| | |
|---|---|
| $m_g := .367\text{kg}$ | Mass of glider |
| $l_{wr} := \frac{23}{24}\cdot 14\text{in} = 0.341\,\text{m}$ | Length of wing chord at root |
| $l_{wt} := \frac{23}{24}\cdot 4\text{in} = 0.097\,\text{m}$ | Length of wing chord at tip |
| $l_{wlr} := l_{wt} = 0.097\,\text{m}$ | Length of winglet chord at root |
| $l_{wlt} := 1.167\text{in} = 0.03\,\text{m}$ | Length of winglet chord a tip |
| $b_w := 22.016\text{in} = 0.559\,\text{m}$ | Span of wing from root to tip |
| $b_{wl} := \frac{23}{24}4\text{in} = 0.097\,\text{m}$ | Span of winglet |
| $\delta := 5\text{mm} = 5 \times 10^{-3}\,\text{m}$ | thickness of wings |
| $A_w := 2\cdot 191.576\text{in}^2 = 0.247\,\text{m}^2$ | Area of both wings |
| $A_{ftop} := 16.357\text{in}\cdot 75\text{mm} = 0.031\,\text{m}^2$ | Cross sectional area of the fuselage from the top |
| $A_{ffront} := 75\text{mm}\cdot 2.629\text{in} = 50.082\cdot\text{cm}^2$ | Coss sectional are of the fuselage from the front |
| $A_{c1} := A_{ffront} + 2\delta\cdot\left(b_w + b_{wl}\right) = 115.74\cdot\text{cm}^2$ | Cross sectional area of glider from the front |
| $A_{c2} := A_{ftop} + A_w = 2.784 \times 10^3\cdot\text{cm}^2$ | Cross sectional area of glider from top or bottom |
| $A_{wfront} := 2\cdot\delta\cdot\left(b_w + b_{wl}\right) = 65.657\cdot\text{cm}^2$ | Cross sectional area of wings and wingelts from the front |
| $\rho_{all} := .0167\frac{\text{kg}}{\text{m}^3}, .0168\frac{\text{kg}}{\text{m}^3} .. 1.225\frac{\text{kg}}{\text{m}^3}$ | Range variable of air density from 100,000ft to sea level |
| $AR := \frac{b_w^{\,2}}{A_w} = 1.265$ | Aspect Ratio of the wings |
| $\lambda := \frac{l_{wt}}{l_{wr}} = 0.286$ | Taper ratio of the wings |
| $l_\mu := \frac{2}{3}\cdot\frac{1 + \lambda + \lambda^2}{1 + \lambda}\cdot l_{wr} = 0.242\,\text{m}$ | mean chord length of swept wings |
| $\rho_{air} := 1.225\frac{\text{kg}}{\text{m}^3}$ | Density of air at sea level |
| $\alpha_0 := 0\cdot° = 0$ | Angle of attack = 0 ° |
| $\alpha_5 := 5° = 0.087$ | Angle of attack = 5 ° |

| | |
|---|---|
| $\alpha := -.175, -.174 .. .175$ | Range variable of angle of attack in radians (-10 ° to 10 °) |
| $g = 9.807\,\frac{m}{s^2}$ | Acceleration of gravity |
| $\mu_{air} := 1.983\cdot 10^{-5}\,Pa\,s$ | Viscocity of air |
| $V_d := 20mph = 8.941\,\frac{m}{s}$ | Desired airspeed of glider determined in simulation |

## Reynalds Number

$$Re := \frac{\rho_{air}\cdot V_d\cdot l_\mu}{\mu_{air}} = 1.334\times 10^{5}$$

Turbulant flow

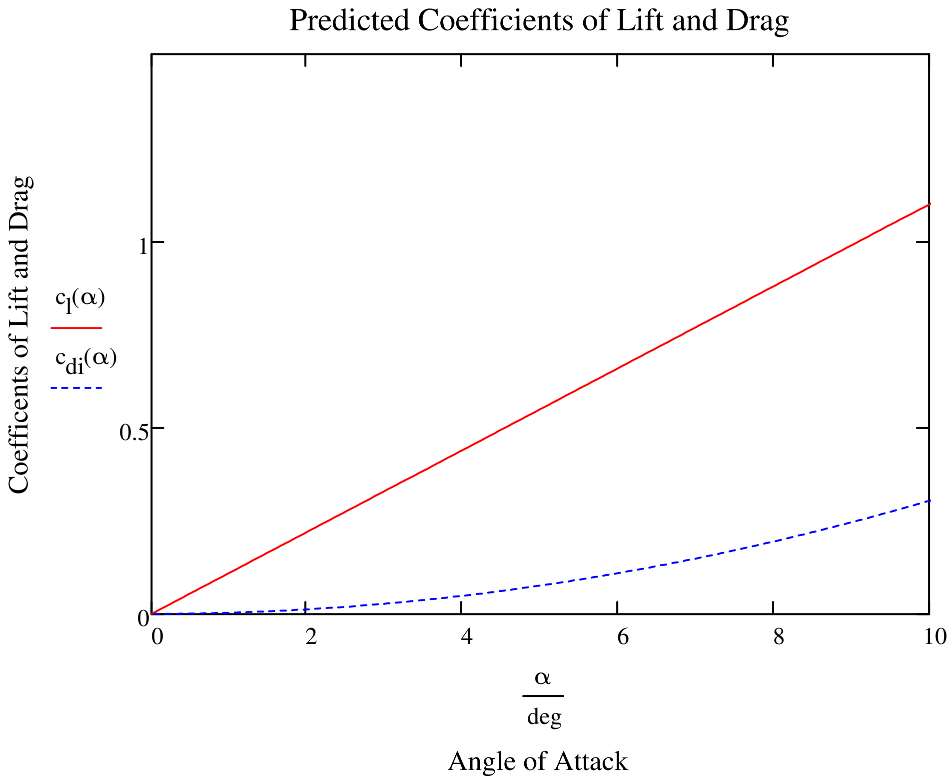
## boundry layer thickness

$$\delta_b := \frac{.382 l_\mu}{Re^{\frac{1}{5}}} = 8.712\cdot mm$$

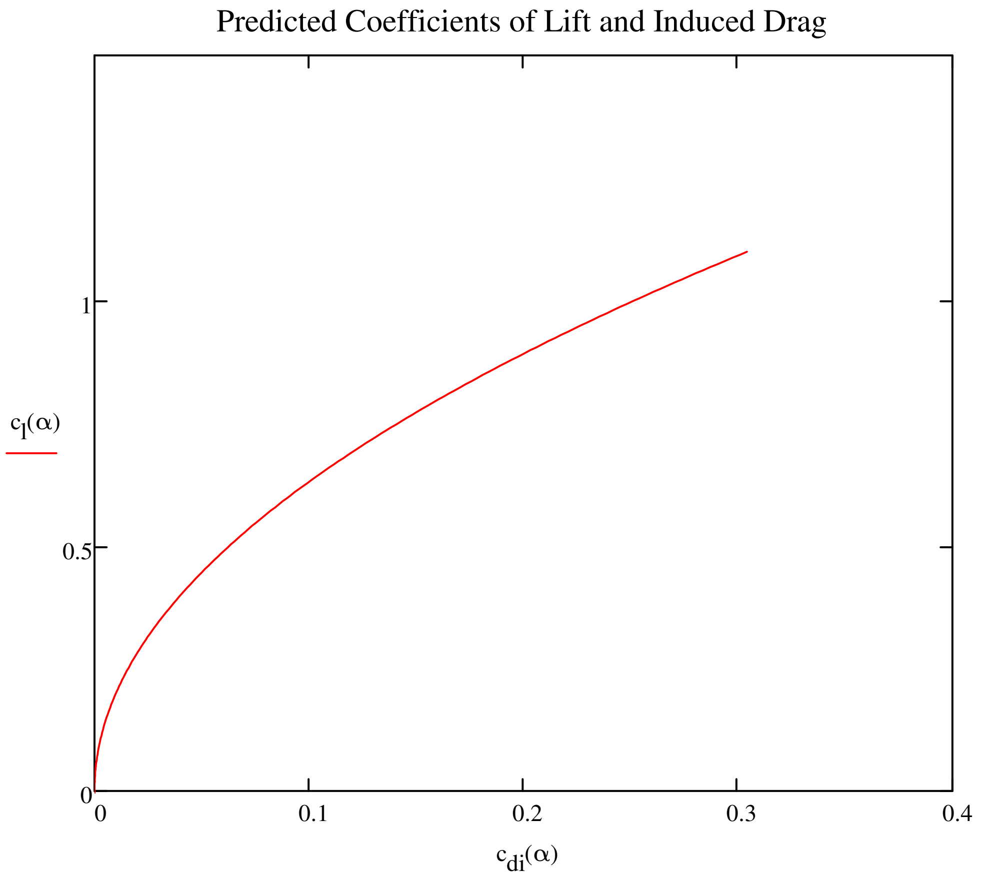
$$WingLoading := \frac{m_g}{A_w} = 1.485\,\frac{kg}{m^2}$$

