NASw-4435

Cabin- Fuselage-Wing Structural Design Concept
with Engine Installation

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AE 421/02/A
Lead Engineer: Scott Ariotti
Team Members:
M. Garner, A. Cepeda
J. Vieira, D. Bolton

Submitted to:
Dr. J. G. Ladesic

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1. **Project Summary**

The purpose of this project is to provide a fuselage structural assembly and wing structural design that will be able to withstand the given operational parameters and loads provided by Federal Aviation Regulation Part 23 (FAR 23) and the Statement of Work (SOW). The goal is to provide a durable lightweight structure that will transfer the applied loads through the most efficient load path. Areas of producability and maintainability of the structure will also be addressed. All of the structural members will also meet or exceed the desired loading criteria, along with providing adequate stiffness, reliability, and fatigue life as stated in the SOW. Considerations need to be made for control system routing and cabin heating/ventilation. The goal of the wing structure and carry through structure is also to provide a simple, lightweight structure that will transfer the aerodynamic forces produced by the wing, tailboom, and landing gear. These forces will be channeled through various internal structures sized for the pre-determined loading criteria. Other considerations were to include space for flaps, ailerons, fuel tanks, and electrical and control system routing. The difficulties encountered in the fuselage design include expanding the fuselage cabin to accept a third occupant in a staggered configuration and providing ample volume for their safety. By adding a third person the CG of aircraft will move forward so the engine needs to be move aft to compensate for the difference in the moment. This required the provisions of a ring frame structure for the new position of the engine mount. The difficulties encountered in the wing structural design include resizing the wing for the increased capacity and weight, compensating for a large torsion produced by the tail boom by placing a great number of stiffeners inside the boom, this problem will result in relocating the fuel tank. Finally, an adequate carrythrough structure for the wing and fuselage interface will be designed to effectively transmit loads through the fuselage.

2. **Description**

2.1 **Fuselage Structure**

The cabin fuselage structure provides a stiff structure that will maintain the proper shape under the applied limit loads as indicated in the FAR 23. The applied loads are distributed through the use of various members such as longerons, ring frames, bulkheads, and skin. These provide paths for bending moments, torsion, and shear flow. The longerons of the fuselage assembly are extruded 7075-T6 Z - channels under the floor and 2024-T3 brake formed C - channels along the sides. Lightening holes are cut in the extruded longerons to reduce weight. In addition, stiffeners have been used along the bottom, in between the Z - channels, to account for torsion produced by the occupants during emergency landing conditions.

2.2 **Nose Cone/Engine Mount Assembly**

The nose cone assembly is fabricated out of a fiberglass /epoxy composite and is installed using flathead screws. The structure can be easily removed to allow for easy access to the nose gear assembly. The powerplant is installed directly to the aft ring frame. A series of longerons leading from this ring frame to the aft bulkhead provide adequate stiffness for the engine during all loading conditions.

2.3 **Spatial Requirements**

The minimum volumetric requirements need to assure adequate occupant safety, seating, and cabin ingress and egress. These areas are provided for by using the Jungle Aviation and Radio Services (JAARS) seat, that is certified under current FAR 23 dynamic crashworthiness conditions, and four point seat belt connections to the fuselage structure. The seat adjustments are relative to pilot physique, and the geometric extent of travel provide acceptable limits for human comfort while maintaining aircraft control.

2.4 **Wing Structure**

2.4.1 **Front Spar and Lug**

The process of designing the front spar began by determining the position within the given airfoil that would allow the height to be a maximum. For the NLF 0414 airfoil the maximum thickness occurs over a range of 25% to approximately 70% of the total chord length, thereby placing the spar at 25% of the chord.

After determining the position, several ideas were considered for construction from a series of extruded I - beams to brake-formed C - channels. Finally, it was decided to use an aluminum shear web of increasing thickness capped on top and bottom by NAS 344 series extruded aluminum T-sections. This built-up member would then be attached to the carrythrough using machined, quadruple shear, 4340 steel lugs and shear bolts.

Using the total lift distribution, shear and bending moment diagrams were generated for the entire wing. From the maneuvering point (A) on the V-n diagram 88% of these loads were applied to the front spar as a result of a chordwise equilibrium analysis. From these loading diagrams a required section modulus was determined at intervals along the span and plotted. The sizes of the spar cap T-section were then chosen, based on the required section modulus and available sizes of stock NAS 344. In order to accommodate the increased loads imposed by the tail boom and landing gear a large ΔS was retained until station 80.

2.4.2 **Rear Spar and Lug**

The design of the rear spar was performed similarly to the front spar. For strength/weight efficiency the spar was designed as a built-up shear web type, incorporating T - sections spar caps attached to a aluminum shear web. The rear spar was located approximately at the 67% chord on the airfoil and was designed to take up 12% of the total lifting load of the entire wing at the maneuvering point (A) on the V-n diagram.

Since the wing was chosen to a NLF 0414 airfoil, the height of the front spar and rear spar were almost identical, thereby allowing the design of the spar caps to depend on the respective shear and bending moment diagrams for each section. The spar caps needed to provide the required moment of inertia and subsequentially the available section modulus (S) is plotted versus the required section modulus. In the case of the available section modulus a NAS 344-32 extruded T - section was chosen as the spar cap at the root of the wing and varied to a NAS 344-2 at the tip. This built up member would then be attached to the carrythrough using machined, quadruple shear, 4340 steel lugs and shear bolts.

2.4.3 **Carrythrough Structure**

The carrythrough structure was initially designed as a continuation of the front and rear spars, but due to the configuration of the tail boom, and the location of the engine the carrythrough was modified significantly to accommodate the loading conditions. The front spar carrythrough incorporated two C - sections riveted together and attached to the firewall of the aircraft. The rear spar carrythrough was designed with a half-hoop C - section connected to another C - section on top of the hoop, this change was necessary due to the location of the engine. (Ref. Dwg. F93-2A-104-7)

3. Loads and Loading

3.1 Fuselage Bending

The highest bending loads are imposed on the structure during the landing condition. In order to comply with FAR Part 23, the structure must be able to withstand a 3g impact under normal landing conditions. Both the bending moment of the passengers, the JAARS system, and the nose gear landing loads are reacted through the floor structure and the longerons to the wing main spar. This moment calculation is shown below.

$$F_C = [3*W_M + 3*W_{JS}]*3g$$
$$= [3(200)+3(30)]*3$$
$$= 2070\,lbs$$
$$M_C = F_C * d_C = 2070(40)$$
$$= 82800\,inlbs$$

The bending moments from the nose gear load are $M_{nose} = 727.2(10^3)$ lb-in and the resulting stress is $f_{bend} = 29.3$ ksi.

$$F_{NG} = W_{ac_{est}} * 3g = 1987(3)$$
$$F_{NG} = 5961\,lbs$$
$$M_{NG} = F_{NG} * d_{NG} = 5961(122)$$
$$M_{NG} = 727200\,inlbs$$

This bending stress is less than the given ultimate stress of 48 ksi, and this results in a margin of safety of 0.363.

3.2 Safety Harness

The JAARS shoulder harness must withstand an impact of 18g's according to FAR Part 23. The attachment bolts have a bearing stress of 99 ksi. In order to accommodate this stress an extruded aluminum T - section was sized under the following conditions:

$f_{allow} = 99$ ksi

$V = 10.91(10^3)$ lbs

$D_{bolt} = 0.5$in.

The resulting thickness of the T - section is 0.220 inches. A similar evaluation of the rear occupant's harness attachment yields similar results with a margin of safety of 1.17.

3.3 Engine Torque

The torque produced by the engine was assumed to be 12,000 in lbs, and is reacted through the engine mounts to the ring frame. This, in turn, translates the torque into shear flow in the surrounding skin panels. The skin thickness and rivet spacing must be determined to withstand the buckling loads imposed by the torque. For a ring frame area of 1,773 in^2, the resulting stress of 136 psi is well under the F_{crit} of 1778 psi prescribed using the method outlined in Niu (Fig 5.4.6 pg 139).

3.4 Landing Load Torque

A torque of 81,300 in-lbs is imposed on the fuselage during emergency landing.

$$T_C = F_{CV} * d_{CV} + F_{CW} * d_{CWV}$$
$$F_{CW} = (100)(1.5) = 1500\,lbs$$
$$T_C = 2700(21.5) + 1500(15.5)$$
$$T_C = 81,300\,inlbs$$

As with the engine torque considerations, the skin thickness and rivet spacing must be determined to withstand the buckling loads caused by the torsional moment. The stress due to shear flow is 8,470 psi. Using Niu's method, the critical stress for buckling of the skin is 13,900 psi. The resulting margin of safety is 0.09.

3.5 Fuselage Bending due to Engine Weight

The weight of the engine applied a bending stress on the fuselage structure which is reacted through a series of 15 inch C - section longerons to the spars. The eleven longerons each carry a stress of 15.6 ksi, thus allowing for a margin of safety of 1.56.

3.6 Buckling Considerations of Engine Longerons

Upon impact during an emergency landing, the engine longerons must withstand an 18g forward buckling load. The C - sections have an ultimate margin of safety of 2.14. The thickness of the members can be changed to allow buckling at different loading conditions. This technique can be used to design a "mechanical fuse" that can steer the engine in a desired direction at impact.

3.7 Rivet Spacing

3.7.1 Lower Skin

Using a 3/16" diameter rivet and assuming a maximum rivet spacing of 1.5 inches, the maximum bearing stress on each rivet is 510 lbs. The bearing and ultimate stresses on each rivet was 675 lbs and 966 lbs, respectively, and the ultimate margin of safety was 0.324 and the bearing margin of safety was 0.263.

3.7.2 Upper Skin

Using a 1/8" diameter rivet and assuming a maximum rivet spacing of 1.0 inch, the maximum bearing stress on each rivet is 3.4 lbs. The bearing and ultimate stresses on each rivet is 281 lbs and 429 lbs, respectively. The ultimate and bearing margins of safety is over 80.

3.8 Engine Mount Bolts

The design of the engine mount requires mounting bolts to withstand the shear load of an

18g impact. With four 1/2" bolts the required shear load each must withstand is 10.8 ksi. A sheet thickness of 0.16 inches is necessary to avoid tear out around the bolts.

3.9 **Nose Gear Landing Load**

The shear stress on the nose gear mounting plate and bolts was calculated using four 1/2" bolts. The supporting C - section is designed to help carry the loads through the fuselage. The thickness of the plate required to avoid a tear out was 0.0951 inches.

3.10 **Wing Structure**

3.10.1 **Lift Reactions**

The first task undertaken in the design of the wing structure was determining the loads induced on the wing at the maneuvering and dive condition.

$$L_A = C_{L_A} * q_A * S_W$$
$$= 1.4(43)(152)$$
$$= 9150.4\,lbs$$
$$D_A = (C_{D_o} + kC_L^2) * q_A * S_W$$
$$= (0.00785 + 0.056 * 1.4^2)(43)(152)$$
$$= 769\,lbs$$
$$N_A = \sqrt{(9150.4^2 + 769^2)}$$
$$= 9182\,lbs$$

From this data the percentage of these loads were decomposed onto the front and rear spar.

$$x_A = \frac{C_M * C}{C_{L_A}} = \frac{-0.07(58.6)}{1.4} = -2.93$$
$$x_D = \frac{C_M * C}{C_{L_D}} = \frac{-0.07(58.6)}{0.45} = -9.12$$

Using a static equilibrium analysis about the front spar position, the force on the rear spar required to balance with the normal force at the center of pressure was calculated to be 1093 pounds. From these calculations, it was shown that the front spar carries 88% of the total wing loading at maneuvering speed while the rear spar carries 12%.

$$M_{FS} = (N_A * x_A) - (F_{RS} * (24.6)) = 0$$
$$= (9182 * 2.93) - (F_{RS} * (24.6)) = 0$$
$$F_{RS} = 1093\,lbs$$
$$F_{FS} = N_A - F_{RS}$$
$$= 9182 - 1093$$
$$F_{FS} = 8089\,lbs$$

At dive speed, the rear spar bears a greater load due to the rearward movement of the center of pressure at low angles of attack. Summing moments about the front spar again yielded the following:

$$M_{FS} = (N_A * x_D) - (F_{RS} * (24.6)) = 0$$
$$F_{RS} = 3404\,lbs$$
$$F_{FS} = N_A - F_{RS}$$
$$F_{FS} = 5778\,lbs$$

The load distribution between the spars at the dive condition was determined to be 63% for the front spar and 37% for the rear. So, the maneuvering condition sized the front spar and the dive condition sized the rear spar.

3.10.2 **Total Wing Lift Distribution**

The external lift distribution on the wing structure was determined using an elliptical and trapezoidal lift distribution. Taking the average of the two gave a close approximation to the actual lift distribution of the wing. This lift distribution is shown in figure 1.

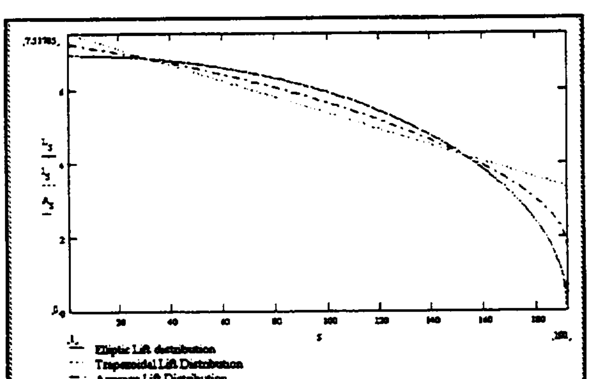


Figure 1

3.10.3 **Spar Shear and Moment Diagrams**

Once the total lift distribution was determined for the half-span of the wing, the shear and moment diagrams were plotted versus wing station.

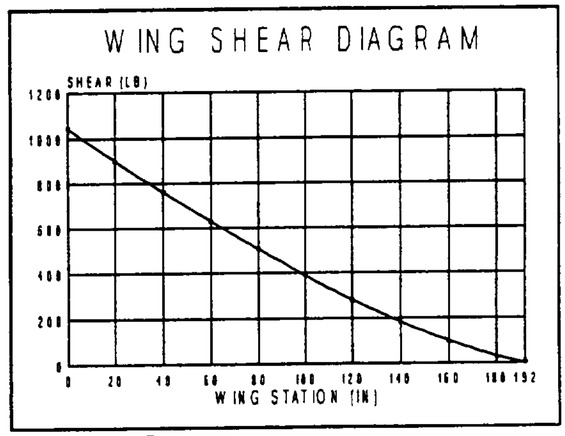


Figure 2

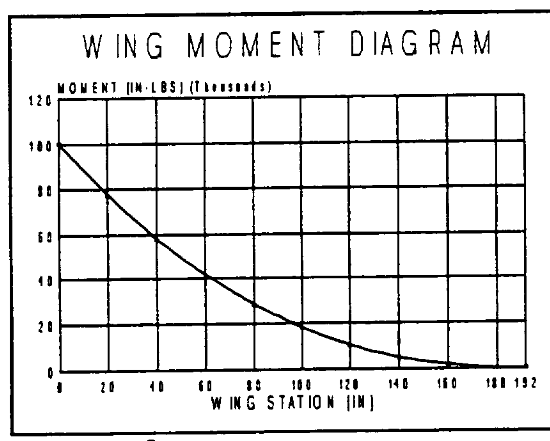


Figure 3

Using the percentages calculated in section 3.10.1, shear and moment diagrams for the respective spars were produced.

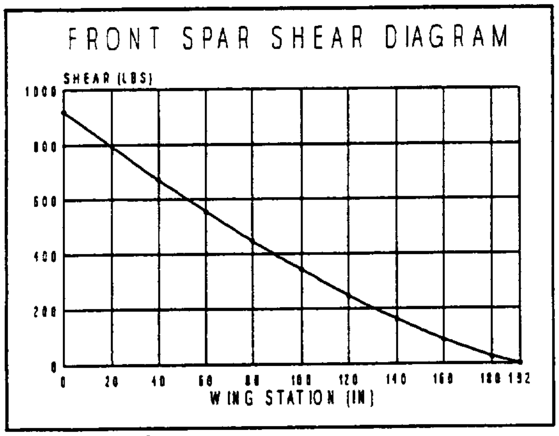


Figure 4

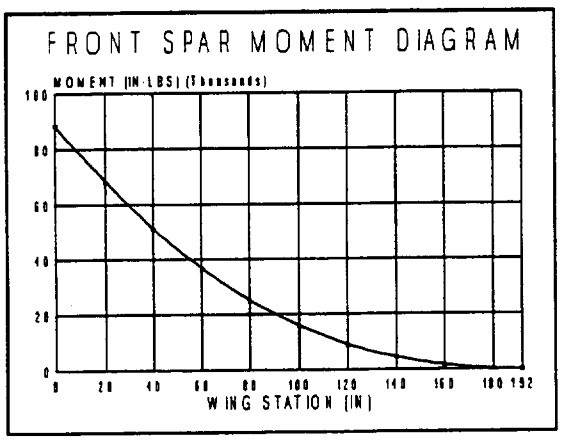


Figure 5

From these diagrams, the attachment lugs were sized based on the loads at the wing-fuselage interface.

3.10.4 **Wing Torsion**

The wing torsion accumulates over the span of the wing through aerodynamic loads and landing loads. The torque produced by aerodynamic loads were calculated for dive, maneuver, and flapped conditions. The aerodynamic change in torque was calculated using the following relation,

$$\Delta T = q * C_M * S_{LOC} * c_{LOC}$$
$$\Delta T = (119.5)(-0.07)(3.07)(2.91)$$
$$\Delta T = -74.73 inlbs$$

over the various flight condition. The boom torque, produced by the tail force, was determined as follows,

$$n_1 \frac{W_{ac}}{S_W} = 4.4 \frac{2091}{152} = 60.5 lb/ft$$

From Fig A5-FAR 23 Appendix A

$$w = 38 lb/ft^2$$
$$L_T = w * S_{HT} = (38)(55.3)$$
$$L_T = 2100 lbs$$
$$M_T = L_T * d_T = (2100)(165)$$
$$M_T = 346500 inlbs$$

The landing loads are also present at the boom and are determined to be,

$$F_f = \mu * W_{ac} * 3g = (0.8)(2091)(3)$$
$$F_f = 5018 lbs$$
$$T_G = F_f * d_v + W_{ac} * (3g) * d_h$$
$$T_G = (5018)(38) + (2091)(3)(21.5)$$
$$T_G = 325600 lbs$$

These torques are plotted versus wing station to produce the torque curve shown below.

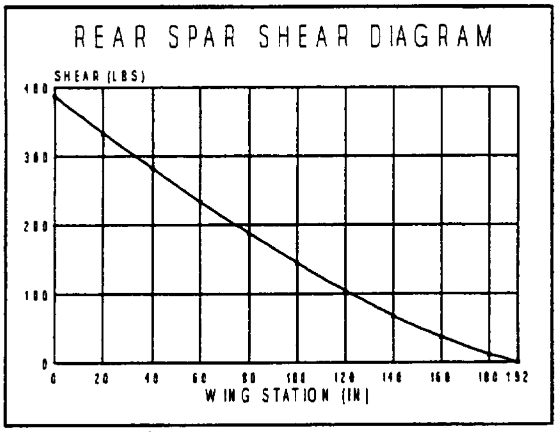


Figure 6

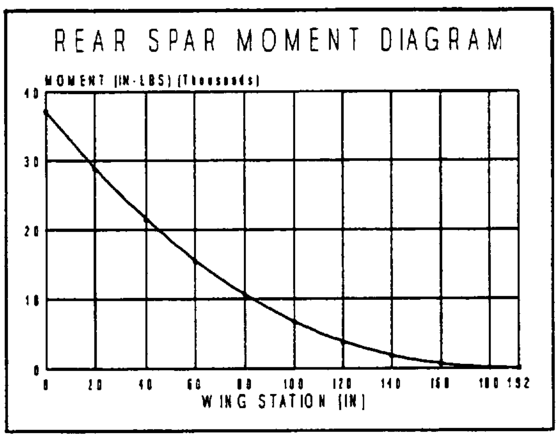


Figure 7

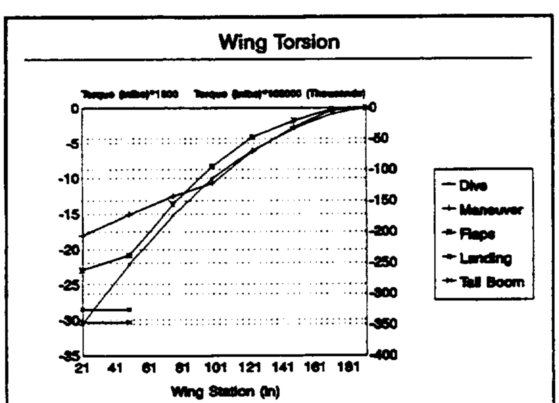


Figure 8

From the torque curve, the rib and stringer spacings were determined using buckling characteristics of the skin panels.

3.11 **Environmental Considerations**

3.11.1 **Temperature**

Cabin ventilation has been incorporated to allow occupant comfort in high temperature conditions. A supplemental heating system will provide warm air to the cabin for operation in cold weather conditions. All structural members and joints are designed taking into account tolerances for thermal expansion for the given thermal operating range(-40degree F to +122degree F) of the aircraft.

3.11.2 **Atmospheric Pressure**

FAR Part 91.211(a) defines the requirements for supplemental oxygen in non-pressurized civil aircraft registered in the United States. The cabin of the aircraft shall be equipped with supplemental oxygen for necessary crew members and passengers for the given conditions, up to 16,000 feet, defined by the FAR's.

3.11.3 **Sand and Dust**

All external surfaces will be coated with chip resistant aircraft paint to withstand dust and sand damage specified in the SOW. Door hinges will be outfitted with plastic washers between moving parts to reduce friction damage caused by particle matter. All windows will be sealed with rubber weatherstrip to keep out excessive sand and dust. Air filters are to be installed in the intake ducts to prevent engine and environmental control system damage by sand and dust.

3.11.4 **Rain**

All windows and doors will be outfitted with weatherstripping and/or sealant to prevent water intrusion and subsequent damage. Filters are to be installed in the engine air intake ducts to prevent excessive water accumulation. Drainage holes are to be made in the air ducts at locations susceptable to water accumulation and icing.

3.11.5 **Humidity**

All external skin panels are manufactured from 2024 aircraft aluminum and resist corrosion under the conditions specified in the SOW. At certain cabin temperatures and pressures, excess humidity is expected to condense on the front windshield and side windows. A heated defrost system has been integrated into the aircraft to eliminate windshield condensation.

3.11.6 **Ice**

All doors are to be equipped with seals designed to prevent water intrusion and subsequent freezing of the latch mechanism. Teflon or plastic washers between metal hinge parts will help prevent icing of the door hinges.

3.2.7 **Snow**

All skin panels and internal supporting members have been designed to support loads in excess of those encountered by the weight of 20 inches of wet snow.

3.11.8 **Salt/Fog Atmosphere**

All external skin panels are manufactured from 2024 aircraft aluminum to resist corrosion from salt and fog. Rivets and interface materials have been chosen that will resist corrosion also. Plastic washers between moving door hinge parts will help to prevent corrosion as well.

3.11.9 **Wind and Gust**

The fuselage structure has been designed to withstand gust loadings as defined by FAR Part

23. All tie-down fittings will be attached to the wings and tail structure.

4. Structural Substantiation

4.1 Floor Structure Bending Due to Occupants

Under aerodynamic loads, the occupants exert a bending moment on the floor members that needs to be counteracted. This is accomplished with the use of 4 NAS346-45 7075-T6 channels (see dwg. F93-2A-102-7). Values for the beams are Ixx = 0.7258 in^4 (4 beams) = 2.9032 in^4.

$$f=\frac{M_C y}{I}=\frac{(82800)(1.5)}{2.9032}$$
$$f=42.8ksi$$

Fty = 76 ksi (MS)ty = 0.78

Ftu = 83 ksi (MS)tu = 0.29

4.2 Fuselage Bending Due to Nose Gear Load

This is a worst case of the aircraft landing with the entire weight on the nose wheel. This bending moment is transferred through fuselage longerons and underfloor members. The simplified model yields a I total= 198.22 in^4 and y = 8.0 in (see dwg F93-2A-102-7 ; Parts 1,2,3).

$$f=\frac{M_{NG} y}{I}=\frac{(727200)(8.0)}{198.22}$$
$$f=29.3ksi$$

Fty = 47 ksi (MS)ty = 0.60

Ftu = 62 ksi (MS)tu= 0.41

4.3 Skin Buckling Due to Landing Load

The torque that the occupants produce during the landing criteria stated in FAR 23 is absorbed by the torque box formed by the under floor carry through (see dwg F93-2A-102-7; Section A-A). The torque produced is 81,300 inlbs through an area of 120 in^2 and skin thickness of 0.040 in.

(MS)bu = 0.09

4.4 Skin Buckling Due to Engine Torque

The torque that is produced by the engine is reacted through the engine ring frame, and into the fuselage skin. The calculation is the same as that in 4.3 with the following changes,

$$q=\frac{T_c}{2A}=\frac{12000}{(2)(1773)}$$
$$q=3.40\,lb/inch$$
$$f=\frac{q}{t}=\frac{3.40}{0.025}=136p.s.i.$$
$$F_{crit}=KE\left(\frac{t}{b}\right)^2=8(10^7)\left(\frac{0.025}{15}\right)^2$$
$$F_{crit}=222p.s.i.$$

(MS)bu = 0.09

4.5 **Fuselage Bending Due to Engine Load**

The moment produced by an acceleration on the engine is reacted through 11 longerons spaced around the periphery of the fuselage (see dwg F93-2A-102-7; Part 15). The stress in each member is,

$$f=\frac{M_{ENG}y}{I_{TOT}}=\frac{(83072)(1.5)}{7.99}=15.5ksi$$

Fty = 47 ksi (MS)ty = 2.0

Ftu = 62 ksi (MS)tu = 1.6

4.6 **Seat Track Fasteners**

The seat tracks are fixed through the floor panel into the NAS346-45 channel with four 5/16 in screws per track.

$$f_S=\frac{V}{A}=\frac{(6210)(1.5)(1.2)}{(\frac{\pi}{4})(\frac{5}{16})^2(4)}$$
$$f_S=36.4ksi$$
$$f_B=\frac{V}{Dt}=\frac{(6210)(1.5)(1.2)}{(\frac{5}{16})(.188)(4)}$$
$$f_B=47.6ksi$$

Fsu = 75 ksi (MS)su = 0.05

Fbu = 251 ksi (for steel) (MS)bu = 5.27

4.7 **Nose Gear Bolt Sizing**

The nose gear is held onto the forward bulkhead with 4 ASN8C-11 bolts (see dwg F93-2A-102-7).

$$f_S=\frac{V}{A}=\frac{(5961)(1.5)(1.2)}{(\frac{\pi}{4})(\frac{1}{2})^2(4)}$$
$$f_S=13.7ksi$$
$$f_B=\frac{V}{Dt}=\frac{(5961)(1.5)(1.2)}{(\frac{1}{2})(0.1)(4)}$$
$$f_B=53.6ksi$$

Fsu = 75 ksi (MS)su = 2.64

Fbu = 118 ksi (MS)bu = 0.46

4.8 **Engine Mount Bolt Sizing**

The engine is connected to the ring frame with four gusset plates. These plates allow the use of AN8C-7 bolts, one per plate, to transfer the engine loads into the ring frame and longerons (see dwg F93-2A-102-7).

$$f_S=\frac{V}{A}=\frac{(8496)(1.5)(1.2)}{(\frac{\pi}{4})(\frac{1}{2})^2(4)}$$
$$f_S=19.5ksi$$
$$f_B=\frac{V}{Dt}=\frac{(8496)(1.5)(1.2)}{(\frac{1}{2})(0.1)(4)}$$
$$f_B=47.8ksi$$

Fsu = 75 ksi (MS)su = 1.56

Fbu = 118 ksi (MS)bu = 0.64

4.9 **Spar Cap Sizing**

The principle of the section modulus was used to generate the spar cap sizing. The caps start off as an NAS344-69 for the front spar and -33 for the rear spar (see dwg F93-2A-104-7). The justification for the sizes is detailed below.

$$S=\frac{I_X}{y}=1.01\left(\frac{M}{F_{TU}}\right)$$
$$S_{REQ}=1.01\left(\frac{88282}{78000}\right)=5.03\,inches^3$$
$$S_{AVAIL}=\frac{11.55}{4.4}=5.63\,inches^3$$

The section modulus plotted over the span of the wing for the both spars yields the following curves.

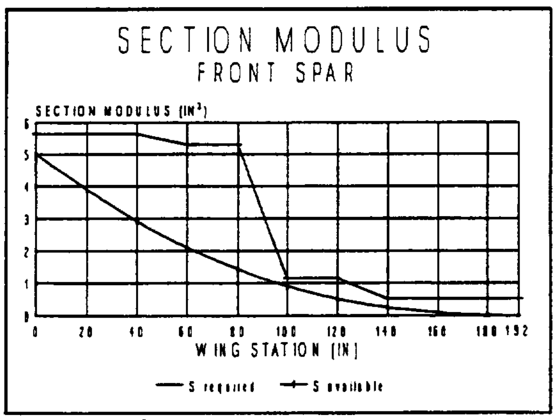


Figure 9

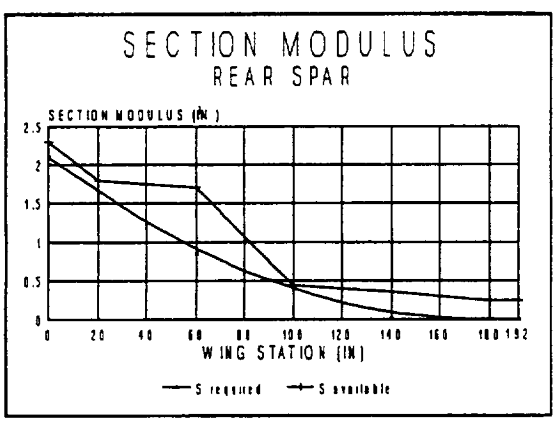


Figure 10

4.10 **Spar Cap Fatigue**

Fatigue estimations were performed on the spar caps using half of the weight of the aircraft distributed over the spars as designated in 3.10.1. Pfs=920 lbs and Prs=324lbs at a distance of 44 inches. Using the S-N curve from Fig 3.7.4.1.8(g) from MIL-HDBK-5E and the following,

$$f_{mean}=\frac{Pl}{S}=\frac{(940)(44)}{(5.63)}$$
$$f_{mean}=7190\,p.s.i.$$
$$f_{max}=2.2f_{mean}=2.2(7190)$$
$$f_{max}=1518\,p.s.i.$$
$$N=2(10^5)\,cycles$$

Similarly, the rear spar yields a count of $5(10^5)$ cycles.

4.11 **Shear Web Sizing**

Shear flow is determined from the torque diagram and the web thicknesses are justified by meeting the buckling criteria. A sample calculation is shown below.

$$q=709\,lb/inch$$
$$f_{L.L.}=\frac{q}{t}=\frac{709}{.125}$$
$$f_{L.L.}=5672\,p.s.i.$$
$$f_{UL}=f_{L.L.}(1.5)=5672(1.5)$$
$$f_{UL}=8508\,p.s.i.$$
$$F_{crit}=KE\left(\frac{t}{b}\right)^2=5(10^7)\left(\frac{0.125}{8.8}\right)^2$$
$$F_{crit}=10{,}088\,p.s.i.$$

(MS)b = 0.19

(see dwg F93-2A-104-7 for remaining thicknesses).

