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AIRCRAFT WING STRUCTURAL DETAIL DESIGN

(Wing, Aileron, Flaps and Subsystems)

Statement of Work Number 2
14 April 1993

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(NASA-CR-195487) AIRCRAFT WING
STRUCTURAL DETAIL DESIGN (WING,
AILERON, FLAPS, AND SUBSYSTEMS)
(Embry-Riddle Aeronautical Univ.)
40 p

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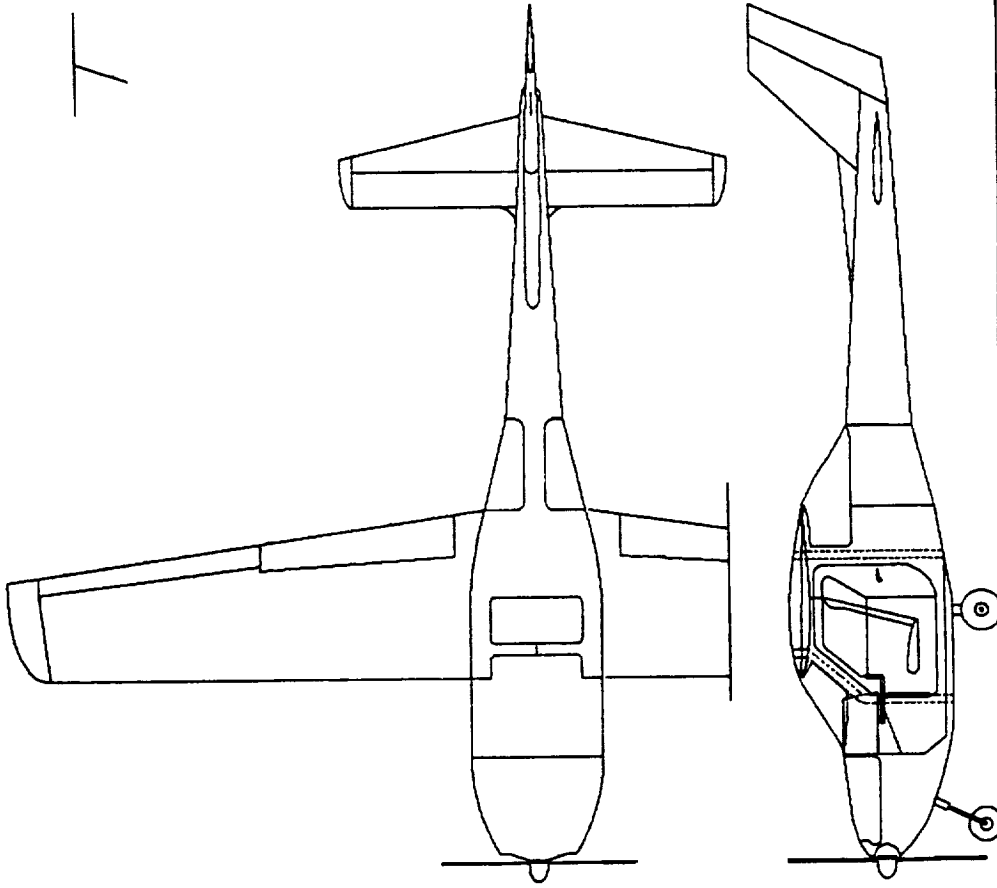
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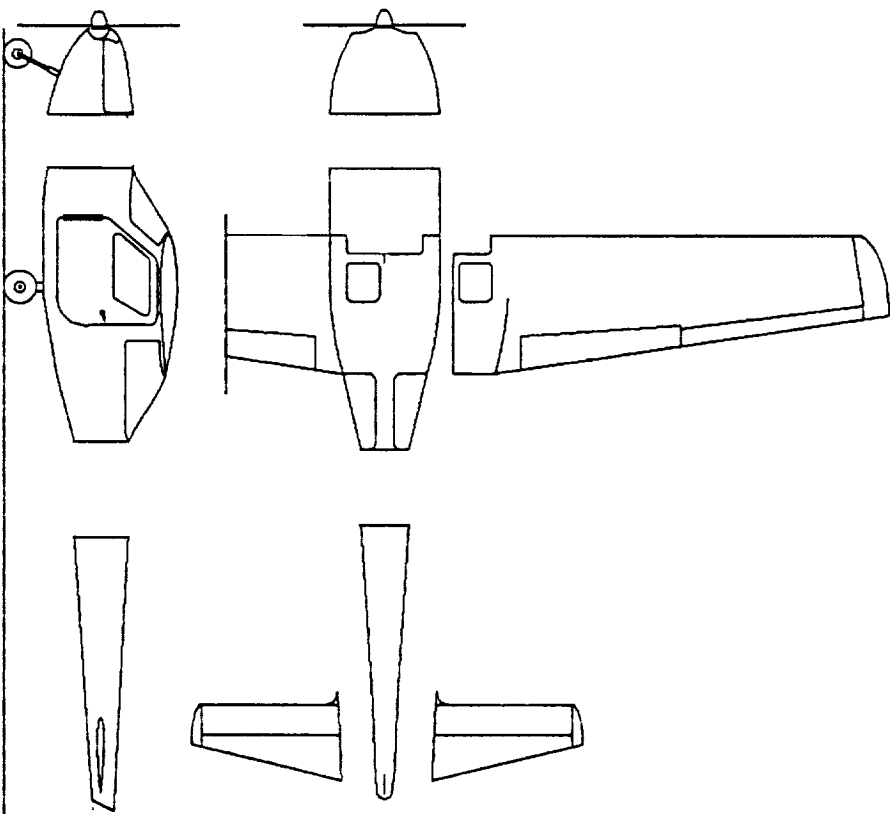
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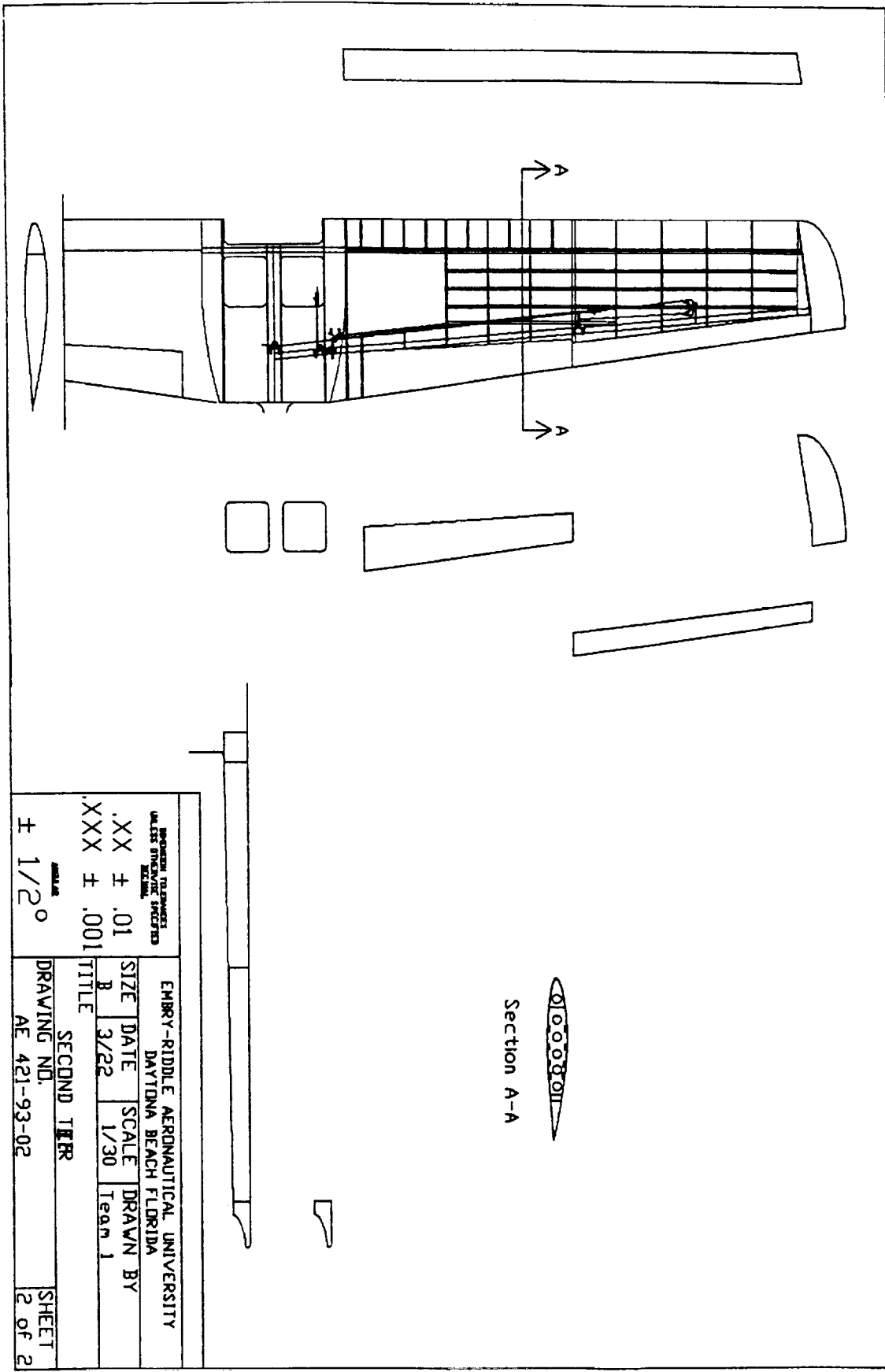
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TRITON





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± 1/2°	TITLE		DRAWN BY
	FIRST TIER		Team 1
	DRAWING NO.		SHEET
	AE 421-93-02		1 of 2




 Section A-A

REVISIONS UNLESS OTHERWISE SPECIFIED		EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA	
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DRAWING NO. AE 421-93-02	DRAWING NO. AE 421-93-02		

1. PROJECT SUMMARY

1.1 Design Goals

The goal of this project was to design, in detail, the wing, flaps, and ailerons for a primary flight trainer. Integrated in this design are provisions for the fuel system, the electrical system and the fuselage/cabin carry-through interface structure.

