MEM 425
Aircraft Design And Performance

Final Report

Group A

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Description

The project is to enable the students to design, build, and fly an electric model airplane to demonstrate the knowledge they have gained in the course. Flying the model will be accomplished by professional pilots, and is not the responsibility of the students. Prior to flying, the students should provide estimates of certain performance characteristics of their airplane.

Details:

A- Design

- Wing (high-wing configuration, with 10° dihedral)
- Span – constant chord airfoil ribs will be provided.
- Angle of incidence – airfoil characteristics will be provided
- Longitudinal location
- Horizontal stabilizer (flat plate)
- Longitudinal location – variable in 1 inch steps
- Span - constant chord airfoil ribs will be provided.
- Angle of incidence – airfoil characteristics will be provided
- Elevators
- Vertical stabilizer (flat plate)
- Location, and rudder

B - Performance Estimation

- Recommended cruise velocity
- Cruise payload weight for flight test
- Maximum payload (flight WILL NOT be with maximum payload)

Objective

Each group has was tasked with designing a wing using 6 inches ribs either of the Clark Y airfoil design or NACA 6409 design. In addition to the wing, we were to build a tail using the same balsa wood used for the wing. The size, location and dimensions of both the wing and tail were entirely dependent on each group with the main focus being on ensuring longitudinal stability. The wing and tail were also designed to be able to attain the greatest possible payload whilst maintaining longitudinal stability at the greatest possible static margin for which our elevator's angle changes could consistently restore a level flight. As such, a thorough analysis and calculations were undertaken for weight, balance, cruise velocities, center of gravity location and longitudinal stability. The wings and tail designed were to be attached to a fuselage of the dimensions and characteristics listed below in Table 1.

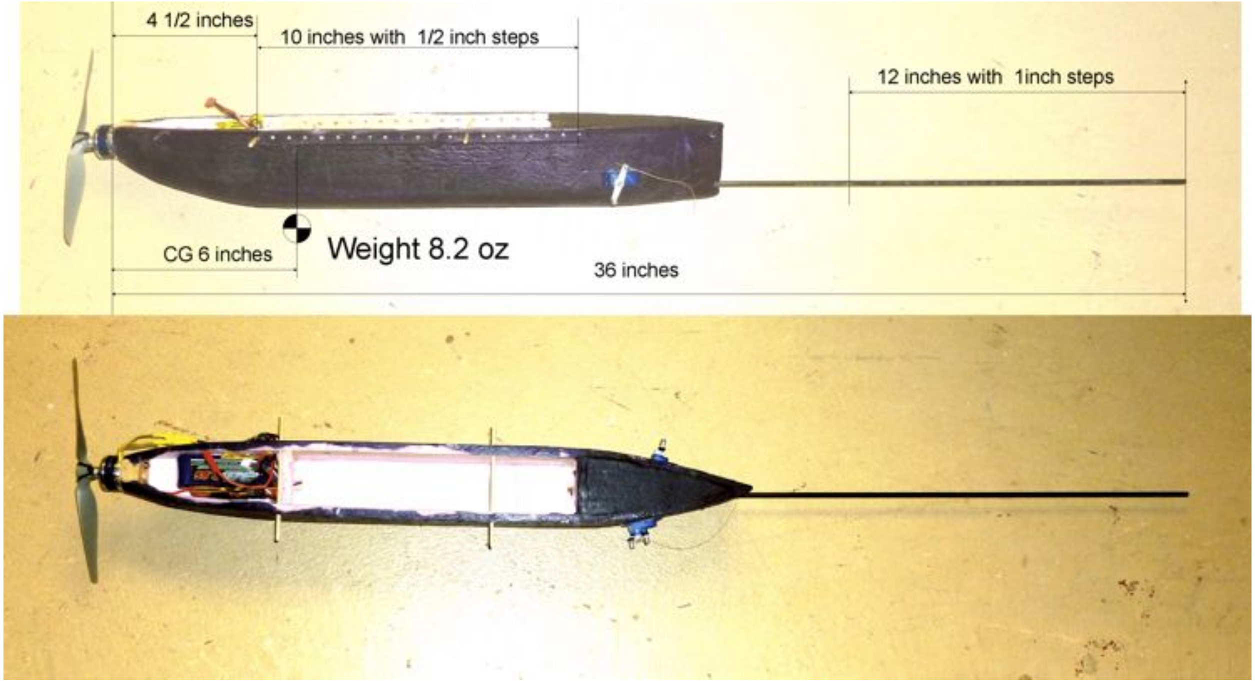


Figure 1: Fuselage for which wings and tail were designed

Table 1: Dimensions of fuselage wings and tail were designed

Dimension	Corresponding value
Fuselage length (in)	36
Weight (oz)	8.2

Center of gravity (in)	6
Diameter of APC propeller (in)	7

Design

Justification on selection of airfoil

As stated earlier we had the choice between two airfoils, being either the NACA 6409 or Clark Y design. Based on our performance analysis on both airfoils we decided to use the NACA 6409 as our calculations and analysis proved it to be superior for this particular RC Aircraft. With a smaller take off and landing distance, as well as a higher cruise velocity we decided it was the better option. It must be said that for this flight, take off and landing distance were not entirely relevant. Thus, those were not necessarily that main reasons we chose the NACA 6409 airfoils but rather the fact that the NACA 6409 had a slightly larger thrust available at lift off velocity and larger max coefficient of lift. This analysis and the corresponding results can be seen in the graphs below.

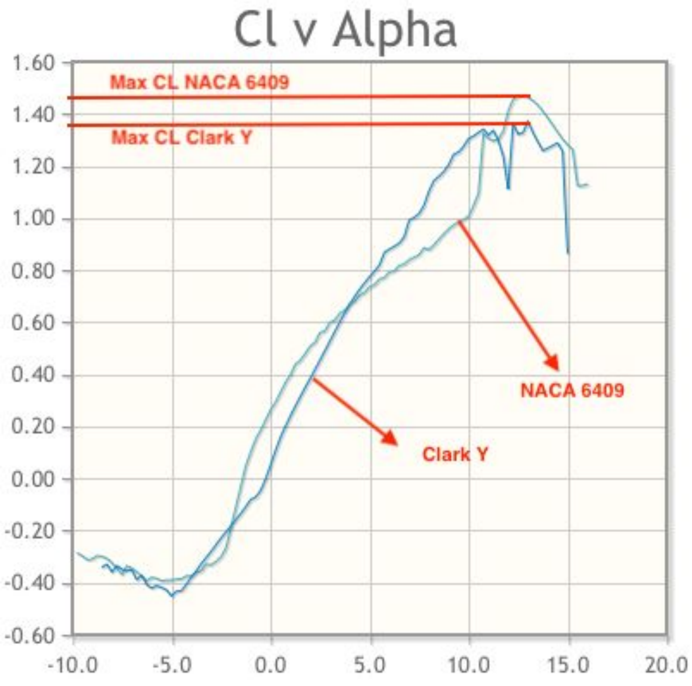


Figure 2: Comparison between Cl vs α for Clark Y and NACA 6409

Determination of wing span

A research was done in order to determine the wing span. Aspect ratio is the ratio of the wingspan to the wing chord. The higher the aspect ratio, the better the glide slope. However, this does not mean that it stalls at a slower speed, only that it floats farther forward in losing the same amount of altitude, compared to a lower aspect ratio wing. Below is a general list of the aspect ratios that are generally used for different planes:

Table 2: List of wing aspect ratio and application

Aspect Ratio	**Application**
>3	Fun flys, and other special application planes, with very fast roll rates
3-4	Combat planes, and some sport planes
4-5	Sport planes
5-6	Trainers
>6	Gliders

Our team decided to choose aspect ratio greater than 6 which fits the application for gliders. This was perfect as it would help increase the stability of the plane due to the fact that the extra wing length would help balance the aircraft by "accommodating" more mass on either side. At the same time, a higher aspect ratio like what we cose would decrease induced drag at the tips of the wings, i.e the wing tip vortices created would be much less than for a shorter aspect ratio as long narrow wings have less end edges. This can be seen from the equation below for induced drag where the induced drag ($C_{D,I}$) is seen to be inversely proportional to aspect ratio (AR).

$$C_{d,i} = \frac{{C_L}^2}{\pi e AR}$$

Table 3: Selected wing characteristics for design

Wing Characteristics	$in,\ in^2$	$ft,\ ft^2$
Chord Length, c	6	0.5
Wingspan, b	36.8	3.067

Wing Area, S	220.8	1.53
Aspect Ratio, AR	6.13	

In the Table 3, we're using the constant chord airfoil ribs that were being provided for us with a chord length of 6 inches. We have decided to choose 18.4 inches as the length of one of the wings, resulting the total wingspan to be 36.8 inches. The wing area was calculated by multiplying the wingspan with the chord length. The calculated aspect ratio, AR, agrees with our target which is to have an aspect ratio > 6 to obtain gliders flight advantages.

$$Wing\ Area,\ S = Wingspan,\ b\ \times Chord\ Length,\ c$$

To begin the calculation for the longitudinal location of the wing, the wing mean aerodynamic center, $m_{ac,wing}$, have to be calculated first. According to the textbook, the wing aerodynamic center, $m_{ac,wing}$, is assumed to be 25% of the mean aerodynamic chord, c, and the wing center of gravity, $(c.g.)_{wing}$ is at 40% of the mean aerodynamic chord, c. The airplane center of gravity location without wing, $(c.g.)_{airplane}$, was given as 6 inches from the nose of the airplane.

$$m_{ac,wing} = 0.25 \times 6 = 1.5\ inches$$
$$(c.g.)_{wing} = 0.4 \times 6 = 2.4\ inches$$

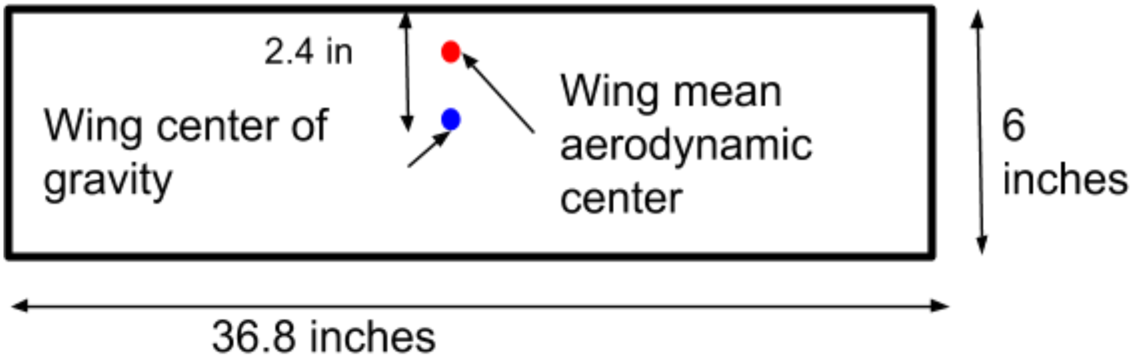


Figure 3: Overview of the locations on the wing

To locate the airplane center of gravity location with wing, we first assumed the weight of the wing and the additional part (horizontal and vertical tail) to be 2 oz (0.125 lb). Thus, the new center of gravity location with wing can be easily calculated using summation of moments at the nose of the airplane.

$$Airplane\ Center\ of\ Gravity,\ (c.g.)_{airplane} = \frac{(8.2*6)+(2*(6+(2.4-1.5)))}{8.2+2} = 6.1764\ inches$$

The longitudinal location of the wing from the nose of the airplane can be easily calculated by adding the wing center of gravity, $(c.g.)_{wing}$, and the reserved location for propeller motor and the battery.

$$Wing\ Longitudinal\ Location = 2.4 + 4.5 = 6.9\ inches$$

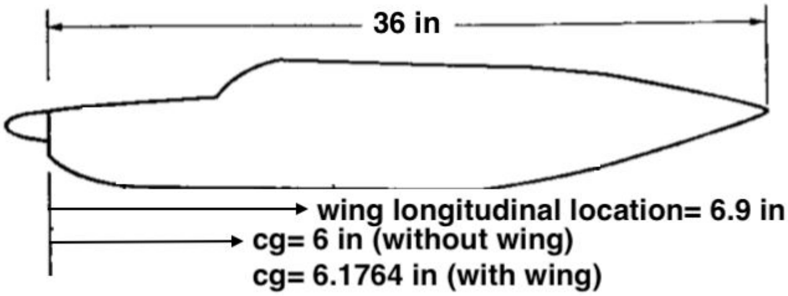


Figure 4: Locations of the center of gravity and the longitudinal location of the wing for the airplane

Stabilizer size and location

The primary function of the horizontal tail is to provide longitudinal stability, while the control surface on the horizontal tail (elevator) provides the longitudinal control and trim. The following tail volume ratio formula for horizontal tail will be used in our calculation.

