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# THE TRITON

## Design Concepts and Methods

Designed By

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All equations used in this report are taken from:

Raymer, D.P., Aircraft Design: A Conceptual Approach, AIAA Education Series, 1989.

# DESIGN SPECIFICATIONS

## Design Requirements

- Design must conform to FAR Part 23, including crashworthiness standards.
- Two to four occupants
- FAA certified engine
- VFR equipment required, allow upgrade to IFR
- Certification category at least utility
- Capable of either of two training missions
  - -- Climb to 5000 ft., cruise 500 n.mi. plus reserve, land
  - -- Climb to 1000 ft. then descend, 10 cycles, climb to 3000 ft., maneuver at 2g for 30 min., cruise 100 n.mi., land
- Cruise speed at least 120 knots
- Runway length not over 3000 ft.
- Cost goal $50,000, not including avionics, for production of 1000 airplanes over a five year period

## Triton Specifications

| | |
|---|---|
| **General:** | |
| Engineering Firm | C & P Aerospace |
| Certification Category | Utility |
| Primary Mission | Flight training |
| Number of Occupants | 2 |
| Structure | Aluminum/composites |
| List Price | $46,020 |
| Operating Cost | $45/hr |
| Overall Height | 7.9 ft. |
| Overall Length | 28.0 ft. |
| Fuselage Width (External/Internal) | 50 in./46 in. |
| **Wing:** | |
| Type | Cantilever high wing |
| Planform Area | 150.6 sq. ft. |
| Span | 33.8 ft. |
| MAC | 4.57 ft. |
| Taper Ratio | 0.561 |
| Airfoil | NACA $64_1$A212 |
| Aileron Type | Frise |
| Aileron Area | 10% planform |
| Max. Aileron Deflection | $+30^\circ/-10^\circ$ |
| Flap Type | Single-slotted |
| Max. Flap Deflection | $30^\circ$ |
| **Horizontal Stabilizer:** | |
| Area | 25.0 sq. ft. |
| Span | 12.25 ft. |
| Elevator Area | 45% stab. area |
| Max. Elevator Deflection | $\pm 20^\circ$ |
| Airfoil | NACA 0009 |
| **Vertical Stabilizer:** | |
| Area | 12.8 sq. ft. |
| Height | 4.0 ft. |
| Rudder Area | 45% stab. area |
| Max. Rudder Deflection | $\pm 20^\circ$ |
| Airfoil | NACA 0009 |
| **Powerplant:** | |
| Engine Type | Lycoming O-235 |
| Rated Horsepower (sea level static) | 118hp @ 2700 rpm |
| Propeller Type | 74 x 64 cruise |

| | |
|---|---|
| **Landing Gear:** | |
| Type | Fixed, tricycle |
| Wheel Track | 8.1 ft. |
| Wheel Base | 7.0 ft. |
| Main Gear Tire Size | 17.5x6.00-6 |
| Nose Gear Tire Size | 14.25x5.00-5 |
| **Weights and Capacities:** | |
| Empty Weight | 1261 lbs. |
| Gross Weight | 1903 lbs. |
| Baggage Capacity | 100 lbs. |
| Max. Fuel Capacity | 39.5 US gallons |
| Wing Loading | 12.6 lb/sq.ft. |
| **Performance:** | |
| Cruise Altitude | 5000 ft. |
| Max. Speed | 128 knots |
| Cruise Speed (83% power) | 120 knots |
| Stall Speed (clean) | 53 knots |
| Stall Speed (flaps) | 45 knots |
| Max. Rate of Climb (sea level) | 1012 fpm |
| Best Rate of Climb Speed | 55 knots |
| Max. Range | 943 n. mi. |
| Range at Cruise | 790 n. mi. |
| Takeoff Distance | 1047 ft. |
| Landing Distance | 909 ft. |

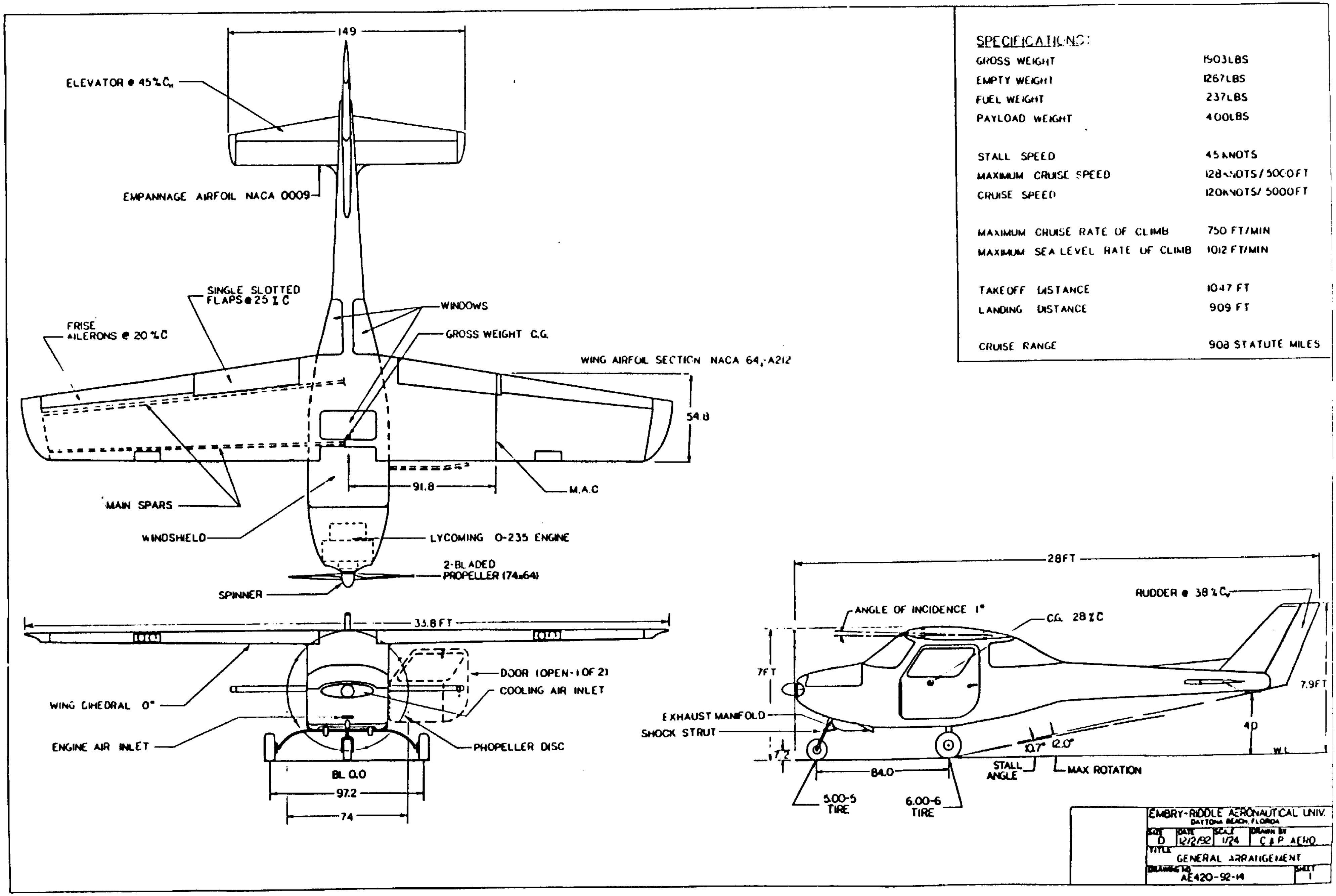



**Figure I:** General Arrangement

## SUMMARY STATEMENT

During the design of the C & P Aerospace Triton, few problems were encountered that necessitated changes in the configuration. After the initial concept phase, the aspect ratio was increased from 7 to 7.6 to produce a greater lift to drag ratio (L/D=13) which satisfied the horsepower requirements (118 hp using the Lycoming O-235 engine). The initial concept had a wing planform area of 134 sq. ft. Detailed wing sizing analysis enlarged the planform area to 150 sq. ft. without changing its layout or location. The most significant changes, however, were made just prior to inboard profile design. The fuselage external diameter was reduced from 54 to 50 inches to reduce drag to meet the desired cruise speed of 120 knots. Also, the nose was extended 6 inches to accommodate landing gear placement. Without the extension, the nosewheel received an unacceptable percentage (25%) of the landing weight. The final change in the configuration was made in accordance with the stability and control analysis. In order to reduce the static margin from 20 to 13 percent, the horizontal tail area was reduced from 32.02 to 25.0 sq. ft.

The Triton meets all the specifications set forth in the design criteria. If time permitted another iteration of the calculations, two significant changes would be made. The vertical stabilizer area would be reduced to decrease the aircraft lateral stability slope since the current value was too high in relation to the directional stability slope. Also, the aileron size would be decreased to reduce the roll rate below the current 106°/second. Doing so would allow greater flap area (increasing $C_{L_{max}}$) and thus reduce the overall wing area. C & P would also recalculate the horsepower and drag values to further validate the 120 knot cruising speed.

# 1.0 DESIGN CONSIDERATIONS

## 1.1 Aircraft Mission

The first and foremost design consideration was the mission of the aircraft, which was determined by the design specifications. The Triton was to be designed under FAR Part 23 to be used as a primary flight trainer. C & P Aerospace, however, has designed the aircraft for applications other than pilot training to increase the marketability (see 1.10 Mission Versatility). All aspects of the design were done in accordance to FAR Part 23 to ensure airworthiness, certification, and marketing.

## 1.2 Number of Persons

The Triton was designed for two side-by-side occupants and baggage. The side-by-side arrangement allows for good forward visibility for both occupants and little movement of the aircraft's c.g.. The decision for two occupants was also determined by the design specification of 120 knot cruise. Although consideration was given to a three-passenger configuration in the concept phase, marketability was questionable. Weight and balance calculations (see 2.9 Weight and Balance), however, show that a three-passenger configuration is possible with respect to c.g. movement (see also 1.10 Mission Versatility). Four passengers were not considered because of extensive c.g. travel, meeting the design requirements, marketability, and cost.

## 1.3 Engine Selection

C & P Aerospace decided to use the Lycoming O-235 engine for the Triton to accommodate several desired characteristics: low cost, light weight, easy maintenance, and availability. Calculations prove this engine is sufficient for the aircraft to attain 120 knots cruising speed at 83% power with the payload used throughout the design calculations (two 170-lb passengers and 60 lbs of baggage). For other missions, the Triton was designed to accept the Lycoming O-320 engine.

### 1.4 Wing Placement

After comparing the advantages of low wing and high wing aircraft, C & P decided to equip the Triton with a high wing. This choice was based on the most significant advantages of downward visibility, easy ingress and egress, longitudinal and lateral stability, and a gravity-fed fuel system. The wing is mounted cantilever to eliminate excess parasite drag from wing struts and add to the aesthetics. Structural and manufacturing considerations of this concept are discussed in section 2.4 Wing Design.

### 1.5 Landing Gear Configuration

The Triton was equipped with fixed, tricycle landing gear. Most general aviation aircraft have tricycle gear and training in an aircraft with commonly used gear would make transition between aircraft easier for the pilot. Visibility over the nose while taxiing was also considered and is usually better for tricycle gear aircraft than for conventional gear.

### 1.6 Certification Category

The Triton was designed to meet the requirements of the utility category. This decision, supported by the aircraft's cantilever high wing, increases both mission versatility and clientele. The spar arrangement inside the wing enables the Triton to withstand the g-loads for this category (+4.4/-1.8). Details about the spar arrangement are given in section 2.4 Wing Design.

## 1.7 Instrumentation

With consideration to flight training, the Triton was designed with IFR instruments and capacity for all-weather instrumentation. Since the student learning VFR flight in the Triton will be familiar with the aircraft, learning IFR operations will be less demanding. The expandability to all-weather instrumentation primarily concerns those who will use the aircraft for missions such as freight hauling and extensive cross-country flying.

## 1.8 Manufacturing and Maintenance

### 1.8.1 Manufacturing

In every aspect of the Triton's design, manufacturing and maintenance were considered. The aircraft is designed to be produced using common manufacturing methods such as stamping and flat-wrapping. Although the aircraft could have been designed requiring more complex and expensive techniques (compound curvatures, forging, etc.), the desired cost limit of $50,000 and ease of repair would be difficult to meet. Aircraft aluminum and steel assembled with rivets has proven sufficient for most general aviation aircraft and was therefore chosen for the Triton. Components such as wingtips and fairings are made of plastic, and the Triton's moveable surfaces are to initially be made of composites (most likely fiberglass around a foam core). This decision was made to test the manufacturing cost and complexity of such components compared to those made from aluminum.

### 1.8.2 Maintenance

Several concepts were used in the Triton to facilitate easy access to all components for repair or inspection. The wing was designed to be assembled in three sections, allowing the outboard panels to be removed if damaged. A large, forward-opening hood over the engine provides access to all components in front of the firewall. Panels underneath the wing and aft fuselage allow for inspection of electrical and control routing. Floor panels are also removable for control linkage and fuel tank selector valve repair.

