UCI Samueli School of Engineering | Department of Mechanical and Aerospace Engineering

Project: Drag Calculator

Course: MAE 158 - Aircraft Performance

Advisor: Prof. Huynh

Author: Triet Ho

Student ID: 88807201

Constants:

$b_{wing} = 97$ [ft]

$c_{r0,wing} = 19$ [ft]

$\sigma_{wing} = 0.26$ [1]

$\Lambda_{c/4} = 6°$

$L_f = 92$ [ft]

$D_f = 11$ [ft]

$S_{wet,fuselage} = 2543.43$ [ft^2]

November 5, 2024

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Nomenclature List

SYMBOL	DEFINITION
b_{wing}	Wing span
$c_{r0,wing}$	Root chord at center line of the wing
σ_{wing}	Taper ratio of the wing
$\Lambda_{c/4}$	Swept angle of the wing at quarter chord
L_f	Length of the fuselage
D_f	Diameter of the fuselage
$S_{wet,fuselage}$	Wetted surface of the fuselage
$S_{wet, wing}$	Wetted surface of the wing
t/c_{wing}	Thickness to chord ratio of the wing
Z	Compressibility correction factor, used in calculating the form factor K
K	Form factor, used to calculate equivalent profile drag area
t/c	Thickness to chord ratio
R^2	Accuracy of the curve fitting
e	Oswald efficiency
Re	Reynolds number
C_f	Skin friction coefficient
C_{Dp}	Parasite drag coefficient
W	Weight of the airplane

SYMBOL	DEFINITION
L	Lift force of the airplane
q	Dynamic pressure
S_{ref}	Reference area of the airplane
C_L	Lift coefficient
AR	Aspect ratio
C_{Di}	Induced drag coefficient
D_p	Parasite drag force
D_i	Induced drag force
D_{total}	Total drag force
L/D	Lift to drag ratio
$C_{r,e,wing}$	Wing exposed root chord
CAD	Computer-aided design
SolidWorks	Software to create parts, assemblies, and drawings of 3D models

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1. Introduction

In aerodynamics, calculating airplane drag is crucial for both design and optimizing operating conditions. In this project, a specific airplane configuration is analyzed through hand calculation, MATLAB coding, and Solidworks. The objective of the project is to find the velocity where lift to drag ratio reaches maximum.

The foundation of this drag calculator project is Shevell's Problem 11.1. In this

11.1. A twin turbofan transport airplane is cruising at 31,000 ft pressure altitude at a Mach number of 0.78. Outside air temperature is −60°F. The airplane gross weight is 98,000 lb. The airplane has unsealed aerodynamically balanced control surfaces. Following are the airplane dimensional data:

Wing

Span	= 93.2 ft
Planform area	= 1000 ft^2
Average t/c	= 0.106
Sweepback angle	= 24.5 deg
Taper ratio	= 0.2
Root chord	= 17.8 ft
Wing area covered by fuselage	= 17%

Horizontal Tail

Exposed planform area	= 261 ft^2
t/c	= 0.09
Sweepback	= 31.6 deg
Taper ratio	= 0.35
Root chord	= 11.1 ft

Pylons

Total wetted area	= 117 ft^2
t/c	= 0.06
Sweepback	= 0 deg
Taper ratio	= 1.0
Chord	= 16.2 ft

Fuselage

Length	= 107 ft
Diameter	= 11.5 ft
Wetted area	= 3280 ft^2

Vertical Tail

Exposed planform area	= 161 ft^2
t/c	= 0.09
Sweepback	= 43.5 deg
Taper ratio	= 0.80
Root chord	= 15.5 ft

Nacelles

Total wetted area	= 455 ft^2
Effective fineness ratio	= 5.0
Length	= 16.8 ft

Flap Hinge Fairings

Δf	= 0.15 ft^2

Determine

(a) Incompressible parasite drag coefficient and equivalent flat-plate area.
(b) Induced drag coefficient.
(c) Total incompressible drag coefficient.
(d) Total incompressible drag in pounds.
(e) Ratio of lift to drag, neglecting compressibility.

Figure 1: Problem 11.1 in Shevell book

problem, an airplane configuration was given with different parts' metrics. With these values, question (a), (b), and (c) gradually lead to the final drag force in question (d). By plotting drag force with velocity, the optimal velocity for minimum drag force can be found. Also, question (e) asks for the lift to drag ratio which is the objective of this project.

Two solutions to Problem 11.1 can be found from Recommended Homework #5 Solution [2] by T.A. Jordi Ventura Siches and Drag Calculator Project Prompt [3].

#	Component	L_{eff}	Re	C_f	K	S_{wet}	C_{D_P}
1	Wing	11.1263	2.4631E+07	0.002843	1.2135	1693.2	5.8419/1000
2	Fuselage	107	2.3687E+08	0.001997	1.1055	3280	7.243/1000
3	H. Tail	8.0715	1.7868E+07	0.003013	1.1602	532.44	1.861/1000
4	V. Tail	14.0074	3.1009E+07	0.002731	1.1281	328.44	1.0117/1000
5	Pylons	16.2	3.5863E+07	0.002663	1.1347	117	0.3536/1000
6	Nacelles	16.8	3.7192E+07	0.002647	1.2885	455	1.5517/1000

Table 1: Results for Problem 11.1 in Drag Calculator Project Prompt [3].

Everything so far is Problem 11.1 [1] with fixed velocity at 765 [ft/s]. When this velocity is varied from 230 - 880 [ft/s], the Drag Calculator Project Prompt gives two plots in Figure 2 [3].

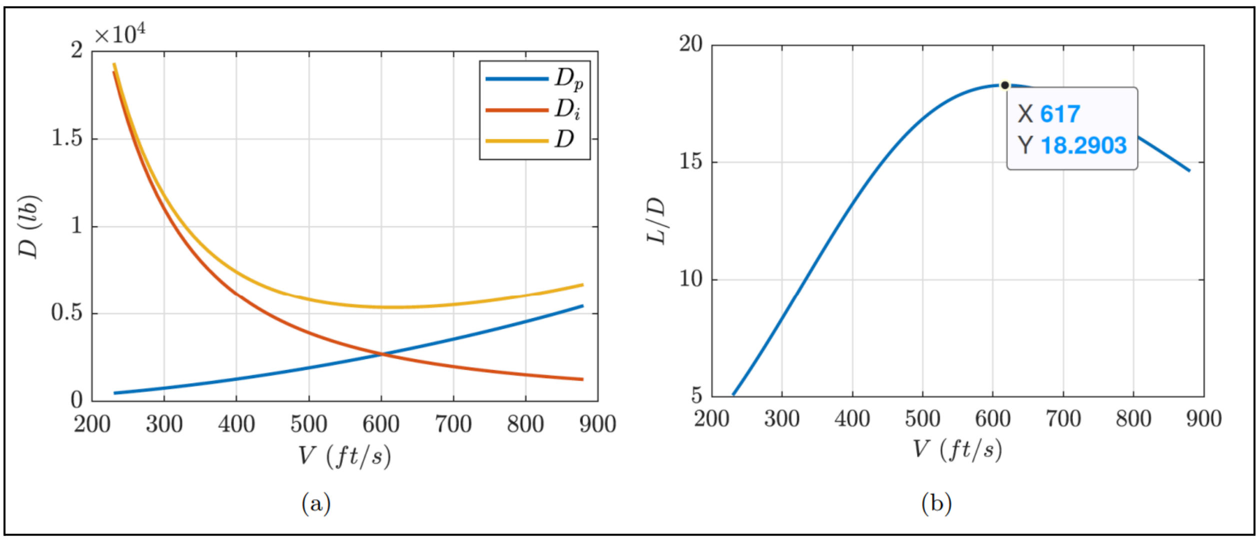


Figure 2: (a) Drag forces versus velocity; (b) Lift to drag ratio versus velocity [3].

For this project, the goal is to repeat these results but with the specific given airplane configuration. The coding process, however, is not just simply replacing the new airplane configuration parameters with the old values. There are sections where it is necessary to use

SolidWorks and data analysis tools like Google Sheet to help determine the important parameters.

2. Drag Calculation Process

The process of creating the drag calculator has 4 main steps:

1. Solve Problem 11.1 [1] in MATLAB.
2. Loop step 1 with varying velocity.
3. Modifying step 2 with the given airplane configuration.

The most work-intensive step was step 1 because it builds up the majority of the code. Step 2 simply adds a loop on top of step 1, so it is particularly easy. Step 3, however, is the most challenging step because it requires incorporating close-to-real-life data, specifically $S_{wet,\ wing}$ and t/c_{wing} through Solidworks.

2.1 Plot Digitalization

MATLAB cannot directly read figures. Therefore, the images of the figures are first digitized. Then these figures are curve fitting through Google Sheets. These curve fitting will give numerical functions so that MATLAB can understand.

$$M_0 = 0.5$$

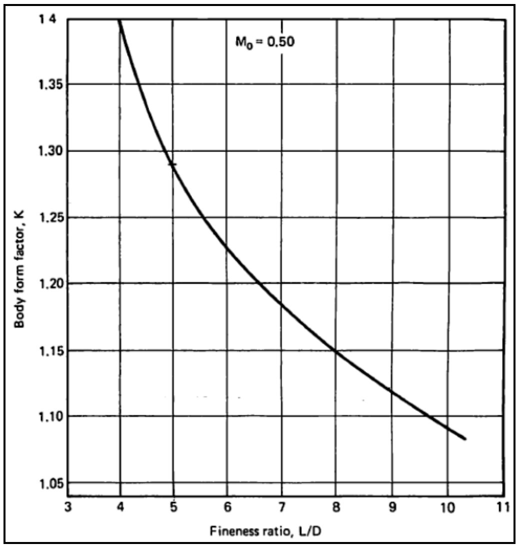
$$K = [1 + Z(t/c) + 100(t/c)^4]$$

where

$$Z = \frac{(2 - M_0^2)\cos\Lambda_{C/4}}{\sqrt{1 - M_0^2\cos^2\Lambda_{C/4}}}$$

Figure 3: Aero surface form factor [1]

Figure 3 shows the calculation for form factor, K, from thickness to chord ratio, t/c. This is the easiest figure to incorporate into the code because they are already given in numerical function form. In the code, there are two lines: one calculates Z as a function of swept angle and one calculates form factor, K, from Z and thickness to chord ratio, t/c.



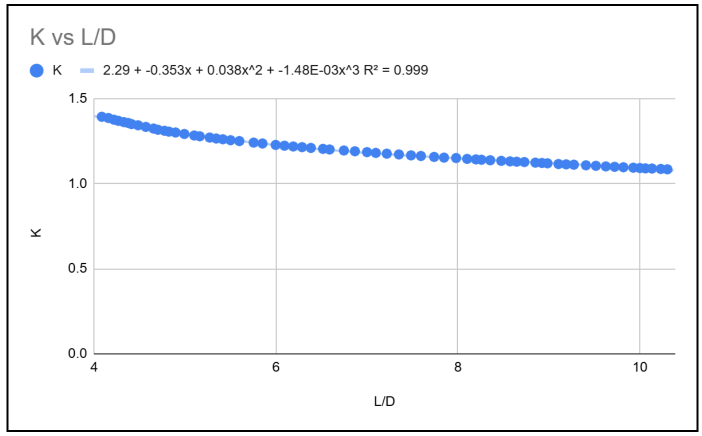
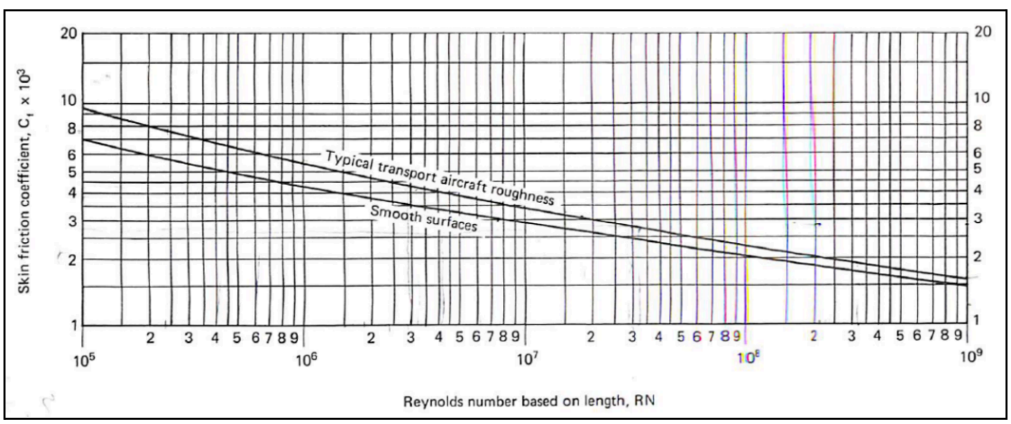


Figure 4: Fineness ratio vs. form factor. Left is from [1] and right is from Google Sheets.

