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AIRCRAFT EMPENNAGE STRUCTURAL DETAIL DESIGN

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1.0 PROJECT SUMMARY

The purpose of this project is to provide an empennage structural assembly that will withstand the operational loads defined in FAR Part 23, as well as those specified in the statement of work i.e. snow, rain, humidity, tiedown forces etc. The goal is to provide a simple yet durable lightweight structure the will transfer the aerodynamic forces produced by the by the tail surfaces through the most efficient load path to the airframe. The structure should be simple and cost-effective to manufacture and repair. All structures meet or exceed loading and fatigue criteria. The structure provides for necessary stiffness and ease of maintenance. The following tables represent a summary of the critical detail design parts.

Components	f_{max} (psi)	$M.S._{UL}$	Page Number
Vertical Tail Assembly			
Spar Stringer	5059	.16	9
Hinges	4938	2.75	10
Skin Buckling	479	3.83	8
Rudder skin	2222	4.33	12
Vertical Stabilizer Stringer	8139	2.09	13

Table # 1.1.1: Vertical Tail Assembly Stresses

Components	f_{max} (psi)	M.S.$_{UL}$	Page Number
Horizontal Tail Assembly			
Skin Panels	350	.7	15
Hinges	4938	2.75	15
Front Spar	3900	.05	16
Rear Spar	1800	.25	16
Ribs	1546	1.01	16

Table # 1.1.2: Horizontal Tail Assembly Stresses

Components	f_{max} (psi)	M.S.$_{UL}$	Page Number
Tailcone Assembly			
Stringer	102.3	.04	20
Skin	5817	.17	19
Tie Down	25000	23.5	23

Table # 1.1.3: Tailcone Assembly Stresses

2.0 DESCRIPTION

2.1 Vertical Tail

The vertical tail assembly provides a lightweight stiff structure that will maintain the proper airfoil shape under the applied limit loads. The aerodynamic load is a distributed load on the surface of the skin that provides, shear torsion and bending moment at the fuselage interface. The vertical tail assembly uses flat wrapped skin and brake formed spars fastened with aircraft type rivets. All sheet metal and rivets are 2024-T3 Aluminum. The skin has been beaded in order to eliminate panel buckling, and reduce the rib weight. The vertical tail assembly has been designed for rapid removal for ease of repair. The fuselage interface consists of two spar mounted castings, that bolt to the fuselage bulkhead. The fuselage interface skins provide a fillet between the vertical tail and the empennage. These are secured using AN 366DF632 nut plates. The vertical tail assembly can be installed or removed in minutes, by simply unscrewing the fillet skin, unbolting the torque tube from the control push-pull rod, and unbolting the mount castings from the fuselage bulkheads. A tip mounted plastic housing contains a rotating beacon and white tail position light. These and all fasteners are vendor supplied and approved.

2.2 Horizontal Tail

The horizontal tail assembly was designed using the loading conditions specified in FAR Part 23. This structure places a an aerodynamic distributed load on the skin surface, that produces shear and bending moments at the fuselage interface. The structure is composed of front and rear spars, four ribs, hinges for the trim tab and elevator assembly, and attachment plates at the interface. No lighting is required on the horizontal tail assembly. The spars and ribs will be riveted together using universal rivets that are aircraft certified. The control surfaces will then be attached followed by the final phase of skin wrapping. This tail structure was designed with maintenance and ease of removal in mind. The horizontal tail assembly can be removed independently from the vertical tail. All parts used to manufacture the horizontal tail assembly are vendor supplied and certified.

2.3 Tailcone Assembly

The tail cone assembly for the Viper aircraft, consists of thirteen stringers, 6 frames (one aluminum casting frame), skin and rivets. The tailcone assembly is designed is such a way as to join the main fuselage and the main fuselage and tailcone on one large frame. The skins of both the main fuselage and tail assemblies will then be riveted to the frame. The vertical and horizontal tail assemblies will then be attached to the first three frames beginning from the point of the tailcone. The first frame is going to be a cast aluminum part due to the strength requirements. It will have a small hoop molded to into the bottom which will act as a tie down. The stringers at the tailcone will be stepped down in height in between the second and third frames. However, the cross sectional area of the stringers will be maintained by increasing the thickness. The rearward stringers will also be attached to the cast frame by L-shaped brackets.

3.0 Loads and Loading

3.1.1 Vertical Tail Section

The loads in the vertical tail are a result of the distributed load acting on the surface. This load will provide a shear load V_y, bending moment M_x and a torsional component M_z. The distributed load can be represented as a 393 lb resultant force acting at the MAC. The chordwise location of this force will vary according to the gust or maneuvering case. The load paths to react these forces and moments are the skin, spar flanges and spar webs.

$V_y = 393$ lbs $\quad M_x = 6428$ in-lbs $\quad M_z = 2468$ in-lbs

3.1.2 Rudder

The rudder also has a distributed load that acts at the MAC and can be represented by a point load of 160 lbs. This is reacted externally by the hinges and torque tube. It is reacted internally by the skin, spar flanges and spar web. The majority of the load passes through the middle hinge. The hinge reactions are as follows.

$\Sigma M = 0 \quad \Sigma F = 0$

$R_1 = 28.25$ lbs $\quad R_2 = 82.3$ lbs $\quad R_3 = 49.7$ lbs

The torque tube reacts a torsion of 191.9 in-lbs.

The shear flow in the skin is assumed to react the entire torsion M_z.

$q = T/2A = 6.38$ lbs/in

The spar flanges are assumed to react the entire bending moment M_x.

M_x = 3006 in-lbs

The spar web is assumed to react the entire shear V_y.

V_y = 160 lbs

3.2 Horizontal Tail Assembly

The Viper horizontal tail assembly has two major parts, the elevator and horizontal stabilizer structures. The skin provides a loadpath for the torsional loads created by the horizontal tail structure, while the spare react the bending moment. Each of these structures are composed of ribs, spars, skins and fasteners, which were analyzed using the two loading conditions, A and B, specified in FAR Part 23. For condition A, the resultant distributed load, located at .56c was calculated to be 78.98 psf, while the resultant load for condition B, was located at .25c with a magnitude of 39.98 psf. Each of these loads were analyzed using the same methods described in the vertical tail assembly loading section.

Using the height of the airfoil and an initial thickness of t = 20, the moment of inertia was determined. This value was then iterated until acceptable results were obtained. The skin thickness was determined to be t= 0.20 using clad 2024-T3 Aluminum. The front and rear spars are .55 and .14 in thick respectively. The front spar has both an inboard and outboard portion for reinforcement. See horizontal tail assembly drawing for reference figures. All spars are C-channels.

3.3. Tailcone

This analysis is divided into two categories:

1. Bending
2. Torsion

The first topic discussed is the bending. It is assumed that all bending is reacted by the fuselage stringers. The bending is produced by the lift on the horizontal and vertical tails, with the worst case being 100% lift on each lifting surface. This results in a moment M_y, which is reacted by the axial loads in the stringers. The total lift is 1152.9 lbs. This is multiplied by the length of the fuselage at each section, with the worst case being the cabin interface. These moments are as follows:

$$M_y = (\text{Lift x Arm}) = (1039.5)(128) = 47572 \text{ in-lbs}$$

$$M_z = (444.6)(107) = 133056 \text{ in-lbs}$$

The worst case torsional load occurs with 100% lift on the vertical tail and a 100%/65% lift on the horizontal tail. This is a uniform load throughout the length of the fuselage. The torsion is solved in the following equation.

$$\Sigma M_x = 0 \qquad 630*29 - 409*29 + 444.6*26 = 17954 \text{ in-lbs}$$

4.0 STRUCTURAL SUBSTANTIATION

4.1 Vertical Tail Assembly

4.1.1 Rudder

The following tables and calculations substantiate that the skin, spars, webs, torque tube and hinges will not buckle under the loadings listed in section 3.0 of this document. The first area of discussion is the vertical tail skin. The skin is assumed to carry all of the torsional load M_z. Panel buckling is the criteria used for substantiation. The skin has been beaded in order to reduce the panel area.

Skin			
$f_{shearLL}$ (psi)	$f_{shearULT}$ (psi)	F_{shear} (ksi)	$M.S._{LL}$
319.2	478.8	38.0	118

Table 4.1.1: Vertical Tail Skin Characteristics

Buckling		
f_{crit} (psi)	$M.S._{LL}$	$M.S._{UL}$
2311	6.24	3.83

Table 4.1.2: Buckling Results

Spars					
	f_{LL} (psi)	f_{ULT} (psi)	f_{crit} (psi)	$M.S._{LL}$	$M.S._{UL}$
Stringers	2906	4358	5059	.74	.16
Webs	796	1193	N/A	46.7	30.8

Table 4.1.3: Spar Results

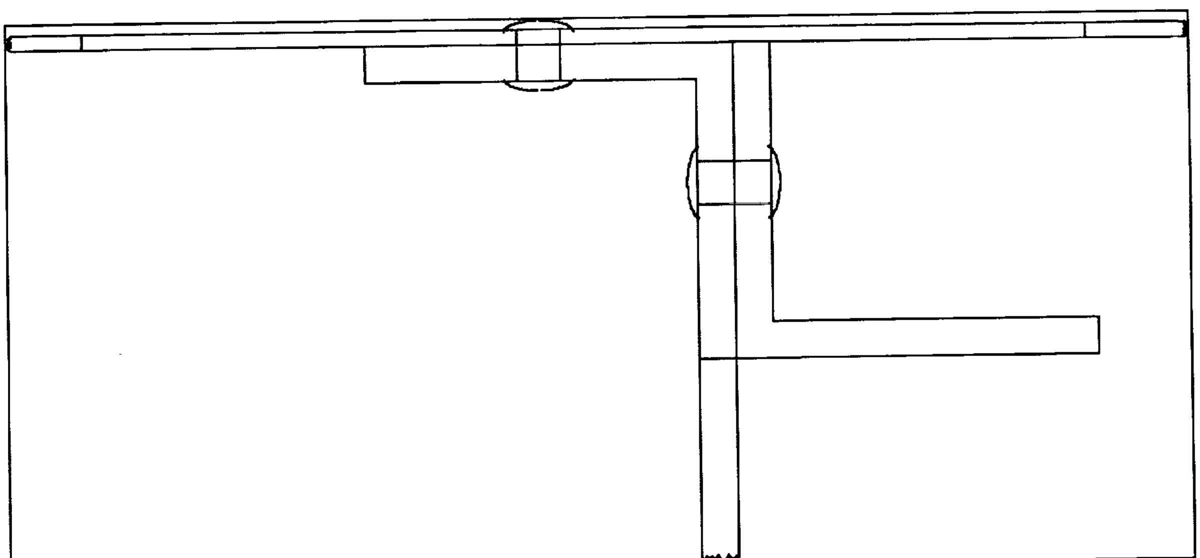
Figure 4.1.1.2: Stringer Cross Scetion

The torque tube provides the external reaction to the torsion M_z. The tube mount is riveted to the root rib of the rudder. The critical stress in bearing stress due to the rivets.

