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Viper
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Preliminary Design / Section 2
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1.0 Design Specification

1.1 The design of the Primary Flight Trainer must comply with FAA Regulations Part 23 unless specifically exempted by the following.

1.2 The primary mission of the aircraft is a primary flight trainer. This requires but is not limited to two seats.

1.3 The aircraft must be redesigned with a new airfoil and engine to verify whether or not improvements can be made to the baseline configuration.

1.4 NASA Natural Laminar Flow airfoils should be used.

1.5 Engines that are not currently FAA certified may be used as a means of introducing new technology to the design process. Approval must be granted from the Contract Monitor.

1.6 An intercom and headset must be provided for all occupants to enhance safety and to lessen the probability of long term hearing loss due to noise exposure.

1.7 The range must be at least 500 nautical miles with VFR reserves.

1.8 A cruise speed of 120 knots is desired, if achievable.

1.9 Runway length over 50 foot obstacle must be under 2500 feet.

1.10 The aircraft must be capable of a sustained turn at a load factor of 2, safely above the accelerated stall speed.

1.11 A cost goal of $50,000 (not including liability insurance) is desired for a production run of 500 aircraft, spread over 5 years.

1.12 VFR instrumentation is required, along with one navigation and communication radio and a Mode C transponder.

1.13 The new FAR Part 23 seat crash worthiness requirements and head injury criteria are to be incorporated into the design process.

2.0 Summary Statement

Throughout the redesign process of the Viper, several changes were made to the configuration to meet all the requirements that were set forth at the beginning of the project. The process began by replacing the baseline aircraft's airfoil (NACA 65_2-415) with an NLF 0414 Natural Laminar Flow airfoil. The general arrangement drawing is shown in figure 2.1 for clarity in the following discussion.

The next major change was replacing the baseline aircraft's Lycoming 0-235 engine with a Teledyne Continental GR-36 rotary combustion engine. These changes were made with hopes that they would increase the baseline aircraft's overall performance. However, to meet the design goals of a 100 knot cruise speed further modifications were necessary.

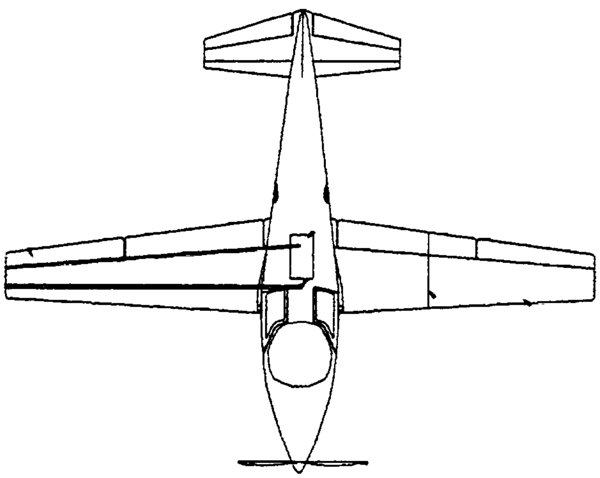
First, both the wing and the horizontal tail areas were decreased to reduce drag. However, the fuselage height and width had to be slightly increased (16.1% and 8%, respectively) to accommodate the FAA certified crash worthy seats as well as to give the occupants sufficient headroom. The inboard profile drawing is shown in Figure 2.2 below showing the JAARS seats as well as physical space for the different components of the airplane. Figure 2.3 shows a cross section of the fuselage through the cockpit displaying the headroom and crew "elbow room."

Further changes were made to meet stability and control requirements. The area of the horizontal tail had to be re enlarged, but remained 20% smaller than the baseline. This provided the required static margin of 14%.

Finally, to increase directional stability, the vertical tail was swept back to an angle of 22 degrees about the quarter chord, which required enlarging the rudder 34% to achieve adequate spin recovery.

Since the design process was based on an existing aircraft, there were few changes that could be made to the existing configuration. Had there been more time the only changes that would have been made would be to redesign the aircraft's door arrangement. This would have allowed for a less complex structure to support the door resulting in a further reduction of weight and inconvenience to the crew. Also, move the Teledyne Continental engine forward into the nose of the aircraft where it should be due to it's small size.

As it is, the redesigned Viper meets all of the design specifications outlined in the design specifications above and would be a viable addition to the primary flight trainer market.



SPECIFICATIONS	
GROSS WEIGHT	1382 LBS.
EMPTY WEIGHT	822 LBS.
FUEL WEIGHT	180 LBS.
PAYLOAD WEIGHT	380 LBS.
STALL SPEED @ SLS, FLAPS UP	56 KNOTS
STALL SPEED @ SLS, FLAPS DOWN	47 KNOTS
MAXIMUM SPEED @ 5000 FEET	124 KNOTS
CRUISE SPEED @ 5000 FT, 70% POWER	100 KNOTS
MAXIMUM SLS RATE OF CLIMB	672 FPM
MAXIMUM CRUISE RATE OF CLIMB	687 FPM
TAKEOFF DISTANCE OVER 50 FT OBSTABLE	776 FEET
LANDING DISTANCE OVER 50 FT OBSTACLE	1458 FEET
CRUISE RANGE	625 NM
MAXIMUM RANGE	783 NM

