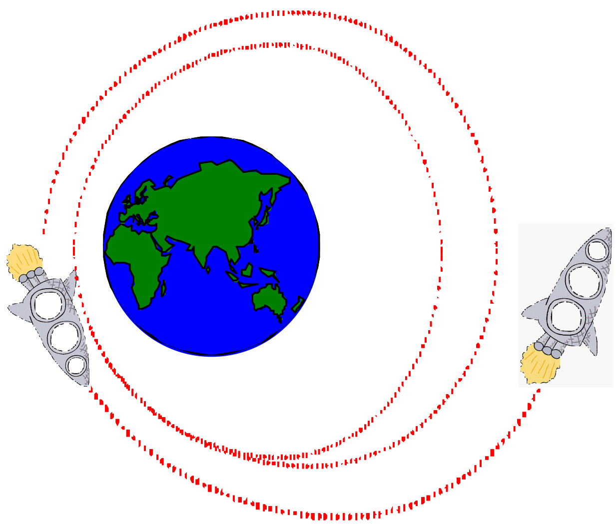
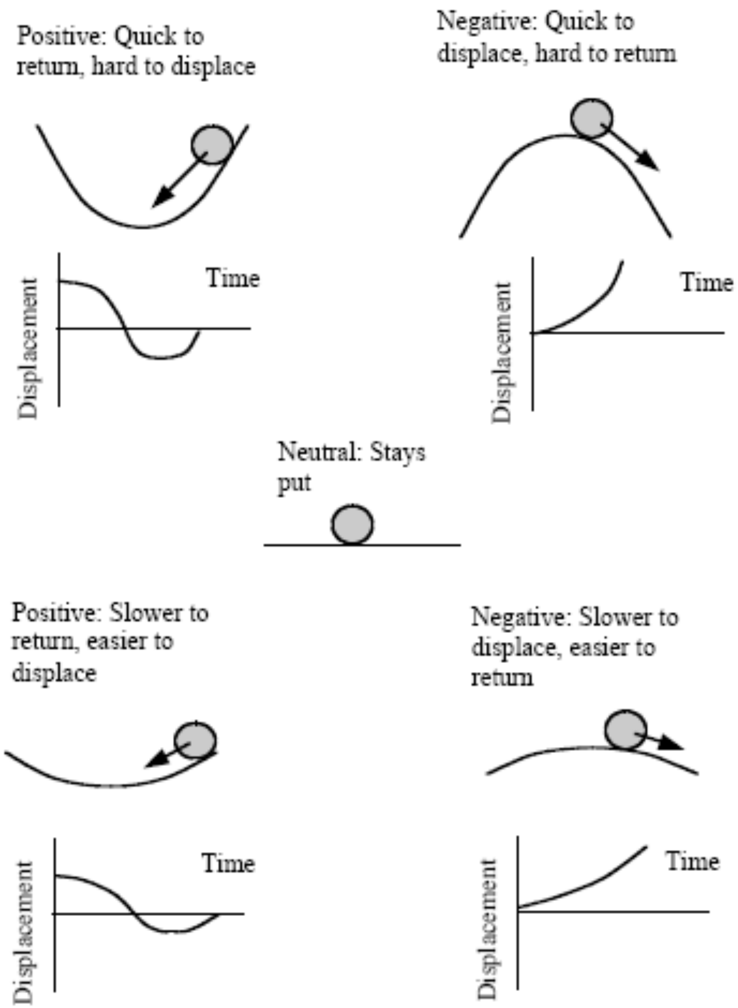
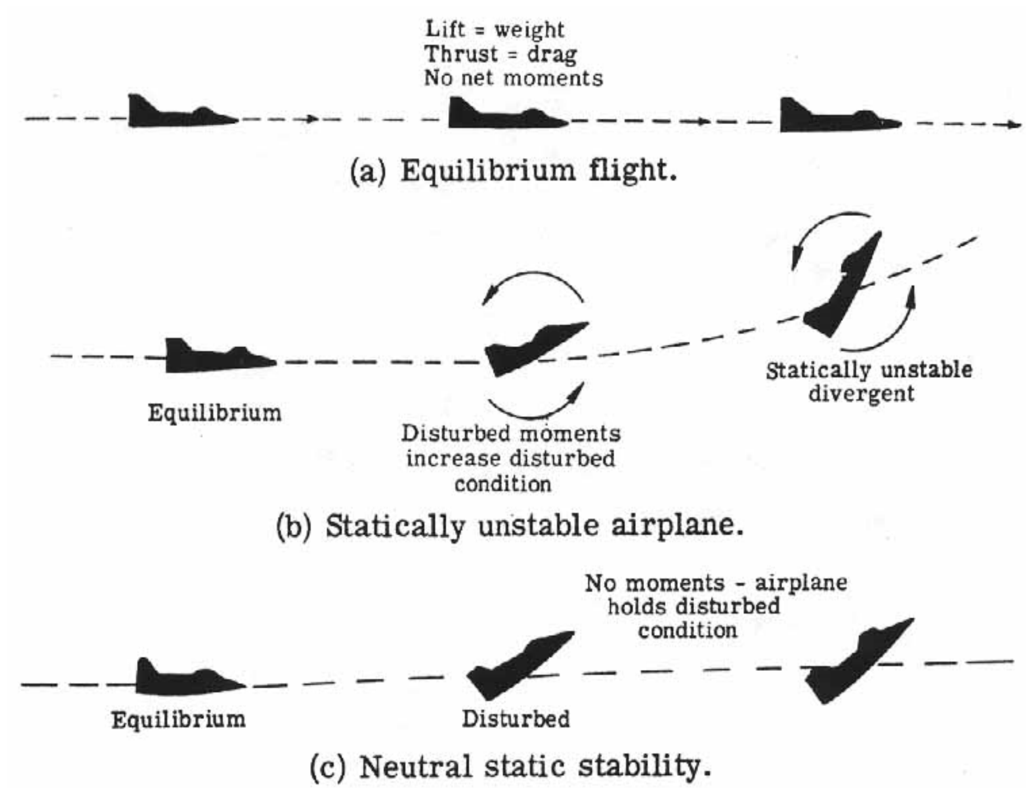
Longitudinal Vehicle (Pitch) Dynamics, Static and Dynamic Stability



Sellers: Chapter 12

Vehicle Stability

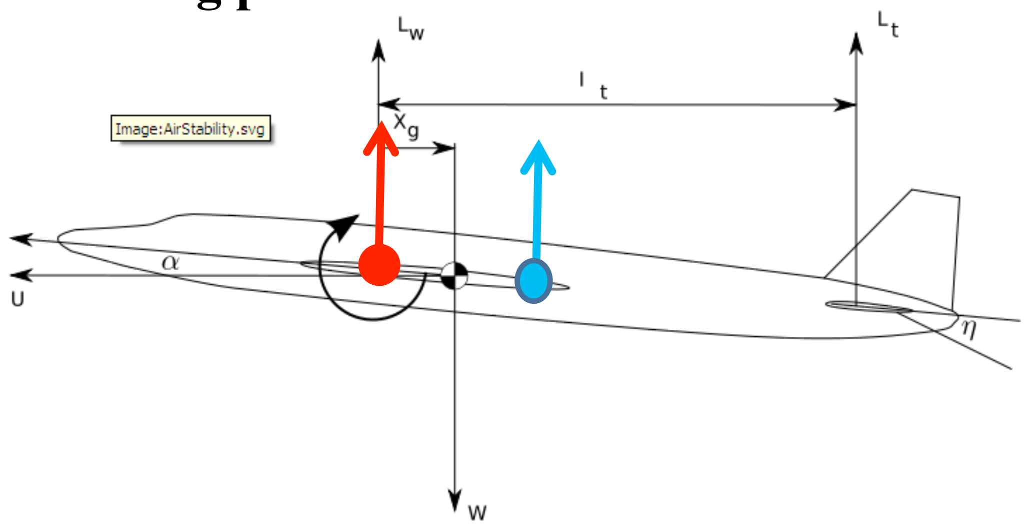




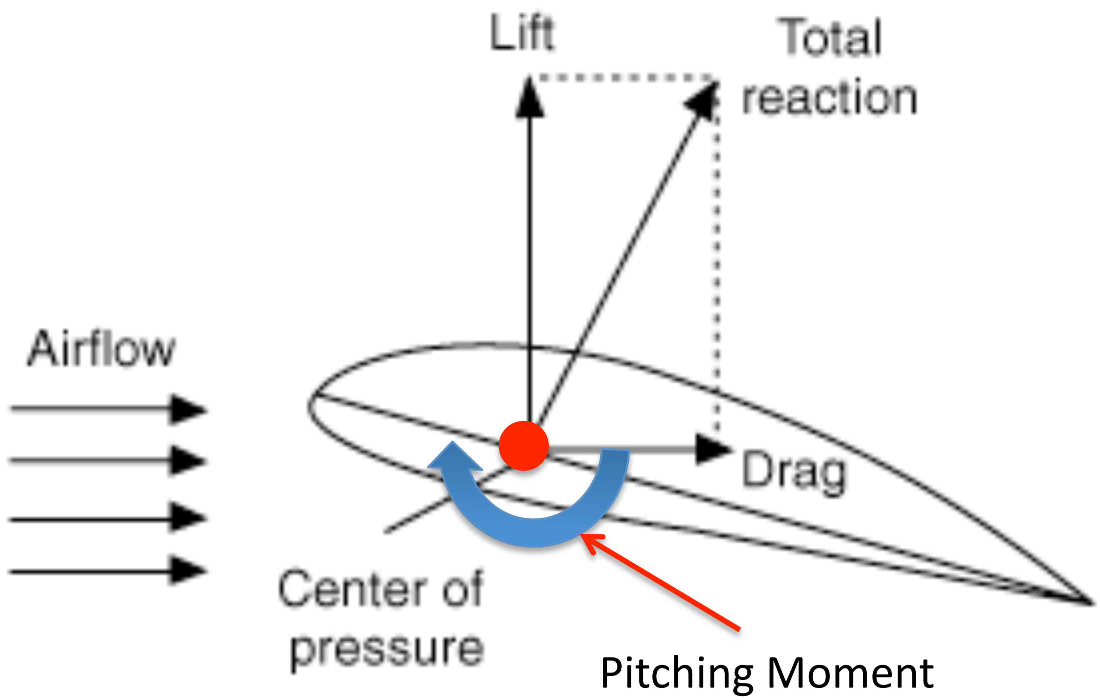
Vehicle Stability (2)

If center of gravity (*cg)* is forward of the (*cp)* , vehicle responds to a disturbance by producing aerodynamic moment that returns Angle of attack of vehicle towards angle that existed prior to the disturbance.

If CG is behind the center of pressure, vehicle will respond to a disturbance by producing an aerodynamic moment that continues to drive angle of attack further away from starting position.



Center of Pressure



•Aerodynamic *Lift, Drag, and Pitching Moment* Can be though of acting at a single point … the **Center of Pressure (*cp*)** of the vehicle

•Sometimes (*and not quite correctly*) referred to as the **Aerodynamic Center (AC)**

•For our purposed AC and CP are synonymous

Vehicle Stability, Rocket Flight Example

During rocket flight small wind gusts or thrust offsets can cause the rocket to "wobble", or change its attitude in flight.

Rocket rotates about its center of gravity *(cg)*

Lift and drag both act through the center of pressure *(cp*) of the rocket

When *cp* is behind *cg*, aerodynamic forces provide a "restoring force" … rocket is said to be "statically stable"

When *cp* ahead of *cg*, aerodynamic forces provide a "destabilizing force" … rocket is said to be "unstable"

Condition for a statically for a stable rocket is that center of pressure must be located behind the center of gravity.

Vehicle Stability, Rocket Flight Example (2)

During flight small wind gusts or thrust offsets cause the rocket to "wobble" … change attitude

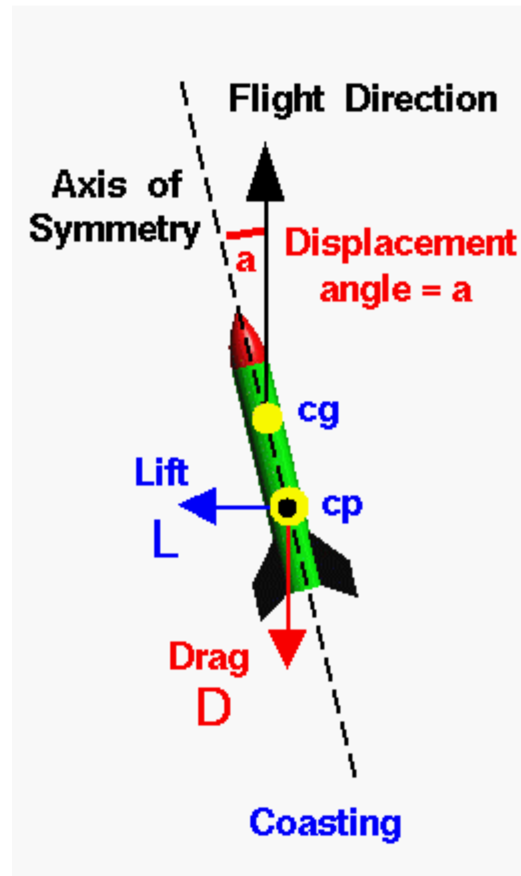
Rocket rotates about center of gravity *(cg)*

Lift and drag both act through center of pressure *(cp)*

When *cp* is behind *cg*, aerodynamic forces provide a "restoring force" … rocket is said to be "statically stable"

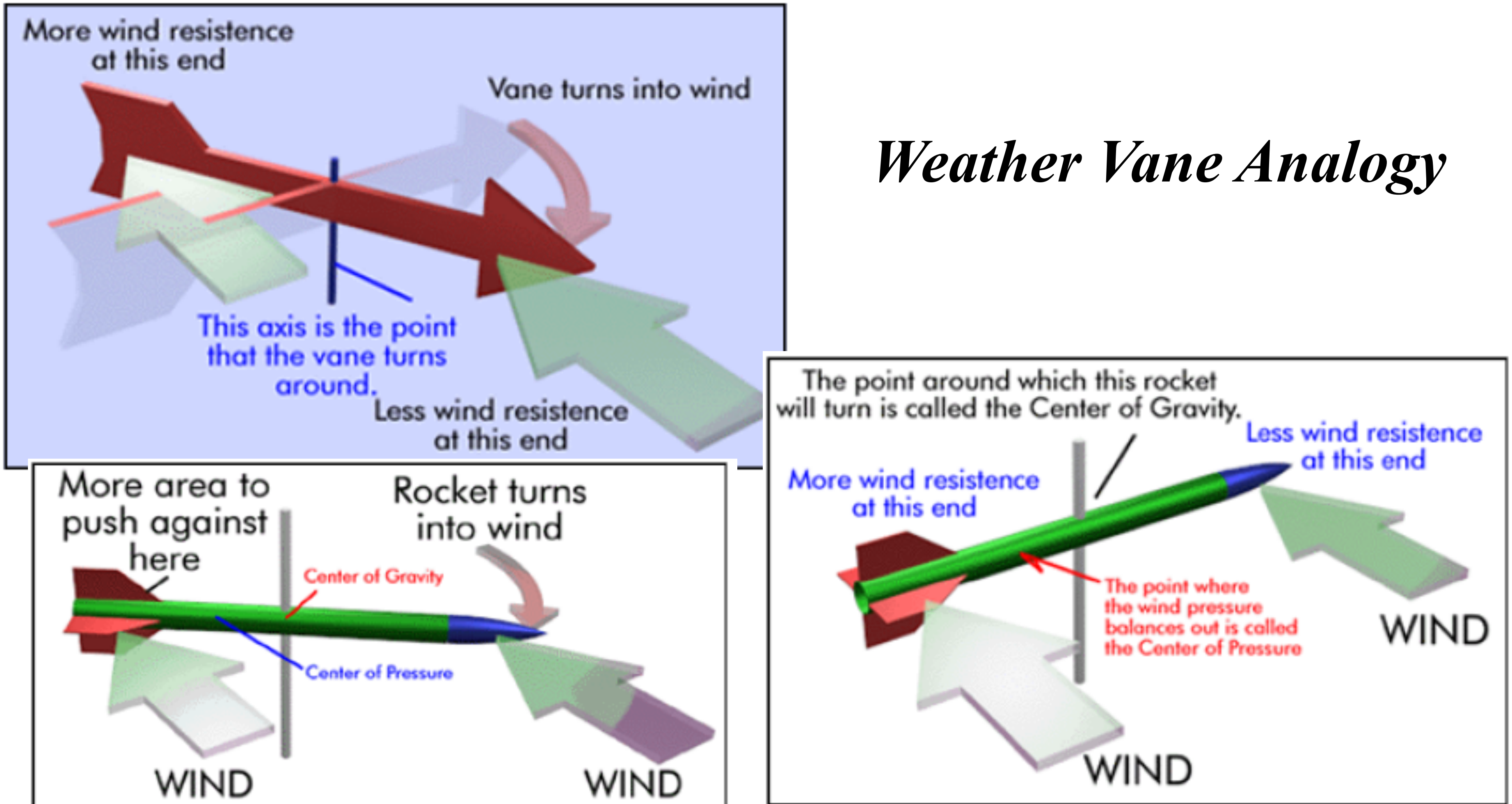
When *cp* ahead of *cg*, aerodynamic forces provide a "destabilizing force" … rocket is said to be "unstable"

Condition for a statically for a stable rocket is that center of pressure must be located behind longitudinal center of gravity.