Figure 4 also helps get to form factor, K, but from fineness ratio instead. This is the most successful digitized plot with linear scale axes and a simple 2D curve. In the Google Sheet plot, the accuracy, R^2, is 0.999 = 99.9% which is really good. The curve fitting function is a second degree polynomial, so the computation cost is not a problem.



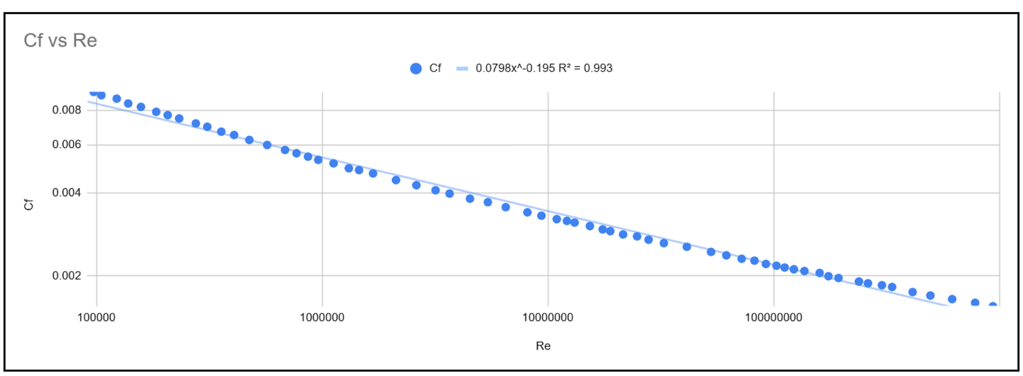


Figure 5: Flat plate skin friction coefficient. Top is from [1] and bottom is from Google Sheets.

In Figure 5, there are two curves: one for typical transport aircraft roughness and one for smooth surfaces. For the purpose of this project, the digitized curve is the top curve “typical transport aircraft roughness.” This is one of the problematic figures because both axes are in log scale and the figure image quality is also not very good. After digitalization, the curve was fitted by a power series function with an accuracy of 99.3%. Although 99.3% sounds high, it was not that good because when comparing the parameters with the solution in Table 1, C_f is not as close to the solution as other values.

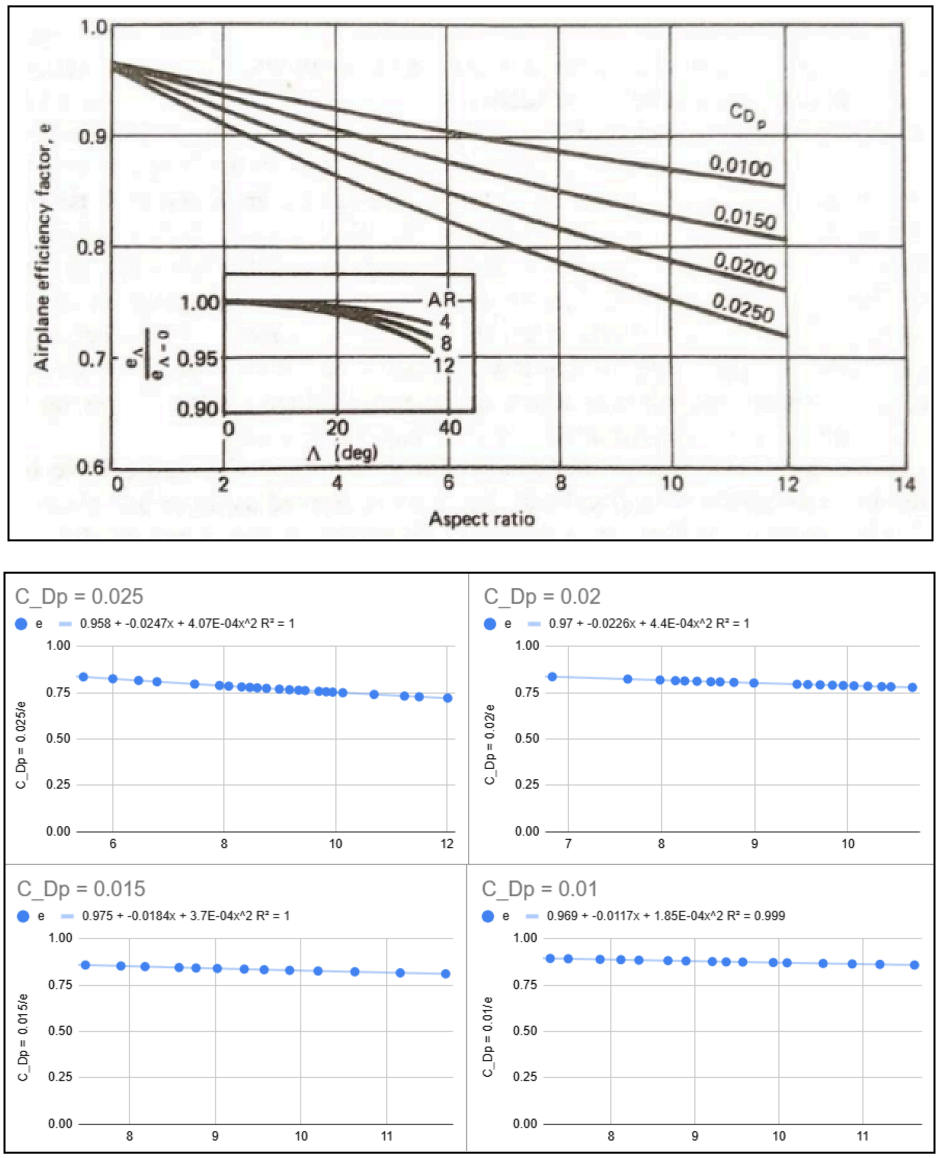


Figure 6: Airplane efficiency factor, e. Top is from [1] and bottom is from Google Sheets.

Figure 6 is the most challenging figure to digitalize. The main problem with it is that there are 3 dimensions presented in a 2D plot. The digitization process can brute force the numerical function of a 2D curve. However, it is impossible to do that infinite amount of time to get to the third dimension. To solve this problem, interpolation is used when C_{Dp} is not the exact value given in the figure. The code block below shows how this works in the MATLAB code.

```
% Oswald efficiency e
if 0.0100 < C_Dp_total & C_Dp_total < 0.0150
  e1 = 0.969 + -0.0117*AR + 1.85E-04*AR^2;
  e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
  e = (e15 - e1) * (C_Dp_total - 0.0100) / (0.0150  - 0.0100) + e1;
elseif 0.0150 < C_Dp_total & C_Dp_total < 0.0200
  e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
  e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
  e = (e2 - e15) * (C_Dp_total - 0.0150) / (0.0200  - 0.0150) + e15;
elseif 0.0200 < C_Dp_total & C_Dp_total < 0.0250
  e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
  e25 = 0.958 + -0.0247*AR + 4.07E-04*AR^2;
  e = (e25 - e2) * (C_Dp_total - 0.0200) / (0.0250  - 0.0200) + e2;
else
  disp('ERROR IN Oswald efficiency e');
end
```

Code Block 1: A snippet using interpolation to find Oswald efficiency e.

2.2 Solve Problem 11.1 in MATLAB

This section is heavily inspired by [2] as most procedures are very similar. Appendix 1 shows the detailed MATLAB code of Problem 11.1. The following sections only discuss the top level key points of the code and skip a lot of smaller details.

2.2.1 Parasite Drag Coefficient

The flow of solving for the Parasite Drag Coefficient is similar for all parts of the airplane:

$$\text{Re and t/c or fineness ratio} \Rightarrow C_f \text{ and } K \Rightarrow C_{Dp}$$

For wing:

$Re = 2.4632*10^7$ and $t/c = 0.106 \Rightarrow C_f = 0.0029$ and $K = 1.2022 \Rightarrow C_{Dp} = 0.0059$

For fuselage:

$Re = 2.3688*10^8$ and fineness ratio = 9.3 ⇒ $C_f = 0.0029$ and K = 1.2022 ⇒ $C_{Dp} = 0.0067$

For horizontal tail:

$Re = 1.7869*10^7$ and t/c = 0.09 ⇒ $C_f = 0.0031$ and K = 1.1548 ⇒ $C_{Dp} = 0.0019$

For vertical tail:

$Re = 3.1010*10^7$ and t/c = 0.09 ⇒ $C_f = 0.0028$ and K = 1.1292 ⇒ $C_{Dp} = 0.0010$

For nacelle:

$Re = 3.7192*10^7$ and fineness ratio = 5 ⇒ $C_f = 0.0027$ and K = 1.29 ⇒ $C_{Dp} = 0.0016$

For pylon:

$Re = 3.59*10^7$ and t/c = 0.06 ⇒ $C_f = 0.0026$ and K = 1.1225 ⇒ $C_{Dp} = 0.0002941$

The total parasite drag coefficient is the sum of all components' parasite drag coefficients multiplied by 1.1 factor accounting for surface roughness.

$$C_{Dp\ total} = 1.1*(C_{Dp,wing} + C_{Dp,fuselage} + C_{Dp,horizontal\ tail} + C_{Dp,vertical\ tail} + C_{Dp,nacelles} + C_{Dp,pylon})$$

$$C_{Dp\ total} = 1.1*(0.0059 + 0.0067 + 0.0019 + 0.0010 + 0.0016 + 0.0002941)$$

$$C_{Dp\ total} = 0.01913351$$

2.2.2 Induced Drag Coefficient

Lift coefficient:

$$W = L = q * S_{ref} * C_L$$

$$C_L = W / (q * S_{ref})$$

$$C_L = 98000 / (256.153 * 1000)$$

$$C_L = 0.3826$$

Induced Drag Coefficient:

$C_{Dp\ total} = 0.01913351$ and AR = 8.6862 ⇒ e = 0.8157 (Code Block 1)

$$C_{Di} = C_L^2 / (\pi * AR * e)$$

$$C_{Di} = 0.3826^2 / (\pi * 8.6862 * 0.8157)$$

$$C_{Di} = 0.0066$$

2.2.3 Drag Forces, Lift Force, Lift to Drag Ratio

Parasite drag force:

$$D_p = q * S_{ref} * C_{Dp\ total}$$

$$D_p = 256.153 * 1000 * 0.01913351$$

$D_p = 4901.1$ [lb]

Induced drag:

$D_i = q * S_{ref} * C_{Di}$

$D_i = 256.153 * 1000 * 0.0066$

$D_i = 1684.5$ [lb]

Total drag force:

$D_{total} = q * S_{ref} * C_{D\ total}$

$D_{total} = q * S_{ref} * (C_{Dp\ total} + C_{Di})$

$D_{total} = 256.153 * 1000 * (\ 0.01913351 + 0.0066)$

$D_{total} = 6591.7$ [lb]

Lift to drag ratio:

$L/D = C_L / C_{D\ total}$

$L/D = C_L / (C_{Dp\ total} + C_{Di})$

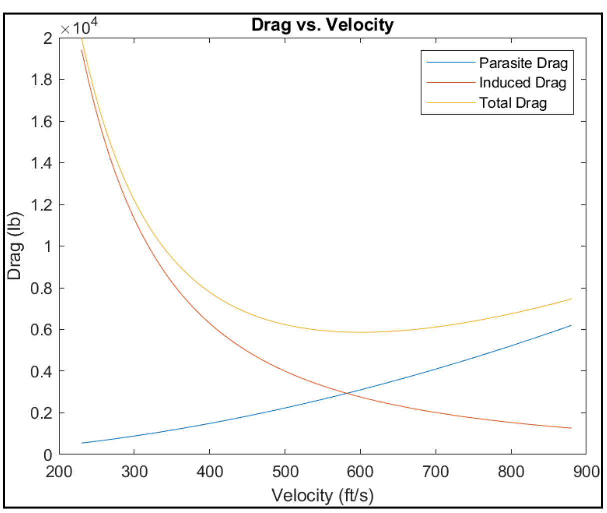
$L/D = 0.3826 / (\ 0.01913351 + 0.0066)$

$L/D = 14.87$

To make sure the code is working as intended, the values generated by the code are compared against the values in Table 1. The only value that is not very close is C_f which is understandable considering the difficulty in digitizing the log scale figure.

