# Project (#1): Aircraft Performance and Airloading

**Due Date: Upload zip folder to TED by 11:58 PM, Friday March 13, 2020**

**<u>Files in your MATLAB Folder:</u>**

| | | | |
|---|---|---|---|
| **Download from TED:** | **SE160A_1_AirLoads_Input.xlsx** | | |
| **Download from TED:** | **SE160A_1_AirLoads_Output.xlsx** | | |
| **Download from TED:** | **SE160A_1_AirLoads.p** | | |
| **Download from TED:** | **createFigure.m** | | |
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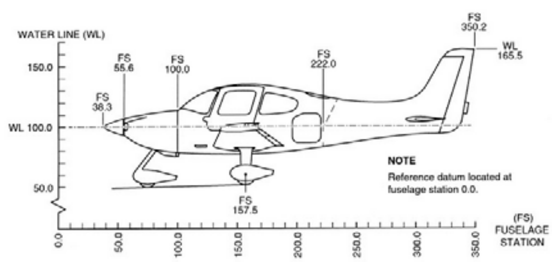
| | **For Graduate Students:** | **[ SE260A_1_LastName_FirstName.m ]** | |

**<u>Problem Answers are saved in a (pdf):</u>**
**For Undergraduate Students:** **[ SE160A_1_LastName_FirstName.pdf ]**
**For Graduate Students:** **[ SE260A_1_LastName_FirstName.pdf ]**

**<u>Upload your (m) file and (pdf) file into a (zip) folder of the same name:</u>**
**For Undergraduate Students:** **[ SE160A_1_LastName_FirstName.zip ]**
**For Graduate Students:** **[ SE260A_1_LastName_FirstName.zip ]**

In order to perform a structural analysis on an aircraft wing one first needs to calculate the aircraft center of gravity (CG) location, then determine the V-n diagram, followed by the calculation of the resultant wing lift (L), drag (D), and moment (M). Finally, these resultant loads are used to determine the distributed lift (l), drag (d), and moment along the wing span. These distributed loads are determined along the aircraft wing quarter-chord (c/4) and presented in both the aerodynamic coordinated frame and the structural coordinate frame.




## Part (1): Aircraft Weight and Center of Gravity

The aircraft weight ($W$) and longitudinal center of gravity ($X_{CG}$) is calculated using the component weights and station location. These components include (1) empty aircraft including all fluids (hydraulics, oil) except fuel, (2) fuel weight (3) Pilot, co-pilot, and passenger weights, and (4) luggage weights. We wish to examine eight different aircraft configurations, where we calculate the total aircraft weight and CG location for each case. These eight cases are plotted with ($X_{CG}$) on the $x$-axis and ($W$) on the $y$-axis.

$$W = \sum_{i=1}^{n} W_i \qquad x_c = \frac{1}{W}\left(\sum_{i=1}^{n} W_i x_i\right)$$

| | **Aircraft Configuration** | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| **Description** | **1** | **2** | **3** | **4** | **5** | **6** | **7** | **8** |
| **Aircraft (empty + oil, etc)** | **X** | **X** | **X** | **X** | **X** | **X** | **X** | **X** |
| **Fuel (40 gallons max)** | | | | | **X** | **X** | **X** | **X** |
| **Pilot** | **X** | **X** | **X** | **X** | **X** | **X** | **X** | **X** |
| **Co-Pilot (front passenger)** | | | **X** | **X** | | | **X** | **X** |
| **Rear passenger** | | | | **X** | | | | **X** |
| **luggage (in cockpit)** | | **X** | **X** | **X** | | **X** | **X** | **X** |
| **luggage (in fuselage)** | | | **X** | **X** | | | **X** | **X** |

## Part (2): V-n Diagram and Gust Diamond

Maneuvering events can put large loads on an aircraft, launch vehicle, and satellite structure. These maneuvers can be classified as "$n$" (or more commonly called "$g$") loading, where every molecule of mass is effectively scaled by a factor of "$n$" during the maneuver. Typical maneuvers include pull-down, pull-up, push-down, and steady level turn. Regulatory agencies and airplane companies use the V-n diagram to set the maximum load factors for different classes of aircraft traveling at different speeds. These diagrams are developed assuming the aircraft is traveling in still air, where a gust diagram can be added to the V-n diagram to account for severe updrafts and downdrafts. This gust diagram is overlaid onto the standard V-n diagram assuming that the pilot is flying the aircraft at ($n$=1) when the gust is encountered. In this section, you will use the maximum aircraft weight from section (A) to calculate the critical speeds and load factors at sea level and flight altitude. The air density at flight altitude is determined using standard air modeling equations (see Appendix 1). Finally, the V-n Diagram and overlaid gust diamond are plotted using sea level aircraft speeds.

**Part (3): Normal Flight (*n=1*) Loads at Altitude**

Using the aircraft geometric and aerodynamic definition, the wing loads and required thrust (T) and horsepower (hp) can easily be determined for four different normal (n=1) flight conditions (stall, stall with flaps, cruise, and dive. Begin by using the air density at altitude to calculate the air loads at altitude. Then calculate the required horsepower (at altitude) using the required thrust (equal to total aircraft drag) and aircraft velocity. The equation is presented in Appendix 2, where the propeller efficiency is used to account propeller losses. The required horsepower at sea level is also calculated where it is recognized internal combustion engines are less efficient as the air density decreases. Thus one needs a more powerful engine at sea level to account for losses at higher altitudes. Finally, the aircraft angle of attack is determined using the lift coefficient and lift curve slope, where these forces can be transformed from the aerodynamic reference frame to the structural reference frame.

**Part (4): Maneuvering Sea Level Loads at Critical V-n Corners and Gust Diamond**

Repeat part (3) for the four corners of the V-n diagram and the gust diamond at a sea level altitude.

**Part (5): Distributed Loads at PHAA**

The distributed lift (l), drag (d), and moment (m) loads are determined at sea level for one specific point on the V-n diagram (PHAA). This distribution is developed using the reader procedures. The results will be presented in six plots over the entire wing span (-b/2 < x < b/2). These six plots include (1) aircraft weight distribution (wing areal weight multiplied by wing chord), (2) wing distributed lift along the quarter-chord using Shrenk's approximation, (3) wing distributed drag acting along the quarter chord with a varying drag function, (4) wing distributed moment, (5) wing distributed loads acting at the quarter-chord pointing in the structural normal direction, and (6) wing distributed loads acting at the quarter-chord pointing in the structural chordwise direction.