4.12 **Rib Sizing**

From the torque diagram the value for torque at WS172 can be determined. The rib is then sized using the buckling criteria used previously. See drawing F93-2A-104-7 for a complete list of thicknesses.

$$q=1.76\,lb/inch$$
$$f_{L.L.}=\frac{q}{t}=\frac{1.76}{.032}=55.1\,p.s.i.$$
$$f_{UL}=f_{L.L.}(1.5)=55.1(1.5)=82.7\,p.s.i.$$
$$F_{crit}=KE\left(\frac{t}{b}\right)^2=7(10^7)\left(\frac{.032}{20}\right)^2=179\,p.s.i$$

(MS)b = 1.2

4.12 **Lightening Hole Justification**

Lightening holes were considered for the ribs inboard of the boom due to there thickness and weight. Using the calculations in Niu and figure 6.2.3, the following dimensions were obtained; D=5.25 inches, b=7.7 inches. Values previously determined; q=709 lb/in, rivet diameter=.375 in., and rivet spacing=5.5 in.

$$f_{s_{web}}=\frac{q}{t}\left(\frac{b}{b-D}\right)$$
$$f_{s_{web}}=\frac{709}{0.125}\left(\frac{7.7}{7.7-5.25}\right)$$
$$f_{s_{web}}=17.8\,ksi$$

$$K_r=\frac{s-d}{s}=\frac{5.5-.375}{5.5}=0.932$$
$$f_{s_{rivet}}=\frac{q}{K_r t}=\frac{709}{(0.932)(0.125)}=6.1\,ksi$$

$$f_{s_{hole}}=\frac{q}{t}\left(\frac{h}{h-D}\right)$$
$$f_{s_{hole}}=\frac{709}{0.125}\left(\frac{8.8}{8.8-5.25}\right)$$
$$f_{s_{hole}}=14.1\,ksi$$

4.13 **Lug Justification**

The lug's purpose is to transfer shear and bending moments to the fuselage carry-through. They are subject to the highest shear and bending moment since they are at the root of the wing.

4.13.1 **Forward Lug**

According to the shear and moment diagram for the front spar the values are 919.6 lbs and 88282 inlbs at the root, respectively. The lugs are 5.8 inches apart which yields a loading, P, of 15,200 lbs. The bolt is in quad shear so the stress is,

$$f_s=\frac{P}{(4(A)}$$
$$f_s=\frac{15,200(1.5)(1.2)(4.4)}{\left(\frac{\pi}{4}\right)(.75)^2(4)}$$
$$f_s=68\,ksi$$

(Fsu)bolt = 75 ksi (MS)su = 0.1

The material for the lug was chosen to be 4340 steel because of its high strength properties. After calculating flange thicknesses required over stresses for bearing, tensile, tear out, and fatigue, the thickest requirement came from tensile stress. The flange thickness needs to be .125 inches to satisfy the tensile stress requirement. Fatigue assessments were made at an infinite life of 10^7 cycles.

4.13.2 **Rear Lug**

Using the same procedure as the front lug the bolt diameter was found to be 7/16 inches and the flange thickness in quad shear was .114 inches. As before the lug thickness was checked for each type of stress and tensile stress was the sizing factor.

4.14 **Carry Through Structure**

The carry through structure for the wing is designed to carry a pure bending moment. The shear reactions are taken up by the fuselage structural members. The pure moment, determined from the moment diagram, is taken up in the moment of inertia (Ifront=31 Irear=10.8 in^4) for the carry through.

5.0 **Manufacturing and Maintenance Provisions**

5.1 **Fuselage Assembly**

The fuselage structure assembly is comprised of four main components: longerons, ring frames, bulkheads, and skin. The floor longerons of the Quest PFT are manufactured from 7075-T6 Aluminum extrusions, while the remaining components are manufactured from 2024-T3 Aluminum sheet brake formed and fitted to shape. The entire fuselage skin is riveted together using AN456DD-4 rivets, while the internal components use AN430DD-8 rivets. Aircraft quality AN designation type bolts are used to mount the engine and nose landing gear. The skin thickness was changed under the belly of the aircraft in order to account for the torsional loading produced by the occupants.

5.2 **Engine Mounts**

The engine mounts to a ring frame attached to the rearward bulkhead by the use of brake formed aluminum sheet c channels. The engine truss is mounted to the ring frame with reinforcing brackets attached to the longerons. Engine maintenance is accomplished through the use of engine cowlings located on the left and right side of the engine.

5.3 **Control System and Ventilation Routing**

The control system installation simply requires allowing space for the routing of this system. This is provided via a channel running between of the four floor longerons. This system will incorporate push-pull rods for control. Access to the control systems are provided by inspection plates located under the fuselage. The venting system runs along the side of the aircraft in between the two side longerons. The blower is located under the engine and fresh air is provided via the air cooling vents on the engine. This system is easily maintained from the same location of the engine cowling doors.

5.4 **Front and Rear Spar**

The front and rear spar is comprised of two main components: the spar caps and shear web. The spar caps are manufactured from 7075-T6 Aluminum extrusions T-sections with varying section properties. The shear web is manufactured from 2024-T3 Aluminum sheet formed and fitted to shape with varying thickness to account for the shear flow. These parts are connected using high quality NAS 1304-4P steel bolts and rivets.

5.5 **Front and Rear Lugs**

The lug assembly is comprised of one piece shaped to the designed configuration. It is composed of a high strength steel (4340 steel), which may have to be cadmium plated to resist corrosion. The lug is connected to the spar cap on the respective spars with the use of 0.25" diameter NAS bolts.

5.6 **Wing Carrythrough Structure**

The wing carrythrough structure was one of the difficulties encountered in the design of the wing. The front spar carrythrough was assembled using two 2024 aluminum sheet brake formed c-sections attached to the firewall by 24 bolts sized according to the moment produced at the root of the wing. The rear spar was modified slightly, wherein a half-hoop aluminum sheet c-section brake formed was attached to a straight c-section on top of the hoop. This was done in order to compensate for the engine location, while at the same time providing support for the engine.

5.7 **Internal Wing Structure and Skin**

The sizing of the ribs, along with the spacing were designed from the torsion induced from the aerodynamic loads of the aircraft. The ribs are manufactured from 2024 aircraft aluminum brake formed and fitted to shape. Various stiffeners were also placed in the wing to help meet the required buckling criteria. The entire wing skin is riveted together using quality AN rivets, the skin is manufactured from 2024 aluminum sheet of 0.032" thickness outboard of the tail booms, and 0.05" thickness inboard of the tail booms. This was designed due to the large torsion produced by the booms. Finally various access panels or holes were placed at strategic locations in order to provide for maintenance of the wing and control systems.

6.0 **Weight Summary**

The weight of the aircraft is an essential design aspect that must be addressed to allow for maximum stiffness of members for given applied loading conditions, while at the same time providing a light, yet durable structure. Weight savings are accomplished by using aluminum sheet for the majority of the structural members since it provides a relatively high stiffness-to-weight ratio at a low cost. Additional weight savings include punching lightening holes in the extruded aluminum floor longerons. Extruded parts are usually thicker than brake formed members so they tend to be heavier. Rivet weight is compensated for by allowing them to add 1% to overall structural weight. The total weight for the aircraft structure is 144 lbs. The weight for the structure in the two seater version was 131 lbs. An increase in 13 lbs for the structural weight to support the loadings imposed by a third passenger seems to be valid. The total weight of the aircraft wing is 269 lbs. The weight for the two seater version was 253 lbs. Thereby, an increase of 13 lbs, also, for the third passenger seems to be valid.

Wing Weight Estimation			
		GAGE	WEIGHT
PART	QTY	(IN)	(LB)
RIB WS 192	2	0.016	0.701344
RIB WS 172	2	0.016	0.788608
RIB WS 148	2	0.032	1.881024
RIB WS 99	2	0.032	2.495104
RIB WS 87	2	0.032	2.676096
RIB WS 75	2	0.032	2.805376
RIB WS 49	2	0.05	4.9288
RIB WS 42	2	0.125	10.12525
RIB WS 35	2	0.125	10.12525
RIB WS 28	2	0.125	10.12525
RIB WS 21	2	0.125	10.12525
F-SPAR CAP 0-80	2		12.84256
F-SPAR CAP 80-120	2		1.635696
F-SPAR CAP 120-192	2		1.6704
R-SPAR CAP 0-20	2		1.102868
R-SPAR CAP 20-100	2		3.464398
R-SPAR CAP 100-140	2		0.531468
BUCKLE STIFFENER (#9)	44	0.02	3.26634
BUCKLE STIFFENER (#10)	28	0.02	4.010104
BUCKLE STIFFENER (#11)	12	0.02	2.63004
BUCKLE STIFFENER (#12)	4	0.02	1.25846
SPAR SHR WEB (20-49)	4	0.125	12.4432
SPAR SHR WEB (49-76)	4	0.05	4.52682
SPAR SHR WEB (76-172)	4	0.032	7.632691
SPAR SHR WEB (172-192)	4	0.016	0.58176

FRONT LUG	4	1	12.9614
REAR LUG	4	1	6.9618
FRONT CARRY THROUGH	1	0.09	11.69883
REAR CARRY THROUGH	1	0.05	4.70862
SKINS:			0
WING TIP	2	0.032	1.105344
OUTBOARD	2	0.032	16.65126
MID LEADING EDGE	2	0.032	18.68096
MID	2	0.032	30.75571
INBOARD LEADING EDGE	2	0.05	31.5524
INBOARD TRAILING EDGE	2	0.05	12.221
INBOARD BOTTOM	2	0.05	7.373

WEIGHT= 269.0445 LBS

7.0 **Conclusion**

The goal of this design project is to provide a cabin fuselage structure detail design along with an adequate wing structure that meets the loading criterion defined by FAR Part 23. Control systems and ventilation routing have been designed for ease of maintenance and removal. Volumetric cabin requirements were also met that assured adequate spacing and safety of the occupants. Calculations of the torsion, or shear flow, induced by the occupants required a change to the original floor design in which stiffeners were added to counteract these loads. Calculations of the aerodynamic forces induced by various aircraft components such as the twin tail booms, the landing gear, and the wing were performed and decomposed into the sizing of the total wing structure. This included sizing the front and rear spar and lug, the wing skin, ribs, along with other parts. The location of the wing on the mid-fuselage required a different approach to designing the wing carrythrough structure. Also the large torsion produced inboard of the tail booms required a change to the skin thickness to counteract this torque. The cabin was expanded to accommodate a three-seat passenger configuration. The new engine location required providing a ring-frame structure to account for a longer moment arm. Weight estimates and sizes provided by the preliminary design report were for a two-seat configuration aircraft. Certain assumptions needed to be made about the weight of this larger aircraft that will need to be validated. Finally, environmental conditions were addressed for all relevant parts. This design meets all of the requirements, but further optimization may be performed through later iterations.

Appendices

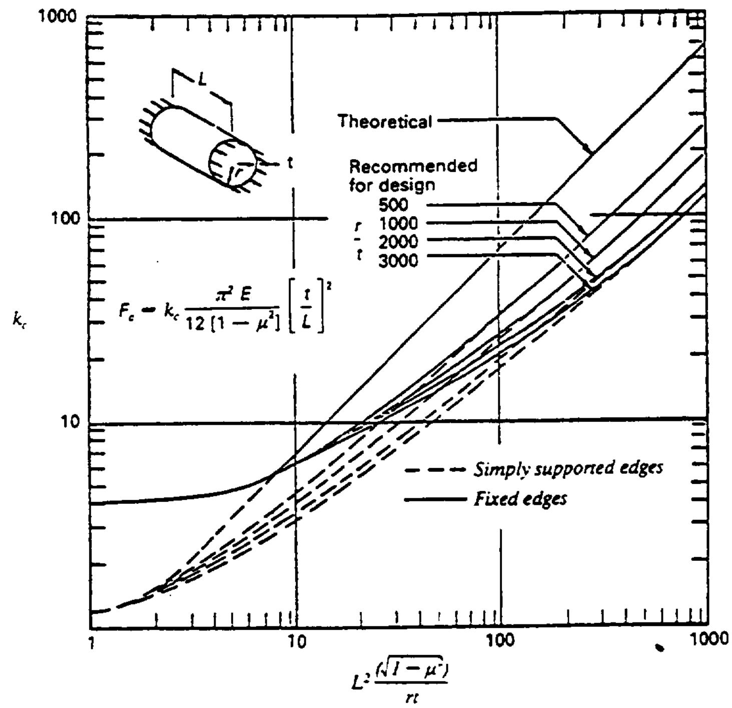


Fig. 5.4.5 Compression buckling coefficients K_c (circular cylinders).

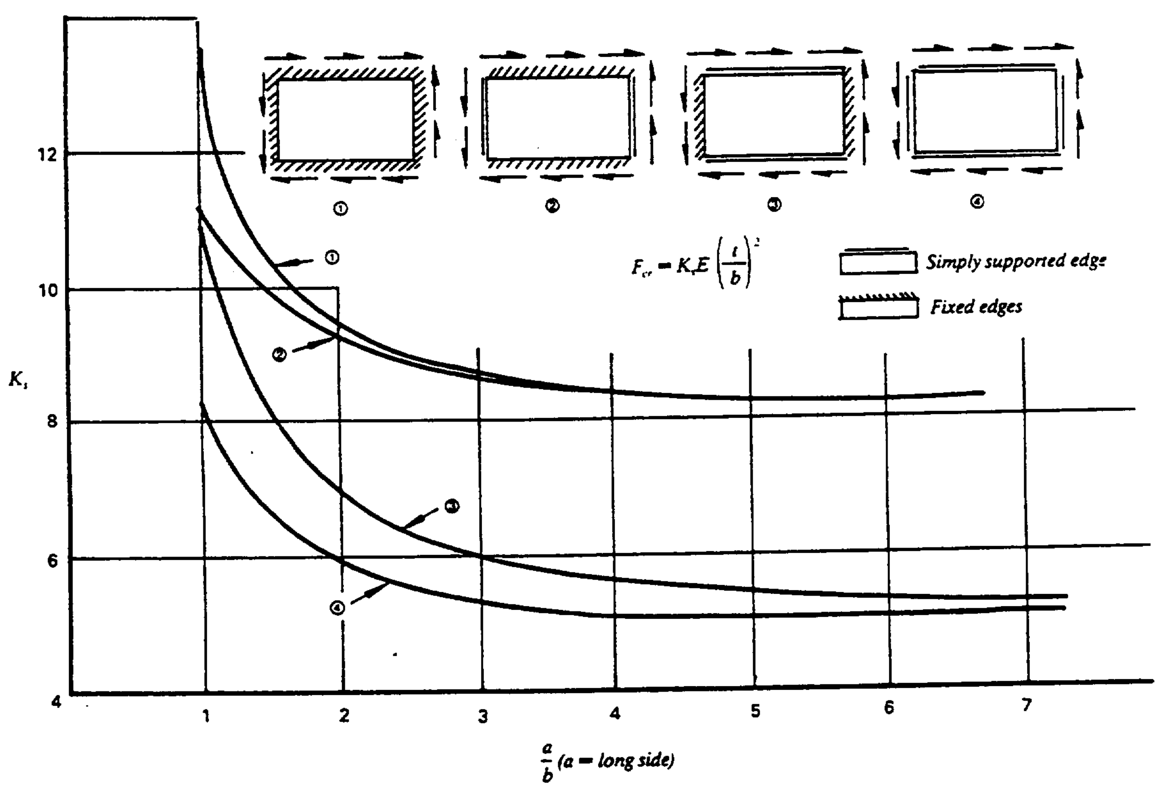


Fig. 5.4.6 Shear buckling coefficients K_s (circular cylinders).

6.2 Lightly Loaded Beams

The ideal construction for most shear-carrying beams is a tension field (or diagonal tension beam per Ref. 6.8). However, in some cases it is advantageous, and in other cases necessary, to incorporate circular, flanged holes in the beam webs. These cases come under two main categories:

- Lightly loaded or very shallow beams. In such cases it may not be practical to construct an efficiently designed tension field beam because of minimum gage considerations and other restrictions due to the small size of the parts involved. It may then be advantageous from a weight standpoint to omit web stiffeners and, instead, introduce a series of standard flanged lightening holes, as shown in Fig. 6.2.1.
- Moderately loaded beams with access holes. Where it is necessary to introduce access holes into the web of a shear-carrying beam, a light, low cost construction is obtained by using a flanged hole with web stiffeners between the holes.

Lightly Loaded or Very Shallow Beams

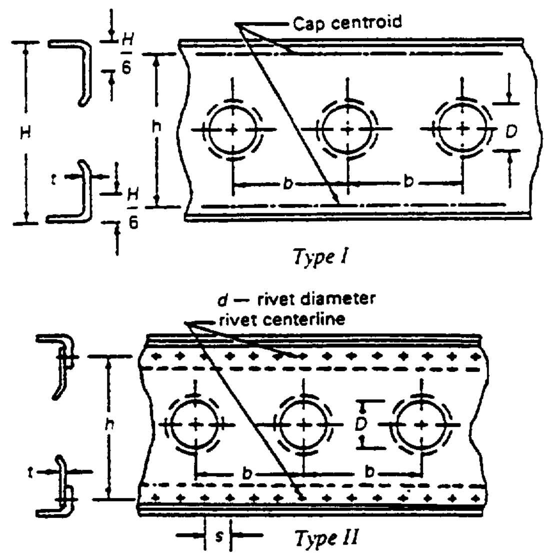
The following two types of beam construction are considered. The standard flanged lightening holes as shown in fig. 6.2.2 are centered and equally spaced.

- The limiting conditions for the design curves is given in Fig. 6.2.3.

$$0.25 \leqslant \frac{D}{h} \leqslant 0.75;$$

web thickness $0.016 \leqslant t \leqslant 0.125$

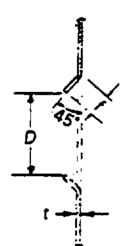
$$0.3 \leqslant \frac{D}{b} \leqslant 0.7; \qquad 40 \leqslant \frac{h}{t} \leqslant 250$$



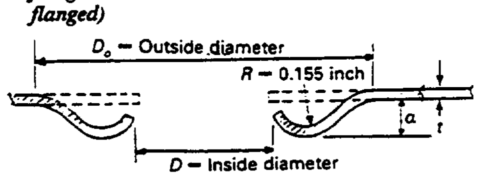
(Note: $\frac{H}{6}$ is the assumed effective depth of beam cap)

Fig. 6.2.2 Lightly loaded or very shallow beams.

D (inch)	2.0	2.5	3.0	3.5	4.0	4.5	5.0	6.0
f (inch)	.25	.3	.4	.45	.5	.5	.5	.55



(a) Lightening holes of typical flanged holes (45° flanged)



D_0 (Inch)	D (Inch)	a (Inch)
1.7	0.8	0.2
1.95	1.05	0.2
2.65	1.7	0.25
3.0	2.05	0.25
3.65	2.7	0.25
3.9	2.95	0.25
4.95	3.8	0.4
5.95	4.8	0.4
6.95	5.8	0.4
7.44	6.3	0.4
7.95	6.8	0.4
8.95	7.8	0.4
9.45	8.3	0.4

(b) Lightening holes with beaded flanged

Fig. 6.2.1 Common flanged lightening holes. (t — 0.032 in — 0.125 in)

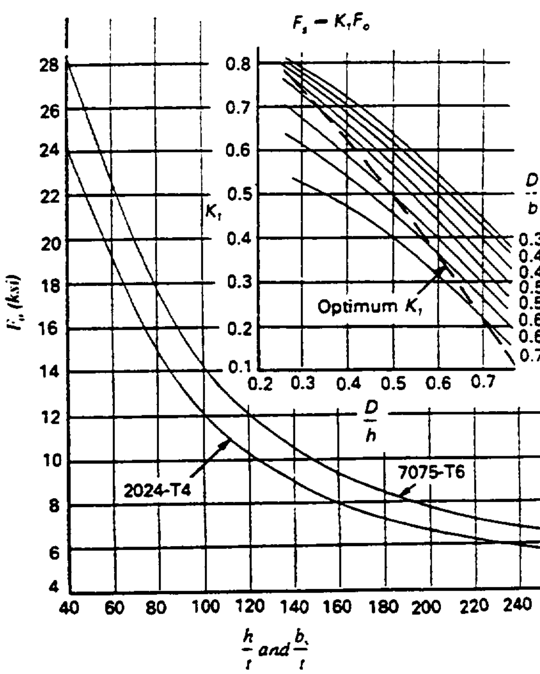


Fig. 6.2.3 Ultimate allowable gross shear stress for aluminum alloy webs with flanged holes as shown in Fig. 6.2.1(a).

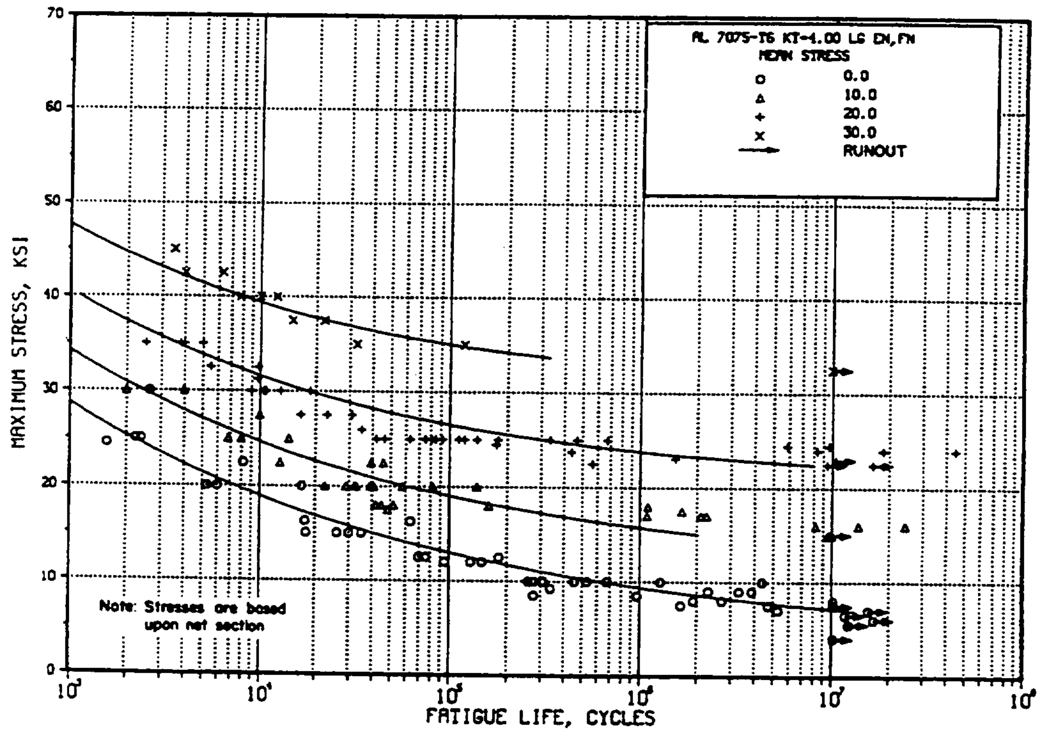


FIGURE 3.7.4.1.8(g). *Best-fit S/N curves for notched, K_t = 4.0, 7075-T6 aluminum alloy sheet, longitudinal direction.*

Correlative Information for Figure 3.7.4.1.8(g)

Product Form: Bare sheet, 0.090 inch

Properties:

TUS, ksi	TYS, ksi	Temp., F
82	76	RT (unnotched)
82	—	RT (notched)

Specimen Details: Notched

Notch Type	Gross Width	Net Width	Notch Radius
Edge	2.25	1.500	0.057
Edge	4.10	1.500	0.070
Fillet	2.25	1.500	0.0195

Surface Condition: Electropolished

Reference: 3.2.3.1.8(b), (f), (g), and (h)

Test Parameters:

Loading – Axial
Frequency – 1100 to 1800 cpm
Temperature – RT
Environment – Air

No. of Heats/Lots: Not specified

Equivalent Stress Equation:

$\log N_f = 10.2 - 4.63 \log (S_{eq} - 5.3)$
$S_{eq} = S_{max} (1-R)^{0.51}$
Standard Error of Estimate = 0.51
Standard Deviation in Life = 1.08
R^2 = 78%

Sample Size = 126

[Caution: The equivalent stress model may provide unrealistic life predictions for stress ratios beyond those represented above]

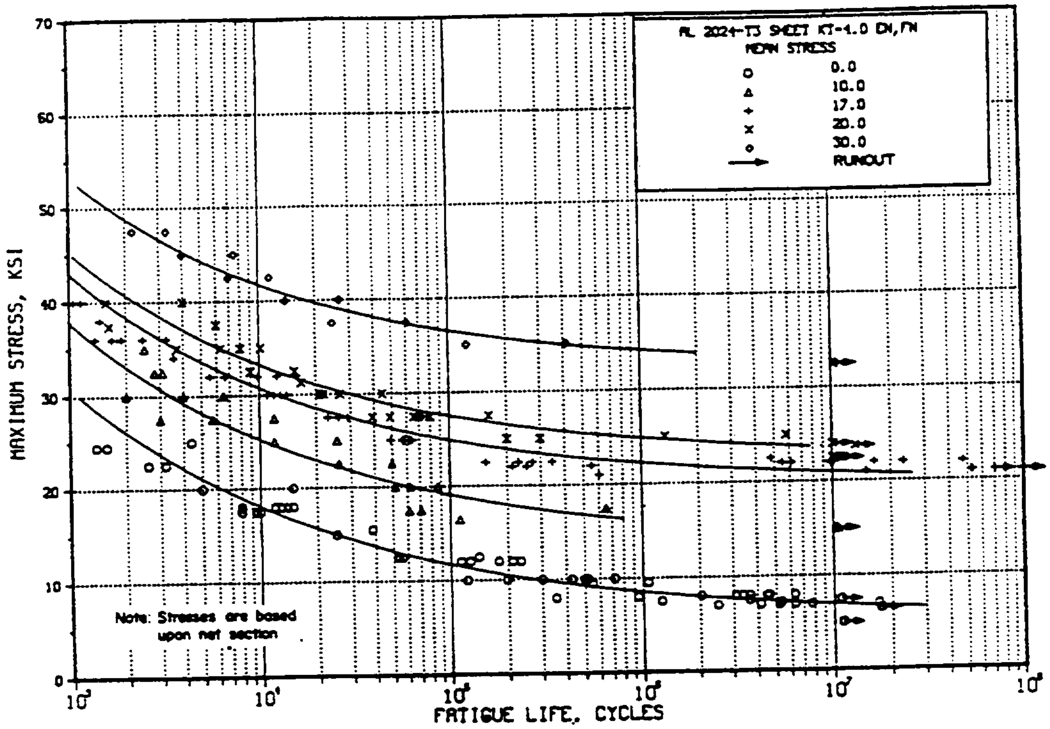


FIGURE 3.2.3.1.8(h). *Best-fit S/N curves for notched, K_t = 4.0 of 2024-T3 aluminum alloy sheet, longitudinal direction.*

Correlative Information for Figure 3.2.3.1.8(h)

Product Form: Bare sheet, 0.090 inch

Properties:	TUS, ksi	TYS, ksi	Temp., F
	73	54	RT (unnotched)
	67	—	RT (notched, K_t = 4.0)

Specimen Details: Notched, K_t = 2.0

Notch Type	Gross Width	Net Width	Notch Radius
Edge	2.25	1.50	0.057
Edge	4.10	1.50	0.070
Fillet	2.25	1.50	0.0195

Surface Condition: Electropolished, machined, and burrs removed with fine crocus cloth

Reference: 3.2.3.1.8(b), (e), (f), (g), and (h)

Test Parameters:

Loading - Axial
Frequency - 1100 to 1800 cpm
Temperature - RT
Environment - Air

No. of Heats/Lots: Not specified

Equivalent Stress Equation:

$\text{Log } N_f = 8.3 - 3.30 \log (S_{eq} - 8.5)$
$S_{eq} = S_{max} (1-R)^{0.66}$
Standard Error of Estimate = 0.39
Standard Deviation in Life = 1.24
R^2 = 90%

Sample Size = 126

[Caution: The equivalent stress model may provide unrealistic life predictions for stress ratios beyond those represented above]

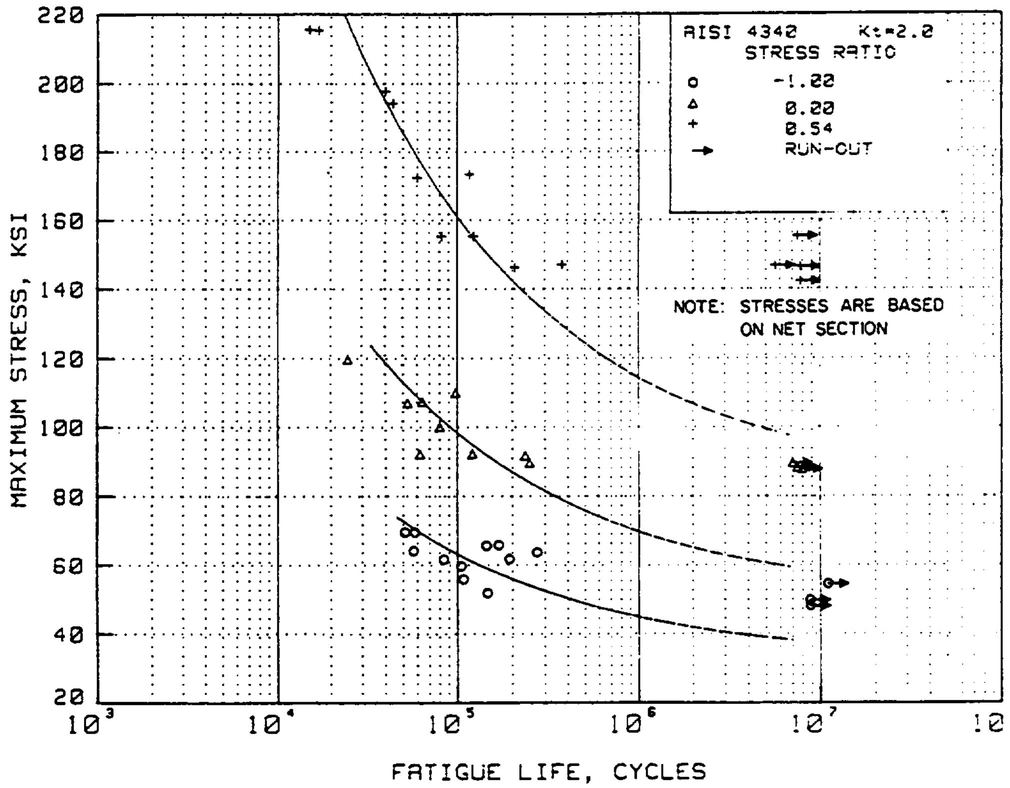


FIGURE 2.3.1.3.8(n). *Best-fit S/N curves for notched, K_t = 2.0, AISI 4340 alloy steel bar, F_{tu} = 260 ksi, longitudinal direction.*

Correlative Information for Figure 2.3.1.3.8(n)

Product Form: Rolled bar, 1-1/8 inches diameter, air melted

Properties:

	TUS, ksi	TYS, ksi	Temp., F
	266	232	RT (unnotched)
	390	—	RT (notched)

Specimen Details: Notched, V-Groove, K_t = 2.0
0.300-inch gross diameter
0.220-inch net diameter
0.030-inch root radius, r
60° flank angle, ω

Surface Condition: Lathe turned to RMS 10

Reference: 2.3.1.3.8(a)

Test Parameters:

Loading - Axial
Frequency - 2000 to 2500 cpm
Temperature - RT
Atmosphere - Air

No. of Heats/Lots: 1

Equivalent Stress Equation:

$\log N_f = 9.46 - 2.65 \log (S_{eq} - 50.0)$
$S_{eq} = S_{max} (1-R)^{0.64}$
Standard Error of Estimate = 0.22
Standard Deviation in Life = 0.34
R^2 = 58%

Sample Size = 30

[Caution: The equivalent stress model may provide unrealistic life predictions for stress ratios beyond those represented above]

