NASA-CR-195484

NASW-4435

IN-05

204233

P-47

DESIGN PROJECT

VIPER

AE 420 - 02

TEAM 2
FALL 1992

DESIGN LEADER:

STEPHEN HAITHCOCK

TEAM MEMBERS:

KYLE KONCAK
RICH NEUFANG
DAVID PAUFLER
RUSS SNOW
FRANK WLAD

(NASA-CR-195484) DESIGN PROJECT: VIPER (Embry-Riddle Aeronautical Univ.) 47 p

N94-25021

Unclas

G3/05 0204233

TABLE OF CONTENTS

DESIGN SPECIFICATIONS

The primary flight trainer:

1. Must conform to F.A.R. 23, including the crashworthiness standards.
2. Is limited to two-four occupants.
3. Engine must be FAA certified.
4. Must comply with FAA standards for VFR and allow for upgrade to IFR flights.
5. Must be at least utility category with good spin recovery characteristics.
6. Must have a structural lifetime of at least 10,000 flight hours.
7. Capable of either of two training missions:
    a. Climb to 5,000 ft., cruise 500 Nm. plus reserve, land.
    b. Climb to 1,000 ft. and descend ten cycles for landing practice, climb to 3,000 ft., maneuver at 2 g's for 15 min., cruise 100 Nm. and land.
8. Must have a cruise speed of at least 120 knots.
9. Must take-off or land on a runway no longer than 3,000 ft.
10. Has a cost goal of $50,000; not including avionics, for production of 1,000 airplanes over a five year period.

SUMMARY STATEMENT

A few changes in the configuration of the aircraft were made in order to meet all of the design specifications. The flap and aileron were moved so that they deflected along the same hinge-line for ease of construction. The horizontal tail was changed from a rectangular to a tapered planform for better spin recovery and for cosmetic reasons. The wheel fairings and the fuselage width at the nose and cockpit sections were slimmed drastically to reduce the drag coefficient of the aircraft in order to meet the optimistic 120 knot cruising speed. The original horizontal tail area was 28.1 ft^2. and X_{acht} was 3.0 ft., but this yielded an unacceptable static margin. The horizontal tail was moved 16 in aft, and the area was increased to 30 ft^2. and the aspect ratio was increased to 4.0. The dihedral was lowered from 7 degrees to 1.6 degrees so that lateral stability $C_{l\beta}$ was close too a -1/2 $C_{N\beta}$.

Changes that may be considered in order to improve the general design beyond meeting the required specifications, include using an NLF-0014 airfoil, installing a rotary engine, and opting to add another passenger seat to accommodate the Gemini Flight Training Program.

SIZING ITERATIONS

The purpose of this task is to conduct a refined sizing analysis of the initial conceptual design. Fixed engine sizing is used to meet certain performance requirements with mission range as the fallout parameter. A new empty-weight fraction is estimated using design variables such as aspect ratio, horse-power-to-weight ratio, wing loading, and maximum speed.

The fuel-weight fraction is determined directly from mission segments, and an iterative process is then used to find the total weight estimate. The required horsepower at cruise is calculated and compared to the available horsepower at cruising altitude. Once the takeoff gross weight has been estimated, the fuselage, wing, and tails can be dimensioned using historical data of single engine aircraft. The primary control surfaces are initially sized in relation to the wing and tail section dimensions in accordance to historical data.

INITIAL VALUES:

Vcr = 120 Kts	$\eta_p)_{loiter}$ = .7
Vst = 46.5 Kts	e = .8
HP/Wo = .0375	C_{Do} = .03
L/D)max = 12.5	A = 7.8
L/D)cr = 8.89	$\eta_p)_{cruise}$ = .8
L/D)loiter = 11.24	Wo = 1565 lb
$C_{BHP loiter}$ = .5 /hr	W_{crew} = 2(190) = 380 lb
$C_{BHP cruise}$ = .4 /hr	
Loiter = 45 minutes	
Range = 500 Nm	

NOTE: All equations are referenced from <u>Aircraft Design: A Conceptual Approach</u>, Raymer.

EMPTY WEIGHT FRACTION

$$\frac{W_e}{W_o} = -.25 + W_o A^{.08} \left(\frac{HP}{W_o}\right)^{.05} \left(\frac{W_o}{S}\right)^{-.05} V_{max}^{.27} = .6459$$

FUEL FRACTION:

engine start, taxi, take off:

$$\frac{W_1}{W_o} = .99 \qquad \text{Eq. 6.8}$$

climb, accelerate:

$$\frac{W_2}{W_1} = \frac{1.0065 - .0325 M_{CR}}{1.0065 - .0325 M_{takeoff}} = .9966$$

cruise:

$$\frac{W_3}{W_2} = e^{-\left[\frac{RC_{BHP}}{550\eta\left(\frac{L}{D}\right)_{CR}}\right]} = .9173 \qquad \text{Eq 6.12}$$

loiter:

$$\frac{W_4}{W_3} = e^{-\left[\frac{RC_{BHP}}{550\eta_p\left(\frac{L}{D}\right)}\right]} = .9862 \qquad \text{Eq 6.15}$$

descent:

$$\frac{W_5}{W_4} = .995 \qquad \text{Eq.6.22}$$

landing and taxi:

$$\frac{W_7}{W_6} = .997 \qquad \text{Eq 6.23}$$

TOTAL:

$$\frac{W_6}{W_o}=\frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{W_3}\frac{W_5}{W_4}\frac{W_6}{W_5}=.8854$$

TOTAL FUEL WEIGHT FRACTION: (with 6% unusable fuel)

$$\frac{W_f}{W_o}=1.06\left(1-\frac{W_6}{W_0}\right)=.1215 \qquad \text{Eq 6.2}$$

TOTAL WEIGHT ESTIMATE:

NOTE: sizing iterations were used in equation below:

$$W_o=\frac{W_{crew}+W_{payload}}{1-\left(\frac{W_f}{W_o}\right)-\left(\frac{W_e}{W_o}\right)}=16421\,lbs$$

$$\frac{HP}{W_o}=.0719$$

REQ'D THRUST/WEIGHT VERIFICATION:

97 HP available at 5000 FT for Lycoming 0-235

$$(HP_{CR})_{req'd}=HP_{takeoff}\frac{W_{CR}}{W_{takeoff}}\frac{1}{\left(\frac{L}{D}\right)_{CR}}\frac{V_{CR}}{550\eta_p}\frac{1}{\left(\frac{HP}{W}\right)_{takeoff}}=83.91HP$$

TOTAL FUEL REQ'D:

$$W_f=\frac{W_f}{W_o}W_o=199.5\,lb$$

MISSION CHECK: [takeoff, climb to 1000ft, descend, land] x 10

-OR- Takeoff, climb to 3000ft, maneuver @ 2g's for 30 min, cruise 100 nm, land.

start, taxi, takeoff:

$$\frac{W_1}{W_o}=.99 \qquad \text{Eq 6.8}$$

climb to 1000ft:

$$\left(\frac{W_2}{W_1}\right)_i = \frac{1.0065 - .0325 M_{END}}{1.0065 - .0325 M_{BEGIN}} = .9981$$

$$\frac{W_2}{W_1} = \left(\frac{W_2}{W_1}\right)_i^{11} = .9793 \qquad \text{Eq 6.9}$$

final cycle climb to 3000ft:(1000 to 3000ft)

$$\frac{W_3}{W_2} = \frac{1.0065 - .0325 M_{end}}{1.0065 - .0325 M_{begin}} = .9995$$

Maneuver at 2g's for 30 min:

$$\frac{W_4}{W_3} = 1 - C\left(\frac{T}{W}\right) d = .9639 \qquad \text{Eq 6.16}$$

cruise for 100nm:

$$\frac{W_5}{W_4} = e^{-\left[\frac{RC}{550\eta_p \left(\frac{L}{D}\right)_{CR}}\right]} = .9813$$

descend:

$$\frac{W_6}{W_5} = .995 \qquad \text{Eq 6.22}$$

land, taxi:

$$\frac{W_7}{W_6} = .997 \qquad \text{Eq 6.23}$$

TOTAL FUEL FRACTION FOR MISSION:

$$\frac{W_7}{W_o}=\frac{W_1}{W_o}\frac{W_2}{W_1}\frac{W_3}{W_2}\frac{W_4}{w_3}\frac{W_5}{W_4}\frac{W_6}{W_5}\frac{W_7}{W_6}$$

$$\frac{W_f}{W_o}=1.06\left(1-\frac{W_7}{W_o}\right)=.09614$$

$$W_f = 157.9 \text{ lb}$$

WING DIMENSIONS:

$$s=\frac{W_o}{\frac{W_c}{s}}=124.4\,ft^2$$

$$A=\frac{b^2}{S}\rightarrow b=31.5\,ft$$

fuselage length:

$$L_{fuselage}=AW_o C\rightarrow (TABLE6.3\,Raymer)\rightarrow L_{fuselage}=24\,ft$$

vertical tail:

$$S_{VT}=\frac{C_{VT}B_w S_w}{L_{VT}}=12.92\,ft^2$$

horizontal tail:

$$S_{HT}=\frac{C_{HT}mac_{wing}S_w}{L_{HT}}=28.07\,ft$$

CONTROL SURFACE SIZING:

vertical tail:

predetermined values: H = 4.5 ft
C_r = 4 ft
$\Lambda_{c/4}$ = 0°

$$S_{VT}=\frac{1}{2}(C_r+C_t)H \rightarrow C_t=1.75ft$$

$$\lambda_{VT}=\frac{C_t}{C_r}=.4375$$

rudder:

assume: H = 3.5 ft
C_r = 1.5 ft

$$S_{rudder}=.3S_{VT}=3.9ft$$

$$S_R=\frac{1}{2}(C_r+C_2)H \rightarrow C_{t_{rudder}}=.75ft$$

NOTE: The following planforms wee determined using semi-spans.

Horizontal tail:

predetermined values: S_{HT} = 14.05 ft
C_r = 4 ft
B = 5.5 ft
$\Lambda_{c/4} = 0°$

$$S_{HT}=\frac{1}{2}(C_r+C_t)B \rightarrow C_t=1.75ft$$

$$\lambda_{HT}=\frac{C_t}{C_r}=.4375$$

elevator:

assume: C_r = 1.33 ft
B = 4.8 ft

$$S_{elev}=.3S_{HT}=4.7ft$$

$$S_{elev}=\frac{1}{2}(C_r+C_t)B \rightarrow C_t=.6ft$$

ailerons:

Typically extend from .5->.9 semi-span, assume hinge at .25c.