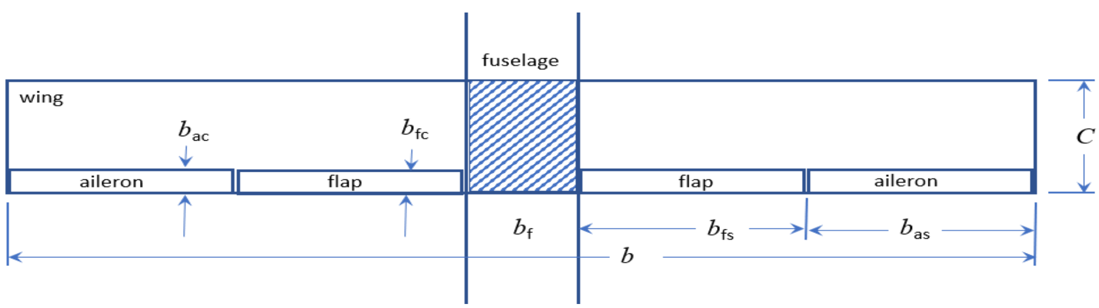



**Figure: Geometry definition of main wing. Defined as (%) of span or chord.**

### *INPUT (See Excel Input File for format restrictions)*

| | |
|---|---|
| Title | Title of project |
| $W, x$ | The weight ($W$) and fuselage station locations ($x$) for the complete aircraft and added weights (crew, fuel, and payload). This is typical information that a pilot must collect and calculate before each flight |
| $S$, $dC_L/d\alpha$ | Aircraft Definition |
| $V$, $n$, $K_g$ | Aircraft Performance Definition |
| $h, T$ | Altitude and Air Temperature (used to calculate air density at altitude) |
| * | *Aircraft geometry and aerodynamic definition* |

### *OUTPUT (See Excel Input File for format restrictions)*

| | |
|---|---|
| Student Name, ID | Name of Student and Student UCSD ID number |
| Title | Title of project |
| Input echo | Echo all input data |
| $W_i$, $x_i$ | Weight ($W$) and the center of gravity location ($x$) of 8 different aircraft configurations |
| $V$, $n$ | Flight speeds at sea level and flight density and load factor ($n$) for the critical locations on the V-n Diagram |
| * | *Aircraft loads, required thrust, and horsepower for normal flight loads (n=1) at sea level, normal loads (n-1) at altitude, and the V-n diagram and gust diamond at sea level.* |

Plots:
(1) Weight versus center of gravity envelope
(2) V-n Diagram and overlaid Gust Diamond
(3) Wing weight distribution versus wing span (-b/2 < x < b/2)
(4) Wing lift distribution at (c/4) versus wing span (-b/2 < x < b/2)
(5) Wing drag distribution at (c/4) versus wing span (-b/2 < x < b/2)
(6) Wing moment distribution versus wing span (-b/2 < x < b/2)
(7) Wing load distribution normal to wing chord at (c/4) versus wing span (-b/2 < x < b/2)
(8) Wing load distribution in the chord direction at (c/4) versus wing span (-b/2 < x < b/2)

## *ANALYTICAL STUDIES*

***Use your MATLAB code to answer question and submit using pdf file. Use the provided experimental Cessna 172x aircraft input file for all questions.***

1) For the provided aircraft input file (Experimental Cessna 172x), calculate the aircraft weight and cg location for the 8 different flight conditions. Plot and compare to the Cessna limits. Which configurations (if any) fall outside the acceptance envelope?

2) Plot the *V-n* Diagram and Gust Diagram for the Cessna 172x configuration.

3) Consider a fully fueled aircraft with a 170 lb pilot and along with 50 lbs of cockpit luggage in the cockpit. Calculate the required horsepower and normal wing resultant for the following flight conditions:

- Maximum structural cruise speed (n=1) at sea level (h = 0 ft) and (T = 70 $^{o}$F)
- Maximum structural cruise speed (n=1) at sea level (h = 5000 ft) and (T = 60 $^{o}$F)
- Maximum structural cruise speed (n=1) at sea level (h = 10,000 ft) and (T = 40 $^{o}$F)
- Maximum structural cruise speed (n=1) at sea level (h = 20,000 ft) and (T = 0 $^{o}$F)

## Appendix 1: Atmospheric Properties

There are a wide variety of atmospheric models for predicting the density of air. These models can depend upon altitude, temperature, dew point, and other variables. Most models are very complex. In this class we will use the perfect gas law to calculate the air density as a function of altitude ($h$ $ft$) and temperature ($T_a$ $^oF$) only. This model has been shown to be very accurate for altitudes up to 36,000 feet.

The standard temperature ($T_{as}$ $^oR$) and pressure ($p_{as}$ $ft/sec^2$) at altitude ($h$) are given by:

$$T_{as}\ (^oR) = T_o\ (^oR) - \left(0.00356616\ \left(\frac{^oR}{ft}\right)\right)\left(h\left(ft\right)\right)$$

$$p_{as}\ (\mathrm{lb/ft^2}) = 2116.22\left(\frac{T_{as}\ (^oR)}{T_o\ (^oR)}\right)^{5.255912}$$

The resulting standard air density ($\rho_a$) and speed of sound ($M_a$) at altitude are found to be:

$$\rho_a = \frac{p_{as}}{RT_a} \quad \rightarrow \quad \rho_a\left(\frac{lb-\sec^2}{ft^4}\right) = \frac{\left(p_{as}\ \mathrm{lb/ft^2}\right)}{\left(1716.5488\,\frac{ft^2}{\sec^2 -{}^o R}\right)\left(T_a(^oF)+459.67\right)}$$

$$M_a = \sqrt{1.4RT_a} \quad \rightarrow \quad M_a\ \left(ft/\sec\right) = \sqrt{1.4\left(1716.5488\,\frac{ft^2}{\sec^2 -{}^o R}\right)\left(T_a(^oF)+459.67\right)}$$

Standard Properties of Air at Sea Level:

| | |
|---|---|
| Temperature ($T_o$): | 59 $^oF$ = 518.67 $^oR$ |
| Pressure ($p_o$): | 14.696 $lb/in^2$ = 2116.224 $lb/ft^2$ |
| Density ($\rho_o$): | 0.0764746 $lb/ft^3$ = 0.00237691 $lb\text{-}sec^2/ft^4$ |
| Speed of Sound, Mach = 1 ($M$) | 1116.45 $ft/sec$ = 761.21 $mph$ |

Constants:

| | |
|---|---|
| Temperature (Rankine): | $T$ ($^oF$) + 459.67 |
| Gravity ($g$): | 32.174 $ft/sec^2$ |
| Molar Mass of Air: | 28.9644 $lb/lb_{mole}$ |
| Specific Gas Constant ($R$ ): | 1545.31 $lb\text{-}ft/lb_{mole}\text{-}^oR$ = 1716.5488 $ft^2/sec^2\text{-}^oR$ |

Example: Calculate the air density ($\rho_a$) and speed of sound ($M_a$) at an altitude of ($h$ = 10,000 feet) and temperature of ($T_a$ = 30 $^oF$). Repeat for a warm day ($T_a$ = 80 $^oF$).