This conceptual design displays the general arrangement of all major components in the wing structure, taking into consideration the requirements set forth by the appropriate sections of Federal Aviation Regulation Part 23 (FAR23) as well as those established in the Statement of Work.

1.2 SOW Requirements

1.2.1

The Statement of Work (SOW) requires that the following key features of the structural design of the Triton wing be shown; spars, ribs, stringers, flaps, ailerons, all attachments and all required subsystems. The SOW also requires that the airworthiness of the wing be addressed as well as provisions for an adequate fuel tank and any needed electrical equipment or navigation lighting. The wing weight is limited to 259 lb. for the full span, including controls, flaps and subsystems. All loading considered is in accordance with FAR23 sections 561 through 629.

1.2.2

The system is to remain operable for temperatures ranging from -40° F to +122° F and to atmospheric conditions experienced up to 10,000 feet. The design will also incorporate provisions

for environmental conditions experienced from sand and dust, rainfall at 4.0 inches per hour with 50 mph winds, humidity up to 100% at +95° F, ice on all external surfaces, 8.0 inches of wet snow, and wind gusts in accordance with FAR23.

1.2.3

The airplane design is also intended to have a twenty year service life with all critical components displaying a safe life of 10^7 load cycles on rational loads adapted from the limit load spectrum and 10,000 operational mission cycles according to the mission profile.

1.3 Design Philosophy

The design philosophy adhered to was to construct a simple, light weight wing with the proper subsystems, while maintaining all requirements set forth by the SOW and the FAR. Standard parts were used whenever possible to keep manufacturing costs down and to ease in the construction process.

1.4 Trouble Spots

The first major change to the design was the repositioning of the aileron control cables so they would not interfere with the rear spar assembly. This was not too difficult once the spacing for the ribs and stringers was determined. The only other hang-up in the design process was in determining how to attach the rear spar to the fuselage/cabin subsystem.

Summary of Critical Detail Parts

Item Name	Load Source	M.S.	Page #
Wing Skin	Torsional Loads	0.27	
Front Spar Caps	Bending Moments	0.02	
Rear Spar Caps	Bending Moments	0.033	
Aileron spar	Torsion	0.51	
Front Spar Connection Bolt	Shear	0.88	
Rear Spar Connection Bolt	Shear	1.35	

Table 1

2.0 Description of Design

2.1 Wing Skin

The wing skin serves as a smooth covering over the wing support structure providing the proper lift required for powered flight. The design of the wing skin followed all procedures set forth by the FAR to maintain proper structural stiffness, including resistance to buckling under worst case conditions. The skin combined with the ribs, stringers, and spars maintain the proper airfoil shape for the entire wingspan. The ribs and leading edge caps are hydropressed aluminum sheet and the stringers are standard aluminum extrusions. The skin for the leading edge is formed by a bending brake.

2.2 Front Spar

The front spar of the wing serves as a primary load path for the bending moments imposed by the lift of the wing. The spar cap carries the portion of the bending moments imposed on the

front spar while the shear web carries the portion of the shear load imposed on the front spar. The spar is a built-up I-beam composed of a standard aluminum extrusion for the caps, another standard extrusion for the stiffeners, and a sheet of 7075-T6 with a thickness of 0.071 inches. These components are riveted together using AD8-10 rivets between the web and the caps and DD4-10 rivets between the stiffeners and the web. Stiffener spacing was kept constant at 6 inches to aid in manufacturing. Rivet spacing on the web-cap joints was kept constant at 0.6 inches. Although smaller spar caps and webs could have been used closer to the tips of the wing, these were kept constant to avoid additional joints in the support structure.

2.3 Rear Spar

The design rationale remains similar for the rear spar. Additional loading was imposed upon the rear spar by the moments about the attachment fittings of the flaps and ailerons. The spar is a built-up I-beam composed of a 7075-T6 extrusion for the spar caps, a standard extrusion for the stiffeners, and a 0.05 inch thick sheet of 7075-T6 for the web. AD6-4 rivets were used to secure the web to the spar caps while AD4-4 rivets were used to attach the stiffener to the web.

2.4 Ailerons

FAR regulations were used for the design of the ailerons and the attachments required for proper aileron deflection. The aileron deflection was $+20^\circ$ and -10° . The aileron type selected to be incorporated with the Triton are Frise ailerons. The area

of each aileron is 15ft^2 . The aileron skin and stiffeners are composed of 2024-T3 while the internal channel and push rod coupler are constructed of 7075-T6.

2.5 Spar Attachments

The front spar is one continuous piece, this is to ease in manufacturing, installation, and transportation of a fully assembled wing. The spar is bolted directly to the fuselage/cabin frame. The bolts used are AN 4-15 and these bolts are placed in shear. The rear spar is not one continuous piece, it is two pieces joined at the center of the fuselage. The bolts used to attach the rear spar to the fuselage/cabin frame are AN 3-14.

3.0 Loads and Loading

3.1 Wing Skin

3.1.1

The design procedure followed incorporates the C_{Mac} multiplied by the dynamic pressure, then moving this moment to .4c on the m.a.c. which causes a nose up pitching moment. Added to this moment are the pitching moments created by the flaps and ailerons (See Fig. 4).

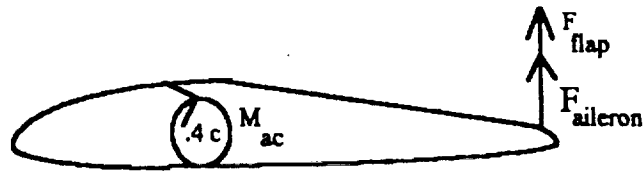


Figure 4

$$\begin{aligned}
 M_{ac} &= C_{M_{ac}} \left(\frac{1}{2}\right) \rho V^2 S \bar{c} \\
 &= (-0.04)(1/2)(.002377) \left(167 \text{ kts} * \frac{1.69 \text{ ft} * \text{s}^{-1}}{\text{kts}}\right) (150.6 \text{ ft}^2) (4.57 \text{ ft}) \\
 &= -419.5 \text{ ft} * \text{lb}
 \end{aligned}$$

Move M_{AC} to leading edge and calculate moment at 0.4c,

$$M_{.4c} = -671.2 \text{ ft} * \text{lb}$$

The next things taken into consideration were the moments created by the flaps and ailerons at their respective hinges about 0.4c. The calculated force created by the flap is 183.3#. This force is divided by two due to the fact that there are two hinges which yields a force of 91.7#.

$$M_{flap_1} = (91.7\#)(1.92') = 176.1 \text{ ft} * \text{lb} \quad M_{flap_2} = (91.7\#)(1.67') = 152.8 \text{ ft} * \text{lb}$$

Next the moments for the ailerons were calculated by the same process. Next these 6 moments were summed to produce a total torsion on the wing.