Using geometry and the locations along the semi-span, the following were determined:

C_t = .666 ft
C_r = 1.1 ft
b/2 = 6.23 ft

location is .5b/2 -> .9b/2

flaps:

Typically extend from fuselage to ailerons (.5b/2), assume hinge line at .25c. The fuselage width = 48 in., therefore the flaps extend from 24 in. (.128b/2) to the ailerons. Using geometry and the above information, the following were determined:

C_t = 1.1 ft
C_r = 1.5 ft
b/2 = 5.8 ft

location is .128b/2 -> .5b/2

CONCLUSION:

All of the design criteria have been met and are summarized in the following table:

REQUIREMENT	RESULT	COMMENTS
CLIMB 1000 FT, DESCEND 10 CYCLES (LANDING PRACTICE) THEN CLIMB TO 3000FT	REQUIREMENT MET	33.3 GALLONS OF FUEL REQUIRED
MANUEVER AT 2G FOR 15 MIN, CRUISE 100nMI, LAND	REQUIREMENT EXCEEDED	MANUEVERED AT 2G FOR 30 MIN WITH 26.3 GAL OF FUEL
$(HP/W)_{REQD\ CR}$ < $(HP/W)_{AVAIL\ CR}$	REQUIREMENT EXCEEDED	97 HP AVAIL, ONLY 84 HP REQ'D (15% SURPLUS POWER)

As seen in the above chart, the landing practice mission is the limmiting factor. The required fuel for the landing practice mission is 33.3 gallons of fuel, which is well below the fuel capacity of other similar aircraft. This would make the aircraft lighter and more cost effective.

The sizing iterations produced an estimated gross takeoff weight of 1642 lbs, which is only a 4.6% increase from the initial gross takeoff weight estimate. The historical data used to dimension the fuselage, wing, and tails also produced reasonable numbers typical of other single engine primary flight trainers.

The vetical tail was sized using the quarter chord sweep of zero degrees and a height of 4.5 feet in an attempt to avoid the blanketing effect of the horizontal tail. The rudder should provide moderate control and good stability being 30% S_{VT}.

The horizontal tail's tapered planform with zero sweep at the quarter chord was chosen to be asthetically pleasing as well as produce good stability characteristics. The elevator area is 30% S_{HT} and should provide moderate control.

The flaps and ailerons were estimated using typical values from other aircraft. The flaps extend from the fuselage to 50% semi-span, and the ailerons extend from the flaps to 90% semi-span. The hinge line for both the flaps and the ailerons is located at the rear spar (75%c) to minimize required structure and for ease of construction.

ENGINE AND FUEL TANK INSTALLATION

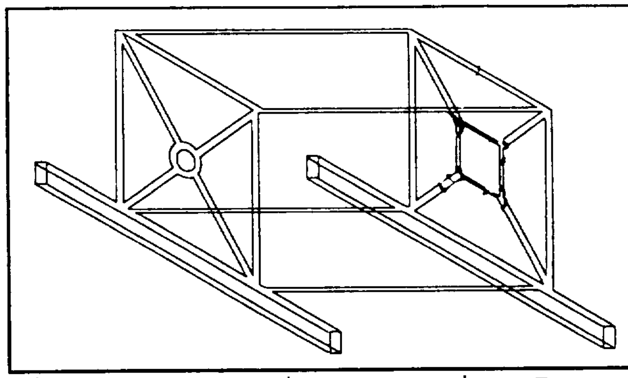


Figure 6.1 Engine Mounting Box

The Lycoming O-235 engine is designed for rear engine mounts which attach to the firewall. Because of the mid-engine design, a mounting box is used to support the engine and transfer it's loads into the aircraft structure. The baggage compartment above the engine box is removable to allow for quick engine extraction. As shown in figure 6.1, the engine box attaches directly to the wing spars and mounts to the front and rear of the engine.

The air inlets for engine cooling are positioned before and above the leading edge of the fuseluge. This placement will prevent turbulent airflow from entering the inlets.

Capture Area:

For adequate engine cooling, a capture area of 30% of engine frontal area is used.

$$A_{EF}=18.75\,inches \times 23.47\,inches \qquad A_{EF}=440\,inches^2$$

$$A_C=.3 \times A_{EF} \qquad A_C=132\,inches^2$$

Inlet ducts are used on both sides.

$$A_{C_{\frac{1}{2}}}=66\,inches^2$$

The opening of the inlets are inset into the lower step door. The maximum hieght is 12 inches.

$$t=\frac{A_{C_{\frac{1}{2}}}}{h} \qquad t=5.5\,inches$$

The airflow for the carburator will be seperated from theses inlets prior of engine heating. To insure an even distribution of cool air, the left inlet will cool the front two clylinder while the right side cools the rear. Secondary cooling consist of window vents on both sides of the plane.

Fuel Tank Approximation:

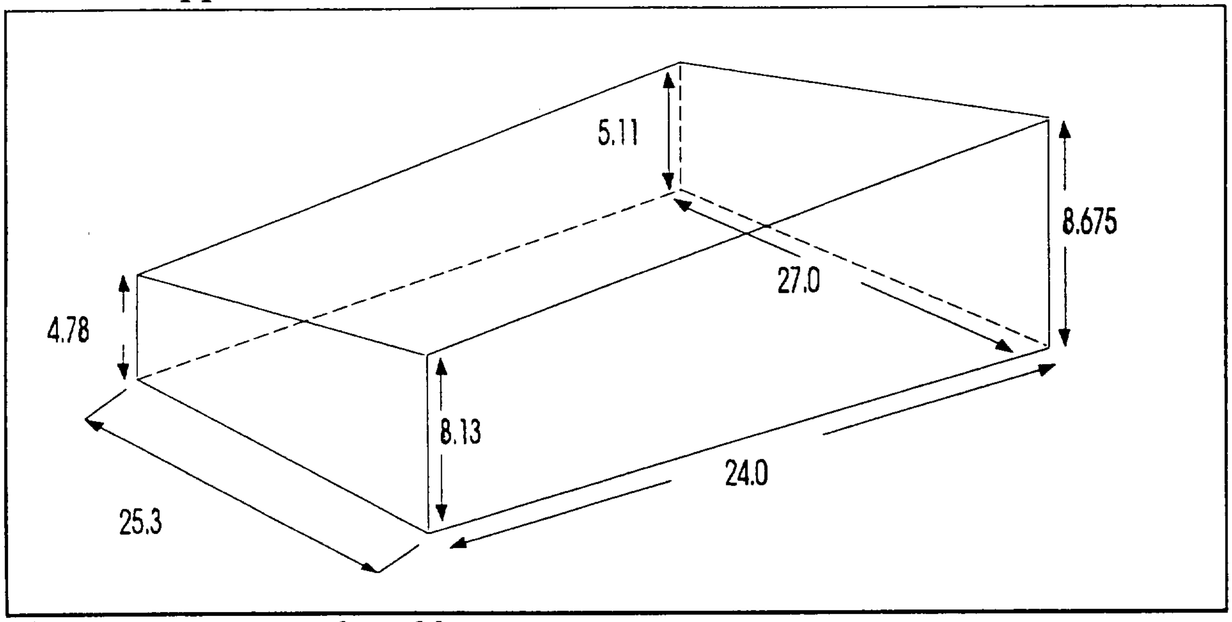


Figure 6.2 Fuel Cell

Volume = 4189 inches^3
Volume = 18.14 Gallons
Total Volume = 36.28 Gallons

Prop Calculation:

Choose standard two bladed prop with length of 74 inches, because of availablity. Checking prop under cruise and take-off conditions.

Cruise

$$V_{TIP} = 3.1415 \ x \ n \ x \ d = 3.1415 \ x \ 2400 \ x \ 74 = 774.9 \frac{ft}{s}$$

$$V_{TIP_H} = \sqrt{V_{TIP}^2 + V_{CRUISE}^2} = 801.0 \frac{ft}{s}$$

801.0 ft/s < 900 ft/s which meets standards set by Raymer.

Take-off

$$V_{TIP_H} = 843.9 \frac{ft}{s}$$

843.9 ft/s < than typical values of aircraft in this class.

Far Field Noise

FAR 36.301 states:

$$dBA < 68 + \frac{W_G - 1320}{165} \quad \therefore \quad dBA < 70\,dB$$

The calculated level is 66.7dB which is less than 70.0dB.

Pitch Angle and Prop Efficiency

Cruise

Using figure 6.5 from Raymer.

$$J = \frac{V}{nD} \quad \therefore \quad J = .822$$

$$C_P = \frac{P}{\rho n^3 D^5} \quad \therefore \quad C_P = .0555$$

Therefore Pitch Angle = 21.5 deg. and η = 86.3%

Manuever

η=83.0%

Stall Speed

η=52%

The propeller selected is optimized for high speed operations because the majority of the mission requirement manuevers are at cruise conditions.

WING DESIGN

Aspect ratio of 7.8 is selected. The initial aircraft concept had an aspect ratio of 8. Table 4.1 from Raymers yields a value of 7.6 for a single engine general aviation aircraft. 7.8 is the average.

The swept at the quarter chord is 0, while the taper ratio is .45. The combination of these two values will give a lift distribution closest to the ideal elliptical lift distribution; which produces the lowest airfoil drag.

The dihedral selected is 1.6 degrees. This value should have adequate stability and prevent dutch rolling. To insure that a stall will begin at the root, a geometric twist of 3 degrees is used. Geometric is choosen instead of aerodynamic twist to keep production cost low and to lessen difficulty.

This aircraft is designed for the training envirnoment, a low stall speed is ideal. A stall velocity of 46.5 knots is selected.

A coefficient of lift with flaps of 1.8 is needed for thr mission requirements. Slotted flaps were choosen for their increased efficiency and low production cost. Figure 5.3 from Raymer verifies this choice with a good margin. The NACA 65_2-415 is used to provide the desired performance and because it has been proven on the Piper Cherokee. This airfoil has a gradual stall, which is a must for the trainer aircraft.

Theory of wing sections gives the following information for the NACA 65_2-425:

$$C_{l_{max_{clean}}} = 1.32 \qquad \alpha_{0_{clean}} = -2.7\ degrees$$

$$C_{l_{flaps}} = 2.2 \qquad \alpha_{0_{flap}} = -13.5\ degrees$$

Known Values:

$A = 7.8$	$b = 31.15$ ft
$M_{stall} = .07035$	$d = 4.16$ ft
$S_{ref} = 124.4$ ft2	$\eta = .95$
$C_r = 5.50$ ft	$\Lambda_{c/4} = 0$ degrees
$C_t = 2.48$ ft	$\lambda = .45$

In order to show a graphical representation of the $wing_{clean}$ and $wing_{flap}$ (C_L Vs. α), the following values are found.

$$C_{L_{MAX_{CLEAN}}} = .9\ x\ c_{l_{max}}\ x\ \cos\Lambda_{\frac{c}{4}} \quad \therefore \quad C_{L_{MAX_{CLEAN}}} = 1.305$$

$$\beta = 1 - M_{stall}^2 \quad \therefore \quad \beta = .9950$$

$$F = 1.07\left(1 + \frac{d}{b}\right)^2 \quad \therefore \quad F = 1.375$$

$$C_{L_\alpha} = \frac{2 \times 3.1415\, A}{2 + \sqrt{4 + \frac{A^2\beta^2}{\eta^2}\left(1 + \frac{\tan^2\Lambda_{\frac{c}{4}}}{B^2}\right)}} \frac{S_{exp}}{S_{ref}} F$$

$$C_{L_\alpha} = 5.15/radian = .08988/^\circ$$

With a value for Λ_{LE} of 2.78 degrees and Δy of 2.9, figure 12.10 from Raymers gives a value of 1.1 so:

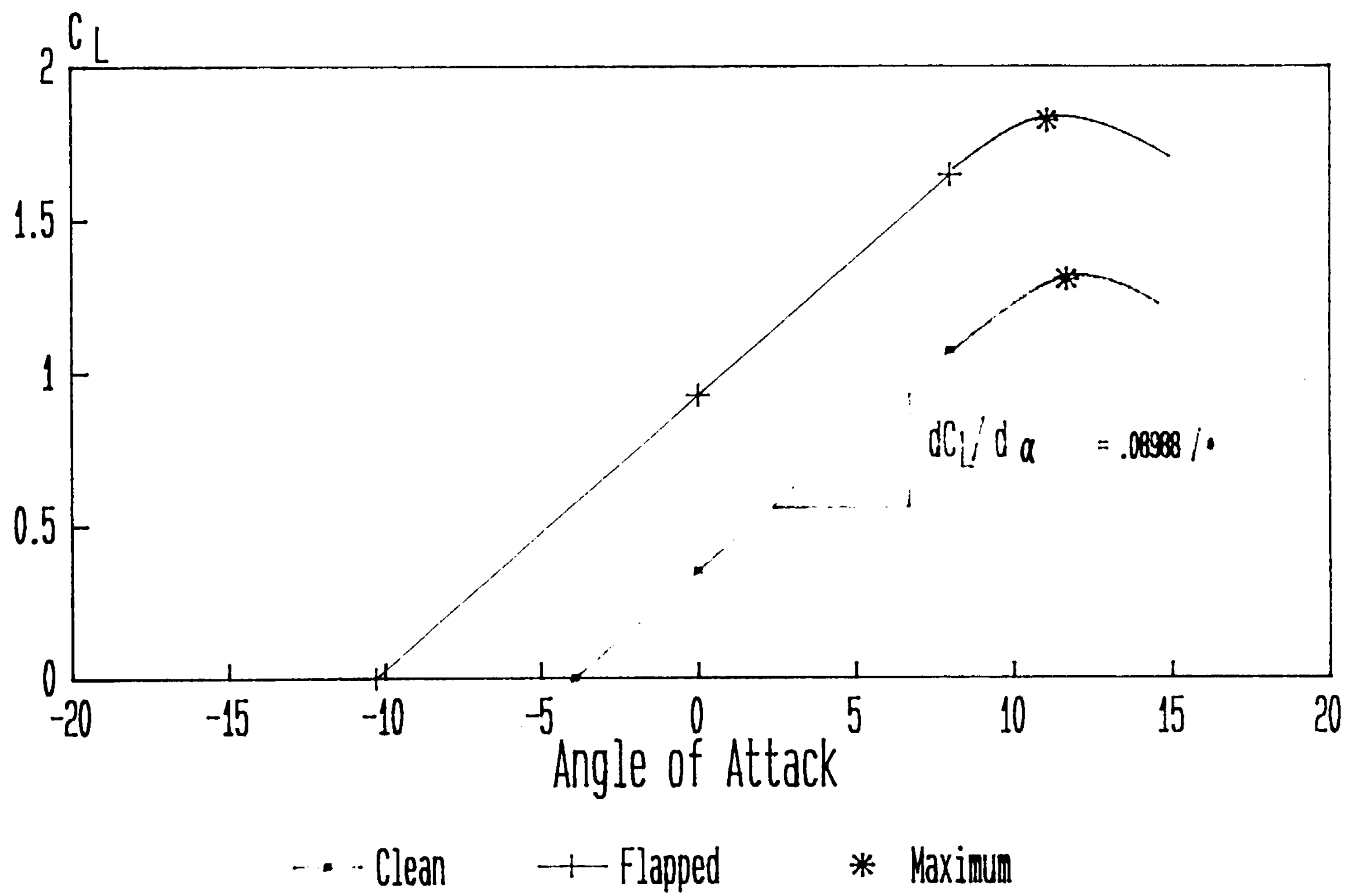
$$\Delta\alpha_{C_{L_{MAX}}} = 1.1\,degrees$$

$$\alpha_{C_{L_{clean}}} = \frac{C_{L_{MAX}}}{C_{L_\alpha}} + \alpha_0 + \Delta\alpha_{C_{L_{MAX}}} = 12.92\,degrees$$

$$\Delta C_{L_{MAX}} = \Delta C_{l_{max}}\left(\frac{S_{flapped}}{S_{ref}}\right)\cos\Lambda \quad \therefore \quad \Delta C_{L_{MAX}} = .9194$$

$$\Delta\alpha_{0L} = \Delta\alpha_{0L_{airfoil}}\left(\frac{S_{flapped}}{S_{ref}}\right)\cos\Lambda = -13.311\,degrees$$

C_L vs. Angle of Attack



LANDING GEAR LAYOUT

Landing Gear Geometry

The landing gear is in a tricycle configuration to enable the student pilot to control the aircraft easier while it is on the ground. The static tail down angle of 12.74 degrees gives sufficient clearance from tail strike and the overturn angle of 43.82 degrees ensures the aircraft won't tip over when making sharp turns of the runway. The roll/wing tip clearance was measured to be 14 degrees which provides sufficent clearance from wing tip strike.

Tire Sizing

The diameter and the width of the main wheels and the nose wheel were calculated using the following:

$$Width = A\,W_{per}^{B}$$

$$Dia = AW_{per}^{B}$$

From Table 11.1

Where A and B are values found in Raymer on page 231 on table 11.1 and W being the percent of the aircraft weight that each particular wheel experiences. The main gear takes 90% of the weight whereas the nose wheel takes 10%. The width and diameter were found to be:

	DIA	WID
NOSE WHEEL	10.6in	3.93in
MAIN WHEEL	15.1in	5.61in

These values may be increased by 30% if the aircraft is to operate from rough unpaved runways.

Static Loads

Static loads were calculated to be 750.8 LBS on each of the main gear and 261.6 LBS on the nose gear. These were calculated using the following :

$$LOAD_{main} = W\frac{N_{aft}}{B} \qquad \text{EQ(11.1)}$$

$$LOAD_{nose} = W\frac{M_{fwd}}{B} \qquad \text{EQ(11.2)}$$

Where N and M are distances from the nose to the aft CG and from the Main gear to the forward CGand B is the distance between the nose wheel and the main gear.

A check for load during dynamic breaking was done on the nose and it was found to be 208.7LBS. The nose was also checked if it would experience too much or too little of the weight of the aircraft. The result was a range from 8.6% - 15.9% of the weight and that was found to be within acceptable limits.

Tires to Accommodate Static Loads

Appropriate tire sizes of 5.00 - 5 and 6.00 - 6 were made for the nose and main gear based on similar aircraft tires, although the estimations calculated from the equations given by Raymer pg. 231 table 11 1 suggest smaller tires may actually be used.

Braking Requirements

The brakes will absorb the kinetic energy of the aircraft at touchdown, less the energy absorbed by aerodynamic drag.

$$KE = \frac{1}{2}\frac{W_{land}}{g}V_{stall}^2 \qquad \text{EQ(11.7)}$$

KE was calculated to be 2.76 ft LBS on each main wheel which is a relatively small kinetic energy so the load requirements will be the determining factor in the diameter of the tires.

Stroke Determination

The required deflection of the shock absorbing system depends on the vertical velocity at touchdown, the shock absorbing material, and the amount of wing lift still available after touchdown.

Using the following equation:

$$S=\frac{V_{vertical}^2}{2g\eta N_{gear}}-\frac{\eta_t}{\eta}S_t \qquad \text{EQ(11.12)}$$

S was calculated to be 7.64 in and was increased to 8.64 to keep a margin of safety.

STRUCTURAL CONCEPT

The design structure was covered simplistically in drawing AE 420-92-08. The following are shown:

- Fuselage bending structure
- Reinforcement around doors and windows
- Fuselage cross-section of floor support and seat attachments
- Wing spars
- Spar attachment structure
- Gear attachment points
- Tail spars.

INBOARD PROFILE

Drawing AE 420-92-09 details and shows workability of the interior design. The pilot's controls consist of a conventional yolk and rudder pedals with toe brakes. The pilot will have 52 degree line of sight over the aircraft nose. Moveable control surfaces will have control lines running under the drive shaft. The throttle control lines will run along the drive shaft top. The horizontal and vertical tails will be 30.0 and 12.9 square feet respectively. A storage compartment is seen above the engine and will house the battery and space for 40 pounds of baggage. Space for 221.3 pounds of fuel is located int the wing structure.

The pilot will enter the aircraft through gull wing door. Maintenance access to the engine will be obtained by removing the horseshoe shaped canopy, including baggage compartment, above it. One should also note the CG range on the MAC from 19.6% to 34.1% of the MAC.

WEIGHT AND BALANCE

This aircraft's weight and balance starts with a detailed weight analysis. Raymer provides statistical weight equations 15.46 to 15.57 to compute component weights, listed respectively. A load factor of 4.4 was used where necessary in order to satisfy FAR 23 for utility aircraft.

$$W_{wing}=0.036S_w^{0.758}W_{fw}^{0.0035}\left(\frac{A}{\cos^2\Lambda}\right)^{0.6}Q^{0.006}\lambda^{0.04}\left(\frac{100\frac{t}{c}}{\cos\Lambda}\right)^{-0.3}(N_z W_{dg})^{0.49}$$

$$W_{horizontal_{tail}}=0.016\,(N_\zeta W_{dg})^{0.414}Q^{0.168}S_{ht}^{0.896}\left(\frac{100\frac{t}{c}}{\cos\Lambda_{ht}}\right)^{-0.12}\times\left(\frac{A}{\cos^2\Lambda_{ht}}\right)^{0.043}\lambda_h^{-0.02}$$

$$W_{vertical_{tail}}=0.073\,(1+0.2\frac{H_t}{H_v})\,(N_\zeta W_{dg})^{0.376}Q^{0.122}S_{vt}^{0.873}\left(\frac{100\frac{t}{c}}{\cos\Lambda_{vt}}\right)^{-0.49}\left(\frac{A}{\cos^2\Lambda_{vt}}\right)^{0}$$

$$W_{fuselage}=0.052S_f^{1.086}\,(N_\zeta W_{dg})^{0.177}L_t^{-0.051}\left(\frac{L}{D}\right)^{-0.072}Q^{0.241}+W_{press}$$

$$W_{mainlanding_{gear}}=0.095\,(N_l W_l)^{0.768}\left(\frac{L_m}{12}\right)^{0.409}$$

$$W_{noselanding_{gear}}=0.125\,(N_l W_l)^{0.566}\left(\frac{L_n}{12}\right)^{0.845}$$

$$W_{installed\ engine_{total}}=2.575W_{en}^{0.922}N_{en}$$

$$W_{fuel\ system}=2.49V_t^{0.726}\left(\frac{1}{1+\frac{V_i}{V_t}}\right)^{0.363}N_t^{0.242}N_{en}^{0.157}$$

$$W_{flight_{controls}}=0.053L^{1.536}B_w^{0.371}\,(N_\zeta W_{dg}\times10^{-4})^{0.80}$$

$$W_{hydraulics}=0.0001\ W_{dg}$$

$$W_{electrical}=12.57\,(W_{fuel\ system}+W_{avionics})^{0.51}$$

$$W_{avionics}=2.117\ W_{uav}^{0.933}$$

Table 10.1 summarizes component weights from the above equations. Values for the engine, propeller, electrical systems, instruments and cabin accommodations were seen as being unrealistic. These values were replaced with component weights taken from The Cessna 152 and Piper Cherokee where appropriate. The avionics weight was also unrealistic. It was replaced by 2% of the empty weight which corresponds to Table 11.6 in Raymer.