Calculate the standard temperature ($T_{as}$) and pressure ($p_{as}$) at ($h$ = 10,000 ft):

$$T_{as} = 483.01\ \ ^oR \qquad p_{as} = 1455.33\ \ lb/ft^2$$

Calculate the air density ($\rho_a$) and mach speed ($M_a$) at the two temperatures:

$$T_a = 30^oF \qquad \rho_a = 0.001731\ \ lb-\sec^2/ft^4 \qquad M_a = 1084.79\ \ \mathrm{ft/sec}$$

$$T_a = 80^oF \qquad \rho_a = 0.001571\ \ lb-\sec^2/ft^4 \qquad M_a = 1138.82\ \ \mathrm{ft/sec}$$

## Appendix 2: Other Required Equations

True Air Speed: Actual aircraft speed at altitude: [ $V_{altitude} = V_{SeaLevel}\sqrt{\dfrac{\rho_{SeaLevel}}{\rho_{altitude}}}$ ]

Required horsepower at altitude (1 *hp* = 550 *lbf-foot/sec)* [ $hp = \dfrac{T\left(V_{altitude}\right)}{\left(\eta/100\right)}$ ]

For normally aspirated engines (non turbocharged, non supercharged) there is a horsepower drop with increasing altitude due to the reduction in air density. Thus, you need a larger horsepower at sea level.

$$[ \quad hp_{SeaLevel} = hp_{altitude}\left(\left(\frac{\rho_{altitude}}{\rho_{SeaLevel}}\right) - \frac{1-\left(\dfrac{\rho_{altitude}}{\rho_{SeaLevel}}\right)}{7.55}\right)^{-1} \quad ]$$

## APPENDIX 3: Normalization of distributed load functions

As you have seen, we can describe the variation of aircraft loads on a wing by applying a distributed load function, such as (for drag) to find the load per unit span-length as a function of $x$:

$$d(x) = qC_D c_o\left(1-\frac{2x}{b}(1-\lambda)\right)f(x)$$

where $q$ is the dynamic pressure $0.5\rho v^2$, $C_D$ is the drag coefficient, $c_o$ is the chord length at the wing root, $b$ is the wingspan (wingtip to wingtip), $\lambda$ is the taper ratio, and $f(x)$ represents the distribution in drag for a uniform chord length, and is given by (for example):

$$f(x) = d_o + \sum_{i=1}^{\infty} d_i\left(\frac{2x}{b}\right)^i$$

The function

$$c_o\left(1-\frac{2x}{b}(1-\lambda)\right)$$

represents the span-wise variation in the chord length.

When we integrate the load function $d(x)$ over the full length of the wingspan, we should recover the total drag on the wing:

$$2\int_0^{b/2} d(x)\,dx = D_{wings}$$

So, if you are given a known total wing drag $D_{wings}$ and a load function such as:

$$f(x) = d_o + d_{10}\left(\frac{2x}{b}\right)^{10}$$

then in order to find the distributed drag function *d(x)*, you must integrate the known functions over the domain:

$$\begin{aligned}
D_{wings} &= 2qC_D c_o \int_0^{b/2} \left(1-\frac{2x}{b}(1-\lambda)\right)\left(d_o + d_{10}\left(\frac{2x}{b}\right)^{10}\right)dx \\
&= 2qC_D c_o \int_0^{b/2} \left[d_o + d_{10}\left(\frac{2x}{b}\right)^{10} - d_o(1-\lambda)\frac{2x}{b} - d_{10}(1-\lambda)\left(\frac{2x}{b}\right)^{11}\right]dx \\
&= 2qC_D c_o \left[d_o x + \frac{d_{10}2^{10}}{b^{10}}\frac{x^{11}}{11} - d_o(1-\lambda)\frac{x^2}{b} - \frac{d_{10}(1-\lambda)2^{11}}{b^{11}}\frac{x^{12}}{12}\right]_0^{b/2} \\
&= 2qC_D c_o \left[\left(\frac{d_o b}{2}\right) + \left(\frac{d_{10} b}{22}\right) - \left(d_o(1-\lambda)\frac{b}{4}\right) - \left(\frac{d_{10}(1-\lambda)b}{24}\right)\right] \\
&= 2qC_D c_o \left[d_o\left(\frac{b}{4}\right)(1+\lambda) + d_{10}\left(\frac{b}{264}\right)(1+11\lambda)\right] \\
&= qC_D c_o b\left[\left(\frac{d_o}{2}\right)(1+\lambda) + \left(\frac{d_{10}}{132}\right)(1+11\lambda)\right]
\end{aligned}$$

$$\boxed{D_{wings} = \frac{qC_D c_o b}{132}\left[66 d_o(1+\lambda) + d_{10}(1+11\lambda)\right]}$$

Solving for $qC_D c_o$, we get

$$qC_D c_o = \frac{132 D_{wings}}{b\left[66 d_o(1+\lambda) + d_{10}(1+11\lambda)\right]}$$

Then plugging this (and our function *f(x)*) back to our first equation, we get our final distributed drag function expressed in terms of the total drag, the wing span, the wing taper, and the drag function:

$$d(x) = \frac{132 D_{wings}}{b\left[66 d_o(1+\lambda) + d_{10}(1+11\lambda)\right]}\left(1-\frac{2x}{b}(1-\lambda)\right)\left(d_o + d_{10}\left(\frac{2x}{b}\right)^{10}\right)$$

# *MATLAB Project (#1) - Aircraft Performance and AirLoads*

***SE-160A Aerospace Structural Mechanics, University of California, San Diego (Copyright J.B. Kosmatka, 2020)***

**Version:** Winter, 2020 (v1)

**Project Title:**

| Variable | Description | Value | Units |
|---|---|---|---|
| *iInput* | Input Units | 1 | 1 = US, 2 = SI |
| *iOutput* | Output Units | 1 | 1 = US, 2 = SI |

| Units Reference | US | SI |
|---|---|---|
| *Power* | *Hp* | *Watts* |
| *Speed (V)* | *ft/sec* | *m/sec* |
| *Mass* | *lb-sec$^2$/inch* | *kg* |
| *Force* | *lb* | *N* |
| *Length* | *inch, feet* | *m* |
| *Temperature* | *oF* | *oC* |