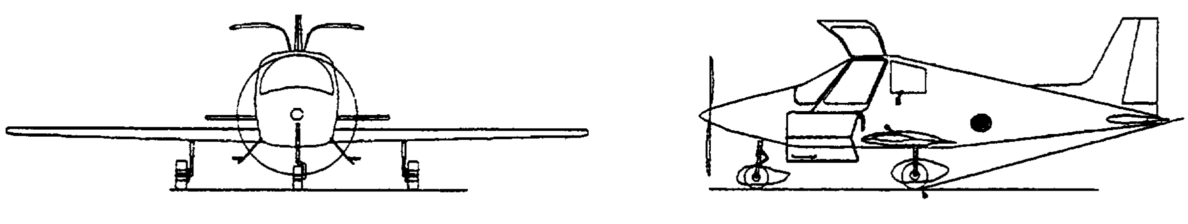


Figure 2.1 General Arrangement Drawing

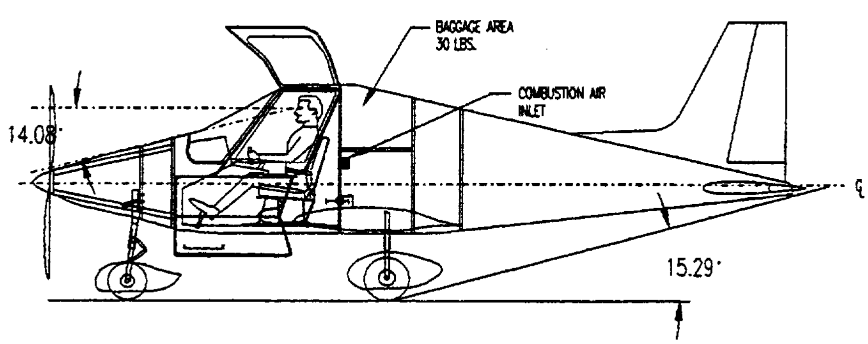


Figure 2.2 Inboard Profile Drawing

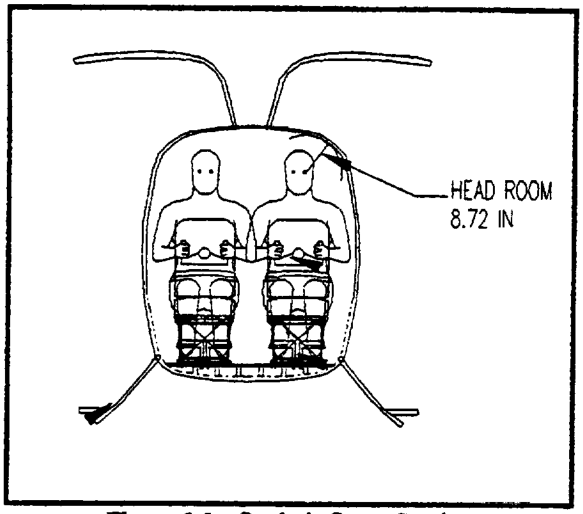


Figure 2.3 Cockpit Cross Section

3.0 Configuration Development

3.1 Initial Sizing

The empty weight of the aircraft was estimated using table 6.2 (Raymer):

$$\frac{W_e}{W_0} := a + bW_0^{C_1} \cdot A^{C_2} \cdot \left(\frac{hp}{W_0}\right)^{C_3} \cdot \left(\frac{W}{S}\right)^{C4} \cdot V_{max}^{C_5} = 0.632$$

Where: $W_0 := W_{0baseline} - \Delta engine\ weight = 1467\ lbs$

$hp := 85 hp$

$V_{max} := 126.6 mph$

$a := 0.25$	$C_1 := 0.2$	$C_4 := -0.05$
$b := 1.14$	$C_2 := 0.08$	$C_5 := 0.27$
$\frac{W}{S} := 13.03 \frac{lb}{ft^2}$	$C_3 := 0.05$	$A := 7.98$

The fuel weight is estimated using the five legs of a 500 nautical mile mission, with 30 minutes loiter time. Figure 3.1 describes the mission profile of the Viper.

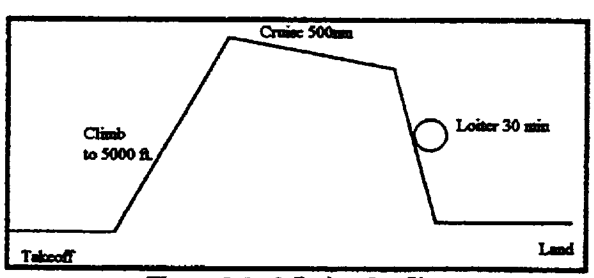


Figure 3.1 Mission Profile

To calculate the fuel fractions of each leg, the mach number at takeoff and climb had to be calculated, as well as for both cruise and loiter. Takeoff velocity was assumed to be 1.10 of the stall velocity. The parasite drag and Oswald efficiency were assumed, and a cruise speed of 100 knots was assigned. As stated in the Project Summary, page 1, a 100 knots cruise velocity was assigned, with the approval of the Contract Monitor, due to the use of Teledyne Continental 85 horsepower rotary engine. The mission leg fuel fractions are summarized in Table 3.1.

Table 3.1 Final Weight Fractions

Engine, Start, Taxi, Takeoff	0.98
Climb/Accelerate	1.001
Cruise	0.931
Loiter	0.994
Land and Taxi	0.995

The product of the individual weight fractions is Wx/Wo. From Raymer's equation 6.2, the total fuel weight fraction is:

$$\frac{W_f}{W_0} := 1.06\left(1 - \frac{W_x}{W_0}\right) = 0.109$$

Where: $\frac{W_x}{W_0} := 0.897$

The total weight estimate is calculated using equation 6.1 from Raymer:

$$W_0 := \frac{W_{crew} + W_{payload}}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}} = 1471\ lbs.$$

Where: $W_{crew} := 340 lb$

$W_{payload} := 40 lb$

The total fuel weight estimation is then calculated by multiplying the total fuel fraction by the estimated gross weight (Wo) of 1471 pounds. This yields a fuel weight of 160.8 pounds or 26.81 gallons. To satisfy this requirement, two fifteen gallon tanks were incorporated into the design of the wing at fuselage station 137 and butt-line 50.

3.2 Thrust-to-Weight Ratio Calculation

It is necessary to verify that the aircraft's engine can provide sufficient power during cruise conditions, as well as in a sustained two-g turn. Applying equation 6.18 for both cruise and a sustained two-g turn:

$$\left[\frac{T}{W}\right]_{req} = \frac{n}{\left(\frac{L}{D}\right)\cdot\frac{550\,\eta_p}{V}}$$

Where, at cruise: $n := 1$

$V_{cruise} := 168.9\frac{ft}{sec}$

$\eta_p := 0.80$

(Eqn. 6.13) $\frac{L}{D} := 10.69$

in 2-g turn (loiter): $n := 2$

$V_{loiter} := 135\frac{ft}{sec}$

$\eta_p := 0.80$

(Eqn. 6.13) $\frac{L}{D} := 12.47$

The actual thrust-to-weight ratio was found using the horsepower available under each condition divided by the estimated gross weight.

$$\left(\frac{T}{W}\right)_{actual} = \frac{Hp_{available}}{W_0}$$

Where, cruise: $Hp_{available} := 59.5\,hp$

2-g turn: $Hp_{available} := 85\,hp$

The results are shown in Table 3.2 below.

Table 3.2 Thrust-to-Weight Ratio Verification

	(T/W)req.	(T/W)actual
Cruise	0.036	0.0404
2-g turn	0.049	0.058

Since in both cases the thrust-to-weight ratio required is less than the actual thrust-to-weight ratio, the aircraft has sufficient power.

3.3 Preliminary Dimensions

The preliminary dimensions are found using the estimated gross weight, the taper and aspect ratios of the baseline configuration. The results of the wing calculations are shown in Table 3.3 below.

Table 3.3 Wing Dimensions

λ	0.45
S_{ref}	112.9 sq. ft.
b_w	30.0 ft
m.a.c.	3.93 ft.

The fuselage length was determined using table 6.3 from. Figure 3.2 shows a fuselage section through the engine compartment showing the engine mounting structure.

$$L := a\cdot W_0^{\,C} \quad = 23.39\,ft$$

Where: $a := 4.37$
$C := 0.23$

The horizontal and vertical tail areas were determined using equations 6.28 and 6.29 from Raymer:

$$S_{ht} := \frac{C_{ht}\cdot c_w\cdot S_w}{L_{ht}} \quad = 22.0\text{ sq. ft.}$$

Where: $C_{ht} := 0.70$

$L_{ht} := 14.1\,ft$

$S_w := 112.9\,ft^2$

$c_w := 3.93\,ft$

$$S_{vt} := \frac{C_{vt}\cdot c_w\cdot S_w}{L_{vt}} \quad = 9.61\text{ sq. ft.}$$

Where: $C_{vt} := 0.04$

$L_{vt} := 14.1\,ft$

$S_w := 112.9\,ft^2$

$c_w := 3.93\,ft$

3.3 Size of Control Surfaces

Assuming the ailerons cover 40% of the half span, figure 6.3 from Raymer was used to approximate the chord of the aileron. Using this and the baseline taper and aspect ratios, and the span of the main wing the following results were determined, shown in Table 3.4.