## Induced Drag

| | |
|---|---|
| $C_l = 2\pi\alpha$ | Equation for coefficient of lift of a flat plate |
| $c_{l0} := 2\pi\alpha_0 = 0$ | Coefficient of lift at 0 ° |
| $c_{l5} := 2\pi\alpha_5 = 0.548$ | Coefficient of lift at 5 ° |
| $c_l(\alpha) := 2\pi\alpha$ | Coefficient of lift at a given angle of attack |
| $C_{di} = \frac{C_l^{2}}{\pi AR}$ | Equation for coefficent of induced drag |
| $c_{di0} := \frac{c_{l0}^{2}}{\pi AR} = 0$ | Coefficent of induced drag at 0 ° |
| $c_{di5} := \frac{c_{l5}^{2}}{\pi AR} = 0.076$ | Coefficent of induced drag at 5 ° |
| $c_{di}(\alpha) := \frac{(2\pi\alpha)^{2}}{\pi AR}$ | Coefficent of induced drag at any given angle of attack |



Does not account for flow seperation at angles of attack above 10*



Form Drag [1]

$c_{dff} := .1$ drag coefficient of long, streamlined body at a=0°

$c_{dfw} := .005$ drag coefficient of flat plate wings at a=0°

Friction (skin) Drag

$$\tau = \mu_{air}\cdot\frac{dV}{dy} = \mu_{air}\cdot\frac{v}{\dfrac{.382l_{\mu}}{Re^{\frac{1}{5}}}}$$ Equation for shear force of air on glider skin

$$c_{dfr} = \frac{\tau}{\frac{1}{2}\rho_{air}\cdot V^2}$$ Equation for coefficent of friction drag

$$A_s := (2\cdot22.118 + 2 + 46.926 + 3.058 + 8.3187 + 23.191 + 4.430 + 14.467 + 4\cdot9.879)in^2 + 4\cdot A_w$$

$A_s = 1.109\,m^2$ Surface area of entire glider

Terminal Velocity

$F_g := m_g \cdot g = 3.599 \cdot N$ Force of gravity on glider

$F_d = \frac{1}{2} \rho_{air} \cdot v^2 \cdot c_d \cdot A_c$ Equation for drag force

$F_g = F_d$ Equation to find terminal velocity

Given

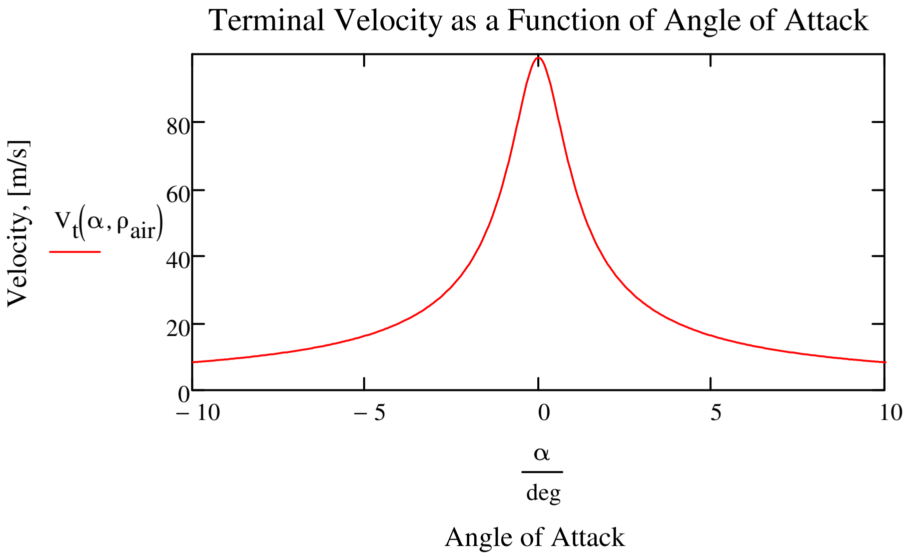
$v := 150mph$ guess for the program

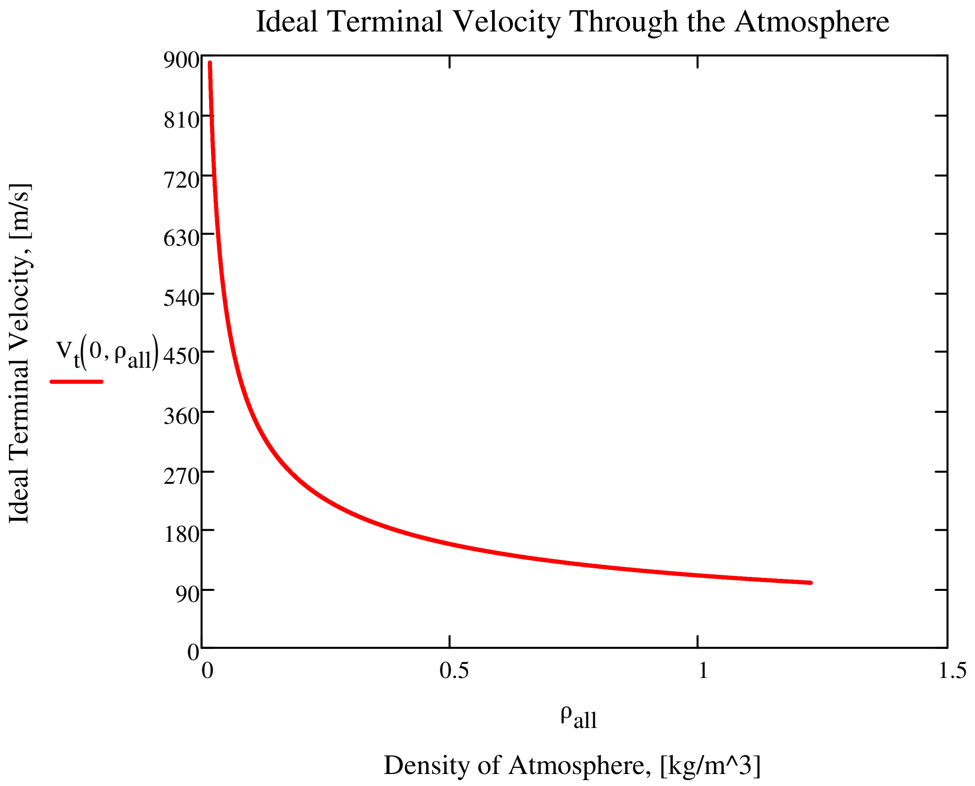
coefficnets of drag added together in proportion to the areas they affect

$$F_g = \frac{1}{2} \rho_{all} \cdot v^2 \left[ c_{di}(\alpha) \cdot \left( \sin\left( \frac{\pi}{2} - \alpha \right) \cdot A_{c1} + \cos(\alpha)\, A_{c2} \right) + c_{dff} \cdot A_{ffront} + c_{dfw} \cdot A_{wfront} + \frac{\mu_{air} \cdot \dfrac{v}{\left[ \dfrac{.382 l_{\mu}}{\left( \dfrac{\rho_{air} \cdot v \cdot l_{\mu}}{\mu_{air}} \right)^{\frac{1}{5}}} \right]}}{\frac{1}{2} \rho_{air} \cdot v^2} \cdot A_s \right]$$