X **Part 1: Aircraft Aerodynamic Definition**

| Variable | Description | Value | Units |
|---|---|---|---|
| $dC_L/da$ | Lift Curve Slope | 0.0836 | 1/degree |
| $a_o$ | Zero Lift Angle | -3 | degree |
| $C_{L-max}$ (+) | Max Lift at Stall (+) | 1.338 | 1 |
| $a_{stall}$ (+) | Stall Angle (+) | 13 | degree |
| $C_{L-max}$ (-) | Max Lift at Stall (-) | -1.338 | 1 |
| $a_{stall}$ (-) | Stall Angle (-) | -19 | degree |
| $C_{mo}$ | Wing Moment Coefficient | -0.045 | 1 |
| $C_{DoA}$ | Parasitic Drag - Fuselage | 0.024 | 1 |
| $C_{DoW}$ | Parasitic Drag - Wing | 0.008 | 1 |
| $C_{DiW}$ | Induced Drag Coefficient | 0.0358 | 1 |
| $d_0$ | Spanwise drag amplification | 1 | 1 |
| $d_{10}$ | Spanwise drag amplification | 0.2 | 1 |
| Hp | Maximum Motor Power | 200 | Hp or Watts |
| $\eta$ | Propeller Efficiency | 85 | % |

X **Part 2: Aircraft Performance at Standard Sea Level**

Note: Based upon maximum aircraft weight and calculated wing area (see below)

**Flaps - Stowed**

| Variable | Description | Value | Units |
|---|---|---|---|
| $V_{s1}$ | Stall Speed | 62 | ft/sec or m/sec |
| $V_{NO}$ | Max Structural Cruise Speed | 150 | ft/sec or m/sec |
| $V_{NE}$ | Dive Speed - Never Exceed | 210 | ft/sec or m/sec |
| $n_p$ | Max Positive Load Factor | 3.8 | 1 |
| $n_n$ | Max Negative Load Factor | -1.52 | 1 |
| $(0 < V < V_C)$ | Gust Updraft | 50 | ft/sec or m/sec |
| $(V_C < V < V_D)$ | Gust Updraft | 25 | ft/sec or m/sec |
| $(0 < V < V_C)$ | Gust Downdraft | -50 | ft/sec or m/sec |
| $(V_C < V < V_D)$ | Gust Downdraft | -25 | ft/sec or m/sec |
| $K_g$ | Gust Alleviation Factor | 1 | 1 |

**Flaps - Extended (60 degrees)**

| Variable | Description | Value | Units |
|---|---|---|---|
| $V_{s0}$ | Stall Speed | 56 | ft/sec or m/sec |
| $V_{FE}$ | Maximum Speed | 100 | ft/sec or m/sec |
| $n_{pf}$ | Positive Load Factor | 2 | 1 |
| $n_{nf}$ | Negative Load Factor | -1.3 | 1 |

X **Part 3: Aircraft Wing and Tail Geometry Definition**

Main Wing Definition

| Variable | Description | Value | Units |
|---|---|---|---|
| $W_w$ | Wing Weight (total) | 250 | lb (or N) |
| $b$ | Span (tip to tip) | 36 | inch or m |
| $b_f$ | Fuselage Width (% of span) | 10 | % |
| $b_{as}$ | Aileron Span Length (% of span) | 45 | % |
| $b_{ac}$ | Aileron Chord Length (% of chord) | 20 | % |
| $b_{fs}$ | Flap Span Length (% of span) | 45 | % |
| $b_{fc}$ | Flap Chord Length (% of chord) | 20 | % |
| $C_R$ | Chord - Root | 4.83 | inch or m |
| $C_T$ | Chord - Tip | 4.83 | inch or m |
| $\alpha_R$ | Incidence Angle - Root | 4.83 | degrees |
| $\alpha_R$ | Incidence Angle - Tip | 4.83 | degrees |
| $\Delta X_{w4}$ | Wing (C/4) Station | 46 | inch or m |

Horizontal Stabilizer Definition (Assume NACA-0012 airfoil)

| Variable | Description | Value | Units |
|---|---|---|---|
| $b_h$ | Span | 36 | inch or m |
| $C_{rh}$ | Chord - Root | 4.83 | inch or m |
| $C_{th}$ | Chord - Tip | 4.83 | inch or m |
| $\Delta X_{h4}$ | Horizontal Stabilizer (C/4) Station | 46 | inch or m |

Vertical Stabilizer Definition (Assume NACA-0012 airfoil)

| Variable | Description | Value | Units |
|---|---|---|---|
| $b_v$ | Span | 36 | inch or m |
| $C_{rv}$ | Chord - Root | 4.83 | inch or m |
| $C_{tv}$ | Chord - Tip | 4.83 | inch or m |

| | | | |
|---|---|---|---|
| $\Delta Xv_4$ | Verical Stabilizer (C/4) Station | 46 | inch or m |

X **Part 4: Aircraft Weight and Balance**

| Station (inch) | Description | Weight (lb) | Aircraft Configuration 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 |
|---|---|---|---|---|---|---|---|---|---|---|
| 39.61 | Aircraft (empty + oil, etc) | 1454 | X | X | X | X | X | X | X | X |
| 48 | Fuel (40 gallons max) | 240 | | | | | X | X | X | X |
| 37 | Pilot | 170 | X | X | X | X | X | X | X | X |
| 37 | Co-Pilot (front passenger) | 170 | | | X | X | | | X | X |
| 73 | Rear passenger | 170 | | | | X | | | | X |
| 95 | luggage (in cockpit) | 96 | | X | X | X | | X | X | X |
| 123 | luggage (in fuselage) | 0 | | | X | X | | | X | X |

X **Part 5: Aircraft Local Environment**

| Variable | Description | Value | Units |
|---|---|---|---|
| h | Altimeter - Altitude (*h*) above SL | 10000 | feet or m |
| T | Temperature | 30 | °F or °C |

**END OF FILE**

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***SE-160A Aerospace Structural Analysis, University of California, San Diego (Copyright J.B. Kosmatka, 2020)***

**Version:** Winter, 2020 (v1)

**Student Name:**

**Student ID:**

**Project Title:**

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Note: Based upon maximum aircraft weight and calculated wing area (see below)

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| $n_{nf}$ | Negative Load Factor | -1.3 | 1 |