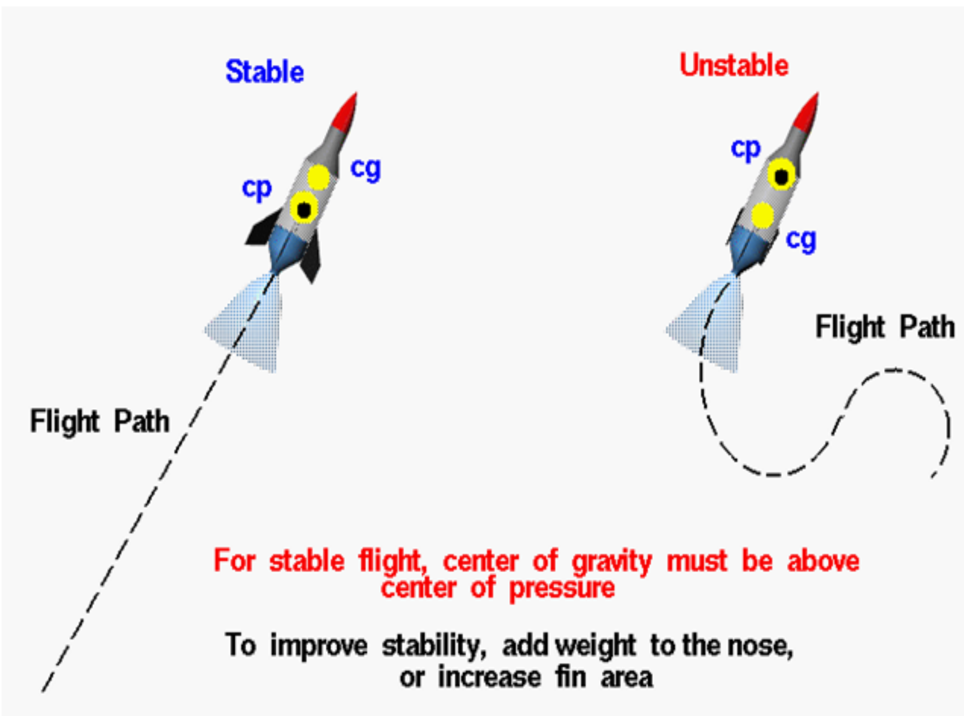


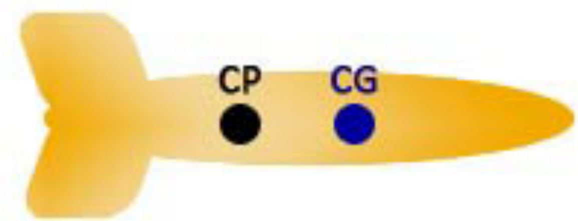
Vehicle Stability, Rocket Flight Example (3)

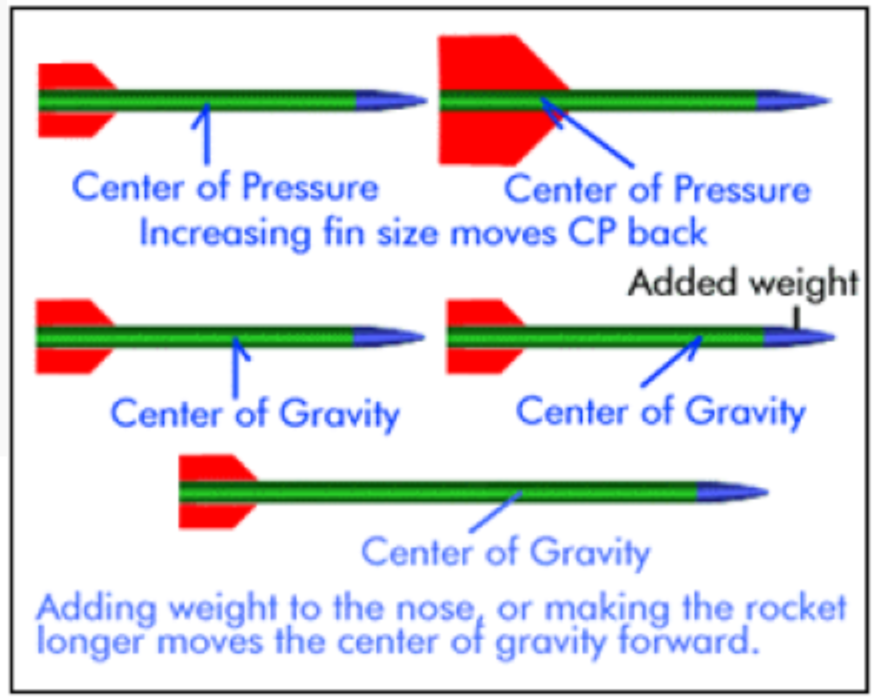
Weather Vane Analogy



Vehicle Stability, Rocket Flight Example (4)







Static Margin and Pitching Moment

Static margin is a concept used to characterize the static stability and controllability of aircraft and missiles.

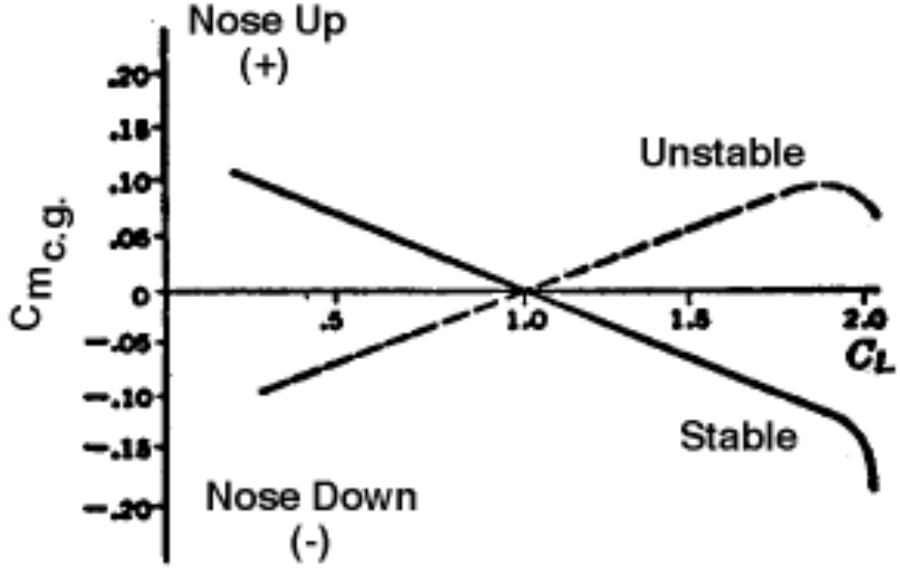
In aircraft analysis, static margin is defined as the non-dimensional distance between center of gravity and aerodynamic center of the aircraft.

In missile analysis, static margin is defined as non-dimensional distance between center of gravity and the center of pressure.

Stability requires that the pitching moment about the rotation point, C_m, become negative as we increase C_L:

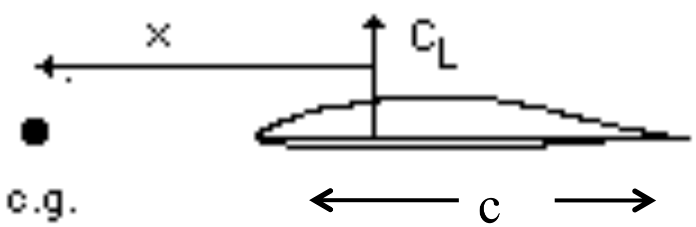
$$\frac{\partial C_m}{\partial C_L} < 0 \rightarrow C_m = C_{m_0} - \frac{x}{c} C_L \rightarrow \frac{\partial C_{m_{c.g.}}}{\partial C_L} = -\frac{x}{c} = -\text{static margin}$$

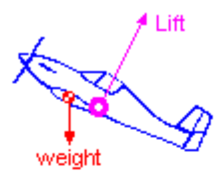
Static Margin and Pitching Moment (2)



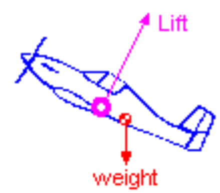
$$\frac{\partial C_{m_{c.g.}}}{\partial C_L} = -\frac{x}{c} = -\text{static margin}$$

For a Rocket Static margin is the distance between the CG and the CP; divided by body tube diameter.





When the CG is ahead of NP the weight tends to correct the upset = Stable



When the CG is behind NP the weight worsens the upset = Unstable

x is the distance from the system's aerodyanmic center to the c.g..

Calculating the Static Margin

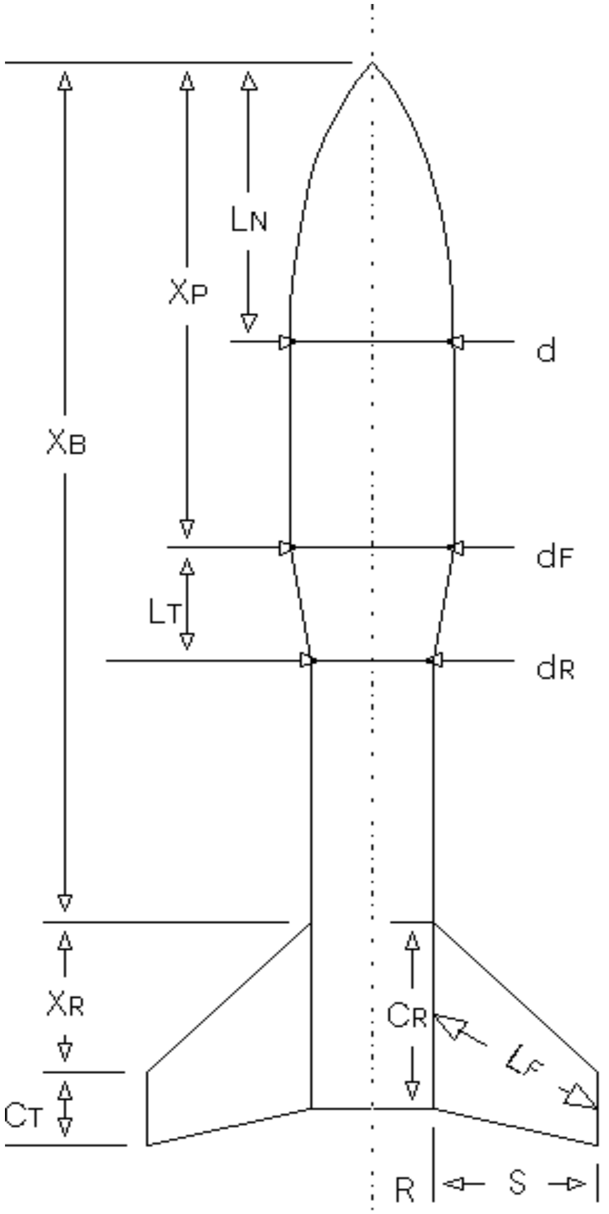
•Key to calculating static margin is to estimate location of longitudinal
•center of pressure at low angles of attack

•Barrowman equations provide simple, accurate technique for Axi-symmetric rockets

•cg is measured as the longitudinal balance point of the rocket.

•As a rule of thumb, CP distance should be aft of the CG by at least one rocket diameter. -- "One Caliber stability”.

Calculating the Static Margin (2)



Parameter Definitions

L_N = length of nose

d = diameter at base of nose

d_F = diameter at front of transition

d_R = diameter at rear of transition

L_T = length of transition

X_P = distance from tip of nose to front of transition

C_R = fin root chord

C_T = fin tip chord

S = fin semispan

L_F = length of fin mid-chord line

R = radius of body at aft end

X_R = distance between fin root leading edge and fin tip leading edge parallel to body

X_B = distance from nose tip to fin root chord leading edge

N = number of fins

Calculating the Static Margin (3)

XN = center of pressure location for nose section

XT = center of pressure location for transitions

(CN)N = normal force coefficient for nose

(CN)T = normal force coefficient for transition

Nose Cone Terms

- $(C_N)_N = 2$
- For Cone: $X_N = 0.666L_N$
- For Ogive: $X_N = 0.466L_N$

Conical Transition Terms

$$(C_N)_T = 2\left[\left(\frac{d_R}{d}\right)^2 - \left(\frac{d_F}{d}\right)^2\right] \rightarrow X_T = X_P + \frac{L_T}{3}\left[1 + \frac{1 - \frac{d_F}{d_R}}{1 - \left(\frac{d_F}{d_R}\right)^2}\right]$$

Calculating the Static Margin (4)

Fin Terms

XF = center of pressure location for fin groups

(CN)F = Normal force coefficient for fin group

$$(C_N)_F = \left[1+\frac{R}{S+R}\right]\left[\frac{4N\left(\frac{S}{d}\right)^2}{1+\sqrt{1+\left(\frac{2L_F}{C_R+C_T}\right)^2}}\right] \longrightarrow X_F = X_B + \frac{X_R}{3}\frac{(C_R+2C_T)}{(C_R+C_T)} + \frac{1}{6}\left[(C_R+C_T) - \frac{(C_R C_T)}{(C_R+C_T)}\right]$$