Table 3.4 Results of Control Sizing

$c_{aileron}$	0.825 ft.
$b_{aileron}$	6.00 ft.
$S_{aileron}$	4.95 sq.^{2}

The length of the elevator and rudder, as stated by Raymer is assumed to cover 90% of the span.

From this, the span of the elevator is 4.22 feet. Similarly, the span of the rudder is 1.748 feet.

3.4 Weight and Balance

One of the most important aspects in the design of an aircraft is the weight and balance analysis. It affects not only the stability and control of the aircraft, but also the way in which the aircraft may be loaded. Much time was spent on the weight and balance calculations, and the process will be summarized in this section.

The weight and balance calculations began by determining the weights of each individual component of the aircraft. These weights were found using either equations from Raymer, in chapter 15 or from existing data supplied by the Contract Monitor. The results are summarized in Table 3.5 below.

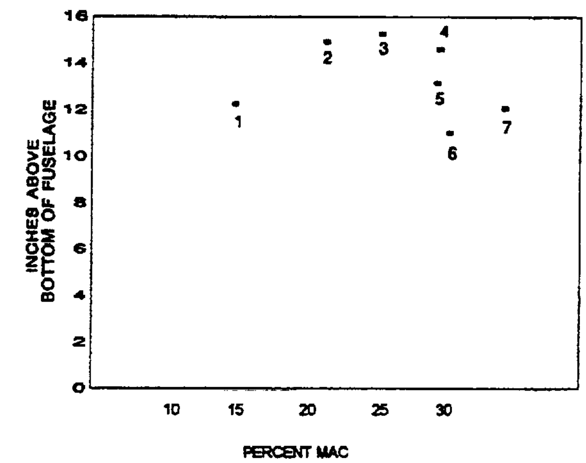
Table 3.5 Component Weights

Component	Weight (pounds)
Wing	184.5
Horizontal Tail	15.8
Vertical Tail	7.8
Fuselage	109.9
Main Gear	87.0
Nose Gear	36.5
Engine	196.0
Fuel System	34.8
Flight Controls	23.2
Electrical System	40.0
Avionics	15.0
Furnishings	51.0
Propeller	20.0
EMPTY	**821.5**
Fuel	180.0
Baggage	40.0
Pilots	340.0
GROSS	**1381.5**

The gross weight of 1381.5 pounds for the aircraft compares to 1607 pounds for the baseline aircraft, a difference of 225 pounds. This difference can be attributed to the use of a much lighter engine, which also allowed for the downsizing for most of the major components of the aircraft, thereby reducing the weight even more.

Once the weights of each component were determined, the horizontal and vertical moment arms of each component were measured relative to fuselage station zero, located at the nose and butt-line zero located at the bottom of the fuselage, respectively. Knowing the moment arms and weights of each component, the vertical and horizontal moment contributions of each component were calculated.

Finally, by summing the moment contributions and dividing by the weight, the center of gravity for several possible flight scenarios were determined. These positions are summarized in Figure 3.2 below.



	FUEL	BAGGAGE	PEOPLE
1	0	0	340
2	180	40	340
3	0	40	100
4	0	0	0
5	180	0	100
6	180	40	100
7	180	0	0

Figure 3.2 Center of Gravity Envelope

As shown, the forward and aft center of gravity limits are 15 % and 33 % (points 1 and 6) of the mean aerodynamic chord, respectively. It will be shown later that these limits satisfy both the stability and the control requirements.

4.0 Aerodynamics

4.1 Wing Design

A Natural Laminar Flow airfoil was decided upon based on the Design Specifications set forth by the Contract Monitor. The NLF-0416 airfoil was originally used, but was replaced by the NLF-0414 in order to achieve more favorable stall characteristics.

According to NASA documents regarding Natural Laminar Flow airfoils, these airfoils provide certain characteristics that older 6-series airfoils lack. For instance, the maximum lift coefficient is reportedly higher with minimal increase in drag.

Since a smaller, and therefore lighter, engine is used to propel the aircraft, the weight of the wing was reduced. However, some aspects of the wing remained the same as the baseline. The aspect and taper ratios of the wing remained the same at 7.98 and 0.45, respectively. For the most part, the other characteristics of the Viper's wing changed dramatically.

From the airfoil data provided by NASA, the maximum lift coefficient was 1.62. Also from airfoil data, the airfoil lift-curve slope was 0.113 per degree.

In converting the two dimensional lift curve slope to the three dimensional lift curve slope, Raymer stated that the maximum lift of the wing is 90% of the maximum lift of the airfoil if the wing is unswept. In the case of the Viper, the wing is unswept. Therefore, the maximum lift coefficient is 1.46 with zero flap deflection (clean).

In converting the lift curve slopes from two dimensional to three dimensional, a more involved process was used. Raymer stated that the wing lift curve slope may be determined in terms of the wing aspect ratio, the area of the wing, and flight velocity using equation 12.6 from Raymer:

$$C_{L\alpha} := \frac{2\cdot\pi\cdot A\cdot\frac{S_{exposed}}{S_{ref}}\cdot F}{\left[2+\sqrt{4+\left(\frac{A^2\cdot\beta^2}{\eta^2}\right)\cdot\left(1+\frac{\tan(\Lambda)}{\beta^2}\right)}\right]}$$

Where: $A := 7.98$

$S_{exposed} := 92.9\,ft^2$

$S_{ref} := 112.9\,ft^2$

$F := 1.37$

$\beta := 0.996$

$\eta := 1.022\cdot\frac{1}{deg}$

$\Lambda := 0\cdot deg$

It was found the wing lift curve slope is significantly less than the airfoil lift curve slope at 0.098 per degree. The three dimensional lift curve slope is shown in Figure 4.1 below.

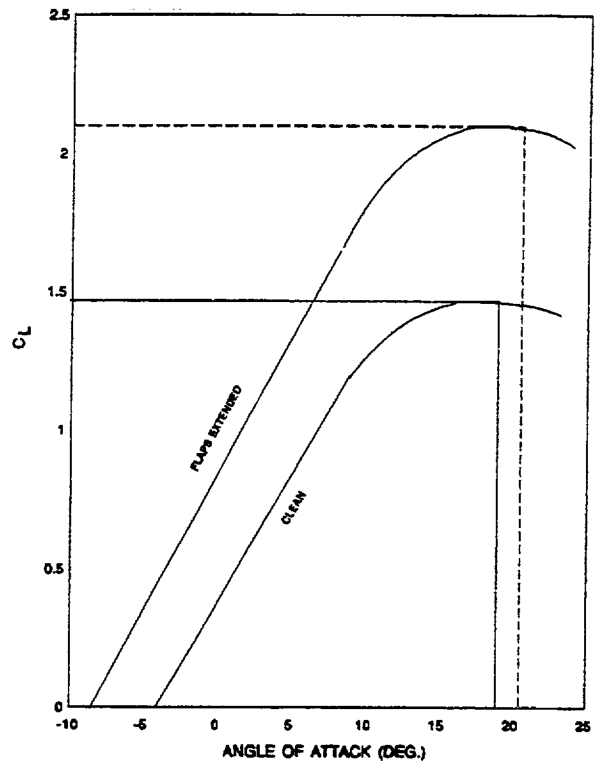


Figure 4.2 Three Dimensional Lift Curve Slope

Due to the extraordinary characteristics of the Natural Laminar Flow wing, the stall angle under clean conditions was calculated to be 18.86 degrees using equation 12.17 from Raymer.