Empty Weight Estimation

Gross Weight = 1560 lbs

$W_e/W_o = A W_o^{C}$ where $A = 2.36$ and $C = -0.18$

$$W_e = \left(\frac{W_e}{W_o}\right) W_o = 2.36\,(1560)^{-0.18}\,(1560)$$

W_e = 980 lbs

Fuel Weight Estimation

$$W_o = W_{crew} + W_{Payload} + W_{fuel} + W_{empty}$$

$$1560 = 350 + 50 + W_{fuel} + 980$$

W_{fuel} = 180 lbs

Payload Trade-off

$$W_o = \frac{200}{0.883 - 2.36\,W_o^{-0.18}}$$

200 lbs Payload

1300 = 855
1100 = 935
1000 = 988

$$W_o = \frac{600}{0.883 - 2.36\,W_o^{-0.18}}$$

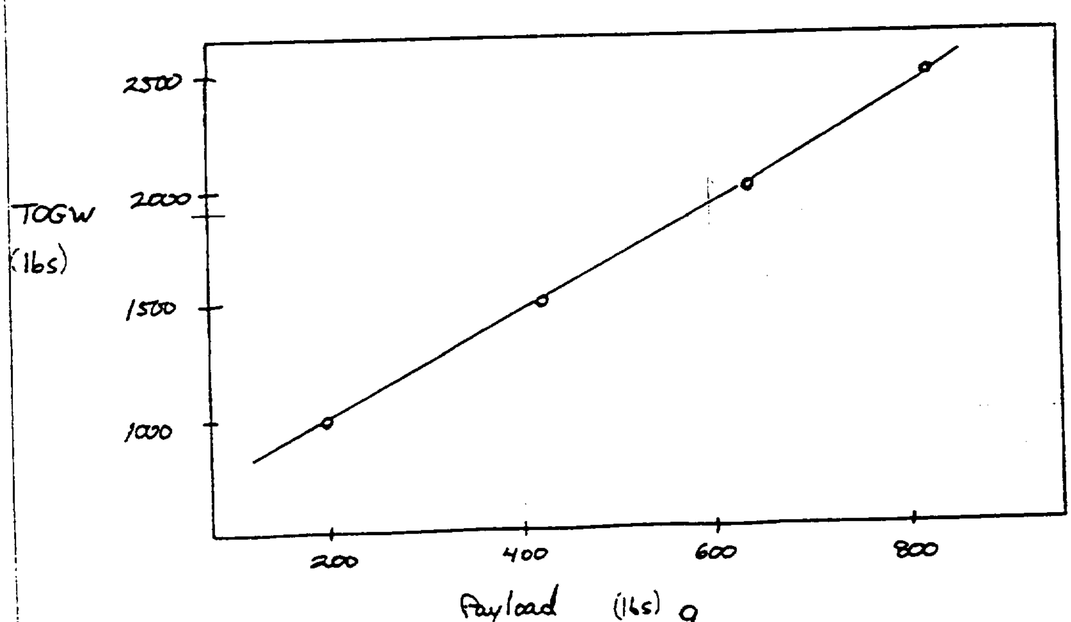
600 lbs Payload

1800 = 2216
1900 = 2169
2020 = 2118
2100 = 2087

$$W_o = \frac{800}{0.883 - 2.36\,W_o^{-0.18}}$$

800 lbs Payload

2400 = 2652
2500 = 2616
2575 = 2590



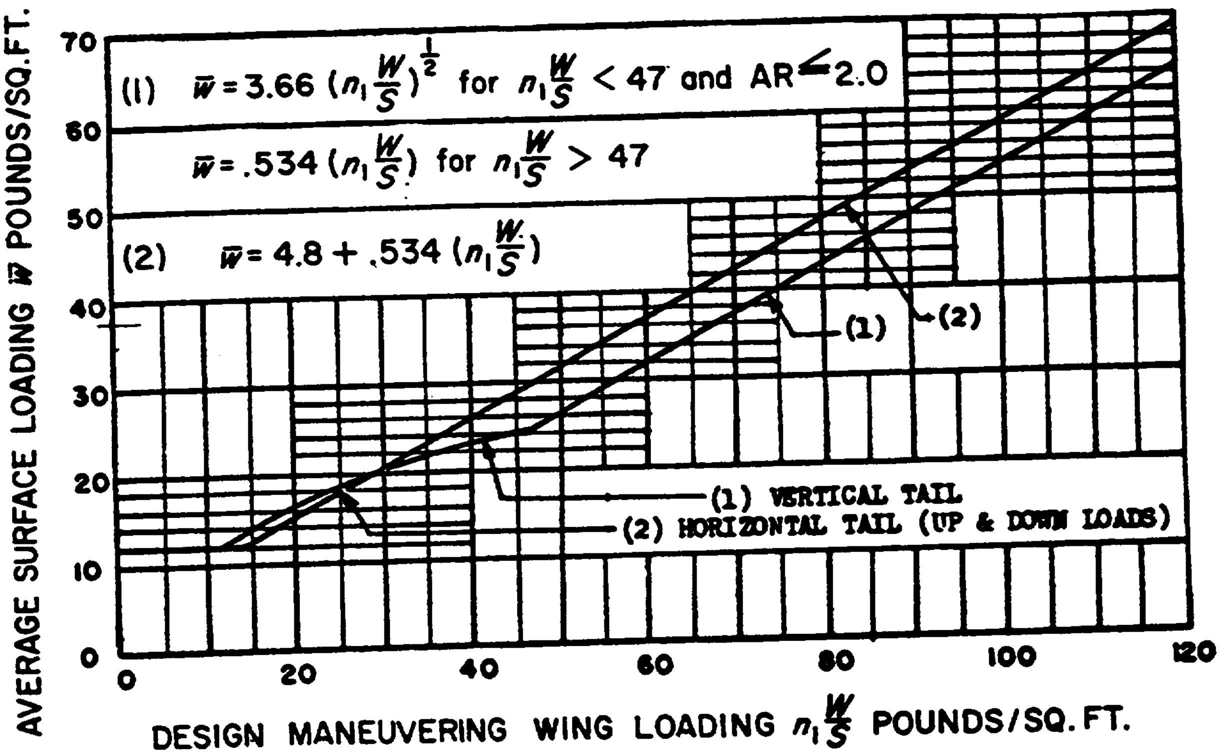


FIGURE A5—Average limit control surface loading.

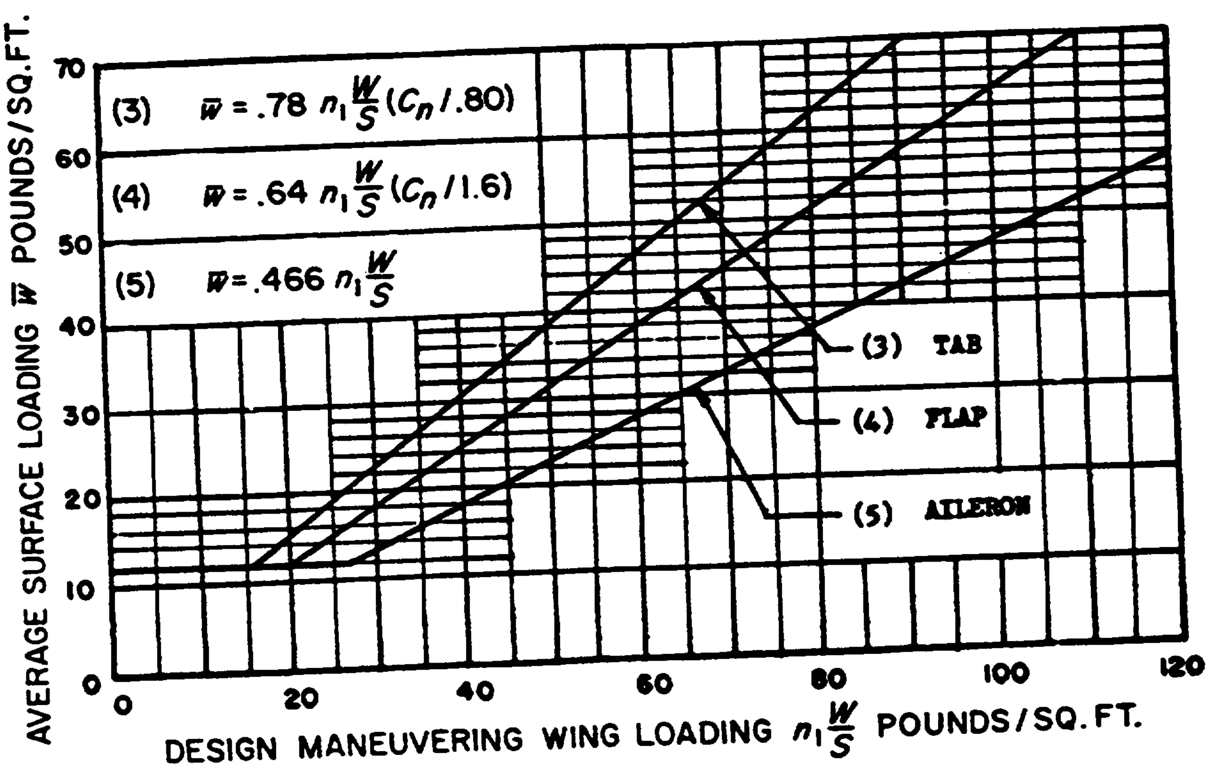
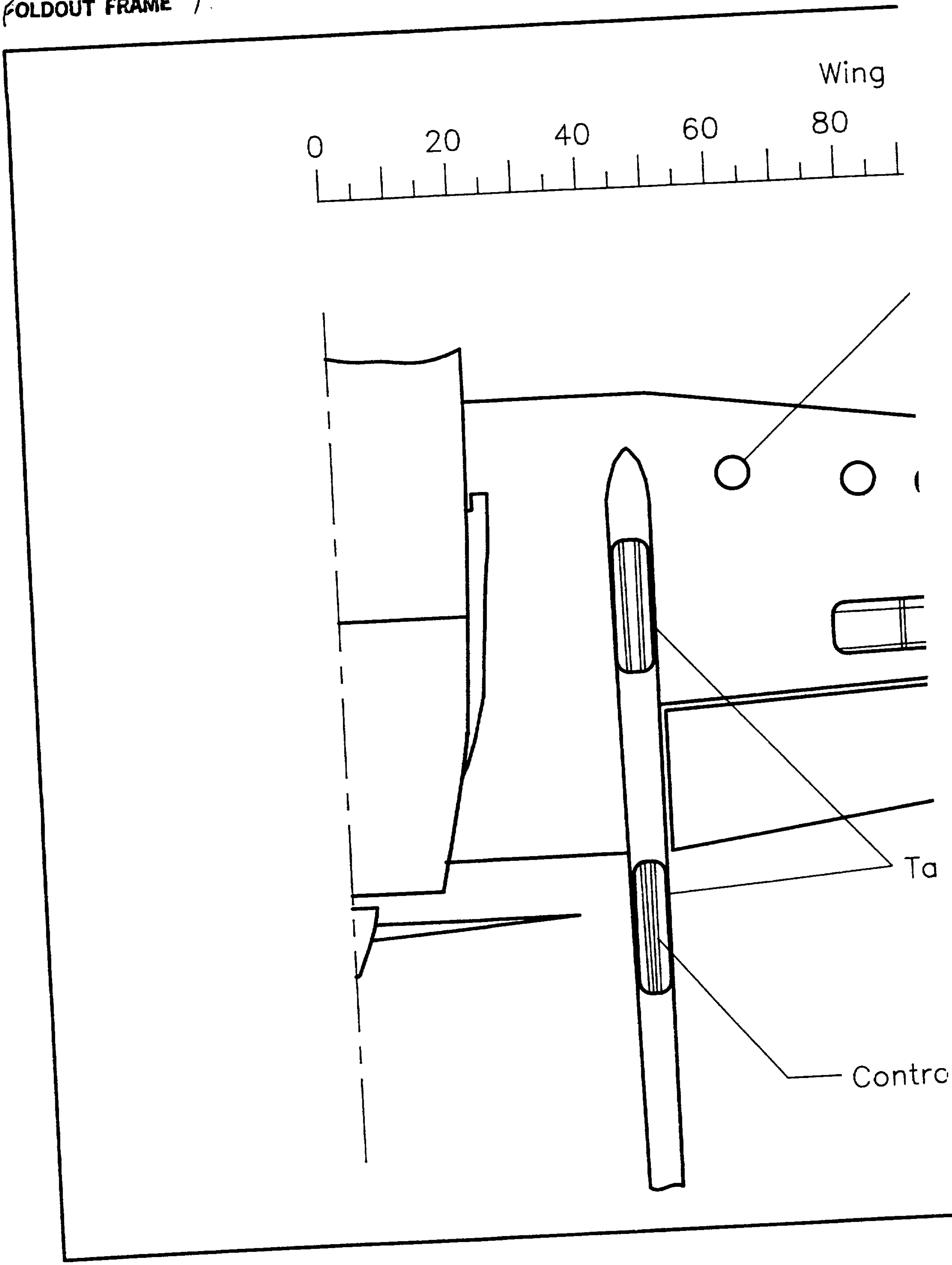
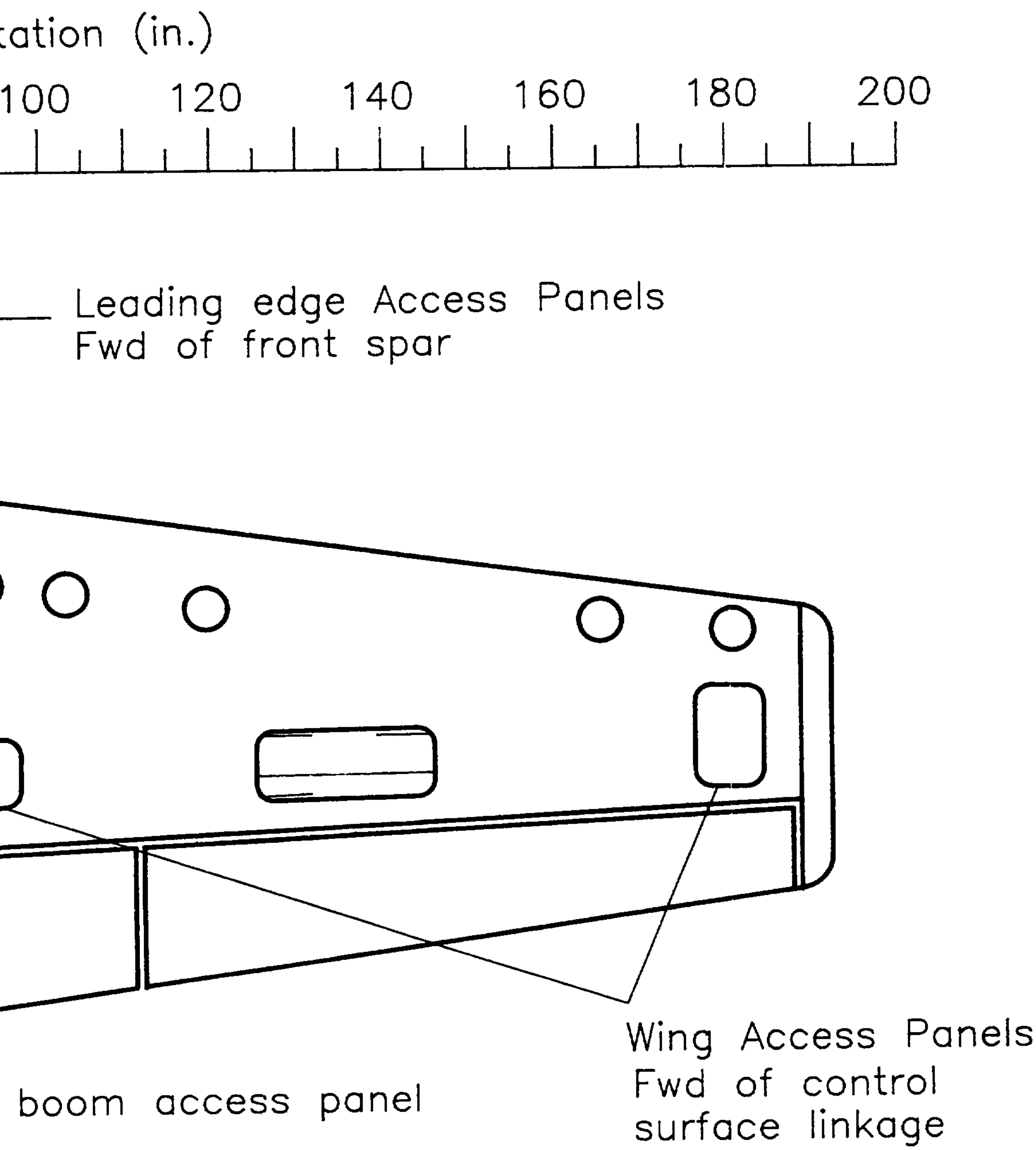


FIGURE A6—Average limit control surface loading.

FOLDOUT FRAME 1

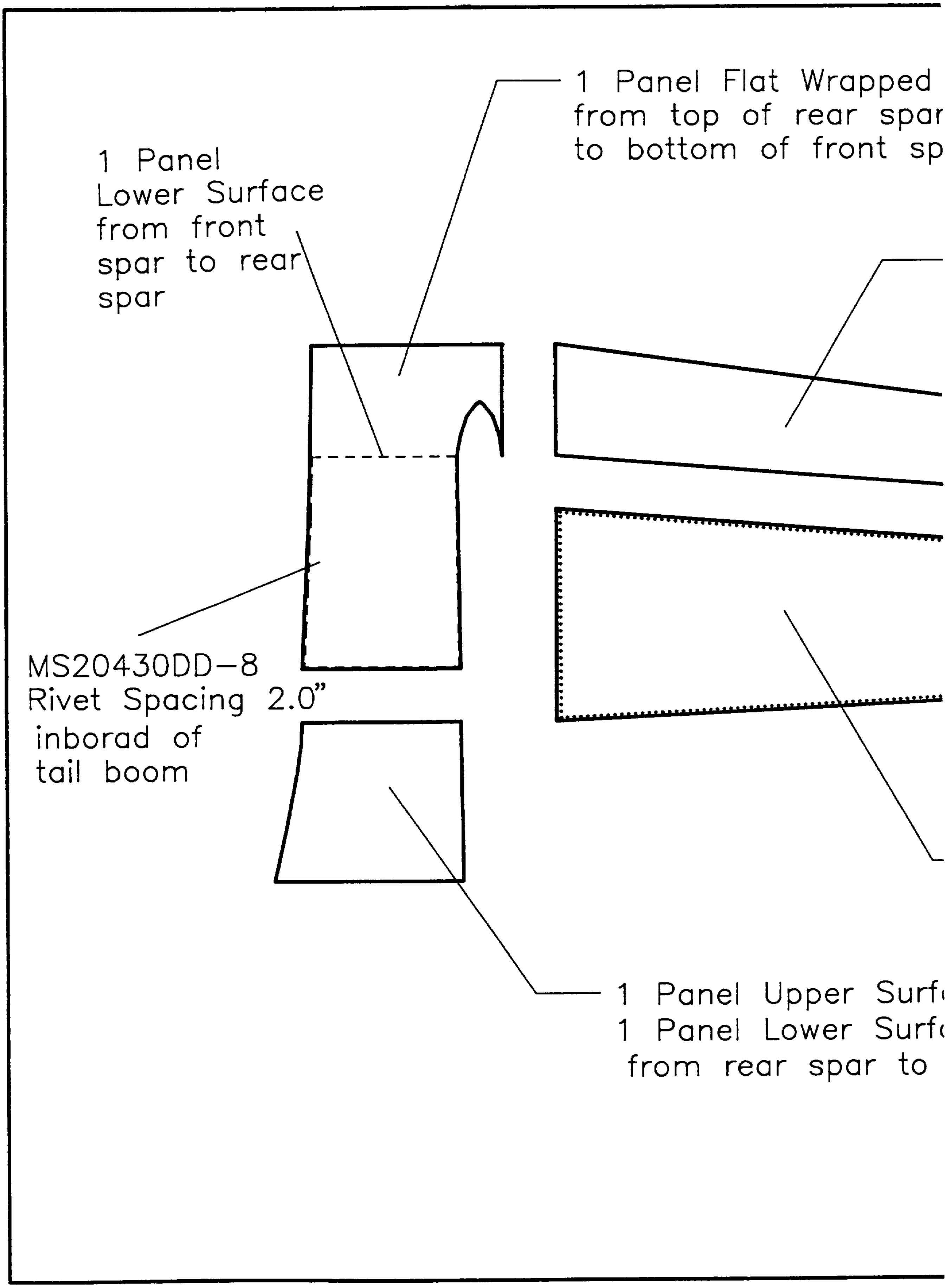


FOLDOUT FRAME 2

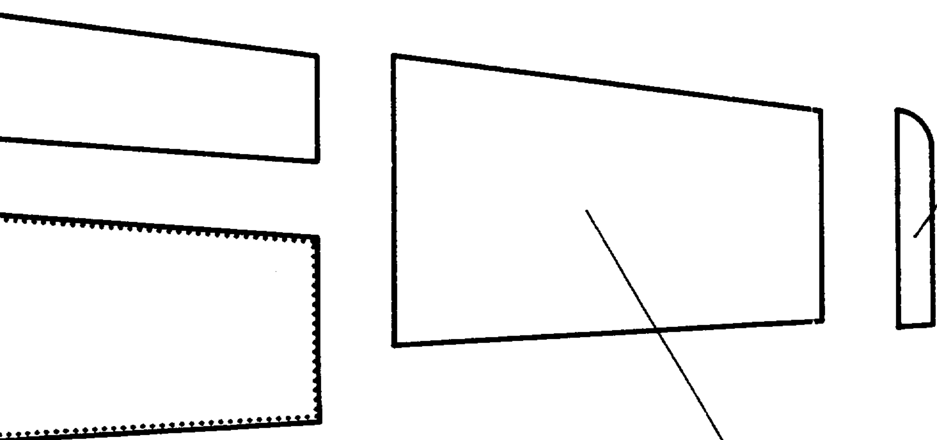


Routing

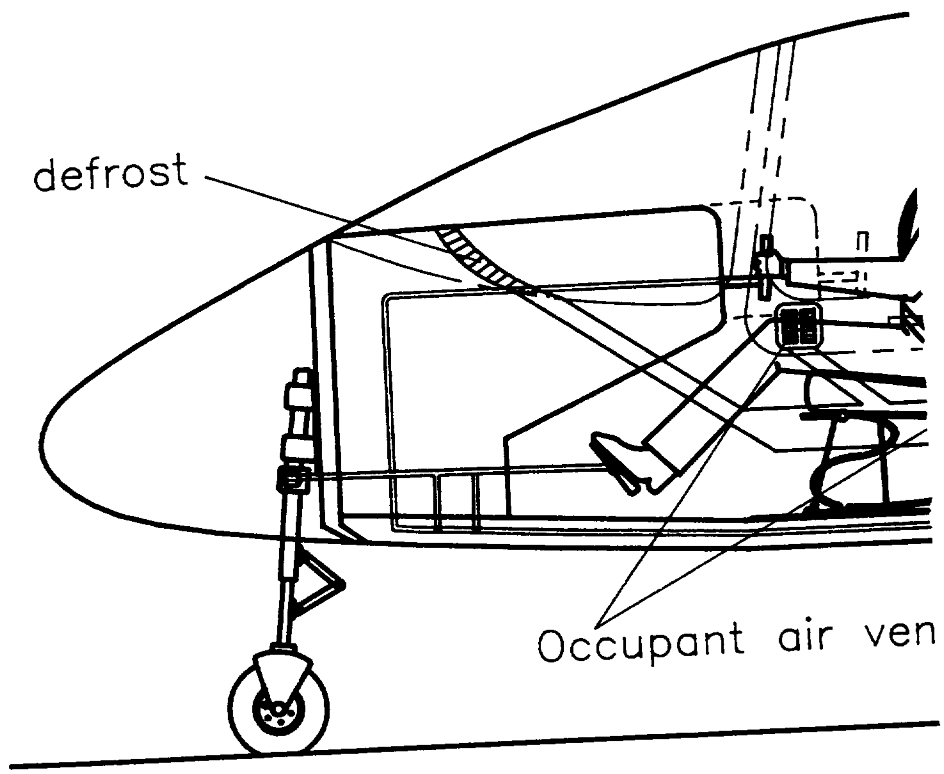
DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
DECIMAL .XX ± .01 .XXX ± .001 ANGULAR ± 1/2°	SIZE B	DATE 12-06	SCALE 1/20	DRAWN BY A. CEPEDA
	TITLE WING ACCESS PANEL LAYOUT			
	DRAWING NO. F93-2A-107-07			SHEET 1/1


1 Panel Flat Wrapped
from top of rear spar
to bottom of front sp
1 Panel
Lower Surface
from front
spar to rear
spar
MS20430DD-8
Rivet Spacing 2.0"
inborad of
tail boom
1 Panel Upper Surf
1 Panel Lower Surf
from rear spar to

NOTE: All wing panels fabricated of 2024–T3 Aluminum. 0.032" skin thickness outboard of tail boom. 0.05" skin thickness inboard of tail boom. No control surfaces show.



DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
DECIMAL .XX ± .01 .XXX ± .001	SIZE B	DATE 12–06	SCALE 1/20	DRAWN BY J. VIEIRA
ANGULAR ± 1/2°	TITLE WING SKIN PANEL LAYOUT			
	DRAWING NO. F93–2A–106–07			SHEET 1/1

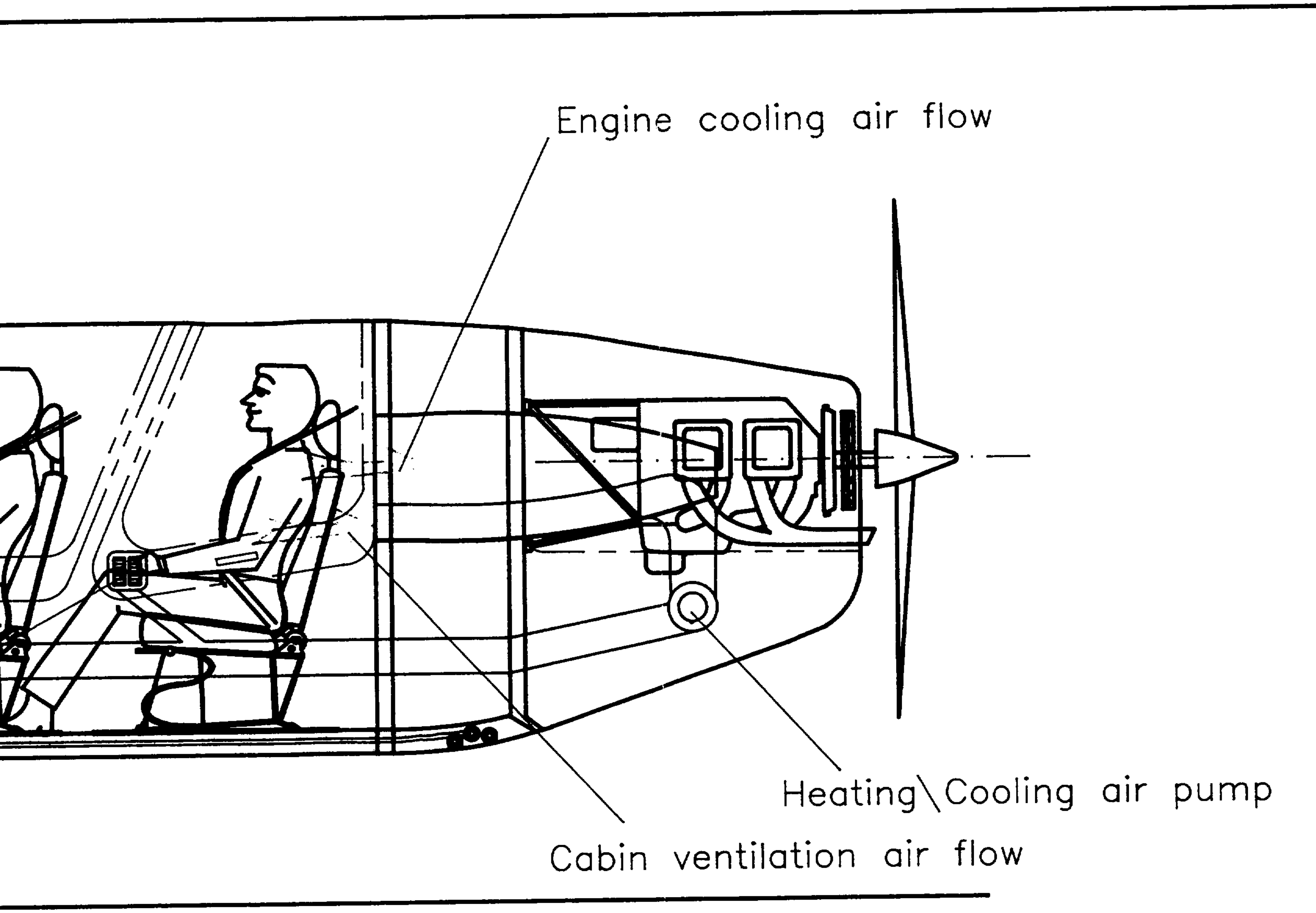


Rudder/Steering

Elevator

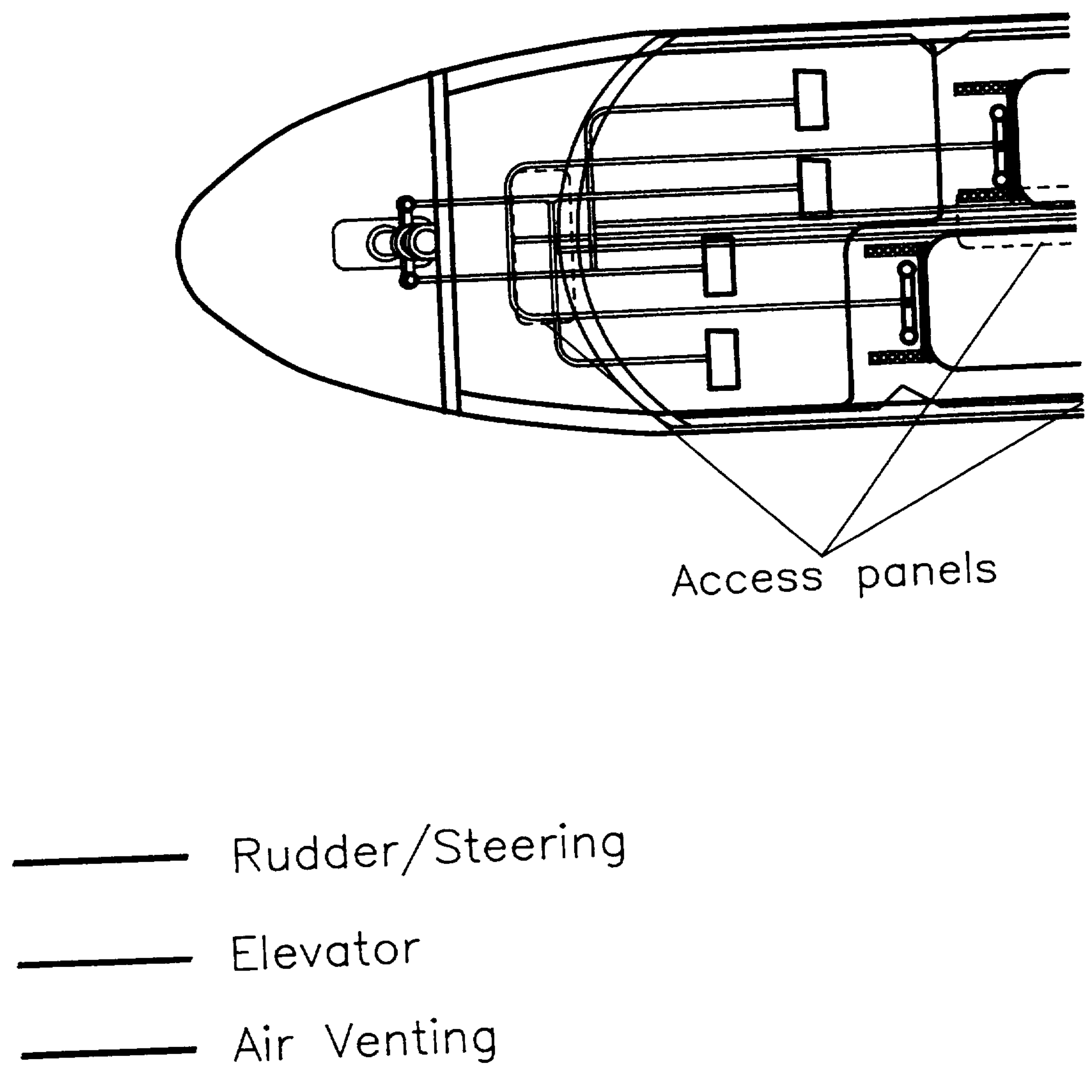
Air Venting

FOLDOUT FRAME 2

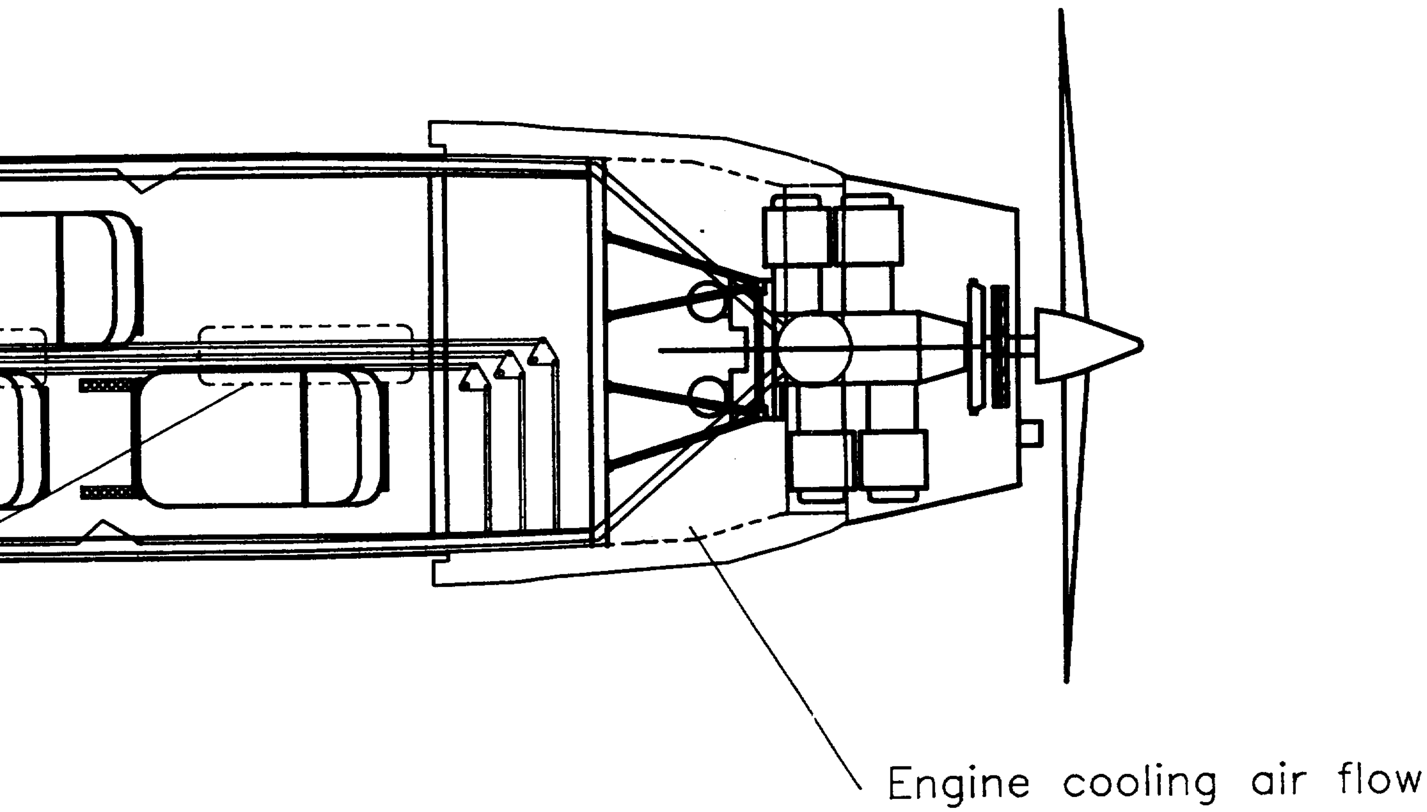


BOARD PROFILE

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL .XX ± .01 .XXX ± .001 ANGULAR ± 1/2°	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
	SIZE B	DATE 10-13	SCALE 1/20	DRAWN BY VIEIRA & CEPEDA
	TITLE CONTROL SYSTEMS LAYOUT			
	DRAWING NO. F93-2A-105-07			SHEET 1/2