### 1.9 Safety and Crashworthiness

C & P Aerospace employed several safety features in the Triton's design. First, the firewall is angled at the bottom so that the aircraft can slide along the ground without scooping into the earth during a forward, falling impact. Angling the firewall also reduces the chance of the fuselage buckling from the increased crash loads imposed by scooping (Raymer, Figure 8.15). Second, the Triton is equipped with energy-absorbing, "s"-frame, JAARS passenger seats. Finally, the seats are rigidly mounted to the structure so that they can properly absorb the crash loads without dislodging from the airframe. The type of occupant harnesses has not yet decided.

### 1.10 Mission Versatility

Reviewing the above considerations, there are numerous design parameters that allow the Triton to fulfill missions other than flight training. The capability for a larger engine, utility category certification, high wing configuration, and baggage capacity enable the aircraft to be used as a freight hauler or general aviation transport. A three-passenger version as preferred for Gemini flight training could also be made without making extensive changes in the airframe (see 2.9 Weight and Balance). Since the high wing allows for better downward visibility than a low wing, the Triton could also be used by forestry, fire, and law enforcement agencies as spotter aircraft. The high wing design also permits conversion from a land-based aircraft to an amphibian.

The Triton, as detailed in this report, is designed to satisfy the requirements for the primary flight trainer mission. All other missions would require further analysis of weight and balance, stability and control, and performance.

## 2.0 DETAILED DESIGN PROCESS

The design requirements for the aircraft specified two different flight scenarios:

1. Take off, climb to 5000 ft., cruise 500 n.mi. at 120 knots, loiter for 45 min. (reserve), land on 3000 ft. (maximum) runway.
2. Take off, climb to 1000 ft. and descend to landing 10 times, climb to 3000 ft., maneuver at 2-g for 30 min., cruise 100 n.mi., land on 3000 ft. runway.

Weight estimation for the conceptual design required that both missions be analyzed. The aircraft was to then be designed according to the flight condition that produced the heaviest aircraft gross weight. For the Triton, the first scenario produced the heavier weight. All calculations in this section are for the first flight scenario and were carried through the design process until refined by more accurate methods.

## 2.1 Preliminary Weight Estimation

The goal for the first part of this analysis was to iterate a takeoff gross weight, $W_0$, using the fuel fraction for each leg to determine the amount of fuel needed to complete the mission.

The first calculation determined the aircraft's estimated empty weight fraction using the following equation:

$$\left(\frac{W_e}{W_o}\right) = a + bW_o^{C1} A^{C2} \left(\frac{HP}{W_o}\right)^{C3} \left(\frac{W_o}{S}\right)^{C4} V_{max}^{C5} \qquad = \quad 0.6739$$

Where: From preliminary conceptual calculations:

| | | |
|---|---|---|
| $A$ | = 7.6 | (aspect ratio) |
| $HP/W_0$ | = 0.07 | (power-to-weight ratio) |
| $W_0/S$ | = 11.7 | (wing loading) |
| $V_{max}$ | = 152 mph | (1.1 x $V_{cruise}$) |

From Raymer Figure 6.2 (general aviation aircraft):

| | | | |
|---|---|---|---|
| a | = −0.25 | C3 | = 0.05 |
| b | = 1.14 | C4 | = −0.05 |
| C1 | = −0.2 | C5 | = 0.27 |
| C2 | = 0.08 | | |

To calculate the fuel fractions of the mission legs, the Mach number at take off and at climb had to be calculated as well as L/D for both cruise and loiter legs. $V_{take\ off}$ was assumed to be about 1.13 x $V_{stall}$. The parasite drag coefficient, Oswald efficiency factor, loiter and stall speeds, and loiter time were all taken from initial concept analysis. For fuel fraction ranges given from historical data, the highest value of the range was chosen due to the aircraft being a light, single-engine, two-seater (e.g. If $W_{i+1}/W_i$ = 0.95 to 0.99, 0.99 would be chosen). The mission leg fuel fractions are summarized in the following table.

| | | |
|---|---|---|
| $(W_1/W_0)$ | Taxi and Take off | 0.990 |
| $(W_2/W_1)$ | Climb to 5000 ft. | 0.996 |
| $(W_3/W_2)$ | Cruise 120 kts - 500 nm | 0.911 |
| $(W_4/W_3)$ | Loiter 45 min | 0.991 |
| $(W_5/W_4)$ | Descent | 0.995 |
| $(W_6/W_5)$ | Landing and Taxi back | 0.997 |

Table 1: Fuel Fraction Summary

The mission weight was determined using the desired payload of $(W_{crew} + W_{payload})$ = 400 lbs. This constitutes two, 170-lb persons and 30 lbs of baggage each. The total fuel fraction was determined by:

$$\left(\frac{W_f}{W_o}\right) = 1.06\left(1-\frac{W_x}{W_o}\right) = 0.1243$$

Where: $(W_x/W_0)$ = 0.8827 (product of all fuel fractions)

The initial estimate of fuel weight was then calculated by multiplying the total fuel fraction by the estimated gross weight $(W_0)$ of 1579 pounds (initial concept value) to yield a fuel weight of 196.27 lbs (32.7 gallons at 6 lbs/gal).

The final aircraft weight was the result of iterating the following formula:

$$W_o = \frac{W_{crew}+W_{payload}}{1-\frac{W_f}{W_o}-\frac{W_e}{W_o}} = 1785 pounds$$

Where:

| | | |
|---|---|---|
| $W_{crew}$ | = 340 pounds | (2 x 170 pounds/person) |
| $W_{payload}$ | = 60 pounds | (30 pounds/person) |
| $(W_f/W_0)$ | = 0.1243 | (total fuel fraction) |
| $(W_e/W_0)$ | = 0.6515 | (refined from iteration) |

The actual fuel weight from the new gross weight was found to be 221.8 lbs (37 gal at 6 lbs/gal).

## 2.2 Thrust-to-Weight Calculation

The equation below was solved for the horsepower required for cruise conditions since take off power (engine BHP) was known from engine data.

$$\left(\frac{HP}{W}\right)_{T.O.} = \left(\frac{V_{cruise}}{550\eta_p}\right)\left(\frac{1}{(L/D)_{cruise}}\right)\left(\frac{W_{cruise}}{W_{T.O.}}\right)\left(\frac{HP_{T.O.}}{HP_{req_{cruise}}}\right)$$

$$HP_{req\ cruise} = 98.9 \text{ hp}$$

| Where: | | | |
|---|---|---|---|
| | $V_{cruise}$ | = 202.54 fps | (120 knots) |
| | $\eta_p$ | = 0.8 | (initial estimation) |
| | $(L/D)_{cruise}$ | = 8.19 | (initial values) |
| | $W_{cruise}$ | = 1760 lbs | (using fuel fractions) |
| | $W_{T.O.}$ | = 1785 lbs | (gross weight) |
| | $HP_{T.O.}$ | = 118 hp | (engine data) |

The $HP_{req\ cruise}$ is lower than the 100 hp available at 5000 ft for the Lycoming O-235 as determined by curve-fitting Raymer Figure 5.2. The above $HP_{req\ cruise}$ to maintain $V_{cruise}$ of 120 knots was calculated at the start of the cruise leg and will decrease as fuel is used (e.g. total fuel weight drops). It was also concluded that since the $(HP/W)_{cruise}$ (calculated to be 0.0562) was slightly less than $(HP/W)_{available}$ (0.0568), the Lycoming O-235 was sufficient to attain cruise.

## 2.3 Fuselage, Wing, and Control Surface Sizing

### 2.3.1 Fuselage Length

From the gross weight, an approximate fuselage length was determined using the following equation:

$$\textit{Fuselage Length} = AW_o^C \quad = \quad 24.5 \textit{ ft.}$$

| | | | |
|---|---|---|---|
| Where: | $W_0$ | = 1785 lbs | (gross weight) |
| | A | = 4.37 | (from Raymer Table 6.3) |
| | C | = 0.23 | (from Raymer Table 6.3) |

2.3.2 Wing Area

The wing area was calculated from the coefficient of lift equation using $(C_l)_{takeoff}$ from the initial concept calculations:

$$S = \frac{W_o}{(1/2)\rho_{S.L.} V_{T.O.}^2 C_{L_{T.O.}}} \quad = \quad 150.6 \textit{ sq.ft.}$$

| | | | |
|---|---|---|---|
| Where: | Wo | = 1785 lbs | (gross weight) |
| | $\rho_{S.L.}$ | = 0.002378 slug/ft$^3$ | (sea level density) |
| | $V_{T.O.}$ | = 86.1 fps | (1.13 x $V_{stall}$ (45 kts)) |
| | $C_{lT.O.}$ | = 1.35 | (initial calculations) |

The wing span was then calculated as 33.8 ft. for an aspect ratio of 7.6 and the MAC assuming a rectangular wing was found to be 4.46 ft using b/S.

2.3.3 Control Surfaces

The tail surfaces were sized using the appropriate tail volume formulas. C & P Aerospace decided that the values for the tail volume coefficients in Raymer Table 6.4 (general aviation, single engine) were satisfactory for the Triton. Also required for these formulas were the moment arm distances for the surfaces. These were approximated from the wing and tail placement on the calculated fuselage length.

2.3.3.1 Vertical Stabilizer Area:

$$S_V = \frac{C_V\, b_W\, S_W}{L_V} = 12.8\ sq.ft.$$

Where: $C_v$ = 0.04 (vert. tail volume coef.)
$b_w$ = 33.8 ft (wing span)
$L_v$ = 15.9 ft (0.65 x fuse length)

2.3.3.2 Horizontal Stabilizer Area:

$$S_H = \frac{C_H\, \bar{c}\, S_W}{L_H} = 32.02\ sq.ft.$$

Where: $C_H$ = 0.7 (horiz. tail volume coef.)
$c_{bar}$ =4.57 ft (wing MAC)
$L_H$ = 14.67 ft (0.6 x fuse length)

2.3.3.3 Aileron Area:

The ailerons of the Triton are 50% of the wing span (0.5 x 33.8 ft = 16.9 ft total). Each aileron has a span of 8.45 ft (0.5 x 16.9 ft) and a chord of 20% that of wing. Each aileron's area is 7.54 sq.ft. making the total aileron area to be 10% of the wing's 150.6 sq.ft.. The 50% was suggested by Raymer Figure 6.3.

## 2.4 Wing Design and Airfoil Selection

### 2.4.1 Planform Geometry

C & P Aerospace decided to use a tapered wing for the Triton mainly for induced drag reduction and increased aesthetics compared to a rectangular wing. To simplify manufacturing and reduce torsional loads at the wing root, a leading edge sweep of 0° was chosen. To attain any reduction in induced drag the wing planform was swept forward at the trailing edge. Stall progression over the span of this type of configuration allows for more effective aileron area during stall. A linear taper ratio, $\lambda$, that would generate the calculated wing area of 150.6 sq.ft. (see 2.3.2 Wing Area) was then determined. Regarding wing twist, it was decided that for manufacturing and calculation purposes, using a single airfoil and geometrically twisting the wing would be the best way to accomplish 3° of washout. The washout is distributed from +1° at the root to -2° at the tip. Vortex-reducing NASA wing tips were added to increase the performance and add to the aesthetics.

A summary of the wing geometry is tabulated in the table below.

| | | |
|---|---|---|
| $c_r$ | Root Chord | 5.7 ft |
| $c_t$ | Tip Chord | 3.2 ft |
| $\lambda$ | Taper Ratio | 0.561 |
| $c_{bar}$ | MAC | 4.57 ft |
| $y_{bar}$ | Distance to MAC | 16.9 ft |
| $\Lambda_{LE}$ | Sweep - Leading Edge | 0° |
| $\Lambda_{c/4}$ | Sweep - Quarter Chord | -2.12° |
| $\Lambda_{c/2}$ | Sweep - Half Chord | -4.23° |
| $\Lambda_{TE}$ | Sweep - Trailing Edge | -8.41° |
| AR | Aspect Ratio | 7.6 |
| S | Wing Area | 33.8 $ft^2$ |

Table 2: Wing Geometry Parameters

2.4.2 Wing/Fuselage Location and Interface

The wing of the aircraft is located on the fuselage so that the quarter MAC coincides with the aircraft's c.g.. This was not difficult to design since there is little distance between the root quarter-chord and the quarter MAC. This is the same location that was used when calculating the tail surface areas (see 2.3.3.1 Vertical Stabilizer Area and 2.3.3.2 Horizontal Stabilizer Area).