**X** **Part 3: Aircraft Wing and Tail Geometry Definition**

Main Wing Definition

| Variable | Description | Value | Units |
|---|---|---|---|
| $W_w$ | Wing Weight (total) | 250 | lb |
| $b$ | Span (tip to tip) | 36 | inch |
| $b_{fs}$ | Fuselage Width (% of span) | 20 | % |
| $b_{as}$ | Aileron Span Length (% of span) | 40 | % |
| $b_{ac}$ | Aileron Chord Length (% of chord) | 20 | % |
| $b_{fs}$ | Flap Span Length (% of span) | 40 | % |
| $b_{fc}$ | Flap Chord Length (% of chord) | 20 | % |
| $C_R$ | Chord - Root | 4.83 | inch |
| $C_T$ | Chord - Tip | 4.83 | inch |
| $\alpha_R$ | Incidence Angle - Root | 4.83 | degrees |
| $\alpha_R$ | Incidence Angle - Tip | 4.83 | degrees |
| $\Delta X_{w4}$ | Wing (C/4) Station | 46 | inch |

Horizontal Stabilizer Definition (Assume NACA-0012 airfoil

| Variable | Description | Value | Units |
|---|---|---|---|
| $b_h$ | Span | 80 | inch |
| $C_{rh}$ | Chord - Root | 50 | inch |
| $C_{th}$ | Chord - Tip | 50 | inch |
| $\Delta X_{h4}$ | Horizontal Stabilizer (C/4) Station | 120 | inch |

Vertical Stabilizer Definition (Assume NACA-0012 airfoil

| Variable | Description | Value | Units |
|---|---|---|---|
| $b_v$ | Span | 50 | inch |
| $C_{rv}$ | Chord - Root | 40 | inch |
| $C_{tv}$ | Chord - Tip | 40 | inch |
| $\Delta Xv_4$ | Verical Stabilizer (C/4) Station | 120 | inch |

X **Part 4: Aircraft Weight and Balance**

| | | | Aircraft Configuration | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| Station (inch) | Description | Weight (lb) | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 |
| 39.61 | Aircraft (empty + oil, etc) | 1454 | X | X | X | X | X | X | X | X |
| 48 | Fuel (40 gallons max) | 240 | | | | | X | X | X | X |
| 37 | Pilot | 170 | X | X | X | X | X | X | X | X |
| 37 | Co-Pilot (front passenger) | 170 | | | X | X | | | X | X |
| 73 | Rear passenger | 170 | | | | X | | | | X |
| 95 | luggage (in cockpit) | 96 | | X | X | X | | X | X | X |
| 123 | luggage (in fuselage) | 0 | | | X | X | | | X | X |

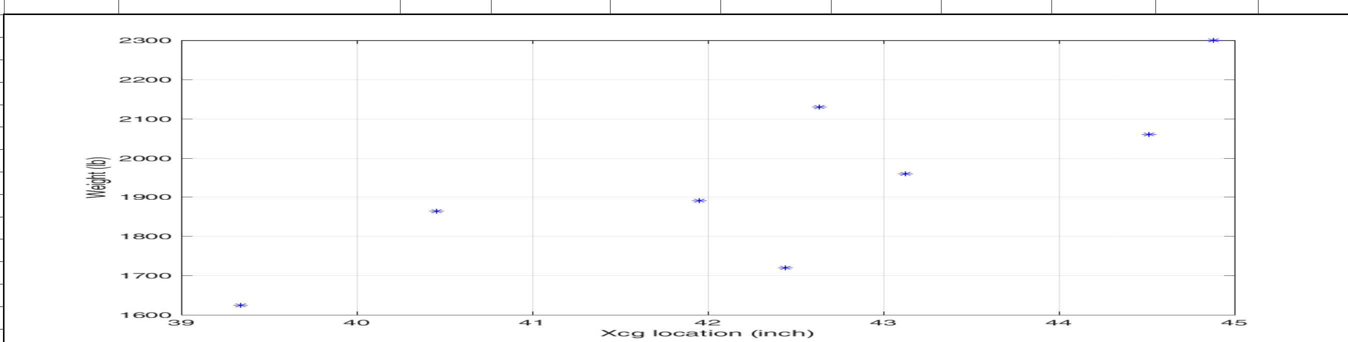
X **Part 5: Aircraft Local Environment**

| Variable | Description | Value | Units |
|---|---|---|---|
| h | Altimeter - Altitude (*h*) above SL | 10000 | feet |
| T | Temperature | 30 | °F |

**OUTPUT:**

**PART 1: Aircraft Weight and Center of Gravity Distribution**

| | | | Aircraft Configuration | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| Station (inch) | Description | Weight (lb) | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 |
| 39.61 | Aircraft (empty + oil, etc) | 1454 | X | X | X | X | X | X | X | X |
| 48 | Fuel (40 gallons max) | 240 | | | | | X | X | X | X |
| 37 | Pilot | 170 | X | X | X | X | X | X | X | X |
| 37 | Co-Pilot (front passenger) | 170 | | | X | X | | | X | X |
| 73 | Rear passenger | 170 | | | | X | | | | X |
| 95 | luggage (in cockpit) | 96 | | X | X | X | | X | X | X |
| 123 | luggage (in fuselage) | 0 | | | X | X | | | X | X |
| | | Aircraft Weight (lb) | 1624 | 1720 | 1890 | 2060 | 1864 | 1960 | 2130 | 2300 |
| | | Aircraft CG Location (inch) | 39.337 | 42.444 | 41.954 | 44.516 | 40.452 | 43.124 | 42.635 | 44.880 |