When flaps are extended, all the variables regarding the aerodynamics of the wing change except the lift curve slope. The zero lift angle of attack increased negatively from -4 to -8.7 degrees.

The effective flap area was estimated to be about 53 square feet, with a small sweep angle of 2.5 degrees. With flaps extended fully, the maximum lift of the wing increased dramatically from 1.46 to 2.07. Likewise, the stall angle of attack increased from 18.9 to 20.4 degrees.

With flaps completely retracted the stall speed was estimated at about 55.8 knots. With flaps fully extended, the stall speed was reduced to 47 knots.

4.2 Propeller Selection

The propeller selection process began with deciding which type of propeller to use for the Viper. A fixed pitch, two bladed propeller was chosen due to the price range and primary use of the aircraft. Next, a diameter of 74 inches was chosen. This decision was made because a 74 inch diameter blade is the standard size for general aviation aircraft. Knowing the diameter of the propeller and the cruise velocity, the efficiency of the propeller at cruise conditions was determined to be 84%. The efficiency was read from the graph of power coefficient versus advance ratio that can be found in NACA War Report 286. A blade angle of 25 degrees was used. This blade angle provided the best efficiency at cruise conditions. The propeller efficiency at stall speed was determined to be 60% for the selected propeller. This analysis resulted in a propeller specification of 74 inch diameter, with a rated pitch of 70 inches.

4.3 Propeller Noise

Once the geometry of the propeller was determined, estimates for the propeller noise were computed using the method described in the Aerospace Information Report 1047. The method involved the following parameters: 1) propeller diameter, 2) number of blades, 3) propellor RPM, 4) power output to the propeller, 5) cruise velocity, 6) temperature, and 7) number of propellers. With these values a series of partial noise levels were calculated and then summed to make a prediction of the propeller noise, summary of this analysis is shown in Table 4.1 below. An A-weighted sound pressure level of 56.5 dBA was calculated for flight directly overhead at an altitude of 1000 feet. This value is below the 68.9 dBA allowable sound pressure level mandated by FAR Part 36.

Table 4.1 Partial Noise Levels

	Noise Level (dB)
Prop Power/Rot. Tip Speed	67
Prop Dia./# of blades	11
Atm. Abs./Sph. Spread	-7.5
Noise Level Adjustment	-14
Total (PNL)	**56.5**

4.4 Drag Polar

4.4.1 Parasite Drag

The parasite drag C_{do} is the inevitable drag produced from roughness of the wetted surfaces of the major components (i.e., the wing, fuselage, and tail surfaces). Other significant sources of parasite drag come from the exposed landing gear parts, leakage's, protuberances, gaps, and physical engine requirements such as cooling.

The task of determing the parasite drag C_{do} was separated into four segments. First, the parasite drag of the four main components (wing, fuselage, vertical tail, and horizontal tail) was determined and summed. Secondly, the miscellaneous drag was found. In this case the only miscellaneous drag was the fixed landing gear. Thirdly, a rough estimate was taken for drag produced by leakage's and protuberances. Finally, engine drag was taken into consideration.

Before any of the component parasite drags could be found, the Reynolds Number, form factors, interference factors, coefficients of friction, and wetted surface area to reference area ratios had to be determined. The flow regimes across the main surfaces were assumed to be turbulent, with the exception of the wing. Because a Natural Laminar Flow (NLF) airfoil was being used, laminar flow was assumed over the first 50% of the mean aerodynamic chord (m.a.c.). After determining the individual parasite drags for the four main components, these were summed to provide the total component parasite drag. Table 4.2 lists the parasite drag for each of the major components.

Table 4.2 Major Component Parasite Drag

Component	**Parasite Drag**
Wing	0.00594
Fuselage	0.0111
Horizontal Tail	0.001729
Vertical Tail	0.000805

Raymer provided the following equation for the component drag:

$$C_{do_c} = \frac{\sum (C_{f_c} FF_c Q_c S_{wet_c})}{S_{ref}}$$

The fuselage and the wing provided the highest amount of parasite drag produced by the Viper. The total component parasite drag was 0.0196 under cruise conditions.

The fixed landing gear was the only component considered for the miscellaneous drag. The frontal areas of the landing gear struts and wheels was determined from the drawings. Raymer provided a list of estimated "drag areas," D/q, for various landing gear components. (The values from table 12.5 which apply to the Vipers landing gear are stated here for completeness [Wheel and tire with fairing; 0.13, Round strut or wire; 0.30]). These values from Raymer were then multiplied by the frontal area of the landing gear to determine the D/q value needed to find the parasite drag. Since the miscellaneous parasite drag consisted only of the landing gear, the drag was determined using the following equation:

$$C_{d_{misc.}} = 1.2\frac{\sum(D/q)}{S_{ref}}$$

As a result, the total miscellaneous parasite drag for the Viper was calculated to be 0.005471.

Raymer simply stated that the parasite drag produced by leakage's and protuberances are five percent of the total parasite drag. Therefore, this drag term was simply added to the parasite drag.

Lastly, the amount of drag produced by engine cooling and components were significant enough that it must be taken into consideration. The drag due to cooling the engine is produced by the momentum loss of the air passing through the air ducts which house the radiators used to cool the engine. This drag is heavily dependent on the velocity v and the break horsepower bhp as represented by the following equation:

$$Cd_{ocooling} := \frac{3\left[4.9\left(10^{-7}\right)\right]\frac{bhp \cdot T^2}{\sigma v}}{S_{ref}}$$

The cooling drag was estimated to be 0.001764 under cruise conditions, which is higher than the amount drag produced by the horizontal tail. The remainder of the engine related drag is due to physical parts of the engine such as the air intake and the exhaust pipe. The resulting contribution to the total parasite drag was small at 0.0004517.

The complete parasite drag of the Viper under cruise conditions was simply the sum of all drag produced by the components. This values have been repeated for completeness in table 4.3. The resulting parasite drag C_{do} was 0.02851, which is lower than a typical primary flight trainer aircraft. This can be attributed in part to the use of an NLF airfoil.