$$T = M_{.4c} + \Sigma M_{flaps} + \Sigma M_{aileron}$$

$$T = 1228 \text{ ft*lb} = 14736 \text{ in*lb}$$

3.2 Front and Rear Spar

3.2.1

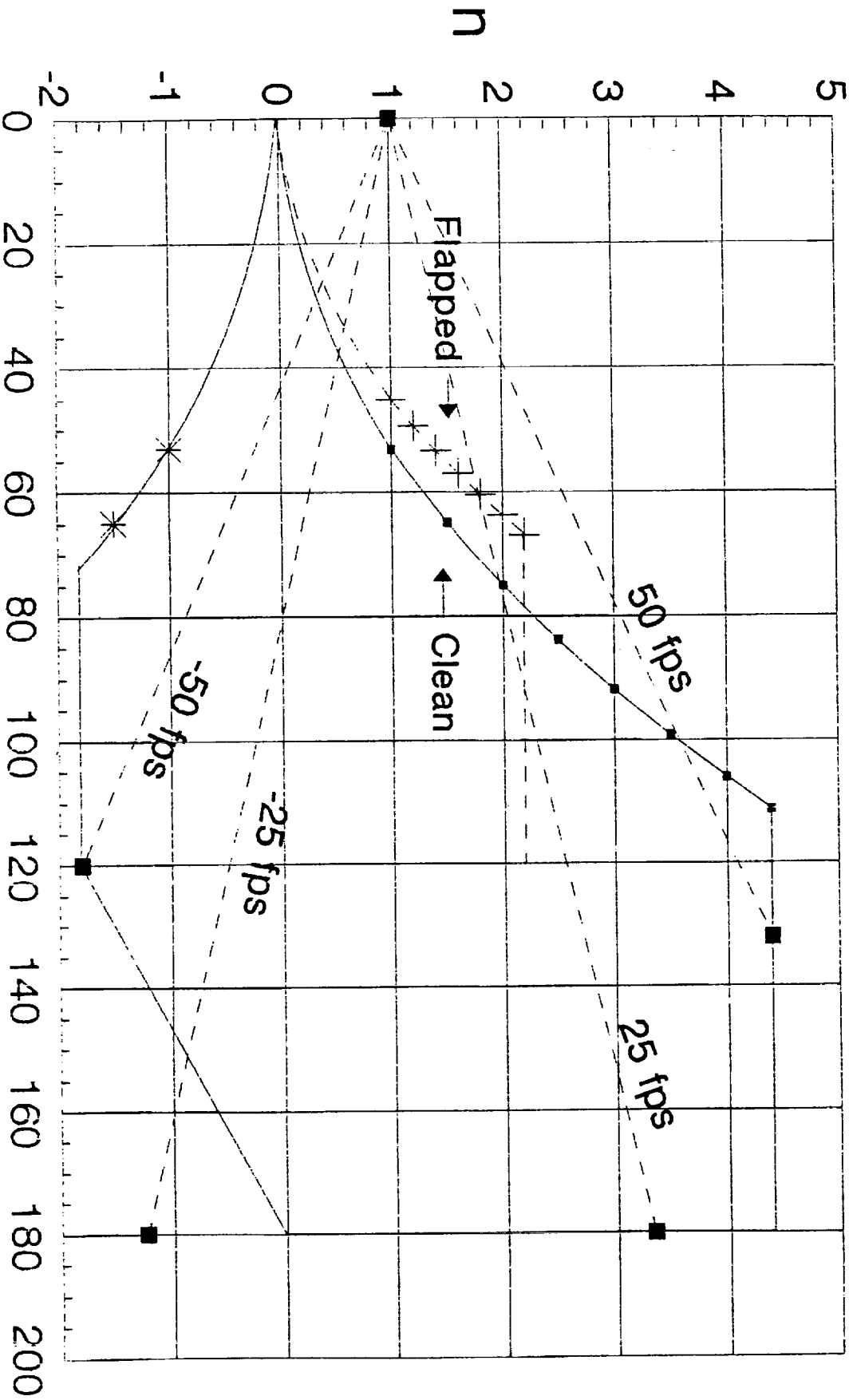
Using the V-n diagram, Figure 5, the worst case loading was determined for each spar. Using the method as outlined in Chapter 3, Sections 4 and 5, and example 6, page 79, of the textbook by Niu the spanwise lift, shear, and moment distribution curves were developed, as seen in Figures 6, 7, and 8. By approximating the center of pressure for both high and low angle of attack and using moment equilibrium the respective loading on the front and rear spars was determined. The forces are listed in Table 2.

High Alpha

Station	Shear on Front Spar	Shear on Rear Spar	Moment on Front Spar	Moment on Rear Spar
26	3543.61bs	580.91bs	25,528ft*lbs	4185ft*lbs
44	3,155.5	469.1	20,228	3,007
62	2,722.5	360.5	12,746	1,688
79	2,320.9	267.2	11,071	1,251
96	1,892.4	190.7	8,123	819
114	1,502.9	121.9	4,885	396
131	1,139.8	68.4	3,321	200
149	763.1	28.5	1,358	50
167	411.3	5.3	695	9.1
185	211.1	2.8	0	0

