AE 721 Aerospace Design Laboratory I
Missile Design I

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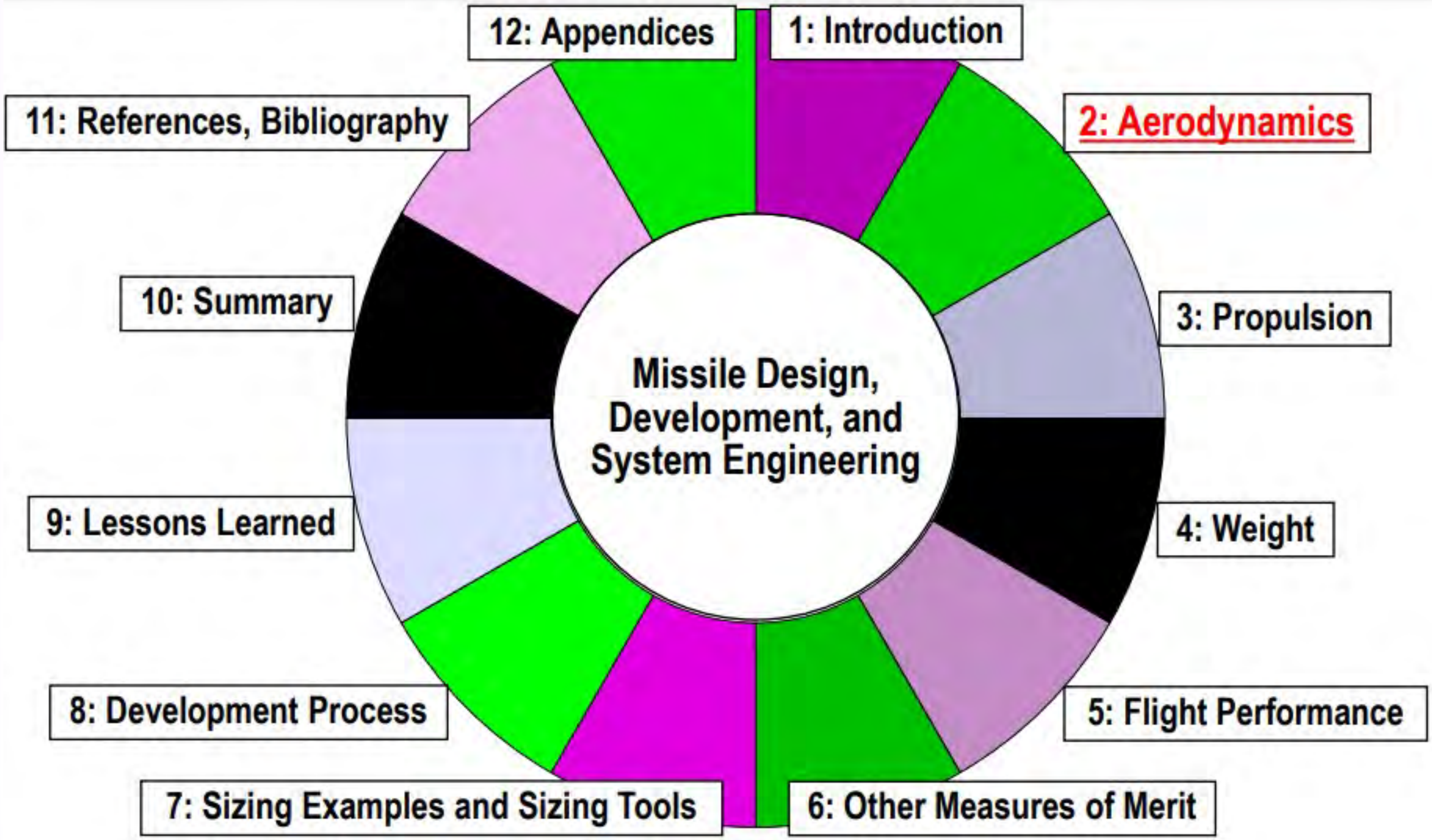
Outline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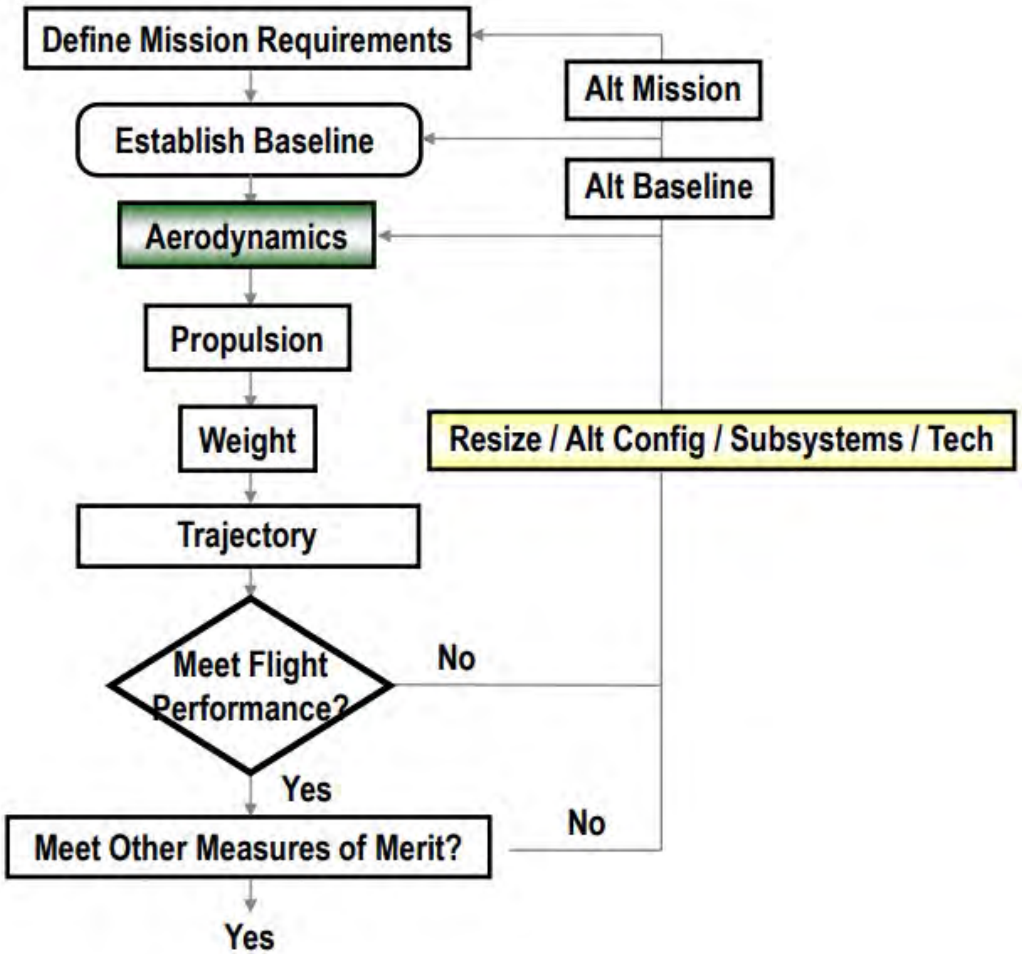
3: Propulsion
Missile Design, Development, and System Engineering
9: Lessons Learned
4: Weight
8: Development Process
5: Flight Performance
7: Sizing Examples and Sizing Tools
6: Other Measures of Merit

Chapter 2: Aerodynamic Considerations in Missile Design, Development, and System Engineering

What You Will Learn

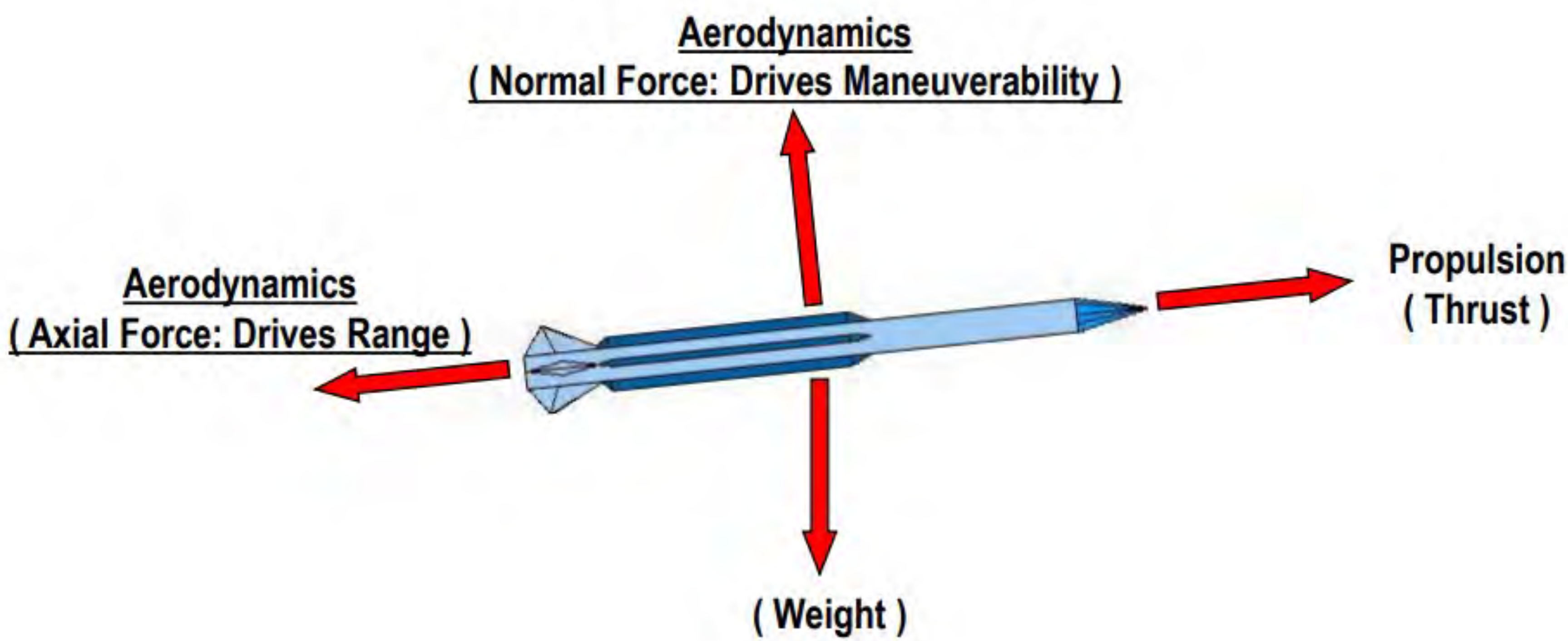
- Aerodynamic Conceptual Design Prediction Methods
- Missile Aerodynamic Design and System Engineering Tradeoffs
- Aerodynamic Stability and Control Guidelines
- Aerodynamic Body Buildup
- Flight Control Alternatives
- Maneuver Law Alternatives
- New Technologies for Missile Aerodynamics

Conceptual Design and System Engineering Require Broad, Creative, Rapid, and Iterative Evaluations

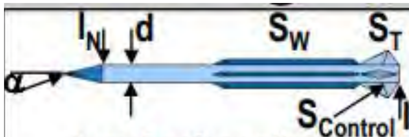


Flight Performance / Trajectory is Driven by Forces (Aerodynamics, Propulsion, Weight) on the Missile

Example of Typical Forces on a Missile



Aero Configuation Sizing / System Engineering has High Impact on Mission Requirements / MOM

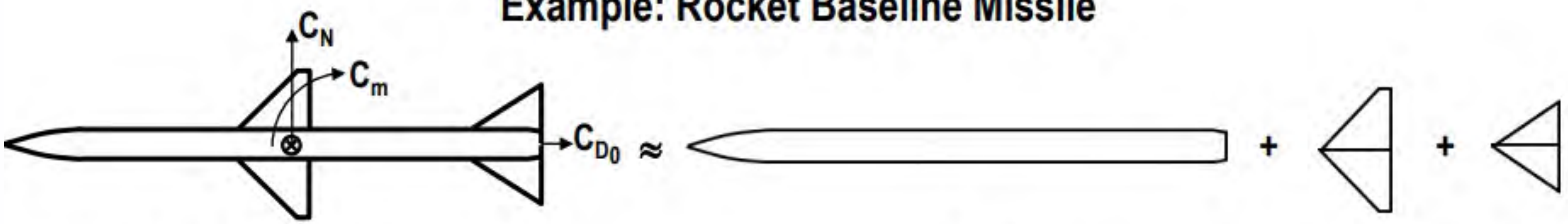


Aero Configuration Sizing / System Engineering Parameter	Typical Impact on Missile Mission Requirements / Measures of Merit (MOM)									
	Flight Performance MOM			Other Typical Measures of Merit						Sys Engr Constraint
	Range	Maneuver-ability	Time to Target	Robust-ness	Lethality	Miss Distance	Observ-ables	Survivability	Cost	Launch Platform
Nose Fineness (l_N / d)	○ ●	○	○ ●	○	○ ●	○ ●	●	●	○	○
Diameter (d)	◒ ●	○	○ ●	○	●	◒	○	○	○ ●	●
Length (l)	◒ ●	○	◒	○	○ ◒	○ ◒	○	○	○ ●	●
Wing Geometry / Size (S_W)	●	●	◒	○	●	●	◒	◒	–	●
Tail Stab Geom / Size (S_T)	◒	●	○	○	◒ ●	◒ ●	◒	◒	–	●
Flight Control Geometry / Size ($S_{Control}$)	◒	●	○	○	●	●	◒	◒	○	●
Flight Conditions (Angle of Attack α, Mach Number M, Altitude h)	●	●	●	●	●	●	●	●	●	○

● Very Strong ◒ Strong ○ Moderate – Relatively Low

Conceptual Design Aerodynamic Methods of This Text are Based on Aero Configurations Buildup

Example: Rocket Baseline Missile



Zero-Lift Drag Coefficient $C_{D_0} \approx (C_{D_0})_{Body} + (C_{D_0})_{Wing} + (C_{D_0})_{Tail}$

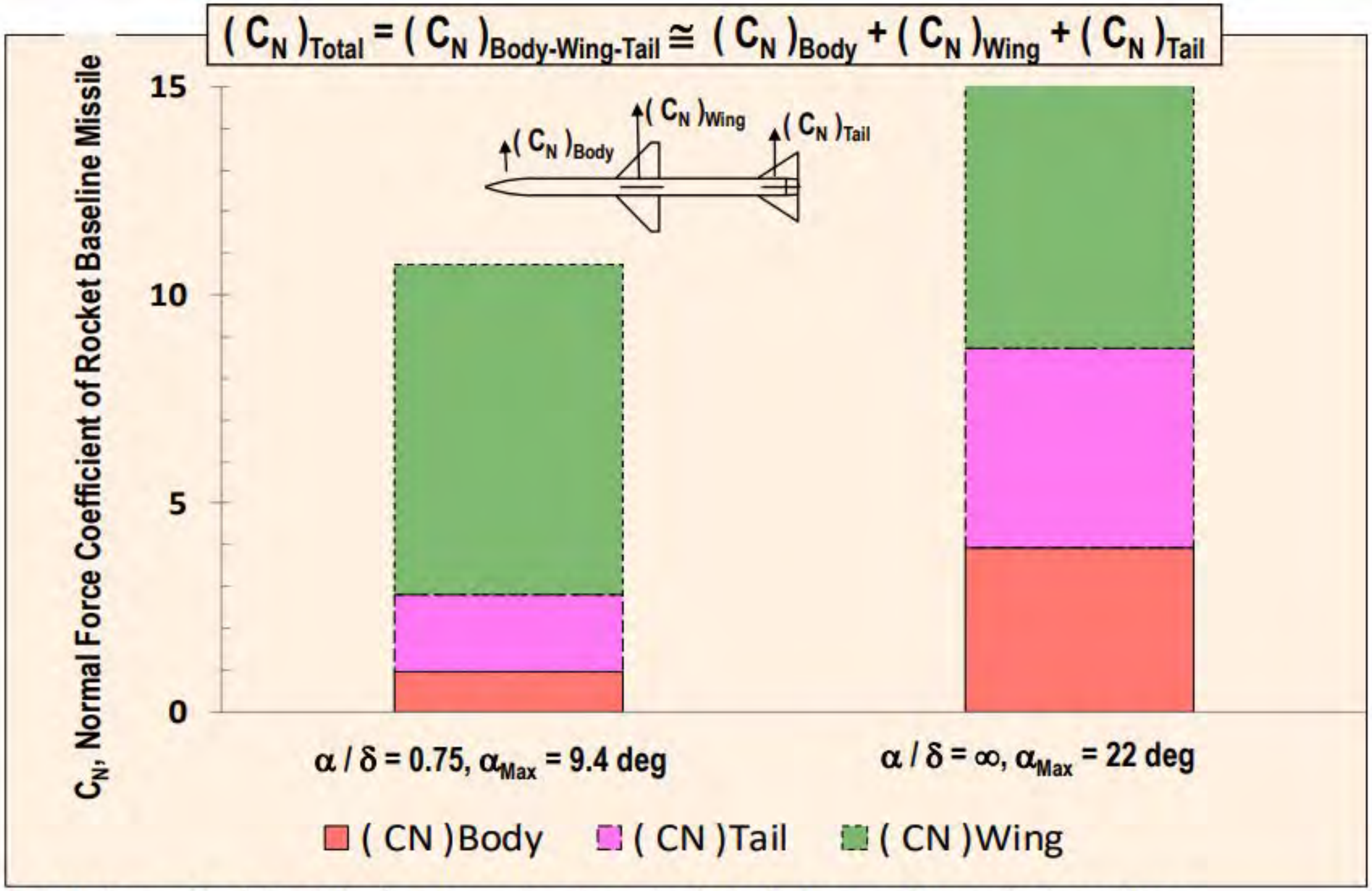
Normal Force Coefficient $C_N \approx (C_N)_{Body} + (C_N)_{Wing} + (C_N)_{Tail}$

Pitching Moment Coefficient $C_m \approx (C_m)_{Body} + (C_m)_{Wing} + (C_m)_{Tail}$

Note:
C_{D_0} = Zero-lift drag / dynamic pressure / reference area, C_N = normal force / dynamic pressure / reference area, C_m = pitching moment / dynamic pressure / reference area.
Conceptual design prediction methods of this text assume independent aerodynamics of body, wing, and tail. These methods do not include aerodynamic interactions of

- Body-wing
- Body-tail
- Wing-body
- Wing-tail
- Wing-wing
- Tail-body
- Tail-wing
- Tail-tail

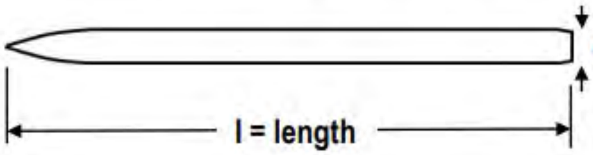
Conceptual Design Total Aerodynmic Force Mat be Estimated by Summing Individual Contributors



Note for figure: M = 2. Prediction from linear wing theory, slender body theory, and Newtonian impact theory.
α = Angle of attack, δ = Flight control deflection.

Missile Diameter is a Tradeoff

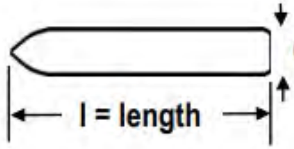
Smaller Diameter*



Benefit

- Launch Platform Diameter Constraint
- Decrease Drag

Larger Diameter**



Benefit

- Launch Platform Length Constraint
- Seeker Range, S/N, Resolution, & Tracking
- Warhead Lethality
- Subsystems Packaging
- Air Breathing Engine Thrust
- Rocket Motor Acoustic Frequency
- Body Bending Frequency

*Smaller diameter missile requires longer length to package subsystems
**Larger diameter missile can package subsystems in shorter length

Note: Typical body fineness ratio is l / d > 5 (Javelin l / d = 8.5) and l / d < 25 (AIM-120 l / d = 20.5)

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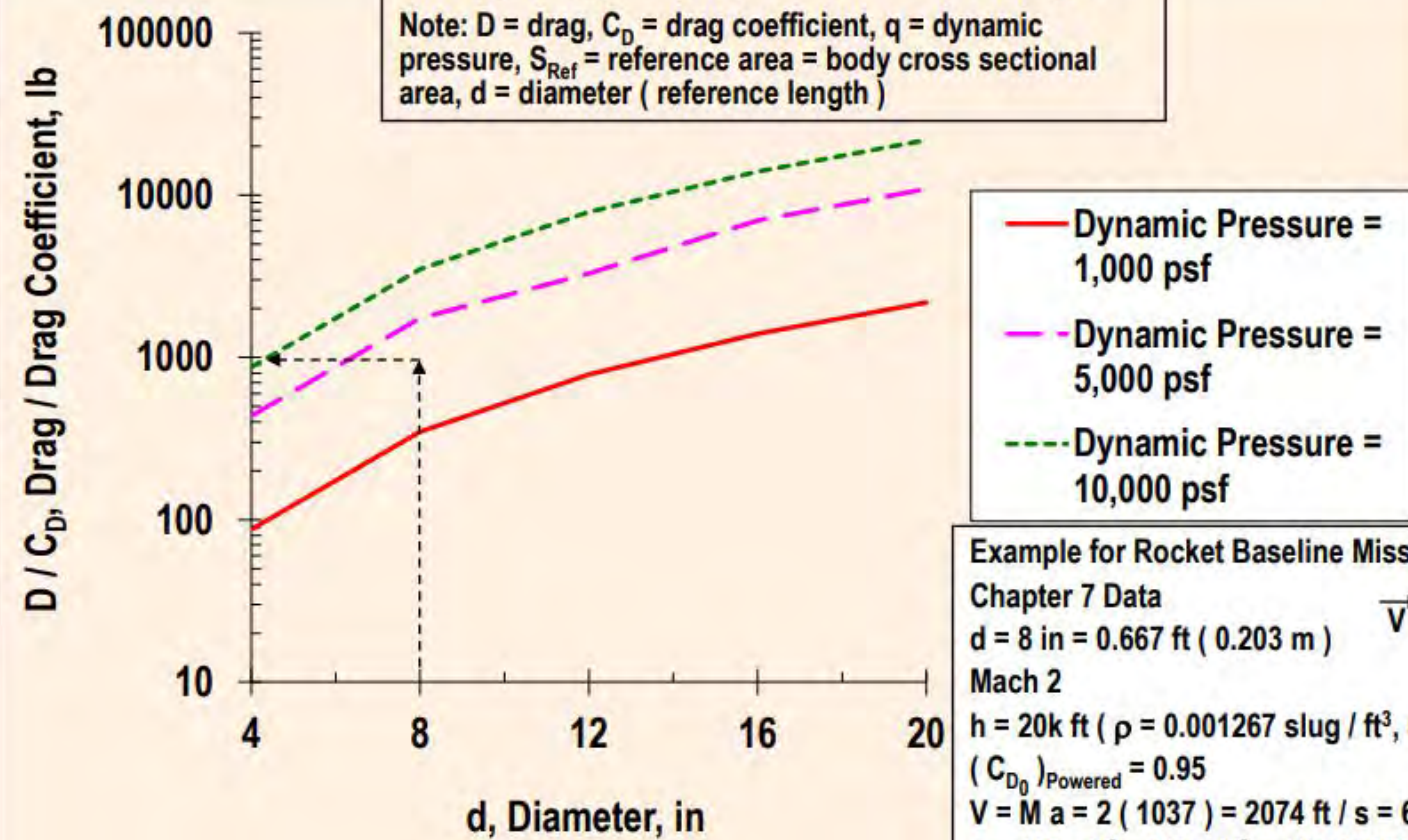
A Small Diameter Missile has Lower Drag