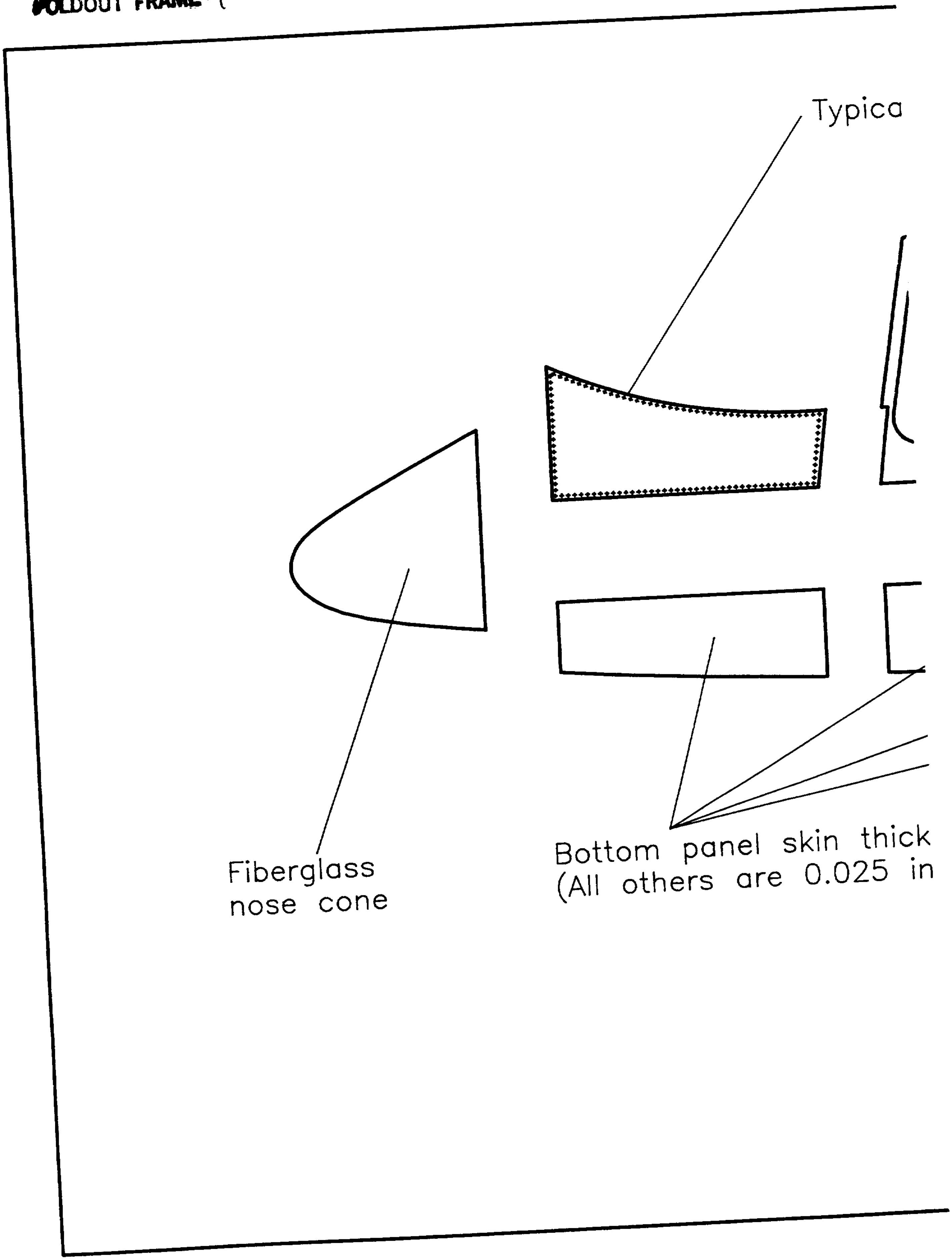

Access panels
Rudder/Steering
Elevator
Air Venting

FOLDOUT FRAME 2

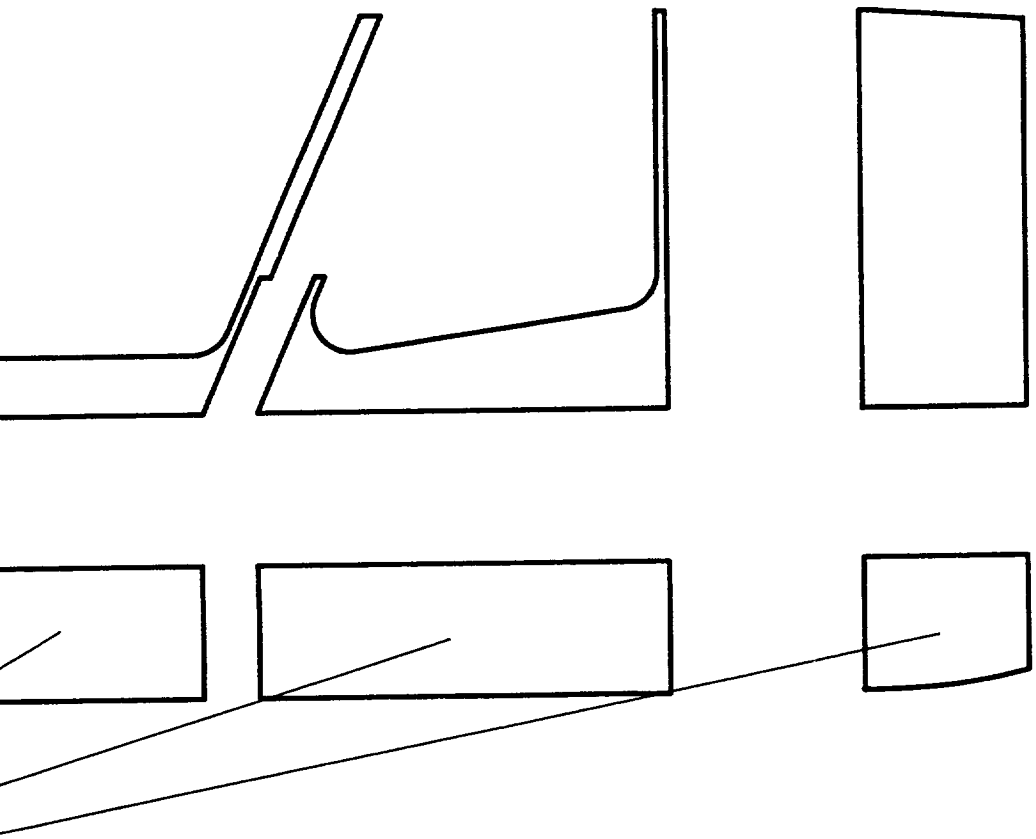


DATE	DWG BY	DRAWING NO.	SHEET
10–13	VIEIRA&CEPEDA	F93–2A–105–07	2/2

FOLDOUT FRAME (



rivet spacing is 1 in



ess is 0.040 in

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	10-13	1/20	A. CEPEDA

TITLE
SKIN PANEL LAYOUT

DRAWING NO.	SHEET
F93-2A-103-07	1/1

			2024–
43	56	MS20430DD–12	2024–
42	488	MS20430DD–5	2024–
41	320	MS20430DD–2	2024–
40	680	MS20430DD–8	2024–
39	2090	MS20426DD–5	2024–
38	4	NAS 1307–15P	STANDAR
37	4	NAS 1308–4P	STANDAR
36	8	NAS 1305–1P	STANDAR
35	4	AN 12–17–J	STANDAR
34	64	NAS 1304–4P	STANDAR
33	2	INBOARD BOTTOM WING PANEL	2024–
32	4	INBOARD TRAILING EDGE PANEL	2024–
31	2	INBOARD LEADING EDGE PANEL	2024–
30	4	CENTER WING SKIN PANEL	2024–
29	2	LEADING EDGE WING SKIN PANEL	2024–
28	2	OUTBOARD WING SKIN PANEL	2024–
27	2	PRESS FORM WING TIPS	2024–
26	1	REAR CARRY THROUGH	2024–
25	1	FRONT CARRY THROUGH	2024–
24	4	REAR LUG	2024–
23	4	FRONT LUG	2024–
22	4	SPAR SHR WEB 0.016" THICK	2024–
21	4	SPAR SHR WEB 0.032" THICK	2024–

ITEM	QTY	DESCRIPTION	MAT'L OR PART #
20	4	SPAR SHR WEB 0.05" THICK	2024-T3
19	4	SPAR SHR WEB 0.125" THICK	2024-T3
18	4	BUCKLE STIFFNER (#12)	2024-T3
17	12	BUCKLE STIFFNER (#11)	2024-T3
16	28	BUCKLE STIFFNER (#10)	2024-T3
15	44	BUCKLE STIFFNER (#9)	2024-T3
14	2	REAR SPAR CAP NAS-344-02	2024-T3
13	2	REAR SPAR CAP NAS-344-32	2024-T3
12	2	REAR SPAR CAP NAS-344-33	2024-T3
11	2	FRONT SPAR CAP NAS-344-10	2024-T3
10	2	FRONT SPAR CAP NAS-344-30	2024-T3
9	2	FRONT SPAR CAP NAS-344-69	2024-T3
8	10	WING RIB 72.0" CHORD	2024-T3
7	2	WING RIB 65.0" CHORD	2024-T3
6	2	WING RIB 61.5" CHORD	2024-T3
5	2	WING RIB 58.0" CHORD	2024-T3
4	2	WING RIB 51.4" CHORD	2024-T3
3	2	WING RIB 44.7" CHORD	2024-T3
2	2	WING RIB 38.0" CHORD	2024-T3
1	2	WING RIB 33.3" CHORD	2024-T3

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01
.XXX ± .001

ANGULAR
± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

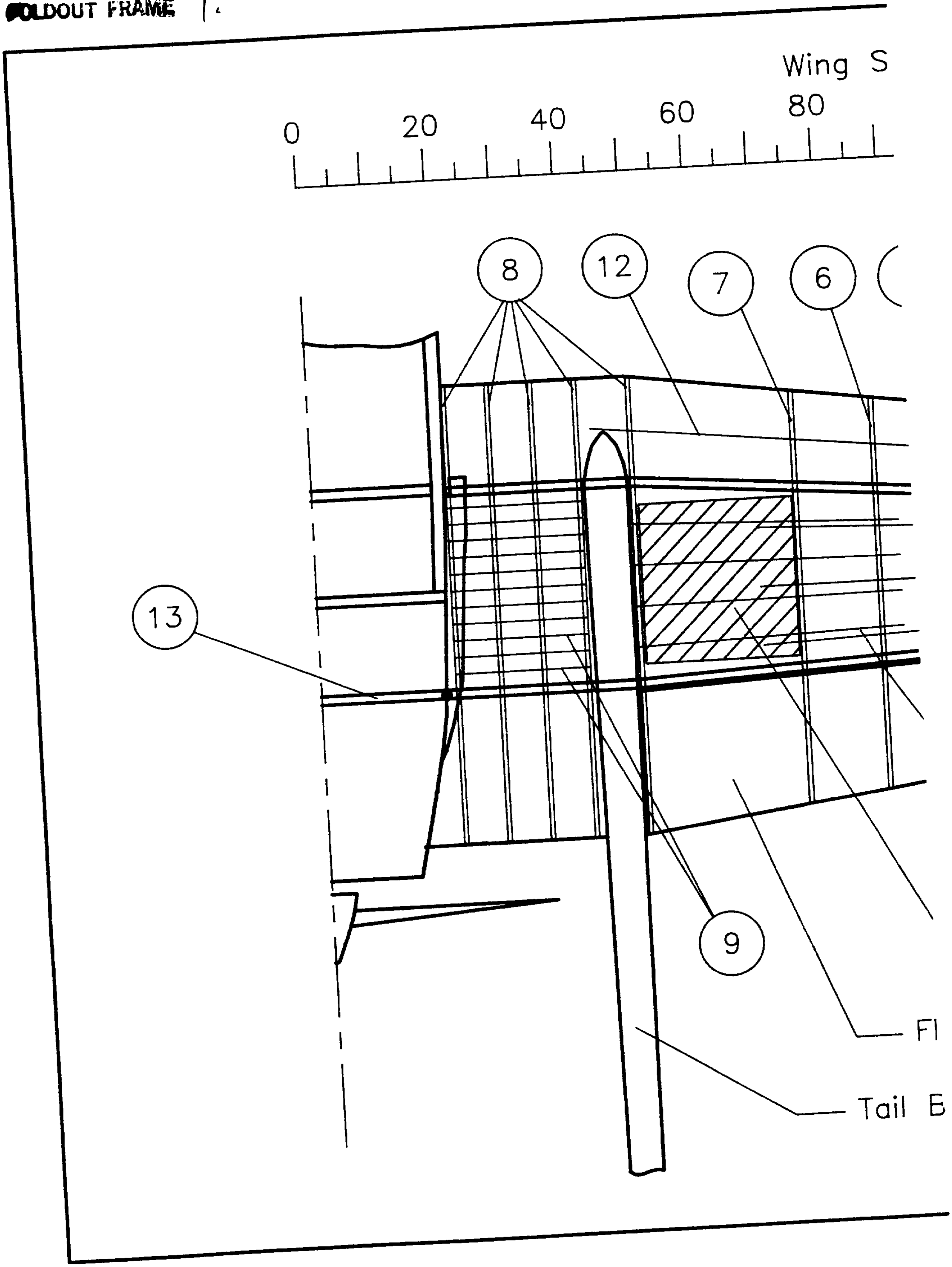
SIZE	DATE	SCALE	DRAWN BY
B	12-06	AS INDICATED	ALPHA TEAM

TITLE
WING STRUCTURAL LAYOUT

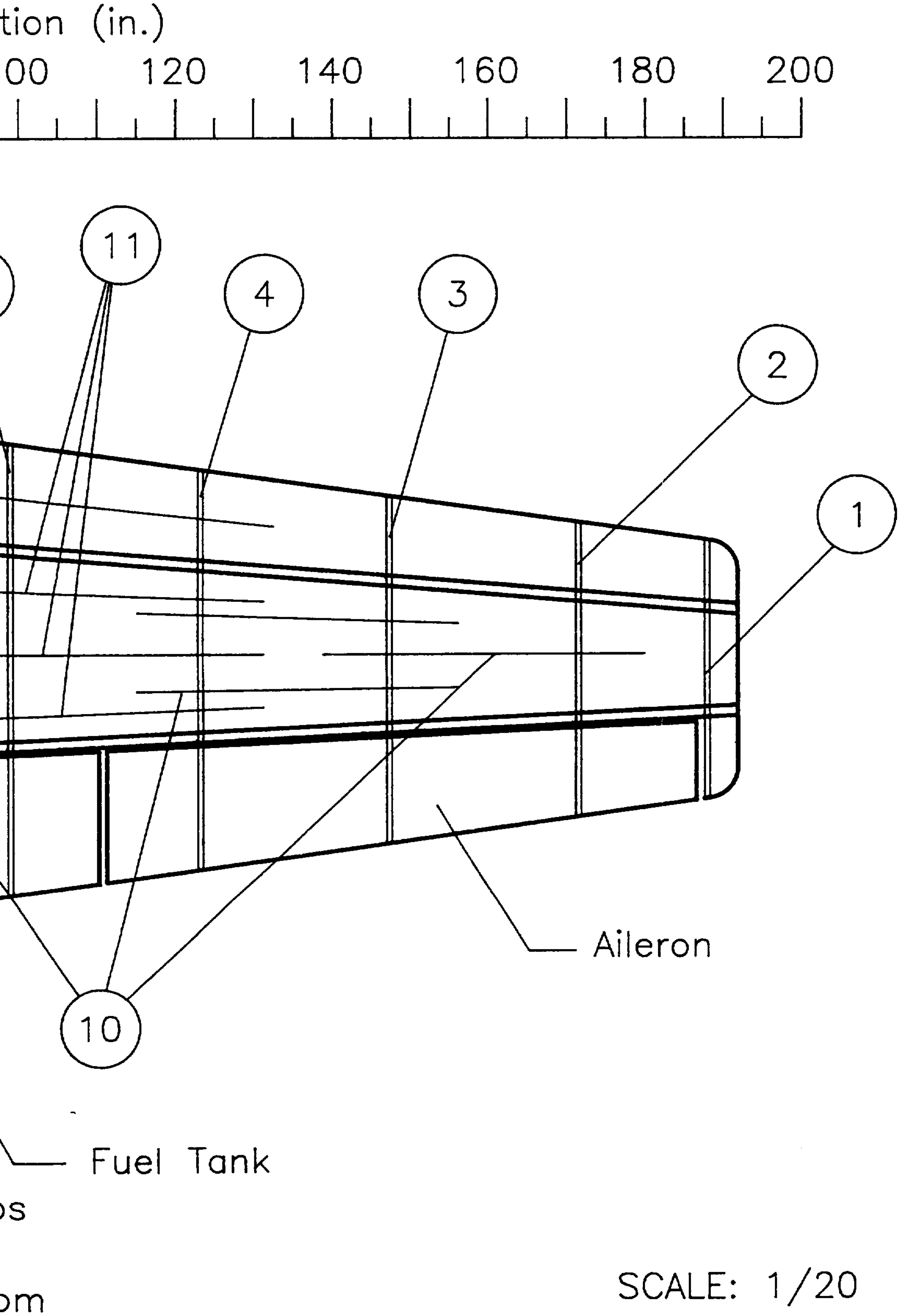
DRAWING NO. F93-2A-104-07

SHEET 1/13

FOLDOUT FRAME 1.



FOLDOUT FRAME 2



SCALE: 1/20

DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-104-07	2/13

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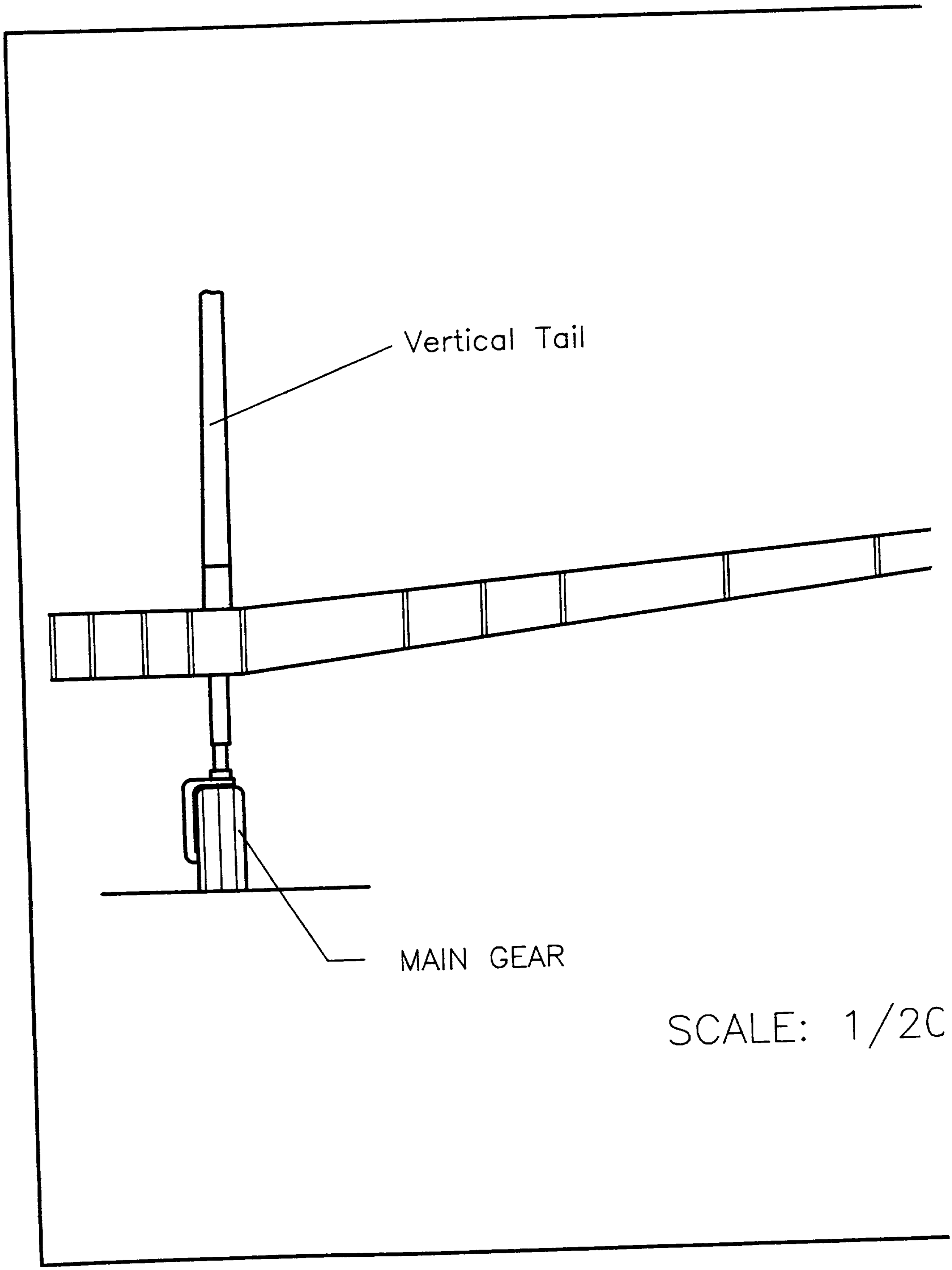
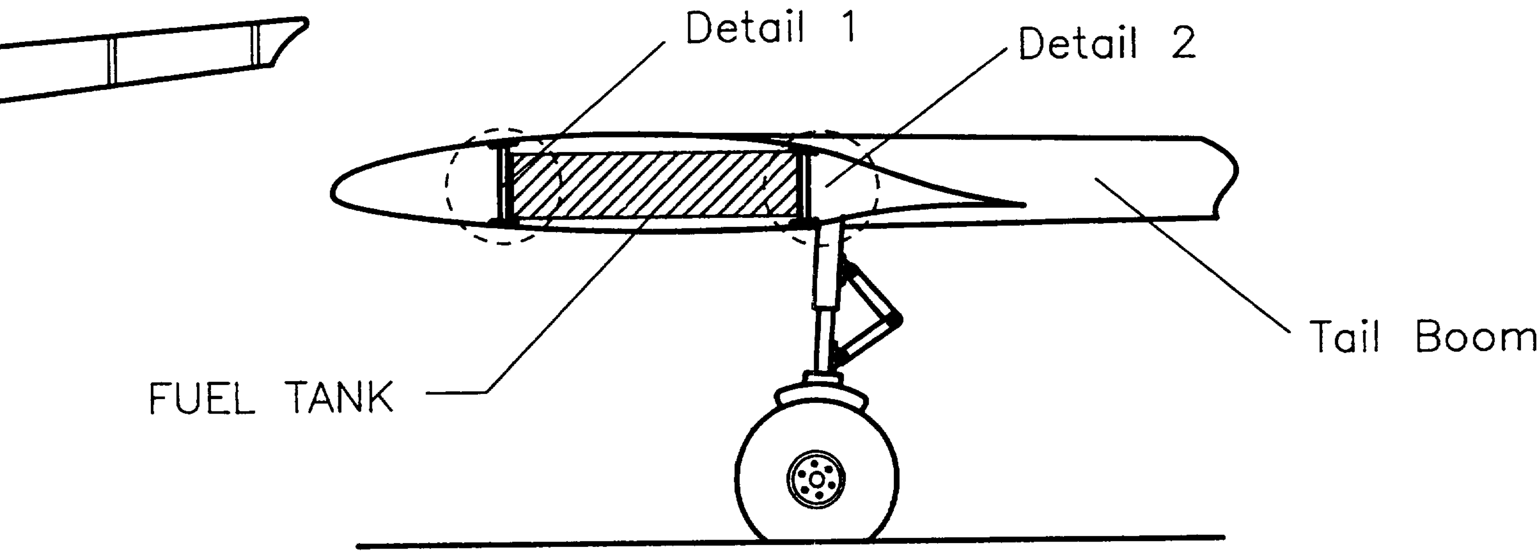
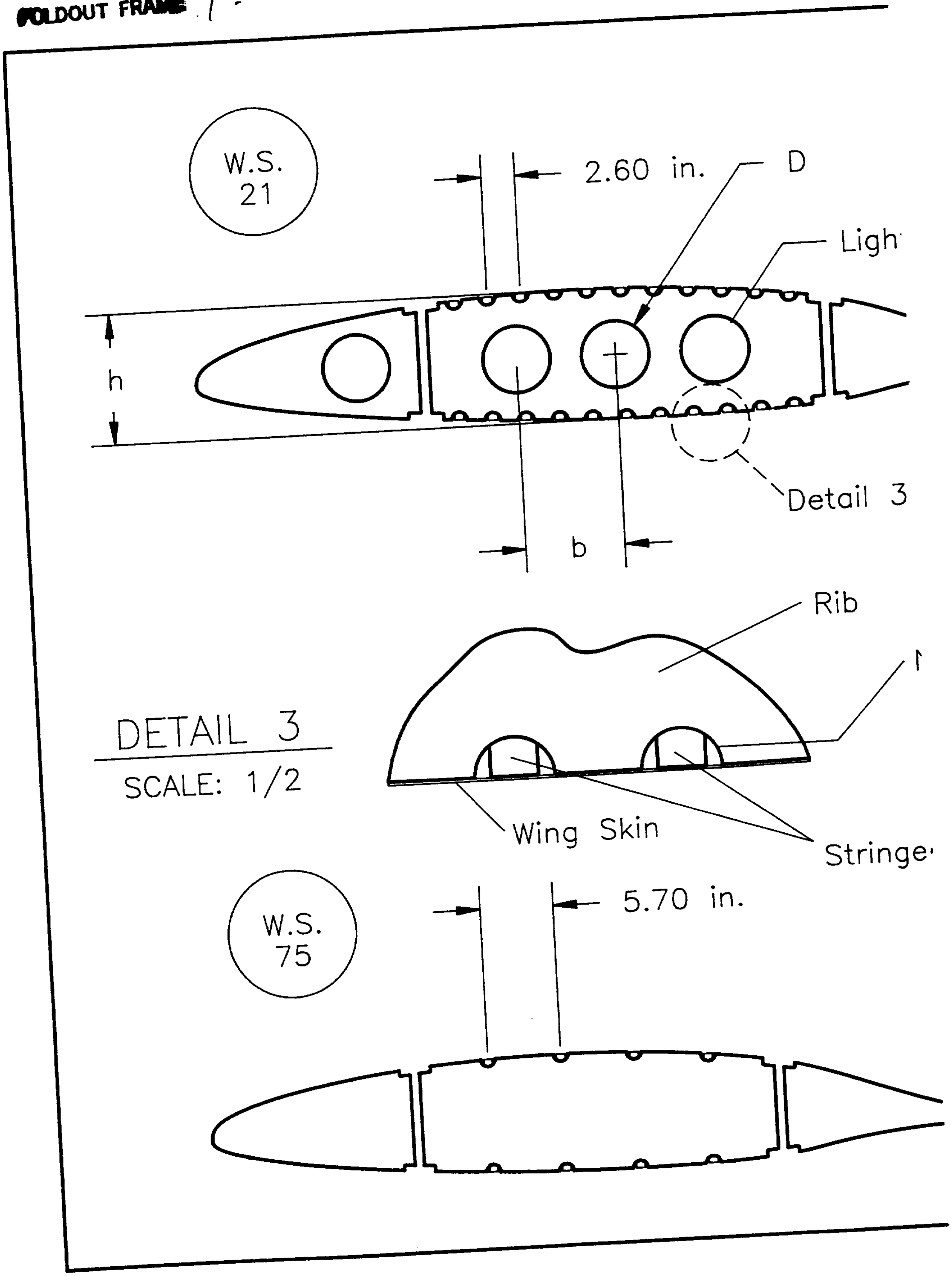