$V_t(\alpha, \rho_{all}) := \text{Find}(v)$

$V_t(0, \rho_{air}) = 98.875 \frac{m}{s}$ (221.177mph) Ideal Terminal Velocity





Drag force at ideal terminal velocity

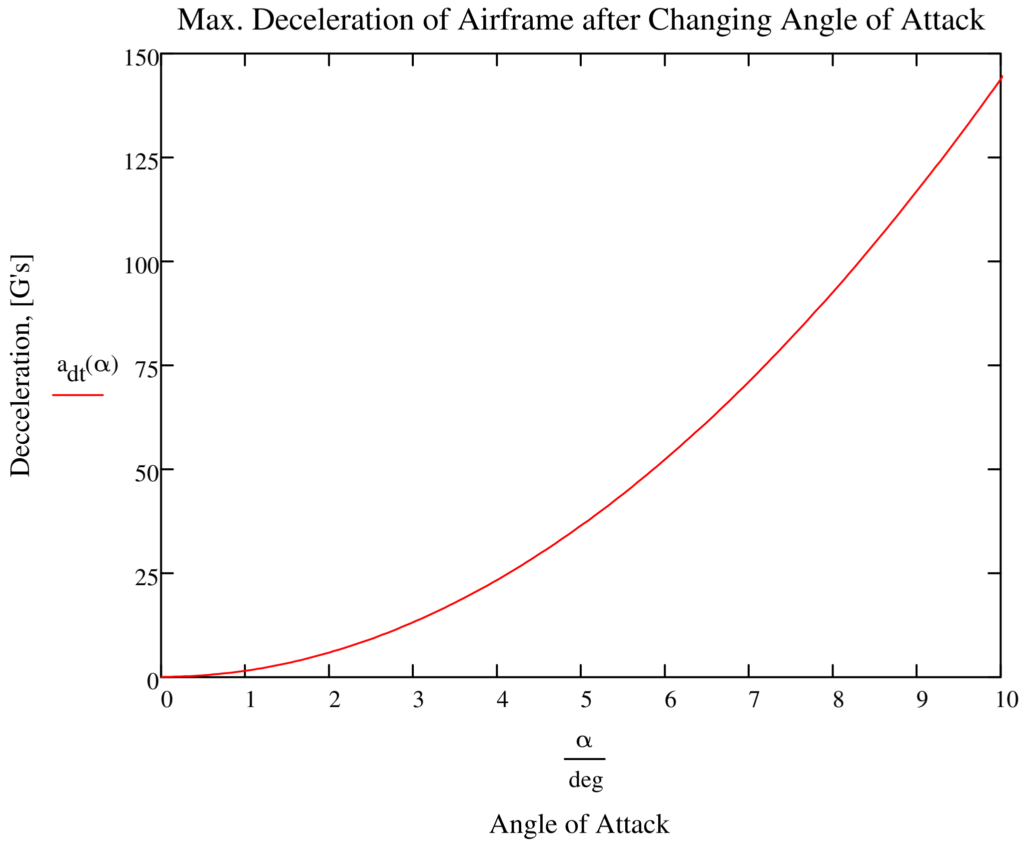
$$F_{dt}(\alpha,\rho_{all}) := \frac{1}{2}\rho_{all}\cdot\left(V_t(0,\rho_{air})\right)^2\left[c_{di}(\alpha)\cdot\left(\sin\left(\frac{\pi}{2}-\alpha\right)\cdot A_{c1}+\cos(\alpha)\,A_{c2}\right)+c_{dff}\cdot A_{ffront}+c_{dfw}\cdot A_{wfront}\cdots + \frac{\mu_{air}\cdot\dfrac{V_t(0,\rho_{air})}{\left[\dfrac{.3821_{\mu}}{\left(\dfrac{\rho_{air}\cdot V_t(0,\rho_{air})\cdot 1_{\mu}}{\mu_{air}}\right)^{\frac{1}{5}}}\right]}}{\frac{1}{2}\rho_{air}\cdot\left(V_t(0,\rho_{air})\right)^2}\cdot A_s\right]$$

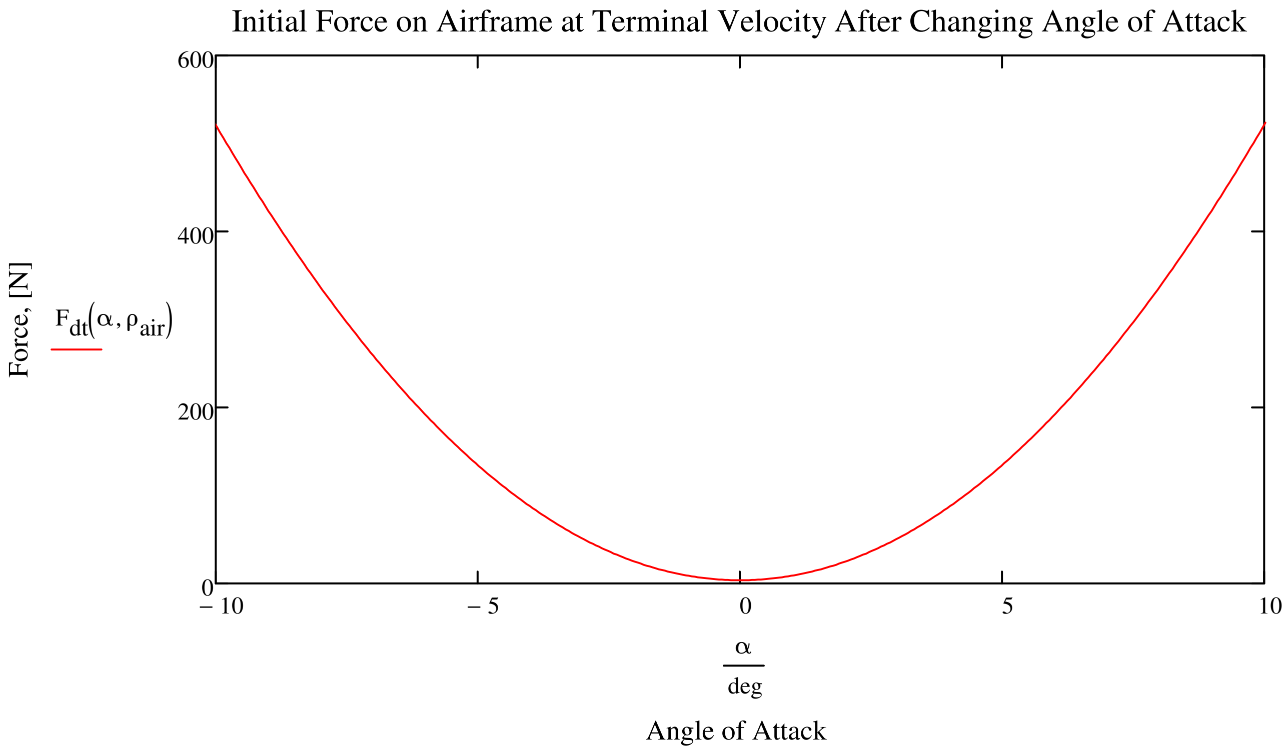
$F_{dt}(5\cdot{}^{\circ},\rho_{air}) = 134.43\,\mathrm{N}$

Inital drag force on airframe
turning 5*

$$a_{dt}(\alpha) := \frac{F_{dt}\left(\alpha, \rho_{air}\right) - F_g}{m_g \cdot g}$$