Torque Tube			
f_{LL} (psi)	f_{ULT} (psi)	$M.S._{LL}$	$M.S._{UL}$
2317	3475	9.9	15.4

Table 4.1.4: Torque Tube Results

Tube Mount			
f_{rivet} (psi)	f_{sheet} (psi)	$M.S._{LL}$	$M.S._{UL}$
429	1147	14.6	9.4

Table 4.1.5: Tube Mount Results

The hinges react the shear V_y. The resultant of the distributed load acts approximately directly on the center hinge, therefore it is assumed to carry the majority of the load, and is used for substantiation.

Hinges			
$f_{brg\ LL}$ (psi)	$f_{t.o.}$ (psi)	$M.S._{LL}$	$M.S._{UL}$
4938	823	4.61	2.75

Table 4.1.6: Hinge Results

Fatigue Analysis					
	f_{LL} (psi)	f_{max} (psi)	f_m (psi)	f_{min} (psi)	f_{alt} (psi)
Torque Tube	2317	1186	527	-105.3	632
Skin	319	157	72.5	-14.5	87
Hinges	6584	3292	1496	-299	1796

Table 4.1.7: Fatigue Analysis

The f_{alt} results were checked with figure 15.4.5 in Aircraft Structural Analysis by Michael Niu, to assure proper lifetime.

4.1.2 Vertical Stabilizer

The vertical stabilizer will be analyzed for shear flow in the skin, shear in the spar webs, and axial stress in the spar flanges. Analysis is performed at a section where the vertical tail interfaces with the fuselage. This is the area of greatest shear and bending moment.

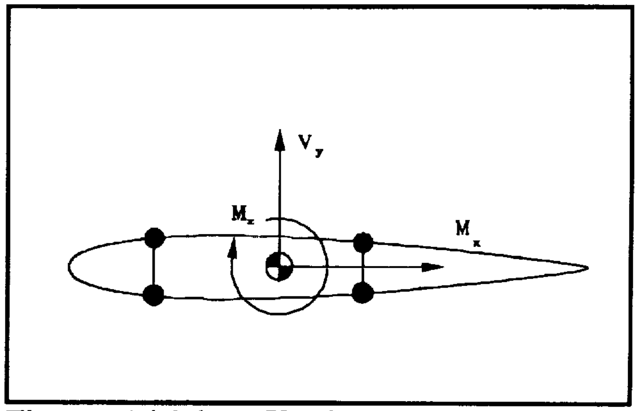

Figure 4.1.2.1: Vertical Stabilizer Cross Section

Shear Flow

Shear stress is found by dividing the average shear flow in the skin by the skin thickness.

$$f_{shear} = 417 \text{ psi}$$

The criterion for substantiation of the skin is panel buckling. It is necessary to utilize beading, to reduce panel area. The beads are spaced three inches apart.

$$f_{crit} = K_s E(t/b)^{\wedge 2} = 2222 \text{ psi}$$

$$M.S._{skin} = 4.33$$

Spar Flange Stress

The bending moment M_x is reacted by the idealized stringers composed of the spar flanges, stiffeners and skin. The flexure formula is used to determine the axial stress in the flanges.

$$f_z = \frac{1}{I_x I_y - I_{xy}^{\wedge 2}}[-(I_y M_x + I_{xy} M_y)y + (I_x M_y + I_{xy} M_x)x]$$

f_{zmax} = 2629 psi

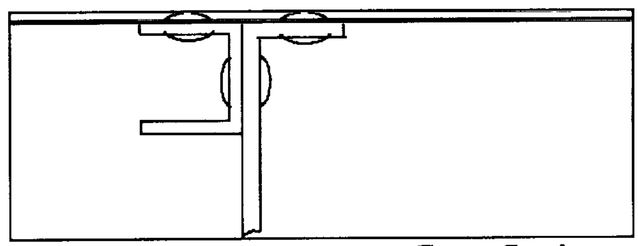

Figure 4.1.2.2: Spar Flange Cross Section

Axial Buckling

The worst case for axial stress is on the forward spar. The critical stress will therefore be evaluated for these stringers.

$f_{crit} = \pi^2 EI/l^2 A$ = 8139 psi

M.S. = 2.09

Stringer Fatigue Analysis

$S_{max} = f_{LL}$*.5 = 3668 psi

$S_m = f_{LL}/4.4$ = 1667 psi

Using figure 15.4.5 from Aircraft Structural Design by Michael C. Niu, the stringers stress falls below the endurance limit, meeting our service life criterion.

Spar Web Shear Analysis

The shear stress in the spar web that reacts the shear load V_y was found to be 1965 psi. From this the ultimate margin of safety was calculated to be 11.9.

Spar Mount Casting

The front and rear spar mount castings, bolt to the fuseluge bulkheads with AN4 bolts. The four bolt holes are spaced three inches apart in a square pattern. It is assumed that each bolt shares the same load. The maximum stress is bearing stress, which is located in the aft casting.

$f_{brg} = P/Dt = 2509$ psi $\qquad M.S._{UL} = 4.31$

4.2 Horizontal Tail Assembly

The horizontal tail assembly was analyzed for shear flow in the skin, ribs and for axial stresses. Analysis was performed at various sections including the interface with the fuselage. The objective if this portion of the analysis was to assure that the spars, ribs skin and hinges would not buckle under the loadings listed in section 3.2 of this report. As the horizontal tail calculations are extremely similar to those of the vertical tail, not all calculations will be repeated.

Shear Flow

The loads over each panel were assumed to be average.

f_{crit} = 350 psi $\quad$ f_{yield} = 595 psi $\quad$ $M.S._{ULT}$ = .7

The K_s values were determined using figure 15.4.6, in Aircraft Structural Dynamics by Michael C. Niu. At t = .02 the panels failed and thus the thickness was increased to .025 to supply a higher tolerance for manhandling. The shear in the leading edge panels were determined using similar analysis, with the exception of figure 15.4.7 in Niu. The front spar has both an outboard and an inboard section whereas the rear spar has only an outboard portion. The results were satisfactory.

The elevator and rudder were designed simultaneously to provide for consistency in parts and maintainability. The elevator is composed of four ribs and is attached to the horizontal stabilizer by means of two hinges. Refer to horizontal tail assembly drawings for detail geometry. The values determined for elevator are shown below.

f_{crit} = 409 psi f_{yield} = 750 psi $M.S._{ULT}$ = .83

The horizontal tail is bolted to the fuselage in the same was as the rudder, for several reasons.

1. Ease of Maintenance
2. Ease of Purchasing Repeated Parts
3. Ease of Removal

At the interface, the horizontal tail will be bolted to the fuselage using a casting and bolts.

Fatigue Analysis

Consideration was given to the service life of the horizontal tail structure as per the statement of work. The following calculations substantiate that the horizontal tail assembly will meet the twenty year service life and exceed the endurance level.

S_{max} = 17500 psi

S_m = 7940 psi

According to figure 15.4.5 in Niu, the horizontal tail assembly meets the endurance criterion.

4.3 Tailcone

This analysis is divided into three categories:

1. Buckling of the tail cone skin
2. Buckling of the stringers
3. Tie down forces

The first topic discussed is the buckling of the tail cone skin. Figure # 4.3.3.1 represents an idealized tailcone cross section.

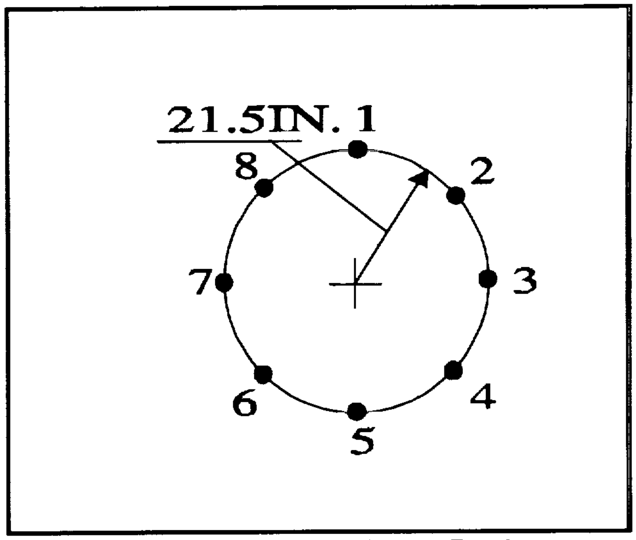

Figure 4.3.3.1: Tailcone Cross Section

The following table summarizes the values used to calculate shear loads, stringer forces, shear flow and buckling.

I_y (in^4)	I_z (in^4)	I_{yz} (in^4)	τ_w (in)	A_s (in^2)	V_y (lb)	V_z (lb)	b	K_s	E	r (in)
27.19	18.12	0	.032	.112	-1040	444.6	14.1	88	10E6	9

Table # 4.3.3.1: Tailcone Cross Section Characteristics

Stringer Force Calculations

$$P_{f1}' = \frac{1}{I_y I_z - I_{yz}^{\wedge 2}}[(I_y V_y - I_{yz} V_z)y + (I_z V_z - I_{yz} V_y)z](A_f)$$

$P_{f1}' = -29.22\ \text{lb/in}^{\wedge 2}$

Table 4.3.3.2 summarizes the stringer force calculations.

Stringer Forces (lb/in^2)	Stringer Force # 1	Stringer Force # 2	Stringer Force # 3	Stringer Force # 4	Stringer Force # 5
	-29.22	52.51	57.83	29.22	52.51

Table # 4.3.3.2: Stringer Force Calculations

Shear Flow Calculations

$q_{out} = p_f' + q_{in}$ thus,

$q_1 = -29.22 + q_0$

$q_2 = 52.51 + q_1$

$q_3 = 57.83 + q_2$

$q_4 = 29.22 + q_3$

$q_5 = 52.51 + q_4$

$\Sigma T = 2A_i q_i$ where $T = 17954$ in-lb

Combining the shear flow calculations and the torque equations, the shear flow can be calculated and are depicted on the following page.

Shear Flow (lb/in)	Stringer # 0	Stringer # 1	Stringer # 2	Stringer # 3	Stringer # 4
	124.1	-67.97	-15.46	42.37	71.59

Table # 4.3.3.3: Shear Flow Results

Shear Buckling

f_{maxLL} (psi)	f_{UL} (psi)	f_{crit} (psi)	$M.S._{LL}$	$M.S._{UL}$
3878	5817	4777	.232	.17

Table # 4.3.3.4: Skin Margins of Safety

The following calculations describe the buckling characteristics of the tailcone stringers. Refer to figure 4.3.3.2, for a general description.

Table 4.3.3.5, located on the following page, represents the values used to calculate the axial loads, moments and buckling loads.