FOLDOUT FRAME 2



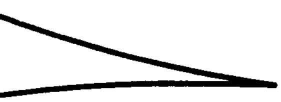
DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	3/13

FOLDOUT FRAME 1



FOLDOUT FRAME 2

ning Hole

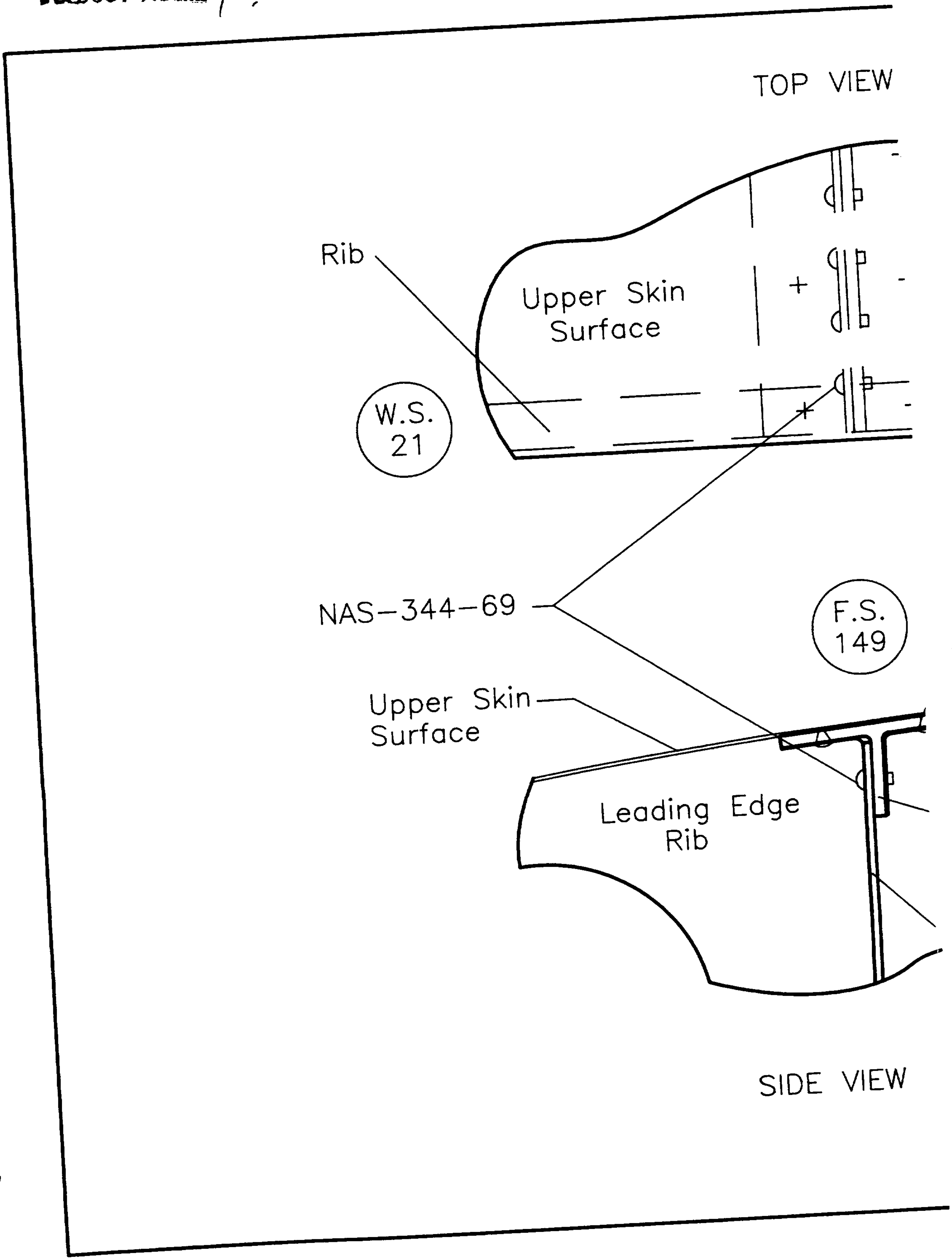


tched Rib

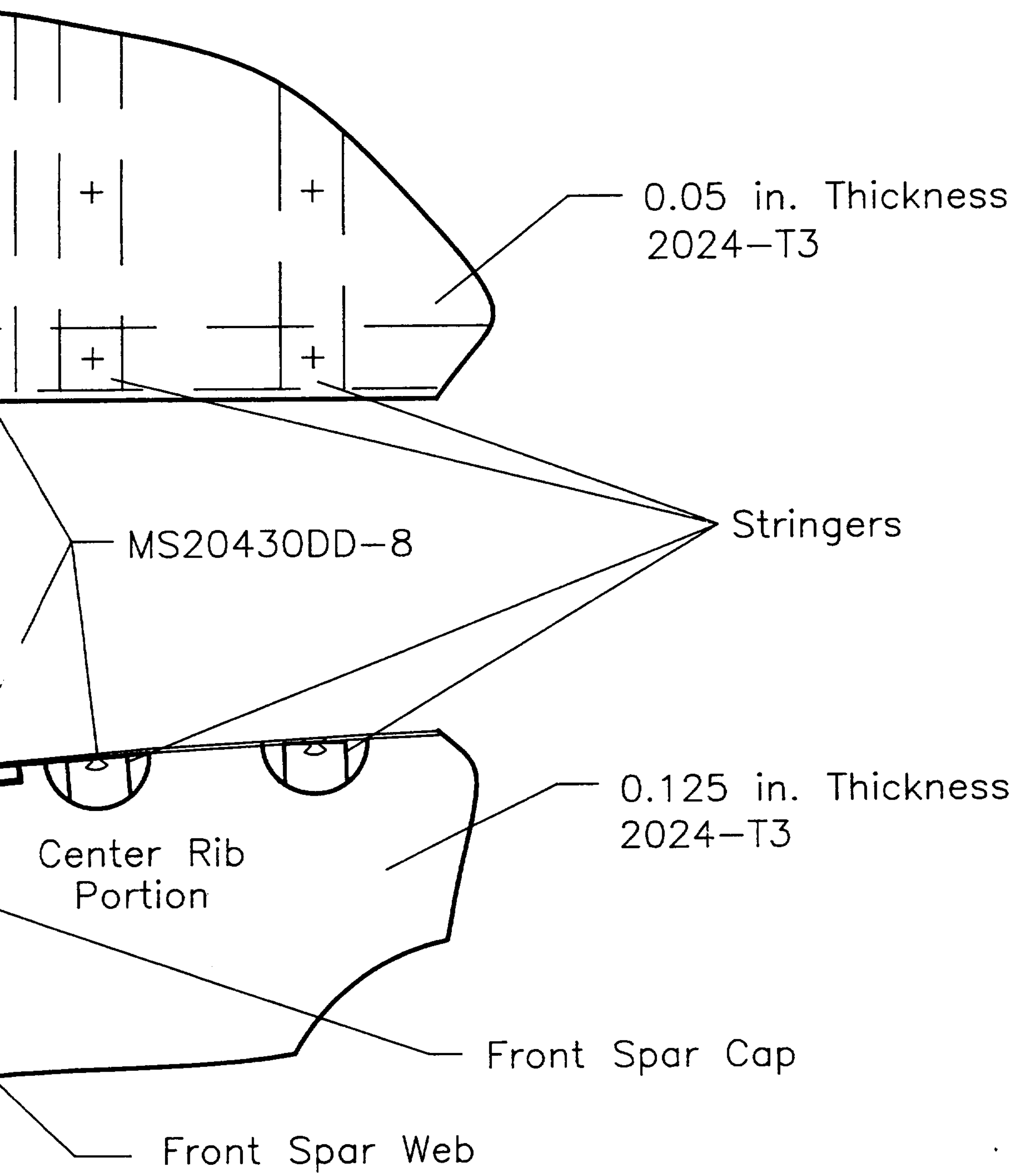
Rib Location and Dimensions for NLF(1)-0414				
W.S. (in.)	Chord (in.)	h (in.)	b (in.)	D (in.)
21.0	72.0	10.08	7.7	5.25
27.0	72.0	10.08	7.7	5.25
35.0	72.0	10.08	7.7	5.25
42.0	72.0	10.08	7.7	5.25
50.0	72.0	10.08	7.7	5.25
75.0	65.0	9.1	--	--
87.0	61.5	8.6	--	--
99.0	58.0	8.1	--	--
124.0	51.4	7.2	--	--
147.0	44.7	6.25	--	--
172.0	38.0	5.3	--	--
187.0	33.3	4.7	--	--

DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	4/13

FOLDOUT FRAME /



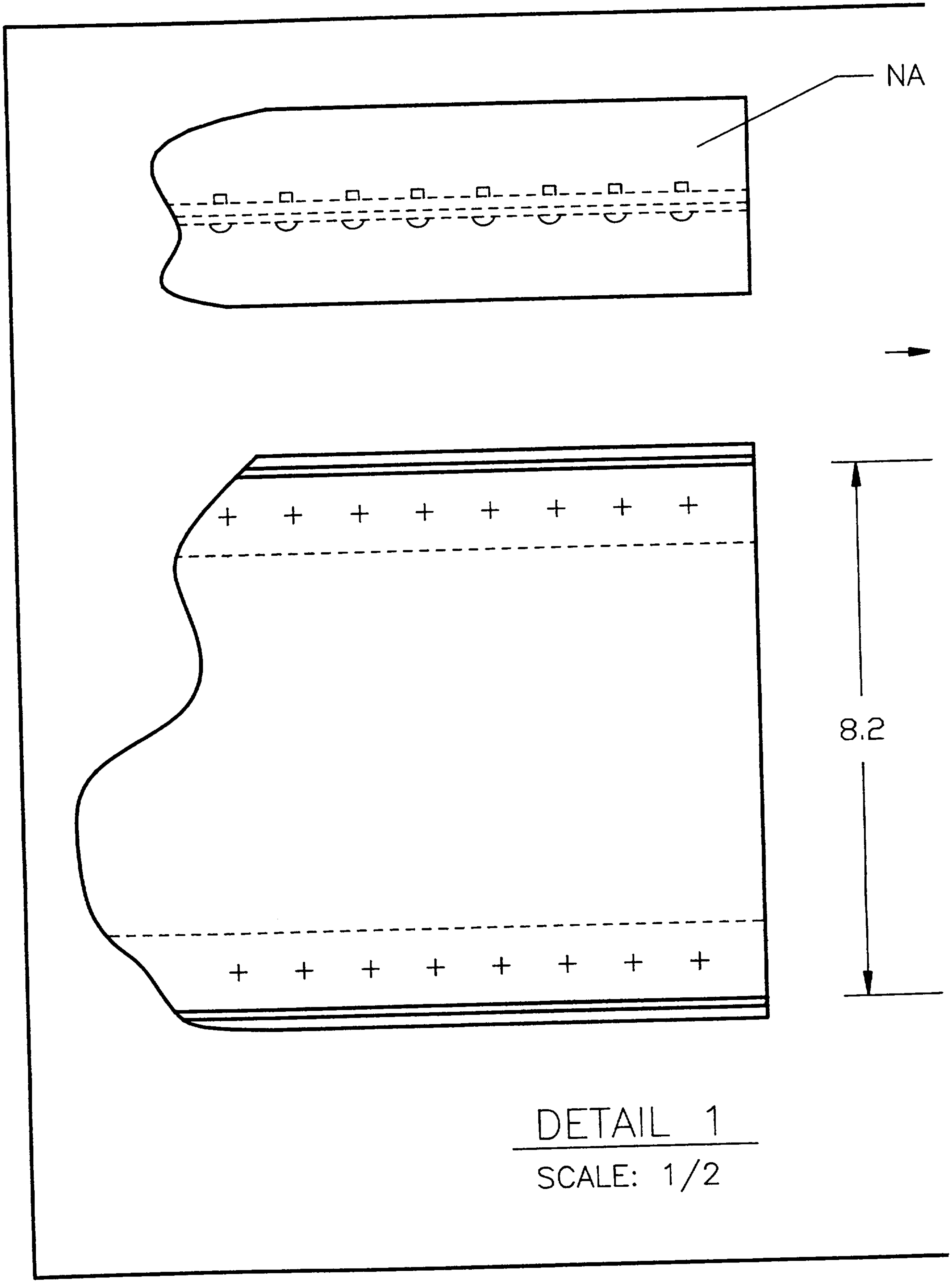
FOLDOUT FRAME 2



SCALE: 1/2

DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	5/13

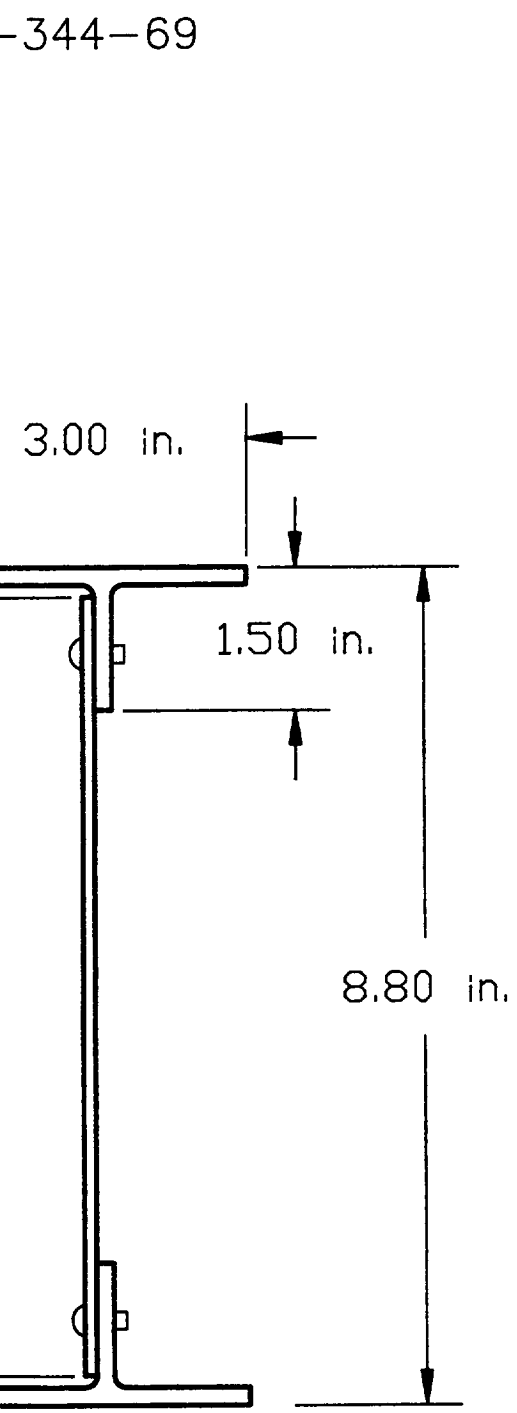
FOLDOUT FRAME 1.



DETAIL 1

SCALE: 1/2

FOLDOUT FRAME 2

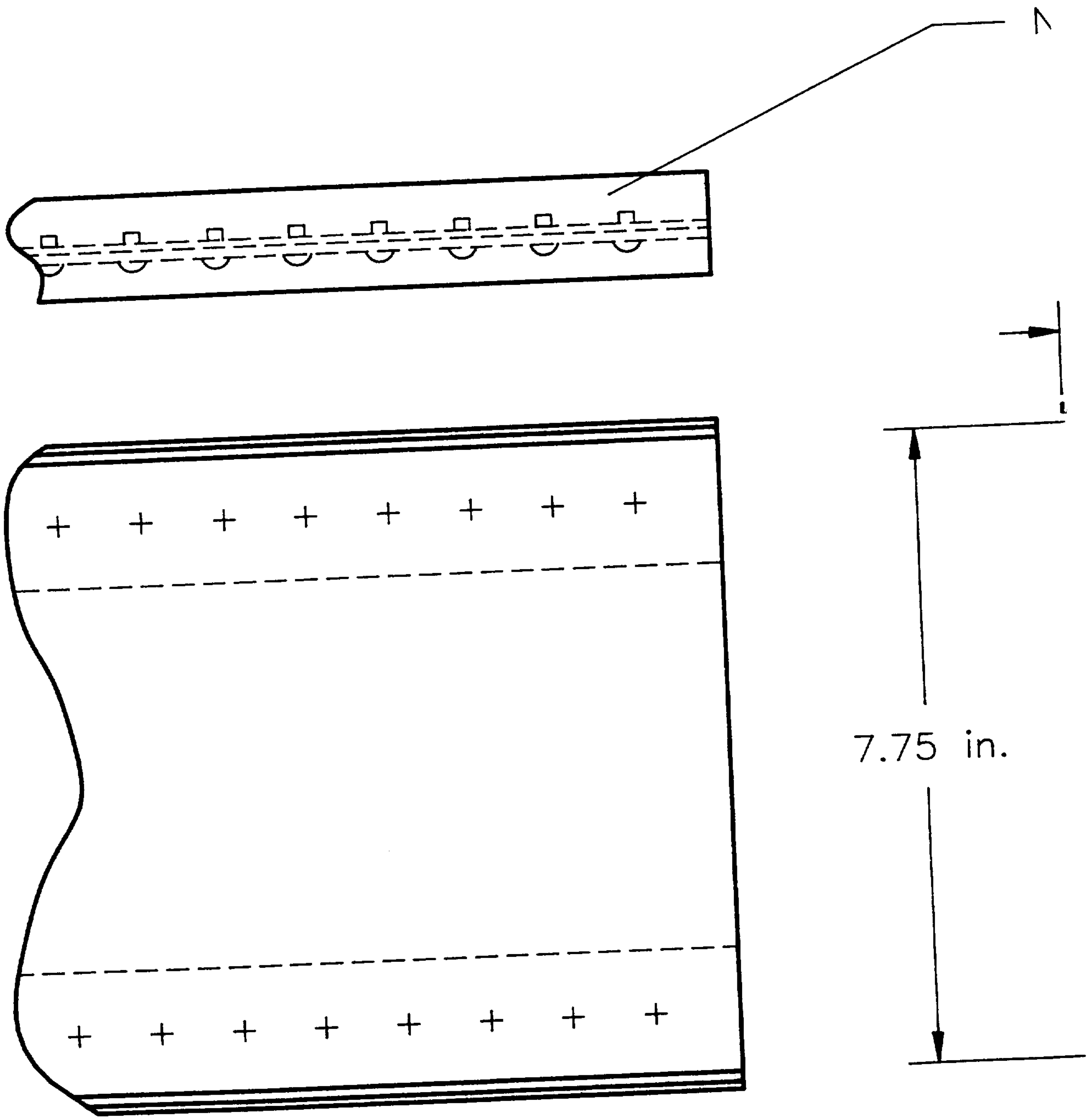


FRONT SPAR CAP SIZE	
Wing Stations	Cap Designation
21 – 101	NAS–344–69
101 – 141	NAS–344–30
141 – 192	NAS–344–10

FRONT SPAR WEB THICKNESS	
Wing Stations	Web Thickness
21 – 41	0.125 in.
41 – 121	0.05 in.
121 – 161	0.032 in.
161 – 192	0.016 in.

DATE	DWG BY	DRAWING NO.	SHEET
12–06	ALPHA TEAM	F93–2A–104–07	6/13

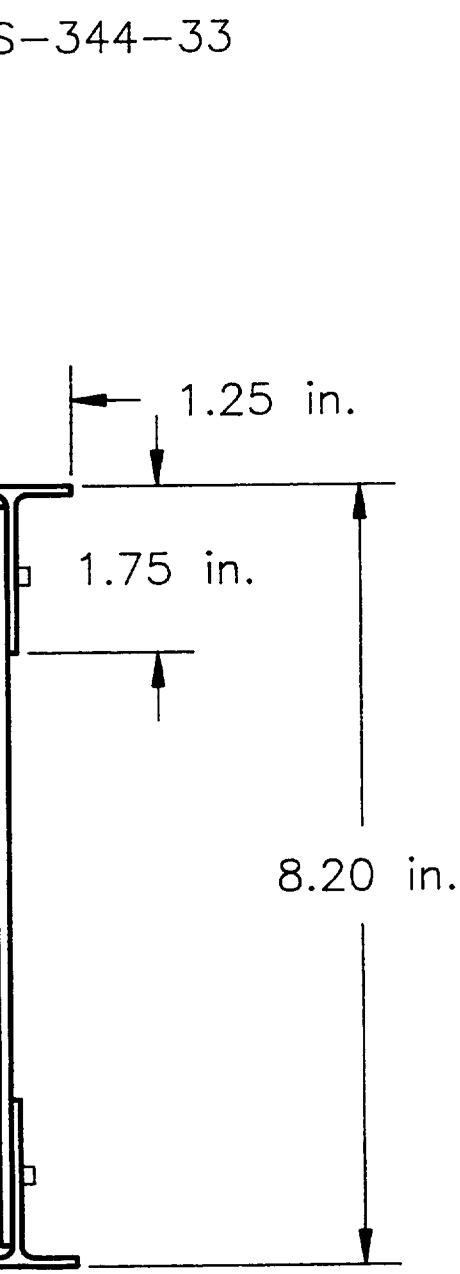
FOLDOUT FRAME 1



DETAIL 2

SCALE: 1/2

FOLDOUT FRAME 2

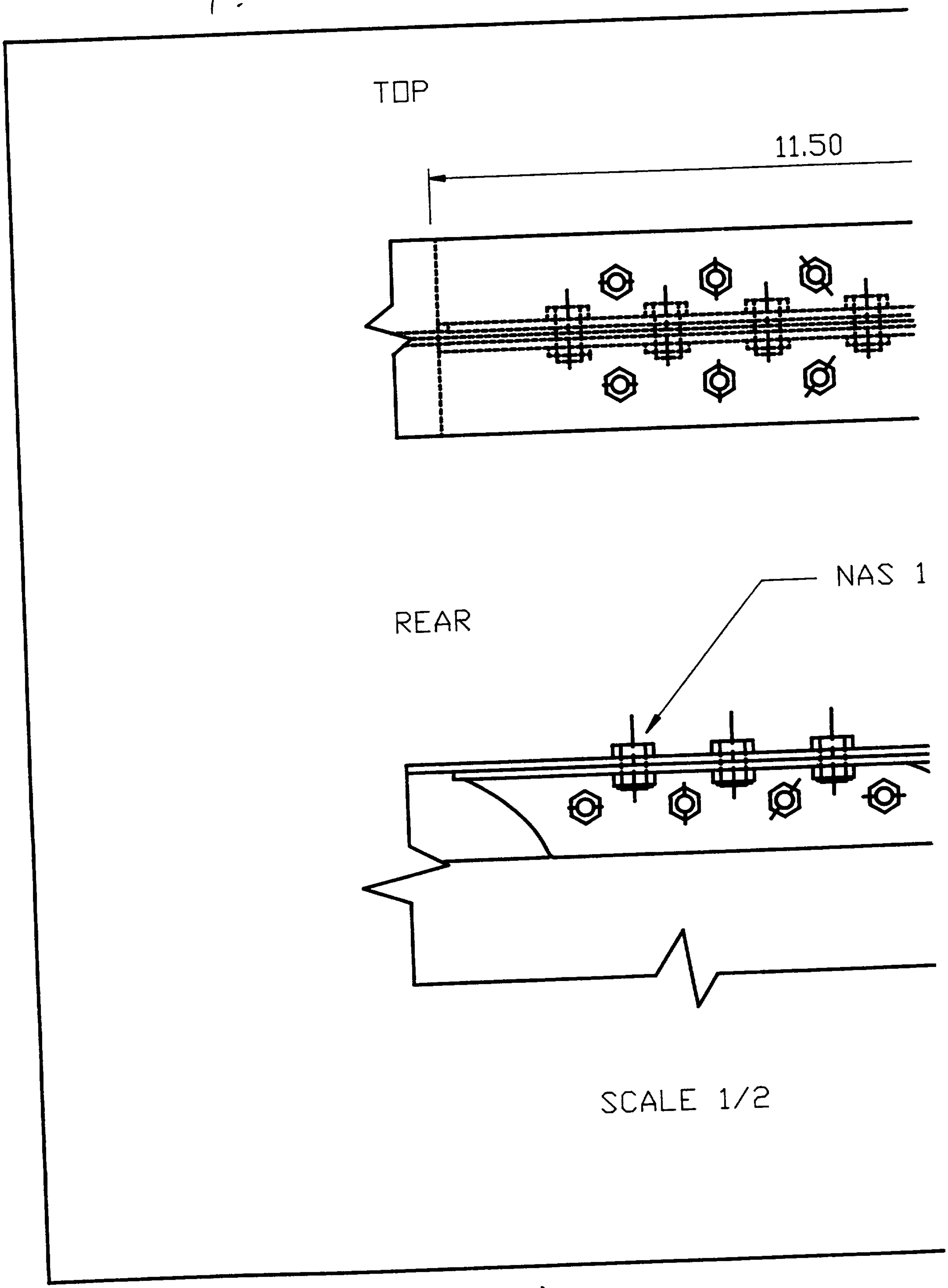


Rear Spar Cap Size	
Wing Stations	Cap Designation
21 – 41	NAS-344-33
41 – 121	NAS-344-32
121 – 192	NAS-344-02

Rear Spar Web Thickness	
Wing Stations	Web Thickness
21 – 41	0.125 in.
41 – 121	0.05 in.
121 – 161	0.032 in.
161 – 192	0.016 in.

DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	7/13

FOLDOUT FRAME 1



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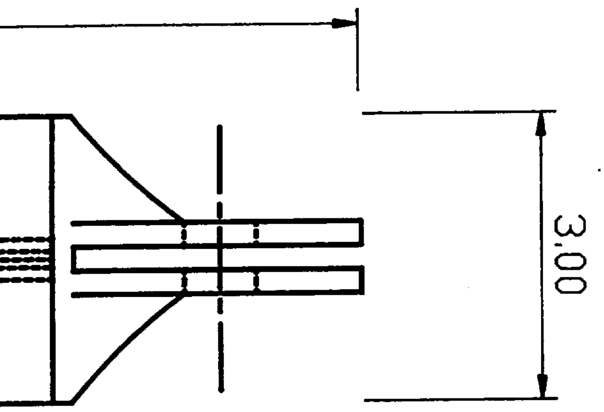
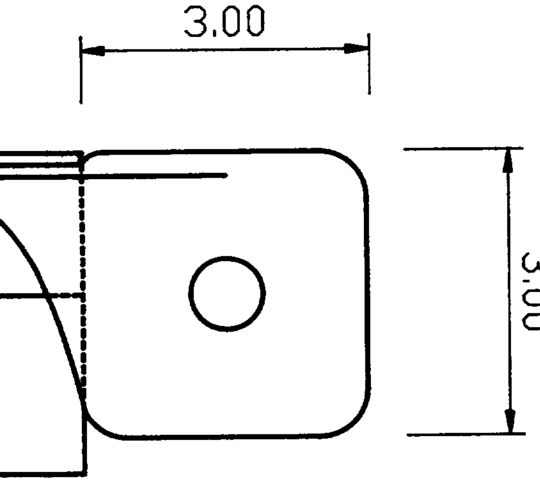


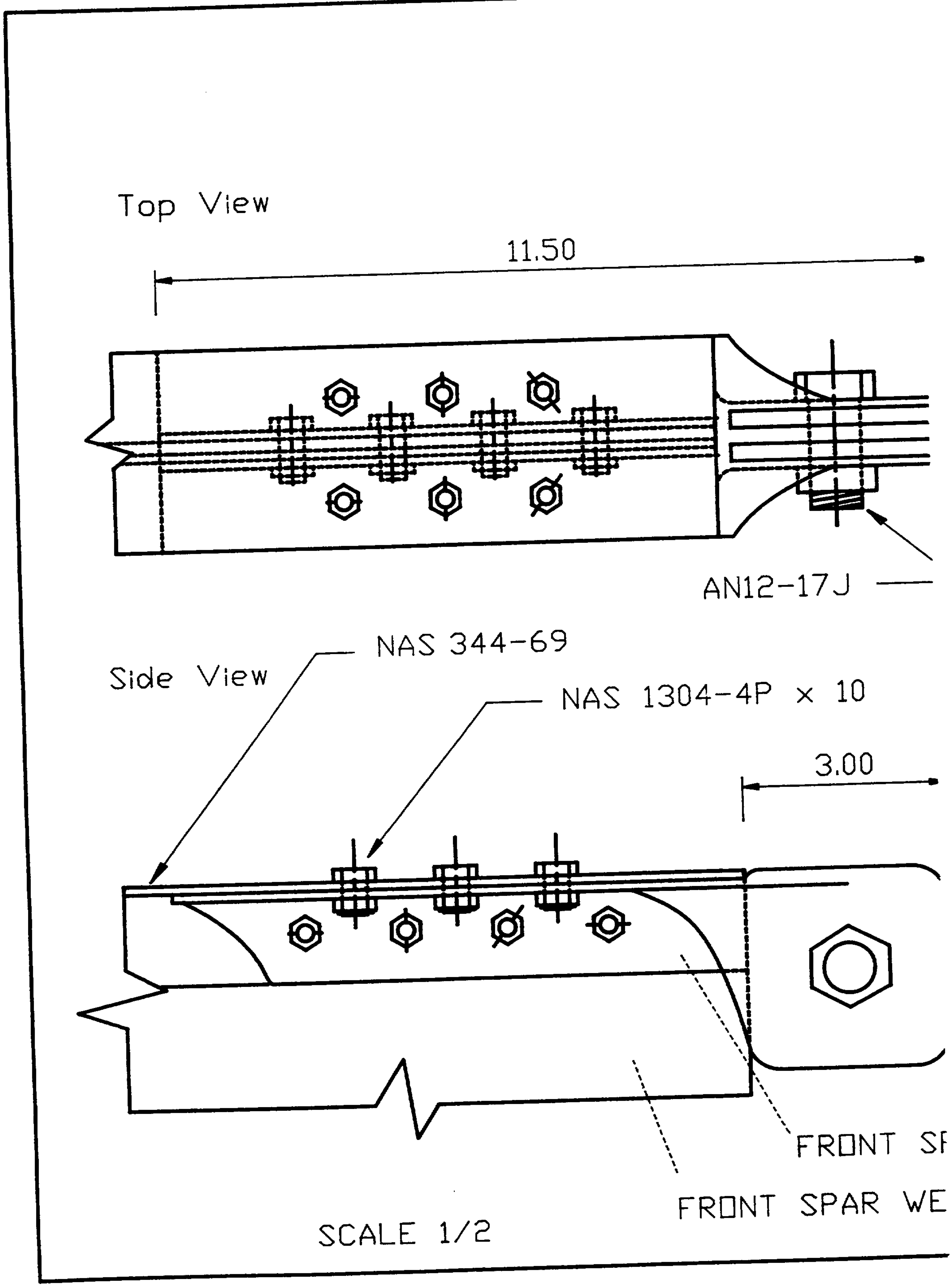
04-4P x 10



FRONT LUG
CARRYTHROUGH
ATTACHMENT

DATE	DWG BY	DRAWING NO.	SHEET
12-06-93	D. BOLTON	F93-2A-104-07	8/13

FOLDOUT FRAME /.