Table 4.3 Summary Parasite Drag

Component	Parasite Drag
Wing	0.00594
Fuselage	0.0111
Horizontal Tail	0.001729
Vertical Tail	0.000805
Miscellaneous	0.005471
Leaks & Protuberances	0.001364
Cooling	0.001764
Engine	0.0004517
Total	0.02862

For comparisons sake, the wing parasite drag was recalculated assuming turbulent flow over 100% of the mean aerodynamic chord. The parasite drag of the wing at cruise increased to 0.012, thereby increasing the overall parasite drag to 0.0405, a relatively significant increase. It can be said that, should an NLF airfoil be used, it would be worthwhile to keep it clean in order to take advantage of its reduced drag capabilities.

4.4.2 Induced Drag

The induced drag C_{di} is the drag produced as a by-product of lift. This drag is highly dependent on the amount of lift produced. Because lift is dependent on the velocity at which the wing travels through the air, the induced drag also must be dependent on the velocity. Therefore, the induced drag may be expressed in terms of lift coefficient, C_L or velocity v. After determining the induced drag, this value may be combined with the parasite drag value to determine the drag polar of the aircraft. This drag polar may be expressed in terms of lift coefficient or velocity.

The induced drag was then calculated using Raymer's methods. Before the induced drag may be determined, the "drag-due-to-lift" factor had to be calculated, where K is:

$$K = \frac{1}{\pi A e}$$

The only unknown variable in finding K is the Oswald efficiency, e. Raymer presented a simple estimation method for the Oswald efficiency factor as a function of the wing aspect ratio A. For a straight wing aircraft:

$$e = 1.78(1 - 0.0454A^{0.68}) - 0.64$$

Using this equation, e was calculated to be 0.811, which is typical for a primary flight trainer. Once the Oswald efficiency factor was determined, the estimation for the induced drag was completed using the following equation:

$$C_{D_i} = K \cdot C_L^2$$

4.4.3 Final Drag Polar

Now that all the variables needed to determine the drag polars for the Viper were determined, the drag polars in terms of lift coefficient and velocity, respectively, are as follows:

$$C_d = 0.02862 + 0.04919 C_L^2$$

and

$$C_d = 0.02862 + \frac{7.103(10^6)}{V^4}$$

4.5 Performance

Once the drag polar equations were found, the horsepower versus velocity curves were established. Power available at sea level and cruise altitude were determined using the maximum brake horsepower and propeller efficiencies at various velocities. Horsepower required at sea level and cruise altitude was determined by first using the drag polar in terms of velocity to find the total drag of the aircraft. Then using the following relationship between drag and horsepower required, the power required at various velocities were calculated,

$$HP_{required} = \frac{D.V}{550}$$

to find the power required at various velocities. These relationships were then plotted against velocity, and several key performance velocities were read from the graph, refer to Appendix 7.1 for this graph. These velocities are summarized in Table 4.4.

Table 4.4 Key Performance Velocities

	Velocity (knots)
Vmax (Sea Level)	127
Vmax (5000 Ft.)	124
Vmin (Sea Level)	35
Best ROC	80
Best Range Velocity	72
Best Endurance Velocity	55

Finally, using these velocities as well as propeller efficiency and lift-to-drag relationships, the performance parameter for the aircraft were determined. These parameters are summarized in Table 4.5 below.

Table 4.5 Performance Parameters

Maximum ROC (Sea Level)	672 fpm
Maximum ROC (5000 Feet)	687 fpm
Best Climb Angle	6 degrees
Range (Cruise)	625 NM
Range (Best)	783 NM
Takeoff Distance	776 feet
Landing Distance	1458 feet

Note: These parameters were found using Raymer's equations, 17.28 through 17.107.

Finally, performance limitations at high angles of attack, specifically power required were investigated. This was due to the fact that the Natural Laminar Flow airfoil rapidly increases induced drag at high angles of attack since so much lift is being produced. However, this proved to not be a limitation, as there was sufficient power available at these angles of attack.

4.5.1 Load Requirements

The performance calculations were done which showed that the requirements for gust loads were met. The velocity versus load factor curves, at sea level and cruise altitude of 5000 feet are shown in Figures 4.3 and 4.4, respectively.

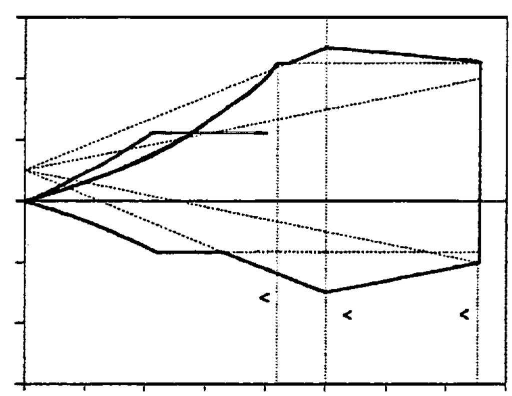


Figure 4.3 Sea Level

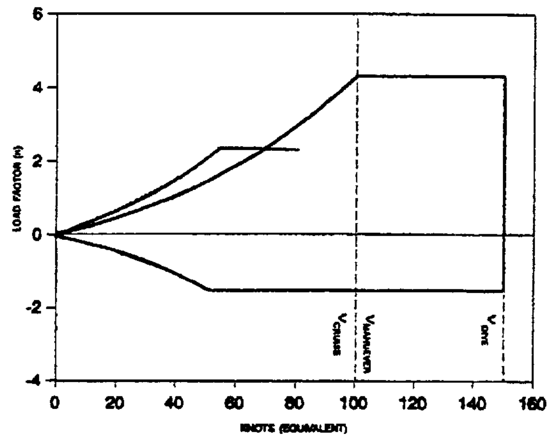


Figure 4.4 Cruise Altitude (5000 Feet)

4.6 Stability and Control

Stability and control calculations were performed using the most forward and aft center of gravity locations. Longitudinal, lateral and directional stability were evaluated as well as control surface deflections, cruise and low speed roll rates, and adequate spin recovery. After the first iteration, the horizontal tail area was increased from 22.01 square feet for longitudinal stability and the vertical tail area was increased from 9.61 square feet to 15.72 square feet for sufficient spin recovery. All calculations were adjusted using the enlarged areas and most recent configurations.

4.6.1 Longitudinal Stability

The first step in evaluating longitudinal stability was calculating the aircraft's neutral point. All distances were measured from the nose and power off conditions were assumed. Equation 16.19 (Raymer) is used.

$$\bar{X}_{np} = \frac{\dfrac{C_{L\alpha} X_{ac_w}}{\bar{c}_w} - C_{m\alpha_{fus}} + \eta_h C_{L\alpha_h} \dfrac{X_{ac_h}}{\bar{c}_w} \dfrac{S_h}{S_w} \dfrac{\partial\alpha_h}{\partial\alpha}}{C_{L\alpha} + \eta_h C_{L\alpha_h} \dfrac{S_h}{S_w} \dfrac{\partial\alpha_h}{\partial\alpha}}$$

Where:

$C_{L\alpha} := 0.098 \frac{1}{deg}$

$X_{acw} := 9.45 ft$

$Cm_{\alpha fus} := 0.016 \frac{1}{deg}$

$\eta_h := 0.90$

$S_h := 22.01 ft^2$

$S_w := 112.9 ft^2$

$C_{L\alpha h} := 0.10 \frac{1}{deg}$

$\frac{\delta\alpha_h}{\delta\alpha} := 0.59$

$X_{ach} := 21.98 ft$

m.a.c. = 3.928 ft

After the first iteration, the neutral point was 40.5% of the mean aerodynamic chord. However, in order to satisfy a static margin of 10% and a 4% correction for power on conditions, a neutral point located at 44.5% of the mac was required. For this reason, the horizontal tail area was increased to 24.86 square feet, which satisfied both conditions.