$$Horizontal\ tail,\ V_{HT} = \frac{l_{HT}S_{HT}}{cS}$$

V_{HT} = Horizontal tail volume ratio

l_{HT} = Horizontal distance between the c.g. of the airplane, $(c.g.)_{airplane}$, and the aerodynamic center of the horizontal tail, $m_{ac,\ ht}$

S_{HT} = Horizontal tail area

c = Average wing chord

S = Wing area

We are using the suggested value by Raymer for a single-engine aviation airplanes for the horizontal tail volume ratio, V_{HT}, and we also assume the horizontal distance between the c.g. of the airplane, $(c.g.)_{airplane}$, and the aerodynamic center of the horizontal tail, $m_{ac,\,ht}$, to be 28.57 inches.

$$V_{HT} = 0.7$$
$$l_{HT} = 28.57\ inches$$

Based on the assumption that we made, we can easily calculate the required horizontal tail area, S_{HT}, using the horizontal tail volume ratio, V_{HT}.

$$0.7 = \frac{28.57 \times S_{HT}}{6 \times 220.8}$$
$$S_{HT} = 32.45\ inches^2$$

The primary function of vertical tail is to provide directional (yawing) stability, while the control surface on the vertical tail (rudder) provides the directional control for the airplane. We used the tail volume ratio for vertical tail to find the design that meet our needs.

$$Vertical\ tail,\ V_{VT} = \frac{l_{VT} S_{VT}}{bS}$$

V_{VT} = Vertical tail volume ratio

l_{VT} = Horizontal distance between the c.g. of the airplane, $(c.g.)_{airplane}$, and the aerodynamic center of the vertical tail, $m_{ac,\,vt}$

S_{HT} = Vertical tail area

b = Wingspan

S = Wing area

The same assumptions were made where we used the Raymer's suggested value for single-engine aviation airplanes for the vertical tail volume ratio, V_{VT}, and we also assume the horizontal distance between the c.g. of the airplane, $(c.g.)_{airplane}$, and the aerodynamic center of the vertical tail, $m_{ac,\,vt}$, to be 26.68 inches.

$$V_{VT} = 0.04$$
$$l_{VT} = 26.68\ inches$$

The corresponding vertical tail area, S_{HT}, can be found using the vertical tail volume ratio formula.

$$0.04 = \frac{26.68 \times S_{VT}}{36.8 \times 220.8}$$
$$S_{VT} = 12.178\ inches^2$$

The horizontal and vertical tail was designed to have lower aspect ratio, AR, than the wing because it will give the tail advantages when dealing at higher angles of attack. The lower the aspect ratio, AR, for the tail, the higher the stall angle of the tail. For the horizontal tail, we use aspect ratio, AR, equals to 4. We would also choose 0.5 as the taper ratio for both vertical and horizontal tail.

$$Horizontal\ tail\ aspect\ ratio,\ AR = 4$$
$$Taper\ ratio,\ \lambda = 0.5$$

Thus, we can calculate the span of the horizontal tail, b_t, using the following formula.

$$b_t = \sqrt{(S_{HT})AR} = \sqrt{32.45 \times 4} = 11.39\ inches$$

The tail root chord, c_{rt}, and the tail tip chord, c_{tt}, can be calculated from the taper ratio, λ, and the following formula.

$$Tail\ root\ chord, c_{rt} = \frac{2S_{HT}}{(\lambda+1)b_t} = \frac{2 \times 32.45}{(0.5+1)11.39} = 3.798\ inches$$
$$Tail\ tip\ chord, c_{tt} = \lambda c_{rt} = 0.5 \times 3.798 = 1.899\ inches$$

The spanwise location of the mean aerodynamic chord for horizontal tail is found as follows.

$$\breve{y}_{HT} = \frac{b_t}{6}\frac{1+2\lambda}{1+\lambda} = \frac{11.39}{6}\frac{1+2(0.5)}{1+0.5} = 2.53\ inches$$

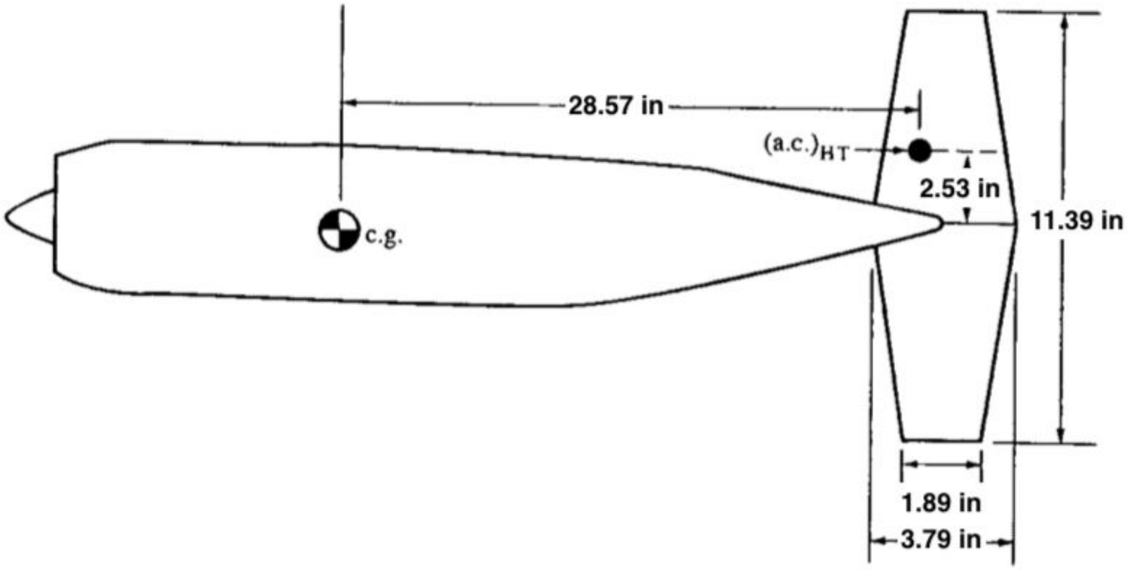


Figure 5: Plan view of the fuselage and the horizontal tail

For the vertical tail, we design the aspect ratio, AR, equals to 1.5 and use the same method to find the vertical tail high, h_{VT}.

$$AR_{VT} = 1.5$$
$$h_{VT} = \sqrt{(AR_{VT})S_{VT}} = \sqrt{1.5 \times 12.178} = 4.27\ inches$$

The root chord, tip chord for the vertical tail, and the vertical location of the mean aerodynamic chord of the vertical tail can be found as follows.