2.3 Loop step 1 with varying velocity.

This section is quite simple as all it does is put the code in section 2.3 in a for-loop with velocity, v, as the loop variable from 230 - 880 [ft/s].



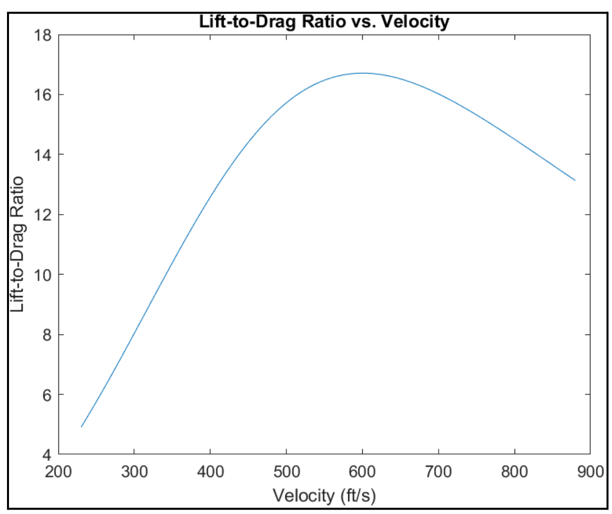


Figure 7: Repeat of Figure 2 using Appendix 2 code.

While the overall shape of the plots are similar, it is worth noting that the optimal velocity is 601 [ft/s] which is at 97.4% of the expected optimal velocity 617 [ft/s] in the prompt. This slightly off value is mainly due to the C_f values not being close to the values in Table 1.

2.4 Modifying step 2 with the given airplane configuration.

The specific given airplane configuration has the following parameters:

1. Wing has BOEING 737 ROOT AIRFOIL at the root and the BOEING 737 OUTBOARD AIRFOIL at the tip
2. b_{wing} = 97 [ft]
3. $c_{r0,wing}$ = 19 [ft]
4. σ_{wing} = 0.26 [1]
5. $\Lambda_{c/4}$ = 6°
6. $L_{fuselage}$ = 92 [ft]
7. $D_{fuselage}$ = 11 [ft]
8. $S_{wet,fuselage}$ = 2543.43 [ft^2]

Because the code is written with variables, most changes can be easily reflected without manually changing every calculation. However, there are some parameters that need to be manually changed including:

1. Wing exposed root chord ($C_{r,e,wing}$).
2. Wing wetted area ($S_{wet,wing}$).
3. Average wing thickness to chord ratio (t/c_{wing})

2.4.1 Wing exposed root chord

To address these parameters in the most realistic value possible, a CAD version of the wing was made in Solidworks. From Figure 8, after inputting all the constraints of the new wing, Solidworks can easily give the length of the exposed root chord as 17.34 [ft].

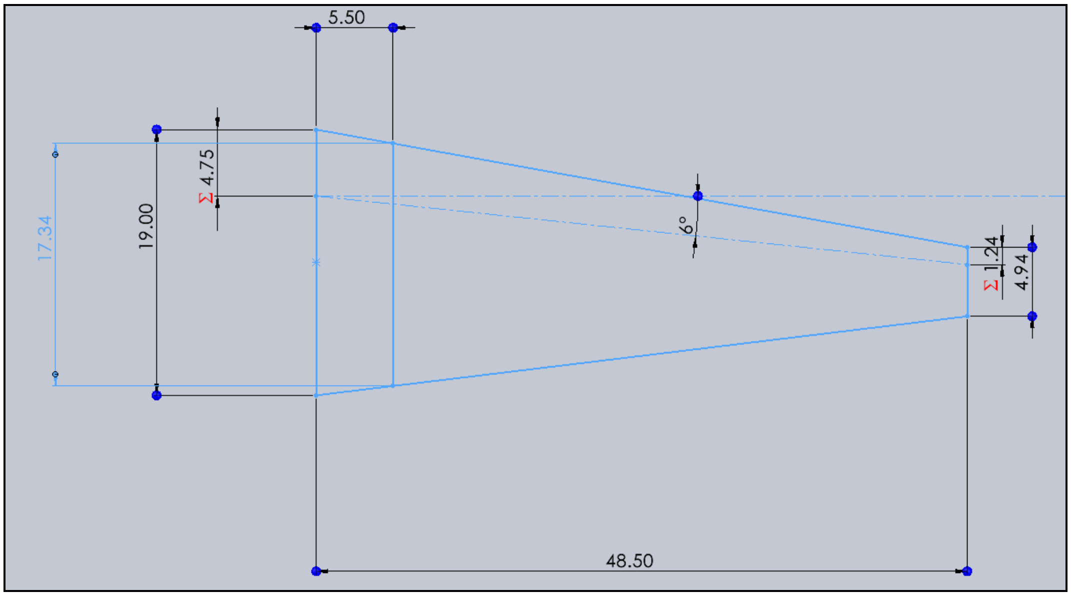


Figure 8: Top down wing sketch with parameters in the given airplane configuration

2.4.2 Wing wetted area

For the wing wetted area, an actual 3D wing was made following [4] but with BOEING 737 AIRFOIL instead. However, just the surface area of the 3D CAD wing is not enough for the wetted area.

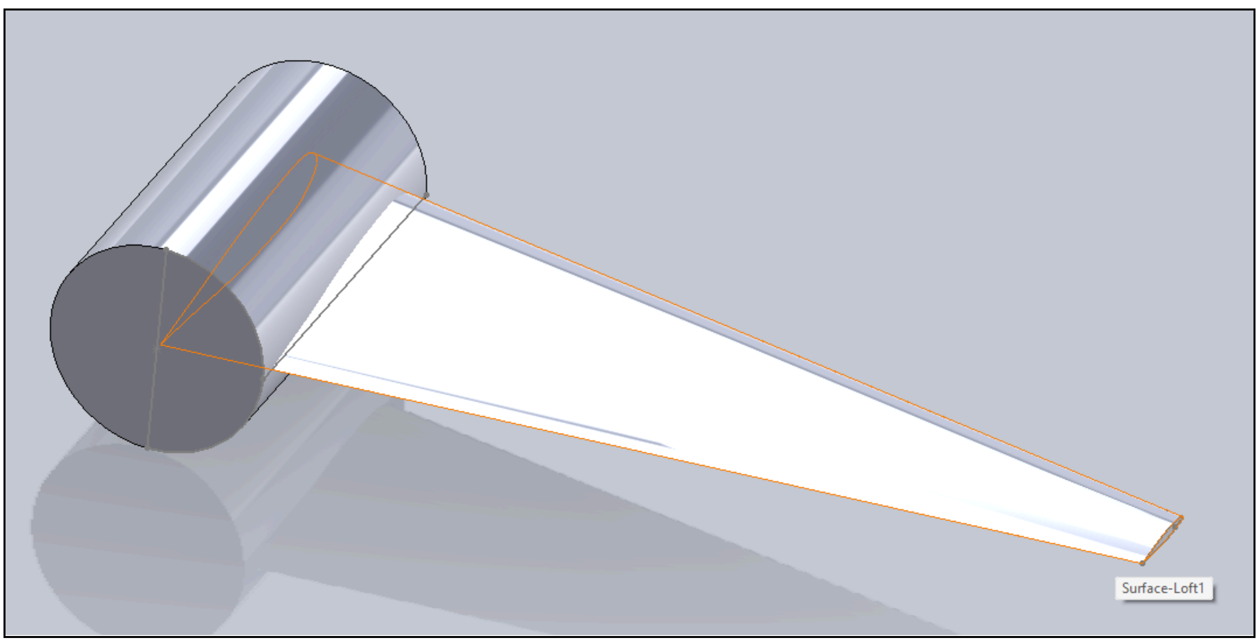


Figure 9: Solidworks CAD of fuselage and wing together.

Figure 9 shows how the fuselage can interfere with the wing and decrease the wetted area. In this CAD, the fuselage was roughly estimated as a cylinder with the given fuselage diameter, 11 [ft]. In order to get the actual wetted area, the fuselage is subtracted from the wing CAD and then SolidWorks evaluate tool is used to measure the exposed surface area of the wing.

2.4.3 Average wing thickness to chord ratio

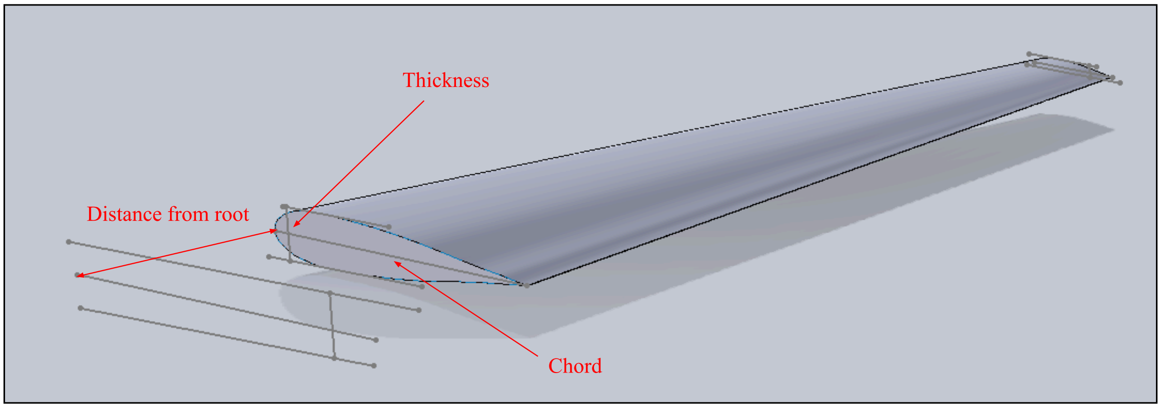


Figure 10: Measure thickness to chord ratio in SolidWorks wing CAD.

To calculate the average wing thickness to chord ratio, the wing was divided into 10 equal sections. In SolidWorks, At every cut, the thickness and chord are measured. These values are then input into table 2 to calculate the thickness to chord ratio at that specific cut. After repeating the process for 11 times, the final value for thickness to chord ratio is simply the average of the previous calculated value.