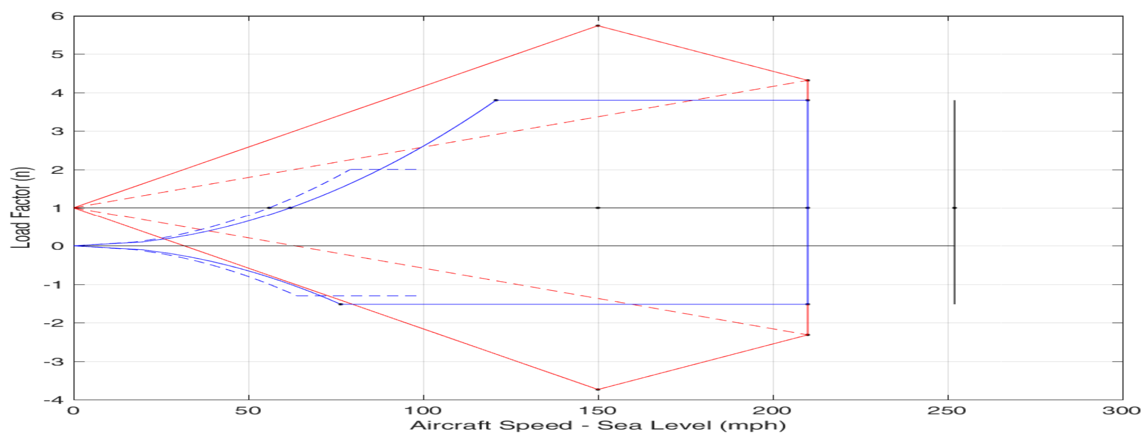



**PART 2: V-n Diagram and Gust Diamond at maximum weight**

| Variable | Description | Value | Units |
|---|---|---|---|
| | Air Density (at altitude) | 0.0017314 | lb-sec$^2$/ft$^4$ |

| | | at Sea Level | | at Altitude | |
|---|---|---|---|---|---|
| Variable | Description | V (mph) | n | V (mph) | n |
| | Stall – flaps down (n=1) | 56.00 | 1.00 | 65.61 | 1.00 |
| | Stall – flaps up (n=1) | | 1.00 | 72.64 | 1.00 |
| | Cruise (n = 1) | 150.00 | 1.00 | 175.75 | 1.00 |

| | | | | |
|---|---|---|---|---|
| Dive (n=1) | 210.00 | 1.00 | 246.05 | 1.00 |
| Flutter (n=1) | 252.00 | 1.00 | 295.26 | 1.00 |
| PHAA | 120.86 | 3.80 | 141.61 | 3.80 |
| PLAA | 210.00 | 3.80 | 246.05 | 3.80 |
| NHAA | 76.44 | -1.52 | 89.56 | -1.52 |
| NLAA | 210.00 | -1.52 | 246.05 | -1.52 |
| + Gust (cruise) | 150.00 | 5.74 | 175.75 | 5.04 |
| + Gust (dive) | 210.00 | 4.32 | 246.05 | 3.83 |
| - Gust (cruise) | 150.00 | -3.74 | 175.75 | -3.04 |
| - Gust (dive) | 210.00 | -2.32 | 246.05 | -1.83 |



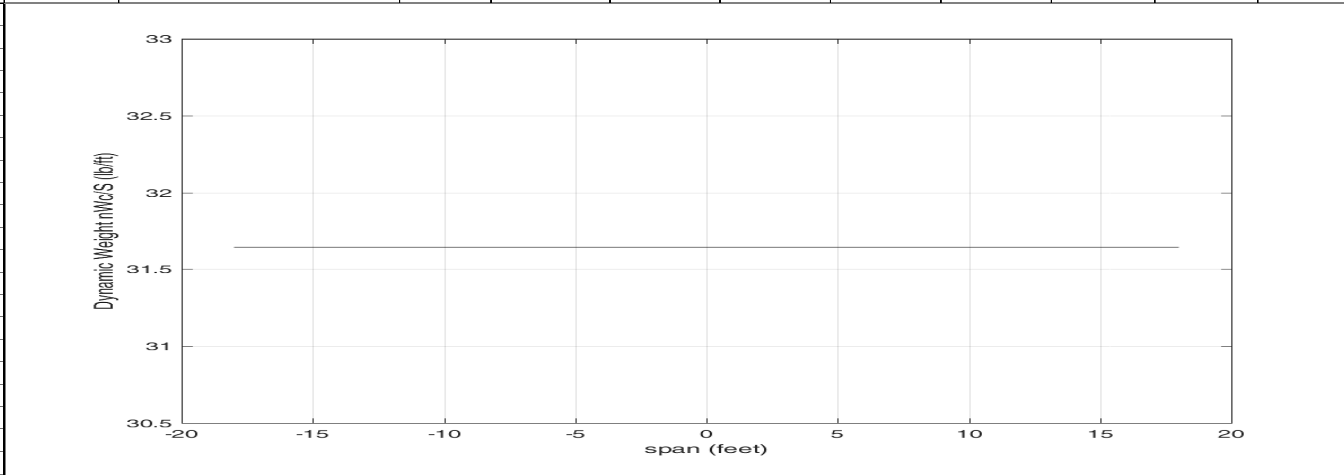

**PART 3:** **Maneuvering Loads at Normal Flight (n = 1)**

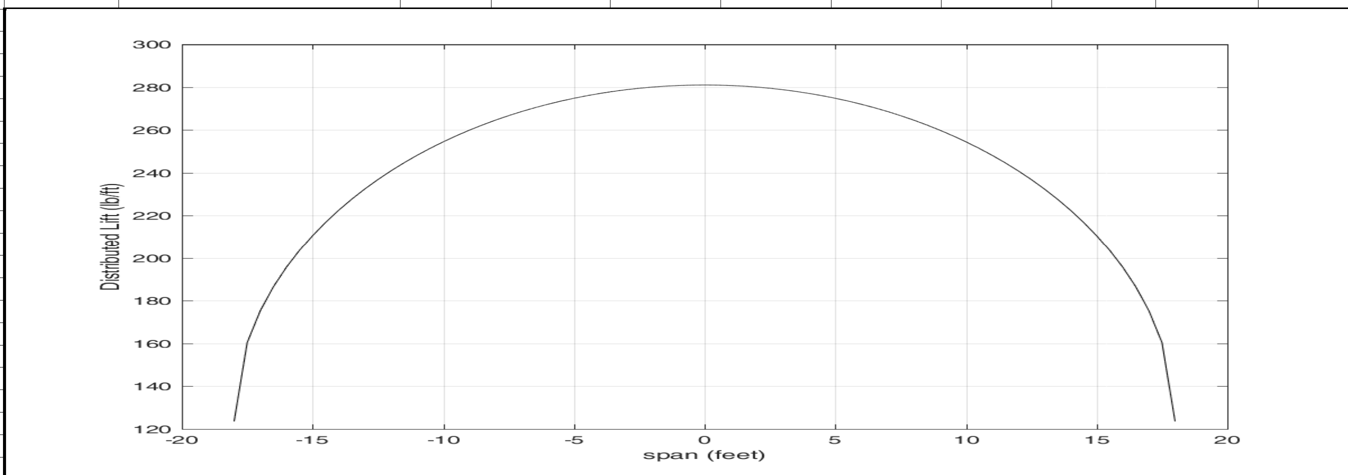
| Variable | Description | Stall, Flaps Extended | Stall, Flaps Stowed | Cruise | Dive | Units |
|---|---|---|---|---|---|---|
| $V$ | Air Speed (at altitude) | 65.61 | 72.64 | 175.75 | 246.05 | mph |
| $q$ | Dynamic Pressure (at altitude) | 8.02 | 9.83 | 57.52 | 112.74 | $lb/ft^2$ |
| $n$ | Load Factor | 1.00 | 1.00 | 1.00 | 1.00 | 1 |
| $M_A$ | Wing Moment | -303.20 | -371.65 | -2175.39 | -4263.76 | lb-ft |
| $L_W$ | Wing Lift | 2337.00 | 2341.89 | 2470.72 | 2619.89 | lb |
| $C_L$ | Lift Coefficient | 1.68 | 1.37 | 0.25 | 0.13 | 1 |
| $L_T$ | Tail Lift | -37.00 | -41.89 | -170.72 | -319.89 | lb |
| $D_W$ | Wing Drag | 151.32 | 128.50 | 101.90 | 169.46 | lb |
| $D_F$ | Fuselage Drag | 33.48 | 41.04 | 240.21 | 470.81 | lb |
| $T$ | Thrust | 184.80 | 169.54 | 342.11 | 640.27 | lb |
| $Power$ | Required Power at altitude | 38.04 | 38.64 | 188.63 | 494.24 | hp |
| $Power$ | Required Power at SL | 54.94 | 55.80 | 272.41 | 713.74 | hp |
| $\alpha$ | Wing Angle of Attack | 17.04 | 13.38 | -0.05 | -1.40 | degrees |
| $F_N$ | Wing Load Resultant (normal) | 1991.95 | 2016.21 | 2170.66 | 2315.08 | lb |
| $F_C$ | Wing Load Resultant (chordwise) | -452.22 | -347.59 | 103.69 | 226.19 | lb |