Next, the incidence of the wing and horizontal tail were computed at trim conditions. Using a downwash gradient of 0.41, a lift coefficient of 0.42, a zero-lift angle of attack of -4 degrees, and the three-dimensional lift curve slope of the main wing, the incidence was computed using equation 16.11 (Raymer):

$$i_w := \alpha_{oL} - \alpha + \frac{C_L}{C_{L\alpha}} = 0.281 \text{ degrees}$$

The incidence of the horizontal tail was computed using the derived equation below. All distances were measured with respect to fuselage station zero, which is located at the nose of the aircraft, and the most forward center of gravity limit was used.

$$i_h := \frac{C_{Lh}}{C_{L\alpha.ht}} - \left(\theta_{tipback} + i_w\right) \cdot \frac{\delta\varepsilon}{\delta\alpha} + \alpha_{oLh} + i_w$$

i_h = -2.80 degrees

Applying these values, the required elevator deflection under stall conditions (flaps down) was computed. Incidentally, the tip-back angle of 15.19 degrees was used as the critical stall angle, since it is less than the stall angle-of-attack which is 20.4

degrees. Solving equation 16.14 (Raymer) for the deflection of the elevator, a value of -17.9 degrees was found. Since this is less than 20 degrees, there is sufficient elevator.

4.6.2 Directional and Lateral Stability

The directional stability slope, C_{N_β} was found using equations 16.41, 16.47 and 16.36 (Raymer) for the wing, fuselage and vertical stabilizer, respectively. The results of these calculations are shown in Table 4.6 below.

Table 4.6 Directional Stability Parameters

Component	C_{N_β} (per degree)
Wing	0.002
Fuselage	-0.000796
Vertical Stabilizer	0.003

The total directional stability slope for the entire aircraft was computed using equation 16.39 (Raymer):

$$C_{N_\beta} = C_{N_{\beta w}} + C_{N_{\beta f}} + C_{N_{\beta vt}} = 0.004 \, / \, \text{degree}$$

This value is relatively high which can be attributed to the fact that the vertical tail area was increased to provide sufficient spin recovery. This will be discussed in detail later.

Rudder deflection necessary to maintain directional control in an 11.5 degree crosswind was calculated using equations 16.45 and 16.46 (Raymer). A value of 9.17 degrees was found, which is much less than the 20 degree maximum deflection designed into the rudder.

The lateral stability slope C_{l_β} was computed using figure 16.21 and equations 16.34 and 16.42 (Raymer) for the vertical stabilizer, wing/fuselage and wing dihedral respectively. The results are shown in Table 4.7 below.

Table 4.7 Lateral Stability Parameters

Component	C_{l_β} (per degree)
Wing/Fuselage	$-2.20\,(10^{-3})$
Vertical Stabilizer	-0.0125
Dihedral (1.6°)	$1.04(10^{-5})$

The total lateral stability of the aircraft was then found using equation 16.44 (Raymer):

$$C_{l_\beta} = C_{l_{\beta wf}} + C_{l_{\beta \Gamma}} + C_{l_{\beta vt}} = -0.002/\text{degree}$$

Although both the lateral and directional stability are relatively high, the calculated value of C_{l_β} is exactly half the value of C_{N_β}.

4.6.3 Roll Rates

In order to be considered a Class I aircraft under MIL-F-8785 B, the aircraft must have the ability to roll 60 degrees in 1.3 seconds (0.806 radians/second). Using cruise conditions, the deflection of the ailerons required to achieve this was computed to be 5.45 degrees. This is well below the 20 degree maximum deflection angle designed into the aileron.

The low speed roll rate was computed using an average of the maximum up and down deflection of the ailerons. Setting $\delta a_{max.}$ = 17.5 degrees, a roll rate of 0.23 radians/second and a minimum velocity of 52.56 feet/second was computed. Since the minimum velocity is less than the stall velocity, the aircraft has adequate aileron deflection even under stall conditions.

4.6.4 Spin Recovery

Spin recovery is the last stability and control calculation. Using figure 16.31 (Raymer), a minimum tail damping power factor (TDPF) of 2.0 (10^{-4}) was computed. After the first iteration, the vertical tail area was increased due to insufficient TDPF. Using the enlarged area, however, a value of $3.88(10^{-4})$ was computed. This nearly twice the minimum required TDPF which ensures good spin recovery.

5.0 Cost

The cost analysis for the aircraft was performed by using the "Litecost" and the "Liteops" programs. These two computer programs use statistical data to estimate both the sale price and the operating cost for a specific aircraft. The results of each program will be discussed separately in this section.

5.1 Litecost Analysis

The "Litecost" program used to determine the sale price of an aircraft, relies on 14 different parameters for a given aircraft, ranging from airframe weight to the number of aircraft produced. The calculations that had to be made before running the program were determining the airframe weight and the cost of the avionics to be used in the aircraft.

The airframe weight was calculated by summing the weights of the wing, horizontal and vertical tails, fuselage, furnishings, and landing gear (minus the weight of the wheels, tires and brakes). The airframe weight was found to be 452 pounds. The cost of the avionics used is summarized below in Table 5.1.

Table 5.1 Avionics Cost Summary

Component	Price (Dollars)
Terra TRT-250 Mode C Transponder	1095.00
Merl 79007-AP ELT	350.00
RST-571 Comm. Radio	800.00
Total	**$2245.00**

Once these calculations were performed, the program was used to find the cost of the aircraft for productions runs of 100, 500, 1000, and 5000 aircraft. The results are summarized in Table 5.2.

The cost goal of $50,000 for 500 aircraft was just exceeded, and could actually be met with a slight decrease (4 - 5 %) in profit.

Table 5.2 Litecost Program Summary

Aircraft Produced	Aircraft/ Month	Factory Cost	Sale Price
100	10	$101,412	$116,624
500	10	45,503	52,329
1000	10	33,390	38,398
5000	10	17,642	20,289

Note: 15% profit margin was used for these calculations.

5.2 Liteops Analysis

The operating cost for the Viper was found using the "Liteops" program. Again, this program requires inputting several values which are specific to a given aircraft. The key values that were varied from case to case include the yearly flight hours, loan value, and whether the aircraft is to be privately owned or company owned. The parameters that remained constant for each case included the following; a 12 % interest rate for a loan period of five years, a fuel cost of $2.00 per gallon, a labor cost of $40.00 per hour, and a storage fee of $25.00 per month (The last parameter corresponding to a tie-down storage arrangement). Table 5.3 summarizes the operating costs for various scenarios.