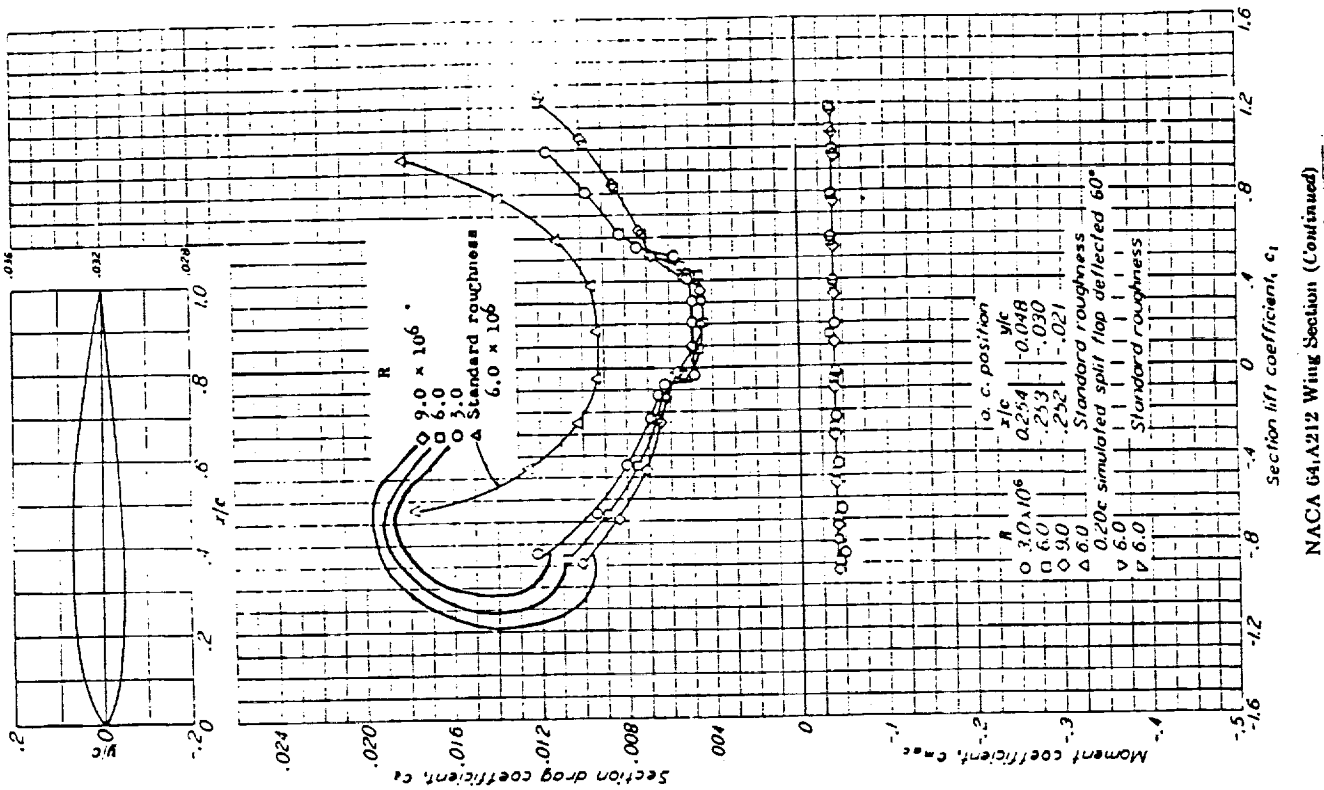
To eliminate the external struts common to most high winged aircraft, the wing has a front and rear spar that continue over the top of the fuselage. The fuselage of the Triton is suspended from the spars by four large, shear pins (two on each side of the aircraft) passing through the neutral axis of each spar into primary airframe crossmembers. A visual description of this is given in section 2.8 Structural Arrangement.

### 2.4.3 Airfoil Selection

The airfoil selected for the Triton's main wing is the NACA $64_1$A212 (Figure 1). Although there are a myriad of airfoils applicable to such an aircraft, several considerations were made to narrow the choice down to one. These considerations were stall behavior, $C_{lmax}$ and design $C_l$, $C_{Mac}$, $C_d$, manufacturing complexity, and section thickness.

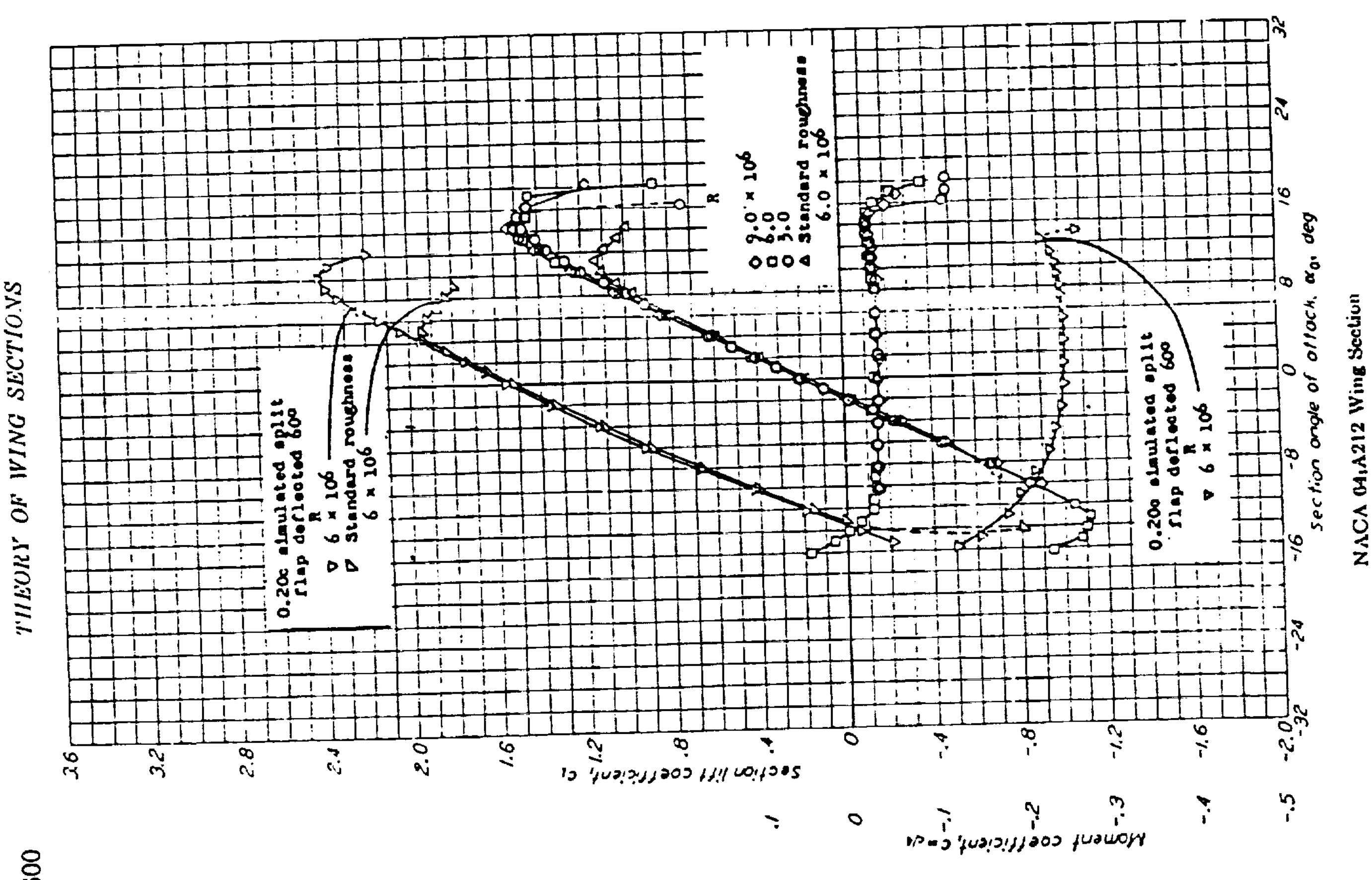
With respect to stall behavior, an airfoil was chosen with a stall that was not too abrupt, yet not too gentle. If a student pilot encounters a sharp stall in another aircraft after learning how to contend with smooth, gentle stalls, the pilot may panic, under-correct, and make matters worse (especially on landing). Conversely, if the student is used to reacting quickly in sharp stalls, the gently-stalling aircraft will most likely not behave as the sharp-stalling aircraft and the student could possibly over-correct, again, making matters worse. Examination of these instances led to an airfoil with a stall range of about $4^\circ$ angle of attack and a stall angle of between $10^\circ$ and $12^\circ$.

From the initial conceptual calculations, the assumed $C_{lmax}$ (flapped) was 1.7 and $C_{lcruise}$ was calculated to be 0.28. An airfoil was selected that would satisfy both values. From examining the NACA airfoils, an airfoil with $C_{ldesign}$ of 0.2 or 0.4 was thought sufficient. $C_{Mac}$ should be low enough (less than -0.1) so that a large horizontal tail area is not needed to counteract the wing




NACA $64_1$A212 Wing Section (Continued)






NACA $64_1$A212 Wing Section

**Figure 1: NACA $64_1$A212 Airfoil Data**

pitching moment, and $C_d$ should be low at cruise $C_l$.

Since the airfoil has to be easy to manufacture, any section with a sharp trailing-edge cusp or small leading-edge radius was not considered. With respect to section thickness, there had to be enough space inside the wing for fuel tanks, control linkages, flap mechanisms, and spars. A 12% thick airfoil could be used since there is no landing gear to retract into the wing structure.

The NACA $64_1$-212 had all of the desired qualities. A closer look at the airfoil, however, revealed a thin trailing edge that would be difficult to manufacture. Therefore, the "A" version of this airfoil was selected without compromising any characteristics other than the trailing edge. The "A" designation indicates that the trailing edge of the airfoil was straightened out for easy construction. The use of a natural laminar flow (NLF) airfoil was discussed, but C & P Aerospace decided to use an airfoil that was already accepted in the general aviation community.

### 2.4.4 Wing Aerodynamic Characteristics

The aerodynamic analysis of the wing was carried out after the configuration was designed. The values generated by these calculations were used extensively throughout the stability and control evaluation.

The first values, the wing lift curve slope, was determined as follows. Since the maximum speed of the aircraft does not exceed Mach 0.3, the compressibility term, ß, was approximately 1.0. The airfoil efficiency term, $\eta$, relates ß to the airfoil's 2-D lift curve slope, $C_{l\alpha}$, by $C_{l\alpha}/(2\pi/\beta)$. $C_{l\alpha}$ from the airfoil data was 0.1/degree. This gave a $\eta$ value of 0.0153/degree. The fuselage width, d, at the wing root was 4.17 ft which determined the wing/fuselage interference factor, F, by 1.07(1 + (d/b)). F was found to be 1.35. The lift curve slope equation is below.

$$C_{L_\alpha} = \frac{2\pi AR}{2+\sqrt{4+\frac{AR^2\beta^2}{\eta^2}\left(1+\frac{\tan^2\Lambda_{t\,max}}{\beta}\right)}}\left(\frac{S_{exposed}}{S_{ref}}\right)F \qquad = 0.113/degree$$

| | | | |
|---|---|---|---|
| Where: | $AR$ | = 7.6 | (initial calculations) |
| | $\beta$ | = 1 | (compressibility factor) |
| | $\eta$ | = 0.0153/degree | (airfoil efficiency term) |
| | $\Lambda_{t\,max}$ | = −3.2 degrees | (sweep at airfoil max. thick) |
| | $S_{exposed}$ | = 125.4 sq.ft. | (exposed wing area) |
| | $S_{ref}$ | = 150.6 sq.ft. | (total wing area) |
| | $F$ | = 1.35 | (fuselage lift factor) |

Using the calculated wing lift curve slope $C_{l\,max\,clean}$ was found by:

$$C_{L\,max_{clean}} = 0.9\ C_{l\,max}\cos\Lambda_{c/4} \qquad = 1.35$$

| | | | |
|---|---|---|---|
| Where: | $C_{l\,max}$ | = 1.5 | (from airfoil data) |
| | $\Lambda_{c/4}$ | = −2.12 degrees | (wing geometry) |

Using the values for $C_{L\,max}$ and $C_{L\alpha}$ determined from above, the stall angle of attack was found by:

$$\alpha_{stall} = \frac{C_{L\,max}}{C_{L_\alpha}} + \alpha_o + \Delta\alpha_{C_{L\,max}} \qquad = 10.7°$$

| | | | |
|---|---|---|---|
| Where: | $\alpha_0$ | = −2 degrees | (airfoil data) |
| | $\Delta\alpha_{CL\,max}$ | = 0.7 degrees | (Raymer Figure 12.10) |

To minimize design complexity, C & P Aerospace decided to equip the Triton with single-slotted flaps. The change in the zero lift angle and in $C_{L\,max}$ can be computed using the following equations. In both equations, $S_{flapped}$ is determined by the wing area affected by flap deployment, 25.2 sq.ft..

Change in zero lift angle:

$$\Delta\alpha_{oL} = \Delta\alpha_{oL_{airfoil}}\left(\frac{S_{flapped}}{S_{ref}}\right)\cos\Lambda_{H.L.} \qquad = \quad -3.97\,°$$

Where: $\Delta\alpha_{oLairfoil} = -12°$ (wing data)
$S_{flapped} = 25.2$ sq.ft. (from wing geometry)
$S_{ref} = 150.6$ sq.ft. (total wing area)
$\Lambda_{H.L.} = -7.5°$ (sweep at flap hinge line)

Change in $C_{Lmax}$:

$$\Delta C_{L\,max} = \Delta C_{lmax}\left(\frac{S_{flapped}}{S_{ref}}\right)\cos\Lambda_{H.L.} \qquad = \quad 0.43$$

Where: $\Delta C_{lmax} = 1.3$ (Raymer Tbl 12.2 - slttd flap)

From these values the plot of Coefficient of Lift vs. Angle of Attack (Figure 2) was made.