Table 10.1

Component	Weight (lbs)	Horizontal Arm (in)	Moment Increment (in lb)	Vertical Arm (in)	Moment Increment (in lb)
Engine	270	158	42660	41	11070
Propeller	60	253	15180	41	2460
Nose Gear	33.3	251	8358.3	16	532.8
Main Gear	77.4	160	12384	16	1238.4
Wing	203.1	165	33511.5	27	5483.7
Horizontal Tail	22	30	660	41	902
Vertical Tail	19.3	33	636.9	70	1351
Electrical	40	182	7280	42	1680
Instruments	20	224	4480	43	860
Cabin Accom.	50	201	10050	44	2200
Avionics	20	233	4660	42	840
Hydraulics	1.6	145	232	27.5	44
Fuel (max)	221.3	163	36071.9	28	6196.4
Fuel Systems	30.9	159	4913.1	34	1050.6
Fuselage	159	167	26553	43	6837
Crew (max)	380	196	74480	45.5	17290
Baggage (max)	40	161	6440	60	2400

Summing component moments vertically and horizontally yields a horizontal and vertical CG of 170.4 inches from the tail and 36.3 inches from the ground. This gives a CG location of 33.5% MAC of the wing.

From here a CG envelope is obtained. Table 10.2 Shows the different cases examined.

Table 10.2

Case Description	CG x (in)	CG in %MAC	CG y (in)
1 190lb person, full tank	172.7	29.0	36.2
1 190lb person, empty tank	174.6	25.2	37.8
2 190lb persons, full tank	175.5	23.4	37.3
2 190lb persons, empty tank	177.4	19.6	38.8
1 80lb person, full tank, 40lb baggage	170.5	33.4	36.1
1 60lb person, empty tank, 40lb baggage	171.5	31.4	37.7
1 60lb person, empty tank	171.9	30.6	36.8
1 190lb person, full tank, 40lb baggage	172.4	29.6	36.9
1 190lb person, empty tank, 40lb baggage	174.1	26.2	38.5
full tank, 40lb baggage	168.8	36.7	-

Figure 10.1 shows the plotted CG envelope with a forward CG limit of 19.6% MAC and aft CG limit of 33.4% MAC. This yields a CG range of 13.8% MAC. It should be noted that the total aircraft payload cannot exceed 380lbs. The overturn angle was recalculated to be 45.71 degrees which is less than the 63 degree limit set by Raymer. The maximum aft CG of 36.7%

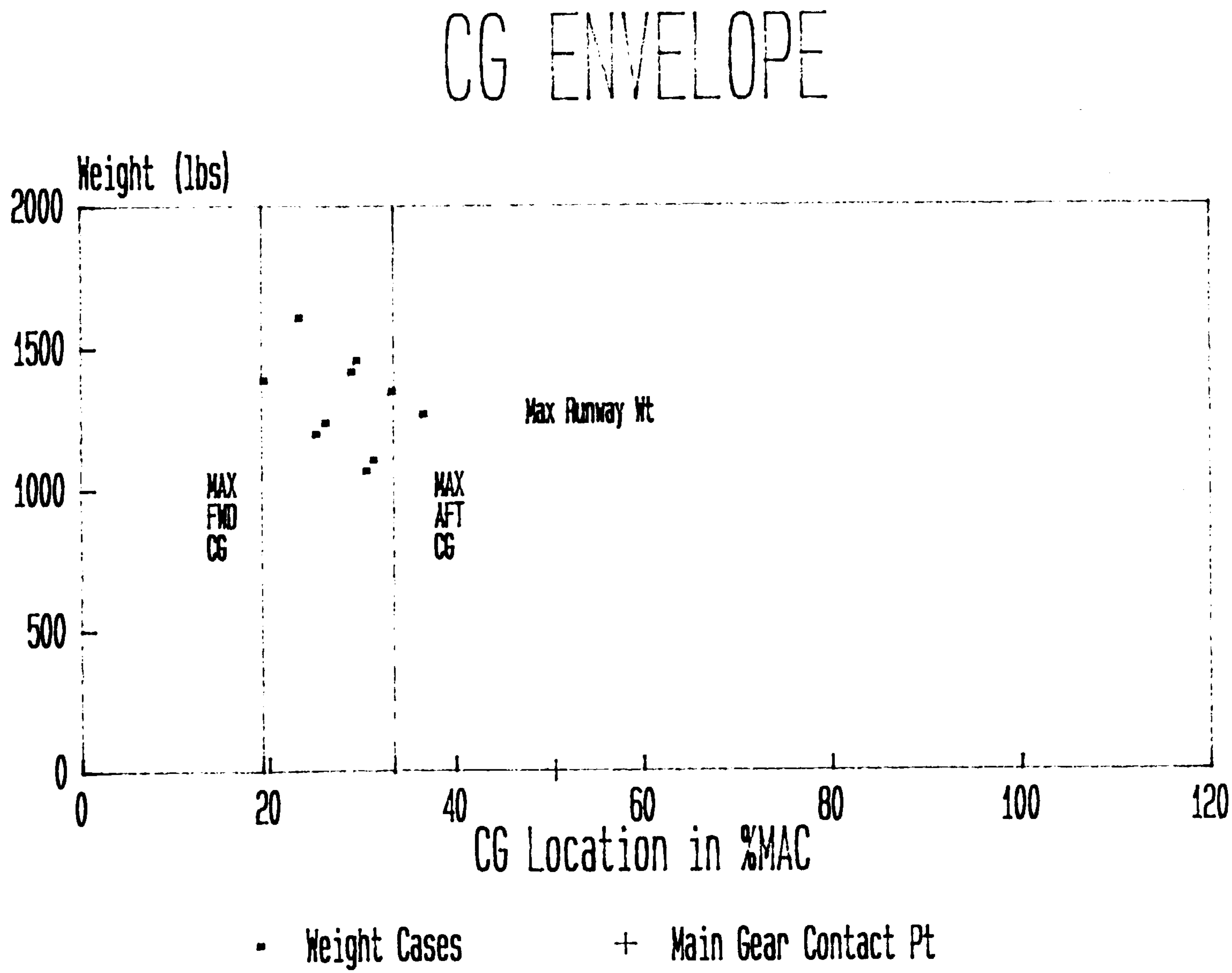

CG ENVELOPE
Weight (lbs)
2000
1500
1000
500
0
0
20
40
60
80
100
120
CG Location in %MAC
MAX FWD CG
MAX AFT CG
Max Runway Wt
Weight Cases
Main Gear Contact Pt

Figure 10.1

STABILITY AND CONTROL

Longitudinal Stability

It is necessary that a plane possess the natural tendency to return to its original attitude, yaw, bank, and speed after a disturbance, without corrections by the pilot. In order to determine the pitch stability of the aircraft, it is necessary to find the location of the C.G. which would cause neutral static stability. This point is called the neutral point, and can be found by solving equation 16.9.

$$\bar{X}_{np}=\frac{C_{L_\alpha}\bar{X}_{ac_w}-C_{M_{\alpha fus}}+\eta_h\frac{S_h}{S_w}C_{L_{\alpha h}}\frac{\partial\alpha_h}{\partial\alpha}\bar{X}_{ac_h}+\frac{F_{P_\alpha}}{qS_w}\frac{\partial\alpha_p}{\partial\alpha}\bar{X}_p}{C_{L_\alpha}+\eta_h\frac{S_h}{S_w}C_{L_{\alpha h}}\partial\frac{\alpha_h}{\partial\alpha}+\frac{F_{P_\alpha}}{qS_w}} \qquad \text{Eq. 16.9}$$

The pitching moment contribution can be approximated in a simplified manner using equation 16.22.

$$C_{M_{\alpha fus}}=\frac{k_f W_f^2 L_f}{cS_w}(57.3^\circ/rad) \qquad \text{Eq. 16.22}$$

The only contribution which is significantly stabilizing is the horizontal lift force.

$$C_{L_{\alpha h}}=\frac{2\pi A_h}{2+\sqrt{4+\frac{A^2B^2}{\eta^2}(1+\frac{\tan^2\Lambda}{(\beta^2)})}}\left(\frac{S_{exposed}}{S_{ref}}\right)(F) \qquad \text{Eq. 12.6}$$

The pitching contribution of the propeller can be found using equation 16.26, with the aid of Figs. 16.15 and 16.16 from Raymer.

$$F_{P_\alpha}=qN_\beta A_p\left(\frac{\partial C_{N_{blade}}}{\partial\alpha}\right)f(T) \qquad \text{Eq. 16.26}$$

The angle of attack seen by the horizontal tail is effected by the downwash of the wing and the propeller. To take this in to account, the downwash derivative can be obtained as:

$$\frac{\partial\alpha_h}{\partial\alpha}=1-\frac{\partial\epsilon}{\partial\alpha}-\frac{\partial\epsilon_p}{\partial\alpha}$$

where the propeller downwash gradient is:
and the downwash gradient of the wing is found from Fig 16.12 in Raymer.

$$\frac{\partial \epsilon_p}{\partial \alpha}=K_1+K_2 N_\beta \frac{\partial C_{N_{blade}}}{\partial \alpha}\left(\frac{\partial \alpha_p}{\partial \alpha}\right) \qquad \text{Eq. 16.27}$$

In addition to change in angle of attack seen by the tail caused by the downwash, the tail term is also affected by the increased dynamic pressure caused by the slipstream of the propeller. This effect can be estimated by simple momentum theory.

$$\eta_h=\eta_{h_{T=0}}\left(1+\frac{pT}{qA_p}\right)$$

With these values, the neutral point is now calculated:

X_{NP}= 0.434 (43.4% M.A.C.)

The static margin is an indication of the degree of stability of the aircraft, and is the percent MAC distance between the neutral point and the most aft C.G. location. Typically for a light trainer type aircraft static margins are of the order of 10%.

$$S.M._{actual}=\overline{X}_{np}-\overline{X}_{CG_{aft}}$$

With the most aft C.G. location at 0.334MAC, 10% is the exact static margin of this aircraft.