The results show that under company ownership, the operating cost per hour ranges from $23 to $53, with an average value of $38 per hour, depending on the flight hours per year and the value of the loan on the aircraft. The average value is similar to current aircraft rental rates, showing that the Viper could be a competitive entry into the primary flight trainer market.

6.0 Conclusions

By changing two major components--the engine and the airfoil--the baseline Viper became a somewhat different, but not necessarily improved aircraft. The Viper became smaller in most respects; the gross weight decreased by 16%, the wing planform area decreased by 10%, and the horizontal tail area decreased by 20%. However, the vertical tail area increased by 18% due to stability considerations.

These reductions in areas, combined with the reduced drag of the Natural Laminar Flow airfoil, decreased the overall drag of the aircraft, which should have increased the performance characteristics of the Viper. However, due to the reduced horsepower available from the Teledyne Continental engine, the performance remained about the same. Cruise velocity was lowered by 26%, which was expected at the outset due to the decrease in horsepower.

The maximum range for the Viper went up slightly by 4%, however, the maximum rate of climb at sea level was reduced by 70%, a rather large decrease which can be attributed to the decreased power available. The total takeoff distance was reduced by 19%, while the landing distance was increased by 30%. Finally, the sale price of the Viper for a production run of 500 aircraft went up by 14% over the previous design. This can be attributed to the fact that composite materials were used in the design. The average operating cost per hour remained the same.

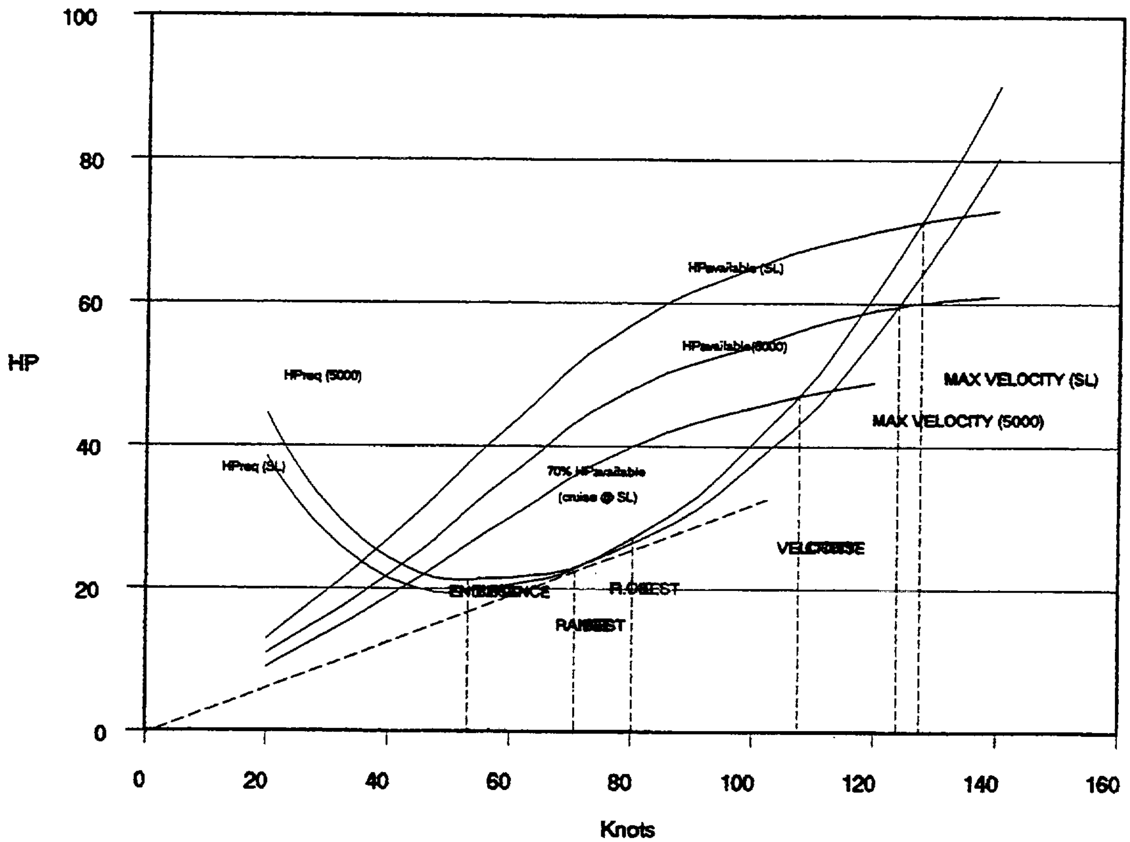
After examining the data, it is difficult to determine if the redesigned Viper is a "better" aircraft. Using the Natural Laminar Flow airfoil allowed a change to a smaller, lighter engine while keeping approximately the same performance characteristics. However, the added complexity and cost of utilizing the new airfoil negates any previous advantages. It has been shown that the redesigned configuration is possible. Table 6.1 summarizes the configuration changes.

Table 6.1 Comparison of Configuration Changes

		Baseline	Redesign
Areas	Wing	124.4	112.9
(sq. ft.)	Horizontal Stabilizer	30.0	24.9
	Vertical Stabilizer	12.9	15.7
	Elevator	9.0	5.5
	Aileron	12.9	10.5
	Rudder	3.87	5.9
Span	Wing	31.15	30.0
(ft.)	Horizontal Stabilizer	10.95	9.98
	Vertical Stabilizer	4.5	4.87
Airfoil		NACA 642-415	NLF-0414
Engine		118 HP Lycoming 0-235-L	85 HP TC GR-36
Performance	max.	140 knots	127 knots
	Cruise	136 knots	100 knots
	Stall Flaps	46.5 knots	46.5 knots
	Stall Clean	55 knots	56 knots
	ROC	1140 fpm	687 fpm
	Range @Cruise	689 Nm	625 Nm
	Takeoff Dist.	920 ft.	776 ft.
	Landing Dist.	1023 ft.	1458 ft.
Price		$45160	$52329

