NASw-4435

**Viper Cabin-Fuselage
Structural Design Concept
with Engine Installation
and Wing Structural Design**

F93-1C-2R3

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1.0 Project Summary

In response to the requirements of the "Cabin-Fuselage Structural Design Concept with Engine Installation" (421F93ADP01-2) Statement of Work (SOW) and its Addendum, "Wing Structural Design," the following report was prepared. This report describes the process and considerations in designing the cabin, nose, drive shaft, and wing assemblies for the "Viper" concept aircraft. Interfaces of these assemblies, as well as interfaces with the sections of the aircraft aft of the cabin, are also discussed. The results of the design process are included.

The goal of this project is to provide a structural design which complies with FAR 23 requirements regarding occupant safety, emergency landing loads, and maneuvering loads. The design must also address the interfaces of the various systems in the cabin, nose, and wing, including the drive shaft, venting, vacuum, electrical, fuel, and control systems. Interfaces between the cabin assembly and the wing carrythrough and empennage assemblies were required, as well. In the design of the wing assemblies, consistency with the existing cabin design was required.

The major areas considered in this report are materials and construction, loading, maintenance, environmental considerations, wing assembly fatigue, and weight. The first three areas are developed separately for the nose, cabin, drive shaft, and wing assemblies, while the last three are discussed for the entire design.

For each assembly, loading calculations were performed to determine the proper sizing of major load carrying components. Table 1.0 lists the resulting margins of safety for these key components, along with the types of the loads involved, and the page number upon which they are discussed.

2.0 Description of the Design

2.1 Nose Assembly

The nose assembly was designed to reduce the impact load on the cabin. This was accomplished by designing the nose into two sections, as may be seen in drawing F93-1C-128-2. The forward section accounts for 60% of the nose length and crumples at

Table 1.0: Summary of Critical Detail Parts

Component	M.S.	Load	Page
Center Longeron	0.05	bending	12
2" Longeron	0.05	bending	12
1" Longeron	0.05	bending	12
Center Longeron	7.16	normal	11
Center Longeron	8.71	buckle	11
MS20470DD-4	0.82	shear	12
Floor skin	2.25	shr bkl	12
MS20470DD-8	5.0	shear	12
Floor Ribs	4.62	buckle	12
Primary Shaft	0.81	buckle	13
Secondary Shaft	2.68	buckle	13
Primary Shaft	2.13	shear	13
Secondary Shaft	2.13	shear	13
Front Spar	0.231	bending	13
Rear Spar	0.231	bending	13
Wing Skin	0.10	torsion	14
Wing Rib	0.19	torsion	14
Stringers	0.062	torsion	14
Front Carrythrough	0.698	bending	15
Rear Carrythrough	1.81	bending	15
Wing Attachments	0.05	shear	16

a load of 8g's. The second part of the nose assembly crumples at 17g's. The load carrying members which determine the crumpling are the longerons in the nose assembly. They were designed specifically to buckle

under the given loadings.

The forward nose longerons are mounted to the prop bulkhead and a nose frame, which acts to split the loading into two stages. The aft nose longerons are mounted to the nose frame and nose bulkhead. The longerons are mounted using a bracket on either end of each longeron, which connects to the interface surface (i.e. a bulkhead or frame).

An access panel was included in the nose assembly, on the port side of the nose. The access panel runs the length of the nose, and is mounted with AN526C-6-32 screws to the prop and nose bulkheads, as well as the nose frame. The skin is mounted on the remaining surface area of the nose. MS20470DD-4 rivets are used to fasten the skin to the longerons, the prop and nose bulkheads, and the nose frame. These rivets are also used to mount half inch angles to the inside of the nose skin, parallel to the longerons. These channels are to stiffen the nose skin against aerodynamic loads from the prop.

2.2 Cabin Assembly

2.2.1 Longeron, Cabin Side, and Roof Design

Before doing the actual structural design of the cabin, the volume constraints imposed by FAR 23 were considered. A Spatial Requirements Specification Document that complies with governing regulations was prepared to address the volume constraints. This document determined the minimum volume required in the cabin for the pilots and JAARS crew seats, as may be seen in Appendix A. The cabin structure was then ready to be designed around these volume constraints.

In approaching the cabin structural design, it was determined that the cabin would undergo normal, bending, buckling, and shear loads. These loads are due to maneuvering and emergency landing conditions, as discussed further in section 3.0.1. The first considerations made were for normal, bending, and buckling loads which were designed to be taken in the longerons of the cabin structure. These considerations were further split into examining the center structure longerons of the drive shaft support box and the longerons run in the floor structure. By designing the longerons to handle the previously mentioned loads, the requirements of the SOW regarding occupant safety demands of FAR 23 are met in part. The remaining part of the requirements will be satisfied in the floor assembly.

The four center longerons are designed to carry the entire forward impact load of the aircraft, without failing in compression or by buckling. The bending loads were designed to be distributed between the center longerons and the floor longerons. The layout of the longerons may be seen in drawing F93-1C-129-2. The center longerons are actually part of the drive shaft assembly and may be seen in detail in drawing F93-1C-150-2.

There are a total of six major load carrying longerons, all mounted as part of the floor structure. The longerons running through the base of the door frame assembly are actually split into four separate longerons in order to interface with the door frames. The longerons are all mounted at their ends to their respective interface surfaces (i.e. bulkheads or door frames) by mounting brackets that are shown in detail in Figure 2.2.1.
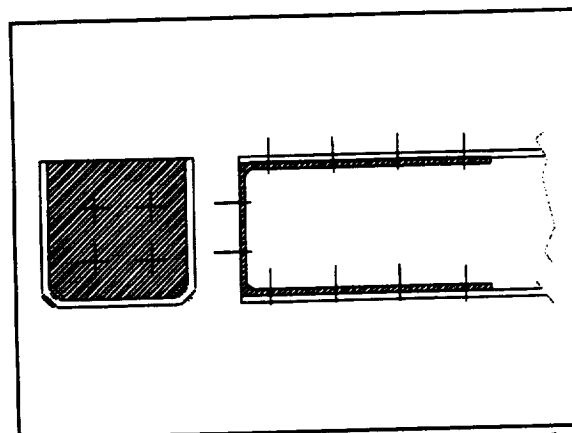

Figure 2.2.1: Standard Longeron Mounting Bracket

The door frame assemblies have been sized accurately. Actual detail design of these assemblies was conducted in order to obtain the sizes. However, these details are beyond the requirements of the SOW, therefore, they are not included in this report. The resulting dimensions of this design process are applied to the cabin structure volume constraints and layout.

Two-inch longerons, similar to the floor longerons are mounted at about the middle of the height of the cabin. These are to help carry the overall loadings on the cabin, but do not carry any major loads. The same is true for the half inch angles mounted between these upper two-inch longerons and the floor longerons (refer to drawing F93-1C-129). They are riveted to the skin to increase the stiffness, but are considered to carry no major load.

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Additional one-inch longerons are mounted in the bottom of the floor structure to aid in carrying the bending loads on the structure. The two-inch longerons are fastened with brackets similar to the two-inch floor longerons, but the half inch angles and one-inch longerons require no bracket mounts.

The nose and aft firewall are supported by angles which run around the fuselage cross section, sandwiching the bulkhead and firewall between the two rings of these angles. This may be seen in drawing F93-1-129-2. The forward bulkhead was designed solely to act as a close out between the cabin area and the control systems and nose gear mountings. This is to keep loose items in the cabin area (such as pencils, etc.) from rolling into the control system and nose gear area. This bulkhead is mounted to its angle frame with AN526C-6-32 screws to allow removal of the bulkhead for access to the control systems and nose gear. Cutouts are made in the bulkhead to allow the longerons to pass through, uninterrupted.

The roof structure and the aft door frames interface with the aft firewall and angle as shown in drawing F93-1C-129-2. The roof structure consists of three channels mounted together with the typical longeron bracket mounts. This structure is also mounted to the top of the two door frame assemblies.

The other major components of the cabin assembly consist of the instrument panel, cabin light, door, windshield, JAARS seat, drive shaft, and floor assemblies. The floor assembly will be explored in further detail in the next section. The drive shaft assembly will be discussed in detail in section 2.3.

2.2.2 Floor Assembly

The floor assembly was designed to withstand the loads imposed by the pilots and their seats on impact, in addition to withstanding the loads imposed on the aircraft while maneuvering. All torque in the aircraft is designed to be carried by the floor structure.

In approaching the layout of the floor structure, considerations were first made for the shear flow throughout the cross section. The floor skin and longerons were designed to take the torque loads, without buckling from the resulting shear flow. The floor skin was designed to be 0.06 inches thick, with two minor angles spaced 4.6 inches apart and mounted on the floor skin between the floor ribs. This

may be seen in the floor cross section view in drawing F93-1C-127-2.

The function of the floor and seat support ribs is mainly to absorb the downward load of the JAARS seats on impact. They also aid in providing bending and torsion support, but are not designed to carry these loads directly as major load paths. The floor ribs are the ribs running perpendicular to the longitudinal axis of the aircraft. They are designed to carry a full 18g downward load from the seat and pilot, as the worst possible loading case. The seat support ribs, which run parallel to the aircraft longitudinal axis, are designed for the same load. The longerons mounted to the bottom of the floor structure are to run through cutouts in the floor ribs. In order to accommodate the floor skin thickness, MS20470DD-8 rivets are used in the ribs to mount the ribs to the floor skin. However, MS20470DD-4 rivets are used to mount the ribs to the bottom of the floor structure, which is the 0.03 inch thick fuselage skin.

The remaining floor components are the one and two-inch longerons. The longerons were discussed in detail in section 2.2.1 and are riveted to the bottom of the floor structure with MS20470DD-4 rivets, as seen in drawing F93-1C-127-2.

2.3 Drive Shaft System

2.3.1 Design and Crashworthiness

The unique design of this aircraft is the mounting of the engine on top of the wing box. This called for an extensive design of a drive system to span the 108 inches separating the engine and the propeller. The design inherently required significant considerations to the size, weight, vibration, occupant safety, manufacturability, and maintenance requirements of the shaft.

The primary function of the drive system is to transmit the torque produced by the engine efficiently and safely. One of the first considerations was safety during the event of an emergency landing where in either the shaft or engine would shear from its attachments and strike the occupants. It was decided to place a support system directly below the shaft as a primary load path and to ensure sufficient support for the shaft in the event of a crash. The four members which assemble to form the central structure were sized to carry bending loads through the cabin and also to provide substantial support for the 18g loads experienced by the shaft during emergency landings. These members also define the outside edges of the drive

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shaft housing that runs through the cabin area.

The housing itself provides the area to run necessary equipment from the engine to the cabin compartment. The control systems for the elevator and rudder run along the floor directly underneath the drive shaft. The electrical and vacuum tubes are attached to the structural member supporting the shaft. Venting for cabin heat also runs through this channel. A plastic skin is all that is required to cover the sides and top of the channel. This skin is held on by quick release fasteners so as to provide easy access to the internal parts. Allocation of space for each subsystem is denoted in drawing F93-1C-152-2.

The entire drive shaft assembly includes a primary and secondary shaft, three bearings, and two universal joints. Bearings are located at stations 1, 37, and 103. Universal joints reduce vibration and forces due to misalignment and are located at stations 5 and 108. The universal connections also allow the drive system to operate free of bending loads resultant from aerodynamic, inertia or engine loading. The relation of each universal joint with the spline gear connections of the primary and secondary shaft along with that of the union of the engine mounting shaft can be seen in drawing F93-1C-151-2.

The spline connection at the rear bulkhead allows for thermal expansion and easy removal of the drive shaft and may be seen in Figure 2.3.1.1. The Second spline system is located in the nose cone just aft of the forward bulkhead. This spline system incorporates a shear pin connection for emergency landing safety. The Shear pin is specially designed to fail at 8g's, allowing the secondary shaft to collapse within the primary shaft and therefore eliminating the chance of either shaft shearing apart and injuring an occupant. This design is similar to that of a collapsible steering system used in cars today. The collapsing nose assembly, in addition, absorbs the shock of impact and therefore further protects the occupants. Once the nose assembly has completely crumpled, the shaft acts as a support to hold the engine aft of the cabin area.

From this design, the drive shaft holds a three-fold crash worthiness agenda. Therefore, the only possibility for the engine to enter the cabin area is for the engine to rotate around the universal, up into the cabin area. This is to be prevented by mounting an engine support to the front, bottom of the engine. This support would then be connected to the center
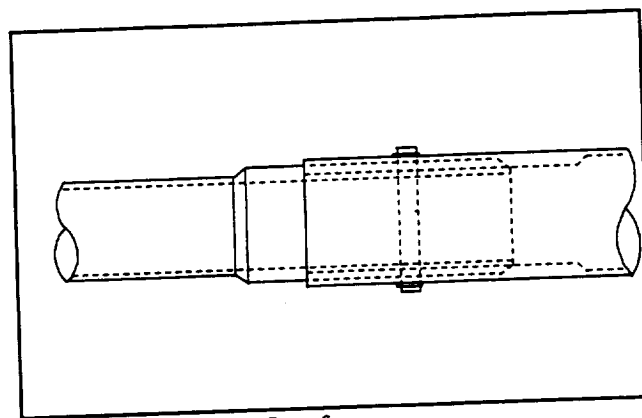

Figure 2.3.1.1: Spline Interface

longeron structure under the drive shaft. The support would then prevent the engine from rotating about the universal, thus completing assurance that there is no possible way for the engine to enter the cabin area in crashes.

This support can not be designed at this point, as research into the mounting points on the O-235 engine resulted in no existing engine mounts at the required points. However, there are bolt locations in the required area which may be picked up in order to mount the support. Coordination of efforts with Lycoming would allow the mount to be made in this required location. Once the support location is confirmed from this process, it will be possible to size the support accordingly. This is recommended for future completion of the Viper aircraft overall design.

2.3.2 Bearings
Bearing location and type plays an important function in the drive shaft design. The first bearing is located at the forward bulkhead. The bearing is a thrust bearing designed to hold the thrust of the propeller and distribute the load into the structure of the aircraft. The second bearing is located just aft of the instrument panel and inside the cabin area. The bearing's location allows support for the weight of the shaft. The bearing itself is mounted onto the structural member running in the middle of the cabin area. This bearing is also responsible for holding the shaft in place during an emergency landing. The third bearing is located at the firewall and holds the end of the drive shaft. This bearing is to take out engine vibration. The bearing is mounted to stiffeners running from longerons along the firewall. These same stiffeners hold the structural member.

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Bearings are all sealed to keep out dust and dirt. The bearings are also all greased bearings. An oil system was considered, but was found to be costly and hard to maintain. The grease system is adequate for the engine rpm expected. The bearings will have to be greased as part of the 100 hour maintenance check.

To provide access to the bearings, access panels in the drive shaft assembly housing are provided at the firewall and at the base of the instrument panel. Access to the forward bearing can be made through the skin of the nose cone.

2.4 Wing Assembly

2.4.1 Spar Design

The front spar is located at the quarter chord, carrying a maximum of 91% of the moment due to lift. The rear spar lies on the 70% chord and is responsible for up to 21% of the total moment.

Both the front and rear spar are 2024-T3 aluminum alloy sheets, which have been blank pressed and brake formed to their respective shapes. The principal form of each cross section is a "c" channel which faces to the interior of the wing.

On the front spar, the flange width initiates at 2 inches to accommodate the fuselage interface. Between stations 43 and 55 this width is reduced to 1 inch, which provides sufficient strength. The design of the rear spar is slightly different with the flange starting at 3 inches and reducing to 1 inch at station 90. Lightening holes have been employed to reduce weight where the design section modulus exceeded the requirements.

2.4.2 Wing Skin and Ribs

The wing skin, ribs, and stringers were designed to withstand the torsional loads applied during flight and landing conditions.

In approaching the layout of the rib spacing, the resulting shear flow is distributed through the skin panels, ribs, and stringers. The skin panel and rib thicknesses vary across the span from 0.02 inches at the tip to 0.032 inches across the center to 0.071 inches thick over the fuel tank. Three stringers are to be spaced at 36.7%, 48.4%, and 60.2% of the mean aerodynamic chord to reduce the shear flow in the skin panels. Seven ribs are spaced across the span

and can be seen in Drawing F93-1C-160-2.

The ribs are designed in two parts. The leading edge portion is riveted to the front spar and the leading edge skin panels are riveted to this portion of the rib. The wing box portion of the rib is riveted to the front and rear spars. The ribs are designed to allow the stringers to pass through uninterrupted. Therefore, cut outs are made at the appropriate positions in the ribs (refer to Drawing F93-1C-160-2).

2.4.3 Carrythrough

The carrythrough structure was designed to carry the lift and drag loads on the wing for worst case load conditions. The carrythrough assembly was also designed to include interfaces between the cabin floor structure and the empennage structure.

In designing the carrythrough structures, the size constraints due to the size of the wing spars at the root were imperative to consider. The location of the carrythrough structures relative to the firewall and rear bulkhead were also of concern, as the manufacturing process at these locations determined the shape of the structures. Another important factor to be considered was fatigue performance. Upon consideration of all of these factors, the carrythrough structure cross sections depicted in drawing F93-1C-170-2 were designed.

The front carrythrough structure, consisting of the front plate and front channel, is mounted to the firewall. The rear carrythrough structure, consisting of the rear plate and rear channel, is mounted to the rear bulkhead. This may be seen in drawing F93-1C-170-2 and will be discussed in detail in section 5.4.3. The cross sections of the structures allow for mounting of the wing attachment fittings within the front and rear channels.

Two-inch longerons are run between the front and rear carrythroughs at the same height as the floor longerons in the cabin structure. The two upper outboard longerons are mounted to the carrythrough channels with standard brackets. The two lower inboard longerons are mounted to the flanges of the carrythrough channels with standard brackets. This may be seen in drawing F93-1C-170-2 and will be discussed in detail in section 5.4.3. One-inch longerons are also placed in line with the floor ribs to continue those load paths from the fuselage to the empennage.

Due to the design of the cabin structure, the cross section of the Viper was increased, thus also requiring an increase

in the diameter of the empennage. Therefore, the empennage will need to be redesigned to account for this. During this redesign, it is planned that the longerons in the empennage may be placed such that they coincide with the placement of the longerons in the carrythrough assembly. This would complete the load path through all longerons throughout the aircraft.

2.4.4 Wing Attachments
The wing fitting attachments consist of the conventual quadruple shear lug and the double shear lug. The front spar attachments consist of a top and bottom quadruple shear lug on both sides of the aircraft. The front spar lugs taper into c-channels which are riveted into the carry through structure and the front spar. The c-channel slides between the carry through structure and the firewall. The other side of the lug is fitted inside the c-channel spar. The rear spar attachment consist of a single double shear lug tapering into an I-beam, which again is riveted to the carrythrough and spar. Again the I-beam is fitted inside the carry through structure and is fitted inside the rear spar. The front spar lugs have 2024 aluminum NAS1314 bolts and the rear spar has 2024 aluminum NAS1310 bolts.

2.5 Structural Decomposition
Appendix D illustrates the total structural decomposition for the cabin-fuselage and wing assembly structural design concepts. This overall concept combines the separate assemblies which are discussed in sections 2.1 through 2.4.

3.0 Loads and Loading

3.0.1 Loading Constraints
In considering the required loading constraints, FAR part 23 was consulted. The worst case loads were determined to be 4.4g's up, due to maneuvering; 2.2g's down, due to maneuvering; 18g's forward, due to emergency landing; and 4.5g's sideways, due to emergency landing. These loads were determined by examining the flight and crash load criteria in FAR A23 and FAR 23.561(b)(3), which were found to be the highest required loadings. The flight loads caused by gusts (n_3 and n_4) were checked using FAR A23 and were determined to be no greater than the maneuvering limit load factors (n_1 and n_2).

Before designing the front and rear spars and carrythroughs, it was necessary to determine the maximum percentages of lift experienced by each component. This was done for the front spar by finding the x_{cp} position for the maneuver condition. This was then divided by the distance between the front spar and rear spar to give the percentage of lift carried there. The result was a maximum of 91% of the lift required to be carried by the front spar. The process was done similarly for the rear spar, but the x_{cp} for the dive condition was used, as this condition produces the worst case load for the rear spar. The result was a maximum of 21% of the lift to be carried by the rear spar.

3.1 Nose Assembly

Since the nose assembly was designed to fail in two parts, the design of the longerons were considered in two parts. A forward load factor of 18 was used in the calculations, as was discussed in section 3.0.1. The longerons were then designed to buckle at 8g's and 17g's by determining the required moment of inertia of the longerons, as will be discussed in section 4.1. The forward load is assumed to be carried through the longerons only, as the skin will buckle before longeron failure.

3.2 Cabin Assembly

3.2.1 Longeron, Cabin Side, and Roof Design
In first approaching the cabin longeron design, the yield strengths in tension, compression, and shear were determined for 2024-T3 aluminum. They appear in Table 3.2.1.1. These were used, assuming a margin of safety (MS) of 0.05, to determine maximum allowable stresses for each failure mode, as shown in Table 3.2.1.2. The general equation used was MS = $(F_y)/(f_{allow})$ - 1.