From Root [ft]	Thickness [ft]	Chord [ft]	t/c [1]
0.0000	2.9294	19.0499	0.1538
4.8500	2.6817	17.6405	0.1520
9.7000	2.4343	16.2311	0.1500
14.5500	2.1876	14.8217	0.1476
19.4000	1.9416	13.4124	0.1448
24.2500	1.6835	11.9248	0.1412
29.1000	1.4556	10.5942	0.1374
33.9500	1.2174	9.1856	0.1325
38.8000	0.9823	7.7778	0.1263
43.6500	0.7543	6.3736	0.1183
48.5000	0.5358	4.9750	0.1077
Average t/c [1]			0.1374

Table 2: Calculate average thickness to chord ratio of the wing.

3. Results

After applying the given airplane configuration their affected parameters, the following results are as follow

3.1 Hand calculations for given airplane configuration at v = 765 [ft/s]

Reynolds number over Characteristic length:

$$\frac{Re}{\ell} = \frac{\rho V}{\mu} = \frac{8.754 \times 10^{-4} \times 765}{3.025 \times 10^{-7}} = 2.2138 \times 10^{6} \left[\frac{1}{ft}\right]$$

Dynamic pressure:

$$q = \frac{1}{2} \rho V^2 = \frac{1}{2} \times 8.754 \times 10^{-4} \times 765^2 = 256.153 \ [lb/ft^2]$$

Parasite drag

WING:

$$\boxed{\begin{aligned} b &= 97 \ [ft] \\ S_{REF} &= 1161.09 \ [ft^2] \\ t/c &= 0.1793 \ [1] \\ \Lambda_{c/4} &= 6^{\circ} \\ \sigma_o &= 0.26 \\ C_{r,o} &= 19 \ [ft] \end{aligned}}$$

Exposed root chord:

Solidworks $\rightarrow$ $C_{r,e} = 17.4056 \ [ft]$

Tip chord:

$$C_T = \sigma_o C_{r,o} = 0.26 \times 19 = 4.94 \ [ft]$$

Mean areodynamic cord:

$$MAC = \frac{2}{3}\left(C_{r,e} + C_T - \frac{C_{r,e} \times C_T}{C_{r,e} + C_T}\right)$$

$$MAC = 12.3318 \ [ft]$$

Reynolds number:

$$Re = 2.2138 \times 10^{6} \ MAC = 2.2138 \times 10^{6} \times 12.3318 = 2.73 \times 10^{7} \ [1]$$

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 2.8 \times 10^{-3}$

Form factor:

Figure 11.3 $\Rightarrow$ $K = 1.46$

Wetted area:

SolidWorks $\Rightarrow$ $S_{wet} = 1990.5 \ [ft^2]$

Parasite drag coefficient:

$$C_{DP} = \frac{K \; C_f \; S_{wet}}{S_{REF}} = \frac{1.46 \times 2.8 \times 10^{-3} \times 1990.5}{1161.09} = 0.007 \quad [1]$$

FUSELAGE

$$\boxed{\begin{array}{l} L = 92 \; [ft] \\ D = 11 \; [ft] \\ S_{wet} = 2543.4 \; [ft^2] \end{array}}$$

Reynolds number:

$$Re = 2.2138 \times 10^6 \times L = 2.2138 \times 10^6 \times 92 = 2.64 \times 10^8 \quad [1]$$

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 2 \times 10^{-3}$

Form factor:

$L/D = 8.36$ & Figure 11.4 $\Rightarrow$ $K = 1.135$

Parasite drag coefficient:

$$C_{DP} = \frac{K \; C_f \; S_{wet}}{S_{REF}} = \frac{1.135 \times 2 \times 10^{-3} \times 2543.4}{1161.09} = 0.004972 \quad [1]$$

HORIZONTAL TAIL

Horizontal Tail	
Exposed planform area	= 261 ft^2
t/c	= 0.09
Sweepback	= 31.6 deg
Taper ratio	= 0.35
Root chord	= 11.1 ft

Mean aerodynamic cord:

$$MAC = \frac{2}{3} c_R \left(1 + \sigma - \frac{\sigma}{1+\sigma}\right) = \frac{2}{3} \, 11.1 \left(1 + 0.35 - \frac{0.35}{1+0.35}\right)$$

$$MAC = 8.07 \; [ft]$$

Reynolds number: $Re = 2.2138 \times 10^6 \times 8.07 = 1.7865 \times 10^7$ [1]

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 3 \times 10^{-3}$

Form factor:

$t/c = 0.09$ & Figure 11.3 $\Rightarrow$ $K = 1.155$

Parasite drag coefficient:

$$C_{DP} = \frac{K\ C_f\ S_{wet}}{S_{REF}} = \frac{1.155 \times 3 \times 10^{-3} \times 2 \times 1.02 \times 261}{1161.09} = 0.0016 \quad [1]$$

VERTICAL TAIL

Vertical Tail	
Exposed planform area	= 161 ft^2
t/c	= 0.09
Sweepback	= 43.5 deg
Taper ratio	= 0.80
Root chord	= 15.5 ft

Mean areodynamic cord:

$$MAC = \frac{2}{3} C_R \left(1 + \sigma - \frac{\sigma}{1+\sigma}\right) = \frac{2}{3} 15.5 \left(1 + 0.8 - \frac{0.8}{1+0.8}\right)$$

$$l = \bar{c} = MAC = 14 \quad [ft]$$

Reynolds number:

$$Re = 2.2138 \times 10^6\ l = 2.2138 \times 10^6 \times 14 = 3.1 \times 10^7 \quad [1]$$

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 2.75 \times 10^{-3}$

Form factor:

$t/c = 0.09$ & Figure 11.3 $\Rightarrow$ $K = 1.155$

Parasite drag coefficient:

$$C_{DP} = \frac{K\ C_f\ S_{wet}}{S_{REF}} = \frac{1.155 \times 2.75 \times 10^{-3} \times 2 \times 1.02 \times 161}{1161.09} = 8.8207 \times 10^{-4} \quad [1]$$

NACELLES

Nacelles	
Total wetted area	= 455 ft^2
Effective fineness ratio	= 5.0
Length	= 16.8 ft

Reynolds number:

$$Re = 2.2138 \times 10^6\ l = 2.2138 \times 10^6 \times 16.8$$

$$Re = 3.72 \times 10^7 \quad [1]$$

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 2.75 \times 10^{-3}$

Form factor:

$L/D = 5$ & Figure 11.4 $\Rightarrow$ $K = 1.28$

Parasite drag coefficient:

$$C_{DP} = \frac{K\, C_f\, S_{wet}}{S_{REF}} = \frac{1.28 \times 2.75 \times 10^{-3} \times 455}{1161.09} = 0.00138 \quad [1]$$

PYLONS

Pylons	
Total wetted area	$= 117\ ft^2$
t/c	$= 0.06\ ft$
Sweepback	$= 0\ °$
Taper Ratio	$= 1$
Chord	$= 16.2\ ft$

Reynolds number:

$$Re = 2.2138 \times 10^6\, \ell = 2.2138 \times 10^6 \times 16.2 = 3.59 \times 10^7 \quad [1]$$

Friction coefficient:

Figure 11.2 $\Rightarrow$ $C_f = 2.6 \times 10^{-3}$

Form factor:

$t/c = 0.06$ & Figure 11.3 $\Rightarrow$ $K = 1.1225$

Parasite drag coefficient:

$$C_{DP} = \frac{K\, C_f\, S_{wet}}{S_{REF}} = \frac{1.1225 \times 2.6 \times 10^{-3} \times 117}{1161.09} = 2.941 \times 10^{-4} \quad [1]$$

Total parasite drag coefficient:

$$C_{DP} = 1.1 \sum C_{DP} = 1.1 (0.007 + 0.004972 + 0.0016 + 8.8207 \times 10^{-4} + 0.00138 + 2.941 \times 10^{-4})$$

$$C_{DP} = 0.01774$$

Induced drag

Lift coefficient:

$$C_L = \frac{W}{q S_{REF}} = \frac{98000}{256.153 \times 1161.09} = 0.3295 \text{ [1]}$$

Aspect ratio:

$$AR = \frac{b^2}{S_{REF}} = \frac{97^2}{1161.09} = 8.1036 \quad \text{[1]}$$

Effiency:

Figure 11.8 $\Rightarrow$ $e = 0.83$

Induced drag coefficient:

$$C_{Di} = \frac{C_L^2}{\pi \, AR \, e} = \frac{0.3295^2}{\pi \times 8.1036 \times 0.83} = 0.005138 \quad \text{[1]}$$

Forces

Parasite drag force:

$$D_p = q \, S_{REF} \, C_{Dp} = 256.153 \times 1161.09 \times 0.01774 = 5276.5 \text{ [lb]}$$

Induced drag force:

$$D_i = q \, S_{REF} \, C_{Di} = 256.153 \times 1161.09 \times 0.005138 = 1528.127 \text{ [lb]}$$

Total drag force:

$$D = q \, S_{REF} \, (C_{Dp} + C_{Di}) = 256.153 \times 1161.09 \times (0.01774 + 0.005138)$$

$$D = 6864.6 \quad \text{[lb]}$$

Lift to drag ratio:

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{0.3295}{0.01742 + 0.005138} = 14.6 \quad \text{[1]}$$

These values are double checked with the code in section 2.4 and Appendix 3. In the MATLAB code, the results are:

$$D_p = 4992.8 \text{ [lb]}$$

$$D_i = 1513.8 \text{ [lb]}$$

$$D_{total} = 6506.6 \text{ [lb]}$$

$$L/D = 15.0616 \text{ [1]}$$

Compare the values in the code and the hand calculation the difference are:

$$D_p \sim 5.6\%$$

$$D_i \sim 0.9\%$$

$$D_{total} \sim 4.6\%$$

$$L/D \sim 4.5\%$$

Again, the main difference is in the log scale figure for C_f.

3.2 Loop given airplane configuration for V = 230 - 880 [ft/s]

Taking the verified code in section 3.1 and loop it with V = 230 - 880 [ft/s], Figure 2 can be reproduced with the given airplane configuration.

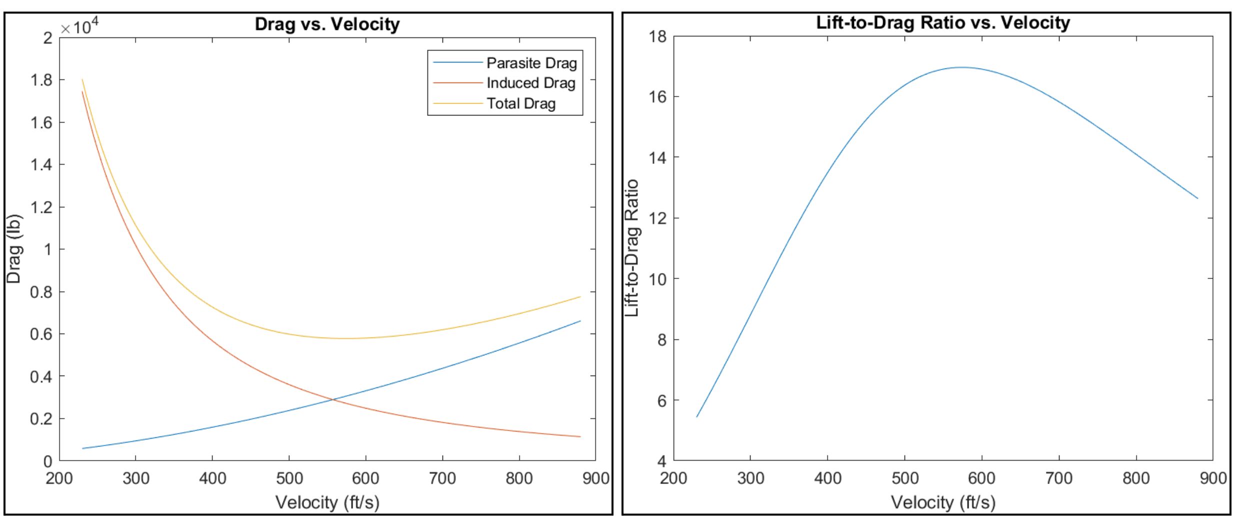


Figure 11: Repeat of Figure 2 using given airplane configuration

3.3 Optimum operating condition of given airplane configuration

From Figure 7, the optimum operating condition is at V = 575 [ft/s]. At this velocity, lift to drag ratio reaches its maximum at 16.9578 and total drag force reaches its minimum at 5779 [lb]

4. Conclusions

Initially, at lower speeds, induced drag is higher than parasite drag. As velocity increases, induced drag decreases while parasite drag increases. At the optimal operating condition, the induced drag and parasite drag are equal. When the velocity is higher than the optimal operating condition, the parasite drag is now higher than the induced drag. The drag forces behave like this with increasing velocity because the parasite drag is proportional with velocity square $D_p \sim V^2$ while the induced drag is inversely proportional with velocity square $D_i \sim 1/V^2$. As velocity increases, D_p increases and D_i decreases. When these two values are combined to get the total drag $D_{total} = D_p + D_i$, the total drag is at its minimum when the two lines D_i and D_p intersect.