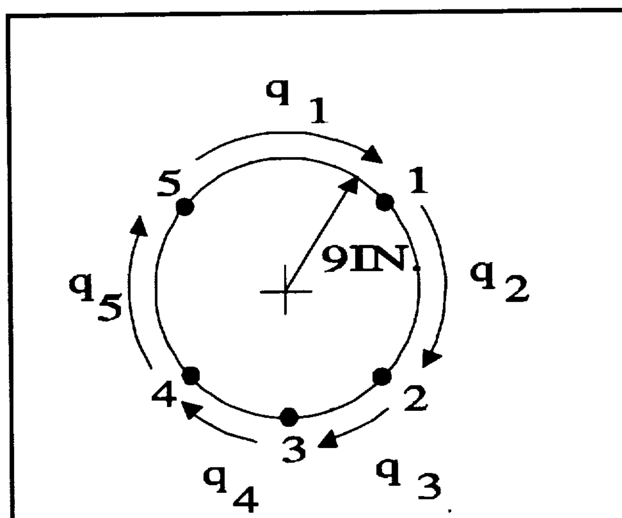

Figure 4.3.3.2: Tailcone Stringers

I_y (in^4)	I_z (in^4)	I_{yz} (in^4)	M_y (lb-in)	M_z (lb-in)	E (psi)	A (in^2)	L (in)	I (in^4)
5.3	155.3	0	47572	133056	10.6E6	.112	137	.015

Table # 4.3.3.5: Tailcone Cross Section Characteristics

Axial Stresses

$$f = P/A + \frac{1}{I_y I_z - I_{yz}^2}[-(I_y M_z + I_{yz} M_y)y + (I_y M_y + I_{yz} M_z)z]A_f$$

Table 4.3.3.6 summarizes the axial loading results.

Axial Stress (lb/in^2) Stringer #	# 1	# 2	# 3	# 4	# 5	#6	#7	# 8
	18420	-8366	6580	17679	18420	8366	-6580	17679

Table # 4.3.3.6: Axial Stress Results

Axial Buckling

f_{maxLL} (psi)	f_{UL} (psi)	f_{crit} (psi)	$M.S._{LL}$	$M.S._{UL}$
18420	27360	21240	.143	.04

Table # 4.3.3.7: Stringer Margins of Safety

Stringer Fatigue Analysis

f_{LL} = 18420 psi

$f_{UL} = f_{LLmax} * 1.5$

F_{yield} = 45000 psi

f_{UL} = 27630 psi

F_{ULT} = 60000 psi

$f_{LL} \leq F_{yield}$

$f_{ULT} \leq F_{ULT}$

thus,

$f_{mean} = f_{avg} = f_{LL}/4.4$ → f_{mean} = 4186 psi

$f_{max} = .5\ f_{LL}$ → f_{max} = 9210 psi

$f_{min} = -f_{max} + 2f_{avg}$ → f_{min} = -838 psi

$f_{alt} = f_{max} - f_{min}/2$ → f_{alt} = 5024 psi

Using figure 15.4.5, form Aircraft Structural Design by Michael Niu, determines that the stringers will function for 10^7 cycles before failure.

Tie Down

The critical cases are defined as;

1. Straight headwind ± 10° (Elevator Lift on H-Tail)
2. 0° elevation, C_{Lmax} (More lift on the V-Tail)
3. 0° elevation, 90° (Drag on V-Tail)

Case # 1: Lift on the Horizontal Tail

τ*	A	a_0	C_L	V (ft/s^2)	S (ft)	ρ (slug/ft^3)
.003	3.86	2π	.723	176	31.1	.00238

* τ obtained from fig. 5.18 Fundamentals of Aerodynamics

Table 4.3.3.8: Horizontal Tail Characteristics

$L_{HT} = .5*\rho*V^{2}*S*C_L$

$\underline{L_{HT} = 829 \text{ lbs}}$

Case # 2: Lift on the Vertical Tail

The calculation of the lift for the horizontal tail, follows the same procedure as case # 1.

$\underline{L_{VT} = 381 \text{ lbs}}$

Case # 3: Drag on the Vertical Tail

$C_D = 2.0$ (flat plate)

$D_{VT} = .5*\rho*V^{2}*S*C_D$

$\underline{D_{VT} = 1092 \text{ lbs}}$

Shear Stress

f_{maxLL} (psi)	f_{UL} (psi)	F_{shear} (psi)	$M.S._{LL}$	$M.S._{UL}$
681	1091.5	25000	35.7	23.5

Table # 4.3.3.9: Tie Down Margins of Safety

5.0 MANUFACTURING AND MAINTENANCE

5.1 Vertical Tail

The vertical tail section of the Viper PFT is manufactured largely from 2024-T3 Aluminum sheet. The vertical stabilizer, rudder and dorsal fin use flat wrapped skin sections over brake formed sheet metal spars. The ribs, made of sheet metal, are manufactured using the hydropress formed method. The entire assembly is fastened with MS20470DD rivets. The assembly is bolted to the fuselage bulkheads through spar attached castings made from 355.0- T3 aluminum. Aircraft quality AN designation type bolts are used in the hinges and at the bulkhead mounting points. Access to the control systems and bolts, are provided by inspection plates. The entire vertical tail assembly is easily removed independently from the horizontal tail.

5.2 Horizontal Tail

The Viper's horizontal tail was designed for ease of production. The skin of the horizontal stabilizer, elevator, trim tab are flat wrapped. The front and rear spars combine both the brake forming and the flat wrapping methods. To alleviate stress in the ribs, the ribs will be notched and manufactured by brake forming and hydropressing. The horizontal tail assembly will be attached to the fuselage bulkhead structure with spar made from castings. All bolts and rivets are aircraft quality, for structural integrity. The horizontal tail was designed to detach independently from the empennage for ease of maintenance. In accordance, their are access panels available for maintenance purposes.for ease of production. The control systems will be attached before the final skin wrap.

5.3 Tailcone Assembly

The Viper tailcone will consist of thirteen stringers, 6 frames and aluminum skin. The material used in all but five of the stringers will be .032 2024-T3 Aluminum, for ease of purchasing. The five stringers are made from .063 2024-T3 Aluminum. The material has been proven in several designs as durable, lightweight and easy to use. The stringers will be cut and brake formed from sheet stock aluminum. The frame at the intersection of the tailcone and fuselage will be notched appropriately for the stringers, brake formed into a flange in shape and then contour rolled and welded at the ends. The other three frames, due to their reduction in size will be cut, notched and then pressed formed into a leg-in type shape.

The tailcone will be assembled by first using an L-shaped bracket to rivet the stringers to the frame. Then the stringer-frame assembly, will be flat wrapped and riveted. This tailcone assembly will then be attached to the main fuselage. It is important to note that the external dimensions of the main body stringers, will be equal to the internal dimensions of the tailcone stringers so that the ends of the stringers marry together at the interface. In addition, the five stringers between frames A and B, will be stepped down in height to yield additional space in the rear of the tailcone.

4.4 Production Assembly

The following diagrams depict the production assembly order.

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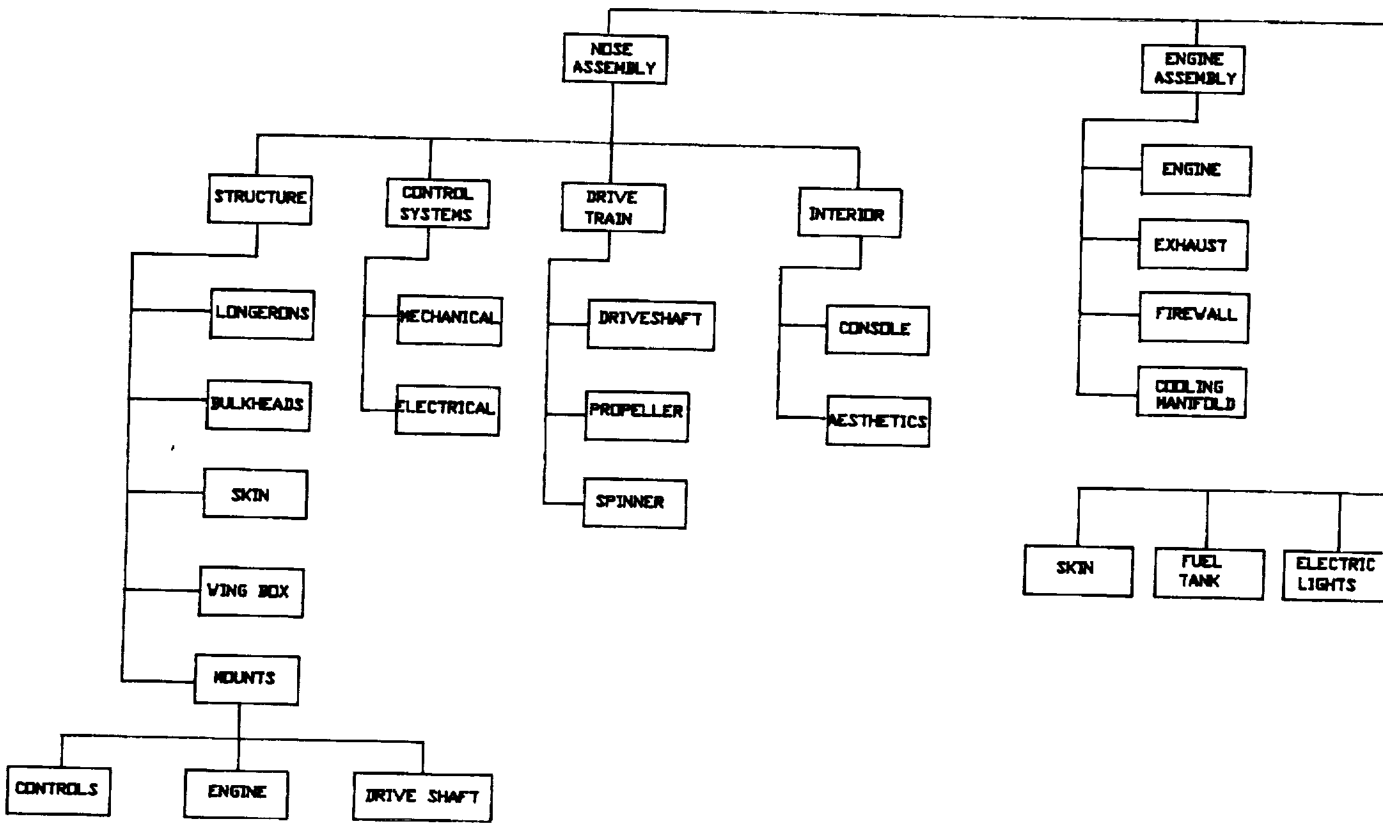