7.0 Appendix

7.1 Appendix 1: Horsepower Versus Velocity Graph



7.2 Appenix 2

GENERAL

NO. OF OCCUPANTS	2
STRUCTURAL MATERIAL	ALUMINUM
LANDING GEAR CONFIG.	TRICYCLE
CERTIF. CATEGORY	UTILITY

REVISIONS

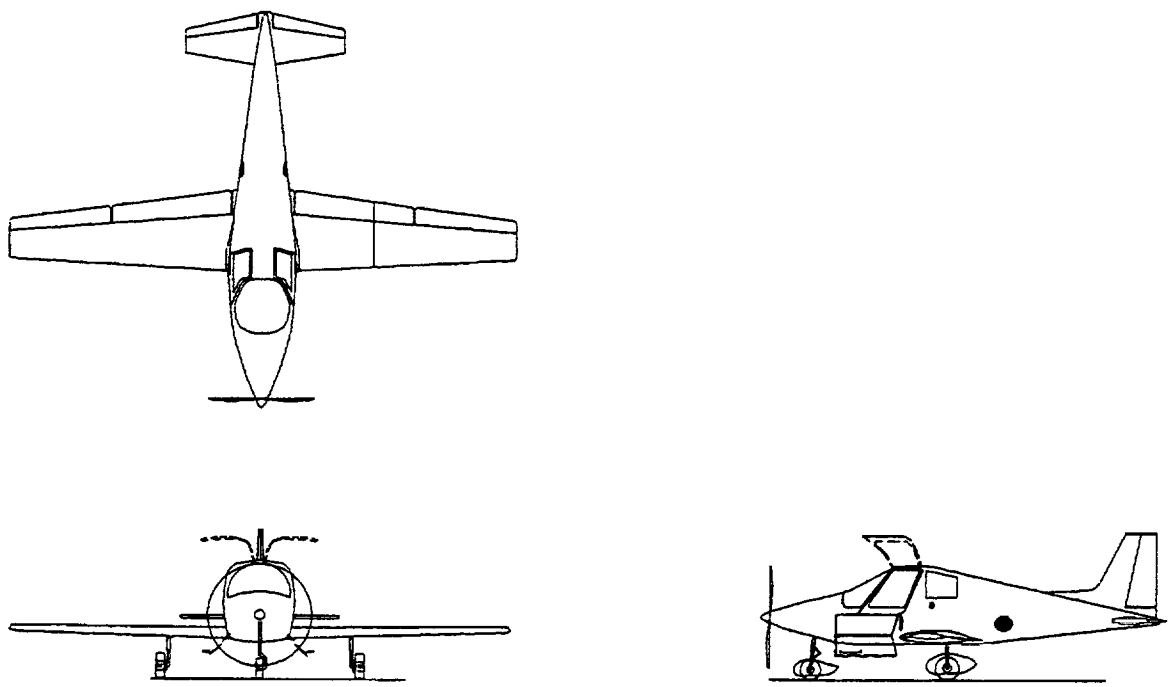
LTR	DESCRIPTION	APPROV.
	TC GR-36 ROTARY ENGINE	
	NLF-0414 AIRFOIL	

COST

ACQUISITION COST:	$ 50,084	(NO AV.)
AVIONICS COST:	$ 2250	
BASE YEAR		
QUANTITY PRODUCED:	500	
PROFIT (IN %)	20	

AIRCRAFT CONFIGURATION

OVERALL HEIGHT		106.8	IN.
OVERALL LENGTH		286.8	
WING SPAN		360.0	
HORIZ. TAIL SPAN		119.7	
VERT. FIN HEIGHT		58.4	
FUSELAGE WIDTH:	INT.	48.6	
	EXT.	50.4	
WHEEL BASE		96.5	
WHEEL TRACK		141.4	

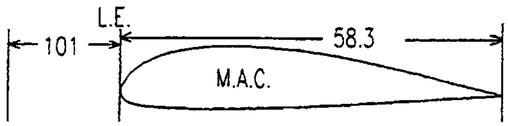


EMBRY-RIDDLE AERONAUTICAL UNIVERSITY			
DRAWN BY: JRM	SUMMARY OF GENERAL DATA		
	AIRCRAFT: VIPER		
DATE: 12-8	A	DRAWING NO.: F93-2B	SHEET: 1 OF 2

7.3 Appendix 3

WING PARAMETERS

HORIZONTAL DATUM (STA 0.0)



ROOT CHORD	58.27	IN.
TIP CHORD	27.96	
SWEEP:	LE 3 .25c̄ 0	DEG.
DIHEDRAL:	1.6	DEG.
A.C. LOCATION:	STA 113 WL 79	

VELOCITIES

DESIGN SPEED	V(KTS)	q(PSF)
CRUISING (Vc)	100	33.9
DIVE (Vd)	150	76.33
MANEUVERING (Va)	85	24.5
FLAP (V_f)	80	21.71
NEVER EXCEED (Vne)	150	76.32
STALL-FLAP UP (Vs)	56	10.64
STALL-FLAP DN (Vsf)	47	7.49
ECONO. CRUISING	55	10.26
BEST R.O.C.	80	21.71

AREAS

WING (INCL. AREA THRU FUS.)	112.87	FT^2
FLAPS	17.5	
AILERONS	10.5	
HORIZONTAL TAIL		
STABILIZER	24.9	
ELEVATOR (INCL. TAB)	5.5	
ELEVATOR TAB	N/A	
VERTICAL TAIL		
FIN	15.8	
RUDDER	5.9	
RUDDER TAB	N/A	

PERFORMANCE

CRUISE ALTITUDE	5000	FT
MAX. RATE OF CLIMB (FT/MIN)	687	
MAX. RANGE (N. MI.)	783	
RANGE AT DSN. CRUISE VEL.	625	
TAKE-OFF DISTANCE (G.T.O.W.)	776	FT
LANDING DISTANCE (G.W.)	1458	FT

CONTROL SURFACE MOVEMENTS

(+OR- 2 DEG.)	UP	DOWN
AILERONS	20	15
FLAPS	N/A	20
ELEVATORS	20	20
ELEVATOR TAB	N/A	N/A
RUDDER (PROJ. TO W.L.)	20 L	R 20

AERODYNAMIC CHARACTERISTICS

AIRFOILS:		
WING ROOT/TIP	NLF-0414 NLF-0414	
HORIZONTAL TAIL	NACA 0012	
VERTICAL TAIL	NACA 0012	
ASPECT RATIO:		
WING	7.98	
HORIZONTAL TAIL	4.00	
VERTICAL TAIL	1.57	
ANGLE OF INCIDENCE:		
WING ROOT/TIP	0 / 0	DEG.
STABILIZER	-2.8	DEG.
HORIZ. TAIL A.C. LOCATION:	STA 0.25 c̄	
AIRCRAFT EFF. FACTOR (e)	0.811	
ZERO-LIFT DRAG COEFF. (C_{d0})	0.0285	

WEIGHTS

DESIGN GROSS	1382	LB
EMPTY:	822	
EST. MIN. FLY. WT.	942	

C.G. POSITIONS

	%MAC	STA
GROSS WT.	25	113
MIN. FLY WT.	27	114
EMPTY WT.	31	116
MOST FORWARD	12	107
MOST AFT	33	117

CRITICAL LIMIT LOAD FACTORS

CONDITION	n	WEIGHT
FLAPS UP		
MANEUVERING	+4.4	1382 LB
	- 1.76	1382
GUST	+ 5.0	921
	- 3.0	921
LANDING	2.0	1382
FLAPS DN. (DEG.)		
MANEUVERING	+ 2.2	1382
GUST	- 3.0	1382

PROPULSION SYSTEM

ENGINE	TCM GR-36 ROTARY ENG
SFC	0.54 LB/HP/HR
RATED POWER	85 hp AT 7000 RPM
FUEL CAPACITY	30 GAL. 180 LB
PROPELLER SIZE	74 X 70
PROPELLER EFFICIENCY:	84%

MISCELLANEOUS

DATE:	DRAWING NO.:	TITLE:	SHEET:
12-8	F93 - 2B	SUMMARY OF GENERAL DATA	2 OF 2