Table 2

Figure 5



V-N Diagram
Sea Level

Spanwise Additional Lift Distribution Curves

$$\frac{C_{la}}{C_L} \frac{C}{C_{av}}$$

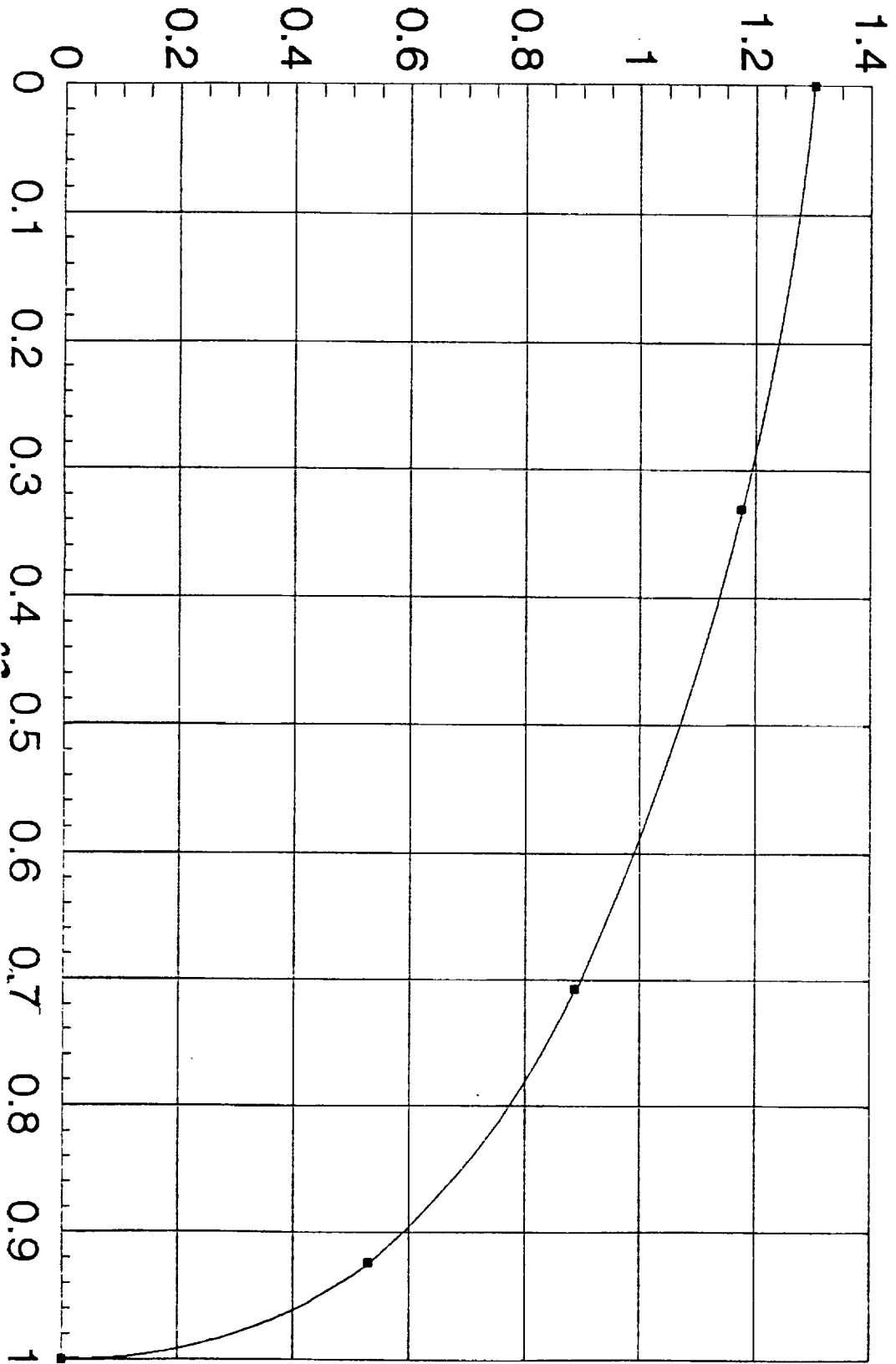


Figure 6

Figure 7

**Spanwise Shear
Distribution Curve**

$$\frac{S_z}{C_L q (\frac{s}{2})}$$

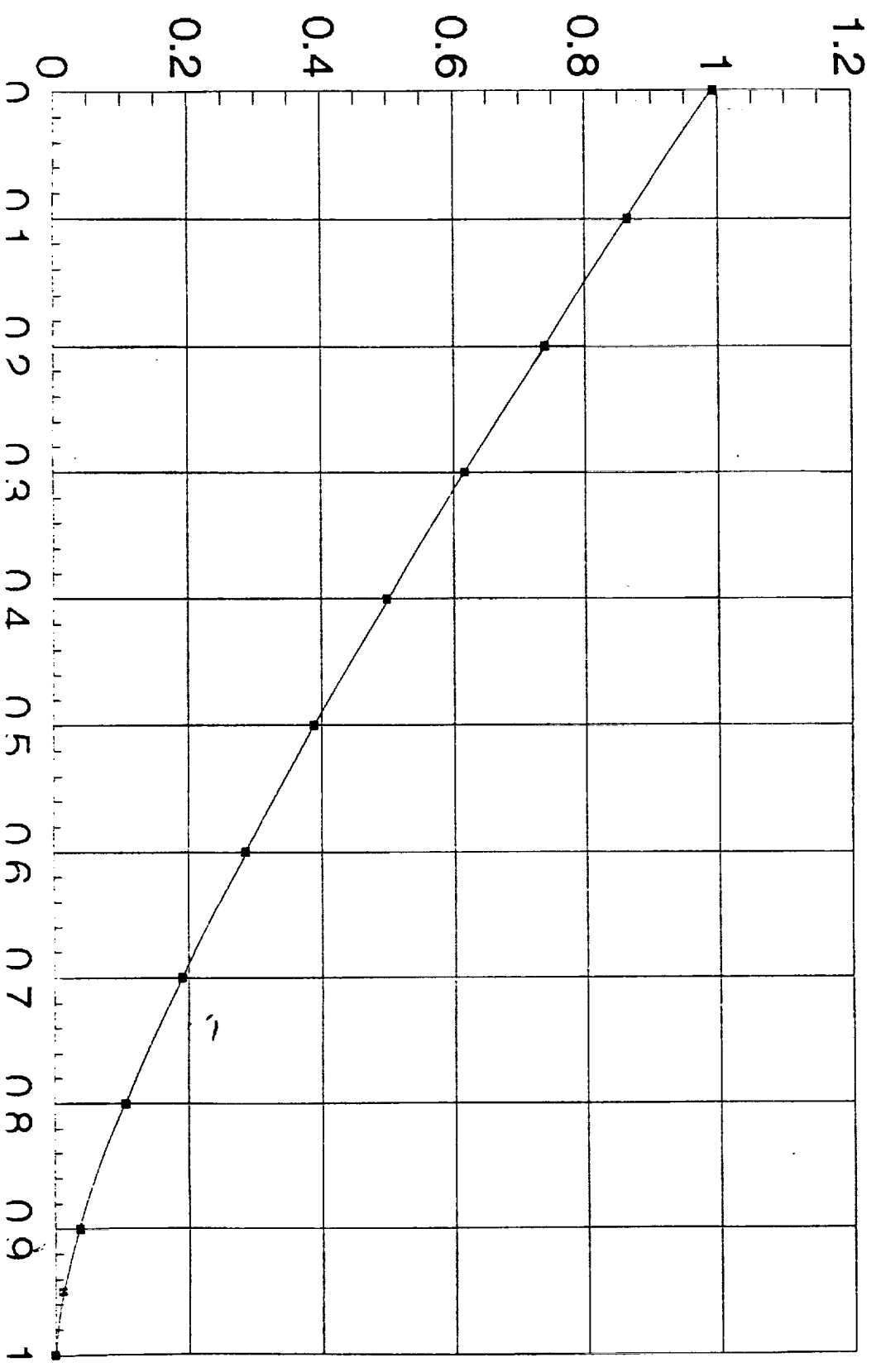
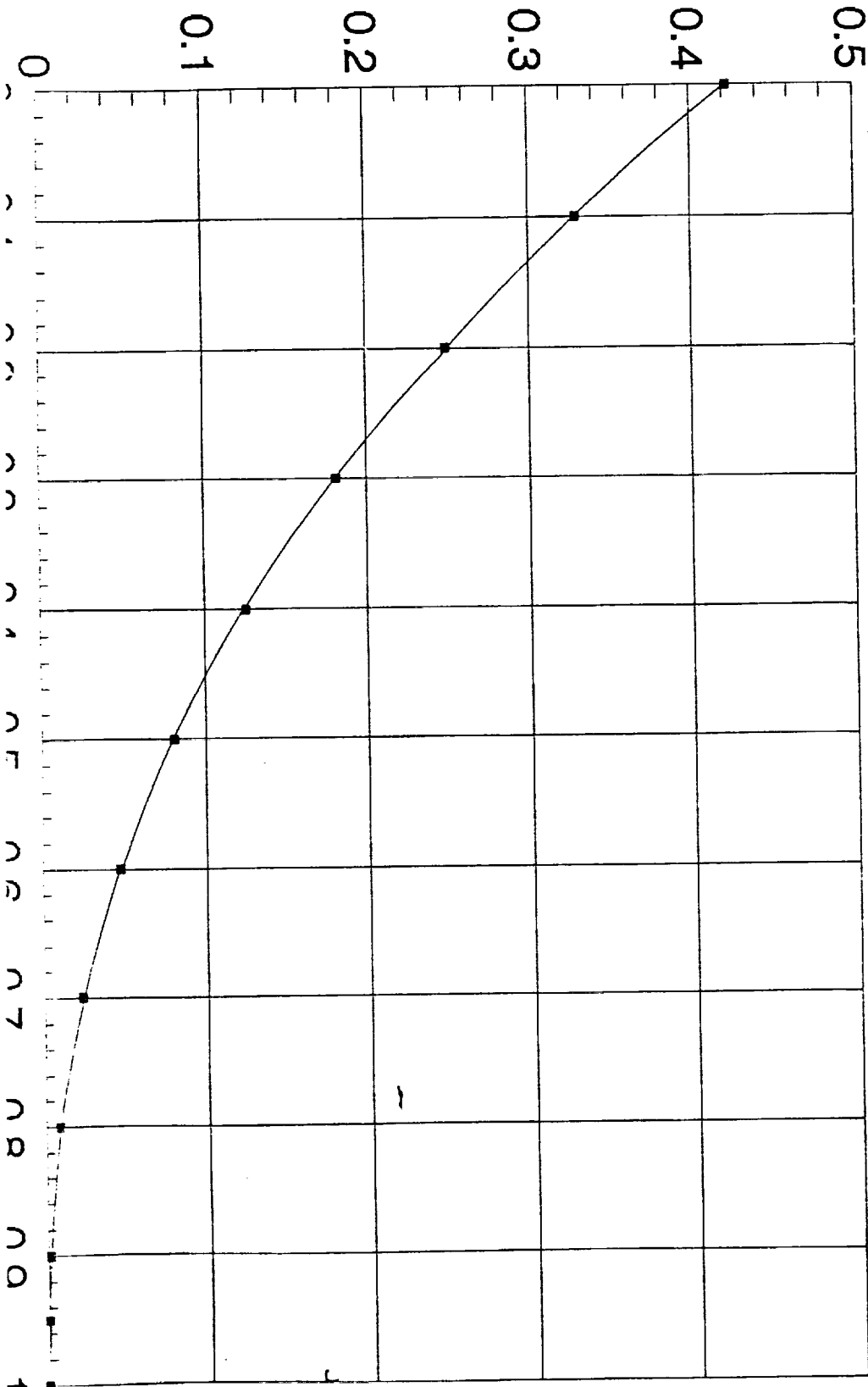


Figure 8

$$\frac{M_z}{C_L q \left(\frac{s}{2}\right) \left(\frac{b}{2}\right)}$$

**Spanwise Moment
Distribution Curve**



Low Alpha

Station	Shear on Front Spar	Shear on Rear Spar	Moment on Front Spar	Moment on Rear Spar
26	3,090.6lbs	1,050	22265ft*lbs	7561ft*lbs
44	2,760.9	877.5	17,698	5,625
62	2,389.8	704.9	13,645	4,025
79	2,040.9	552	9,732	2,632
96	1,679.9	411.4	7,211	1,765
114	1,339.8	291.2	4,354	946
131	1,020.6	192.2	2,974	560
149	686.5	108.1	1,221	192
167	371.9	46.3	629	78
185	192	17.1	0	0

Table 3

3.2.2

Additional moments caused by ailerons were determined using the coefficient of normal force. Aileron deflection at low alpha produced the highest loading of 580 lbs. This produces the highest moment at the root, equal to 11,346 ft*lbs.

3.4 Aileron

The Loads on the aileron components must adhere to FAR requirements as stated in Part 23.397. The loading on the aileron is equal to the force generated times the area.

$$F_{\text{aileron}} = W * A$$

$$F = 25.83\text{lb*ft}^2 * 7.5\text{ft} = 193.71\text{b}$$

This force was then multiplied by a 3.277" moment arm to generate the high moment which was 634.8in*lb. This moment was then multiplied by 1.25 as stated in FAR for a moment of 793.5in*lb. This moment was then transformed into a shear flow and shear stress to determine the sizing of all necessary parts.

The forces acting on the aileron spar were calculated by the use of Lotus. Several examples of the calculations from this program can be seen in the Appendix. The reason for using Lotus was that the C-channel used for the spar is not a standard extrusion. The moment of inertia and the centroids had to be calculated. The overall outcome of this program was that shear force was found to be 25,371 lbs for a thickness of 0.032in, 32,387 lbs for a thickness of 0.025in, and 40,406 lbs for a thickness of 0.020in. Using these results 7075-T6 sheet was used. This yielded a margin of safety equal to 0.51.