$$Vertical\ tail\ root\ chord, c_{rvt} = \frac{2S_{VT}}{(\lambda+1)h_{VT}} = \frac{2\times12.178}{(0.5+1)4.27} = 3.8\ inches$$
$$Vertical\ tail\ tip\ chord, c_{tvt} = \lambda c_{rvt} = 0.5 \times 3.8 = 1.9\ inches$$
$$\check{z}_{VT} = \frac{2h_{VT}}{6}\frac{1+2\lambda}{1+\lambda} = \frac{2(4.27)}{6}\frac{1+2(0.5)}{1+0.5} = 1.89\ inches$$

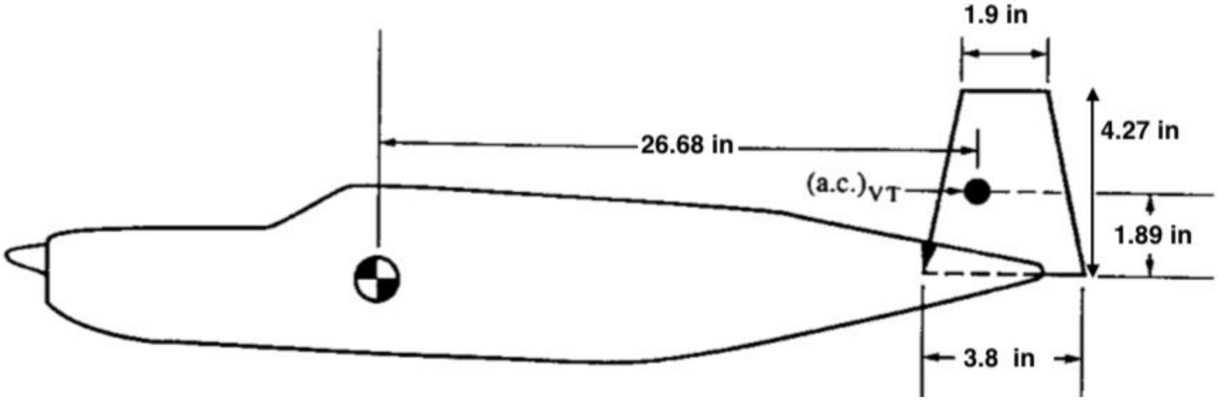


Figure 6: Side view of the fuselage and vertical tail

Longitudinal Stability Margin

"For longitudinal static stability, the position of the center of gravity must always be forward of the neutral point"

This is an excerpt taken from the notes given out in the class, and from this statement we can identify few criterias to reach longitudinal stability for our flight.

Longitudinal stability requirements:

- $C_{M,0} > 0$
- $\frac{\partial C_{M,cg}}{\partial \alpha_a} < 0$
- Static margin > 0

For our design process, we'll only considered calculating the static margin of the airplane as the experimental data for $C_{M,0}$ and $\frac{\partial C_{M,cg}}{\partial \alpha_a}$ were absent at the time of designing process. We continue the calculation of our airplane design process by evaluating the neutral point of the airplane. A neutral point is a point frozen somewhere in the airplane. It is a fixed quantity for a given airplane design, and independent of the location of the airplane center of gravity. For conventional generation aviation airplanes, the static margin should be in between 5% and 10%. We are going to use static margin equals to 10% in our design process. The neutral point of the airplane was calculated as follows.

$$Static\ margin = \frac{x_n - x}{c}$$

x_n = neutral point location

x = location of the airplane center of gravity

c = wing chord length

$$0.1 = \frac{x_n - 6.1764}{6}$$

$$x_n = 6.7764\ inches$$

$$Static\ margin = x_n - x = 6.7764 - 6.1764 = 0.6\ inches$$

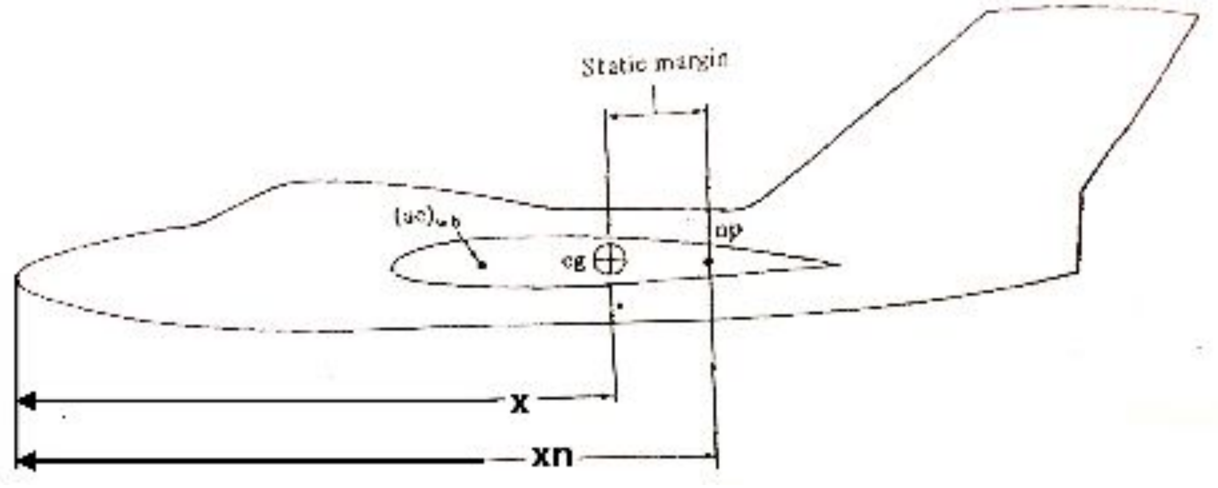


Figure 7: The corresponding static margin location for the airplane design

Based on the calculations performed for the location of the neutral point, we can see that our airplane design will achieve longitudinal stability in flight. The summary of the result for airplane design can be seen in the appendices along with the calculation for different aspect ratio, AR.

Performance Analysis

In order to evaluate the design that was created, it is important to do some performance analysis. The key performance parameters are wing loading, $W/S = 0.6375/1.533 = 0.416 lb/ft^2$ and power loading, $W/P = 0.6375/0.134 = 4.76\ lb/hp$. The aerodynamic coefficients are $C_{D,0} = 0.0172$, $K = 0.0865$, $(L/D)_{max} = 27.1$.

