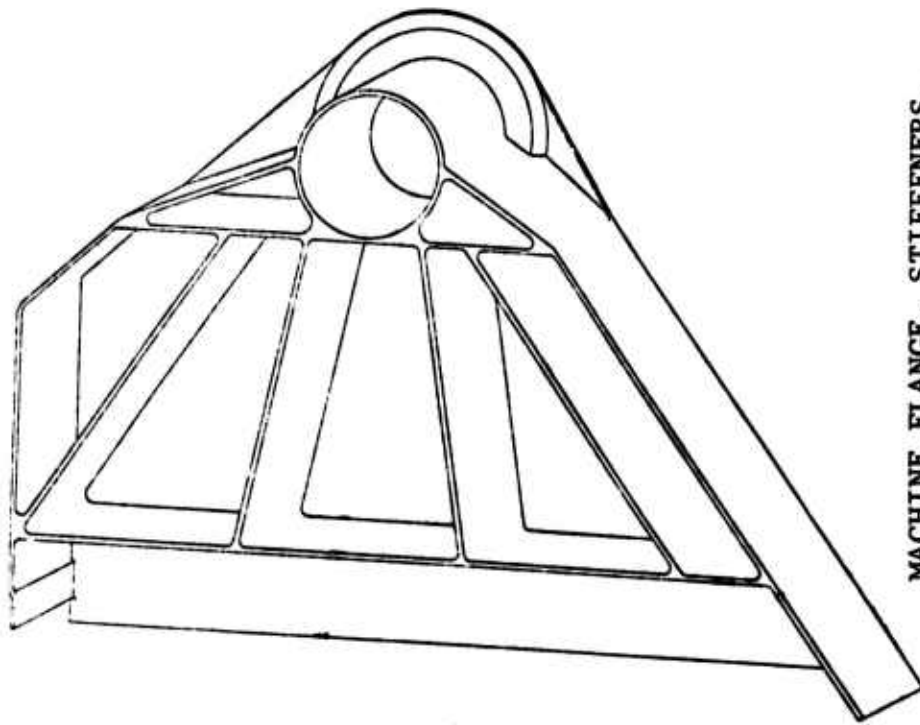
The failsafe straps and bushings are machined structure made from HY-180 steel plate and forgings respectively.

9.2.2.9 Assembly of Lower Surface

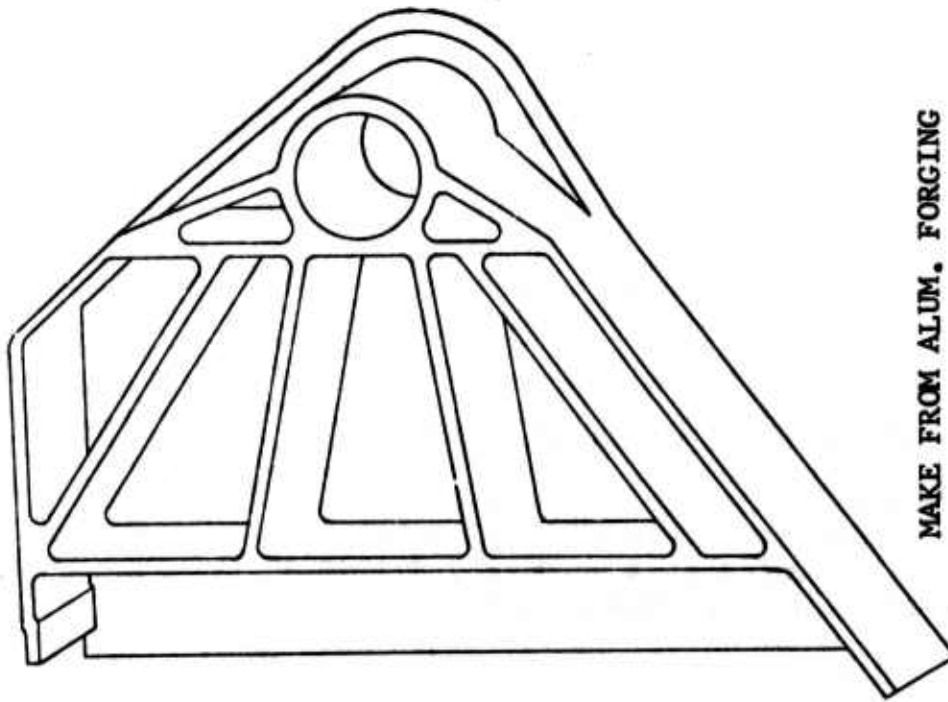
This assembly consists of the aluminum skin sheets laminated and joined to the lower spar caps which were step machined in detail to provide overlap of each individual skin laminate with the spar cap flange. This total assembly is bonded in one bond cycle (Fig. 123).

9.2.2.10 Assembly of Lower Surface and Lower Pivot Plate

This assembly consists of the laminated lower skin panel (skin sheets/spar caps) joined to the lower pivot plate (Fig. 124). This assembly is bonded as a secondary operation to the bonding operation described in paragraph 9.2.2.9.

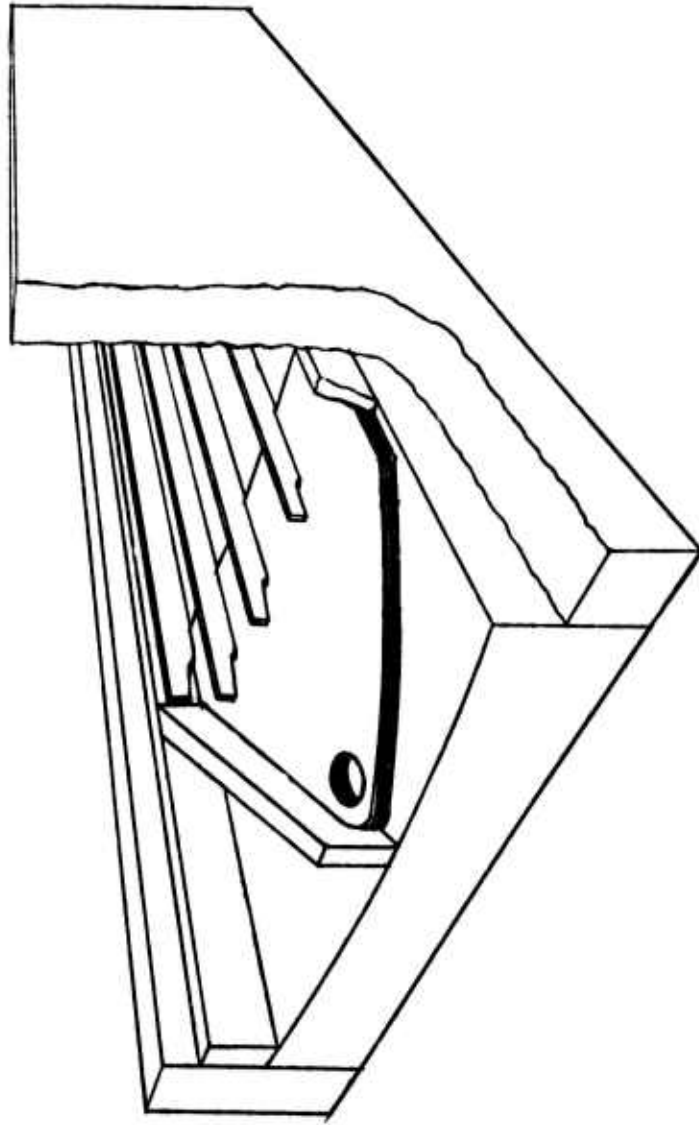


**MACHINE FLANGE, STIFFENERS,
AND CONTOUR**



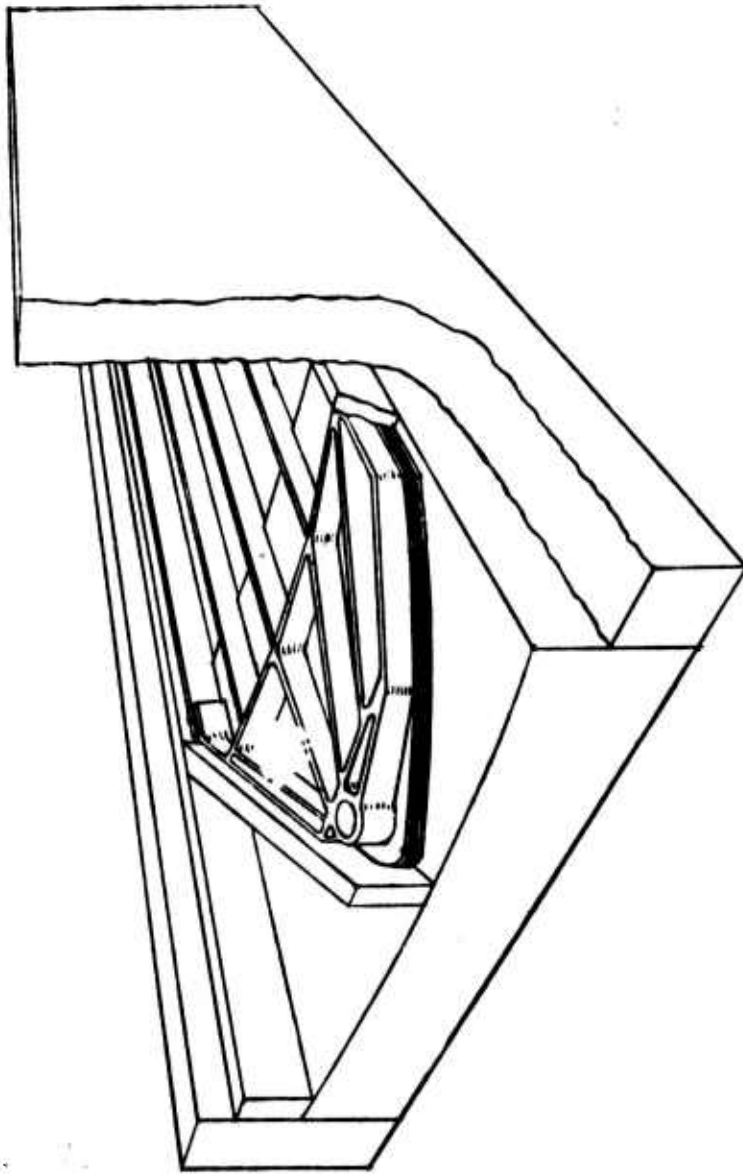
MAKE FROM ALUM. FORGING

Figure 122 Basic Manufacturing Lower Pivot Plate



1. LOCATE FRONT, INTERMEDIATE, AND REAR LWR SPAR CAPS FROM LOCATORS ON FIXTURE
2. LOCATE LAMINATES FROM TOOLING HOLES AT ENDS OF DETAILS
3. APPLY ADHESIVE BONDING TAPE
4. ADHESIVE BOND AND CURE LAMINATED LOWER SKIN PANEL

Figure 123 Secondary Manufacturing Lower Surface Laminated Skin Panel Adhesive Bonded Assembly



1. LOCATE LOWER SKIN PANEL AND PIVOT PLATE FROM LOCATORS ON FIXTURE
2. APPLY ADHESIVE BONDING TAPE
3. ADHESIVE BOND AND CURE LAMINATED LOWER SKIN PANEL & PIVOT PLATE

Figure 124 Secondary Manufacturing Lower Skin Panel & Lower
Pivot Plate Adhesive Bonded Assembly

9.2.2.11 Final Assembly of Wing Box

A major assembly and drill fixture commonly known as a "wing buck" will be required to perform final assembly operations of the box.

The fixture will first be used to receive and locate the basic lower skin panel assembly, pivot fitting, bulkheads, and spars. The fixture will then be used to provide a method of positioning, holding, and locating hole drilling tooling for drilling holes common to the upper skin, understructure, and lower skin.

Upon completion of the hole drilling operations on the lower skin and understructure all faying surfaces formed by the front and rear spars will have fuel sealing applications and all understructure will be permanently attached to the lower skin with bolts (Fig. 125). Following the hole drilling operation on the upper skin all faying surfaces formed by the upper skin will have fuel sealing applications and all understructure will be permanently attached to the upper skin with blind fasteners (Fig. 126).

9.2.3 Preliminary Manufacturing Plan for 633-RW003

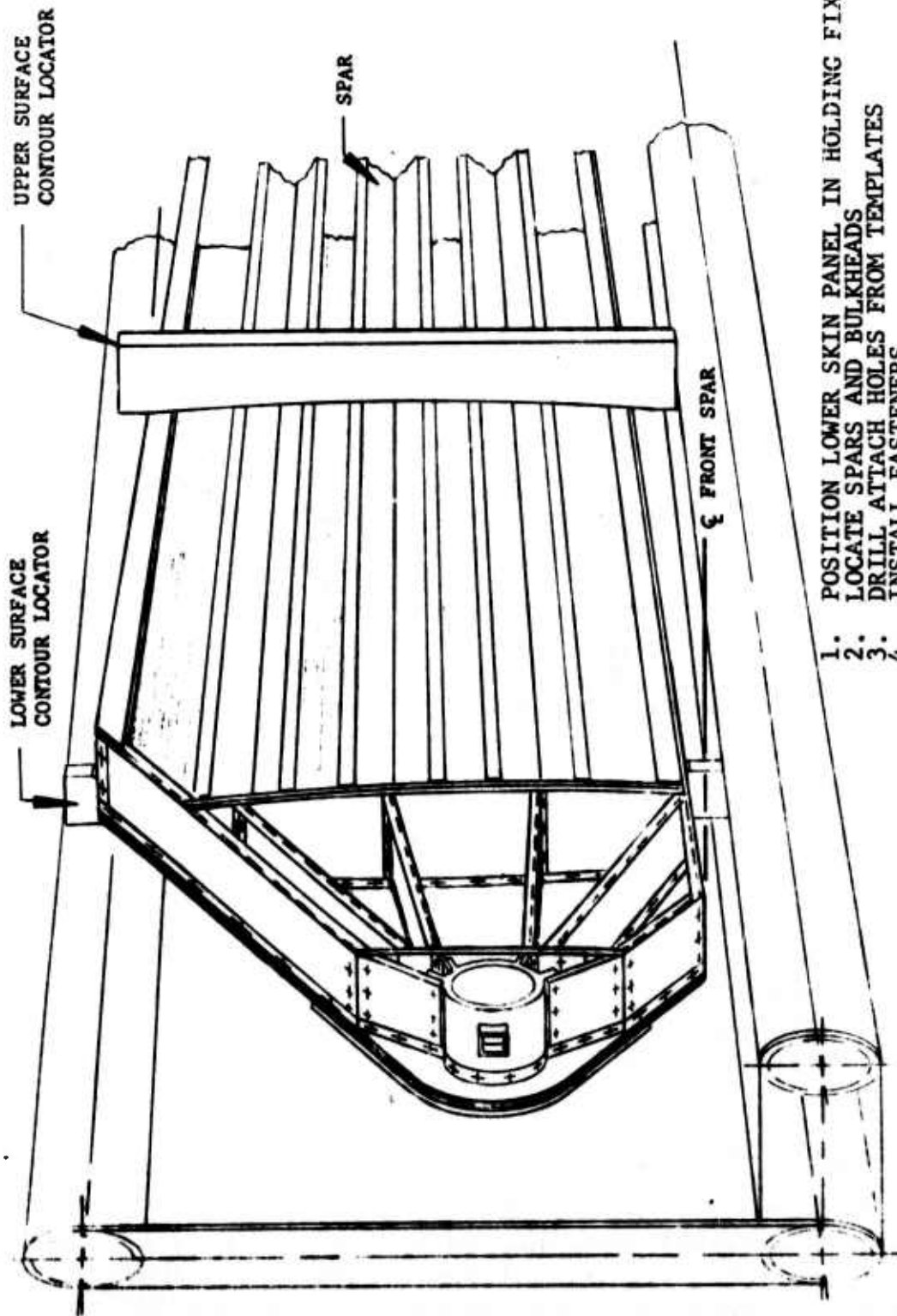
The manufacturing approach for the composite wing design is discussed below. The processes are shown schematically in Figures 127 through 131.

9.2.3.1 Front and Rear Spar Fabrication

The front and rear spars are made of graphite/epoxy material with fiberglass material for the upper cap sealing grooves. The spars are manufactured by draping machine-layed, "dinked-out" pieces into a hard female tool and cured with a male rubber punch in place. In this manner, skin/spar cap faying surface geometry can be well maintained (Fig. 127).

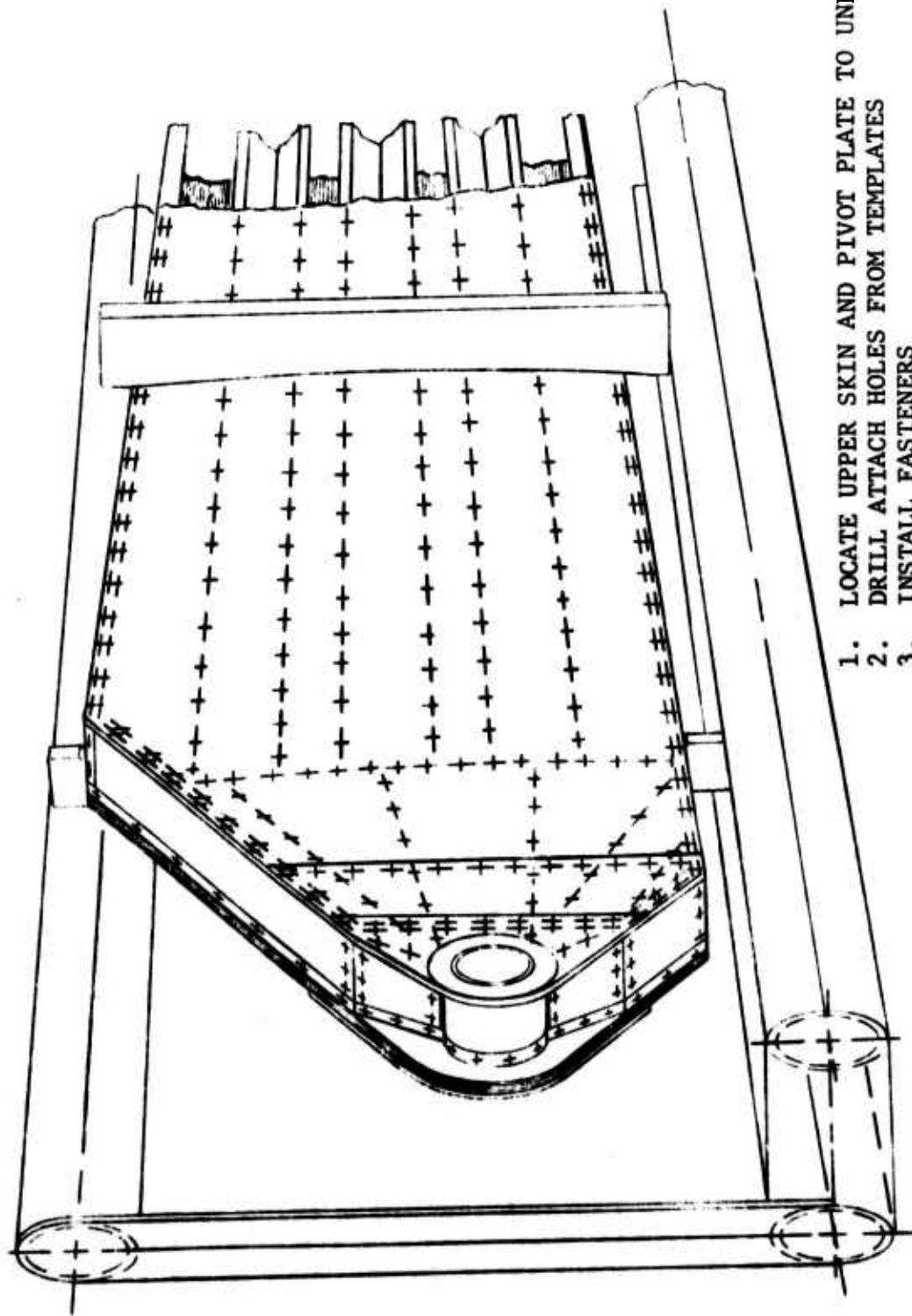
9.2.3.2 Intermediate Spar Fabrication

The intermediate spars are made of graphite/epoxy material and nomex cloth material with chopped fiberglass/epoxy pre-cast material for the lower skin transition members. The spars are machine-layed, "dinked-out", and draped over the tool and cured (Fig. 128).



1. POSITION LOWER SKIN PANEL IN HOLDING FIXTURE
2. LOCATE SPARS AND BULKHEADS
3. DRILL ATTACH HOLES FROM TEMPLATES
4. INSTALL FASTENERS

Figure 125 Secondary Manufacturing Lower Skin Panel and Understructure Assembly



1. LOCATE UPPER SKIN AND PIVOT PLATE TO UNDERSTRUCTURE
2. DRILL ATTACH HOLES FROM TEMPLATES
3. INSTALL FASTENERS

Figure 126 Secondary Manufacturing Final Assembly Upper Skin and Understructure

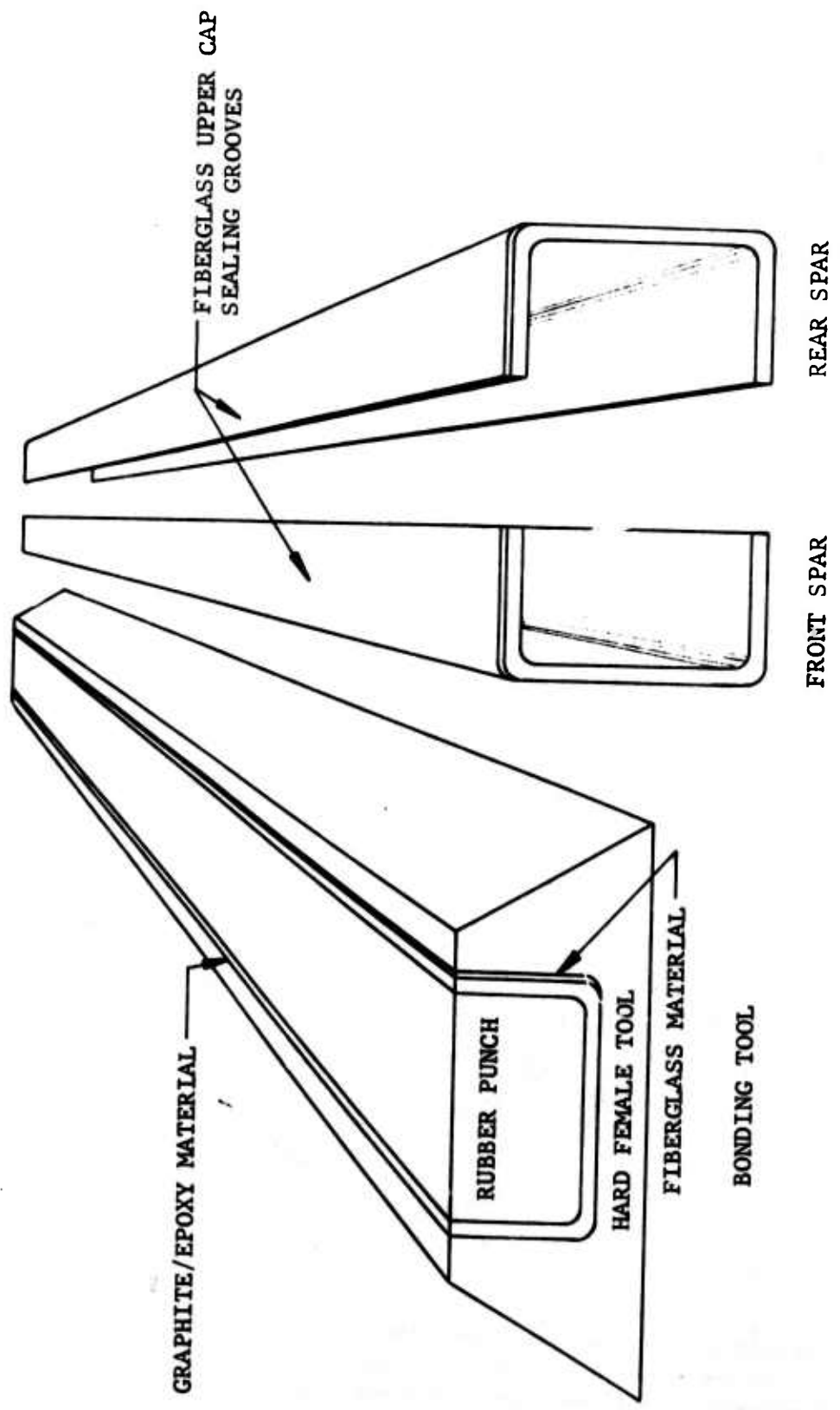


Figure 127 Basic Manufacturing Front and Rear Spars

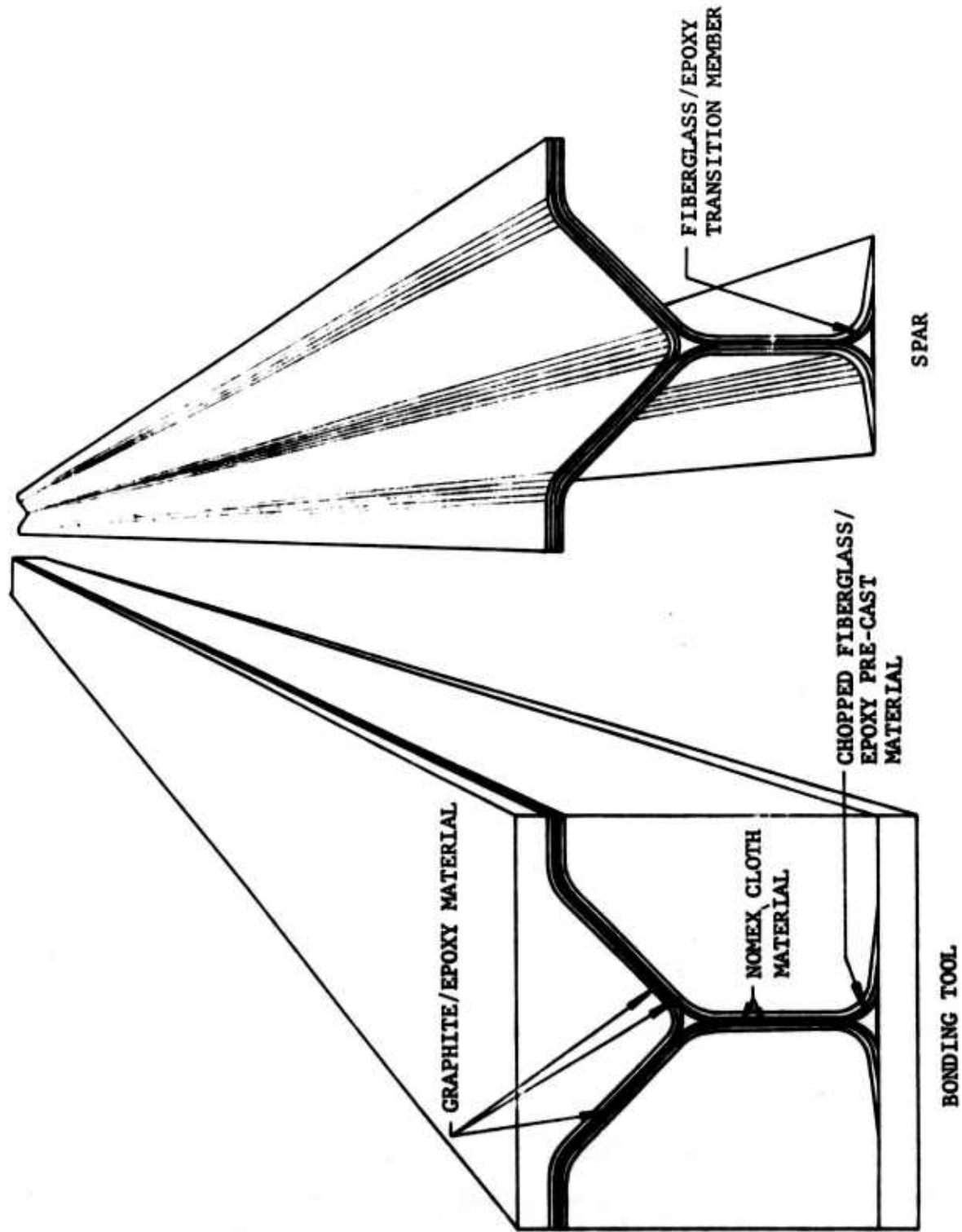


Figure 128 Basic Manufacturing Intermediate Spars

9.2.3.3 Fabrication of Bulkheads

The bulkheads are made of graphite/epoxy material. The bulkheads are made by draping machine-layed, "dinked-out" pieces into a hard female tool and cured with a male rubber punch in place to maintain outer surface contour.

9.2.3.4 Fabrication of Closure Bulkhead

The closure bulkhead is a one-piece, machined casting made from A356-T6 aluminum casting alloy.

9.2.3.5 Pivot Fitting Fabrication

The pivot fitting consists of upper and lower skin transition doublers, a shear fitting, root spars, and steel bushings.

The upper and lower skin transition doublers are machined structure made from 6Al-4V Beta annealed titanium. The outer surfaces of the doublers are contour machined.

The shear fitting is a one-piece integrally machined structure made from 6Al-4V Beta annealed titanium.

The root spars are integrally machined structure made from 6Al-4V Beta annealed titanium.

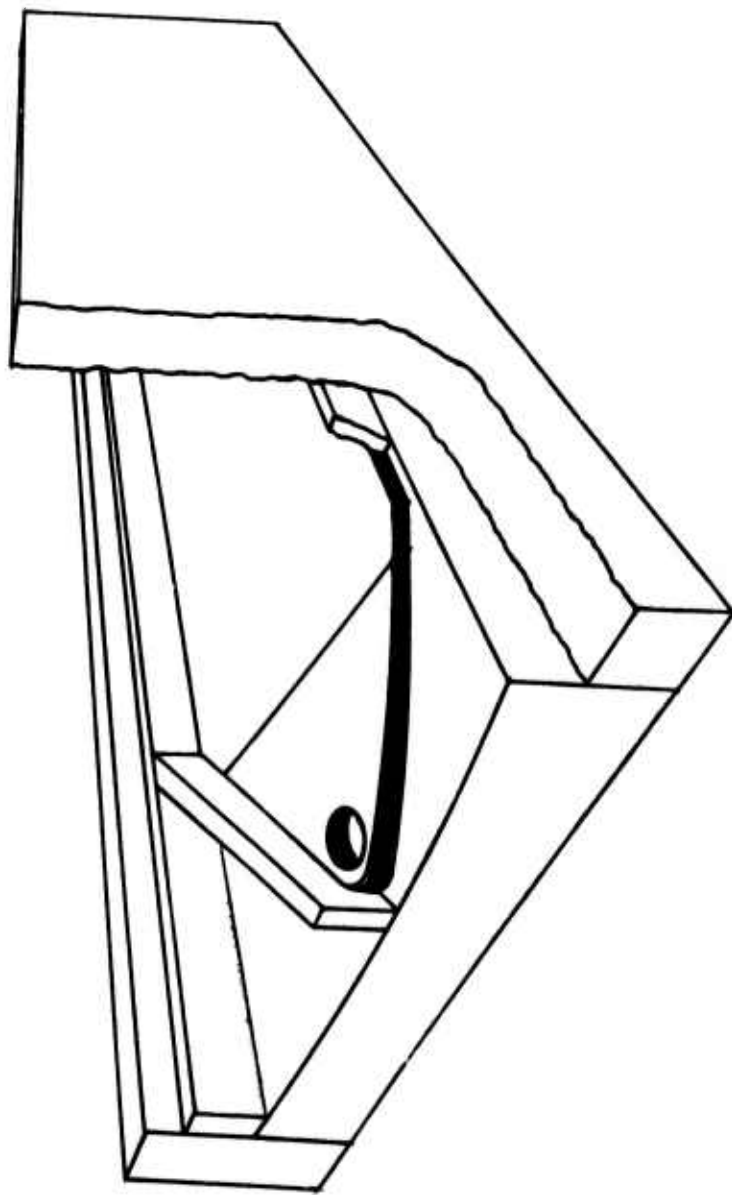
The bushings are machined structure made from 17-4PH steel forgings.

9.2.3.6 Assembly of Upper Skin

The upper skin is a bonded assembly consisting of a graphite/epoxy skin and machined titanium transition doublers (Pivot Area). The graphite/epoxy skin material is layed-up (machine-layup) and co-cured with the transition doublers (Fig. 129). It is assumed that the machine-layup of the graphite/epoxy material requires no pre-bleeding.

9.2.3.7 Assembly of Lower Skin and Intermediate Spars

The lower skin is a bonded assembly consisting of a graphite/epoxy skin, the intermediate spars, and machined titanium transition doublers (Pivot Area). The skins' graphite/epoxy outer plies



1. LOCATE TRANSITION DOUBLERS AND GRAPHITE/EPOXY SKIN MATERIAL ON FIXTURE
2. CURE SKIN PANEL

Figure 129 Secondary Manufacturing Upper Skin Bonded Assembly

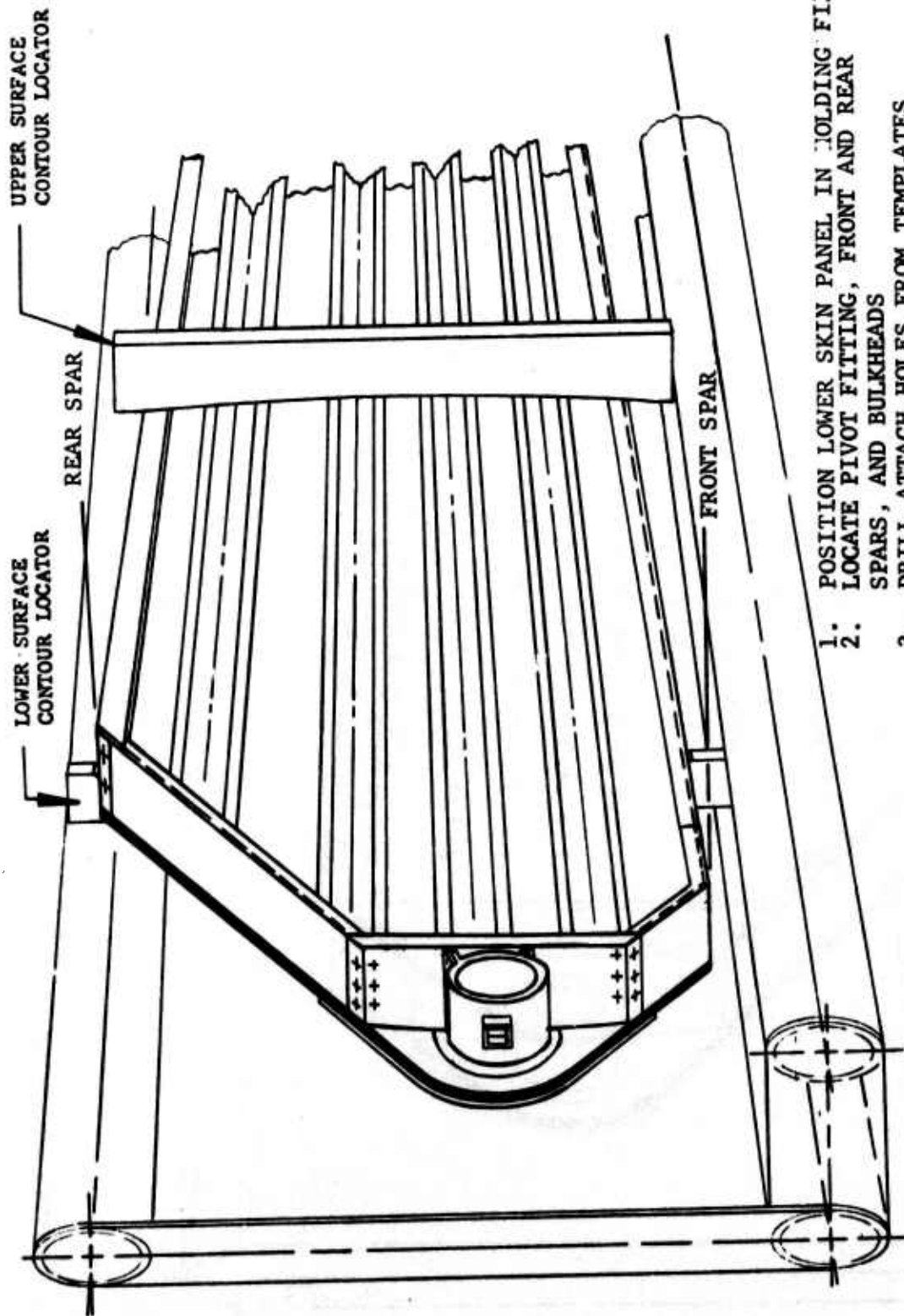
are first layed-up (machine-layup) and draped and the intermediate spars located. The skins' inner plies are then layed-up and draped and the entire lower skin assembly is co-cured. It is assumed that the machine-layup of the graphite/epoxy material requires no pre-bleeding; i.e., the graphite/epoxy tape is "net material".

9.2.3.8 Final Assembly of Wing Box

A major assembly and drill fixture commonly known as a "wing buck" will be required to perform final assembly operations of the box.

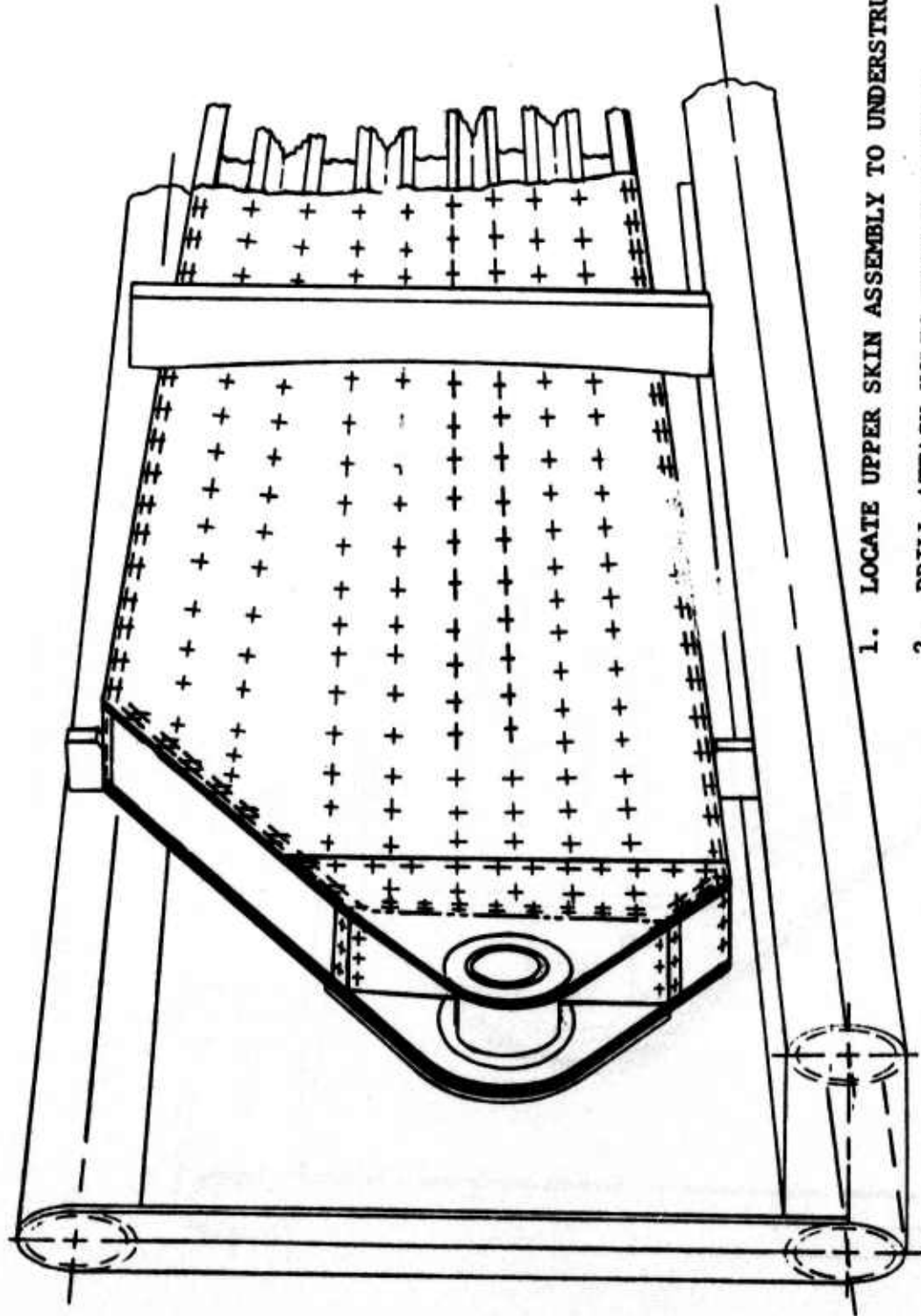
The fixture will first be used to receive and locate the basic lower skin panel assembly and understructure (pivot fitting, bulkheads, and front and rear spars). The fixture will then be used to provide a method of positioning, holding, and locating hole drilling tooling for drilling holes common to the upper skin, understructure, and lower skin assembly.

Upon completion of the hole drilling operations on the lower skin and understructure all faying surfaces formed by the front and rear spars will have fuel sealing applications and all understructure will be permanently attached to the lower skin with bolts (Fig. 130). Following the hole drilling operation on the upper skin all faying surfaces formed by the upper skin will have fuel sealing applications and all understructure will be permanently attached to the upper skin with blind fasteners (Fig. 131).



1. POSITION LOWER SKIN PANEL IN FOLDING FIXTURE
2. LOCATE PIVOT FITTING, FRONT AND REAR SPARS, AND BULKHEADS
3. DRILL ATTACH HOLES FROM TEMPLATES
4. INSTALL FASTENERS

Figure 130 Secondary Manufacturing Lower Skin Panel and Understructure Assembly



1. LOCATE UPPER SKIN ASSEMBLY TO UNDERSTRUCTURE
2. DRILL ATTACH HOLES FROM TEMPLATES
3. INSTALL FASTENERS

Figure 131 Secondary Manufacturing Final Assembly Upper Skin and Understructure

SECTION X
CONCLUSIONS

The overall goal of the program was to establish the applicability and payoffs, in terms of acquisition cost and weight, of innovative structural concepts to a variable camber supercritical wing box configuration. The program has met this goal by developing preliminary designs that provide cost savings up to 12.3% and weight savings up to 27%.

10.1 COST EVALUATIONS

Preliminary design configurations were developed and analyzed that produce the following results:

633RW001	12.3% savings	
633RW002	10.7% savings	
633RW003	29% increase for Vendor A 13.9% increase for Vendor B 4.4% increase for \$20/Lb	See paragraph 10.1.2 & Table XXVIII

The two metallic configurations produce a cost savings while the composite configuration gives a cost increase. Detailed examination of the items that make up the cost for the composite box as compared to the baseline is shown in Table XXVII. It is evident from this table that the total cost for the composite design is dominated by the material cost.

10.1.1 Factors That Influence Costs

The costs shown in Table XXVII above were derived by the Fort Worth Plant Industrial Engineering, Tooling, Material, and Division Estimating Departments. The costs used in this section are those defined by the groundrules listed in Section VIII and excludes profit. The numbers for 1980 year dollars reflect the professional opinions of the estimating personnel in these departments with regard to the effect that certain economic factors will have on wing box prices in 1980 year dollars. The economic factors considered to be most significant in increasing costs are: in-house labor rates, vendor labor rates, transportation, energy cost changes, and general economic inflation. Significant decreasing costs at the Fort Worth Plant are: diminishing overhead and administrative costs, learning, technology advancement and production quantity increases. It should be noted however, that all of the designs costed were estimated by applying the same procedure to each design.

**TABLE XXVII COST BREAKDOWN
(200 Unit Average in 1980 Dollars)**

CONFIGURATION	MATERIAL	TOOLING	FABRICATION	TOTAL
633RW000-1	42903	13413	195838	252154
633RW001-1	42932	13622	164737	221291
633RW002-1	36044	13584	175736	225364
633RW002-3	35257	13584	184777	233618
633RW003-1	188875	14203	137708	340786
633RW003-3	183072	14800	127976	325848

General Dynamics recognizes that other assumptions could have been made in predicting the 1980 prices, particularly in the area of composite materials.

10.1.2 Composite Material Costs

The cost of the composite material was based upon \$.95 per foot for 3 inch tape in 1975 dollars (\$66.50/lb). The projection for 1980 used a basic tape cost of \$.55 per foot in 1975 dollars (38.50/lb). Currently, there are other sources of material that are not qualified to our specification but that may meet our requirements in the future. One of these materials, (designated as Vendor "B" in Table XXVIII) is quoted at about 3/4 the cost in 1980 of our qualified material. Another popular figure being quoted by many for the 1980 cost is \$20 per pound. In light of these variations in predictions the cost of the 633RW003-3 configuration using these various estimates is shown in Table XXVIII.

10.1.3 Other Costs

A breakdown of the total material costs for each of the preliminary designs is shown in Table XXIX. It is evident from this table that the increased cost of the 633RW003 design is not entirely due to the composites. Part of the cost increase can be attributed to the titanium pivot fitting that was used to be able to get the 27% weight savings. A bolted steel fitting would have yielded a better cost picture, however, the weight savings would not have been outstanding. It would have been desirable to have conducted a study of a steel bolted configuration in order to have established the boundaries of the designs for maximum weight savings and the minimum cost composite designs.

A comparison of the fabrication hours for each of the preliminary designs is shown in Table XXX. This table shows that there is a reduction in factory manhours for the composite design over the metallic designs.

10.2 WEIGHT EVALUATIONS

The preliminary design configurations all saved weight over the baseline as follows:

633RW001	15%
633RW002	12%
633RW003	27%

TABLE XXVIII COMPOSITE MATERIAL PRICE VARIATION ANALYSIS
(AVERAGE COST FOR 200 UNITS IN 1980 DOLLARS)

COST ITEM	633RW003-3		BASELINE 633RW000
	VENDOR A	VENDOR B	
			\$20/#
MATERIAL	183K	145K	121K
MFG	128K	128K	128K
TOOLING	14K	14K	14K
TOTAL	325K	287K	263K
			43K
			196K
			13K
			252K

TABLE XXIX MATERIAL COST BREAKDOWN

MATERIAL	633R000 BASELINE		633R001 ADV. MET.		633R002-1 ADV. MET.		633R002-3 ADV. MET.		633R003-1 COMPOSITE		633R003-3 COMPOSITE	
	*BUY WT. (LBS.)	FLY WT. (LBS.)	**MATERIAL COST	BUY WT. (LBS.)	FLY WT. (LBS.)	MATERIAL COST	BUY WT. (LBS.)	FLY WT. (LBS.)	MATERIAL COST	BUY WT. (LBS.)	FLY WT. (LBS.)	MATERIAL COST
STEEL	1530	738	1893	1174	296	8857	1174	296	8857	1010	351	7624
ALUMINUM	16350	1942	26306	7936	1913	24248	6830	1990	17059	6962	2121	17477
TITANIUM												
FIBERGLASS S. NOMEK												
GR/EP												
HARDWARE		73	13731		74	8838		75	9142		76	9170
MISC***		6			58			60			60	
TOOLING MATERIAL			973			989			986			1031
TOTALS PER SIDE		2759	42903		2341	42932		2421	36044		2608	33257
										2248		188875
											2006	183072

* BUY WT. IS: RAW STOCK WT. PLUS ATTRITION

** MATERIAL COSTS ARE AVERAGE COST PER UNIT OF 200 UNITS IN 1980 DOLLARS

*** COST INCLUDED IN FACTOR FOR ALL ALLOCATED MATERIALS

**TABLE XXX MANUFACTURING COMPARISONS
(Unit Average for 200 Units)**

CONFIGURATION	MFG. (HOURS)	QA (HOURS)	TOTAL (HOURS)
633RW000	7041	1084	8125
633RW001	5924	912	6836
633RW002-1	6319	973	7292
633RW002-3	6644	1023	7667
633RW003-1	4901	808	5709
633RW003-3	4554	751	5305

One significant item became apparent during the program. The 633RW003 composite wing box indicated a potential of approximately 40% savings in the analytical assembly phase. When trying to splice this wing into the existing wing pivot fitting, much of the weight savings was going to disappear. The importance of tailoring the attachment for the structural concept was clear; therefore, a new pivot fitting was designed and evaluated for cost and weight savings.

A second important item is that while the composite concepts do not rank above the baseline by using the merit rating system it does offer an attractive weight savings on a dollar per pound savings basis. Table XXXI shows the potential for the various composite materials.

10.3 FINAL EVALUATION SUMMARY

As part of the groundrules of the program all designs were evaluated by the merit rating system. The design that provides the best balance of all the priorities established by this system is to be considered the best design. The concept chosen using this criteria is the 633RW001. This configuration has a bonded laminated aluminum lower skin, etched aluminum "Y" intermediate spars, machined aluminum front & rear spars, straight tapered aluminum upper skin, and a 10 Ni fail safe wing pivot fitting. It is our recommendation at this point that we proceed into a "proof of concept" test program as outlined in Appendix D.

**TABLE XXXI COST PER POUND OF WEIGHT SAVINGS
(AVERAGE COST FOR 200 UNITS IN 1980 DOLLARS)**

MATERIAL COST	COST PER POUND SAVINGS VS BASELINE	COST PER POUND SAVINGS VS 633RW001
VENDOR A	\$96.95	\$310.45
VENDOR B	\$46.48	\$197.01
\$20/#	\$14.61	\$125.37

APPENDIX A

DESIGN LOADS DATA
FOR THE
VARIABLE CAMBER SUPERCRITICAL
WING PROGRAM

Abstract

Preliminary design loads data are presented for conceptual design of a supercritical wing with leading and trailing edge variable camber devices. The baseline wing is Advanced Transonic Wing ATW-4, integrated into a study configuration of the FB-111 airplane for a strategic mission. The preliminary design conditions are presented with criteria used to establish condition parameters. A brief discussion of basic data and analysis procedures is included. Static design loads distributions and related data are presented for design of the wing box and leading and trailing edge variable camber devices. The final section presents a fatigue loads spectrum of wing bending moment and the data for developing fatigue loads spectra for the variable camber devices.