Static Margin (X_{sm}) =

$(X_{cp} - X_{cg})/D_{max}$

Finding the Center of Pressure

- Sum up coefficients:

$$(C_N)_R = (C_N)_N + (C_N)_T + (C_N)_F$$

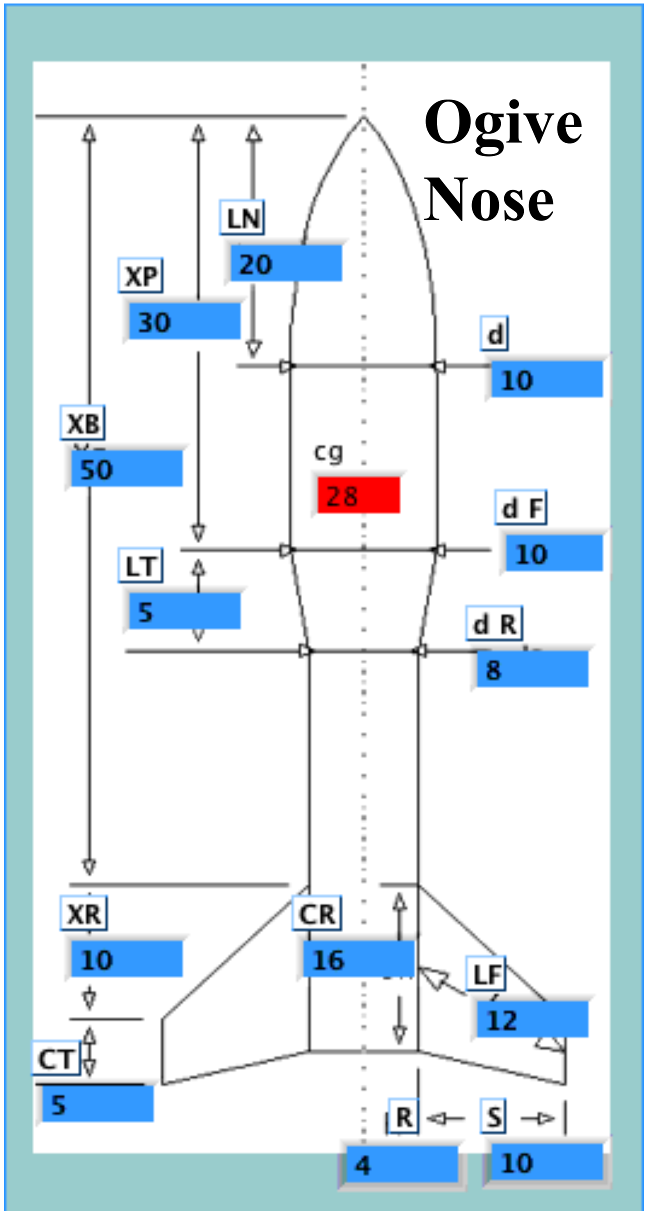
(CN)R = total normal force coefficient

- Find CP Distance from Nose Tip:

$$X_{cp} = \frac{(C_N)_N X_N + (C_N)_T X_T + (C_N)_F X_F}{(C_N)_R}$$

X – measured aft from nose of vehicle

Example: Stable Static Margin Vehicle



L_N = *20 cm*

d = *10 cm*

d_F = *10 cm*

d_R = *8 cm*

L_T = *5 cm*

X_P = *30 cm*

C_R = *10 cm*

C_T = *5 cm*

S = *10 cm*

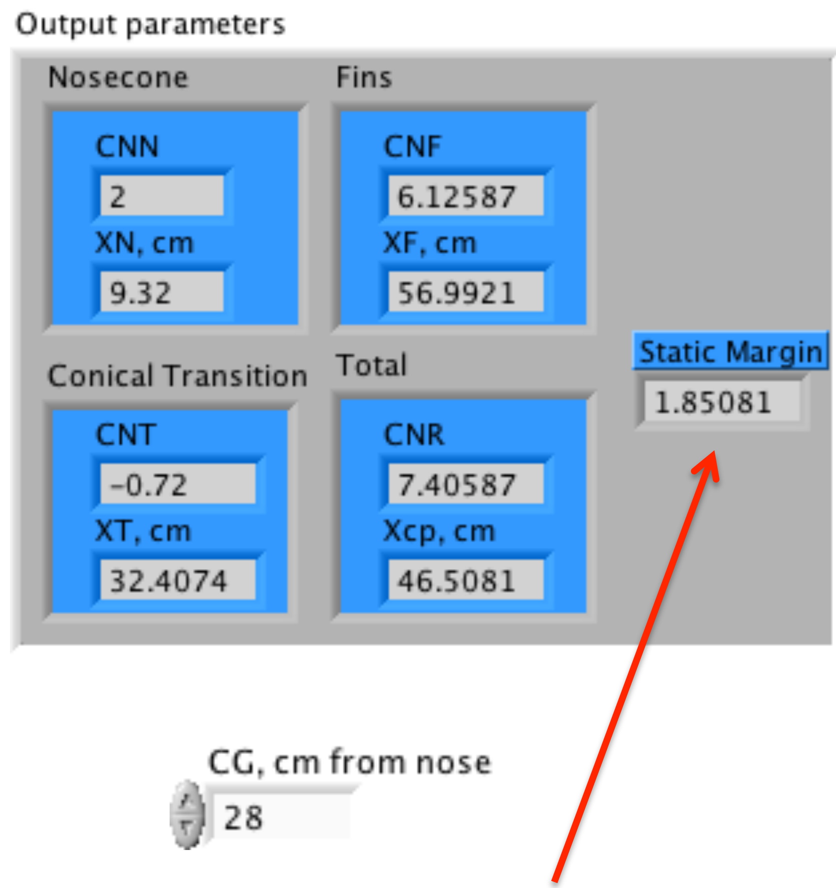
L_F = *12 cm*

R = *4 cm*

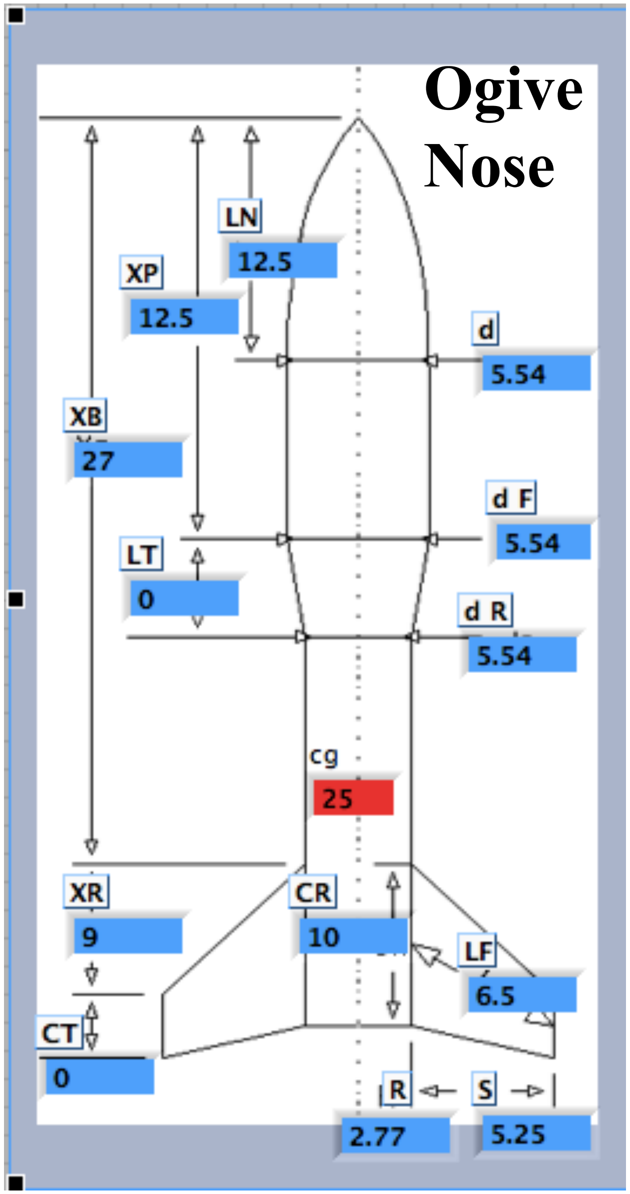
X_R = *16 cm*

X_B = *50 cm*

N = *3*



Example: Unstable Static Margin Calculation



L_N = *12.5 cm*

d = *5.54 cm*

d_F = *5.54 cm*

d_R = *5.54 cm*

L_T = *0 cm*

X_P = *12.5 cm*

C_R = *10 cm*

C_T = *0 cm*

S = *5.25 cm*

L_F = *6.5 cm*

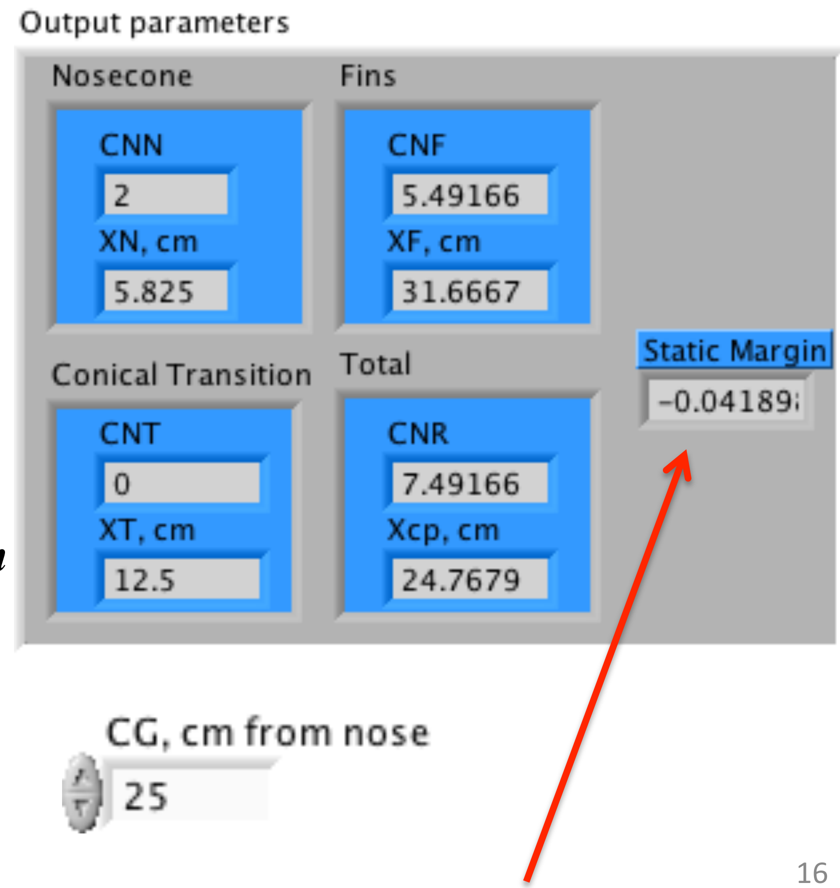
R = *2.77 cm*

X_R = *9 cm*

X_B = *27 cm*

N = *3*

Constant diameter tube (No transition section)



CHIMAERA

Utah State University

Mechanical & Aerospace Engineering

Effect of Static Margin on Launch Conditions

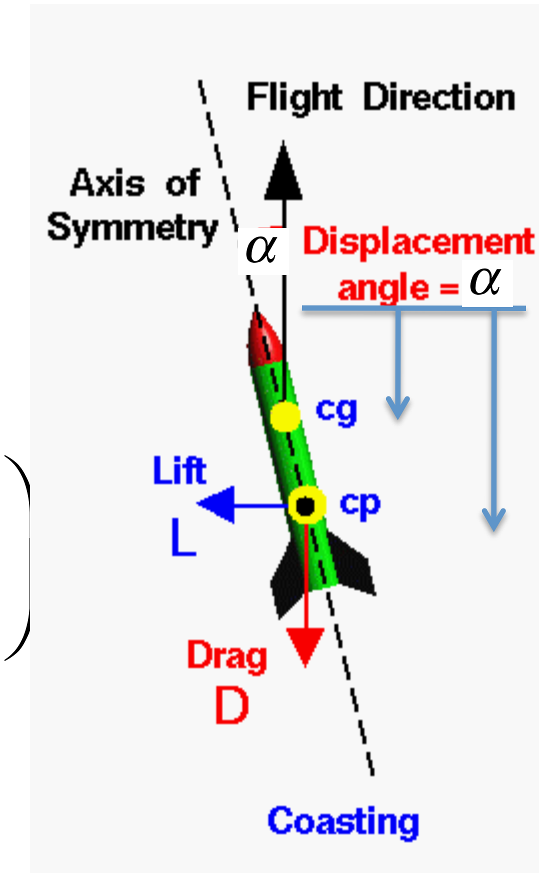
$$C_m = (C_m)_{\alpha=0} - \left(\frac{x_{cp} - x_{cg}}{D_{max}}\right) \cdot C_L$$

$$\rightarrow \frac{x_{cp} - x_{cg}}{D_{\max}} = \text{"}static\ margin\text{"}$$

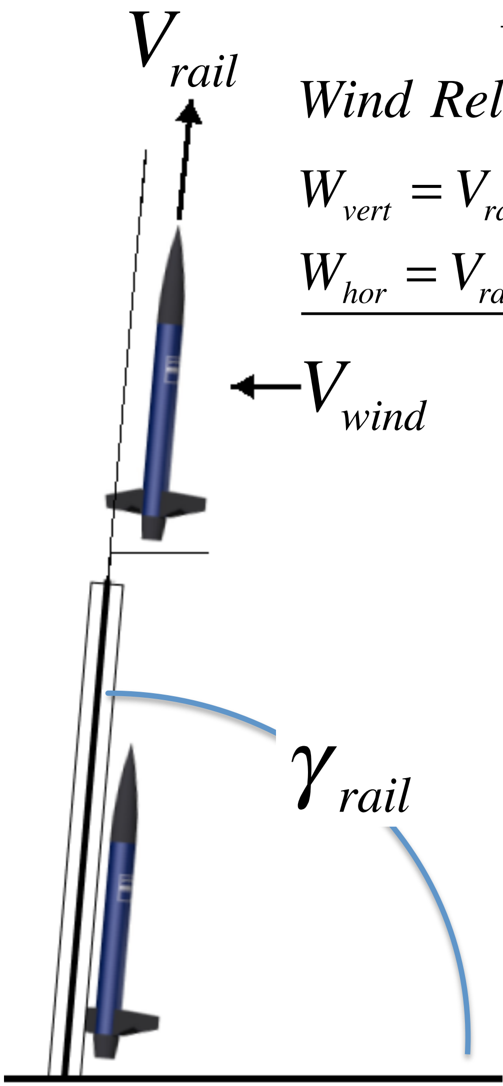
$$Static\ Stability \rightarrow \frac{\partial C_m}{\partial C_L} < 0 \approx -\left(\frac{x_{cp} - x_{cg}}{D_{max}}\right)$$

$$\boxed{1 \leq Desired\ Static\ Margin \leq 2}$$

$$(C_m)_{\alpha=0} \approx 0 ....for\ axi - symmetric\ rocket$$



Effect of Static Margin on Launch Conditions (2)



Wind Relative Velocity :

$$W_{vert} = V_{rail} \cdot \sin\gamma_{rail}$$
$$W_{hor} = V_{rail} \cdot \cos\gamma_{rail} + V_{wind}$$
$$\rightarrow \gamma_{wind} = \tan^{-1}\left(\frac{V_{rail} \cdot \sin\gamma_{rail}}{V_{rail} \cdot \cos\gamma_{rail} + V_{wind}}\right)$$

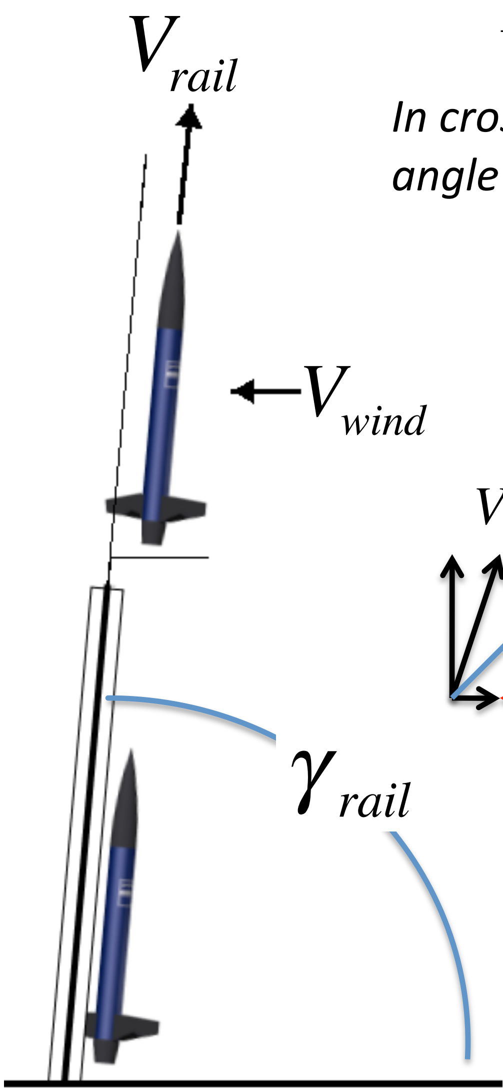
$$\gamma_{wind} = \tan^{-1}\left(\frac{\sin\gamma_{rail}}{\cos\gamma_{rail} + \frac{V_{wind}}{V_{rail}}}\right)$$