Table 3.2.1.1: Yield Strengths of 2024-T3 Aluminum

$F_{y,t}$	61 ksi
$F_{y,c}$	41 ksi
$F_{y,s}$	32 ksi

These stresses were used to size the cabin structure longerons through determining the maximum loads permitted on each longeron. The maximum allowable loads determined for the longerons are discussed in detail in sections 3.2.1.1 through 3.2.1.3 and are listed in table 3.2.1.3.

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Table 3.2.1.2: Allowable Stresses for Longeron Sizing (2024-T3 Al & MS=.05)

$f_{allow,tension}$	57.95 ksi
$f_{allow,compression}$	38.95 ksi
$f_{allow,shear}$	30.4 ksi

Table 3.2.1.3: Allowable Loads for Longerons

P_{norm}	28,800 lb	center struct
$P_{ctr\ long}$	579.6 lb	bending
$P_{2"\ long}$	131.4 lb	bending
$P_{1"\ long}$	20.5 lb	bending

3.2.1.1 Normal Loads

In approaching the loading for the longeron structures, three separate areas were considered. They are normal, bending, and buckling. The first load types considered were normal loading and buckling on the cabin structure. All of the normal and buckling loads were designed to be carried by the center longerons, due to the large size of the center structure. The normal load, P_{norm}, was calculated to be 28,800 pounds, due to an 18g impact load, n_{fwd}, using $P_{norm} = (n_{fwd})(W_{gross\ take\ off})$. This normal load was used to determine the compression and buckling loads on the center longerons.

3.2.1.2 Bending Loads

The next load considered was for the bending induced on the cabin structure by maneuvering loads. The maneuvering loads act at the cg of the aircraft, which is located at the firewall. Each maneuvering load then results in a reaction load from the weight of the forward fuselage acting in the opposite direction of the maneuvering load. The worst case loads are due to an upward acceleration of 4.4g's and a sideward acceleration of 4.5g's. The structure was designed entirely for the very worst case load of 4.5g's. The resultant bending load, $P_{bending}$, was determined to be 3,429 pounds, by multiplying the weight of the forward fuselage (762 pounds, including all components forward of the aircraft cg and the JAARS seats and pilots) by the 4.5 load factor. The moment carried by the total bending structure was found to be 277,749 inch-pounds. This was found from $M_{bending} = (P_{bending})(L)$, where L is the length of the longerons

(which is the worst case moment arm from the cg to the applied load).

The total bending load was then designed to be distributed between the center longeron and the floor longerons. The allowable load on each center longeron was first determined. This was done by finding the allowable moment, $M_{center\ long}$, on a center longeron from $f_{allow,s} = (M_{center\ long})(c)/(I)$, using c and I properties calculated for one center longeron's cross section. A corresponding load, $P_{center\ long}$, of 579.6 pounds was found by dividing this moment by the length of the longeron. This load is the greatest load that may be applied to a center longeron without failure. Next, $P_{center\ long}$ was divided by $P_{bending}$ to determine the percentage of the total bending load that one center longeron can carry. This corresponds to 16.9% of the total load. Therefore, the four center longerons carry a total of 67.6% of the total load.

The remaining bending load was designed to be carried by six two-inch and four one-inch longerons. The loads carried by these longerons were found iteratively using the *Mathematica* (Wolfram, 1988) computer program. The two-inch longeron loads were found first. This was accomplished by programming moment of inertia equations with corresponding variable cross sectional dimensions into *Mathematica*. This generated "c" and "I" values for inputted cross sectional dimensions. These "c" and "I" values were then imported into the second part of the program which solves the equation $f_{allow,s} = (M_{2"\ long})(c)/(I)$ for $M_{2"\ long}$. The $M_{2"\ long}$ value was then divided by the length of the longeron to determine the maximum allowable load on the longeron, $P_{2"\ long}$. Various cross sectional dimensions were inputted into the program until an optimal trade between load capacity and structural volume constraints was determined. The resulting value was $P_{2"\ long} = 131.4$ pounds, corresponding to 3.8% of the total bending load. Therefore, six of these longerons carry a total of 22.8% of the total bending load.

The same process used for the two-inch longerons was repeated for a smaller cross section and the one-inch longeron cross section was determined. The one-inch longeron was determined to carry 20.5 pounds ($P_{1"\ long}$), resulting in a capacity of 3.5% of the total bending load for four of these longerons. The remaining 6% of the bending load is assumed to be carried by the floor, door frame, and roof structures.

3.2.1.3 Additional Load Paths and Interfaces

In addition to carrying part of the bending load, the two-

inch longerons act as load paths to the nose and engine/wing box stations. The nose longerons are fastened through the nose bulkhead to the upper two-inch longerons and the two-inch floor longerons. These same two-inch longerons are then fastened through the firewall to interface with the structure in the engine/wing box section of the aircraft. This allows for a continuous flow of bending and normal loads throughout the fuselage.

The loads on the interface fasteners were determined assuming a worst case scenario in which the weight of the aircraft components aft of the cg acts in shear on the fasteners. The weight (840 pounds) is distributed over the 16 interface fasteners. Therefore, the load on each fastener, $P_{interface}$, was determined by multiplying the aft components' weight times the 4.5 load factor and dividing by 16. The resulting value was 236 pounds for $P_{interface}$.

The center longeron structure is connected with the two-inch floor longerons by channels, as seen in drawing F93-1C-150-2. These channels allow the center structure to be connected to the longeron structure, thus creating a load interface between the two structures. Another load path is created at the door frame assembly interface with the two inch floor longerons and at the roof assembly. In this manner, the door frames and roof are integrated into the total flow of loads throughout the cabin structure.

3.2.2 Floor Assembly

As previously mentioned, the floor structure was designed for two main purposes, torque carrying capacity and downward loading of the JAARS seats. The loads are discussed in detail in sections 3.2.2.1 and 3.2.2.2.

3.2.2.1 Torque

Torque is applied to the fuselage in two possible ways. Torque is created by the difference in lift on the wings when turning and by the inertial side load on items of mass in the cabin. The wing torque was the first to be determined. This was done by assuming a difference in lift of 30% of the total lift (70% lift on one wing versus 100% on the other, when banking) acting at the aerodynamic center of the wing. The aerodynamic center was assumed to be at 50% of each wing, as a worst case position. This distance from the centerline of the aircraft was used as the moment arm for the lift. It was multiplied by 30% of the total lift to determine the resultant torque,

T_{wing}, of 51,360 inch-pounds.

The inertial torque was next found. The first step in doing this was to find the resultant cg position for the seats, pilots, and instrument panel. This cg position was referenced to the center of the floor cross sectional area. This arm was then multiplied by the combined weight of the seats, pilots, and instrument panel to find the resultant inertial torque, $T_{inertial}$, of 10,105 inch-pounds.

The wing and inertial torques act opposite of one another, therefore the higher torque is chosen for shear flow determination. The shear flow, q_{floor}, resulting from the higher wing torque was found by solving $T_{wing} = 2(A_{floor \, x-sec})(q_{floor})$. The result was $q_{floor} = 120$ pounds per inch. This shear flow will be used to determine the maximum allowable shear buckling stress in the floor cross section. It was also used to find the fastener bearing load, resulting from the shear flow, in the floor fasteners. This was done by multiplying q_{floor} by the floor rivet spacing of two inches, resulting in $P_{br. \, floor \, fastener} = 1,440$ pounds.

3.2.2.2 Buckling Loads

The last floor loading consideration was buckling due to the downward load of the JAARS seat and pilot. This was calculated assuming a worst case scenario of a full 18g acceleration down on the floor and seat support ribs. The smallest length rib was also chosen to provide the highest loading criterion. The load was found by multiplying the weight of the seat and pilot by the 18g load factor, resulting in $P_{floor, \, buckle} = 3,420$ pounds. The load must be resisted in buckling by the floor and seat support ribs.

3.3 Drive Shaft Assembly

The primary load consideration for the drive shaft is in the transmission of power through the eight foot span separating the engine and propeller. The drive shaft itself is not responsible, nor expected, to carry any loads resultant from aerodynamic or inertial forces. Such loads will be expected to be transmitted by the cabin structure and the central beam assembly supporting the shaft by the bearings. Sizing of the primary and secondary shaft is, therefore, dependent on the critical strength needed to resist shear or buckling under maximum loading.

Limits set forth by FAR 23 require the drive shaft to support loads without detrimental or permanent deformation under the torque produced by the maximum rated takeoff power and related prop speed. Noting that

both shear and buckling restrictions are dependent on the maximum torque carried by the drive shaft and because of the importance of pilot safety in the unique design of the Viper, the frequency of the propeller was taken at a value 400 rpm slower than anticipated along with the use of a factor of safety of 1.5. Through the direct relationship between power, torque and angular velocity, the maximum torque on the drive shaft was computed as $Torque_{max}$ = $(Power_{max})(FS)/(Angular\ Velocity_{min})$. The Avco-Lycoming O-235 is certified at a max power output of 125 hp. Coupled with an expected 700 rpm frequency at initialization of take off, the maximum torque placed on the shaft assembly is 1406 foot pounds.

Iteration with size and stress levels indicated the required wall thickness of the primary and secondary shaft to be 0.0625 inches. Methods provided by NACA TN 3783 (Gerard and Becker) indicated the critical shear strength for buckling to be well below that required for shear fracturing to occur. Prediction of the critical buckling stress under pure torsional load was made as follows:
$T_{critical}=(K_t*Pi^2*E/(12*(1-.3)^2)*(t/L)^2$ where K_t is a factor based on the parameter Z_t as denoted in the technical note and its figure 26. Shear stress in the shaft was calculated as $Shear\ stress$ = $(Torque_{max})(Radius_{outer})/(Polar\ moment\ of\ Inertia)$.

Transmission of torque between the primary and secondary shaft is concentrated in the spline gear union. The shear pin which is designed for failure at 8g's, ten times the expected load encountered under engine torque, is not used for this function. The primary purpose is to restrict crumpling of the nose assembly until the design impact level has been reached.

Secondary load considerations of the drive shaft include forces produced due to vibration and misalignment of the shaft. In order to hold such loads to a minimum, the drive shaft design has incorporated a support system of three aluminum housed steel bearing in a combination with two universal joints at each extreme of the shaft assembly.

A variation of the operating temperature of the shaft in the range of 250 F results in a thermal expansion of approximately 0.25 inches over the 100 inch span. Aft of the mid-bearing, this expansion is to be absorbed by the rod extending into the aft spline gear union. In the section between the nose and mid-

bearing, thermal expansion forces are placed on the frontal thrust bearing and into the nose structure.

The final sizing consideration is in the ability for the shaft to remain centered within the housing from the point of the spline gear union aft. Motion in both lateral and longitudinal directions is limited to the shear strength of the central bearing mounting bolts. Sizing of the bolts to meet the 18g crash worthiness requirement in the forward direction ensures substantial strength to resist travel from side-to-side. Eight bolts, as seen in drawing F93-1C-151-2, are responsible for carrying the 28,800 pound force in shear experienced in emergency deceleration. Assuming equal distribution among bolts indicates a selection of AN5C4 bolts. Sizing of the bolts at 5/16 of an inch provides a margin of safety of: $MS_{mounting\ bolts}$ = 0.59 .

3.4 Wing Loads

3.4.1 Spar Design
The wing loading was calculated by assuming two simple pressure distributions: a trapezoidal and an elliptical. The average of the two curves was then taken to obtain the final loading seen in Figure 3.4.1.1. The moment as a function of buttock line location was then calculated using the estimated loading distribution. Figure 3.4.1.2 illustrates the total moment distribution over the half span of the wing. Location of the front and rear spars on the wing planform established the percent of total moment distribution carried by each structure.
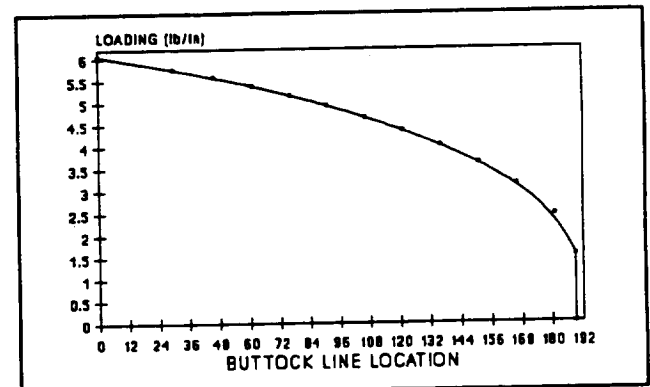

Figure 3.4.1.1: Wing Load Distribution

The front spar location at 0.25c requires that it carry a maximum of 91 percent of the total load on the wing. The rear spar, located at 0.70c, is required to carry at most only 21 percent. The maximum percentage of the moment

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handled by the front spar was found to be at maneuvering speed at a high angle of attack. The limiting design criteria for the rear spar was found to be at dive speed and low angle of attack.
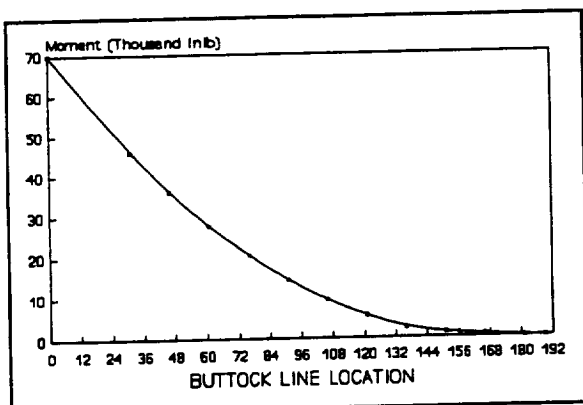

Figure 3.4.1.2: Wing Moment Distribution

Sizing of the spar cross section was determined by the section modules requirements needed to achieve an endurance limit on cyclic loading of 10^7 cycles. The S-n curves for 2024-T3 aluminum alloy placed the maximum allowable stress at 27.5 ksi. The required section modules was therefore calculated as follows:
$S_o = (M)$(maximum cyclic l.f.)$/f_{allowable}$
where $f_{allowable} = F/1.01$, providing a margin of safety of 0.01. The maximum cyclic load factor was taken as 2.2.

A portion of skin, averaging 3 inches in width, was added to the cross section of the spar to obtain the effective moment of inertia in bending. Lightening holes employed in the spar structure allowed reduction of the overall weight while maintaining approximately 90% of the section modules. Though sufficient room was allotted for stress concentrations, further analysis may need to be done on this area.

3.4.2 Wing Skin and Ribs

In approaching the loading for the skin and ribs, torsion was determined to be the sizing criteria for the skin panels, ribs, and stringers. A spanwise torsional graph (shown in Appendix C) was constructed according to FAR 23A. The flight loads at dive, maneuver, flap speed, and flap speed with ailerons deflected are determined by the equation $T = qC_m c_{loc} S$. C_m and the dynamic pressure, q, are determined according to the above configurations and the speeds are taken from the v-n Diagram shown in Appendix

B. the loads applied to the wing from the flaps and ailerons are accounted for by the difference in C_m. C_m for the clean wing, using 65_2-415 airfoil, was determined in the preliminary design report to be -0.0473 and $C_{m,flap}$ is -0.3 using a conservative 60° split flap.

To account for skin buckling during landing, the force of one main wheel upon touchdown is calculated by $F = knW$, where k is a correction factor, n is the load factor, and W is the aircraft gross weight. From FAR 23, K=0.25 and n=2.67 and the force computed was 1167.5 pounds. This force multiplied by the distance from the landing gear attachment point to the ground caused a counter-clockwise torque (as seen from the tip) of 32689 inch pounds. This torque is distributed through the rib to the spars and the inboard to the fuselage. The design of the skin and ribs is determined by the curve that represents the maximum local torque.

3.4.3 Carrythrough

As mentioned in section 2.4.3, the carrythrough structures were designed to carry the wing loads. The moment due to lift acting on the wing at a distance of half of the semi span from the center of the wing, was first determined. This was done at the worst case of 4.4g's, as required by FAR 23 for utility category aircraft, resulting in a moment of 365,541 inch-pounds. The percentages discussed in section 3.0.1 were then applied to the front and rear carrythroughs to determine the worst case load experienced by each. These loads were then to be used to determine the required section moduli of the carrythroughs to withstand the loads, as will be further discussed in section 4.4.3. A similar process was conducted for drag. The moment due to drag was found to be 21,470 inch-pounds, corresponding to a drag of 452 pounds, calculated at the dive condition.

The only other load to be considered in the carrythrough structure was the shear load due to lift, which must be carried by the fasteners. This was found to be 3,848 pounds in each half of the carrythrough structure. The landing loads were not considered in any part of the design, as these loads are less than the 4.4g loads experienced in flight. The snow load condition was also checked, as will be discussed in section 7.1.5, and found to be insignificant in comparison to flight loads, as well.

3.4.4 Wing Attachments

There were several conditions looked at in order to size the lugs for the front and rear spar attachments. One of the loads considered is the landing loads. Two types of landing conditions were considered the direct moment of

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the load to bend the wing up and the torsional load of twisting the wing off. The other loads considered wear flight loads. The conditions how high angle of attack and high velocity were calculated. After calculating all these loads the highest value was found to be the moment due to lift at 4.4 g's. The proportion of the loads on the spars turned out to be 91% on the front spar and 21% on the rear spar. The load found to be on the front spar was 55.8 thousand pounds and on the rear 19.4 thousand pounds. A margin of safety of 1.02 was used to calculate the shear of the bolt and then the dimensions of the lug.

4.0 Structural Substantiation

4.1 Nose Assembly

4.1.1 Loading
From the loading considerations discussed in section 3.1, a required moment of inertia, I, for the nose longerons was found. This was accomplished by solving $P = pi^2 EI/L^2$ for I, where E for 2024-T3 aluminum and longeron length, L, were used. The P value used was the failure load factor times the weight of the aircraft. This was done for both the 8g and 17g crumpling loads. Both loads resulted in similar I values. An extrusion cross section, whose properties matched the loading requirements, was then selected.

4.1.2 Fastener Spacing
Once the sizing was completed for the longerons, the fastener spacing for each longeron was considered. The fasteners were sized to be four times the skin thickness of 0.03 inches, resulting in 0.125 inch fasteners. The skin thickness also determined that the longerons be 0.09 inches thick (three times the skin thickness). The minimum edge distances for the fasteners were then found to be 0.25 inches (two times the fastener diameter). Minimum and maximum spacing for the fasteners were determined to be 0.5 inch (four times the fastener diameter) and 1.0 inches (eight times the fastener diameter), respectively. These dimensions were used to lay out the fastener patterns shown in drawing F93-1C-128-2.

4.2 Cabin Assembly

4.2.1 Longeron, Cabin Side, and Roof Design
In response to the loading criteria discussed in section 3.2.1, various sizing requirements were found for the

cabin components. The material selected for the longerons was 2024-T3 aluminum. Sections 4.2.1.1 through 4.2.1.4 discuss the longeron and interface related structural substantiation results. Table 4.2.1 lists the margins of safety which were determined in this process.

Table 4.2.1: Cabin Margins of Safety

MS_{norm}	7.16	normal
MS_{buckle}	8.71	normal
$MS_{bending}$	0.05	bending
$MS_{interface}$	0.82	shear
$MS_{flr tqe}$	2.25	shr bkl
$MS_{flr fast}$	5.0	shear
$MS_{flr bkl}$	4.62	buckle

4.2.1.1 Normal Loads
The first loading considered was the normal loading constraint. From the constraint, it was calculated that a required cross sectional area, A_{req}, of 0.739 square inches must be used to resist normal load failure. This was found by solving $f_{allow,c} = (P_{norm})/(A_{req})$ for A_{req}. The cross sectional area of one center longeron, $A_{center long}$, was designed to be 1,433 square inches, thereby showing that the center structure alone easily supports the normal load. The corresponding stress in the total center structure, f_{norm}, was found to be 5,024 pounds per square inch from $f_{norm} = (P_{norm})/(A_{center long})$. The resulting margin of safety was found to be 7.16 from $MS_{norm} = (Fyc)/(f_{norm}) - 1$.

The buckling of the center structure was next examined from the normal loading condition. A required moment of inertia of 0.00377 in^4 was determined from the loading constraint. This was found by solving $P_{norm} = pi^2 E(I_{req})/L^2$ for I_{req} using E for 2024-T3 and the longeron length, L. The moments of inertia of one center longeron were next calculated and the lower of the two values was selected for the design. The design stress for the total center structure, f_{buckle}, was found from $f_{buckle} = pi^2 E(I_{req})/(A_{center long})L^2$ to be 4,223 pounds per square inch. The resulting margin of safety was 8.71 from $MS_{buckle} = (Fyc)/(f_{buckle}) - 1$.