Calculations were done to maximize the efficiency of the skin thickness. The final thickness was determined to be 0.020in. The material used for this application was chosen to be 2024-T3. This sheet is thicker than necessary in order to use standard thickness.

The calculation of the rivet spacing yielded values which were unacceptable large. The calculations indicated a spacing of 13.7 inches. This was reduced to an assumed spacing of 7 diameters from the center of each rivet. The rivet chosen for this application was a blind rivet. The part number for this piece is CR 3213-4-4. This was chosen due to the thickness of the sheet through which it would be placed.

3.5 Attachments

The front spar is attached by bolts which are placed in shear. These bolts are under 3543.6 in*lb. After factor of safety and fitting factors were taken into consideration, both being 1.5, the final shear was 6112 in*lb. It was decided to use two AN5 bolts to minimize failure possibilities plus the single shear for one of these bolts is 5750in*lb. The same routine was followed for the rear spar, and two AN3 bolts will do the job.

4.0 Structural Substantiation

4.1 Wing Skin

4.1.1 The total torsion was then used to compute the shear flow in the wing skin, which was then divided by different standard thickness' to determine the force in the skin in the form of pounds per square inch.

$$T = 2Aq = 1228 \text{ ft*lb}$$

$$q = \frac{(1228\text{ft*lb})(12\frac{\text{in}}{\text{ft}})}{(2)(240\text{in}^2)} = 30.7 \text{ lb*in}$$

$$\frac{q}{t} = \frac{30.7}{.032} = 959\text{psi} \quad , \quad \frac{30.7}{.025} = 1228\text{psi} \quad , \quad \frac{30.7}{.020} = 1535\text{psi}$$

These forces, along with different rib and stringer spacing, were then employed in an iterative process to determine the critical buckling force in the skin.
For .020" thick skin:

$$F_{cr} = K_s E \left(\frac{t}{b}\right)^2$$

K_s and E are determined from Figure 5.4.6 from the design text.
 b is the spacing between stringers.

$$F_{cr} = (9)(10^7)\left(\frac{.020}{6}\right)^2 = 1000\text{psi}$$

This allowable force was too low for the force experienced by a thickness of .020".

For .025" thick skin:

$$F_{cr} = (9)(10^7)\left(\frac{.025}{6}\right)^2 = 1563 \text{ psi}$$

These critical forces were then compared to the initial forces calculated by dividing the shear flow by standard thickness' to determine which thickness met the requirements. It was determined that the skin thickness at the root to station 78.5 is .032" 2024-T3 Aluminum sheet. The next 48 inch spanwise section will have a thickness of .025" with the remainder of the wing being constructed of .020" thick aluminum. The leading edge skin is constructed of .025" thick 2024-T3 along the entire span.

4.1.2

The spacing between the ribs was calculated to be 16 inches for the exception of the rib at the flap/aileron division which is only 13 inches outboard from station 94.5. The ribs and leading edge are caps constructed of .020" 2024-T3 aluminum. The calculated spacing for the stringers is 6.4 inches. The stringers are standard extrusion NAS 346-11.

4.1.3

The next aspect of the wing skin design process was to determine the rivet spacing. To do this the airload moment from the additional lift was used, along with the moment about the ac and the moments created by the flaps and ailerons at different spanwise stations, to determine the total torsion on the wing. The worst case torsion, which was calculated at the wing root, was employed to determine a shear flow through the skin. This

number was then divided into the allowable single shear forces for given rivets to calculate a desirable spacing. The rivets used are MS20442AD-4-4 spaced at .6 inches. See Figure 9.

$$q = \frac{T}{2A} = \frac{366957 \text{ in} \cdot \text{lb}}{(2)(240 \text{ in}^2)} = 705 \text{ lb} \cdot \text{in}^{-1}$$

$$\text{Spacing} = \frac{f_{\text{allowable}}}{q} = \frac{388 \text{ lb}}{705 \text{ lb} \cdot \text{in}^{-1}} = .6 \text{ in}$$

4.1.4 Fatigue Life

The fatigue life of the wing skin was calculated to be in excess of 200,000 flight hours.

4.2 Front and Rear Spar (Figures 11 and 12)

4.2.1 Web Analysis

The web analysis for both front and rear spars is very similar. As such, only one calculation is shown. The following example illustrates the most critical loading of the rear web. By first finding the shear flow, the maximum allowable shear in the web can be determined. This value must be greater than the actual shear, f_s .

Web at flap hinge 2:

$$H=4.1", \text{ Lightening hole } D=$$

$$q = \frac{V}{h} = \frac{1650.7 \text{ lb}}{4.1 \text{ in}} \times 1.5 = 414.40 \text{ lb/in}$$

Figure 9

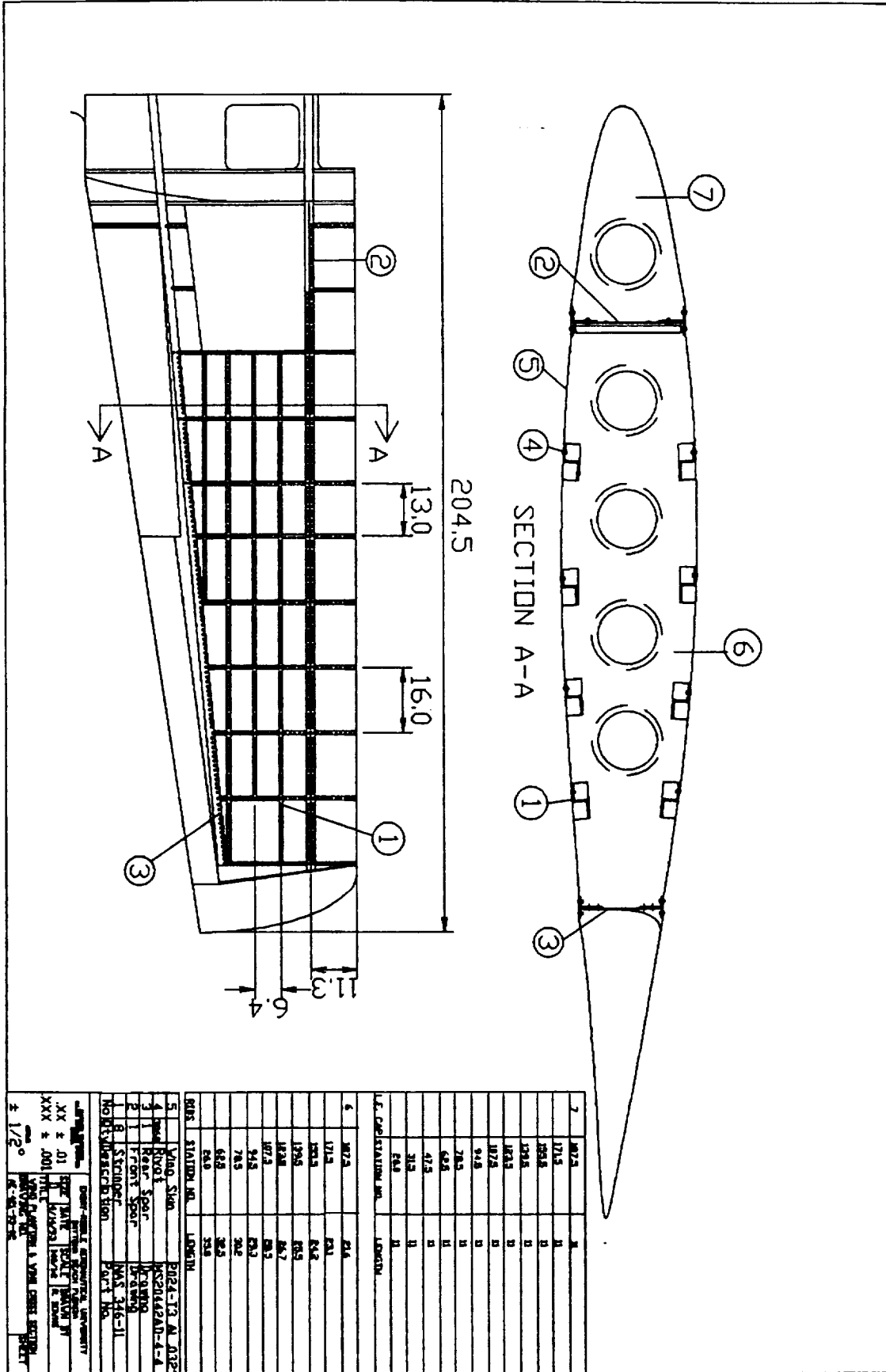
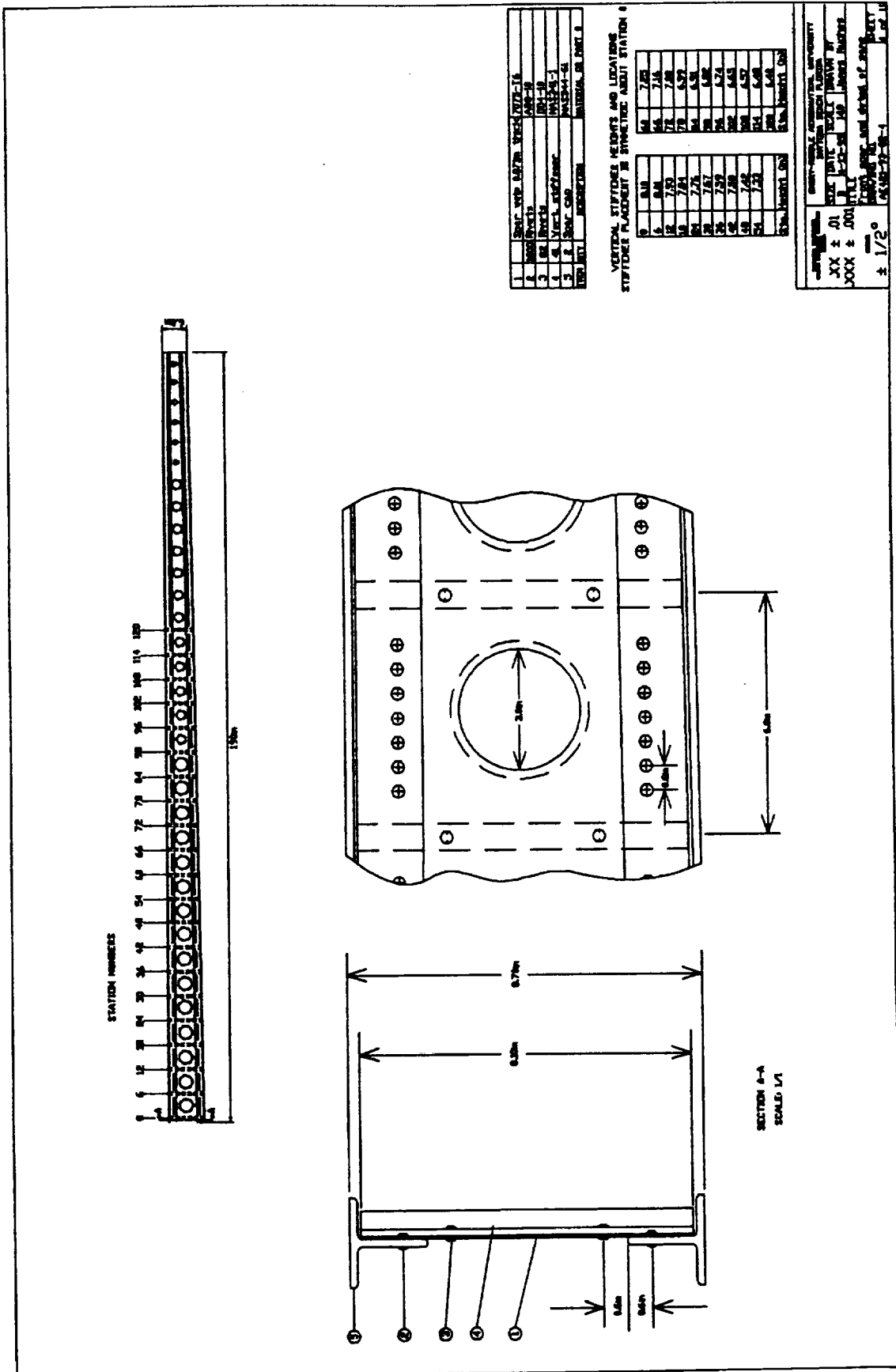