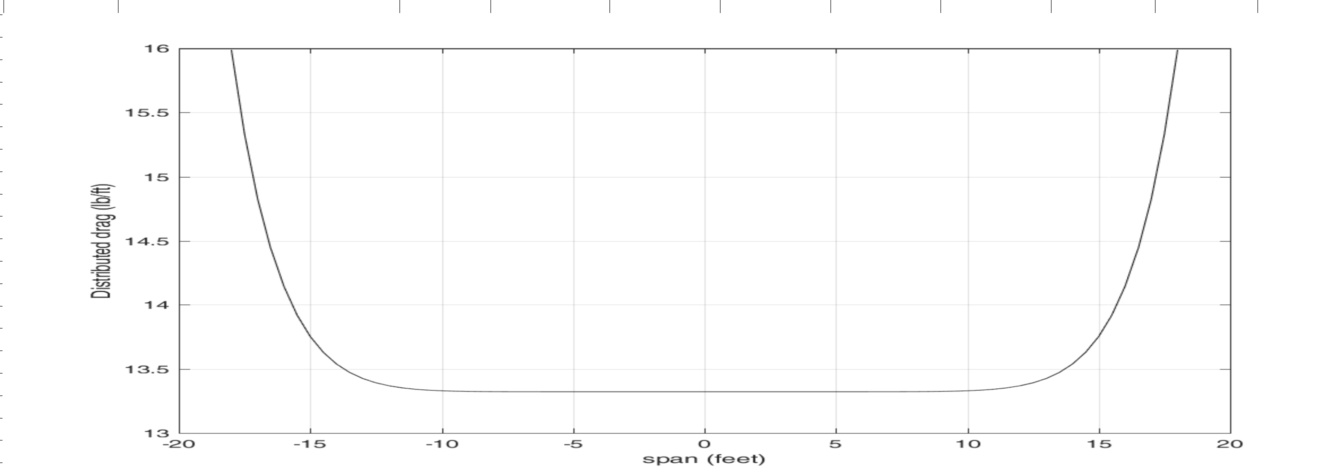
**PART 4:** **Maneuvering Loads at Critical V-n Diagram Corners and Gust Diamond Corners**

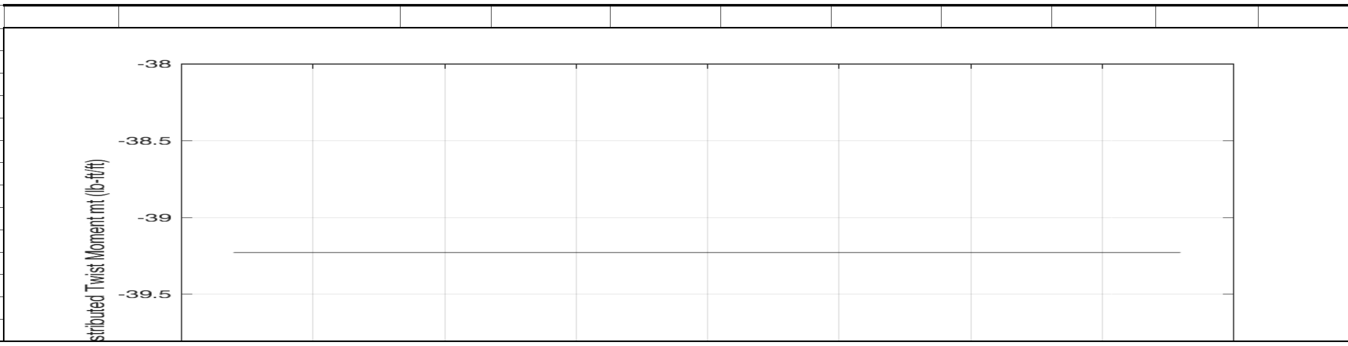
| Variable | Description | V-n Corner Points PHAA | PLAA | NHAA | NLAA | Gust Diamond +gust (cruise) | +gust (dive) | -gust (cruise) | -gust (dive) | Units |
|---|---|---|---|---|---|---|---|---|---|---|
| $V$ | Air Speed (at sea level) | 120.86 | 210.00 | 76.44 | 210.00 | 150.00 | 210.00 | 150.00 | 210.00 | mph |
| $q$ | Dynamic Pressure (at sea level) | 37.34 | 112.74 | 14.94 | 112.74 | 57.52 | 112.74 | 57.52 | 112.74 | $lb/ft^2$ |
| $n$ | Load Factor | 3.80 | 3.80 | -1.52 | -1.52 | 5.74 | 4.32 | -3.74 | -2.32 | 1 |
| $M_A$ | Wing Moment | -1412.28 | -4263.76 | -564.91 | -4263.76 | -2175.39 | -4263.76 | -2175.39 | -4263.76 | lb-ft |
| $L_W$ | Wing Lift | 8899.17 | 9102.85 | -3478.97 | -3214.76 | 13439.05 | 10297.72 | -8497.60 | -5057.94 | lb |
| $C_L$ | Lift Coefficient | 1.37 | 0.46 | -1.34 | -0.16 | 1.34 | 0.52 | -0.85 | -0.26 | 1 |
| $L_T$ | Tail Lift | -159.17 | -362.85 | -17.03 | -281.24 | -243.39 | -370.76 | -98.06 | -269.03 | lb |
| $D_W$ | Wing Drag | 488.32 | 308.15 | 187.50 | 175.80 | 726.09 | 350.46 | 338.35 | 203.62 | lb |
| $D_F$ | Fuselage Drag | 155.95 | 470.81 | 62.38 | 470.81 | 240.21 | 470.81 | 240.21 | 470.81 | lb |
| $T$ | Thrust | 644.26 | 778.96 | 249.88 | 646.61 | 966.29 | 821.27 | 578.56 | 674.43 | lb |
| $Power$ | Required Power at SL | 244.28 | 513.20 | 59.92 | 426.00 | 454.73 | 541.07 | 272.26 | 444.33 | hp |
| $\alpha$ | Wing Angle of Attack | 13.38 | 2.55 | -19.01 | -4.96 | 13.06 | 3.28 | -13.16 | -6.08 | degrees |
| $F_N$ | Wing Load Resultant (normal) | 7661.59 | 7968.76 | -2919.19 | -2763.67 | 11578.94 | 9008.31 | -7259.93 | -4360.18 | lb |
| $F_C$ | Wing Load Resultant (chordwise) | -1320.83 | -46.51 | -807.47 | -63.40 | -1940.93 | -165.09 | -1349.42 | -259.97 | lb |