## 2.5 Preliminary Design of Fuselage Shape and Cross-Sections

Informal scaled drafts of the fuselage shape and size were made to accommodate the engine, passengers, baggage, landing gear, instrument panel, and other significant internal components based on the calculated wing geometry and placement, tail surface sizes, and fuselage length. During refinement of the fuselage shape, cross-sections at different stations were considered while taking into account manufacturing, weight, cost, and load distribution. After pro and con analysis of several types of cross-sections, C & P Aerospace decided to use a conical tail section transitioning to a rectangular cabin area based on aesthetics.

## 2.6 Engine Installation and Propeller Sizing/Noise Level

### 2.6.1 Engine Installation

The engine compartment and firewall section were proportioned to accommodate the Lycoming O-320 engine and accessories. This also included designing the

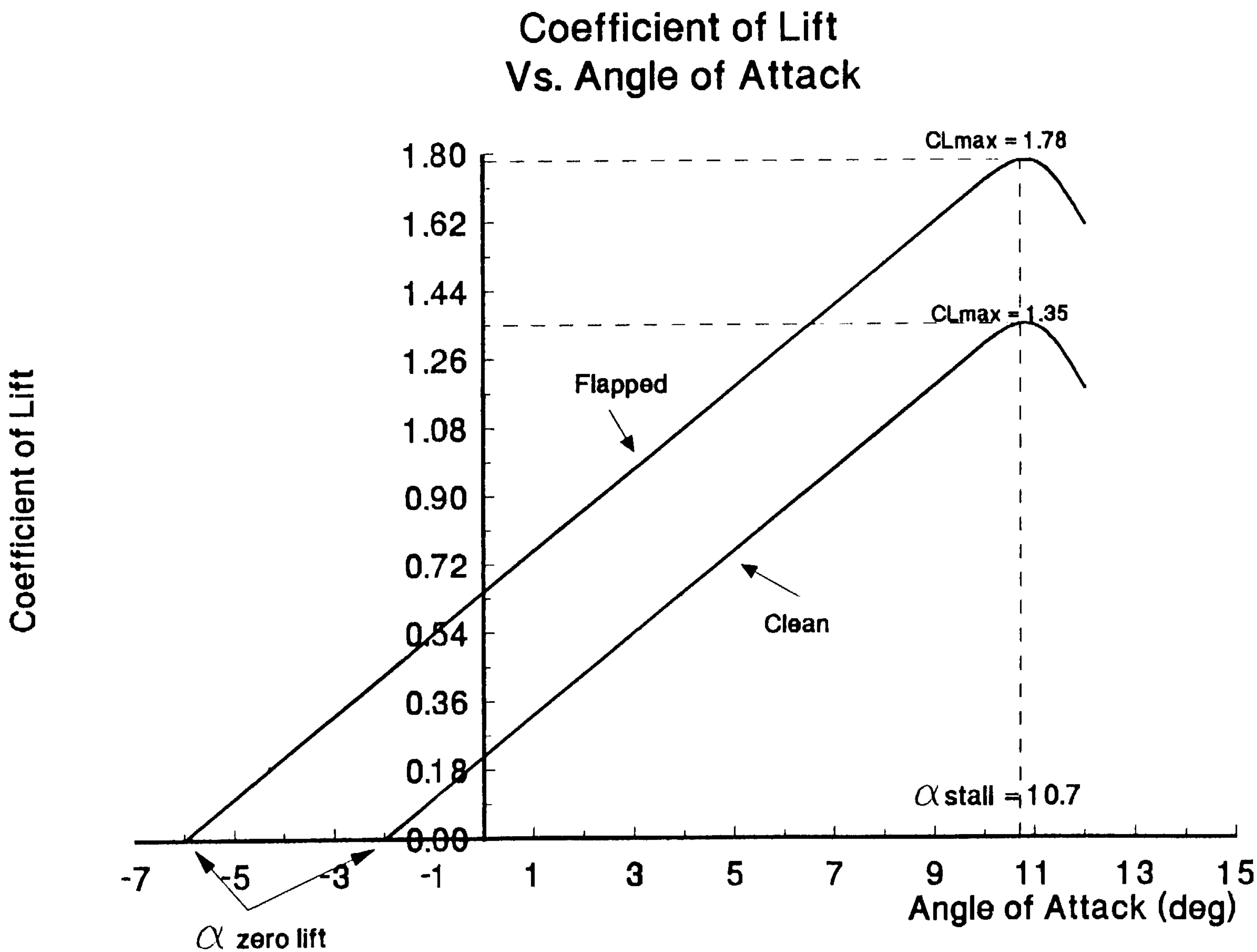

Coefficient of Lift
Vs. Angle of Attack
Coefficient of Lift
1.80
1.62
1.44
1.26
1.08
0.90
0.72
0.54
0.36
0.18
0.00
-7
-5
-3
-1
1
3
5
7
9
11
13
15
Angle of Attack (deg)
CLmax = 1.78
CLmax = 1.35
Flapped
Clean
α stall = 10.7
α zero lift

Figure 2

engine mount, exhaust tubes, carburetor intake duct, and cooling air vents. This proved that the fuselage shape ahead of the firewall approximated in section 2.5 provided sufficient space.

After determining the precise location of the engine in the compartment, the engine mount was designed. The dynafocal, symmetrical mount made of 0.5-inch O.D. 4130 steel tubing was designed to transfer torsional and lateral loads to the airframe. The mount was also designed to provide adequate space for accessories between the firewall and the rear of the engine. Later modifications were made to the mount to account for nosewheel attachment.

The exhaust pipes were designed next. Simplicity and routing were of primary concern. The exhaust system is comprised of two independent assemblies, one for each side of the engine, that extend from the bottom of the cowling with no bends or curves. There was a deliberate change in diameter from the manifold tubes to the exhaust tubes to reduce backpressure into the engine. No muffler was used in the system due to inherent power reduction, but cabin heat was provided by channeling exhaust into the heater assembly.

For the sake of simplicity and aesthetics, a single, large, elliptical, cooling intake was made to allow unobstructed airflow over the cylinders. The exit vent was simply a large cutout at the bottom of the cowling through which the exhaust tubes pass. Due to the location of the carburetor with respect to the engine, C & P Aerospace made the carburetor intake duct a small, square opening located in the lower cowling.

2.6.2 Propeller Sizing and Noise

This analysis entailed sizing a propeller to satisfy the performance specified in the mission and ensuring the propeller noise level generated conformed to FAR part 36.

2.6.2.1 Propeller Sizing

The initial diameter, d, was first calculated as 6.04 feet (72 inches) using the equation for a two-bladed propeller of d = $22(\sqrt[4]{HP})$. This was then increased to 6.166 feet (74 inches) due to general aviation standards and availability. In order to approach and maintain the 120 knot cruising speed, a cruise propeller instead of a climb propeller was used on the Triton. To determine the number of blades and appropriate blade angle for the propeller, the power coefficient, $C_p$, and advance ratio, J, were determined.

Power Coefficient, $C_p$:

$$C_p = \frac{550\ BHP}{\rho_{alt}\ n^3 D^5} = 0.0294$$

Where:

| | | | |
|---|---|---|---|
| BHP | = 98.9 hp | (estimated cruise horsepower) |
| $\rho_{alt}$ | = 0.002054 slug/ft$^3$ | (density at cruise altitude) |
| n | = 38.33 rps | (assumed 2300 rpm/60) |
| D | = 6.166 ft | (propeller diameter) |

Advance Ratio, J:

$$J = \frac{V}{n\ D} = 0.857$$

Where: V = 202.54 fps (cruise velocity)

Using the calculated values of $C_p$ and J, the most applicable graph was the Hamilton-Standard graph for a two-bladed propeller (a two-bladed propeller was also chosen because of weight, cost, and availability). The propeller corresponding to t he determined values of $C_p$ and J had a 20° blade angle ß (at 75% radius) and an 86% efficiency $\eta_p$ at cruise conditions. At stall speed

(45 knots) $\eta_p$ drops to 45%. The prop size is then determined using the rated pitch equation:

$$\textit{Rated Pitch} = \pi\ D\ (0.75)\ \tan\beta_{75\%} \quad = \ 63.5\ \textit{inches}$$

Where: ß = 20° (blade angle)
D = 6.166 ft (propeller diameter)

The final cruise propeller size for the aircraft was 74 x 64.

2.6.2.2 Propeller Noise

The far-field noise level was determined using SAE report AIR 1407. The calculations in this report determined the far-field sound pressure level for the propeller used and provided a comparison for FAR Part 36.301. The total sound pressure level (SPL) for the designed 74 x 64 propeller was calculated as 61.7 dBA. FAR Part 36 states that:

$$\textit{Far-Field Noise Level} \le 68 + \left[\frac{W_o - 1320}{165}\right] \quad \le \ 70.8\ dBA$$

Where: $W_0$ = 1785 lbs (aircraft gross weight)

Since 61.7 dBA ≤ 70.8 dBA, the propeller satisfied FAR noise requirements.

## 2.7 Landing Gear Layout

Before any equations could be used approximation of the longitudinal and vertical c.g. location of the aircraft was done. For longitudinal location, it was assumed that our c.g. was located at 25% MAC. The aircraft configuration showed that the c.g. could only move slightly forward since there are no variable loads ahead of the 25% point, thus the forward limit was chosen to be 7% MAC ahead of the 25% point. The fuel and baggage, however, are located behind the 25% point and would cause a greater shift. The aft c.g. shift was then chosen to be 10% MAC behind the 25% point. As for the ground height of the c.g., the pilot's waist location used in the initial fuselage layout was chosen. The angles considered in the gear placement were the tipback angle, the static tail down angle, the maximum stall angle, and the overturn angle.

The tipback angle is measured between the vertical axis and the aft c.g. location with respect to the main gear ground contact point. The static tail down angle is between the ground and the most rearward, bottom point on the aircraft as taken from the main gear contact point. The maximum stall angle is that of the wing but measured from the ground line with respect to the contact point. These angles are related such that if the aircraft is loaded to its aft c.g. limit, approaches for landing, and stalls the tail will not strike the ground and stay there. This dictates that the tipback angle must be slightly greater than the stall angle so that the aft c.g. is still ahead of the contact point even when rotated to stall attitude. The static tail down angle should be greater than the others since this will give tail/ground clearance on landing even at high angles of attack. From Figure 3 below, $\alpha_{stall} = 9.7°$ (approximate landing configuration), $\alpha_{tipback} = 10°$, and $\alpha_{static} = 12°$.

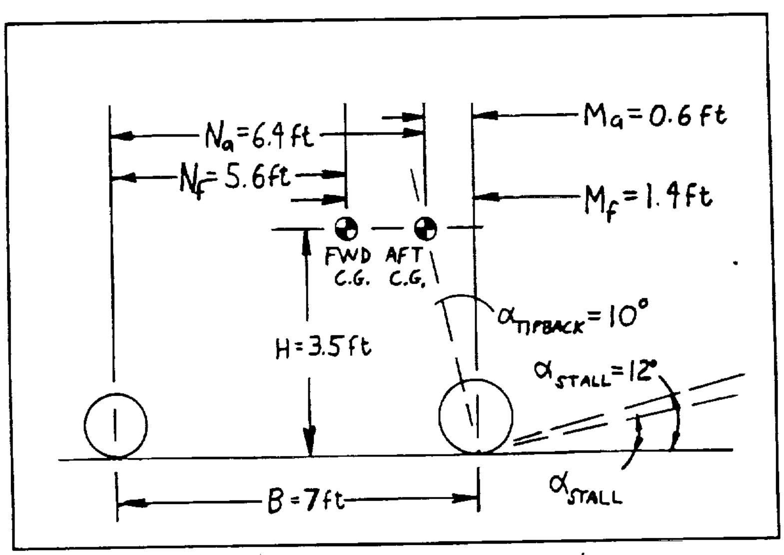



**Figure 3**: Landing Gear Arrangement

The overturn angle, $\Psi$, is measured as the angle required to tip the aircraft over on the nosewheel and a main gear wheel (like a tricycle tipping to one side). This angle is usually encountered during high-speed taxiing turns or crosswind landings and should not exceed 63° (FAR Part 23). From the distances above, this angle was determined by:

$$\Psi = \tan^{-1}\left[\frac{H}{N_f \sin\left[\tan^{-1}\left(\frac{W/2}{N_f\, M_f}\right)\right]}\right] \qquad = 51.3°$$

Where: W = 8.1 ft (estimated wheel track)

Since 51.3° ≤ 63°, this satisfied the FAR Part 23 requirement.