Figure 11



t	b _s /t	F ₀	K ₁	F _S	f _S
0.025	240	7,000	1.3	9,100	16,576
0.03	200	7,750	1.2	9,300	13,813
0.04	150	10,000	0.975	9,750	10,360
0.05	120	12,000	0.8	9,600	8,288

Table 4

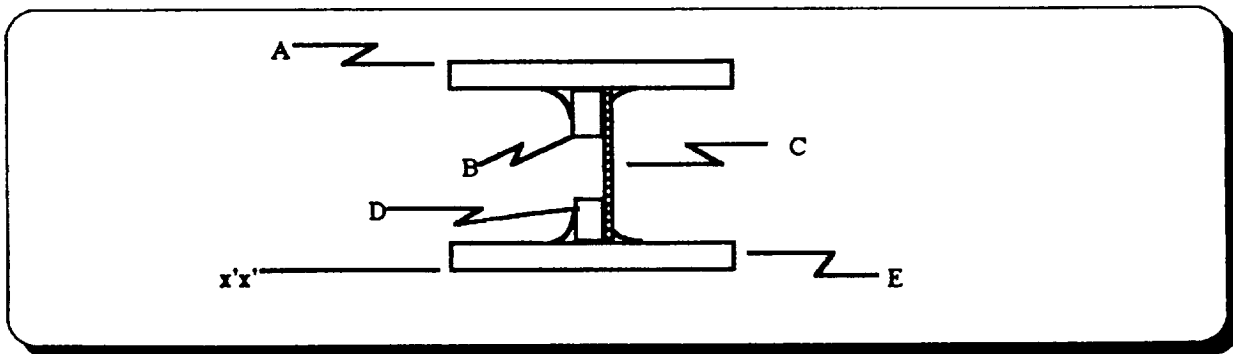
Use thickness of 0.05" for web

$$M.S. = \frac{F_S}{f_S} - 1 = \frac{9600}{8288} - 1 = .1583$$

4.2.2 Spar Cap Analysis

Once again, the process used to analyze the spar cap is very similar for both front and rear spars. The spar cap carries all of the applied bending moment. This moment acts at a distance y above the neutral axis. Using $\frac{My}{I}$, the necessary moment of inertia of the spar is determined. Using the parallel axis theorem, the required spar cap size is found.

Fig. 10



	Area	y	Ay	Ay ²	I _{CX}	I _{X'X'}
A	0.1875	5.82	1.091	6.351	0.00024	6.3512
B	0.1875	5.01	0.939	4.706	0.0352	4.7412
C	0.249	2.94	.732	2.152	0.5146	2.6666
D	0.1875	0.875	0.164	0.1436	0.0352	0.1788
E	0.1875	0.0625	0.012	0.0007	0.00024	0.0009
F1	0.0022	5.73	0.013	0.0722	-	0.0722
F2	0.0022	5.73	0.013	0.0722	-	0.0722
F3	0.0022	0.1473	0.00032	0.00005	-	0.00005
F4	0.0022	0.1473	0.00032	0.00005	-	0.00005
Sum	1.0078		2.964			14.083

Table 5

$$\bar{y} = \frac{\sum Ay}{\text{area}} = \frac{2.964}{1.0078} = 2.49$$

$$I_{XX} = \sum I_{X'X'} + (\sum \text{Area})(\sum Ay)^2 = 5.37 \text{ in}^4$$

$$F = \frac{My}{I_{XX}} = \frac{136,155 \text{ lb} \cdot \text{in}(2.49 \text{ in})}{5.37 \text{ in}^4} = 74,543 \text{ PS}$$

$$\text{M.S.} = \frac{77,000}{74,543} - 1 = .033$$

4.2.3 Composite Analysis

4.2.3.1 Front Spar

Composite analysis of the front web required a total of 4±45 degree plies to carry the shear load. 2 zero and 2 90 degree plies were arbitrarily added for lateral stability and to decrease crack propagation. The overall weight of the composite web was calculated to be 5.20 lbs. Despite the reduced weight it was decided to use an aluminum web to decrease cost of manufacturing.

4.2.3.2 Rear Spar

Composite analysis of the rear web required a total of 4 ± 45 degree plies to carry the shear. 4 zero and 4 90 degree plies were arbitrarily added for lateral stability and to decrease crack propagation, which resulted in a total thickness of 0.06". Although the composite web is slightly thicker than a comparable Aluminum web, the composite web is lighter, weighing 3.14 lbs. However, it is believed that the increased manufacturing difficulty outweighs the slight weight advantage gained by using the composite web.

4.2.4 Fatigue Life

The fatigue life of the front spar was calculated to be in excess of 200,000 flight hours, while the rear spar fatigue life was calculated to be 80,000 flight hours.

4.3 Ailerons

See Section 3.4 for analysis. Figures 13,14,15,16

The fatigue life of the aileron skin was calculated to be 18,018 flight hours.