4.2.1.2 Bending Loads
Bending loads were next examined. Section 3.2.1.2 discussed the process used in determining the loads on

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each longeron in detail. In this section it was mentioned that the bending load is distributed between the center, two-inch, and one-inch longerons. The load percentage for each and how each was determined is discussed in the section. In this procedure, the sizing of each longeron was completed as part of the process. As was mentioned in section 3.2.1.2, the loadings were found using a margin of safety of 0.05. This margin of safety was designed into the $f_{allow,j}$ value which was used in determining the load carried by each longeron. Therefore, the margins of safety for the center, one-inch, and two-inch longerons are all 0.05 and each longeron has a stress of $f_{allow,j}$ by design.

4.2.1.3 Interface Fasteners

In response to the interface fastener shear load, $P_{interface}$, determined in section 3.2.1.3, the fastener strength was considered. Aluminum (2024-T3) fasteners of a 0.125 inch diameter were selected to be used at the interfaces. Table 1, "Shear and Bearing Strengths of Aluminum Alloy Rivets," in *Aerospace Systems Detail Design* (Ladesic, 1993) was used to determine the fastener strengths. The value found from this table was an allowable shear load, $P_{allow,interface}$, of 429 pounds on the fasteners, which is far greater than the $P_{interface}$ value of 236 pounds. The resulting margin of safety was 0.82, found from $MS_{interface} = (P_{allow,interface})/(P_{interface}) - 1$.

4.2.1.4 Fastener Spacing

Once the sizing was completed for the longerons, the fastener spacing for each longeron was considered in the same manner as for the nose assembly. The identical process to that discussed in section 4.1.2 was completed for the cabin longerons. Identical results were determined for the cabin longerons, namely 0.125 inch diameter fasteners through a 0.03 inch skin. The skin thickness also determined that the longerons be 0.09 inches thick. The minimum edge distances for the fasteners were also found to be 0.25 inches and minimum and maximum spacing for the fasteners were determined to be 0.5 inch and 1.0 inches, respectively. These dimensions were used to lay out the fastener patterns in the standard longeron mounting brackets shown in Figure 2.2.1.

4.2.2 Floor Assembly

4.2.2.1 Torque

In order to meet the requirements of the imposed torque on the fuselage, the floor structure must be designed to resist buckling under the torque. The stress resulting from the shear flow was determined with the floor skin thickness, $t_{floor\ skin}$, of 0.06 inches. The shear flow found in section 3.2.2.1 was divided by the floor skin thickness. This produced the maximum allowable shear buckling stress in the floor cross section, $f_{floor\ torque}$, of 2,000 pounds per square inch.

The floor skin thickness was then used to determine the critical shear buckling strength of the floor skin, $F_{floor\ torque}$, which was found from $F_{floor\ torque} = kE(t_{floor\ skin}/b)^2$. In this equation, a k value of 3.6, corresponding to simple supports, and the E value for 2024-T3 aluminum were used. The b value was the spacing between the longerons in the floor, which is 4.5 inches. The resulting $F_{floor\ torque}$ value was 6,492 pounds per square inch. This yielded a margin of safety of 2.25, found from $MS_{floor\ torque} = (F_{floor\ torque})/(f_{floor\ torque}) - 1$.

The fastener bearing load, $P_{br.\ floor\ fastener}$, determined in section 3.2.2.1, was used to check the floor fastener strength. Aluminum (2024-T3) fasteners of a 0.25 inch diameter were selected to be used at the interfaces. Tables 1 and 2, "Shear and Bearing Strengths of Aluminum Alloy Rivets," in *Aerospace Systems Detail Design* (Ladesic, 1993) were used to determine the fastener strengths. The value found from these tables was an allowable shear load, $P_{allow,floor\ fastener}$ of 1,440 pounds on the fastener, considering the 0.06 inch floor skin thickness. This produced a margin of safety of 5.0, found from $MS_{floor\ fastener} = (P_{br,floor\ fastener})/(P_{allow,floor\ fastener}) - 1$.

4.2.2.2 Buckling Loads

The buckling load on the floor produced by the seats and pilots on impact was determined in section 3.2.2.2 to be $P_{floor,\ buckle} = 3,420$ pounds. The stress resulting from this load was found from $f_{floor,\ buckle} = (P_{floor,\ buckle})/(A_{floor\ rib\ x-sec})$ to be 2,714 pounds per square inch. The critical buckling strength of the floor ribs was then required to be found in order to compare with $f_{floor,\ buckle}$. This strength was determined from $F_{floor,buckle} = kE(t_{floor\ rib}/b)^2$. In this equation, a k value of 3.6, corresponding to simple supports, and the E value for 2024-T3 aluminum were used. The b value was the spacing between the longerons in the floor, which is 4.5 inches. The resulting $F_{floor,buckle}$ value was 15,264 pounds per square inch. This yielded a margin of safety of 4.62, found from $MS_{floor,buckle} = (F_{floor,buckle})/(f_{floor,buckle}) - 1$.

4.2.2.3 Fastener Spacing

The same process discussed in section 4.1.2 was completed for the floor fastener sizing and spacing. The resulting fastener diameter for the 0.06 inch floor skin was 0.25 inches. The minimum and maximum spacing were found to be 1.0 and 2.0 inches, respectively, with a minimum edge distance of 0.5 inches. The thickness of the floor ribs was designed to be 0.09 inches to allow for a maximum of three times the 0.03 inch fuselage skin thickness, as well as the 0.06 inch floor skin thickness.

4.3 Drive Shaft Assembly

The drive shaft was sized for buckling and torsion loads. The design for the shaft does not require a transverse load. Calculations were carried out to determine the proper thickness of the shaft to handle buckling and shearing tendencies. A shaft thickness of 0.0625 (1/16) inches was determined as substantial to resist failure in either mode.

Table 4.3.1: Allowable Shear and Buckling Stresses

	Actual Values	Buckling (Allow.)	Shear (Allow.)
Primary Shaft	17.20 ksi	31.2 ksi	54.4 ksi
Secondary Shaft	17.20 ksi	64.0 ksi	54.4 ksi

Table 4.3.1 shows the allowable and calculated stresses in the two shaft systems. By comparing these numbers, it is seen that the shaft will buckle before it actually shears in the primary section only. Using the buckling stress as the critical stress the thickness of the shafts was determined. The thickness was set at 0.0625 inches to provide insurance of design in the primary section and then maintained throughout the secondary sections for producability reasons. Values from Table 4.3.1 indicated the following margins of safety.

4.4 Wing Assembly

4.4.1 Spar Design

Table 4.3.2: Drive Shaft Margins of Safety

	Buckling	Shear
Primary Shaft	0.81	2.13
Secondary Shaft	2.68	2.13

Figures 4.4.1.1 & .2 clearly illustrate the "over-design" of the spar in its ability to meet the required section modulus. The required curve already incorporates a margin of safety of 0.01 in material strength characteristics, and yet the margin of safety between the designed spars and that required does not drop below 0.231.
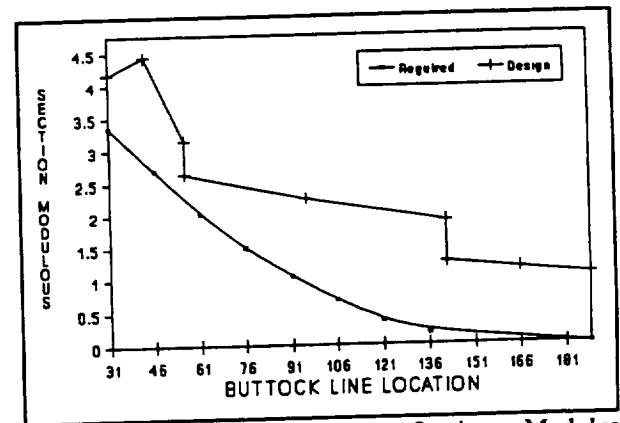

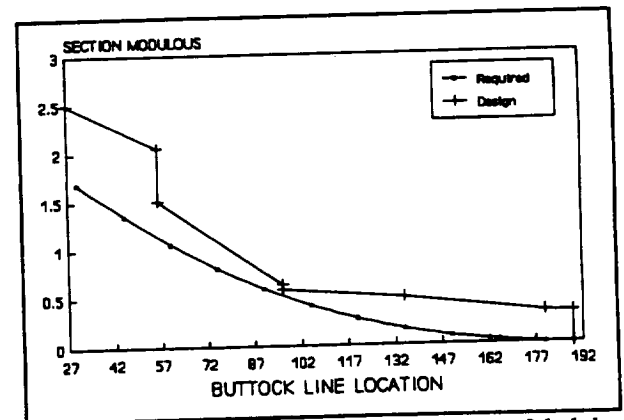Figure 4.4.1.1: Front Spar Section Modulus Requirements

Figure 4.4.1.2 Rear Spar Section Modulus Requirements

Load requirements for the front and rear were taken at the worst case scenario: maneuvering speed and high angle of attack for the front spar, and dive speed, low angle of attack for the rear spar.

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Cutaways in the spar web reduce the overall weight of the structure while at the same time allowing the moment of inertia of the cross section to remain fairly constant. The result is that the section modules remains almost constant. Stress concentration around the lightening holes will reduce the strength of the spar. Further structural substantiation must be considered for these points.

4.4.2 Wing Skin and Ribs

4.4.2.1 Torque

To determine rib spacing, the skin panels are designed not to buckle. This is accomplished by assuming a rib location and obtaining an estimate of its area. This area is used to determine the shear flow in the skin and ribs. Shear flow for both the skin and the ribs is computed from $q=T/2A$. To keep from buckling, the stress is calculated by dividing the shear flow by the appropriate thickness, $f=q/t$, and this is compared to the thin plate critical buckling strength $F_{s,crit}=KE(t/b)^2$ where K is determined from Figure 5.4.6, in the Nui text page 139, using a conservative two side simply supported curve with the appropriate a/b ratio. For the ribs, a/b is shown above. For the skin panels, b is the spanwise dimension between the known rib location and the assumed rib location, and a is the chordwise dimension between the front and rear spars and the stringers placed in between.

To size the skin, an average torque value between ribs is determined from the graph. It is assumed that this torque is distributed evenly across the a/b dimensions discussed above.

To size the rib, the torque is obtained from the graph at the rib location. The torque is assumed to be carried only within the wing box between the front and rear spar. Therefore, only the area within the wing box is used to determine buckling. Once it is determined that the ribs have a positive margin of safety lightening holes can now be addressed.

For ease of manufacturing, a 2 inch diameter flanged hole is used where possible. The holes occur in rib locations of LBL96 outward to rib location LBL170. The holes are to be centered between the stringers which determines D/b needed for Figure 6.2.3 (Nui, page 165). Using the hole diameter divided by the rib height, D/h, and D/b a value of k_1 is determined. This is the value used to adjust $F_{s,crit}$ where

$F_{s,crit}'=k_1F_{s,crit}$. The margin of safety values in Table 4.4.3.1 correspond to $MS_{rib}=F_{s,crit}'/f -1$.

Sections 4.4.2.1 through 4.4.2.3 discuss the skin, rib, and stringer structural substantiation. Results from this process are listed in Table 4.4.2.

Table 4.4.2: Skin and Rib Margins of Safety

$MS_{skin,170-190}$	8.95
$MS_{skin,142-170}$	0.69
$MS_{skin,120-142}$	4.9
$MS_{skin,96-120}$	2.68
$MS_{skin,56-96}$	1.18
$MS_{skin,31-56}$	0.10
$MS_{rib,170}$	1.1
$MS_{rib,142}$	1.1
$MS_{rib,120}$	0.19
$MS_{rib,96}$	4.8
$MS_{rib,56}$	1.6
$MS_{rib,31}$	1.6
$MS_{z\ stringers}$	0.062
$MS_{rivet,142-190}$	16.4
$MS_{rivet,31-142}$	5.4

4.4.2.2 Stringers

To reduce the shear flow in the skin panels, stringers are positioned at chord positions discussed in 2.4.3. It is assumed that the stringers act as beams fused at both ends where the load P is equal to the shear flow times the length of the panel. To size the stringer, the load is set equal to $P_{crit}=pi^2EI_{req}/L^2$ and the required moment of inertia is calculated using the largest shear flow seen by the stringers multiplied by the length of that panel (q=19.08lb/in and l=40in @LBL56-96). The resulting $I_{req}=0.0167$ inches[4].

Knowing the flange requirement of 4 times the rivet diameter equaling 0.5 inches, a z-stringer was chosen from Ladesic. The NAS346-3 equal leg, extruded z-stringer has a flange length of 1.0 inch and $I_{xx}=0.0124$ inches[4]. It is 0.625 inches in height and has a cross sectional area of

0.1968 inches^2. This results in the smallest margin of safety about the length of the stringer to be 0.062.

4.4.2.3 Fastener Spacing

For riveting the skin to the ribs, spars, and z-stringers, two different size rivets are used to maximize spacing and reduce drag. The diameters are determined by 4 times the skin thickness in the appropriate section. The MS20470DD-3 rivet is used from the tip up to and including the rib at location LBL142. The maximum spacing of 8 times the diameter in this section is 0.75 inches. The largest shear flow in this section (rib LBL142) is $q=18.4$ lb/in. this is compared to the q_{allow} which is equal to the allowable shear strength of 241 pounds (Ladesic) divided by the spacing, resulting in $q_{allow}=321$ lb/in. The margin of safety is computed by $MS_{rivet}=q_{allow}/q -1$ and is equal to 16.4.

In a similar manner, the rest of the skin uses MS20470DD-4 rivets. The maximum spacing is 1.0 inch with the largest shear flow of 79.0 lb/in occurring at rib location LBL56. The rivet allowable shear strength is 429 pounds resulting in a $q_{allow}=429$ lb/in. The margin of safety is $5/4$.

4.4.3 Carrythrough

4.4.3.1 Loading Constraints

From the loading constraints discussed in section 3.4.3, a required section modulus for 100% lift capacity on a carrythrough was determined. This was done by first applying a factor of safety of 1.5 to the ultimate tensile strength. The resulting limit load was used with a margin of safety of 0.01 to determine the maximum allowable stress. The stress was then used in $f=Mc/I$ to solve for the section modulus ($s=I/c$), where the moment used was the moment due to lift of 365,541 inch pounds as determined in section 3.4.3. The resulting section modulus was 9.07 cubic inches in the lift direction. The same process was completed for the drag moment of 21,470 inch pounds, resulting in a required section modulus of 0.533 cubic inches in the drag direction.

Next, the computer program *Mathematica* was used to calculate the section moduli of the front and rear carrythrough structures. The resulting moduli for the front carrythrough structure were 15.4 cubic inches in the lift direction and 2.78 cubic inches in the drag direction. For the rear carrythrough structure, the

moduli were 5.35 and 1.31 cubic inches in the lift and drag directions, respectively.

The resulting maximum stresses corresponding to the section moduli were found by dividing each corresponding moment by the section modulus. The resulting stresses for the front carrythrough were 23,736 and 7,723 pounds per square inch in the lift and drag directions, respectively. For the rear carrythrough they were 14,348 and 16,389 pounds per square inch. The resulting margins of safety for each carrythrough in the lift and drag directions were found using the limit load determined previously. This resulted in the margins of safety in table 4.4.3.

4.4.3.2 Fasteners

As mentioned in section 3.4.3, the fasteners are required to carry a shear load of 3,848 pounds distributed over each half of the carrythrough structures. The maximum load required in each fastener was found from *Aerospace Systems Detail Design* (Ladesic, 1993) table 1 on page 151 to be 429 pounds for a 1/8 inch 2024-T3 rivet. Therefore, the minimum number of rivets required to carry the shear load is 9 in each half of the carrythrough, or a total of 18 minimum rivets in the each structure. The minimum spacing requirements for the rivets, however, required a total of 180 rivets total in the structure the resulting margin of safety is 9.0, as seen in table 4.4.3.

4.4.3.3 Fatigue

As mentioned previously, one of the greatest requirements in designing the carrythrough was fatigue performance. In order to withstand the required safe life of 10^7 load cycles, the carrythrough structure cross section was determined to be as appears in drawing F93-1C-170-2. The process used in evaluating the fatigue characteristics of the wing assemblies is discussed in detail in section 7.7.

The front carrythrough was the most critical, as it was found that the calculated maximum stress, due to lift, was 11,869 pounds per square inch, versus a maximum allowable of 12,000 pounds per square inch (found from MIL-HDBK-5E figure 3.2.1.8(h)). The rear carrythrough was found to have a calculated maximum stress of 7,174 pounds per square inch versus a maximum allowable stress of 9,000 pounds per square inch. For drag, the front and rear carrythroughs had calculated maximum stresses of 3,861 and 7,450 pounds per square inch, respectively. The corresponding maximum allowable stresses were 8,000 and 10,000 pounds per square inch, respectively. Therefore, since in all cases the calculated maximum stresses are less than the maximum allowable

stresses, all components possess a minimum safe life of 10^7 load cycles.

Table 4.4.3: Carrythrough Margins of Safety

Component	Load	$f_{maximum}$ (psi)	M.S.
Front Carry-through	Lift	23,736	0.698
Rear Carry-through	Lift	14,348	1.81
Front Carry-through	Drag	7,723	4.27
Rear Carry-through	Drag	16,389	2.48
Fasteners	Lift		9.0

4.4.4 Wing Attachments

The lug design was chosen because of the ease of assembling and disassembling. Only three bolts need to be installed in order to put the wing on. The use of the quadruple shear lug on the front spar allows for a smaller bolt diameter. The channel inside of the front spar carry through was designed to carry the tensile and bending loads into the structure and allows for a double row of rivets to dissipate the load throughout the structure. The I-beam design for the rear spar is oversized for the load, but allows for the easy installation of the lugs into the structure. The lug can be slid into the structure and riveted into place with the double row of rivets on either side of the I-beam web. The web carries the load from the bolt lug to the I-beam structure. The margin of safety for the lug assemblies was selected to be 0.05.

The spar attachments have many cycles applied to them each flight. The lugs are very accessible to fatigue failure. The shear stresses for the bolts and lugs were at a mean stress of 5.3 ksi and below. This gives a maximum stress of 11.7 ksi. At this stress level the part has a life of one hundred million cycles.

5.0 Manufacturing and Maintenance

5.1 Nose Assembly

The nose assembly was designed for simplicity of construction and ease of access to the nose bearing. The longerons are to be Alclad extruded channels of 2024-T3 aluminum, as previously mentioned. The Alclad coating is to protect the longerons from corrosion. These channels are to be contour rolled to create the proper curvature required of the nose assembly. The longerons are then to be mounted to the prop bulkhead and nose frame, and nose bulkhead nose frame. The mounting brackets will already be fastened to the bulkheads and frame. The longerons will be fastened to the brackets. These brackets, as well as the prop bulkhead and nose frame, are to be machined from permanent mold castings of 2024-T3 aluminum, which are also to be Alclad. All fasteners used are cadmium plated MS 204070DD-4.

The skin sheets and access panel are to be made of 0.030 inch thick Alclad 2024-T3 aluminum sheets. Each skin sheet is to have two half inch angles riveted on to it for aerodynamic loading support. These channels are to be Alclad 2024-T3 extrusions which are contour rolled to the proper nose curvature. The skin panels, with angles in place, will then be riveted to the nose longerons. All rivets for the nose assembly are to be cadmium plated MS20470DD-4 rivets, as determined previously.

The access panel is to be mounted using AN526C-6-32 screws. The screws will be fastened directly into the longerons, bulkheads, and nose frame, which all will have holes threaded to suit the screws machined in them. This is to allow the panel to be firmly mounted in place, while allowing for it to be removed with ease. The access panel is to allow easy access to the drive shaft bearing and bearing support mounted on the inside of the prop bulkhead. Sufficient room is made available by the panel opening for easy mounting, inspection, and replacement of the parts.

5.2 Cabin Assembly

5.2.1 Longeron, Cabin Side, and Roof Design

The cabin structure was designed for ease of manufacturing and maintenance, as well as part commonality between the cabin and the existing empennage design. The empennage design used 2024-T3 aluminum in all of its key parts such as longerons, rivets, and the like. The same material, Alclad 2024-T3 aluminum, was used in all components of the cabin design, except for the firewall, which is made of 17-4PH stainless steel.

The cabin is to be manufactured by incorporating the floor, instrument panel, windshield and door assemblies into one structure. This is done by starting with the floor assembly (the floor assembly is explored in detail in section 5.2.2). The drive shaft box assembly is first mounted to the floor assembly. The instrument panel assembly will be mounted to the drive shaft assembly at this point (the drive shaft and instrument panel assemblies are discussed in section 5.3).

The firewall, including the angle frames sandwiching the firewall is mounted to the drive shaft assembly at this point. The firewall is manufactured from 0.02 inch thick 17-4PH stainless steel sheet, in accordance with FAR 23.1191 which requires a minimum 0.015 inch thick stainless steel firewall. The angle frames are each made up of four 0.75 inch by 0.75 inch by 0.09 inch thick extruded angles. The channels are to be contour rolled into their required curvatures. The channels are each alodined on the firewall interface side and riveted to the firewall with cadmium plated MS20470DD-4 rivets. The firewall is also to be mounted to the floor longeron mounts with MS20470DD-4 rivets. However, at this point in assembly, temporary fasteners are to be used to fasten to these mounts. This is to keep the fastener holes open for later riveting to the carrythrough longeron mounts on the other side of the firewall.