Tire loads were determined from the gear geometry and are summarized below.

| Formula | Title | Load (lbs) | $W_0$ Percentage |
|---|---|---|---|
| $(W_0N_a)/B$ | Max. Static Load - Main | 1632 | 91.4 |
| $(W_0M_f)/B$ | Max. Static Load - Nose | 357 | 20.0 |
| $(W_0M_a)/B$ | Min. Static Load - Nose | 153 | 8.6 |

**Table 3: Gear Loading ($W_0$ = 1785 lbs)**

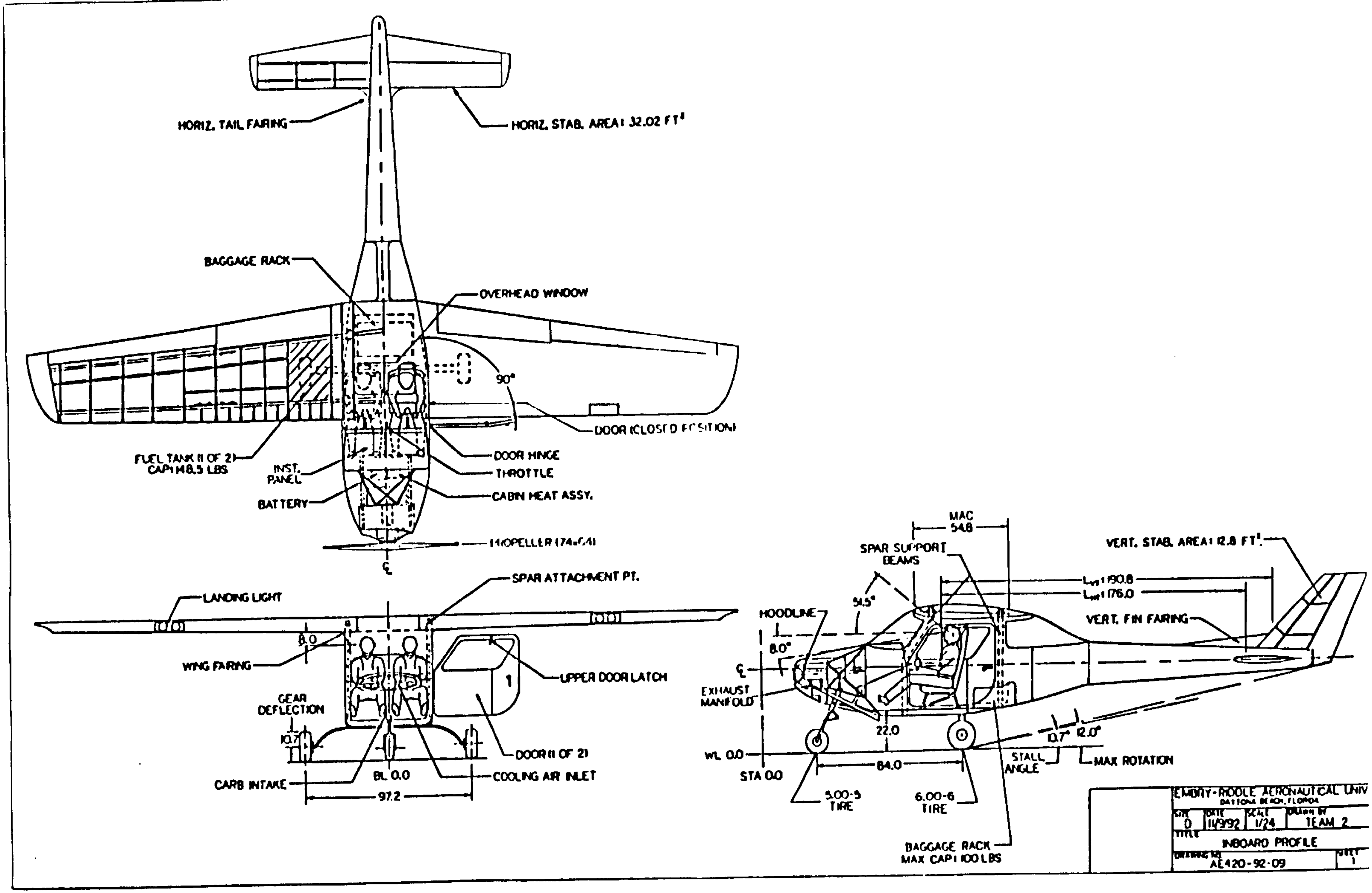
The maximum load on the nosewheel should not exceed 20% of the gross weight for structural and taxiing considerations. Table 3 and Figure 3 are the product of many iterations which eventually resulted in lengthening the aircraft's nose by six inches to achieve the 20% nosewheel load. This also led to redesigning the engine mount to support the nosewheel assembly.

Tire sizes were figured using the loads from Table 3 in conjunction with the wheel size equation: Diameter or Width = $A(W_0/2)^B$ (A and B taken from Raymer Table 11.1). The Triton will use standard 17.5x6.00-6 main gear tires supported on a steel, elliptical cross-sectioned, spring strut (chosen for simplicity, high strength, and low drag) and a smaller 14.25x5.00–5nose gear tire supported by a steerable, oleo strut fixed to the engine mount.

## 2.8 Structure and Inboard Profile Design

After the engine and landing gear location were determined, the fuselage layout and cross-section were used to design the internal structure of the aircraft. The wing and tail surface structures were also developed, as well as their integration into the fuselage. The complete structural arrangement combined with a refined fuselage configuration produced Figure 4, the Inboard Profile. Cost, construction complexity, and crashworthiness were major considerations during this phase of the design.

There were many design aspects that had to be considered while designing the fuselage shape. The initial external width of the fuselage was 54 inches. In the interest of drag reduction, this was decreased to 50 inches without adverselv affecting the internal width. The height of the aircraft above the ground, dictated by landing gear height, determined cabin headroom and floor space for control linkage. The cabin height was then measured to comply with the headroom standard "8 inches from eyes to ceiling". Occupant position in the cabin was determined to provide adequate line-of-sight and viewing angles through the large side windows and sunroof. Consideration was also given to door size and placement. Large doors, framed by surrounding stringers, were designed for easy access to the baggage area. Since the nosewheel oleo and struts were fixed directly to the engine mount, additional supports were needed to distribute the extra loads developed by the nosewheel into the airframe. The two beams which give shape to the cockpit and suspend the fuselage from the spars had to be routed and designed considering ease of


HORIZ. TAIL FAIRING
HORIZ. STAB. AREA: 32.02 FT²
BAGGAGE RACK
OVERHEAD WINDOW
90°
DOOR (CLOSED POSITION)
FUEL TANK (1 OF 2)
CAP: 148.5 LBS
INST. PANEL
BATTERY
DOOR HINGE
THROTTLE
CABIN HEAT ASSY.
PROPELLER (74x64)
℄
LANDING LIGHT
SPAR ATTACHMENT PT.
8.0
WING FAIRING
UPPER DOOR LATCH
1
GEAR DEFLECTION
10.7
DOOR (1 OF 2)
CARB INTAKE
BL 0.0
COOLING AIR INLET
97.2
MAC
54.8
SPAR SUPPORT BEAMS
VERT. STAB. AREA: 12.8 FT²
$L_{VT}$: 190.8
$L_{HT}$: 176.0
VERT. FIN FAIRING
51.5°
HOODLINE
8.0°
℄
EXHAUST MANIFOLD
22.0
WL 0.0
STA 0.0
84.0
10.7°
12.0°
STALL ANGLE
MAX ROTATION
5.00-5 TIRE
6.00-6 TIRE
BAGGAGE RACK
MAX CAP: 100 LBS
EMBRY-RIDDLE AERONAUTICAL UNIV
DAYTONA BEACH, FLORIDA
SIZE D
DATE 11/9/92
SCALE 1/24
DRAWN BY TEAM 2
TITLE INBOARD PROFILE
DRAWING NO. AE420-92-09
SHEET 1

manufacture, load path, cost, and weight. The cabin floor was designed to distribute landing loads from the gear, crew, baggage, and structure such that FAR crashworthiness standards were satisfied. Since the empennage is an aluminum sheet flat-wrapped to form a conical cross section, only a few formers and stringers were required for easy construction, maintenance, load distribution, and tail surface mounting.

Inside the all-aluminum wing, the two box spars are paralleled by two stringers along the top and bottom surfaces stiffen the skin. False ribs were used along the inboard eight feet of each wing panel to strengthen the leading edge and to help retain shape. The fuel section is an area between the spars where an aluminum fuel tank is placed. The tips of the wing and stabilizers are molded plastic to attain the desired compound curves.

The tail surfaces are similar in structure to the wing and rely on stressed skin for rigidity. As mentioned in section 1.8.1 Manufacturing, all of the moveable surfaces including ailerons and flaps were to be made of composites.

### 2.9 Weight and Balance Analysis

Since the exact location of each aircraft component could now be determined from Figure 4, a detailed weight and balance estimation was performed. Equations from Raymer Chapter 15 were used to derive the weights of major structural assemblies and components. Weights for components not included in the equations were estimated. Each component's location and weight were entered into a LOTUS spreadsheet that calculated the aircraft's weight and c.g. position as the removable loads were varied. This was to establish the true weight of the aircraft and to define the actual forward and aft c.g. limits

A summary of the equation results are tabulated in Table 4 below. Following Table 4 are the spreadsheet calculations. The c.g. range graphed with respect to percent MAC is Figure 5.

From the spreadsheet analysis, the Case 3 loading scenario is indicative of the design criteria (two 170 lb passengers with 30 lbs baggage each). The gross weight for this case was found to be 1898 lbs compared to the initial weight calculation (see 2.1 Weight Fraction Estimation) of 1785 lbs. The c.g. landed at 28% MAC (STA 104.7) which was 2.2 inches aft of the estimated 25% MAC location. As Figure 5 shows, the c.g. travel for the various loading conditions allows for a possible three-passenger version of the Triton (justified by Case 1). Calculated c.g. range was between 28% (empty aircraft c.g.) and 31.5% (most aft limit) yielding a 3.5% aft movement. Since this was less than the initial estimation of 10% used in the landing gear layout (section 2.7), the c.g. position was acceptable for the previously determined gear arrangement.

| Component | Weight (lbs) | Raymer Eqn. Number |
|---|---|---|
| Wing Structure | 259.0 | 15.46 |
| Horizontal Stabilizer | 21.6 | 15.47 |
| Vertical Stabilizer | 13.8 | 15.48 |
| Fuselage Structure | 188.8 | 15.49 |
| Main Landing Gear | 163.1 | 15.50 |
| Nose Gear | 39.6 | 15.51 |
| Fuel System | 19.6 | 15.53 |
| Flight Controls | 31.8 | 15.54 |
| Brake System | 1.8 | 15.55 |

Table 4: Component Weight Equation Results

| COMPONENT | WEIGHT : | STA | MOMENT : | WL | MOMENT |
|---|---|---|---|---|---|
| SPINNER | 1.3 | 15.6 | 20.28 | 45.6 | 59.28 |
| PROPELLER | 19 | 16.2 | 307.8 | 45.6 | 866.4 |
| ENGINE | 225 | 28.5 | 6412.5 | 40.8 | 9180 |
| BATTERY | 22.6 | 54 | 1220.4 | 54 | 1220.4 |
| ENG. MOUNT/BOLTS | 10.4 | 50.4 | 524.16 | 40.8 | 424.32 |
| EXHAUST SYS. | 7 | 48 | 336 | 32.4 | 226.8 |
| COWLING (UPPER) | 3 | 44.4 | 133.2 | 50.4 | 151.2 |
| COWLING (LOWER) | 5 | 43.2 | 216 | 34.8 | 174 |
| HINGES* | 3 | 18 | 54 | 49.2 | 147.6 |
| NSW. OLEO | 26 | 39 | 1014 | 24 | 624 |
| NOSEWHEEL | 9.7 | 31.2 | 302.64 | 6 | 58.2 |
| MISC. NSW. STRUTS | 4 | 48 | 192 | 25.8 | 103.2 |
| MAIN GEAR STRUT | 90 | 115.2 | 10368 | 18 | 1620 |
| MAIN GEAR TIRES | 70 | 115.2 | 8064 | 8.4 | 588 |
| MAIN GEAR BRAKES | 3.3 | 115.2 | 380.16 | 8.4 | 27.72 |
| FUSELAGE STRUC. | 188.8 | 156 | 29452.8 | 45.6 | 8609.28 |
| DOORS* | 14 | 108 | 1512 | 39.6 | 554.4 |
| DOOR WINDOWS* | 4 | 108 | 432 | 61.2 | 244.8 |
| WINDSHIELD | 8.4 | 78 | 655.2 | 68.4 | 574.56 |
| REAR WINDOW* | 10 | 162 | 1620 | 63.6 | 636 |
| BAGGAGE RACK | 1.8 | 132 | 237.6 | 30 | 54 |
| SKYLIGHT | 4 | 120 | 480 | 81.6 | 326.4 |
| TOTAL WING | 259 | 114 | 29526 | 78 | 20202 |
| HORIZ. TAIL | 21.6 | 284.4 | 6143.04 | 44.4 | 959.04 |
| VERT. TAIL | 13.8 | 318 | 4388.4 | 58.8 | 811.44 |
| LANDING LIGHTS* | 1 | 95.4 | 95.4 | 79.2 | 79.2 |
| FLAP MECHANISM* | 12 | 125.5 | 1506 | 76.8 | 921.6 |
| FUEL TANKS* | 14.8 | 116.5 | 1724.2 | 77 | 1139.6 |
| WING ATTACH PINS* | 8 | 118.8 | 950.4 | 78 | 624 |
| FUEL SYSTEM | 4.7 | 80.4 | 377.88 | 50.4 | 236.88 |
| INSTRUMENT PANEL | 12 | 82.8 | 993.6 | 56.4 | 676.8 |
| INSTRUMENTS | 62 | 73.2 | 4538.4 | 54 | 3348 |
| CONTROL YOKE* | 10 | 90 | 900 | 49.2 | 492 |
| RUDDER/BRAKE PDLS* | 12 | 69 | 828 | 27.6 | 331.2 |
| SEATS* | 60 | 114 | 6840 | 42 | 2520 |
| CONTROL LINKAGE | 12 | 78 | 936 | 34.8 | 417.6 |
| SEATBELTS* | 4 | 114 | 456 | 40.8 | 163.2 |
| MISC. CONTROLS | 7 | 90 | 630 | 48 | 336 |
| EMER LOCATR. XMTTR | 4 | 186 | 744 | 36 | 144 |
| WING/FUSE FILLETS* | 5 | 119 | 595 | 73.2 | 366 |
| H. TAIL FAIRINGS* | 6 | 271.2 | 1627.2 | 44.4 | 266.4 |
| V. TAIL FAIRINGS | 2 | 279.6 | 559.2 | 56.4 | 112.8 |

| | | | |
|---|---|---|---|
| EMPTY AIRCRAFT: | 1261.2 LBS | 128293.5 IN LB | 60618.32 IN LB |
| CG LOC: | STA 101.7233 | WL 48.064 | |