4.4 Spar Attachments

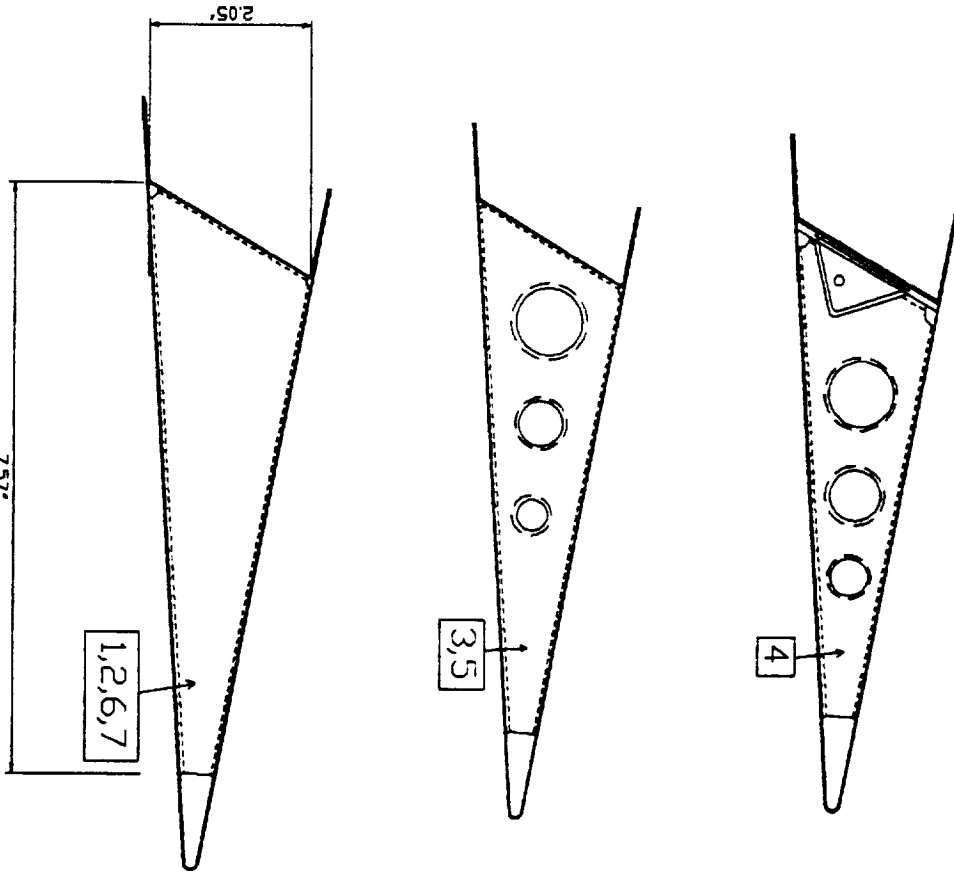
See Section 3.5 for analysis. Figure 17.

5.0 Manufacturing and Maintenance Provisions

5.1 Wing Skin

The leading edge skin for the wing is formed by a brake, while the rivet holes are drilled. The thickness of the skin is mentioned in a previous section of this report. The leading edge caps are hydropressed from a piece of 2024-T3 Al that is .020"

Figure 14

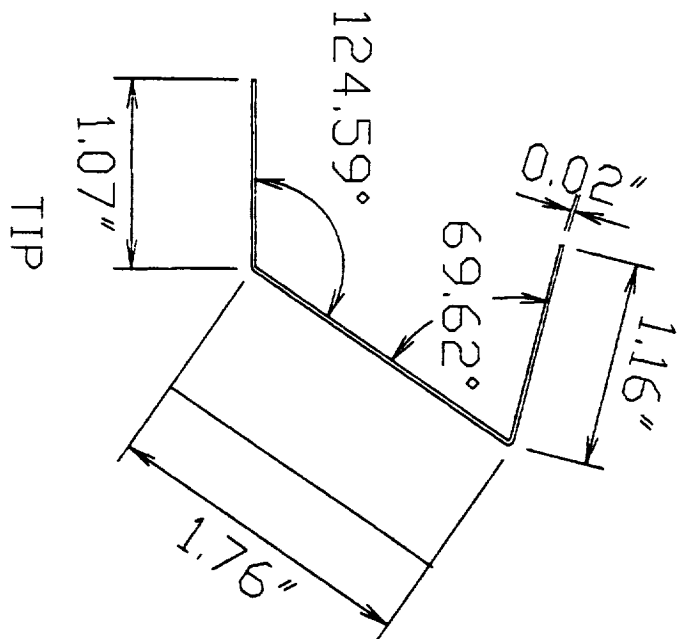
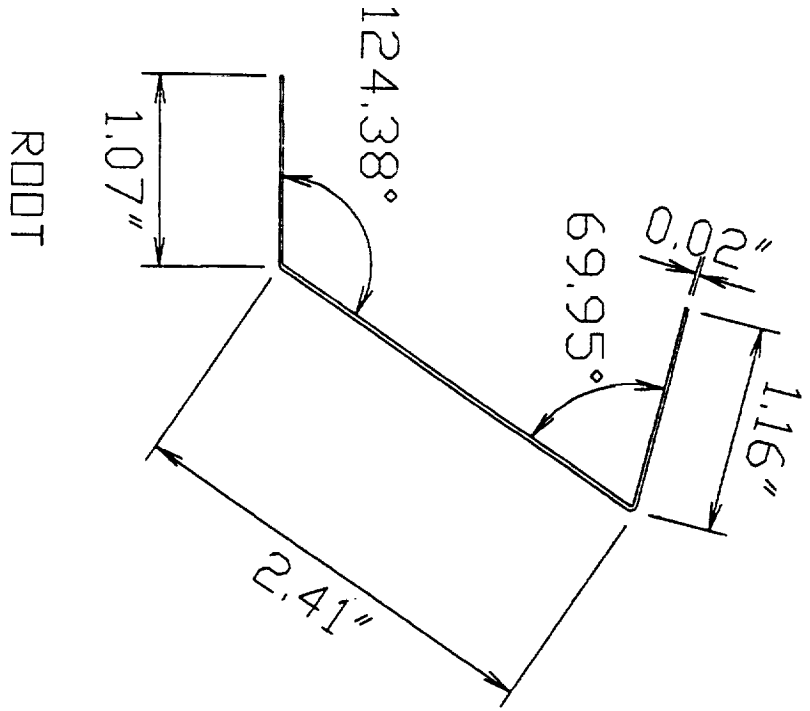


ITEM	QTY	DESCRIPTION	MATERIAL	DR PART #
6,7	1,47			5,30
5	1,52			5,59
4	1,76			6,44
3	1,98			7,30
1,2	2,05			7,58

ITEM QTY	DESCRIPTION	MATERIAL	DR PART #
7	1 Aileron Rib #5 RH	AL 2024-T3	
6	1 Aileron Rib #5 LH	AL 2024-T3	
5	2 Aileron Rib #4	AL 2024-T3	
4	2 Aileron Rib #3	AL 2024-T3	
3	2 Aileron Rib #2	AL 2024-T3	
2	1 Aileron Rib #1 LH	AL 2024-T3	
1	1 Aileron Rib #1 RH	AL 2024-T3	

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SIZE .XX ± .01 XXX ± .001	DATE 4/14	SCALE 1/1	DRAWN BY Kenneth Adriano
TITLE Aileron Ribs		DRAWING NO AL 421-93-02-10	
± 1/2°		SHEET of	

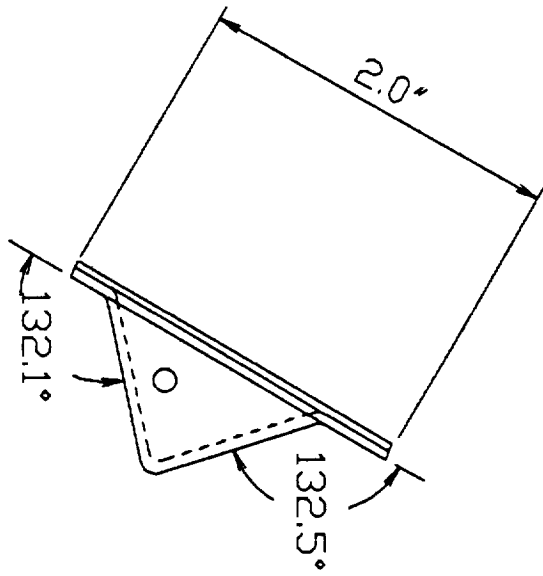
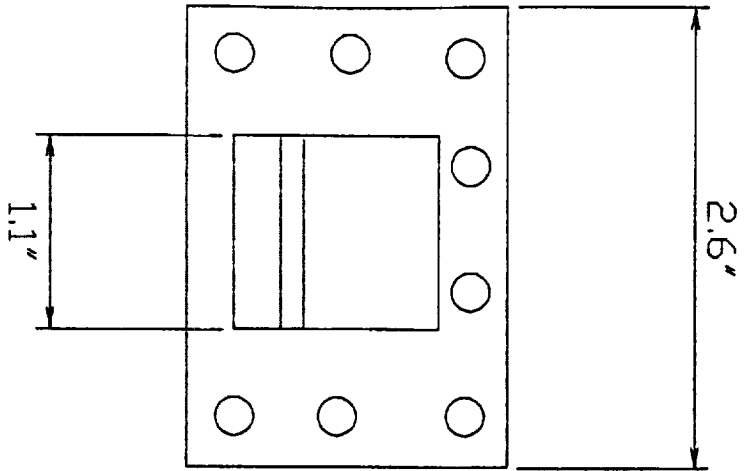
Figure 15



Material AL 7075-T6

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL .XX ± .01 .XXX ± .001 ANGULAR ± 1/2°			
EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
SIZE	DATE	SCALE	DRAWN BY
A	4/14	1/1	Kenneth Adrian
TITLE			
Aileron SPAR			
DRAWING NO.			SHEET
AE 421-93-02-11			OF