$$\alpha_{rail} = \gamma_{rail} - \gamma_{wind}$$

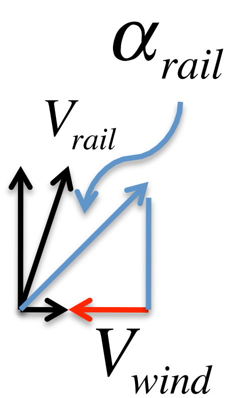
UtahState UNIVERSITY

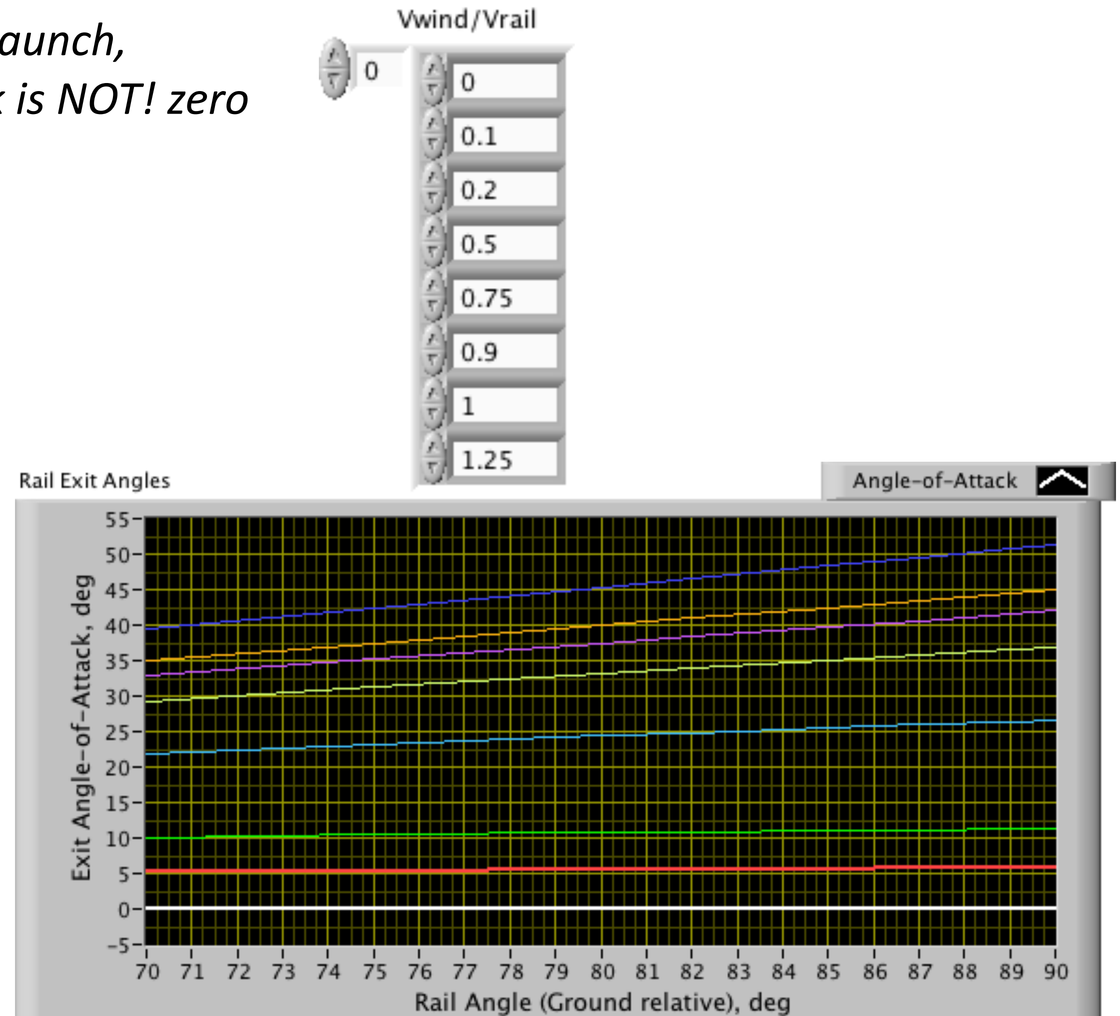
Mechanical & Aerospace Engineering

Effect of Static Margin on Launch Conditions (3)



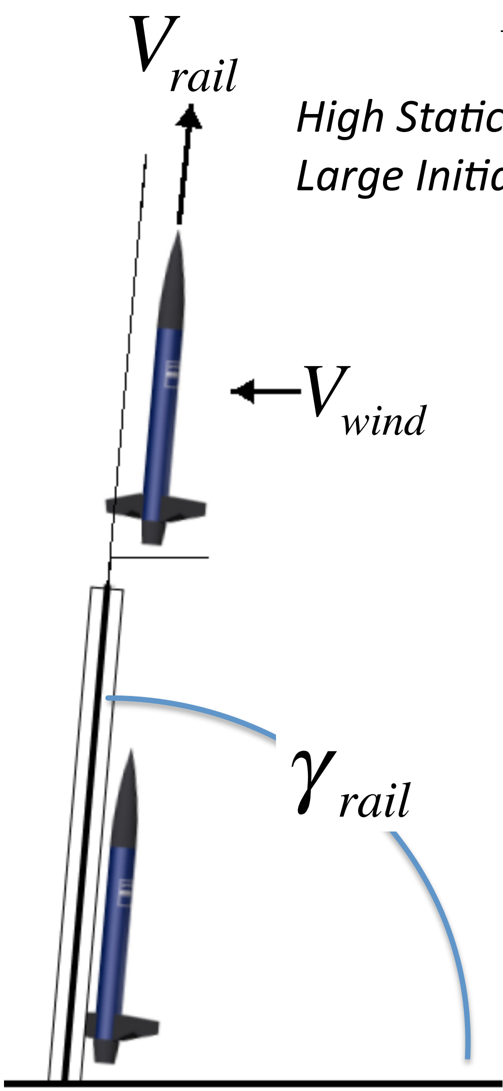
In cross wind Launch,
angle of attack is NOT! zero



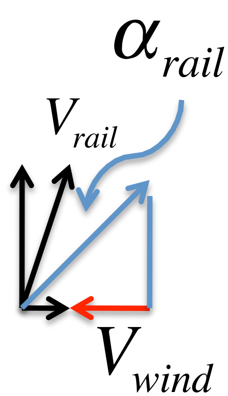


Effect of Static Margin on Launch Conditions (4)

High Static Stability produces Large Initial Pitching Moment



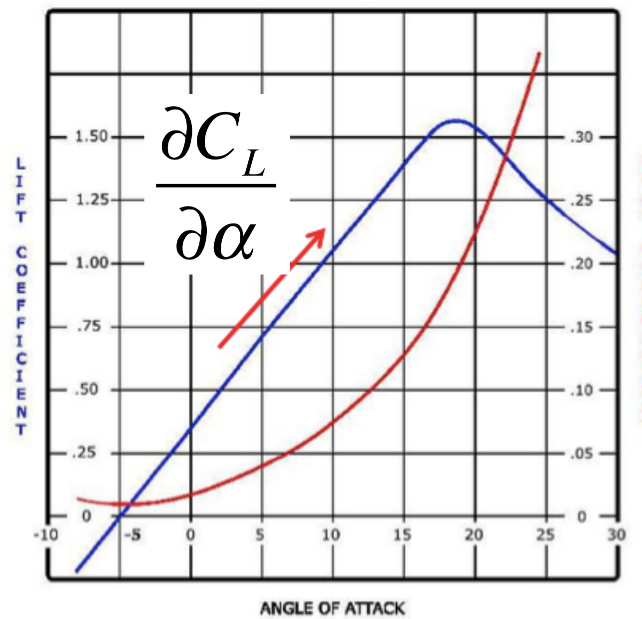
$$\left(C_m\right)_{launch} = \left(C_m\right)_{\alpha=0} - \left(\frac{x_{cp}-x_{cg}}{D_{max}}\right)\cdot C_L \approx -\left(\frac{x_{cp}-x_{cg}}{D_{max}}\right)\cdot\frac{\partial C_L}{\partial\alpha}\cdot\alpha_{rail}$$



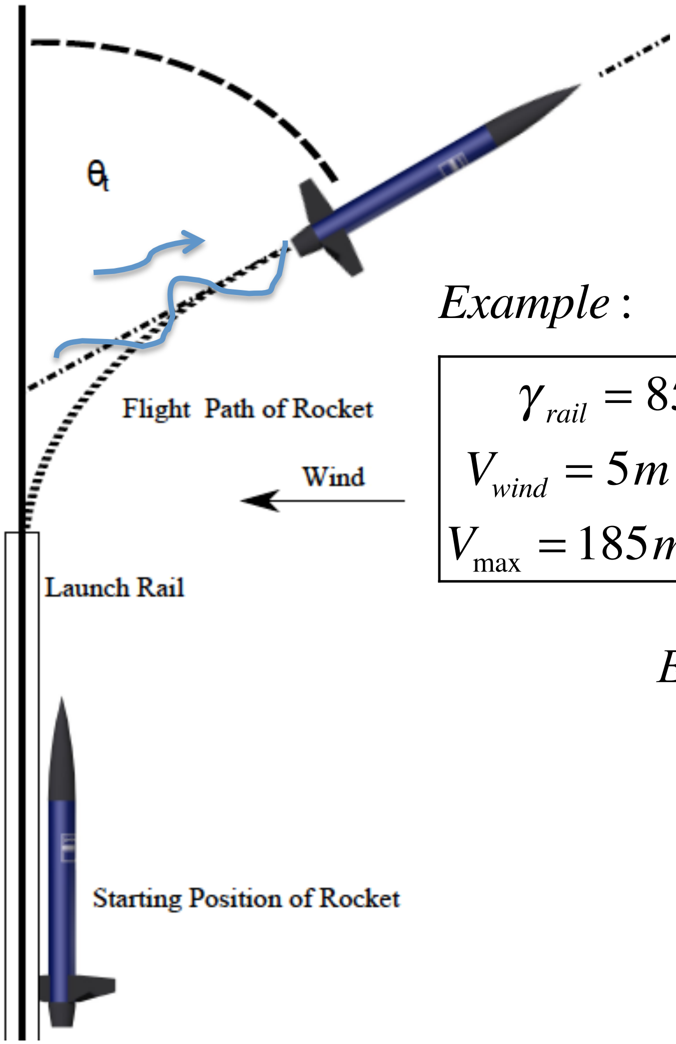
$$C_{L_\alpha} \equiv \frac{\partial C_L}{\partial \alpha}$$

$$\gamma_{wind} = \tan^{-1}\left(\frac{\sin\gamma_{rail}}{\cos\gamma_{rail} + \frac{V_{wind}}{V_{rail}}}\right)$$

$$\alpha_{rail} = \gamma_{rail} - \gamma_{wind}$$



Effect of Static Margin on Launch Conditions (5)



$$\theta_t \approx \gamma_{rail} - \tan^{-1}\left(\frac{\sin\gamma_{rail}}{\cos\gamma_{rail} + \frac{V_{wind}}{V_{\max}}}\right)$$

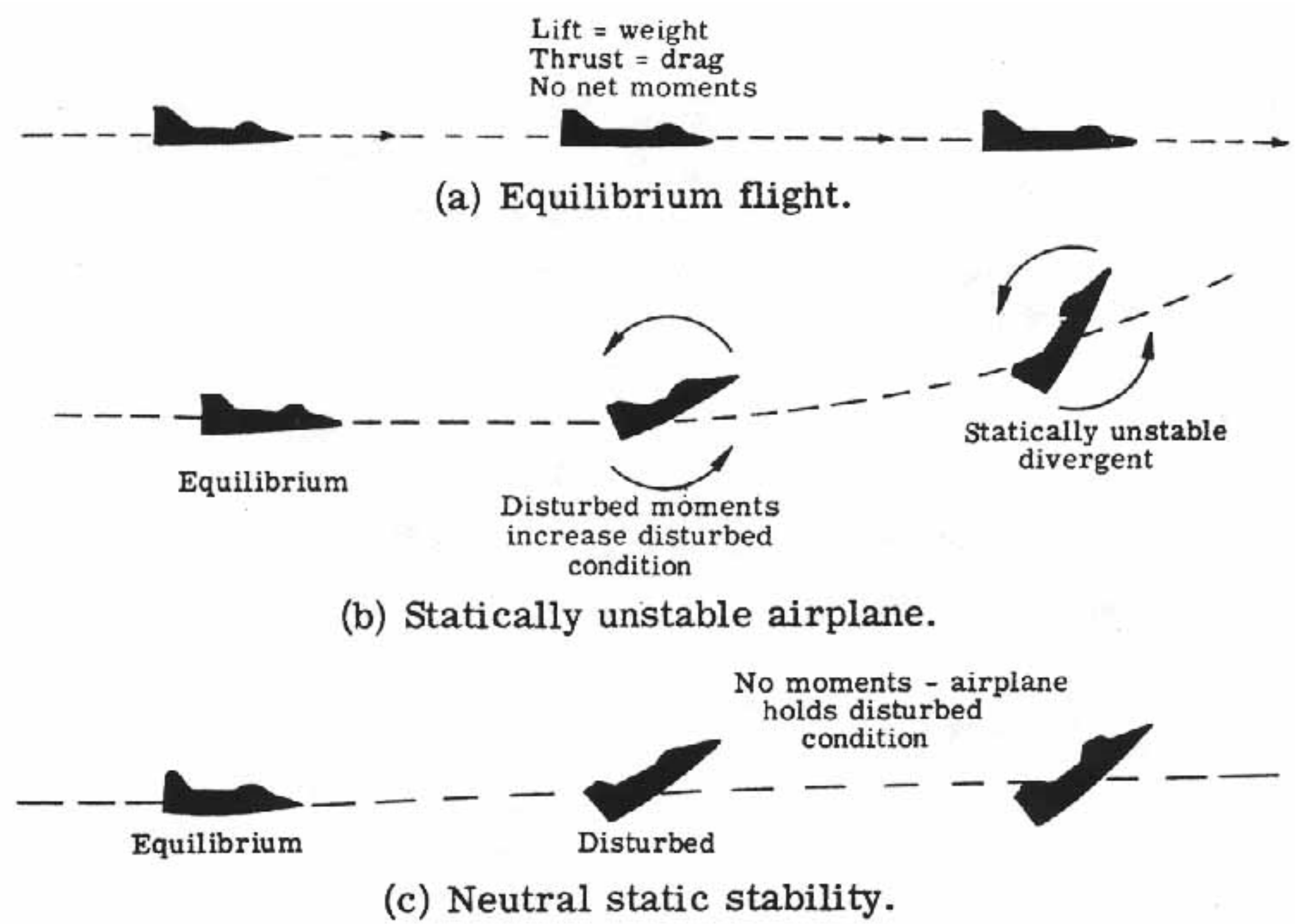
Example :

$\gamma_{rail} = 85^o$
$V_{wind} = 5m/\sec$
$V_{\max} = 185m/\sec$

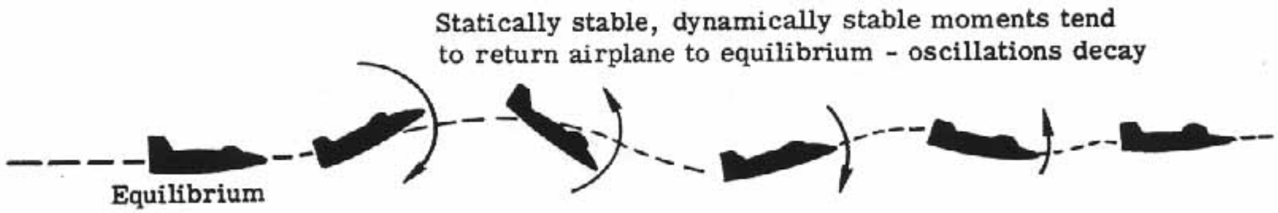
$$85 - \frac{180}{\pi}\,\text{atan}\left(\frac{\sin\left(\frac{\pi}{180}85\right)}{\cos\left(\frac{\pi}{180}85\right) + \frac{5}{185}}\right) \geq 1.54^o$$

Effective Launch Angle $\leq 83.46^o$

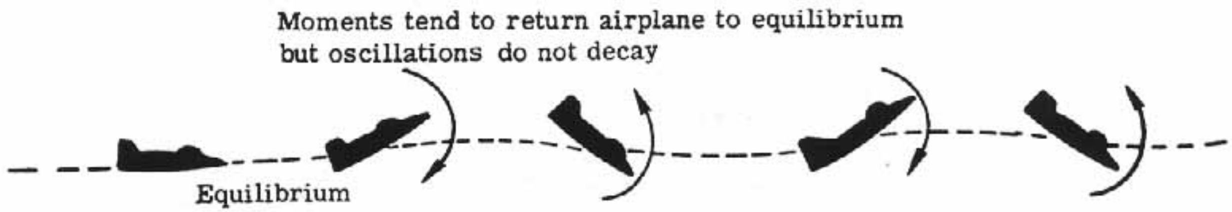
Static Versus Dynamic Stability



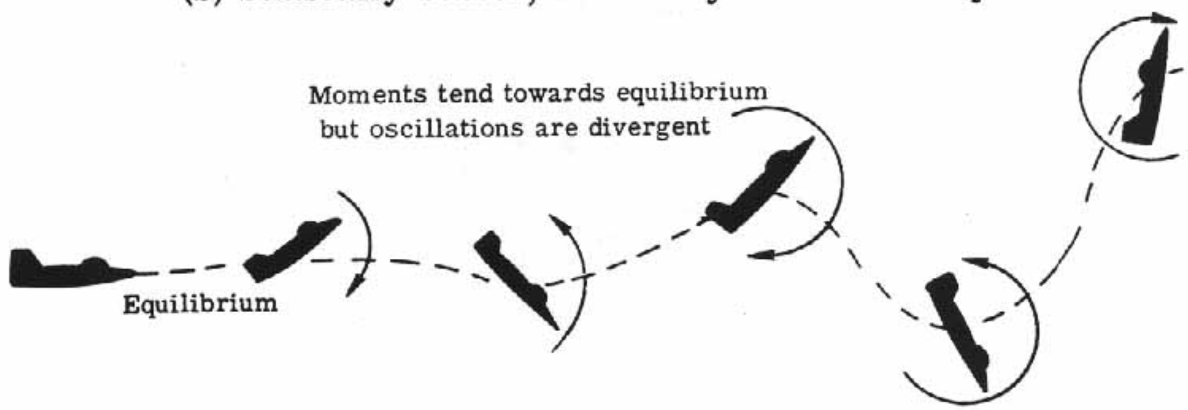
Static Versus Dynamic Stability (2)



(a) Statically and dynamically stable.