$D = C_D\, q\, S_{Ref} = C_D\, q\, (\pi / 4)\, d^2 = 0.785\, C_D\, q\, d^2$

Note: D = drag, C_D = drag coefficient, q = dynamic pressure, S_{Ref} = reference area = body cross sectional area, d = diameter (reference length)



Example for Rocket Baseline Missile:
Chapter 7 Data

d = 8 in = 0.667 ft (0.203 m)

Mach 2

h = 20k ft (ρ = 0.001267 slug / ft^3, a = 1037 ft / s)

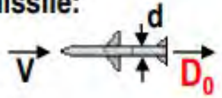
$(C_{D_0})_{Powered} = 0.95$

V = M a = 2 (1037) = 2074 ft / s = 632 m / s

$q = 1/2\, \rho V^2 = 1/2\, \rho V^2 = 1/2\, (0.001267)\, (2074)^2$
$= 2725$ lb / ft^2 = 130,500 N / m^2

$D_0 / C_{D_0} = 0.785\, (2725)\, (0.667)^2 = 952$ lb

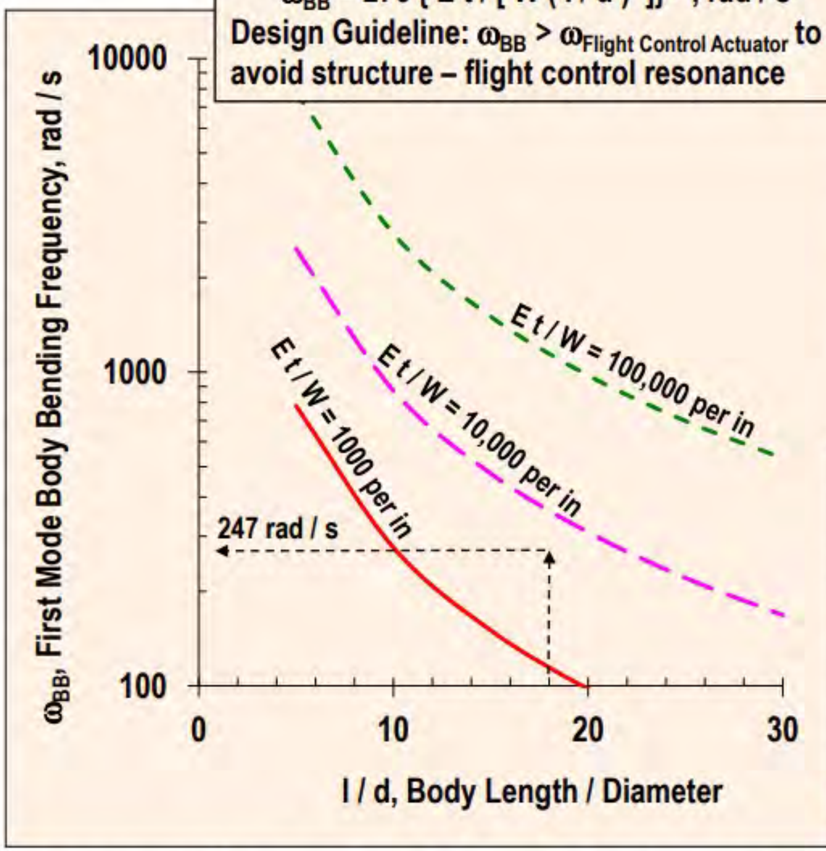
$D_0 = 0.95\, (952) = 900$ lb (4000 N)



Missile Finess Ratio May be Limited by Resonance of Body Bending Frequency with Flight Control

$\omega_{BB} = 276 \{ E t / [W (l / d)^3] \}^{1/2}$, rad / s

Design Guideline: $\omega_{BB} > \omega_{Flight\ Control\ Actuator}$ to avoid structure – flight control resonance

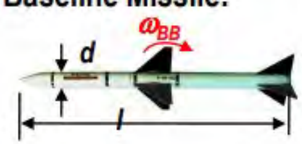


Assumes body cylinder structure, thin skin, high fineness ratio, uniform weight distribution, free-free motion. Neglects bulkheads, joints, wing / tail, insulation, subsystems damping effects.

ω_{BB} = First mode body bending frequency, rad / s
E = Modulus of elasticity in psi
t = Thickness in inches
W = Weight in lb
l / d = Fineness ratio
$\omega_{Flight\ Control\ Actuator}$ = Flight control actuator bandwidth, rad / s

Example for Rocket Baseline Missile:

l / d = 18
$E_{AVG} = 19.5 \times 10^6$ psi
$t_{AVG} = 0.12$ in
W = 500 lb

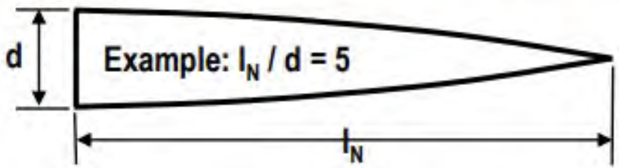


$E t / W = 19.5 \times 10^6 (0.12) / 500 = 4680$ per in
$\omega_{BB} = 276 (4680 / 18^3)^{1/2} = 247$ rad / s = 39.3 Hz
$\omega_{Flight\ Control\ Actuator}$ = 100 rad / sec = 16 Hz
$\omega_{BB} > \omega_{Flight\ Control\ Actuator}$ (satisfies design guideline)

Derived from: AIAA Aerospace Design Engineers Guide, American Institute of Aeronautics and Astronautics, 2012.

Nose Fineness and Geometry is a Tradeoff

High Fineness Nose Ideal for Less Drag at Supersonic Mach Number and Low Radar Cross Section (RCS)



Hemi Nose Ideal for Seeker, Length for Propellant / Subsystems

l_N / d = 0.5 (hemisphere)

l_N / d ~ 0.5 (w/o spike)

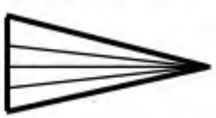
Video of Trident Extending Aerospike

High Fineness Nose Geometry Alternatives

<u>Conventional (e.g., Tangent Ogive)</u>

Low Manufacturing Cost

<u>Faceted</u>



Low Distortion, Low RCS, Low Drag

<u>Window</u>

Low RCS, Low Distortion

<u>Multi-lens</u>

Low Dome Error Slope

Note: Missile nose fineness ratio is typically l_N / d ~ 2 if M > 1.

Faceted and Flat Window Domes can Provide Low Distortion, Low Drag, & Low Radar Cross Section

Faceted Dome (Mistral) Video

Body Maximum Zero-Lift Drag Coefficient Occurs Near Mach 1

$(C_{D_0})_{Body} = (C_{D_0})_{Body,Friction} + (C_{D_0})_{Base} + (C_{D_0})_{Body,\ Wave}$

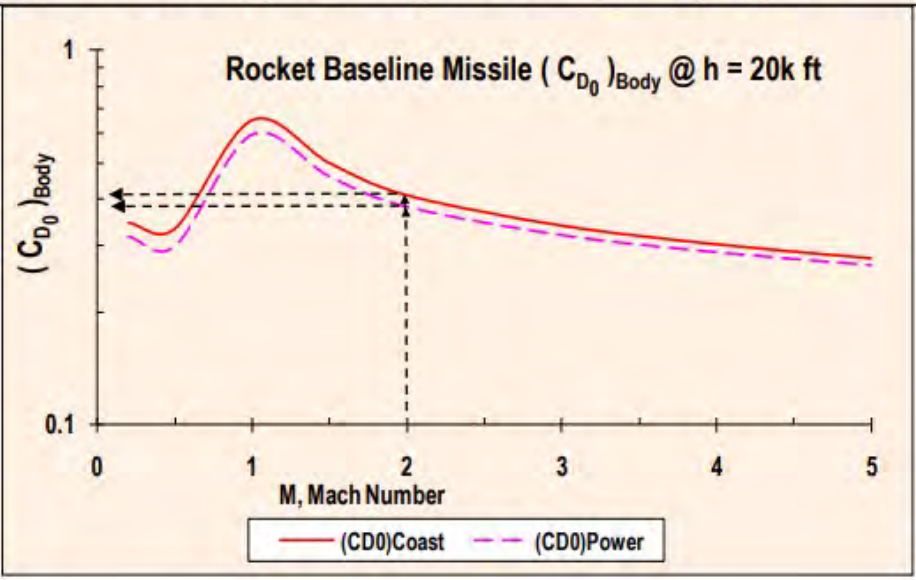
$(C_{D_0})_{Body,Friction} = 0.053\ (l / d)\ [M / (q\ l)]^{0.2}$. Based on Jerger reference, turbulent boundary layer, q in psf, l in ft.

$(C_{D_0})_{Base,Coast} = 0.25 / M$, if $M > 1$ and $(C_{D_0})_{Base,Coast} = 0.12 + 0.13\ M^2$, if $M < 1$

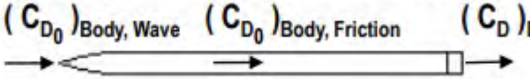
$(C_{D_0})_{Base,Powered} = (1 - A_e / S_{Ref})\ (0.25 / M)$, if $M > 1$ and $(C_{D_0})_{Base,Powered} = (1 - A_e / S_{Ref})\ (0.12 + 0.13\ M^2)$, if $M < 1$

$(C_{D_0})_{Body,\ Wave} = (1.59 + 1.83 / M^2)\ \{\tan^{-1}[0.5 / (l_N / d)]\}^{1.69}$, for $M > 1$. Based on Bonney reference, $\tan^{-1}$ in rad.

Nomenclature: $(C_{D_0})_{Body,Wave}$ = body zero-lift wave drag coefficient, $(C_{D_0})_{Base}$ = body base drag coefficient, $(C_{D_0})_{Body,\ Friction}$ = body skin friction drag coefficient, $(C_{D_0})_{Body}$ = body zero-lift drag coefficient, l_N = nose length, d = body diameter, l = body length, A_e = nozzle exit area, S_{Ref} = reference area, q = dynamic pressure, $\tan^{-1}[0.5 / (l_N / d)]$ in rad.



Example for Rocket Baseline Missile:



$l_N / d = 2.4$, $A_e = 11.22\ in^2$, $S_{Ref} = 50.26\ in^2$, $M = 2$, $h = 20k$ ft, $q = 2725$ psf, $l / d = 18$, $l = 12$ ft

Calculate:

$(C_{D_0})_{Body,\ Friction} = 0.053\ (18)\ \{(2) / [(2725)\ (12)]\}^{0.2} = 0.14$

$(C_{D_0})_{Base\ Coast} = 0.25 / 2 = 0.13$

$(C_{D_0})_{Base\ Powered} = (1 - 0.223)\ (0.25 / 2) = 0.10$

$(C_{D_0})_{Body,\ Wave} = 0.14$

$(C_{D_0})_{Body,\ Coast} = 0.14 + 0.13 + 0.14 = 0.41$

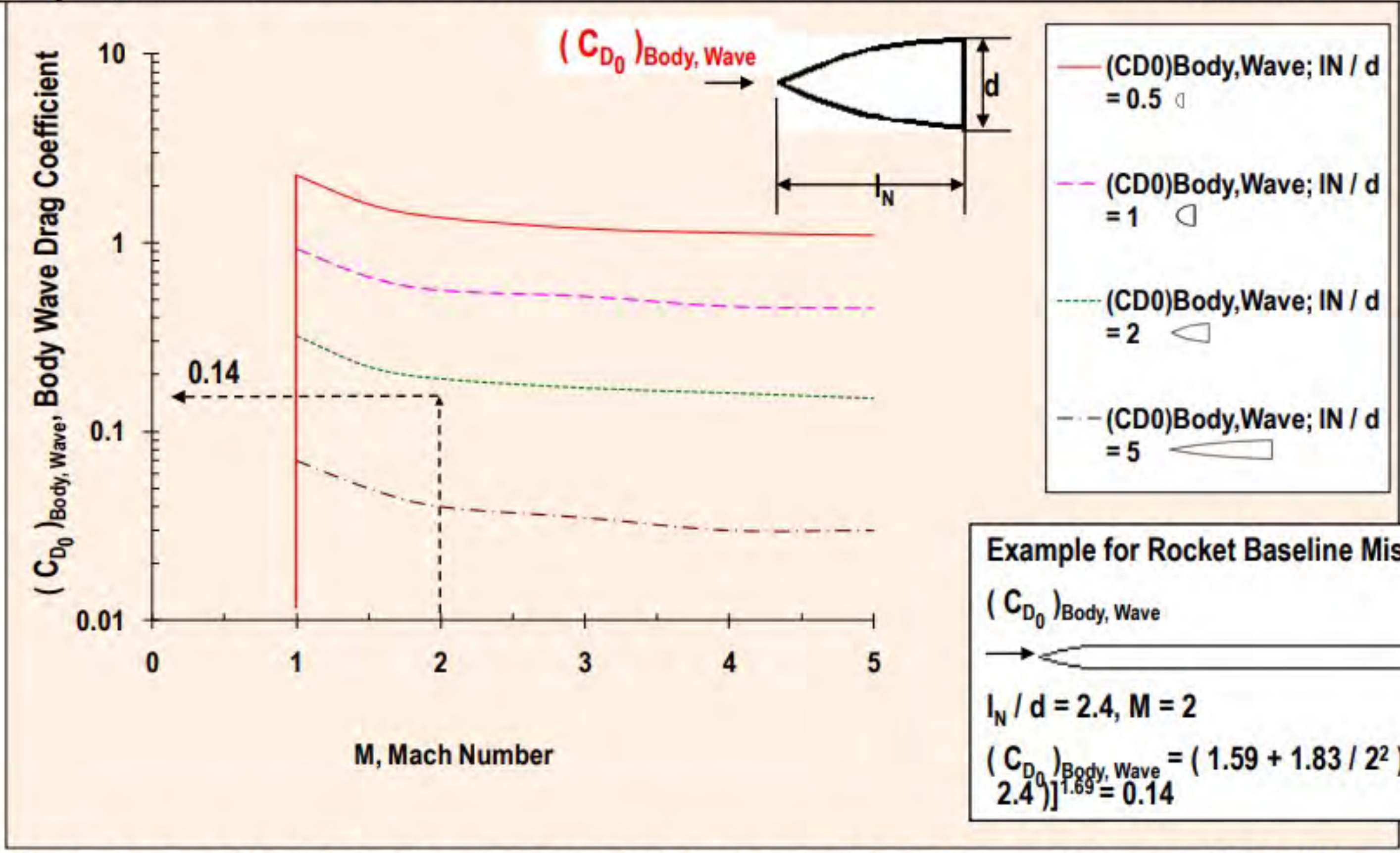
$(C_{D_0})_{Body,\ Powered} = 0.14 + 0.10 + 0.14 = 0.38$

Chapter 2: Aerodynamics

Supersonic Body Wave Drag is Driven by Nose Fineness

$$(C_{D_0})_{\text{Body, Wave}} = (1.59 + 1.83 / M^2)\{\tan^{-1}[0.5/(l_N/d)]\}^{1.69}, \text{ for } M > 1$$

Note: $(C_{D_0})_{\text{Body,,Wave}}$ = body zero-lift wave drag coefficient, l_N = nose length, d = body diameter, $\tan^{-1}[0.5/(l_N/d)]$ in rad.



Example for Rocket Baseline Missile:

$(C_{D_0})_{\text{Body, Wave}}$

$l_N / d = 2.4$, $M = 2$

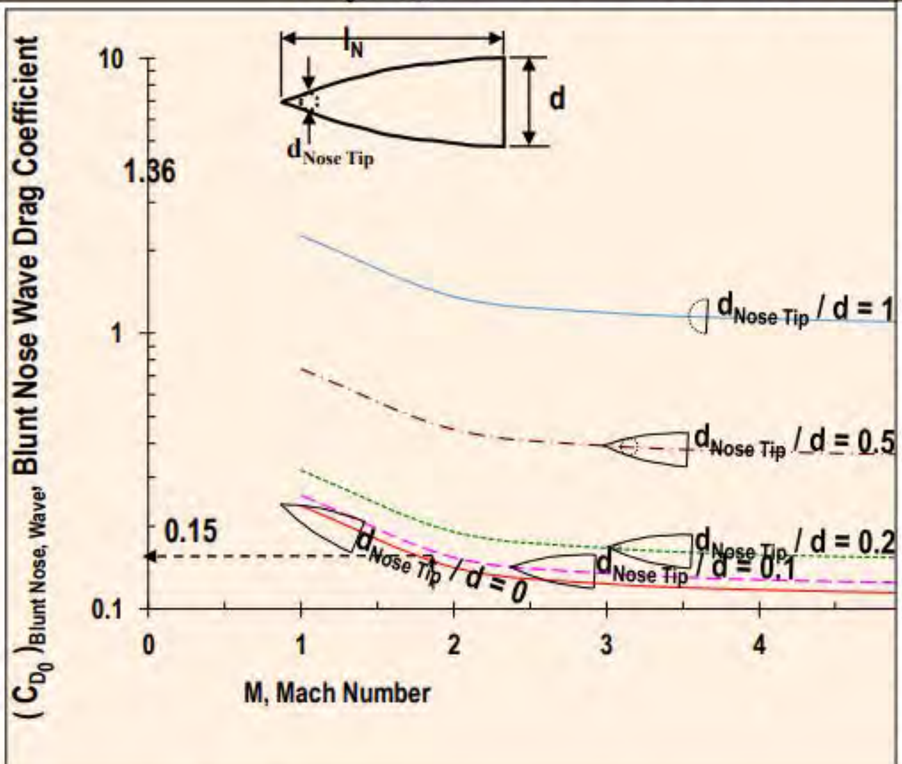
$(C_{D_0})_{\text{Body, Wave}} = (1.59 + 1.83/2^2)[\tan^{-1}(0.5/2.4)]^{1.69} = 0.14$

Source. Bonney, E.A., et al, Aerodynamics, Propulsion, Structures, and Design Practice, "Principles of Guided Missile Design"

Moderate Nose Tip Blutness Causes a Relatively Small Increase in Supersonic Drag

$$(C_{D_0})_{Blunt\ Nose,\ Wave} = (C_{D_0})_{Sharp\ Nose,\ Wave} [1 - (d_{Nose\ Tip} / d)^2] + (C_{D_0})_{Hemi,\ Wave} (d_{Nose\ Tip} / d)^2$$

$$(C_{D_0})_{Body,\ Wave} = (1.59 + 1.83 / M^2) \{\tan^{-1} [0.5 / (l_N / d)]\}^{1.69}, \text{ for } M > 1.$$



Example: Rocket Baseline Missile @ M = 2:
l_N = 19.2 in, d_{Ref} = 8 in ⇒ l_N / d = 2.4, 10% Nose Tip Bluntness

1. Compute $(C_{D_0})_{Sharp\ Nose,\ Wave}$:
$(C_{D_0})_{Wave,\ Sharp\ Nose} = (1.59 + 1.83 / 2^2) [\tan^{-1} (0.5 / 2.4)]^{1.69} = 0.140$ based on body reference area

2. Compute $(C_{D_0})_{Hemi,\ Wave}$ of hemispherical nose ($l_{NoseTip} / d_{NoseTip} = 0.5$) with $d_{NoseTip} = 0.10\ d = 0.10\ (8) = 0.8$ in:
$(C_{D_0})_{Wave,\ Hemi} = (1.59 + 1.83 / 2^2) \{[\tan^{-1} (0.5 / (0.5))]\}^{1.69} = 2.05$, based on nose tip area

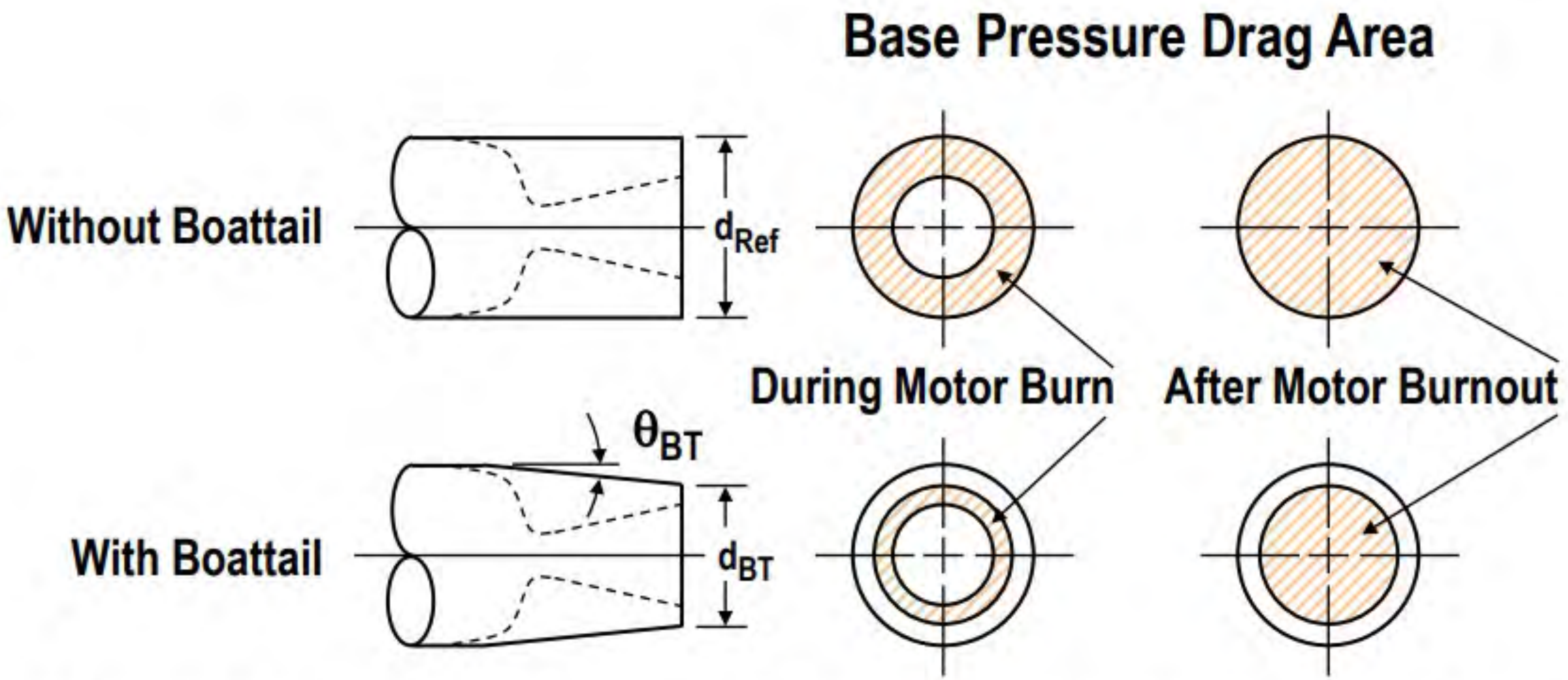
3. Finally, compute
$(C_{D_0})_{Blunt\ Nose,\ Wave} = (C_{D_0})_{Sharp\ Nose,\ Wave} [1 - (d_{Nose\ Tip} / d)^2] + (C_{D_0})_{Hemi,\ Wave} (d_{Nose\ Tip} / d)^2$
$= 0.141 ([1 - 0.10)^2 + 2.05 (0.10)^2 = 0.140 + 0.013 = 0.153$