At this point, the JAARS seats are to be installed on their mounts (which are already a part of the floor assembly). The seat belt (a four point harness) may also be installed at this point. Next, the nose bulkhead is mounted to the floor longeron mounts, similarly to the firewall mounting. This is to keep the fastener holes open for later riveting to the nose assembly longeron mounts on the other side of the nose bulkhead. The nose bulkhead is cut from 0.020 inch thick Alclad 2024-T3 aluminum sheet and is assembled to its angle frame in the same manner as the firewall. The door frame assemblies may now be mounted to the floor assembly at the floor attachment points.

Next, the upper two-inch longerons are to be mounted between the nose bulkhead and door frame and between the door frame and firewall. They may be riveted to the door frame, but temporary fasteners must be used in the nose bulkhead, similarly to the floor longeron mountings. The longeron mounting brackets are to be Alclad permanent mold castings of 2024-T3 aluminum, which are shown in Figure 2.2.1. These brackets are common to mounting all of the two-inch longerons in the cabin and floor assemblies, and are also referred to as "standard" mounts due to this fact.

The roof assembly may now be constructed. The roof frame is constructed of three channels. Two are connected with standard mounts to the firewall frame. These mounts are to be attached with temporary fasteners at his point, later to be riveted to the upper engine/wing box longerons. The third channel is to be mounted between the two side channels. This channel is mounted with standard mounts at either end, each end riveted to the sides of the side channels. This frame now forms a box, with the forward channel at the front, the two side channels on the sides, and the firewall frame forming the back.

The forward bulkhead may be installed at this time. The bulkhead is formed similarly to the nose bulkhead. Its angle frame is formed in the same manner as the firewall frame. Yet, only one angle frame is constructed, not a two channel sandwiched frame like the firewall. This is to allow the bulkhead, which is actually an access panel, to be screwed into place on the angle frame. AN526C-6-32 screws are to be used here, as were used in the nose access panel. Due to this construction, the bulkhead may be easily removed for access to the control systems and the nose gear. This access is extremely easy, as it is a large opening, thus allowing for ease of maintenance and or inspection of these areas.

Once the nose assembly and carrythrough assembly have been riveted into place at the longeron mounts, the skin may then be installed on the cabin. The skin sheets are to be 0.03 inch thick Alclad 2024-T3 aluminum sheets. Half inch angles will be riveted to the skin panels that are mounted on the side of the cabin, similar to what was done on the nose skin. These angles are to placed such that they are mounted half way between the upper two-inch longerons and the two-inch floor longerons. The door, windshield, and cabin light assemblies may now be mounted to the fuselage. Once the carrythrough, wing, and empennage sections are joined to the forward fuselage assembly, the aircraft will be painted with white enamel paint to further protect the skin from corrosion. Various colored enamel paints may then be used for detail graphics on the aircraft.

5.2.2 Floor Assembly
Again, keeping in accordance with the rest of the cabin structural design, all materials in the floor assembly are Alclad 2024-T3 aluminum. The commonality of parts is

maintained once again, as the same standard mounts for the two-inch longerons are used. The cadmium plated MS20470DD-4 rivet is also used again throughout the floor structure, similar to the nose, cabin, and empennage assemblies.

The first components of the floor assembly to be constructed are the floor and seat support ribs. These are to be cut from 0.030 inch Alclad 2024-T3 aluminum sheet. The cutout pattern includes cutouts for the longerons to pass through in the floor ribs, when installed, as shown in drawing F93-1C-128. The cutout patterns are then hydropressed into the required shapes. The seat support ribs are then to be riveted to the floor ribs with MS204070DD-4 rivets.

The second major floor component is the two-inch channel longeron. These longerons are brake formed from 0.09 inch thick Alclad 2024-T3 aluminum and then contour rolled into the required curvature for assembly. One-inch floor longerons are also formed for the floor. These are extruded and contour rolled to the required assembly curvature.

Once the ribs are formed, they will be mounted to the 0.06 inch thick floor skin with cadmium plated MS20470DD-8 rivets. Cutouts in the floor skin are provided toward the aft of the floor, through which the elevator trim and elevator control linkages are to pass. The JAARS seat track is fastened through the floor skin and the seat support rib's flanges at this point with MS20509DD-6 countersunk rivets (the seat may be slid into place on the tracks later in the cabin assembly, as spaces are cut in the track for seat removal). Half inch angles will now be fastened to the underside of the floor skin with MS20470DD-4 rivets. The angles are to be spaced 4.6 inches apart, as shown in drawing F93-1C-128-2, to provide extra stiffness in the floor. The two-inch channels must now be riveted into place, above and below the floor skin and into the floor ribs with MS204070DD-4 rivets, thus forming the frame of the floor assembly. The top of the floor assembly is now complete.

The one and two-inch floor longerons will now be riveted to the bottom skin sheets, thus forming the bottom of the floor assembly. All attachments are made with cadmium plated MS204070DD-4 rivets. The standard mounts must all be installed at this point in the ends of all of the two-inch channels, using the technique mentioned previously. The bottom floor assembly may now be attached to the top of the floor

assembly. This is done by blind riveting MS204070DD-4 rivets through the bottom skin into the floor and seat support rib flanges. This completes the floor assembly.

5.3 Drive Shaft Assembly

The drive shaft itself is made of 4130 steel alloy. It is 3.25 inches in diameter and 0.0625 inches thick. Manufacturing of the drive shaft sections should begin with extrusion of steel tubing of the indicated outer diameter and thickness which would then be trued to reduce vibration while spinning. Both male and female halves of the spline interface should be bored separately and plasma welded to the trued shaft. Spline sections are also 4130 steel alloy. Similar metals are needed to ensure strong weld connections and the hard alloy (in comparison to an aluminum alloy) is needed for proper spline interface. Extreme care should be taken in the balancing of each shaft assembly, since this will be the primary source of vibration in transmission of torque to the prop.

A secondary drive shaft is connected to the primary shaft in an almost identical fashion as at the shaft to engine interface. The forward spline gear connection, however, uses a 0.46 inch diameter aluminum shear pin to aid in engine torque transmission and to ensure resistance to crumpling of the front nose assembly until an 8 g impact has been reached. This assembly includes the thrust bearing attachment which is bolted to the nose cone bulkhead.

The drive shaft uses three steel bearings with aluminum housing and mounting brackets to cut down on weight. The bearings and housings can be subcontracted to find a light weight durable bearing. New composite materials may also provide a lighter more effective design. Bearing design, however, should incorporate use of split journal bearings. This will aid in assembly of the drive shaft.

The universal joint is a standard universal MS20271 which can be joined to the rear spline section. The universal meets military specifications and should last the life of the aircraft. It has a permanent lubrication and should not require any maintenance.

The universal joint is connected to the engine by a spline gear attachment. The bearings and mounts are attached to the drive shaft and are inserted together through the front of the aircraft with the nose cone removed. The bearings are then mounted to the firewall and center structural members.

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Each structural member is made up of two channels. The channels are made from 0.125 inch thick 2024-T3 aluminum. Each channel is to be brake formed with .5 inch minimum bend radii. All four channels are identical in dimensions and shape and, therefore, allow for part commonality.

All electrical, vacuum, and control systems are to be installed prior to installation of the drive shaft assembly. In order to simplify assembly, installation of the instrument panel and seat assemblies should follow that of the primary shaft.

For routine maintenance of the bearings, the upper shield which covers the primary shaft, as seen in drawing F93-1C-152-2, can be removed for access from within the cabin.

The electrical system and control systems can be accessed through the side panels of the drive shaft channel. These panels are attached with quick release fasteners to allow maintenance entry into the channel. The top of the channel is also removable, so the drive shaft itself can be inspected and maintained.

Drive shaft removal would have to be done in a procedure similar to that of a reverse installation. The nose cone would be removed by quick release fasteners on the skin.

5.4 Wing Assembly

5.4.1 Spar Design
The front and rear spars are both manufactured in similar fashions. 2024-T3 aluminum alloy sheets of the appropriate thickness, 0.08 and 0.06 inches, are to be blank pressed to the specified pattern. The blanking operation will identify the flange edges and the lightening cutaway's. The flanges are then formed by use of a power brake. Because of the sheet size limitation, each spare is made of a 96 inch inboard and a 67 inch outboard section. The outboard section is fish-mouthed at the 5 inch overlap and joined by 10 MS20470DD-4 rivets.

5.4.2 Wing Skin and Ribs
All materials within the wing box and skin panels are Alclad 2024-T3 aluminum. The ribs are formed in similar fashion to the floor ribs discussed in 5.2.2 using the appropriate thicknesses described in 3.4.2. In addition, lightening holed are stamped into the ribs

with the exception of the ribs surrounding the fuel tank. Four 2 inch diameter holes are centered between the z-stringers. The holes provide a reduction in weight and room for maintenance and inspection. The ribs are riveted to the front and rear spar using MS20470DD-4 rivets from LBL31 up to LBL142 and MS20470DD-3 rivets are used from LBL142 to the tip.

Once the ribs are attached, the stringers can be riveted to the top and bottom of the ribs at LBL56 and the tip positioned at 36.7%, 48.4% and 60.2% chord. The stringers are equal leg, extruded aluminum z-stringers (NAS346-3) and are riveted using the rivets as discussed above.

Finally, the skin can be flat wrapped and riveted into position. Again, the skin used is Alclad 2024-T3 aluminum and comes in standard thicknesses and in 48 inch wide sheets. All joints are overlapped and all of the skin panels must be in position before riveting can begin.

First, the upper and lower skin panels are temporarily held into position while the leading edge skin panel is positioned over them. This is done across the entire semi-span until all panels are into position.

From the tip inboard to the rib positioned at LBL142, a 0.02 inch thick sheet is riveted using the MS20470DD-4 rivets. Two panels of 0.032 inch thickness are used from LBL142 inboard to LBL96 and from LBL56 inboard to LBL31 and 0.071 inch thick sheet is used over the fuel tank. All three panels are riveted using the MS20470DD-4.

Consideration must be given to cutouts for the fuel tank access, landing gear, and maintenance access panels. These concepts were not specifically addressed in this assignment.

5.4.3 Carrythrough
The carrythrough assembly was designed to be simply constructed and utilize parts common to the existing cabin structure. The major load carrying items are the forward and rear channels and plates. The plates are to be cut from 0.1 inch thick 2024-T3 aluminum sheet. The channels are to be extruded 2024-T3 aluminum in the cross sections depicted in drawing F93-1C-170-2. The one and two inch longerons used in the assembly are the same as those used in the cabin structure, brake formed from 2024-T3 aluminum and contour rolled to the required curvatures. The longeron mounting brackets are also identical to those used in the cabin structure, 2024-T3

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permanent mold castings.

The first items to be assembled are to mount the wing attachment lugs to the forward and rear channels, as discussed in section 5.5. Next, the upper outboard two-inch longerons are to be riveted to the front and rear channels with the mounting brackets, as shown in drawing F93-1C-170-2. Mounts may also be installed in the lower inboard two inch longerons at this point.

The front plate and channel, with the two-inch longerons attached, may now be riveted to the firewall. The rivet locations of the floor longerons will be picked up. Therefore, the temporary fasteners in those holes (discussed in the cabin structure manufacturing section) may now be removed and replaced with MS204070DD-4 rivets which will fasten the interface between the floor longerons and the front carrythrough structure through the firewall. The same will be done to install the inboard two inch longerons. Their mounts will be riveted to the lower flange of the front channel at the locations of the floor longerons, thus picking up the fastener locations there. As before, the temporary fasteners in the floor structure will be removed and replaced with MS204070DD-4 rivets at this point, thus completing the interface from the floor longerons through the firewall to the front carrythrough and inboard two inch longerons.

Next, the inboard two inch longerons will be fastened to the lower flange of the rear channel with temporary fasteners through the longeron mounts, rear channel, and rear plate. Temporary fasteners will also be installed at all rivet locations in the rear channel and plate to fasten the plate and channel together. At installation of the empennage and rear bulkhead, these temporary fasteners will be removed and permanent fasteners will fasten the interface of the rear carrythrough structure through the rear bulkhead to the empennage longerons.

The skin is now ready to be fastened to two inch longerons to form the outer shell of the fuselage surrounding the carrythrough structure. Once the skin in is place, the one inch longerons may be riveted to the skin at the positions shown in drawing F93-1C-170-2. The exact skin thickness and corresponding rivet size has not been determined by the requirements of this report, but these will be determined in future efforts. The carrythrough assembly is now complete.

5.4.4 Wing Attachments

The lugs are machined out of 2024 bar stock. The machining of the lug will allow for a higher shear stress allowable than a cast part. The hole for the bolt will have to be drilled very precisely to make sure the bolt fits tightly. The tight fit of the bolt is necessary to maintain the strength desired of the lug and bolt. The channels on the front spar attachments are tapered at the ends to not allow a stress concentration at the end of the attachment. The same is true with the I-beam on the rear spar in which the web is fish-mouthed.

There is no maintenance required for the attachments except for the continuous inspection of the parts for corrosion or fatigue displays. As shown by the fatigue calculations, the parts should last more than one hundred million cycles, but this is for normal flight. Therefore, for safety reasons the lugs should be inspected.

6.0 Weight Summary

The weights of all components used in the construction of each assembly were calculated individually and then summed up for each total assembly weight. The resulting assembly weights are shown in Table 6.0. In addition, the center of gravity for the cabin-fuselage assembly and the occupant groupings are included in the Spatial Requirements Specification Document (refer to Appendix A).

In the design of the cabin structure, there is no place where weight could further be reduced, as all components were designed to be as small as possible while still carrying the required loads. However, in the cases of the drive shaft and wing assemblies, further weight reductions are possible, as will be discussed below.

The high margins of safety of Table 4.3.2 indicate a small sacrifice on the designer's part in respect to the drive shaft's overall weight. The purpose of the high values, however, is twofold. The first is that priority was placed on occupant safety, and the second is that an increase in drive shaft thickness is expected as further research is done on the fatigue characteristics of the shaft.

Due to the load distribution on the wings, the curves determining the spar section moduli required the outer portion of the wing to be over designed. This was necessary to retain the desired taper in the spars. In the

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front carrythrough, fatigue determined that the size of the front carrythrough structure could not be reduced. The rear carrythrough size could be reduced as far as loading and fatigue constraints are concerned. It was over designed in these areas due to the desire to maintain similar construction as the front carrythrough for ease of manufacturing and cost reduction. The size requirements for interface with the rear spar were also a contributor to the design selected.

In comparison of the preliminary weight estimation to the total weight listed in table 6.0, the preliminary estimation was 117 pounds less. This is due to the fact that the preliminary estimation did not account for the weight incurred by the drive shaft assembly. This comparison is made to the listings for fuselage, cabin systems, and wing weight entries in the preliminary report versus the values of table 6.0.

Table 6.0: Subassembly Weights

Nose	31.42 lbs
Floor	67.21 lbs
Cabin	92.31 lbs
Drive Shaft	109.57 lbs
Carrythrough	37.52 lbs
Front Spar	20.0 lbs
Rear Spar	17.0 lbs
Skin & Ribs	145.7 lbs
Wing Attachment Fittings	50.0 lbs
TOTAL	570.7 LBS

7.0 Environmental Considerations

7.0.1 Considerations
In designing the various components of the overall deign in this report, several environmental considerations were regarded. The areas considered were temperature, atmospheric pressure, sand and dust, rain, humidity, ice, snow, salt/fog, wind gusts, and fatigue performance. Each is discussed in the following sections.

7.1 Temperature

With respect to temperature, all parts are designed to operate without degradation from -40 F to 122 F. The 2024-T3 aluminum and stainless steel parts used in the construction all easily withstand this temperature regime without material degradation. With regards to the cabin environment in this temperature range, the heating and ventilation systems account for pilot comfort. No actual insulation is to be included in the cabin structure, beyond the skins, longerons, and interior walls (standard plastic wall coverings). However, the heating system is to keep the pilot sufficiently warm, when he is properly dressed, within the cold extremes. The venting system is to cool the pilots at the upper temperature extremes. These concepts are the same as those currently applied in today's existing primary trainers.

7.2 Atmospheric Pressure

The Viper is designed to operate up to an altitude of 10,000 feet, which does not require pressurization or supplemental oxygen (per FAR 91.211(a)). Therefore, the structure as it has been developed will be capable of operation up to this flight level.

7.3 Sand and Dust

External surfaces, mechanisms, hinges, and associated items have been designed to endure up to 150 microns in size and in combinations of sand and dust in concentrations up to 0.041 grams per cubic foot without degradation. This has been accounted for by having the drive shaft bearings properly fitted to disallow dust interference. The most critical point for this is at the nose interface with the prop. The prop bulkhead seals off the shaft area, thus keeping dust and sand from entering the drive shaft area. The housing around the drive shaft in the cabin area also serves this purpose. In the wing assembly structural designs, there are no critical moving parts, and the lugs for wing attachment are protected, as they are housed within the wing skin.

7.4 Rain, Humidity, and Salt/Fog

All external surfaces have been coated in various manners, as discussed in section 5. These surface coatings prevent corrosion due to exposure to rain and up to 100% relative humidity at 95 F without degrading. They also deter corrosion due to exposure to a salt/ fog atmosphere, as may be encountered in coastal areas. There are no cavities designed into the concept which allow for containment of water which leads to corrosion problems. Again, the prop bulkhead prevents intrusion of water into

the drive shaft area, including rainfall up to a 4.0 inch/hour rate with net wind velocities up to 150 miles per hour. All other structural interfaces and skins are constructed to easily withstand these conditions without intrusion of water into the structural interior.

7.5 Ice and Snow

The coatings mentioned in section 7.1.4 are also responsible for withstanding ice and snow at temperatures as low as -40 F without degradation of materials. At loadings of 10 inches of wet snow accumulation, the cabin structure easily holds without buckling. Snow accumulation on the wings does not effect the loading on the nose gear, as the wings are placed aft of the aircraft cg.

The snow load was also checked for the wing assemblies and was found to be 1,117 pounds by multiplying the wing area by the 10 inch snow depth and the density of wet snow (10 pounds per cubic foot). This load is far less than the load imposed at 4.4 g's. Therefore, the various wing assemblies are designed to easily withstand the snow load.

7.6 Wind and Gust

The structural design for the cabin and wing assemblies has been completed for flight conditions, which include loads higher than would be experienced at tie down. Winds of 120 miles per hour and 50 mile an hour gusts do not impose loads at tie down greater than the required inflight design loads. Therefore, the design is capable of withstanding these tie down conditions without degradation.

7.7 Fatigue Performance

As discussed in previous sections, fatigue was a major consideration in the design of the wing assemblies. The cabin and engine interface structural designs were not evaluated for fatigue. However, all of the wing assemblies were found to display a safe life of 10^7 load cycles, as required by the SOW. The process to determine if each component can endure a maximum of 10^7 load cycles used the stress on the component at 1 g as the mean stress. The resulting maximum stress was then found as 2.2 times the mean stress. Next, the mean stress was checked at 10^7 cycles on MIL-HDBK-5E figure 3.2.1.8(h) for 2024-T3

aluminum. The resulting maximum stress was found from the figure and must have been greater than the calculated maximum stress in order to provide the minimum required safe life.

8.0 Conclusions

In conclusion, this report has been prepared to answer the requirements of the 421F93ADP01-2 SOW and its Addendum. The requirements of both have been met, regarding occupant safety and volume constraints. A structural concept which withstands the demands of FAR 23 has been completed within this report. This was done by heavily considering part commonality and ease of manufacturing and maintenance. All of the parts used in the cabin structural design match the materials used in the existing empennage design. The same holds true for the wing structural design with respect to the cabin and empennage structural designs. Also, a number of detail parts, such as rivets, brackets, and longerons are used repeatedly to reduce the amount of different part types required.

Perhaps the greatest challenge of designing the Viper cabin-fuselage structure was insuring that the drive shaft and engine would cause no harm to occupants in a crash. This was achieved, as has been discussed in this report. The drive shaft and its supports have been designed to account for this. Indeed, it has been shown in this report that the drive shaft and engine will not cause harm to occupants upon crashing.

In order to meet the restraints of the project, the size of the fuselage was required to be increased, thus moving the wings by two inches outboard to either side of the preliminary design fuselage. This size increase was due to lack of proper space attributed to realistic structural volume requirements, as well as not providing for the JAARS seat installation, in the preliminary design. Due to the fuselage size increase, the empennage must now also be redesigned to match the new diameter of the Viper fuselage. It is recommended that when this is done, the empennage longerons should be placed in line with the existing longerons in the cabin and carrythrough designs. This would allow continuous load paths through the longerons throughout the entire aircraft.