MOST AFT ALLOWABLE CG STA: 108.712

---- CASE 1 ----

| REMOV. LOADS | WEIGHT : | STA | MOMENT : | WL | MOMENT |
|---|---|---|---|---|---|
| PILOT | 100 | 104.4 | 10440 | 45.6 | 4560 |
| PASSENGER | 0 | 104.4 | 0 | 45.6 | 0 |
| BAGGAGE | 220 | 132 | 29040 | 36 | 7920 |
| FUEL | 237 | 114 | 27018 | 78 | 18486 |

FINAL CG: STA 107.1342 WL 50.37087 A/C WGT: 1818.2

---- CASE 2 ----

| REMOV. LOADS | WEIGHT : | STA | MOMENT : | WL | MOMENT |
|---|---|---|---|---|---|
| PILOT | 220 | 104.4 | 22968 | 45.6 | 10032 |
| PASSENGER | 280 | 104.4 | 29232 | 45.6 | 12768 |
| BAGGAGE | 0 | 132 | 0 | 36 | 0 |
| FUEL | 12 | 114 | 1368 | 78 | 936 |

FINAL CG: STA 102.5612 WL 47.5718 A/C WGT: 1773.2

---- CASE 3 ----

| REMOV. LOADS | WEIGHT : | STA | MOMENT : | WL | MOMENT |
|---|---|---|---|---|---|
| PILOT | 170 | 104.4 | 17748 | 45.6 | 7752 |
| PASSENGER | 170 | 104.4 | 17748 | 45.6 | 7752 |
| BAGGAGE | 60 | 132 | 7920 | 36 | 2160 |
| FUEL | 237 | 114 | 27018 | 78 | 18486 |

FINAL CG: STA 104.6926 WL 50.97899 A/C WGT: 1898.2

---- CASE 4 ----

| REMOV. LOADS | WEIGHT : | STA | MOMENT : | WL | MOMENT |
|---|---|---|---|---|---|
| PILOT | 170 | 104.4 | 17748 | 45.6 | 7752 |
| PASSENGER | 0 | 104.4 | 0 | 45.6 | 0 |
| BAGGAGE | 45 | 132 | 5940 | 36 | 1620 |
| FUEL | 237 | 114 | 27018 | 78 | 18486 |

FINAL CG: STA 104.4825 WL 51.64389 A/C WGT: 1713.2

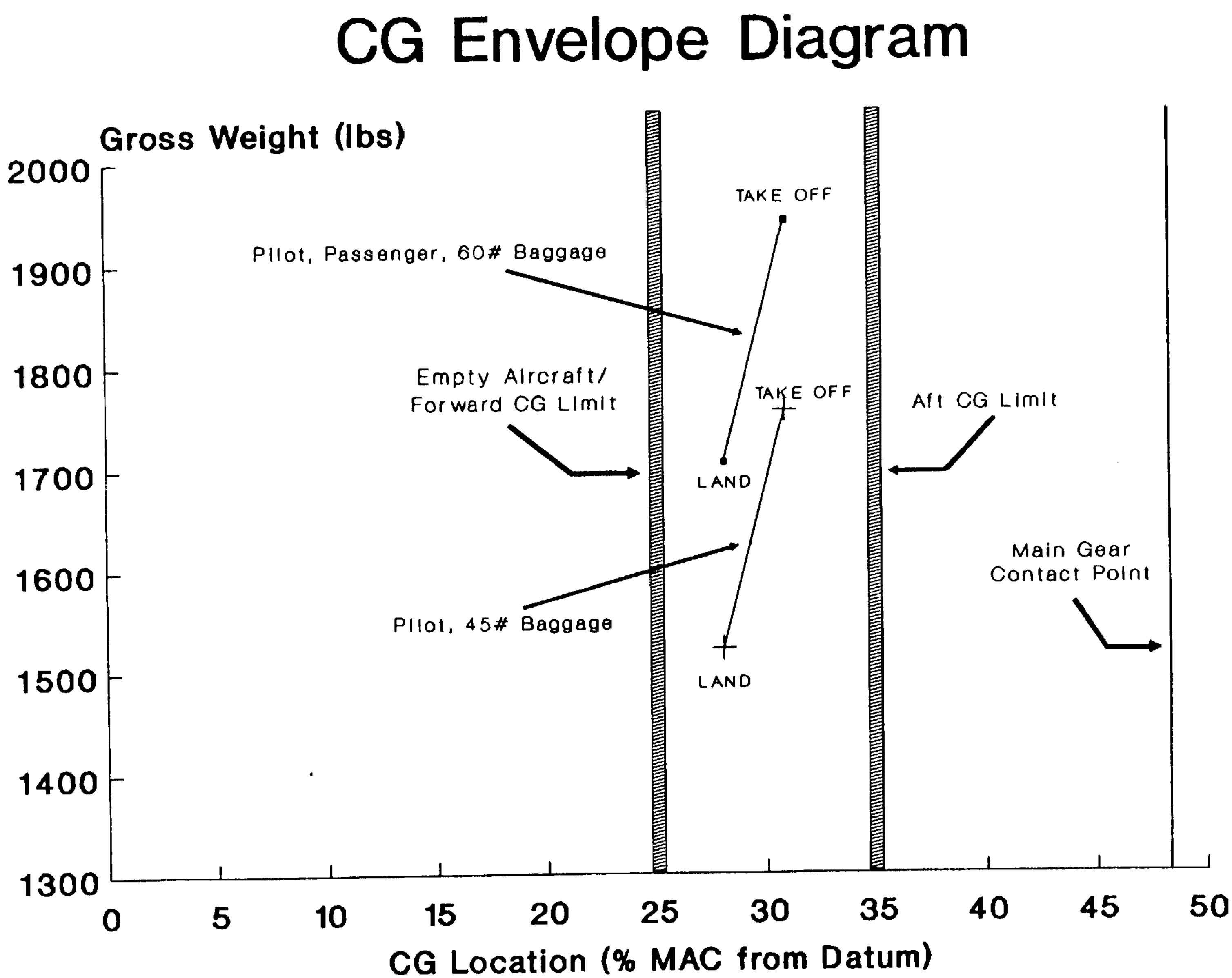

CG Envelope Diagram
Gross Weight (lbs)
2000
1900
1800
1700
1600
1500
1400
1300
0
5
10
15
20
25
30
35
40
45
50
CG Location (% MAC from Datum)
TAKE OFF
LAND
TAKE OFF
LAND
Pilot, Passenger, 60# Baggage
Empty Aircraft/
Forward CG Limit
Pilot, 45# Baggage
Aft CG Limit
Main Gear
Contact Point

Figure 5

## 2.10 Stability and Control

The Triton's static stability and control parameters were then calculated. Longitudinal, directional, and lateral stability were evaluated, as well as roll rate and spin recovery. Only the most significant parameters from the calculations will be described in detail.

For longitudinal stability, the three most significant calculated parameters are the neutral point location, static margin, and the horizontal tail incidence. The aircraft's neutral point was determined by taking measurements from the inboard profile (Figure 4) and deriving values for the equation below:

$$\overline{X}_{NP} = \frac{C_{L_\alpha}\overline{X}_{acw} - C_{m_{\alpha fus}} + \eta_h C_{L_{\alpha h}} \overline{X}_{ach}\left(\frac{S_h}{S_w}\right)\left(\frac{\delta\alpha_h}{\delta\alpha}\right) + \overline{X}_p\left(\frac{F_{p_\alpha}}{qS_w}\right)\left(\frac{\delta\alpha_p}{\delta\alpha}\right)}{C_{L_\alpha} + \eta_h C_{L_{\alpha h}}\left(\frac{S_h}{S_w}\right)\left(\frac{\delta\alpha_h}{\delta\alpha}\right) + \left(\frac{F_{p_\alpha}}{qS_w}\right)} \approx 0.156 \textit{ power off}$$

$$X_{NP} = 0.151 \textit{ power on}$$

Where:

| | | | |
|---|---|---|---|
| | $C_{L\alpha}$ | $= 5.115$/rad | (wing lift curve slope) |
| | $X_{acw}$ | $= 0.03$ | (distance: cg → ac) |
| | $C_{M\alpha fus}$ | $= 0.3254$/rad | (fuselage moment) |
| | $\eta_h$ | $= 0.85$ | (horiz. efficiency factor) |
| | $S_h$ | $= 25$ sq.ft. | (horiz. tail area) |
| | $S_w$ | $= 150.6$ sq.ft. | (wing area) |
| | $C_{L\alpha h}$ | $= 4.762$/rad | (horiz. tail lift curve slope) |
| | $(\delta\alpha_h/\delta\alpha)$ | $= 0.623$ | (downwash correction factor) |
| | $X_{ach}$ | $= 3.21$ | (distance: cg → h. tail ac) |
| | $F_{p\alpha}$ | $= 92.35$ | (propeller force factor) |
| | $q$ | $= 41.68$ lbf/ft$^2$ | (dynamic pressure at cruise) |
| | $(\delta\alpha_p/\delta\alpha)$ | $= 1$ (power on) | (propeller downwash factor) |
| | $X_p$ | $= 1.575$ | (distance: cg → prop plane) |

The neutral point locations calculated from above (43.6% MAC power off and 43.1% MAC power on) yielded a power off static margin of 12.8%. This was acceptable for sufficient longitudinal stability.

Table 5 summarizes the results obtained in order to determine the horizontal tail incidence, $i_h$:

| Parameter | Definition | Value | Raymer Eqn. Number |
|---|---|---|---|
| $C_L$ cruise | Lift Coef. at Cruise | 0.3032 | (Lift Equation) |
| $C'_{Lh}$ cruise | Lift Coef. of H. Tail | 0.007 | 16.7 ($C_{Mcg}$) |
| $i_w$ | Wing Incidence Angle | 0.9067° | 16.11 |
| $\epsilon_0$ | Zero-Lift Downwash | 0.754 | 16.18 |
| $C_{L0}$ | Lift Coef. Correction | 0.178 | Stated in Text |
| $\epsilon$ | Total Downwash | 0.9425 | 16.20 |

**Table 5:** Longitudinal Stability Parameters

The equation for the horizontal tail incidence at cruise, $i_h$, is below. $C_{L\alpha h}$ for the horizontal tail was calculated using the same formula as for the wing (section 2.4.4). The aircraft angle of attack, $\alpha$, was determined by $C_{L\alpha}/C_{Lcruise}$.

$$i_h = \frac{C_{L_h}}{C_{L_{\alpha_h}}} + \alpha_{oL_h} + \epsilon + \alpha \qquad = \mathbf{0.5267^\circ}$$

| | | |
|---|---|---|
| Where: | $C_{L\alpha h} = 0.0831/^\circ$ | (horiz. tail lift curve slope) |
| | $\alpha_{oLh} = 0^\circ$ | (from NACA 0009 data) |
| | $\alpha = 0.5^\circ$ | (aircraft angle of attack) |

During landing configuration with the elevator deflected, the $C_{Lh}$ drops to −0.547. Trim flight at stall attitude required 7.3° of elevator deflection. This value was less than the maximum limit of 20° and was calculated based on the size of the elevator (45% of the horizontal stabilizer area).