Note: 10% nose tip bluntness provides 9% increase in wave drag of the sharp nose

Note: Nose tip bluntness increases drag, increases nose tip strength, and decreases nose tip aero heating

Nomenclature: $d_{Nose\ Tip}$ = nose tip diameter, d = body diameter (reference length), l_N = nose length, $\tan^{-1} [0.5 / (l_N / d)]$ in rad

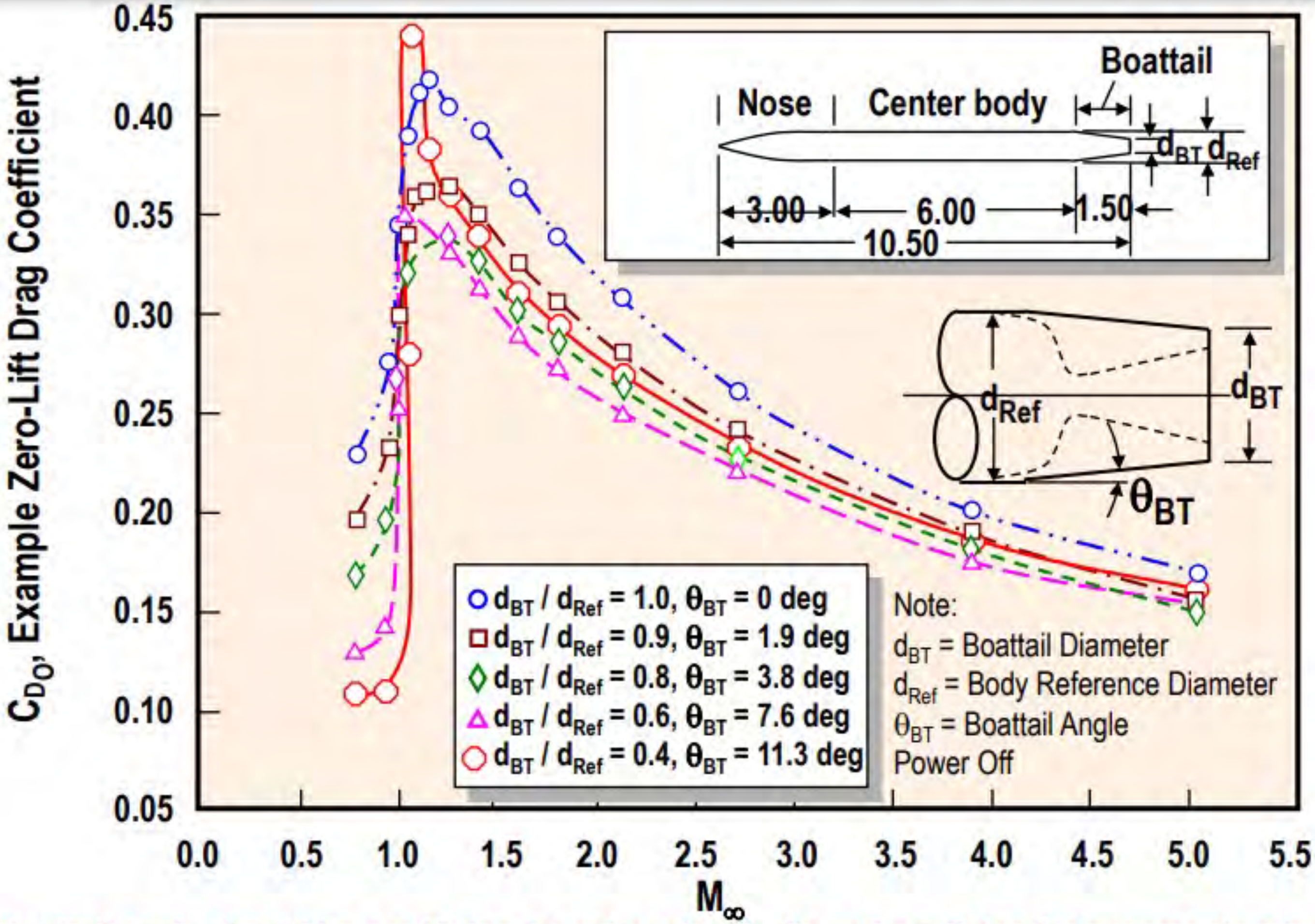
A Boattail Decreases Base Pressure Drag Area



Note: Boattail angle θ_{BT} and boattail diameter d_{BT} limited by propulsion nozzle packaging, tail flight control packaging, and flow separation

Reference: Chin, S. S., *Missile Configuration Design*, McGraw-Hill Book Company, New York, 1961

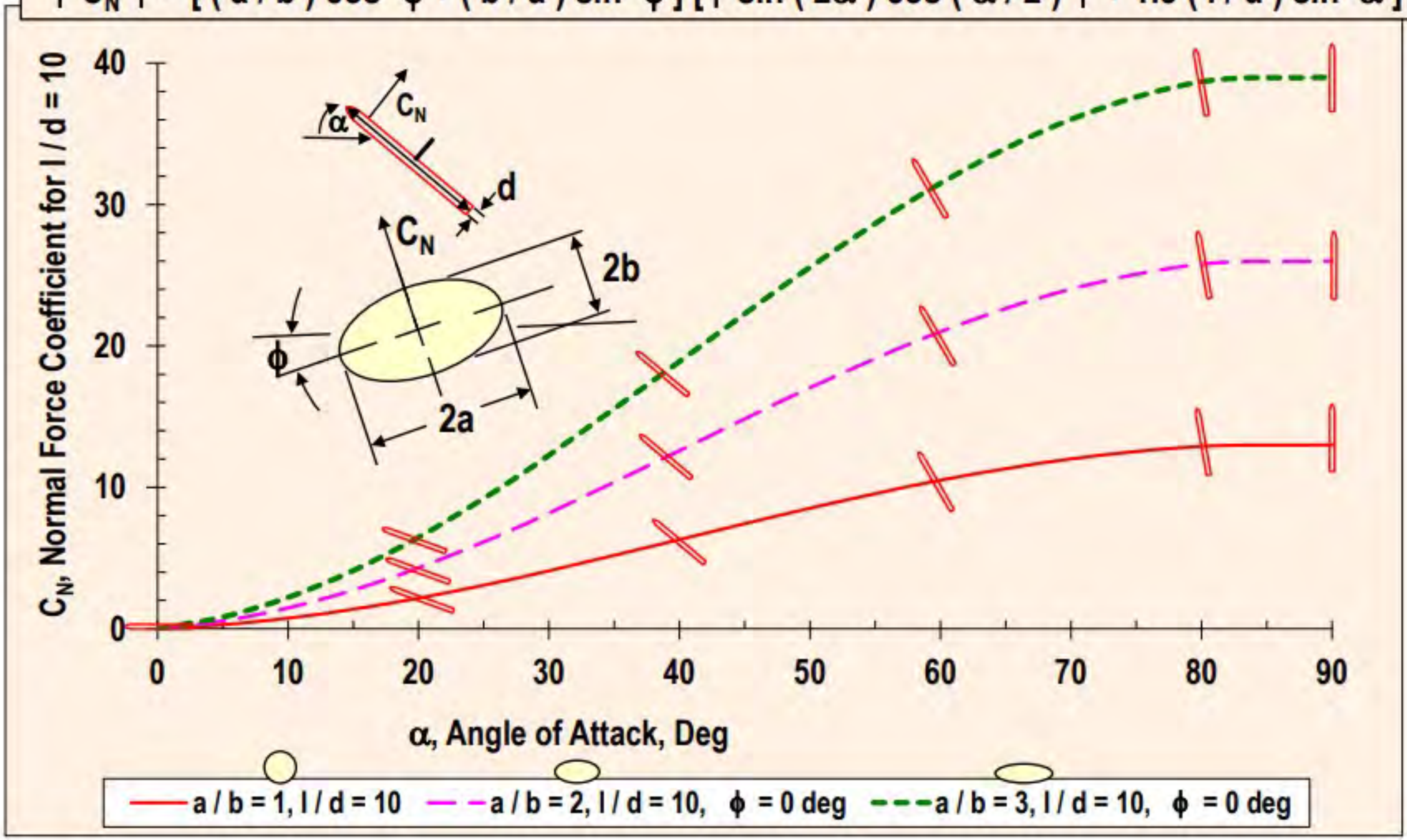
A Boattail is More Effective for a Subsonic Missile



Note: Boattail angle should be < ≈ 12 deg for subsonic missile and < ≈ 7 deg for supersonic missile, to avoid flow separation.

A Lifting Body Has Higher Normal Force

$$|C_N| = [(a/b)\cos^2\phi + (b/a)\sin^2\phi]\,[\,|\sin(2\alpha)\cos(\alpha/2)| + 1.3\,(l/d)\sin^2\alpha\,]$$

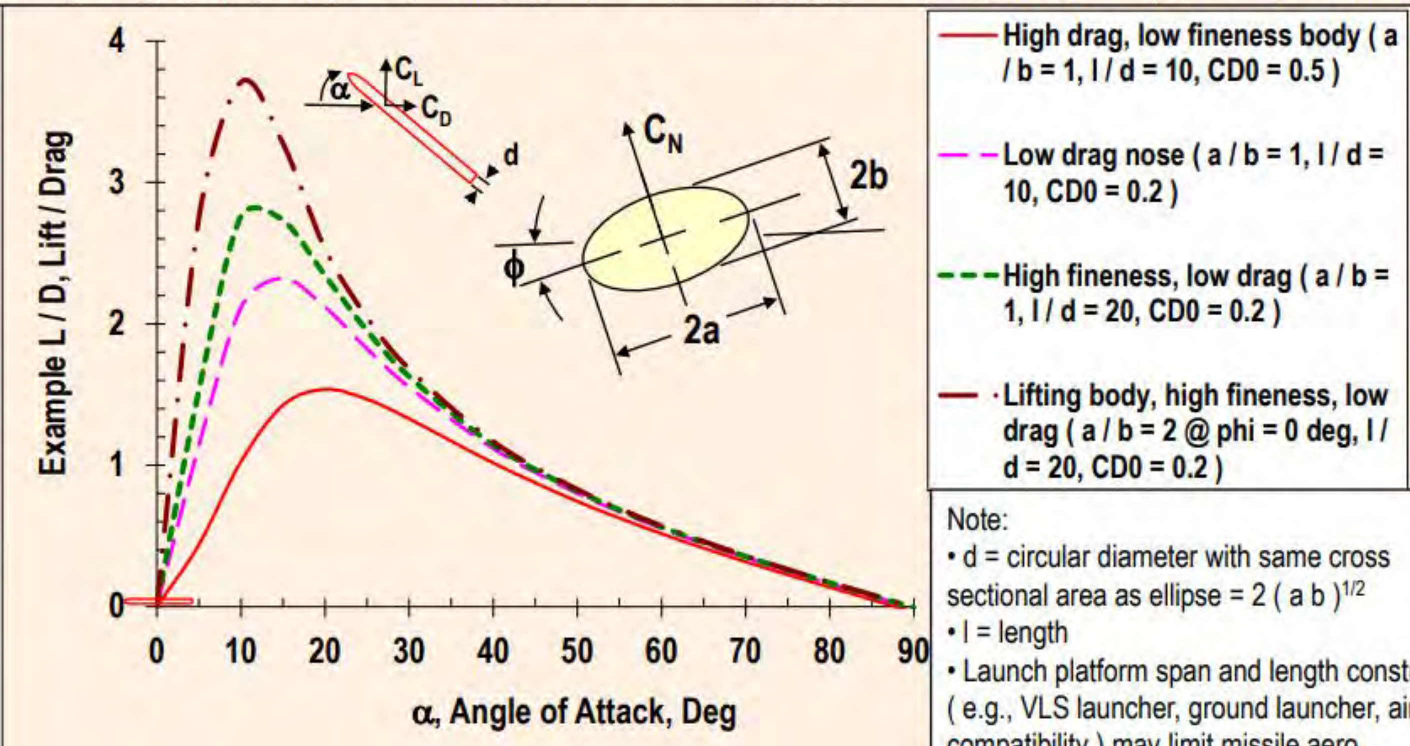


Note: Based on slender body theory (Pitts, et al) and cross flow theory (Jorgensen) references. Valid for l / d > 5. $d = 2\,(a\,b)^{1/2}$

Body Lift-to-Drag Ratio is Impacted by Angle of Attack, C_{do}, Body Fineness, and Cross Section Geometry

$L / D = C_L / C_D = (C_N \cos\alpha - C_A \sin\alpha) / (C_N \sin\alpha + C_A \cos\alpha) \approx (C_N \cos\alpha - C_{D0} \sin\alpha) / (C_N \sin\alpha + C_{D0} \cos\alpha)$

For lifting body, $| C_N | = [(a / b) \cos^2 (\phi) + (b / a) \sin^2 (\phi)] [| \sin (2\alpha) \cos (\alpha / 2) | + 1.3 (l / d) \sin^2\alpha]$



Note:

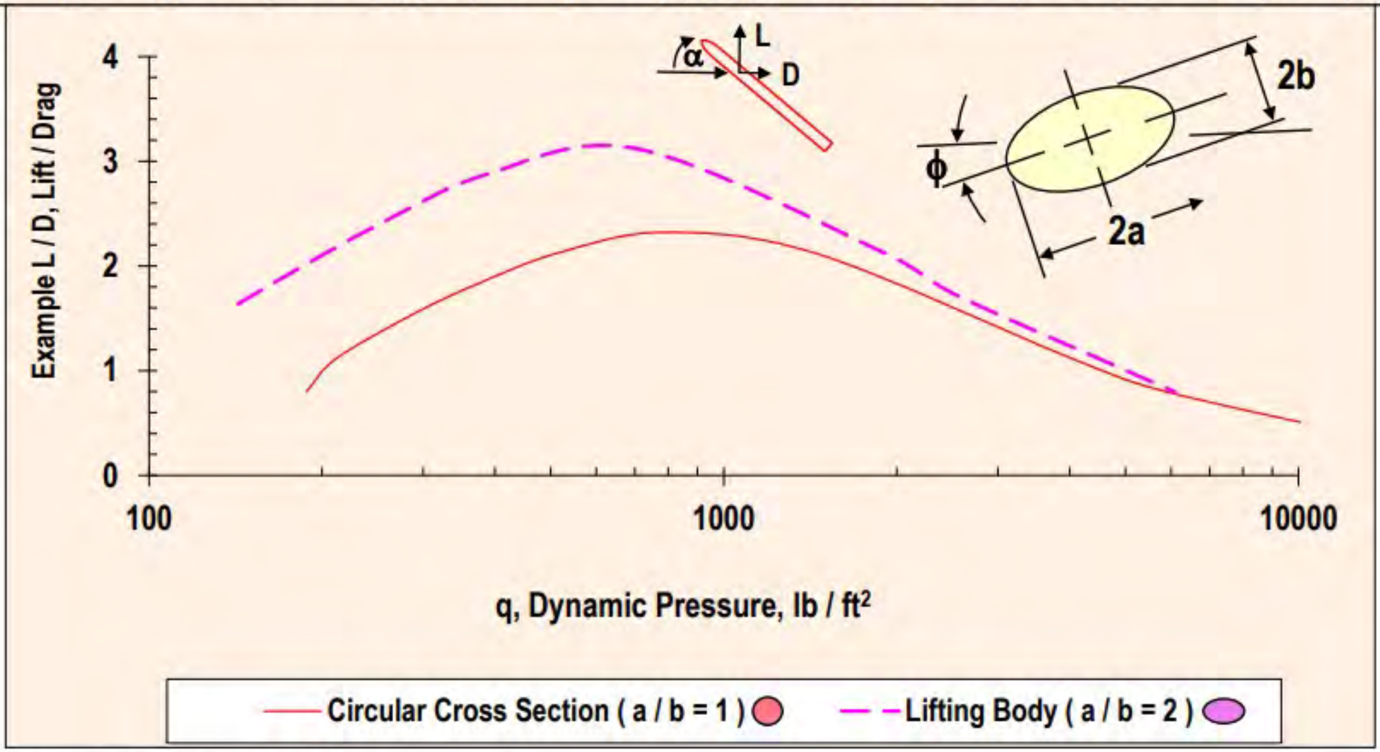
- d = circular diameter with same cross sectional area as ellipse = $2 (a b)^{1/2}$
- l = length
- Launch platform span and length constraints (e.g., VLS launcher, ground launcher, aircraft compatibility) may limit missile aero configuration enhancements

Note: Based on slender body theory (Pitts, et al) and cross flow theory (Jorgensen) references. Valid for l / d > 5

A Lifting Body Required Flight at Relatively Low Dynamic Pressure to Achieve High Lift-to-Drag Ratio

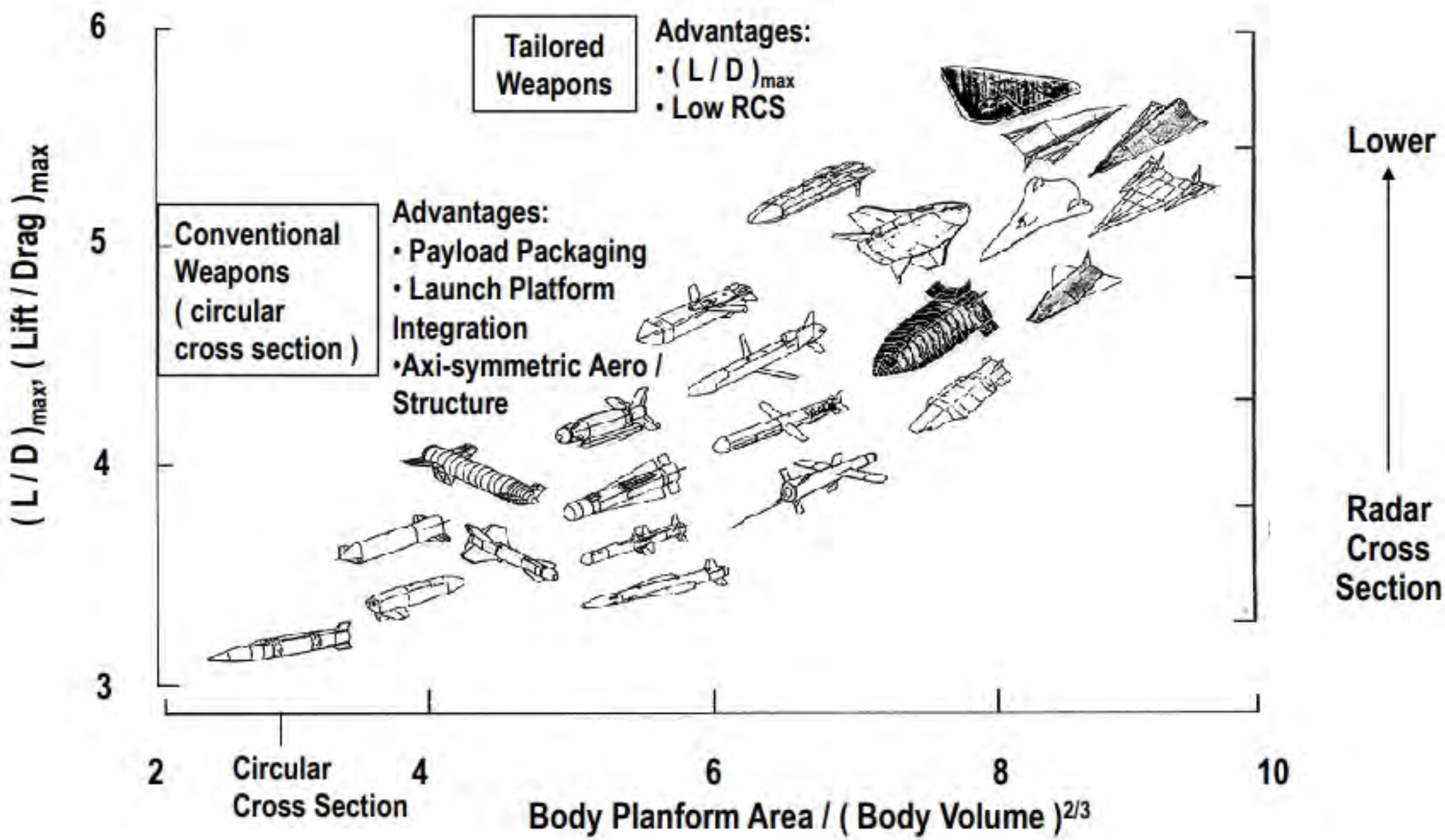
$$L / D = C_L / C_D \approx (C_N \cos\alpha - C_{D_O} \sin\alpha) / (C_N \sin\alpha + C_{D_O} \cos\alpha)$$

$$| C_N | = [(a / b) \cos^2 (\phi) + (b / a) \sin^2 (\phi)] [| \sin (2\alpha) \cos (\alpha / 2) | + 1.3 (l / d) \sin^2 \alpha]$$