<u>Trim -incidence angles</u>

For trim at any flight condition, the sum of the moments about the C.G. of the aircraft must be zero. The sum of the moment contributions is given in equation 16.7. In trimmed flight, the most forward C.G. location (19.6% MAC) is the most critical case, since this makes it relatively more difficult to hold up the nose of the plane.

$$C_{M_{CG}}=C_L(\overline{X}_{CG}-\overline{X}_{ac_w})+C_{M_w}+\frac{\partial C_{M_w}}{\partial \delta_f}\delta_f+C_{M_{FUS}}-\eta_h\frac{S_h}{S_w}C_{L_h}(\overline{X}_{ac_h}-\overline{X}_{CG})+\frac{T}{qS_w}\overline{Z}_T+\frac{F_p}{qS_w}(\overline{X}_{CG}-\overline{X}_p)$$

$$C_{M_w}=C_{M_{o_{airfoil}}}A\frac{\cos^2\Lambda}{A+2\cos\Lambda}=-0.478 \qquad \text{Eq. 16.16}$$

$$C_L=C_{L_\alpha}(\alpha+i_w-\alpha_{0L}) \qquad \text{Eq 16.11}$$

For cruise, assume the plane flies level (α=0°)

Obtain an approximate value of i_w by assuming the entire lift to be carried by the

$$C_{L_h} = C_{L_{\alpha_h}}[\alpha + i_h - \epsilon - \alpha_{0L_h}] \qquad \text{Eq.16.12}$$

wing in level cruising flight. This analysis produced an incidence of the wing of, $i_w \approx 0°$. Plugging this into equation 16.12, and putting the result into equation 16.7, yields an incidence angle of the horizontal tail, $i_h = +0.865°$. This combination of incidence angles results in a total lift coefficient that is slightly higher than the desired value for cruise, but this only requires the plane to fly at a negative angle of attack of less than 1/2 degree, and this will balance out since the weight during cruise is not the full gross weight.

Trim - elevator control

There must be enough elevator control to trim the aircraft during the worst case scenario, which occurs when at stall speed, stall AOA, flaps down, most forward C.G., power on (but neglecting propeller face force, which causes a nose up moment). Under these conditions plugging into equation 16.7 yields $C_L = 1.8225$, at $\alpha_w = 11.28°$. The lift increment due to flap deflection must be accounted for by a reduction in the zero-lift angle ($\Delta\alpha_{0L} = -10.3°$), which can be calculated using equation 16.14.

$$\Delta\alpha_{0L} = -\frac{1}{C_{L_\alpha}}\frac{\partial C_L}{\partial \delta_f}\delta_f \qquad \text{Eq. 16.14}$$

Where

$$\frac{\partial C_L}{\partial \delta_f} = 0.9K_f\left(\frac{\partial C_l}{\partial \delta_f}\right)\frac{S_{flapped}}{S_{ref}}\cos\Lambda_{H.L.} \qquad \text{Eq. 16.15}$$

Figures 16.6 and 16.7 are used to find theoretical lift increment for large flap deflections ($\delta_f = 30°$). The lift increment due to flap deflection, $\Delta C_{Lmax} = 0.5157$ (from equation 12.21), which yields $\alpha_{CLmax} = 11.70°$ (clean) and 11.08° (flapped).

$$\Delta C_{L_{max}} = \Delta C_{l_{max}}\frac{S_{fl}}{S_{ref}}\cos\Lambda_{H.L.} \qquad \text{Eq. 12.21}$$

Plugging into equation 16.7 yields the required elevator deflection for trim at stall to be $\delta_e = -11.78°$. Two important considerations in which greater elevator deflection is required to trim the aircraft are in ground effects ($\delta_e = -17.55°$), and during takeoff rotation ($\delta_e = -22.55°$). The required elevator deflection exceeds 20° by a small amount because calculations were made from hypothetical worst case scenarios in which the aircraft takes off without fuel and waits until stall speed to begin rotation.

Directional Stability

The aircraft must have directional and lateral stability as well as longitudinal. For directional stability, the major stabilizing contribution comes from the vertical tail. There must be sufficient vertical tail area in order to have an adequate level of directional stability.

$$C_{N_\beta}=C_{N_{\beta_w}}+C_{N_{\beta_{fus}}}+C_{N_{\beta_v}}-\frac{F_{p\beta}}{qS_w}\frac{\partial\beta_p}{\partial\beta}(\bar{X}_{cg}-\bar{X}_p) \qquad \text{Eq. 16.39}$$

where the wing,fuselage and vertical tail contributions are given by

$$C_{n\beta_w}=C_L^2[\frac{1}{4\pi A}-[\frac{\tan\Lambda}{\pi A(A+4\cos\Lambda)]}[\cos\Lambda-\frac{A}{2}-\frac{A^2}{8\cos\Lambda}+\frac{6(X_{acw}-X_{cg})\sin\Lambda}{A}]]$$

$$C_{n_{\beta_{fus}}}=-1.3\frac{volume}{S_w b}(\frac{D_f}{W_f}) \qquad \text{Eq. 16.47}$$

$$C_{n_{\beta_v}}=C_{F_{\beta_v}}\frac{\partial\beta_v}{\partial\beta}\eta_v\frac{S_v}{S_w}(\bar{X}_{acv}-\bar{X}_{cg}) \qquad \text{Eq. 16.36}$$

where the local dynamic pressure ratio and side slip derivative can be approximated using equation 16.48.

$$\frac{\partial\beta_v}{\partial\beta}\eta_v=.724+\frac{3.06\frac{S_{vs}'}{S_w}}{1+\cos\Lambda}-0.4\frac{Z_{wf}}{D_f}+0.009A_w \qquad \text{Eq. 16.48}$$

The effect of the propeller side force is given by equation 16.26 as with the pitch case. The result obtained from using these equations is a sideslip moment derivative (directional stability) of,

$$C_{N_\beta}=-0.08324/rad$$

This falls within the limits given in Fig 16.20 (Raymer) of .06 and .11. This should provide adequate directional stability for a primary flight trainer.

Lateral Stability

For lateral, or roll stability, which is coupled with directional stability, the dihedral of the wing and height of the a.c. of the vertical tail are the main stabilizing factors. The rolling moment derivative is the sum of the wing contributions and the vertical tail contribution.

$$C_{l_\beta}=C_{l_{\beta_w}}+C_{l_{\beta_v}} \qquad \text{Eq. 16.40}$$

The wing term is comprised of the sweep, dihedral, and wing-fuselage interference effects. The sweep effect is obtained from Fig 16.21 (Raymer). The dihedral and wing-fuselage effects are found by:

$$C_{l_{\beta_\Gamma}}=-\frac{C_{L_\alpha}\Gamma}{4}\left[\frac{2(1+2\lambda)}{3(1+\lambda)}\right] \qquad \text{Eq. 16.42}$$

$$C_{l_{\beta_{wf}}}=-1.2\sqrt{A}\frac{Z_{wf}(D_f+W_f)}{b^2} \qquad \text{Eq 16.43}$$

The vertical tail term is found using the same equation as used for the directional moment derivative, with moment arm as the height of the a.c. of the vertical tail above the fuselage reference line in percent wing span.
With a dihedral of 1.6°, these equations produce a rolling moment derivative of:

$$C_{l_\beta}=-0.04191$$

This falls in the range suggested by Perkins&Hage of approximately half of the directional moment derivative. This can be accomplished with a relatively small amount of dihedral for this plane due to the high location of the a.c. of the vertical tail. The result is 3.657 degrees of "effective dihedral, which agrees with Perkins&Hage's statement of 3 to 4 degrees effective as a maximum.

Directional Control

There must be sufficient rudder size to maintain a steady sideslip angle of 11.5°. This insures that the pilot can land the aircraft in a crosswind. The required rudder size is determined by setting the sum of the directional moments equal to zero and solving for the change in zero-lift angle of the horizontal tail. This then dictates the size needed for the rudder. This is verified by finding the change in zero-lift angle caused by deflecting the rudder, and comparing it to the result obtained from Eq. 16.14 and 16.15 with a reasonable deflection of the rudder (10 degrees).

$$C_N=C_{N_{\beta_w}}\beta+C_{N_{\delta_a}}\delta_a+C_{N_{\beta_{fus}}}\beta+C_{N_{\beta_v}}(\beta+\Delta\beta_{0L})=0 \qquad \text{Eq. 16.35}$$

Roll Control

There must be sufficient aileron size to roll aircraft at a steady rate of 46.2 degrees per second at low speeds. The roll rate is found using the aileron control power (Eq. 16.45), and the roll damping factor. The roll damping factor, C_{l_p} is found from Fig. 16.25 (Raymer), and is -0.44 for this plane.

$$P = -\left(\frac{C_{l_{\delta_a}}}{C_{l_p}}\right)\delta_a \qquad \text{Eq. 16.61}$$

$$C_{l_{\delta_a}} = \frac{2\Sigma K_f\left(\frac{\partial C_L}{\partial \delta_f}\right) Y_i S_i \cos\Lambda_{H.L.}}{S_{wb}} \qquad \text{Eq. 16.45}$$

This yields a roll rate p of 82.5 degrees per second, which is well over the minimum required value. This suggests that perhaps the aileron size could be decreased.

$$p = P\left(\frac{2V}{b}\right)$$

Spin Recovery

The spin recovery of the aircraft depends on the amount of rudder area left unblanketed by the horizontal tail (usually the area outside of a line drawn at 60° from the leading edge of the horizontal tail). The spin recovery criteria can be obtained from Fig 16.31 (Raymer) using the moments of inertia of the aircraft. The calculated spin recovery criteria is four times the required value taken from Fig 16.31 at 5000 ft. Therefore, this aircraft has excellent spin recovery characteristics.

PERFORMANCE

Having the final specifications for the plane, it is now necessary to determine if it has adequate performance characteristics. In order to determine the performance, the drag polar equation must be obtained. The drag polar is found using the drag break down procedure given by Raymer. Wherever possible, worst case scenarios will be assumed. This will insure at least the expected performance.

$$C_{D_o}=\frac{\Sigma(C_{f_c}FF_cQ_cS_{wet_c})}{S_{ref}}+C_{D_{misc}}+C_{D_{L\cdot P}} \qquad \text{Eq. 12.24}$$

$$C_f=\frac{0.455}{(\log_{10}R)^{2.58}(1+0.144M^2)^{0.65}} \qquad \text{Eq. 12.27}$$

$$FF=[1+\frac{0.6}{(x/c)_m}(\frac{t}{c})+100(\frac{t}{c})^4][1.34M^{0.18}(\cos\Lambda)^{0.28}] \qquad \text{Eq 12.30}$$

for wing, tail, and strut.

$$FF=(1+\frac{60}{f^3}+\frac{f}{400}) \qquad \text{Eq 12.31}$$

for fuselage and canopy.

$$f=\frac{l}{d}=\frac{l}{\sqrt{(4/\pi)A_{max}}} \qquad \text{Eq 12.33}$$

Q = 1.2 for fuselage,and tails

Q = 1.4 for low wing

S_{wet} = 180.5 ft^2 S_{ref}= 124.4 ft^2

$$C_{D_{misc}}=\frac{\Sigma(D/q)}{S_{ref}}Q_c \qquad C_{D_o}=.08(\Sigma C_{D_{o(comp)}})$$

Tbl 1- Zero lift drag at cruise

C_{D_o} = 0.0267

Component	C_{D_o}
FUSELAGE	.00679
WING	.0103
V TAIL	.00108
H TAIL	.00252
MISC.	.004026
LEAK&PROTUB	.001568

The drag polar consists of two parts - the zero-lift drag, and the lift induced drag. C_{D_0}= 0.0267. The lift induced part is $C_{D_i} = KC_L^2$, where K = 1/(πAe) = 0.05. Thus,

$$C_D=0.0267+0.05C_L^2$$

From this, the power required can be obtained.

$$P=\frac{1}{2}\rho V^3 S C_{D_o}+\frac{KW^2}{\frac{1}{2}\rho VS} \qquad \text{Eq. 17.17}$$

The power available is the output horsepower of the engine at the particular altitude multiplied by the propeller efficiency. The power available from the Lycoming O-235 L is 118 hp at sea level, and 97 hp at 5000 ft. The power vs. velocity curves are plotted in Fig. 12.1.