In the context of this unique airplane configuration, the optimal operating condition is at V = 575 [ft/s]. Therefore, if the airplane is flying at V = 765 [ft/s], it is on the right of the optimal point. This is more stable because if the airplane velocity temporarily decreases due to the environment, the required thrust decreases.

To maximize the flight speed, the parasitic drag has to decrease. From Figure 7, as the parasite drag curve moves downward, the intersection with induced drag curve moves to the right, leaving higher optimal flight velocity. This also leads to the total drag curve moving downward and to the right. When the available thrust curve intersects with the total drag curve, the intersection on the right side is the maximum flight velocity. With the total drag curve moving downward and to the right, the maximum flight velocity will increase.

5. Acknowledgments

I want to thank Prof. Huynh for teaching and lecturing about the necessary theory in order to grasp the purpose and scope of this project. I would also like to thank T.A. Jordi Ventura Siches for the Recommended Homework #5 Solution.

6. References

[1] R. S. Shevell, Fundamentals of flight, 2nd ed. Upper Saddle River, NJ: Prentice Hall, 1989.

[2] HW 5 Solution.pdf

[3] Drag_Calculator_Project_2024.pdf

[4] SOLIDWORKS Tapered Wing - MAE 158

7. Appendices:

Appendix 1: MATLAB Code for Problem 11.1

```
clc;clear;close all;
% Given Parameter
% Environment condition
rho = 0.0008754; % slug/ft^3
T = 400; % °R
mu = 3.025E-7; % lb*s / ft^2
% Geometry parameter
W = 98000;
% Wing ----------------------------Change later
b_wing = 93.2; % ft
S_ref = 1000; % ft^2
t_c_wing = 0.106;
swept_angle_wing = 24.5;
sigma_wing = 0.2;
C_r_0_wing = 17.8; % ft
expose = 1 - 0.17;
% Fuselage ----------------------------Change later
L_f = 107;
D_f = 11.5;
S_wet_f = 3280;
% Horizontal tail
S_ref_ht = 261;
t_c_ht = 0.09;
swept_angle_ht = 31.6;
sigma_ht = 0.35;
C_r_ht = 11.1;
% Vertical tail
S_ref_vt = 161;
t_c_vt = 0.09;
swept_angle_vt = 43.5;
sigma_vt = 0.8;
C_r_vt = 15.5;
% Pylons
S_wet_p = 117;
t_c_p = 0.06;
swept_angle_p = 0;
sigma_p = 1;
c_p = 16.2;
```

```
% Nacelles
S_wet_n = 455;
Ln_Dn = 5;
L_n = 16.8;
% Prepare list for Graph
D_p = [];
D_i = [];
D_total = [];
L_D = [];
v = 765;
% $$$$$$$$$$$$$$$$ (a) Parasite drag coefficient $$$$$$$$$$$$$$$$$$$$$$$$
% Reynolds number over characteristic length (Re/L)
R_L = (rho*v) /mu;
% Dynamic pressure
q = 0.5*rho*v^2;
% $$$$$$$$$$$$$$$$ Wing $$$$$$$$$$$$$$$$
% Tip cord
C_t_wing = sigma_wing*C_r_0_wing;
% Root cord from Solidworks file
(Wing_MAC.SLDPRT)----------------------------Change later
C_r_e_wing = 16.04291846; % ft
% Mean aerodynamic chord   ----------------------------Change later
MAC_wing = (2/3) * (C_r_e_wing + C_t_wing - (C_r_e_wing *
C_t_wing)/(C_r_e_wing + C_t_wing) ); % ft
% Reynolds number
Re_wing = R_L * MAC_wing;
% Skin friction coefficient Cf
C_f_wing = 0.0798*Re_wing^-0.195;
% Form factor K
Z = ( (2-0.5^2)*cos(deg2rad(swept_angle_wing)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_wing)))^2);
K_wing = 1 + Z*t_c_wing + 100*t_c_wing^4;
% Wetted area
------------------------------------------------------------Change later
S_wet_wing = 2*1.02*S_ref*expose;
% Equivalent profile drag area of the wing
f_wing = K_wing*C_f_wing*S_wet_wing;
% Parasite drag coefficient
```

```
C_Dp_wing = f_wing/S_ref;
% $$$$$$$$$$$$$$$$ Fuselage $$$$$$$$$$$$$$$$
% Reynolds number
Re_f = R_L * L_f;
% Skin friction coefficient Cf
C_f_f = 0.0798*Re_f^(-0.195);
% Form factor K
Lf_Df = L_f/D_f;
K_f = 2.29 + -0.353*Lf_Df + 0.038*Lf_Df^2 + -1.48E-03*Lf_Df^3;
% Equivalent profile drag area of the wing
f_f = K_f * C_f_f * S_wet_f;
% Parasite drag coefficient
C_Dp_fuselage = f_f/S_ref;
% $$$$$$$$$$$$$$$$ Horizontal Tail $$$$$$$$$$$$$$$$
% Mean aerodynamic chord
MAC_ht = (2/3) * C_r_ht * (1 + sigma_ht - sigma_ht / ( 1 + sigma_ht)); % ft
% Reynolds number
Re_ht = R_L * MAC_ht;
% Skin friction coefficient Cf
C_f_ht = 0.0798*Re_ht^-0.195;
% Form factor K
Z_ht = ( (2-0.5^2)*cos(deg2rad(swept_angle_ht)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_ht)))^2);
K_ht = 1 + Z_ht*t_c_ht + 100*t_c_ht^4;
% Wetted area
S_wet_ht = 2*1.02*S_ref_ht;
% Equivalent profile drag area of the wing
f_ht = K_ht*C_f_ht*S_wet_ht;
% Parasite drag coefficient
C_Dp_horizontal_tail = f_ht/S_ref;
% $$$$$$$$$$$$$$$$ Vertical Tail $$$$$$$$$$$$$$$$
% Mean aerodynamic chord
MAC_vt = (2/3) * C_r_vt * (1 + sigma_vt - sigma_vt / ( 1 + sigma_vt)); % ft
% Reynolds number
Re_vt = R_L * MAC_vt;
% Skin friction coefficient Cf
C_f_vt = 0.0798*Re_vt^-0.195;
% Form factor K
```

```
Z_vt = ( (2-0.5^2)*cos(deg2rad(swept_angle_vt)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_vt)))^2);
K_vt = 1 + Z_vt*t_c_vt + 100*t_c_vt^4;
% Wetted area
S_wet_vt = 2*1.02*S_ref_vt;
% Equivalent profile drag area of the wing
f_vt = K_vt*C_f_vt*S_wet_vt;
% Parasite drag coefficient
C_Dp_vertical_tail = f_vt/S_ref;
% $$$$$$$$$$$$$$$$ Nacelles $$$$$$$$$$$$$$$$
% Reynolds number
Re_n = R_L * L_n;
% Skin friction coefficient Cf
C_f_n = 0.0798*Re_n^(-0.195);
% Form factor K
K_n = 2.29 + -0.353*Ln_Dn + 0.038*Ln_Dn^2 + -1.48E-03*Ln_Dn^3;
% Equivalent profile drag area of the wing
f_n = K_n * C_f_n * S_wet_n;
% Parasite drag coefficient
C_Dp_nacelles = f_n/S_ref;
% $$$$$$$$$$$$$$$$ Pylon $$$$$$$$$$$$$$$$
% Reynolds number
Re_p = R_L * c_p;
% Skin friction coefficient Cf
C_f_p = 0.0798*Re_p^(-0.195);
% Form factor K
Z_p = ( (2-0.5^2)*cos(deg2rad(swept_angle_p)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_p)))^2);
K_p = 1 + Z_p*t_c_p + 100*t_c_p^4;
% Equivalent profile drag area of the wing
f_p = K_p * C_f_p * S_wet_p;
% Parasite drag coefficient
C_Dp_pylon = f_p/S_ref;
% TOTAL Parasite drag coefficient
C_Dp_total = 1.1*(C_Dp_wing + C_Dp_fuselage + C_Dp_horizontal_tail +
C_Dp_vertical_tail + C_Dp_nacelles + C_Dp_pylon);
% $$$$$$$$$$$$$$$$ (b) Induced drag coefficient $$$$$$$$$$$$$$$$$$$$$$$$
% Lift coefficient
```

```matlab
C_L = W / (q*S_ref);
% Aspect Ratio
AR = b_wing^2 / S_ref;
% Oswald efficiency e
if 0.0100 < C_Dp_total & C_Dp_total < 0.0150
   e1 = 0.969 + -0.0117*AR + 1.85E-04*AR^2;
   e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
   e = (e15 - e1) * (C_Dp_total - 0.0100) / (0.0150  - 0.0100) + e1;
elseif 0.0150 < C_Dp_total & C_Dp_total < 0.0200
   e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
   e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
   e = (e2 - e15) * (C_Dp_total - 0.0150) / (0.0200  - 0.0150) + e15;
elseif 0.0200 < C_Dp_total & C_Dp_total < 0.0250
   e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
   e25 = 0.958 + -0.0247*AR + 4.07E-04*AR^2;
   e = (e25 - e2) * (C_Dp_total - 0.0200) / (0.0250  - 0.0200) + e2;
else
   disp('ERROR IN Oswald efficiency e');
end
% Induced drag coefficient
C_Di = C_L^2 / (pi*AR*e);
% $$$$$$$$$$$$$$$$ Forces $$$$$$$$$$$$$$$$$$$$$$$$
% Parasite drag
D_p= q*S_ref*C_Dp_total
% Induced drag
D_i = q*S_ref*C_Di
% Total Drag
C_D_total = C_Di + C_Dp_total;
D_total = q*S_ref*C_D_total
% Lift to Drag ratio
L_D = C_L/C_D_total
```