Cruise Velocity

In order to determine the recommended cruise velocity, it is assumed that the aircraft need to has a minimum drag. Flying for minimum drag corresponds to flying with maximum $C_L = C_D$. In this case, there is a few pertinent relations:

$$C_D = 2C_{D,0}$$

$$C_L = \sqrt{C_{D,0} / K}$$

$$V^2 = \frac{2W}{\rho S}\sqrt{\frac{K}{C_{D,0}}}$$

Substituting numerical values yield drag coefficient, $C_D = 0.0344$, lift coefficient, $C_L = 0.446$ and cruise velocity, $V = 28.01\ ft/s$. In these calculations, $K = 1/(\pi e AR) = 0.0865$, with $AR = 6.13$ has been used. In order to balance the drag, the thrust required is $T_R = D_{min} = 0.5 \varrho S V^2 S C_D = 0.049\ lb$.

$$V_{\max} = \left\{ \frac{[(T_A)_{\max}/W](W/S) + (W/S)\sqrt{[(T_A)_{\max}/W]^2 - 4C_{D,0}K}}{\rho_\infty C_{D,0}} \right\}^{1/2}$$

Equation above allows the direct calculation of the maximum velocity. By substituting the numerical value, the max velocity is $V_{max} = 37.73\ ft/s$.

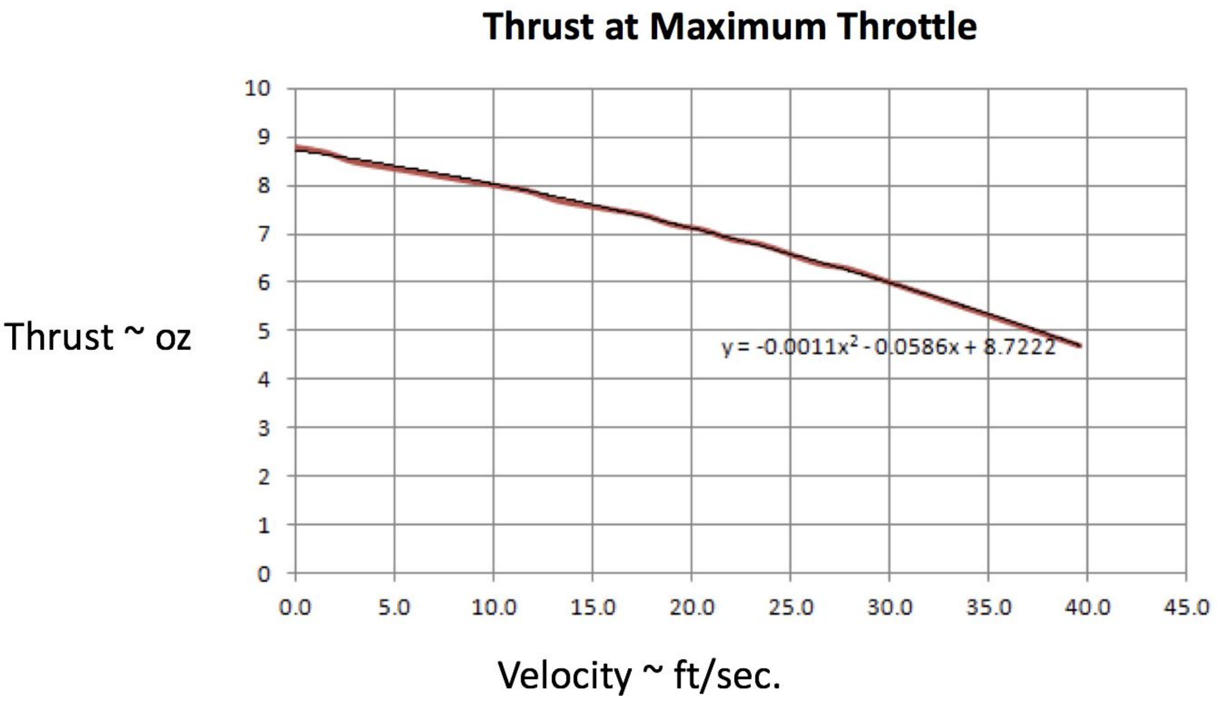


Figure 8: Thrust at Maximum Throttle

Above is a figure and equation of thrust at maximum throttle. From the equation given, thrust available can be calculated by putting the velocity value. Below is the figure of Thrust Available vs. Velocity that was calculated.

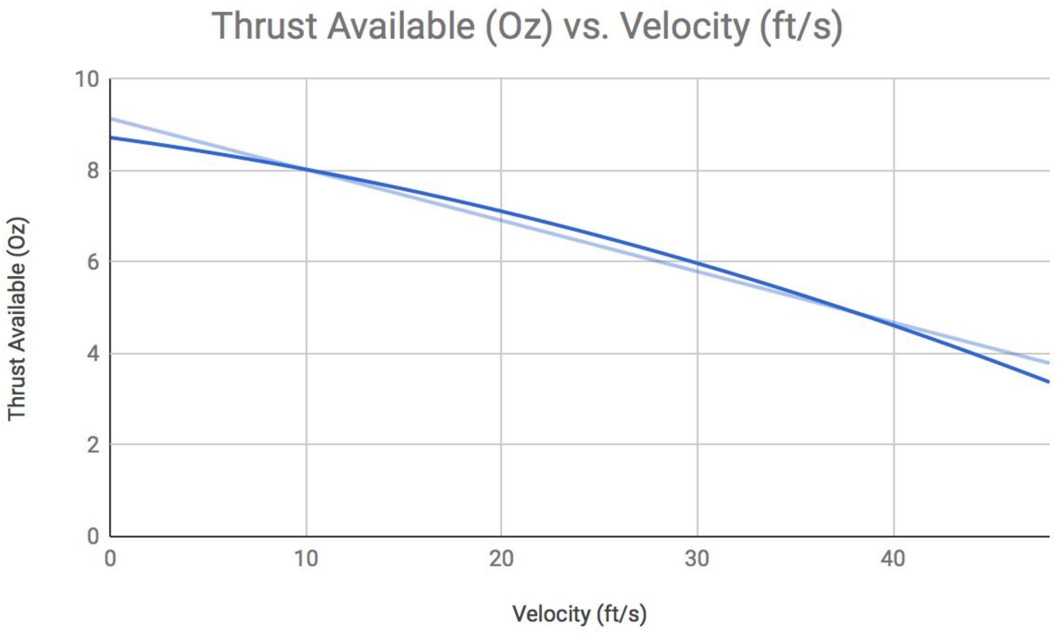


Figure 9: Thrust Available vs. Velocity

Next, power is calculated by using equation $Power = Thrust\ x\ Velocity$. Following is a figure of Power vs. Velocity.