The directional stability slope, $C_{N\beta}$, was then determined using:

| | | | |
|---|---|---|---|
| $C_{N\beta w}$ | Wing Dirn'l Stability | 0.00084634/deg | 16.41 |
| $C_{N\beta fus}$ | Fuselage Dirn'l Stability | −0.00070928/deg | 16.47 |
| $C_{N\beta v}$ | Vert. Stab. Dirn'l Stab. | 0.001225/deg | 16.36 |

**Table 6:** Directional Stability Parameters

The values from Table 6 were then used to find $C_{N\beta\ aircraft}$:

$$C_{N_\beta}\ aircraft = C_{N_{\beta_w}} + C_{N_{\beta_{fus}}} + C_{N_{\beta_v}} - \frac{F_{p_\beta}}{qS_w}\left(\frac{\delta\beta_p}{\delta\beta}\right)(\overline{X}_{cg} - \overline{X}_p) \qquad = 0.0013078/degree$$

| | | | |
|---|---|---|---|
| Where: | $F_{p\beta}$ | = 92.35 | (same as $F_{p\alpha}$) |
| | $(\delta\beta_p/\delta\beta)$ | = 1 | (same as $(\delta\alpha_p/\delta\alpha)$) |
| | $X_{cg}$ | = 0 | (from reference point) |
| | $X_p$ | = 0.213 | (distance: cg → prop plane) |

Raymer Figure 16.20 indicates that this value is within the suggested $C_{N\beta}$ range. Rudder deflection to maintain directional control in an 11.5° crosswind was calculated to be 4.4°. This value was also less than the 20° maximum.

The lateral stability slope, $C_{l\beta}$, was then calculated as follows:

$$C_{l_\beta} = \left(\frac{C_{l_{\beta_w}}}{C_L}\right)C_L + C_{l_{\beta_\Gamma}} + C_{l_{\beta_{wf}}} - C_{F_{\beta_v}}\left(\frac{\delta\beta_v}{\delta\beta}\right)\eta_v\left(\frac{S_v}{S_w}\right)\overline{Z}_v \qquad = -0.00163/degree$$

| | | | |
|---|---|---|---|
| Where: | $(C_{l\beta w}/C_L)$ | = −0.04 | (Raymer Figure 16.21) |
| | $C_L$ | = 0.3032 | (cruise lift coefficient) |
| | $C_{l\beta\Gamma}$ | = 0 | (Raymer eqn. 16.42) |
| | $C_{l\beta wf}$ | = −0.0715 | (Raymer eqn. 16.43) |
| | $C_{F\beta v}$ | = 1.69 | (vert. tail lift force coef.) |
| | $(\delta_{\beta v}/\delta\beta)\eta_v$ | = 0.6754 | (Raymer eqn. 16.48) |
| | $S_v$ | = 12.8 sq.ft. | (vertical tail area) |
| | $Z_v$ | = 0.1006 | (distance: cg ↑ v.tail ac) |

The calculated value of $C_{l\beta}$ was greater than recommended. The magnitude should be about half that of $C_{N\beta}$. For roll rate at stall speed (45 knots), an aileron deflection averaging 15° develops a roll rate of 106°/sec because the Triton has a large aileron area. As expressed in the summary statement, these values would be corrected if another iteration of the design were permitted.

Regarding spin recovery, the Triton can recover from a spin using both rudder and elevator. This was determined using Raymer Figure 16.31 after calculating required values for the graph in the text.

## 2.11 Performance Analysis

The performance analysis involved generating the Triton's drag polar equation, power-required curves, climb/cruise performance, and V-n diagrams. To develop the drag polar equation, the parasite drag for each major component of the aircraft had to be calculated using the equation below:

$$C_{D_o} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{LP}} \qquad = 0.026$$

Where: $\Sigma(C_{fc} FF_c Q_c S_{wetc})$ is determined from the table below:

| Component | $C_f$ $(10^{-3})$ | $FF_c$ | $Q_c$ | $S_{wetc}$ | $C_{fc} FF_c Q_c S_{wetc}$ |
|---|---|---|---|---|---|
| Fuselage | 2.546 | 1.477 | 1.00 | 229.91 | 0.8646 |
| Wing | 3.332 | 1.183 | 1.00 | 301.20 | 1.1873 |
| Horiz. Tail | 3.684 | 1.169 | 1.00 | 64.04 | 0.2758 |
| Vert. Tail | 3.512 | 1.088 | 1.00 | 25.60 | 0.0978 |
| | | | | $\Sigma$ | 2.4255 |

$C_{Dmisc}$ = 1.205 (drag from landing gear)
$C_{Dlp}$ = 0.00208 (leakage & prot. (0.08 x $C_{Do}$))

The induced drag coefficient, k, was then determined by:

$$k = \frac{1}{\pi AR\ 1.78(1-0.045AR^{0.68}) - 0.64} \qquad = 0.05096$$

Where: AR = 7.6 (wing aspect ratio)

The Oswald efficiency factor, e = 1.78 (1 - 0.045 $AR^{0.68}$), for the aircraft was found to be an acceptable 0.822.

Using the k factor from above, the drag polar equation can be written as:

$$C_D = C_{Do} + kC_L^2$$

The simplified drag polar equations for the Triton are as follows:

$$\textit{Sea level:} \qquad C_D = 0.026 + \frac{717344}{V^4} \qquad V = \textit{knots}$$

$$\textit{Cruise alt. (5000 ft):} \qquad C_D = 0.026 + \frac{971072}{V^4}$$

$C_{Do}$ was thought to be much higher because of the 50-inch wide fuselage, but the calculations yielded results typical for this type of aircraft.

The horsepower/velocity relationships were then evaluated. The $C_D$ equations were used to determine the drag of the aircraft and in turn the horsepower required for different velocities. The horsepower required for at both sea level and cruise altitude (5000 ft) were plotted with the horsepower available (from engine data and propeller efficiency) as Figure 6. Values from the graph were used to determine some of the Triton's performance parameters:

| | | Raymer Eqn. Number |
|---|---|---|
| Best Climb Rate - S.L. | 55 knots | 17.19 |
| Best Climb Rate - Alt. | 60 knots | 17.19 |
| Best Climb Angle - S.L. | 9.7° | 17.42 |
| Best Climb Angle - Alt. | 7.2° | 17.42 |
| Best Range Velocity | 78 knots | 17.25 |
| Best Endurance Velocity | 60 knots | 17.19 |
| Max. Rate of Climb - S.L. | 1012 ft/min | 17.43 |
| Max. Rate of Climb - Alt. | 750 ft/min | 17.43 |
| Range at Cruise Velocity | 790 n.mi. | 17.28 |
| Range at Best Speed | 943 n.mi. | 17.14 |
| Take off Distance | 1047 ft | 17.99 |
| Lndng Distance - Brakes | 909 ft | 17.99 |
| Lndng Distance - Rollout | 3188 ft | 17.99 |

Table 7: Performance Parameter Results

# Horsepower Required Versus Airspeed

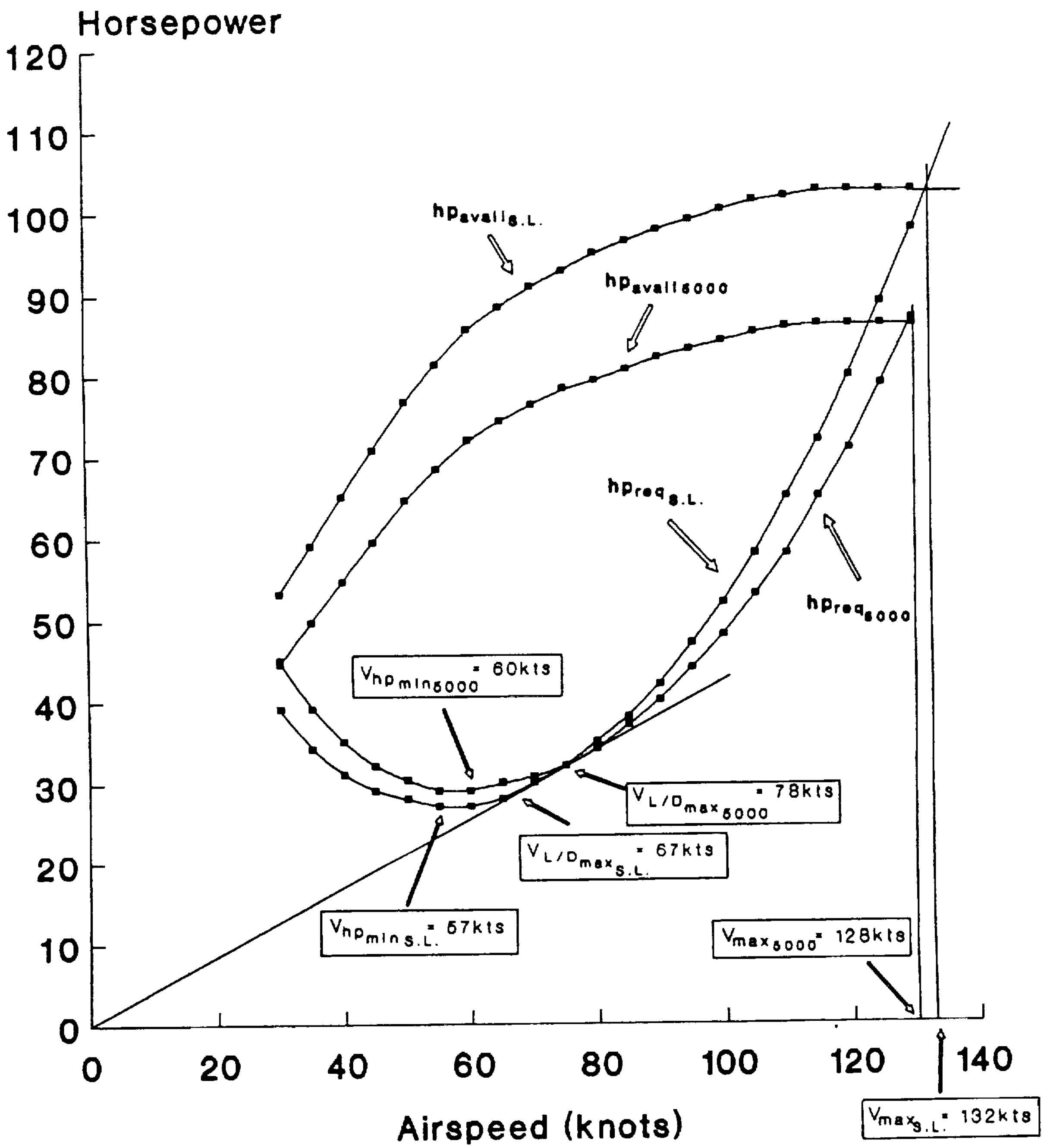



**Figure 6**

All of the above values are within acceptable ranges for this type of aircraft. The take off and landing distances (with exception to the conservative rollout value) meet the mission's runway requirement of 3000 ft.

Most of the information required to develop the flight envelope using V-n diagrams was already determined and is tabulated in Table 8.

| Parameter | $C_{L_{max}}$ | $-C_{L_{max}}$ | $\rho_{5000}$ | $V_{stall}$ cln | $V_{stall}$ flpd | $V_{cruise}$ |
|---|---|---|---|---|---|---|
| Value | 1.35 | −1.12 | 0.002054 | 52.67 kts | 45.67 kts | 120 kts |

Table 8: Known Parameters for the V-n Diagrams

FAR Part 23 stipulates minimum values of certain velocities (Table 9) based on the maximum limit load factor (4.4 for the Triton) and the wing loading (12.5 lbs/ft²). These values are used on both V-n diagrams.

| | | | Sea Level | 5000 ft. |
|---|---|---|---|---|
| $V_A$ | Maneuvering Vel. | $15\sqrt{(4.4x12.5)}$ | 112 knots | 104 knots |
| $V_C$ | Min. Cruise Vel. | $17\sqrt{(4.4x12.5)}$ | 127 knots | 118 knots |
| $V_D$ | Design Dive Vel. | $24\sqrt{(4.4x12.5)}$ | 180 knots | 167 knots |
| $V_F$ | Max. Flap Vel. | $11\sqrt{(4.4x12.5)}$ | 87 knots | 76 knots |

Table 9: FAR Minimum Velocities

The values from Table 8 and Table 9 were combined to form the initial V-n diagrams (Figures 7 and 8). The intersection of the ±50 fps and ±25 fps gust load lines with the diagrams determined the actual flight envelope. Analysis of the sea-level envelope shows that the maximum n-load of +4.4 (for utility category) is reached when a +50 fps gust is encountered at cruise velocity.