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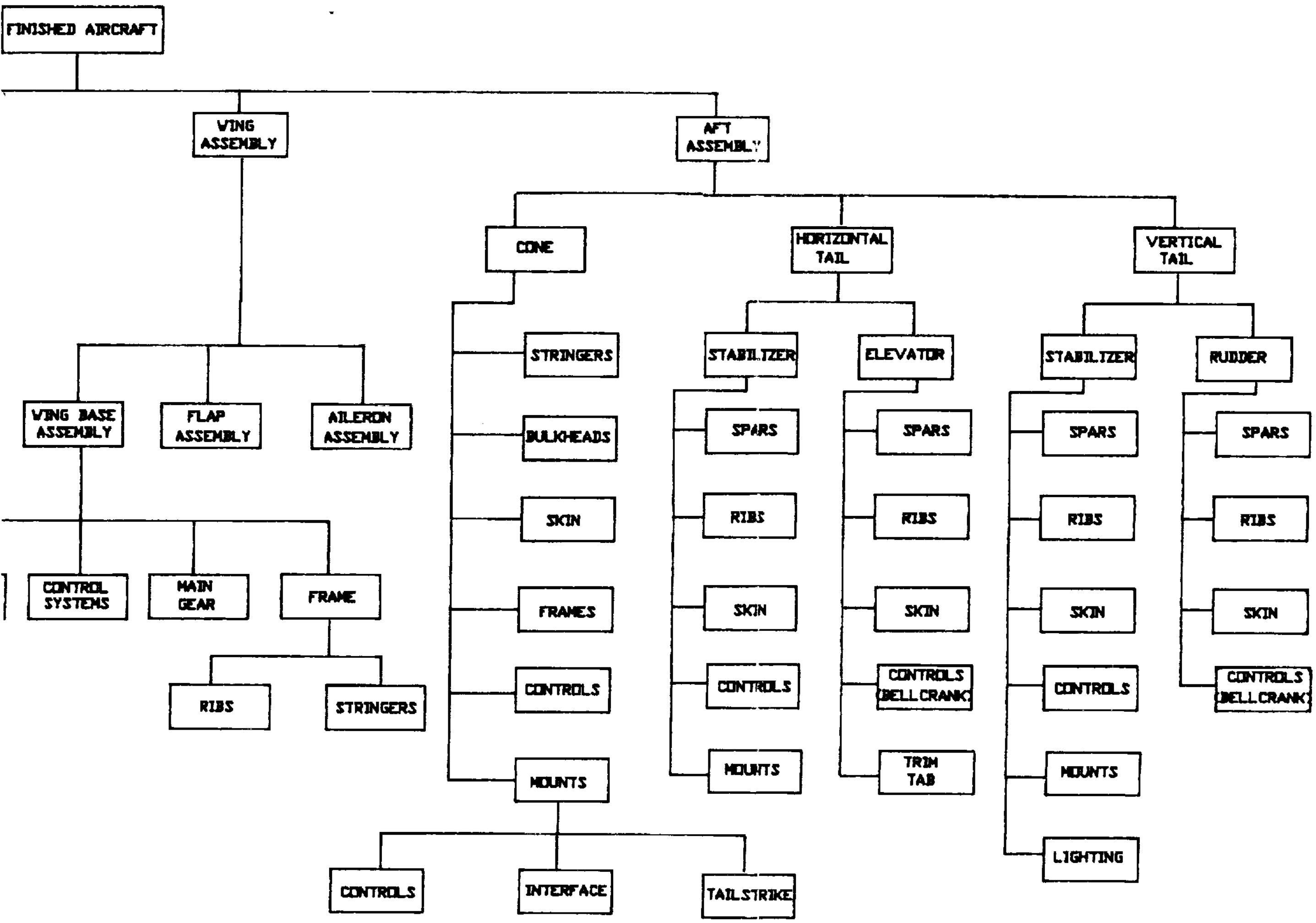

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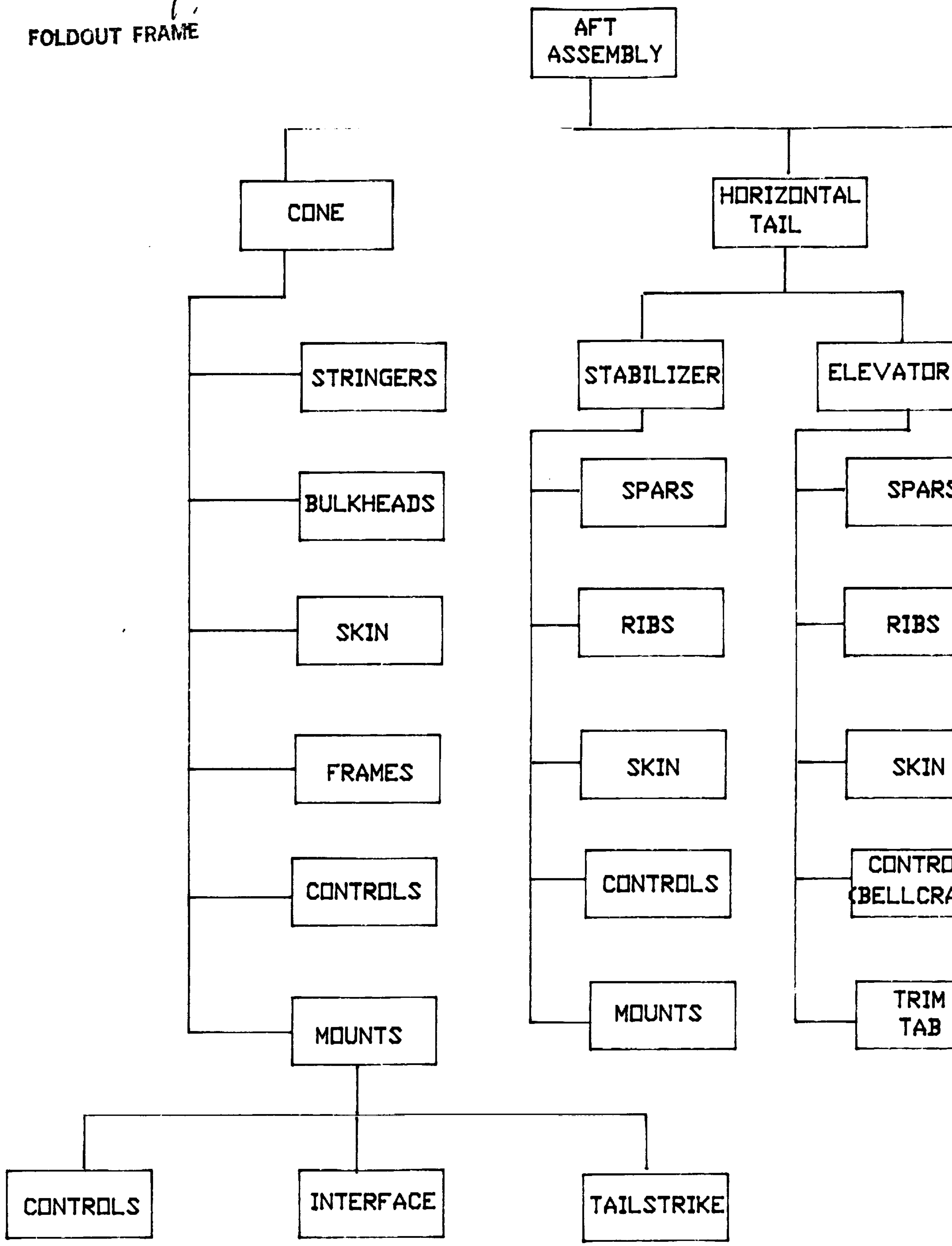

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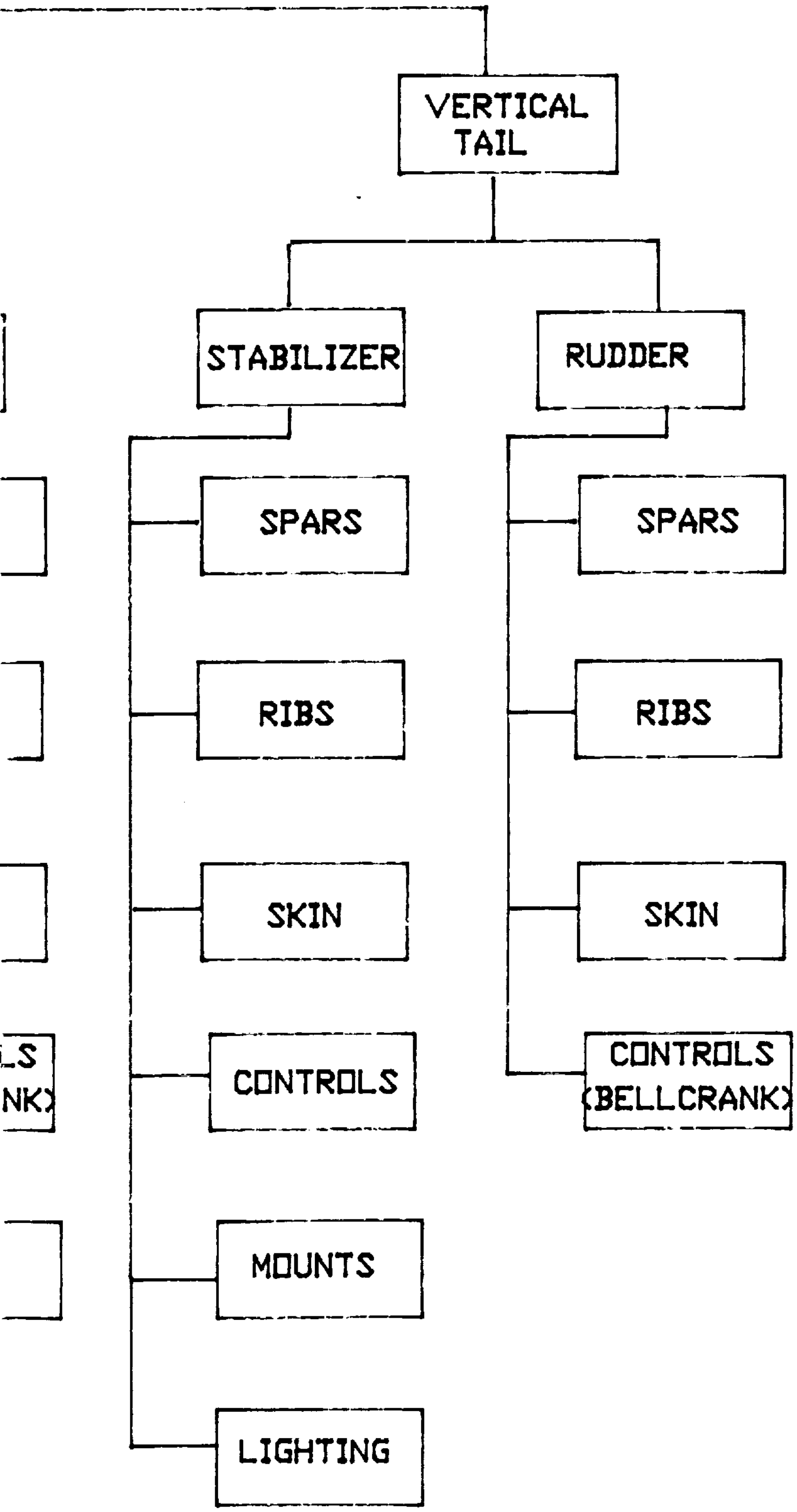


6.0 WEIGHT SUMMARY

COMPONENT	WEIGHT (LBS)
Vertical Tail Assembly	
Skin	8.6
Spars	3.9
Ribs	1.0
Castings	2.0
Torque Tube	.14
Fasteners	1.0
Total Vertical Tail Assembly Weight	16.64
Horizontal Tail Assembly	
Skin	14.8
Spars	4.2
Ribs	1.0
Castings	2.0
Fasteners	1.2
Total Horizontal Assembly Weight	23.2
Tailcone Assembly	
Skin	61.1
Stringers	2.8
Frames	7.5
Castings	1.5
Bolts and Nuts	.04
Rivets	14.8
Total Tailcone Assembly Weight	87.74
Total Empennage Weight	127.58