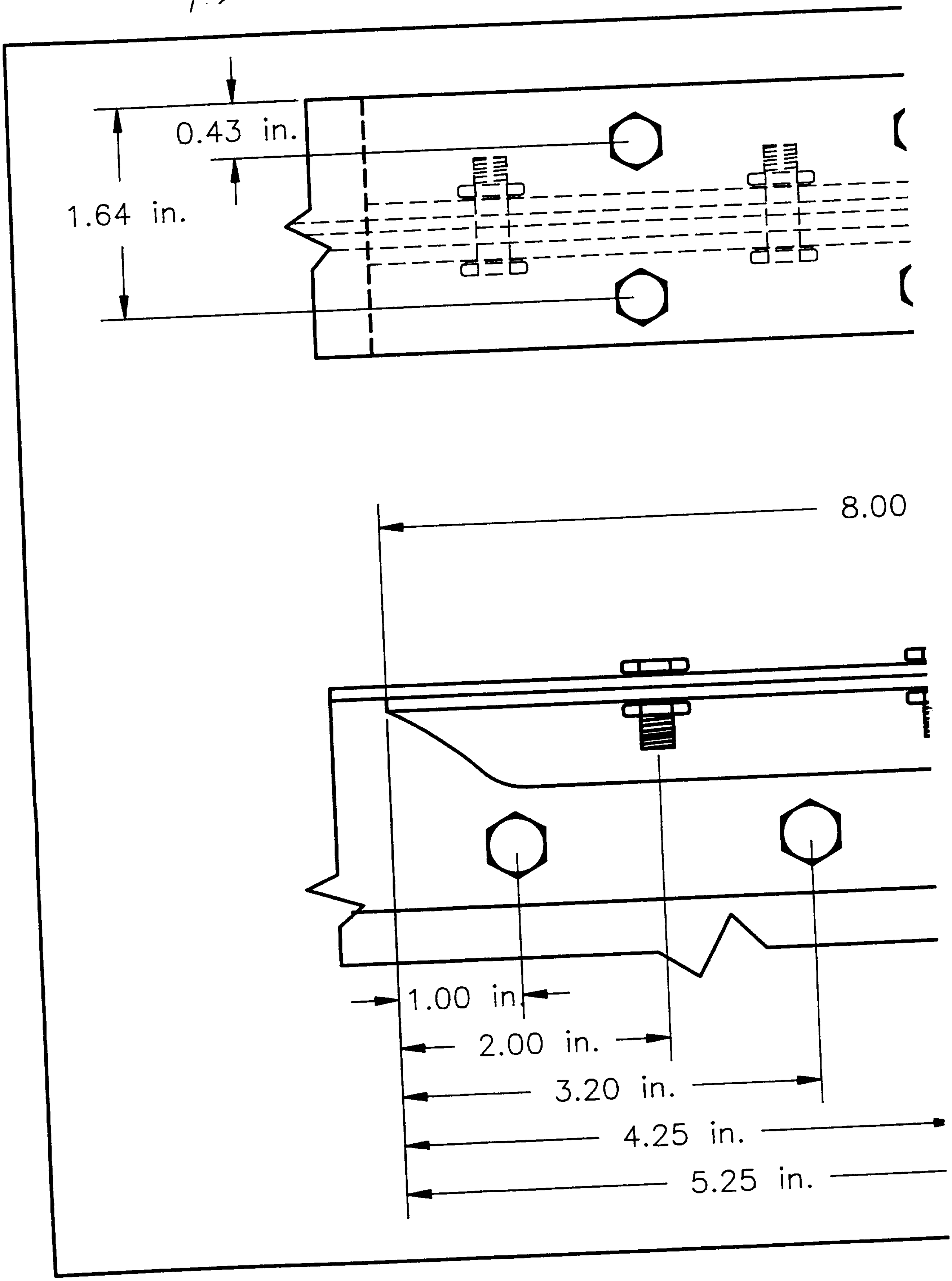


FRONT LUG

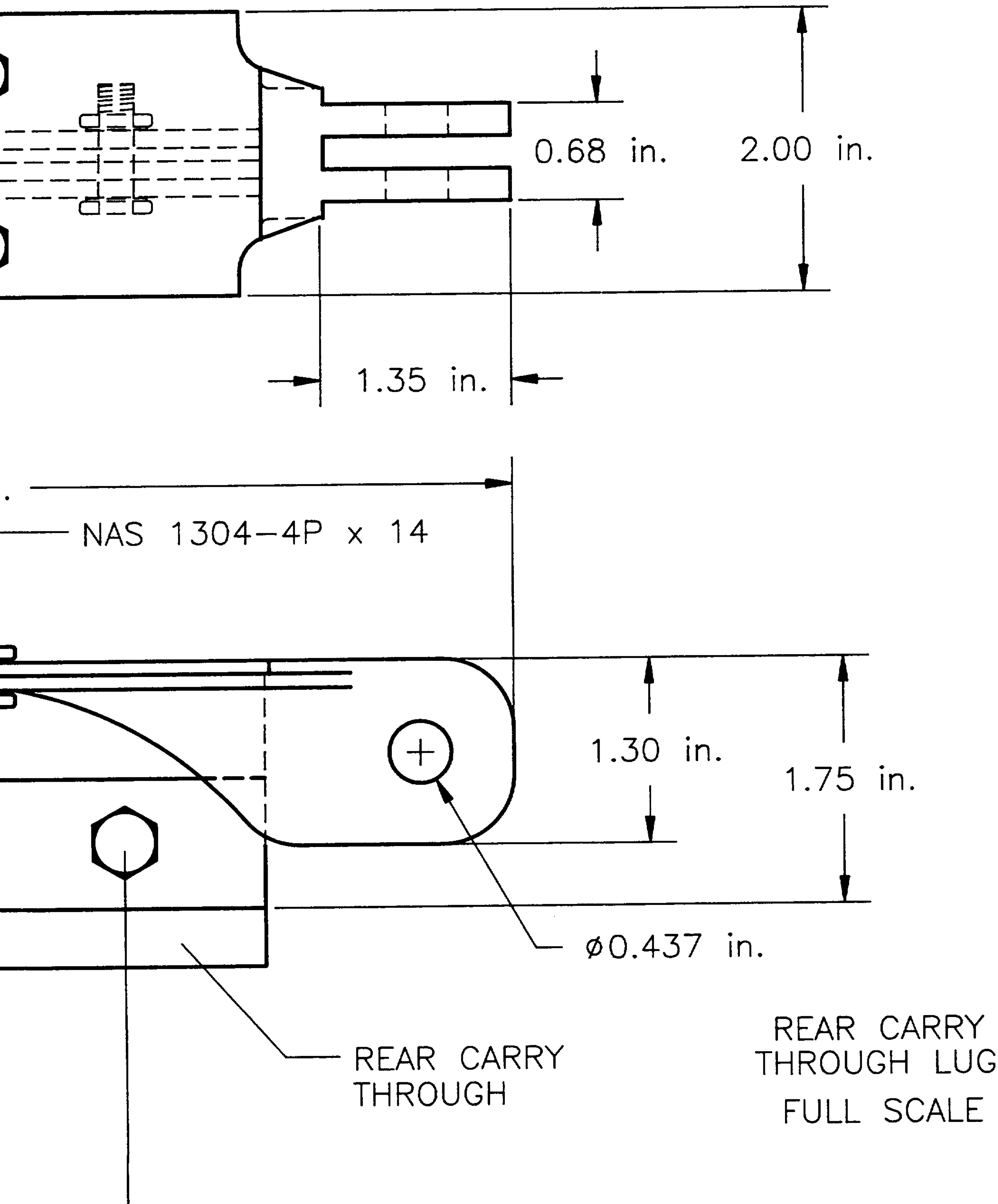
WING ATTACHMENT

R CAP

DATE	DWG BY	DRAWING NO.	SHEET
12-06-93	D. BOLTON	F93-2A-104-07	9/13



0.43 in.
1.64 in.
8.00
1.00 in.
2.00 in.
3.20 in.
4.25 in.
5.25 in.



DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	10/13

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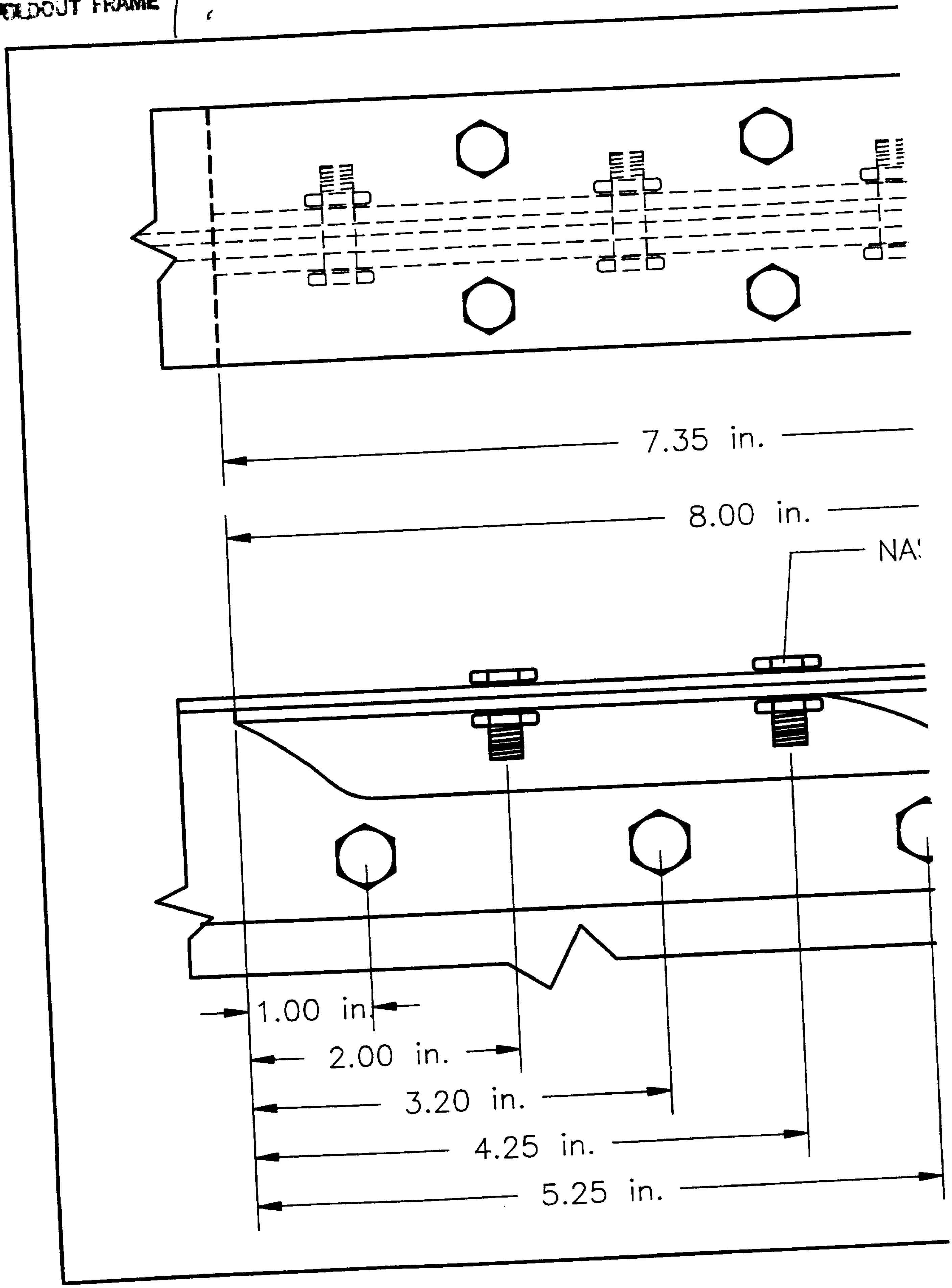
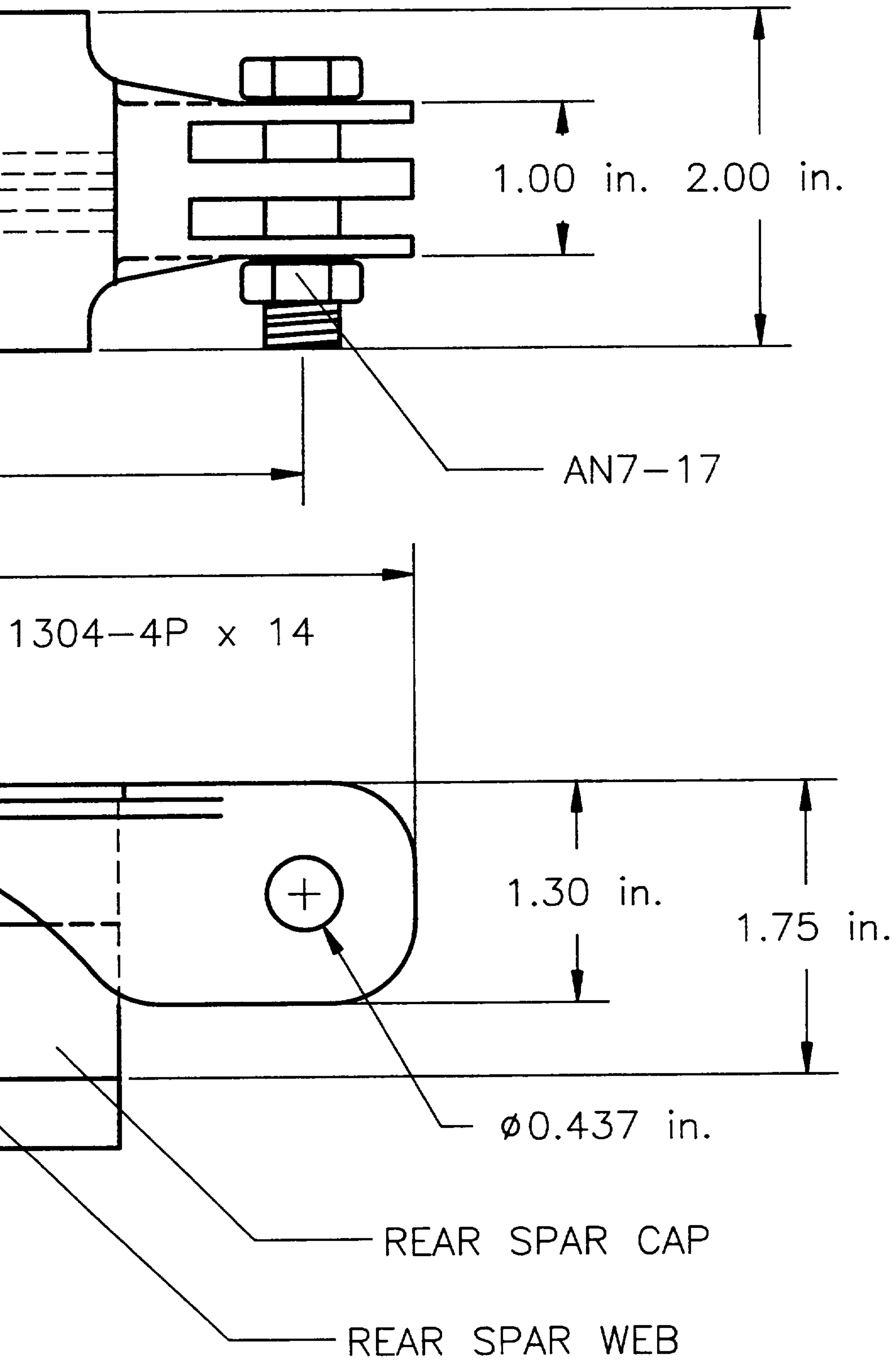


FOLDOUT FRAME 2.

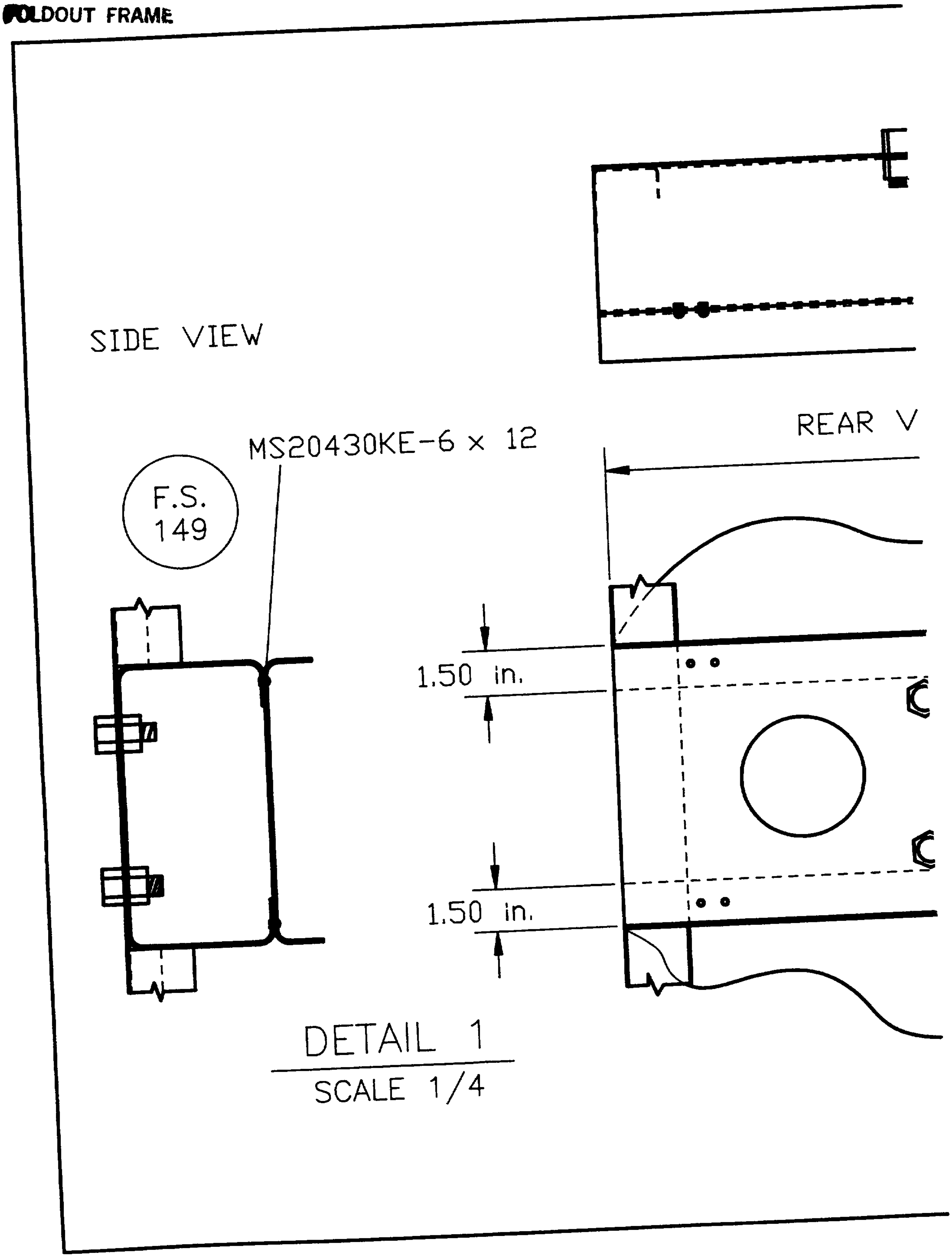


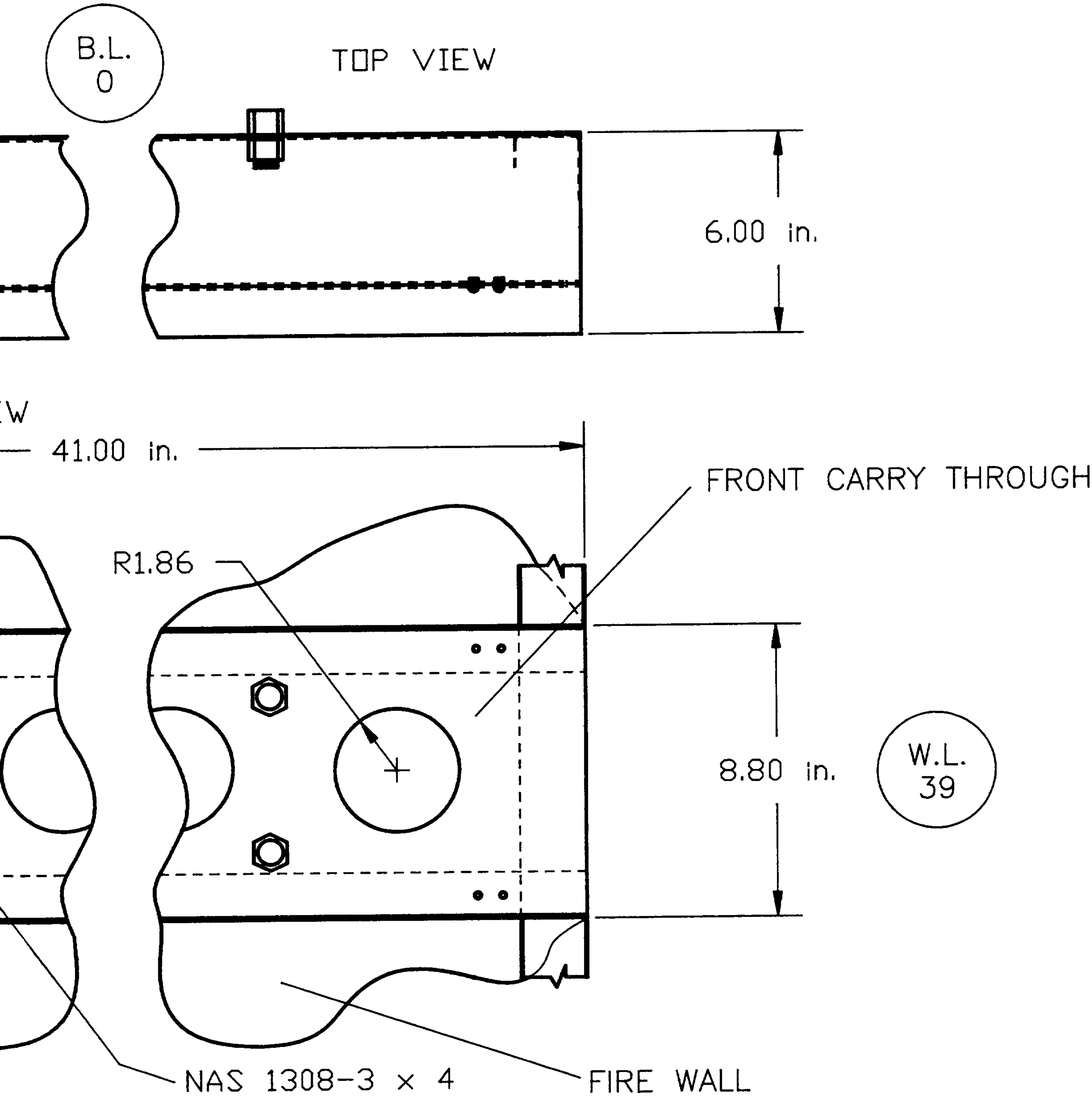
REAR SPAR LUG
FULL SCALE

DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	11/13

1.

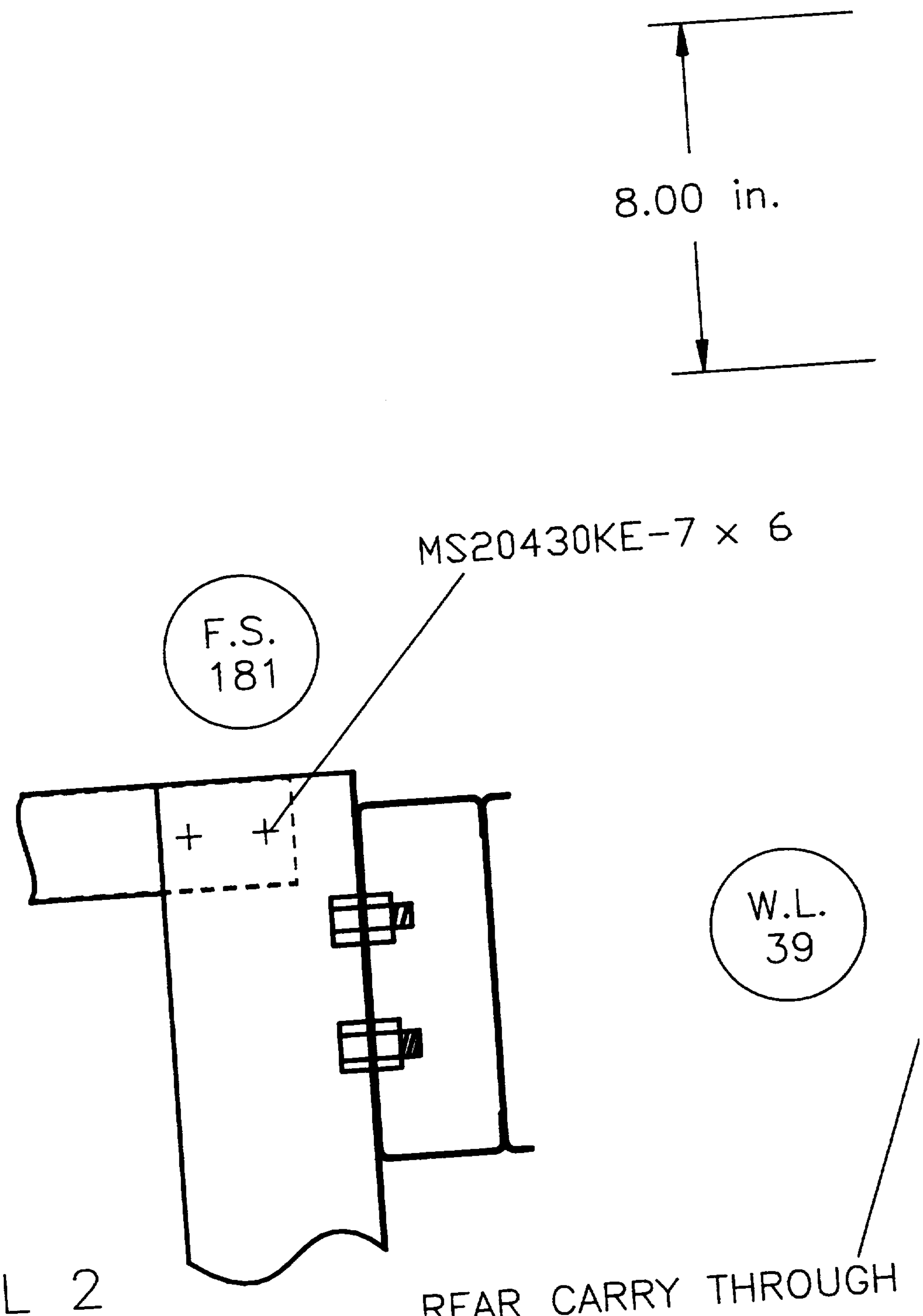
FOLDOUT FRAME





DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	12/13

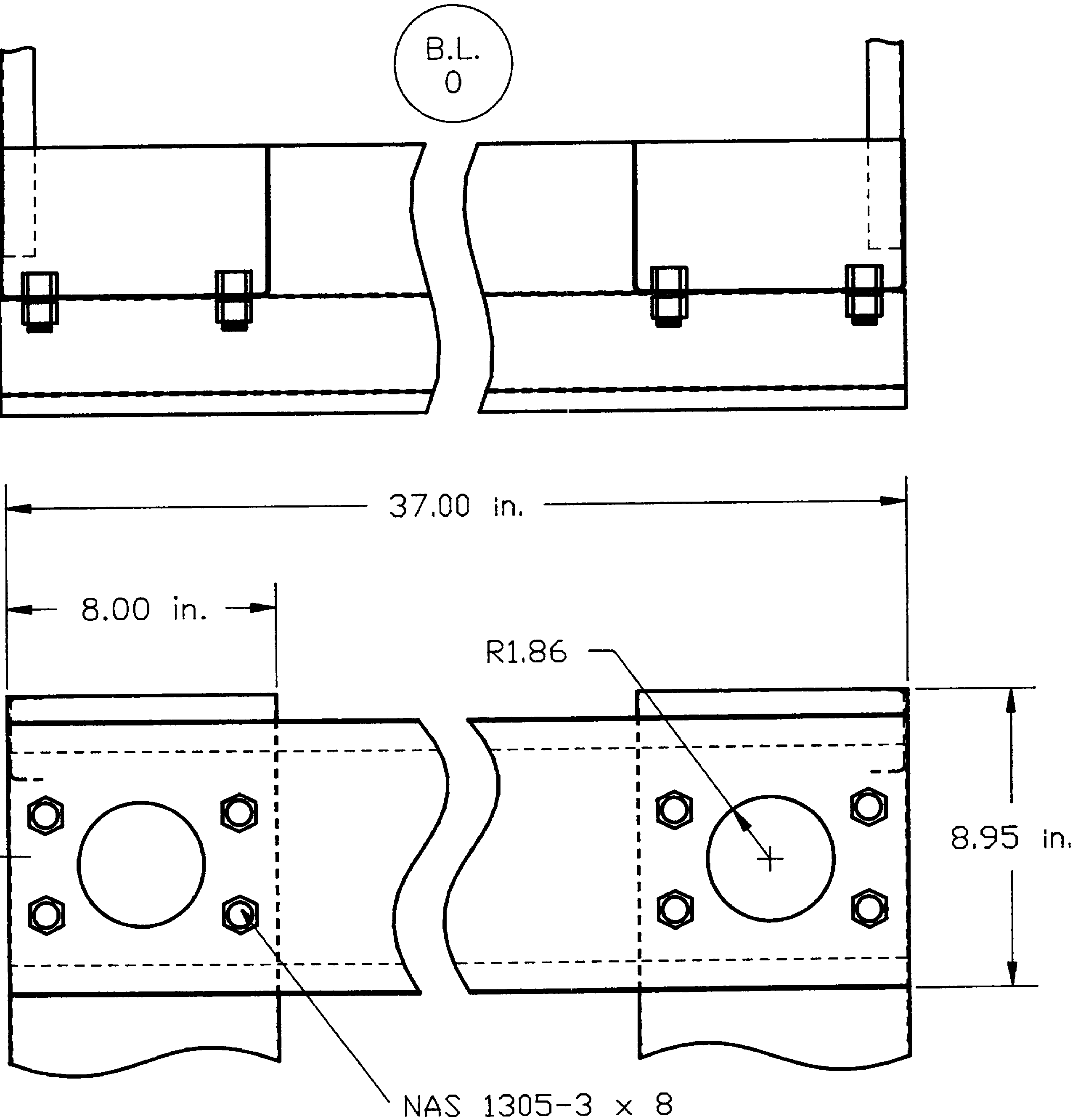
FOLDOUT FRAME 1.