Appendix 2: MATLAB Code for Problem 11.1 in a For Loop

```
clc;clear;close all;
% Given Parameter
% Environment condition
rho = 0.0008754; % slug/ft^3
T = 400; % °R
mu = 3.025E-7; % lb*s / ft^2
% Geometry parameter
W = 98000;
% Wing ----------------------------Change later
b_wing = 93.2; % ft
S_ref = 1000; % ft^2
t_c_wing = 0.106;
swept_angle_wing = 24.5;
sigma_wing = 0.2;
C_r_0_wing = 17.8; % ft
expose = 1 - 0.17;
% Fuselage ----------------------------Change later
L_f = 107;
D_f = 11.5;
S_wet_f = 3280;
% Horizontal tail
S_ref_ht = 261;
t_c_ht = 0.09;
swept_angle_ht = 31.6;
sigma_ht = 0.35;
C_r_ht = 11.1;
% Vertical tail
S_ref_vt = 161;
t_c_vt = 0.09;
swept_angle_vt = 43.5;
sigma_vt = 0.8;
C_r_vt = 15.5;
% Pylons
S_wet_p = 117;
t_c_p = 0.06;
swept_angle_p = 0;
sigma_p = 1;
c_p = 16.2;
```

```
% Nacelles
S_wet_n = 455;
Ln_Dn = 5;
L_n = 16.8;
% Prepare list for Graph
D_p = [];
D_i = [];
D_total = [];
L_D = [];
for v = 230:880 % True air speed
% $$$$$$$$$$$$$$$$ (a) Parasite drag coefficient $$$$$$$$$$$$$$$$$$$$$$$$
% Reynolds number over characteristic length (Re/L)
R_L = (rho*v) /mu;
% Dynamic pressure
q = 0.5*rho*v^2;
% $$$$$$$$$$$$$$$$ Wing $$$$$$$$$$$$$$$$
% Tip cord
C_t_wing = sigma_wing*C_r_0_wing;
% Root cord from Solidworks file
(Wing_MAC.SLDPRT)----------------------------Change later
C_r_e_wing = 16.04291846; % ft
% Mean aerodynamic chord   ----------------------------Change later
MAC_wing = (2/3) * (C_r_e_wing + C_t_wing - (C_r_e_wing *
C_t_wing)/(C_r_e_wing + C_t_wing) ); % ft
% Reynolds number
Re_wing = R_L * MAC_wing;
% Skin friction coefficient Cf
C_f_wing = 0.0798*Re_wing^-0.195;
% Form factor K
Z = ( (2-0.5^2)*cos(deg2rad(swept_angle_wing)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_wing)))^2);
K_wing = 1 + Z*t_c_wing + 100*t_c_wing^4;
% Wetted area
------------------------------------------------------------Change later
S_wet_wing = 2*1.02*S_ref*expose;
% Equivalent profile drag area of the wing
f_wing = K_wing*C_f_wing*S_wet_wing;
% Parasite drag coefficient
```

```
C_Dp_wing = f_wing/S_ref;
% $$$$$$$$$$$$$$$$ Fuselage $$$$$$$$$$$$$$$$
% Reynolds number
Re_f = R_L * L_f;
% Skin friction coefficient Cf
C_f_f = 0.0798*Re_f^(-0.195);
% Form factor K
Lf_Df = L_f/D_f;
K_f = 2.29 + -0.353*Lf_Df + 0.038*Lf_Df^2 + -1.48E-03*Lf_Df^3;
% Equivalent profile drag area of the wing
f_f = K_f * C_f_f * S_wet_f;
% Parasite drag coefficient
C_Dp_fuselage = f_f/S_ref;
% $$$$$$$$$$$$$$$$ Horizontal Tail $$$$$$$$$$$$$$$$
% Mean aerodynamic chord
MAC_ht = (2/3) * C_r_ht * (1 + sigma_ht - sigma_ht / ( 1 + sigma_ht)); % ft
% Reynolds number
Re_ht = R_L * MAC_ht;
% Skin friction coefficient Cf
C_f_ht = 0.0798*Re_ht^-0.195;
% Form factor K
Z_ht = ( (2-0.5^2)*cos(deg2rad(swept_angle_ht)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_ht)))^2);
K_ht = 1 + Z_ht*t_c_ht + 100*t_c_ht^4;
% Wetted area
S_wet_ht = 2*1.02*S_ref_ht;
% Equivalent profile drag area of the wing
f_ht = K_ht*C_f_ht*S_wet_ht;
% Parasite drag coefficient
C_Dp_horizontal_tail = f_ht/S_ref;
% $$$$$$$$$$$$$$$$ Vertical Tail $$$$$$$$$$$$$$$$
% Mean aerodynamic chord
MAC_vt = (2/3) * C_r_vt * (1 + sigma_vt - sigma_vt / ( 1 + sigma_vt)); % ft
% Reynolds number
Re_vt = R_L * MAC_vt;
% Skin friction coefficient Cf
C_f_vt = 0.0798*Re_vt^-0.195;
% Form factor K
```

```
Z_vt = ( (2-0.5^2)*cos(deg2rad(swept_angle_vt)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_vt)))^2);
K_vt = 1 + Z_vt*t_c_vt + 100*t_c_vt^4;
% Wetted area
S_wet_vt = 2*1.02*S_ref_vt;
% Equivalent profile drag area of the wing
f_vt = K_vt*C_f_vt*S_wet_vt;
% Parasite drag coefficient
C_Dp_vertical_tail = f_vt/S_ref;
% $$$$$$$$$$$$$$$$ Nacelles $$$$$$$$$$$$$$$$
% Reynolds number
Re_n = R_L * L_n;
% Skin friction coefficient Cf
C_f_n = 0.0798*Re_n^(-0.195);
% Form factor K
K_n = 2.29 + -0.353*Ln_Dn + 0.038*Ln_Dn^2 + -1.48E-03*Ln_Dn^3;
% Equivalent profile drag area of the wing
f_n = K_n * C_f_n * S_wet_n;
% Parasite drag coefficient
C_Dp_nacelles = f_n/S_ref;
% $$$$$$$$$$$$$$$$ Pylon $$$$$$$$$$$$$$$$
% Reynolds number
Re_p = R_L * c_p;
% Skin friction coefficient Cf
C_f_p = 0.0798*Re_p^(-0.195);
% Form factor K
Z_p = ( (2-0.5^2)*cos(deg2rad(swept_angle_p)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_p)))^2);
K_p = 1 + Z_p*t_c_p + 100*t_c_p^4;
% Equivalent profile drag area of the wing
f_p = K_p * C_f_p * S_wet_p;
% Parasite drag coefficient
C_Dp_pylon = f_p/S_ref;
% TOTAL Parasite drag coefficient
C_Dp_total = 1.1*(C_Dp_wing + C_Dp_fuselage + C_Dp_horizontal_tail +
C_Dp_vertical_tail + C_Dp_nacelles + C_Dp_pylon);
% $$$$$$$$$$$$$$$$ (b) Induced drag coefficient $$$$$$$$$$$$$$$$$$$$$$$$
% Lift coefficient
```

```
C_L = W / (q*S_ref);
% Aspect Ratio
AR = b_wing^2 / S_ref;
% Oswald efficiency e
if 0.0100 < C_Dp_total & C_Dp_total < 0.0150
   e1 = 0.969 + -0.0117*AR + 1.85E-04*AR^2;
   e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
   e = (e15 - e1) * (C_Dp_total - 0.0100) / (0.0150  - 0.0100) + e1;
elseif 0.0150 < C_Dp_total & C_Dp_total < 0.0200
   e15 = 0.975 + -0.0184*AR + 3.7E-04*AR^2;
   e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
   e = (e2 - e15) * (C_Dp_total - 0.0150) / (0.0200  - 0.0150) + e15;
elseif 0.0200 < C_Dp_total & C_Dp_total < 0.0250
   e2 = 0.97 + -0.0226*AR + 4.4E-04*AR^2;
   e25 = 0.958 + -0.0247*AR + 4.07E-04*AR^2;
   e = (e25 - e2) * (C_Dp_total - 0.0200) / (0.0250  - 0.0200) + e2;
else
   disp('ERROR IN Oswald efficiency e');
end
% Induced drag coefficient
C_Di = C_L^2 / (pi*AR*e);
% $$$$$$$$$$$$$$$$ Forces $$$$$$$$$$$$$$$$$$$$$$$$
% Parasite drag
D_p(end+1) = q*S_ref*C_Dp_total;
% Induced drag
D_i(end+1) = q*S_ref*C_Di;
% Total Drag
C_D_total = C_Di + C_Dp_total;
D_total(end+1) = q*S_ref*C_D_total;
% Lift to Drag ratio
L_D(end+1) = C_L/C_D_total;
end
v_list = 230:880; % Velocity range for plotting
% Plot Parasite, Induced, and Total Drag vs Velocity
figure;
plot(v_list, D_p, v_list, D_i, v_list, D_total);
xlabel('Velocity (ft/s)');
ylabel('Drag (lb)');
```

```
legend('Parasite Drag', 'Induced Drag', 'Total Drag');
title('Drag vs. Velocity');
% Plot Lift-to-Drag Ratio vs Velocity
figure;
plot(v_list, L_D);
xlabel('Velocity (ft/s)');
ylabel('Lift-to-Drag Ratio');
title('Lift-to-Drag Ratio vs. Velocity');
```

Appendix 3: MATLAB Code for Given Airplane Configuration

```matlab
% Clear workspace and close figures
clc; clear; close all;
% ===================== Environment Conditions =====================
rho = 0.0008754; % Air density (slug/ft^3)
T = 400;         % Temperature (°R)
mu = 3.025E-7;   % Dynamic viscosity (lb*s/ft^2)
% ===================== Geometry Parameters =====================
% Aircraft weight
W = 98000; % Weight (lb)
% Wing geometry
b_wing = 97;                % Wingspan (ft)
S_ref = 580.545 * 2;        % Reference area (ft^2)
t_c_wing = 0.1374;    % Thickness-to-chord ratio
swept_angle_wing = 6;       % Sweep angle (degrees)
sigma_wing = 0.26;          % Taper ratio
C_r_0_wing = 19;            % Root chord length (ft)
% Fuselage geometry
L_f = 92;                       % Fuselage length (ft)
D_f = 11;                       % Fuselage diameter (ft)
S_wet_f = 0.8 * pi * D_f * L_f; % Wetted area (ft^2)
% Horizontal tail geometry
S_ref_ht = 261;            % Horizontal tail area (ft^2)
t_c_ht = 0.09;             % Thickness-to-chord ratio
swept_angle_ht = 31.6;     % Sweep angle (degrees)
sigma_ht = 0.35;           % Taper ratio
C_r_ht = 11.1;             % Root chord length (ft)
% Vertical tail geometry
S_ref_vt = 161;            % Vertical tail area (ft^2)
t_c_vt = 0.09;             % Thickness-to-chord ratio
swept_angle_vt = 43.5;     % Sweep angle (degrees)
sigma_vt = 0.8;            % Taper ratio
C_r_vt = 15.5;             % Root chord length (ft)
% Pylon geometry
S_wet_p = 117;          % Wetted area (ft^2)
t_c_p = 0.06;           % Thickness-to-chord ratio
swept_angle_p = 0;      % Sweep angle (degrees)
sigma_p = 1;            % Taper ratio
c_p = 16.2;             % Chord length (ft)
```

```
% Nacelle geometry
S_wet_n = 455;          % Wetted area (ft^2)
Ln_Dn = 5;              % Length-to-diameter ratio
L_n = 16.8;             % Nacelle length (ft)
% ===================== Arrays for Graphing Results =====================
D_p = [];        % Parasite drag
D_i = [];        % Induced drag
D_total = [];    % Total drag
L_D = [];        % Lift-to-drag ratio
% ===================== Loop over Velocity Range =====================
for v = 230:880 % True airspeed (ft/s)
   % ===== (a) Parasite Drag Coefficient Calculation =====
   R_L = (rho * v) / mu;        % Reynolds number over characteristic length
   q = 0.5 * rho * v^2;         % Dynamic pressure (lb/ft^2)

   % ----- Wing Parasite Drag -----
   C_t_wing = sigma_wing * C_r_0_wing;      % Tip chord
   C_r_e_wing = 17.4056;                    % Root chord from model (ft)
   MAC_wing = (2/3) * (C_r_e_wing + C_t_wing - (C_r_e_wing *
C_t_wing)/(C_r_e_wing + C_t_wing)); % Mean aerodynamic chord (ft)
   Re_wing = R_L * MAC_wing;                % Reynolds number (wing)
   C_f_wing = 0.0798 * Re_wing^-0.195;      % Skin friction coefficient (wing)
   Z = ((2 - 0.5^2) * cosd(swept_angle_wing)) / sqrt(1 - 0.5^2 *
cosd(swept_angle_wing)^2);
   K_wing = 1 + Z * t_c_wing + 100 * t_c_wing^4;  % Form factor
   S_wet_wing = 2*(498.95 + 496.3);         % Wetted area from model (ft^2)
   f_wing = K_wing * C_f_wing * S_wet_wing; % Equivalent profile drag area
   C_Dp_wing = f_wing / S_ref;              % Parasite drag coefficient (wing)
   % ----- Fuselage Parasite Drag -----
   Re_f = R_L * L_f;                        % Reynolds number (fuselage)
   C_f_f = 0.0798 * Re_f^(-0.195);          % Skin friction coefficient
(fuselage)
   Lf_Df = L_f / D_f;                       % Length-to-diameter ratio
   K_f = 2.29 - 0.353 * Lf_Df + 0.038 * Lf_Df^2 - 1.48E-03 * Lf_Df^3; % Form
factor (fuselage)
   f_f = K_f * C_f_f * S_wet_f;             % Equivalent profile drag area
   C_Dp_fuselage = f_f / S_ref;             % Parasite drag coefficient
(fuselage)
```