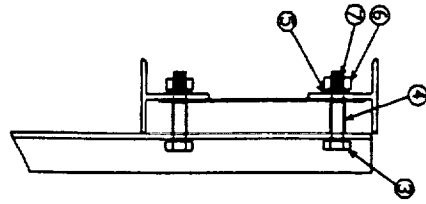
Figure 16



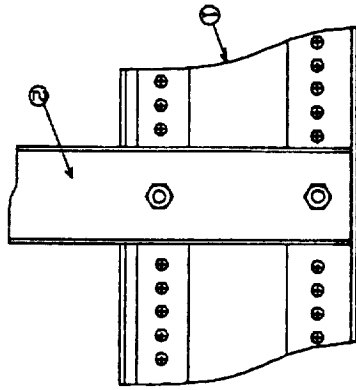
Material 7075-T6

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL		EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA	
.XX	± .01	SIZE	DATE
.XXX	± .001	A	4/14
ANGULAR ± 1/2°		SCALE	1/1
		DRAWN BY	Kenneth Adrian
TITLE			
PUSH ROD BRACKET			
DRAWING NO.			SHEET
AE 421-93-02-12			of

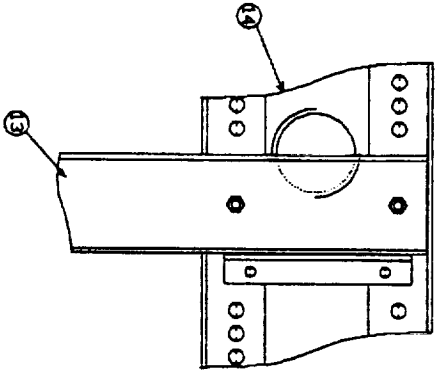
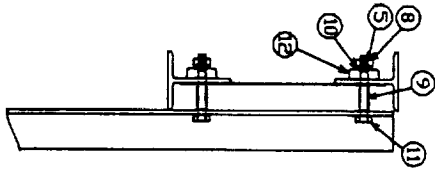
Figure 17



Front Spar Attachment



Rear Spar Attachment



ITEM QTY	DESCRIPTION	MATERIAL OR PART #
1	1 Front Spar	AE421-93-02-04
2	1 Front Fuselage Frame	Table A316 sect 13
3	4 Frt Attachment Bolt	ANS-15
4	4 Spacer	3/8x.028x3/4-C
5	4 Washer	ANS60-516
6	4 Nut, Castellated	ANS10-5
7	4 Cotter Key	MS24665-132
8	4 Nut, Castellated	ANS10-3
9	4 Spacer	1/4x.028/5/8 - C
10	4 Washer	ANS60-10
11	4 Rear Att. Bolt	ANS-14
12	4 Wedge Washer	AE421-93-02-08
13	1 Rear Spar	AE421-93-02-05
14	1 Rear Fuselage Frame	Table A316 sect 13

SIZE .XX ± .01 DATE 4/14/93 SCALE 1/2 DRAWN BY M. Zolme TITLE Wing Attachment Support	EMERY-RIDBLE AERONAUTICAL UNIVERSITY MAYTOWN BEACH FLORIDA
DRAWING NO. AE421-93-02-07 SHEET 7 OF 13	1/20

thick. The ribs, which are the same thickness as the leading edge caps, are formed by the same process. The main wing skins are standard sheets of aluminum as mentioned in the previous sections. The rivet holes are drilled in these sheets the same as they are in the leading edge skins.

5.2 Front and Rear Spars

5.2.1 Front Spar

The spar caps and vertical stiffeners are formed from a standard extrusion to simplify construction. The caps are then drilled to accommodate rivets. The web is trimmed by use of a shear to the proper shape, while the stiffeners are cut to their proper height by sawing. Rivet holes are also drilled through the web and stiffeners to allow for attachment. Access to the interior of the wing is provided through lightening holes in the spar web.

5.2.2 Rear Spar

The spar cap is constructed from extruded 7075-T6 Al. This piece is then drilled to allow for riveting the web to the spar caps. The vertical stiffeners are made of a standard extrusion, and are cut to their proper height by sawing. Rivet holes are drilled through the stiffeners and web to allow them to be attached together. The web is a standard thickness Al sheet which is shaped using a shear. Access to the interior of the wing is provided by lightening holes in the shear web.

5.3 Ailerons

The skin of the ailerons follows the same manufacturing processes as the wing skin. The ribs in the aileron are

constructed of 2024-T3 which was hydropressed into form with the lightning holes stamped in the proper locations. The C-channel which forms the aileron spar is a custom extrusion of 7075-T6.

6.0 Cost Summary

Item	Cost Merit
Wing Skin	4.2
Front Spar	3.99
Rear Spar	4.75

Table 6

6.1 Wing Skin

The wing skin cost merit was calculated for one method of production, because the only real machining done to it is the drilling process. It is estimated that 3060 rivets are needed to fasten the upper and lower skins to both wings. The drilling process will remove approximately 86.5 in³ from the total volume of aluminum.

7.0 Weight Summary

Table

Part	Weight Estimate
Wing Skin	88.5 Lbs
Wing Stringers, Ribs	22.5
Front Spar	68.5
Rear Spar	57.6
Aileron	14.7
Total	251.8

Table 7

The estimated weight of 251.8 lbs stays within the specified limit of 259 lbs given in the Statement of Work.

8.0 Summary

The calculations and drawings which were done in this report indicate that the proposed design is a usable system which will allow the aircraft to function properly for a minimum cost due to manufacturing manufacturing and maintainence. Some slight modifications were necessary with repect to the original design, but all major systems are the same as originally proposed. The calculations, as seen in this report, indicate that the system designed here not only conforms to FAR Part 23 requirements but also to the Statement of Work 2 requirements. The drawings, included, show that the structural systems deisnged will fit into the space available with no interface interference between structure and control linkages. Therefore, the consensus of the deisgn team is that this is a workable and reliable design of the main wing need for the Triton aircraft.

Appendix A

Calculations for the C-spar in the Aileron

bo1= 1.07 in
bo2= 1.767 in
bo3= 1.16 in
theta2= 34.4 deg 0.60039 rad
theta3= 14.2 deg 0.24784 rad
Thickne 0.032 in

	AREA	X	Y
SECT1	0.03424	0.535	0.016
SECT2	0.05654	1.569	0.729
SECT3	0.03712	1.506	1.600

centroid x 1.27398
centroid y 0.79098

Moment of inertia in X1 0.0187
Moment of inertia in X2 0.01671
Moment of inertia in Y2 0.00067
Moment of inertia in XY2 -0.0012
Moment of inertia in X2' 0.01267
Moment of inertia in X3 0.08222
Moment of inertia in Y3 0.09015
Moment of inertia in XY3 0.08408
Moment of inertia in X3' 0.04271

MOMENT OF INTERIA ABOUT X 0.07408

Stress due to bending

sigma= $My/I=$ 25371.6
y= 0.667
I= 0.07408
Mdown= 2817.8

Calculations for the C-spar in the Aileron

bo1= 1.07 in
bo2= 1.767 in
bo3= 1.16 in
theta2= 34.4 deg 0.60039 rad
theta3= 14.2 deg 0.24784 rad
Thickne 0.025 in

	AREA	X	Y
SECT1	0.02675	0.535	0.013
SECT2	0.04418	1.569	0.729
SECT3	0.029	1.506	1.600

centroid x 1.27398
centroid y 0.79004

Moment of inertia in X1 0.01461
Moment of inertia in X2 0.01311
Moment of inertia in Y2 0.0005
Moment of inertia in XY2 -0.0009
Moment of inertia in X2' 0.00992
Moment of inertia in X3 0.06424
Moment of inertia in Y3 0.0692
Moment of inertia in XY3 0.06509
Moment of inertia in X3' 0.03358

MOMENT OF INTERIA ABOUT X 0.05811

Stress due to bending

sigma= $My/I=$ 32387.7
y= 0.66794
I= 0.05811
Mdown= 2817.8

Calculations for the C-spar in the Aileron

bo1= 1.07 in
bo2= 1.767 in
bo3= 1.16 in
theta2= 34.4 deg 0.60039 rad
theta3= 14.2 deg 0.24784 rad
Thickne 0.02 in

	AREA	X	Y
SECT1	0.0214	0.535	0.010
SECT2	0.03534	1.569	0.729
SECT3	0.0232	1.506	1.600

centroid x 1.27398
centroid y 0.78937

Moment of inertia in X1 0.01169
Moment of inertia in X2 0.01052
Moment of inertia in Y2 0.00038
Moment of inertia in XY2 -0.0007
Moment of inertia in X2' 0.00795
Moment of inertia in X3 0.05139
Moment of inertia in Y3 0.05466
Moment of inertia in XY3 0.05172
Moment of inertia in X3' 0.02699

MOMENT OF INTERIA ABOUT X 0.04663

Stress due to bending

sigma= $My/I=$ 40406
y= 0.6686
I= 0.04663
Mdown= 2817.8