With the power required and power available now known for each velocity, the critical velocity performance values can be taken from the graph.

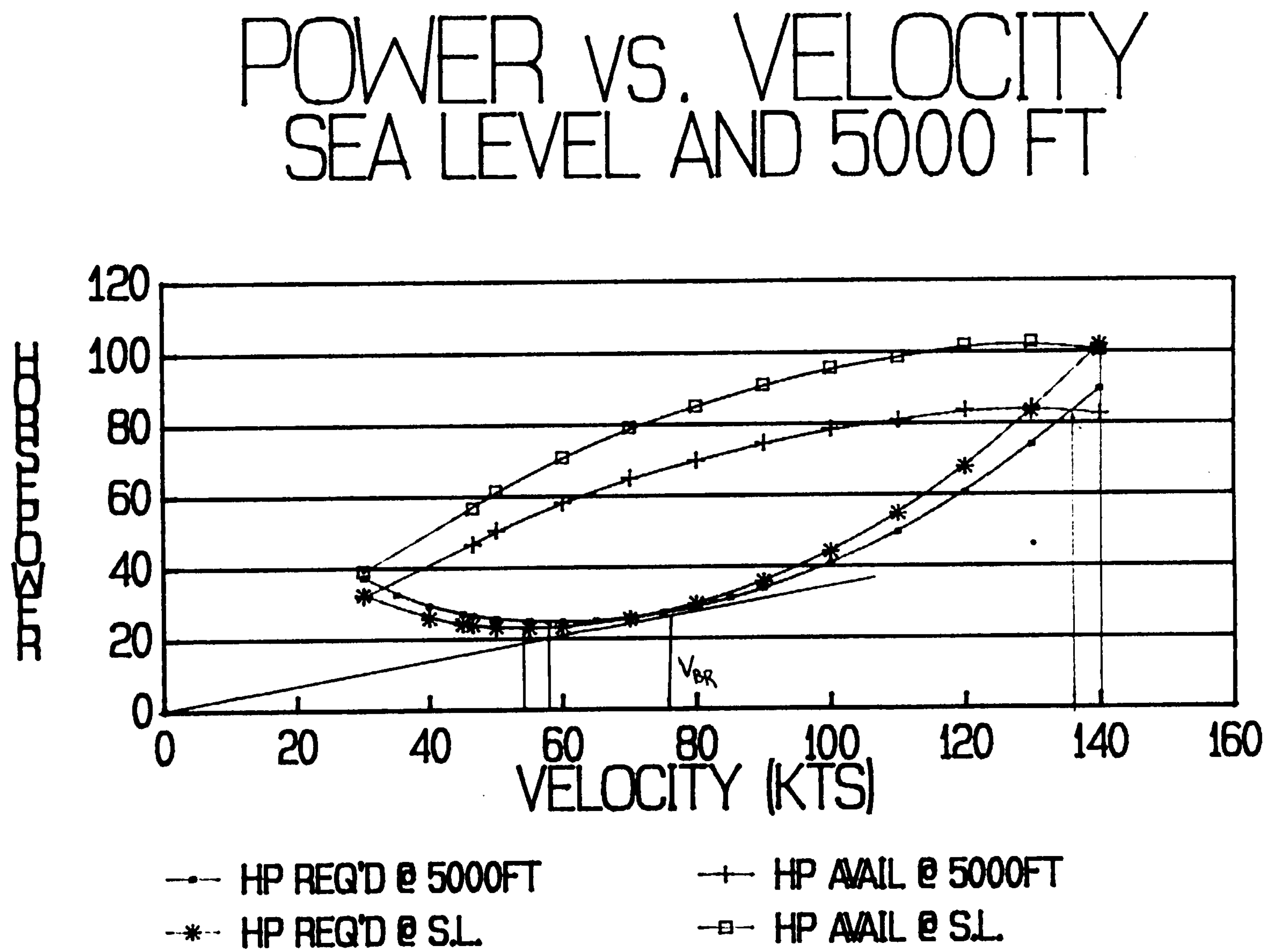
Best range velocity comes from the tangent to the power required curve drawn from the origin. Maximum endurance velocity comes from the lowest point of this curve. Maximum velocity is the velocity at which the power required equals the power available (right intersection of curves). Likewise, minimum velocity comes from the left most intersection, unless this is lower than the stall speed, in which case, the stall speed is the minimum velocity. The rate of climb is determined by:

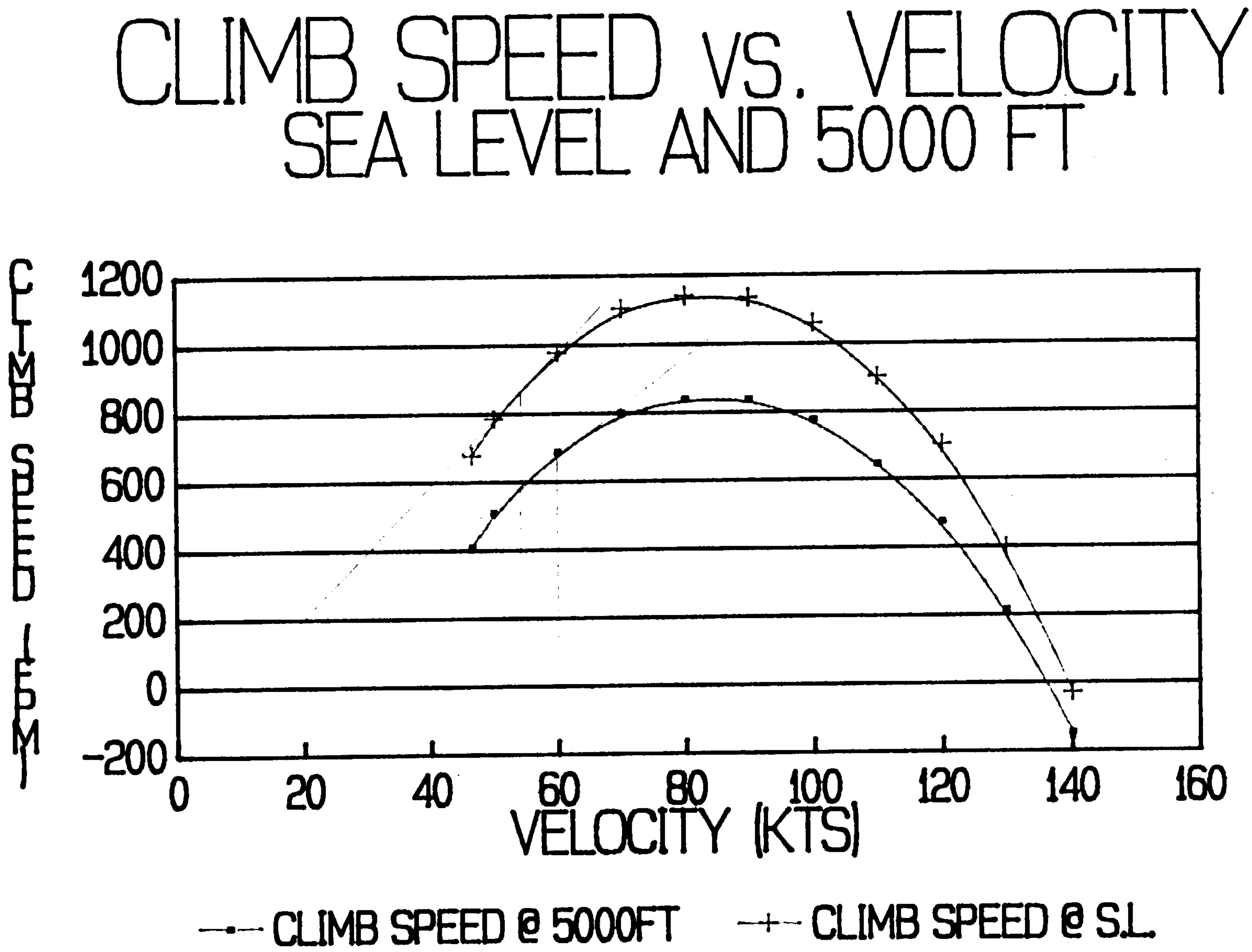
$$V_v=\frac{(P_{avail}-P_{req'd})550}{W}(60\frac{\text{sec}}{\text{min}})$$

A plot of the climb speed vs. forward velocity is shown in Fig. 12.2. The critical velocity performance values obtained from the graph are shown in Fig. 12.3.

Fig. 12.3

$V_{max\ S.L.}$	140 kts	$V_{max\ range}$ (S.L & 5000)	77 kts
$V_{max\ 5000}$	136 kts	$V_{max\ climb}$ (S.L & 5000)	80 kts
$V_{min\ S.L.}$	46.5 kts	$V_{v\ max\ 5000}$	840.3 kts
$V_{max\ end\ 5000}$	58 kts	$V_{v\ max\ S.L.}$	1140.5kts


POWER vs. VELOCITY
SEA LEVEL AND 5000 FT
HORSEPOWER
0
20
40
60
80
100
120
0
20
40
60
80
100
120
140
160
VELOCITY (KTS)
V_{BR}
HP REQ'D @ 5000FT
HP REQ'D @ S.L.
HP AVAIL @ 5000FT
HP AVAIL @ S.L.


CLIMB SPEED VS. VELOCITY
SEA LEVEL AND 5000 FT
CLIMB SPEED (FPM)
1200
1000
800
600
400
200
0
-200
0
20
40
60
80
100
120
140
160
VELOCITY (KTS)
CLIMB SPEED @ 5000FT
CLIMB SPEED @ S.L.