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APPENDIX A
DESIGN LOADS DATA

A.1 BASELINE WING GEOMETRY

The baseline wing used for this data package is Advanced Transonic Wing ATW-4 as configured for a strategic mission aerodynamic and performance evaluation in Reference 1. The leading and trailing edge variable camber surface hinge lines were taken along the reference wing 15 and 65 percent chord lines respectively. A load reference axis was taken along the reference wing 33.64 percent chord line passing through the wing pivot center. The reference wing geometry is shown in the Figure A.1

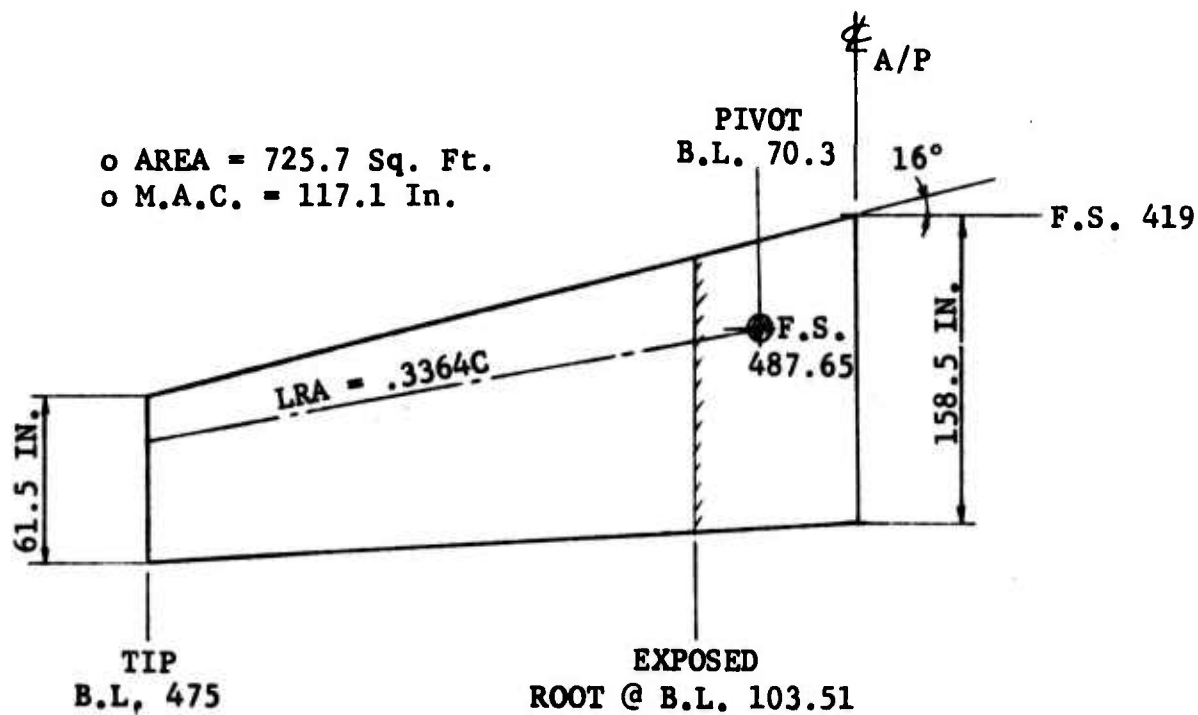


Figure A.1 Wing Geometry

A.2 PRELIMINARY DESIGN CONDITIONS

Conditions for which preliminary design loads data have been developed are presented in Table A-I. These conditions are thought to be representative of likely high load conditions suitable for preliminary design of the wing structural box and/or variable camber surfaces. Their selection was based, to a great extent, on past supercritical wing preliminary design studies at GD/FW.

Table A-I - Preliminary Design Conditions

Condition Number	1	1A	2	3	4
Description	Take-off	Take-off	Dash	Refuel (Sym.Man.)	Refuel (Roll Control)
Wing Sweep, Deg.	16	16	65	16	16
Mach No.	.35	.51	.85	.80	.80
Altitude, Ft.	SL	SL	SL	20,000	20,000
Gross Weight, Lbs.	128,500	128,500	110,400	110,400	110,400
Center of Gravity, % \bar{c}	-	-	36.4	25.14	25.14
Angle of Attack, Deg. ^(c)	-	-	10.78	6.68	4.8
Load Factor, g's	2.0	(Fig. 12)	3.0	2.6	1.9
L.E. Deflection, Deg. ^(a)	20.0	20.0	12.5	4.0	10.0
T.E. Deflection, Deg. ^(a)	15.0	7.5	-3.1	3.5	3.5 \pm 5.0 ^(b)

(a) Plus sign is for surfaces deflected downward.

(b) Linear variation 0° at root to \pm 5° at tip.

(c) Measured with respect to the manufacturing chord plane which has a 1 deg. positive inclination to the waterline reference at the wing pivot.

A.3 CRITERIA

A.3.1 Maneuver Load Factor

The general approach was to establish limit maneuver load factor for each condition based either on wing carry-through box allowable load limits or on a desired maneuver capability goal. The carry-through box allowable load limits for the F-111A series airplanes were used. The following symmetric maneuver capability goals were established:

- (a) Take-off and Landing Conditions; $n_z = 2.0g$
- (b) All Other Conditions; $n_z = 3.0g$

Asymmetric maneuver load factor goals are equal to $1.0g + 2/3$ (n_z symmetric - 1.0).

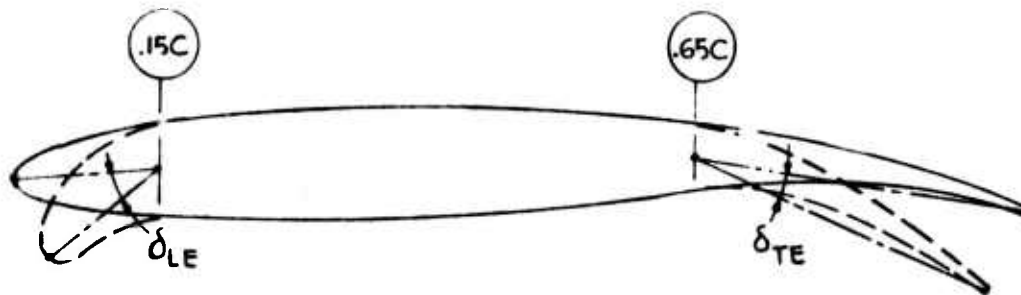
As shown on Table A-I the maneuver load factor goals were achieved for conditions 1 and 2, while the load factors for conditions 3 and 4 are somewhat less than the goals (3.0g for condition 3 and 2.33 g for condition 4). The analysis resulting in lower maneuver load factors is conservative having been based on linear aerodynamic theory and, very likely, conservative trailing edge camber settings.

A.3.2 Airplane Weight and Balance

Airplane gross weight and center of gravity values are based on a GD/FW procedure R5C weight study for the FB-111F with the ATW-4 wing, dated 5-30-75. The fuel burn sequence used assumes that wing fuel is used first. Wing net loads for gross weights with and without wing fuel were checked for relative criticalness to establish design maneuver load factor levels. As a result, the gross weight used for conditions 2, 3, and 4 on Table A-I is for wing tanks empty of fuel (produces the lowest allowable maneuver load factor). Full wing tanks was assumed for the take-off condition. The design gross weight for this condition is set somewhat greater than the maximum from procedure R5C in order to develop net wing pivot loads at 2.0 g's load factor which are equal to the carry-through box allowable load limits. The design center-of-gravity was set at 1.5 percent M.A.C. forward of the actual location.

A.3.3 Variable Camber Settings

The leading and trailing edge variable camber surface deflection angles used for developing airload distributions are preliminary estimates of 1.0g trim flight camber settings for optimum performance. These 1.0g trim camber settings were also assumed to be held constant during maneuvering flight conditions. Leading and trailing edge deflections are measured as illustrated below.



DEFLECTIONS WITH RESPECT TO THE BASIC SECTION

Endpoints and vertices of the chordal angle reference lines are located on the mean camber line(s). The chordal angles specified are for the wing in the reference position (i.e., 16 degrees leading edge sweep angle).

The trailing edge trim camber setting used for Conditions 3 and 4 is most likely conservative for structural loads. A constant deflection angle of 3.5 degrees was assumed along the span for these conditions. A more likely deflection schedule would have a linearly varying angle along the span, i.e., 3.5 degrees at the inboard end to 0.0 degrees at the outboard end.

A.3.4 Thermal Criteria

Static strength properties of external structure shall be established taking into consideration the effects of 56 total hours exposure at 2.0 Mach number at 35,000 feet altitude.

A 4 PROCEDURES AND BASIC DATA

Analyses required to evaluate the relative criticalness of the complete operational loads environment were beyond the intended scope of this program. Therefore, the critical structural design loads conditions for the ATW-4 wing box and variable camber devices were postulated based on past supercritical wing design studies (B-1/W11 wing and F-111/TACT wing), and preliminary usage criteria for the variable camber system. The conditions selected are summarized in Table A-I. Preliminary design loads were not developed for supersonic flight conditions based on the following considerations:

1. The B-1/W-11 wing box supersonic condition was considerably less critical than subsonic flight conditions.
2. Lifting surface leading edge airloads for supersonic flight conditions are typically of smaller magnitude than loads for subsonic/transonic conditions.
3. Expected use of the variable camber system for trim control, while configured to aft wing sweep positions, results in relatively low net differential pressure load normal to the trailing edge variable camber surface.

Wing local pressure distributions are strongly influenced by variable camber settings (deflection angles), and thereby, unique load distributions are produced at any given flight condition. Geometric similarity (supercritical wing planform and flap settings) of the low speed high lift configurations enabled the development of preliminary design loads data for the ATW-4 wing conditions 1 and 1A by scaling load distributions which were available from the B-1/W-11 wing design study program. Equations used for scaling airloads accounted for effects of differences in wing area, span, wing loading, exposed wing root location, pivot location, etc. The flight conditions for the ATW-4 configuration conditions 1 and 1A were established at a higher Mach number and dynamic pressure than used for the corresponding B-1/W11 conditions due to a slightly higher wing loading. Using this approach, airplane trim angles of attack were not predicted to develop the ATW-4 wing data; therefore, values are not given for angle of attack in Table A-I for conditions 1 and 1A.

Variable camber settings cause the load distributions for conditions 2, 3 and 4 to be unique for the ATW-4 wing; therefore, the required data for these conditions were developed using theoretical prediction techniques. Rigid airframe pressure distributions and airplane force data (lift and pitching moment) were predicted using the finite element theory and procedures presented in Reference 2. In addition to basic camber and angle of attack data, pressure distributions and force data were also predicted for leading and trailing edge variable camber deflections. These rigid aerodynamic data were corrected for static aeroelasticity, based on preliminary estimates of wing box structural stiffness (EI and GJ distributions), using the procedure described in Reference 3. The resulting flexible aerodynamic data were used to determine trim airplane angle of attack and to build up wing airload distributions.

Spanwise load distribution data are presented with respect to the load reference axis (LRA) as shown in Figure 1. This implies local strip loads are on chords oriented perpendicular to the LRA. The wing basic aerodynamic distribution data (local pressure, c_l and c_m) were predicted for streamwise chords. As an expediency in developing design load distributions for the conditions having 16 degrees wing sweep (conditions 3 and 4), it was assumed that local streamwise aero data at given span stations could be applied to chords oriented perpendicular to the LRA at that station. This assumption is invalid for higher wing sweep conditions. Therefore, for condition 2, streamwise pressures were interpolated onto the exposed wing panel (i.e., outboard of B.L. 103.51) chords oriented perpendicular to the LRA. This accounts for the truncated appearance of the chordwise pressure distributions inboard of .4738 ETA for condition 2.

A.5 DESIGN LOADS DATA

Preliminary design loads data for the wing box and variable camber surfaces are presented on the figures which follow. All loads data are design limit values. Data presented are summarized below:

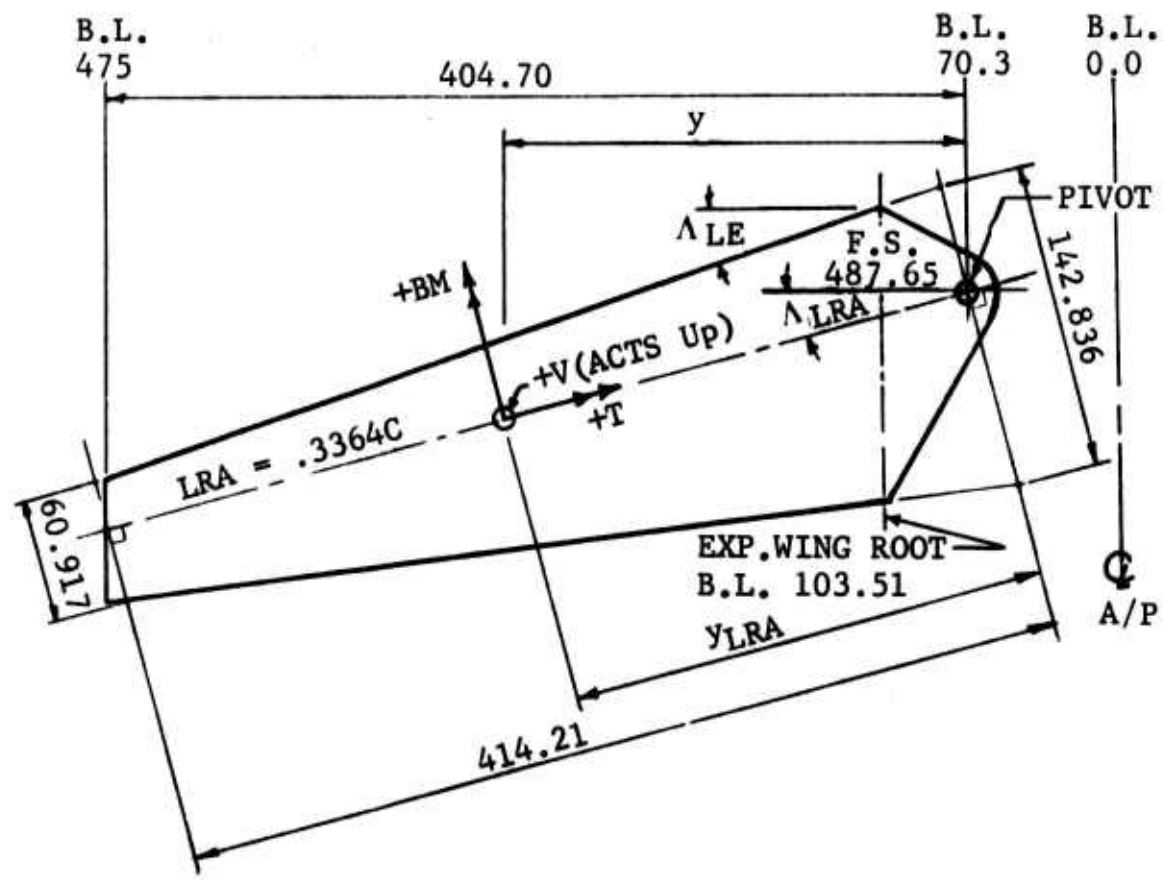
<u>Figure Number</u>	<u>Data</u>
A.2	Loads Reference Axis and Sign Convention
A.3 and A.4	Wing Unit Inertia Distributions
A.5 thru A.14	Loads Data for Condition 1
A.15	Loads Data for Condition 1A
A.16 thru A.27	Loads Data for Condition 2
A.28 thru A.43	Loads Data for Condition 3
A.44 thru A.58	Loads Data for Condition 4

Loads data for Conditions 1, 2, 3 and 4 consist of:

- (1) Wing spanwise shear, bending moment and torsion distributions.
- (2) Leading and trailing edge camber surface loads.
- (3) Chordwise pressure distributions.
- (4) Non-dimensional aerodynamic wing span load distributions for panel local lift force and torsion.

Loads data for condition 1A (see Figure A-12) consist of one distribution each for the leading edge and trailing edge variable camber surfaces. Comparable conditions are included in the B-1/W-11 data package. This condition was not critical for W-11 wing box design, therefore, the associated ATW-4 wing box spanwise load data were not developed for this design loads package.

Some of the structural loads data presented are obviously not critical for determining structural static strength. For example, the trailing edge flap load for condition 2 is quite low compared with other conditions. In this case, the flap is used to augment airplane trim (i.e., the surface is deflected up from the basic position) and thereby produces a tendency to washout the usually dominant local pressures from basic camber, as evidenced by the net chordwise pressure distributions. The flap load for this condition is included in this report for use in development of variable camber surface fatigue loads spectra.



NOTES :

- (1) RIGHT HAND RULE FOR MOMENT VECTORS
- (2) LRA = LOAD REFERENCE AXIS
- (3) $\Delta_{LRA} = 12.301^\circ$ WHEN $\Delta_{LE} = 16^\circ$
- (4) $\eta_{LRA} = Y_{LRA}/414.21$
- (5) LOCAL CHORD \perp TO LRA: $C = 142.836 - 81.919 \eta_{LRA}$
- (6) AVG CHORD \perp TO LRA: $C_{AVG} = 101.876$ IN.
- (7) $S/2 = \frac{(LRA) \times C_{AVG}}{144} = 293.042$ FT²

144

Figure A.2 Sign Convention and Reference Axis for Wing Loads - ATW-4 Wing

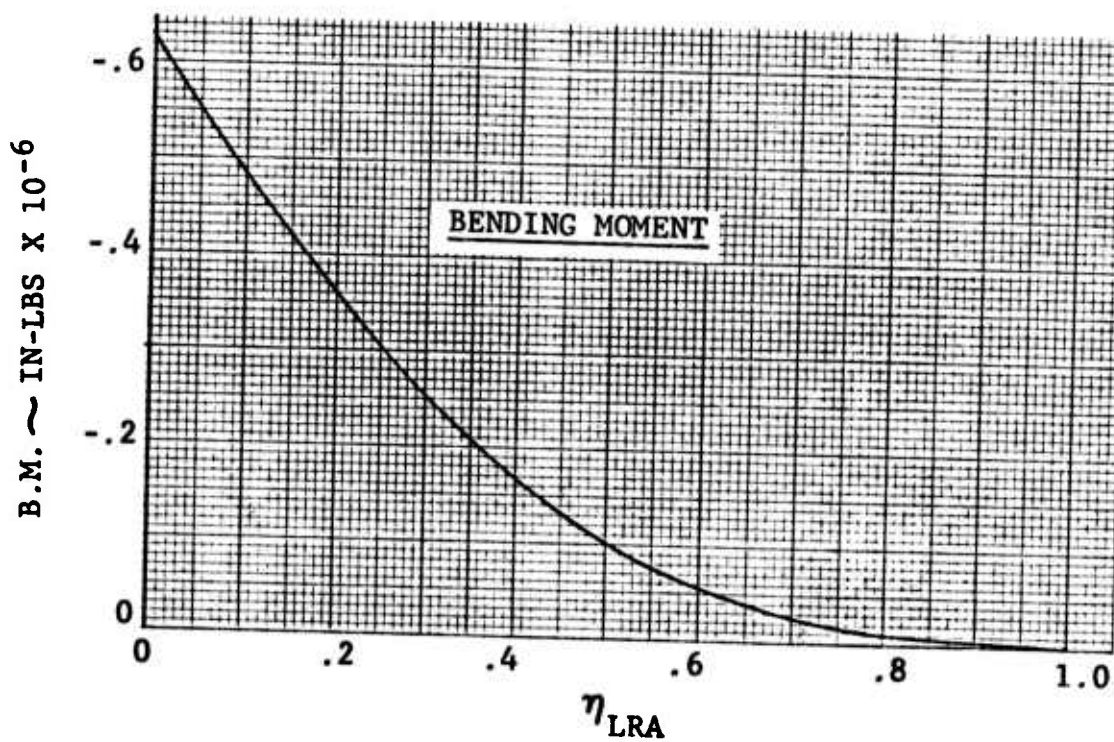
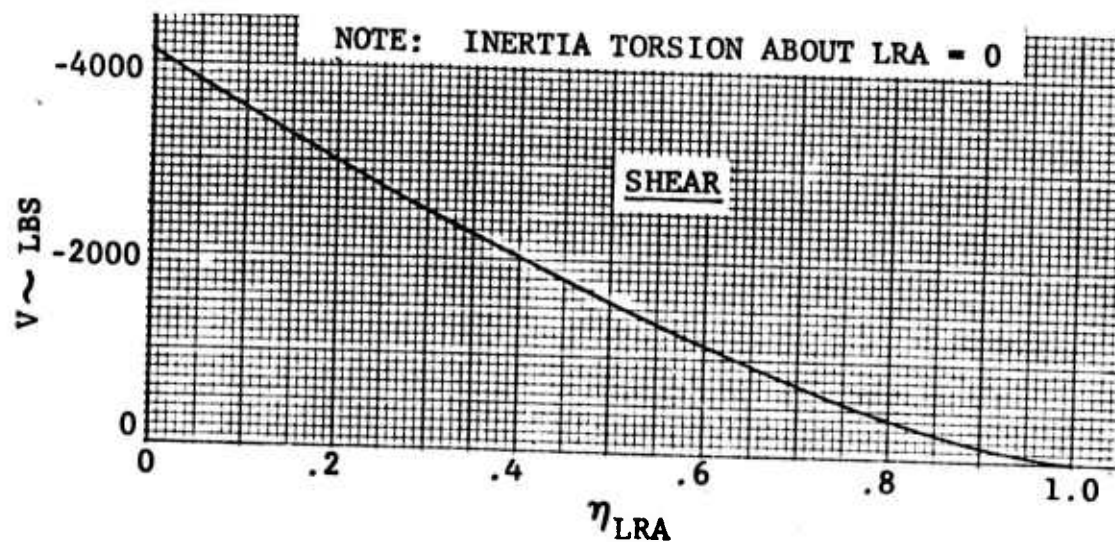


Figure A.3 ATW-4 Wing Structure Inertia Shear and Bending Moment Distributions ($N_z = 1.0g$)

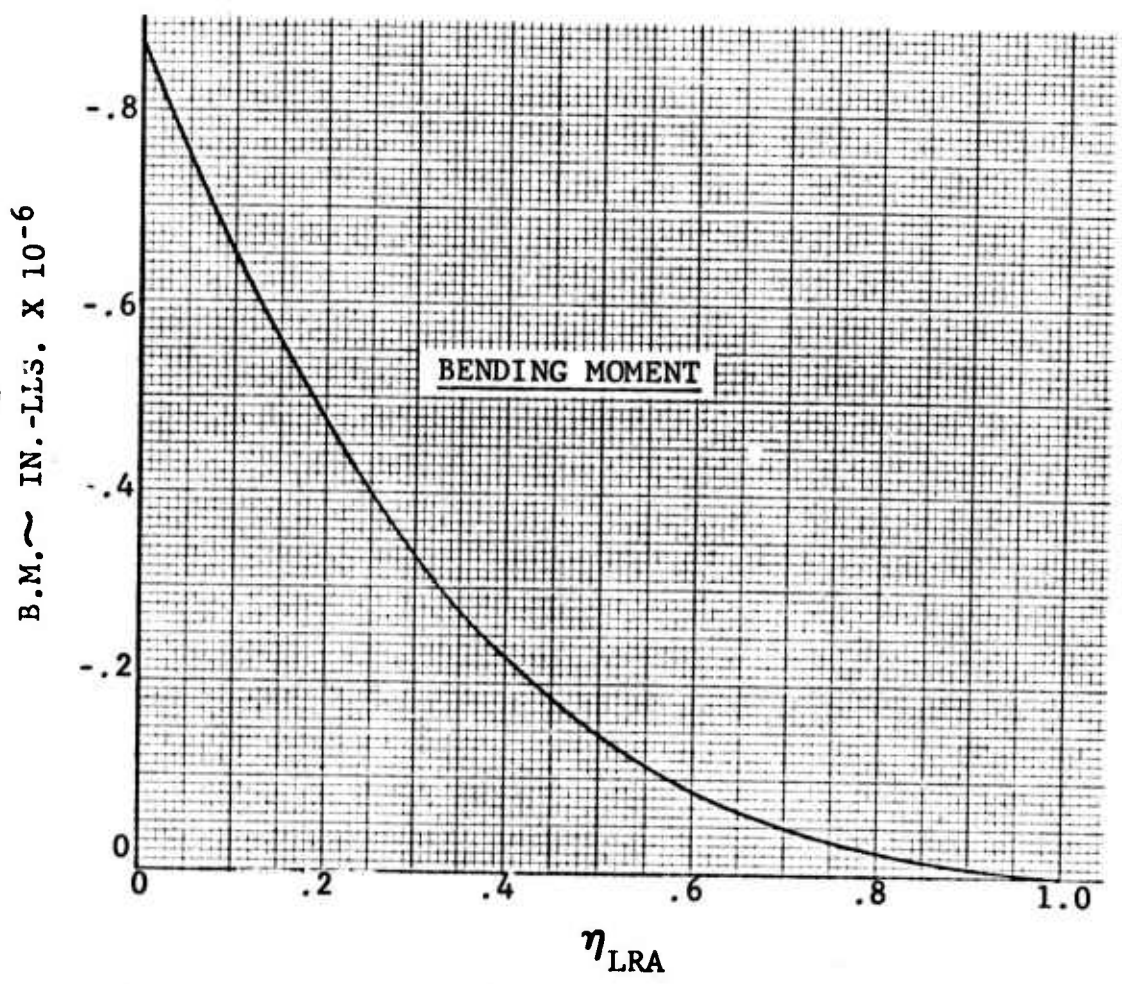
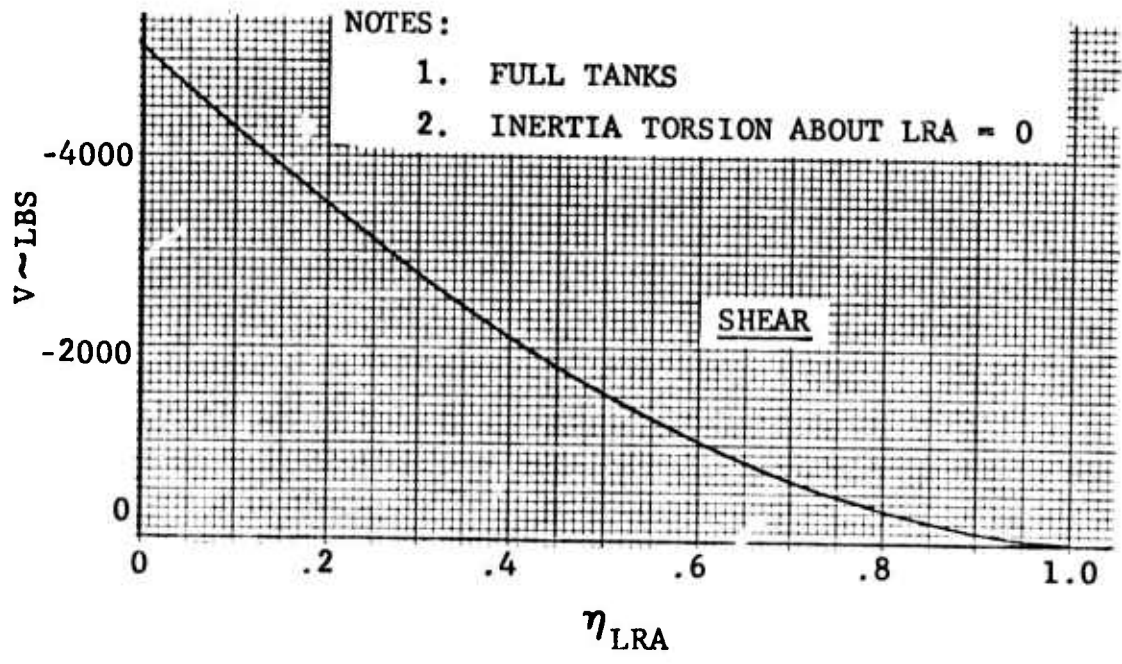


Figure A.4 ATW-4 Wing Fuel Inertia Shear and Bending Moment Distributions ($N_z = 1.0g$)

A.5.1 Condition 1 - Take-Off with Full Flaps

Figures A.5 through A.14 define preliminary design loads for condition 1, take-off with full flaps.

$\Lambda = 16^\circ$
 $M = .35$
 ALT = SL
 GW = 128,500 LBS
 $n_z = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

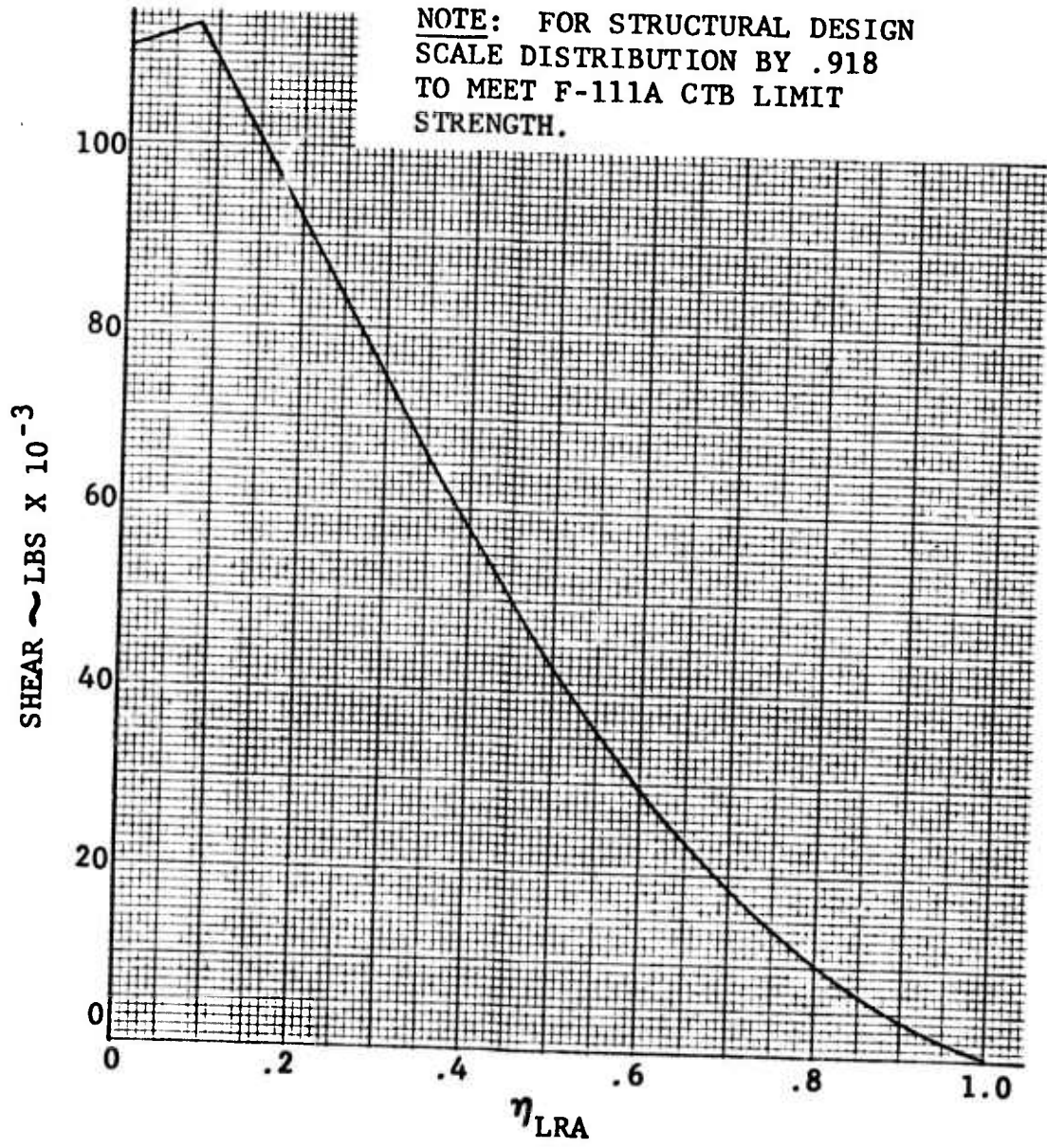


Figure A.5 ATW-4 Wing Shear Distribution - Condition 1

$\Lambda = 16^\circ$
 $M = .35$
 ALT = SL
 $GW = 128,500 \text{ LBS}$
 $n_z = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

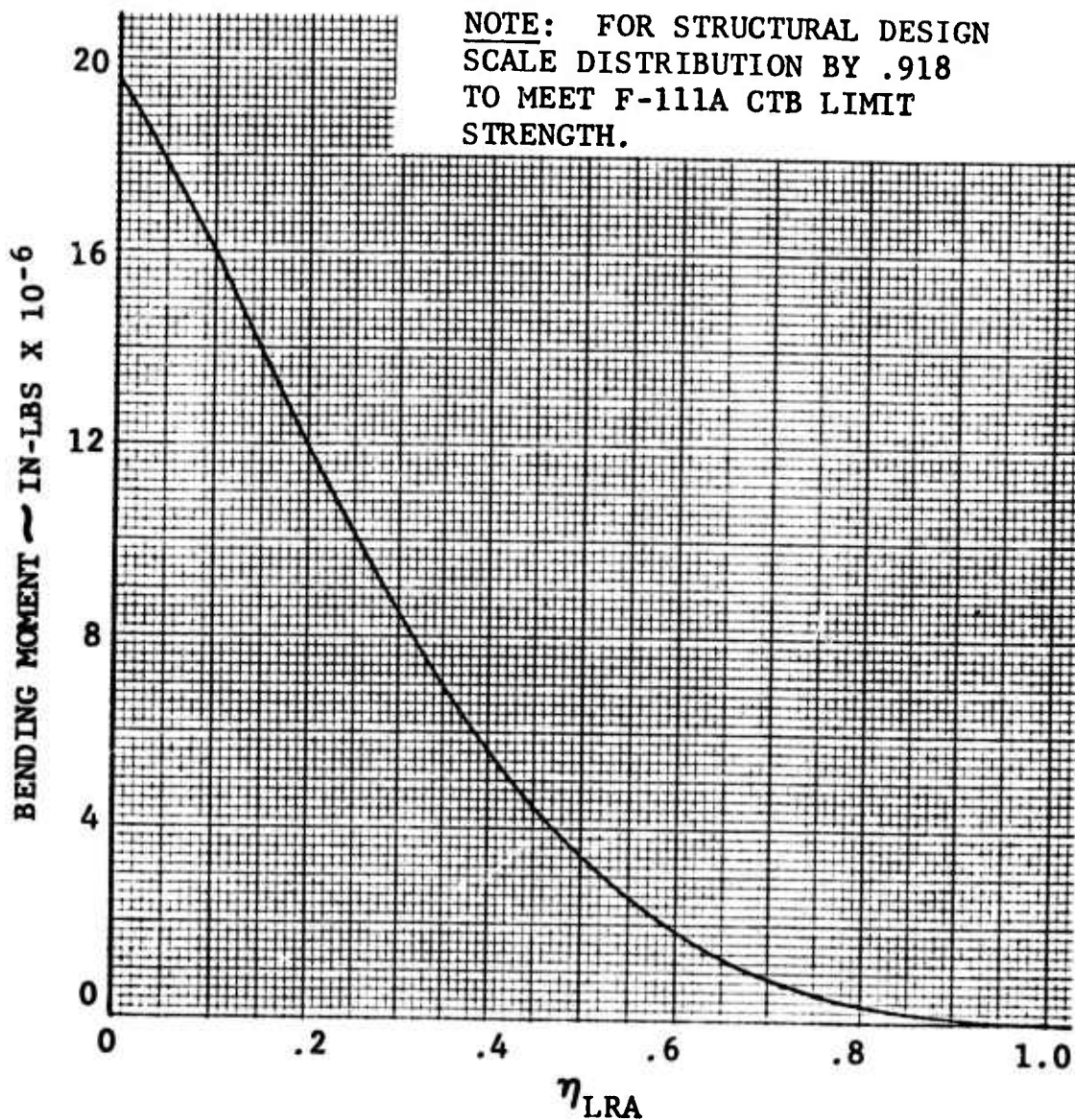


Figure A.6 ATW-4 Wing Bending Moment Distribution - Condition 1

$\Lambda = 16^\circ$
 $M = .35$
 ALT = SL
 $GW = 128,400 \text{ LBS}$
 $n_z = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

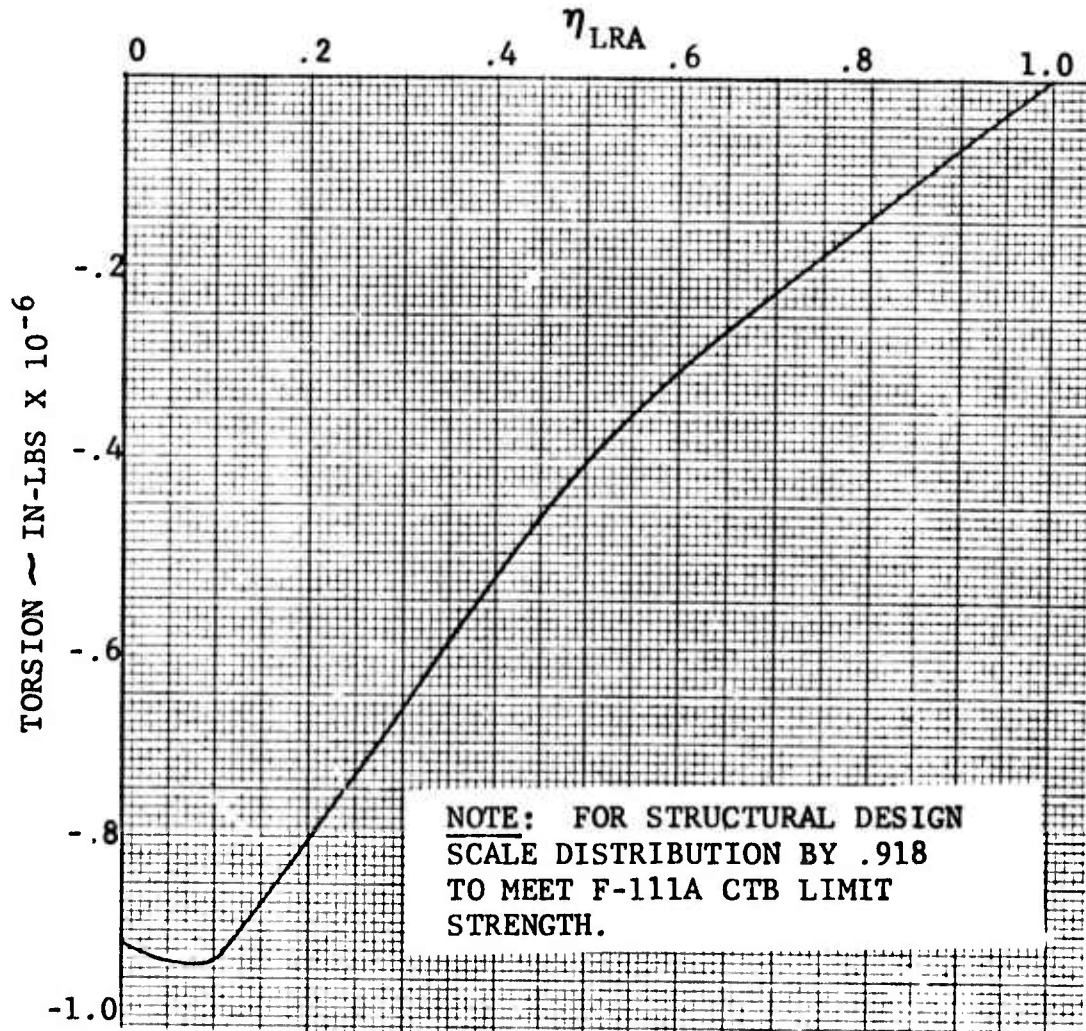


Figure A.7 ATW-4 Wing Torsion Distribution - Condition 1

$\Lambda = 16^\circ$
 $M = .35$
 ALT = SL
 $GW = 128,500 \text{ LBS}$
 $n_z = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

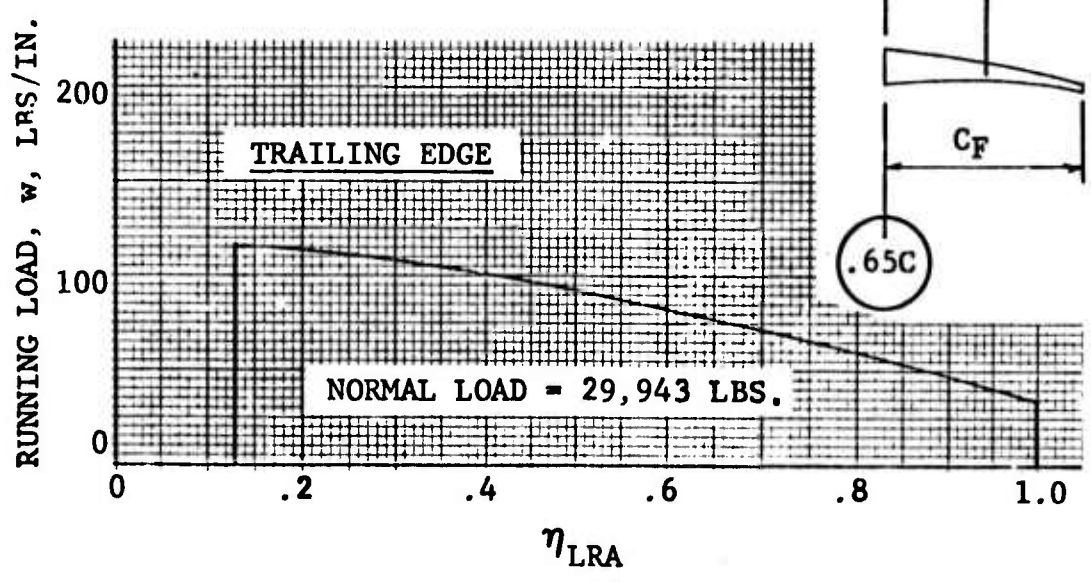
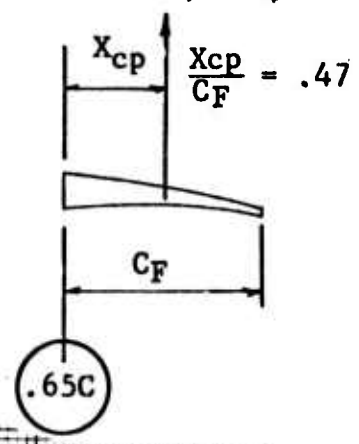
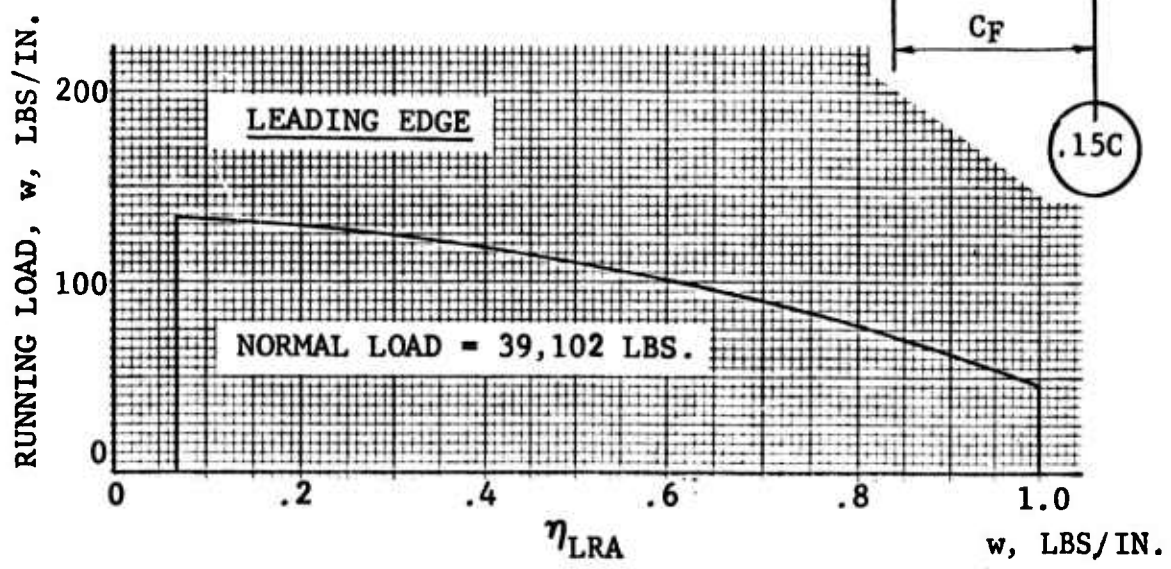
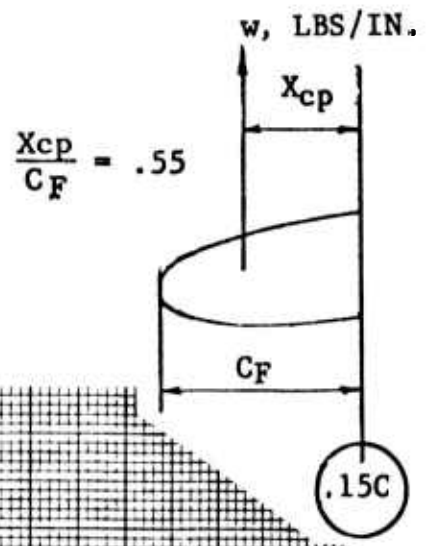


Figure A.8 ATW-4 Wing Leading and Trailing Edge Flap Running Loads - Condition 1

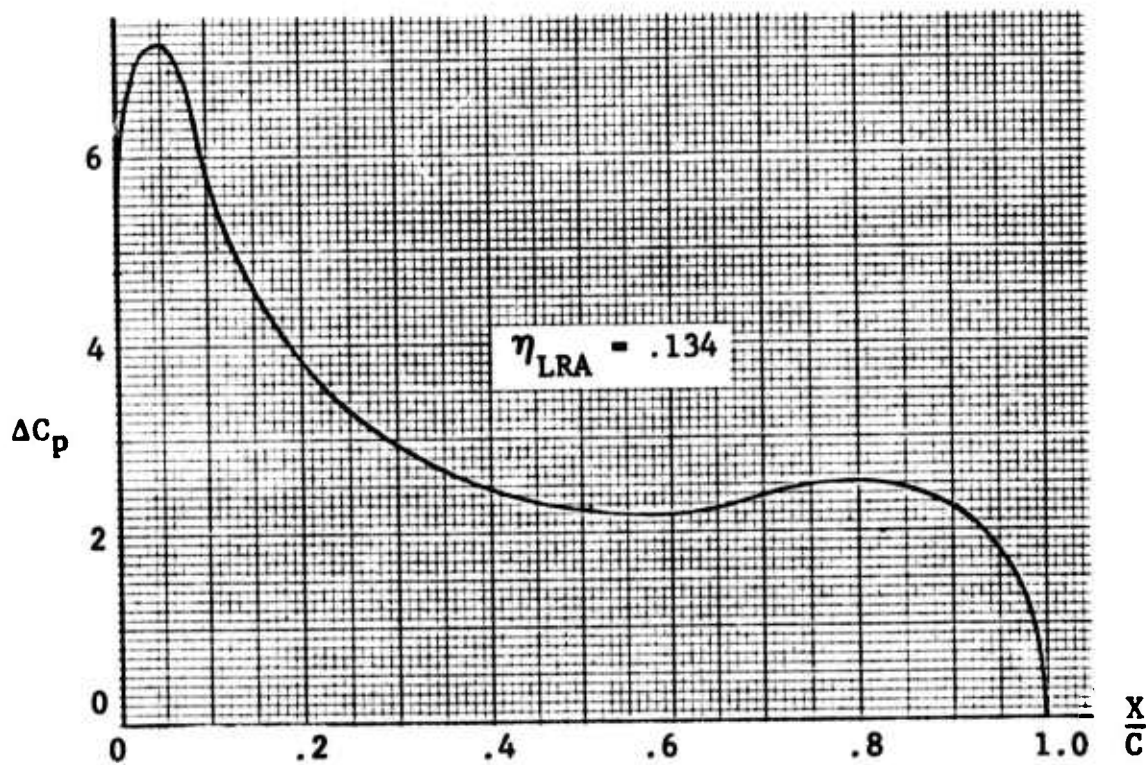
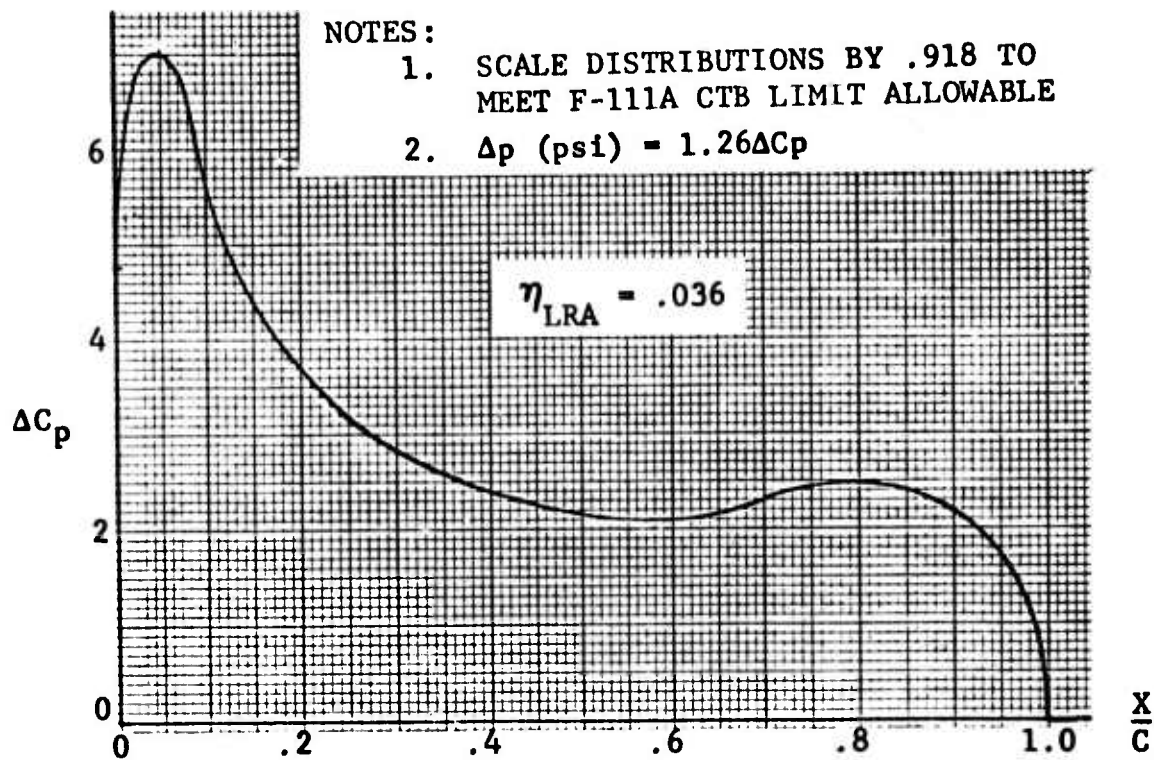


Figure A.9 ATW-4 Wing Chordwise Pressure Distributions for .251 and .369 η_{LRA} - Condition 1

NOTES:

1. SCALE DISTRIBUTIONS BY .918 TO MEET F-111A GTB LIMIT ALLOWABLE
2. Δp (psi) = 1.26 ΔC_p

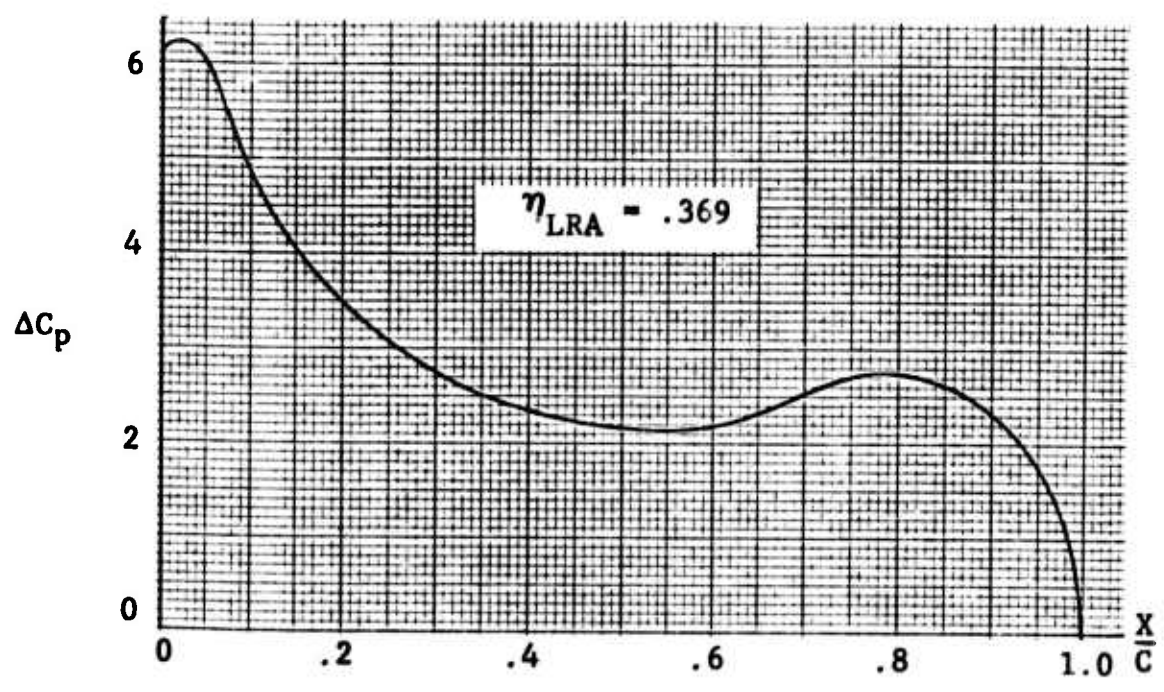
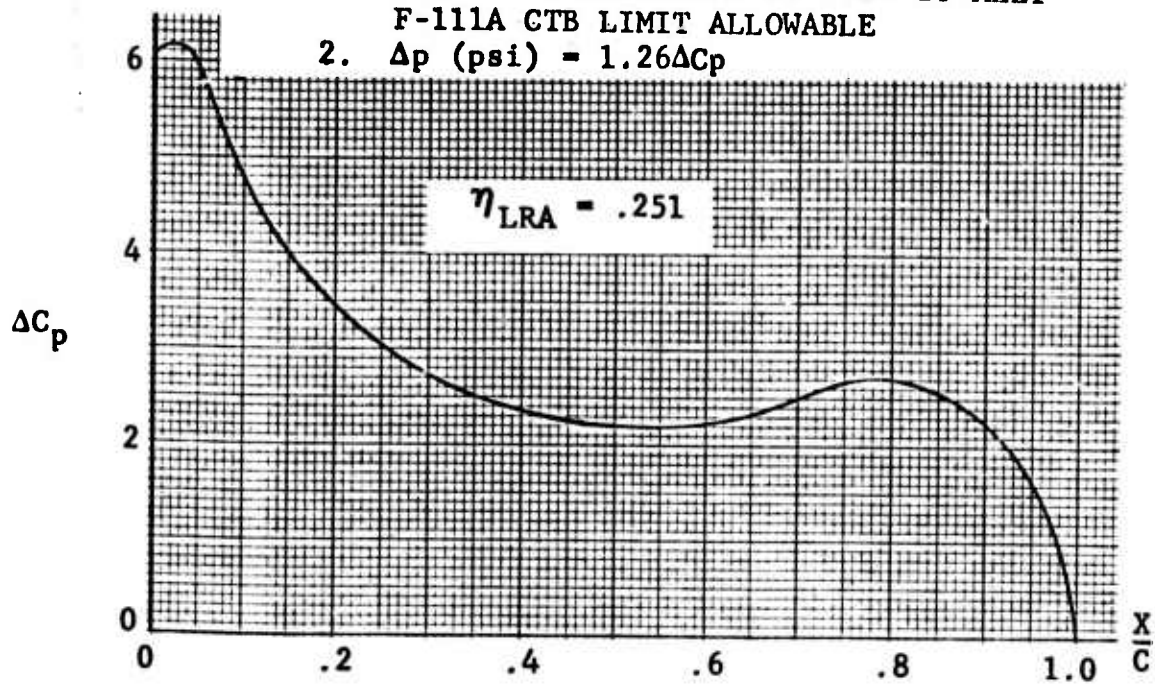


Figure A.10 ATW-4 Wing Chordwise Pressure Distributions for .251 and .369 η_{LRA} - Condition 1

NOTES:

1. SCALE DISTRIBUTIONS BY .917 TO MEET F-111A CTB LIMIT ALLOWABLE
2. Δp (PSI) = 1.26 ΔC_p

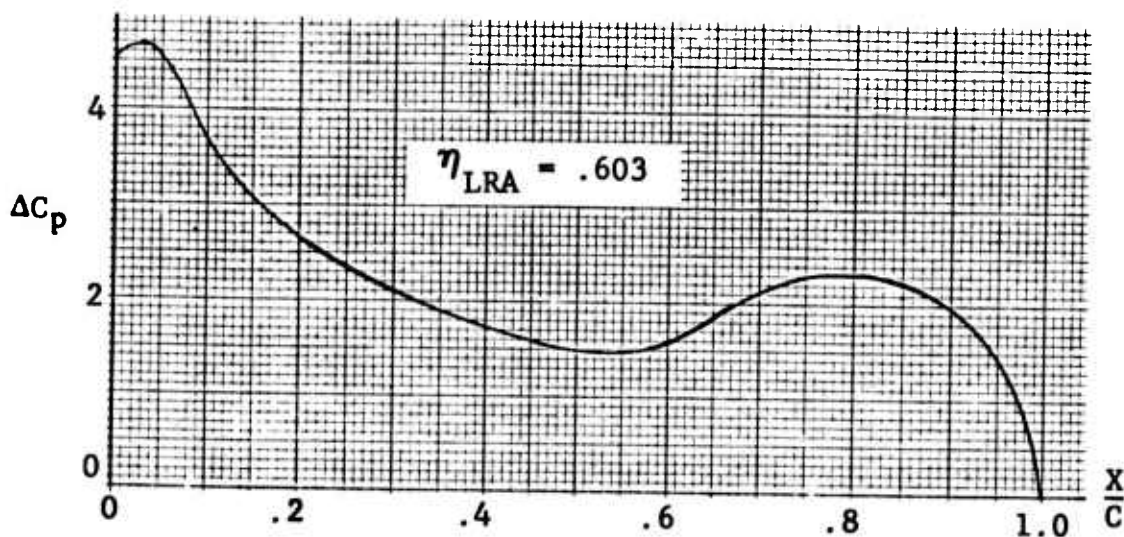
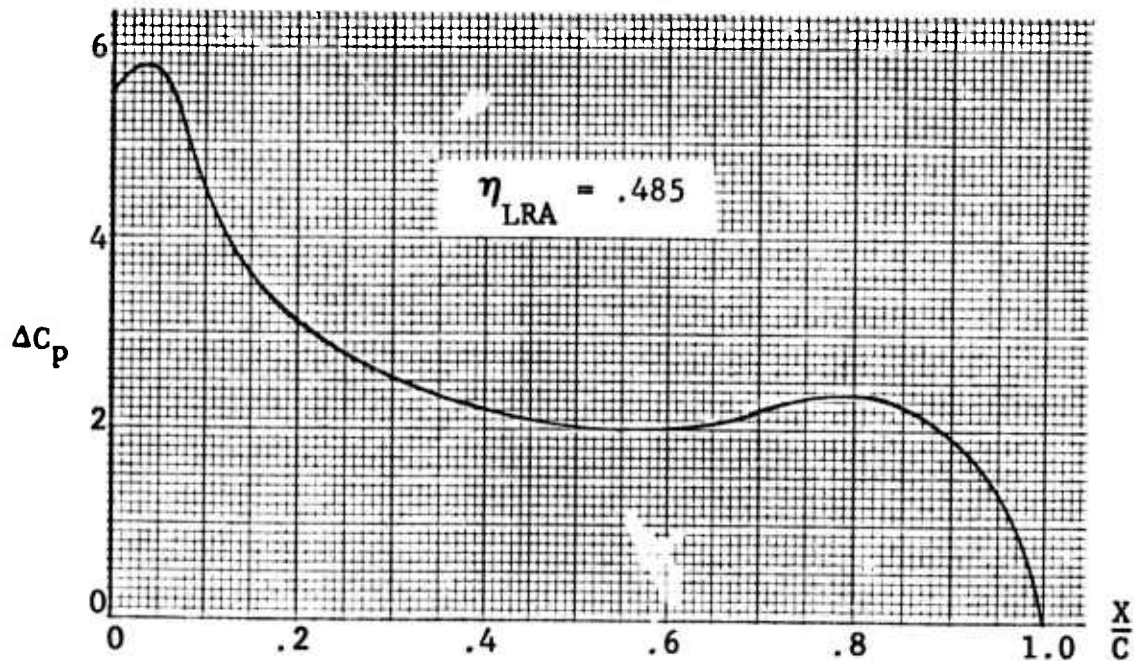


Figure A.11 ATW-4 Wing Chordwise Pressure Distributions for .485 and .603 η_{LRA} - Condition 1

NOTES:

1. SCALE DISTRIBUTIONS BY .918 TO MEET F-111A CTB LIMIT ALLOWABLE
2. Δp (psi) = 1.26 ΔC_p

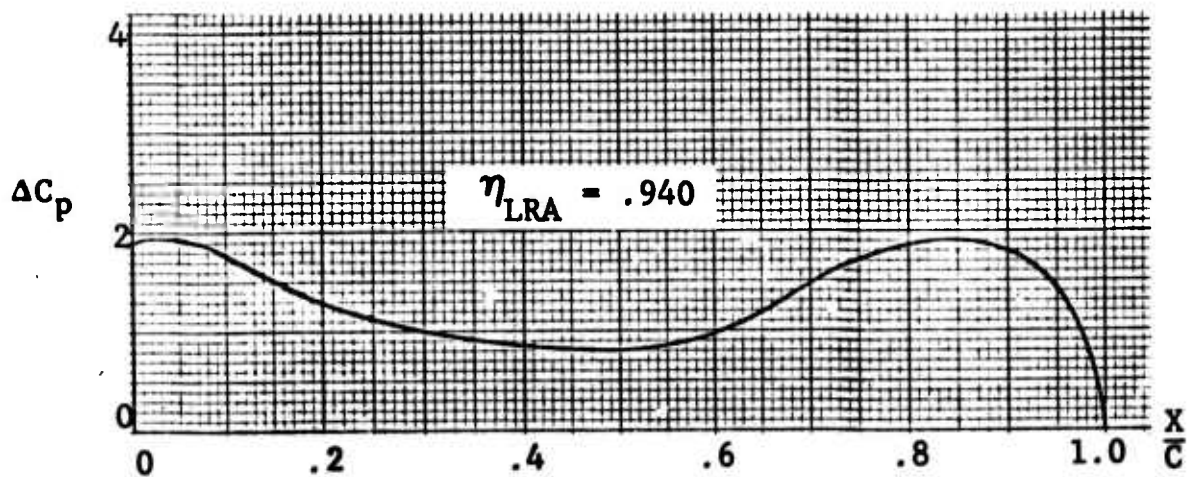
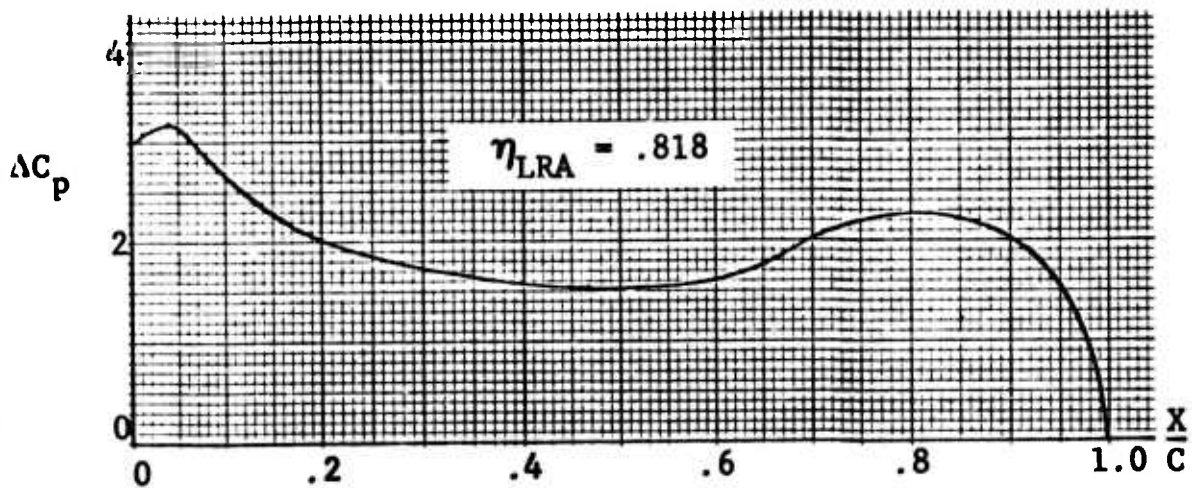


Figure A.12 ATW-4 Wing Chordwise Pressure Distributions for .818 and .940 η_{LRA} - Condition 1

NOTES:

1. LOCAL LIFT CN CHORDS \perp TO LRA
2. SEE FIGURE A.1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA
3. SCALE ORDINATES BY .918 TO MEET F-111A CTB LIMIT STRENGTH

$\Lambda = 16^\circ$
 $M = .35$
 $ALT = SL$
 $GW = 128,500 \text{ LBS}$
 $nZ = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

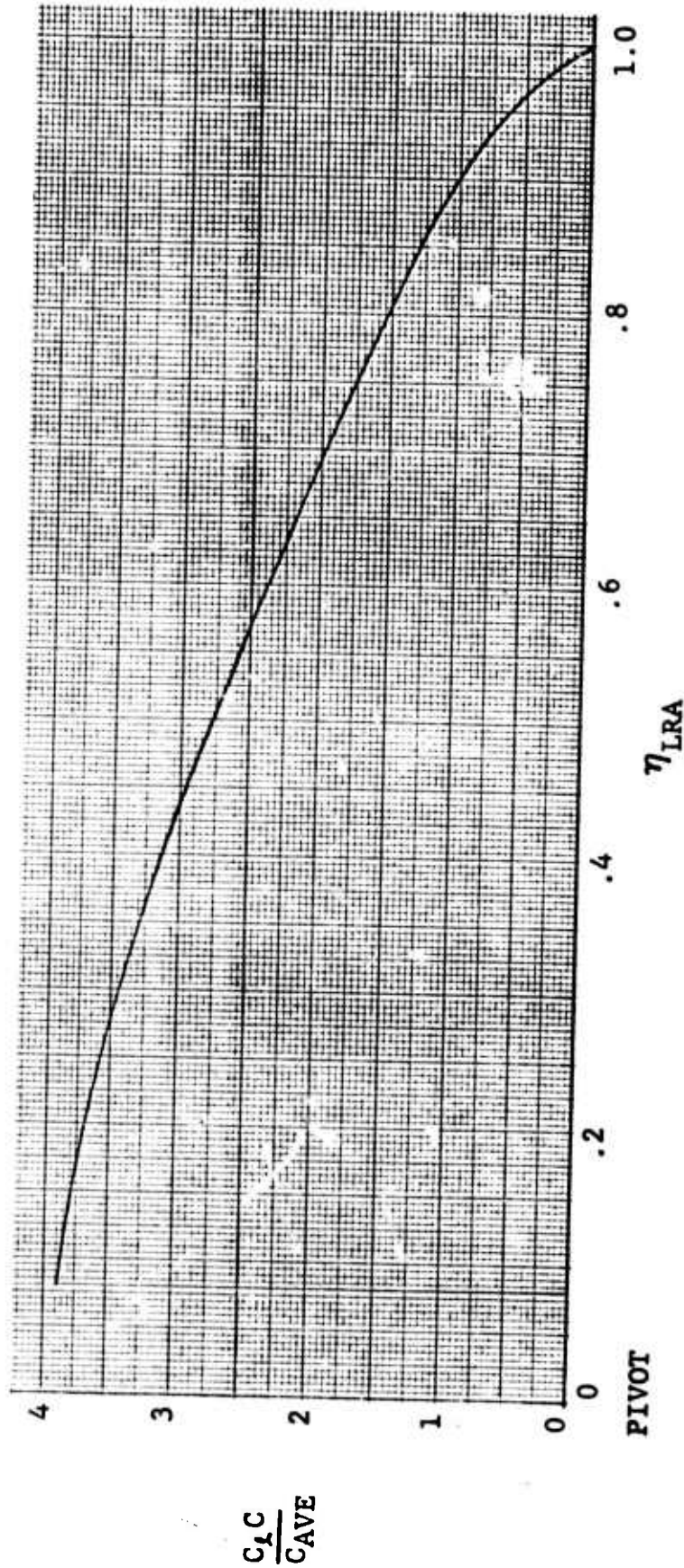


Figure A.13 ATW-4 Wing Local Lift Spanwise Distribution - Condition 1

$$\frac{C_L}{C_{CAVE}}$$

NOTES:

1. LOCAL TORSION ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA
3. MOM. REF. IS LRA
4. $\bar{C} = 117.1$ IN. (THEORETICAL REF. WING MAC)
5. SCALE ORDINATES BY .918 TO MEET F-111A CTB LIMIT STRENGTH

$\Lambda = 16^\circ$
 $M = .35$
 $ALT = SL$
 $GW = 128,500$ LBS
 $n_z = 2.0g$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 15^\circ$

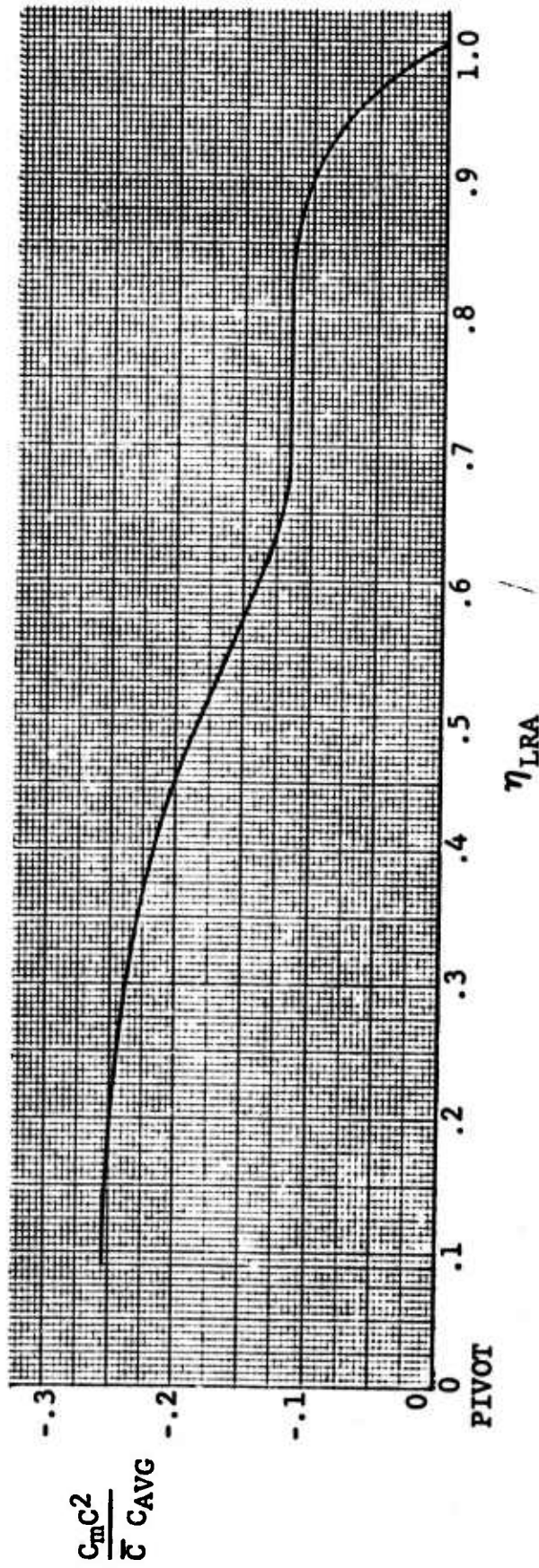


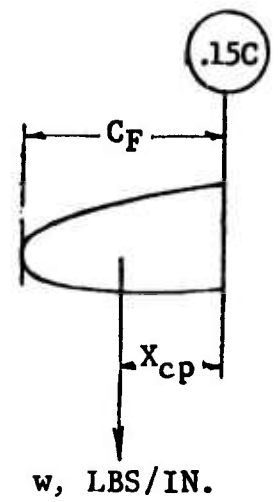
Figure A-14 ATW-4 Wing Local Torsion Spanwise Distribution - Condition 1

A.5.2 Condition 1A - Take-Off with Half Flaps

Figure A.15 shows the wing leading and trailing edge running loads for the take-off condition with half flaps. This condition is not critical; therefore, the chordwise pressure distributions were not plotted.

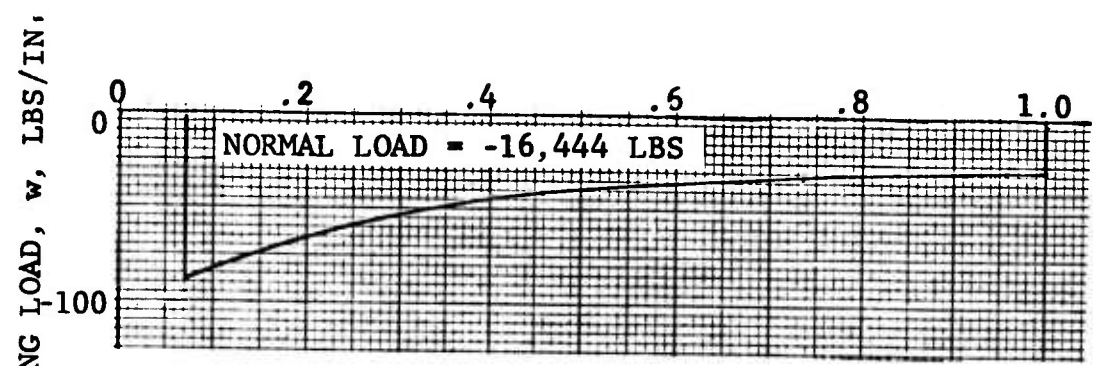
$\Lambda = 16^\circ$
 $M = .51$
 ALT = SL
 GW = 128,500 LBS
 $n_z = \text{AS NOTED}$
 $\delta_{LE} = 20^\circ$
 $\delta_{TE} = 7.5^\circ$

$\frac{x_{cp}}{C_F} = .66$



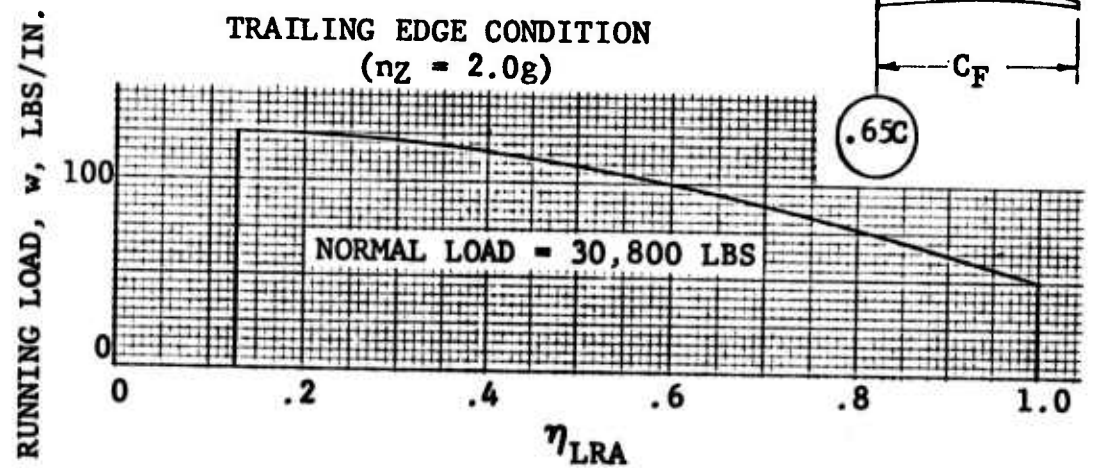
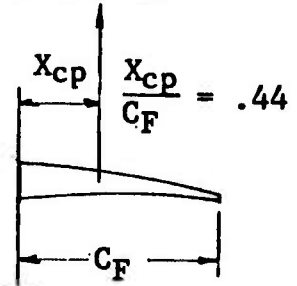
η_{LRA}

w, LBS/IN.



LEADING EDGE CONDITION
($n_z = 0g$)

w, LBS/IN.



TRAILING EDGE CONDITION
($n_z = 2.0g$)

Figure A.15 ATW-4 Wing Leading and Trailing Edge Flap Running Loads - Condition 1A

A.5.3 Condition 2 - Dash

Figure A.16 through A.27 define preliminary design loads for condition 2, the sea level dash condition.

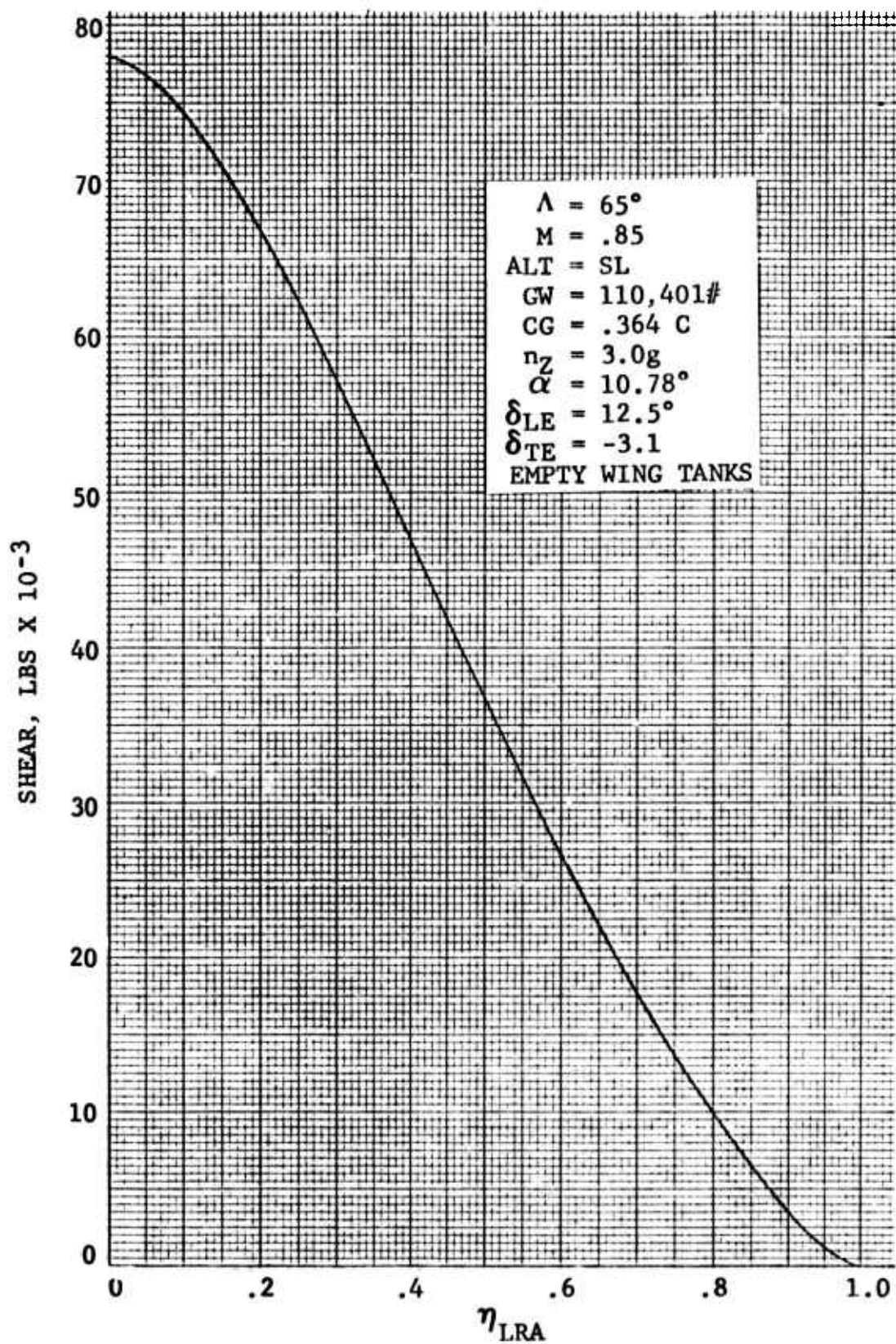


Figure A.16 ATW-4 Wing Shear Distribution - Condition 2

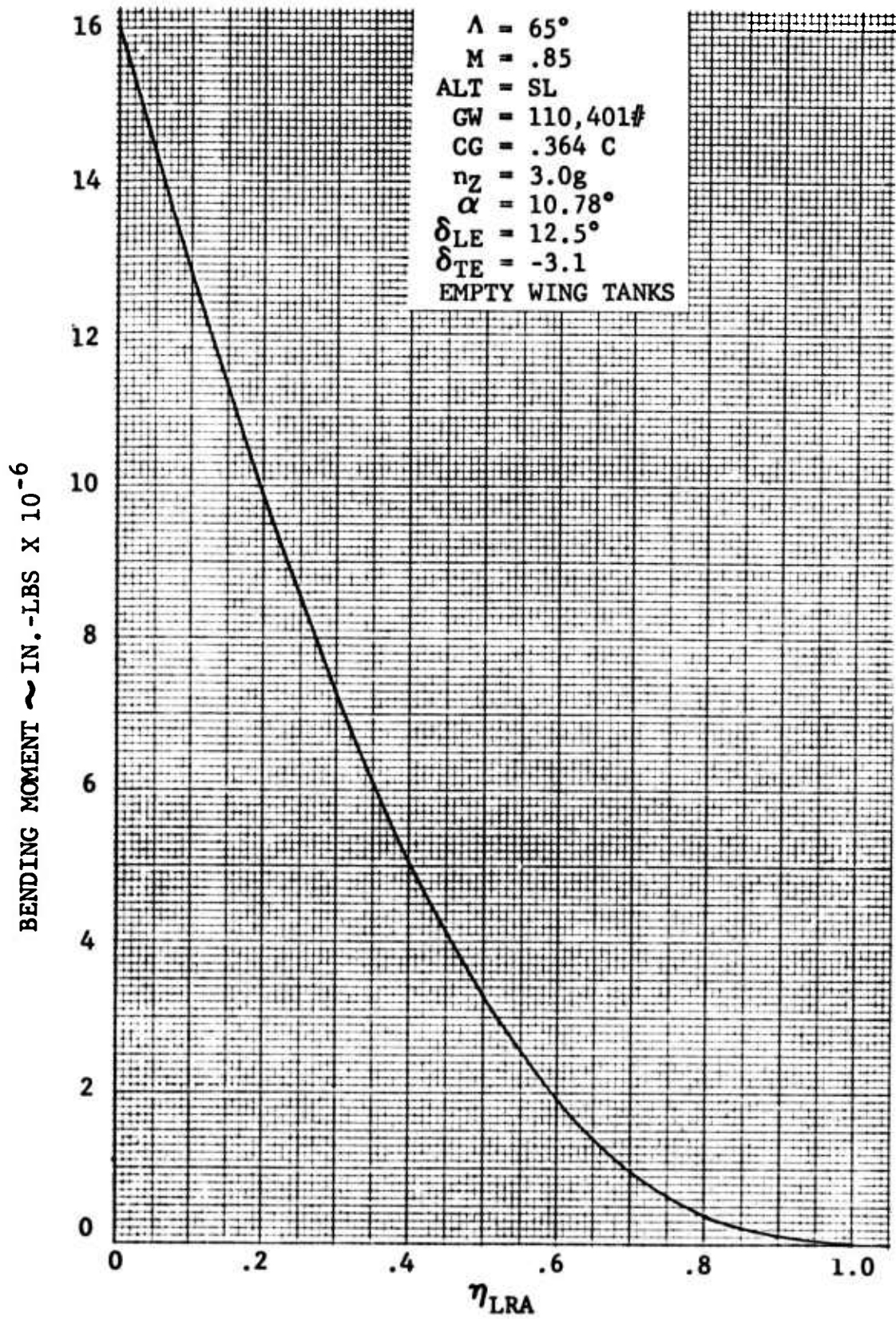


Figure A.17 ATW-4 Wing Bending Moment Distribution - Condition 2

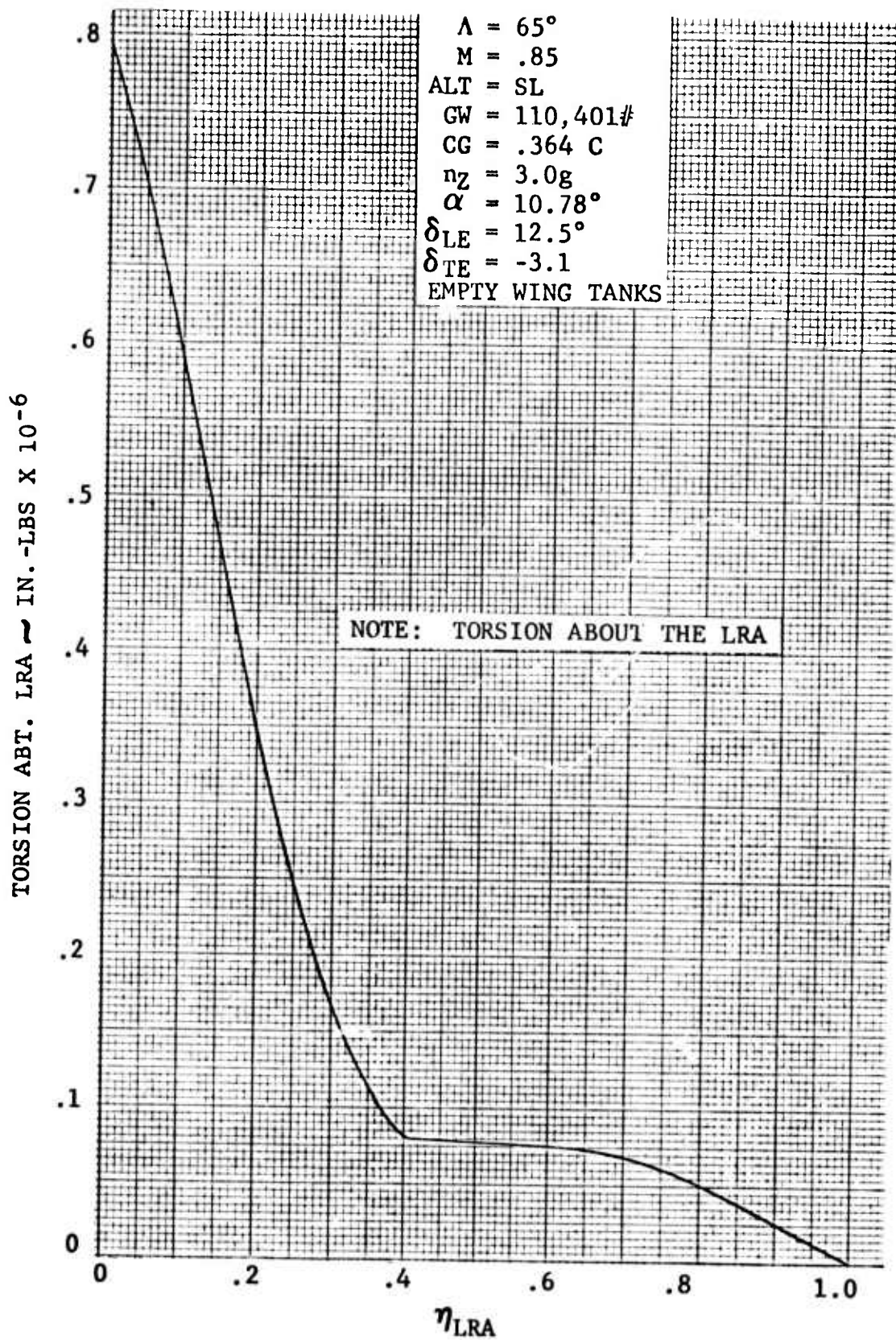


Figure A.18 ATW-4 Wing Torsion Distribution - Condition 2

$\Lambda = 65^\circ$
 $M = .85$
 ALT = S.L.
 $GW = 110,401 \text{ LBS}$
 $n_z = 3.0g$
 $\delta_{LE} = 12.5^\circ$
 $\delta_{TE} = -3.1^\circ$

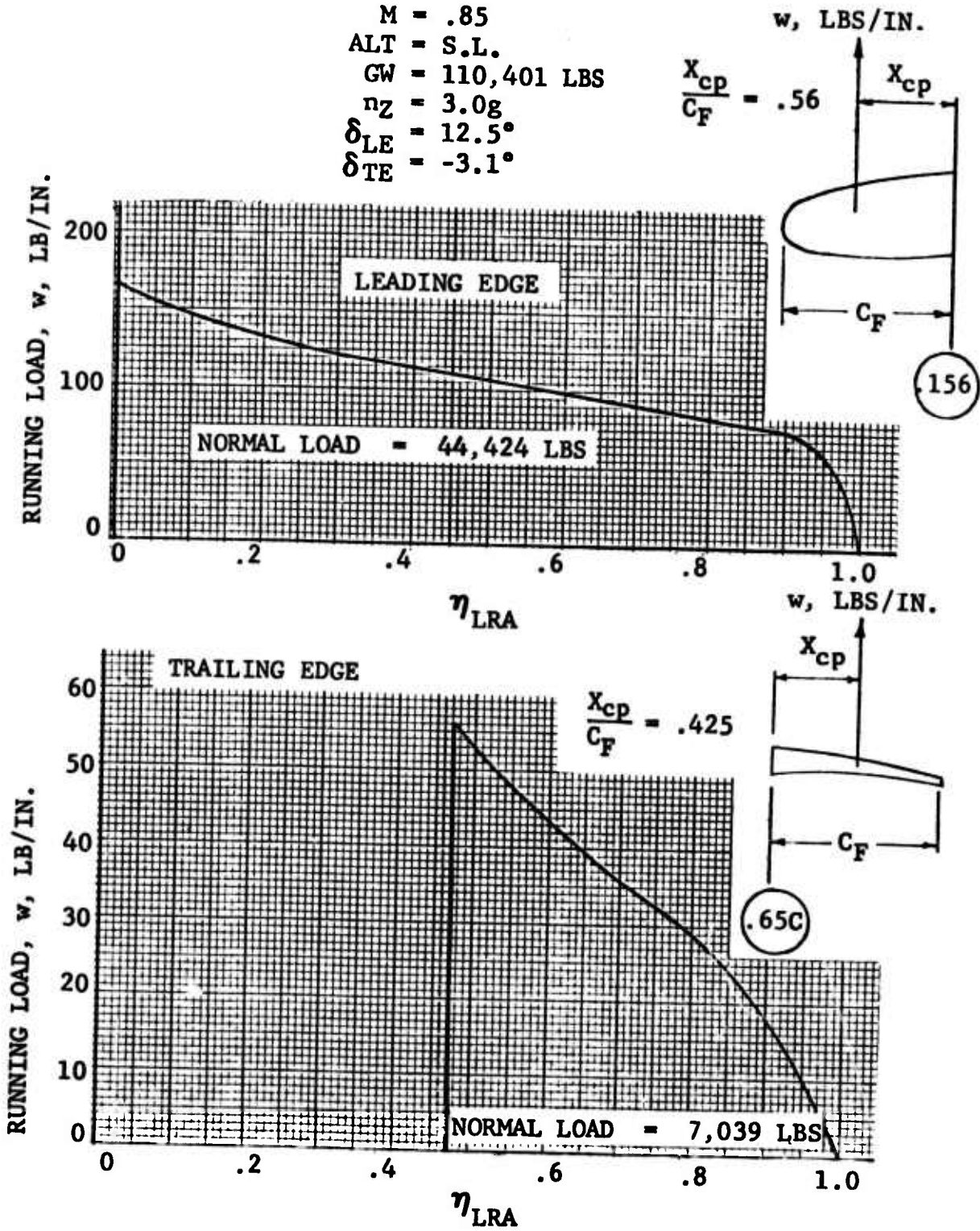


Figure A.19 ATW-4 Wing Leading and Trailing Edge Flap Running Loads - Condition 2

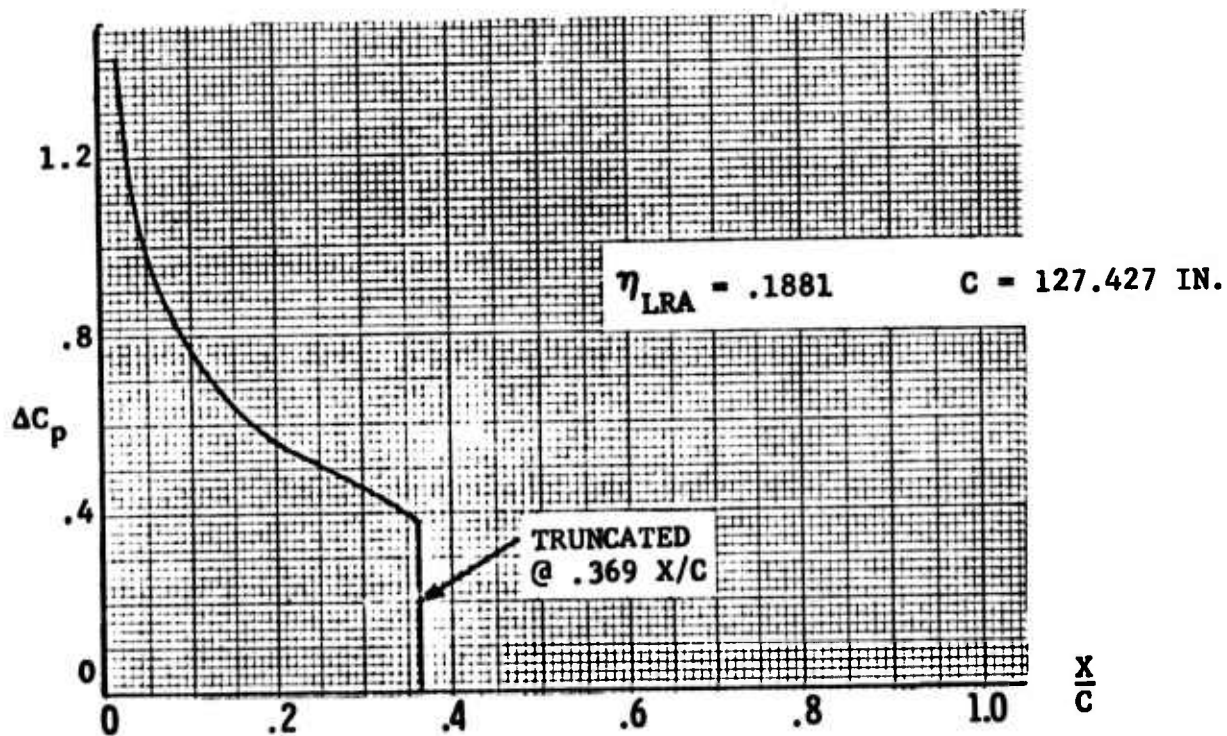
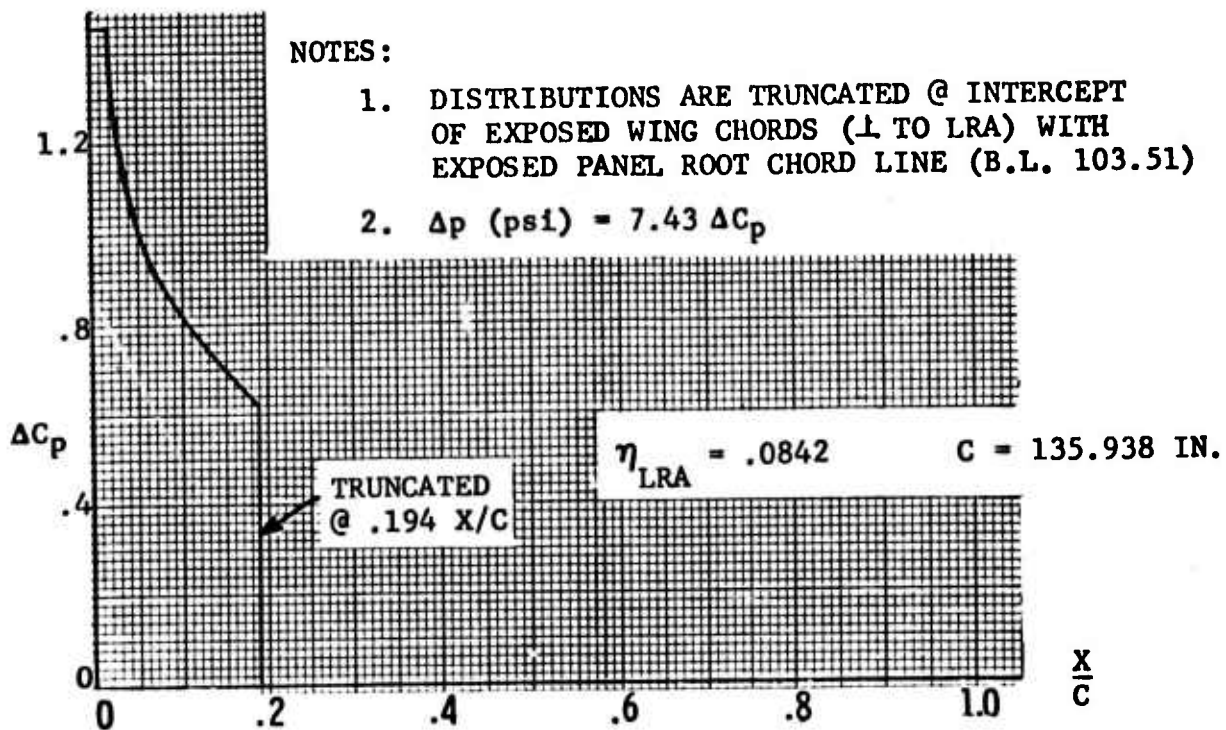


Figure A.20 ATW-4 Wing Chordwise Pressure Distributions for .0842 and .1881 η_{LRA} - Condition 2

NOTES:

1. DISTRIBUTIONS ARE TUNCATED @ INTERCEPT OF EXPOSED WING CHORD (\perp TO LRA) WITH EXPOSED PANEL ROOT CHORD LINE (B.L. 103.51)
2. Δp (psi) = 7.43 ΔC_p

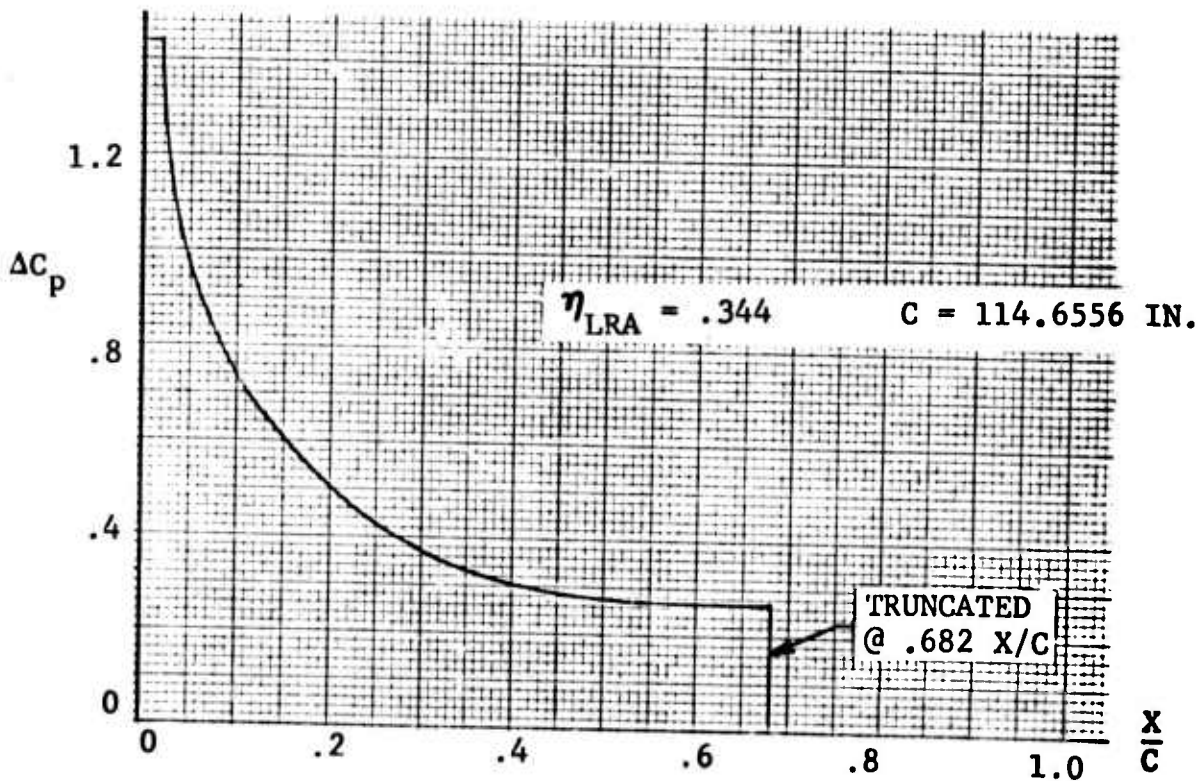
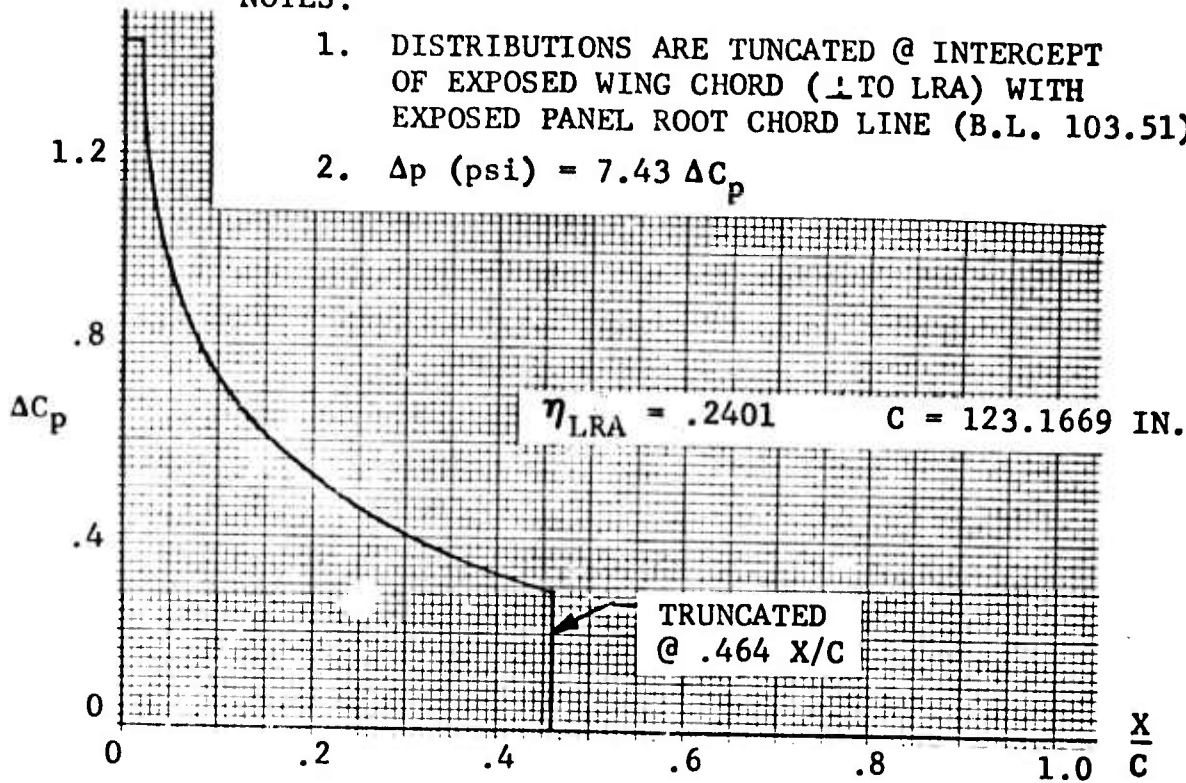