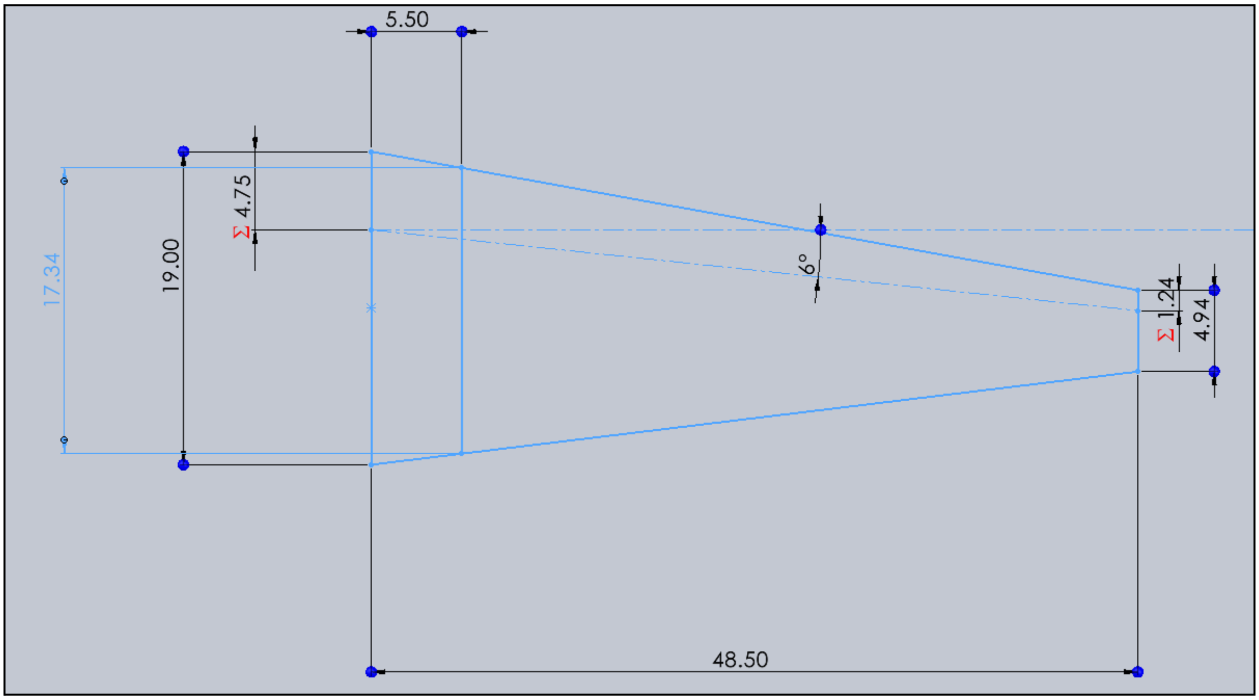
```
  % ----- Horizontal Tail Parasite Drag -----
  MAC_ht = (2/3) * C_r_ht * (1 + sigma_ht - sigma_ht / (1 + sigma_ht)); %
Mean aerodynamic chord (ft)
  Re_ht = R_L * MAC_ht;                  % Reynolds number (horizontal tail)
  C_f_ht = 0.0798 * Re_ht^-0.195;        % Skin friction coefficient
(horizontal tail)
  Z_ht = ((2 - 0.5^2) * cosd(swept_angle_ht)) / sqrt(1 - 0.5^2 *
cosd(swept_angle_ht)^2);
  K_ht = 1 + Z_ht * t_c_ht + 100 * t_c_ht^4; % Form factor
  S_wet_ht = 2 * 1.02 * S_ref_ht;        % Wetted area (ft^2)
  f_ht = K_ht * C_f_ht * S_wet_ht;       % Equivalent profile drag area
  C_Dp_horizontal_tail = f_ht / S_ref;   % Parasite drag coefficient
(horizontal tail)
  % ----- Vertical Tail Parasite Drag -----
  MAC_vt = (2/3) * C_r_vt * (1 + sigma_vt - sigma_vt / (1 + sigma_vt)); %
Mean aerodynamic chord (ft)
  Re_vt = R_L * MAC_vt;                  % Reynolds number (vertical tail)
  C_f_vt = 0.0798 * Re_vt^-0.195;        % Skin friction coefficient
(vertical tail)
  Z_vt = ((2 - 0.5^2) * cosd(swept_angle_vt)) / sqrt(1 - 0.5^2 *
cosd(swept_angle_vt)^2);
  K_vt = 1 + Z_vt * t_c_vt + 100 * t_c_vt^4; % Form factor
  S_wet_vt = 2 * 1.02 * S_ref_vt;        % Wetted area (ft^2)
  f_vt = K_vt * C_f_vt * S_wet_vt;       % Equivalent profile drag area
  C_Dp_vertical_tail = f_vt / S_ref;     % Parasite drag coefficient
(vertical tail)
  % ----- Nacelle Parasite Drag -----
  Re_n = R_L * L_n;                     % Reynolds number (nacelle)
  C_f_n = 0.0798 * Re_n^-0.195;         % Skin friction coefficient (nacelle)
  K_n = 2.29 + -0.353*Ln_Dn + 0.038*Ln_Dn^2 + -1.48E-03*Ln_Dn^3;         %
Form factor based on length-to-diameter ratio
  f_n = K_n * C_f_n * S_wet_n;          % Equivalent profile drag area for
nacelle
  C_Dp_nacelles = f_n / S_ref;          % Parasite drag coefficient (nacelle)
  % ----- Pylon Parasite Drag -----
  Re_p = R_L * c_p;                     % Reynolds number (Pylon)
  C_f_p = 0.0798*Re_p^(-0.195);         % Skin friction coefficient (Pylon)
```

```
    Z_p = ( (2-0.5^2)*cos(deg2rad(swept_angle_p)) ) / sqrt(1 -
0.5^2*(cos(deg2rad(swept_angle_p)))^2);
    K_p = 1 + Z_p*t_c_p + 100*t_c_p^4;     % Form factor
    f_p = K_p * C_f_p * S_wet_p;           % Equivalent profile drag area for
Pylon
    C_Dp_pylon = f_p/S_ref;                % Parasite drag coefficient (Pylon)
    % ===== Total Parasite Drag Coefficient =====
    C_Dp_total = 1.1 * (C_Dp_wing + C_Dp_fuselage + C_Dp_horizontal_tail +
C_Dp_vertical_tail + C_Dp_nacelles + C_Dp_pylon);
    % ===== (b) Induced Drag Coefficient Calculation =====
    C_L = W / (q * S_ref);                  % Lift coefficient
    AR = b_wing^2 / S_ref;                  % Aspect ratio

    % ----- Oswald Efficiency Factor (e) -----
    if 0.0100 < C_Dp_total && C_Dp_total < 0.0150
        e1 = 0.969 - 0.0117 * AR + 1.85E-04 * AR^2;
        e15 = 0.975 - 0.0184 * AR + 3.7E-04 * AR^2;
        e = (e15 - e1) * (C_Dp_total - 0.0100) / (0.0150 - 0.0100) + e1;

    elseif 0.0150 < C_Dp_total && C_Dp_total < 0.0200
        e15 = 0.975 - 0.0184 * AR + 3.7E-04 * AR^2;
        e2 = 0.97 - 0.0226 * AR + 4.4E-04 * AR^2;
        e = (e2 - e15) * (C_Dp_total - 0.0150) / (0.0200 - 0.0150) + e15;

    elseif 0.0200 < C_Dp_total && C_Dp_total < 0.0250
        e2 = 0.97 - 0.0226 * AR + 4.4E-04 * AR^2;
        e25 = 0.958 - 0.0247 * AR + 4.07E-04 * AR^2;
        e = (e25 - e2) * (C_Dp_total - 0.0200) / (0.0250 - 0.0200) + e2;

    else
        disp('ERROR IN Oswald efficiency e');
    end
    % Induced drag coefficient
    C_Di = C_L^2 / (pi * AR * e);
    % ===== Forces =====
    D_p(end+1) = q * S_ref * C_Dp_total;        % Parasite drag
    D_i(end+1) = q * S_ref * C_Di;              % Induced drag
    C_D_total = C_Di + C_Dp_total;              % Total drag coefficient
```

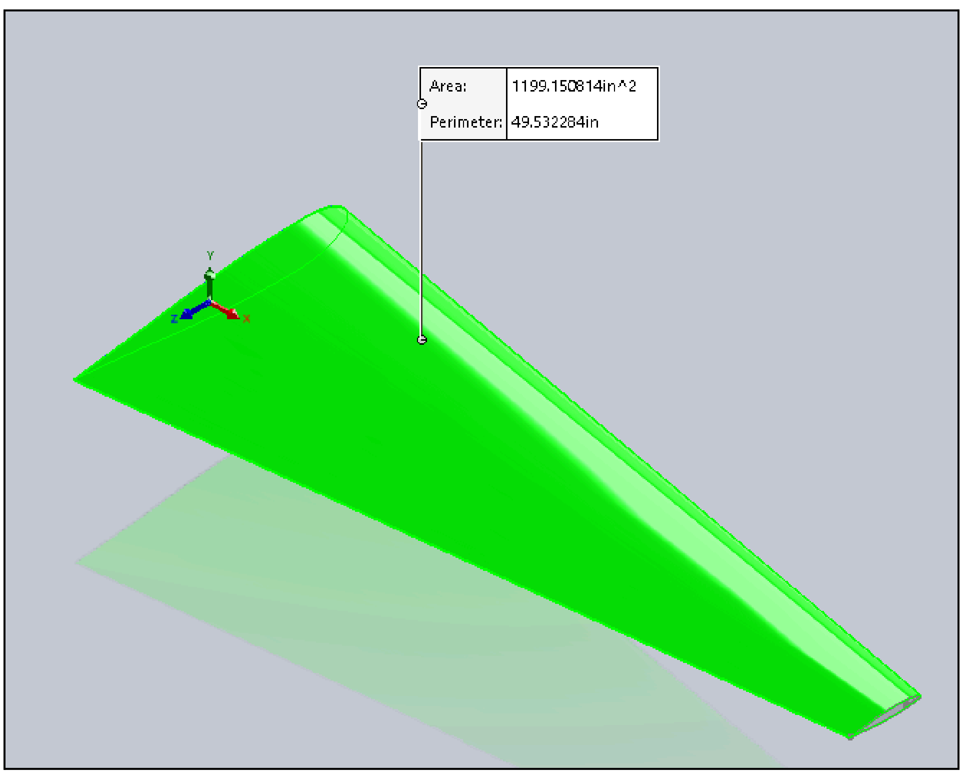
```
    D_total(end+1) = q * S_ref * C_D_total;     % Total drag
    L_D(end+1) = C_L / C_D_total;               % Lift-to-drag ratio
end
% ===================== Plotting Results =====================
v_list = 230:880; % Velocity range for plotting
% Plot Parasite, Induced, and Total Drag vs Velocity
figure;
plot(v_list, D_p, v_list, D_i, v_list, D_total);
xlabel('Velocity (ft/s)');
ylabel('Drag (lb)');
legend('Parasite Drag', 'Induced Drag', 'Total Drag');
title('Drag vs. Velocity');
% Plot Lift-to-Drag Ratio vs Velocity
figure;
plot(v_list, L_D);
xlabel('Velocity (ft/s)');
ylabel('Lift-to-Drag Ratio');
title('Lift-to-Drag Ratio vs. Velocity');
```