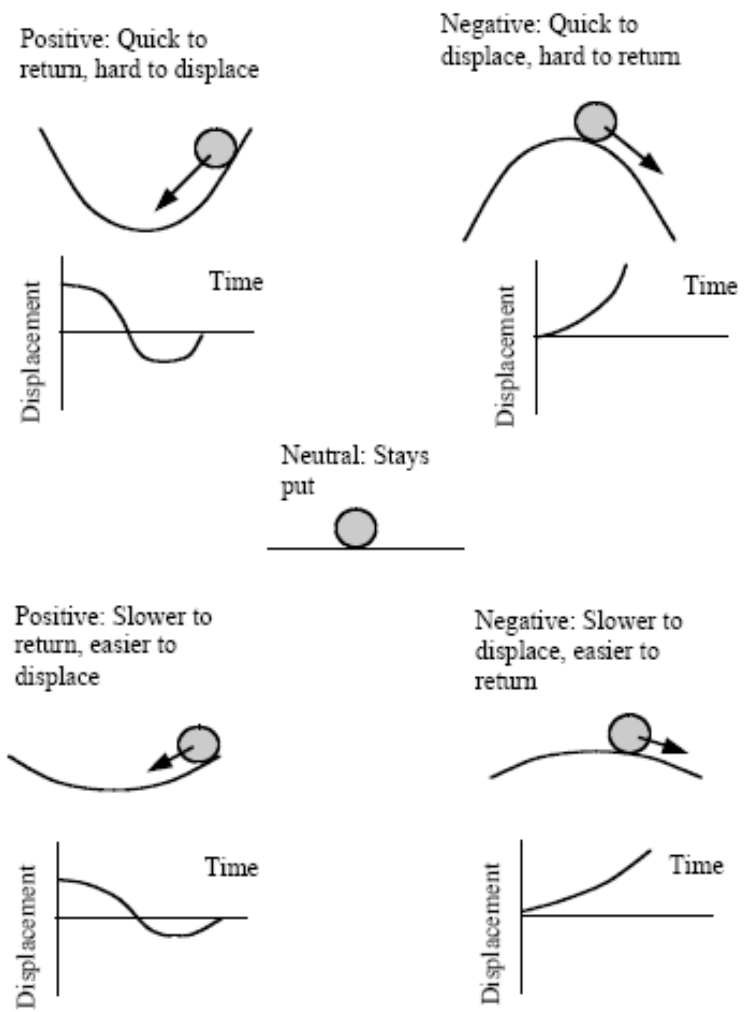


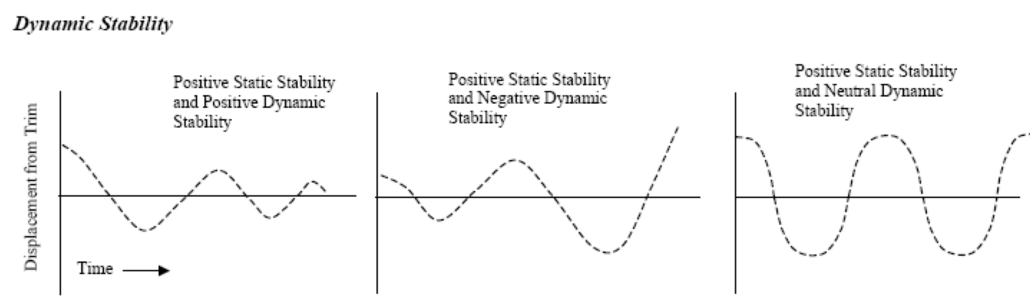
(b) Statically stable; neutral dynamic stability.

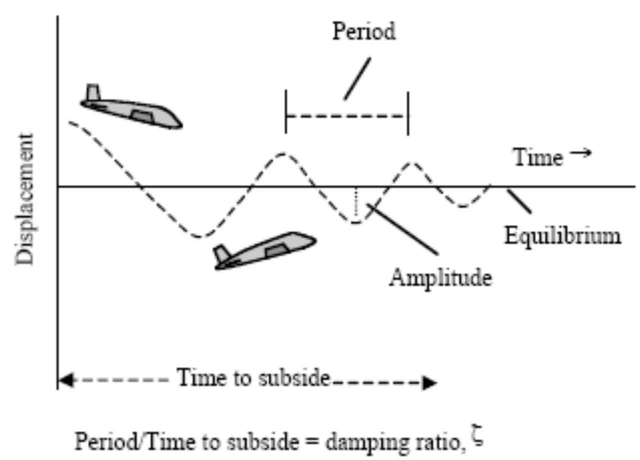


(c) Statically stable; dynamically unstable.

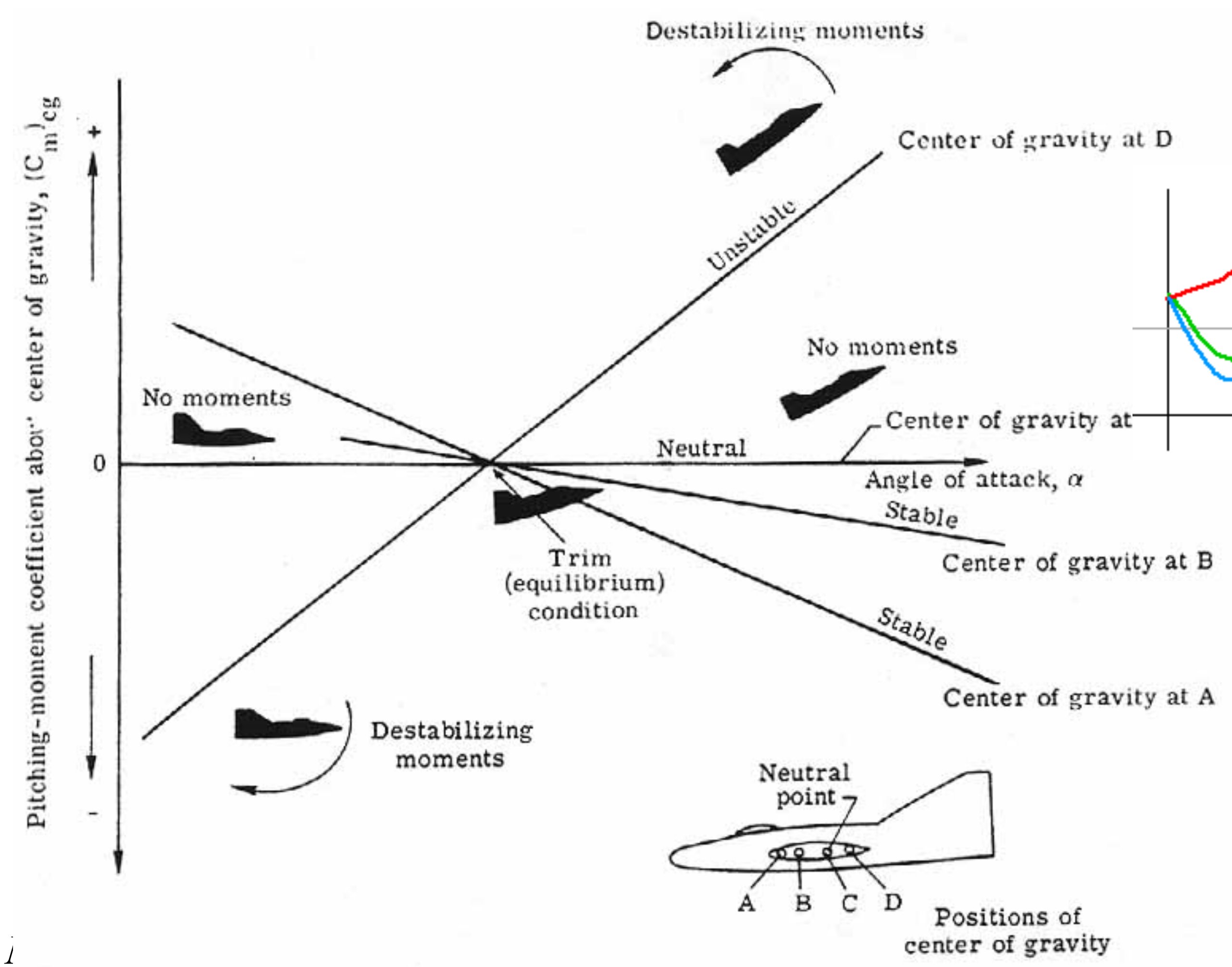
Static Versus Dynamic Stability (3)

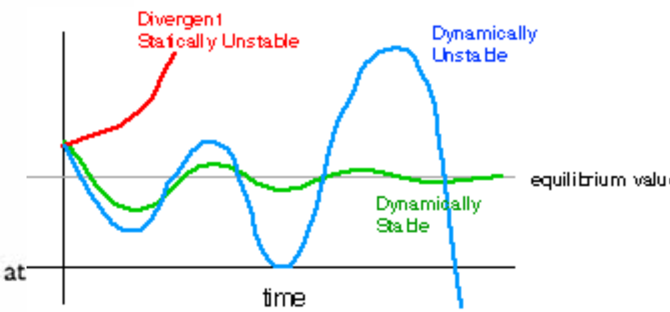






Static Versus Dynamic Stability (4)





Simplified Pitch Axis-Rotational Dynamics

Neglect Cross Products of Inertia

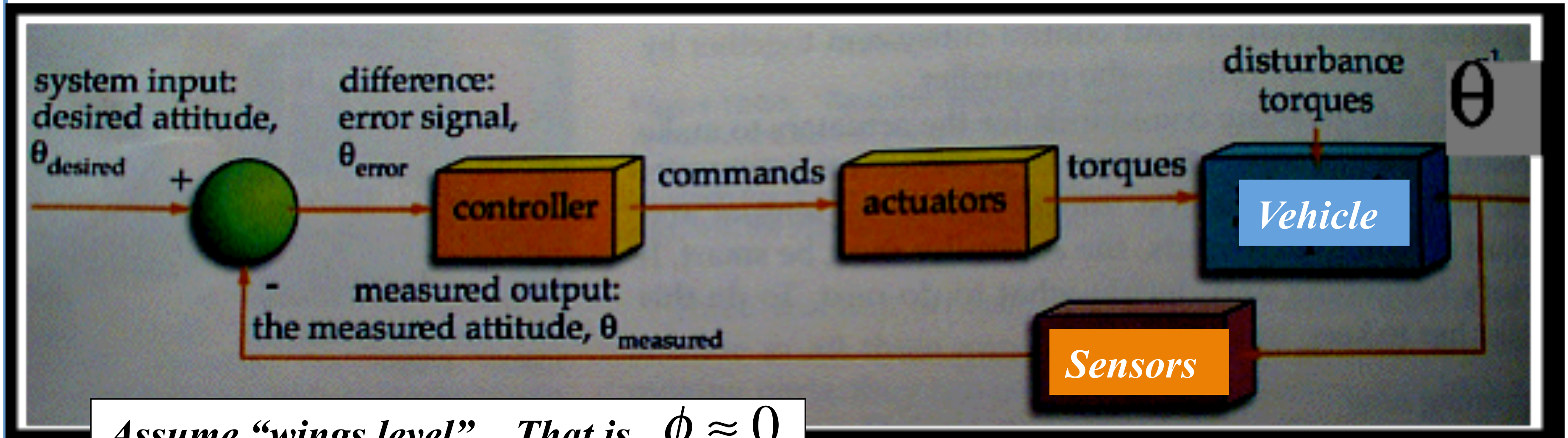
$$\begin{bmatrix} I_x & -I_{xy} & -I_{xz} \\ -I_{xy} & I_y & -I_{yz} \\ -I_{xz} & -I_{yz} & I_z \end{bmatrix} \cdot \begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} = \begin{pmatrix} q \cdot r\left(I_y - I_z\right) + \left(q^2 - r^2\right) I_{yz} + p \cdot q\left(I_{xz}\right) - r \cdot p\left(I_{xy}\right) \\ r \cdot p\left(I_z - I_x\right) + \left(r^2 - p^2\right) I_{xz} + q \cdot r\left(I_{xy}\right) - p \cdot q\left(I_{yz}\right) \\ p \cdot q\left(I_x - I_y\right) + \left(p^2 - q^2\right) I_{xy} + r \cdot p\left(I_{yz}\right) - q \cdot r\left(I_{xz}\right) \end{pmatrix} + \begin{bmatrix} M_x \\ M_y \\ M_z \end{bmatrix}$$

$$\rightarrow \boxed{\dot{q} = \ddot{\theta} = \frac{M_y}{I_y} + r \cdot p \frac{\left(I_z - I_x\right)}{I_y}}$$

Forcing moment

Second order Disturbance torque (neglected when r, p are small)

Simplified Pitch Axis-Rotational Dynamics (2)



Assume "wings level" .. That is $\phi \approx 0$

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{pmatrix} 1 & \sin\phi\tan\theta & \cos\phi\tan\theta \\ 0 & \cos\phi & -\sin\phi \\ 0 & \dfrac{\sin\phi}{\cos\theta} & \dfrac{\cos\phi}{\cos\theta} \end{pmatrix} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \rightarrow \begin{bmatrix} \dot{\theta} = q \\ \ddot{\theta} = \dot{q} \end{bmatrix}$$

Simplified Pitch Axis-Rotational Dynamics (3)

Neglecting Disturbance torques

$$\ddot{\theta} = \dot{q} = \frac{M_y}{I_y} \rightarrow M_y \approx \text{"}\mathit{pitching\ moment}\text{"}$$

$$\text{"}\mathit{pitching\ moment\ coefficient}\text{"} \equiv C_m = \frac{M_y}{\bar{q} \cdot A_{ref} \cdot c_{ref}}$$

$$\bar{q} = \left(\frac{1}{2} \cdot \rho \cdot V^2 \right)$$

$$A_{ref} = \frac{\pi}{4} \cdot D_{ref}^{\ 2}$$

$$\text{"}\mathit{reference\ length}\text{"} \rightarrow c_{ref} = D_{\max}$$

Check Units

$$\frac{\bar{q} \cdot A_{ref} \cdot c_{ref}}{I_y} \sim \frac{\frac{Nt}{m^2} \cdot m^2 \cdot m}{kg \cdot m^2} \sim$$

$$\left(\frac{kg - m}{\sec^2 m^2} \cdot m^2 \cdot m \right) \cdot \frac{1}{kg \cdot m^2} \sim \frac{1}{\sec^2}$$