Figure A.21 ATW-4 Wing Chordwise Pressure Distributions for .2401 and .344 η_{LRA} - Condition 2

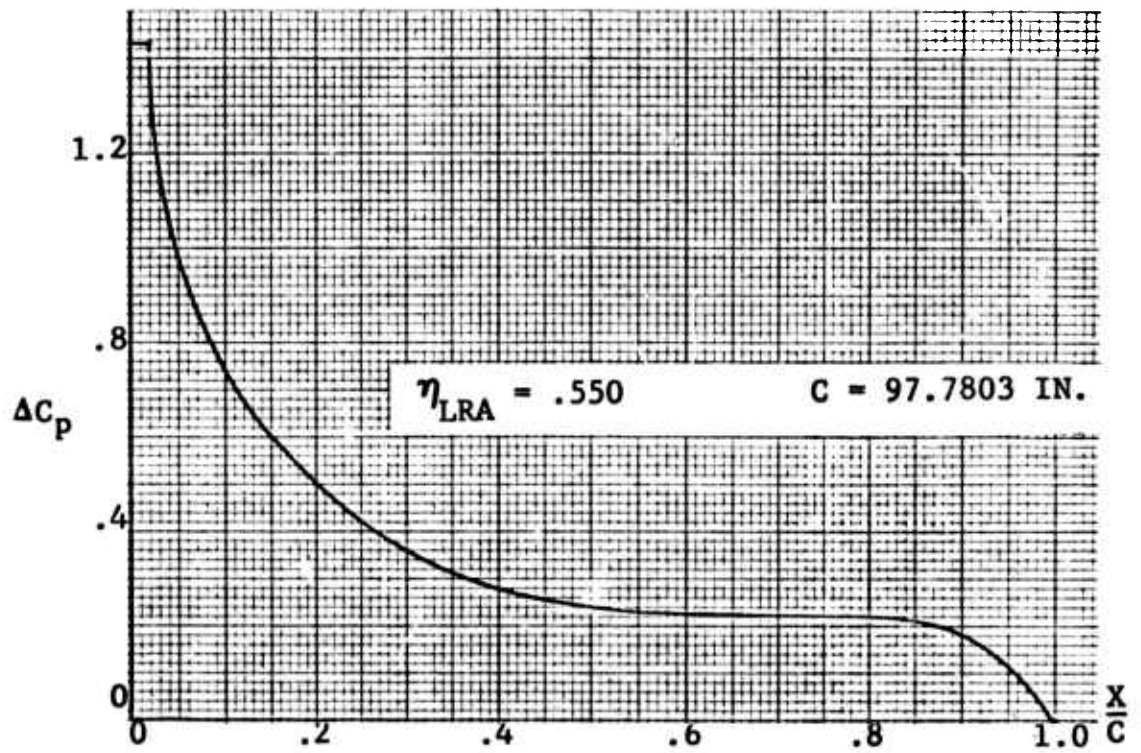
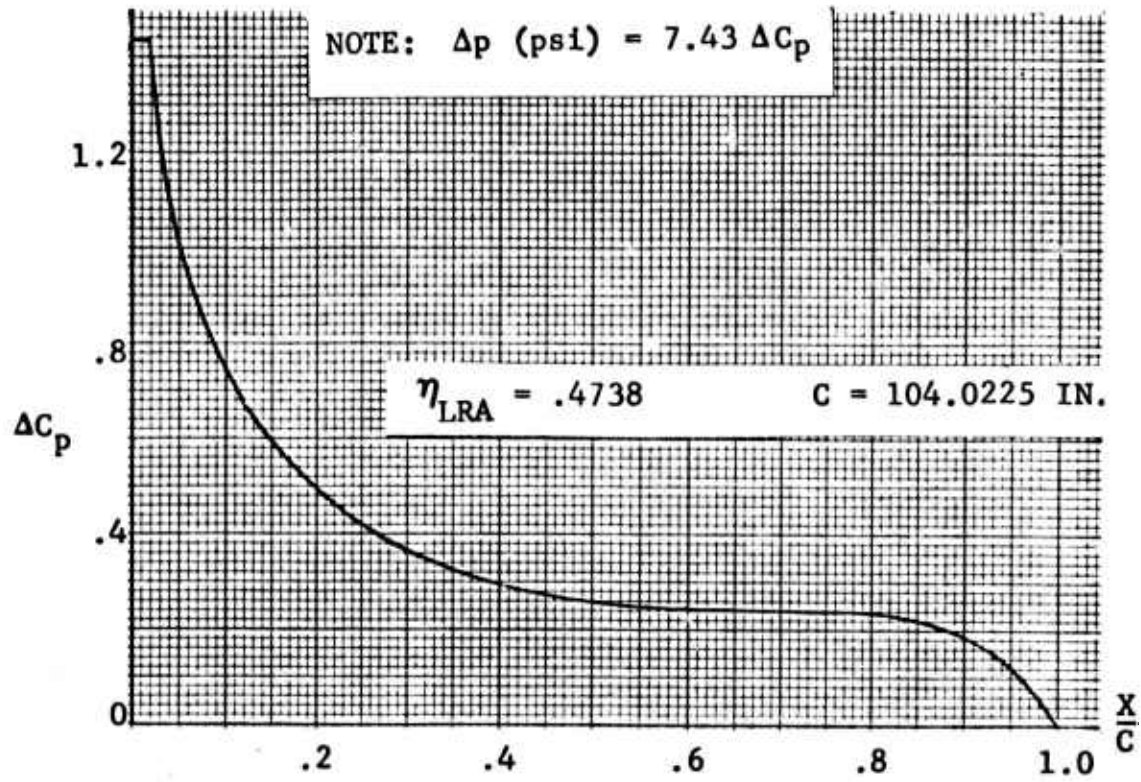


Figure A.22 ATW-4 Wing Chordwise Pressure Distributions for .4738 and .550 η_{LRA} - Condition 2

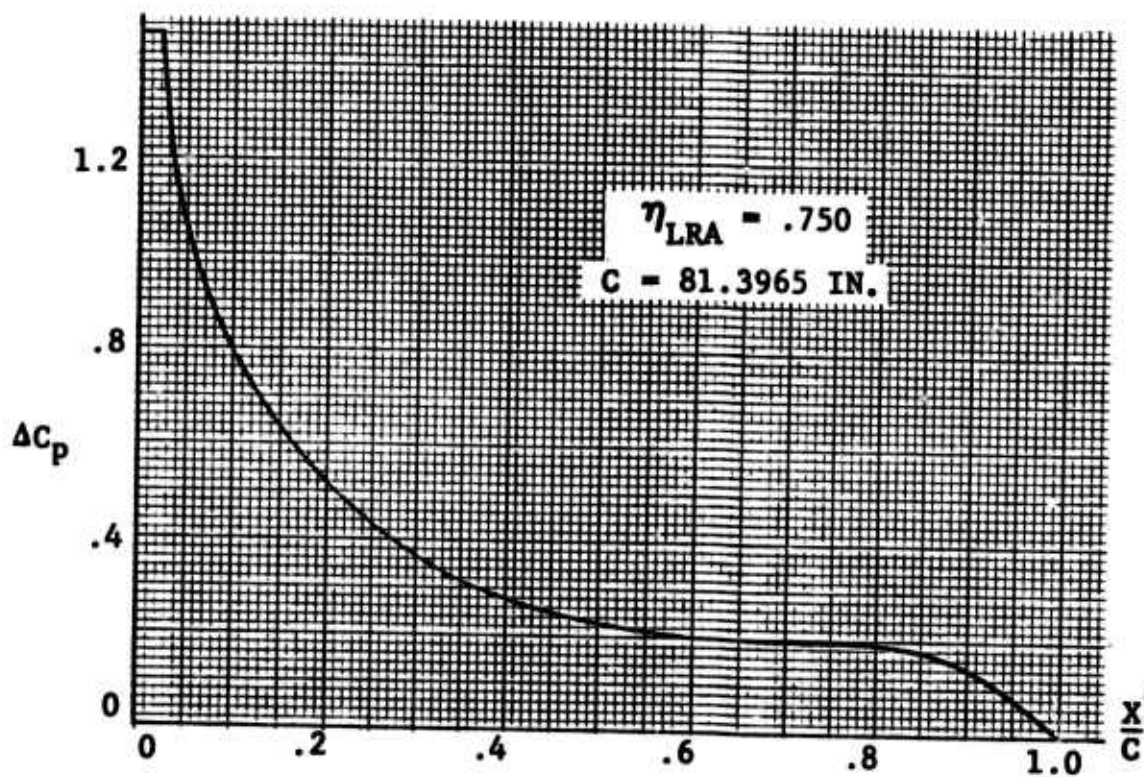
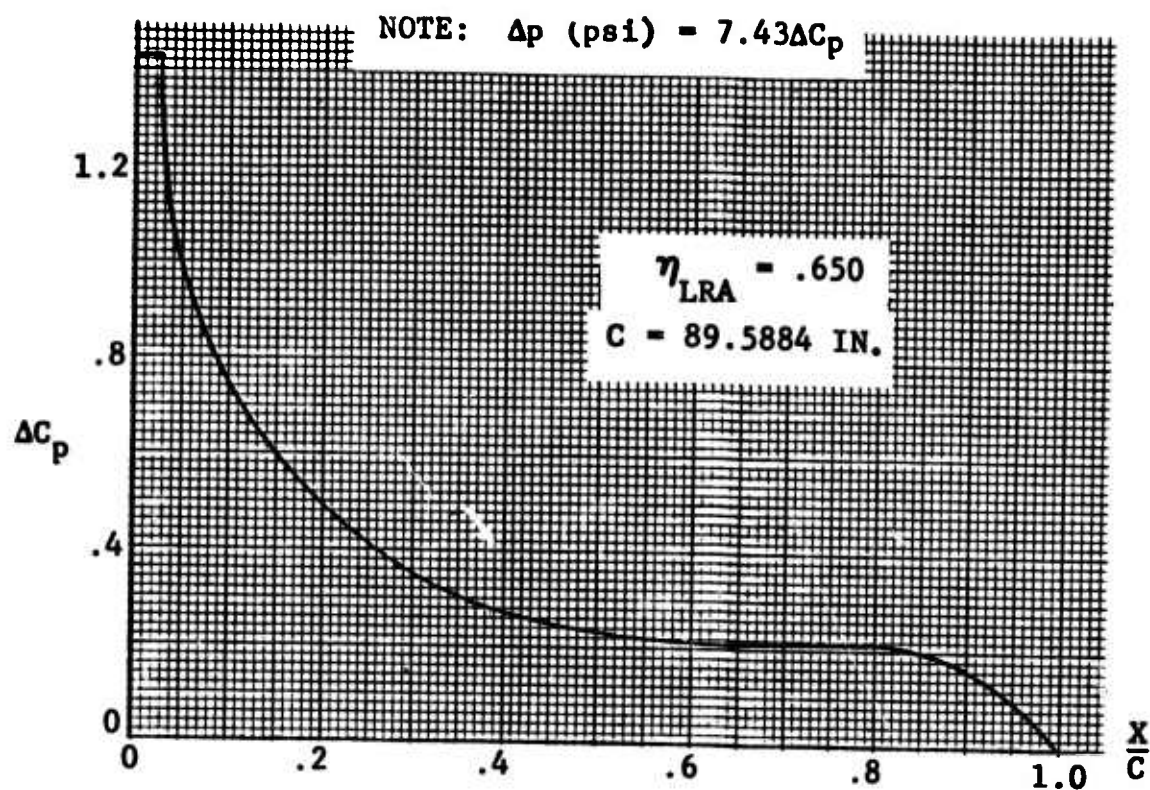


Figure A-23 ATW-4 Wing Chordwise Pressure Distributions for .650 and .750 η_{LRA} - Condition 2

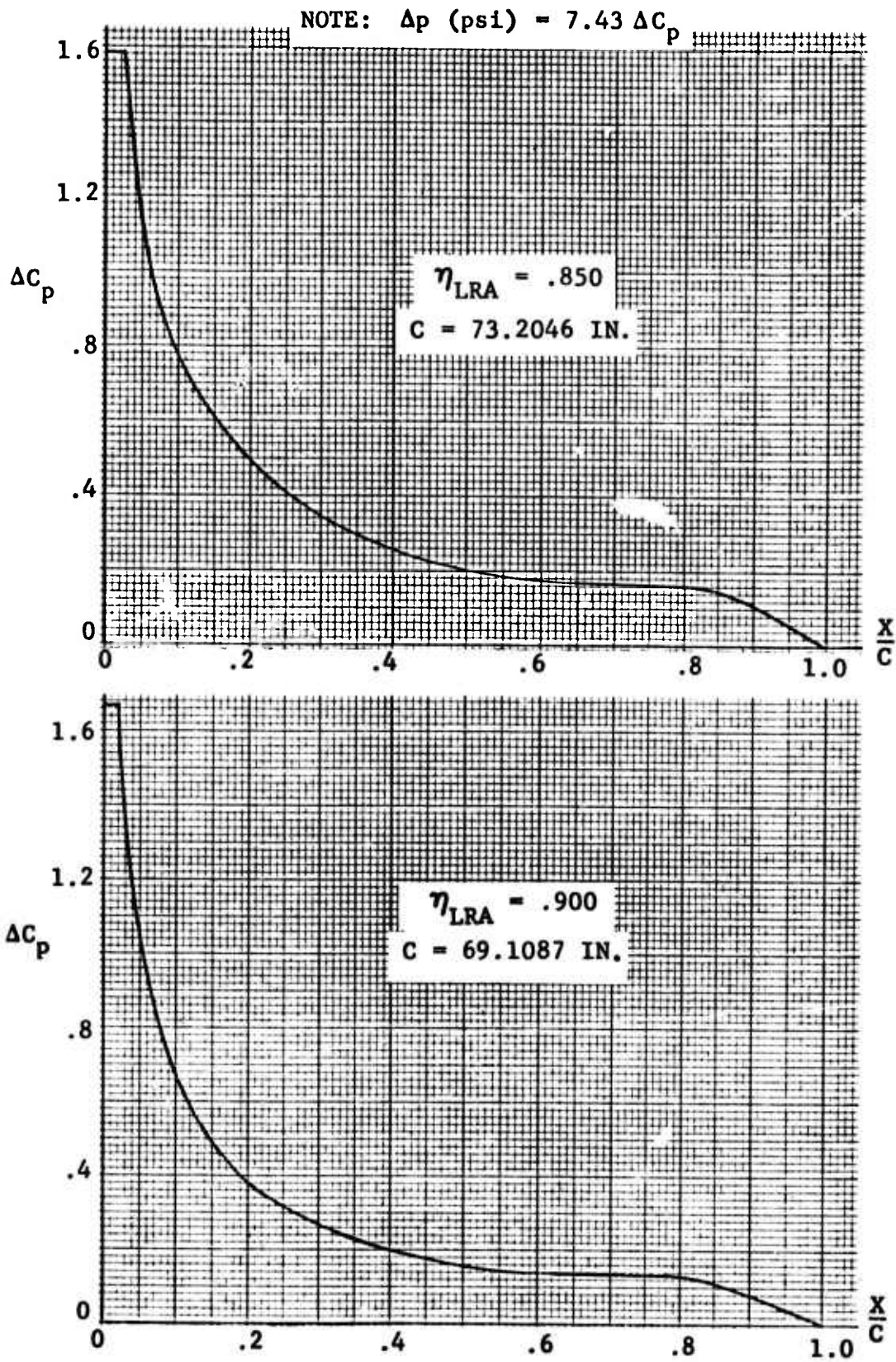


Figure A.24 ATW-4 Wing Chordwise Pressure Distributions for .850 and .900 η_{LRA} - Condition 2

NOTE: Δp (psi) = 7.43 ΔC_p

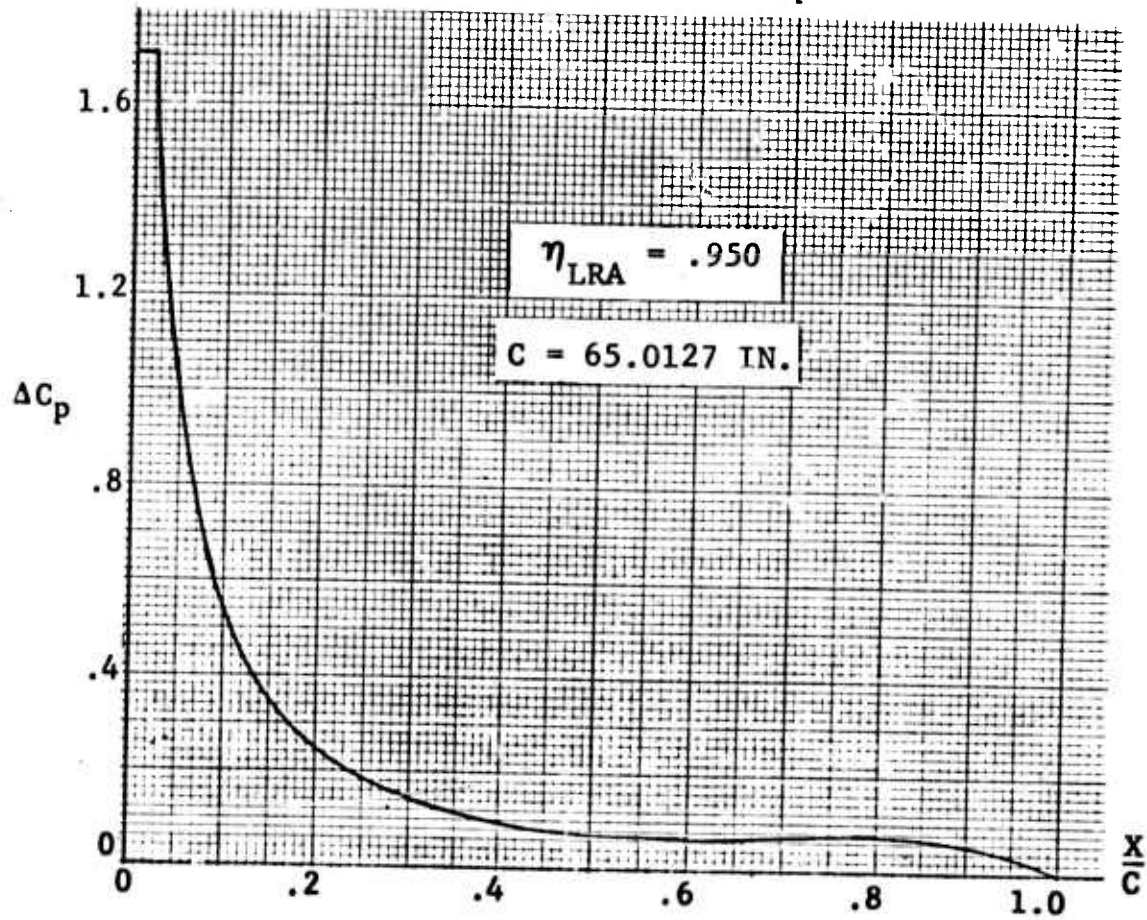


Figure A.25 ATW-4 Wing Chordwise Pressure Distribution for .950 η_{LRA} - Condition 2

NOTE:

1. LOCAL LIFT ON CHORDS \perp TO THE LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA

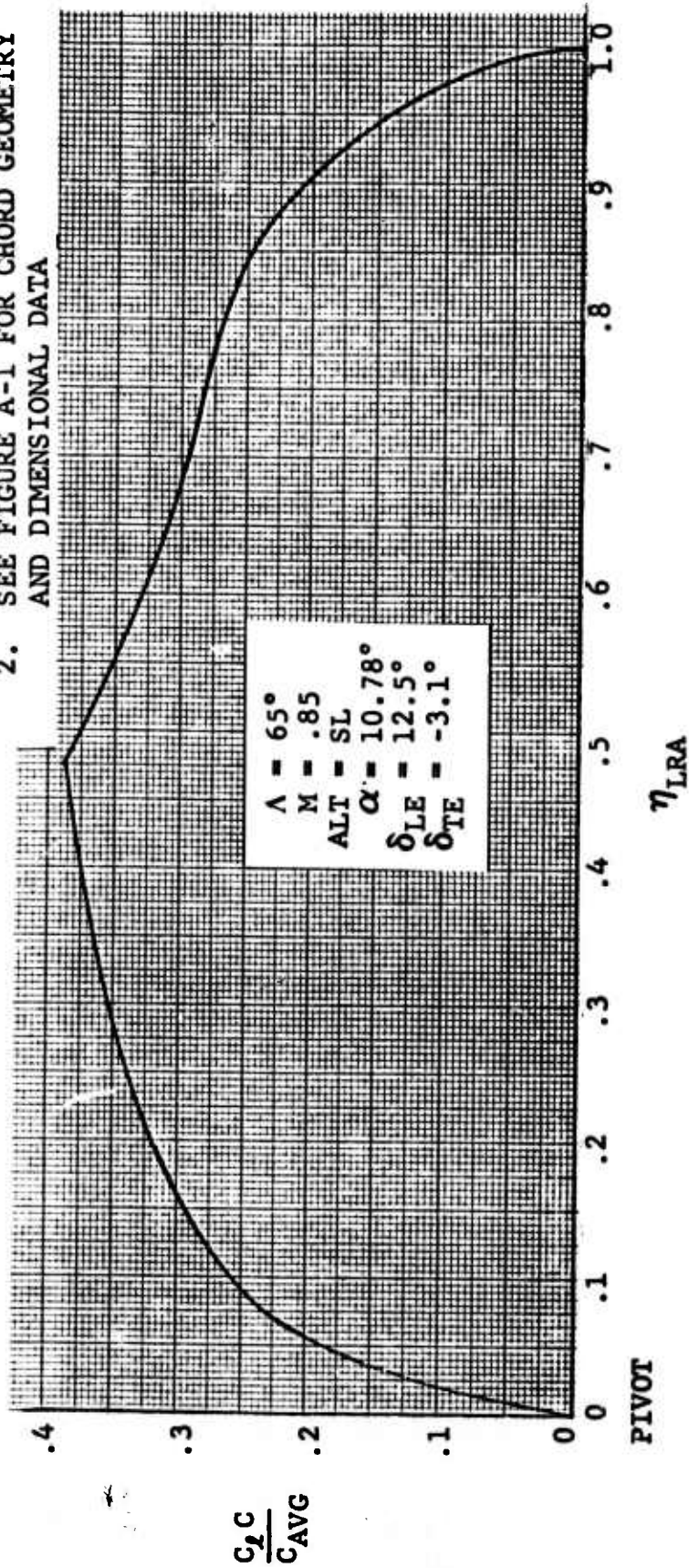


Figure A-26 ATW-4 Wing Local Lift Spanwise Distribution - Condition 2

NOTES:

1. LOCAL TORSION ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA.
3. MOMENT REFERENCE IS LRA
4. $\bar{c} = 117.1$ IN. (THEORETICAL REF. WING MAC)

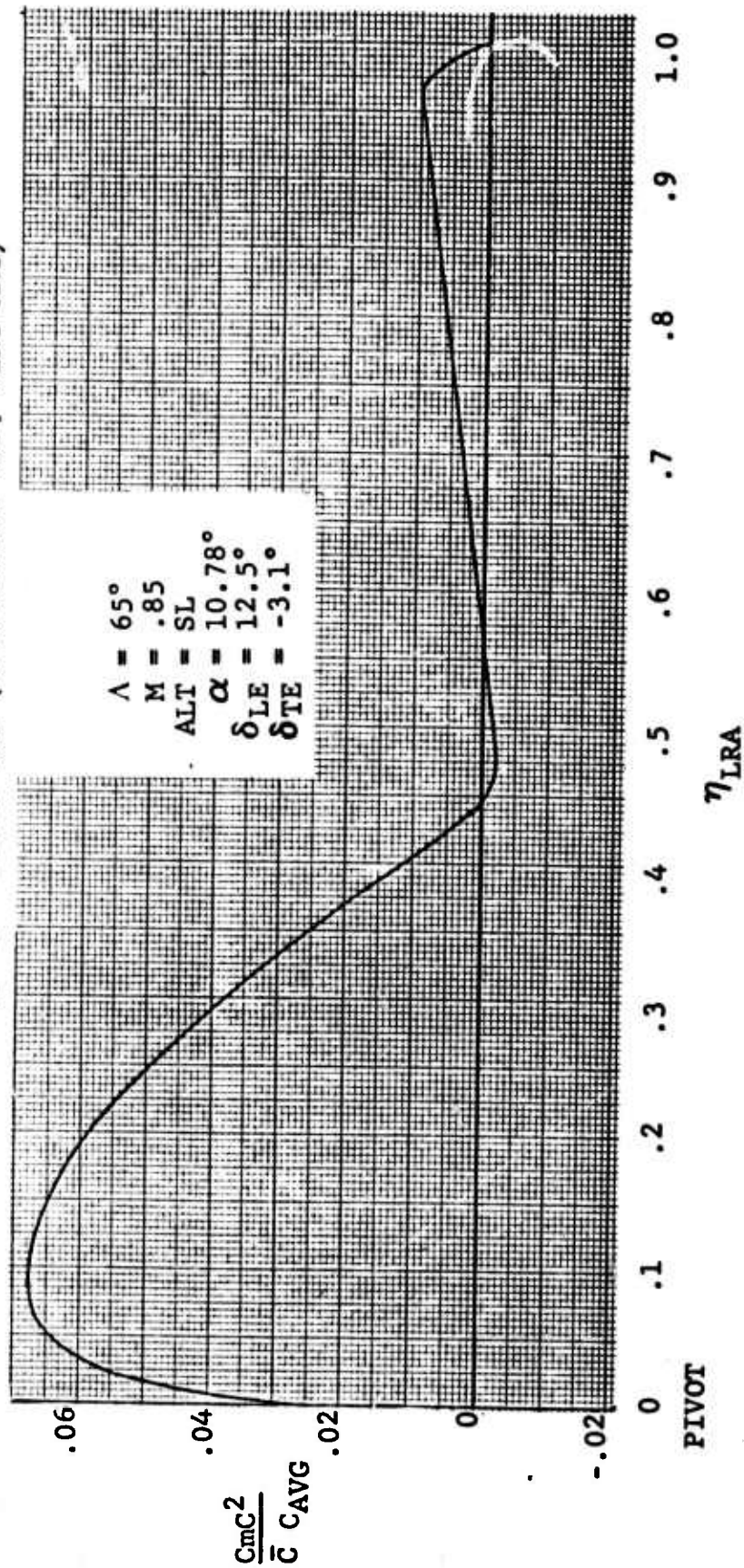


Figure A.27 ATW-4 Wing Local Torsion Spanwise Distribution - Condition 2

A.5.4 Condition 3 - Refuel with Symmetrical Maneuver

Figures A.28 through A.43 define the preliminary design loads for the refuel mission segment with symmetrical maneuver loads applied.

$\Lambda = 16^\circ$
 $M = .80$
ALT = 20,000 FT.
GW = 110,400 LBS
 $n_z = 2.58g$

$\delta_{LE} = 4^\circ$
 $\delta_{TE} = 3.5^\circ$
WING TANK EMPTY

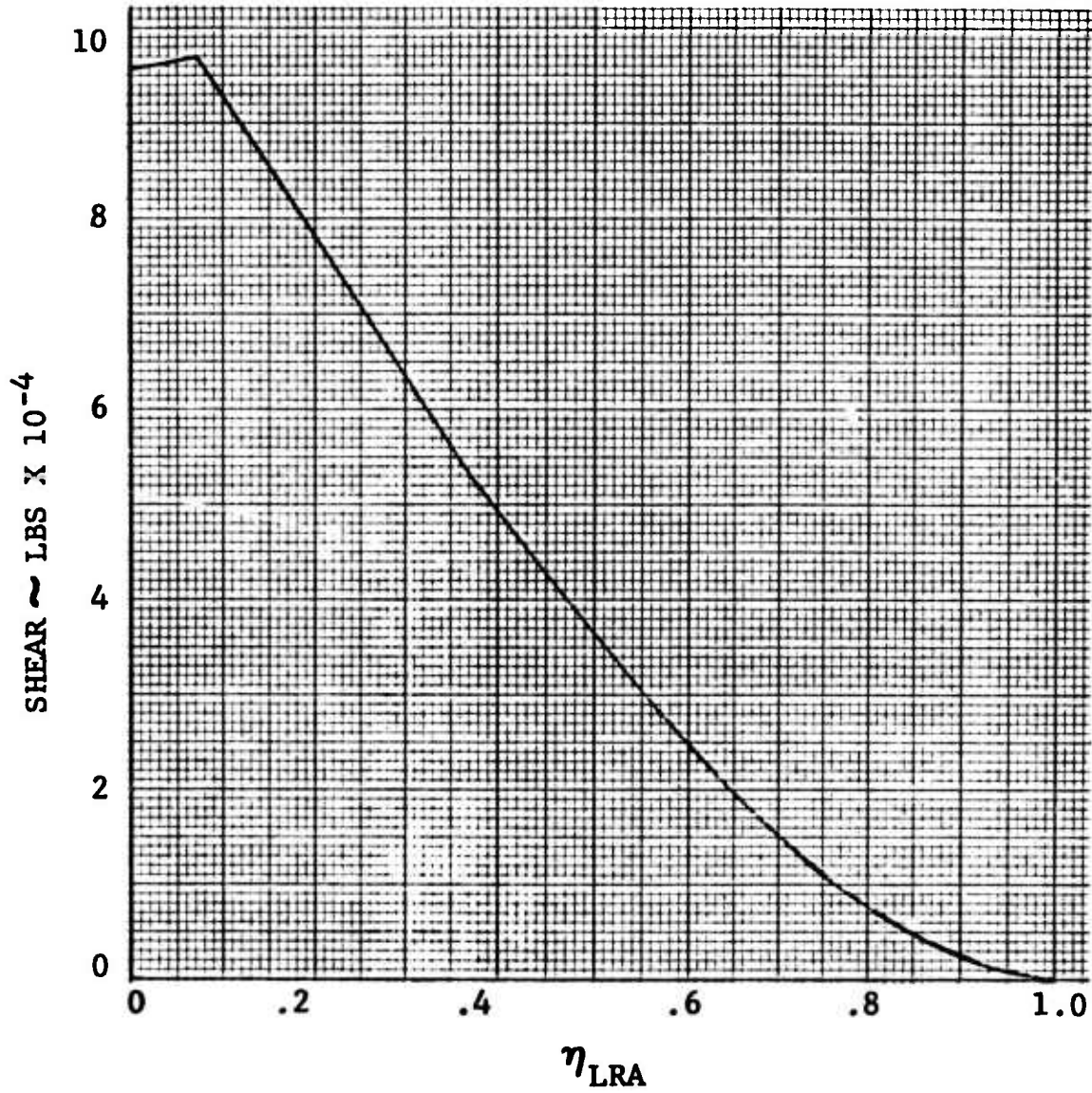


Figure A.28 ATW-4 Wing Shear Distribution - Condition 3

$\Lambda = 16^\circ$
 $M = .80$
ALT = 20,000 FT
GW = 110,400 LBS
 $n_z = 2.58g$

$\delta_{LE} = 4^\circ$
 $\delta_{TE} = 3.5^\circ$
WING TANKS EMPTY

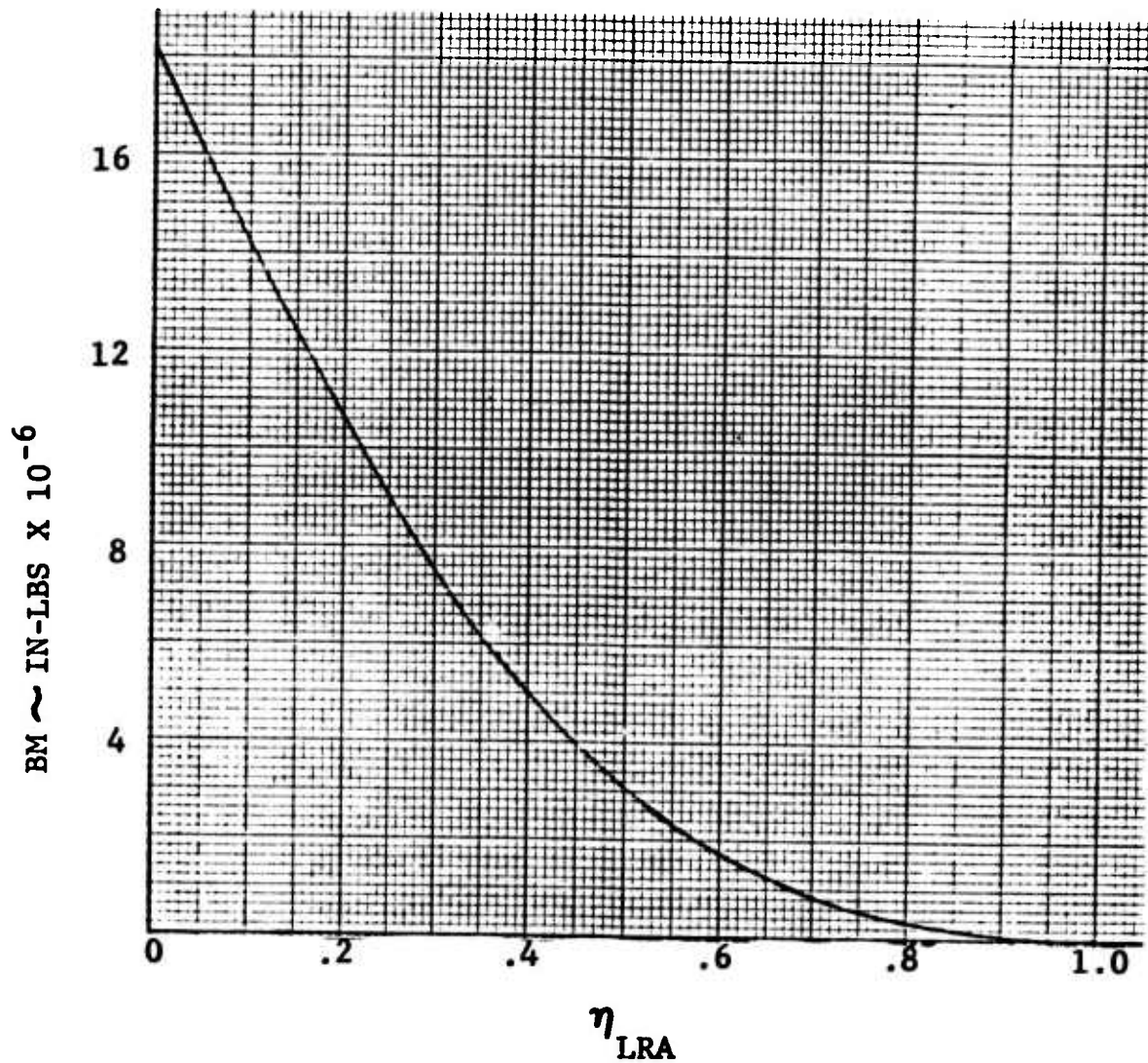


Figure A.29 ATW-4 Wing Bending Moment Distribution - Condition 3

$\Lambda = 16^\circ$	$\delta_{LE} = 4^\circ$
$M = .80$	$\delta_{TE} = 3.5^\circ$
ALT = 20,000 FT.	WING TANKS EMPTY
GW = 110,400 LBS	
$n_z = 2.58g$	

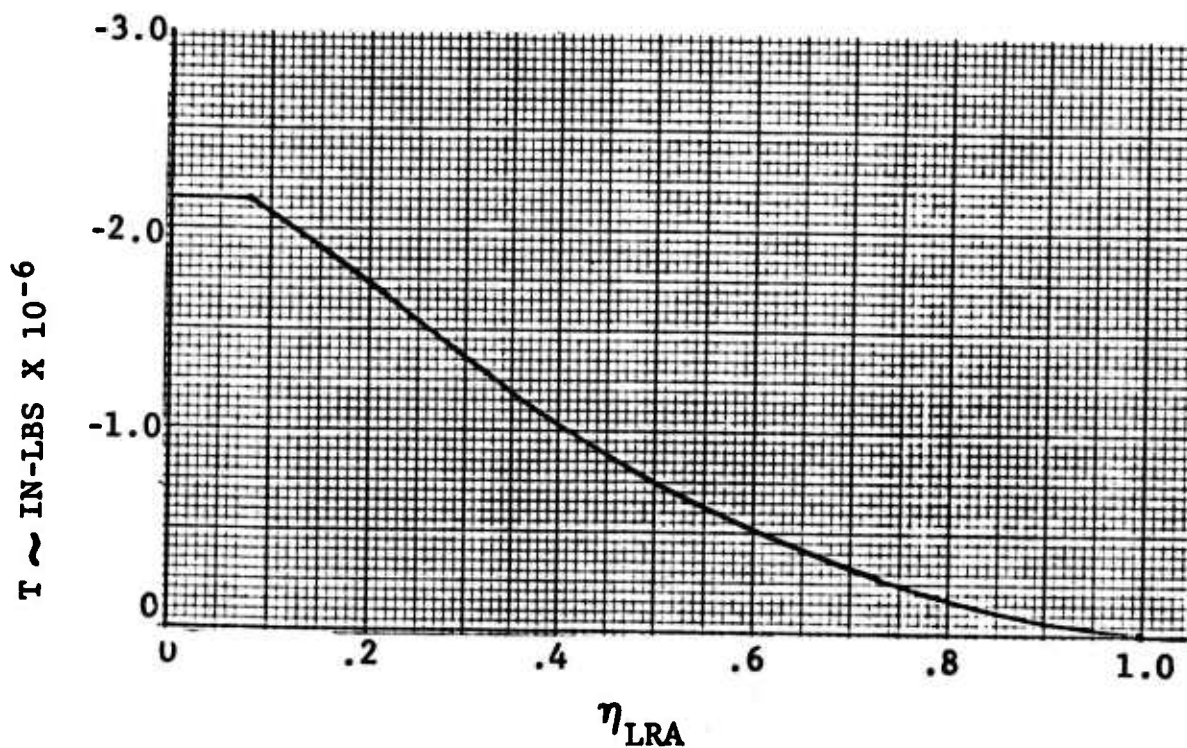


Figure A.30 ATW-4 Wing Torsion Distribution - Condition 3

NOTES:

1. X_{cp} DISTANCE FWD. OF .15C FOR L.E. FLAP C.P.
2. X_{cp} DISTANCE AFT OF .65C FOR T.E. FLAP C.P.
3. C_F IS LOCAL FLAP CHORD

$\Lambda = 16^\circ$
 $M = .80$
 ALT = 20,000 FT.
 GW = 110,400 LBS.
 $n_z = 2.6g$
 $\delta_{LE} = 4^\circ$
 $\delta_{TE} = 3.5^\circ$

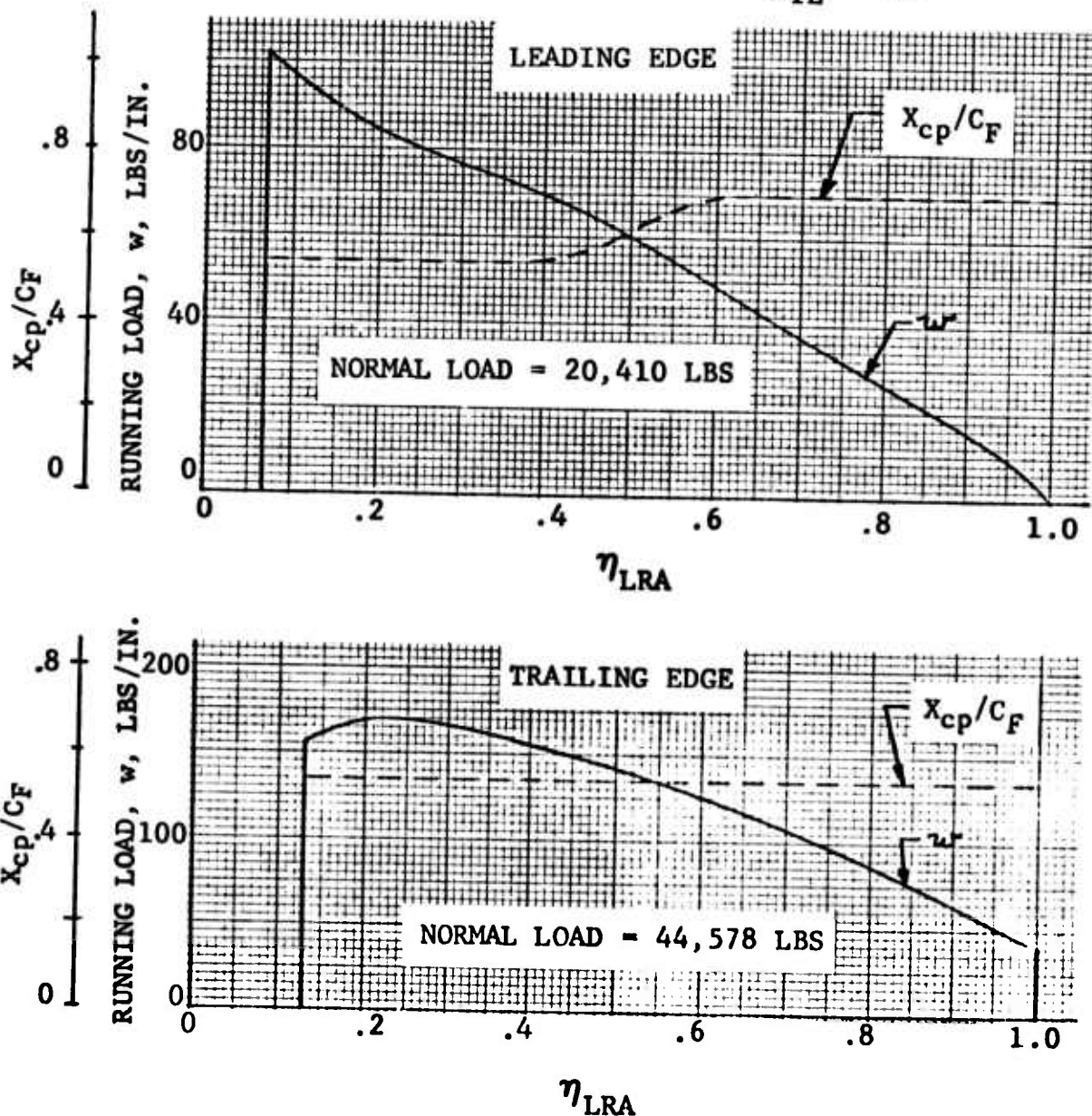


Figure A.31 ATW-4 Wing Leading and Trailing Edge Flap Running Loads - Condition 3

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

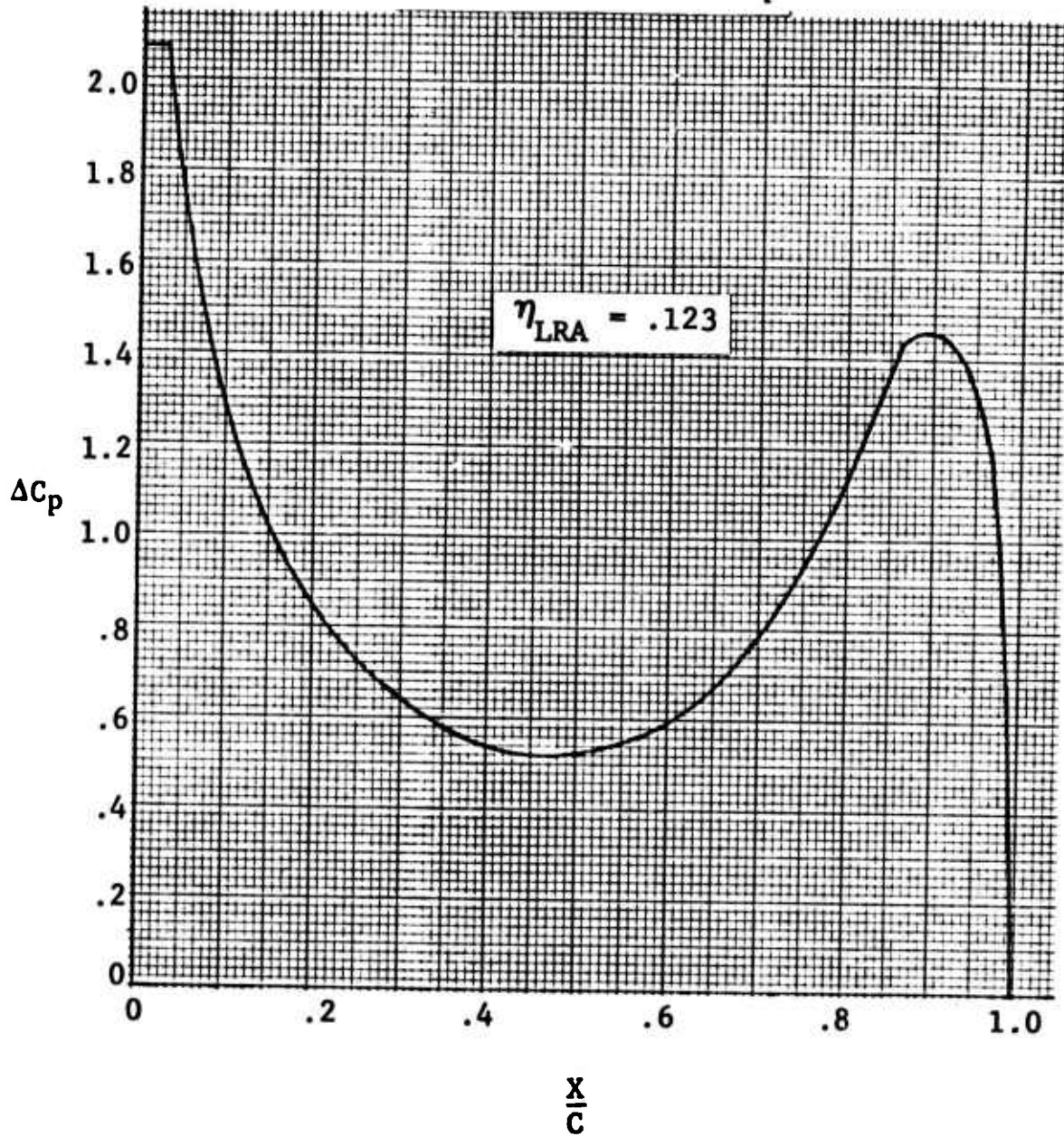


Figure A.32 ATW-4 Wing Chordwise Pressure Distribution for .123 η_{LRA} - Condition 3

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

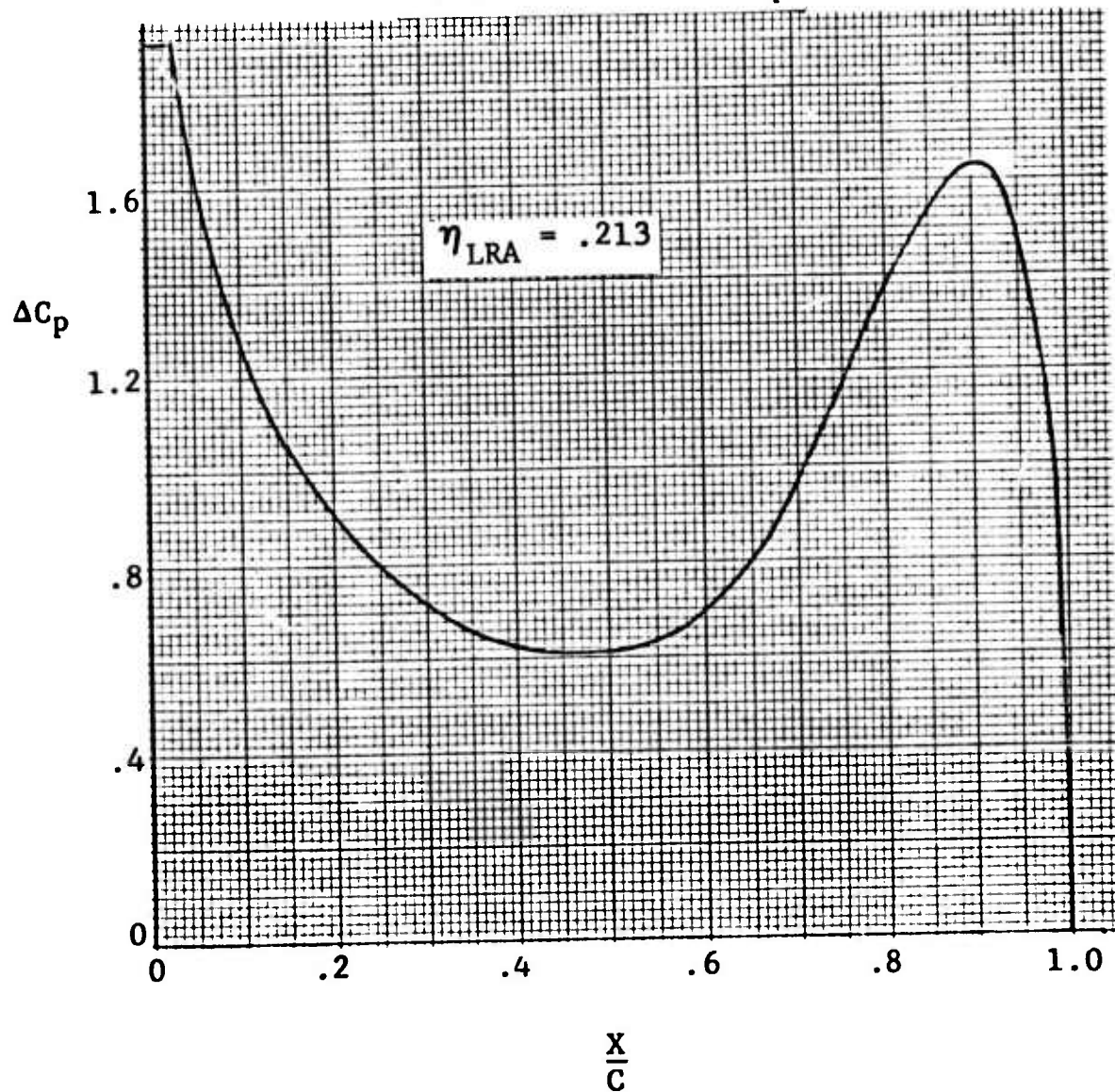


Figure A.33 ATW-4 Wing Chordwise Pressure Distribution for
.213 η_{LRA} - Condition 3