Range-

In addition to the velocity performance, it is also required to find the range of the aircraft.

$$R=\frac{550\eta_p}{C_{bhp}}\frac{L}{D}\ln\frac{W_i}{W_f} \qquad \text{Eq. 17.28}$$

At cruise, propeller efficiency is 86.3%, C_{bhp} = .4 /hr, L = W, C_L =.315. Thus, D = 165.5 lbs using the drag polar equation.
The weight fraction for the cruise leg can be determined using the weight fractions for the other legs a assumed earlier, and the weight of fuel available for the cruise mission. At the beginning of cruise, the weight is 1585.5 lbs, and 1445 lbs at the end of cruise.
Using these numbers, the range is 688 NMi at cruise altitude.

$$R_{cr} = 688 \text{ NMi}$$

The maximum range will occur when the induced drag is minimum.

$$C_{L_{min}}=\sqrt{\frac{C_{D_o}}{K}}=.7308 \qquad \text{Eq. 17.14}$$

$$C_D = .0534$$

With these values and the propeller efficiency of 67% at the maximum range speed of 77 kts, the maximum range is found to be 752.8 NMi.

$$R_{max} = 752.8 \text{ NMi}$$

Flight Envelopes

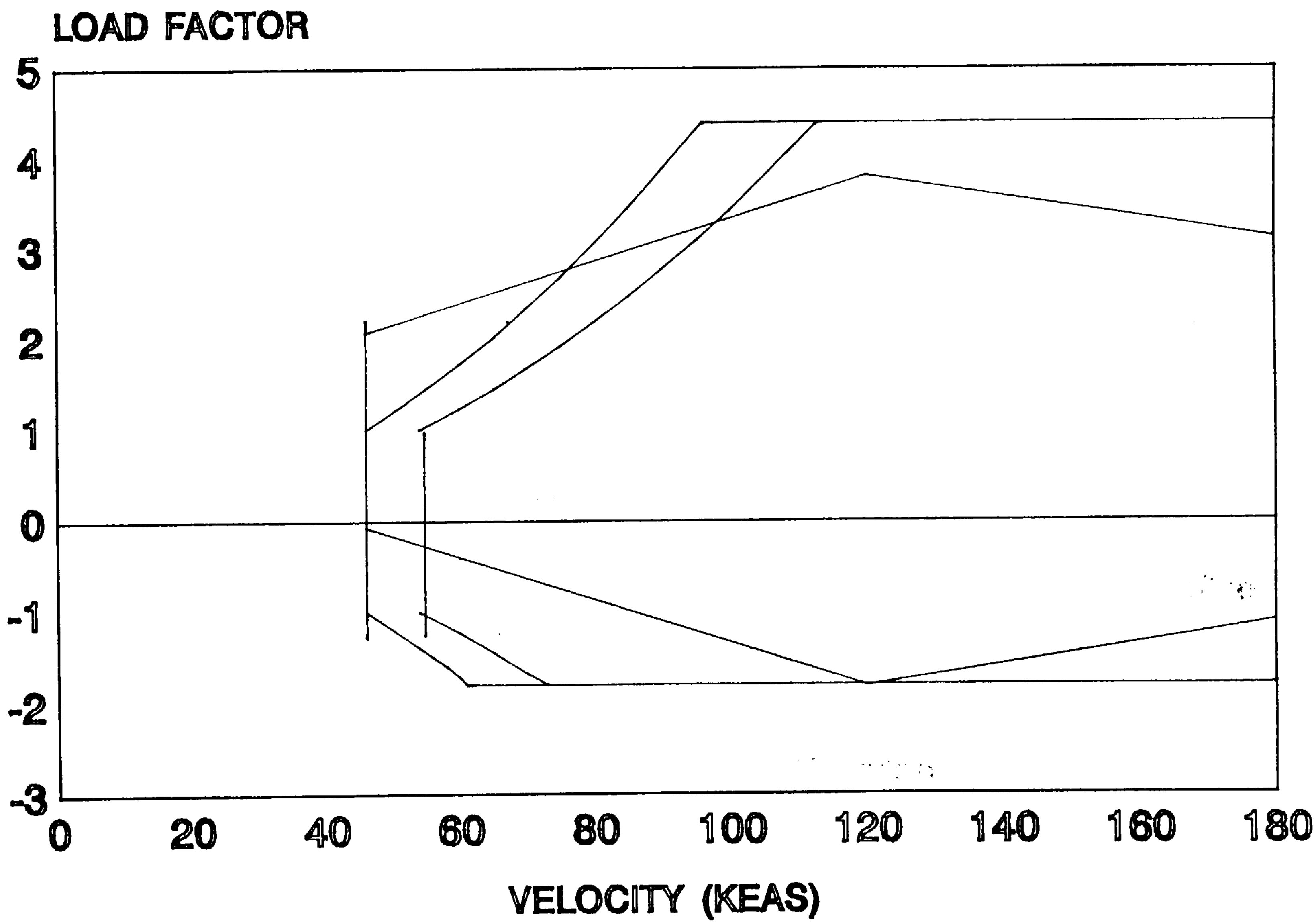
Stall-

In order to design the structure of the aircraft, the maximum loads which the aircraft will be subjected to must be determined. For a utility class plane, the limit load factor must be at least 3.8, with the ultimate load being 1.5 times the limit load.
The loads at stall will determine one boundary of the flight envelope. They can be determined by calculating the lift produced at various velocities for positive and negative high angle of attack.

$$n=\frac{L}{W}=\frac{C_L\frac{1}{2}\rho V^2 S_w}{W}$$

These curves, starting from n=1 at zero velocity, along with the stall speed, determine the left boundary of the V-n diagram.

V-N Diagram
SEA LEVEL



Gust load factors-

The top and bottom limits are determined by a combination of the limit load and gust load factors. The limit load factors are -1.76 and 4.4. The gust load factors must be calculated with the procedure specified in the F.A.R.'s.

$$\Delta n = \frac{\rho U V C_{L_\alpha}}{2\frac{W}{S}} \qquad \text{Eq. 14.4}$$

where, U is the gust velocity - $U = KU_{de}$. U_{de} is the flight test derived equivalent airspeed, and is 50 ft/s at cruise, and falls off to 25 ft/s at design dive speed.

$$K = \frac{0.88\mu}{5.3+\mu} \qquad \text{Eq. 14.6}$$

for subsonic flight regime.

$$\mu = \text{"}massratio\text{"} = \frac{2(\frac{W}{S})}{\rho g \bar{c} C_{L_\alpha}} \qquad \text{Eq.14.8}$$

These formulas yield the following equations for this plane:

$$\Delta n = 0.01388V \text{"}forgustloadatV<120kts\text{"}$$
$$\Delta n = 0.00694V \text{"}forgustloadsatV>120kts\text{"}$$

The far right hand side of the flight envelopes are determined by the design dive speed. These limitations combine to form the envelopes seen in Fig. 12.4 and Fig. 12.5, for sea level and 5000 ft respectively.

Take-off Distance

The total take-off distance is the sum of the ground roll, transition, and climb distances.

Ground Roll-

The ground roll covers the distance required for the plane to accelerate from a velocity of zero.

Assume a fairly high coefficient of rolling friction (.05)

$$S_G = \frac{1}{2gK_A} \ln \frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}$$ Eq. 17.99

$$K_T = \frac{T}{W} - \mu$$ Eq. 17.100

$$T = \frac{550 bhp \eta_p}{V}$$ Eq. 17.5

$$V_{avgr} = \frac{1}{\sqrt{2}} V_{T-O}$$

V_{avgr} = 61 ft/s
T = 553.25 lb

$$K_A = \frac{\rho}{2(\frac{W}{S})}(\mu C_L - C_{D_o} - KC_L^2)$$ 17.101

$S_{G\ TO}$ = 448.8 ft

Transition-

In the transition portion of the take off run, the aircraft is accelerating from take-off velocity to climb speed along a circular path. The radius of rotation is $R = .205\ V^2_{stall}$, and the final climb angle, $\gamma_{cl} = \sin^{-1}(T/W - 1(L/D))$. The distance covered in the transition run is the product of these two values:

$$S_T = R \sin \gamma_{CL}$$ Eq. 17.104

Climb-

F.A.R. 23 requires clearing a 50 ft obstacle. The distance covered in climbing to this angle is simply the change in height divided by the tangent of the climb angle.

$$S_c = \frac{h_{obst} - h_{TR}}{\tan \gamma_{CL}}$$ Eq. 17.109

$$S_{T-O} = S_{gr} + S_{tr} + S_c = 920.1 FT$$

Landing Distance-

With a conservative approach angle of 3° and a flaring radius of rotation given by equation 17.104, the flare height is 2 ft, and the distance covered in the flare is 75.8 ft.

The distance covered on the approach from a 50 ft obstacle is found the same way as for take off. Likewise, the ground roll after braking is found using equations 17.99 and 17.100, with $V_i=1.15V_{st}$, and a coefficient of braking friction of -0.3 (fairly conservative). In addition, there is usually a 1-3 sec delay before the pilot applies the brakes on touch down. During this time, a distance equal to the touch down velocity multiplied by the delay time is covered. (Use 3 sec to be conservative.)

$$R=\frac{(1.23V_{st})^2}{g(.2)} \qquad \text{Eq. 17.104}$$

$$h_{fl}=R(1-\cos\gamma_a) \qquad \text{Eq. 17.107}$$

$$S_{land}=S_{approach}+S_{flare}+S_{freeroll}+S_{gr}=2291.5FT$$

COST ANALYSIS

Airframe Weight

The airframe weight was calculated by taking the empty weight and subtracting the following components:

COMPONENT	WEIGH
1. Wheel,Brakes,Tires	45.0
2. Engine	248.0
3. Cooling Fluid	NONE
4. Rubber Fuel Cells	NONE
5. Propeller	28.7
6. Shaft	30.0
7. Aux. Power Plant Unit	NONE
8. Instruments	7.0
9. Battery & Generator	35.2
10. Electronics	NONE
11. Turrets & Mounts	NONE
12. Cabin Heat/Defrost	3.5
13. Cameras	NONE
14. Trapped Fuel & Oil	NONE
15. Avionics	11.5
TOTAL	409.5

Table 13.1

This total subtracted from the empty weight yields an airframe weight of 597.1 LBS.

Avionics Cost

There are many options for which avionics could be placed in the aircraft. On the next page is a list of prices of several instruments.

INSTRUMENT	COST ($)
ADF Davtron 701B-2	247.00
ELT Merl 79007-P	300.00
Encoding Altimeter Davtron M650	225.00
Horizontal Situation Indicator Aeronautics 8131	4620.00
Intercom System Concept ATC-2	130.00
Transponder Narco AT-150	1145.00
VHF Nav Reciever/Communication Transceiver	800.00
Distance Measuring Equipment Bendix/King KN 64	1995.00

Table 13.2

For IFR flight rules the avionics package would further include ADF, DME, Transponder, and VHF Transciever. This would increase the cost to a total of $11,654.

Production Rate and Profit Percentage

The production rate would depend on the demand, so listed below are several different production rates.