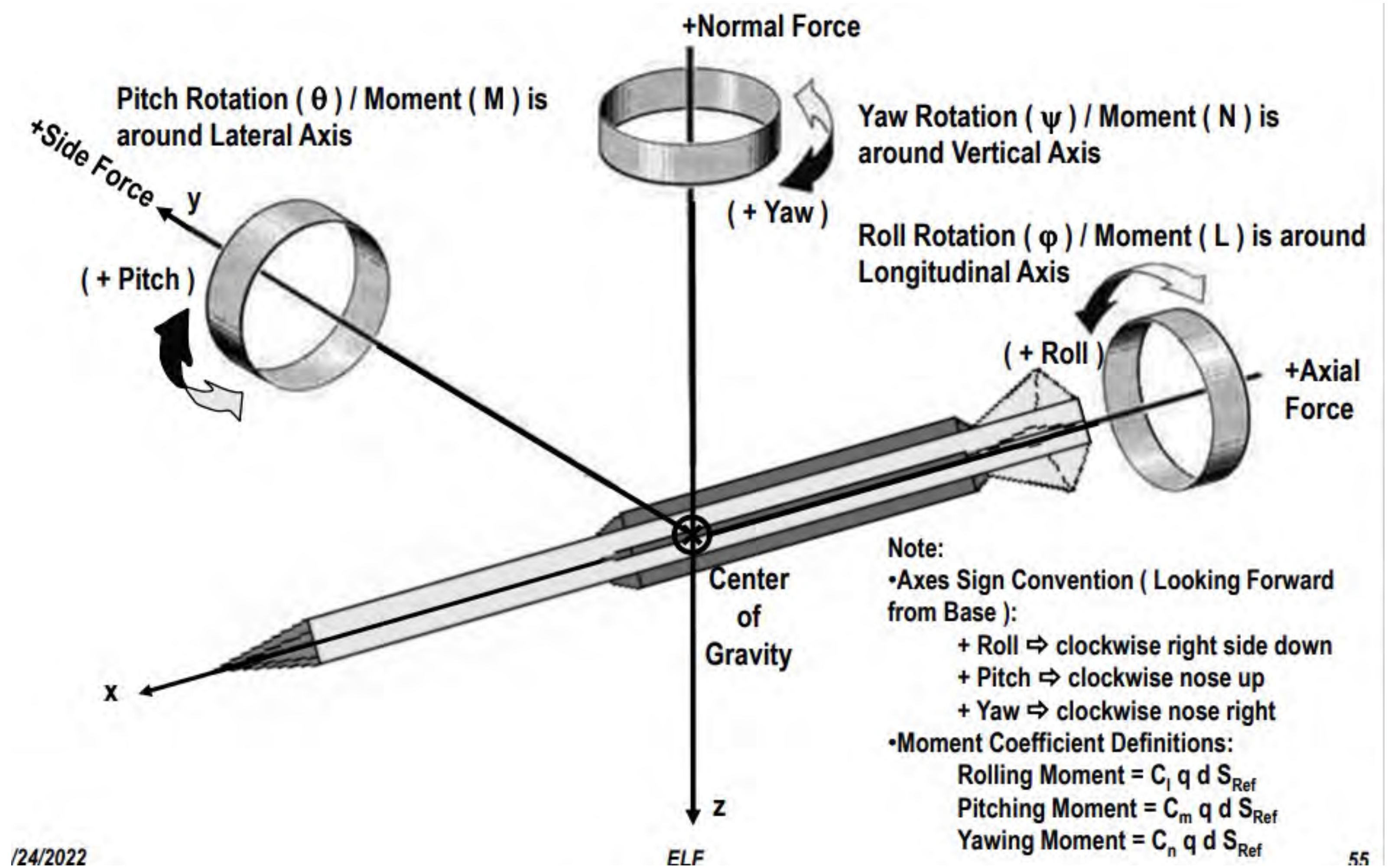


Note. Example figure based on following assumptions:
Body lift only (no surfaces), slender body theory + cross flow theory, cruise flight (lift = weight), W = L = 2000 lb, d = circular diameter with same cross sectional area as ellipse = $2 (a b)^{1/2}$, S = cross sectional area = 2 ft^2, l / d = 10, $C_{D_0} = 0.2$, $\varphi = 0$ deg

Tradeoff of Low Observable Missle Planform and (L/D)Max Versus Volumetric Efficiency

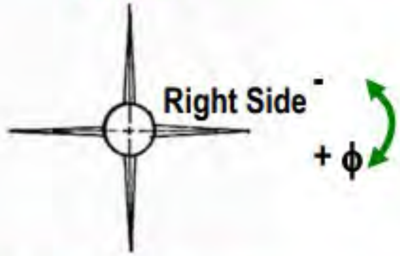


Sign Convention of Forces, Moments, and Axes for This Material

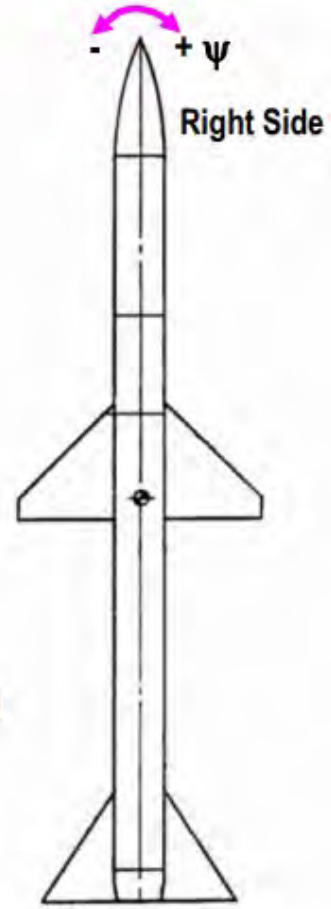


Pitch, Yaw, and Roll Animation

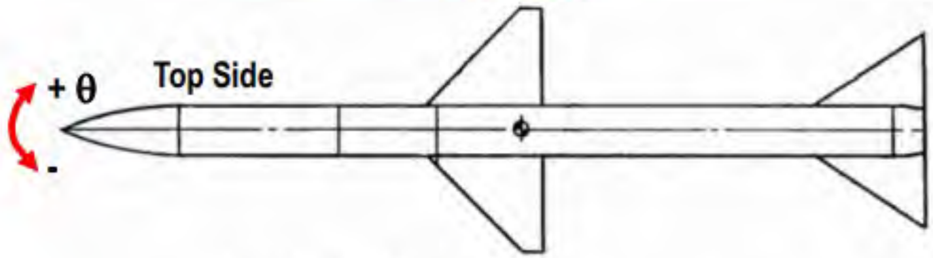
Rear View Animation of Roll



Top View Animation of Yaw

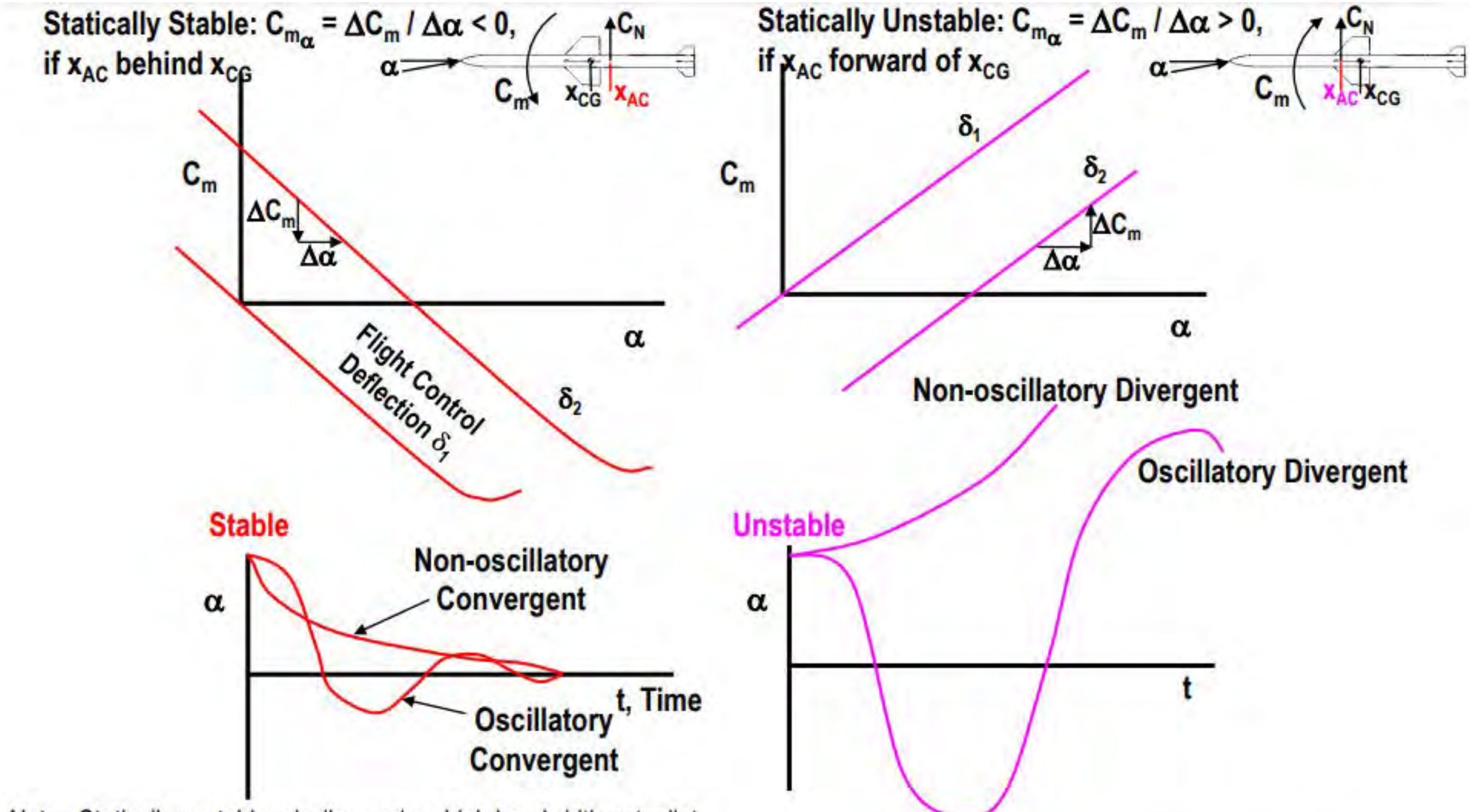


Side View Animation of Pitch



- Axes Sign Convention:
  - + Roll ⇨ clockwise right side down (looking forward from base)
  - + Pitch ⇨ clockwise nose up
  - + Yaw ⇨ clockwise nose right (looking forward from base)
- Moment Coefficient Definitions:
  - Rolling Moment = $C_l \, q \, d \, S_{Ref}$
  - Pitching Moment = $C_m \, q \, d \, S_{Ref}$
  - Yawing Moment = $C_n \, q \, d \, S_{Ref}$

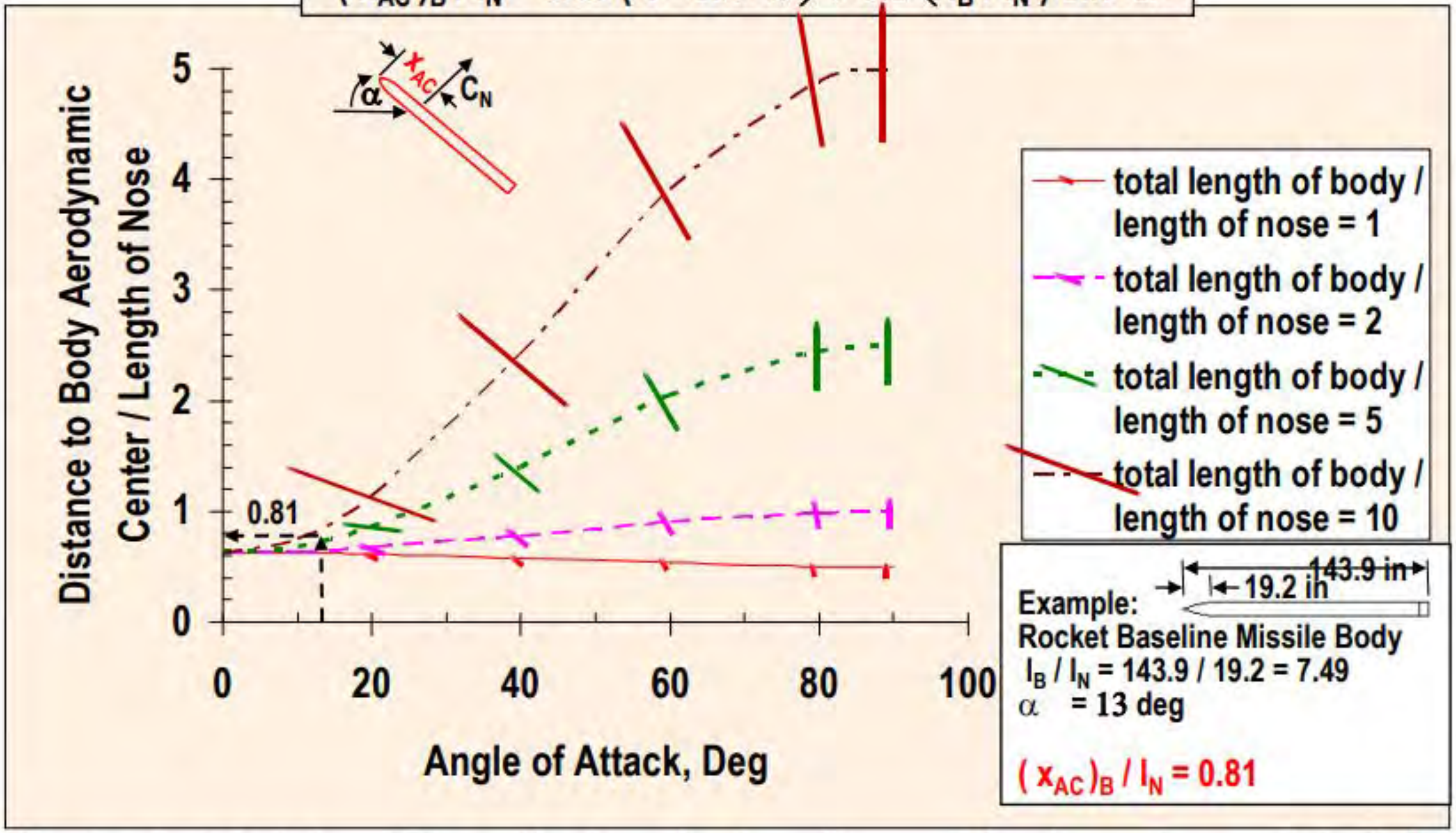
Oitch Moment Stability Δ Cm / Δ α and Static Margin $(X_{AC} - X_{CG})$ Define Pitch Static Stability



Note: Statically unstable missile requires high bandwidth autopilot.
Autopilot negative rate feedback provides stability augmentation.
Static Margin in Diameters = (Aerodynamic Center – Center of Gravity) / Diameter = $(x_{AC} - x_{CG}) / d = - (\Delta C_m / \Delta\alpha) / (\Delta C_N / \Delta\alpha)$

Body Aerodynamic Center is Driven by Angle of Attack, Nose Length, and Body Length

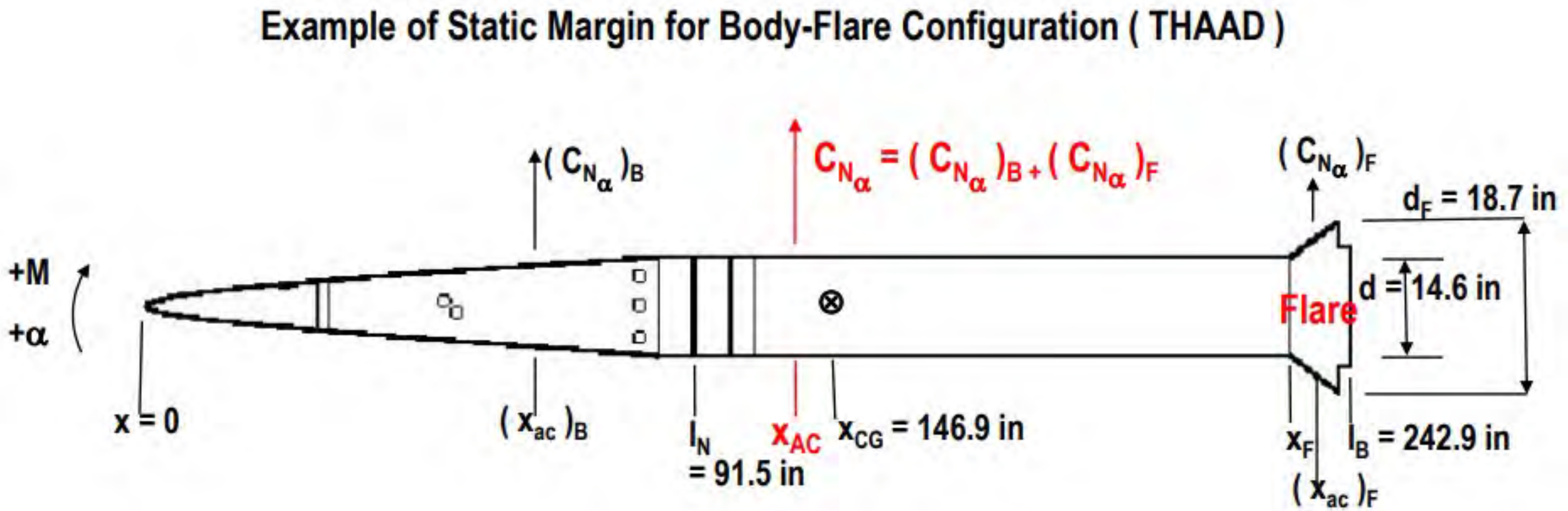
$$(x_{AC})_B / l_N = 0.63\,(1 - \sin^2\alpha) + 0.5\,(l_B / l_N)\,\sin^2\alpha$$



Note: Based on slender body theory (Pitts, et al) and cross flow theory (Jorgensen) for axisymmetric nose-cylinder. No flare.

$(x_{AC})_B$ = location of body aerodynamic center, l_N = length of nose, α = angle of attack, l_B = total length of body.

An Aft Flare Increases Static Stability



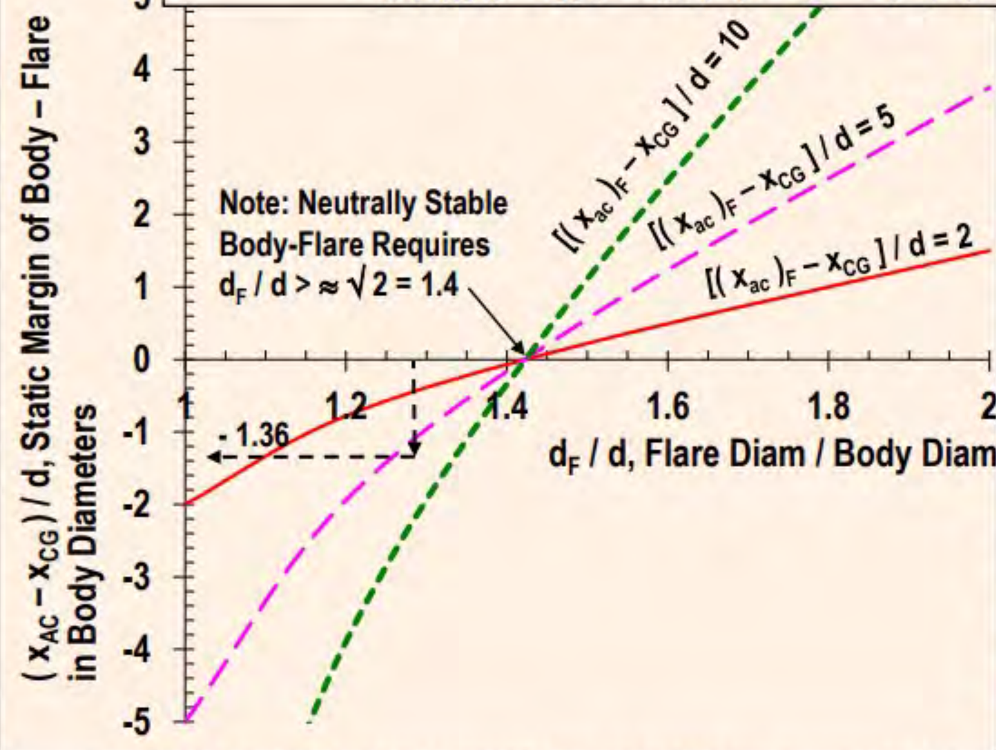
Nomenclature:.

M = pitching moment, α = angle of attack, x_{AC} = location of aerodynamic center, x_{CG} = location of center of gravity, d = diameter of body, $(C_{N_\alpha})_B$ = normal force coefficient variation with angle of attack for body, $(x_{AC})_B$ = location of aerodynamic center for body, l_N = length of nose, $(C_{N_\alpha})_F$ = normal force coefficient variation with angle of attack for flare, $(x_{AC})_F$ = location of aerodynamic center for flare, l_B = length of body, d_F = diameter of flare

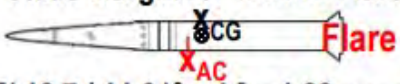
An Aft Flare Increases Static Stability (cont)

$$(x_{AC} - x_{CG}) / d = - \{(C_{N_\alpha})_B \{[x_{CG} - (x_{AC})_B] / d\} + (C_{N_\alpha})_F \{[x_{CG} - (x_{AC})_F] / d\}\} / [(C_{N_\alpha})_B + (C_{N_\alpha})_F]$$

$$\approx \{[(x_{AC})_F - x_{CG}] / d]\}[(d_F / d)^2 - 2] / (d_F / d)^2, \text{ for Body-Flare Configuration}$$



Example of Static Margin for THAAD Flare at Launch



$(C_{N_\alpha})_F = 2 [(18.7 / 14.6)^2 - 1] = 1.28$ per rad

$(x_{ac})_F = x_F + 0.33\, l_F [2 (d_F / d) + 1] / [(d_F / d) + 1] = 230.9 + 0.33 (12.0)[2 (18.7 / 14.6) + 1] / [(18.7 / 14.6) + 1] = 237.1$ in

$(x_{ac})_B = 0.63 (91.5) = 57.7$ in

$(x_{CG})_{Launch} = 146.9$ in

From More Precise Equation, Compute:

$(x_{AC} - x_{CG})_{Launch} / d = - \{2 \{[146.9 - 57.7] / 14.6\} + 1.28 \{[146.9 - 237.1] / 14.6\}\} / [2 + 1.28] = - 1.31$

From Previous Figure of THAAD:

$d_F / d = 18.7 / 14.6 = 1.28$

$[(x_{ac})_F - x_{CG}] / d = (237.1 - 146.9) / 14.6 = 6.18$

For Approximate Equation, Compute:

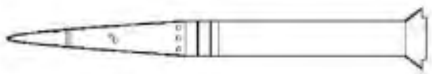
$(x_{AC} - x_{CG})_{Launch} / d = [(237.1 - 146.9) / 14.6][(18.7 / 14.6)^2 - 2] / (18.7 / 14.6)^2 = - 1.36$

Note: THAAD is statically unstable at launch

Note: Based on slender body theory (Pitts, et al) for axisymmetric body.