An additional required modification is that the existing rudder pedals must be moved 6.3 inches aft and the yoke must be moved 5.1 inches up and 2 inches aft. These changes are to allow for the placement of the JAARS seat

and proper volume safety constraints, which were not accounted for in the preliminary design. Otherwise, the cabin structure is designed entirely around the existing control systems.

One final modification made to the preliminary design was in removing the vents from the doors, as these vents were of insufficient size. They were also placed in an area that is not conducive to flow into the vents. The engine venting was moved to an intake below the fuselage, as may be seen in drawing F93-1C-129-2. This placement allows for much improved air flow, as well as a larger intake area.

Certain fuselage components, such as the door frames, windshield, door, and cabin light assemblies were not explored in complete detail. They were included as considerations in the design. These components may be examined in detail in the future. Also, as was discussed in section 2.3.1, the engine support and mounts for mounting to the front of the O-235 must be further examined in cooperation with Lycoming. This will allow for final sizing of the engine support between the front of the engine and the center longeron structure under the drive shaft.

With regards to aerodynamic loads, such as the prop wash on the nose assembly, or gust loads on the cabin and wings in general, it is recommended that further studies be done. They should be in the form of wind tunnel testing to ensure proper reactions to aerodynamic loadings without buckling.

Crash and fatigue tests are also recommended to substantiate the loading calculations made for the design. This should be done in future studies to ensure a twenty year service life without component failure. Actual fatigue tests are recommended to ensure accurate results.

Regarding cost, a formal cost summary was never completed, due to lack of time. However, manufacturing, maintenance, and related costs were heavily considered in the design process. It is recommended that once the entire Viper aircraft concept has been completed, that a cost analysis then be run. This would ensure the most accurate cost estimates.

In general, the design presented in this report is a viable production design, once the recommended detail studies have been completed. The design was created entirely using existing technology, as many components were modeled similarly to actual aircraft components. The Cessna 152, in particular, was examined for this purpose, as its pilot operating handbook was used as a reference, as well as crash worthiness tests performed on it by NASA. Approximate dimensions for certain components were also taken from measurements on a Cessna 152 structure at ERAU's AMT facility. These factors, coupled with the results discussed above, indicate that the design presented in this report is indeed a believable concept.

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Appendix A

Spatial Requirements Specification Document

To: Dr. J.G. Ladesic
From: F93-1C
Date: October 20, 1993
Subject: Spatial Requirements Specification Document (F93-1C-2R1)

Introduction

This report contains the occupant safety requirements set forth in Statement of Work 421F93ADP01-2. Under Federal Aviation Regulations (FAR) Part 23, the light aircraft certification requirements specify minimum volumetric constraints for occupant safety. This report is to fulfill the need to address cabin sizing at the conceptual design level for a utility category, conventional, single mid-engine, low wing, tricycle landing gear airplane configuration. Integration of future design concepts must take into account these constraints to meet certification.

Requirements

FAR 23.562 specifies emergency landing dynamic conditions. This requirement calls for a complete dynamic test for seat/restraint system to ensure compliance for crashworthiness. At present, Jungle Aviation And Radio Service (JAARS) has dynamically tested and certified one of the only seats that meets requirements specified by FAR 23.562. To meet aircraft certification requirements, the JAARS crew seat was selected for this project and FAR 23.562 is met by rational analysis.

In the event of a crash, the occupant must be restrained from coming in contact with any portion of the cabin and not to exceed the head injury criteria (HIC) value of 1000 (refer to FAR 23.562). Due to the limitation of performing this dynamic test at this stage of development, volume approximations are made to ensure that the occupant's head does not impact any cabin object.

A standard occupant in a normal seated position, shown in Figure 1 A, must have a minimum of 2.0 inches from the top of the head to the cabin's inner structure. Upon forward impact, the body's range of motion, shown in Figure 1 B, is restrained by a four point shoulder harness allowing only the head to rotate about the base of the neck. This head rotation creates an arc from the base of the neck extending 2.0 inches above the top of the head. At the forward seat adjustment position, the windshield must be positioned at an angle clear of this region. In addition, the bottom of the instrument panel must be at a minimum height of 2.0 inches above the knees. This is to ensure that the occupant has adequate room for cabin ingress/egress.

Due to the general fuselage shape, the minimum clearance is defined as the distance from the occupant's head to the cabin's inner structure occurring at a diagonal. This is displayed in Figure 1 C. Upon side impact, the head's side-to-side motion again creates an arc from the base of the neck extending 2.0 inches above the top

of the head. Sufficient space must be allowed for the curvature of the fuselage and the door structure.

The JAARS seat used in this configuration can be adjusted 6 inches forward and aft of the nominal position. This adjustment creates the spacial envelope that is the sum of the regions to be void of any obstruction as shown in Figure 2.

Finally, this minimal volumetric envelope needs to account for the cabin structure (refer to Figures 3 and 4). If the design configuration dose not include a canopy, a minimum of two inches must be allocated for the cabin's roof structure. This two inch consideration takes into account the required structural sizing for the external skin, roof structure, and headliner to maintain structural integrity in event of an 18g impact. Additionally, another five inches needs to be allocated for the floor. The floor structure is a major load path and must house longerons and ribs sized to meet the same 18g impact criteria. Also, a minimum of three inches must be allotted for door frame structure.

Conclusions

This document develops the minimum spacial requirements for aircraft certification under FAR 23. A seat/restraint system certified by dynamic testing under FAR 23.562 must be used. To meet HIC using the JAARS crew seat, all dimensions of the spacial envelope, shown in Figure 4, must be the minimum dimensions used. Realistic structural volume constraints must be considered in the cabin design, as well. It is left to the conceptual designer to provide additional room for adequate occupant comfort.

Cabin inner structure

2.0 in nominal clearance

(A)

Cabin inner structure

2.0 in nominal clearance

(B)

(B)

(C)
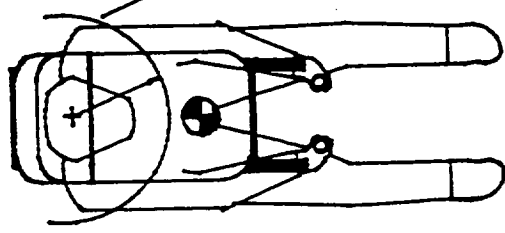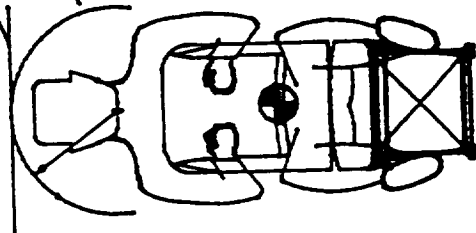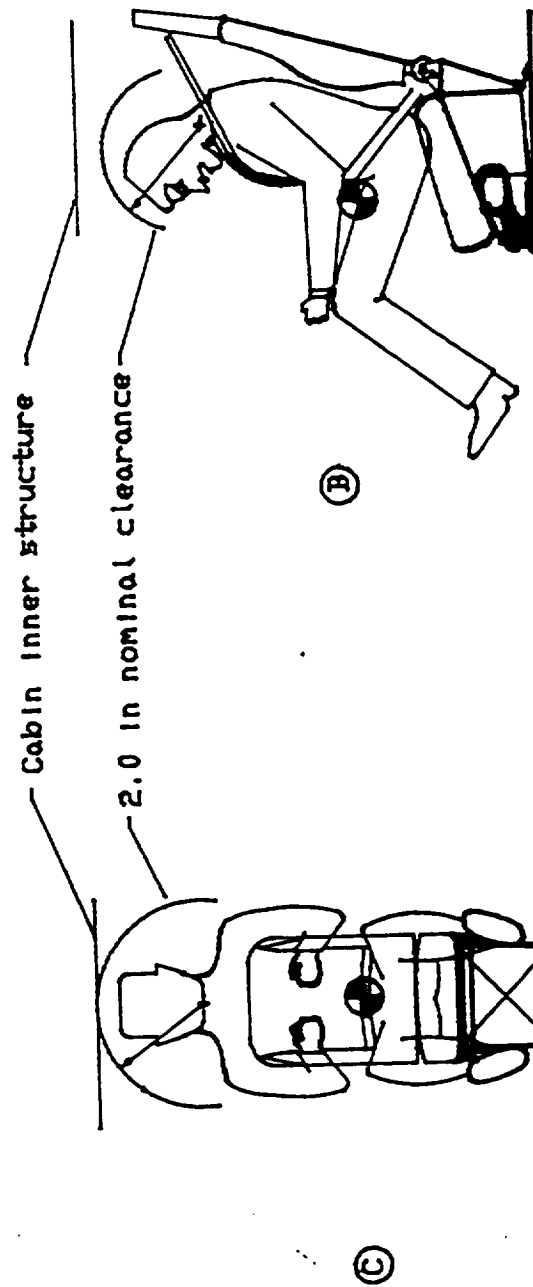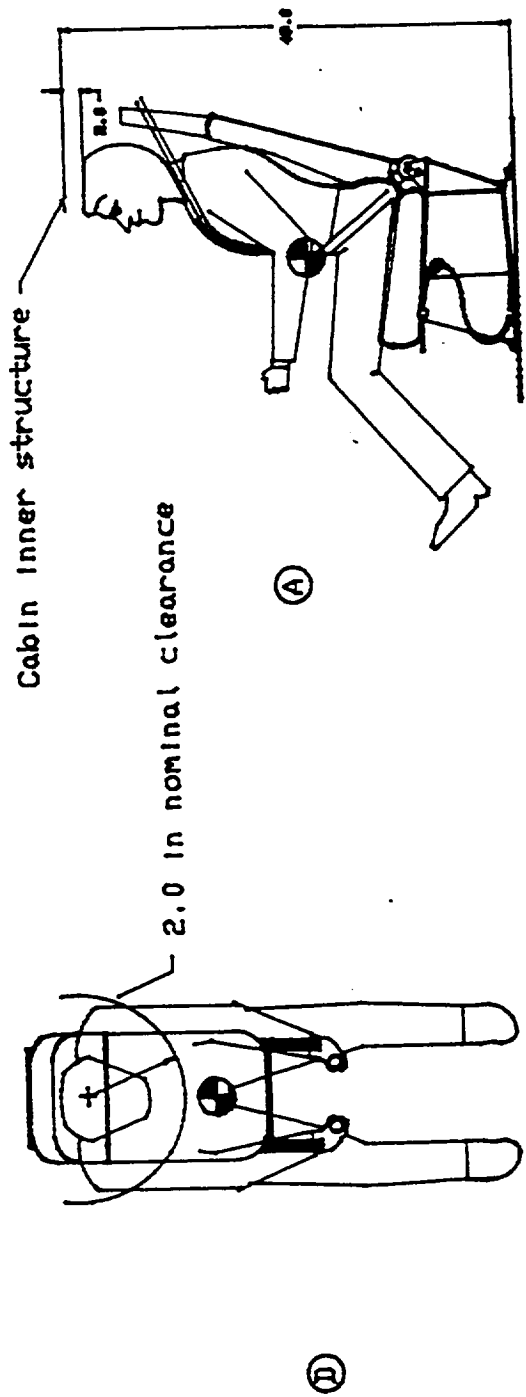
FIGURE 1

HEAD INJURY CONSIDERATIONS

20.0

Forward most position

Control panel

Cabin Inner structure

Aft most position

29.0

26.0

18.0

49.0

5.0

55.0

Control panel

18.0

2.0

11.5
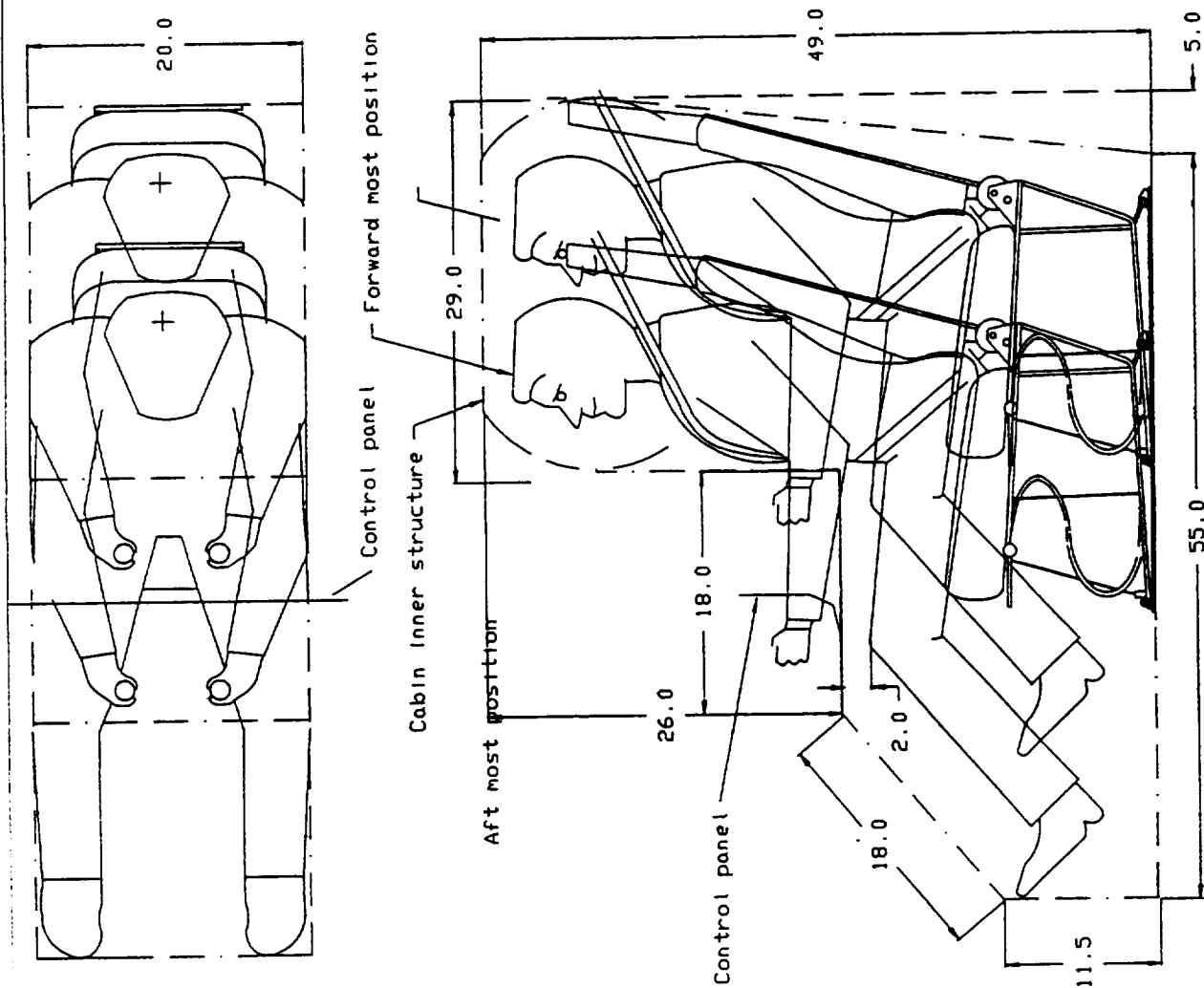
Figure 2

Occupant Volumetric Enevelope

Aft Most Position

Forward Most Position

Instrument Panel

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5.0
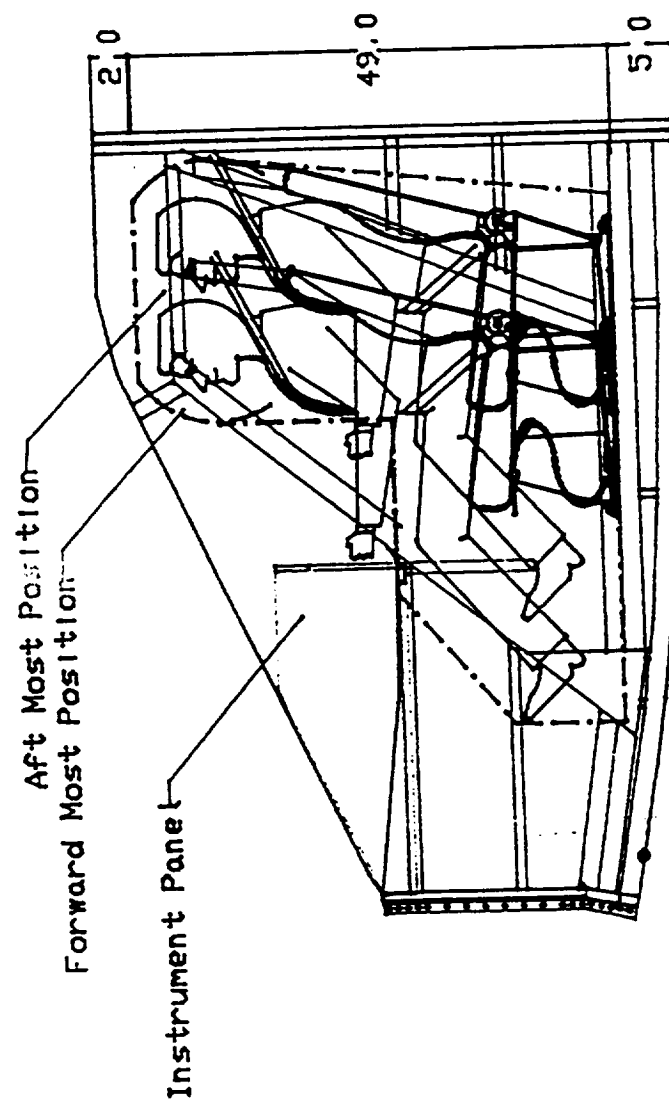
Figure 3

Occupant Volummetric Requirements
With Cabin Structure Cinsiderations
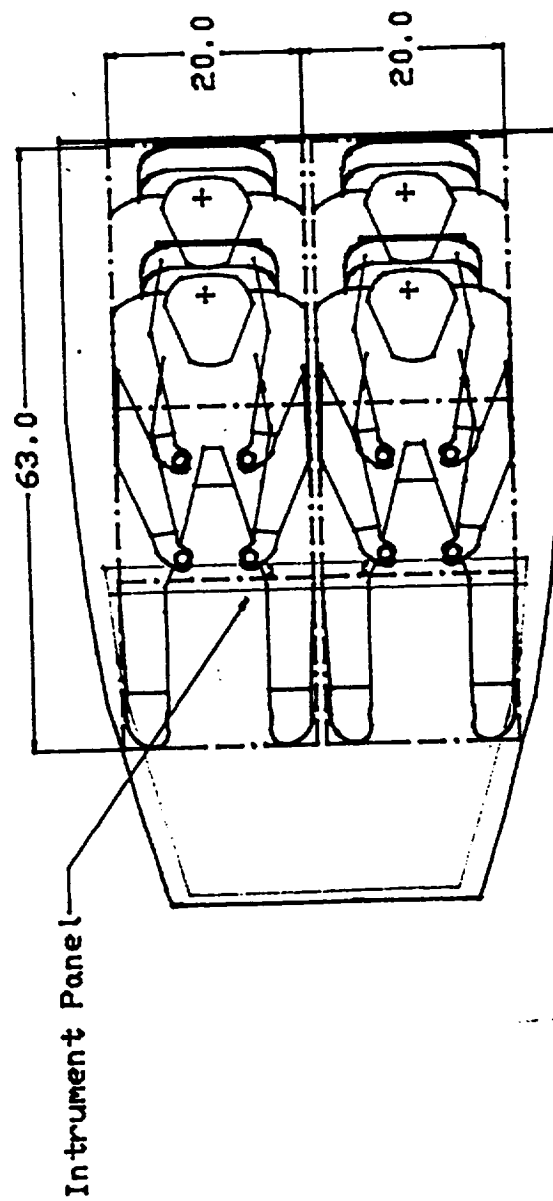
Figure 4
Occupant Volummetric Requirements
With Cabin Structure Considerations

Intrument Panel

Appendix B

V-N Diagram

V-n Diagram

Load Factor

Velocity (kts)

Vsf Vs Vf Va Vd
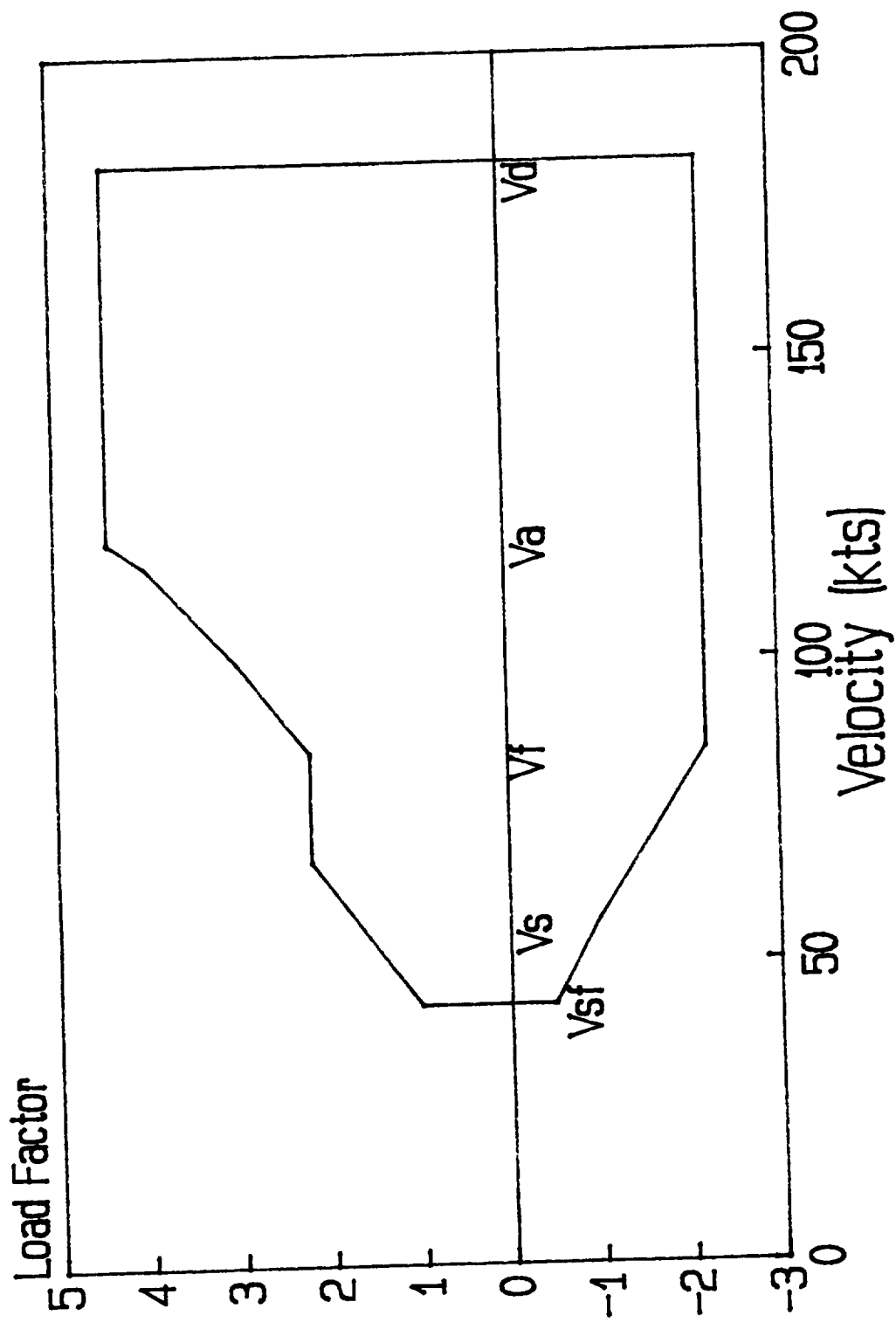
Appendix C

Torsional Diagram

Wing Torsion

Torsion (in. lb.) (Thousands) vs Span (in.)