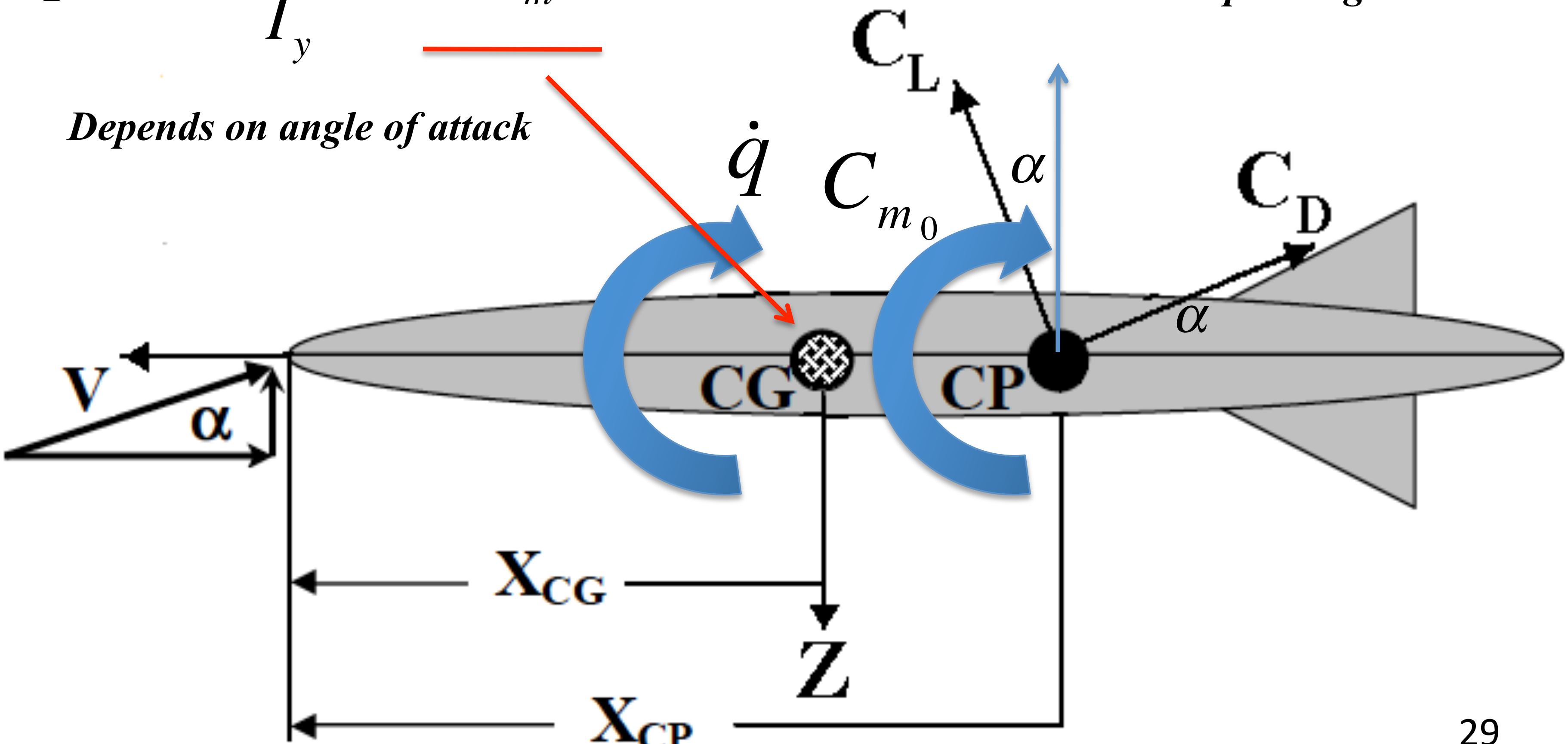
Simplified Pitch Axis-Rotational Dynamics (4)

Pitching moment about cg

$$\dot{q} = \frac{\bar{q} \cdot A_{ref} \cdot c_{ref}}{I_y} \cdot C_m$$

Depends on angle of attack

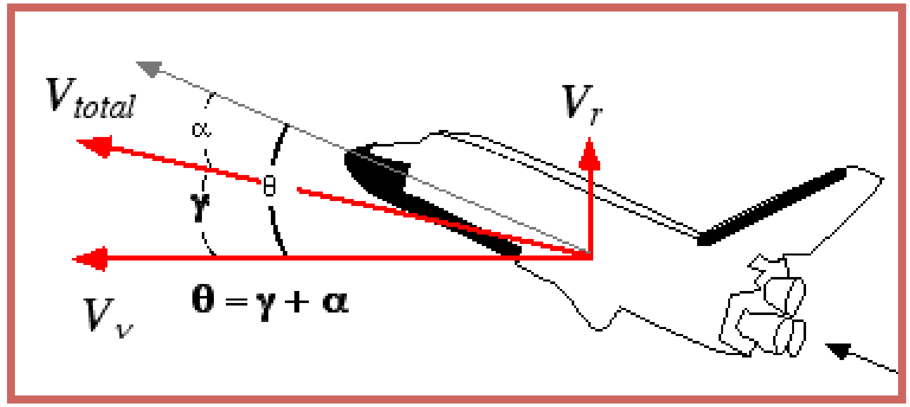
Gravity acts at CG and Cannot induce pitching moment



Simplified Pitch Axis-Rotational Dynamics (4)

$$\ddot{\theta} = \dot{q} = \frac{\bar{q} \cdot A_{ref} \cdot c_{ref}}{I_y} \cdot C_m$$

Depends on angle of attack + Control inputs



It can also be shown that … (NASA RP-1168 pp 10-22) that for angle of attack

$$\dot{\alpha} = -\frac{\bar{q} \cdot A_{ref}}{m \cdot V} \cdot C_L + q + \frac{g_{(r)}}{V} \cos(\theta - \alpha) - \frac{F_{thrust} \sin\alpha}{m \cdot V}$$

Simplified Pitch Axis-Rotational Dynamics (5)

Collected, simplified pitch dynamics equations

$$\begin{bmatrix} \dot{\alpha} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} -\frac{\bar{q} \cdot A_{ref}}{m \cdot V} \cdot C_L + q + \frac{g_{(r)}}{V} \cos(\theta - \alpha) - \frac{F_{thrust} \sin\alpha}{m \cdot V} \\ \frac{\bar{q} \cdot A_{ref} \cdot c_{ref}}{I_y} \cdot C_m \\ q \end{bmatrix}$$

Collected Longitudinal Equations of Motion

$$\begin{bmatrix} \dot{V}_r \\ \dot{V}_v \\ \dot{r} \\ \dot{v} \\ \dot{x} \\ \dot{\alpha} \\ \dot{q} \\ \dot{\theta} \\ \dot{m} \end{bmatrix} = \begin{bmatrix} \left(\frac{V_v{}^2}{r}\right) + \left(\frac{\bar{q}\cdot A_{ref}}{m}\right)\cdot\left(C_L\cos\gamma - C_D\sin\gamma\right) + \left(\frac{F_{thrust}\sin\theta}{m}\right) - g_{(r)} \\ -\left(\frac{V_r\cdot V_v}{r}\right) - \left(\frac{\bar{q}\cdot A_{ref}}{m}\right)\cdot\left(C_L\sin\gamma + C_D\sin\gamma\right) + \left(\frac{F_{thrust}\cos\theta}{m}\right) \\ V_r \\ \frac{V_v}{r} \\ V_v \\ -\frac{\bar{q}\cdot A_{ref}}{m\cdot V}\cdot C_L + q + \frac{g_{(h)}}{V}\cos(\gamma) - \frac{F_{thrust}\sin\alpha}{m\cdot V} \\ \frac{\bar{q}\cdot A_{ref}\cdot c_{ref}}{I_y}\cdot C_m \\ q \\ -\frac{F_{thrust}}{g_0\cdot I_{sp}} \end{bmatrix} \rightarrow$$

$$g_{(r)} = \frac{\mu}{r^2}$$

$$\gamma = \tan^{-1}\frac{V_r}{V_v} = \theta - \alpha$$

$$\dot{x} = r\cdot\dot{v}$$

$$V = \sqrt{V_r{}^2 + V_v{}^2}$$

$$\bar{q} = \frac{1}{2}\cdot\rho_{(h)}\cdot V^2$$

$$h = r - R_{earth}$$

How do we control this "mess"

.. With ***"F_{thrust}, C_m, C_L, and C_D"***

Fixed Wing Aircraft Controls

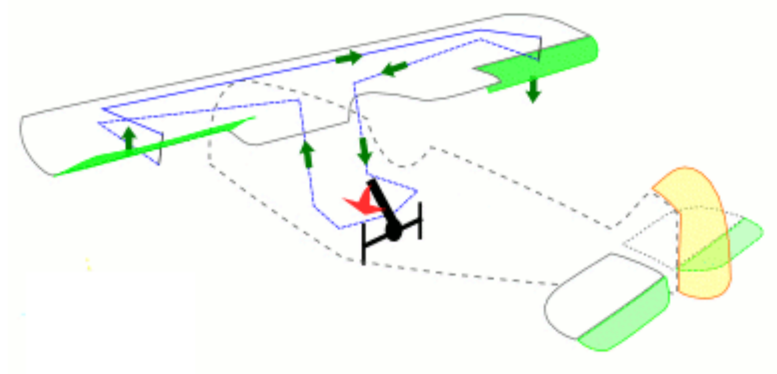
-- Stick Roll Control (ailerons)

-- Stick Pitch Control (elevator)

-- Pedal Yaw Control (Rudder)

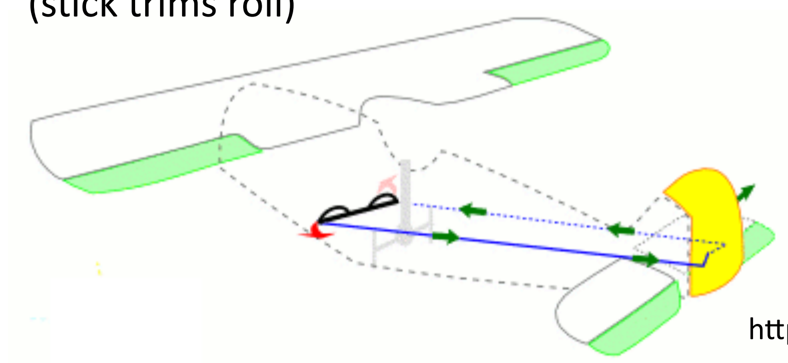
-- Throttle (Thrust)

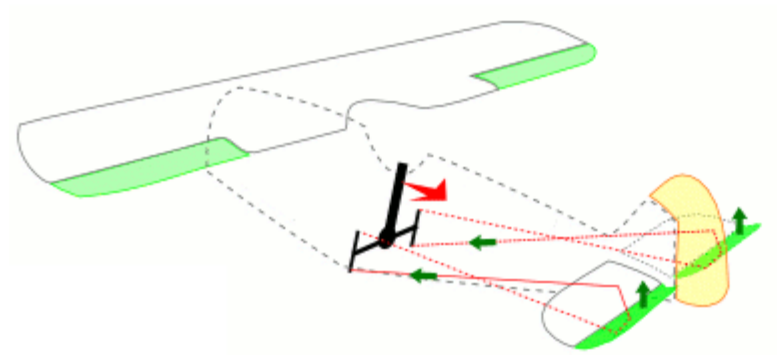
Roll Control From Ailerons (rudder "coordinates" turn)



Pitch Control From Elevator (throttle "coordinates" climb)

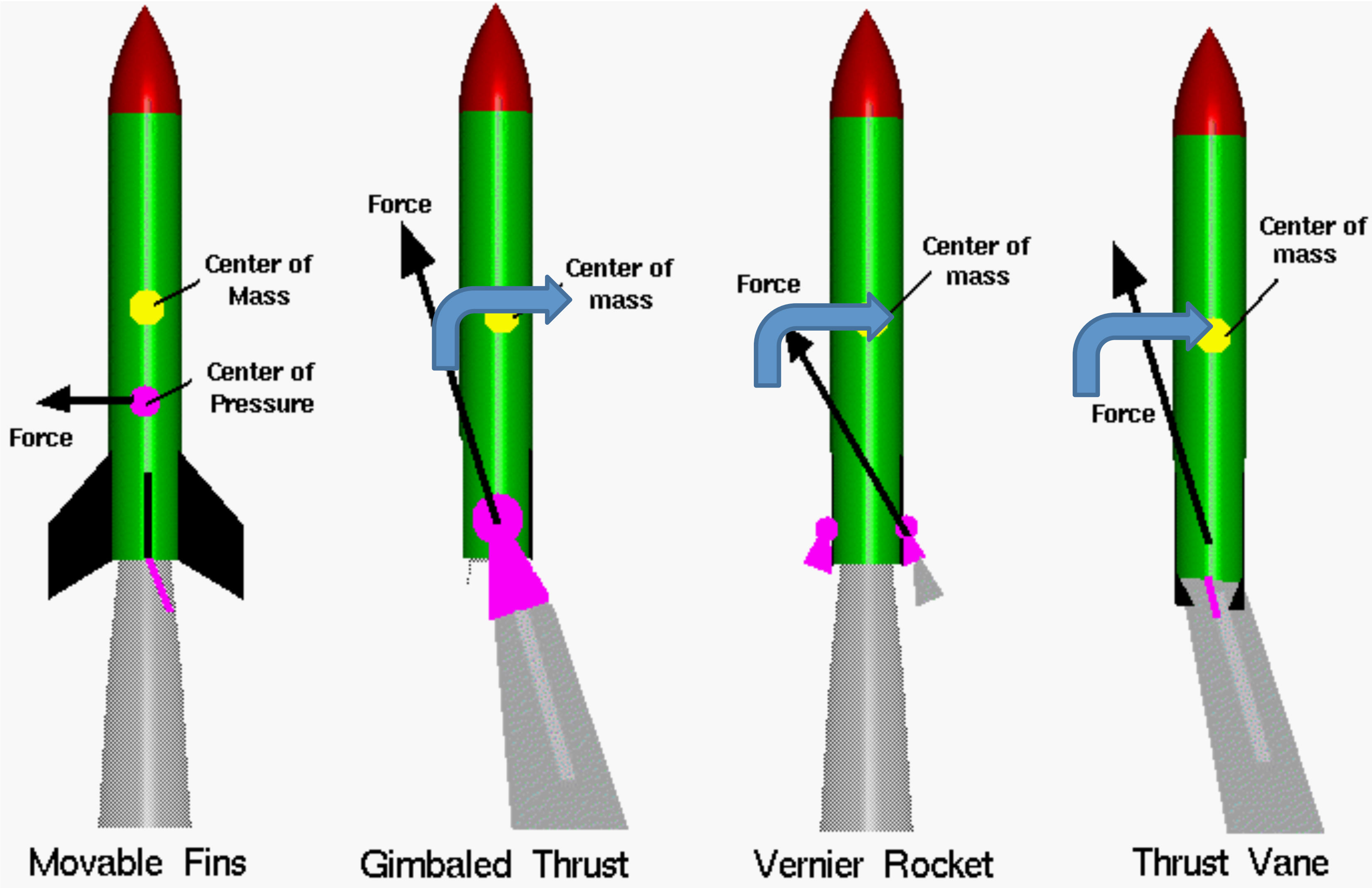
Yaw Control From Rudder
(stick trims roll)