Nomenclature: x_{AC} = location of aerodynamic center, x_{CG} = location of center of gravity, d = diameter of body, $(C_{N_\alpha})_B$ = normal force coefficient variation with angle of attack for body = 2 / rad, $(x_{AC})_B$ = location of aerodynamic center for body = 0.63 l_N, l_N = length of nose, $(C_{N_\alpha})_F$ = normal force coefficient variation with angle of attack for flare = $2 [(d_F / d)^2 - 1]$, $(x_{AC})_F$ = location of aerodynamic center for flare = $x_F + 0.33\, l_F [2 (d_F / d) + 1] / [(d_F / d) + 1]$, d_F = diameter of flare

Tail Stabilizer Advantages: Drag & Flight Control, Flare Stabilizer Advantages: Aero Heating and Stability

Type Stabilizer	Drag	Span	Heating	Stability Variation	Flight Control
Flare (e.g., THAAD) 	–	◒	◒	●	–
Tails (e.g., Standard Missile) 	◒	○	–	–	●

Note: ● Superior ◒ Good ○ Average – Poor

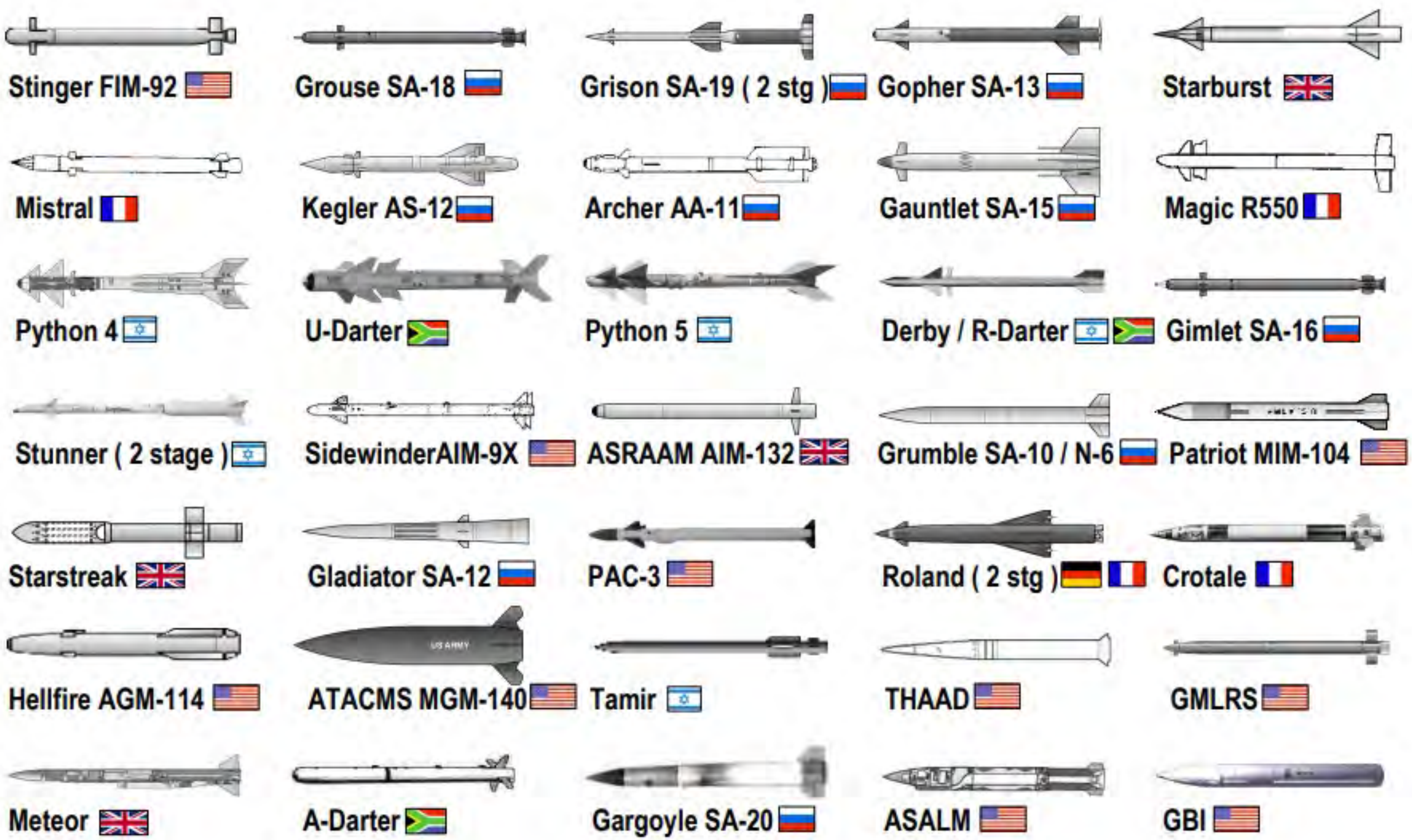
Most Supersonic Missiles Do Not Have Wings

Supersonic Missiles Usually Do Not Have Wings (e.g., PAC-3) 

- Longer Range for M > 4
- Longer Range for M > 2 @ Low Altitude*
- Higher Max Angle of Attack
- Better Fitment with Launch Platform

*Adding wing may be best @ higher altitude

Most Supersonic Missiles Do Not Have Wings (cont)



Most Subsonic Cruise Missiles Have Relatively Large Wings

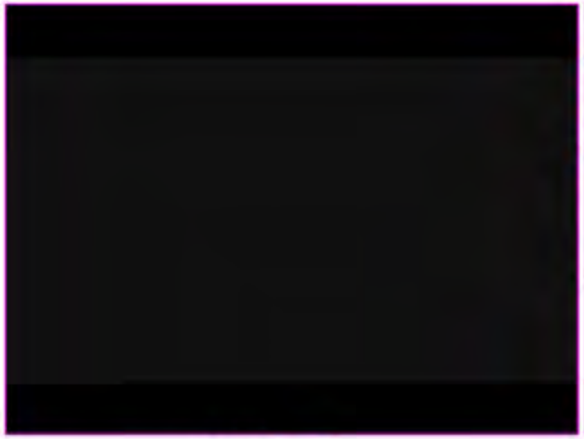
Subsonic Cruise Missiles Usually Have Large Wings (e.g., Taurus) 

- Longer Range for M < 1*
- Lower Guidance Time Constant**
- Higher Normal Acceleration / Maneuverability**
- Less Seeker Error from Dome Error Slope***
- Less Wipe Velocity for Warhead at Impact***
- Less Gimbal Requirement for Seeker***

*Relatively small wing may be best @ sea level
**Based on assumption of aero flight control
***Lower angle of attack required for maneuver

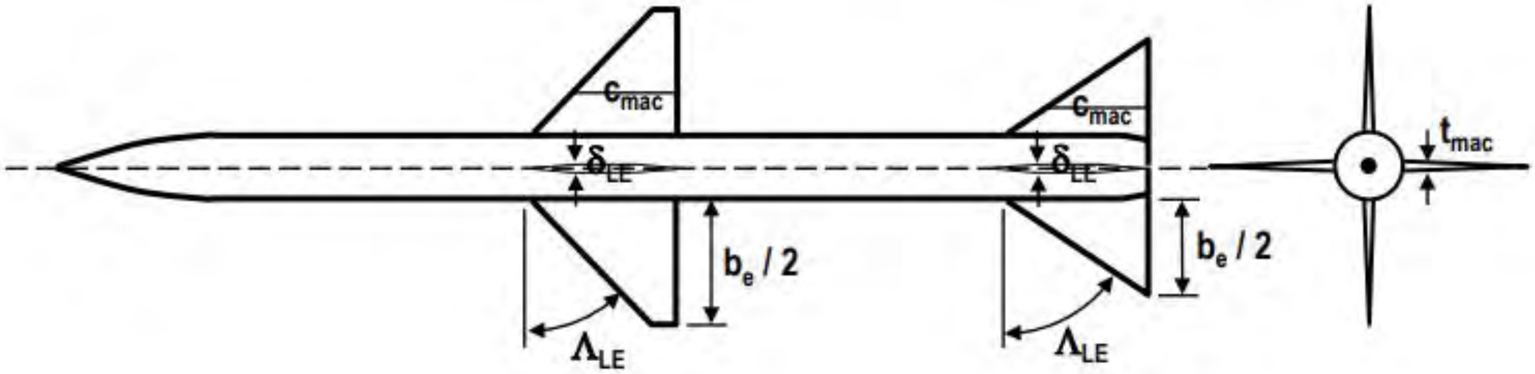
Most Subsonic Cruise Missiles Have Relatively Large Wings (Cont)

Examples of Guided Bombs That Have Wings for Extended Range



Videos: GBU-39 SDB KGK Spice (Courtesy of Defense Update)

Definition of Planar Aerodynamic Surfrace Geometry Parameters

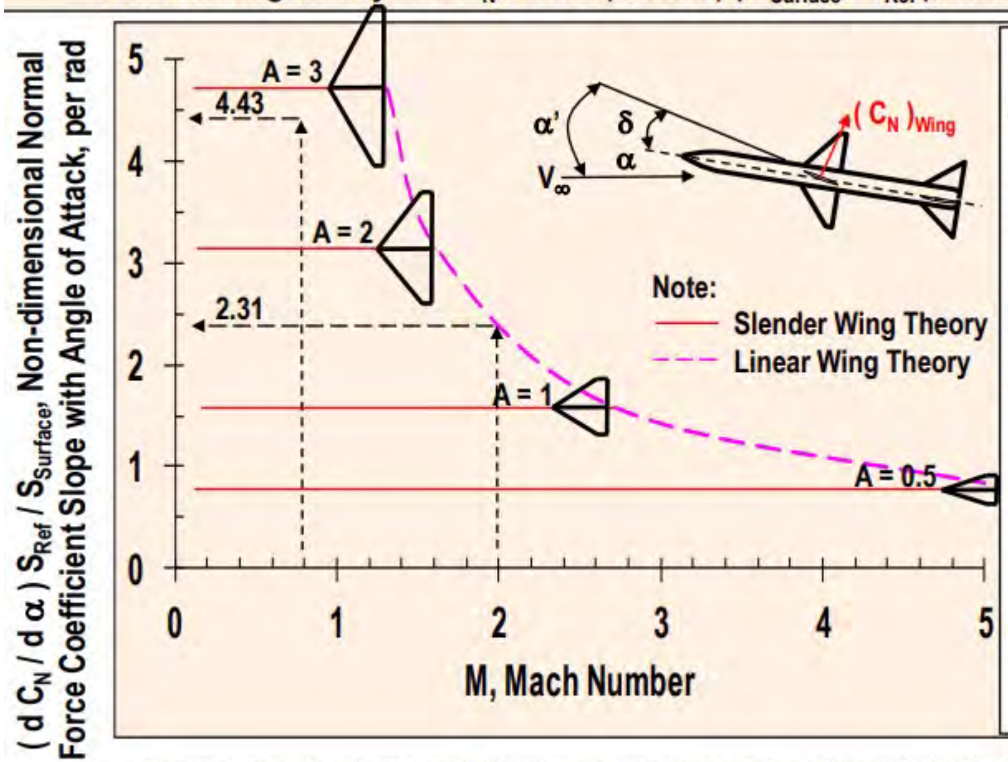


- c_{mac} = mean aerodynamic chord length
- Λ_{LE} = leading edge sweep angle
- δ_{LE} = leading edge section total angle
- t_{mac} = max thickness of mean aerodynamic chord
- b_e = span of exposed planform
- S_e = area of exposed planform
- $A_e = b_e^2 / S_e$ = aspect ratio of exposed planform

Normal Force Coefficient of a Planar Surface (Wing, Tail, Canard) is Higher at Low Mach

Linear Wing Theory ⇨ $d C_N / d\alpha \approx [4/(M^2-1)^{1/2}](S_{Surface}/S_{Ref})$, if $\alpha' < \approx 10$ deg, $M > \{1+[8/(\pi A)]^2\}^{1/2}$

Slender Wing Theory ⇨ $d C_N / d\alpha \approx (\pi A/2)(S_{Surface}/S_{Ref})$, if $\alpha' < \approx 10$ deg, $M < \{1+[8/(\pi A)]^2\}^{1/2}$



Example for Rocket Baseline Missile Wing:

$A_W = b^2 / S_W = 2.82$

$S_W = 2.55\ ft^2$

$S_{Ref} = 0.349\ ft^2\ (0.0324\ m^2)$

Assume M = 0.8

Test: $(M_{max})_{Slender\ Wing\ Theory} = \{1+[8/(\pi A)]^2\}^{1/2} = 1.35$

M = 0.8 < 1.35, use slender wing theory

Compute

$(d C_N / d\alpha)\, S_{Ref}/S_{Surface} \approx [\pi(2.82)/2] = 4.43$ per rad

$(d C_N / d\alpha) = 4.43(2.55)/0.349 = 32.4$ / rad

Assume M = 2

M = 2 > 1.35, use linear wing theory

Compute

$(d C_N / d\alpha)(S_{Surface}/S_{Ref}) \approx [4/(2^2-1)^{1/2}] = 2.31$

Note for Rocket Baseline Missile: $(d C_N / d\alpha)_{wing} \approx d C_N / d\delta$

Note: Linear wing theory and slender wing theory equations from USAF Stability and Control DATCOM

Note: Slender wing theory good accuracy limited to A < ≈ 2 for small effects of compressibility, wing sweep, and taper ratio

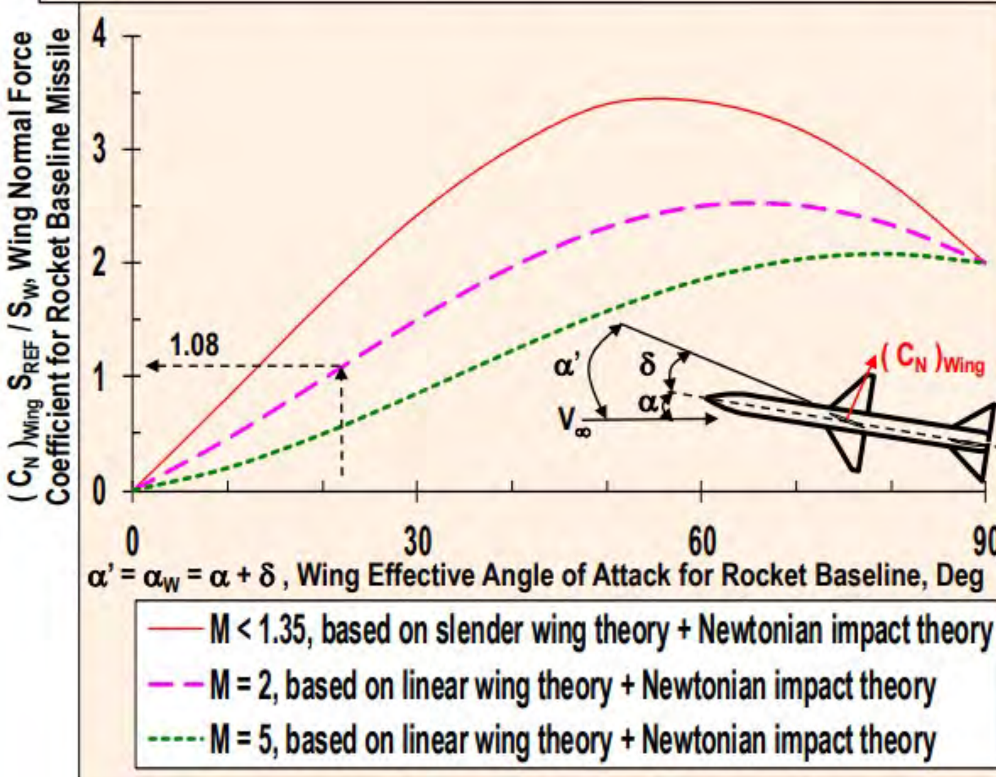
Nomenclature: A = Aspect ratio, $S_{Surface}$ = Surface planform area, S_{Ref} = Reference area, α = Angle of attack

High Normal Force for a Planar Surface Occurs at High Local Angle of Attack

$$|(C_N)_{Surface}| = \left(\frac{4\,|\sin\alpha'\cos\alpha'|}{(M^2-1)^{1/2}} + 2\sin^2\alpha'\right)(S_{Surface}/S_{Ref}),\ \text{if } M > \{1+[8/(\pi A)]^2\}^{1/2}$$

$$|(C_N)_{Surface}| = [(\pi A/2)\,|\sin\alpha'\cos\alpha'| + 2\sin^2\alpha'](S_{Surface}/S_{Ref}),\ \text{if } M < \{1+[8/(\pi A)]^2\}^{1/2}$$

Nomenclature: A = Aspect Ratio, $S_{Surface}$ = Surface Planform Area, S_{Ref} = Reference Area, α' = Effective Angle of Attack, M = Mach Number

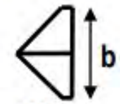


Example for Rocket Baseline Missile Wing

$A_W = b^2 / S_W = 2.82$, $S_W = 2.55\ ft^2$, $S_{Ref} = 0.349\ ft^2$

$\delta = 12.6$ deg, $\alpha = 9.4$ deg

$\alpha' = \alpha_W = \alpha + \delta = 22$ deg



Assume M = 2

$\{1 + [8/(\pi A)]^2\}^{1/2} = 1.35 < M$

Since M > 1.35, use linear wing theory + Newtonian theory

$(C_N)_{Wing}\ S_{Ref} / S_W = 4 \sin 22 \text{ deg} \cos 22 \text{ deg} / (2^2 - 1)^{1/2} + 2 \sin^2 22 \text{ deg} = 0.80 + 0.28 = 1.08$

$(C_N)_{Wing} = 1.08\ (2.55) / 0.349 = 7.91$

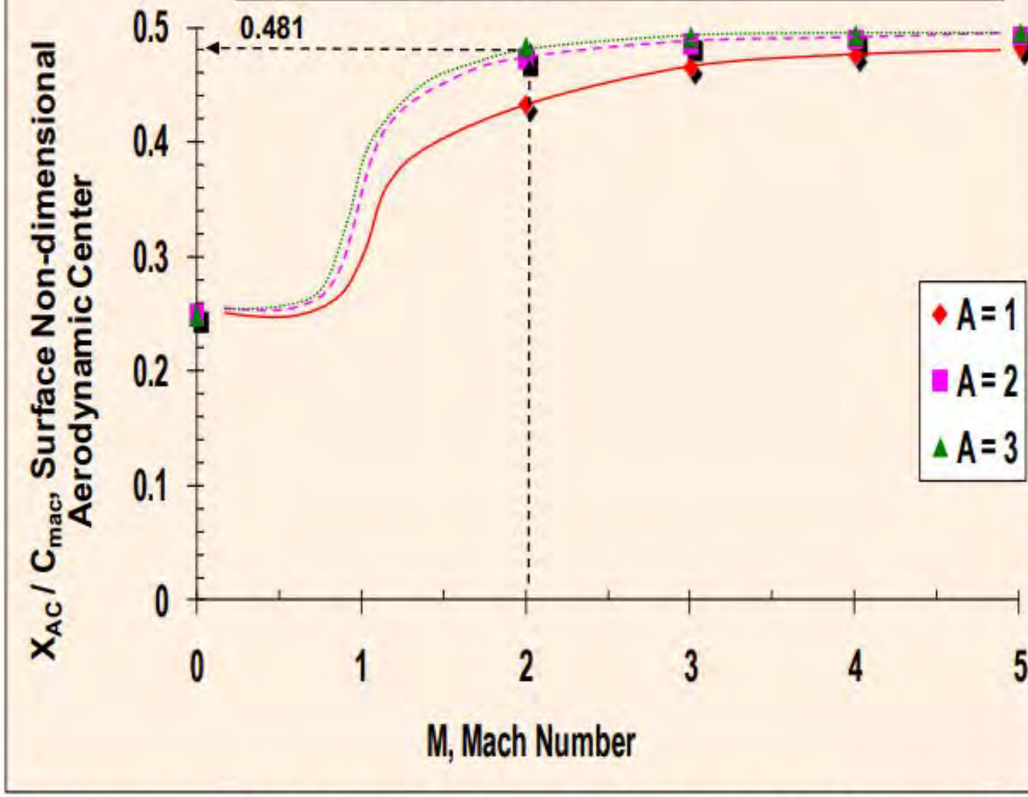
Note: Rocket baseline missile wing typically has much larger C_N than that of the body. For example at $\alpha = 9.4$ deg: $(C_N)_{Body} = \sin(2\alpha)\cos(\alpha/2) + 1.3\,(l/d)\sin^2\alpha = 0.95$

$(C_N)_{Body}$ is 12% of $(C_N)_{Wing}$ (0.95 vs 7.91)

Aerodynamic Center of a Planar Surface Moves Aft with Increasing Mach Number

$$(x_{AC} / c_{mac})_{Surface} = \frac{A (M^2 - 1)^{1/2} - 0.67}{2A (M^2 - 1)^{1/2} - 1}, \text{ if } M > \sim 2$$

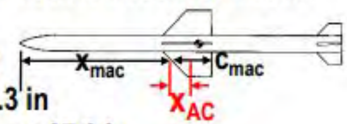
$$(x_{AC} / c_{mac})_{Surface} \approx 0.25, \text{ if } M < \sim 0.7$$



Reference: Chin, S.S., *Missile Configuration Design*, McGraw-Hill Book Company, 1961

Note: Based on linear wing theory
Thin wing ⇒ $M (t / c) << 1$, Angle of attack $\alpha < 10$ deg
t = thickness, α = Angle of attack
$(x_{AC})_{Surface}$ = Surface aerodynamic center distance from leading edge of c_{mac}
c_{mac} = Mean aerodynamic chord
A = Aspect ratio = b^2 / S

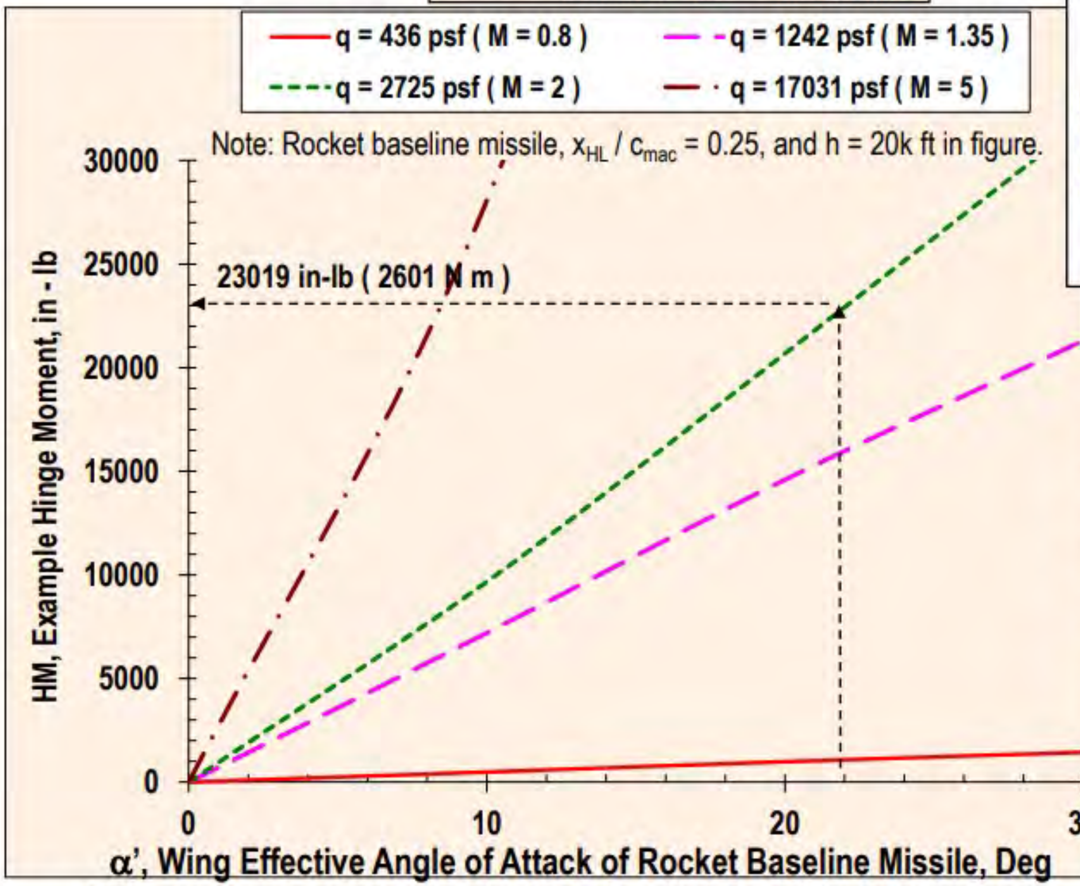
Example: Rocket Baseline Missile Wing



$A = 2.82$
$c_{mac} = 13.3$ in
$(x_{MAC})_{Wing} = 67.0$ in
$M = 2$, $\alpha < 10$ deg
$(t / c)_{mac} = 0.044 \Rightarrow M (t / c) = 2 (0.044) = 0.088 << 1$ (thin wing)
Calculate:
$(x_{AC} / c_{mac})_{Wing} = [2.82 (2^2 - 1)^{1/2} - 0.67] / [2 (2.82)(2^2 - 1)^{1/2} - 1] = 0.481$
$(x_{AC})_{Wing} = 0.481 (13.3) = 6.4$ in from mac leading edge = 73.4 in from nose tip

Hinge Moment Increases with Dynamic Pressure and Effective Angle of Attack

$$HM = N_{Surface} (x_{AC} - x_{HL})_{Surface}$$



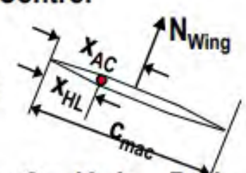
Note: $x_{AC} \leq X_{HL}$ requires increased actuator bandwidth but has lower hinge moment.