Legend:
- Dive
- Maneuver w/ailerons
- Maneuver
- Flaps/ailerons
- Flaps
- Landing gear
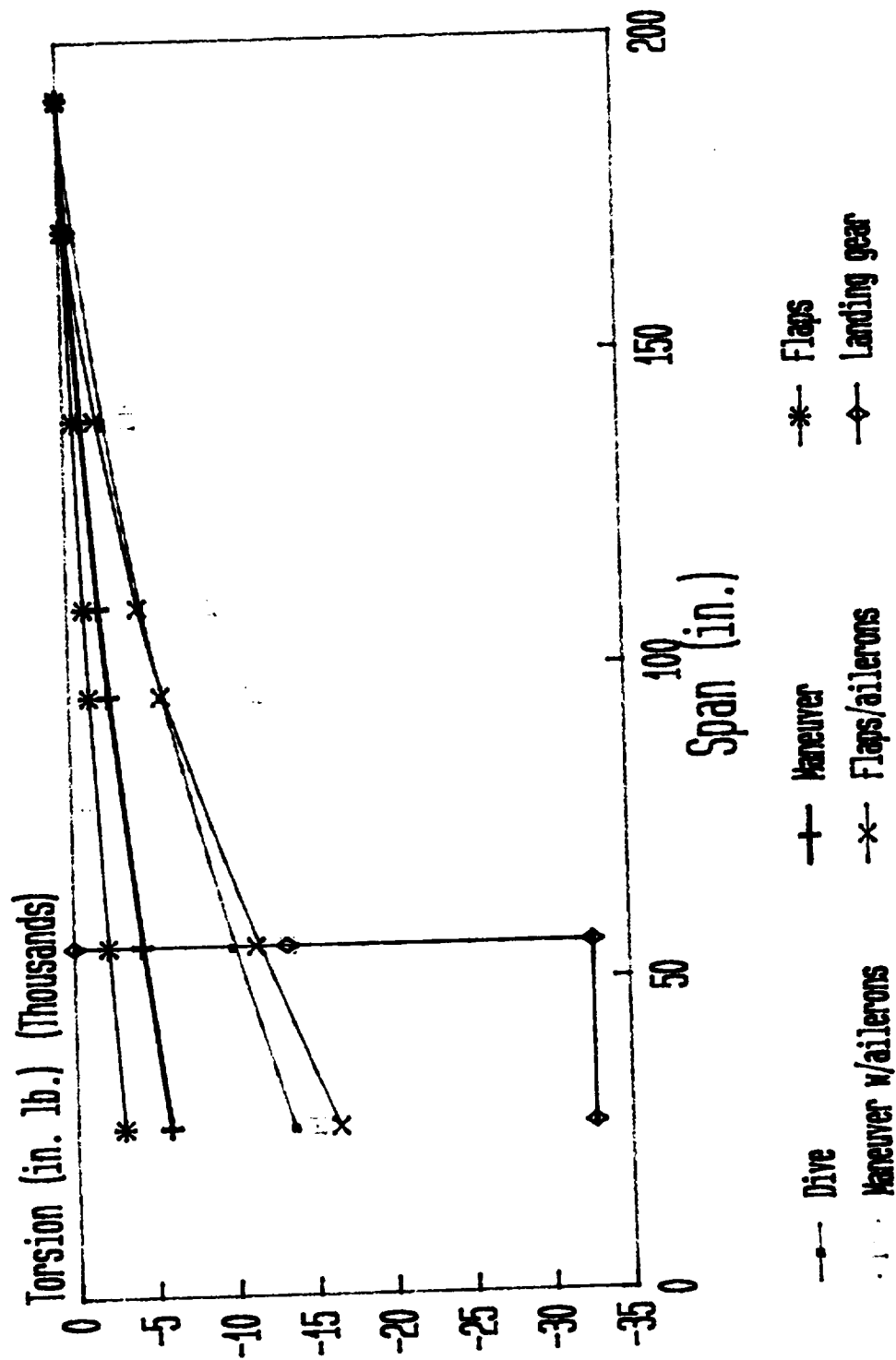
Appendix D

Structural Decomposition

Forward Assembly

Nose Assembly
- Nose Bulkhead
- Nose Longerons
- Nose Longeron Mounts
- Nose Frame
- Prop Bulkhead
- Nose Skin
- Nose Access Panel

Cabin Assembly
- Instrument Panel Assembly
- Forward Bulkhead
- Firewall
- L-Channels
- Longerons
- Longeron Mounts
- Cabin Light Assembly
- Door Assembly
- Windshield Assembly
- Nose Bulkhead
- Door Frame Assembly
- JAARS Seat Assembly
- Skin
- Floor Assembly
- Longerons
- Longerons Mounts
- Nose Gear Access Panel
- Floor Ribs
- Seat Support Ribs
- Floor Skin

Drive Shaft Box Assembly
- Drive Shaft
- Universal Joint
- Spline Connection Assembly
- Forward Bearing Assembly
- Center Bearing Assembly
- Aft Bearing Assembly
- C-Channel
- C-Channel Mounts

Wing
- Spars
- Ribs
- Skin Sections
- Leading Edge
- Tip Caps
- Fuselage Fairings
- Carry-Through Structure
- Landing Gear
- Fuel Systems
- Ailerons
- Flaps
- Wing Attachments
- Lights

Appendix E

Drawings

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FRONT SPAR ATTACHME

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REAR SPAR ATTACHMENT

NOTE: ALL PARTS ALCL
BREAK ALL SHAR

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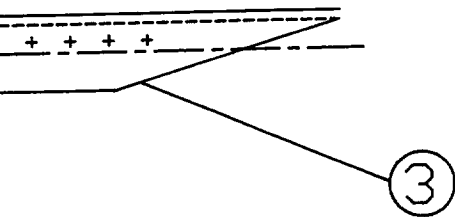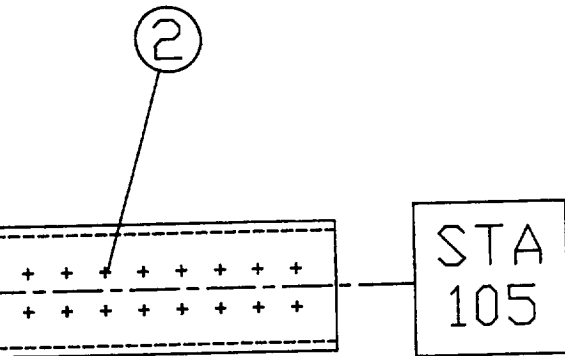
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10	64	RIVET	MS20430A-8-10
9	4	NUT-AIRCRAFT-SHEAR	AN320-D16
8	2	NUT-AIRCRAFT-SHEAR	AN320-D12
7	2	FUSELAGE ATTACHMENT LUG	2024-T4 AL
6	2	REAR SPAR ATTACHMENT LUG	2024-T4 AL
5	2	BOLT SHEAR, HEXAGON HEAD	NAS 1310-8
4	4	FRONT SPAR ATTACHMENT LUG	2024-T4 AL
3	4	FUSELAGE ATTACHMENT LUG	2024-T4 AL
2	264	RIVET	MS20430A-8-12
1	4	BOLT SHEAR, HEXAGON HEAD	NAS1314-31

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01

.XXX ± .001

ANGULAR
± 1/2°

EDGES & BURS

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

SIZE B	DATE 12/2/93	SCALE 1/5	DRAWN BY CRAIG STEVENS

TITLE   WING/SPAR ATTACHMENTS

DRAWING NO.   F93-1C-180-2

SHEET 1 OF 1

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FLOOR LONGERONS (REF)

ENGINE VENT
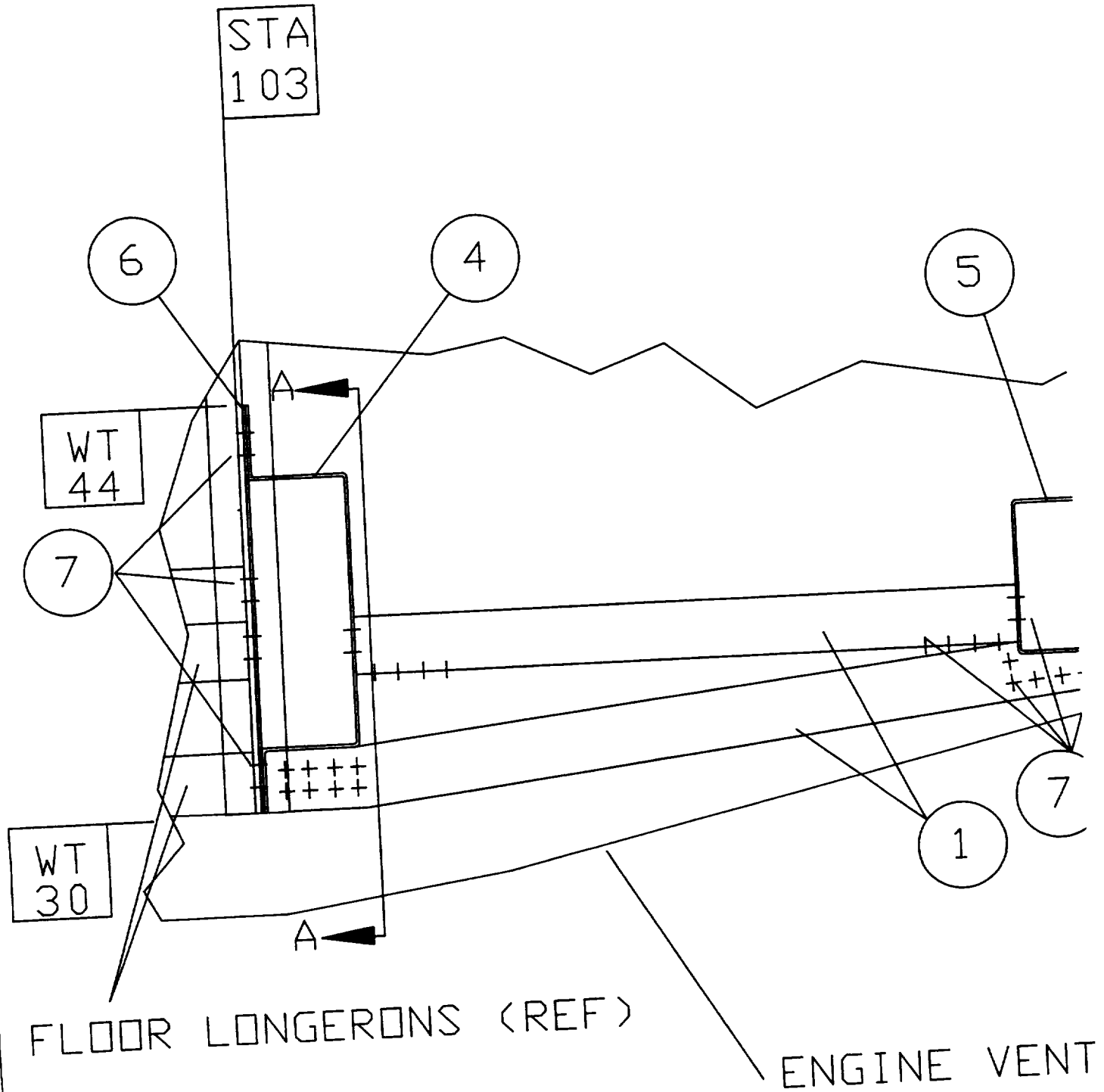
NOTES: 1   ALL   MATERIALS ALCLAD 2024

UNLESS OTHERWISE SPECIFIED

2   REMOVE ALL SHARP EDGES AND

STA
133

REAR BULKHEAD (REF)

⑧

(REF)

T3

BURRS

ITEM	QTY	DESCRIPTION	PART#
8	1	REAR PLATE	F93-1C-178
7	490	RIVET	MS20470DD-4
6	1	FRONT PLATE	F93-1C-176
5	1	REAR CHANNEL	F93-1C-175
4	1	FRONT CHANNEL	F93-1C-174
3	8	2 INCH LONGERON MOUNTS	F93-1C-110
2	4	1 INCH LONGERON	F93-1C-109
1	4	2 INCH LONGERON	F93-1C-108

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01
.XXX ± .001

ANGULAR
± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

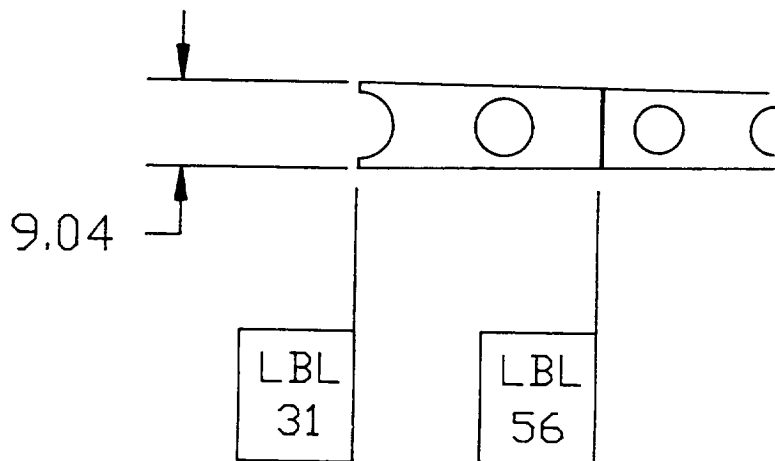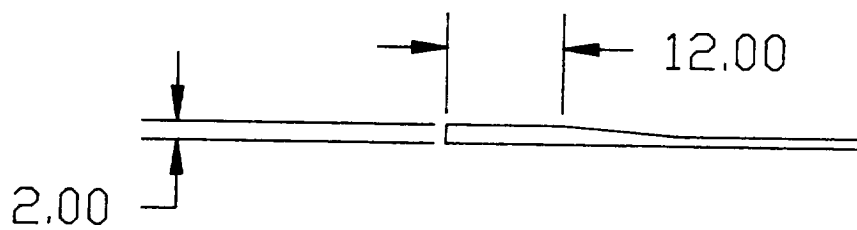| SIZE	DATE	SCALE	DRAWN BY
B	11/93	1/5	MARCHESSEAULT

TITLE
CARRYTHROUGH ASSEMBLY

DRAWING NO.
F93-1C-170-2

SHEET
1 OF 3

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FLOOR LONGERONS (REF)

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REAR BULKHEAD (REF)

CL

(2)

(5)

(8)

(1)

(3)

(7)

DOWN AT WT 44

DATE	DWG BY	DRAWING NO.	SHEET
11/93	MARCHESSEAULT	F93-1C-170-2	2 OF 3

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ENGINE VENT (REF

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DATE	DWG BY	DRAWING NO.	SHEET
11/93	MARCHESSEAULT	F93-1C-170-2	3 OF 3

NOTES:

ALL COMPONENTS ARE ALCLAD 2024-T3

REMOVE ALL SHARP EDGES AND BURRS

ITEM	QTY	DESCRIPTION	PART#
15	908	RIVET	MS20470DD-3
14	2060	RIVET	MS20470DD-4
13	6	Z STRINGER	NAS346-3
12	3	.063 IN RIB	F93-1C-211
11	2	.032 IN RIB	F93-1C-210
10	1	.02 IN RIB	F93-1C-209
9	1	.02 IN TIP RIB	F93-1C-208
8	1	.071 LEADING EDGE SKIN	F93-1C-207
7	2	.032 LEADING EDGE SKIN	F93-1C-206
6	1	.02 LEADING EDGE SKIN	F93-1C-205
5	2	.071 IN SKIN	F93-1C-204
4	4	.032 IN SKIN	F93-1C-203
3	2	.02 IN SKIN	F93-1C-202
2	1	REAR SPAR	F93-1C-201
1	1	FRONT SPAR	F93-1C-200

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01

.XXX ± .001

ANGULAR

± 1/2°

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

SIZE B	DATE 12/93	SCALE 1/20	DRAWN BY CARR

TITLE LEFT WING ASSEMBLY

DRAWING NO. F93-1C-160-2

SHEET 1 OF 4

FUSELAGE LINE

NOTES:

△1 SKIN THICKNESS BL27 TO BL56 IS .071 IN

△2 SKIN THICKNESS BL56 TO BL96 IS .032 IN

△3 SKIN THICKNESS BL96 TO BL189 IS .02 IN

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LBL
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1.00

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LBL
96

LBL
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DATE	BY	DRAWING NO.		SHEET
12/93	CARR		F93-1C-160-2	3 OF 4

DETAIL A

DETAIL A

DATE	BY	DRAWING NO.		SHEET
12/93	CARR		F93-1C-160-2	4 OF 4

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NOTE: 1) ALL PARTS AL(
2) BREAK ALL ED(

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4	4	BOLT – MACHINE, AIRCRAFT	AN4-5
3	4	BOLT – MACHINE, AIRCRAFT	AN8-5
2	1	PROP BULKHEAD	QQ-A-225/6
1	1	THRUST BEARING	F93-1C-197-2
ITEM	QTY	DESRIPTION	MATERIAL OR PART #