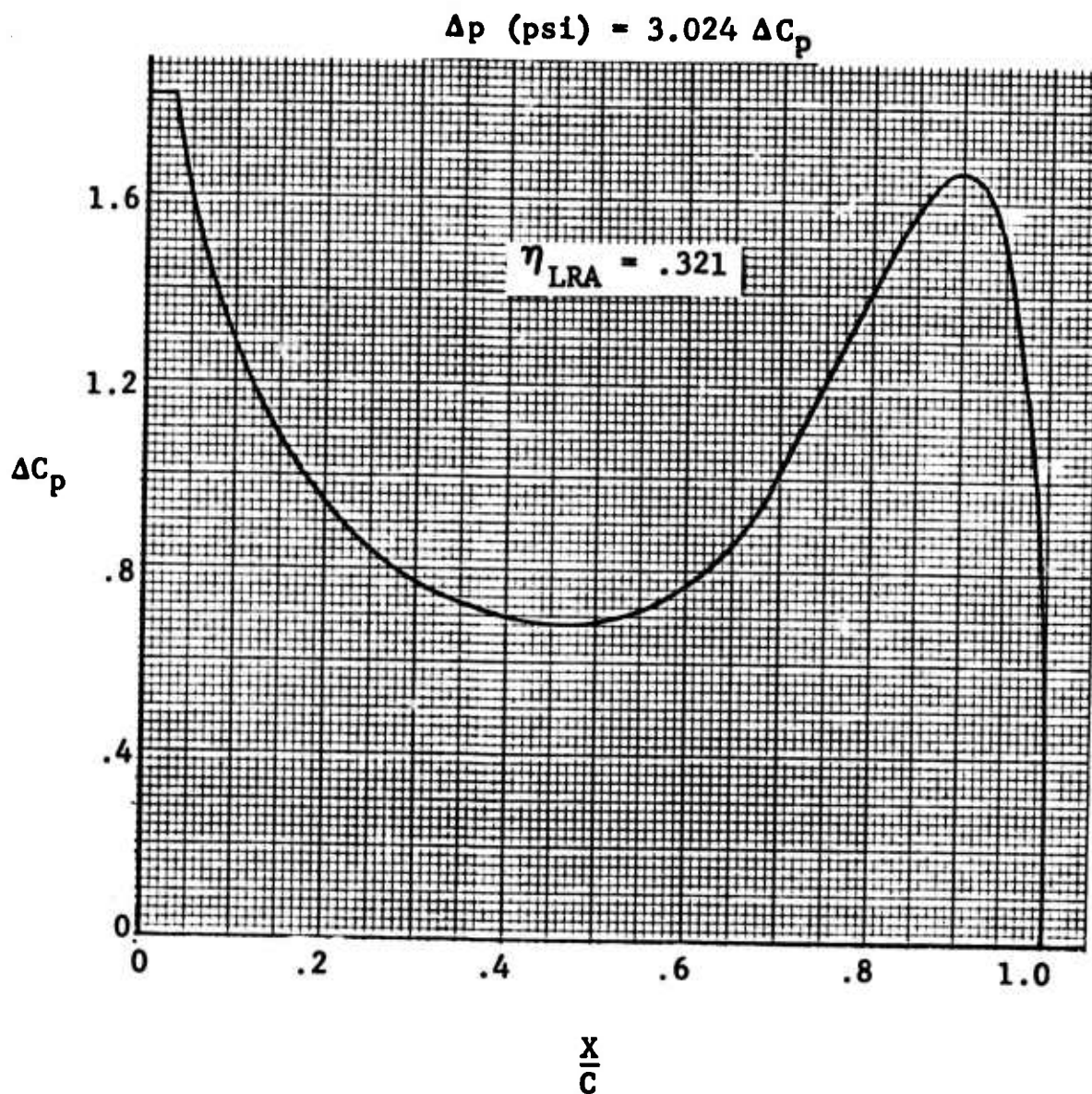


Figure A.34 ATW-4 Wing Chordwise Pressure Distribution for
 .321 η_{LRA} - Condition 3

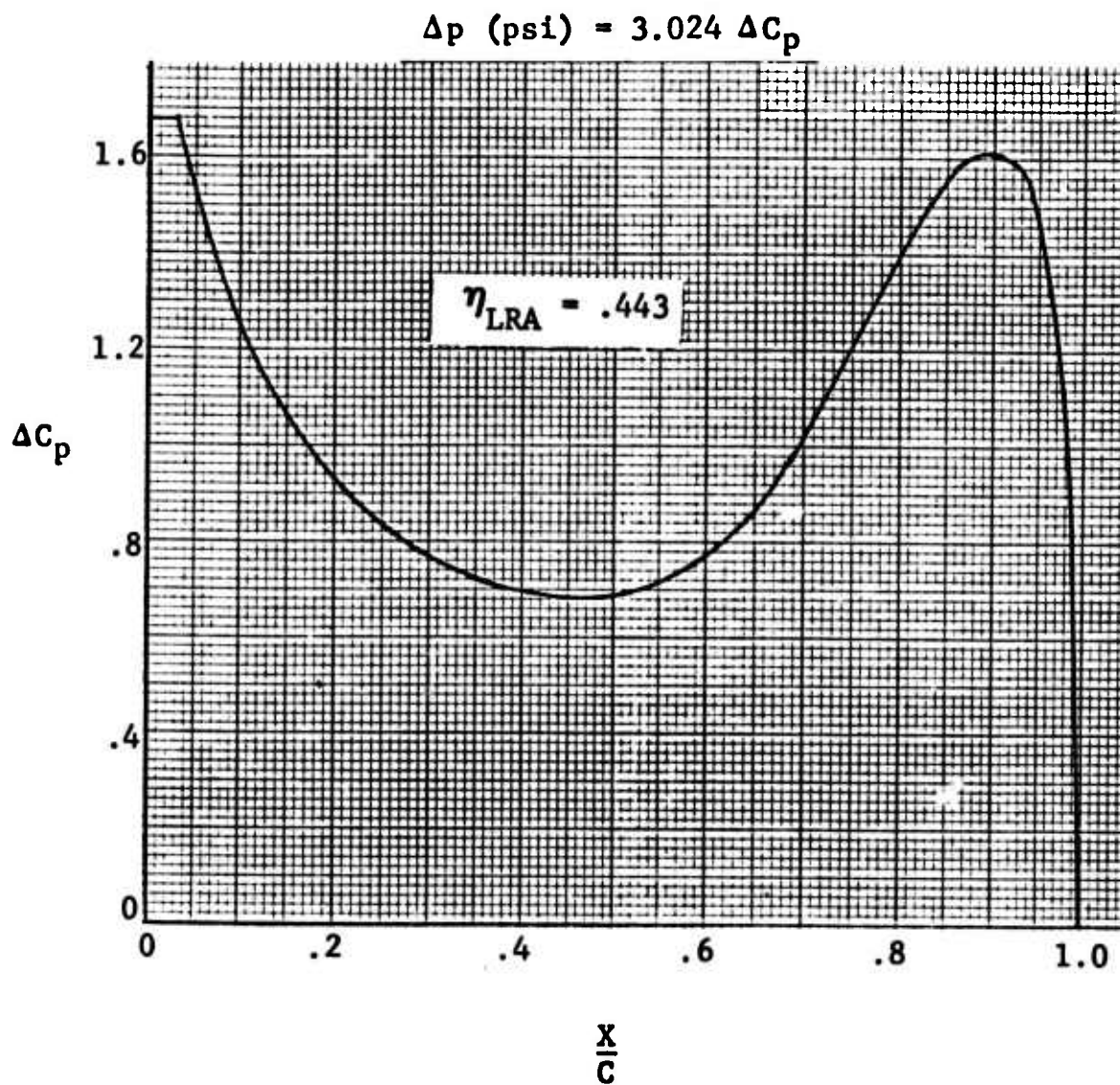


Figure A.35 ATW-4 Wing Chordwise Pressure Distribution for
 .443 η_{LRA} - Condition 3

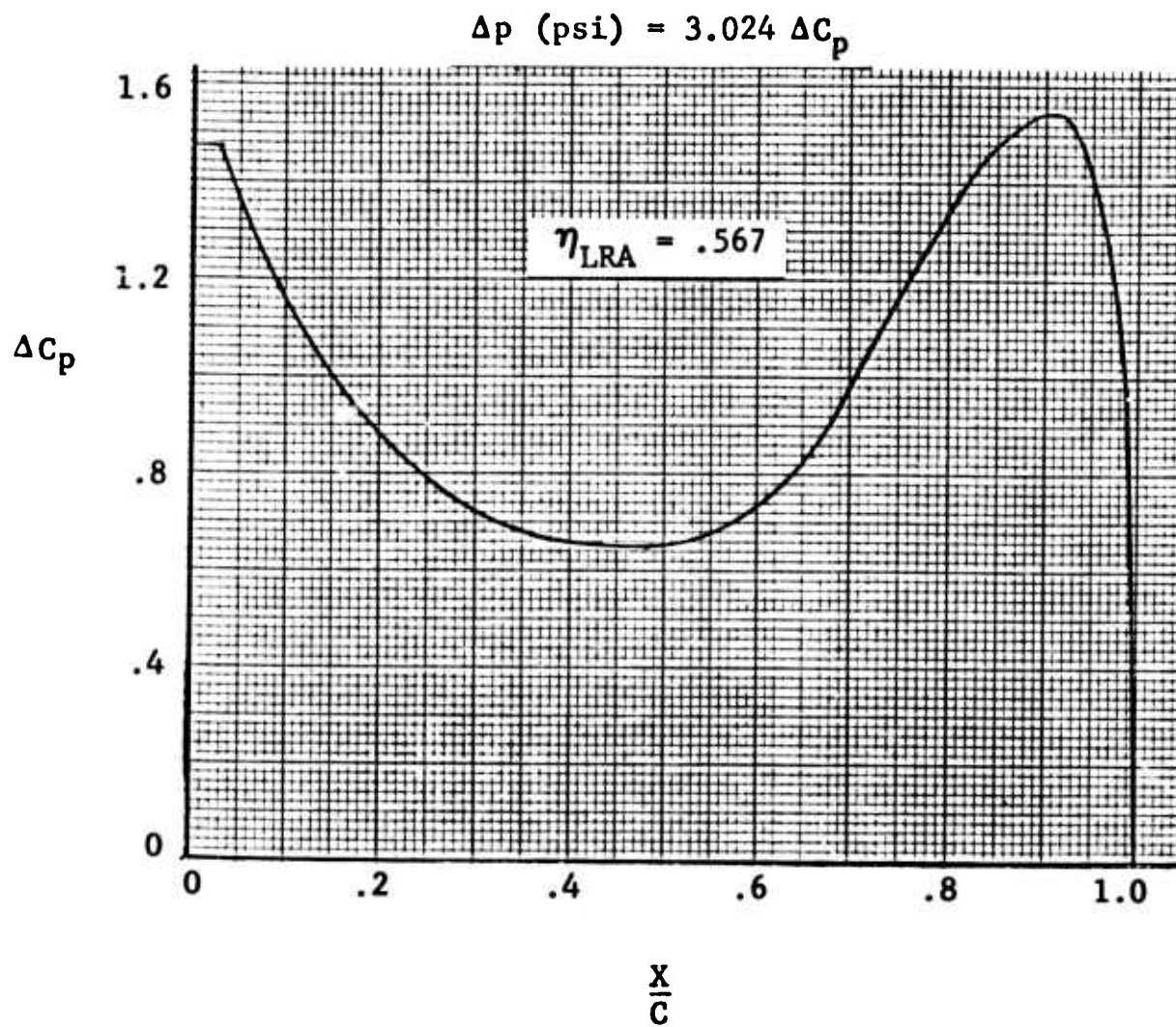


Figure A.36 ATW-4 Wing Chordwise Pressure Distribution for .567 η_{LRA} - Condition 3

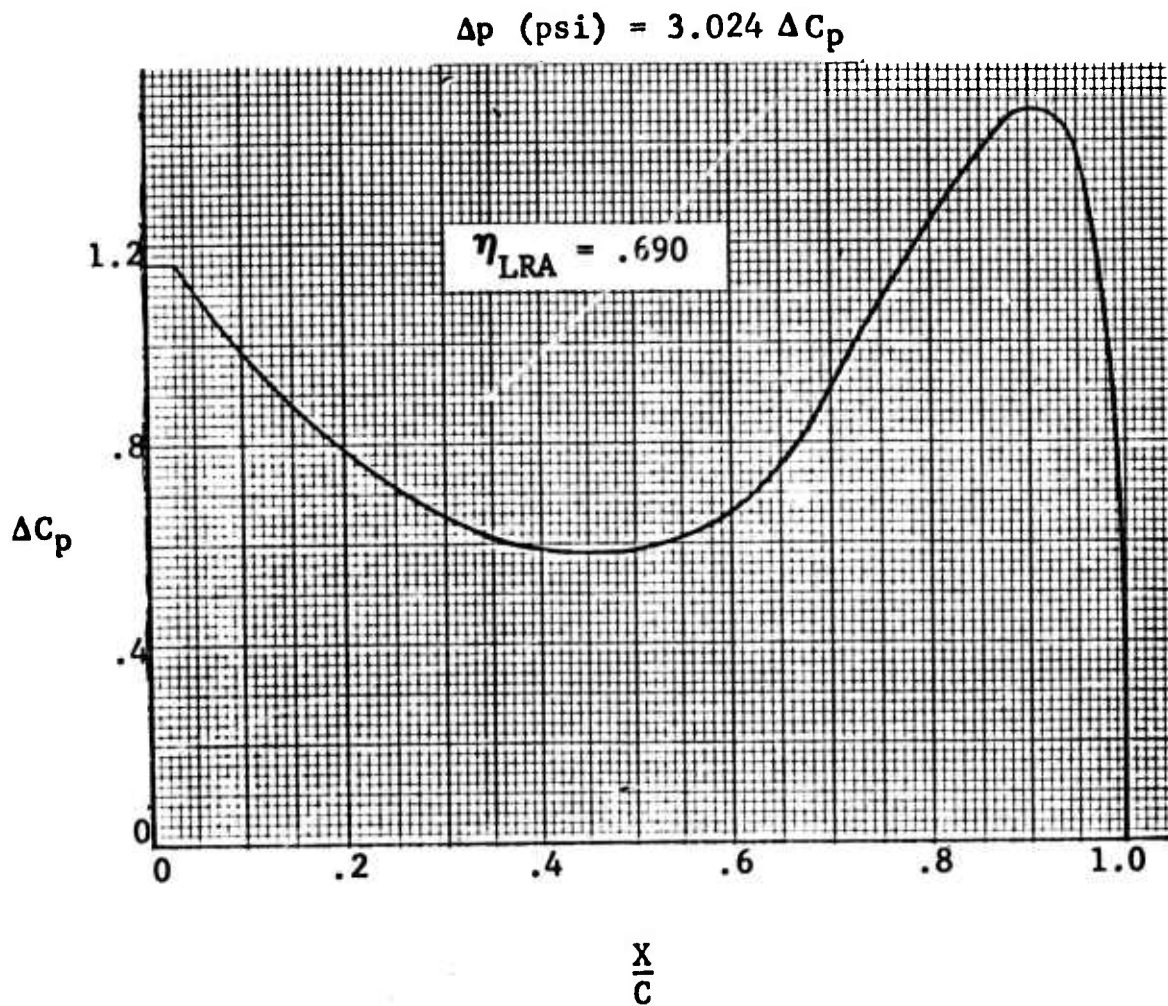


Figure A.37 ATW-4 Wing Chordwise Pressure Distribution for .690 η_{LRA} - Condition 3

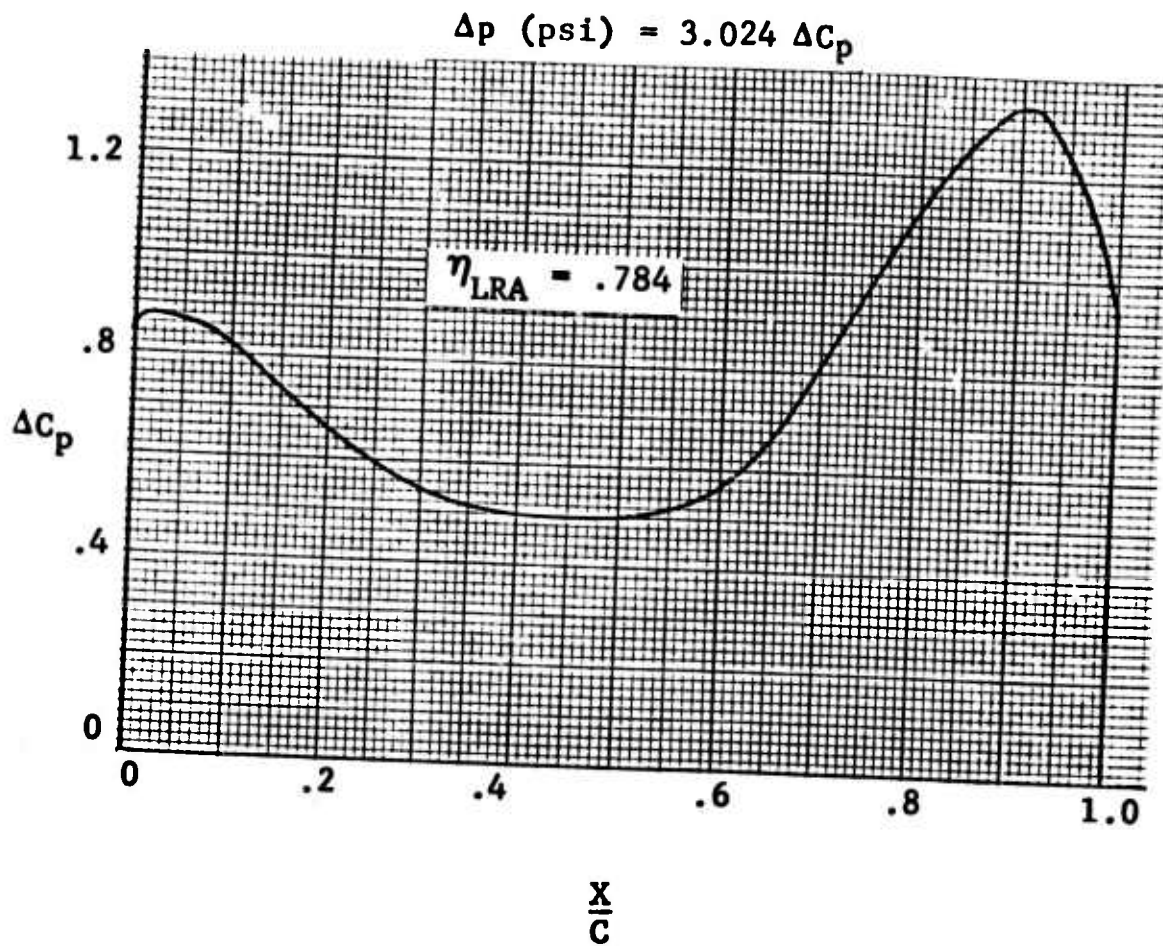


Figure A.38 ATW-4 Wing Chordwise Pressure Distribution for .784 η_{LRA} - Condition 3

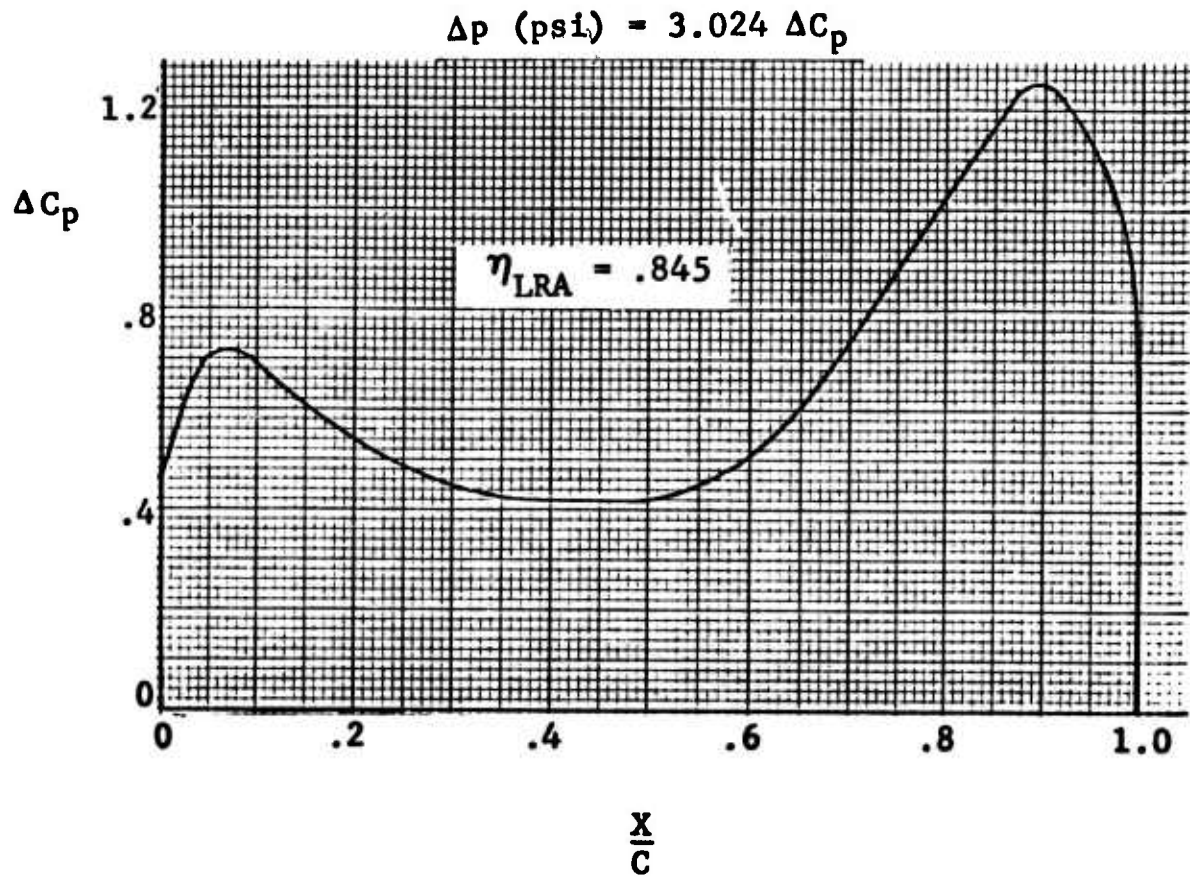


Figure A.39 ATW-4 Wing Chordwise Pressure Distribution for .845 η_{LRA} - Condition 3

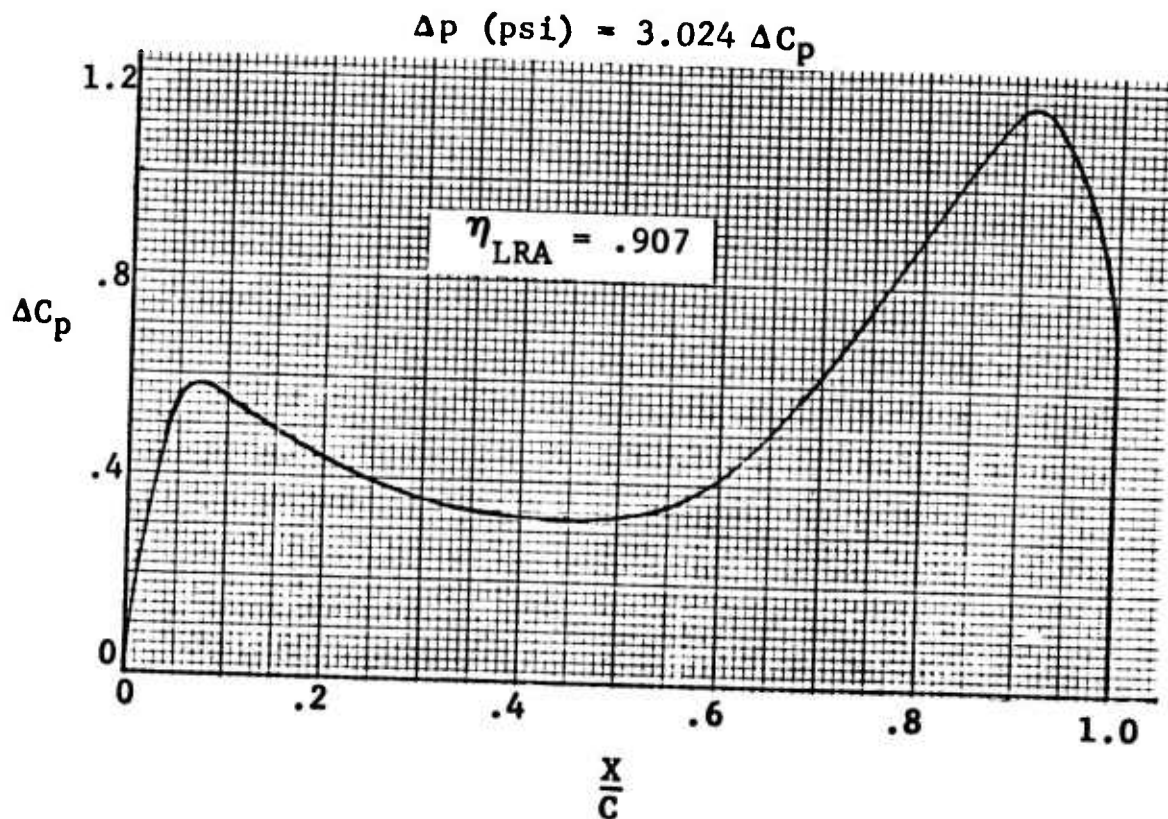


Figure A.40 ATW-4 Wing Chordwise Pressure Distributions
for .907 η_{LRA} - Condition 3

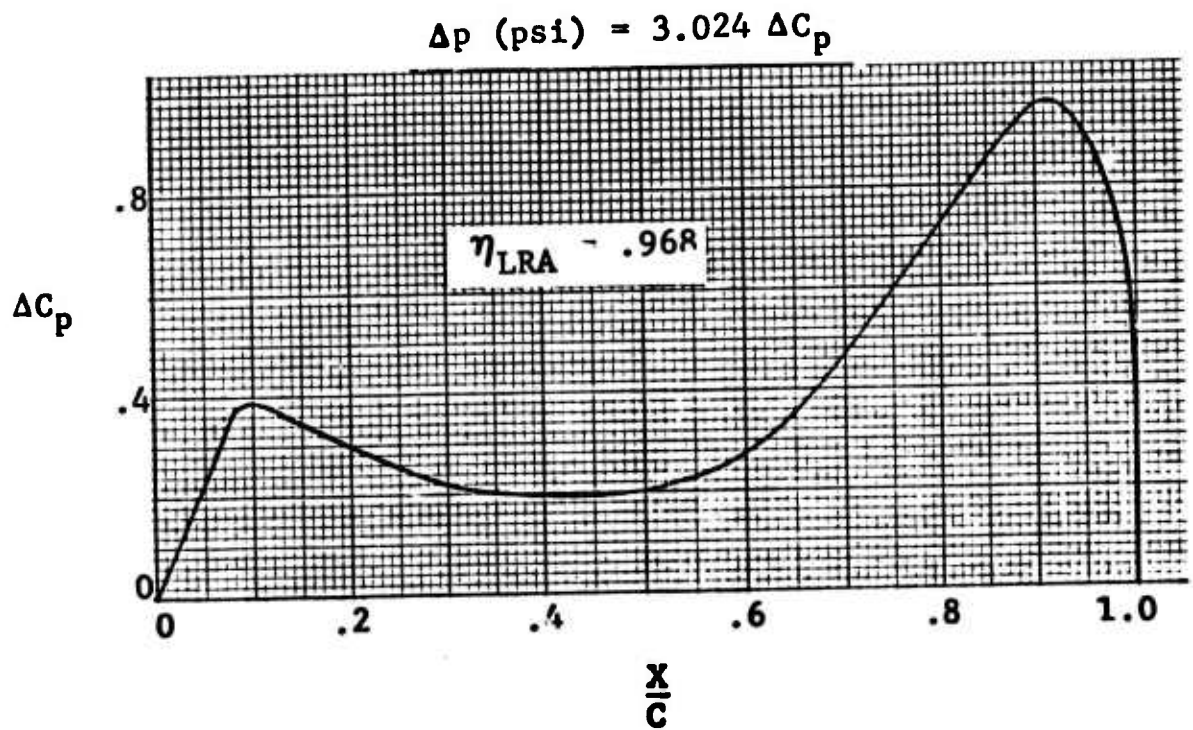


Figure A.41 ATW-4 Wing Chordwise Pressure Distribution for .968 η_{LRA} - Condition 3

NOTES:

1. LOCAL LIFT ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA

$\Lambda = 16^\circ$
 $M = .80$
 $ALT = 20,000 \text{ FT}$
 $GW = 110,400 \text{ LBS}$
 $\alpha = 6.68^\circ$
 $\delta_{LE} = 4.0^\circ$
 $\delta_{TE} = 3.5^\circ$

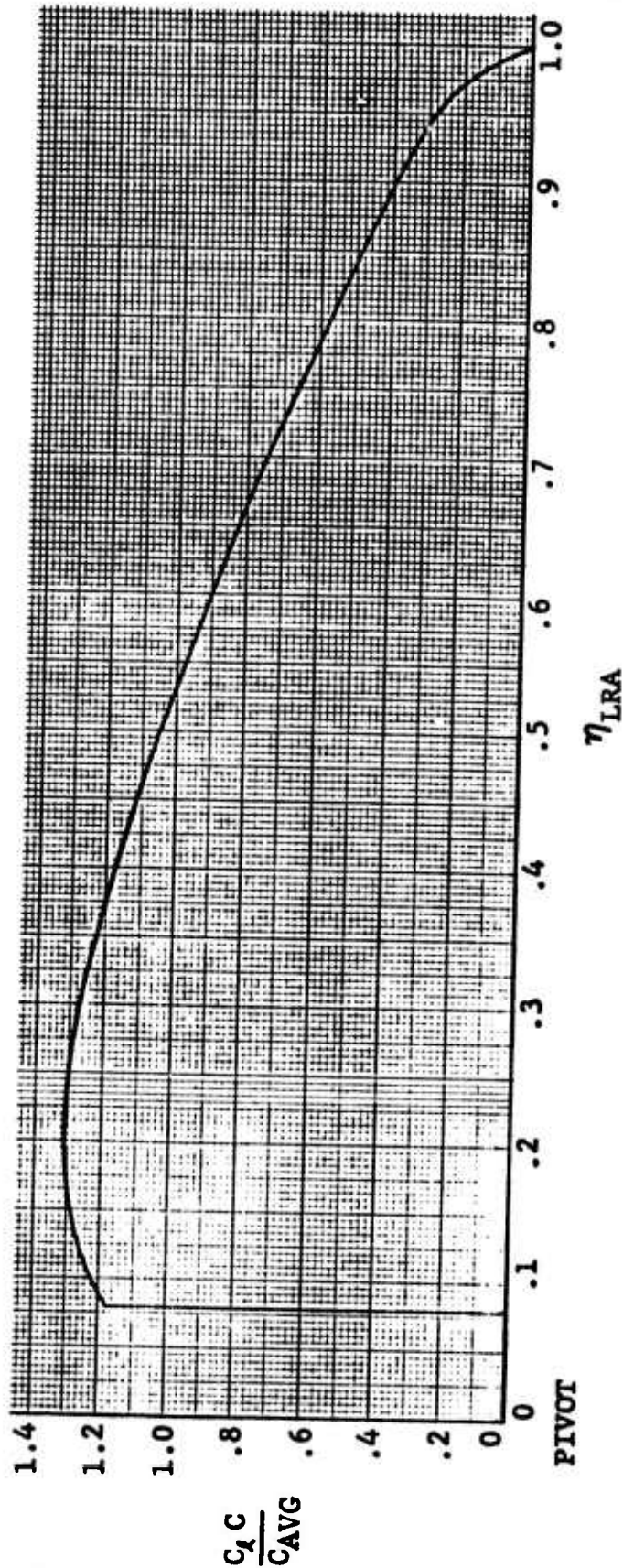


Figure A.42 ATW-4 Wing Local Lift Spanwise Distribution - Condition 3

$\Lambda = 16^\circ$
 $M = .80$
 $ALT = 20,000 \text{ FT}$
 $GW = 110,400 \text{ LBS}$
 $\alpha = 6.68^\circ$
 $\delta_{LE} = 4.0^\circ$
 $\delta_{TE} = 3.5^\circ$

NOTES:

1. LOCAL TORSION ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA
3. MOM. REF. IS LRA
4. $\bar{c} = 117.1 \text{ IN. (THEORETICAL REF. WING MAC)}$

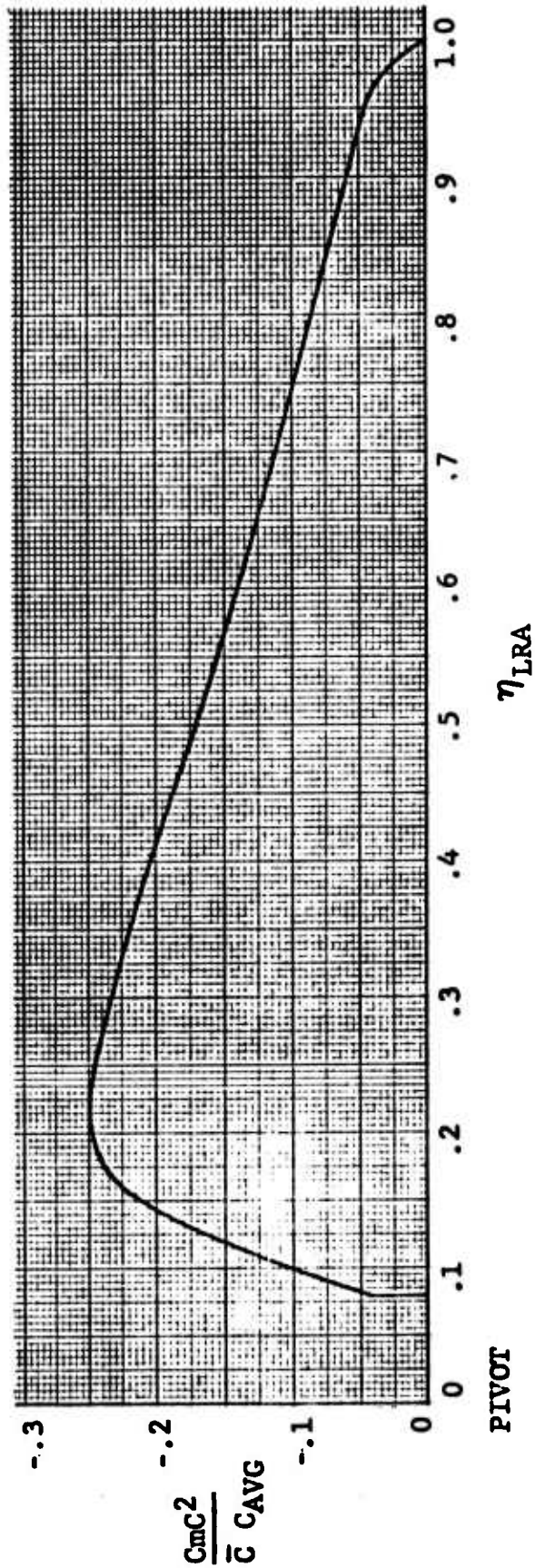


Figure A.43 ATW-4 Wing Local Torsion Spanwise Distribution - Condition 3

A.5.5 Condition 4 - Refuel with Roll Maneuver

Figures A.44 through A.58 define the preliminary design loads for the refuel mission segment with roll maneuver loads applied.

$\Lambda = 16^\circ$
 $M = .80$
 ALT = 20,000 FT
 GW = 110,400 LBS (EMPTY WING TANKS)
 $n_z = 1.92g$
 $\delta_{LE} = 10^\circ$
 $\delta_{TE} = 3.5^\circ$ @ I/B END TO 8.5° @ O/B END

NOTE: EFFECTS OF ROLL DAMPING AND ROLL ACCELERATION ARE NEGLECTED.

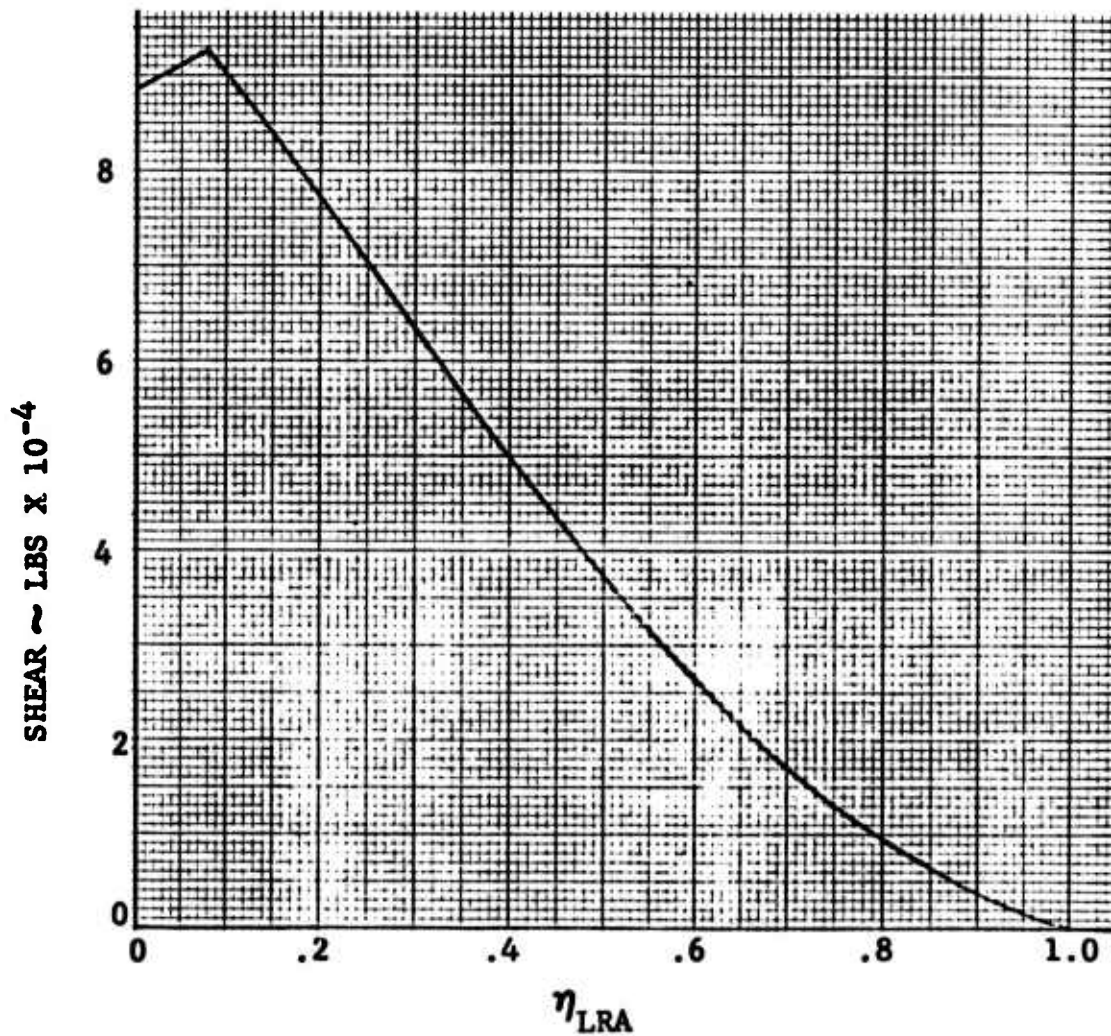


Figure A.44 ATW-4 Wing Shear Distribution - Condition 4

$\Lambda = 16^\circ$
 $M = .80$
 ALT = 20,000 FT
 GW = 110,400 LBS (EMPTY WING TANKS)
 $n_z = 1.92g$
 $\delta_{LE} = 10^\circ$
 $\delta_{TE} = 3.5^\circ @ I/B \text{ END TO } 8.5^\circ @ O/B \text{ END}$

NOTE: EFFECTS OF ROLL DAMPING AND ROLL ACCELERATION ARE NEGLECTED.

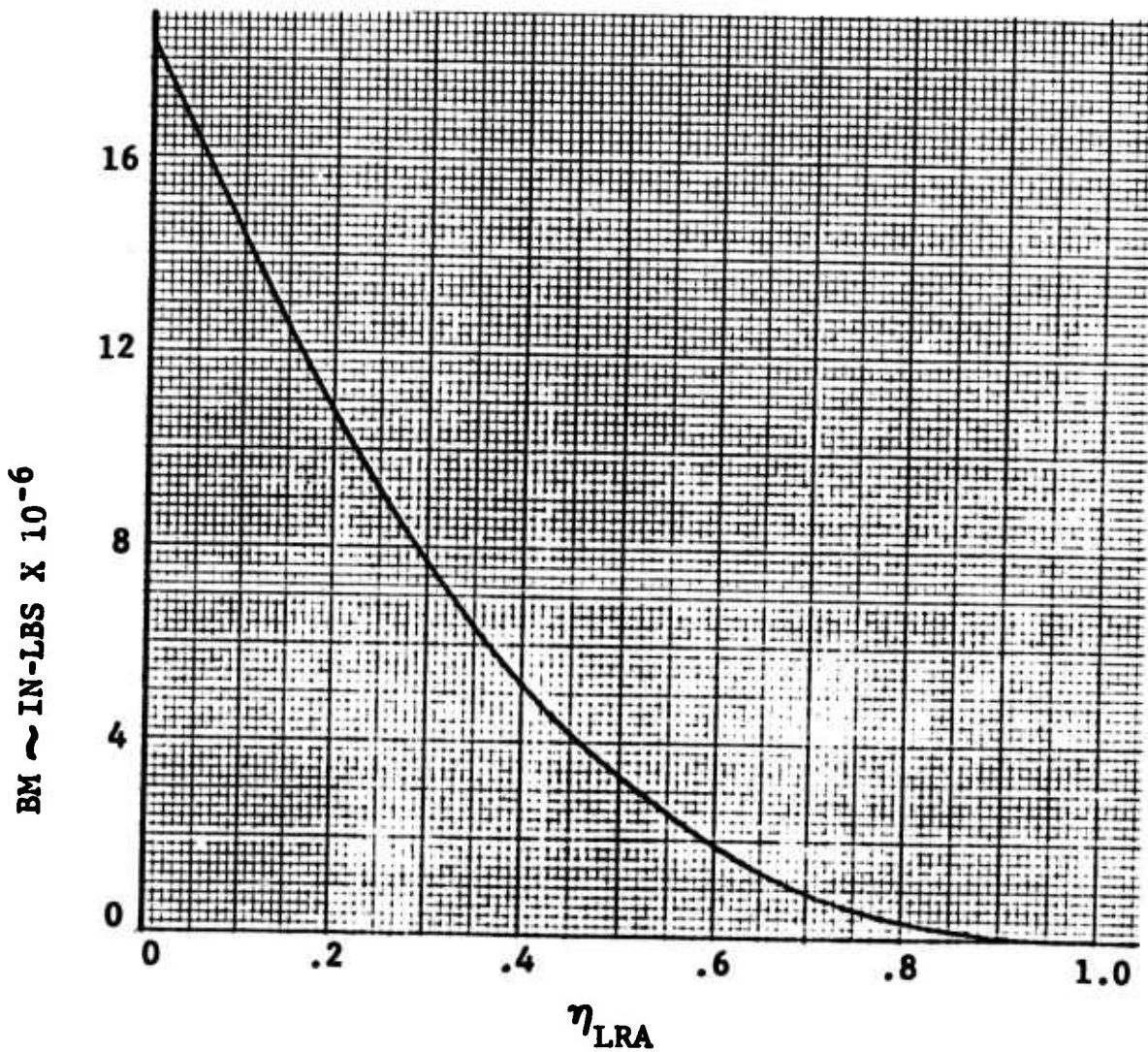


Figure A.45 ATW-4 Wing Bending Moment Distribution - Condition 4

$\Lambda = 16^\circ$
 $M = .80$
 $ALT = 20,000 \text{ FT}$
 $GW = 110,400 \text{ LBS (EMPTY WING TANKS)}$
 $n_z = 1.92g$
 $\delta_{LE} = 10^\circ$
 $\delta_{TE} = 3.5^\circ \text{ @ I/B END TO } 8.5^\circ \text{ @ O/B END}$

NOTE: EFFECTS OF ROLL DAMPING AND ROLL ACCELERATION ARE NEGLECTED

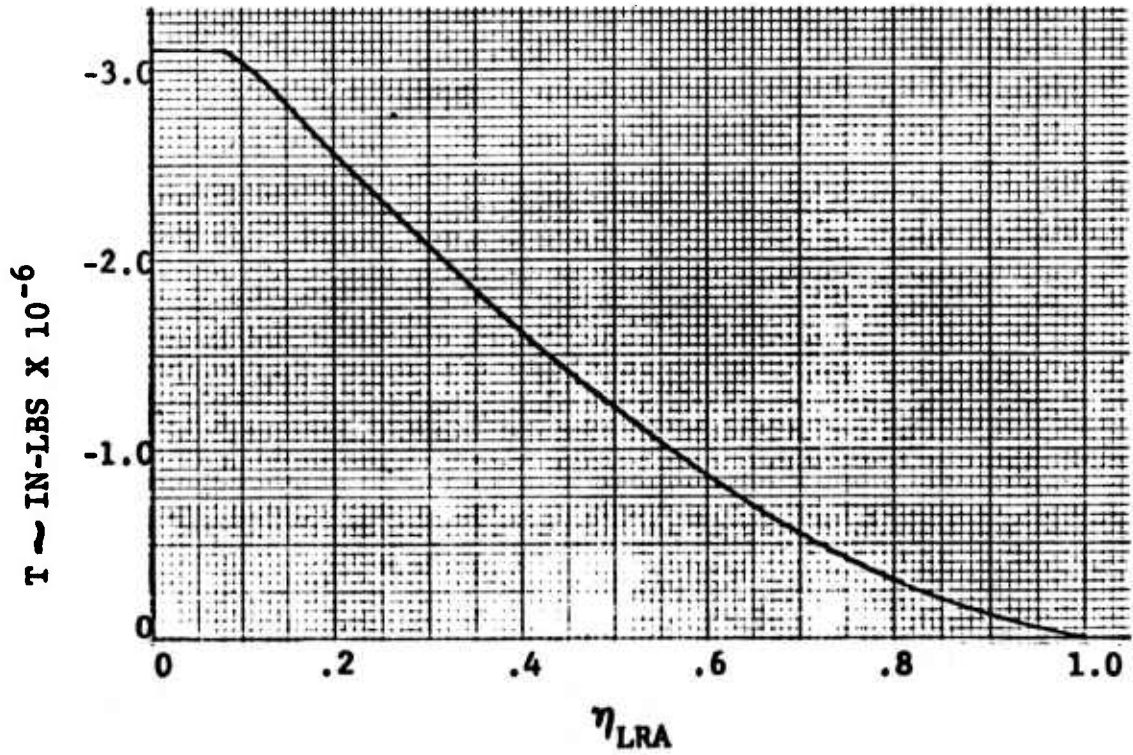


Figure A.46 ATW-4 Wing Torsion Distribution - Condition 4

$\Lambda = 16^\circ$
 $M = .80$
 ALT = 20,000 FT.
 GW = 110,400 LBS
 $n_z = 1.9g's$
 $\delta_{LE} = 10^\circ$
 $\delta_{TE} = 3.0^\circ$ @ I/B END TO 8.5° @ O/B END

NOTES:

1. X_{cp} MEAS. FWD OF .15C @ L.E. FLAP
2. X_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F = LOCAL FLAP CHORD

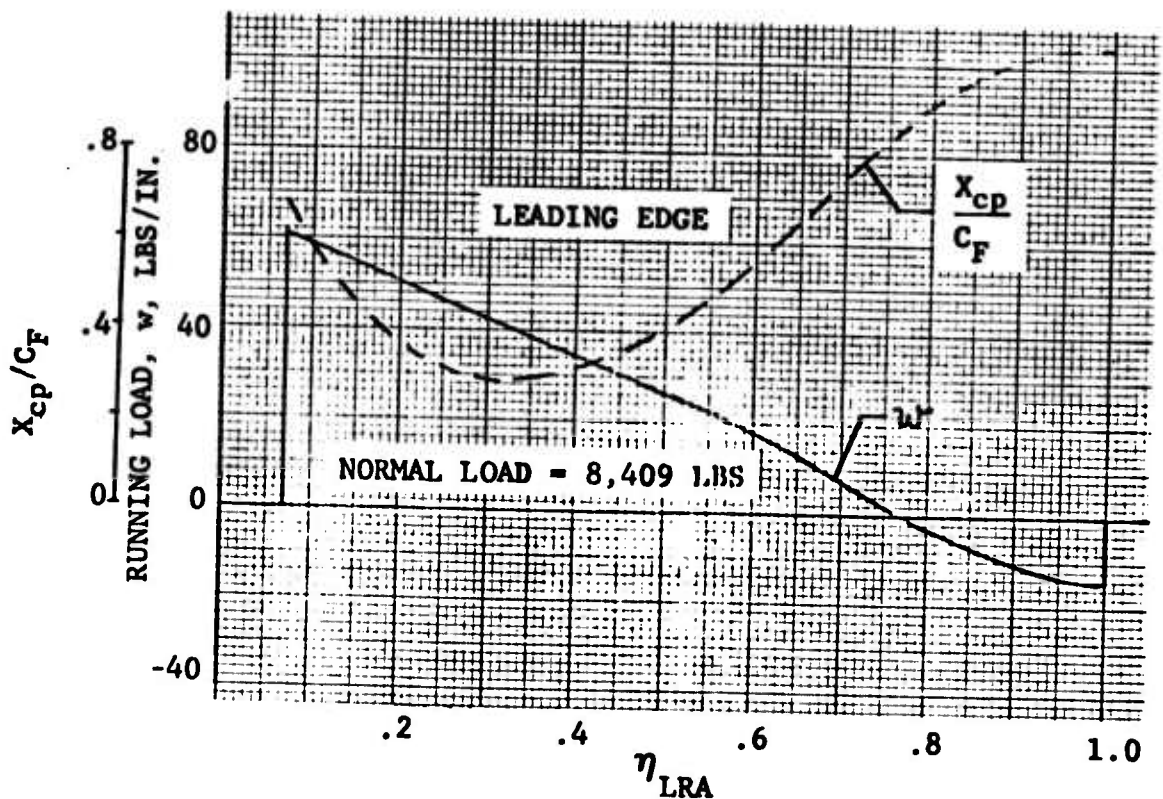
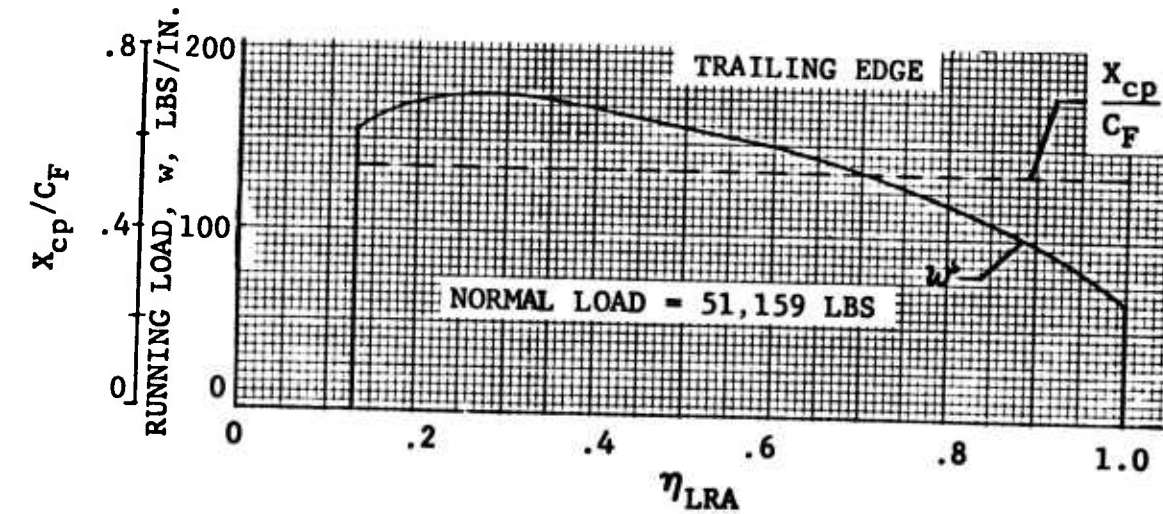


Figure A.47 ATW-4 Wing Leading and Trailing Edge Flap Running Loads - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

$$\eta_{LRA} = .123$$

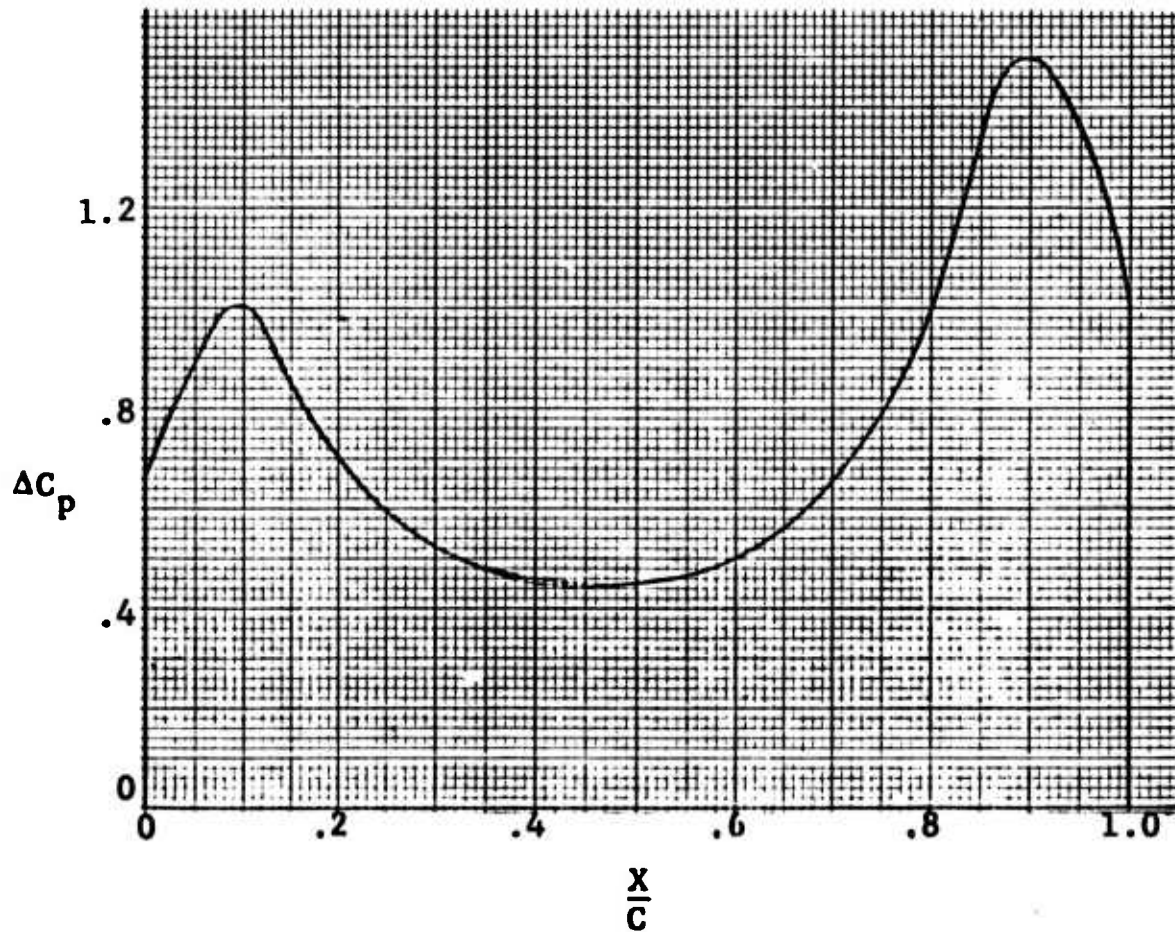


Figure A.48 ATW-4 Wing Chordwise Pressure Distribution for
.123 η_{LRA} - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

$$\eta_{LRA} = .213$$

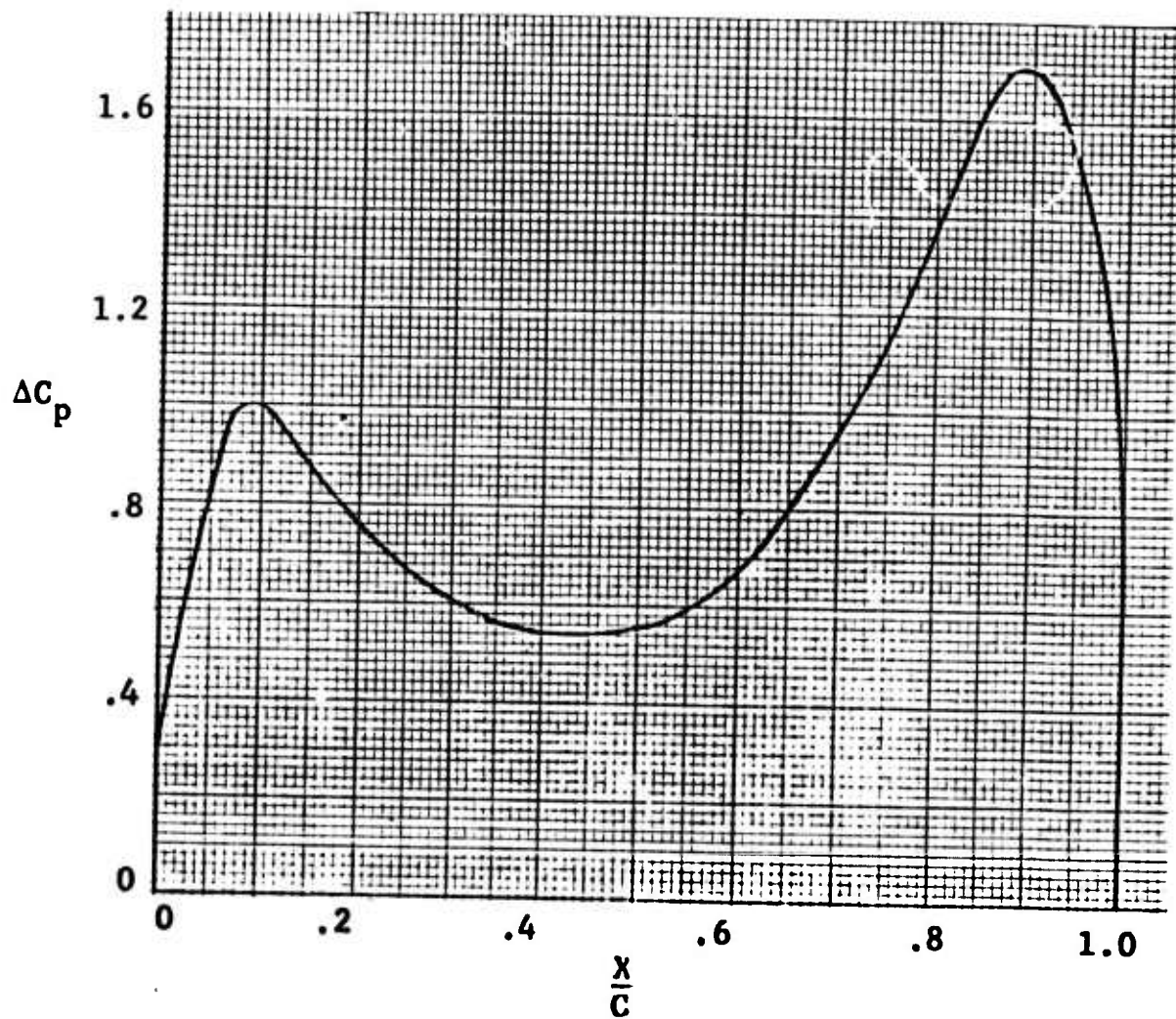


Figure A.49 ATW-4 Wing Chordwise Pressure Distribution for
.213 η_{LRA} - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

$$\eta_{LRA} = .321$$

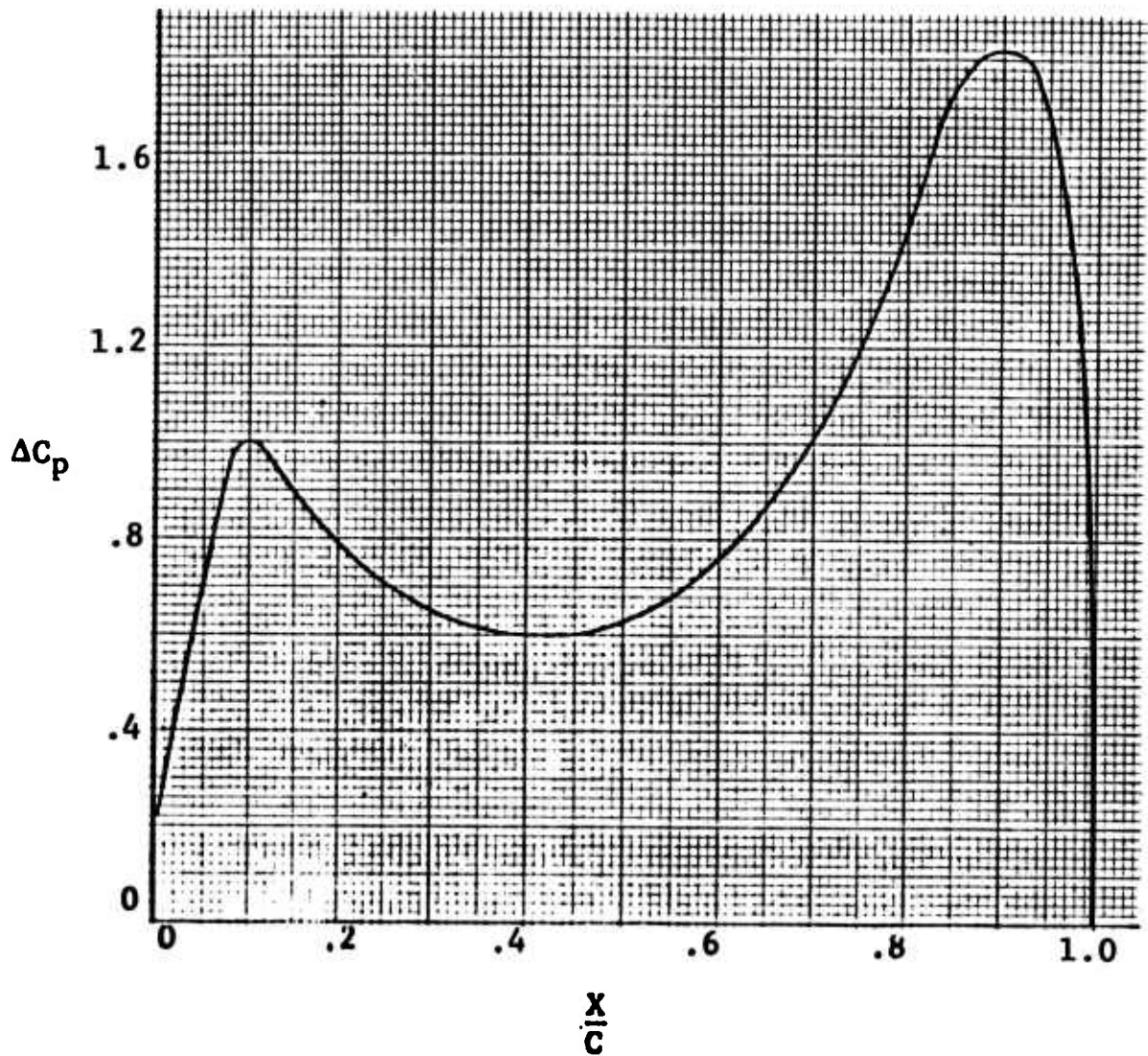


Figure A.50 ATW-4 Wing Chordwise Pressure Distribution for .321 η_{LRA} - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

$$\eta_{LRA} = .443$$

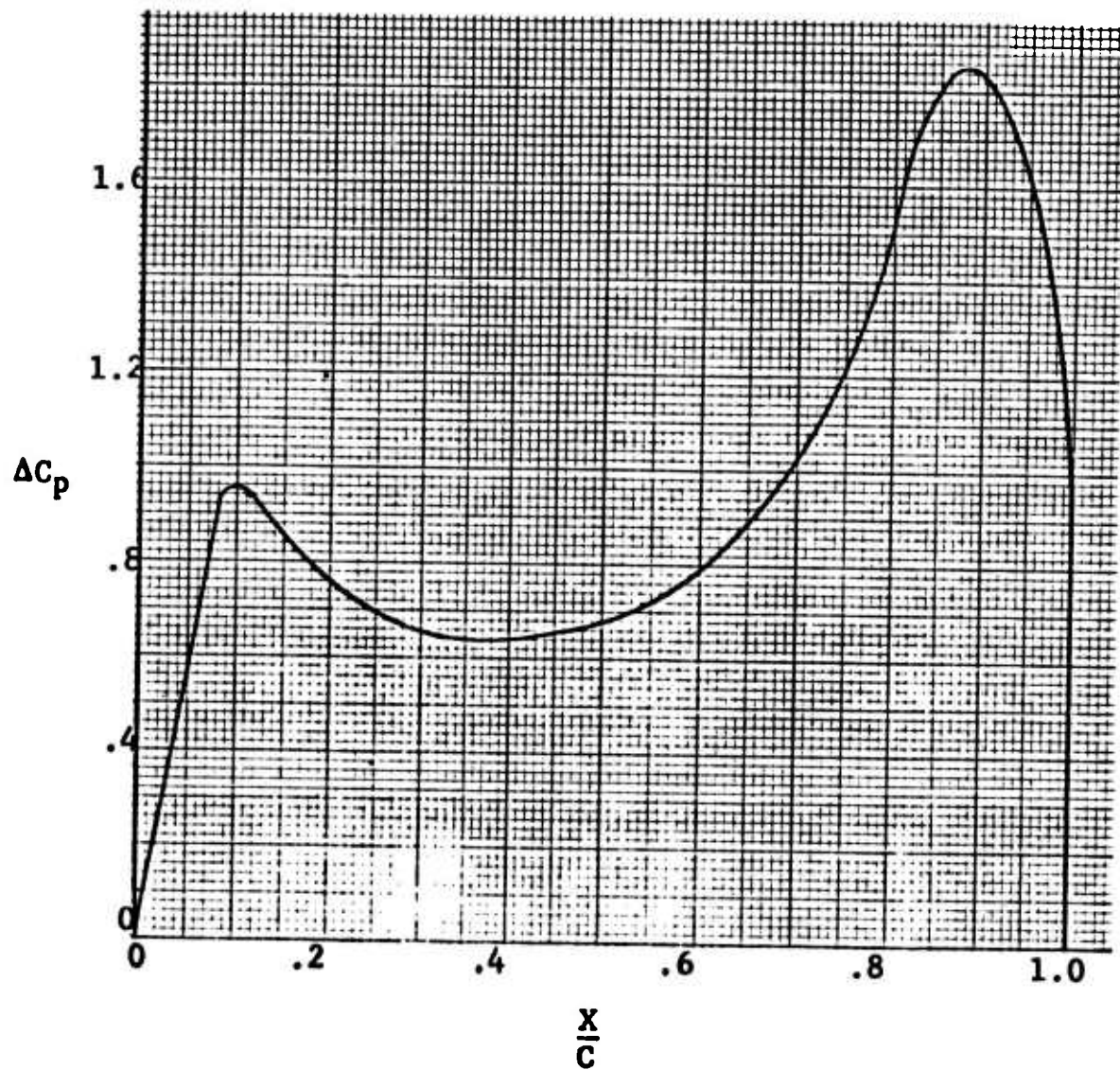


Figure A.51 ATW-4 Wing Chordwise Pressure Distribution for
.443 η_{LRA} - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$

$$\eta_{LRA} = .567$$

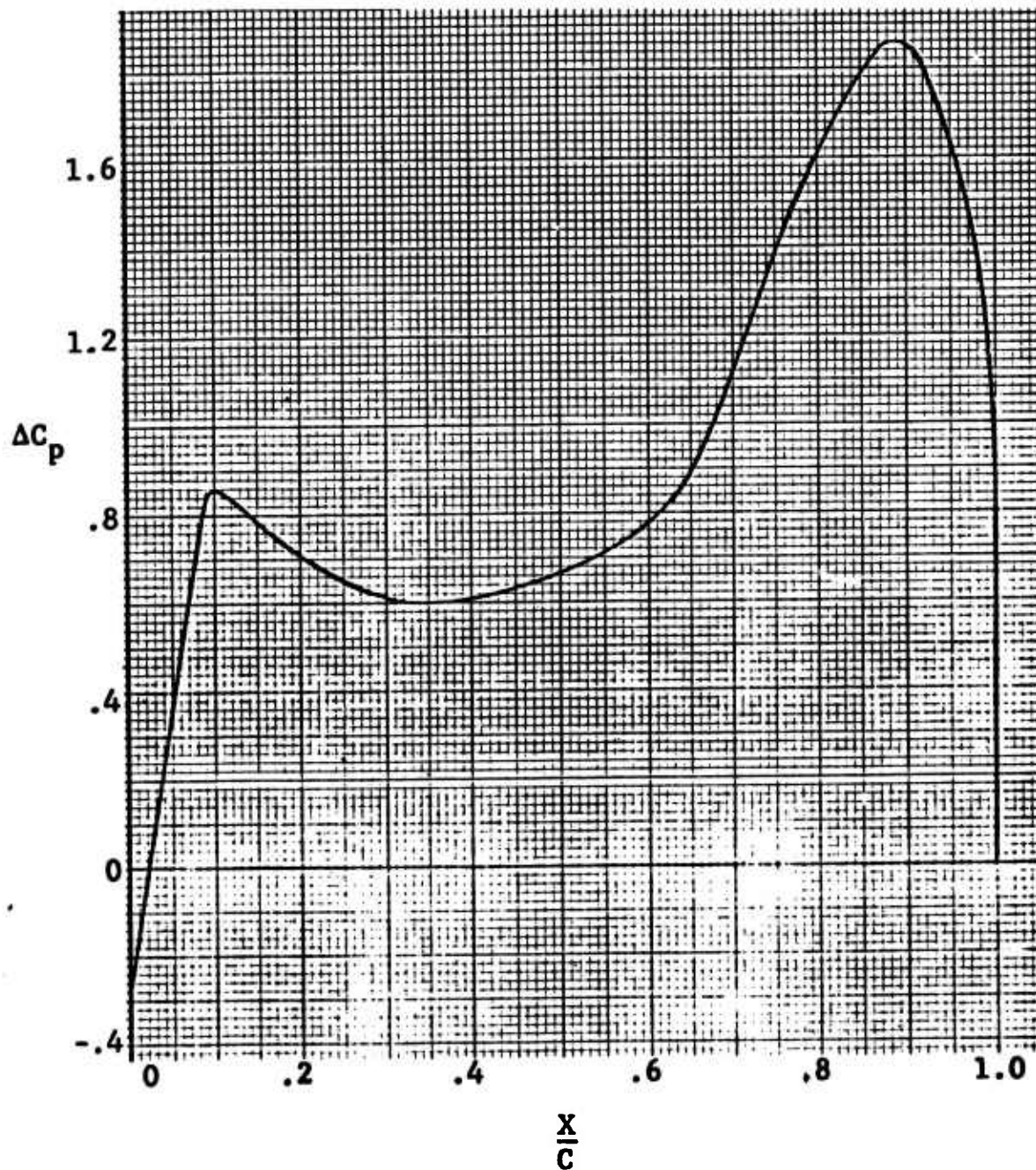


Figure A.52 ATW-4 Wing Chordwise Pressure Distribution for .567 η_{LRA} - Condition 4

$$\Delta p \text{ (psi)} = 3.024 \Delta C_p$$
$$\eta_{LRA} = .690$$

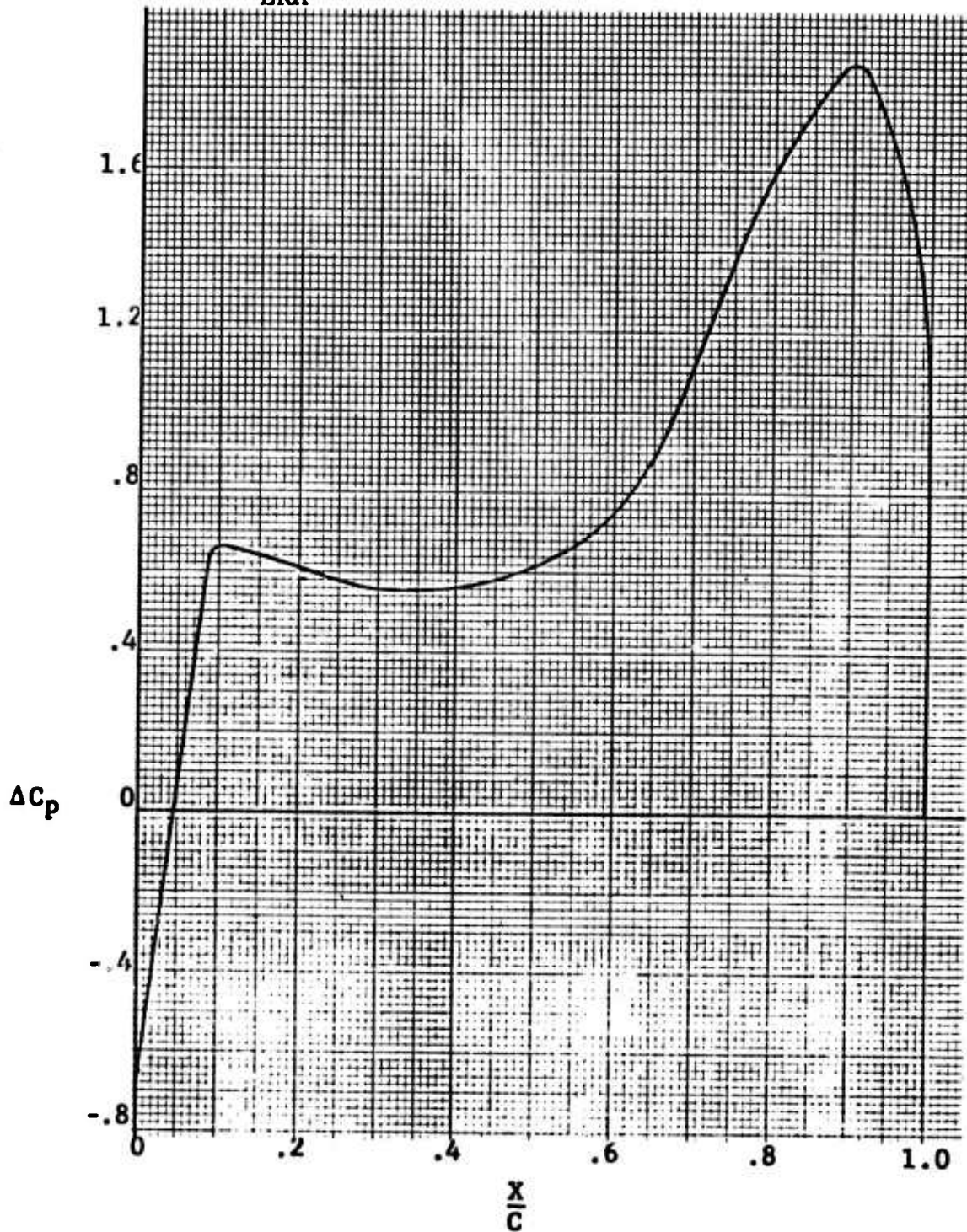


Figure A.53 ATW-4 Wing Chordwise Pressure Distribution for .690 η_{LRA} - Condition 4

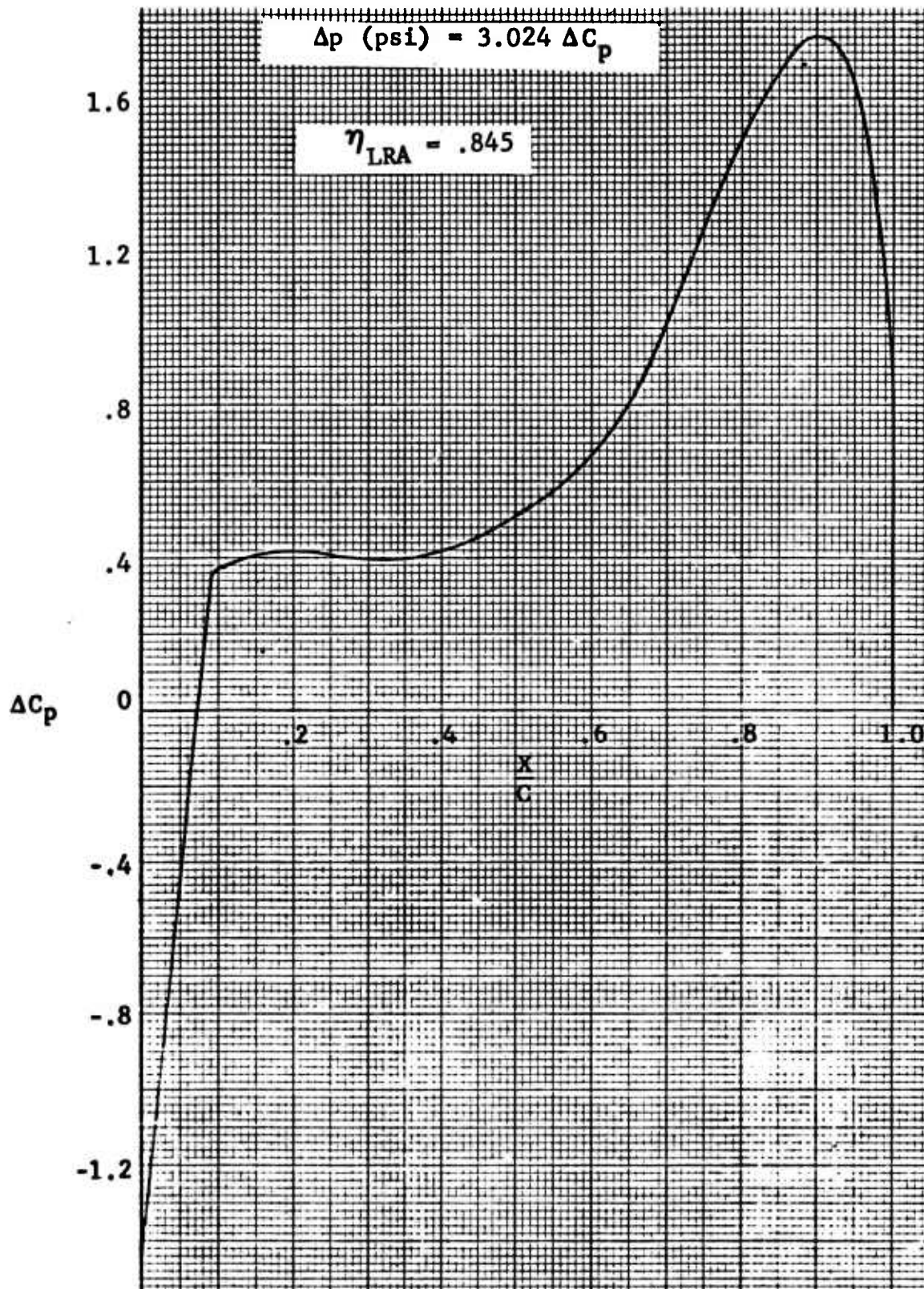


Figure A.54 ATW-4 Wing Chordwise Pressure Distribution for .845 η_{LRA} - Condition 4

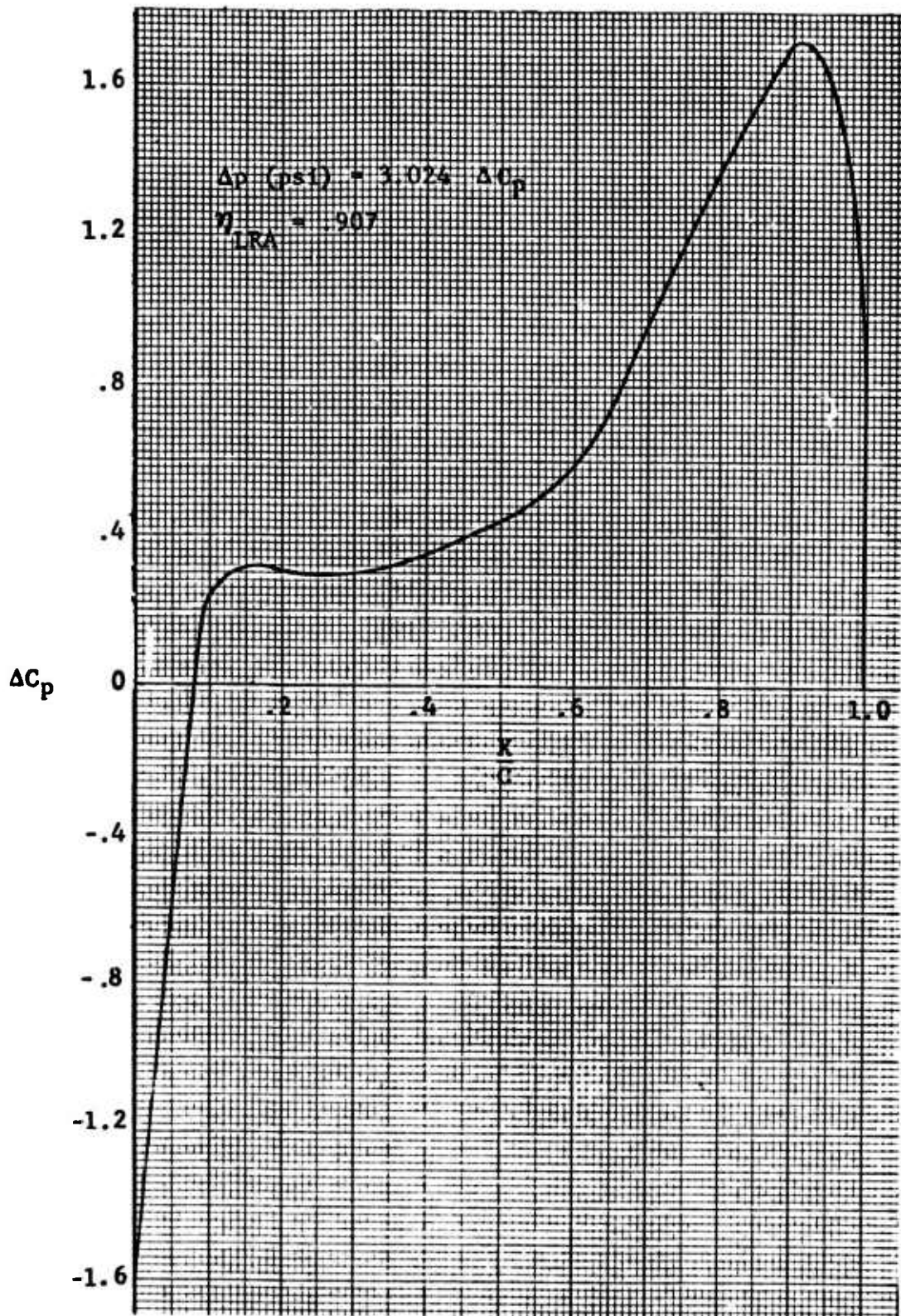


Figure A.55 ATW-4 Wing Chordwise Pressure Distribution for .907 η_{LRA} - Condition 4

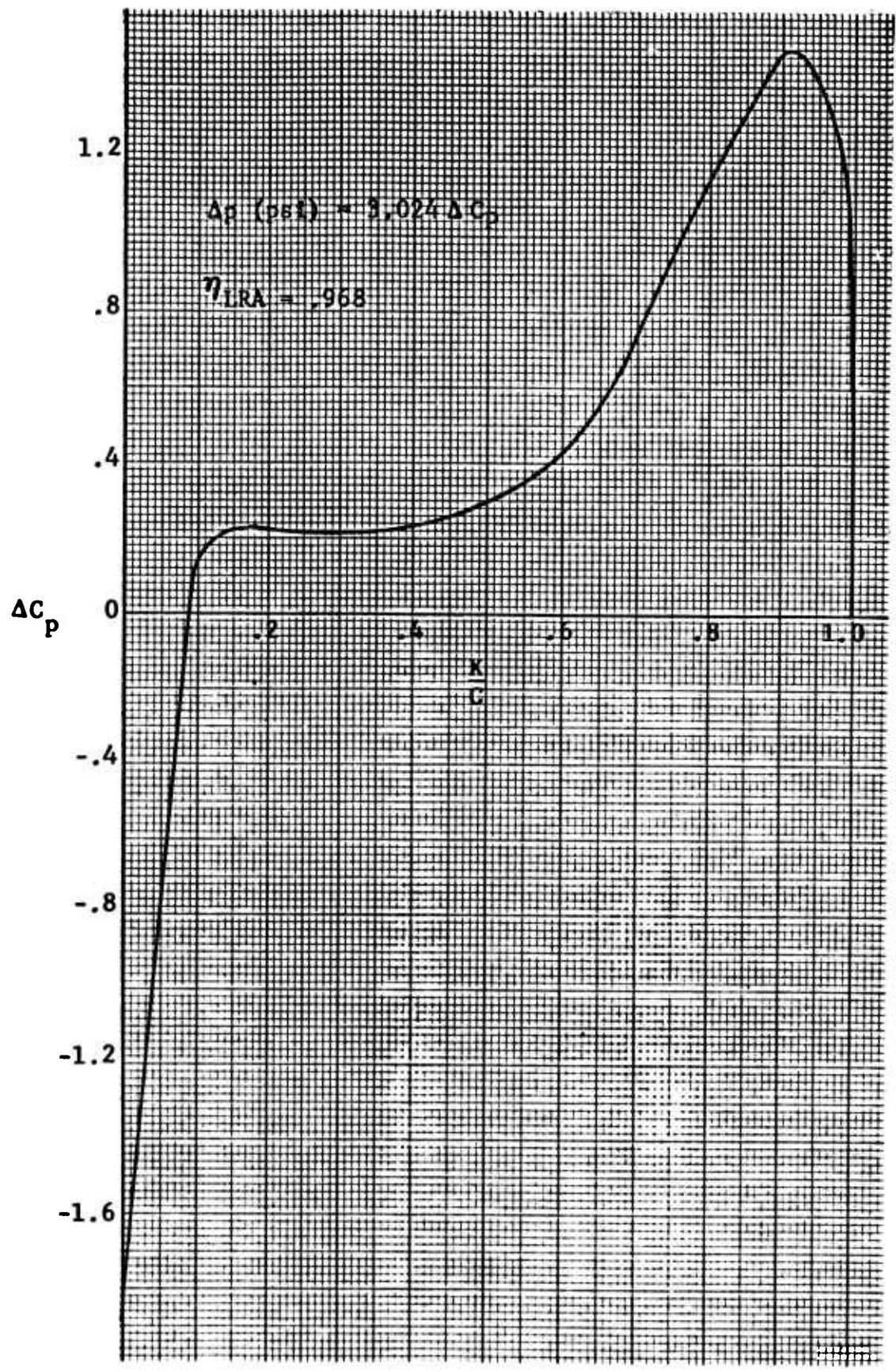


Figure A.56 ATW-4 Wing Chordwise Pressure Distribution for .968 η_{LRA} - Condition 4

NOTES:

1. LOCAL LIFT ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA

$\Lambda = 16^\circ$
 $M = .80$
 $ALT = 20,000 \text{ FT}$
 $GW = 110,400 \text{ LBS}$
 $\alpha = 4.8^\circ$
 $\delta_{LE} = 10^\circ$
 $\delta_{TE} = 3.5^\circ \text{ @ I/B END TO } 8.5^\circ \text{ @ O/B END}$

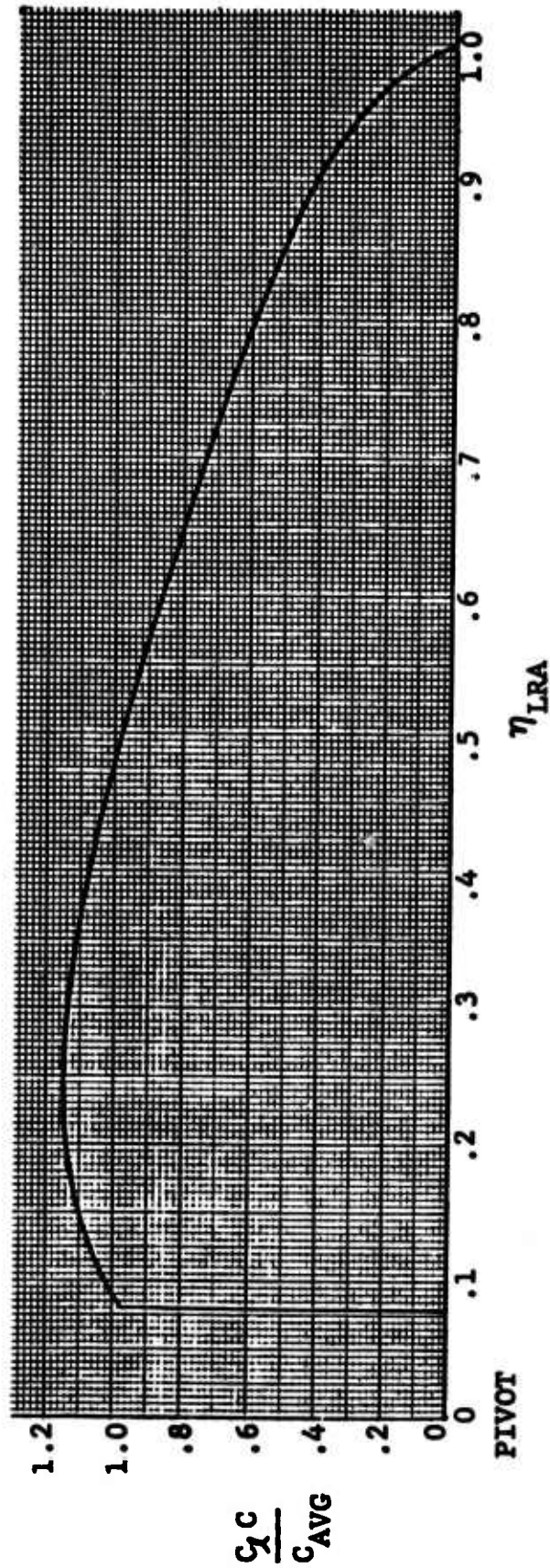


Figure A.57 ATW-4 Wing Local Lift Spanwise Distribution - Condition 4

NOTES:

1. LOCAL TORSION ON CHORDS \perp TO LRA
2. SEE FIGURE A-1 FOR CHORD GEOMETRY AND DIMENSIONAL DATA
3. MOM. REF. IS LRA
4. \bar{c} = 117.1 IN. (THEORETICAL REF. WING MAC)

Λ = 16°
 M = .80
 ALT = 20,000 FT
 GW = 110,400 LBS
 α = 4.8°
 δ_{LE} = 10°
 δ_{TE} = 3.5° @ I/B END TO
 8.5° @ O/B END

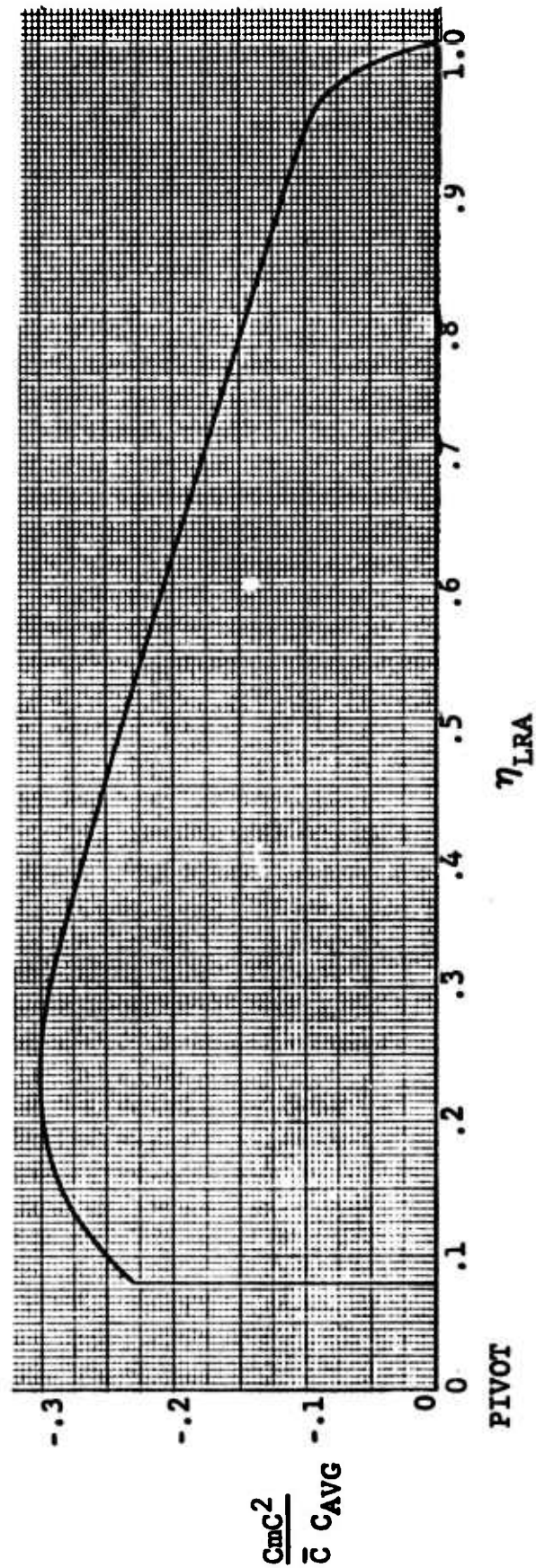


Figure A.58 ATW-4 Wing Local Torsion Spanwise Distribution - Condition 4

A.6.0 FATIGUE LOADS SPECTRA

A.6.1 Wing Box

The spectrum of wing bending moment is presented in Table A-II (2 pages). This spectrum is based on the B-1 flight-by-flight composite mission analytic wing bending moment spectrum. ATW-4 wing bending moments are used in lieu of B-1 wing values. Also, the percent of condition max/min values for load steps 9 through 15 and 33 were adjusted upward. This was done because the magnitude of the ATW-4 wing 1.0g bending moment, expressed in terms of percent design limit bending moment of baseline condition 3, is higher than the corresponding percentage ratio for the B-1 wing. Therefore, the upward adjustment of cyclic peaks provides a more reasonable distribution of the bending moment spectrum with respect to the 1.0g bending moment. The column headed "BM_p/BM(68)" is the ratio of wing net bending moment at the pivot to the bending moment at the conceptual design section located 68 inches outboard of the pivot along the load reference axis (LRA). The ATW-4 wing design data were utilized as listed below:

<u>Mission Segment</u>	<u>ATW-4 Wing Data</u>
Ground -----	Unit Inertia (Full Tanks), 2.0g's
Post Takeoff -----	Condition 1 (91.8%)
Climb, Cruise, Refuel ---	Condition 3 (With Condition 4)
Fly-Up -----	Condition 2
Terrain-Following -----	Condition 2
Prelanding -----	Condition 1 (91.8%)
Ground -----	Unit Inertia (Full Tanks), 2.0g's
Takeoff -----	Condition 1
Climb -----	Condition 3 (With Condition 4)
Prelanding -----	Condition 1 (91.8%)
Ground -----	Unit Inertia (Full Tanks), 2.0g's

Condition 4 loads are for a roll maneuver at the same flight condition as symmetric maneuver condition 3. The wing bending moment in the vicinity of the pivot is essentially the same for both conditions since the carry-through-box allowable limit is developed in each case (compare Figures A-26 and A-41). However, the bending moment on the outer portion of the wing panel is slightly higher for condition 4, therefore, the loading for this condition should be included in the spectrum for fatigue evaluations on the outer portion of the wing panel. For this purpose it is assumed that 50% of the maneuver occurrences (designated "M" on Table A-II) are symmetric and 50% are roll maneuvers.

A.6.2 Variable Camber Devices

Cyclic loads spectra for the leading and trailing edge variable camber devices are assumed to be comprised of:

1. Maneuver and gust occurrences as shown for the composite mission spectrum on Table A-III (3 pages).
2. Load cycles related to different trim settings in changing from one mission segment to the next while progressing through the composite mission on Table A-III.

The flap loads spectra data in Table A-III are based on the wing bending moment spectra data in Table A-II. In addition to maneuver and gust load cycles per mission, the percent of condition (wing bending moment) max/min values shown on Table A-II have been retained on Table A-III. These wing bending moment data were used to estimate the delta N_2 spectra max/min values on Table A-III. The leading and trailing edge flap unit loads data for developing loads spectra are referenced on Table A-III by figure numbers associated with given mission segments. Finally, the load steps on Table A-II have been resequenced on Table A-III to accommodate roll maneuvers and additional load cycles from changing trim settings between mission segments. The latter spectra are presented in supplementary Table A-IIIA, and integrated into the total spectrum according to load step numbers. Three cycles per mission are assumed for each load step on Table A-IV.

The unit loads data referenced on Table A-III are presented in Figures A.59 through A.62 for four baseline conditions. These conditions correspond with preliminary design condition 1 through 4 presented on Table A-I with the exception of values for load factor and angle of attack. For each of these conditions, leading and trailing edge flap spanwise running loads and center-of-pressure distribution data are presented for the following cases:

1. $N_z = 1.0g$
2. $\Delta N_z = 1.0g$

Leading and trailing edge flap loads for maneuver and gust loads spectra are developed using the following equations for running load and center of pressure distributions:

$$w_p = w_{1.0g} + \frac{\Delta w}{g} (N_z - 1.0)$$

$$w_p \left(\frac{X_{CP}}{C_F} \right)_p = w_{1.0g} \left(\frac{X_{CP}}{C_F} \right)_{1.0g} + \frac{\Delta w}{g} (N_z - 1) \left(\frac{X_{CP}}{C_F} \right)_{\Delta N_z = 1}$$

where

- 1) Values for w and X_{CP}/C_F appearing on the right hand side of the equations are read from Figures A.59 through A.66 at given ETA stations.
- 2) $(N_z - 1.0) = \Delta N_z$ values given for load steps on Table A-III.
- 3) Subscript p is attached to cyclic peak load distribution.

For the "Climb/Cruise/Refuel" mission segment it is assumed that 50% of the maneuver occurrences are symmetric maneuvers (baseline condition 3) and 50% are roll maneuvers (baseline condition 4).

The leading and trailing edge flap load distributions related to trim change cycles correspond with the 1.0g distributions referenced by figure number on Table A-IV.