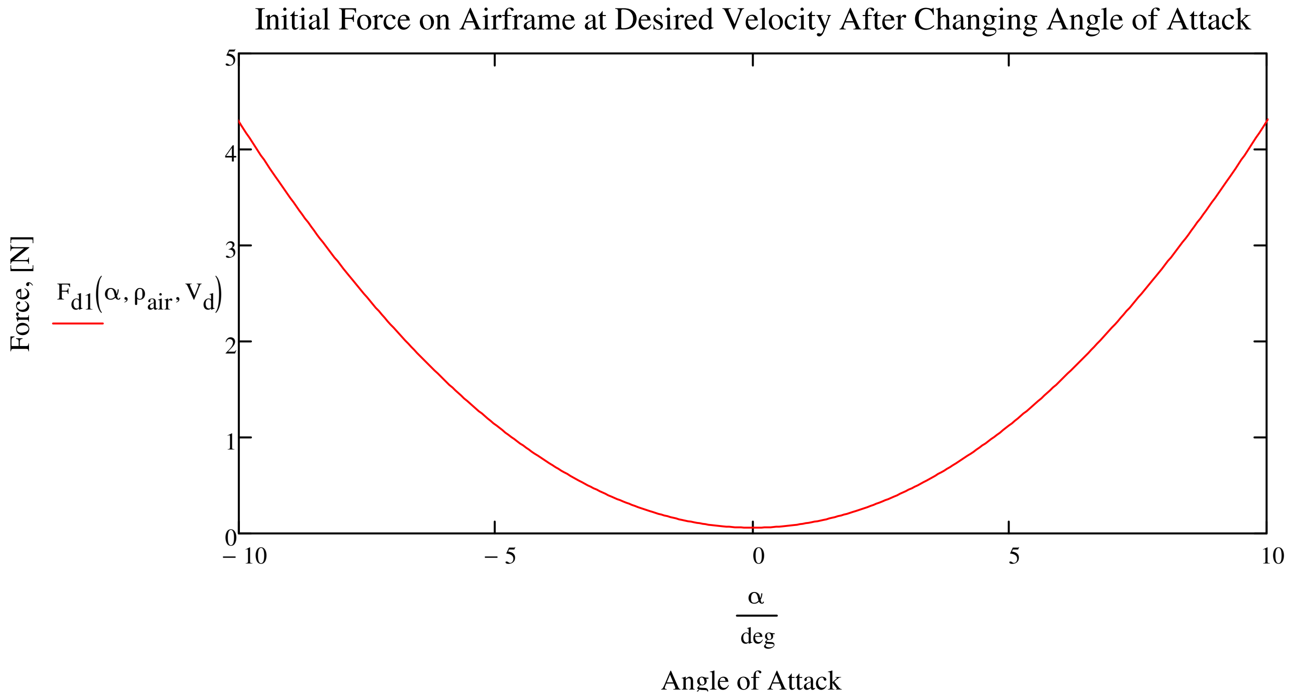
Initial decceleration on airframe at terminal velocity (in G's)




Initial Force on Airframe at Terminal Velocity After Changing Angle of Attack

Drag force at any angle of attack, air density, and velocity:

$$F_{d1}(\alpha, \rho_{all}, v) := \frac{1}{2}\rho_{all}\cdot(v)^2\left[c_{di}(\alpha)\cdot\left(\sin\left(\frac{\pi}{2}-\alpha\right)\cdot A_{c1}+\cos(\alpha)\,A_{c2}\right)+c_{dff}\cdot A_{ffront}+c_{dfw}\cdot A_{wfront}\cdots + \frac{\mu_{air}\cdot\frac{v}{5.823mm}}{\frac{1}{2}\rho_{air}\cdot(v)^2}\cdot A_s\right]$$


Initial Force on Airframe at Desired Velocity After Changing Angle of Attack

Bamboo Properties [2]

$\sigma_{bb} := 20.27\frac{N}{mm^2} = 20.27\cdot MPa$ Ultimate bending stress of bamboo

$r_b := .125in = 3.175\,mm$ Radius of bamboo spars

Carbon fiber Properties [3]

$\sigma_{bc} := 89000psi = 613.633\cdot MPa$ Ultimate bending stress of carbon fiber

$r_c := \frac{3}{32}in = 2.381\,mm$ Radius of carbon fiber spars

$\sigma = \frac{FL}{\pi r^3}$ $\quad F_b = \frac{\sigma\cdot\pi\cdot r^3}{L}$ Equation for maximum bending force on wings

Forces on differnt bamboo spar lengths

$$F_{bb1} := \frac{\sigma_{bb}\cdot\pi\cdot r_b^{\,3}}{9.5in} = 8.447\,N \qquad F_{bb2} := \frac{\sigma_{bb}\cdot\pi\cdot r_b^{\,3}}{12in} = 6.687\,N$$

Forces on different carbon fiber spar lengths

$$F_{bc1} := \frac{\sigma_{bc}\cdot\pi\cdot r_c^{\,3}}{9.5in} = 107.874\,N \qquad F_{bc2} := \frac{\sigma_{bc}\cdot\pi\cdot r_c^{\,3}}{12in} = 85.4\,N \qquad F_{bc3} := \frac{\sigma_{bc}\cdot\pi\cdot r_c^{\,3}}{24in} = 42.7\,N$$

$F_{bbmax} := F_{bb1} + 3F_{bb2} = 28.507\,N$ Force to break bamboo spars (4 spars)

$F_{bcmax} := F_{bc1} + F_{bc2} + F_{bc3} = 235.974\,N$ Force to break carbon fiber spars (3 spars)

$W_{bb} := \frac{F_{bbmax}}{g} = 2.907\,kg$ Equivalent mass on airframe to break bamboo

$W_{bc} := \frac{F_{bcmax}}{g} = 24.063\,kg$ Equivalent mass on airframe to break carbon fiber

$V_{cs} := 0, .1\frac{m}{s} .. V_t(0, \rho_{air})$ Range variable of velocities from 0 to ideal terminal velocity

$\sigma_{bps} = \frac{3F \cdot L}{2 \cdot b \cdot d^2}$ Equation to find bending stress of polystyrene

Experimental results:

$b_{bps} := 1\text{cm}$ Width of polystyrene foam board test articles

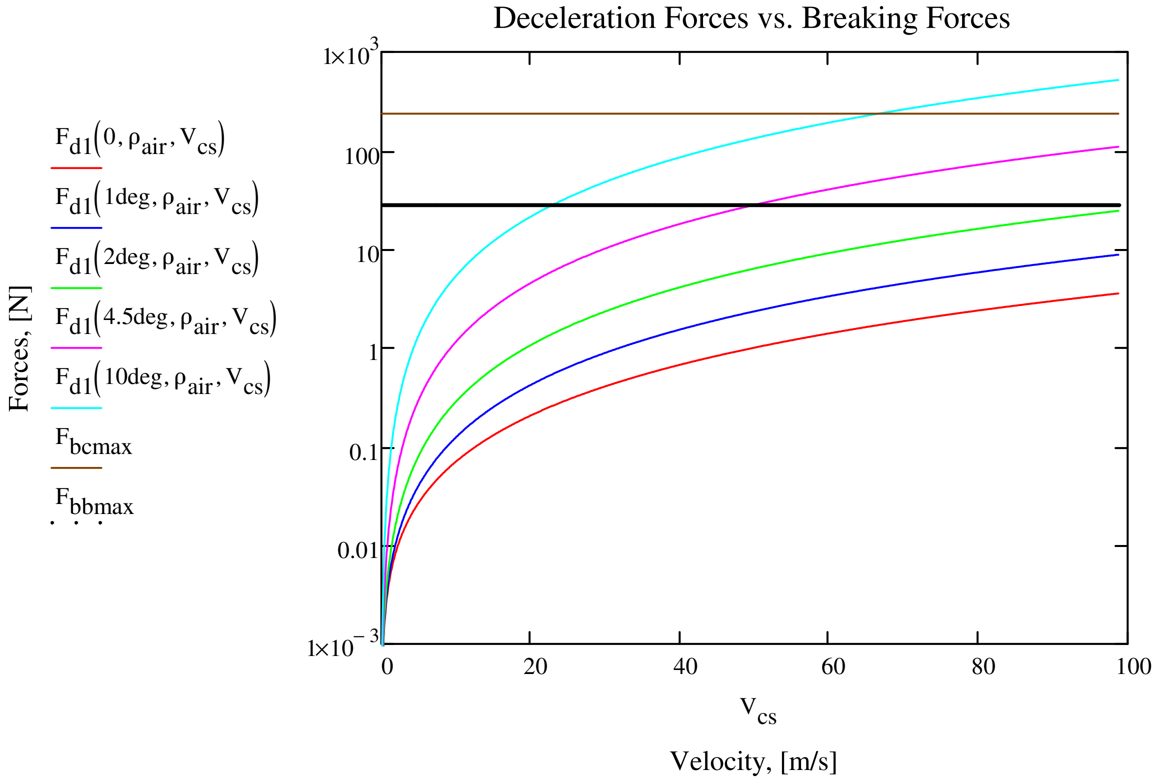
$L_{bps} := 5.3\text{cm}$ Length of polystyrene foam board test articles