DIMENSION TOLERANCES
UNLESS OTHERVISE SPECIFIED
DECIMAL

.XX ± .01

.XXX ± .001

ANGULAR
± 1/2°

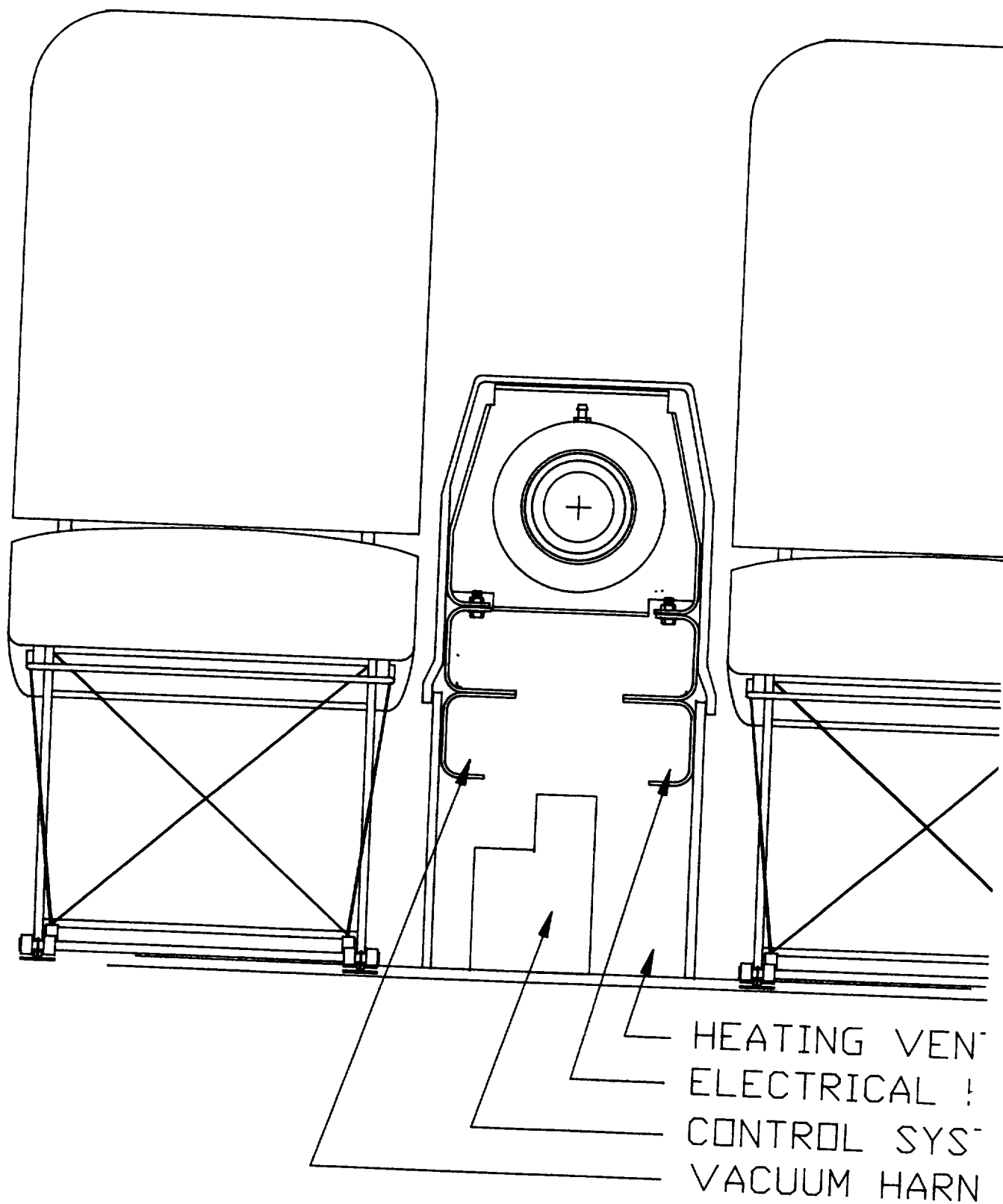**EMBRY-RIDDLE AERONAUTICAL UNIVERSITY**
DAYTONA BEACH FLORIDA

SIZE	DATE	SCALE	DRAWN BY
B	12/3/93	1/2	CRAIG STEVENS

TITLE
PROP BULKHEAD

DRAWING NO.	SHEET
F93-1C-158-2	1 OF 1
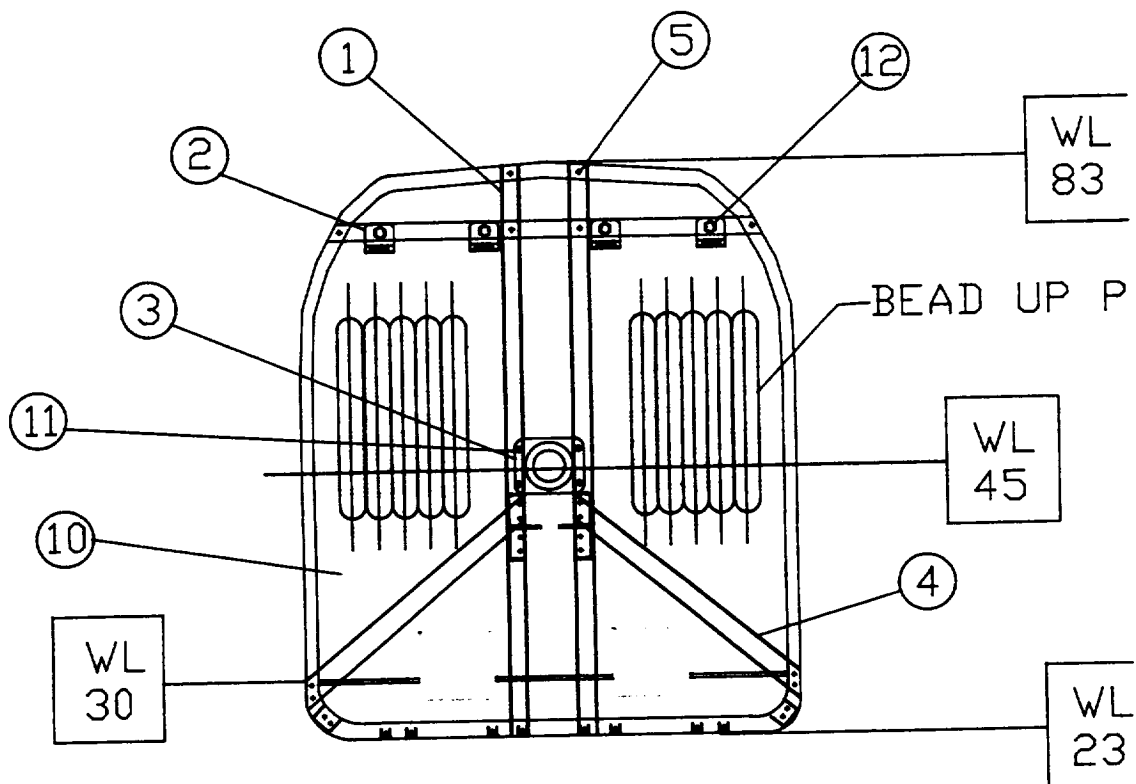
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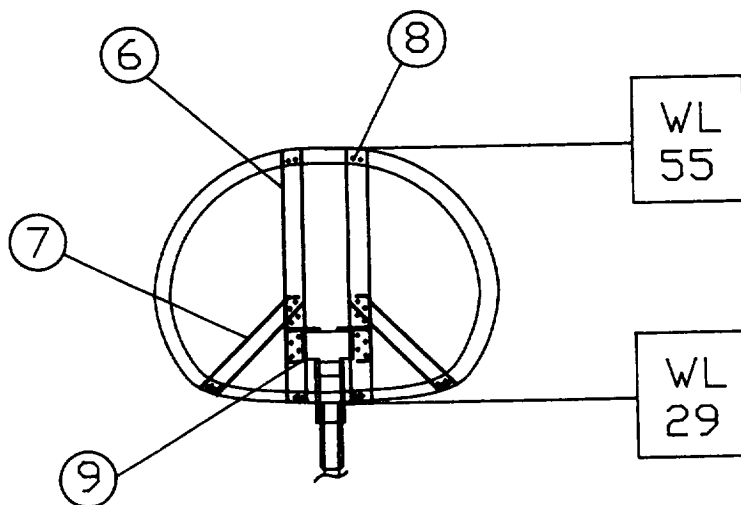
STA 23

NOTE: 1) ALL PARTS ALCL
2) BREAK ALL EDGE

MC152-6

ITEM	QTY	DESCRIPTION	MATERIAL OR PART #
12	4	BOLT - MACHINE, AIRCRAFT	AN12-5
11	4	BOLT - MACHINE, AIRCRAFT	AN4-5
10	1	FIREWALL	AMS 5604
9	1	LANDING GEAR ATTACHMENT	QQ-A-225/6
8	28	RIVET	MS20430A-4-6
7	2	C-CHANNEL	QQ-A-200/3
6	2	C-CHANNEL	QQ-A-200/3
5	18	RIVET	MS20430A-4-7
4	2	C-CHANNEL	QQ-A-200/3
3	1	BEARING	F93-1C-198-2
2	4	SEAT BELT ATTACHMENT	QQ-A-250/5
1	1	C-CHANNEL	QQ-A-200/3

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

.XX ± .01

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ANGULAR

± 1/2°

& BURS

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

SIZE	DATE	SCALE	DRAWN BY
B	12/2/93	1/20	CRAIG STEVENS

TITLE

BULKHEADS

DRAWING NO.   F93-1C-157-2

SHEET 1 OF 1

HEATING VENT
ELECTRICAL !
CONTROL SYS
VACUUM HARN

CROSS SECTION
STATION 28

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DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
.XX ± .01	SIZE	DATE	SCALE	DRAWN BY
	B	10/12/93	1/10	T.S. MCCORKLE
.XXX ± .001	TITLE			
	ELECTRIC, HEATING, AND VACUUM CONCEPT			
ANGULAR	DRAWING NO.			SHEET
± 1/2°	F93-1C-152			1 OF 1

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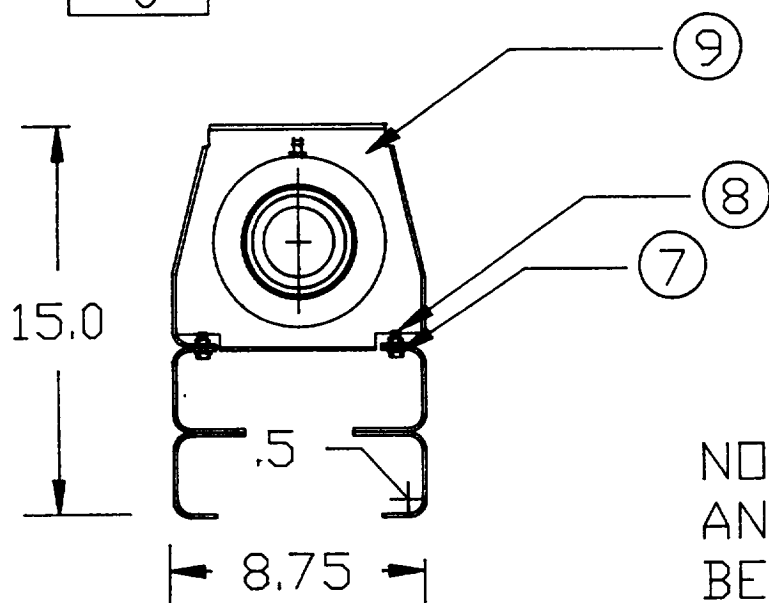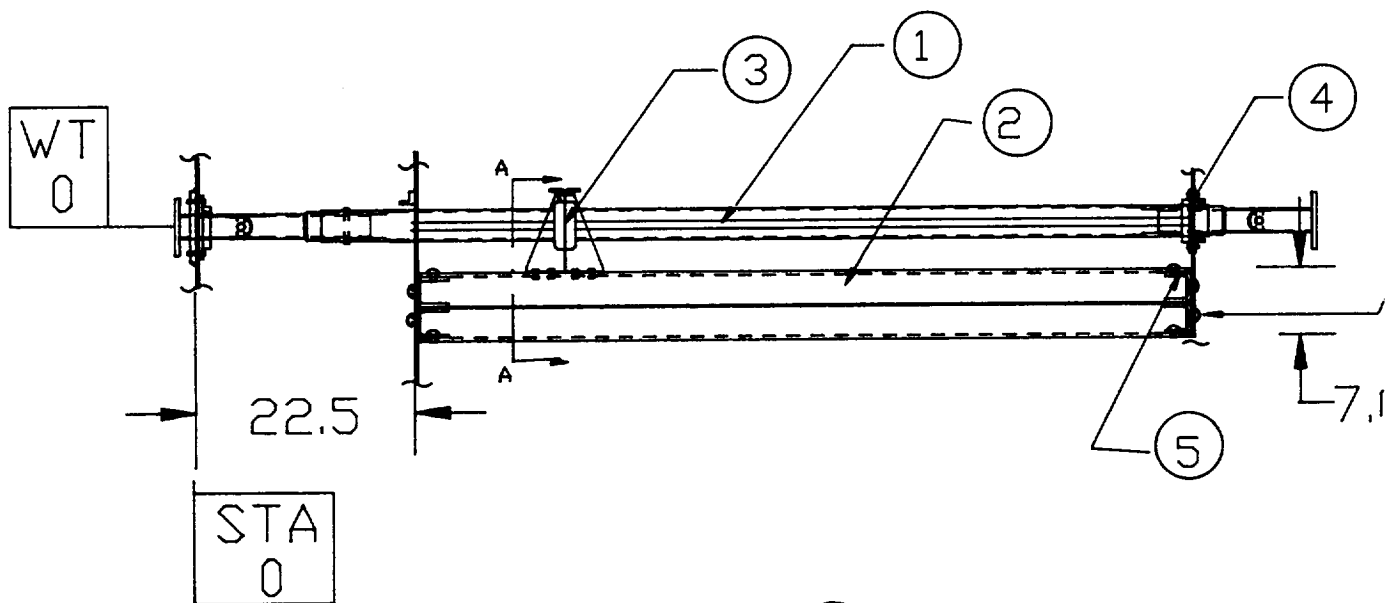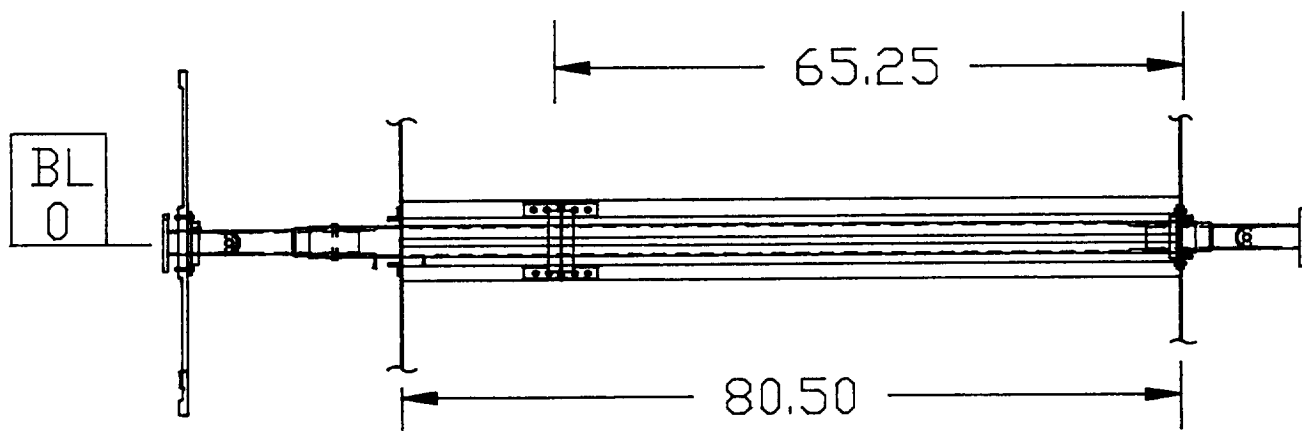
NOTE: ALL SHARP
AND CORNERS SHOU
BE ROUNDED OFF.

SECTION A-A

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ITEM	QNTY	PART DESCRIPTION	NUMBER OR MATERIAL
9	2	BEARING PLATE	2024-T3
8	8	NUT	AN320-5
7	8	BOLT	AN5C4
6	20	BOLT	AN6-12
5	8	"C" BRACKET	2024-T3
4	2	END BEARING	2024-T3
3	1	JOURNAL BEARING	2024-T3
2	4	"C" CHANNEL	2024-T3
1	1	DRIVE SHAFT	F93-C01-150

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DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL			EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA		
.XX ± .01 .XXX ± .001	SIZE B	DATE 10\12\93	SCALE 1/20	DRAWN BY T.S.M & G.S.	
	TITLE DRIVE SHAFT\BEAM ASSEMBLY				
ANGULAR ± 1/2°	DRAWING NO. F93-1C-151			SHEET 1 OF 1	

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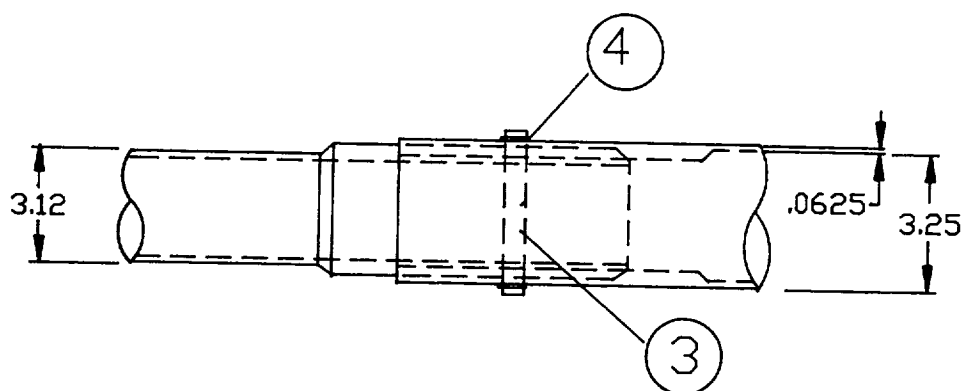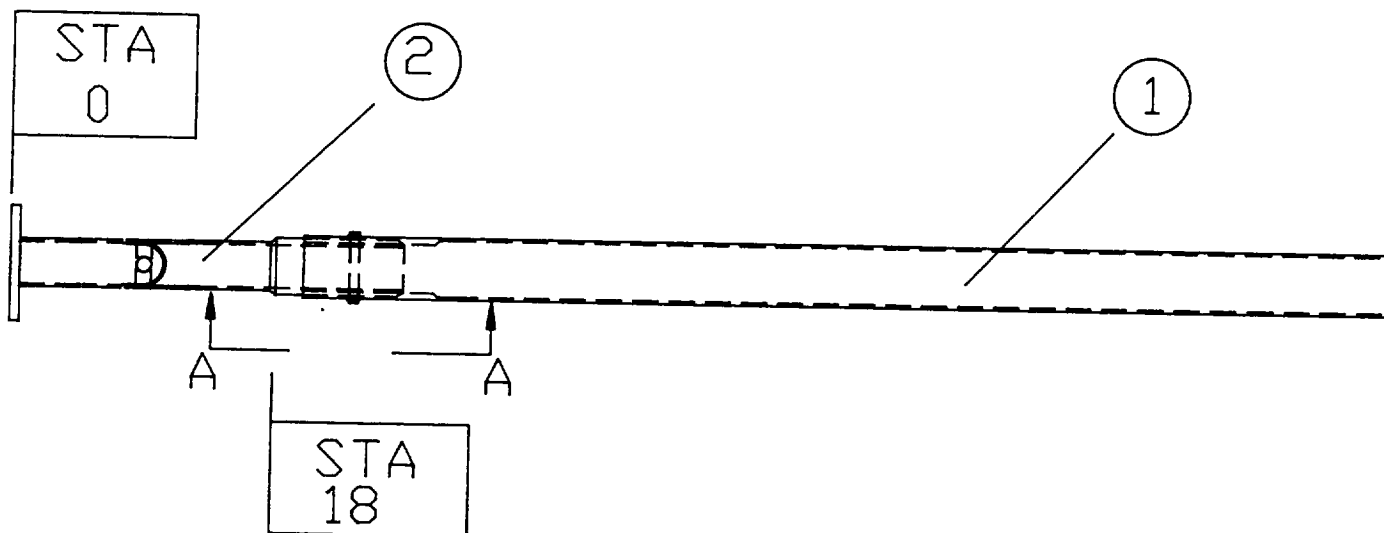
VIEW A-A
1/2 FULL SCALE

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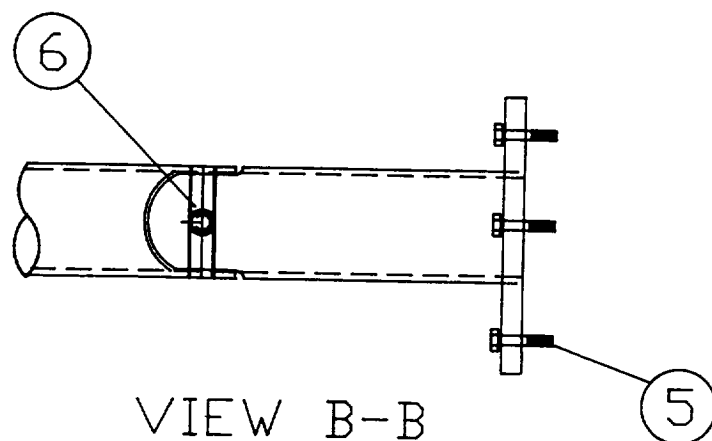
VIEW B-B
1/2 FULL SCALE