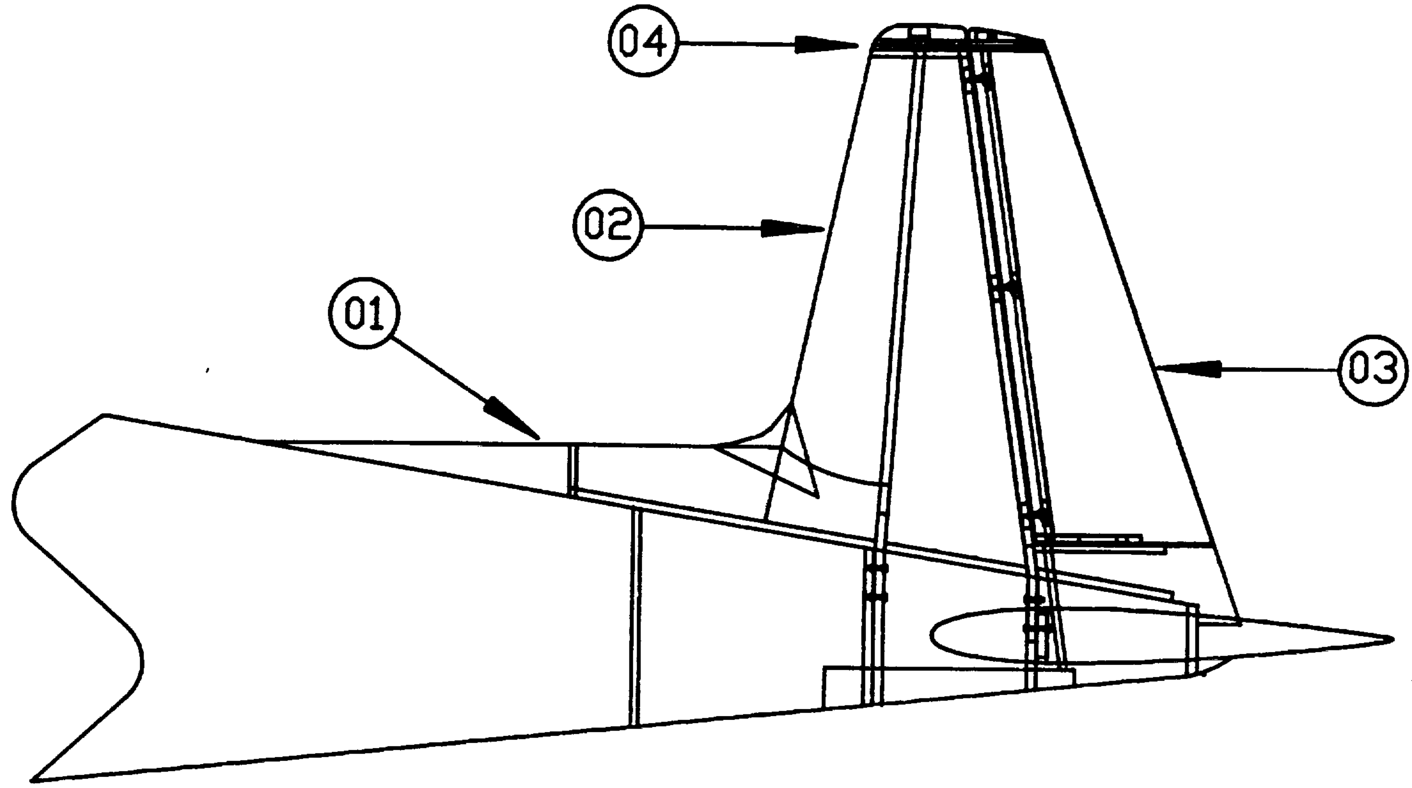
Table 6.1.1.1: Weight Summary

7.0 CONCLUSION

The goal of this design project is to provide an empennage detail design that meets the loading criterion outlined in FAR Part 23. Additional loading criterion include; snow, tie down forces of 120 mph and resistance to rain, humidity and ice. Fatigue analysis was performed to assure a service life of twenty years. In the interest of maintenance, the control surfaces have been designed for easy removal. While the design meets all requirements, further optimization, through weight reduction could be performed in later iterations.

APPENDIX

FOLDOUT FRAME



2'

FOLDOUT FRAME

ITEM	QTY	DESCRIPTION	PART#
04	01	TIP ASSY.	421S9303D234
03	01	RUDDER ASSY.	421S9303D232
02	01	VERTICAL STAB. ASSY.	421S9303D233
01	01	DORSAL ASSY.	421S9303D231

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01
.XXX ± .001

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

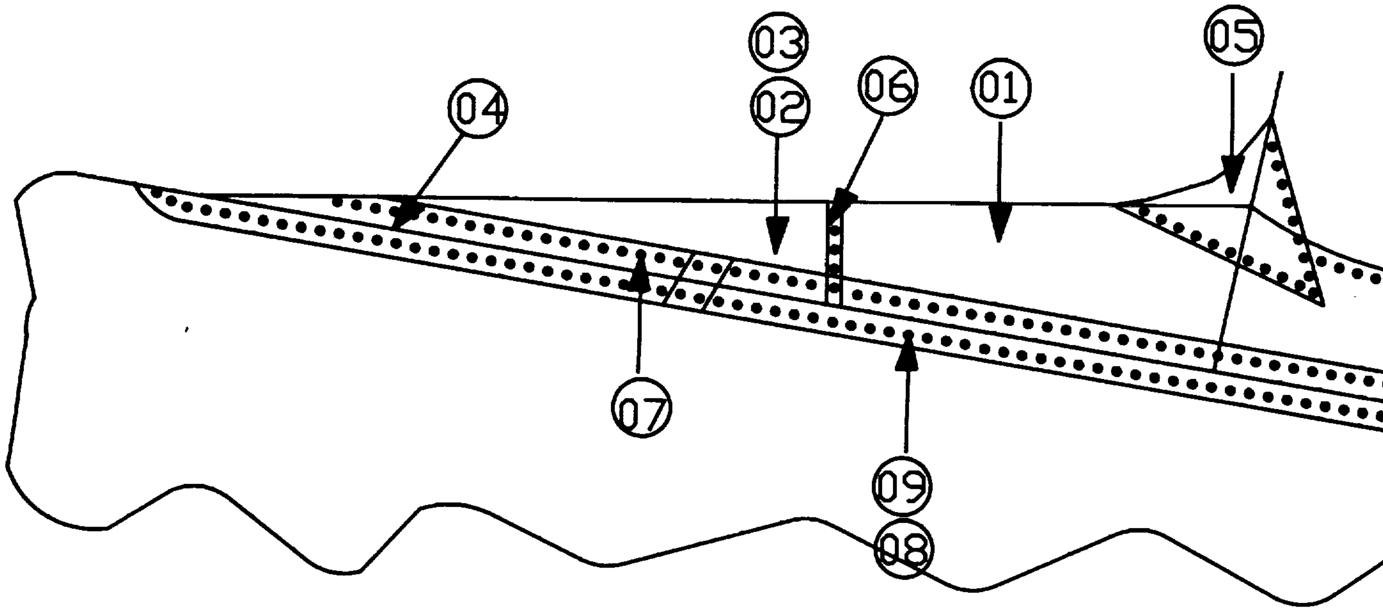
SIZE	DATE	SCALE	DRAWN BY
B	4-16	1/20	RUSS SNOW

TITLE
VERTICAL TAIL ASSY.

DRAWING NO.	SHEET
421S9303D230	01

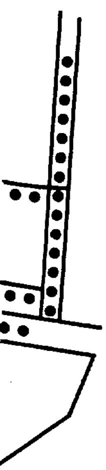
/,

FOLDOUT FRAME



2.

FOLDOUT FRAME



ITEM	QTY	DESCRIPTION	PART #
09	40	MACHINE SCREW	AN526DD-632-6
08	40	NUT PLATE	AN366DF-632
07	173	RIVETS	MS20470DD-4
06	01	DORSAL RIB	421S9303D294
05	01	FILLET SKIN	421S9303D295
04	01	INTERFACE (FRONT)	421S9303D296
03	01	INTERFACE (R)	421S9303D297
02	01	INTERFACE (L)	421S9303D298
01	01	DORSAL SKIN	421S9303D299

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED

DECIMAL

.XX ± .01

.XXX ± .001

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

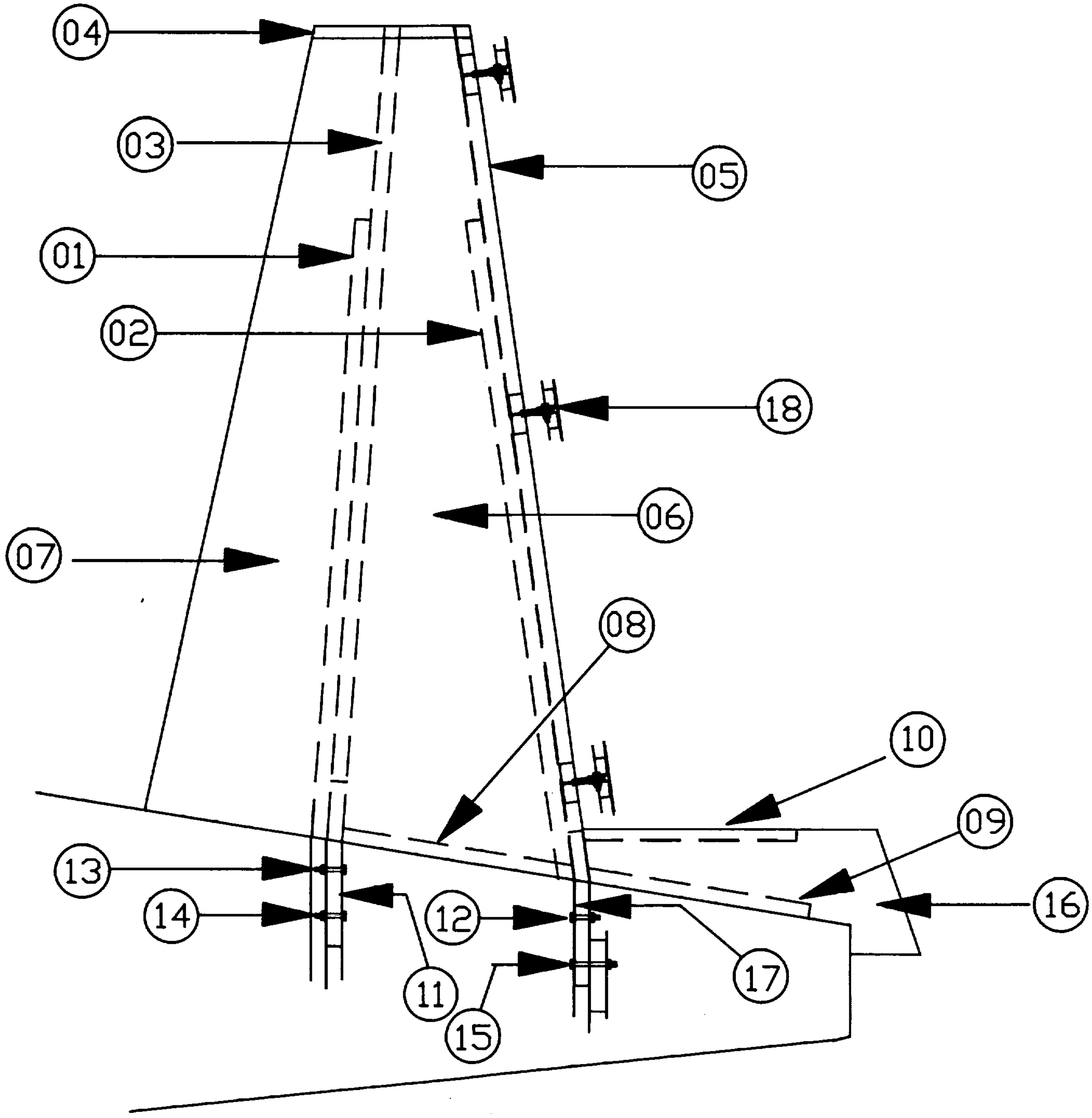
SIZE	DATE	SCALE	DRAWN BY
B	4-16	1/10	RUSS SNOW

TITLE
DORSAL FIN ASSY.