Note: Conceptual design equations in figure are based on linear wing theory, slender wing theory, and thin wing (M (t / c) << 1)

$N_{Surface}$ = Normal force on a surface (two panels)

$(x_{AC} - x_{HL})_{surface}$ = Distance from surface aerodynamic center to hinge line of surface

Example for Rocket Baseline Missile Wing Control



c_{mac} = 13.3 in

x_{HL} = 0.25 c_{mac}

S_{Ref} = 0.349 ft²

S_W = 2.55 ft²

Wing deflection δ = 13 deg, Body angle of attack α = 9 deg

$\alpha' = \alpha_W = \alpha + \delta = 22°$

M = 2, h = 20k ft, q = 2725 psf

Compute:

$N_{Wing} = C_{N_{Wing}}\, q\, S_{Ref}$ = 7.91 (2725) (0.349) = 7525 lb

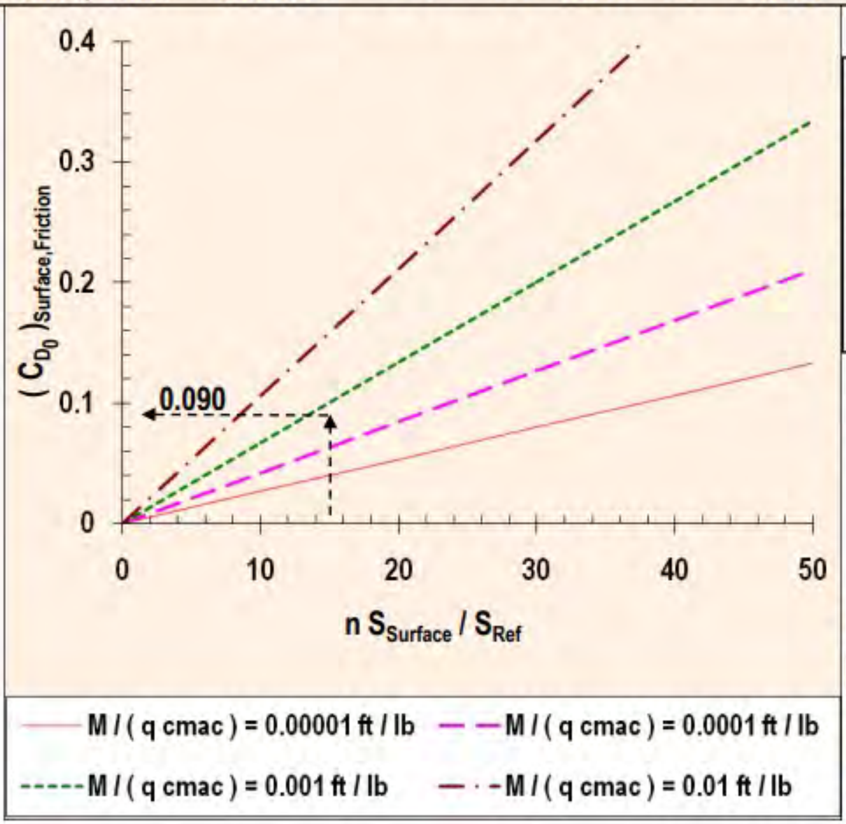
x_{AC} / c_{mac} = 0.48

HM = 7525 (0.48 – 0.25) (13.3) = 23019 in–lb for two panels

Skin Friction Drag is Lower for Small Surface Area

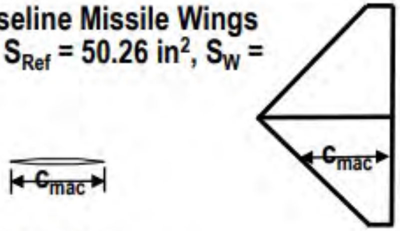
$$(C_{D_0})_{Surface} = (C_{D_0})_{Surface,Friction} + (C_{D_0})_{Surface,Wave}$$

$$(C_{D_0})_{Surface,Friction} = n_{Surface}\{0.0133\,[M/(q\,c_{mac})]^{0.2}\}(2\,S_{Surface}/S_{Ref})$$



Nomenclature:
C_{D_0} = zero-lift drag coefficient
$n_{Surfaces}$ = number of surface planforms (cruciform = 2)
q = dynamic pressure in psf
c_{mac} = length of mean aero chord in ft
M = Mach number
$S_{surface}/S_{ref}$ = surface area / reference area

Example for Rocket Baseline Missile Wings ($n_W = 2$, $c_{mac} = 1.108$ ft, $S_{Ref} = 50.26\ in^2$, $S_W = 367\ in^2$)



Assume:
M = 2, h = 20k ft (q = 2725 psf) ⇒
$n\,S_{Surface}/S_{ref} = 2(367)/50.26 = 14.60$
$M/(q\,c_{mac}) = 2/[2725(1.108)] = 0.000662$ ft / lb

Compute from equation:
$(C_{D_0})_{Wing,Friction} = 2\{\{0.0133\{2/[(2725)(1.108)]\}^{0.2}\}\,[2(367)/50.26] = 0.090$

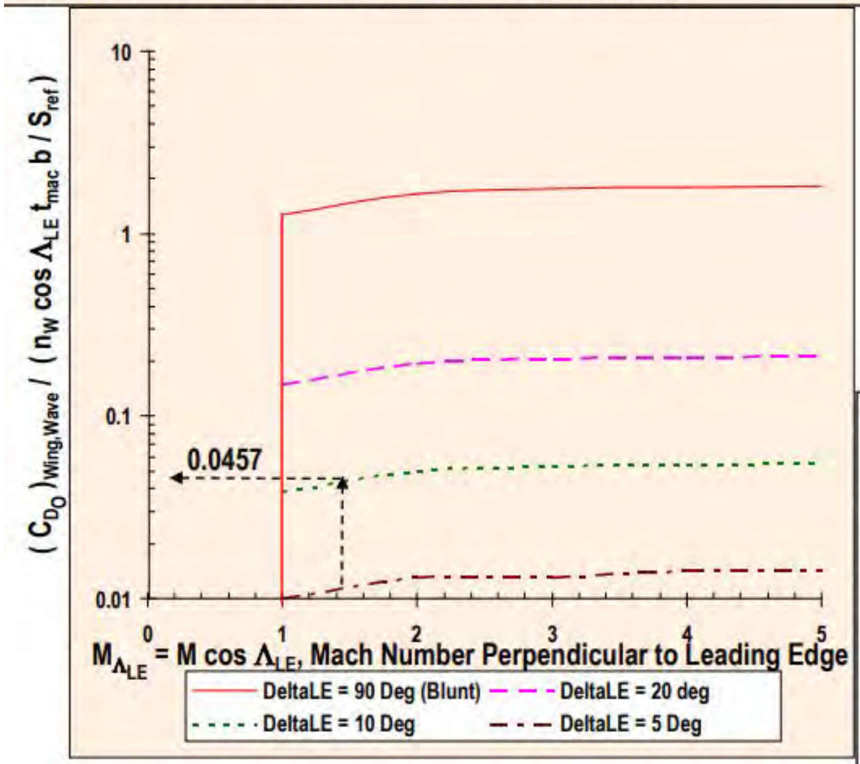
Based on Jerger reference

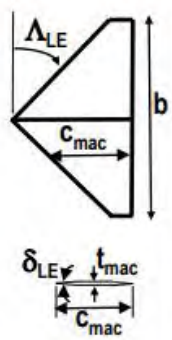
Supersonic Drag of Planar Surface is Smaller if Leading Edge Has Sweep and Small Section Angle

$$(C_{D_0})_{Surface} = (C_{D_0})_{Surface,Wave} + (C_{D_0})_{Surface,Friction}$$

$$(C_{D_0})_{Surface,Wave} = n_{Surface} (1.429 / M_{\Lambda_{LE}}^2)\{(1.2 M_{\Lambda_{LE}}^2)^{3.5} \left(\frac{2.4}{2.8 M_{\Lambda_{LE}}^2 - 0.4} \right)^{2.5} - 1 \} (\sin^2 \delta_{LE} \cos \Lambda_{LE}\ t_{mac}\ b / S_{Ref})$$

Equation based on modified Newtonian impact theory





Note:
C_{D_0} = zero-lift drag coefficient
n_W = number of wings (cruciform = 2)
δ_{LE} = leading edge section angle
Λ_{LE} = leading edge sweep angle
t_{mac} = max thickness of mac
b = span
S_{Ref} = reference area

Example for Rocket Baseline Missile Wing:

δ_{LE} = 10.01 deg, Λ_{LE} = 45 deg, t_{mac} = 0.585 in, b = 32.2 in, S_{Ref} = 50.26 in^2, M = 2 ⇒ $M_{\Lambda_{LE}}$ = 2 cos 45 deg = 1.41

$(C_{D_0})_{Wing,Wave} / (n_W \cos \Lambda_{LE}\ t_{mac}\ b / S_{Ref}) = 0.0457$

$(C_{D_0})_{Wing,Wave} = 0.0457 (2)(0.707)(0.585)(32.2) / 50.26 = 0.024$

From previous figure, $(C_{D_0})_{Wing,Friction} = 0.090$

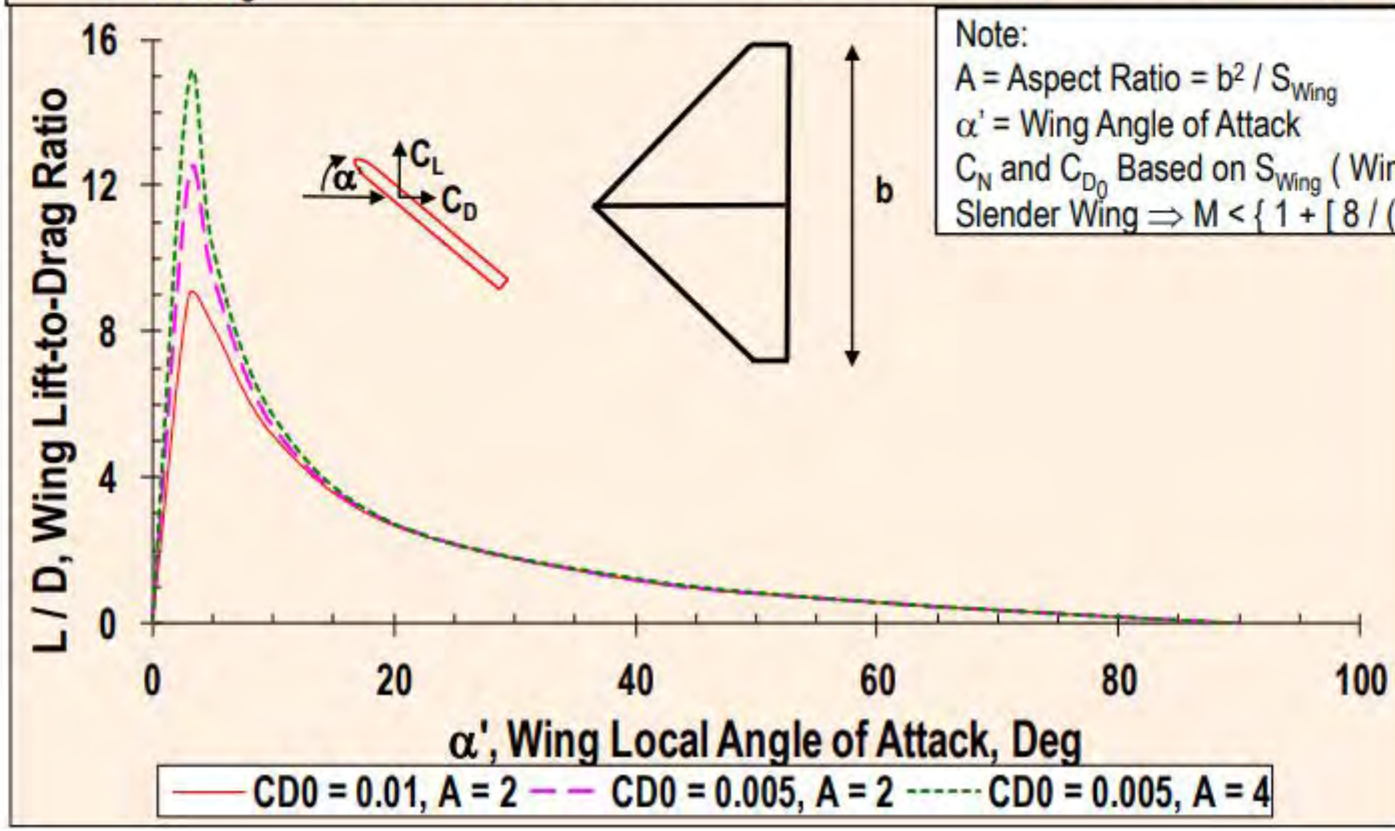
$(C_{D_0})_{Wing} = 0.024 + 0.090 = 0.11$

Note: Most of rocket baseline missile wing drag is skin friction drag .

Wing Subsonic Aero Efficiency L/D is Driven by Angle of Attack, C_{D_0}, and Aspect Ratio

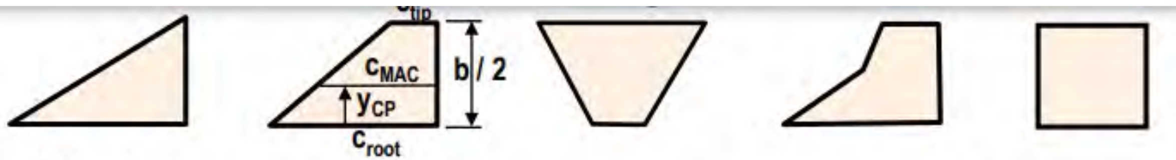
$$L / D = C_L / C_D = \frac{C_N \cos\alpha - C_A \sin\alpha}{C_N \sin\alpha + C_A \cos\alpha} \approx \frac{C_N \cos\alpha - C_{D_0} \sin\alpha}{C_N \sin\alpha + C_{D_0} \cos\alpha}$$

$$\left|(C_N)_{Wing}\right| = [\,(\pi A/2)\left|\sin\alpha' \cos\alpha'\right| + 2\sin^2\alpha'\,]$$, if slender wing



Note:
A = Aspect Ratio = b^2 / S_{Wing}
α' = Wing Angle of Attack
C_N and C_{D_0} Based on S_{Wing} (Wing Planform Area)
Slender Wing $\Rightarrow M < \{ 1 + [8 / (\pi A)]^2 \}^{1/2}$

Planar Surface (Wing, Tail, Canard) panel Geometry is a Tradeoff with Many Considerations



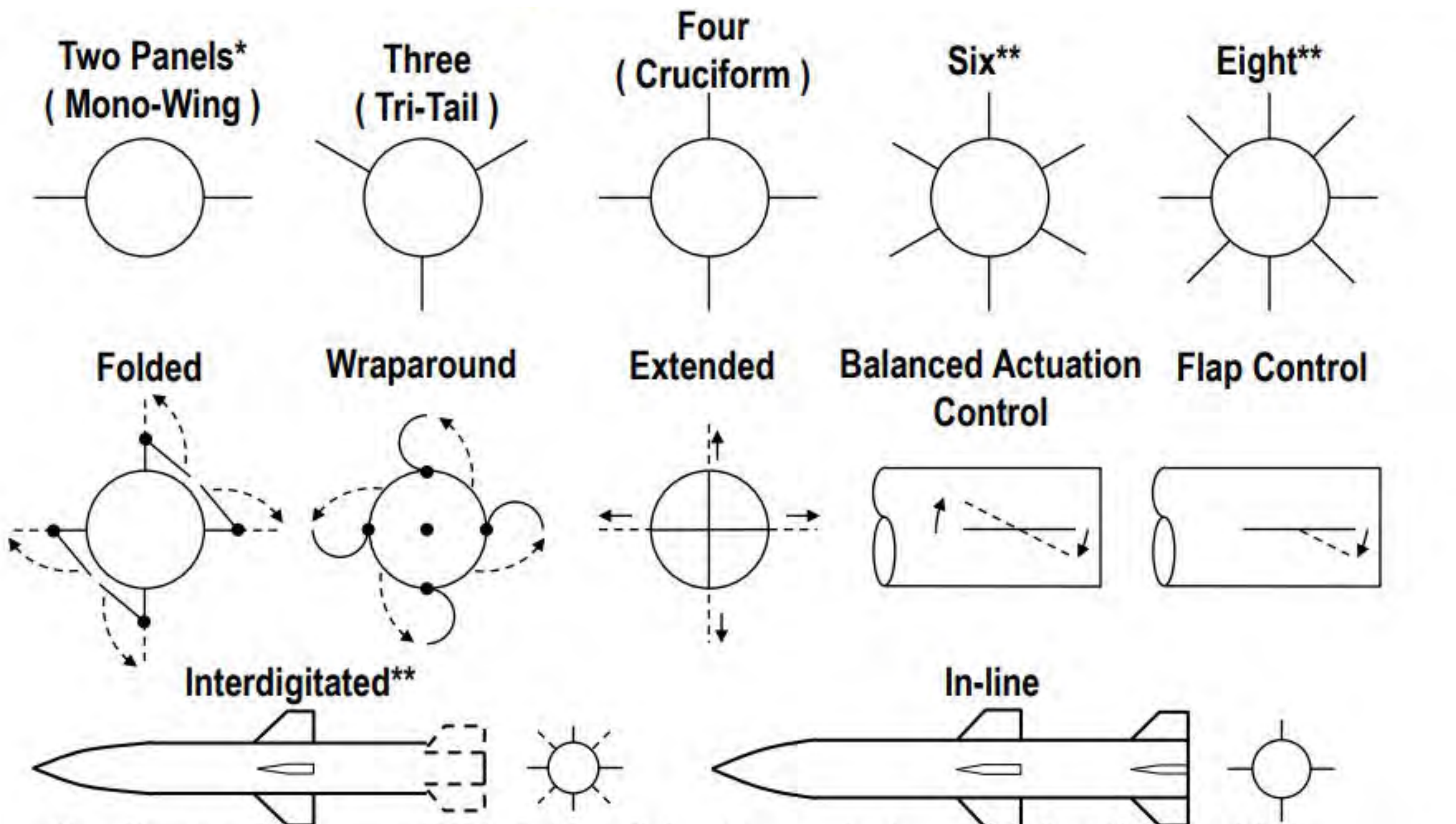
Parameter	Triangle (Delta)	Aft Swept Leading Edge	Forward Swept Leading Edge	Double Swept Leading Edge	Rectangle
Δx_{AC}, Δ Hinge Moment	–	○	○	●	◒
y_{CP} (Bending / Friction)	●	◒	–	○	–
M > 1 Drag	●	◒	◒	○	–
Radar Cross Section	◒	◒	●	○	–
Span Constraint	–	○	○	○	◒
Stability & Control	○	●	◒	●	○
Aeroelastic Stab.	●	◒	–	◒	○
Weight	●	◒	○	◒	–

Δx_{AC} = Change in aerodynamic center location with Mach number
λ = Taper ratio = c_{tip} / c_{root}
A = Aspect ratio = $b^2 / S = 2\,b / [(1 + \lambda)\,c_{root}]$
y_{CP} = Outboard center-of-pressure = $(b/6)(1 + 2\lambda)/(1 + \lambda)$
c_{MAC} = Mean aerodynamic chord = $(2/3)\,c_{root}\,(1 + \lambda + \lambda^2)/(1 + \lambda)$

Note: Superior ● Good ◒ Average ○ Poor –

Note: Based on equal surface area and equal span. Surface area often has more impact than geometry.

Examples of Wing / Stabilizer / Flight Control Surface Arrangements and Alternatives



*Note: Negative rolling moment with sideslip ($C_{l_\beta} < 0$) for high mono-wing (more statically stable). $C_{l_\beta} > 0$ for low mono-wing (less statically stable). $C_{l_\beta} \approx 0$ is desirable.