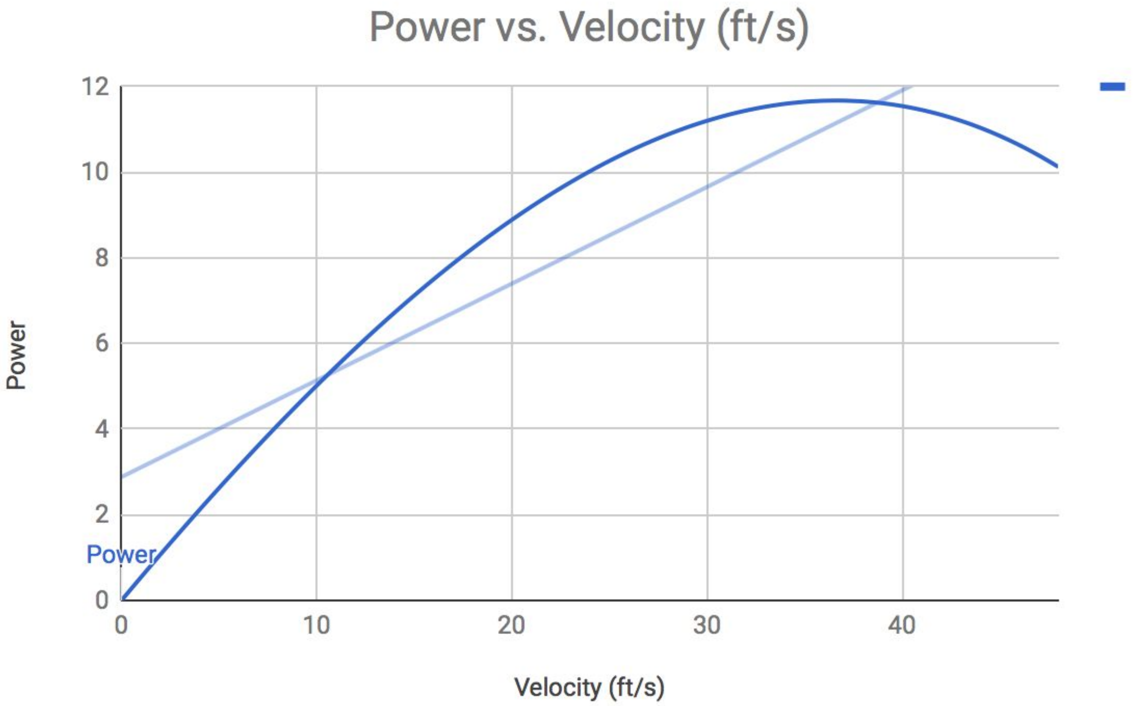


Figure 10: Power vs. Velocity

Take off

The theoretical take off distance (S_g) was calculated using the equation $S_g = \frac{1.21W^2}{\rho g S T C_{Lmax}}$, where g is the force of gravity, ρ the density, S the wing area, W, T and C_{Lmax} the weight, thrust force and max coefficient of lift respectively. This value upon calculation was determined to be *6.108 ft.* However, considering the fact that this is a model aircraft, this take off distance is nothing more than theoretical as the aircraft will not actually accelerate off the ground but will be launched straight into the air by hand. As such our equivalent of what a take off distance would be in terms of getting our aircraft of the ground was our hand launch velocity.This hand launch velocity, which is the velocity required to lift the weight of the plane. In determining the hand launch-launchability we had to consider the stall velocity in relation to the mass of the aircraft. Limiting the stall velocity of the aircraft was hence essential for launching the aircraft by hand and this was calculated with Newton's Law of Motion taken into consideration. Ultimately we determined the hand launch velocity as seen below.

$$F = mg = 0.5\rho S_w (v_{stall})^2 C_l$$

$$V_{hl/stall} = \sqrt{\frac{2mg}{\rho S_w C_l}}$$

With the necessary values placed into the equation, we obtained a hand-launch velocity of *160 ft/s or 48.8 m/s* , where the mass being considered was the wing, tail and engine. Also with the maximum altitude of an rc aircraft being 400 ft above sea level, the density used was that at 400 ft which was $0.0023495\ slug/ft^3$

Climb

For the climb performance, it is assumed that the aircraft experienced a steady (unaccelerated) climb. To make the analysis less complex, it is also assumed that the thrust line is parallel to the direction of flight. In regard to climb performance, power and and the weight of the airplane are considered as an important parameters to achieve the best rate of climb for the aircraft. This is denoted by equation 5.77, 5.78, 5.79, and 5.80:

$$R/C = V \sin \theta \ (equation\ 5.77)$$

$$R/C = \frac{TV - DV}{W} \ (equation\ 5.78)$$

$$TV - DV = excess\ power\ (equation\ 5.79)$$

$$R/C = \frac{excess\ power}{W}$$

With *TV* corresponds to the power available and *DV* corresponds to power required. Based on the information of the battery to operate the aircraft, the power available of the aircraft, *TV,* was calculated based on the equation:

$$P = IV$$

$$I = 950\ mA\ and\ V = 11\ V$$

Thus, the values obtained for power available is $TV = 7.707\ ft.lb/sec$.

During climbing, less lift is needed compared to level flight as part of the weight of the airplane is supported by thrust. Because of that, this will affect drag as less lift will cause less drag due to lift. Therefore, the drag during climbing will be less than drag during level flight. The value of drag can be denoted by equation 5.84:

$$D = qSC_{D,0} + \frac{kW^2 cos^2\theta}{qS}$$

To obtain an accurate results of rate of climb and angle of rate of climb, it is assumed that $cos\,\theta \approx 1$. According to Mair and Birdsall, this assumption leads to an accurate climb performance for angles as large as 50°. By assuming $cos\,\theta \approx 1$ for drag expression, the error in calculated rate of climb would be 3% or less, and the error in calculated climb angle would be 2.5° or smaller. Hence, the power required will be denoted as power required during steady, level flight. After plugging in all numerical values, the calculated power required obtained is $DV = 1.694\, ft.lb/sec$. from equation 5.77, its corresponding angle of climb was $\theta = 19.681\ degrees$. Ultimately, this leads to the rate of climb of $R/C = 9.432\, ft/sec$. Based on the result, the achievable climb angle exceeded normal conventional airplanes climb angles, which are usually less than 15°.

Maximum Endurance

The endurance for normal fuel-burning aircraft has been analyzed and verified in various types of flying conditions. However, in this case, the aircraft was operated by using a powered battery, Turnigy nano-tech 950mAh Lipo-pack. Given that the mass of the aircraft is constant in all condition, the calculation for maximum endurance of the aircraft is simplified. Based on the equation below, the maximum endurance of the aircraft is determined by the time to drain the propulsive power required by the battery:

$$E = \frac{mE^*}{P_{battery}}$$

Where m is the mass of the battery (lb) and E^* is the mass specific content energy of the battery (Wh/lb). Substituting all the numerical values, the maximum endurance for the aircraft to operate is 4.55 hours.