DETAIL 2

SCALE 1/4

FOLDOUT FRAME 2



DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-104-07	13/13

FOLDOUT FRAME /

46	3	TEE-FIREWALL BOLT	AN8-
45	1	DOUBLER	2024-
44	2	NAS341-26	2024-
43	6	BUCKLE STIFFENERS	2024-
42	6	BUCKLE STIFFENERS	2024-
41	1	FWD WINDSHIELD JOINT	2024-
40	20	FASTENING SCREWS	AN507DD
39	50	SPLICING RIVETS	AN430D
38	1	REAR DOOR LATCH C CHANNEL	2024-
37	924	LOWER SKIN RIVETS	AN456D
36	1200	UPPER SKIN RIVETS	AN456D
35	4	NOSE GEAR MOUNTING BOLTS	AN8C-
34	4	ENG MOUNTING BOLTS	AN8C
33	4	ENG MOUNTING BRACKETS	2024-
32	1	NOSE CONE	GLASS/E
31	6	UNDER FLOOR CARRY THROUGHS	2024-
30	1	FIRE WALL	2024-
29	1	LOWER NOSE GEAR REINFORCER	2024-
28	1	UPPER NOSE GEAR REINFORCER	2024-
27	1	RT AFT FUSELAGE SKIN	2024-
26	1	LOWER AFT FUSELAGE SKIN	2024-
25	1	UPPER LT AFT FUSELAGE SKIN	2024-
24	1	UPPER RT MIDDLE FUSELAGE SKIN	2024-
23	1	LOWER MIDDLE FUSELAGE SKIN	2024-
22	1	UPPER LT MIDDLE FUSELAGE SKIN	2024-
21	1	UPPER RT FWD FUSELAGE SKIN	2024-

FOLDOUT FRAME 2

ITEM	QTY	DESCRIPTION	MAT'L OR PART #
20	1	LOWER FWD FUSELAGE SKIN	2024-T3
19	1	UPPER LEFT FWD FUSELAGE SKIN	2024-T3
18	1	FRONT BULKHEAD	2024-T3
17	1	ENG RING FRAME	2024-T3
16	1	ENG RING FRAME	2024-T3
15	1	AFT RING FRAME SECTION	2024-T3
14	1	AFT RING FRAME SECTION	2024-T3
13	1	MID RING FRAME SECTION	2024-T3
12	1	MID RING FRAME SECTION	2024-T3
11	8	RING FRAME BOTTOM CORNERS	2024-T3
10	1	FWD RING FRAME SECTIONS	2024-T3
9	1	FWD RING FRAME SECTIONS	2024-T3
8	1	FLOOR PANEL	2024-T3
7	2	LONGERON BRAKE FORM C	2024-T3
6	1	CEILING SUPPORT NAS 344-60	7075-T6
5	1	HINGE REINFORCER	2024-T3
4	11	ENG LONGERON BRAKE FORM C	2024-T3
3	4	LONGERON BRAKE FORM C	2024-T3
2	4	LONGERON BRAKE FORM C	2024-T3
1	4	FLOOR BRACE NAS 346-45	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
B	10-13	AS INDICATED	ALPHA TEAM

TITLE
STRUCTURAL LAYOUT AND DETAILS

DRAWING NO. F93-2A-102-07

SHEET 1/10

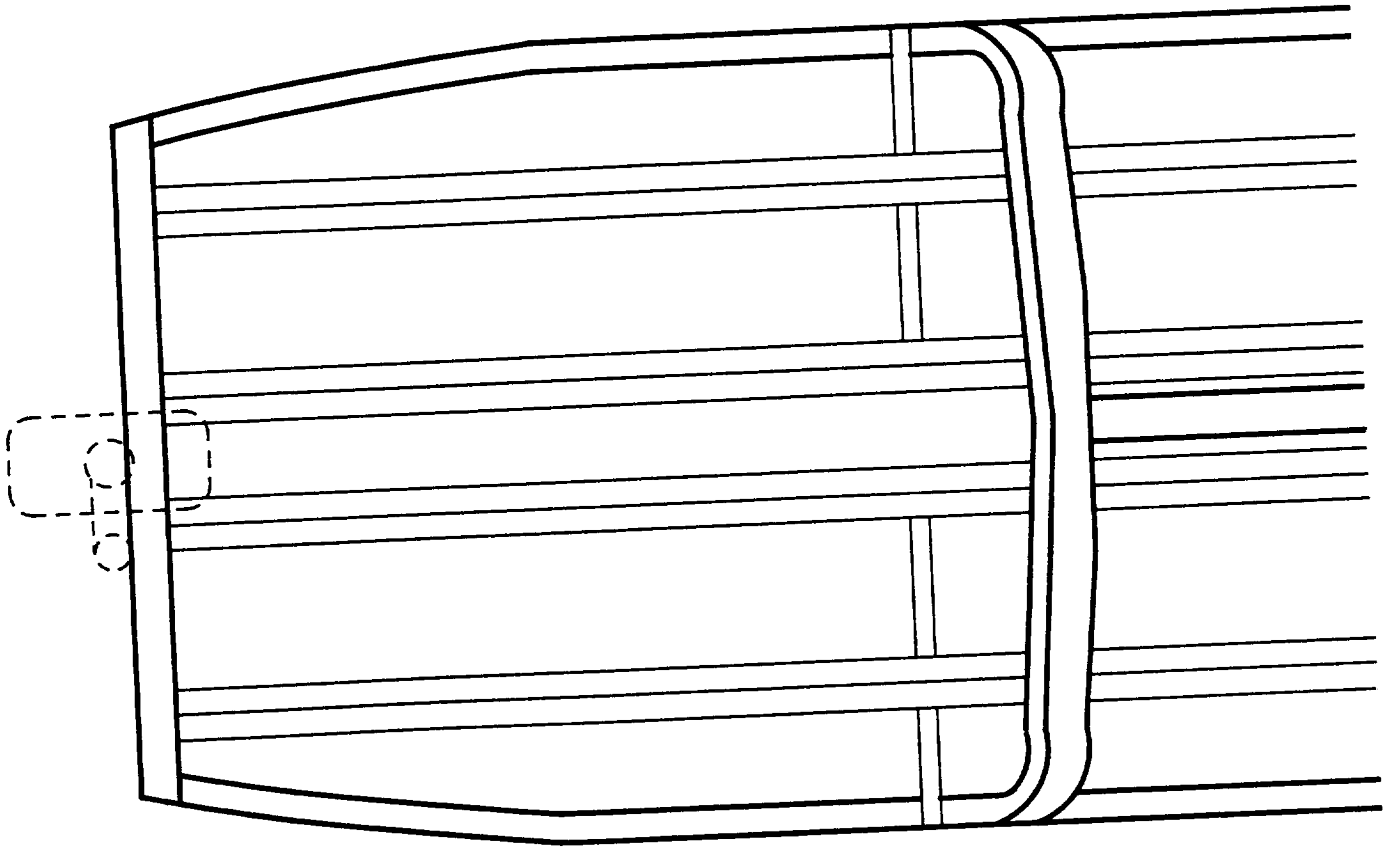
FOLDOUT FRAME 1.

[redacted]

ITEM	QTY	DESCRIPTION	MAT'L OR PART #
53	1	FRONT CARRY THROUGH FWD C	2024-T3
52	1	FRONT CARRY THROUGH AFT C	2024-T3
51	1	REAR CARRY THROUGH AFT C	2024-T3
50	1	REAR CARRY THROUGH FWD C	2024-T3
49	4	SUPPORT LONGERON	2024-T3
48	1	CARRY THROUGH SUPPORT	2024-T3
47	2	SUPPORT LONGERON	2024-T3

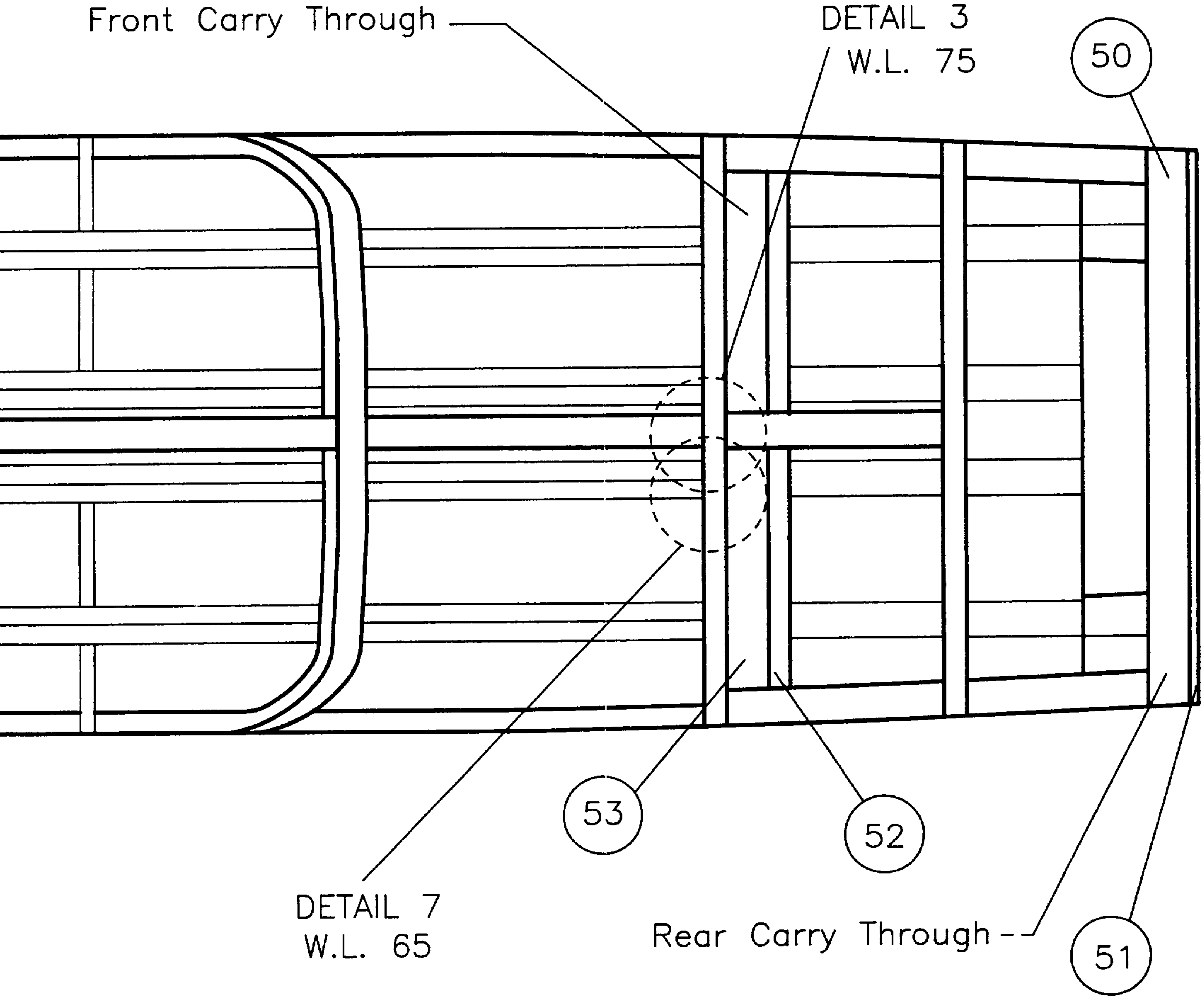
DATE	DWG BY	DRAWING NO.	SHEET
12-06	ALPHA TEAM	F93-2A-102-07	2/10

FOLDOUT FRAME /



TO

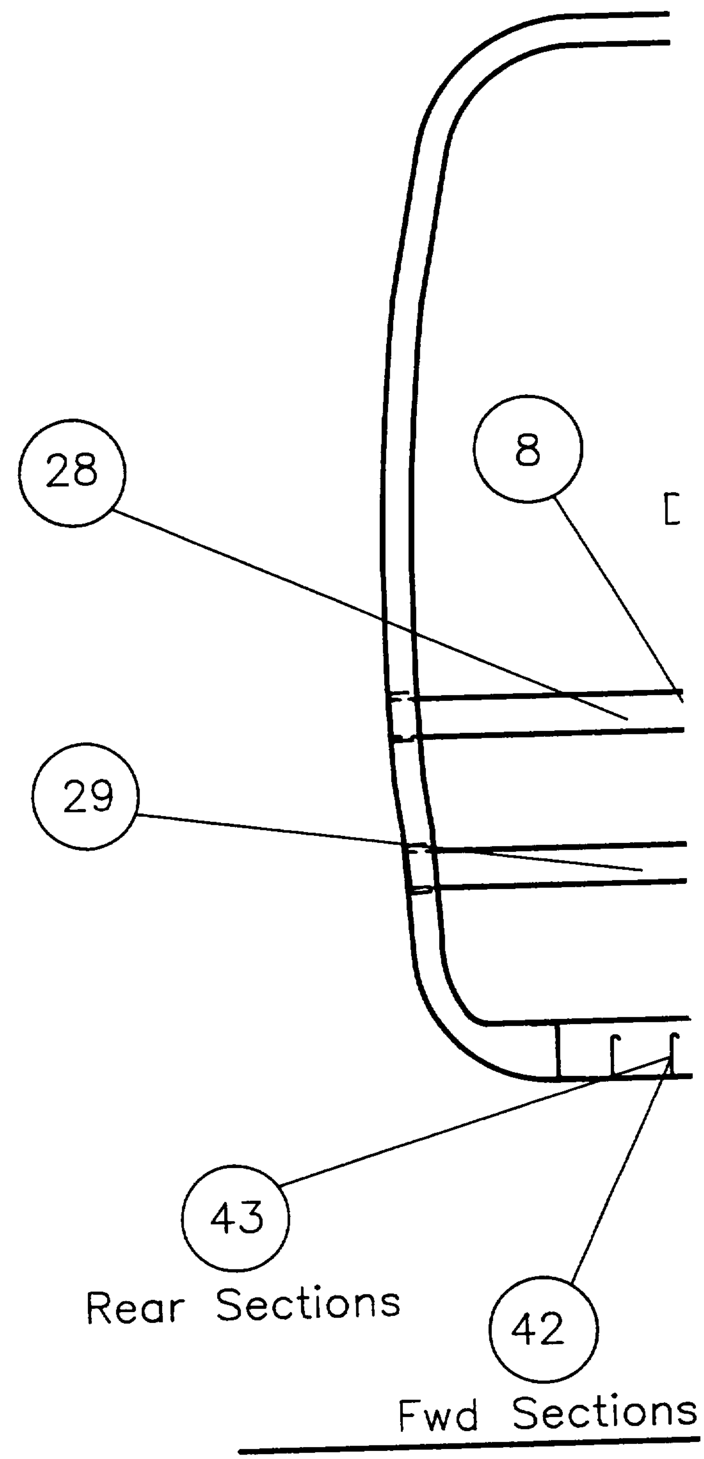
SC



P VIEW

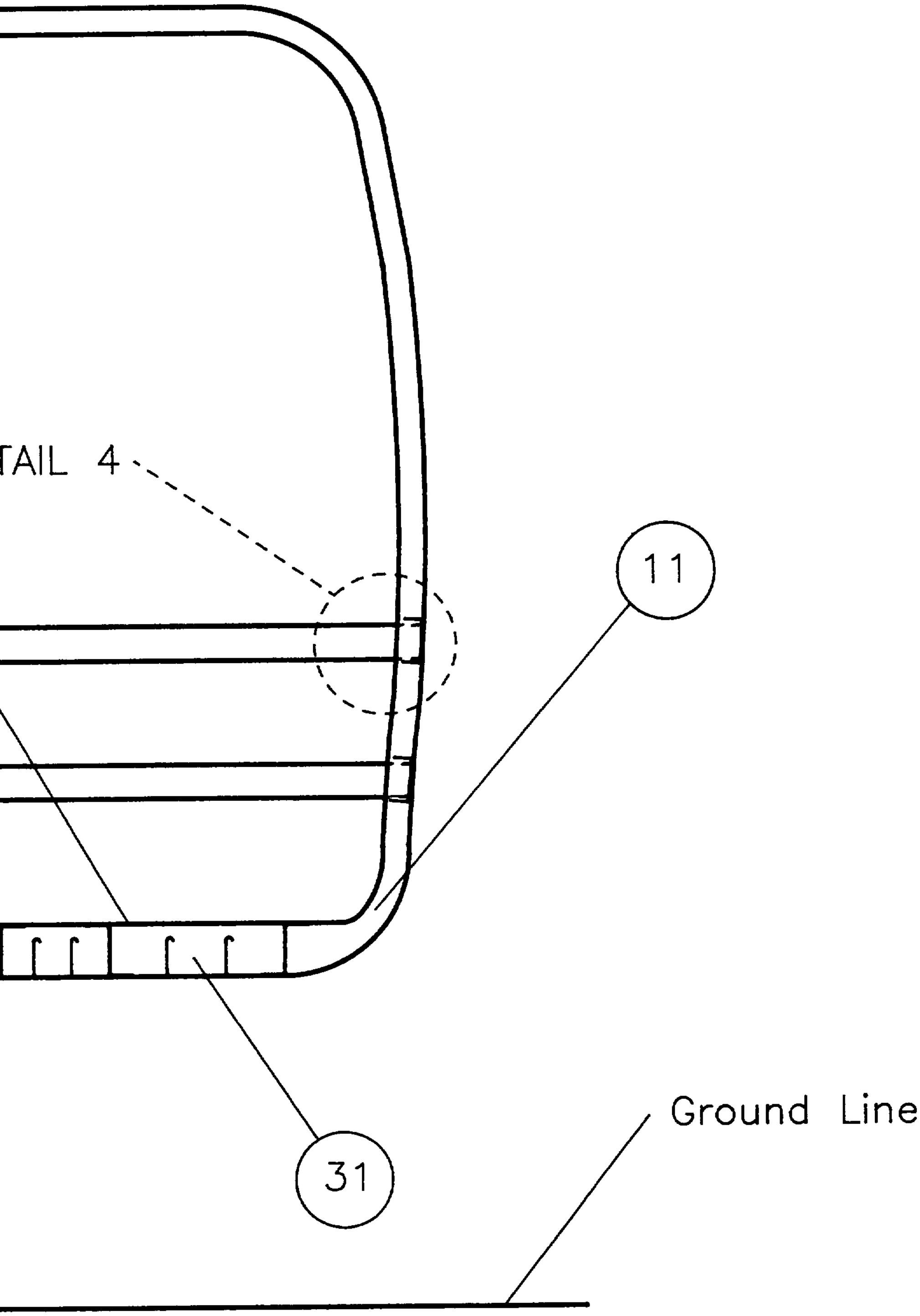
LE: 1/10

DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	3/10



FR

S

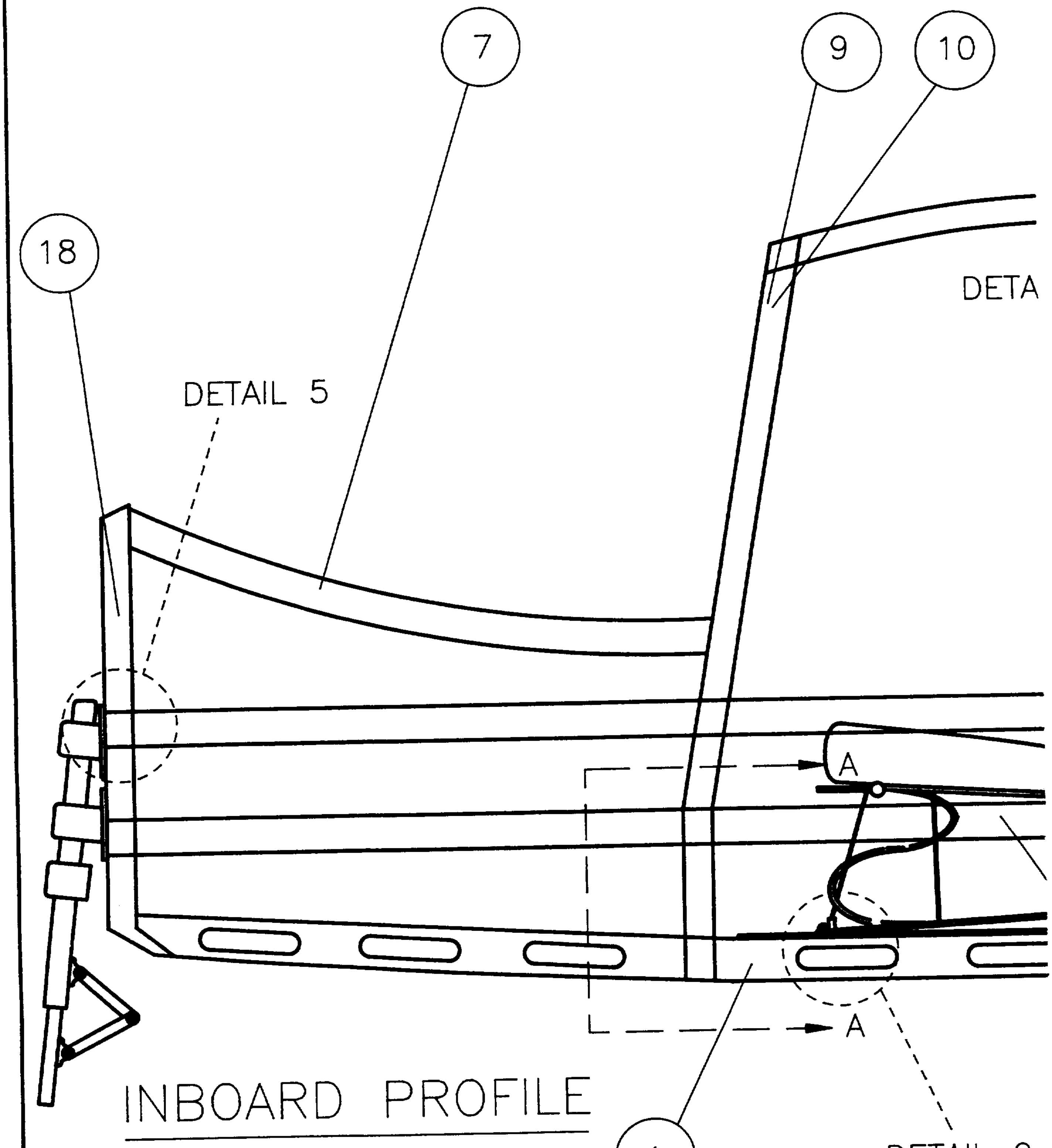


NT VIEW

ALE: 1/10

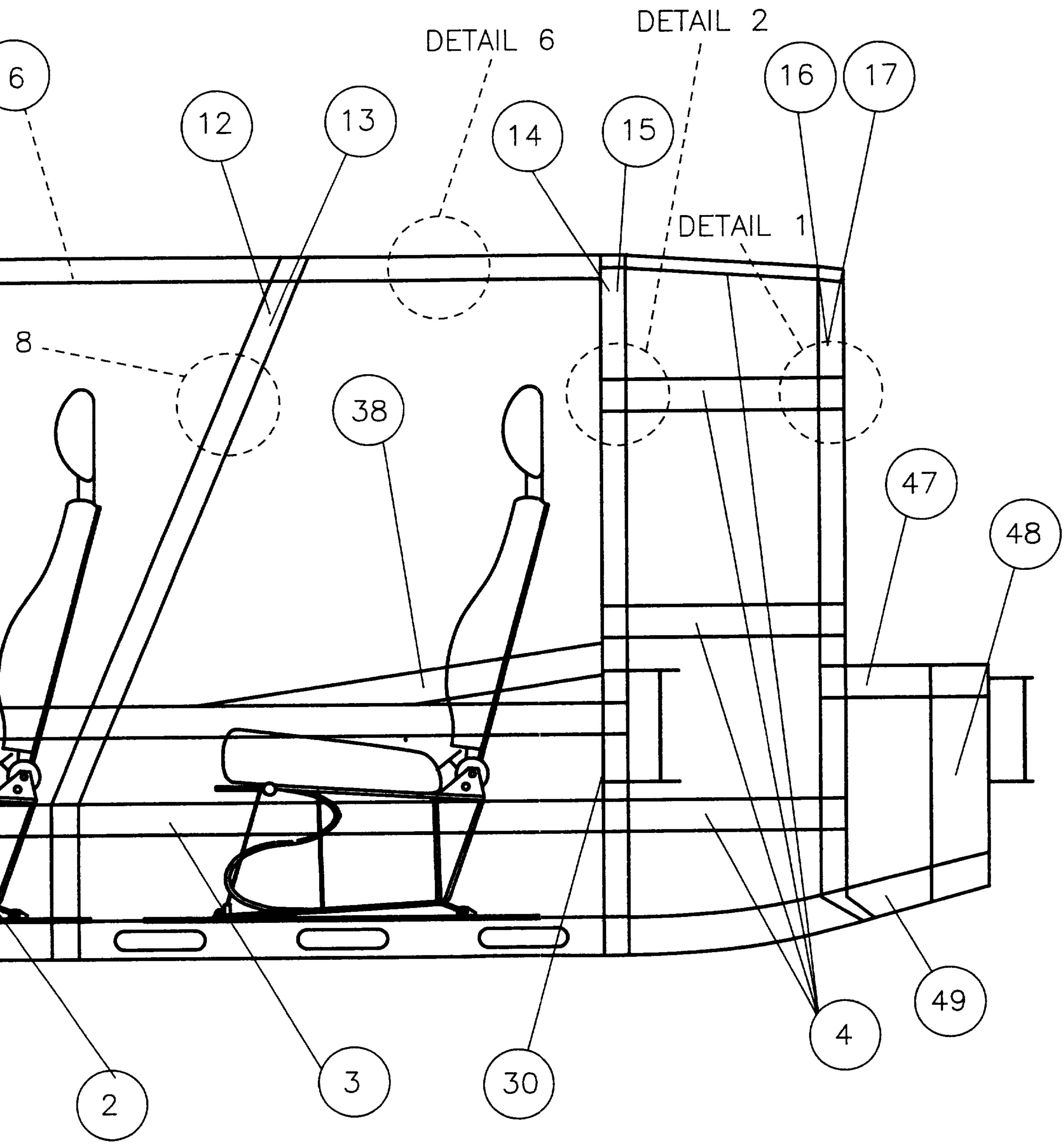
DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	4/10

FOLDOUT FRAME 1

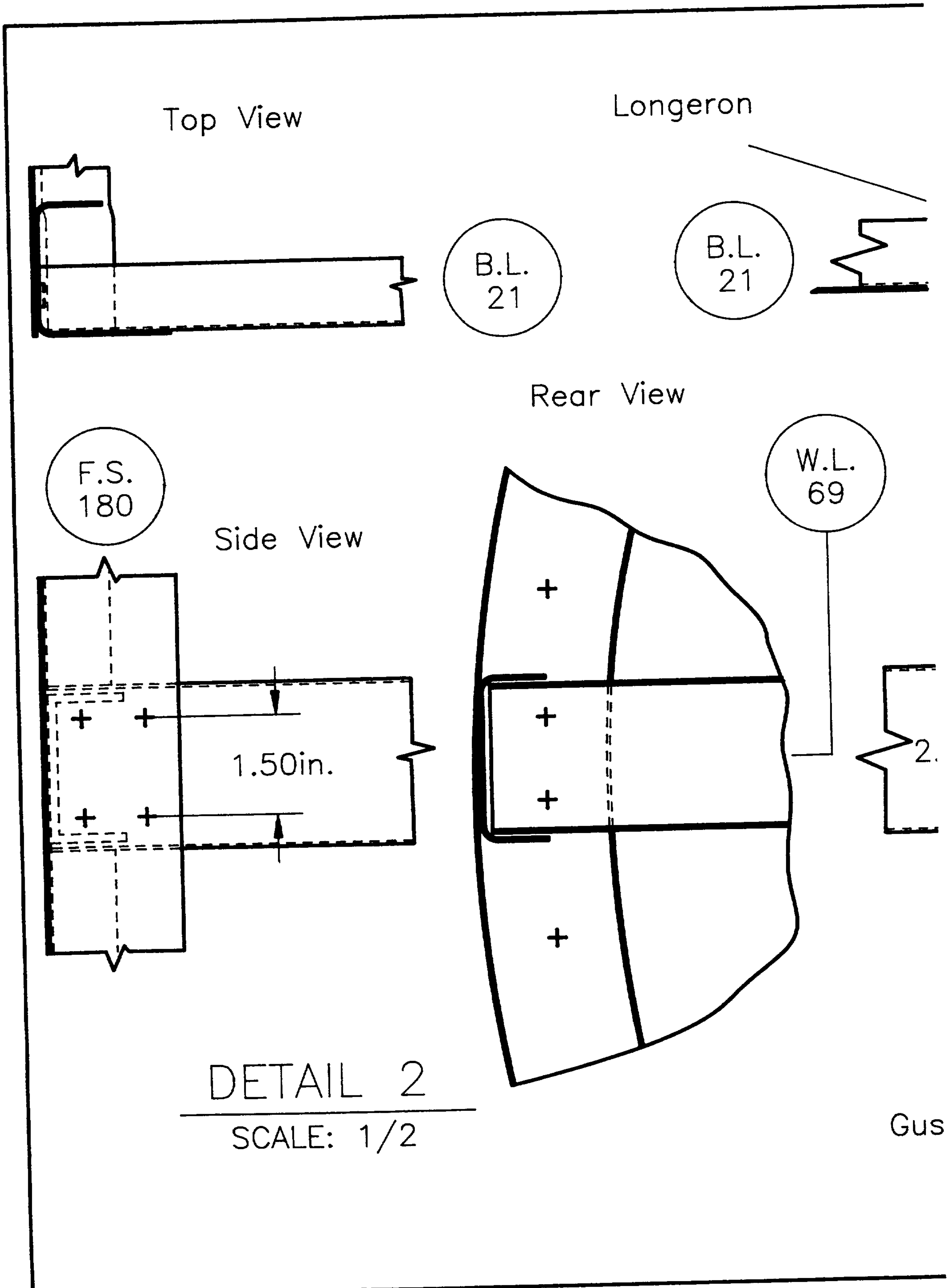


INBOARD PROFILE

SCALE: 1/10



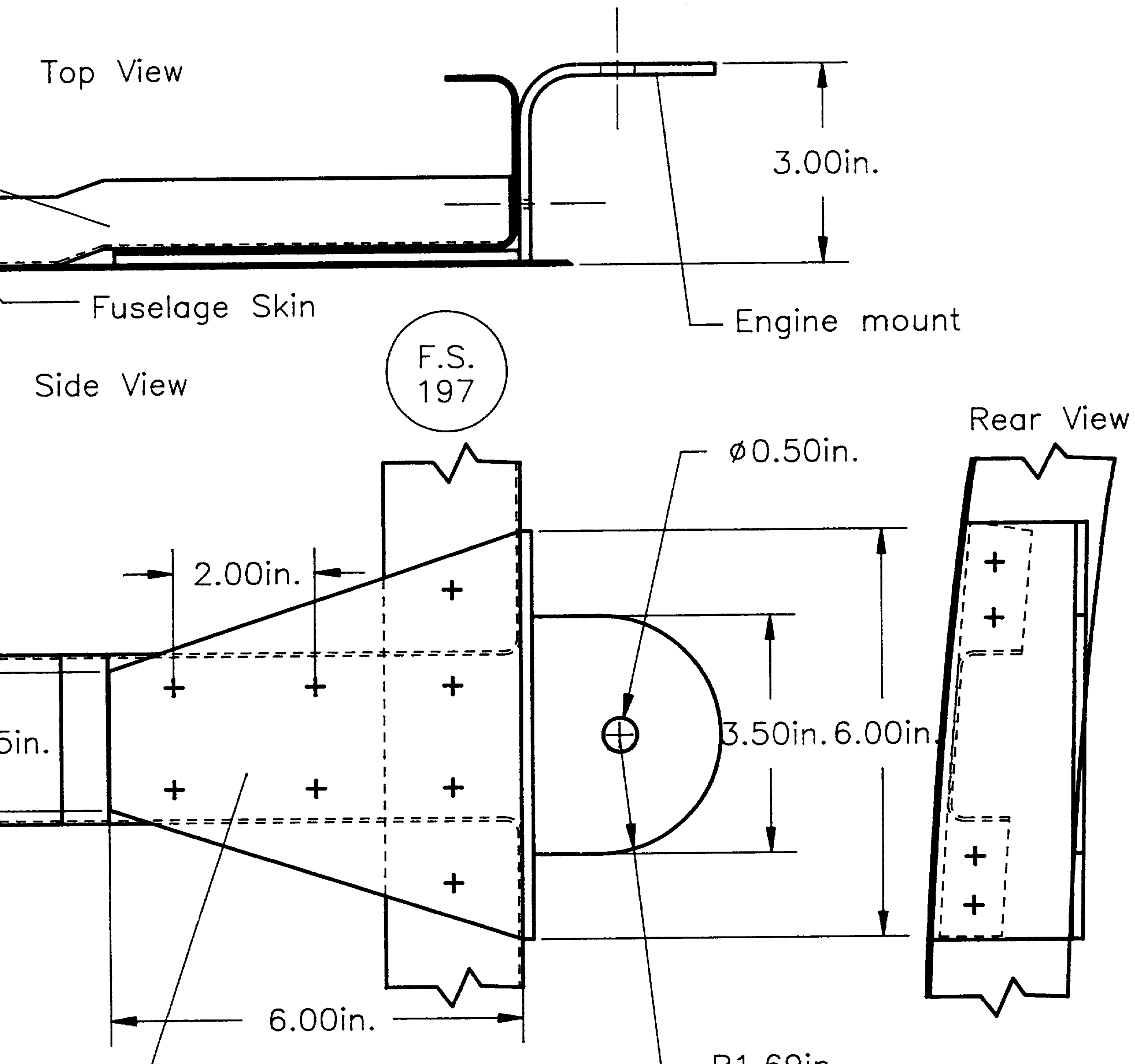
DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	5/10



DETAIL 2

SCALE: 1/2

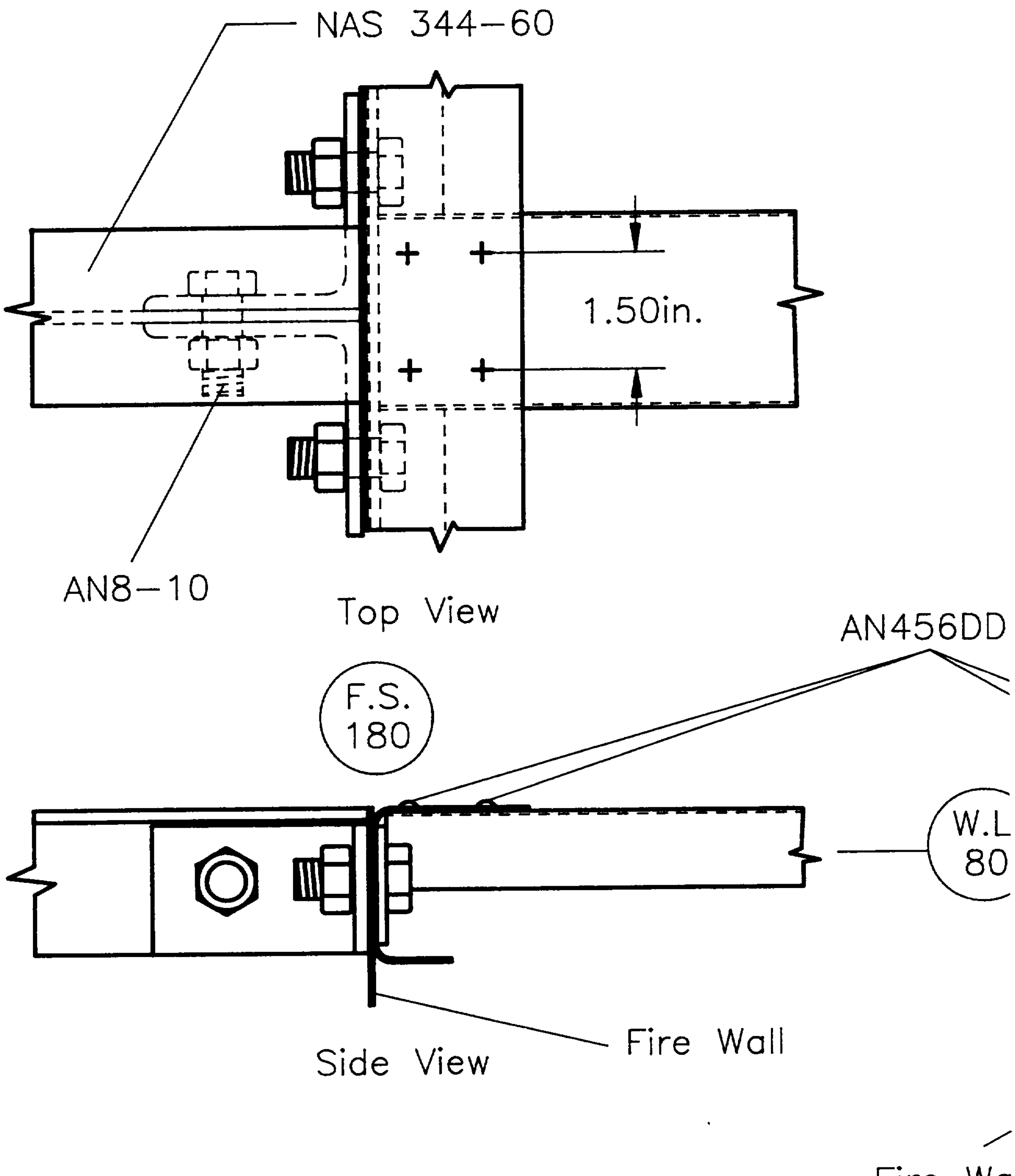
FOLDOUT FRAME 2.