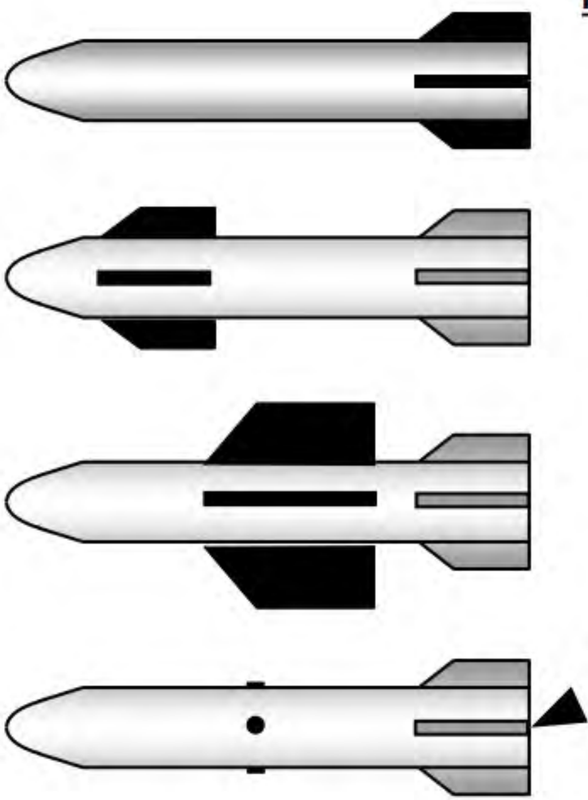
**Note: More than four tails have lower induced roll C_{l_ϕ} and more pitch / yaw stability for span limit (e.g., JSOW). Free-to-roll tails and interdigitated surfaces also have lower induced roll C_{l_ϕ}.

Most Missiles have Aero Flight Control with Four Control Surfaces Providing Pitch, Yaw, Roll Control

Control Integ	Control Surfaces	Example	Control Effect	Cost	Packaging
♦Pitch / Yaw	2	Stinger FIM-92	–	●	●
♦ Pitch / Roll	2	ALCM AGM-86	–	●	●
♦Pitch / Roll + Yaw	3	JASSM AGM-158	–	◒	●
♦Pitch / Yaw / Roll	3	SRAM AGM-69	–	◒	◒
♦Pitch / Yaw / Roll (Cruciform Most Common Type for Missiles)	4	Adder AA-12	○	○	○
♦Pitch + Yaw + Roll	5	Kitchen AS-4	●	–	–
♦Pitch / Yaw + Roll	6	Derby / R-Darter	◒	–	–
♦Pitch / Yaw / Roll (Blended Canard – Tail Control)	8	Stunner	●	–	–

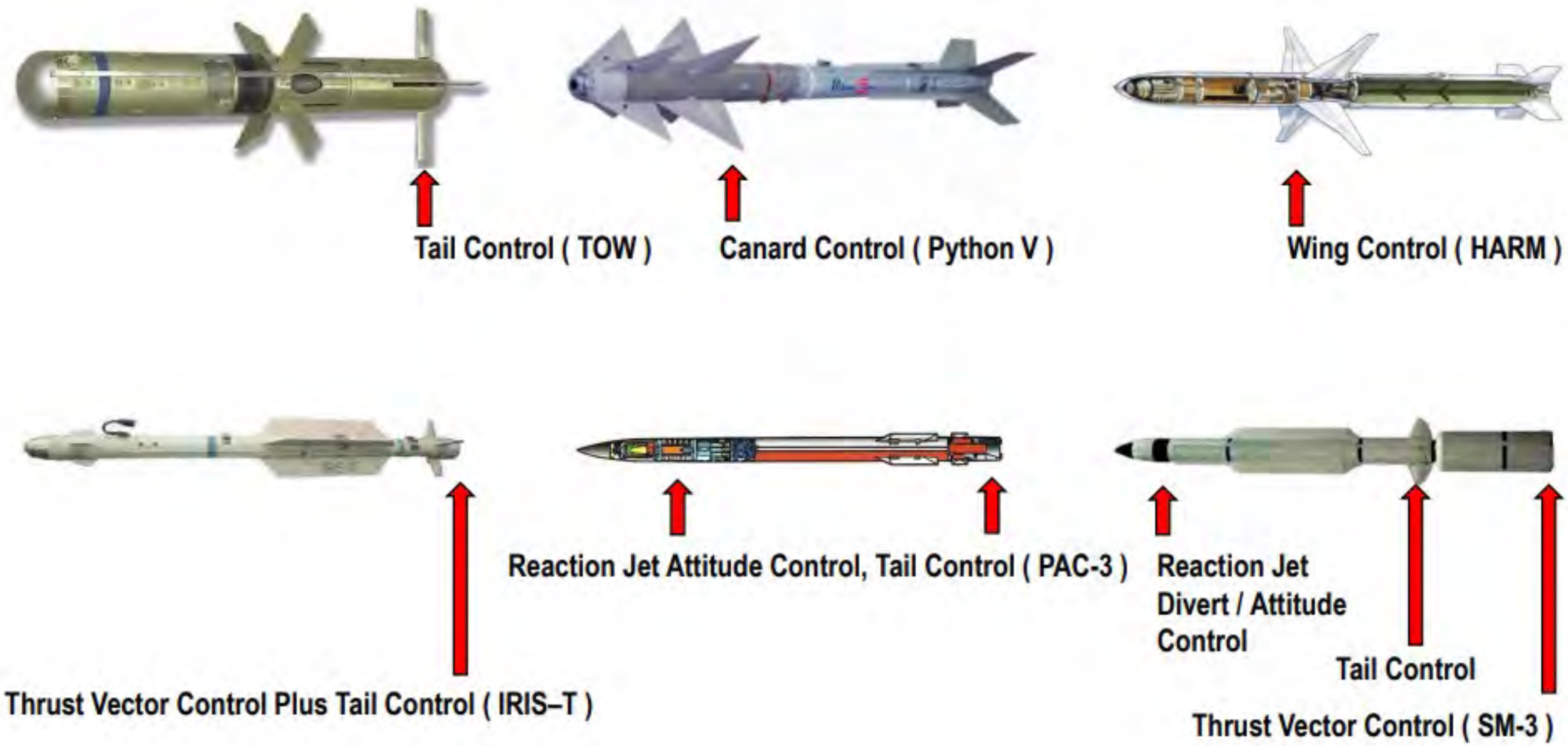
Note: ● Superior ◒ Good ○ Average – Poor

There are Many Flight Control Aerodynamic Configuration Alternatives

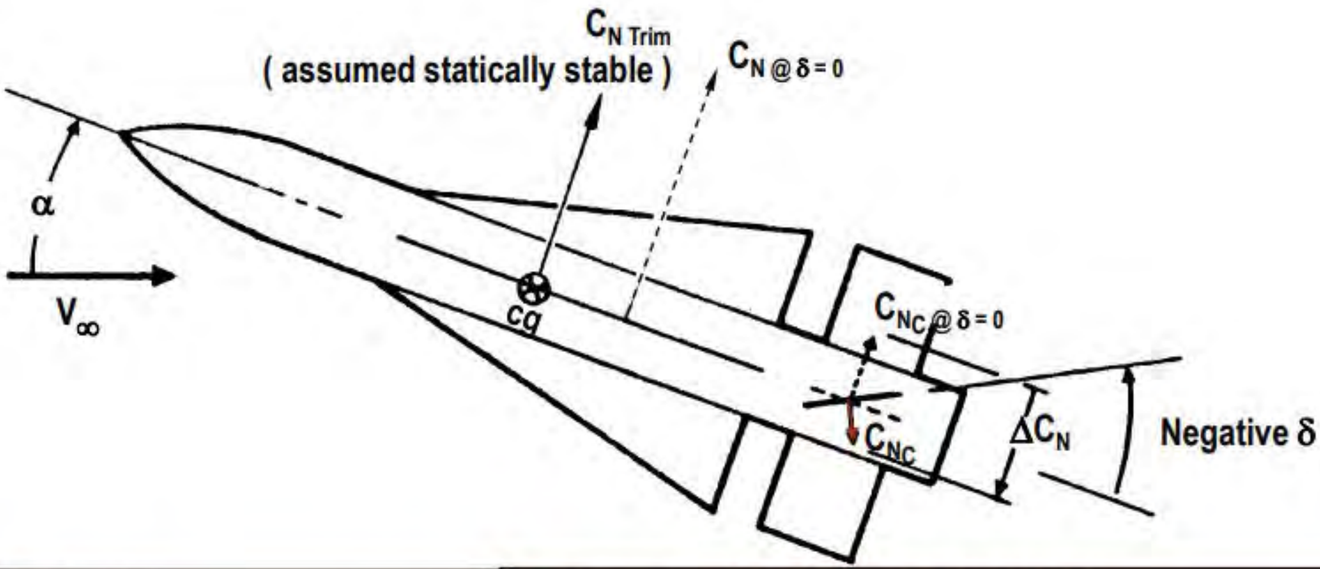


Flight Control	Control Design Alternatives	Fixed Surface Alternatives
Tail	Cruciform (4) Tri-tail (3) Not Compressed Folded Wraparound Switchblade	Wingless Wing Strake / Canard In Line with Controls Interdigitated with Controls Number (2, 3, 4)
Canard	Above Rolling Airframe (2)	Tail (3, 4, 6, 8) Tail + Wing In Line with Controls Interdigitated with Controls
Wing	Above	Tail (3, 4, 6, 8) Strake / Canard & Tail In Line with Controls Interdigitated with Controls
Thrust Vector Control (TVC) or Reaction Jet Control	Movable Nozzle Jet Tab Jet Vane Axial Plate Secondary Injection Normal Reaction Jet Spanwise Reaction Jet	Tail (3, 4, 6, 8) Tail + Canard / Strake Tail + Wing

Missile Flight Control Alternatives are Driven by Maneuverability, Packaging Efficiency, and Cost



Tail Flight Control is Efficient at High Angle of Attack



☺ Efficient Packaging ☺ Low Hinge Moment / Actuator Torque ☺ Low Induced Rolling Moment ☺ Efficient at High α	☹ Decreased Lift if Statically Stable ☹ Initial Motion May Be in Wrong Direction if Statically Stable

Note:
$C_{N\ Trim}$ = Trim normal force coefficient
$C_{N\ @\ \delta=0}$ = Normal force coefficient at zero control deflection
C_{N_C} = Normal force coefficient from control deflection
$\Delta C_N = C_{N_C} - C_{N_C\ at\ \delta=0}$

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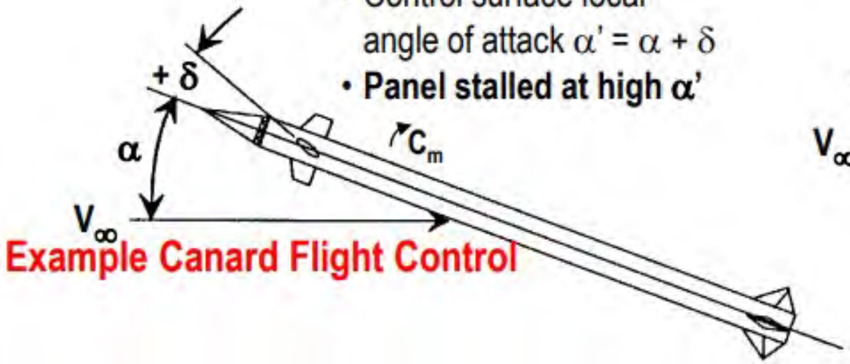
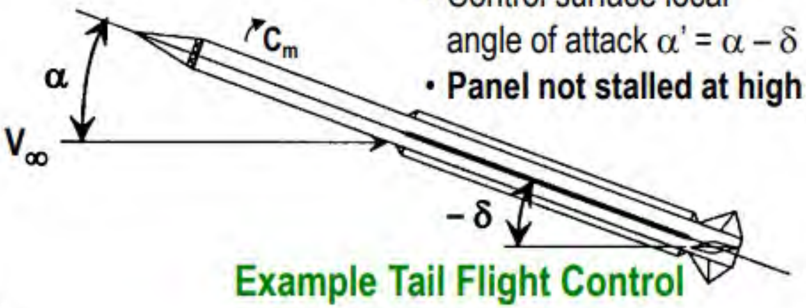
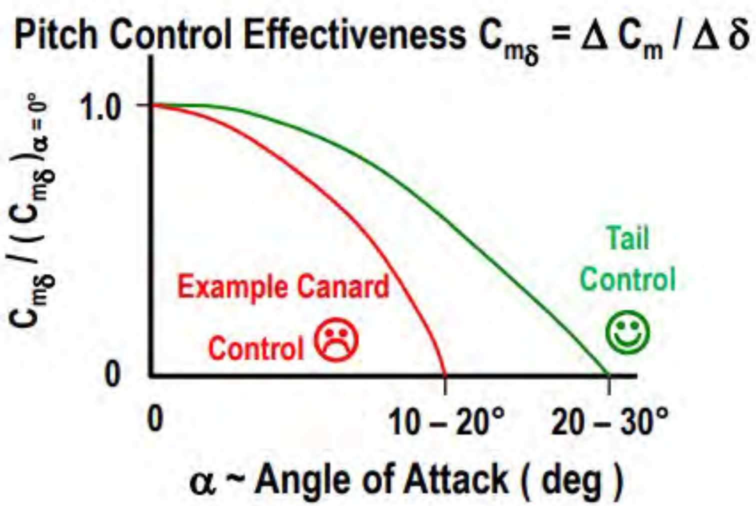
Tail Flight Control Can Usually Operate at Higher Angle of Attack Than Canard Flight Control

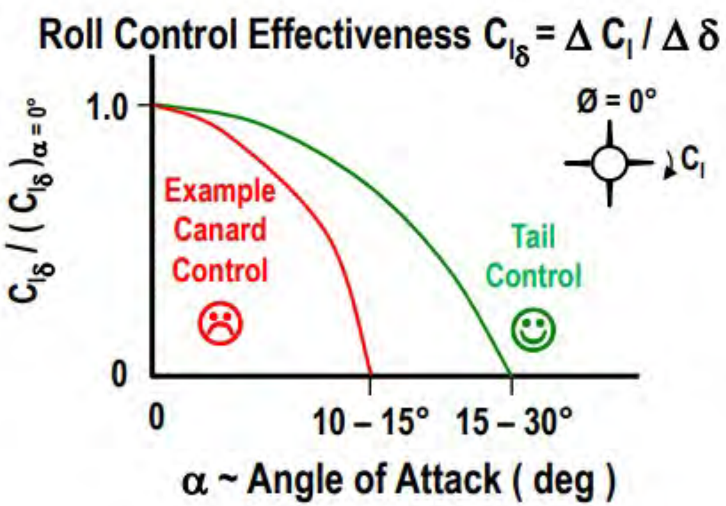
- Assumed static stability
- Control surface local angle of attack $\alpha' = \alpha + \delta$
- **Panel stalled at high α'**



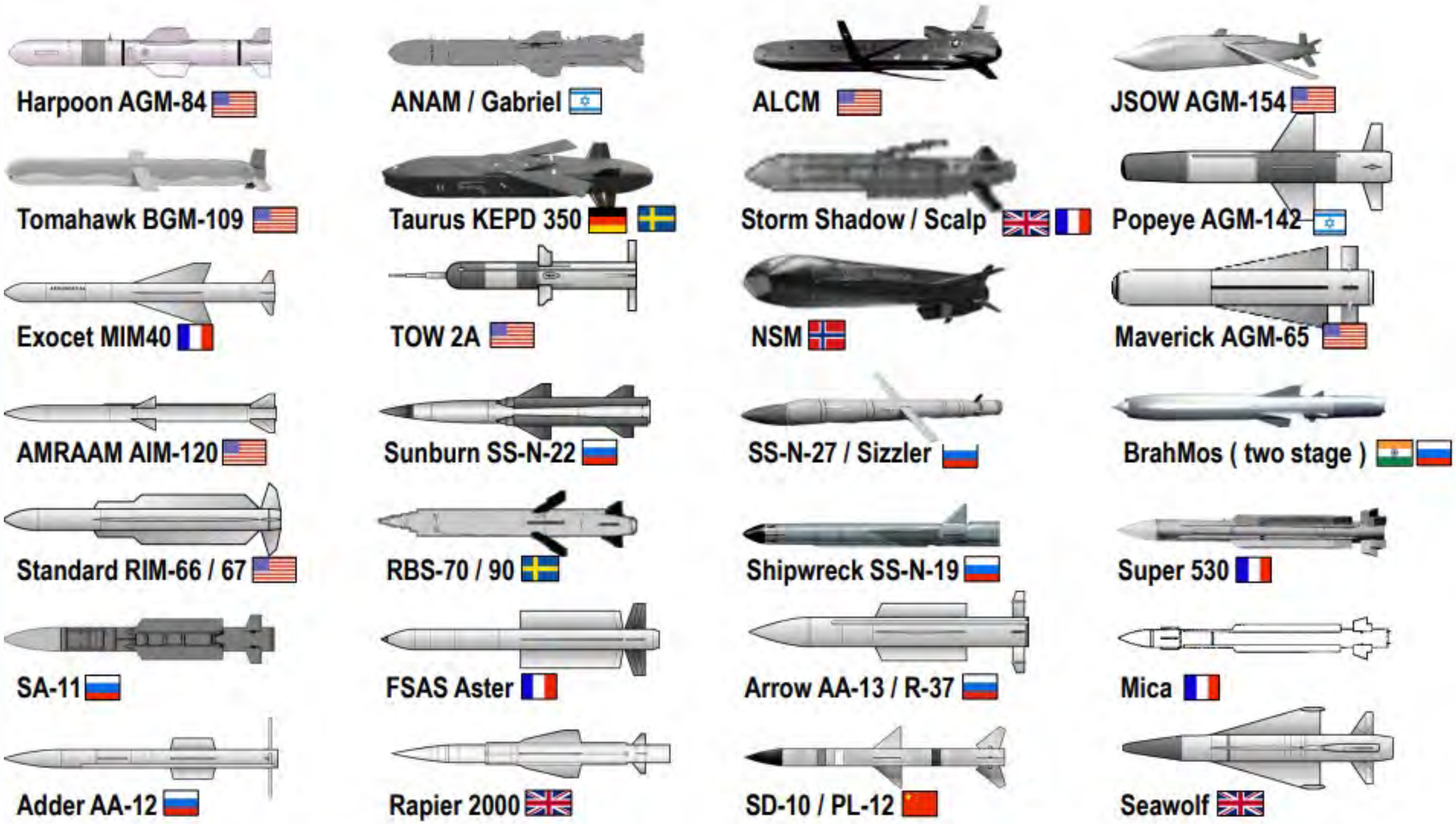
- Assumed static stability
- Control surface local angle of attack $\alpha' = \alpha - \delta$
- **Panel not stalled at high α**





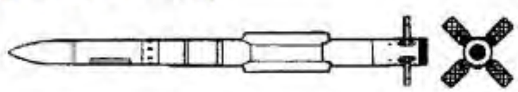


About 70% of Tail Flight Control Missiles have Wings



Note: Wing aero center should be near missile center of gravity (≈ 50% of missile length), to avoid large shift in static margin.

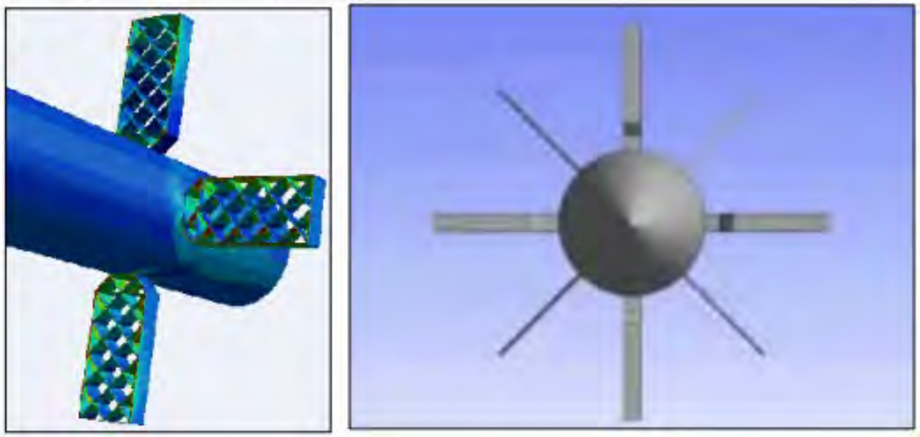
Tail Flight Control Alternatives: Conventional Balanced Actuation Fin, Flap, and Lattice Fin

Type of Tail Flight Control	Control Effectiveness	Drag	Hinge Moment	Radar Cross Section
♦ Conventional Balanced Fin (e.g., ASRAAM AIM-132) 	Good	Superior	Good	Good
♦ Flap (e.g., Hellfire AGM-114) 	Average	Good	Poor	Good / Superior
♦ Lattice Fin (e.g., AA-12 / R-77) 	Superior	Good	Superior	Poor

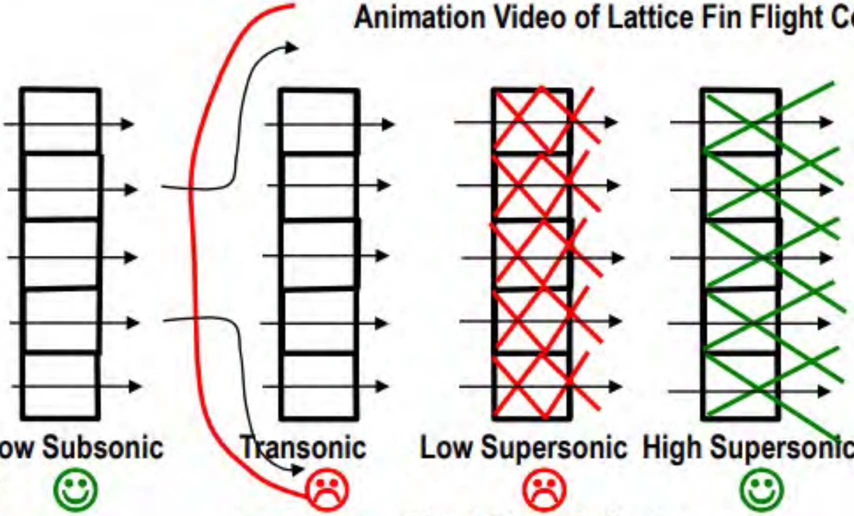
Note: ● Superior ◒ Good ○ Average – Poor

Lattice Fin Flight Control has Advantages for Low Subsonic and High Supersonic Missiles

- **Advantages** ☺
  - High control effectiveness at low subsonic and high supersonic Mach number
  - Low hinge moment
  - Short chord length
- **Disadvantages** ☹
  - High radar cross section (cavities, normal leading edges)
  - High drag at transonic Mach number (choked flow)
  - High drag at low supersonic Mach number (shock wave – boundary layer interaction)
  - Higher leading edge heat transfer at supersonic Mach number