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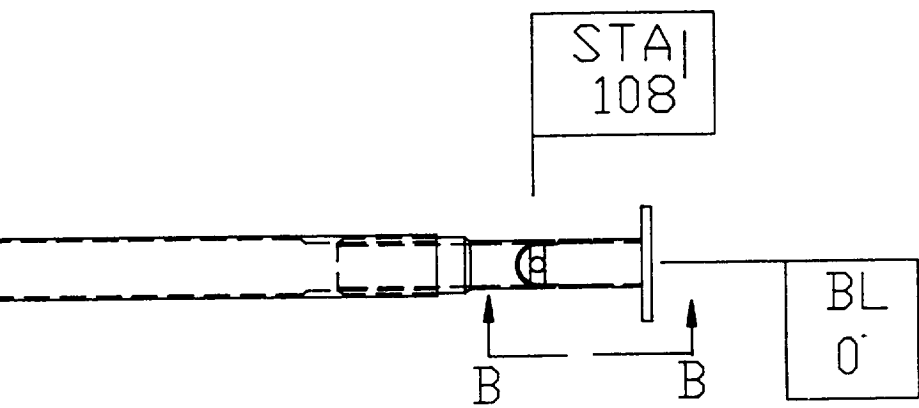
6	1	UNIVERSAL JOINT	MS20271
5	8	BOLT	AN6-12
4	1	CIR CLIP	2024-T3
3	1	SHEAR PIN	2024-T3
2	1	SECONDARY SHAFT	4130
1	1	PRIMARY SHAFT	4130
ITEM	QNTY	DESCRIPTION	PART #

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\3\93	1/10	T.S. MCCORKLE

TITLE
DRIVE SHAFT ASSEMBLY

DRAWING NO.	SHEET
F93-1C-150	1 OF 1
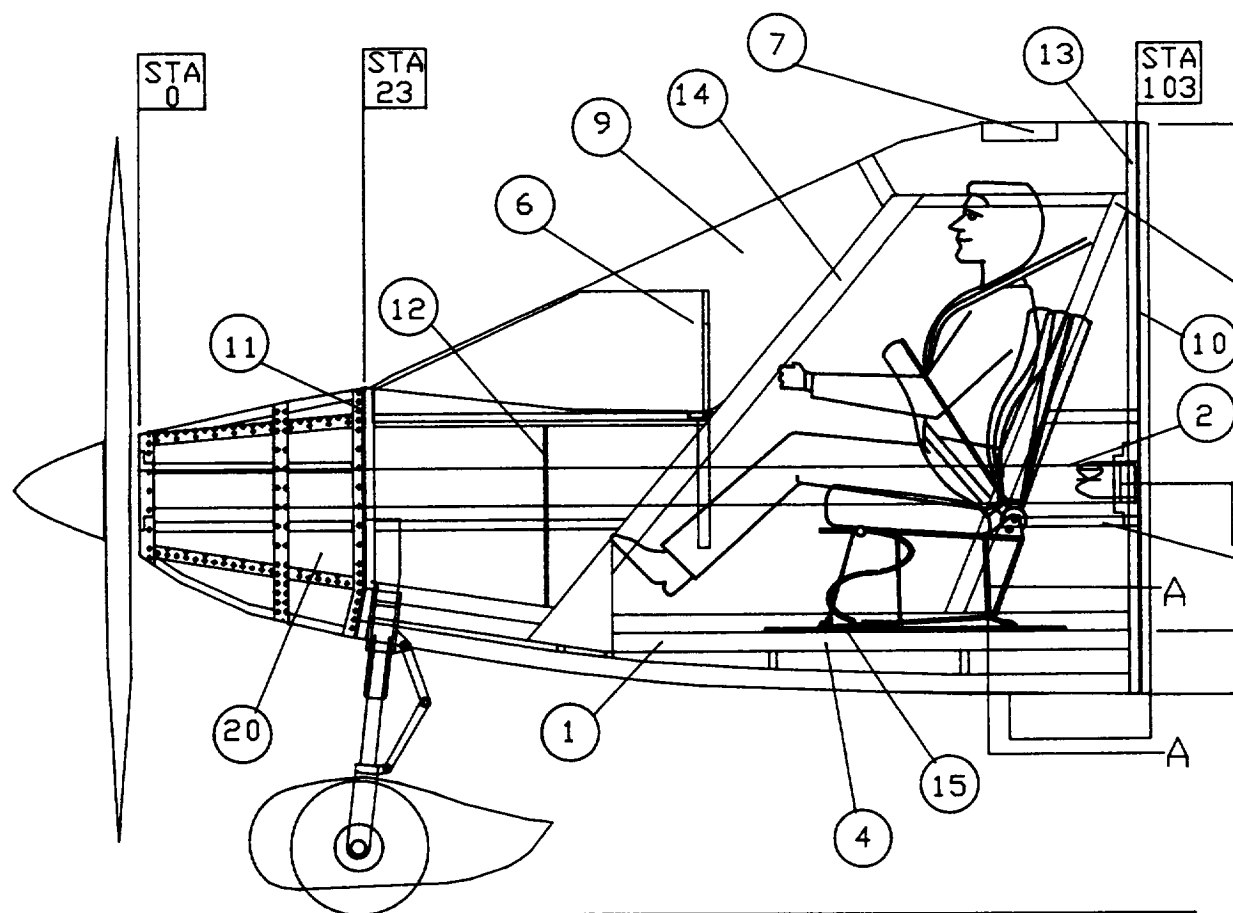
STA 0
STA 23
STA 103

NOTE:

ALL COMPONENTS ARE ALCLAD 2024-T3
UNLESS OTHERISE NOTED

ALL SHARP EDGES AND BURRS REMOVED

ITEM	QTY	DESCRIPTION	PART#
20	1	NOSE ASSEMBLY	F93-1C-123
19	30	TRUSSHEAD SCREW	AN526C-6-32
18	7	0.03 INCH SKIN	F93-1C-106
17	1500	RIVET	MS20470DD-4
16	4	SEAT BELT MOUNTS	F93-1C-122
15	2	JAARS SEAT ASSEMBLY	F93-1C-121
14	2	DOOR FRAME ASSEMBLY	F93-1C-120
13	16	.75 INCH ANGLE	F93-1C-119
12	1	FORWARD BULKHEAD	F93-1C-118
11	1	NOSE BULKHEAD	F93-1C-117
10	1	17-4PH STAINLESS STEEL FIREWALL	F93-1C-116
9	1	WINDSHEILD ASSEMBLY	F93-1C-115
8	2	DOOR ASSEMBLY	F93-1C-114
7	1	CABIN LIGHT ASSEMBLY	F93-1C-113
6	1	INSTRUMENT PANEL ASSEMBLY	F93-1C-111
5	14	2 INCH LONGERON MOUNTS	F93-1C-110
4	7	2 INCH LONGERONS	F93-1C-108
3	4	1/2 INCH ANGLE	F93-1C-105
2	1	DRIVE SHAFT BOX ASSEMBLY	F93-1C-130
1	1	FLOOR ASSEMBLY	F931-C-127
ITEM	QTY	DESCRIPTION	PART#

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	10/93	1/20	CARR

TITLE

CABIN ASSEMBLY

DRAWING NO. F93-1C-129-2	SHEET 1 OF 2

SECTION
SCALE

F93-1C-116 (REF)

F93-1C-121 (

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F93-1C-110 (REF)

NOTE:

⚠1 ENGINE VENT INTAKE

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F93-1C-110 (REF)

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F93-1C-119 (REF)

DATE	BY	DRAWING NO.	SHEET
12/93	CARR	F93-1C-129-2	2 OF 2

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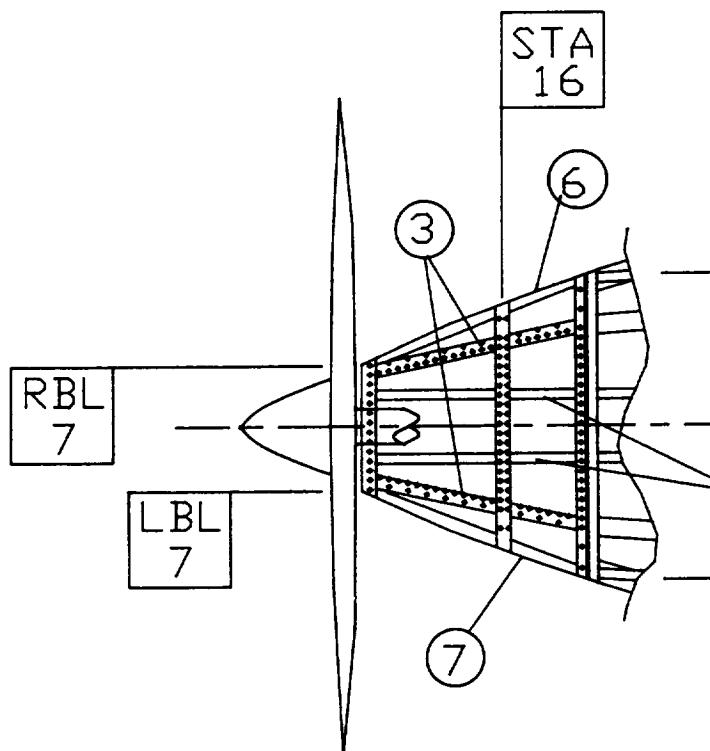
NOTE:

ALL COMPONENTS ARE
ALCLAD 2024-T3

ALL SHARP EDGES
AND BURRS REMOVED

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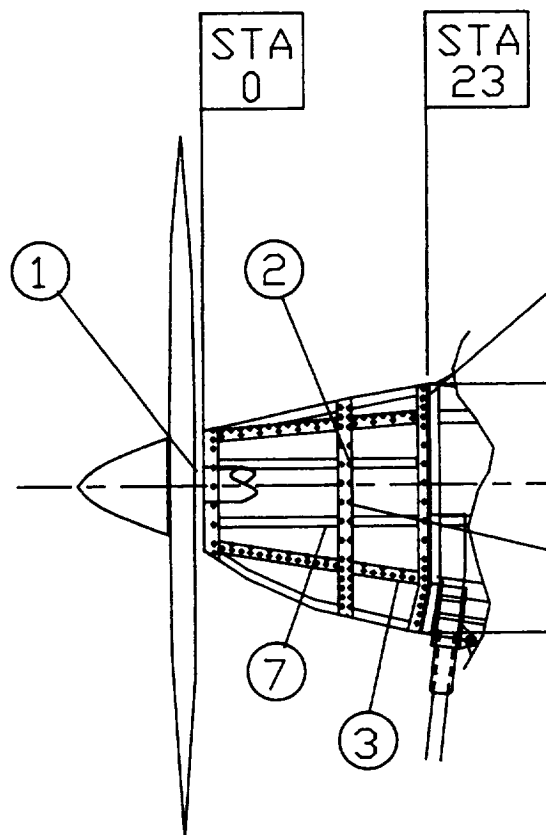
ITEM	QTY	DESCRIPTION	PART#
9	512	RIVET	MS20470DD-4
8	19	TRUSSHEAD SCREW	AN526C-6-32
7	1	NOSE ACCESS PANEL	F93-1C-107
6	3	0.03 SKIN	F93-1C-106
5	8	1/2 IN ANGLE	F93-1C-105
4	16	NOSE LONGERON MOUNTS	F93-1C-104
3	8	NOSE LONGERON	F93-1C-103
2	1	NOSE FRAME	F93-1C-102
1	1	PROPELLER BULKHEAD	F93-1C-101

DIMENSION TOLERANCES
UNLESS OTHERWISE SPECIFIED
DECIMAL

$.XX \pm .01$

$.XXX \pm .001$

ANGULAR

$\pm 1/2°$

EMBRY-RIDDLE AERONAUTICAL UNIVERSITY
DAYTONA BEACH FLORIDA

SIZE B	DATE 10/93	SCALE 1/20	DRAWN BY CARR

TITLE
NOSE ASSEMBLY

DRAWING NO.
F93-1C-128-2

SHEET
1 OF 1

STA 23

STA 65

STA 103

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RBL 26

LBL 16

LBL 26

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F93-1C-118 (REF)
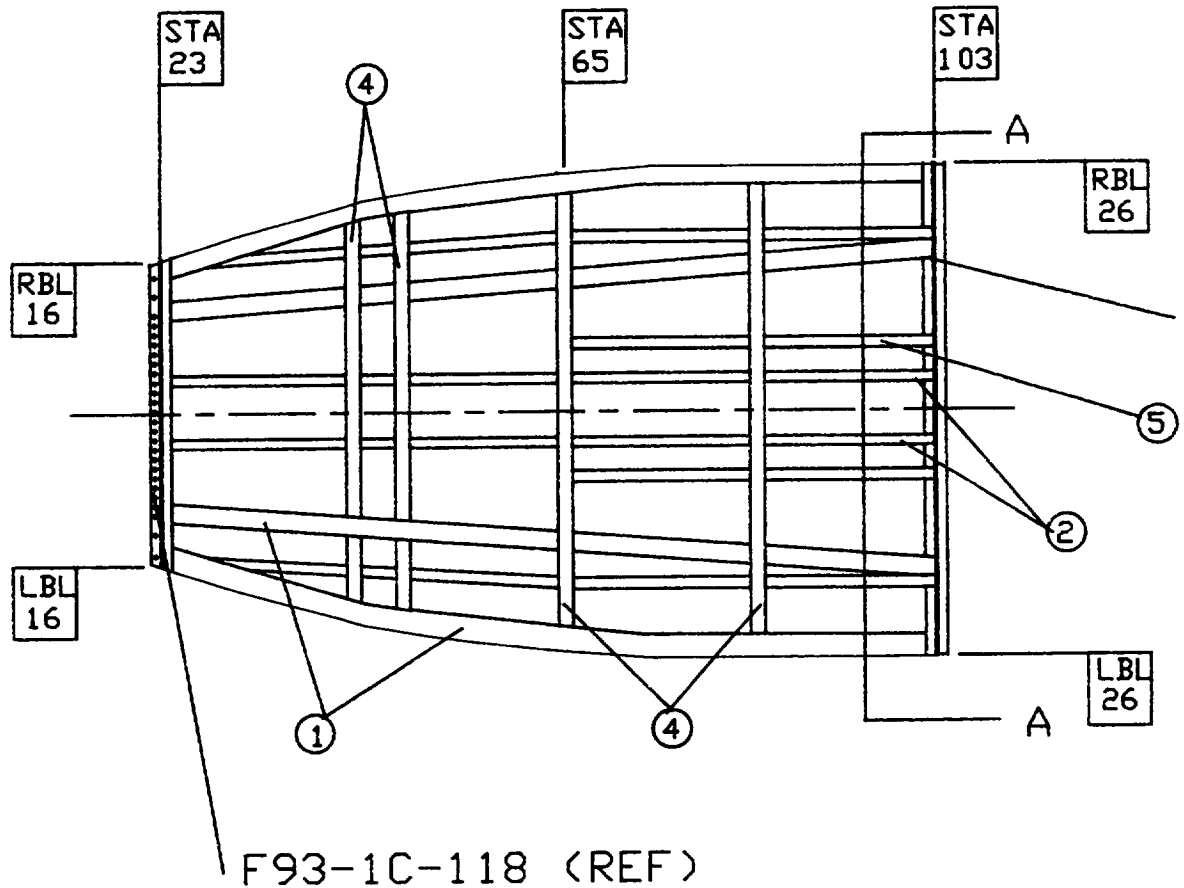
NOTE:

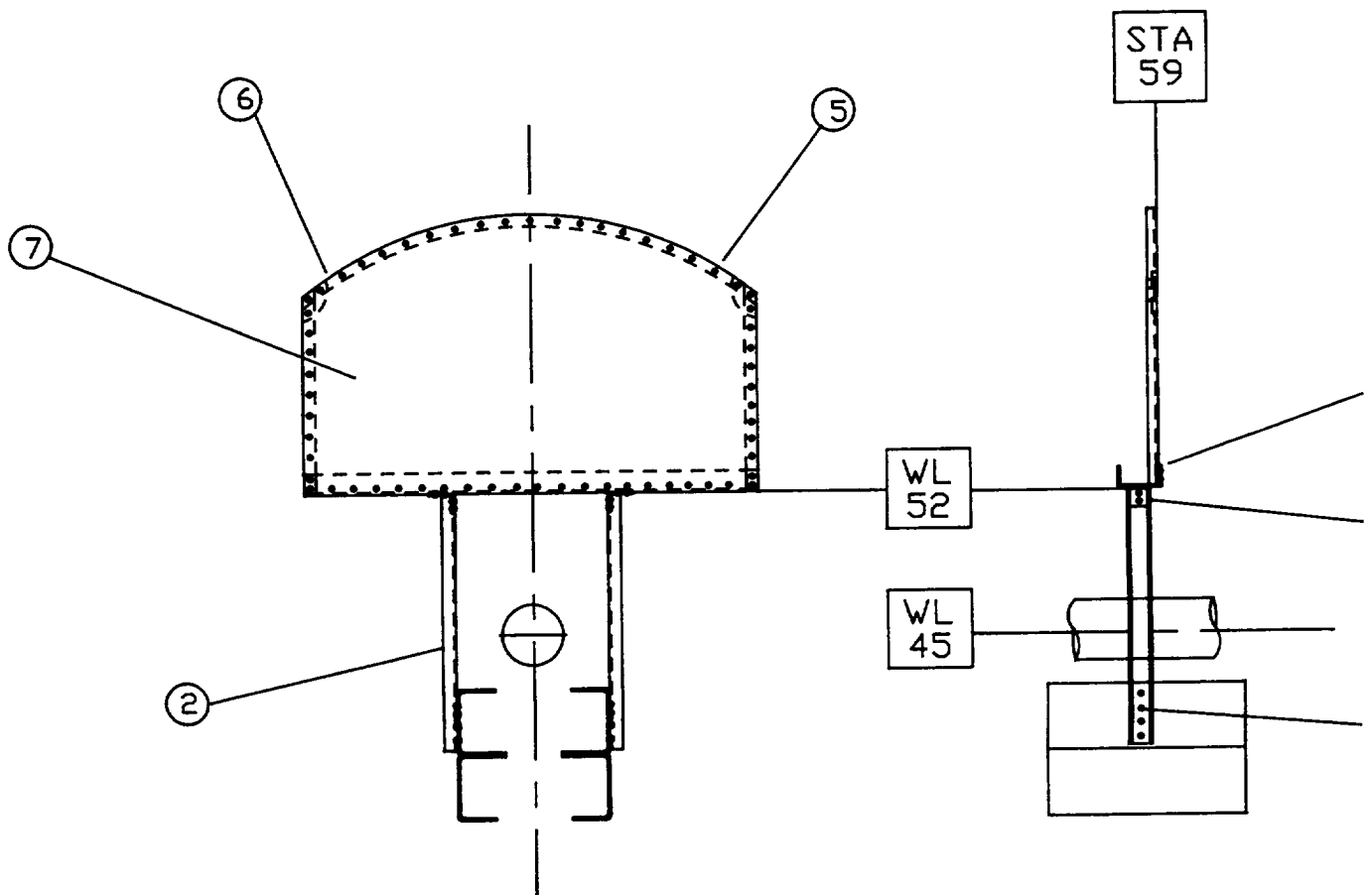ALL COMPONENTS ARE ALCLAD 2024-T3
UNLESS OTHERWISE NOTED

ALL SHARP EDGES AND BURRS REMOVED

UPPER FLOOR SKIN IS
.06 THICKNESS

93-1C-116 (REF)

9	8	1/2 INCH ANGLE	F93-1C-105
8	200	RIVET	MS20470DD-8
7	3100	RIVET	MS20470DD-4
6	2	0.06 INCH FLOOR SKIN	F93-1C-126
5	8	SEAT SUPPORT RIB	F93-1C-125
4	4	FLOOR RIB	F93-1C-124
3	36	2 INCH LONGERON MOUNTS	F93-1C-110
2	4	1 INCH LONGERON	F93-1C-109
1	18	2 INCH LONGERON	F93-1C-108
ITEM	QTY	DESCRIPTION	PART#

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
.XX ± .01 .XXX ± .001	SIZE B	DATE 10/93	SCALE 1/20	DRAWN BY CARR
	TITLE FLOOR ASSEMBLY			
ANGULAR ± 1/2°	DRAWING NO. F93-1C-127-2			SHEET 1 of 1

⑥　⑤　⑦　②

STA 59

WL 52

WL 45

NOTE: ALL PARTS ALCLAD FINISHED
      BREAK ALL SHARP EDGES AND BL

①

③

④

①	1	C-CHANNEL	NO. 14 2024 AL
②	2	C-CHANNEL	NO. 10 2024 AL
③	2	ANGLES EQUAL LEG	NAS341-14
④	16	RIVET ROUND HEAD	MS20430A-6-8
⑤	61	RIVET ROUND HEAD	MS20430A-2-4
⑥	3	ANGLES EQUAL LEG	NAS341-4
⑦	1	INSTRUMENT PANEL FACE	QQ-A-255
ITEM	QTY	DESCRIPTION	MATERIAL OR PART #

DIMENSION TOLERANCES UNLESS OTHERWISE SPECIFIED DECIMAL	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
.XX ± .01 .XXX ± .001	SIZE B	DATE 10/10/93	SCALE 1/10	DRAWN BY R. CRAIG STEVENS
	TITLE INSTRUMENT PANEL STRUCTURE			
ANGULAR ± 1/2°	DRAWING NO. F93-1C-111			SHEET 1 OF 1