Maximum Payload

To ensure steady and level flight of the aircraft, the lift-to-drag ratio to the reciprocal of the thrust-to-weight ratio can be used as below:

$$\frac{L}{D} = \left(\frac{T_R}{W}\right)^{-1} \qquad \text{Steady, level flight}$$

To obtain the lift-to-drag ratio, we can use C_L/C_D. By calculating this, we get L/D = 12.9629733. Base on our initial calculation, the thrust required for our wing design is 0.04917853221lb. This will result in a maximum payload weight of 0.6375lb.

An approach used to prove this maximum payload weight is by noting that the lift-to-drag ratio is a function of velocity. The equation that can be used is as followed:

$$\boxed{\frac{L}{D} = \left(\frac{\rho_\infty V_\infty^2 C_{D,0}}{2(W/S)} + \frac{2K}{\rho_\infty V_\infty^2}\frac{W}{S}\right)^{-1}}$$

By substituting values $p\infty$=0.0023769 slug/ft^3, V∞=28.00914256ft/s, $C_{D,O}$=0.0172, S=1.533ft^2 , W = 0.6375 (obtained above) and K=0.08649725168, we get the value of L/D = 12.9630 which is similar to the L/D obtained earlier.

References

1. John D. Anderson, Jr., *Aircraft Performance and Design,* TATA Mcgraw-Hill. 2010.
2. "NACA 6409 9% (n6409-il)", *Airfoil Tools*, 2017.
3. "CLARK-Y AIRFOIL (clarky-il)" *Airfoil Tools,* 2017.
4. "Turnigy nano-tech 950mah 3S~25 50C Lipo Pack" *HobbyKing.com,* 2017. https://hobbyking.com/en_us/turnigy-nano-tech-950mah-3s-25-50c-lipo-pack.html?___store=en_us.
5. Hepperle, Martin, Electric Flight-Potential and Limitations, 2012.

Appendix

Figure 11: Top and side view of wing

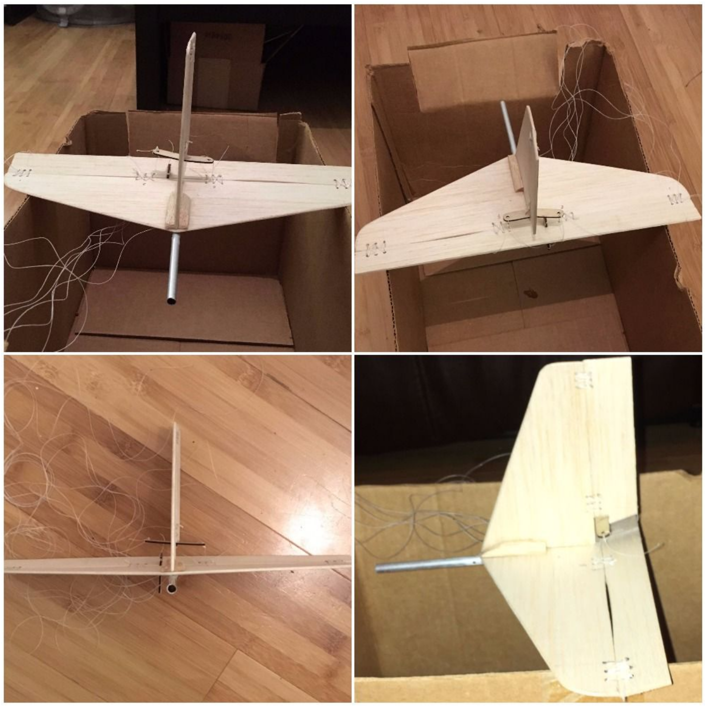


Figure 12 :Top, side and front view of tail

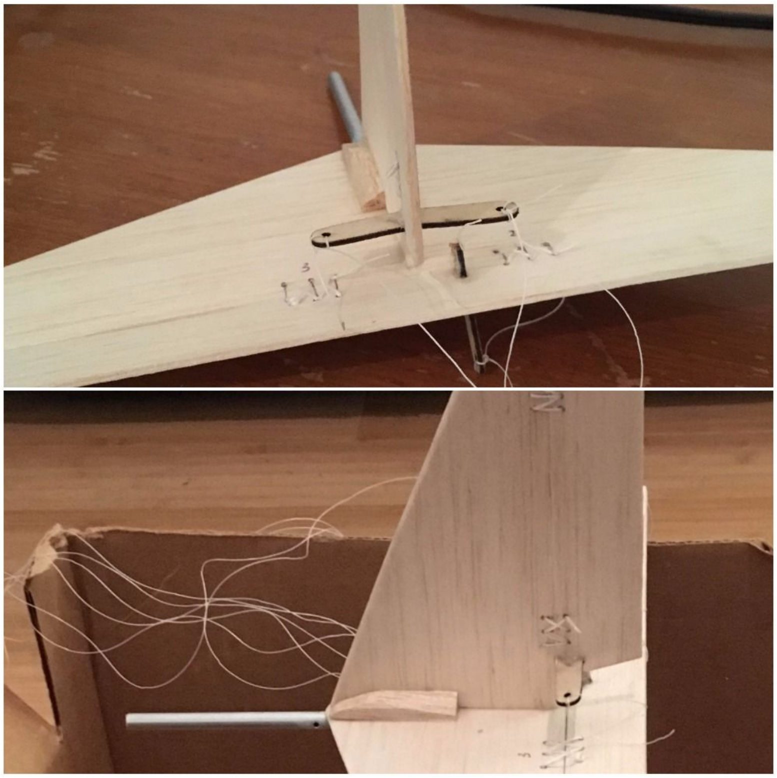


Figure 13: View of elevator controls

Assumption		**convert to ft,lb, ft2**	
Constant Chord length (inches)	6	0.5	ft
Wing span, b (inches)	36.8	3.066666667	ft
Wing area, S (inches^2)	220.8	1.533333333	ft2
Aspect Ratio, AR	6.133333333		
Wing loading (W/S)		0.4157608696	lb/ft2
Power loading (W/P)		4.757462687	lb/hp
Fuselage weight, oz	8.2	0.5125	lb
Additional weight (wing, tail), oz [2]	2	0.125	lb
Total gross weight, oz	10.2	0.6375	lb
Engine weight, g	32.6	0.0718707	lb
mass specific energy content (Wh/lb)	50.66772001		
mass of battery (lb)	0.152119		
Oswald efficiency, e [4]	0.6		
Swet/S [5]	4		
Skin friction Coefficient, Cfe [7]	0.0043		
Cdo	0.0172		
K	0.08649725168		
pi	3.14159		