DRAWING NO. 421S9303D231

SHEET 01

FOLDOUT FRAME



NOTE: VERTICAL STAB. IS BEADED EVERY 3
NOT SHOWN FOR CLARITY

		FOLDOUT FRAME	
18	03	HINGE ASSY.	421S9303D248
17	01	REAR SPAR CASTING	421S9303D247
16	02	REAR SKIN	421S9303D246
15	02	BOLT	AN4-20
14	08	NUT	AN363-428
13	08	WASHER	AN960-XC416
12	06	BOLT	AN4-14
11	01	FORWARD SPAR CASTING	421S9303D245
10	01	RUDDER INTERFACE RIB	421S9303D244
09	01	AFT INTERFACE RIB	421S9303D243
08	01	FORWARD INTERFACE RIB	421S9303D242
07	01	L.E. SKIN	421S9303D241
06	02	SIDE SKIN	421S9303D240
05	01	AFT SPAR	421S9303D238
04	01	TIP RIB	421S9303D239
03	01	FORWARD SPAR	421S9303D237
02	02	AFT SPAR STIFFENER	421S9303D236
01	02	FORWARD SPAR STIFFENER	421S9303D235
ITEM	QTY	DESCRIPTION	PART#

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED

DECIMAL

.XX ± .01

.XXX ± .001

IN.

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

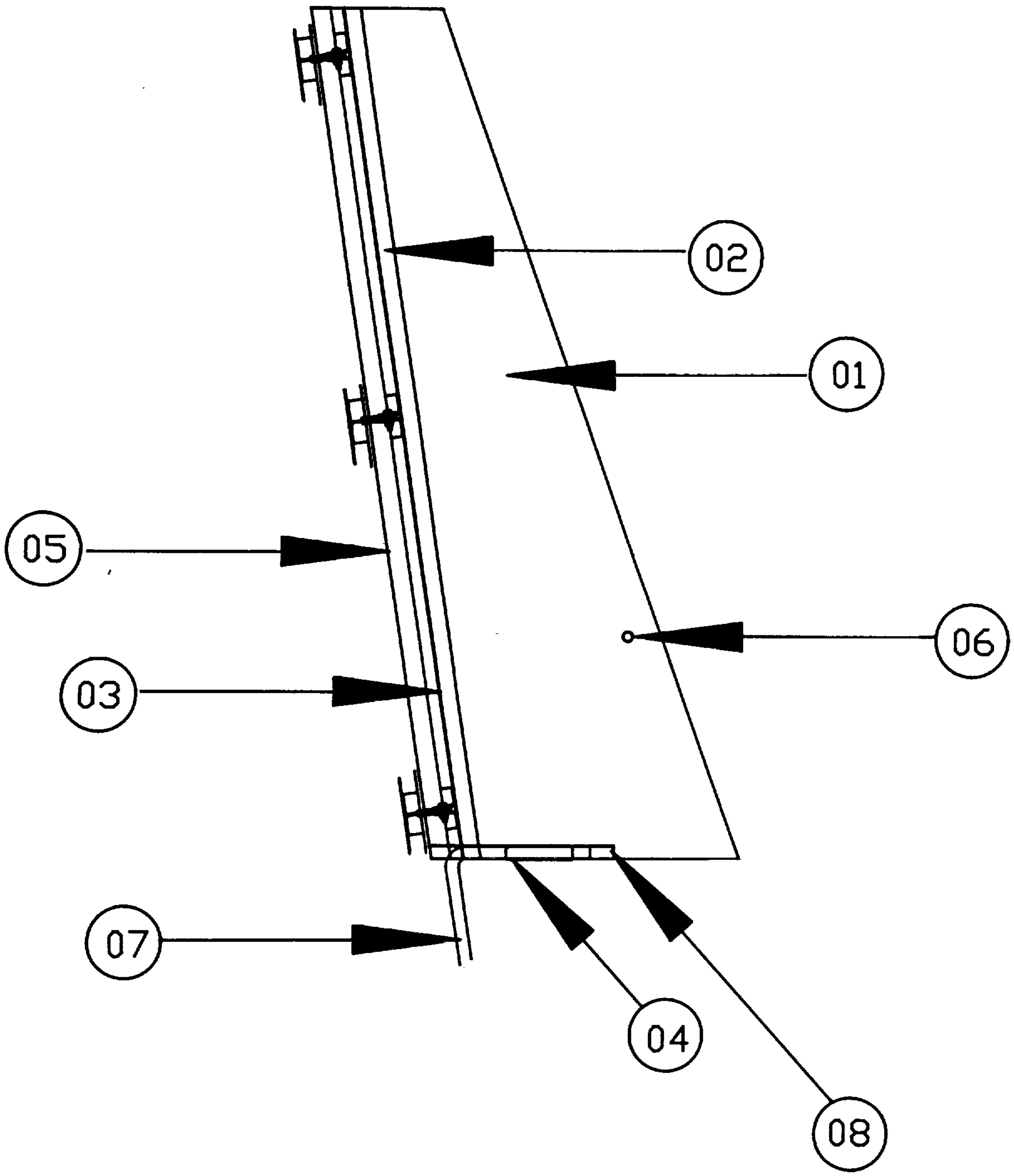
SIZE	DATE	SCALE	DRAWN BY
B	4-16	1/10	RUSS SNOW

TITLE
VERTICAL STAB. ASSY.

DRAWING NO.	SHEET
421S9303D233	01

1.

FOLDOUT FRAME



2

FOLDOUT FRAME

ITEM	QTY	DESCRIPTION	PART#
08	01	RUDDER ROOT RIB	421S9303D255
07	01	TORQUE TUBE	421S9303D254
06	136	RIVET	MS20470DD-4
05	01	L.E. SKIN	421S9303D253
04	01	TORQUE TUBE MOUNT	421S9303D252
03	01	SPAR	421S9303D251
02	02	SPAR STIFFENER	421S9303D250
01	02	SIDE SKIN	421S9303D249

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
B	4-16	1/10	RUSS SNOW

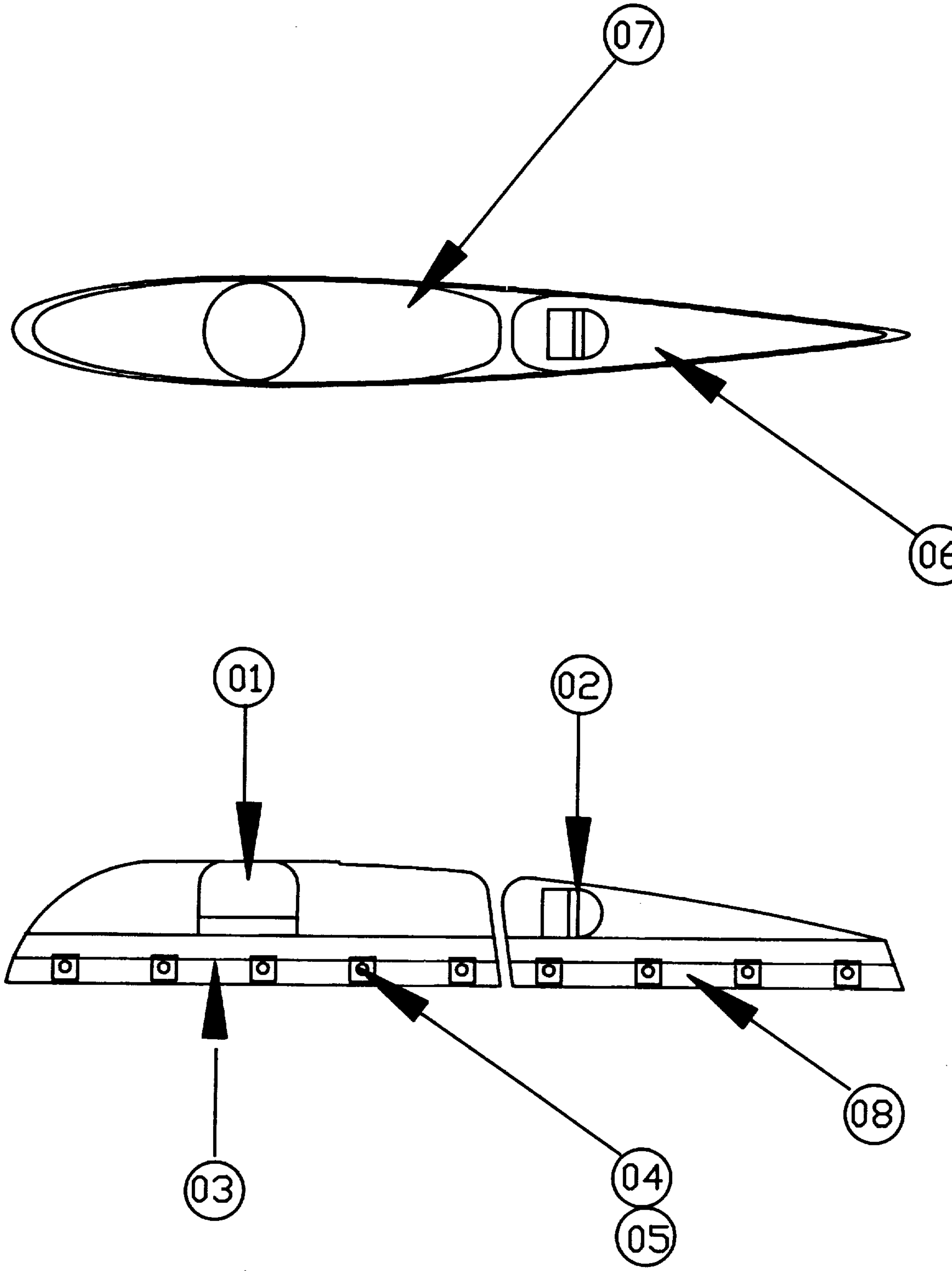
TITLE
RUDDER ASSY.

DRAWING NO. 421S9303D232

SHEET 01

1.

FOLDOUT FRAME



2.