$F_{bps} := 4.2\text{N}$ Force applied when polystryene foam board failed

$\sigma_{bps} := \frac{3F_{bps} \cdot L_{bps}}{2 \cdot b_{bps} \cdot \delta^2} = 1.336 \times 10^3 \cdot \text{kPa}$ Ultimate bending stress of polystyrne foam baord.

$F_{bwing} := \int_{l_{wt}}^{l_{wr}} \frac{2\sigma_{bps} \cdot \delta^2}{3 \cdot b_w} \, dl = 9.69 \cdot \text{N}$ Bending force to break wing at center of span

The wing technically will break before the bamboo or carbon fiber spars, however since the bending moment of both wings is centered on midpoint of the spars (inside the bulkhead), the wings experience much less bending force.



$\sigma_{Tp} := 43.6 \dfrac{N}{mm^2}$ Tensile strength of craft paper [4]

$\delta_{paper} := .12mm$ Thickness of paper

$l_{paper} := 19.349in = 0.491\,m$ Length of control surface

$A_{cpaper} := \delta_{paper} \cdot l_{paper} = 0.59 \cdot cm^2$ Area of control surface connection

$F_{tpaper} := \sigma_{Tp} \cdot A_{cpaper} = 2.571 \times 10^3 \cdot N$ Force required to tear off control surface

Asuming force on control surfaces is akin to a fluid jet striking an angled, flat plate

$A_{cs} := 19.215in \cdot 1.772in = 0.022\,m^2$ Area of 1 control surface

$\theta_{max} := 60° = 1.047$ Max deflection of control surface

$\phi := 90° - 83.2° = 0.119$ Sweep angle of control surfaces

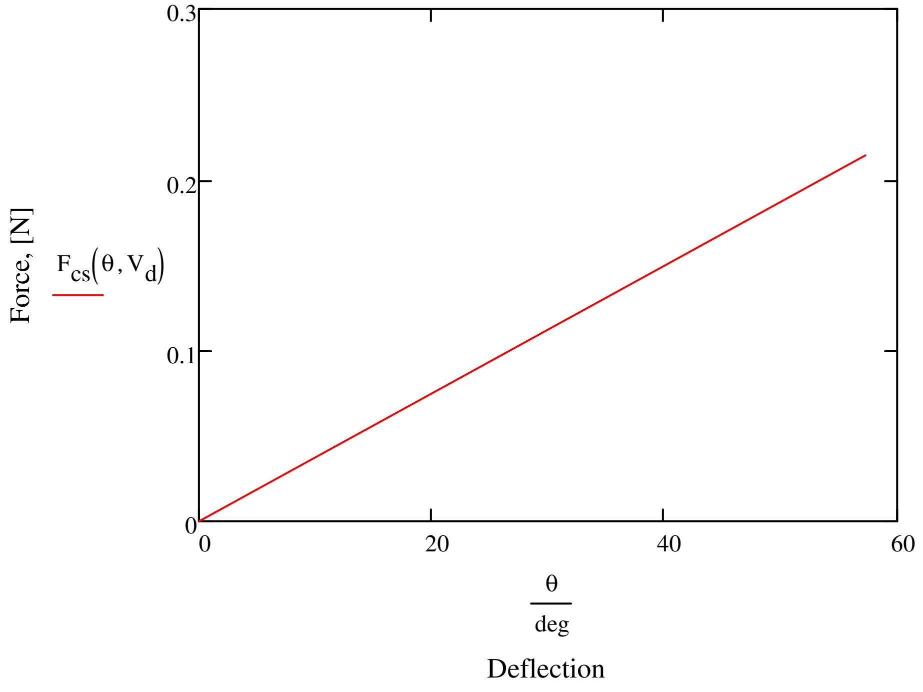
$F_{csmax}(V_{cs}) := \rho_{air} \cdot A_{cs} V_{cs}^{2} \cdot \sin(\theta_{max}) \cdot \sin(\phi)$ Force on 1 control surface at maximum deflection

$F_{csmax}(20mph) = 0.221\,N$ Force on 1 control surface at 20mph

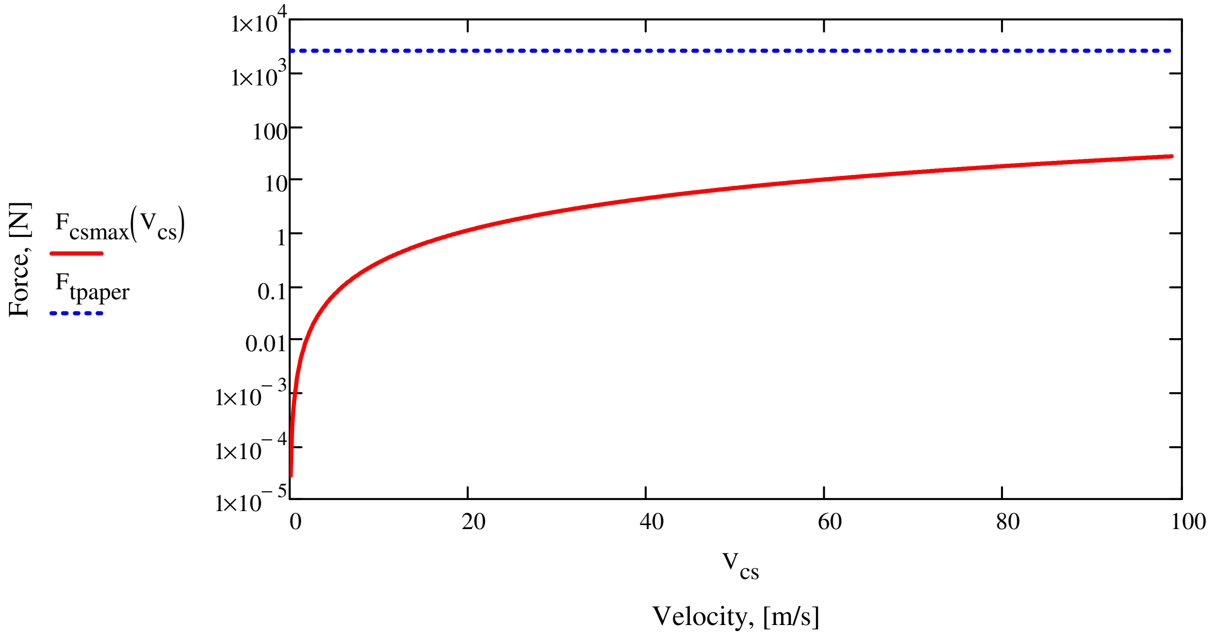
$\theta := 0 .. 1.047$ Range variable of control surface deflection, from 0° to 60°