http://en.wikipedia.org/wiki/File:ControlSurfaces.gif

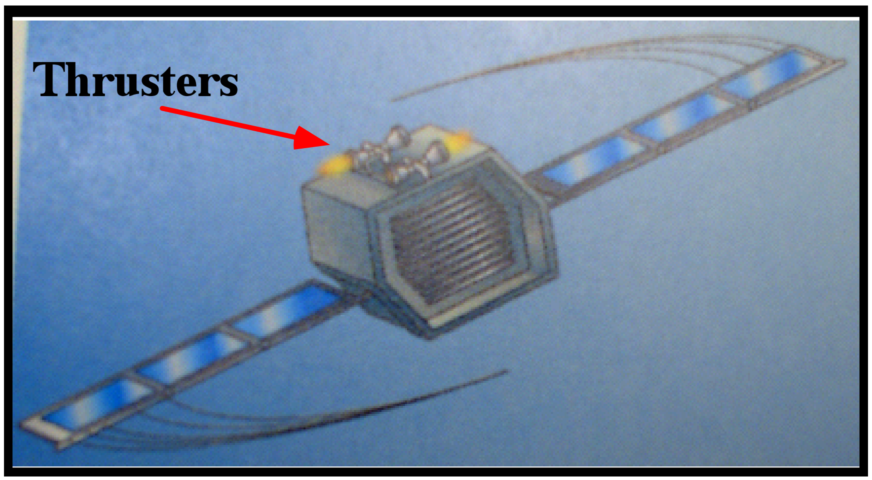
Launch (Rocket) Controls



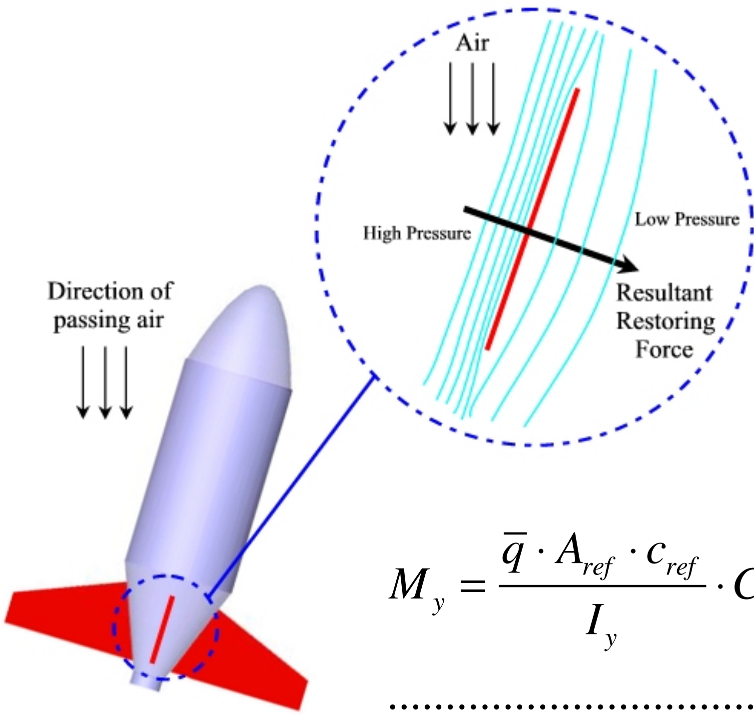
Spacecraft Reaction Controls (RCS)

- **The spacecraft propulsion system provides controlled impulse for:**
  - Orbit insertion and transfers
  - Orbit maintenance (station keeping)
  - Attitude Control
- **Propulsion Types**
  - Cold gas, monopropellant, bipropellants, ion

Thruster rockets apply force at some distance away from center of mass, causing a torque that rotates the spacecraft



Pitching Moment Control of Vehicle

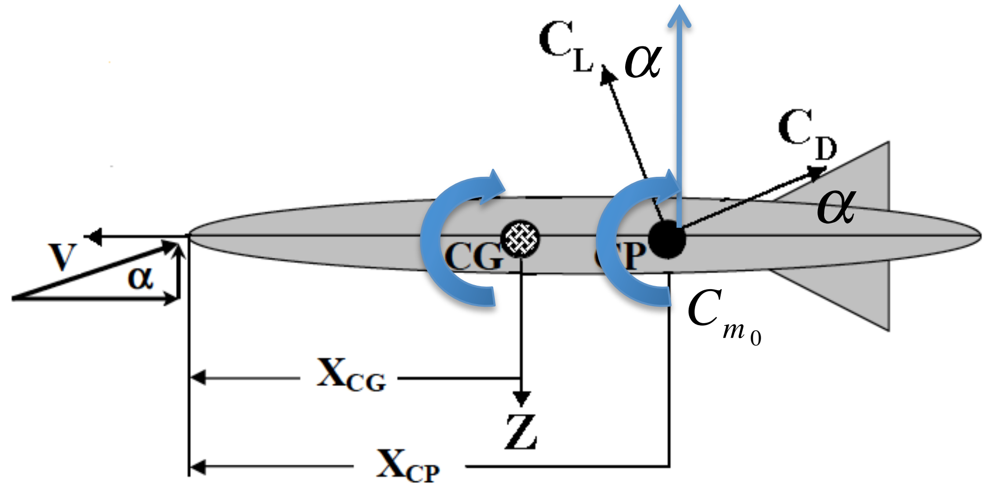


$$\begin{bmatrix}\dot{\alpha}\\ \dot{q}\\ \dot{\theta}\end{bmatrix} = \begin{bmatrix} -\frac{\bar{q}\cdot A_{ref}}{m\cdot V}\cdot C_L + q + \frac{g_{(r)}}{V}\cos(\theta-\alpha) - \frac{F_{thrust}\sin\alpha}{m\cdot V} \\ \frac{\bar{q}\cdot A_{ref}\cdot c_{ref}}{I_y}\cdot C_m \\ q \end{bmatrix}$$

$$M_y = \frac{\bar{q}\cdot A_{ref}\cdot c_{ref}}{I_y}\cdot C_m = \frac{\bar{q}\cdot A_{ref}\cdot c_{ref}}{I_y}\cdot\left[C_m(\alpha) + C_m(\delta)\right]$$

....*depends on* α........*depends on control* "δ"

Vehicle "Aerodynamic" Pitching Moment



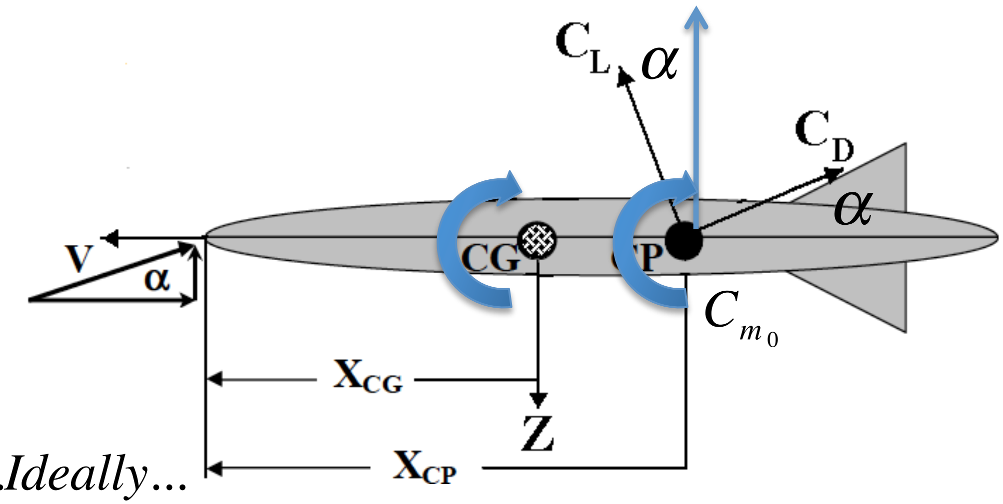
$$C_m(\alpha) = C_{m_0} + \frac{\left(X_{cg} - X_{cp}\right)}{c_{ref}}\left(C_L \cdot \cos\alpha + C_D \cdot \sin\alpha\right) = C_{m_0} - X_{sm}\left(C_L \cdot \cos\alpha + C_D \cdot \sin\alpha\right)$$

$$Small\ \alpha \rightarrow C_m(\alpha) = C_{m_0} - X_{sm}\left(C_L + C_D\alpha\right) \rightarrow \frac{\partial C_m}{\partial\alpha} = -X_{sm}\left(\frac{\partial C_L}{\partial\alpha} + \frac{\partial C_D}{\partial\alpha}\alpha + C_D\right)$$

$$Linearized \rightarrow \boxed{C_m(\alpha) = C_{m_0} - X_{sm}\left(\frac{\partial C_L}{\partial\alpha} + C_D\right)\cdot\alpha \equiv C_{m_0} + \frac{\partial C_m}{\partial\alpha}\cdot\alpha}$$

$$\frac{\partial C_m}{\partial\alpha} \rightarrow "C_{m_\alpha}"$$

Vehicle "Aerodynamic" Pitching Moment (2)



$$\frac{\partial C_m}{\partial \alpha} \rightarrow "C_{m_\alpha}" \text{.....} \textit{Ideally...}$$

$$C_{m_\alpha} = -X_{sm}\left(\frac{\partial C_L}{\partial \alpha} + C_D\right) \rightarrow \left[\begin{matrix}\frac{\partial C_L}{\partial \alpha} \\ C_D\end{matrix}\right] > 0 \rightarrow X_{sm} > 0 \ldots \underline{\textit{static stability}} \rightarrow \boxed{C_{m_\alpha} < 0 \ldots \underline{\textit{static stability}}}$$

Linear Aerodynamic Model

$$\textit{Linear Aerodynamic Model}$$

$$\textit{Lift Coefficient} \rightarrow C_L = C_{L_0} + C_{L_\alpha} \cdot \alpha + C_{L_\delta} \cdot \delta$$

$$\textit{Drag Coefficient} \rightarrow C_D = C_{D_0} + C_{D_\alpha} \cdot \alpha + C_{D_\delta} \cdot \delta$$

$$\textit{Pitching Moment Coefficient} \rightarrow C_m = C_{m_0} + C_{m_\alpha} \cdot \alpha + C_{m_\delta} \cdot \delta$$

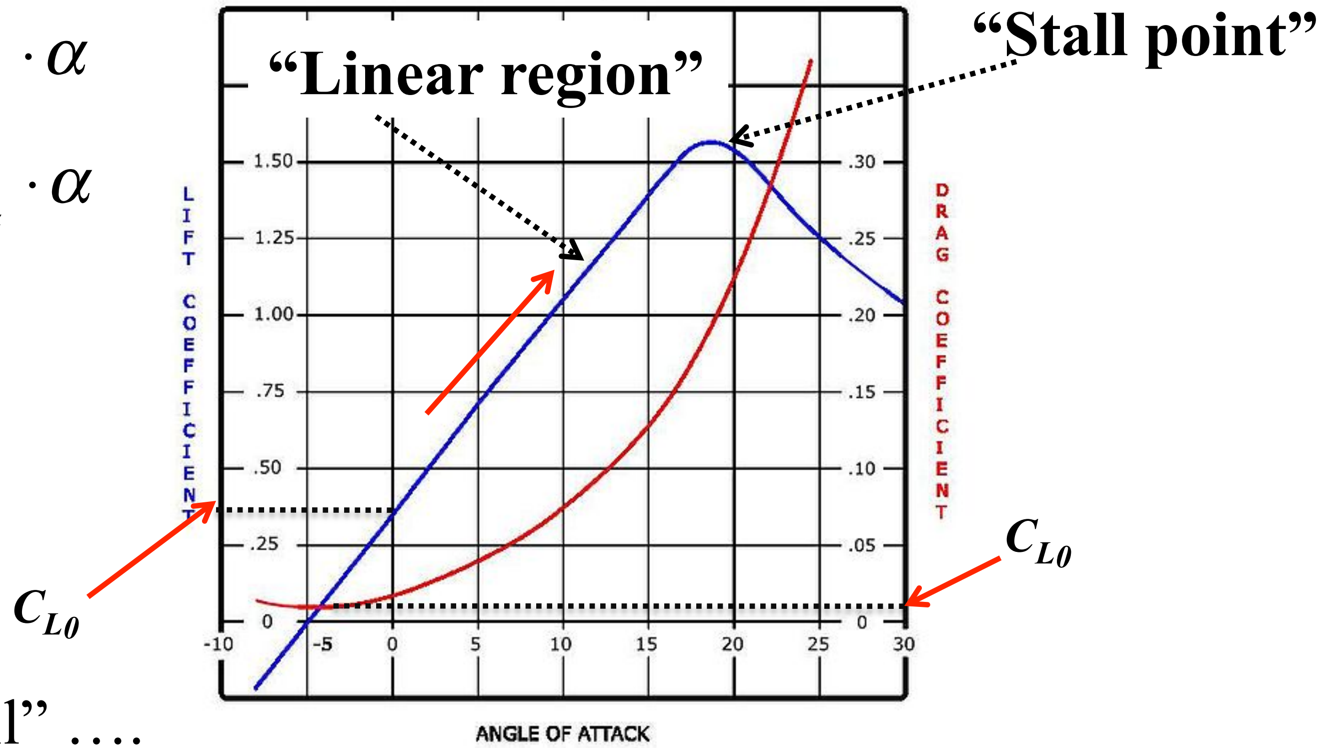
$$\textit{For Rocket (Typically)} \rightarrow \begin{array}{l} \{C_{L_0} \quad C_{m_0}\} \approx 0 \\ C_{m_0} \neq 0 \end{array}$$

$$\delta \rightarrow \textit{Longitudinal Control Actuator}$$

Linear Aerodynamic Model (2)

$$C_L = C_{L_0} + C_{L_\alpha} \cdot \alpha$$

$$C_D = C_{D_0} + C_{D_\alpha} \cdot \alpha$$



For "Thin Airfoil" ….