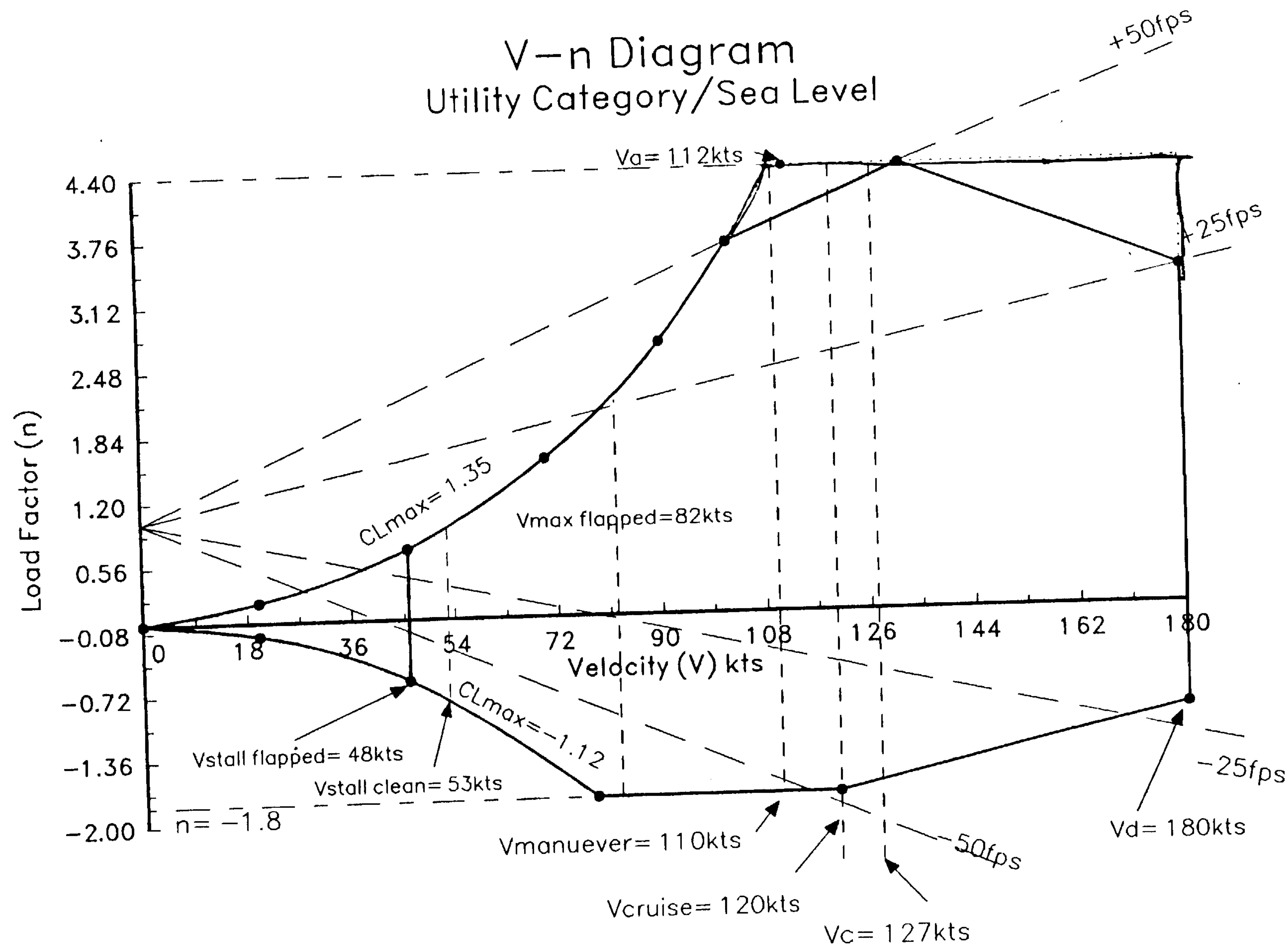

V—n Diagram
Utility Category/Sea Level
Load Factor (n)
4.40
3.76
3.12
2.48
1.84
1.20
0.56
−0.08
−0.72
−1.36
−2.00
Velocity (V) kts
0
18
36
54
72
90
108
126
144
162
180
Va= 112kts
+50fps
+25fps
−25fps
−50fps
CLmax= 1.35
CLmax= −1.12
Vmax flapped=82kts
Vstall flapped= 48kts
Vstall clean= 53kts
n= −1.8
Vmanuever= 110kts
Vcruise= 120kts
Vc= 127kts
Vd= 180kts

**Figure 7**

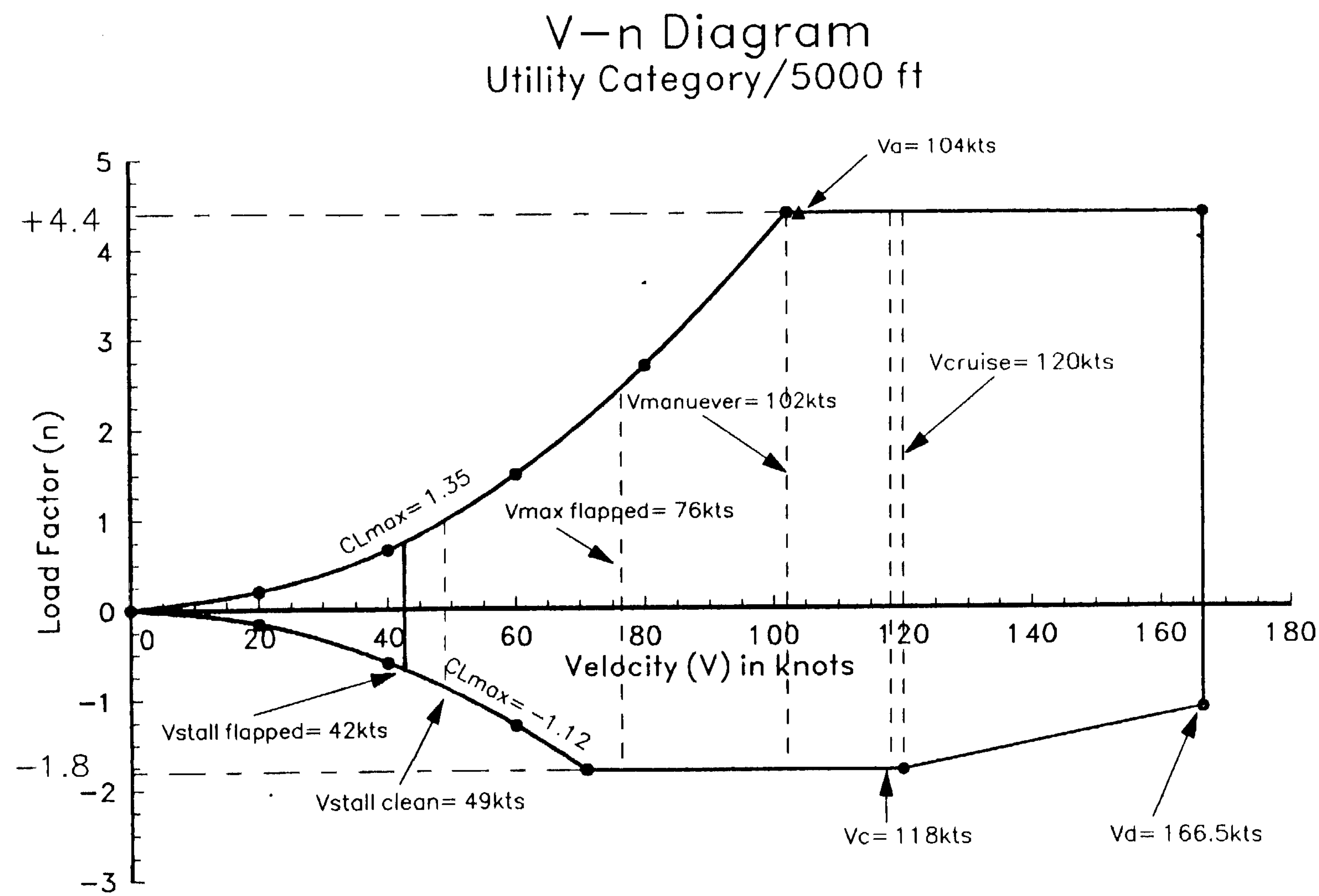

V–n Diagram
Utility Category/5000 ft
Load Factor (n)
Velocity (V) in knots
5
+4.4
4
3
2
1
0
-1
-1.8
-2
-3
0
20
40
60
80
100
120
140
160
180
Va= 104kts
Vcruise= 120kts
Vmanuever= 102kts
Vmax flapped= 76kts
CLmax= 1.35
CLmax= -1.12
Vstall flapped= 42kts
Vstall clean= 49kts
Vc= 118kts
Vd= 166.5kts

Figure 8

## 2.12 Cost Analysis

A cost analysis was performed to show whether the aircraft could be produced within the desired $50,000 limit (less instruments) set forth by the design specifications. The cost was to be determined for a schedule of 1000 aircraft produced over 5 years (17 per month) with a profit margin of 10% (Case F in Table 10). Other schedules were also evaluated for the production cost. The LITECOST program used provided a detailed breakdown of various parameters encountered throughout production. The results of this analysis, in addition to the final sale price of the aircraft, are tabulated in Table 10. Figure 9 depicts the approximate breakdown of total cost per aircraft for Case F shaded in Table 10. All values given below are costs without instruments.

| Case | Production Quantity | Production | Profit | Sale Price |
|---|---|---|---|---|
| A | 100 | 2 per month | 5% | $130,382 |
| B | 500 | 4 per month | 5% | $59,457 |
| C | 1000 | 7 per month | 5% | $43,802 |
| D | 5000 | 14 per month | 5% | $23,109 |
| E | 1000 | 17 per month | 5% | $43,928 |
| F | 1000 | 17 per month | 10% | $46,020 |

**Table 10:** Cost Summary

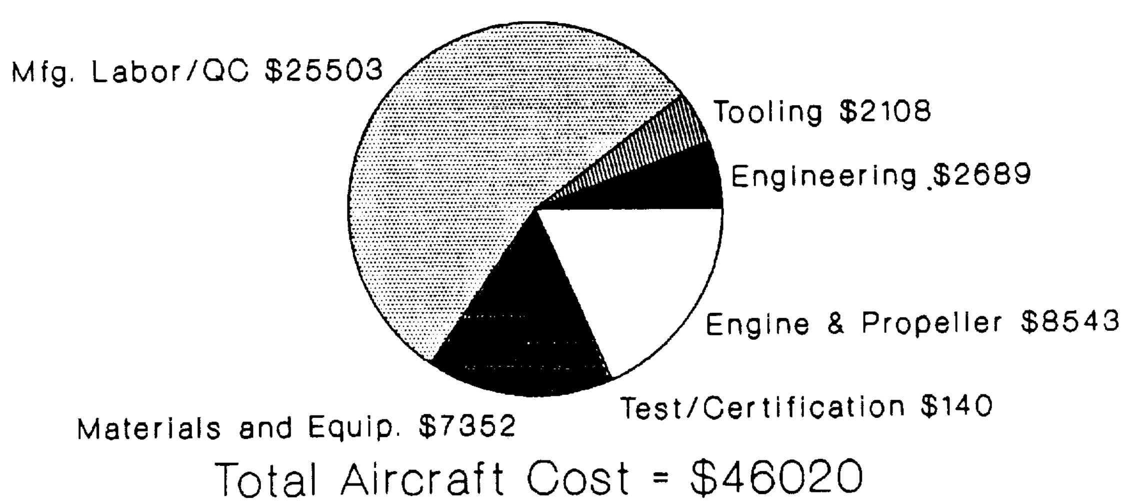



**Figure 9:** Case F Total Aircraft Cost Breakdown

The operating cost was performed for several cases using the LITEOPS cost evaluation program. The significant values that were varied in this program were yearly flight hours and loan value. LITEOPS generated a cost breakdown for parameters such as maintenance, loan payments, and overhaul bank and calculated the approximate operating cost per flight hour. Tables 11 and 12 tabulate the results. This summary shows that under company ownership, the price per flight hour is $36.52. This value compares with current aircraft rental rates and shows that the Triton would be competitive with all primary flight training aircraft.

| Case | Yearly Flight Hours | Loan Value (% of sale price) | Ownership | Price per Flight Hour |
|---|---|---|---|---|
| A | 100 | 0 | Private | $64.18 |
| B | 500 | 0 | Private | $32.02 |
| C | 500 | 90 | Private | $54.12 |
| D | 1000 | 90 | Private | $39.05 |
| E | 1000 | 90 | Company | $36.52 |

**Table 11:** Operation Schedules

| | A | B | C | D | E |
|---|---|---|---|---|---|
| Maint. Hours/FH | .2 | .2 | .2 | .2 | .2 |
| Gal. Fuel/Year | 549 | 2745 | 2745 | 5490 | 5490 |
| Overhaul Bank (2.50/FH) | $300 | $1500 | $1500 | $3000 | $3000 |
| Loan Monthly Payment | 0 | 0 | $920.92 | $920.92 | $920.92 |
| Operations Cost/Year | $6417.95 | $16010 | $27060 | $39050 | $36520 |
| **Operations Cost/FH** | **$64.18** | **$32.02** | **$54.12** | **$39.05** | **$36.52** |

•NOTES•

- Interest Rate for the Loan is 12% paid over 5 years.
- Fuel Cost is calculated using $2.00 per gallon.
- Labor Cost is calculated using $50.00 per Hour.
- Tie Down Storage is used in all cases.

**Table 12:** Operations Parameters

## 3.0 CONCLUDING REMARKS

As expressed in the summary statement, the C & P Aerospace Triton meets all the criteria set forth in the design specifications. The aircraft conforms to FAR Part 23 and utilizes an FAA certified engine. The Triton was designed with IFR equipment and capacity for all-weather instrumentation. The 3000 ft. maximum runway requirement is easily met. The Triton meets the specified cruise speed of 120 knots; however, this requires 83% power. With a list price of $46,020, and an operating cost of roughly $45 per hour, the Triton could be competitive as a primary flight trainer. C & P Aerospace, however, feels that the strongest asset of the Triton is mission versatility, as explained in section 1.10.