FOLDOUT FRAME

ITEM	QTY	DESCRIPTION	PART#
08	01	AFT LENS MOUNT	421S9303D259
07	01	FORWARD LENS	421S9303D258
06	01	AFT LENS	421S9303D257
05	18	NUT PLATE	AN366DF632
04	18	MACHINE SCREW	AN526DD-632-6
03	01	FORWARD LENS MOUNT	421S9303D256
02	01	POSITION LIGHT	B-1943-4
01	01	ROTATING BEACON	157-0001-R

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01
.XXX ± .001

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

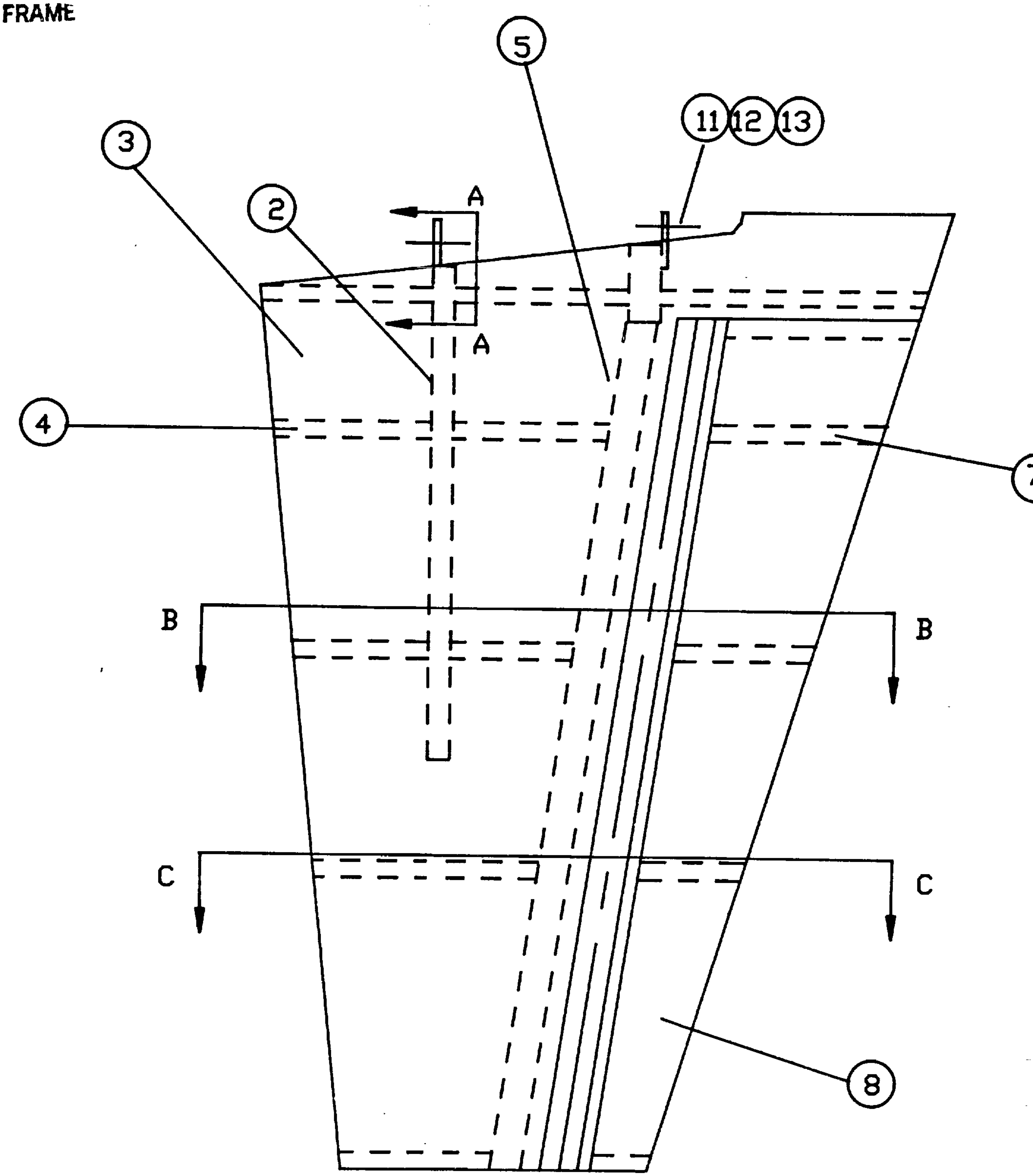
SIZE	DATE	SCALE	DRAWN BY
B	4-16	1/3	RUSS SNOW

TITLE
TIP ASSY.

DRAWING NO. 421S9303D234

SHEET 01

FOLDOUT FRAME



2

FOLDOUT FRAME

ITEM	QTY	DESCRIPTION	MATERIAL OR PART #
1	5	LEADING EDGE RIB	AL 2024-T3 .025
2	1	LEADING EDGE SPAR	AL 2024-T3 .075
3	2	SKIN	AL 2024-T3 .025
4	5	RIBS	AL 2024-T3 .025
5	1	REAR SPAR	AL 2024-T3 .14
6	2	HINGE ASSEMBLY	421S9303D248
7	4	ELEVATOR RIBS	AL 2024-T3 .025
8	1	ELEVATOR SKIN	AL 2024-T3 .025
9	2	ATTACHMENT CASTING	AL 2024-T3 .25
10	295	RIVET	MS20470DD-4
11	8	BOLTS	AN4-14
12	8	WASHER	AN960XC416
13	8	NUTS	AN363-C428

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
B	4/15/93	1/10	DAVE PAUFLER

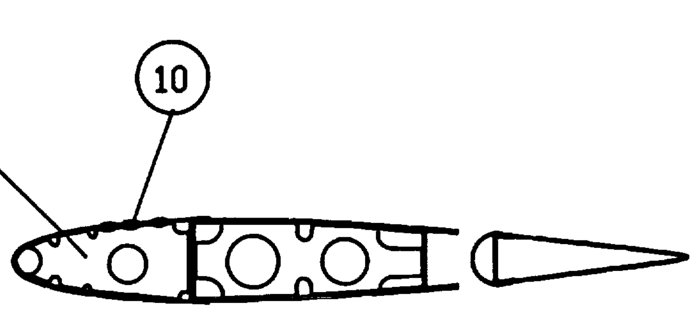
TITLE
HORIZONTAL TAIL ASSEMBLY

DRAWING NO.
421S9303D263

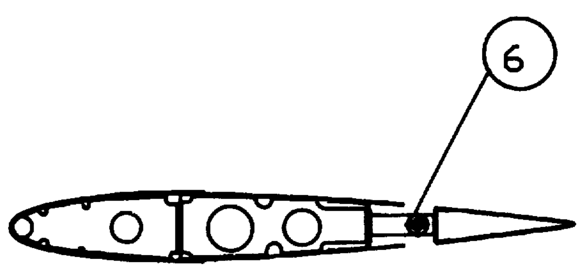
SHEET
1 OF 2

1.

FOLDOUT FRAME



SECT B-B

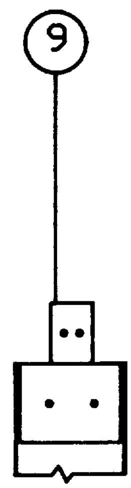


SECT C-C

2.

FOLDOUT FRAME

SECT A-A



NOTE: RIGHT HORIZONTAL STABILIZER WILL BE MIRROR
IMAGE OF THE LEFT HORIZONTAL STABILIZER SEEN HERE.

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
.XX ± .01	SIZE B	DATE 4/15/93	SCALE 1/10	DRAWN BY DAVID PAUFLER
.XXX ± .001	TITLE HORIZONTAL TAIL ASSEMBLY			
ANGULAR ± 1/2°	DRAWING NO. 421S9303D24863			SHEET 2 OF 2

FOLDOUT FRAME

25	80	RIVETS	MS 20426-
24	160	SCREWS	AN 526-I
23	80	NUT PLATE	AN 366-I
22	01	SKIN PANEL A	421S9303
21	01	SKIN PANEL B	421S9303
20	01	SKIN PANEL C	421S9303

FOLDOUT FRAME 2

	ITEM	QTY	DESCRIPTION	PART#
	19	01	SKIN PANEL D	421S9303D17
	18	01	SKIN PANEL E	421S9303D16
	17	01	ACCESS PANEL	421S9303D15
	16	02	STRINGER 2,8	421S9303D14
	15	01	STRINGER 5	421S9303D13
	14	02	STRINGER 4,6	421S9303D12
	13	01	STRINGER 3, 7	421S9303D11
	12	01	STRINGER 1	421S9303D10
	11	01	STRINGER 3A	421S9303D09
	10	02	STRINGER 2A, 4A	421S9303D08
	09	02	STRINGER 1A, 5A	421S9303D07
	08	01	FRAME A	421S9303D206
	07	01	FRAME B	421S9303D205
-DD-4	06	01	FRAME C	421S9303D204
D632	05	01	FRAME D	421S9303D203
F632	04	01	FRAME E	421S9303D202
D	03	01	FRAME F	421S9303D201
D	02	2565	RIVETS	MS20470DD-4
D	01	8	REINFORCMENT STRIP	MS20470DD-4

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED

DECIMAL

.XX ± .01

.XXX ± .001

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

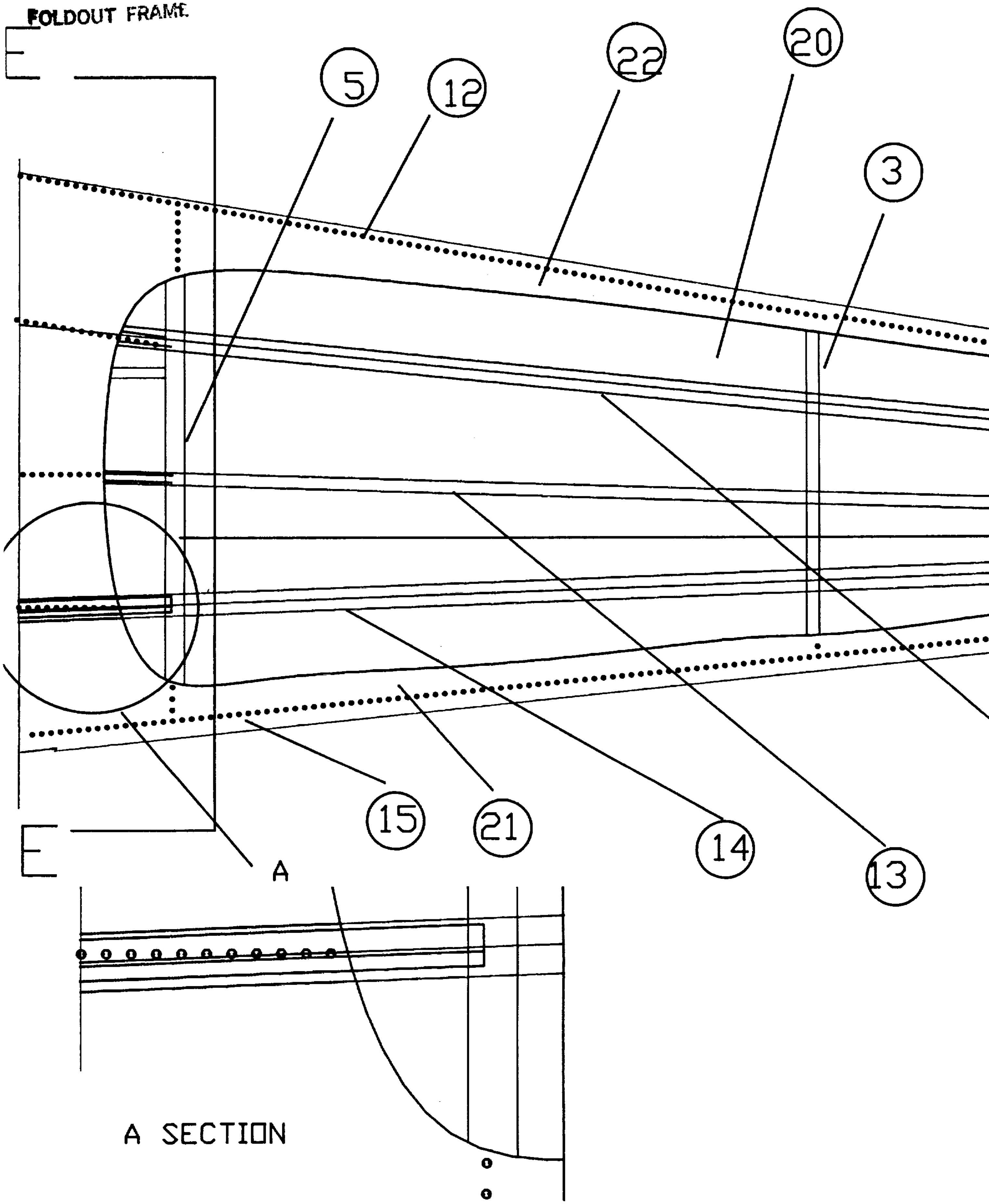
SIZE	DATE	SCALE	DRAWN BY
B	4-16	1/10	ATHOUSAKE ROX.