DETAIL 1

SCALE: 1/2

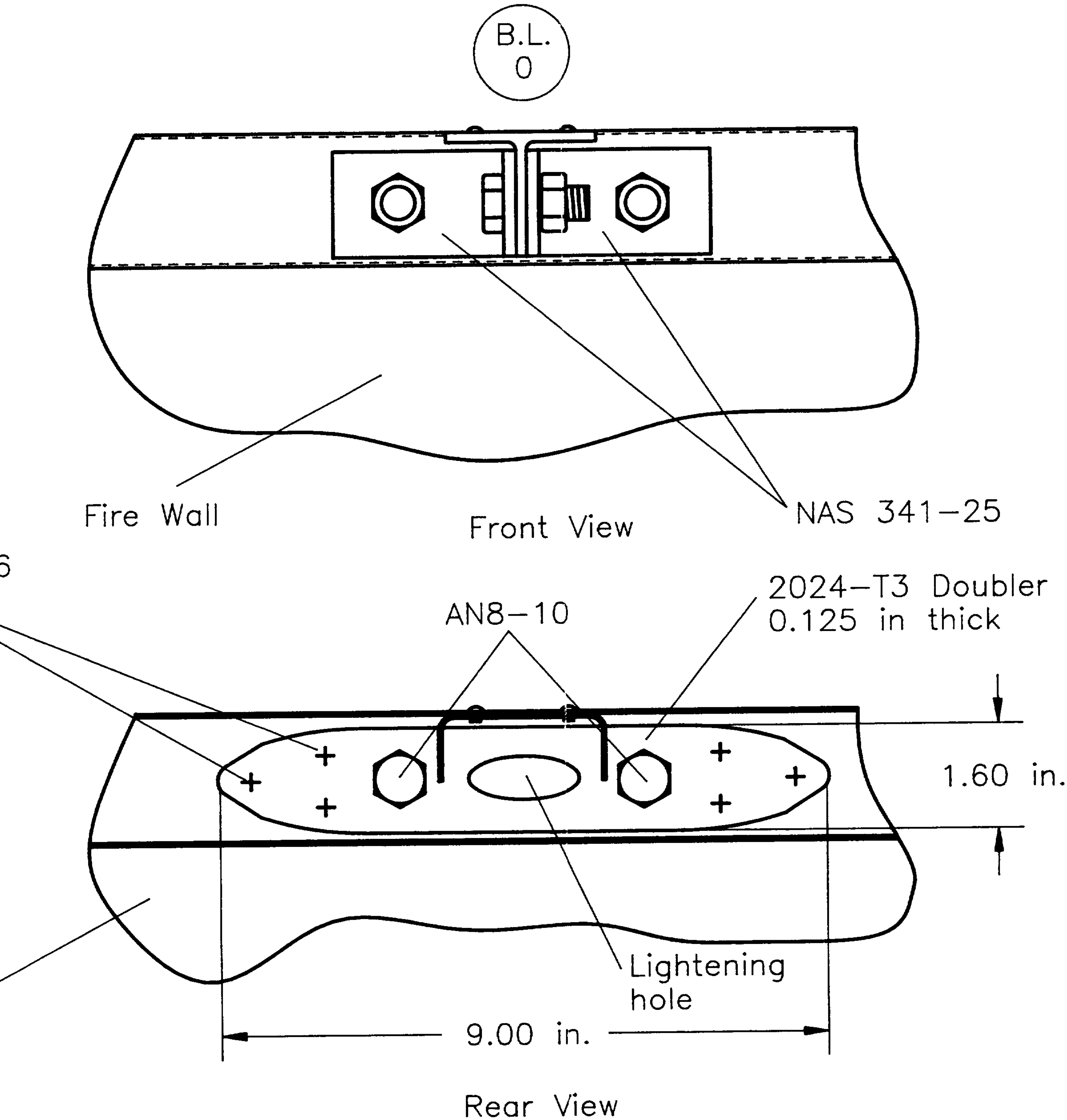
DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	6/10



DETAIL 3

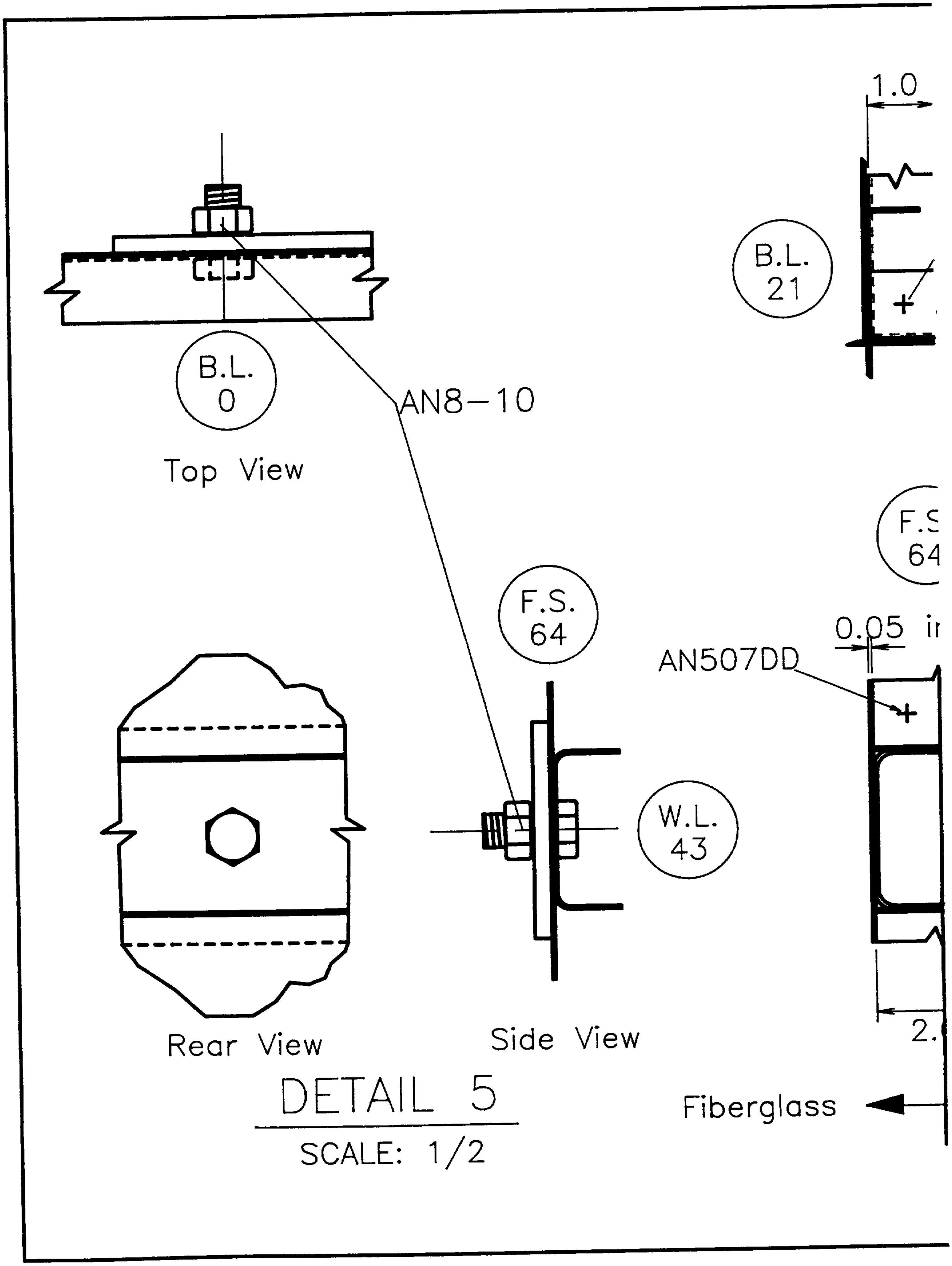
SCALE: 1/2

FOLDOUT FRAME 2



DATE	DWG BY	DRAWING NO.	SHEET
10-13	Alpha Team	F93-2A-102-07	7/10

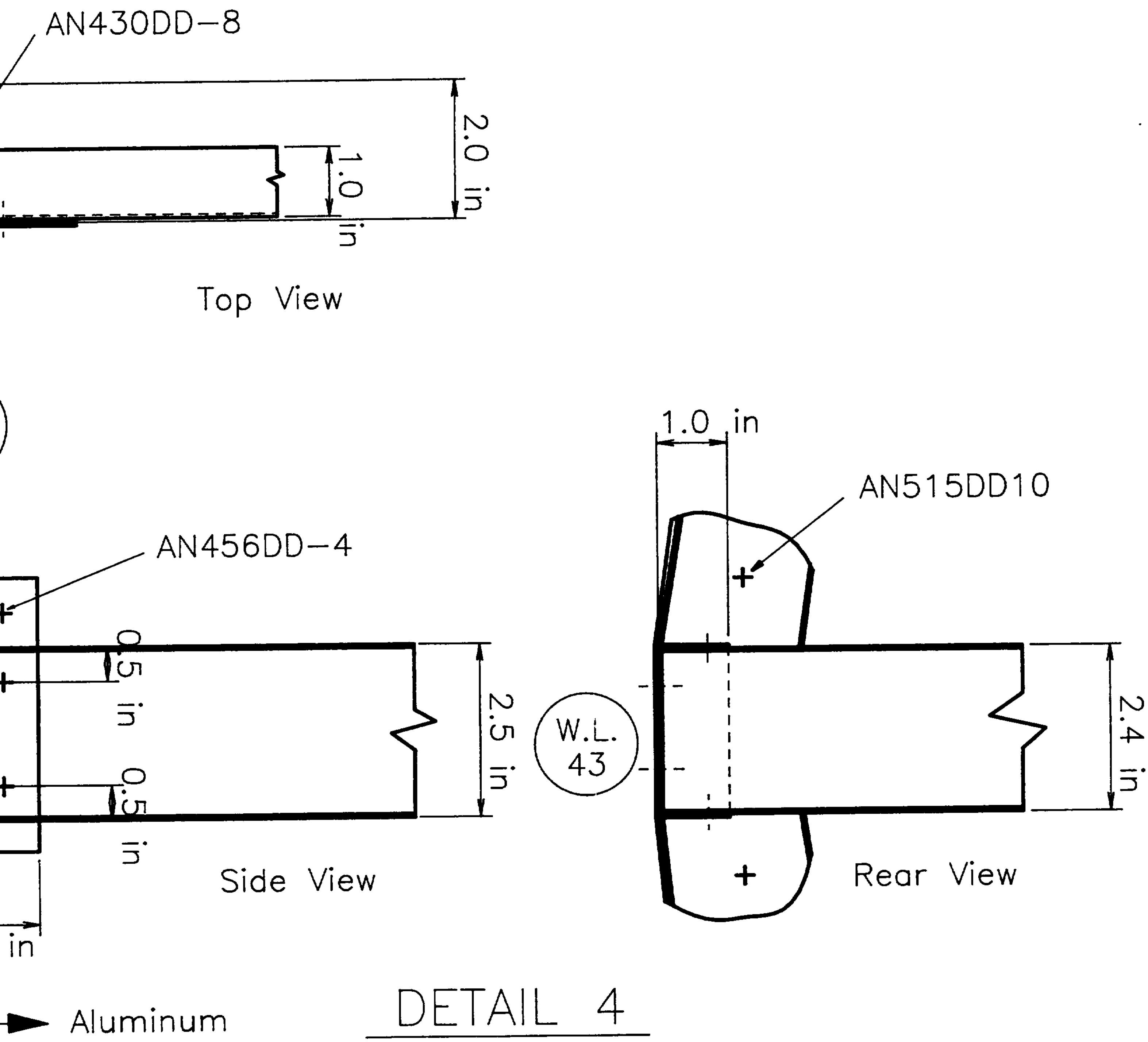
FOLDOUT FRAME 1.



DETAIL 5

SCALE: 1/2

FOLDOUT FRAME 2

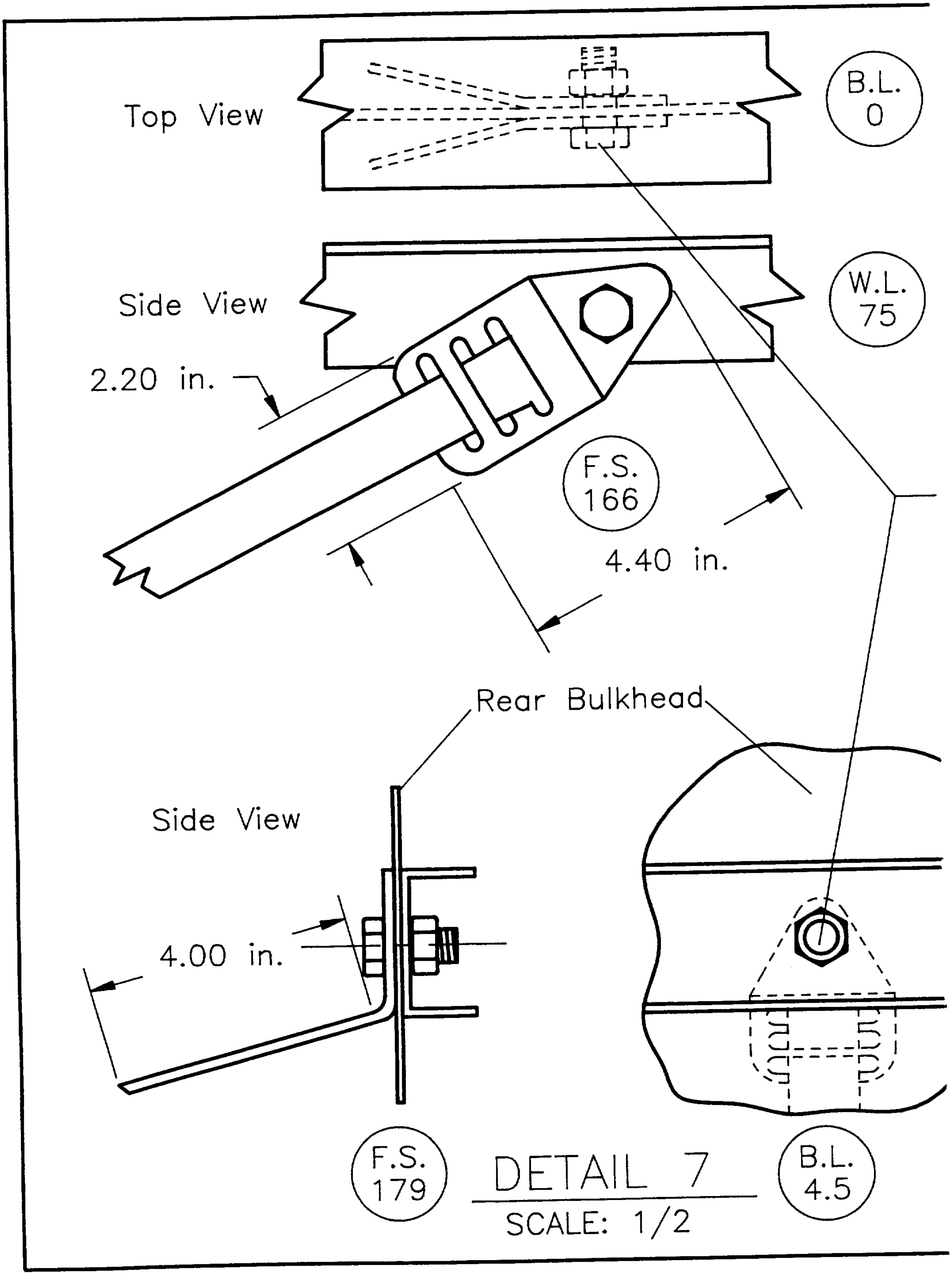


DETAIL 4

SCALE: 1/2

DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	8/10

FOLDOUT FRAME

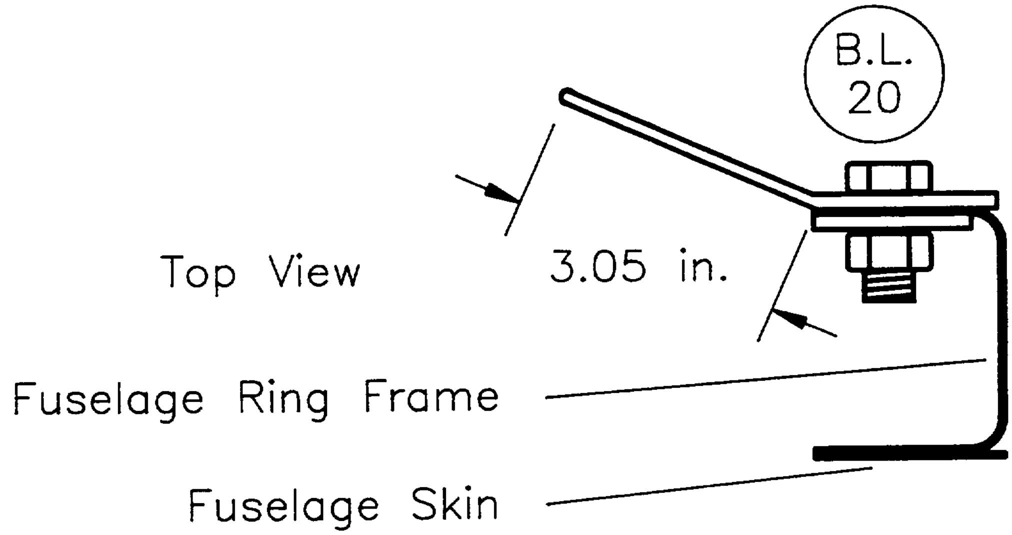


DETAIL 7

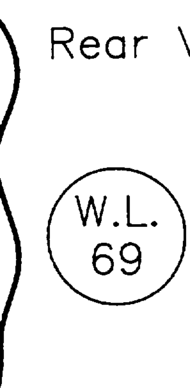
SCALE: 1/2

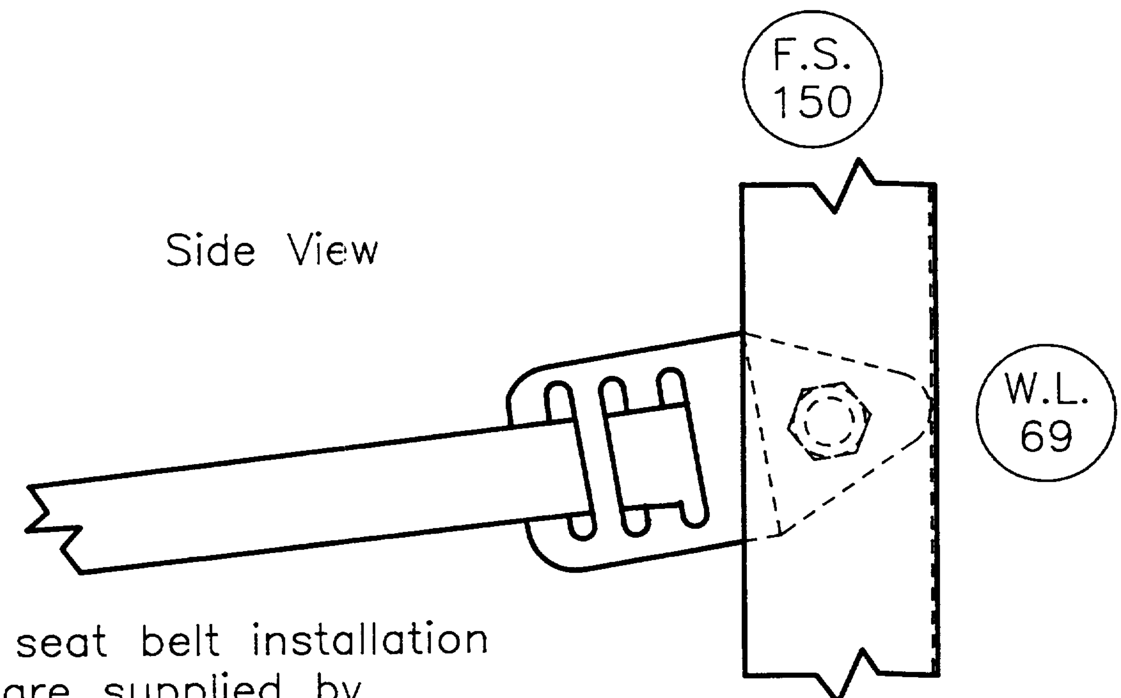
DETAIL 6

SCALE: 1/2



AN8-10
for all seat belt
attachments





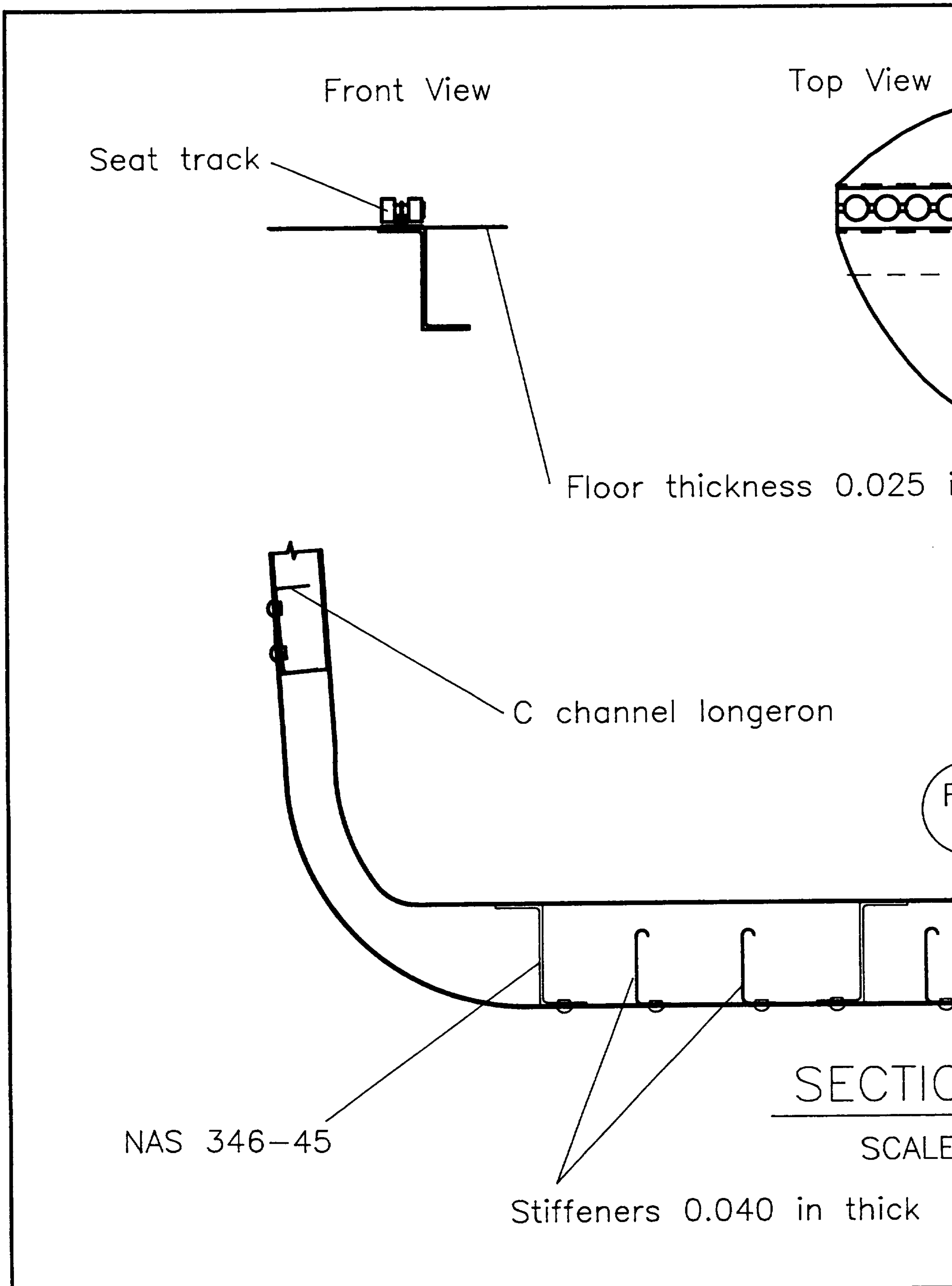
Note: seat belt installation hardware supplied by manufacturer

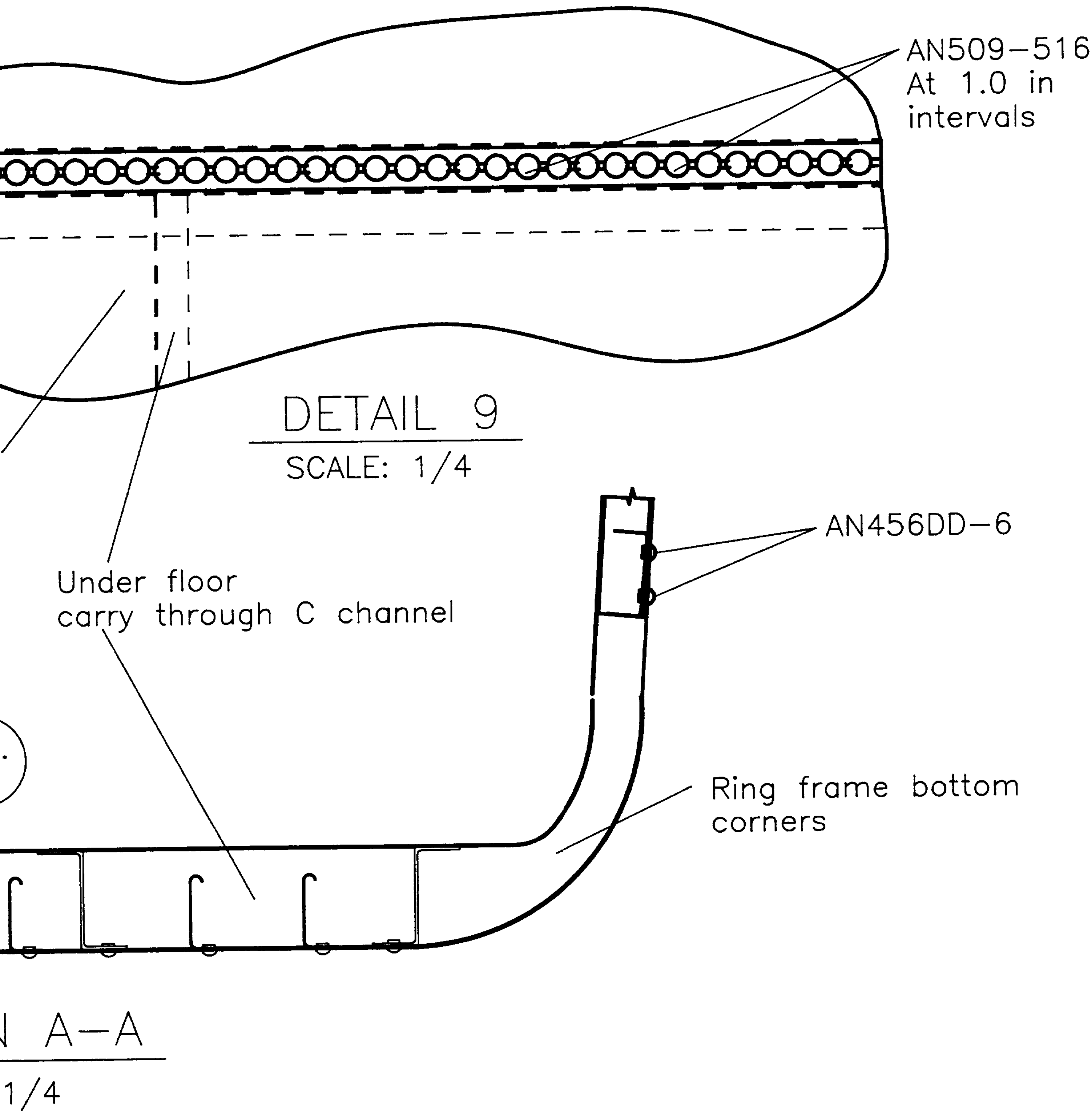
DETAIL 8

SCALE: 1/2

DATE	DWG BY	DRAWING NO.	SHEET
10-13	ALPHA TEAM	F93-2A-102-07	9/10

FOLDOUT FRAME 1.





DATE	DWG BY	DRAWING NO.	SHEET
10-13	J. VIEIRA	F93-2A-102-07	10/10