TABLE A-II

ATW-4 WING BENDING MOMENT SPECTRUM AT WING PIVOT

LOAD STEP	MISSION SEGMENT	BMD BM(68")	BM X 10 ⁶ in-lb LIMITS	TAW(°F)	WING ANGLE	% OF CONDITION		BM X 10 ⁻⁶ MAX	BM X 10 ⁻⁶ MIN	CYCLES/ MISSION	
						MAX	MIN				
1	GROUND	1.6	-2.9624	58.69	16°	11.5	60.8	-1.8	1		
2						85.1	51.5	16.62	10.06	.01	
3						M	76.6	51.5	14.96	10.06	.1
4						59.3	51.5	11.58	10.06	2	
5						51.5	41.4	8.08	10.06	2	
6						60.5	56.8	11.09	11.09	2	
7						G	60.9	36.2	11.89	7.07	1
8						50.5	44.5	8.69	8.69	29	
9	CLIMB, CRUISE & REFUEL	1.5	18.3	38.97	16°	56.0	37.0	10.25	6.77	1	
10						64.5	56.0	11.8	10.25	22	
11						M	56.0	50.5	10.25	9.24	22
12						69.0	46.5	12.63	8.51	1	
13						46.5	31.0	8.51	5.67	1	
14						54.0	46.5	9.88	8.51	58	
15						46.5	42.5	8.51	7.78	58	
16	FLY-UP	1.5	16.106	126.15	65°	68.5	25.5	11.03	4.11	1	
17						46.7	26.5	7.52	4.27	1	
18	TERRAIN FOLLOWING	1.5	16.106	126.15	65°	64.7	48.2	10.42	7.76	.1	
19						55.1	28.0	8.87	4.51	1	
20						45.1	34.5	7.26	5.56	7	
21						G	48.6	8.1	7.83	1.3	1
22						32.1	18.1	5.17	2.92	132	
23						28.6	-4.4	4.61	-71	1	
24						19.6	8.4	3.16	1.35	132	
25						65.4	3.7	10.53	.596	1	
26						M	49.5	6.8	7.97	1.1	9
27						32.1	14	5.17	2.255	95	
28	PRELANDING	1.5	17.828	70.13	16°	92.1	51.7	16.42	9.22	.01	
29						M	82.9	-13.8	14.78	-2.46	.1
30						71.3	51.7	12.71	9.22	1	
31	GROUND TAKE-OFF	1.6	-2.9624	58.69	16°	11.5	60.8	-3.41	-1.8	1	
32						73.2	49.4	14.29	9.65	1	

TABLE A-II (CONTINUED)

LOAD STEP	MISSION SEGMENT	BMP B ₁ (68")	BM X 10 ⁶ in -lb LIMITS	TAW(°F)	WING ANGLE	% OF CONDITION		BM X 10 ⁻⁶ MAX	BM X 10 ⁻⁶ MIN	CYCLES/ MISSION	
						MAX	MIN				
33	CLIMB	1.5	18.3	38.97	16°	80.5	56.0	14.73	10.25	1	
34	PRELANDING	1.5	17.828	70.13	16°	51.7	34.5	9.22	6.15	1	
35						M	59.9	51.7	10.68	9.22	19
36							51.7	46.1	9.22	8.22	19
37							76.8	47.9	13.69	8.54	1
38							65.7	56.2	11.71	10.02	4
39						G	67.0	33.6	11.94	5.99	1
40		60.6	39.9	10.8	7.11	9					
41		57.4	43.1	10.23	7.68	48					
42		53.0	46.1	9.45	8.22	294					
43	GROUND	1.6	-2.9629	58.69	16°	11.5	60.8	-.341	-1.8	8	
44							15.4	56.2	-.456	-1.665	154

NOTES:

- (1) THIS COMPOSITE MISSION TABLE CONTAINS 1143.32 CYCLES PER MISSION AND 1,463,449.6 CYCLES PER LIFE.
- (2) LEGEND: M - MANEUVER LOAD
G - GUST LOAD
- (3) BENDING MOMENT IS IN THE LOAD REFERENCE AXIS SYSTEM.

TABLE A-III

ATW-4 WING VARIABLE CAMBER SYSTEM MANEUVER/GUST LOAD NZ SPECTRUM

LOAD STEP	MISSION SEGMENT	REF. FIG. NO.	WING ANGLE	% OF CONDITION		ΔNZ MAX	ΔNZ MIN	CYCLES/ MISSION
				MAX	MIN			
1	GROUND	-	16°	11.5	60.8	-	-	-
3	POST TAKE-OFF	A.59 & A.60	16°	M	51.5	.74	.16	.01
4		A.59 & A.60			51.5	.60	.16	.1
5		A.59 & A.60			51.5	.29	.16	2
6		A.59 & A.60			41.4	.16	-.02	2
7		A.59 & A.60			56.8	.31	.25	2
8		A.59 & A.60			36.2	.32	-.11	1
9		A.59 & A.60			44.5	.14	.03	29
11	CLIMB, CRUISE & REFUEL	A.63 & A.64	16°	M	37.0	.144	-.48	1
12		A.63 & A.64			56.0	.43	.144	11
13		A.65 & A.66			56.0	-.23	-.50	11
14		A.63 & A.64			50.5	.144	-.038	11
15		A.65 & A.66			50.5	-.50	-.68	11
16		A.63 & A.64			46.5	.586	-.168	1
17		A.63 & A.64			31.0	-.168	-.688	1
18		A.63 & A.64			46.5	.092	-.168	29
19		A.65 & A.66			46.5	-.56	-.81	29
20		A.63 & A.64			42.5	.168	-.298	29
21	A.65 & A.66	42.5	-.81	-.94	29			
23	FLY-UP	A.61 & A.62	65°	68.5	25.5	1.07	-.205	1
24		A.61 & A.62		46.7	26.5	.425	-.175	1

TABLE A-III (CONTINUED)

LOAD STEP	MISSION SEGMENT	REF. FIG. NO.	WING ANGLE	% OF CONDITION		ΔNZ MAX	ΔNZ MIN	CYCLES/ MISSION		
				MAX	MIN					
25	TERRAIN FOLLOWING	A.61 & A.62	65°	64.7	48.2	.98	.47	.1		
26		A.61 & A.62		55.1	28.0	.68	-.13	1		
27		A.61 & A.62		45.1	34.5	.38	.065	7		
28		A.61 & A.62		48.6	8.1	.47	-.715	1		
29		A.61 & A.62		32.1	18.1	0	-.415	132		
30		A.61 & A.62		28.6	-4.4	-.115	-1.09	1		
31		A.61 & A.62		19.6	8.4	-.40	-.715	132		
32		A.61 & A.62		65.4	3.7	.98	-.85	1		
33		A.61 & A.62		49.5	6.8	.5	-.76	9		
34		A.61 & A.62		32.1	14	0	-.55	95		
36		PRE-LANDING		A.59 & A.60	16°	92.1	51.7	.865	.16	.01
37				A.59 & A.60		82.9	-13.8	.71	-.99	.1
38				A.59 & A.60		71.3	51.7	.5	.16	1
40		GROUND			16°	11.5	60.8	-	-	-
42	TAKE-OFF	A.59 & A.60	16°	73.2	49.4	.54	.12	1		
44	CLIMB	A.63 & A.64	16°	80.5	56.0	.807	-.233	1		
46	PRE-LANDING	A.59 & A.60	16°	51.7	34.5	.16	-.14	1		
47		A.59 & A.60		59.9	51.7	.31	.16	19		
48		A.59 & A.60		51.7	46.1	.16	.05	19		
49		A.59 & A.60		76.8	47.9	.60	.09	1		
50		A.59 & A.60		65.7	56.2	.41	.245	4		
51		A.59 & A.60		67.0	33.6	.43	-.16	1		
52		A.59 & A.60		60.6	39.9	.32	-.05	9		
53		A.59 & A.60		57.4	43.1	.26	0	48		
54		A.59 & A.60		53.0	46.1	.18	.06	294		

TABLE A-III (CONTINUED)

LOAD STEP	MISSION SEGMENT	REF. FIG. NO.	WING ANGLE	% OF CONDITION		ΔNz MAX	ΔNz MIN	CYCLES / MISSION
				MAX	MIN			
56	GROUND	-	16°	11.5	60.8	-	-	-
57				15.4	56.2			

LEGEND: M - MANEUVER LOAD
G - GUST LOAD

NOTES:

(1) REFERENCE FIGURES NOTED FOR L.E. AND T.E. FLAPS UNIT LOADS DATA

(2) $N_z = 1.0 + \Delta N_z$

TABLE A-IV

ATW-4 WING VARIABLE CAMBER SYSTEM LOADS SPECTRA FROM TRIM CHANGES

LOAD STEP	WING ANGLE MAX/MIN	REF. FIG. NO. MAX.	REF. FIG. NO. MIN	LOAD (LBS) MAX	LOAD (LBS) MIN	CYCLES/ MISSION
(LEADING EDGE FLAP)						
2	16°/16°	A.59	—	5,685	0	3
10	16°/16°	A.59	A.63	5,685	-3071	3
22	65°/16°	A.61	A.63	11,865	-3071	3
35	65°/16°	A.61	A.59	11,865	5,685	3
39	16°/16°	A.59	—	5,685	0	3
41	16°/16°	A.59	—	5,685	0	3
43	16°/16°	A.59	A.63	5,685	-3071	3
45	16°/16°	A.59	A.63	5,685	-3071	3
55	16°/16°	A.59	—	5,685	0	3
(TRAILING EDGE FLAP)						
2	16°/16°	A.59	—	20,734	0	3
10	16°/16°	A.59	A.63	39,926	20,734	3
22	65°/16°	A.61	A.63	39,226	3,737	3
35	65°/16°	A.61	A.59	20,734	3,737	3
39	16°/16°	A.59	—	20,734	0	3
41	16°/16°	A.59	—	20,734	0	3
43	16°/16°	A.59	A.63	39,226	20,734	3
45	16°/16°	A.59	A.63	39,226	20,734	3
55	16°/16°	A.59	—	20,734	0	3

NOTES: (1) REFERENCE FIGURES NOTED FOR L.E. AND T.E. FLAPS LOADS DISTRIBUTIONS
 (2) INTEGRATE SPECTRA FROM THIS TABLE WITH TABLE III MANEUVER AND GUST LOADS SPECTRA ACCORDING TO LOAD STEP SEQUENCE NUMBER

NOTES:

1. $+x_{cp}$ MEAS FWD OF .15C @ L.E.
FLAP
2. $+x_{cp}$ MEAS AFT OF .65C @ T.T.
FLAP
3. C_F IS FLAP LOCAL CHORD

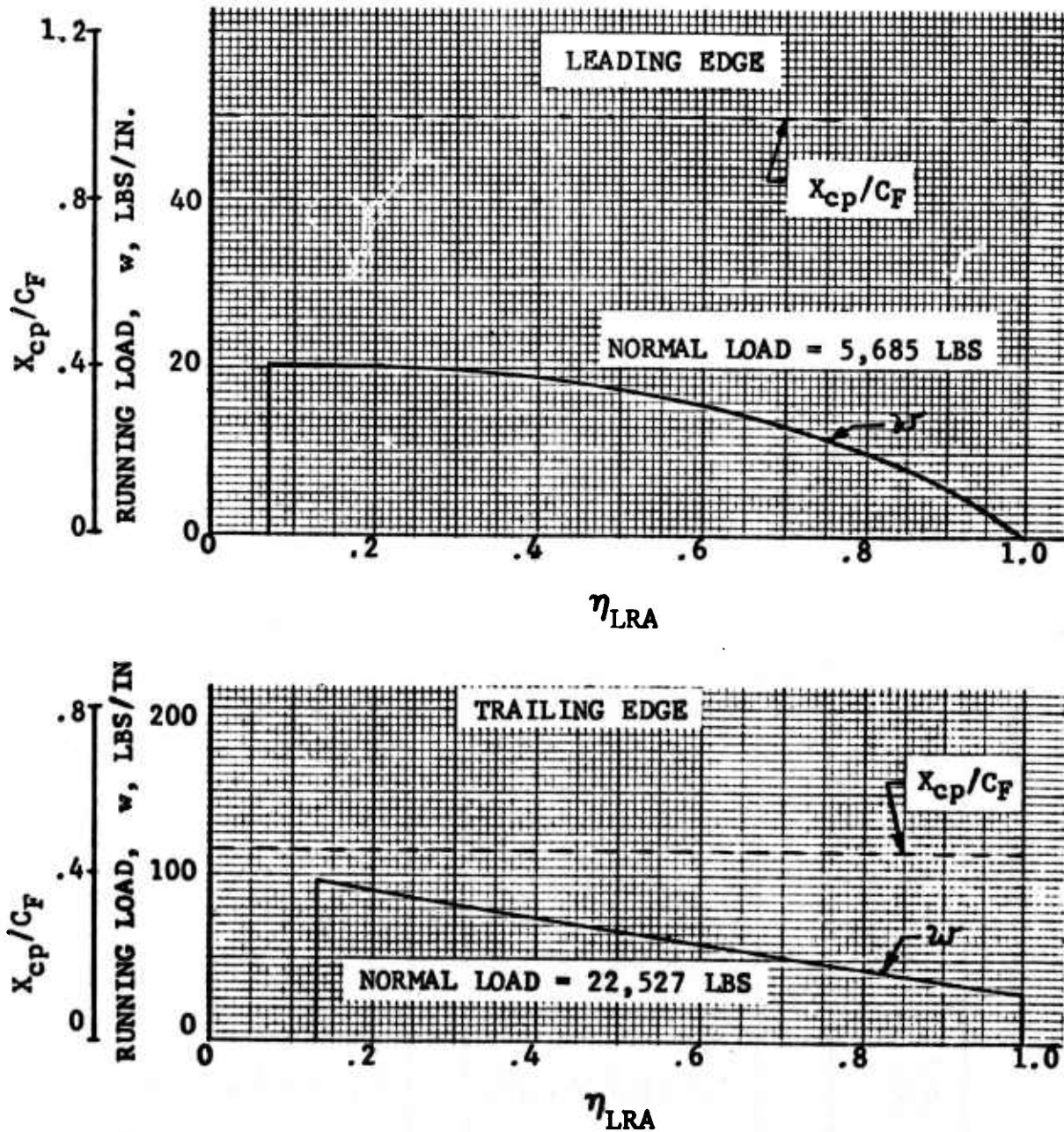


Figure A.59 ATW-4 Wing Leading and Trailing Edge Flap Loads for $N_z = 1.0g$ - Condition 1

NOTES:

1. X_{cp} MEAS. FWD. OF .15C @ L.E. FLAP
2. X_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F IS FLAP LOCAL CHORD

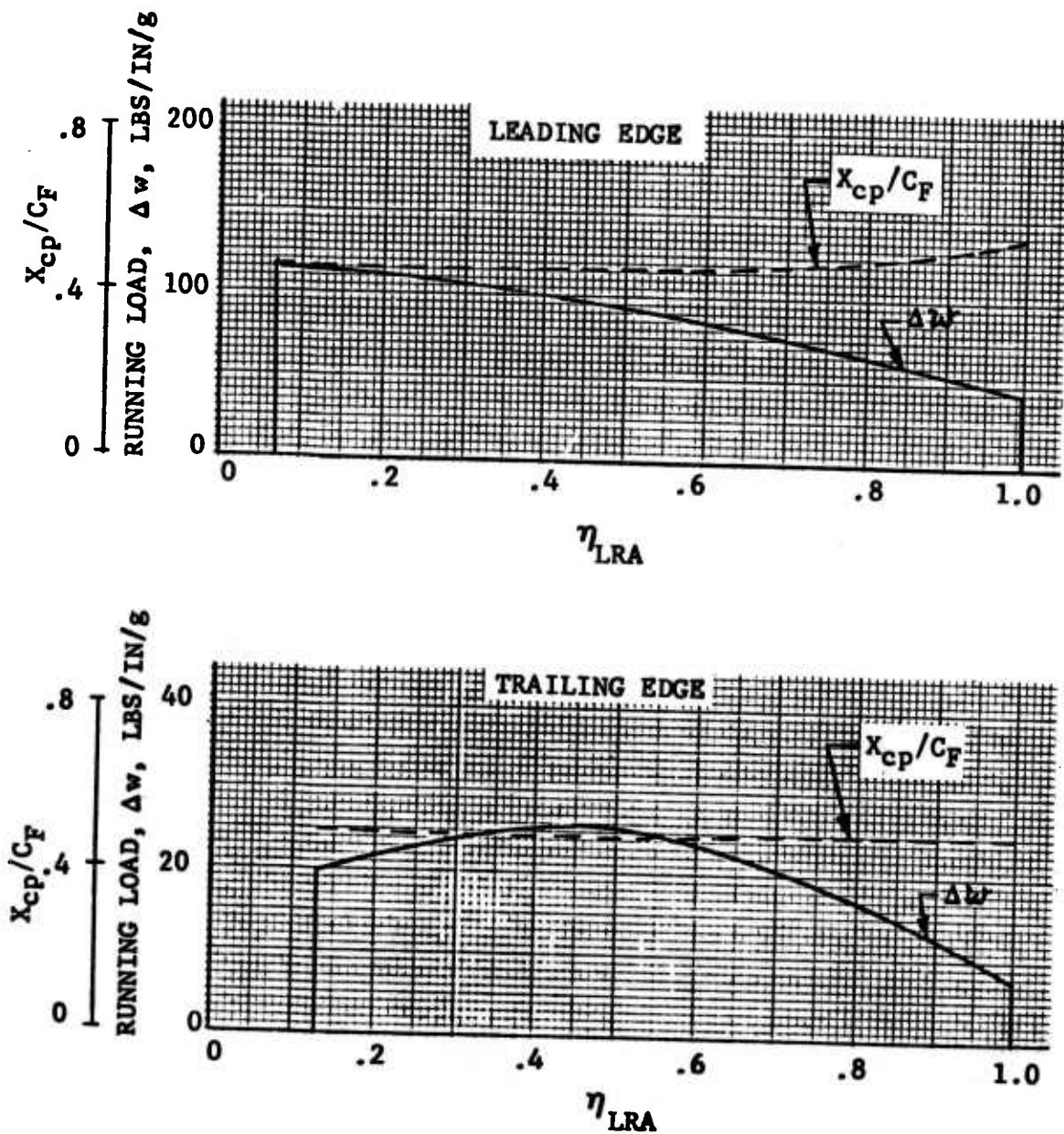


Figure A.60 ATW-4 Wing Leading and Trailing Edge Flap Loads for $\Delta N_z = 1.0g$ - Condition 1

NOTES:

1. X_{cp} MEAS. FWD. OF .15C @ L.E. FLAP
2. X_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F = FLAP LOCAL CHORD

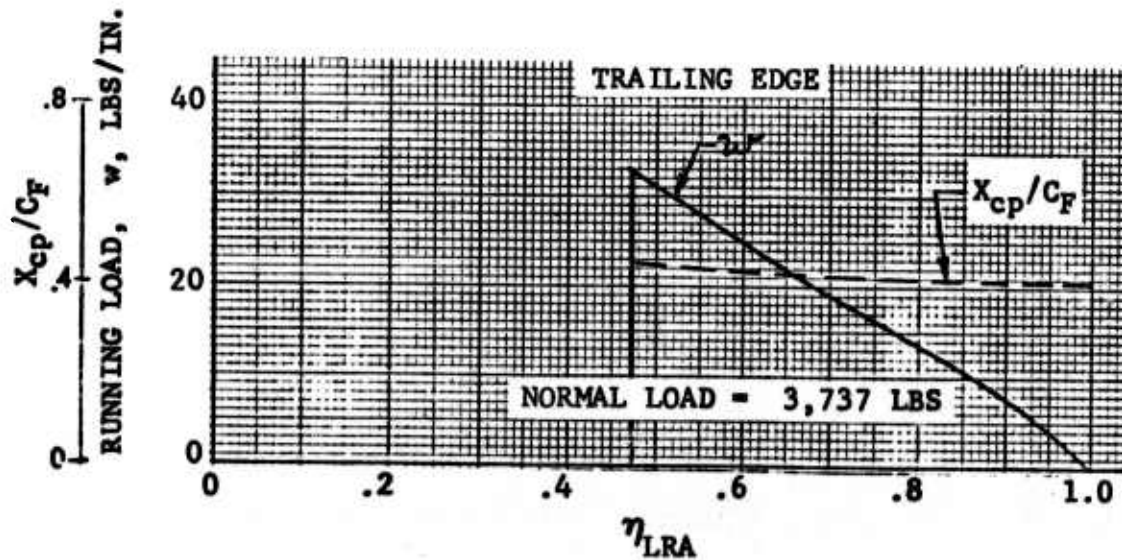
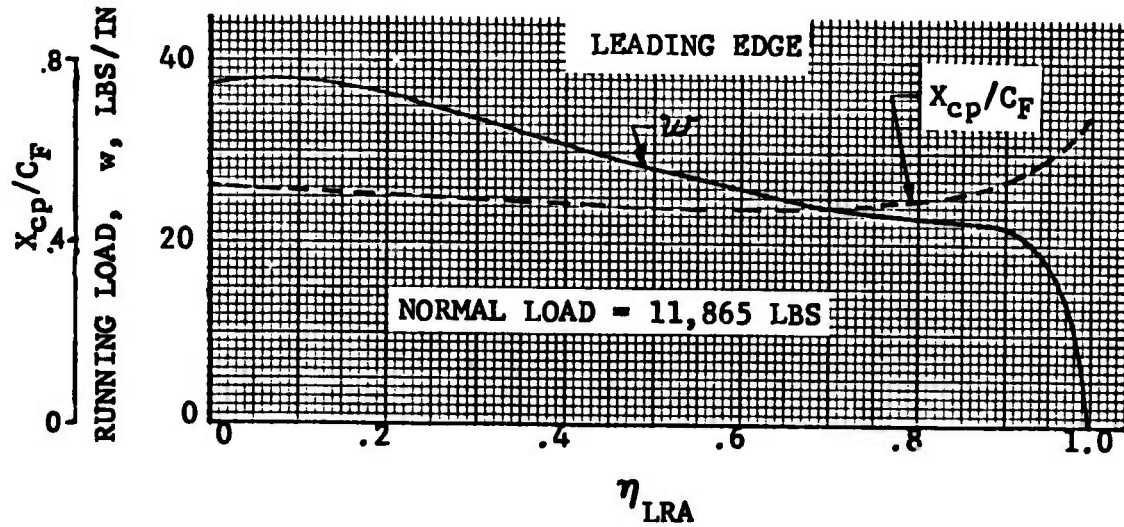


Figure A.61 ATW-4 Wing Leading Edge and Trailing Edge Flap Loads for $N_z = 1.0g$ - Condition 2

NOTES:

1. X_{cp} MEAS. FWD. OF .15C @ L.E. FLAP
2. X_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F - FLAP LOCAL CHORD

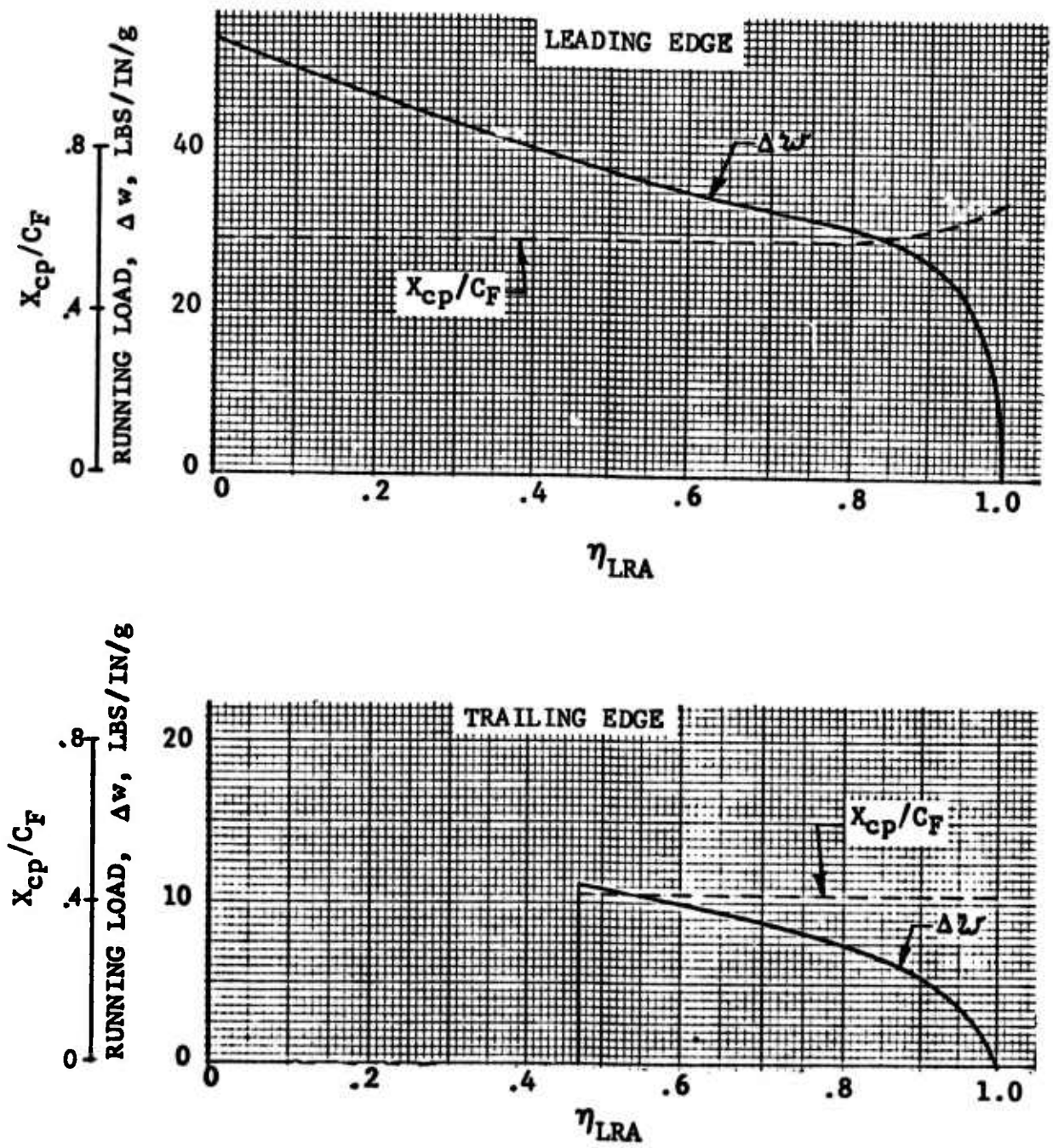


Figure A.62 ATW-4 Wing Leading and Trailing Edge Flap Loads for $\Delta N_z = 1.0g$ - Condition 2

NOTES:

1. $+X_{cp}$ MEAS. FWD. OF .15C @ L.E.
FLAP
2. $+X_{cp}$ MEAS. AFT OF .65C @ T.E. FLAP
3. C_F IS FLAP LOCAL CHORD

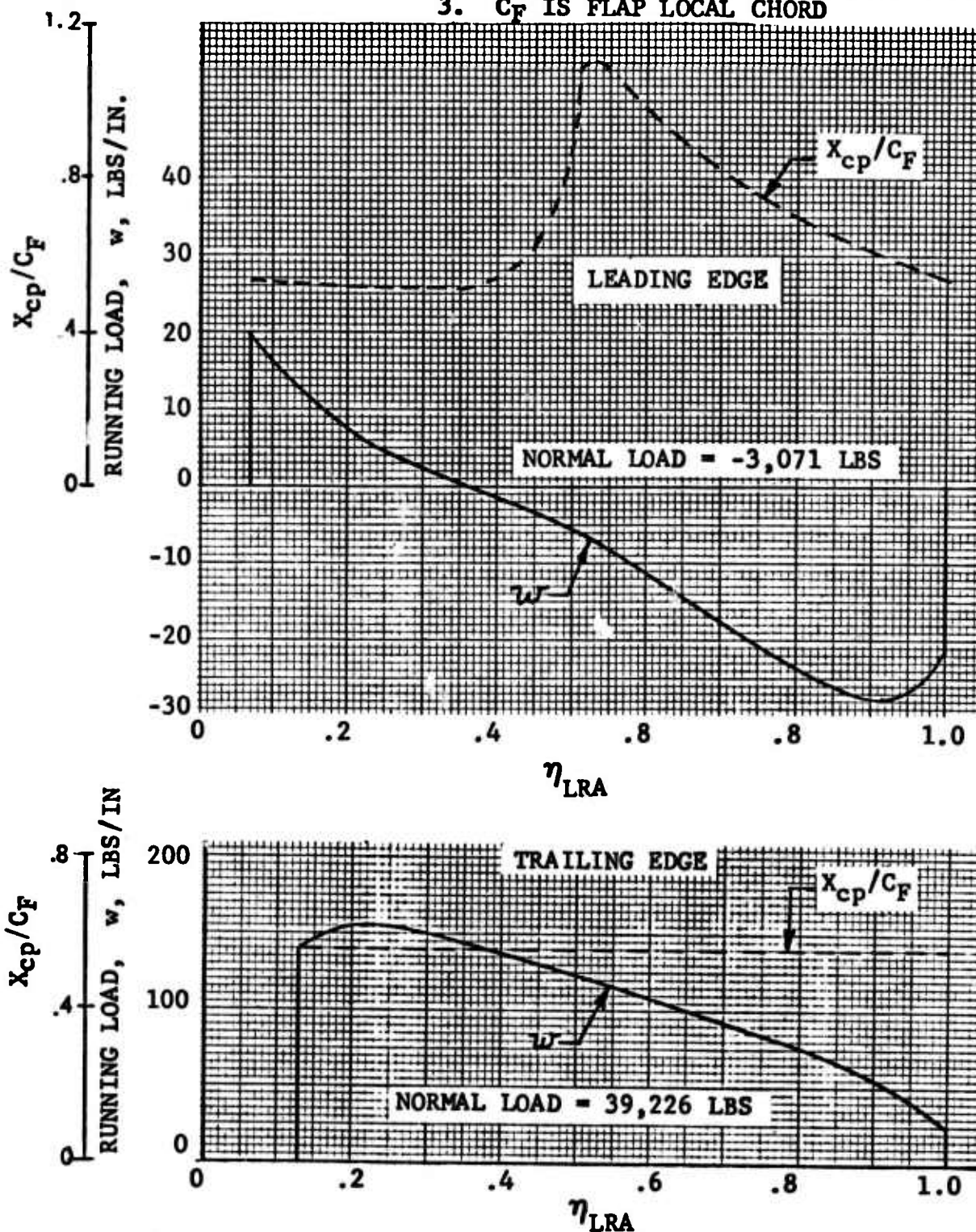


Figure A.63 ATW-4 Wing Leading and Trailing Edge Flap Loads for $N_z = 1.0g$ - Condition 3

NOTES:

1. X_{cp} MEAS. FWD. OF .15C @ L.E. FLAP
2. X_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F IS FLAP LOCAL CHORD

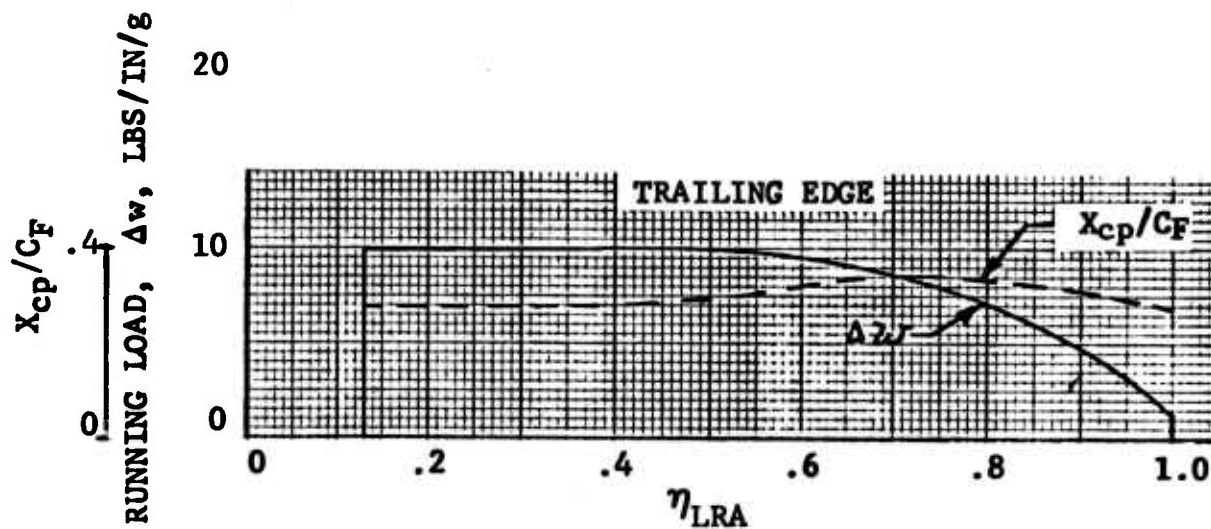
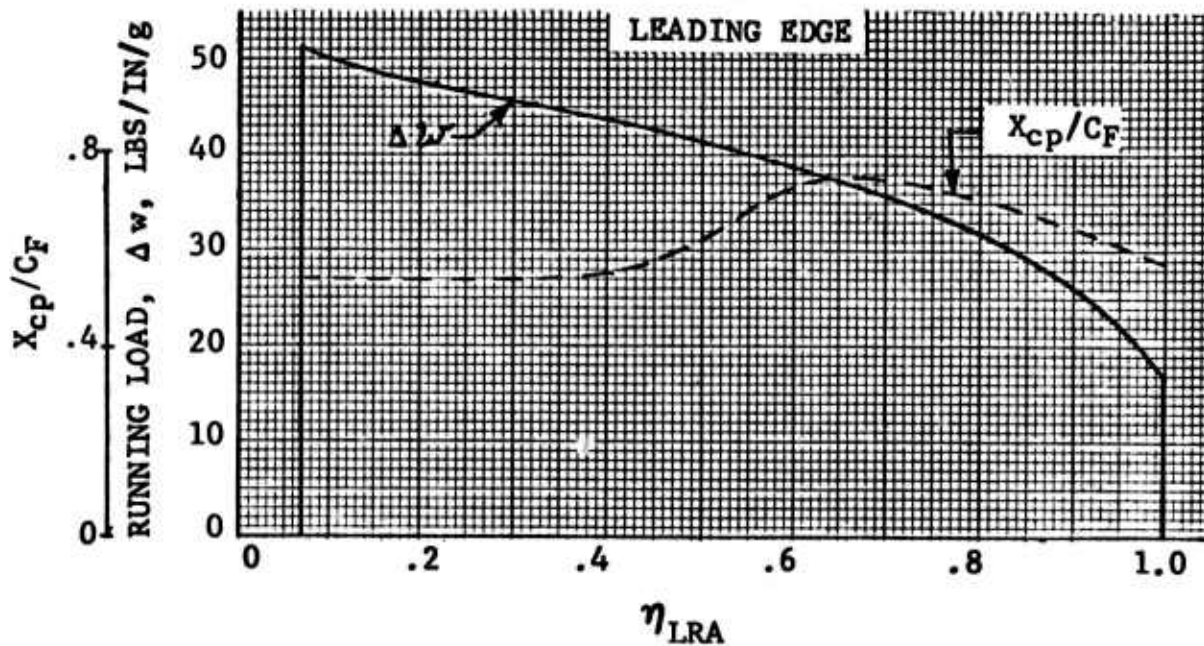


Figure A.64 ATW-4 Wing Leading and Trailing Edge Flap Loads for $\Delta N_2 = 1.0g$ - Condition 3

NOTES:

1. $+x_{cp}$ MEAS. FWD. OF .15C @ L.E. FLAP AND AFT OF .65C @ T.E. FLAP
2. C_F IS FLAP LOCAL CHORD

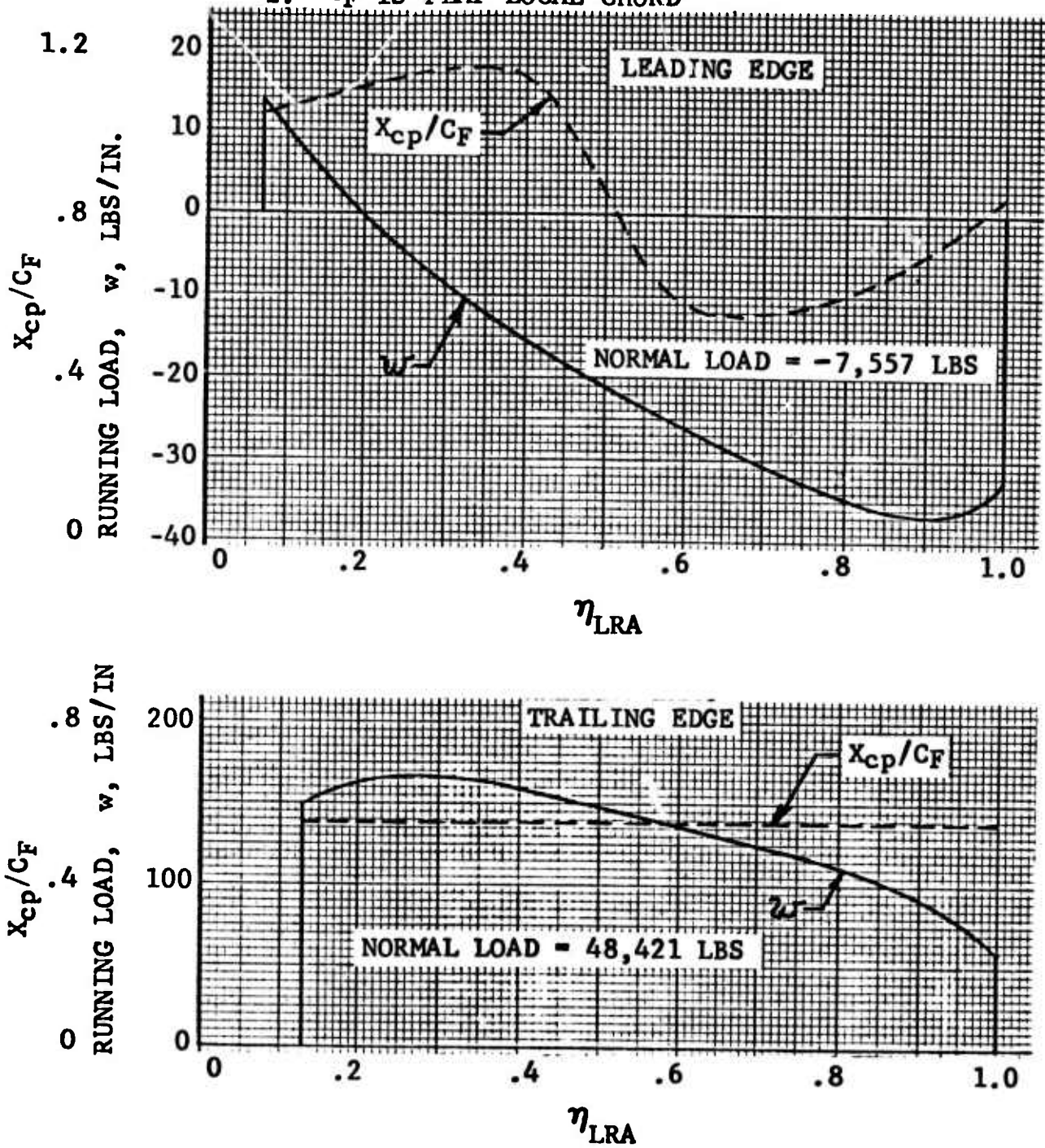


Figure A.65 ATW-4 Wing Leading and Trailing Edge Flap Loads for $N_z = 1.0g$ - Condition 4

NOTES:

1. x_{cp} MEAS. FWD. OF .15C @ L.E. FLAP
2. x_{cp} MEAS. AFT OF .65C @ T.E. FLAP
3. C_F IS FLAP LOCAL CHORD.

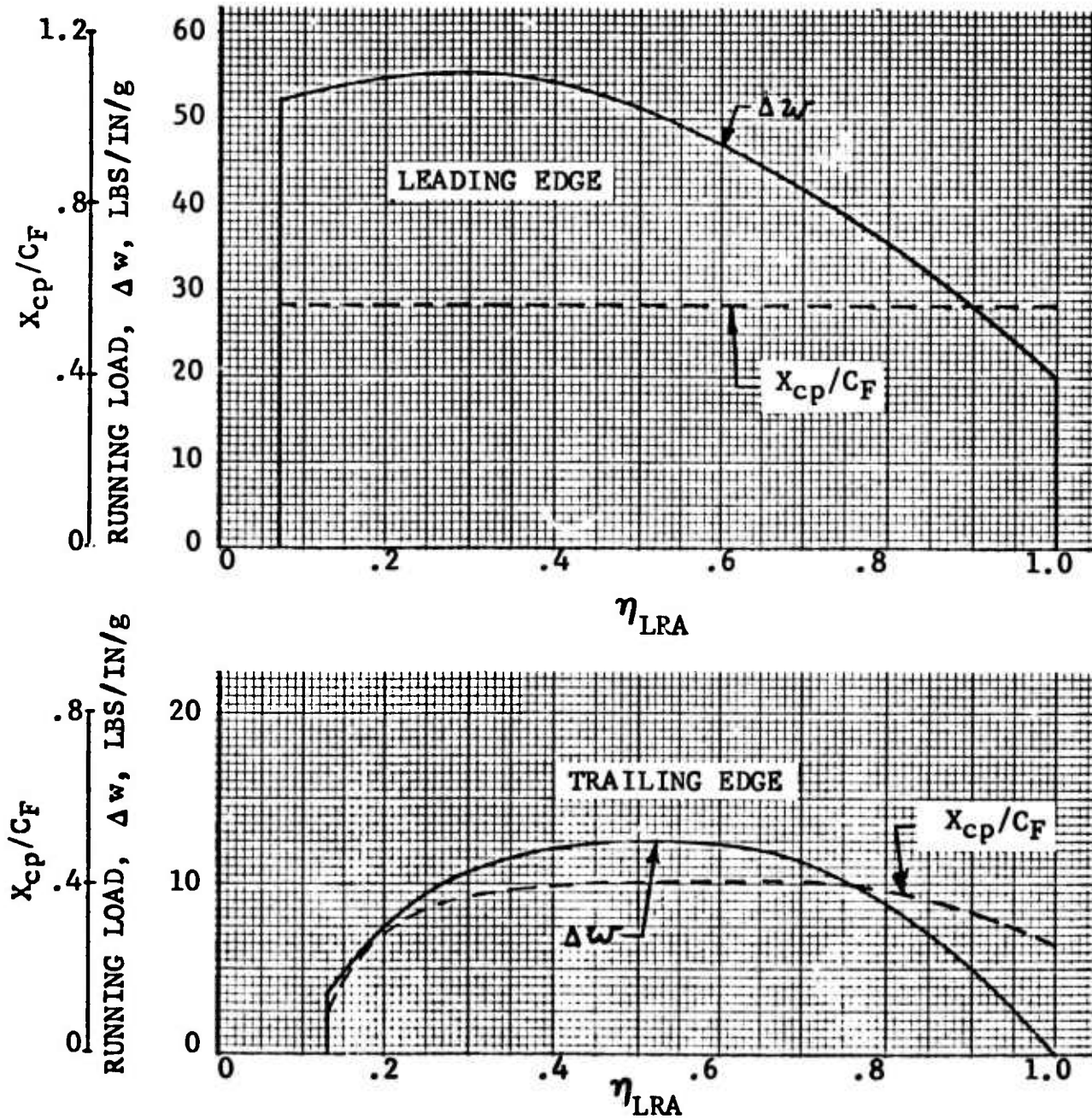


Figure A.66 ATW-4 Wing Leading and Trailing Edge Flap Loads for $\Delta N_z = 1.0g$ - Condition 4

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APPENDIX B

FATIGUE DESIGN ALLOWABLE CURVES

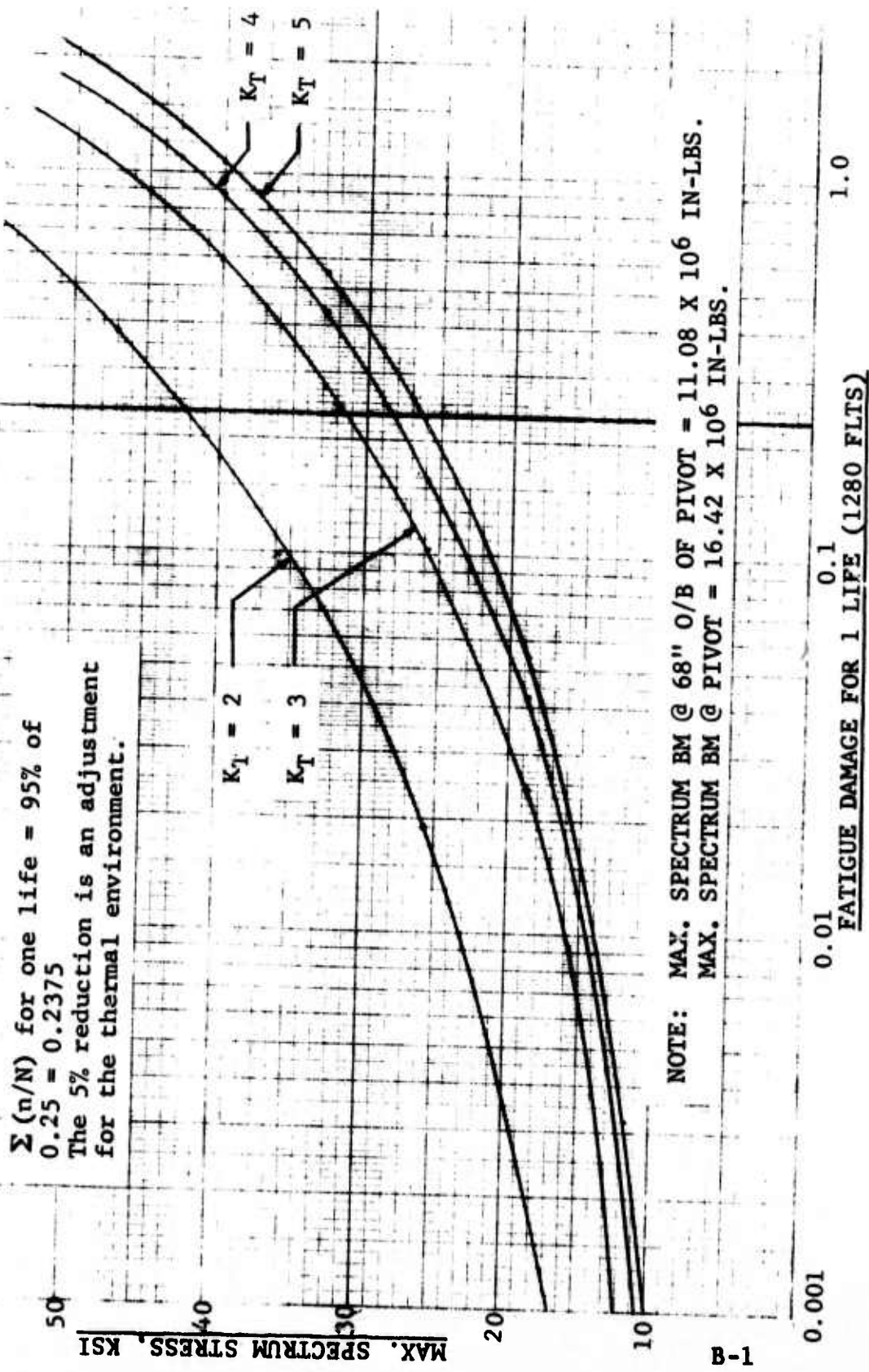


Figure B.1 ATW-4 Wing Box Lower Surface Fatigue Design Allowable Curves (2024-T851 Aluminum)

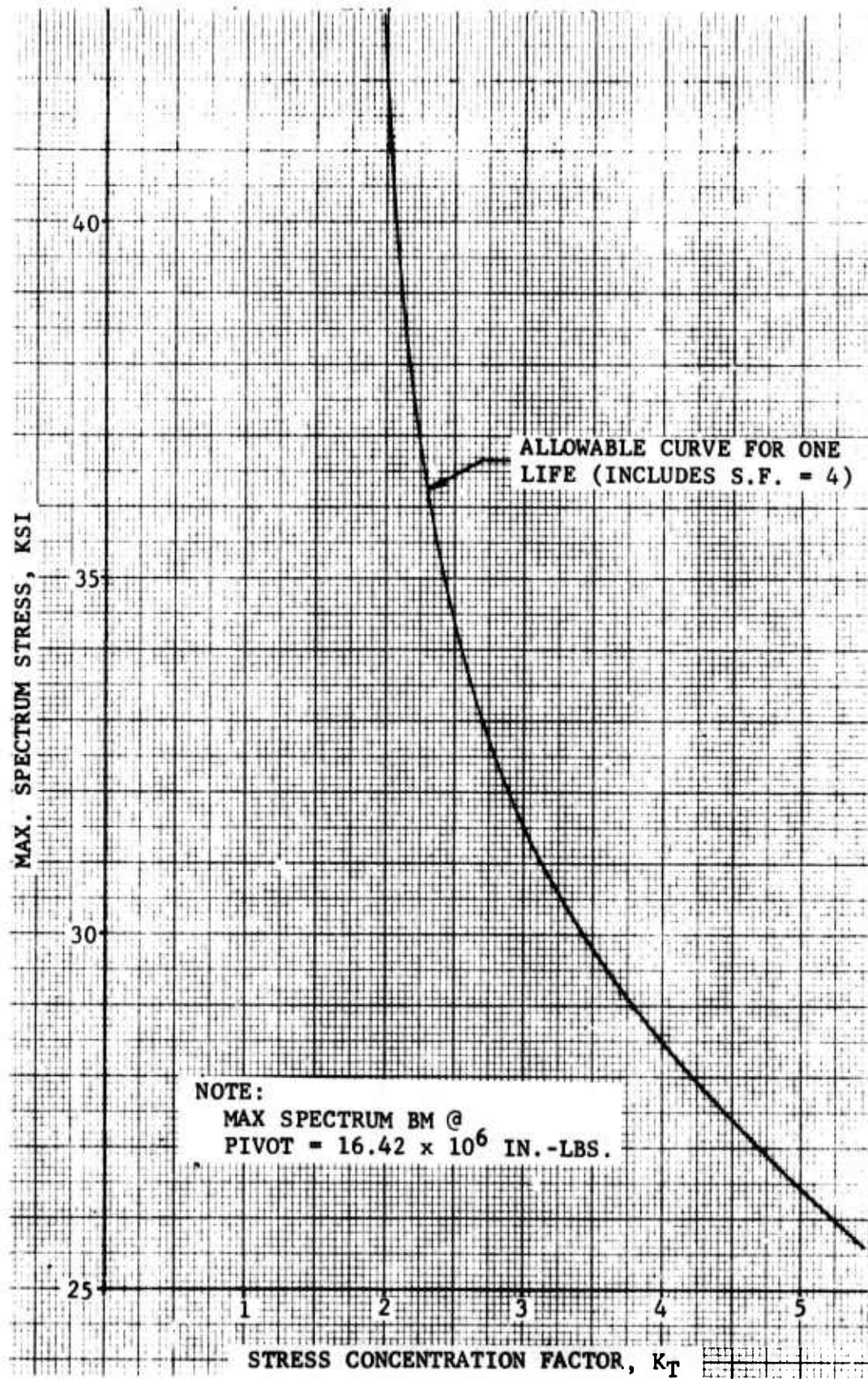


Figure B.2 Fatigue Design Allowable Curve - Wing Box Lower Surface (2024-T851 Aluminum)

NOTE: MAX. SPECTRUM BM @ 68" O/B OF PIVOT = 11.08×10^6 IN-LBS.
 MAX. SPECTRUM BM @ PIVOT = 16.42×10^6 IN-LBS.