Appendix 4: SolidWorks Wing CAD

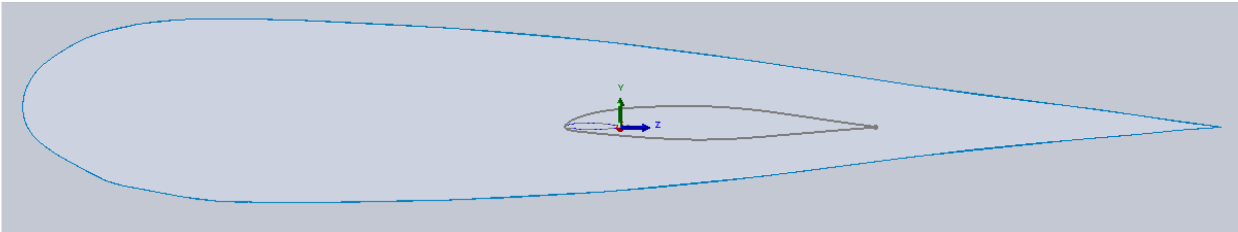
Wing sketch from top-down view. Unit is in [ft]



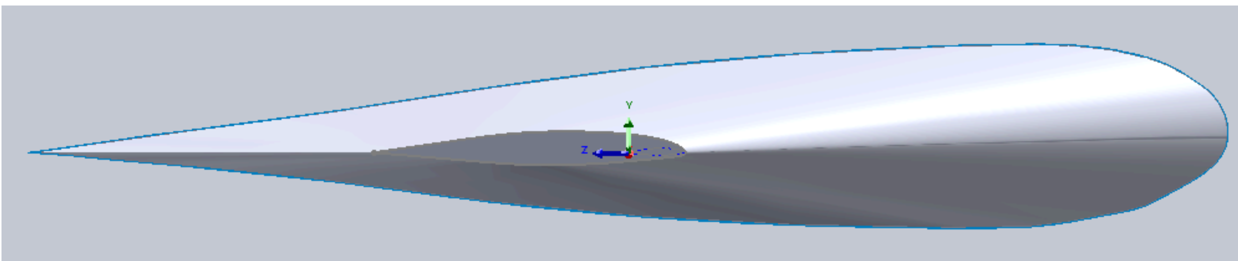
Wing from centerline root chord to tip chord



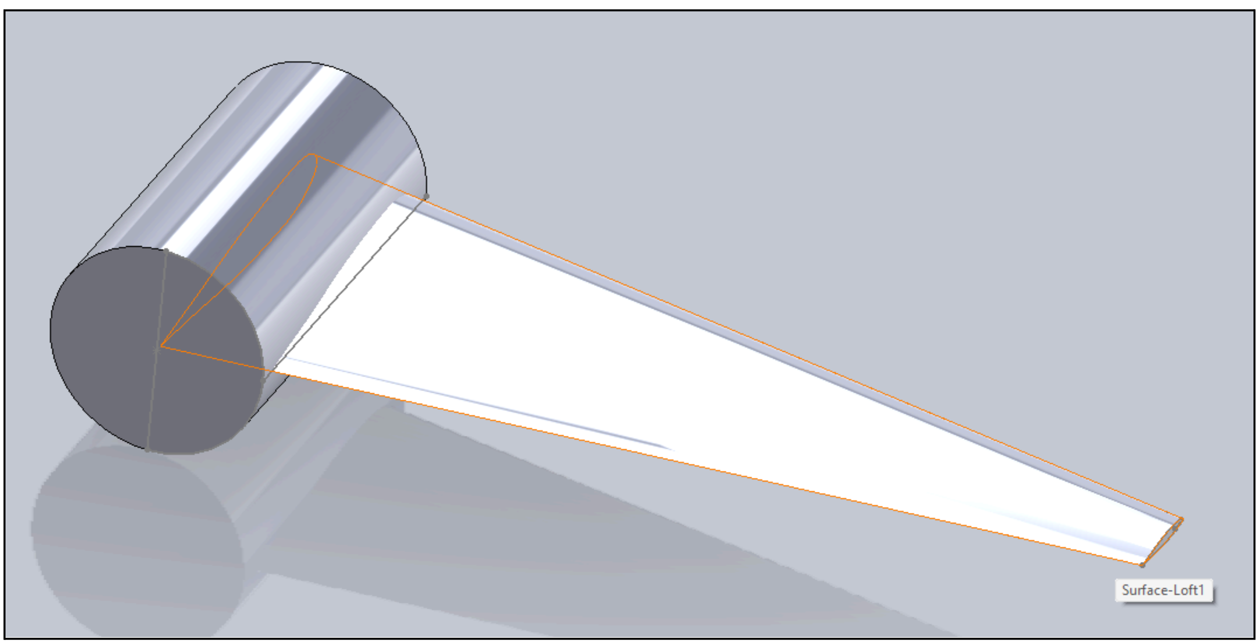
Root Chord



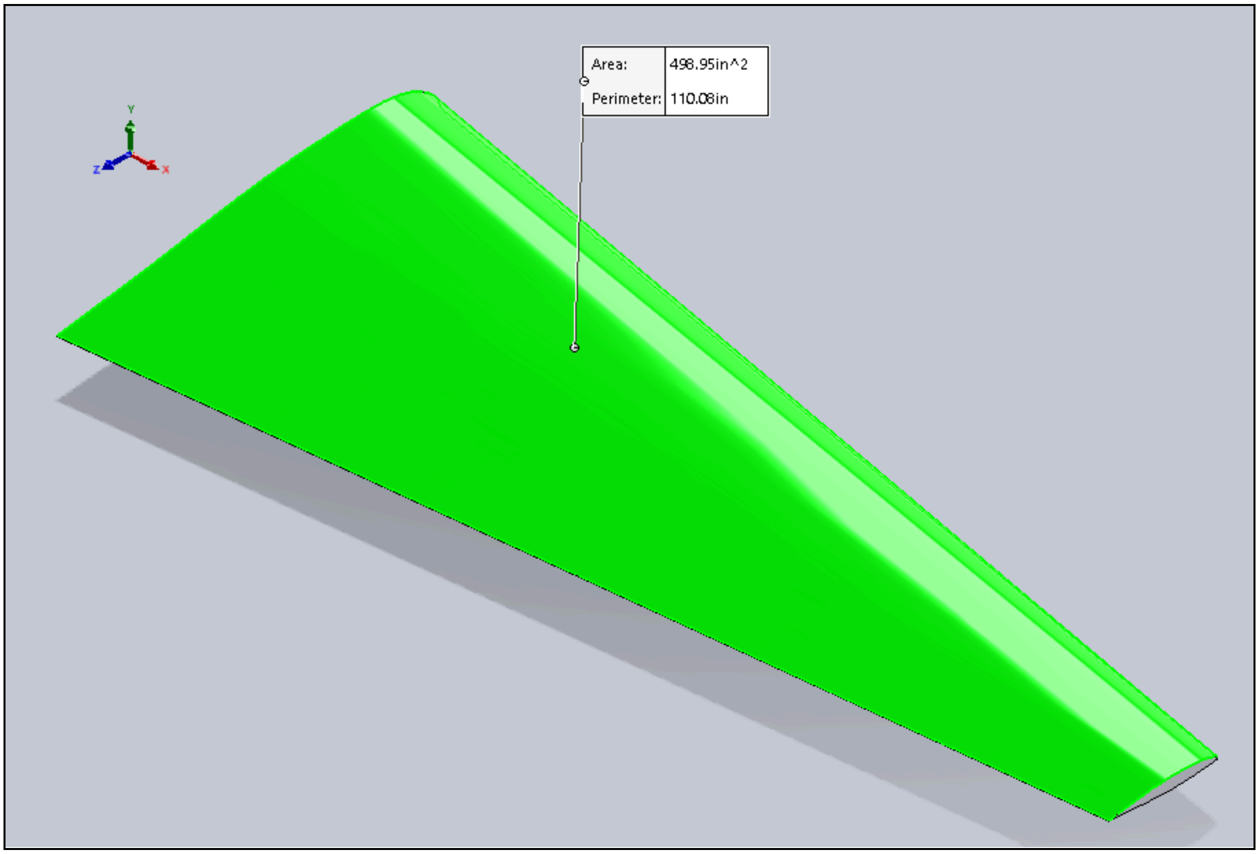
Tip Chord

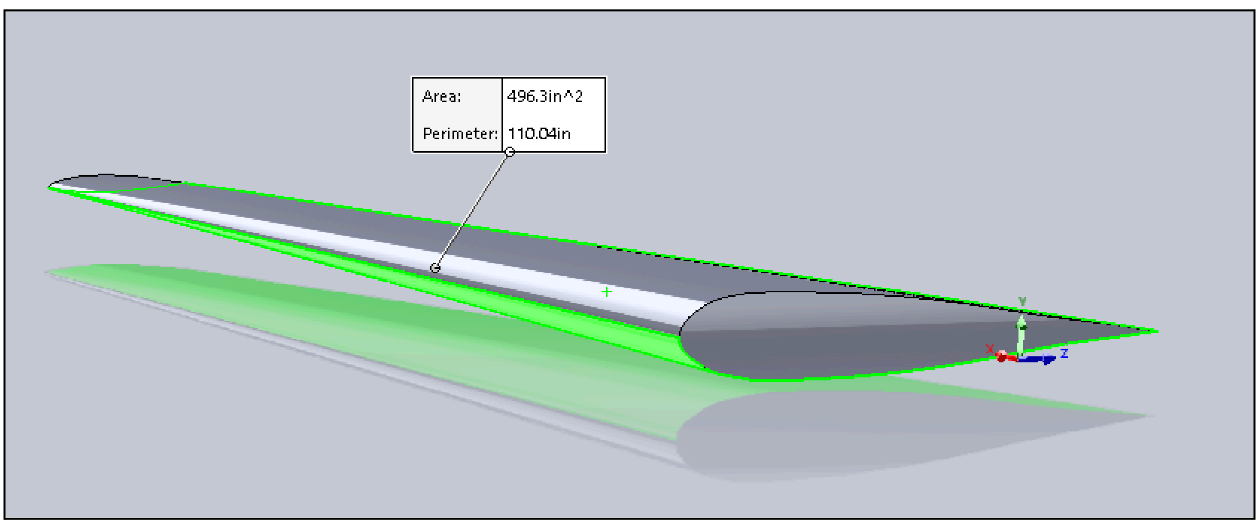


Wing and fuselage together



Wing that is exposed to air only (obtained by subtract fuselage from the full wing)


Area: 498.95in^2
Perimeter: 110.08in


Area: 496.3in^2
Perimeter: 110.04in