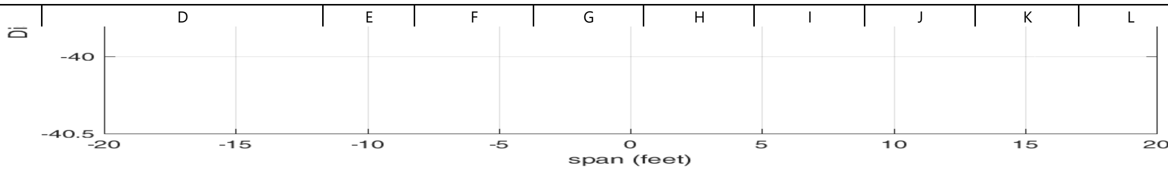
**PART 5:** **Distributed Loads at PHAA**

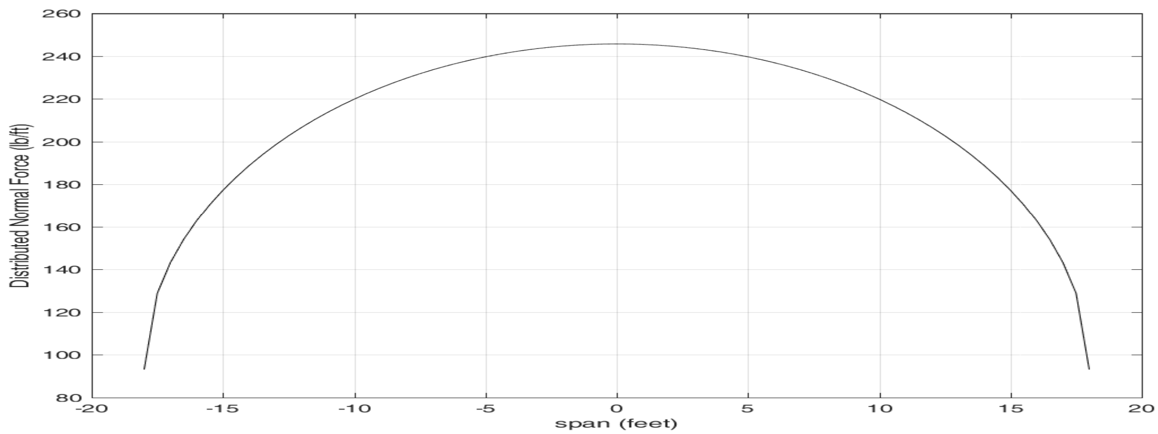

Dynamic Weight nWc/S (lb/ft)
33
32.5
32
31.5
31
30.5
-20
-15
-10
-5
0
5
10
15
20
span (feet)

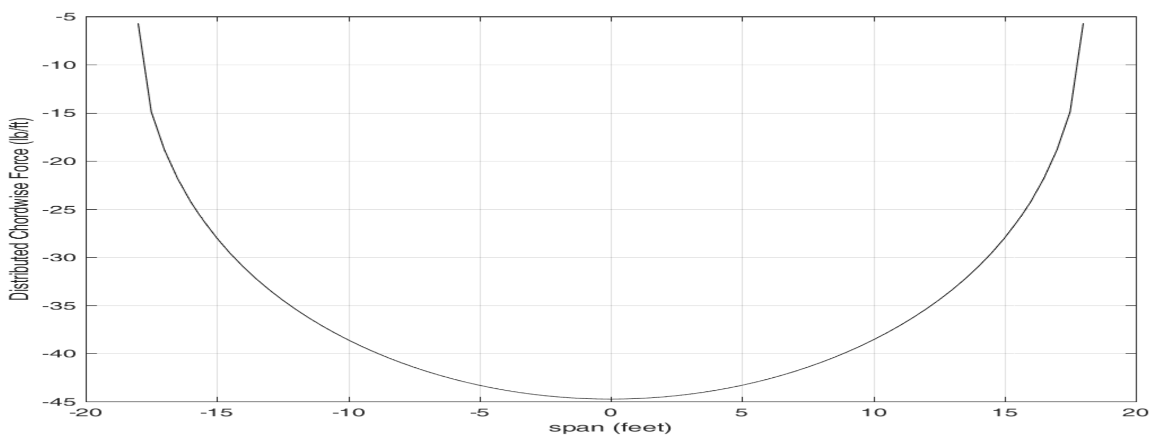

Distributed Lift (lb/ft)
300
280
260
240
220
200
180
160
140
120
-20
-15
-10
-5
0
5
10
15
20
span (feet)


Distributed drag (lb/ft)
16
15.5
15
14.5
14
13.5
13
-20
-15
-10
-5
0
5
10
15
20
span (feet)


stributed Twist Moment mt (lb-ft/ft)
-38
-38.5
-39
-39.5




**END OF OUTPUT**