$$C_D \approx C_{D_0} + \frac{C_L^{\,2}}{\pi \cdot \varepsilon \cdot AR}$$

$$0.85 < \varepsilon < 0.95$$

→

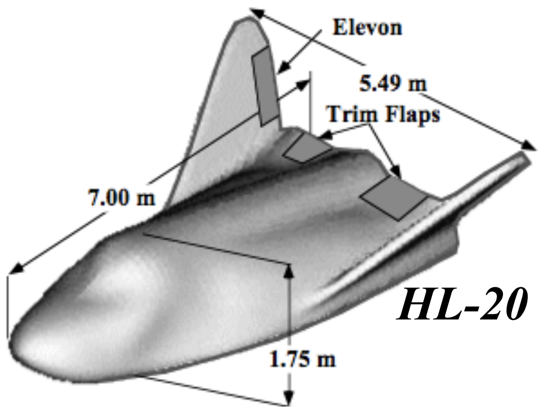
C_{D_0} = *parasite drag*

ε = *Oswald Efficiency Factor*

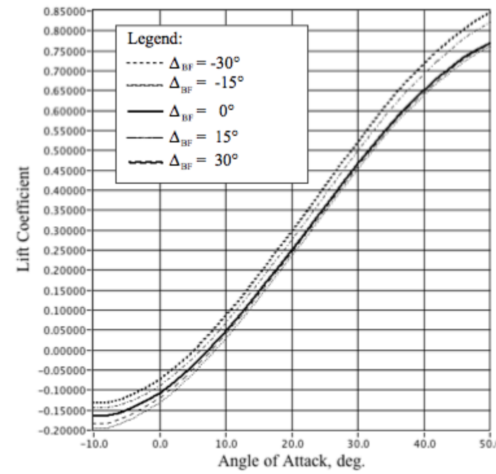
AR → *Aspect Ratio* $= b^2 / A_{ref}$

Supersonic flow also has "wave drag"
… But we won't worry about that here

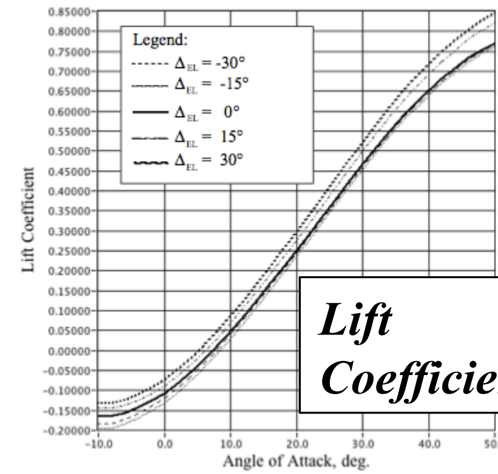
Examples of Aerodynamic Coefficients



Lift Coefficient

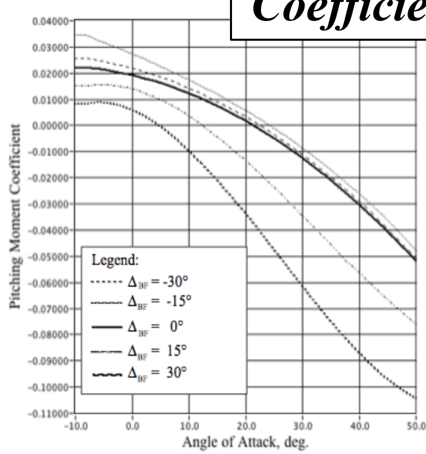


a) Effect of Symmetric Body Flap on Lift Coefficient

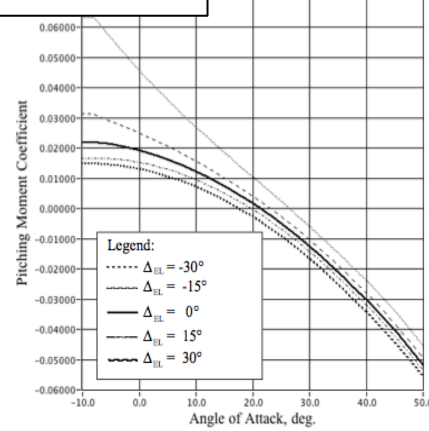


b) Effect of Symmetric Elevon on Lift Coefficient

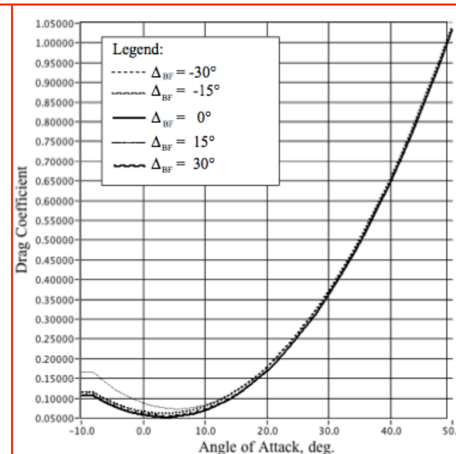
Pitching Moment Coefficient



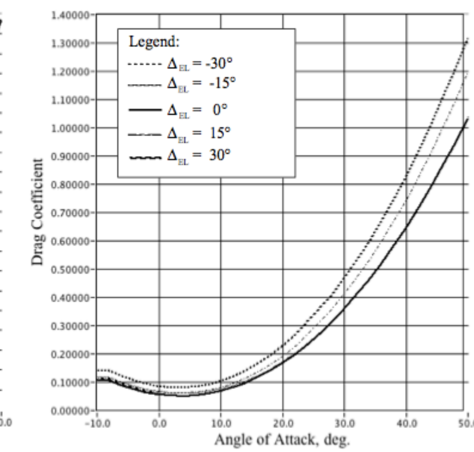
a) Effect of Symmetric Body Flap on Pitching Moment Coefficient



b) Effect of Symmetric Elevon on Pitching Moment Coefficient



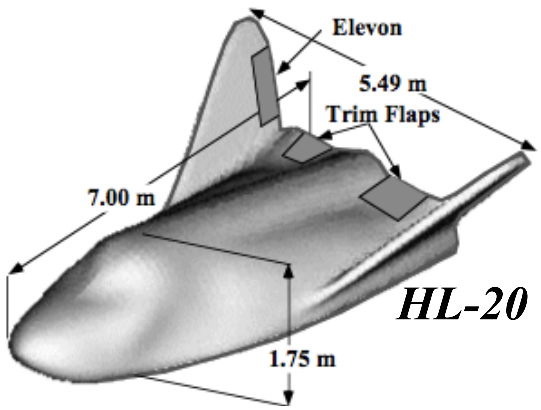
a) Effect of Symmetric Body Flap on Drag Coefficient

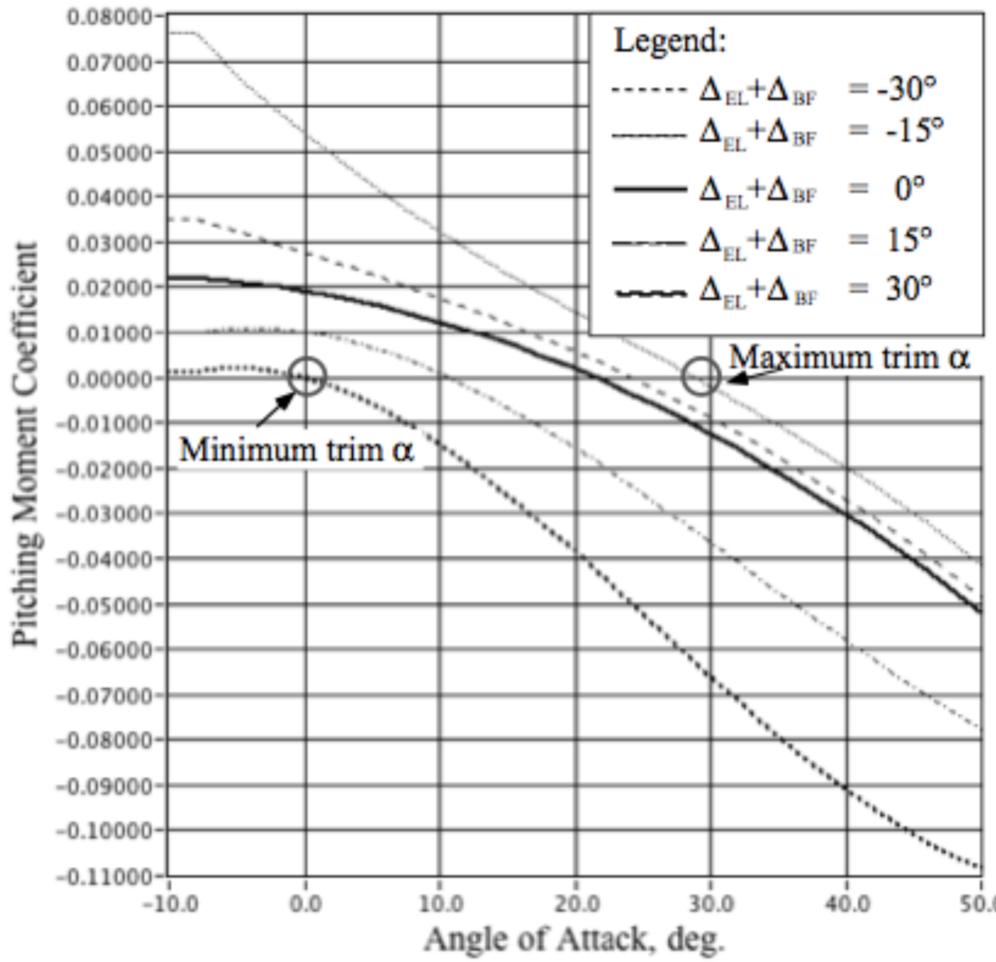


b) Effect of Symmetric Elevon on Drag Coefficient

Drag Coefficient

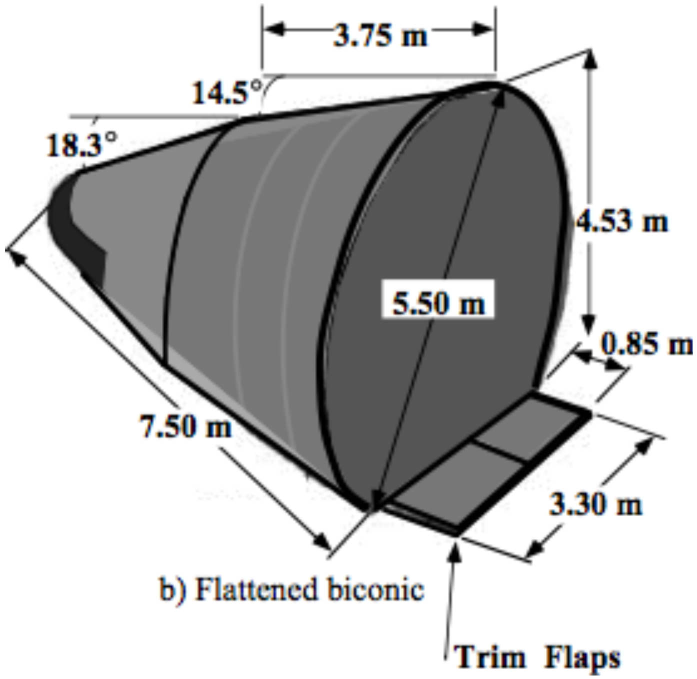
Examples of Aerodynamic Coefficients (2)

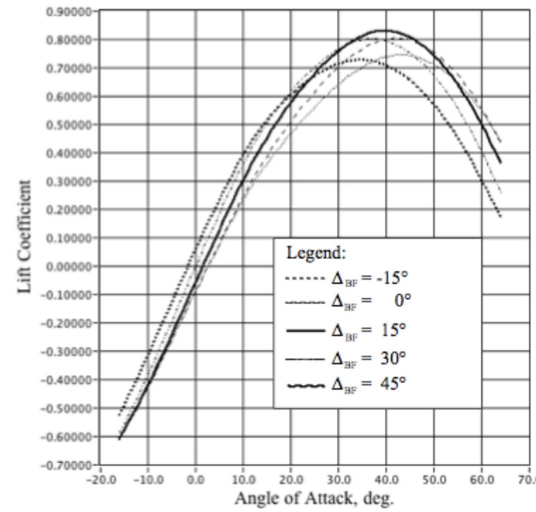




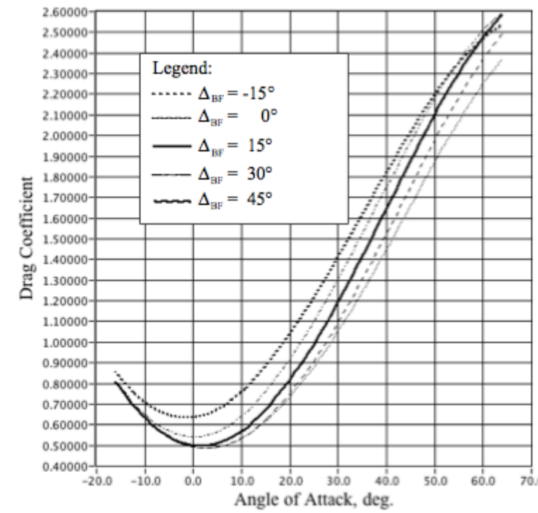
c) Effect of Symmetric Elevon and Body Flap deflections on Pitching Moment Coefficient

Examples of Aerodynamic Coefficients (3)

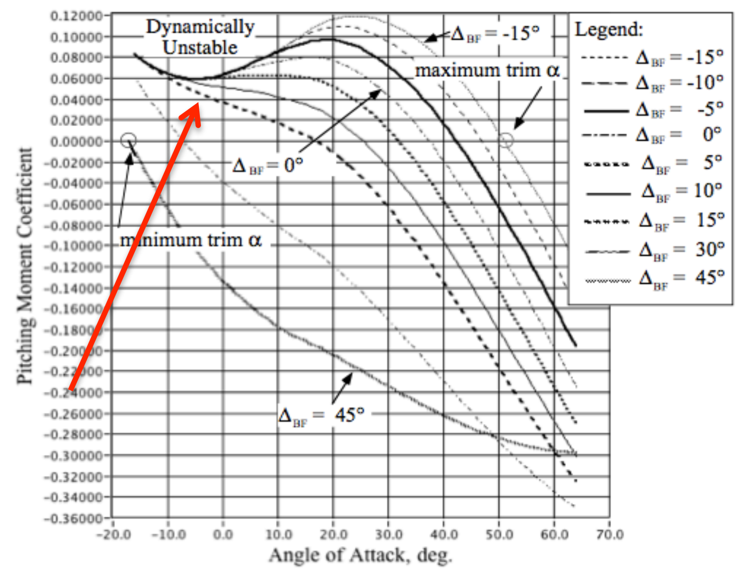




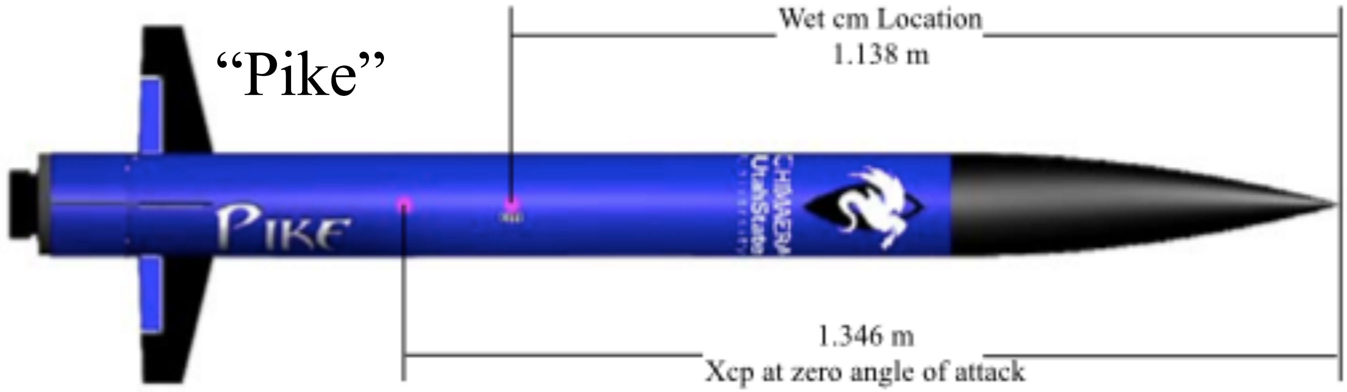
a) Effect of Symmetric Body Flap on Lift Coefficient

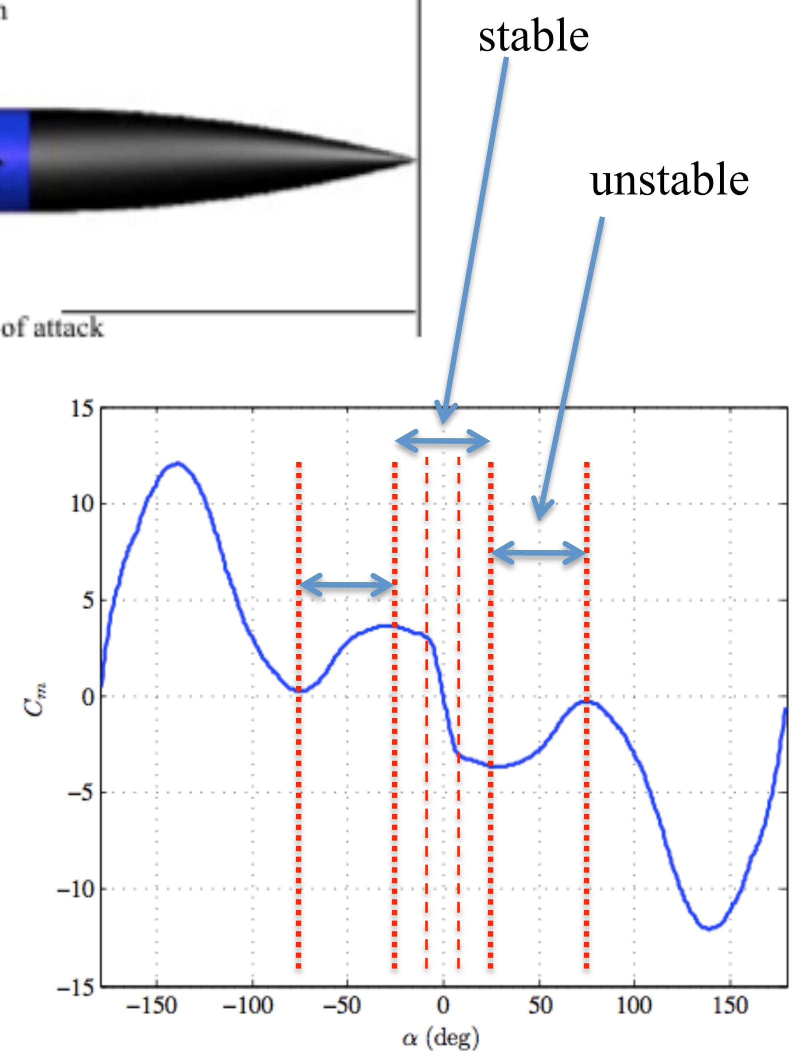


b) Effect of Symmetric Body Flap on Drag Coefficient



Examples of Aerodynamic Coefficients (3)





Even $X_{sm} > 0$ (static stability)
At low angle of attack

"strong stability" region limited to very low angle of attack range


Questions??