NUMBER OF AIRCRAFT	NUMBER OF AIRCRAFT PRODUCED	TIME TO COMPLETE (MONTHS)
100	8	12.5
500	10	50
*1000	17	58
2000	25	80

Table 13.3

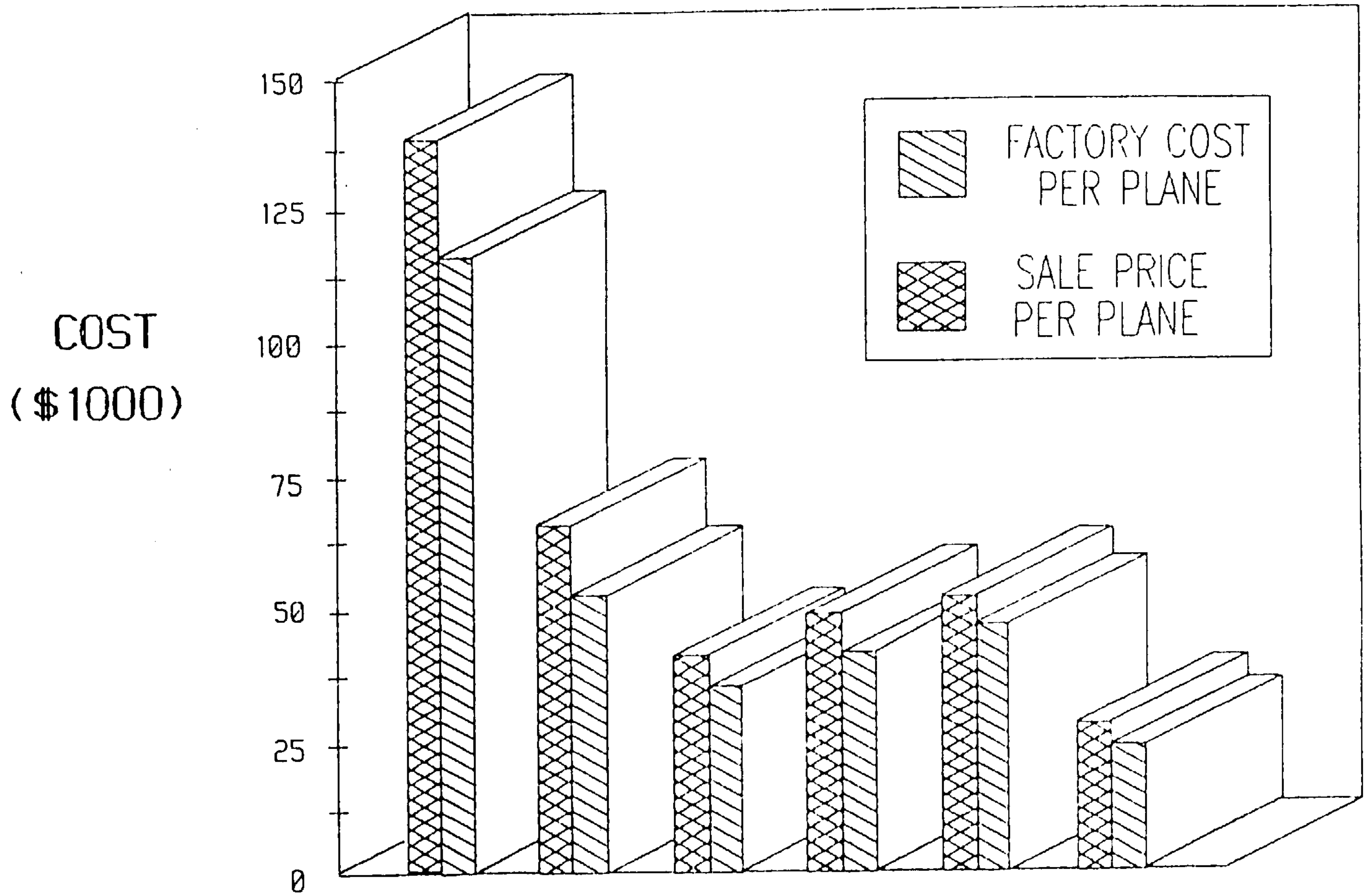
A profit percentage of 15% was chosen to provide a fair return on a large investment and to ensure the buyer a reasonable price. A listing of the costs depending on the number of aircraft produced is provided in Table 13.4. The option which is the most promising is the 1000 aircraft at 17 per month with no avionics for a price of $41,798 which meets the requirement of being less than $50,000.

PRODUCTION COST

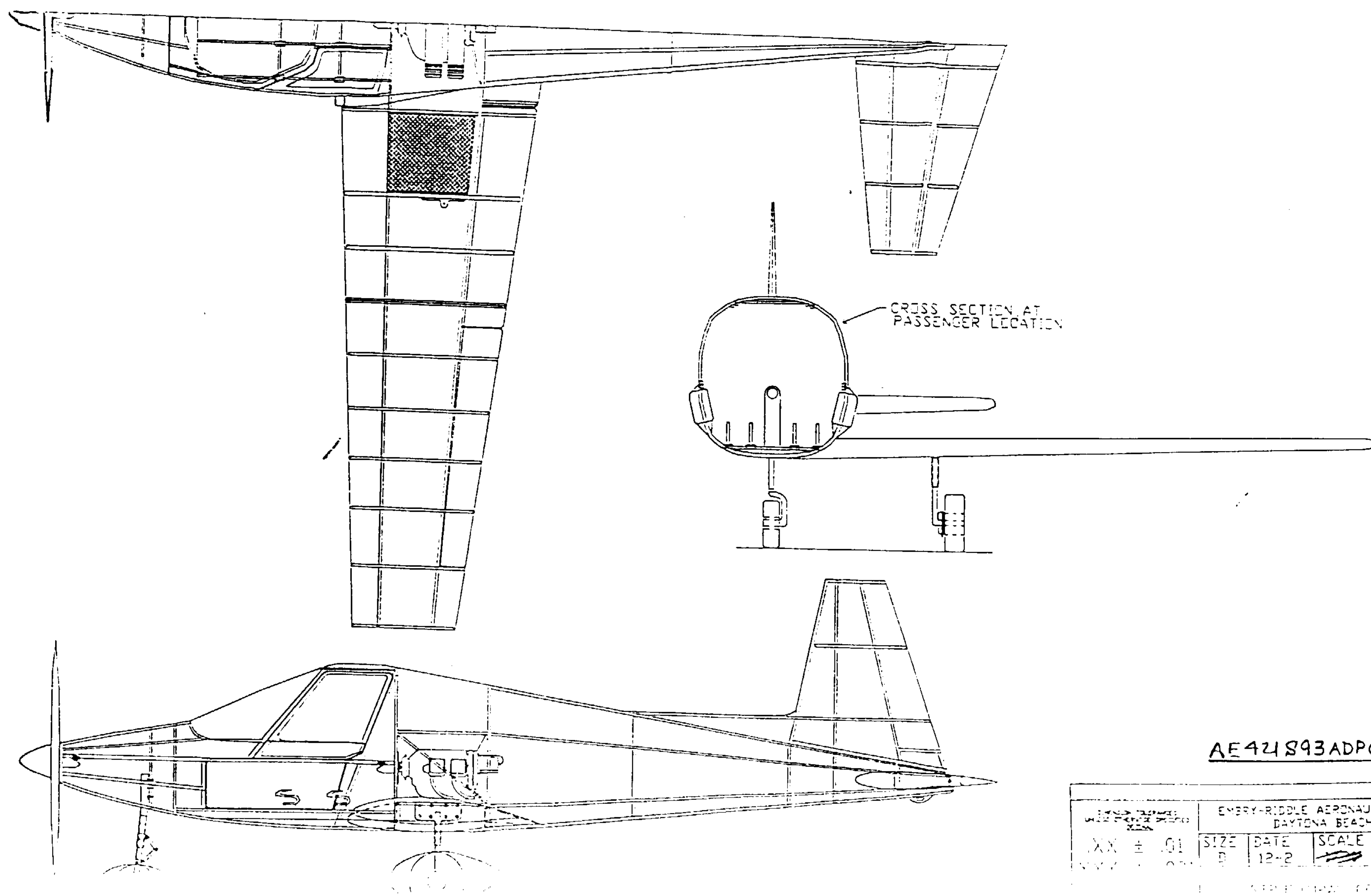
PRODUCTION COST	100/ 8 month	500/10 month	**1000/ 17 month (no avionics)**	1000/ 17 month (VFR)	1000/ 17 mo (IFR)	5000/25 month
ENGINEERING MANHOURS	28014.09	37608.68	**42694.94**	42694.94	42694.94	57317.59
ENGINEERING COST ($)	1589660	2134104	**2422725**	2422725	2422725	3252487
ENGINEERING COST PER PLANE ($)	15896.6	4268.21	**2422.73**	2422.73	2422.73	650.5
CERTIFICATION SUPPORT COST ($)	118374.5	118374.5	**118374.5**	118374.5	118374.5	118374.5
COST PER PLANE ($)	1183.745	236.75	**118.3745**	118.37	118.37	23.67
FLIGHT TEST OPERATIONS COST ($)	8433.91	8433.91	**8433.91**	8433.91	8433.91	8433.91
COST PER PLANE ($)	84.34	16.87	**8.43**	8.43	8.43	1.69
TOOLING MANHOURS	29326.5	39634.45	**46437.18**	46437.18	46437.18	63436.16
TOOLING COST ($)	1109422	1499372	**1756718**	1756718	1756718	2399790
COST PER PLANE ($)	11094.22	2998.74	**1756.72**	1756.72	1756.72	479.96
MANUFACTURING LABOR HOURS	180708.6	419990.3	**603919.3**	603919.3	603919.3	1403586
LABOR MANHOURS PER PLANE	1807.086	839.9806	**603.92**	603.92	603.92	280.71
MANUFACTURING LABOR COST ($)	5696840	13240190	**19038550**	19038550	19038550	44248050
LABOR COST PER PLANE ($)	56968.4	26480.39	**19038.56**	19038.56	19038.56	8849.61
Q C COST ($)	740589.1	1721225	**2475012**	2475012	2475012	5752247
Q C COST PER PLANE ($)	7405.891	3442.45	**2475.01**	2475.01	2475.01	1150.45
MANUFACTURING MAT. & EQUIP. COST ($)	1046540	3744037	**6482700**	6482700	6482700	2319210
MANUFACTURING MAT. & EQUIP. COST PER PLANE ($)	10465.4	7488.07	**6482.7**	6482.7	6482.7	4638.42
ENGINE LIST PRICE ($)	12350.24	12350.24	**12350.24**	12350.24	12350.24	12350.24
ENGINE FACTORY COST ($)	8783.62	7797.39	**7407.52**	7407.52	7407.52	6575.8
PROP LIST PRICE ($)	1891.5	1891.5	**1891.5**	1891.5	1891.5	1891.5
PROP FACTORY COST ($)	1345.255	1194.21	**1134.5**	1134.5	1134.5	1007.12
AVIONICS LIST PRICE ($)	7467	7467	**0**	7467	11654	7467
AVIONICS FACTORY COST ($)	6696.68	5944.77	**0**	5647.53	8814.3	5013.42
FACTORY COST PER PLANE ($)	114590.1	55132.69	**36346.13**	41993.66	45160.42	24397.31
SALE PRICE PER PLANE ($)	131778.6	63402.59	**41798.05**	48292.71	51934.49	28056.91
TOTAL INVESTMENT REQD. TO FINANCE CERT. ($)	2825890	3760284	**4306252**	4306252	4306252	5779085

Table 13.4

COST ANALYSIS:



PLANES PRODUCED: @ /MONTH:	100 8	500 10	1000 17 (NO AVIONICS)	1000(VFR) 17	1000(IFR) 17	5000 25
FACTORY COST $:	114,590	55,133	36,346	41,994	45,160	24,397
SALE PRICE $:	131,779	63,403	41,798	48,293	51,934	28,057



AE421S93ADP01-2

	EMBRY-RIDDLE AERONAUTICAL UNIVERSITY DAYTONA BEACH FLORIDA			
.XX ± .01	SIZE B	DATE 12-2	SCALE	DRAWN BY KYLE KO[illegible]