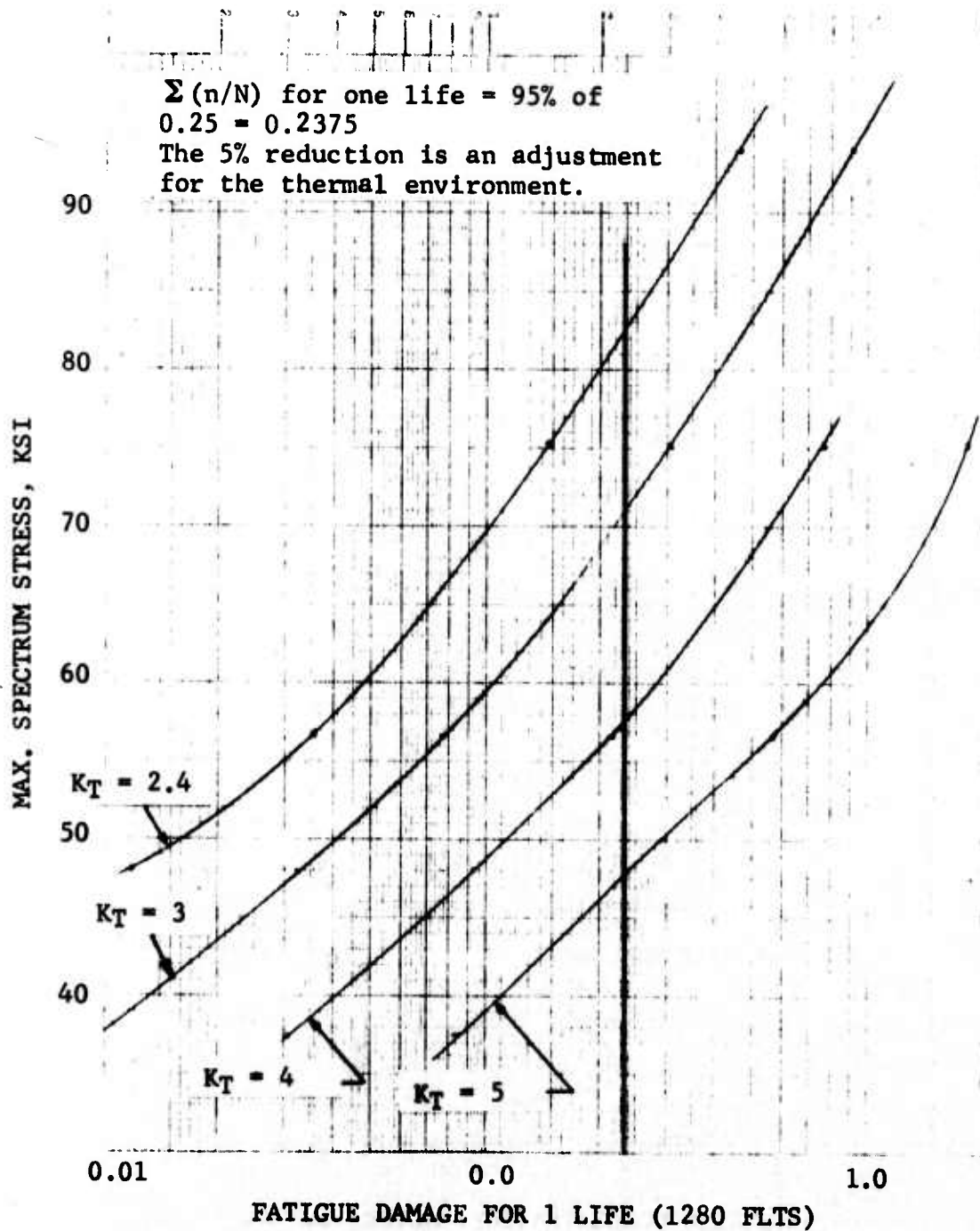


Figure B.3 ATW-4 Wing Box Lower Surface Fatigue Design
 Allowable Curves (6AL-4V Beta Annealed Titanium)

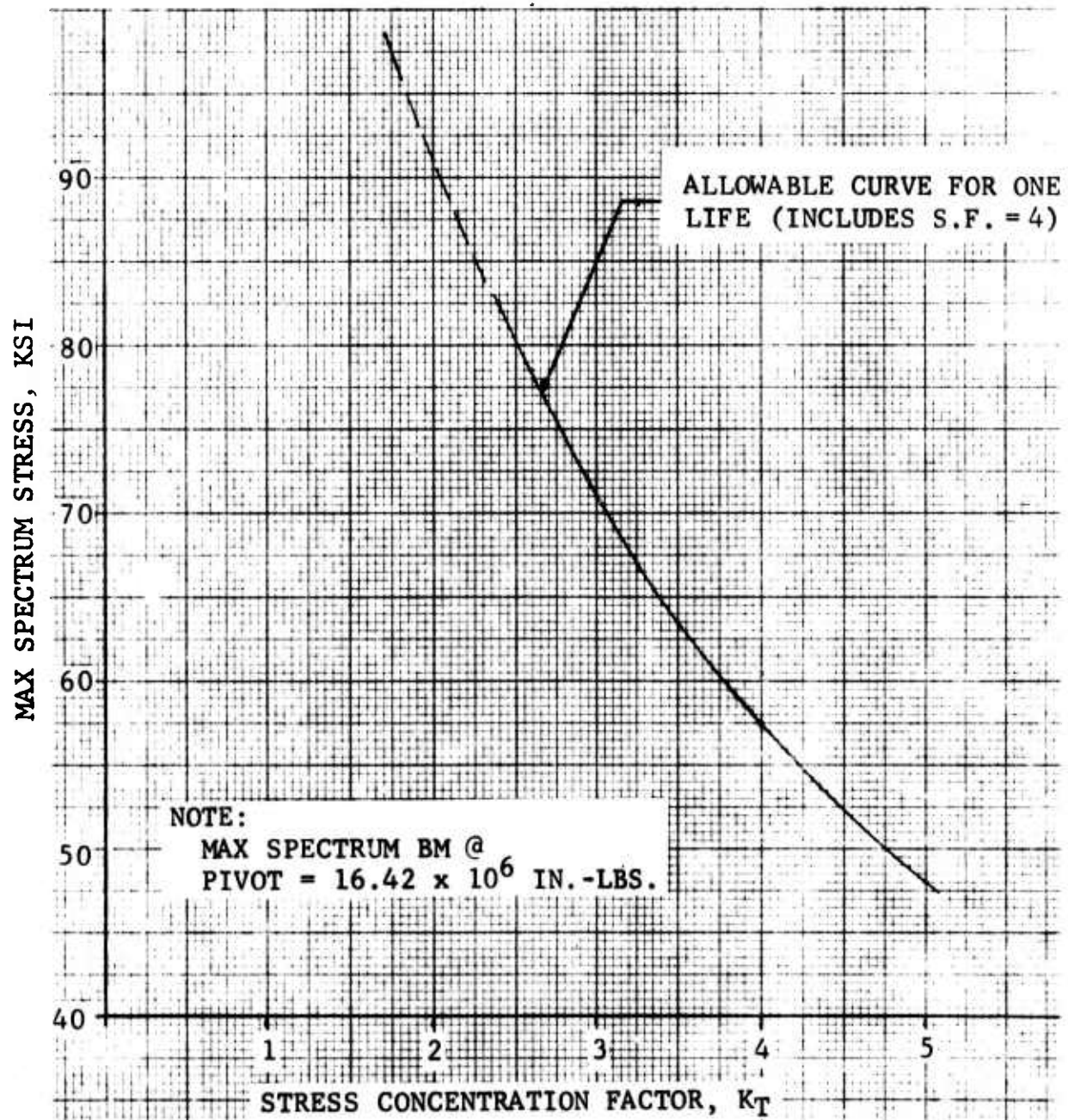


Figure B.4 Fatigue Design Allowable Curve - Wing Box Lower Surface (6AL-4V Beta Annealed Titanium)

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APPENDIX C

FRACTURE DESIGN ALLOWABLE CURVES

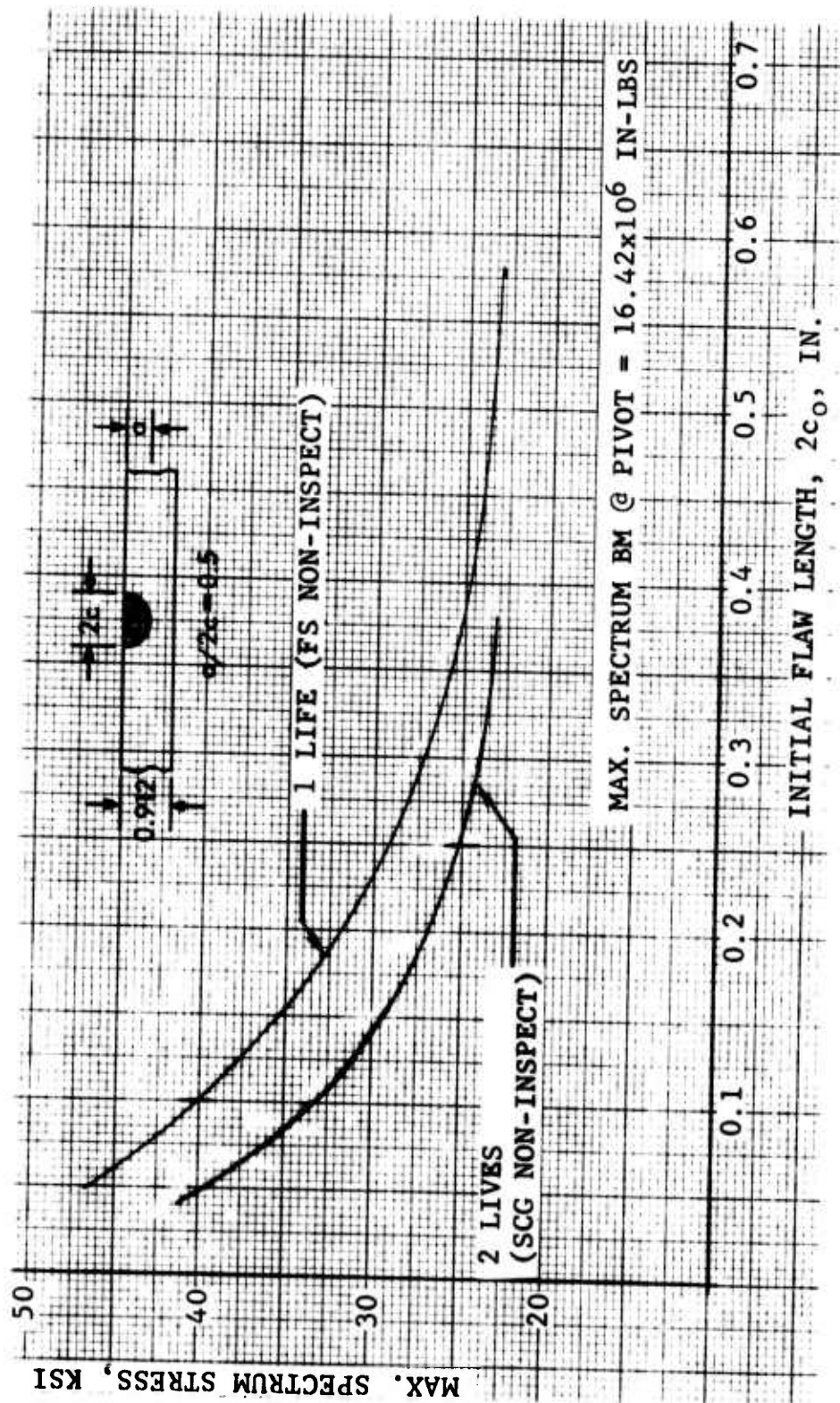
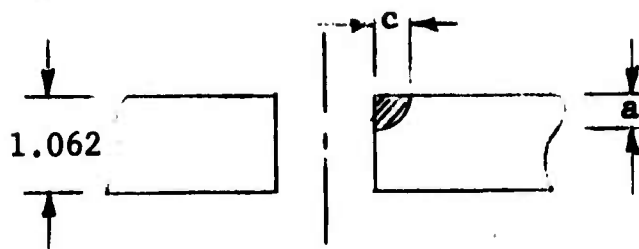


Figure C.1 Fracture Design Allowable Curve - Wing Box
Lwr Surface (2024-T851 Aluminum Surface Flaw)



$$a/2c = 0.5$$

NOTE: MAX. SPECTRUM BM @ PIVOT = 16.42×10^6 IN.-LBS.

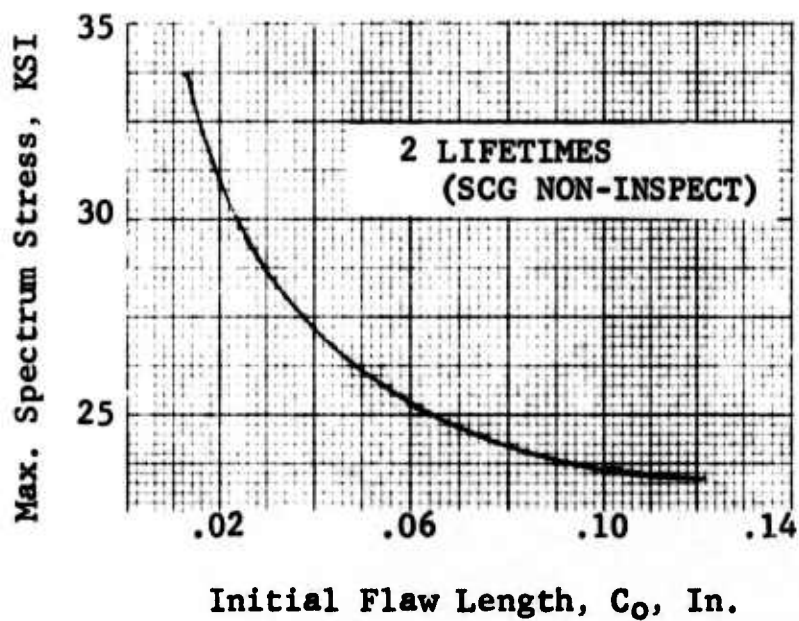


Figure C.2 Fracture Design Allowable Curve - Wing Box Lower Surface (2024-T851 Aluminum Corner Flaw)

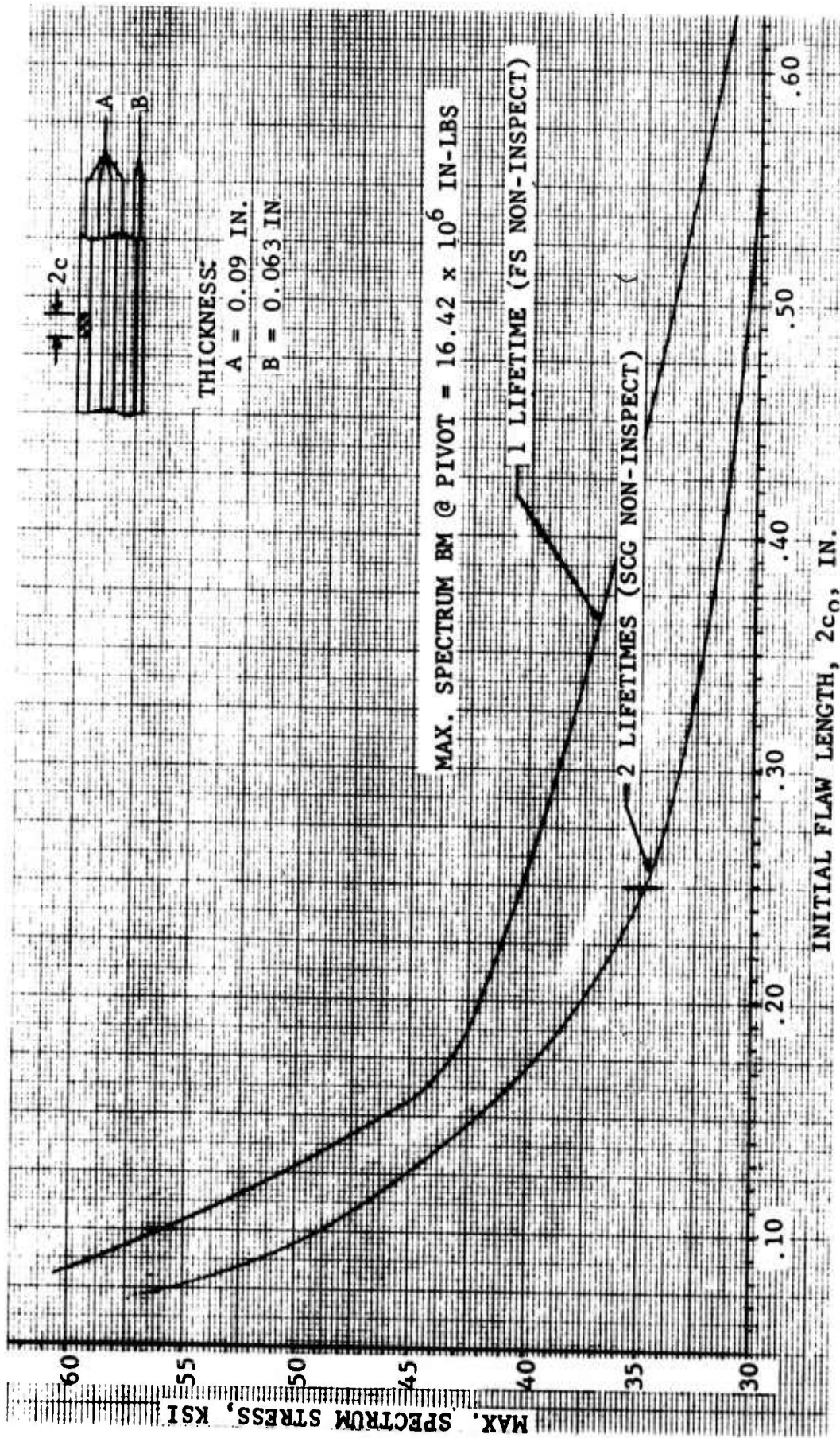


Figure C.3 Fracture Design Allowable Curve - Wing Box
 Lwr Surface (Laminated 2024-T81 Aluminum Surface Flaw)

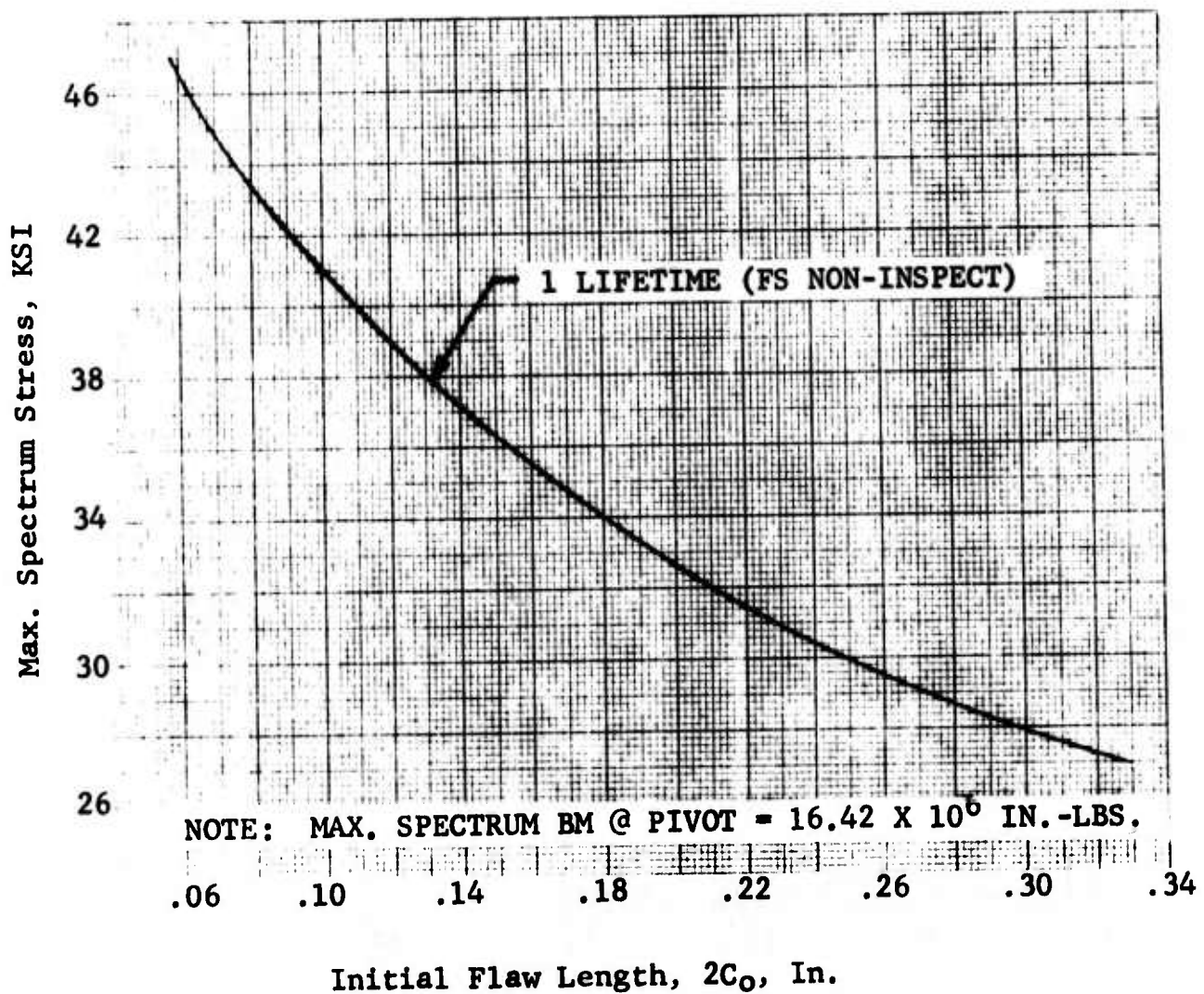
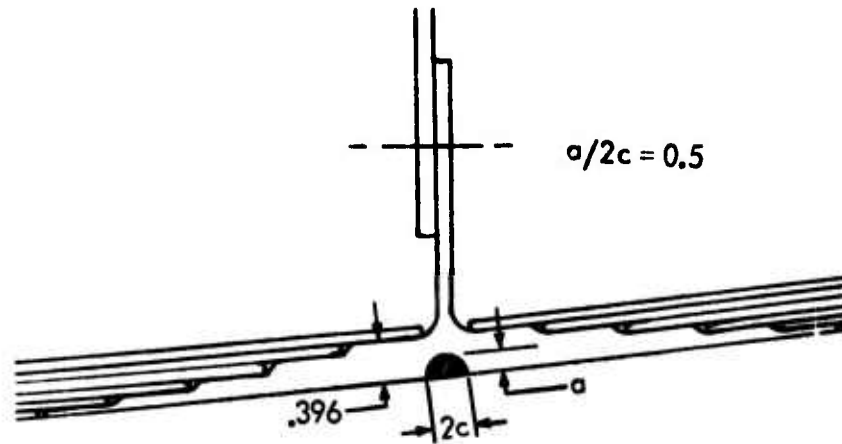
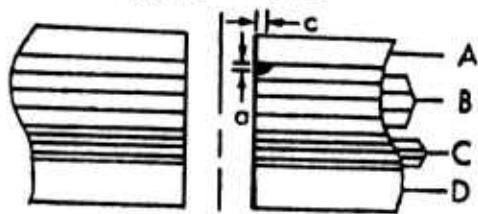


Figure C.4 Fracture Design Allowable Curve - Wing Box
Lwr Spar Cap (2024-T8511 Aluminum Surface Flow)

TAPER-LOK BOLT HOLE
 $a/2c = 0.5$



THICKNESS	MATERIAL
A=0.2 IN	2024-T8511 AL
B=0.09 IN	2024-T81 AL
C=0.063 IN	2024-T81 AL
D=0.31 IN	10 NICKEL STEEL

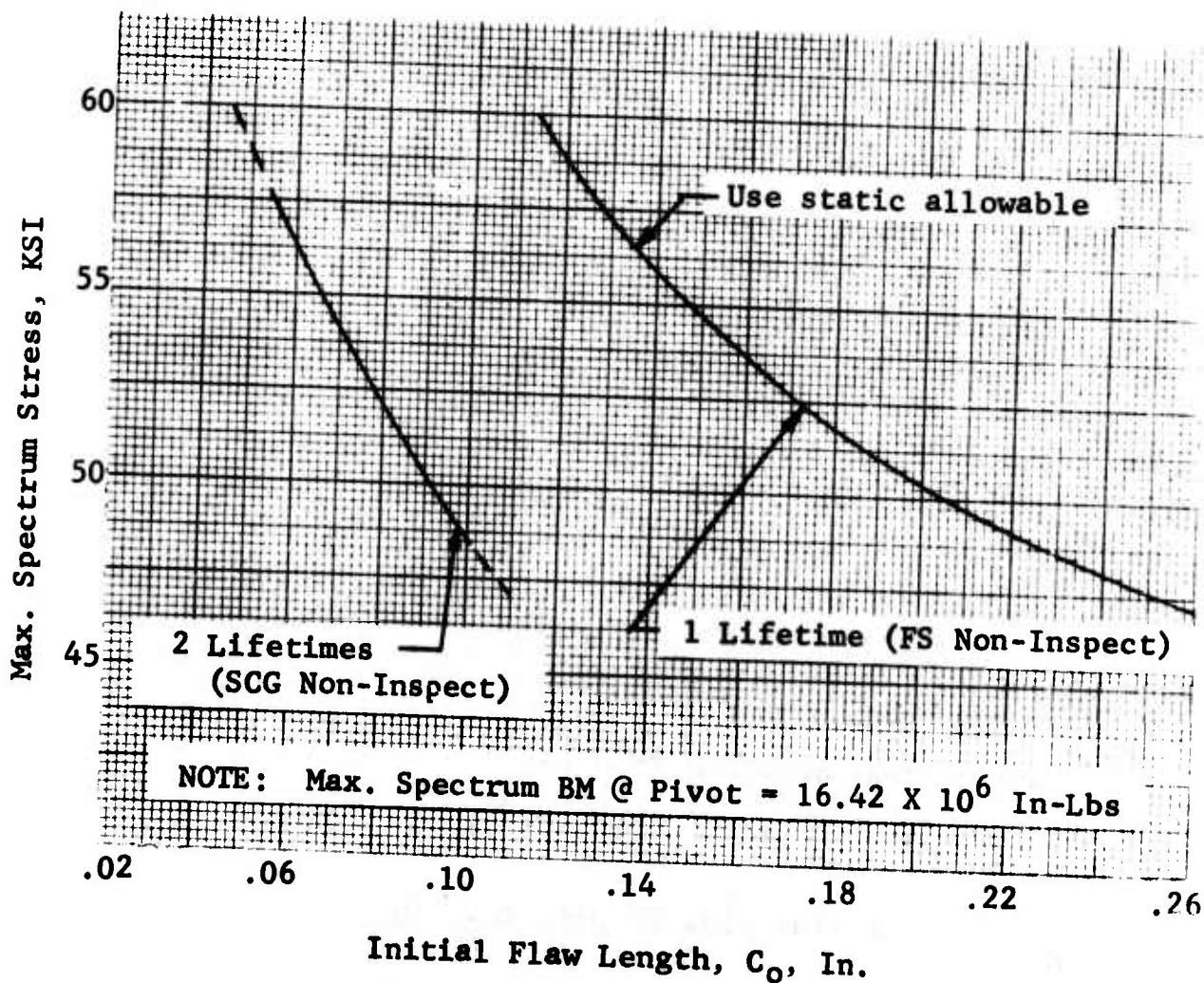
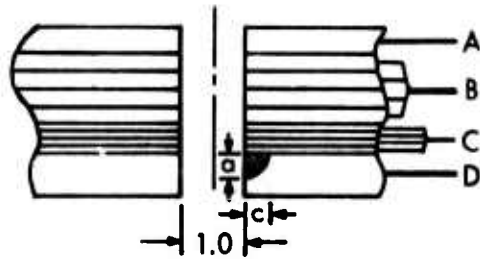


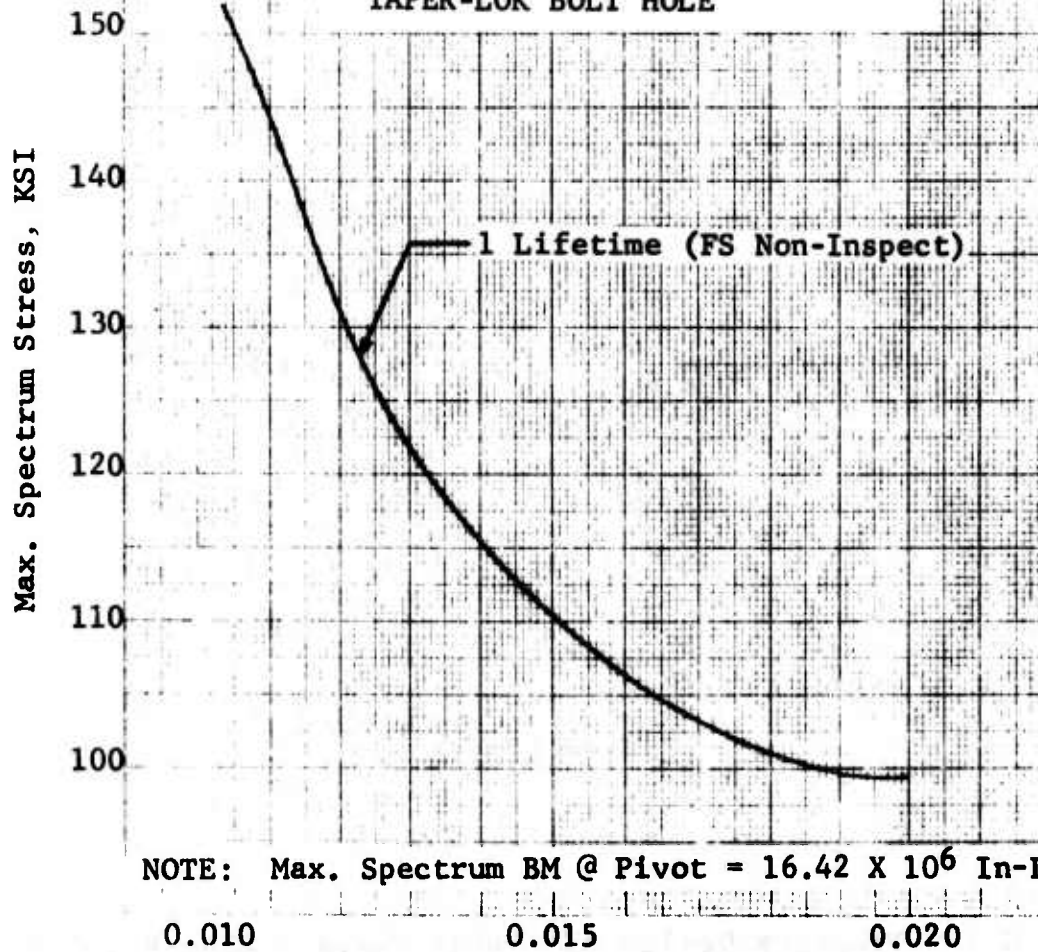
Figure C.5 Fracture Design Allowable Curve - Wing Pivot Fitting (Laminated 2024-T81 Aluminum Corner Flaw)

$$a/2c = 0.5$$



THICKNESS	MATERIAL
A=0.2 IN	2024-T8511 AL
B=0.09 IN	2024-T81 AL
C=0.063 IN	2024-T81 AL
D=0.31 IN	10 NICKEL STEEL

TAPER-LOK BOLT HOLE

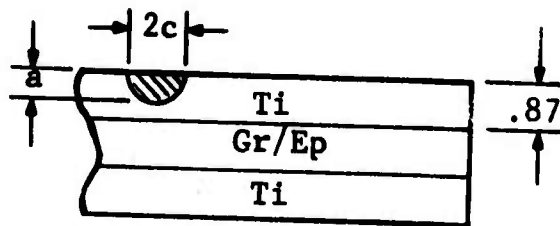


NOTE: Max. Spectrum BM @ Pivot = 16.42×10^6 In-Lbs

Initial Flaw Length, C_0 , In.

Figure C.6 Fracture Design Allowable Curve - Wing Pivot Fitting (10 Nickel Steel Corner Flaw)

$$a/2c = 0.5$$



NOTE: Max Spectrum BM @ Pivot = 16.42×10^6 In-Lbs

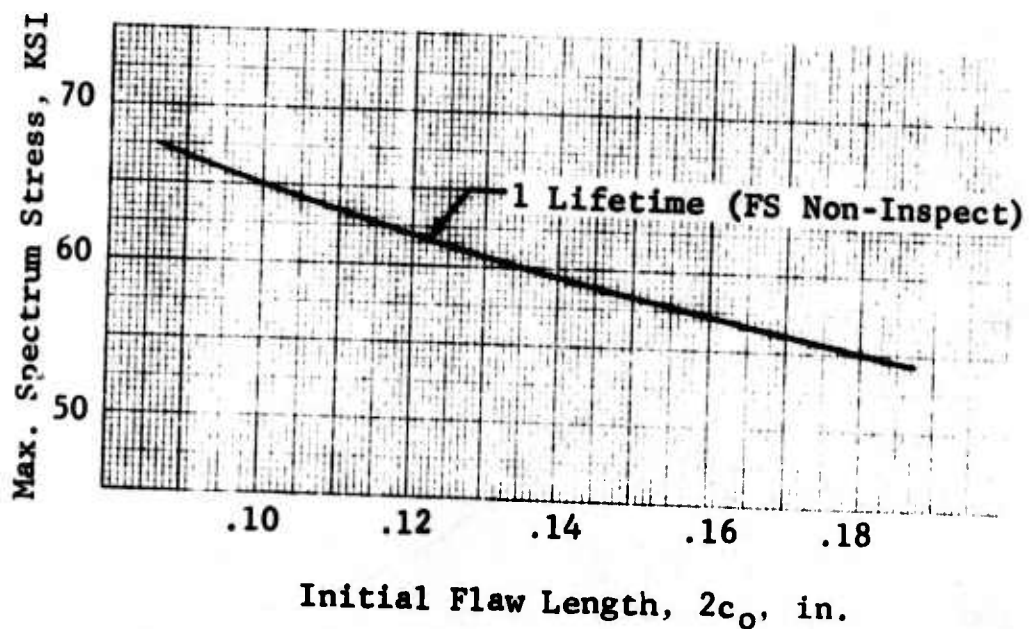


Figure C.7 Fracture Design Allowable Curve - Wing Pivot Fitting (Ti-6AL-4V Surface Flaw)

APPENDIX D
FOLLOW-ON PROGRAM

A P P E N D I X D

F O L L O W - O N P R O G R A M

The logical steps that are needed to fully develop the supercritical wing for use on a future vehicle, such as the ATF, would include the following:

- o Kinematic model that demonstrates the mechanical feasibility of Variable Camber High Lift System
- o Supercritical Wing Structural Component Test Program
- o Full Scale Wing Structural Test Adv. Dev. Program
- o Flight Test Adv. Dev. Program

The first two of the above tasks are described with a statement of work in this section.

D.1 STATEMENT OF WORK FOR DESIGN AND CONSTRUCTION OF VARIABLE CAMBER SYSTEM KINEMATIC MODEL

D.1.1 Introduction

The task defined herein encompasses the design definition and fabrication of one (1) constant section kinematic model that duplicates a production design concept of a leading and trailing edge variable camber system. The task will span a 5 month period. The kinematic model will be used as an instrument to:

- o Promote interest in the variable camber wing concept.
- o Build confidence in leading and trailing edge variable camber devices, particularly in the area of flexible skins and variable camber driving mechanisms.

- o Improve understanding of selected variable camber devices and provide insight into the principles of operation of these devices.
- o Demonstrate the aerodynamic shapes and smoothness obtainable with variable camber devices.

The variable camber devices will be supported structurally by a simulated wing box which will be supported on an attractive pedestal.

D.1.2 Engineering Task

D . 1.2.1 Design

The design task will require the preparation of the following drawings:

D . 1.2.1.1 Leading Edge

- (a) Layout
- (b) Actuation Arm
- (c) Roller
- (d) Leading Edge
- (e) Upper Skin
- (f) Lower Skin
- (g) Front Spar
- (h) Machined Details
- (i) Leading Edge Variable Camber Assembly drawing.

D, 1.2.1.2 Trailing Edge

- (a) Layout
- (b) Upper Skin
- (c) Lower Skin
- (d) Skin Beam Details
- (e) Skin Details, Trailing Edge
- (f) Channel Link Details
- (g) Actuator Beam Details
- (h) Socket Fitting Details
- (i) Sleeve Fitting Details
- (j) Coupling Details
- (k) Swivel
- (l) Beam Pivot Block Details
- (m) Pin and Bolt Details
- (n) Link Details
- (o) Fork, Trailing Edge Actuation
- (p) Slide Members, Trailing Edge
- (q) Bushing Details
- (r) Pivot Fitting Details

D. 1.2.1.3 Electrical Drive & Control

An electrical drive and control drawing will be prepared with a circuit diagram included.

D . 1 . 2 . 1 . 4 Pedestal & Simulated Wing Box

A drawing of the pedestal & simulated wing box will be prepared.

D . 1 . 2 . 2 Stress Analysis

A stress analysis shall be conducted to verify that part sizing and material selection for the model is compatible with airplane criteria required under contract F33615-75-C-3018.

D . 1 . 2 . 3 Engineering Support

Engineering support shall be provided throughout fabrication of the model.

D.1.3 Tooling Task

An overall manufacturing plan shall be made to assure that the tooling and manufacturing task is executed in an efficient manner. Parts planning shall also be accomplished. Parts planning shall make maximum use of drawing references in order to minimize the parts planning task.

Minimum cost tooling shall be provided for the composite, glass fiber, and molded rubber parts. No assembly tools will be required.

D.1.4 Fabrication Task

One (1) leading edge assembly, one (1) trailing edge assembly, and one (1) simulated wing box member shall be fabricated as shown on Figure D.1. The wing model components will be joined with mechanical fasteners to form a wing cross-section member.

The 38 inch long constant cross-section model will be mounted on a wooden pedestal as shown in Figure D.1.

Two 110V, 60 cps, 1/4 HP A.C. reversible motors will be installed in the pedestal and one mechanically linked to the leading edge drive shaft and one to the trailing edge drive shaft. Four protective limit switches will be provided, two on each of

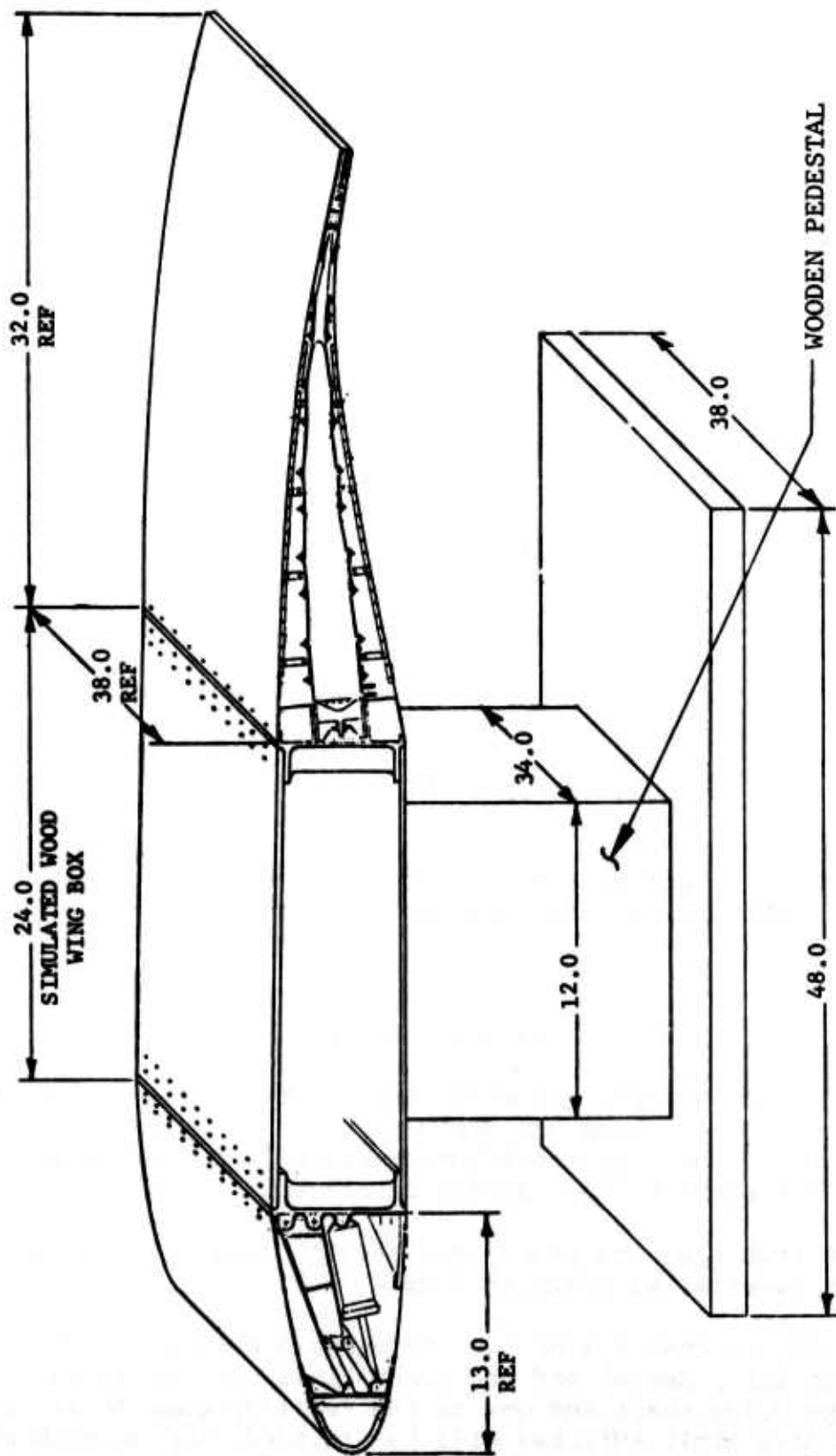


Figure D.1 Variable Camber System Kinematic Model

the leading edge device and two on each of the trailing edge device. Two control switches in separate circuits will be provided so that the leading edge and trailing edge can be operated separately.

D.1.5 Inspection

Inspection will be limited to material receiving inspection only. Final disposition of rejected material will be determined by engineering design personnel.

D.2 STATEMENT OF WORK FOR ADVANCED TRANSONIC WING DEVELOPMENT PROGRAM

D.2.1 Objective

The objective of this program is to evaluate the feasibility of integrating the advanced technologies of the Variable Camber Program and the Supercritical Wing Box Program on to a full scale advanced technology test wing. The overall approach used to accomplish this objective will be to conduct a test program to demonstrate the feasibility of the leading and trailing concepts developed during the Variable Camber Program and the wing box concept developed during the Supercritical Wing Box Program.

D.2.2 Scope

This program will consist of designing, fabricating, and testing a supercritical wing section which incorporates the top ranked leading and trailing edge concepts developed during the Variable Camber Program and the top ranked wing box structural concept developed during the Supercritical Wing Box Program. The task will span a 12 month period.

D.2.3 Program Tasks

The following paragraphs define the basic tasks that are a part of this program

- D. 2.3.1 Develop a test component integrating the selected leading and trailing edge variable camber concept and the selected supercritical wing box concept. This test component will be approximately 72 inches in length and of constant size airfoil shape (ATW-4 airfoil configuration at C.S.S. 140).
- D. 2.3.1.1 Develop detail design drawings of the test component (leading and trailing edge structure, mechanical drive systems and linkages, wing box structure, and load introduction fittings) such that actual hardware can be fabricated and assembled.
- D. 2.3.1.2 Perform stress analysis to size test component.
- D. 2.3.1.3 Perform fatigue analysis to size test component in accordance with MIL-A-008866A.
- D. 2.3.1.4 Perform damage tolerance analysis to size test component in accordance with MIL-A-83444.
- D. 2.3.1.5 Compute detailed weight calculations for test component.
- D. 2.3.1.6 Compute budgetary cost estimate for test component.
- D. 2.3.1.7 Determine the methods of fabrication and quality assurance provisions necessary for the manufacture of the test component. Develop a manufacturing plan for the test component and fabricate all tooling and fixtures required.
- D. 2.3.2 Fabricate test component.
- D. 2.3.2.1 Fabricate one test component for the leading and trailing edge and wing box design.

- D.2.3.2.2 Conduct (NDI) inspection procedures required during fabrication of test component using current state-of-the-art equipment with the necessary refinements for adapting to the specific component.
- D.2.3.2.3 Monitor all costs relevant to the fabrication of the test component. Compare these actual costs with those estimated in paragraph D.2.3.1.6.
- D.2.3.2.4 Determine actual weights of test component and compare with those estimated in paragraph D.2.3.1.5.
- D.2.3.2.5 Develop test plan for static and fatigue testing (wing bending and variable camber deflection) the test component. Include loads, instrumentation, inspection requirements, fatigue spectrum, and safety provisions necessary for testing.
- D.2.3.2.6 Fabricate the test fixtures required for conducting the static and fatigue tests.
- D.2.3.3 Conduct testing of the component in accordance with test plan prepared in paragraph D.2.3.2.5 and summarized in Table D-I. The loads that will be shown in the test plan will simulate those measured on the baseline aircraft. The instrumentation required by the test plan will be adequate to control inputs and measure load distribution and response to the applied loads.
 - D.2.3.3.1 Conduct strain survey of test component at limit load to determine stress distribution.
 - D.2.3.3.2 Conduct fatigue test of component to four lives. During fatigue loading (wing bending loads) perform cyclic operations (deflections) of variable camber systems.

TABLE D-I TEST PLAN SUMMARY

TEST	SPECIMEN	INSTRUMENTATION** AND INSPECTION	LOADING	LOADING CONTROL
Strain Survey	See Figure D.2	Inspect after loading to limit load	1 condition - Max bending @ Limit Load. See Figure D-2	Manually Controlled
Fatigue*	See Figure D.2	Inspect after each lifetime	4 lives @ 142,060 cycles per lifetime of wing bending. 4 lives @ 2,560 cycles per lifetime of vari- able camber deflection. See Figure D-2	Programmed
Static	See Figure D.2	Inspect after failure	1 condition - Max bending to failure. See Figure D-2	Manually Controlled

* Fatigue spectrum based on B-1 Exceedence Spectrum with
ATW-4 Supercritical Airfoil Configuration Loads.

** Assume 8 Rosettes and 4 Axial strain gages on box beam component.
Assume 6 Rosettes and 6 Axial strain gages on leading edge system
and on trailing edge system.

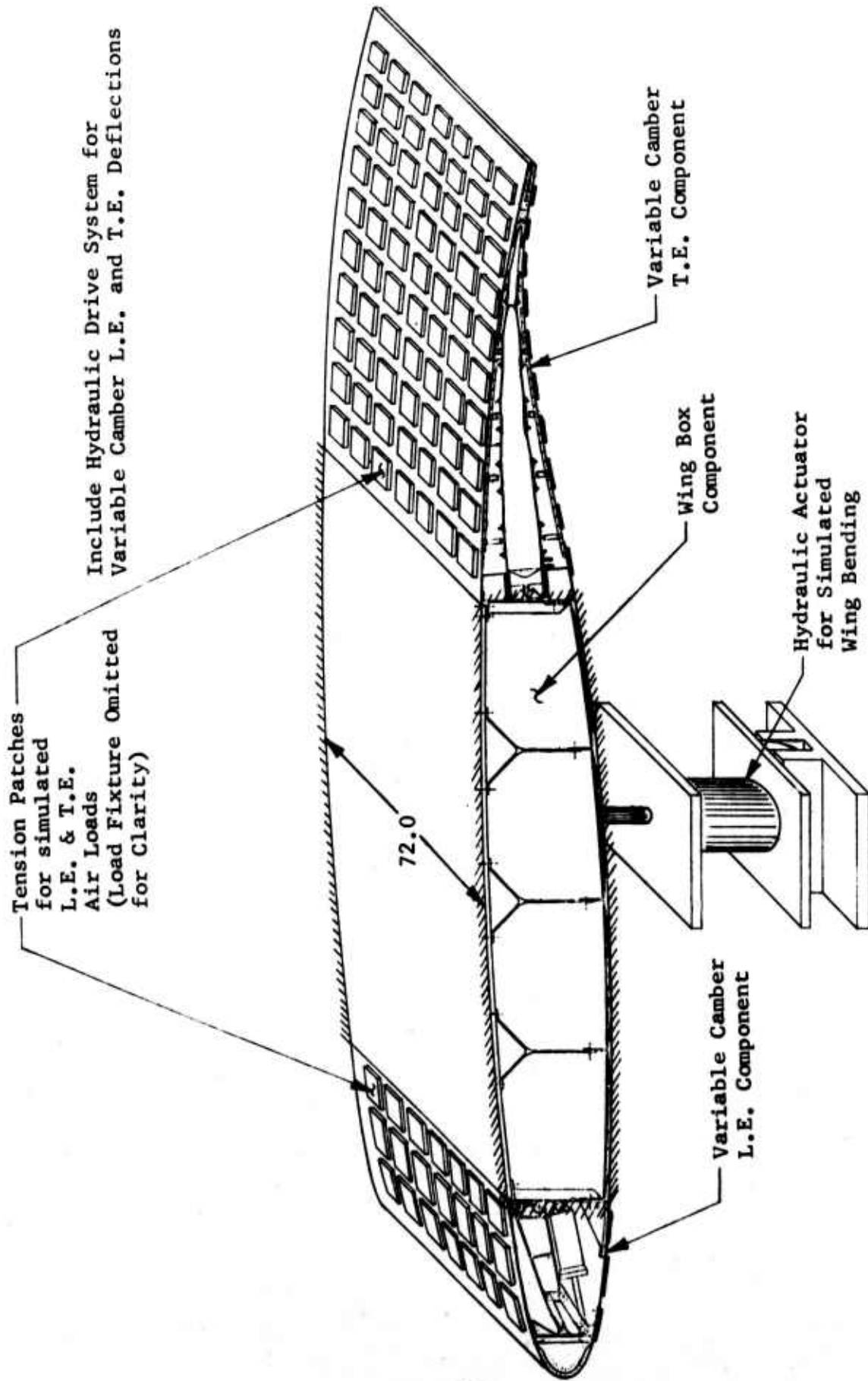


Figure D.2 Static and Fatigue Test "Set-Up" for Advanced Transonic Wing Test Component

- D.2.3.3.3 Conduct static test of component to failure after completion of fatigue testing.
- D.2.3.3.4 Conduct nondestructive evaluation prior to testing to detect and monitor any failure.
- D.2.3.3.5 Reduce recorded results, analyze, and evaluate. Compare with analytical predictions calculated previously.
- D.2.3.4 Evaluate reliability and efficiency of the leading and trailing edge systems and wing box configuration.
- D.2.3.5 Evaluate operational durability of the variable camber systems.
- D.2.3.6 Evaluate internal loads and stress distributions in the leading and trailing edge systems and wing box structure.

References

1. FZP-1718 "A Proposal for an Advanced Technology Wing (ATW) Configuration Design and Analysis Program".
2. ERR-FW-867, Vol. I "A Computer Program for the Aerodynamic Analysis and Design of Wing-Body-Tail Combinations at Subsonic and Supersonic Speeds", by F. A. Woodward and D. S. Hague.
3. MRL-163 "The Effects of Static Aeroelasticity in the Design of Advanced Air Vehicles", by E. E. Cwach and J. J. Hosek.

References

1. FZP-1718 "A Proposal for an Advanced Technology Wing (ATW) Configuration Design and Analysis Program".
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3. MRL-163 "The Effects of Static Aeroelasticity in the Design of Advanced Air Vehicles", by E. E. Cwach and J. J. Hosek.