$F_{cs}(\theta, V_{cs}) := \rho_{air} \cdot A_{cs} \cdot V_{cs}^{2} \sin(\theta) \sin(\phi)$ Force on 1 control surface at any velocity and delfection

Force on Control Surface at a Given Deflection



Force on Control Surface at Max Deflection vs Force to Tear it off.



$$c_d := 1.2$$

Coefficient of drag of a perpendicular flat plate

$$d_{com} := 7.14\text{in}$$

Distance from COM

$$d_{center} := 10.604\text{in} + \frac{75\text{mm}}{2} = 0.307\,\text{m}$$

Distance from center of fuselage

$$M_{com}(\theta, V_{cs}) := d_{com} \cdot 2\,F_{cs}(\theta, V_{cs})$$

Pitch moment

$$M_{center}(\theta, V_{cs}) := d_{center} \cdot 2\,F_{cs}(\theta, V_{cs})$$

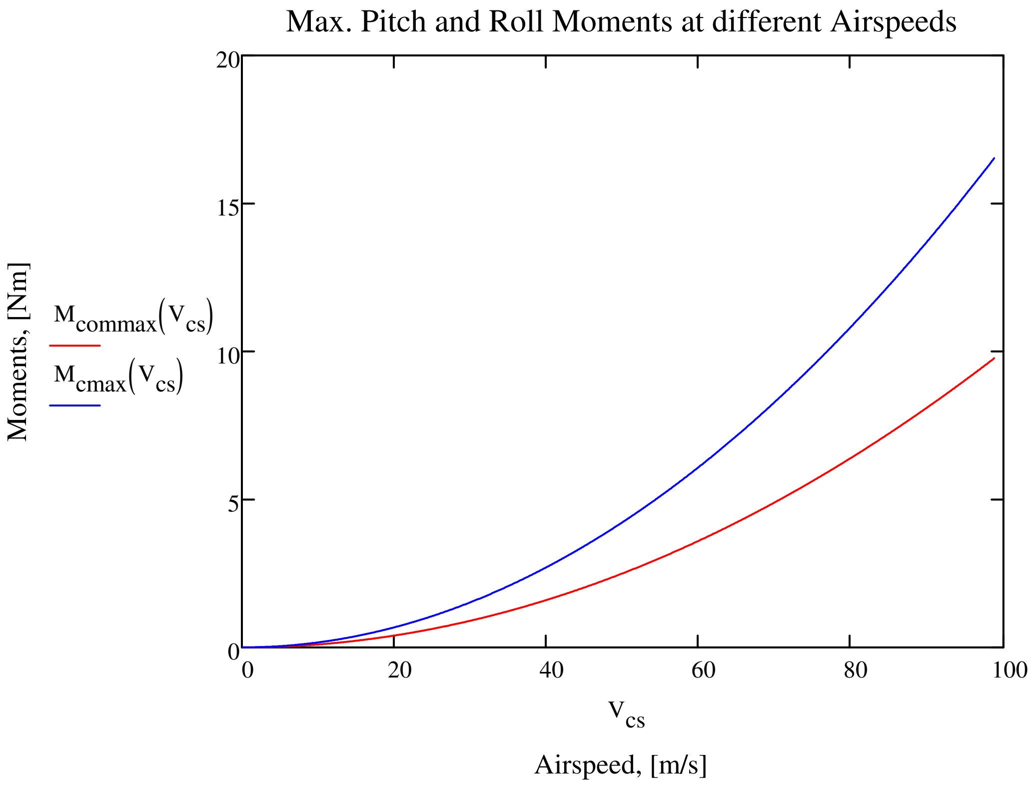
Roll moment

$$M_{commax}(V_{cs}) := d_{com} \cdot 2\,F_{csmax}(V_{cs})$$

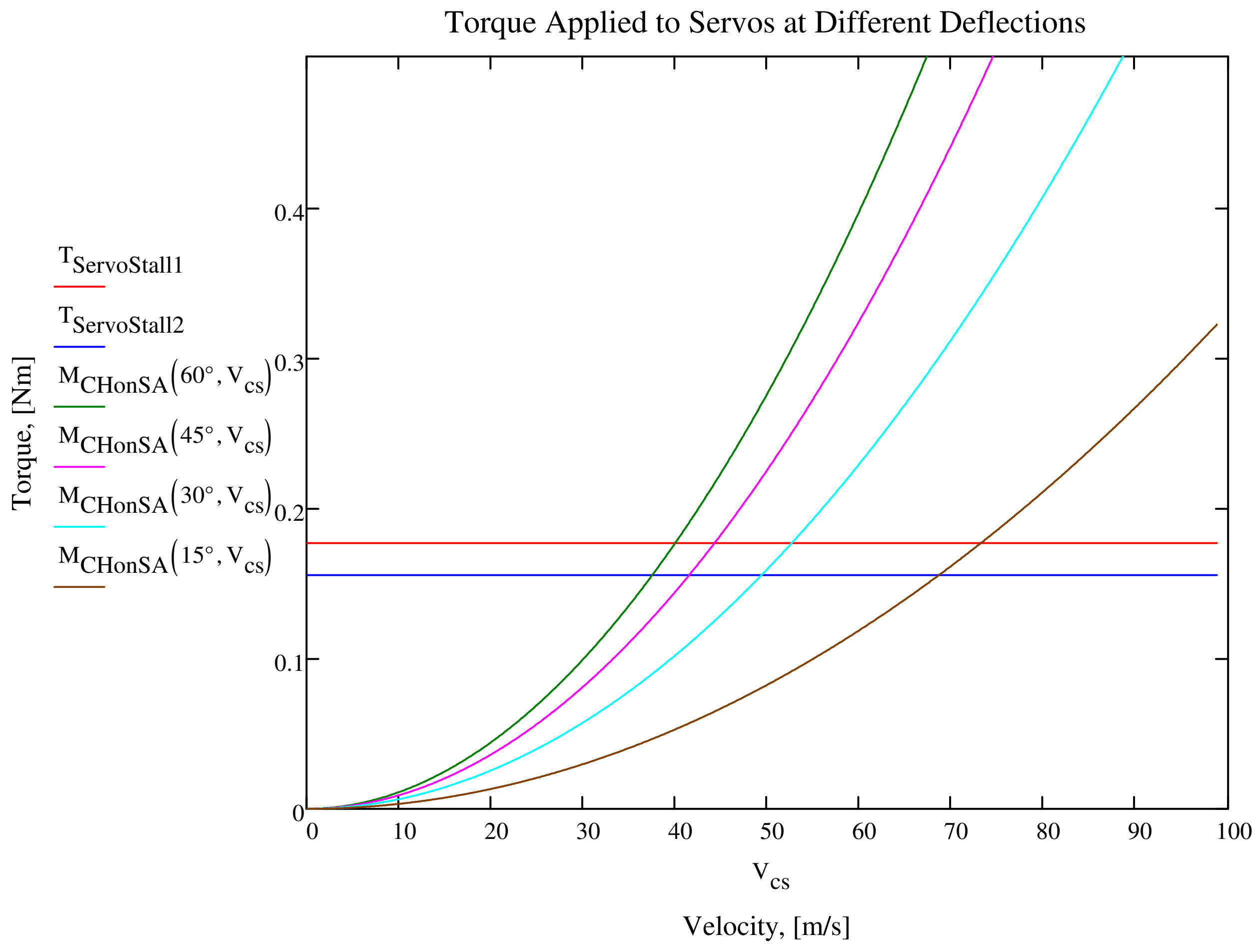
Max pitch moment

$$M_{cmax}(V_{cs}) := d_{center} \cdot 2\,F_{csmax}(V_{cs})$$

Max roll moment



| | |
|---|---|
| $T_{ServoStall1} := 1.8\,kgf \cdot cm = 0.177 \cdot N \cdot m$ | Max torque of SG90 Servos [5] |
| $T_{ServoStall2} := 22\,ozf \cdot in = 0.155 \cdot N \cdot m$ | Max torque of SM22 Servos [6] |
| $l_{ch} := 13.5\,mm$ | Length of control horn |
| $l_{sa} := 14\,mm$ | Length of Servo arm |
| $d_{cs} := 1.633\,in = 41.478\,mm$ | Distance between control surface center of force and control horn |
| $M_{CSonCH}(\theta, V_{cs}) := d_{cs} \cdot F_{cs}(\theta, V_{cs})$ | Moment of control surface on control horn |
| $M_{CHonSA}(\theta, V_{cs}) := M_{CSonCH}(\theta, V_{cs}) \cdot \frac{l_{ch}}{l_{sa}}$ | Moment of control horn on servo arm |