Altitude (ft)	**Density (slug/ft^3)**
0	0.0023769
20000 [1]	0.0012673

Wet grass resistance (brake off)	0.08	**j**	1.15
Wet grass resistance (brake on)	0.2		

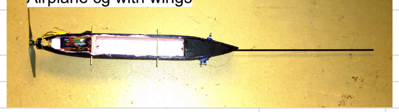
	Max CL/CD		**Thrust Required, Tr (Oz)**	**Max CL**	**Vstall (ft/s)**	**Velocity lift off (ft/s)**	**Thrust avail at Velocity lift off (lb)**	**T/W cruise**	**Power Required (Climb) (ft.lb/sec)**	**Power available (climb) (ft.lb/sec)**
NACA 6409	27.1	at α=10.75°	0.3763837638	1.48	15.37448195	16.91193014	0.4635341359	0.036900369	1.888209822	7.7075229
CLARK Y	29.6	at α=10.75°	0.3445945946	1.38	15.92178684	17.51396553	0.4599042958	0.03378378378	1.888209822	7.7075229

Performance Analysis	**NACA 6409**	**CLARK Y**
(Min Drag)		
Cd	0.0344	0.0344
Cl	0.4459262816	0.4459262816
cruise velocity (ft/s) [3]	**28.00914256**	**28.00914256**
thrust reqd	0.04917853221	0.04917853221
Vmax	**37.72878215**	**37.72878215**
Emax (hr)	4.081920776	4.549670065
excess power	5.819313078	5.819313078
rate of climb, R/C [6]	9.128334241	9.128334241
climb angle	19.02046088	19.02046088
Takeoff Distance (ft)	**6.10799892**	**6.602308854**
Landing Distance (ft)	**24.27061602**	**26.02935631**
L/D	12.9629733	12.9629733
Max Payload Weight	**0.6375 [8]**	**0.6375 [9]**

N/b min velocity dependant on stalling speed

Wing Longitudinal Calculation			
wing mean aerodynamic center (inches) [10]	1.5	**Wing cg longitudinal location from nose (inches)**	6.9
wing center of gravity (inches) [11]	2.4		
airplane center of gravity location without wing (inches) [12]	6		
Reserved area for battery (inches) [13]	4.5		
airplane center of gravity location with wing (inches) [14]	6.176470588		

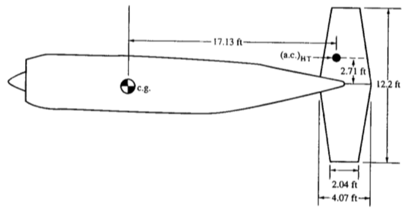
Airplane cg with wings



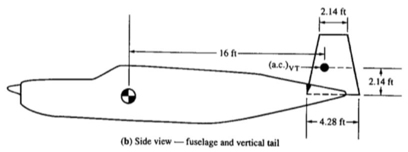
Horizontal and vertical tail size			
VHT [15]	0.7		
VVT	0.04		
total length of fuselage (inches)	36		
lht,aerodynamic center of the horizontal tail (inches) [16]	28.57256416		
SHT (inches^2) [17]	32.45631		
lvt, aerodynamic center of vertical tail (inches) [18]	26.68774236		
SVT (inches^2) [19]	12.17853484		
AR for horizontal tail [20]	4		
tail taper ratio, [21]	0.5		
horizontal tail span (inches) [22]	11.39408794		
tail root chord (inches) [23]	3.798029314		
tail tip chord (inches) [24]	1.899014657	Spanwise location of the mean aerodynamic chord for horizontal tail (inches) [25]	2.532019543
AR for vertical tail [26]	1.5		
Vertical tail height	4.274084961		
vertical root chord (inches) [27]	3.799186632		
vertical tip chord (inches) [28]	1.899593316	Vertical location of the mean aerodynamic chord of the vertical tail (inches) [29]	1.899593316

Horizontal tail $V_{HT} = \frac{l_{HT} S_{HT}}{\bar{c} S}$

Vertical tail $V_{VT} = \frac{l_{VT} S_{VT}}{b S}$



(a) Plan view — fuselage and horizontal tail



(b) Side view — fuselage and vertical tail

Figure 8.22 Layout of **(a)** the horizontal tail and **(b)** the vertical tail.

Longitudinal Stability Margin	
static margin (%) [30]	0.1
neutral point, xn (inches)	6.776470588
xac(wing), inches	6.076470588
location of wing leading edge from the nose airplane (inches)	4.576470588
static margin (inches) [31]	0.6

[1] Altitude assumption that would be comfortable for pressurized cabin with passengers inside

[2] in the book, Raymer suggested the estimation wing weight equal to 2.5*wing area, but the assumption doesnt make sense for rc airplane.

2.5(432)=1080oz=67.5 lb
see page 434

[3] eqn 5.22

[4] Assumption

[5] From figure 2.54, for single engine general aviation plane

[6] eqn 5.8

[7] From fig 2.55, and also based on textbook example page 414

[8] coincidental with total gross weight c11??

[9] coincidental with total gross weight c11??

[10] assume 25% from the leading edge of the wing

textbook page 434

[11] wing center of gravity = 0.4% x mean aerodynamic chord

from textbook page 434

[12] given from the diagram in the powerpoint slide

[13] based on diagram in powerpoint slide, from the nose of the airplane

[14] using sum of moment on the nose of the airplane

[15] suggested tail volume ratio for vertical and horizontal stabilizer from textbook page 436

[16] distance between airplane cg with wing and the ac tail

based on example on textbook page 437

[17] area of the horizontal tail

[18] based on example textbook page 438

[19] area vertical tail

[20] from textbook

[21] from textbook

[22] in diagram is the length = 12.2 ft

[23] this is the length = 4.07 ft in diagram

[24] length = 2.04 ft

[25] This is the length = 2.71 ft in diagram

[26] from textbook

[27] this is the length = 4.28 ft

[28] this is the length = 2.14 ft

[29] this is the length = 2.14 ft in diagram

[30] based on textbook example, conventional airplane should have static margin between 5% and 10%

[31] from figure 7.23 of chapter 7,

if static margin > 0, then the plane is stable

for longitudinal static stability, the position of center of gravity must always be forward of the neutral point

(in our case, xn > airplane center of gravity location with wing (inches))