TITLE TAILCONE ASSEMBLY

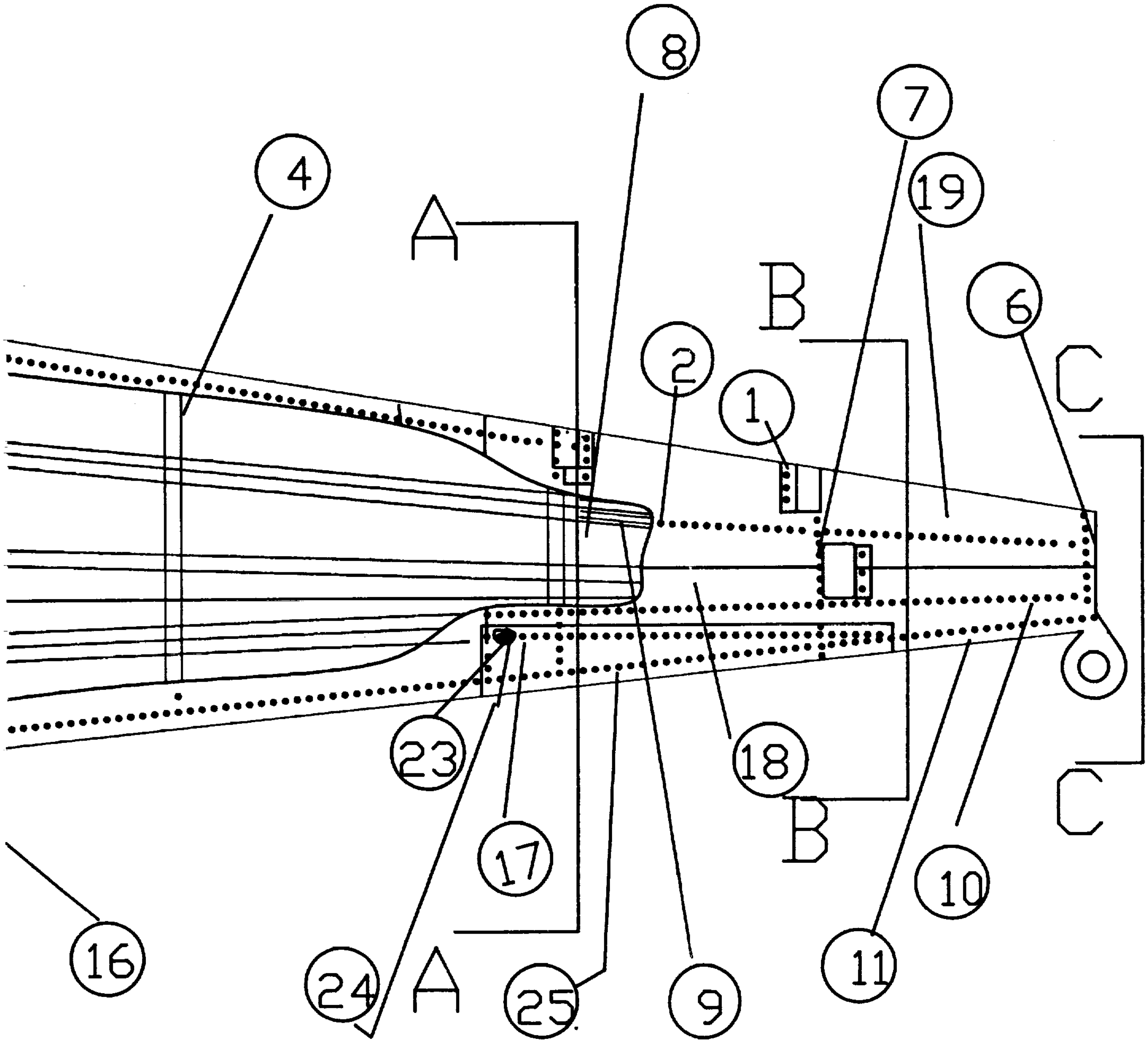
DRAWING NO. 421S9303D233

SHEET 1 OF 3

FOLDOUT FRAME



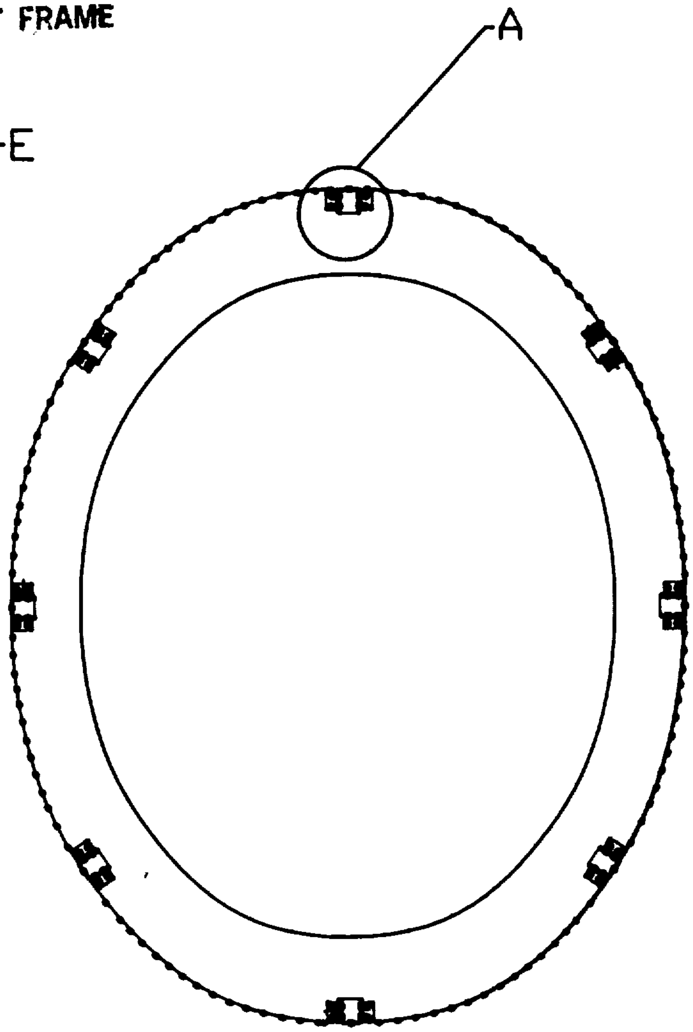
FOLDOUT FRAME

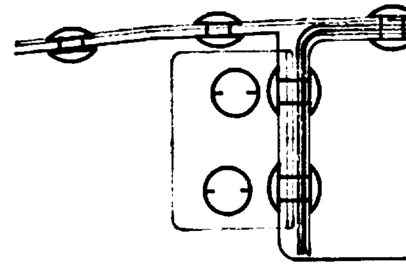


DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL .XX ± .01 .XXX ± .001	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
	SIZE B	DATE 4-16	SCALE 1/10	DRAWN BY ATHOUSAKE ROX.
ANGULAR ± 1/2°	TITLE TAILCONE ASSMEBLY			
	DRAWING NO. 421S9303D233			SHEET 2 OF 3

FOLDOUT FRAME

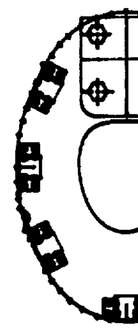
E–E

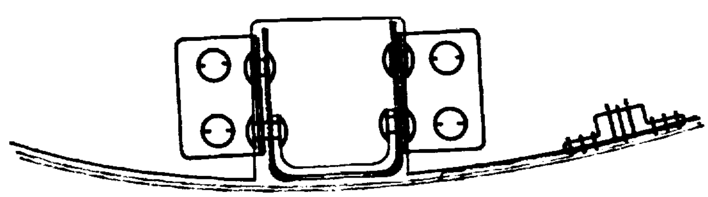




A SECTI

A–A

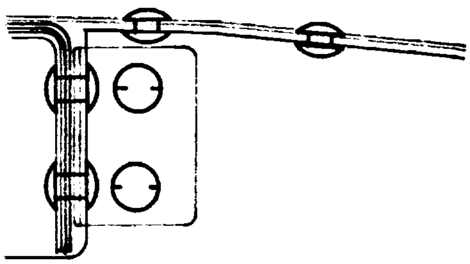




B SECTION

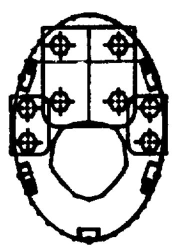
2.

FOLDOUT FRAME

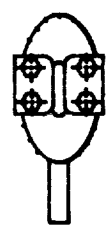


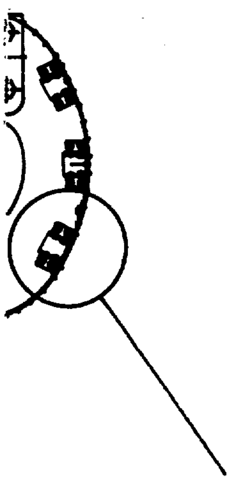
JN

B-B



C-C





B

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL .XX ± .01 .XXX ± .001 ANGULAR ± 1/2°	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
	SIZE B	DATE 4-16	SCALE 1/10	DRAWN BY ATHOUSAKE ROX.
	TITLE TAILCONE ASSEMBLY			
	DRAWING NO. 421S9303D99			SHEET 3 OF 3