Given

$V_{cs} := 100\,\frac{m}{s}$

$T_{ServoStall1} = d_{cs}\cdot F_{cs}(60°, V_{cs})\cdot\frac{l_{ch}}{l_{sa}}$

$V_{servostall60} := Find(V_{cs}) = 39.993\,\frac{m}{s}$ (89.461 mph)

Given

$V_{cs} := 100\,\frac{m}{s}$

$T_{ServoStall1} = d_{cs}\cdot F_{cs}(45°, V_{cs})\cdot\frac{l_{ch}}{l_{sa}}$

$V_{servostall45} := Find(V_{cs}) = 44.259\,\frac{m}{s}$ (99.005 mph)

Given

$V_{cs} := 100\,\frac{m}{s}$

Speeds at which the servos will stall
at a given deflection

$T_{ServoStall1} = d_{cs}\cdot F_{cs}(30°, V_{cs})\cdot\frac{l_{ch}}{l_{sa}}$

$V_{servostall30} := Find(V_{cs}) = 52.633\,\frac{m}{s}$ (117.738 mph)

Given

$V_{cs} := 100\,\frac{m}{s}$

$T_{ServoStall1} = d_{cs}\cdot F_{cs}(15°, V_{cs})\cdot\frac{l_{ch}}{l_{sa}}$

$V_{servostall15} := Find(V_{cs}) = 73.156\,\frac{m}{s}$ (163.645 mph)

Given

$\theta := 5°$

$T_{ServoStall1} = d_{cs}\cdot F_{cs}(\theta, V_t(0, \rho_{air}))\cdot\frac{l_{ch}}{l_{sa}}$

$\theta_{maxt} := Find(\theta) = 8.145\cdot deg$

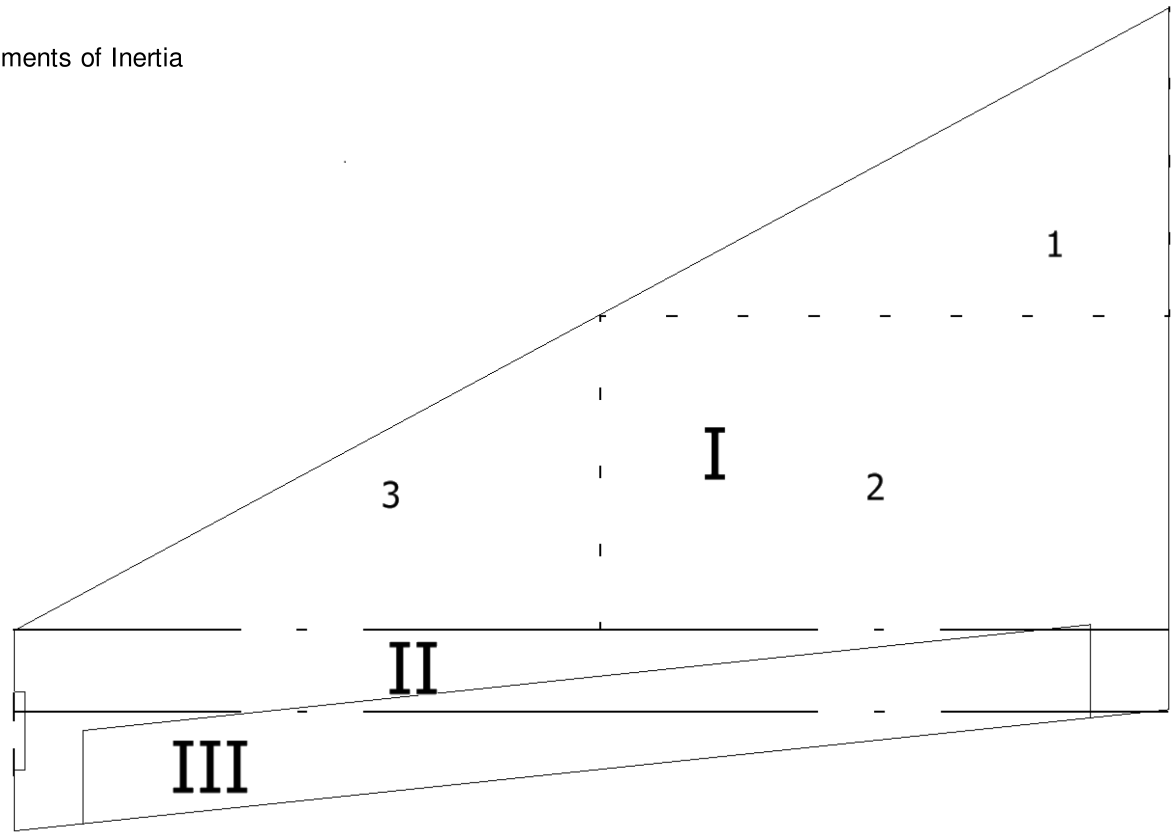
Max deflection at ideal terminal velocity

$M_{com}(\theta, V_{cs}) := d_{com}\cdot 2\,F_{cs}(\theta, V_{cs})$

$M_{com}(\theta_{maxt}, V_t(0, \rho_{air})) = 1.601\cdot N\cdot m$

$V_{cs} := 0, .1\,\frac{m}{s} .. V_t(0, \rho_{air})$ reset $V_{cs}$, $M_{com}$ from calculations

Moments of Inertia



Division of wing for moment of inertia calculations. I, II, and III are the main sections while 1, 2, and 3 are subsections of I

$b_f := 16.357\text{in} = 0.415\,\text{m}$ "Base" of fuselage

$h_f := 75\text{mm}$ "Height" of fusealge

$b_{w1} := 12.01\text{in} = 0.305\,\text{m}$ Base of each section of wing

$b_{w2} := 1.406\text{in} = 0.036\,\text{m}$

$b_{w3} := 2.427\text{in} = 0.062\,\text{m}$

$h_{w1} := 22.213\text{in} = 0.564\,\text{m}$ Height of each section of wing

$h_{w2} := 22.213\text{in} = 0.564\,\text{m}$

$h_{w3} := 22.213\text{in} = 0.564\,\text{m}$

Areas of each section of wing

$A_{w1} := \frac{1}{2} b_{w1} \cdot h_{w1} = 0.086\,\text{m}^2$ $A_{w2} := b_{w2} \cdot h_{w2} = 0.02\,\text{m}^2$ $A_{w3} := \frac{1}{2} b_{w3} \cdot h_{w3} = 8.695 \times 10^{-3}\,\text{m}^2 \cdot 2$

Wing section 1 cut into more sections around COM

$b_{11} := 6.5\text{in} = 0.165\,\text{m}$

Base of each subsection of wing

$b_{12} := 5.51\text{in} = 0.14\,\text{m}$

$b_{13} := 5.51\text{in} = 0.14\,\text{m}$

Height of each subsection of wing

$h_{11} := 12.021\text{in} = 0.305\,\text{m}$

$h_{12} := 12.021\text{in} = 0.305\,\text{m}$

$h_{13} := 10.191\text{in} = 0.259\,\text{m}$

Areas of each subsection of wing

$A_{w11} := .5 \cdot b_{11} \cdot h_{11} = 0.025\,\text{m}^2$

$A_{w12} := h_{12} \cdot b_{12} = 0.043\,\text{m}^2$

$A_{w13} := .5 \cdot b_{13} \cdot h_{13} = 0.018\,\text{m}^2$

$$I_{wx} := \int_{-A_w}^{A_w} 3\left(h_{w1}\right)^2 dA_w + \int_{-A_{ftop}}^{A_{ftop}} \left(\frac{h_f}{2}\right)^2 dA_{ftop} = 0.472\,\text{m}^4$$

Second moment of area of wing around x axis (roll)

$$I_{wy} := 2\cdot\left(\int_0^{A_{w11}} {h_{11}}^2\, dA_{w11}\right) - 2\cdot\left(\int_0^{A_{w12}} {h_{12}}^2\, dA_{w12}\right) - 2\cdot\left(\int_0^{A_{w13}} {h_{13}}^2\, dA_{w13}\right) \ldots = -5.879 \times 10^{-3}\,\text{m}^4$$
$$+ -2\cdot\left(\int_0^{A_{w2}} {b_{w2}}^2\, dA_{w2}\right) - 2\cdot\left(\int_0^{A_{w3}} {b_{w3}}^2\, dA_{w3}\right)$$

Second moment of area around COM (pitch)

$R_{gx} := \sqrt{\dfrac{I_{wx}}{A_{c1}}} = 6.388\,\mathrm{m}$ Radius of gyration about the x axis

$R_{gy} := \sqrt{\dfrac{\left|I_{wy}\right|}{A_{c1}}} = 0.713\,\mathrm{m}$ Radius of gyration about the y axis

$I_{xx} := m_g \cdot R_{gx}{}^2 = 14.974\,\mathrm{m}^2 \cdot \mathrm{kg}$ Moment of inertia about x axis (roll)

$I_{yy} := m_g \cdot R_{gy}{}^2 = 0.186\,\mathrm{m}^2 \cdot \mathrm{kg}$ Moment of inertia about y axis (pitch)

Roll rate [7]

$C_{l\delta a}$ = roll authority
$C_{lp}$ = roll damping
$\delta_a = \theta$
$c_{l\delta a}$=coefficiemt of lift with aileron deflection

$$\frac{p \cdot b_w}{2V_{cs}} = \text{constant}$$

$$p = -\frac{C_{l\delta a}}{C_{lp}} \delta_a \left(\frac{2V_{cs}}{b}\right)$$

$$\frac{p \cdot b_w}{2V_{cs}} = -\frac{C_{l\delta a}}{C_{lp}} \cdot \delta_a$$

$b_1 := \dfrac{75\mathrm{mm}}{2} + 1.006\mathrm{in} = 0.063\,\mathrm{m}$ Inner edge of aileron

$b_2 := \dfrac{75\mathrm{mm}}{2} + 20.213\mathrm{in} = 0.551\,\mathrm{m}$ Outer edge of aileron

$b_{cs} := \dfrac{75\mathrm{mm}}{2} + 10.611\mathrm{in} = 0.307\,\mathrm{m}$ Spanwise location of midpoint of control surface

$$c_{la} := \frac{\pi \cdot AR \cdot .5[1 + \cos[2 \cdot (0°)]]}{1 + \sqrt{1 + \dfrac{AR^2}{4} \cdot \left(1 - .0390997^2\right)\left[\dfrac{1 - \cos[2 \cdot (28.4°)]}{1 + \cos[2 \cdot (28.4°)]} + 1\right]}} = 1.781$$

Coeffcient of lift of aileron [8]

$$c_{l\delta a} := c_{la} \cdot \sqrt{\lambda} \cdot \frac{A_{cs}}{A_w} \cdot \frac{b_{cs}}{b_w} = 0.046$$

Coeffient of lift with aileron deflection

$$C_{l\delta a} := \frac{c_{l\delta a}\cdot l_{wr}}{A_w\cdot b_w}\cdot\left[\left(b_2^{\,2}-b_1^{\,2}\right)+\frac{4(\lambda-1)}{3\cdot b_w}\left(b_2^{\,3}-b_1^{\,3}\right)\right] = 1.74\times 10^{-3}$$

Roll authority

$$C_{lp} := -\frac{\left(c_{la}+c_{dfw}\right)\cdot l_{wr}\cdot b_w}{24\cdot A_w}(1+3\cdot\lambda) = -0.107$$

Roll damping

$$\theta := 0\,..\,1.047$$

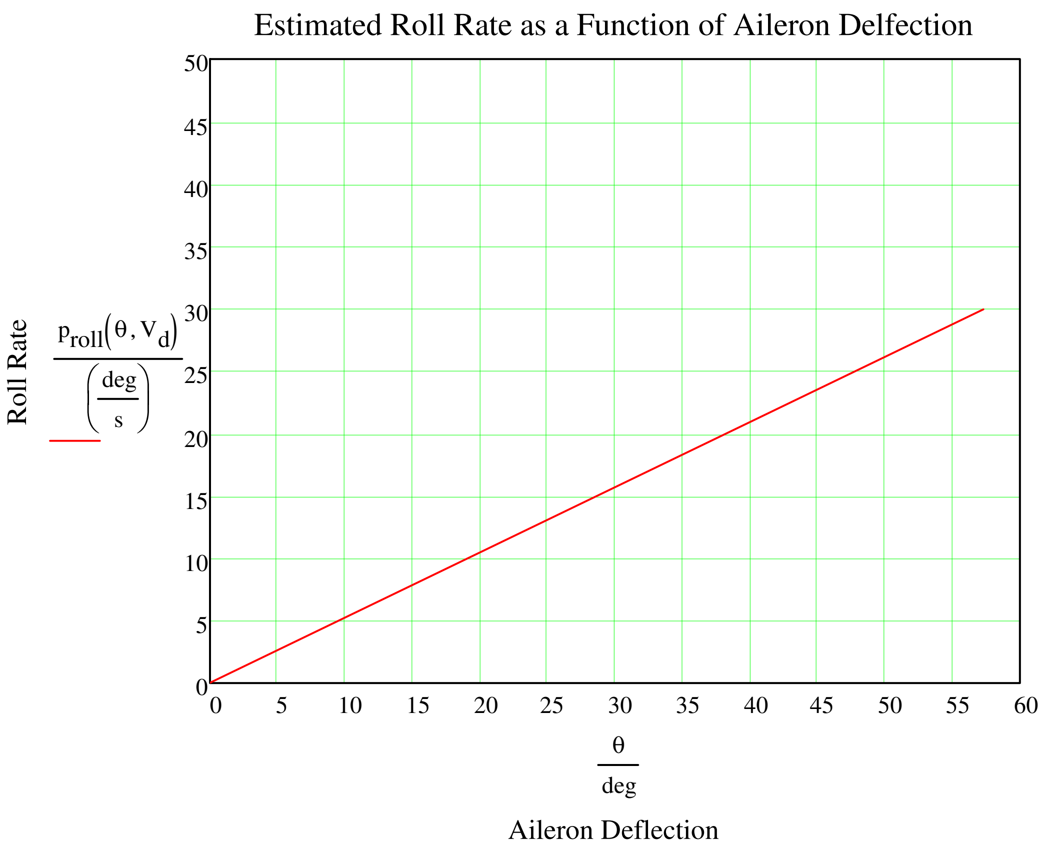
Redefining θ for program

$$p_{roll}\left(\theta, V_{cs}\right) := -\frac{C_{l\delta a}}{C_{lp}}\,\theta\cdot\frac{2\cdot V_{cs}}{b_w}$$

Roll Rate

$$p_{roll}\left(10^{\circ}, V_d\right) = 5.221\cdot\frac{deg}{s}$$

Roll rate with 5 ° deflection at 20mph



$$p_x := -\frac{C_{l\delta a}}{C_{lp}}\cdot\frac{2}{b_w} = 0.05839\,\frac{1}{m}$$

Roll rate constant, multiply by control surface deflection (in degrees) and velocity (in m/s) to get roll rate in deg/s

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