Animation Video of Lattice Fin Flight Control



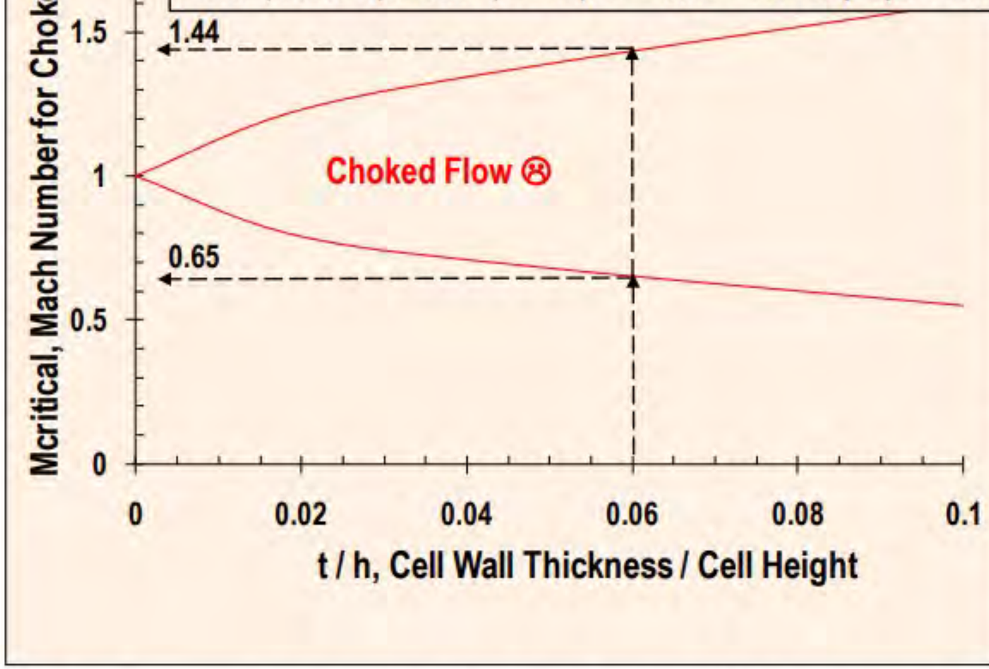
Lattice Fin Chicked Flow is Driven by Lattice Section Thickness and Transonic Mach Number

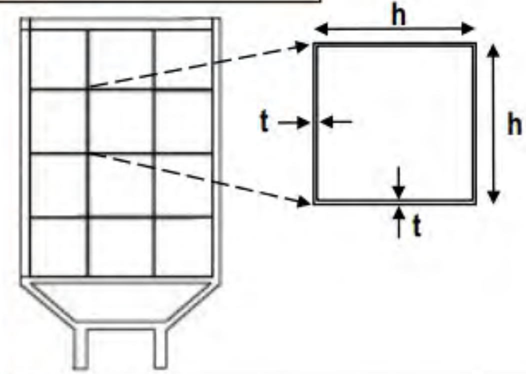
$$A^* / A = [(\gamma + 1)/2]^{(\gamma+1)/[2(\gamma-1)]} M_{critical} \{1 + [(\gamma - 1)/2] M_{critical}^{2}\}^{-(\gamma+1)/[2(\gamma-1)]}$$
$$= (216/125) M_{critical} (1 + M_{critical}^2 / 5)^{-3}$$

Assumptions: Isentropic flow, perfect gas ($\gamma = 1.4$)

$$A^* / A = 1 - 2 (t / h)$$

Assumptions: Square cell ($A = h^2$), $h >> t$, $t >>$ boundary layer thickness

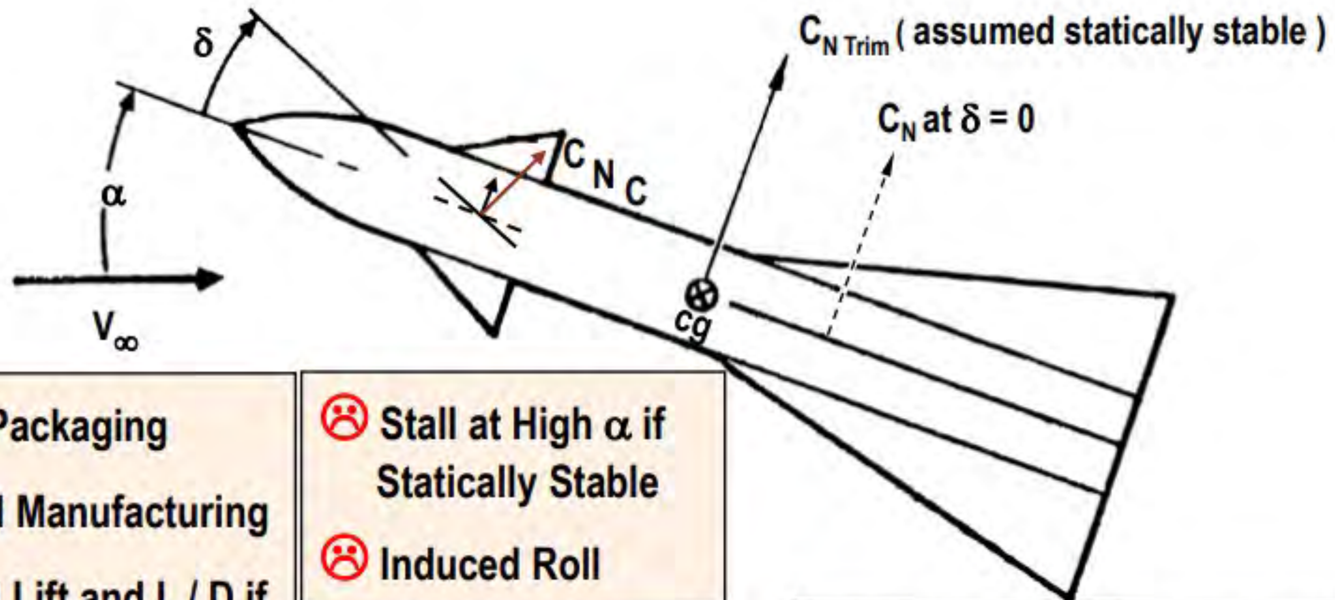




Example:
$h = 1$ in, $t = 0.06$ in (minimum gauge)
$t / h = 0.06 / 1 = 0.06$
$A^* / A = 1 – 2 (0.06) = 0.88 \Rightarrow M_{critical} =$ 0.65, 1.44
Choked Flow (high drag) for $0.65 < M_\infty < 1.44$

Note: A^* = Flow area through inside cell for choked flow (local Mach number = 1), A = Cell outside area, γ = Specific heat ratio = 1.4, $M_{critical}$ = Critical free stream Mach number for choked flow, t = Cell wall thickness, h = Cell height

Conventional Canard Flight Control is Efficient at Low Angle of Attack, but Stalls at High α with Induced Roll



- ☺ Efficient Packaging
- ☺ Simplified Manufacturing
- ☺ Increased Lift and L / D if Statically Stable
- ☺ Initial Motion in Desired Direction if Statically Stable

- ☹ Stall at High α if Statically Stable
- ☹ Induced Roll

Note: Additional forward fixed surface in front of movable canard alleviates stall at high α. Free-to-roll Tail stabilizers alleviate induced roll at high α. Dedicated roll control surfaces avoid roll control saturation and simplify autopilot design. Blended canard – tail control reduces required deflection of canards.

Most Canard Flight Control Missiles are Wingless

Examples of Aerodynamic Approaches that Enhance Maneuverability and Accuracy of Canard Flight Control

Python 4: Split Canards, Dedicated Roll Control, Free-to-Roll Tail Stabilizers

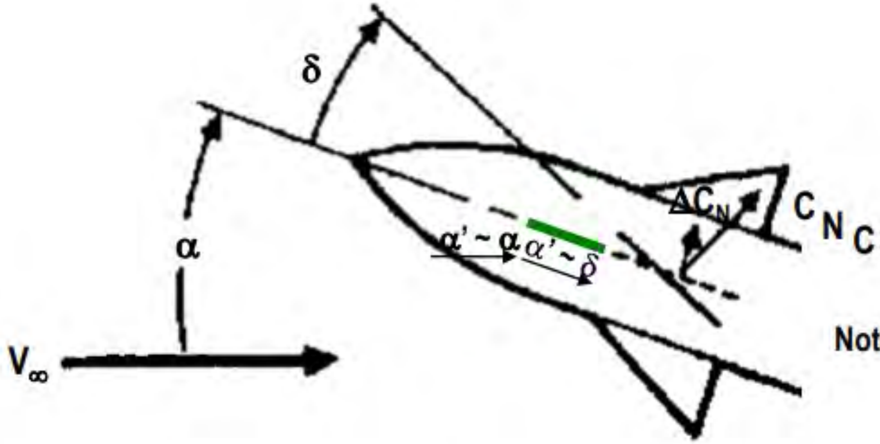
AIM-9M: Rollerons (~ 30,000 rpm)

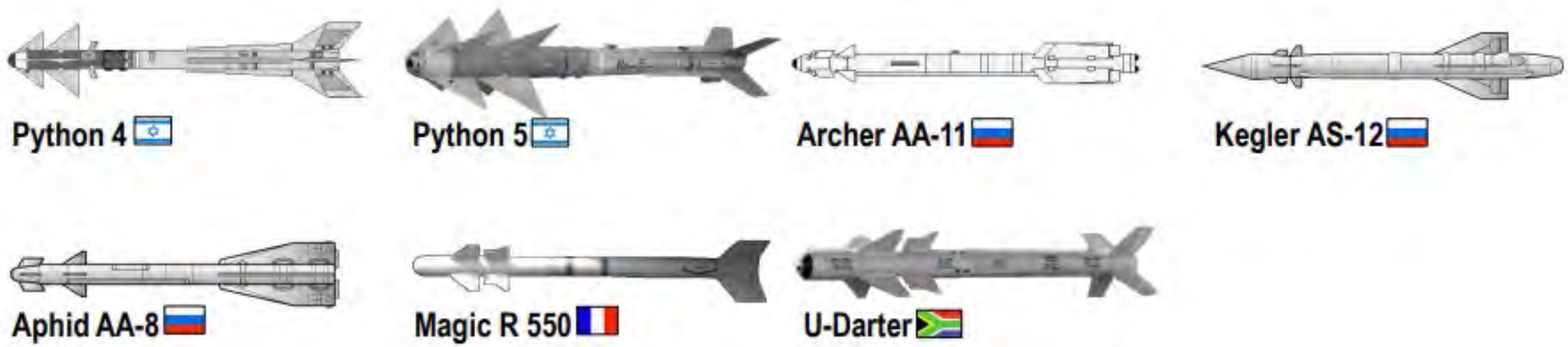
Video of DAGR Free-to-Roll Tail Stabilizers)

SA-8: Free-to-Roll Tails

Stunner: Blended Canard – Tail Control Provides Direct Lift Divert Accuracy and Skid-to-Turn Maneuverability.

Split Canard Flight Control → Maneuverability at High Angle of Attack with Lower Hinge Moment

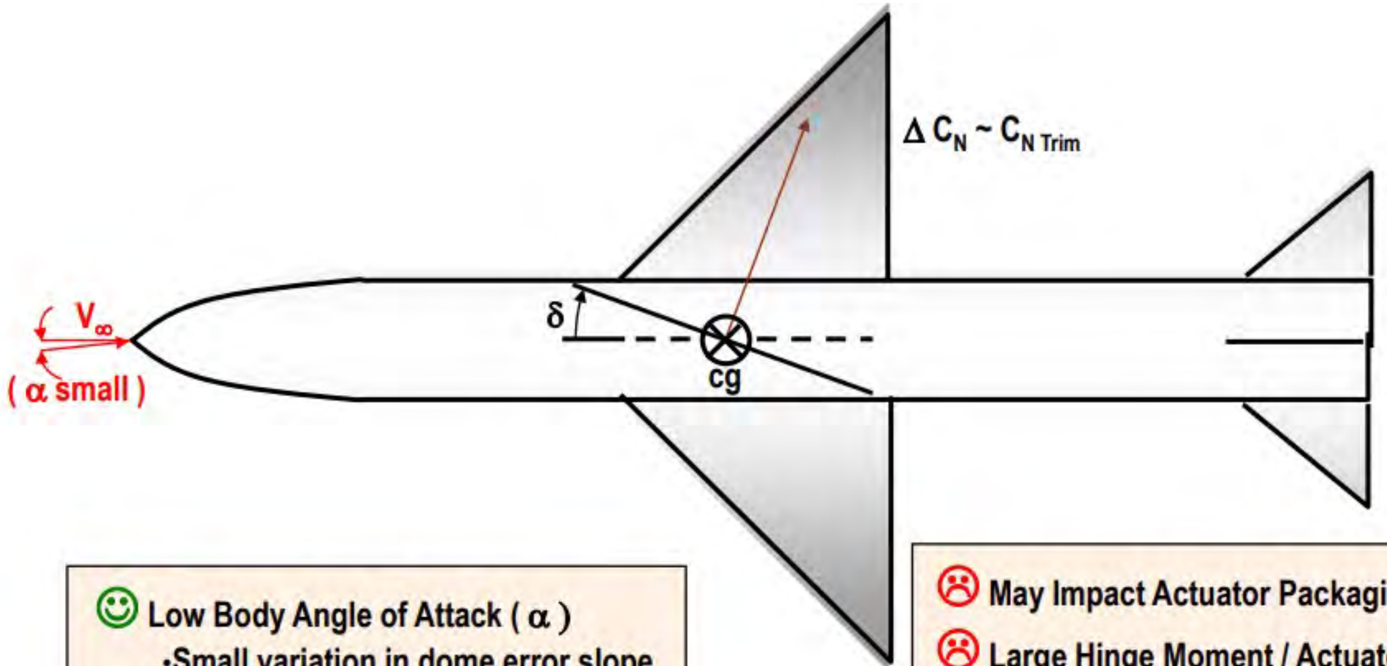




Note: Forward fixed surface reduces local angle-of-attack for movable canard, providing lower hinge moment and higher stall angle of attack. Forward surface also provides a fixed, symmetrical location for vortex shedding from the body.

Python 4 also has free-to-roll tail stabilizers and dedicated roll control ailerons.

Wing Flight Control Advantages: Low Body Rotation. Disadvantages: High Hinge Moment, Induced Roll, Stall



- ☺ Low Body Angle of Attack (α)
  - Small variation in dome error slope
  - Compatible with strap-down seeker
- ☺ Fast Response / Lower Time Constant

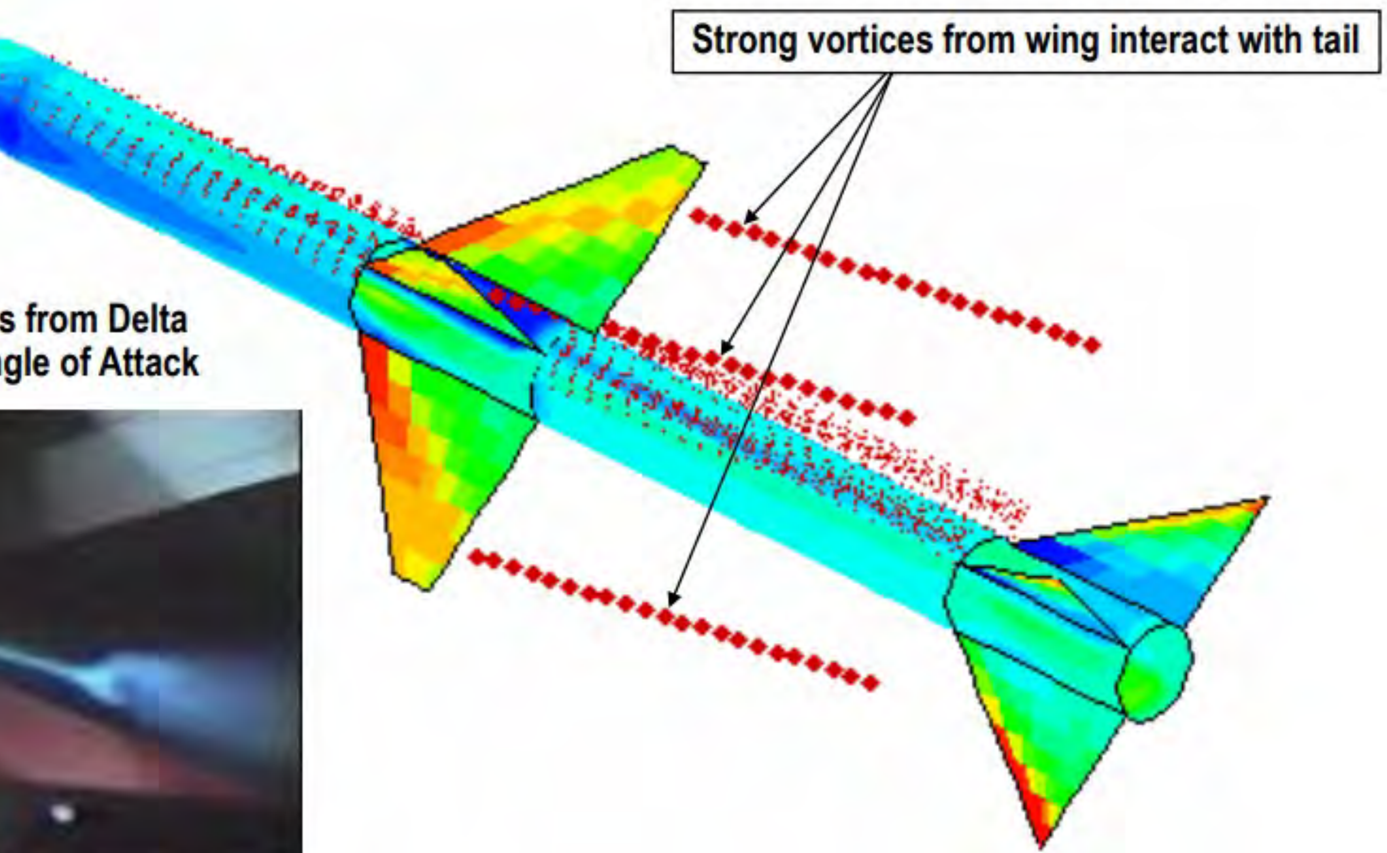
Note: $(C_{N_\delta})_{wing} \approx (C_{N_\alpha})_{wing}$

- ☹ May Impact Actuator Packaging
- ☹ Large Hinge Moment / Actuators
- ☹ Larger Wing Size
- ☹ Induced Roll
- ☹ Wing Stall
- ☹ Larger Variation in Static Margin

Wings are Susceptible to Strong Vortex Shedding

Video of Vortices from Delta Wing at High Angle of Attack

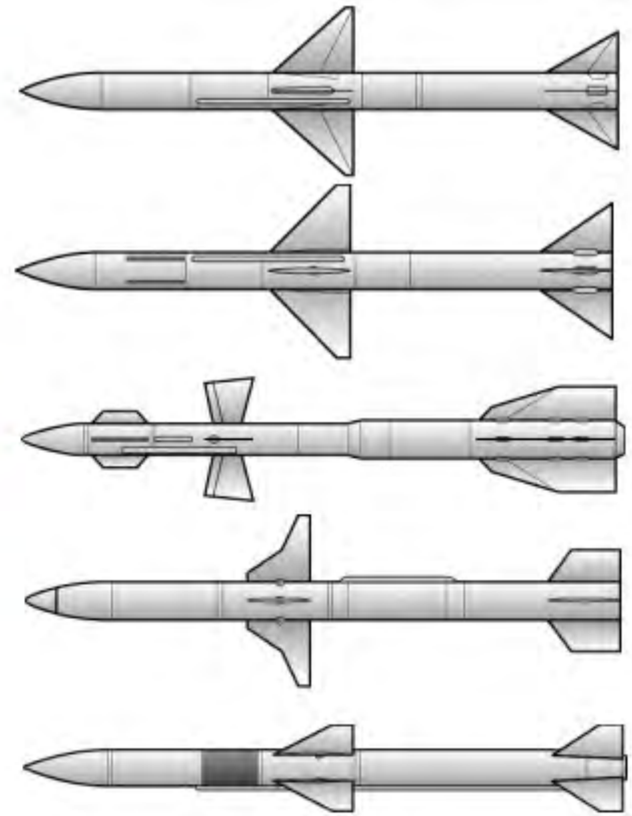
Source: University of Notre Dame



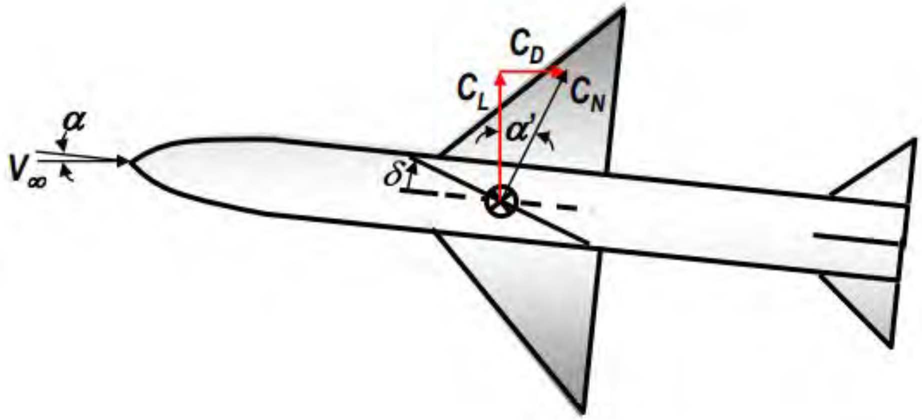
Source: Nielsen Engineering & Research (NEAR)

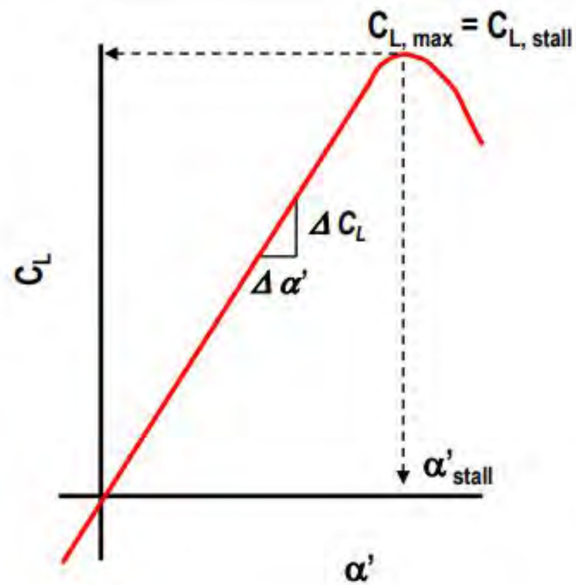
Current Wing Flight Control Missiles are Supersonic and Are Old Technology

- Sparrow AIM-7: IOC 1956
- Skyflash: IOC 1978
- Alamo AA-10 / R-27: IOC 1980
- HARM AGM-88: IOC 1983
- Aspide: IOC 1986



Aerodynamic Flight Control Surfaces Stall at a Surface Local Angle of Attack $\alpha' \approx 22$ Deg





Note:
$C_L \approx C_N \cos(\alpha + \delta)$
$C_D \approx C_N \sin(\alpha + \delta)$
$\alpha' \approx \alpha + \delta$
$\alpha'_{stall} \approx (\alpha + \delta)_{stall} \approx 22$ deg, if $\Lambda_{LE} \leq 45$ deg

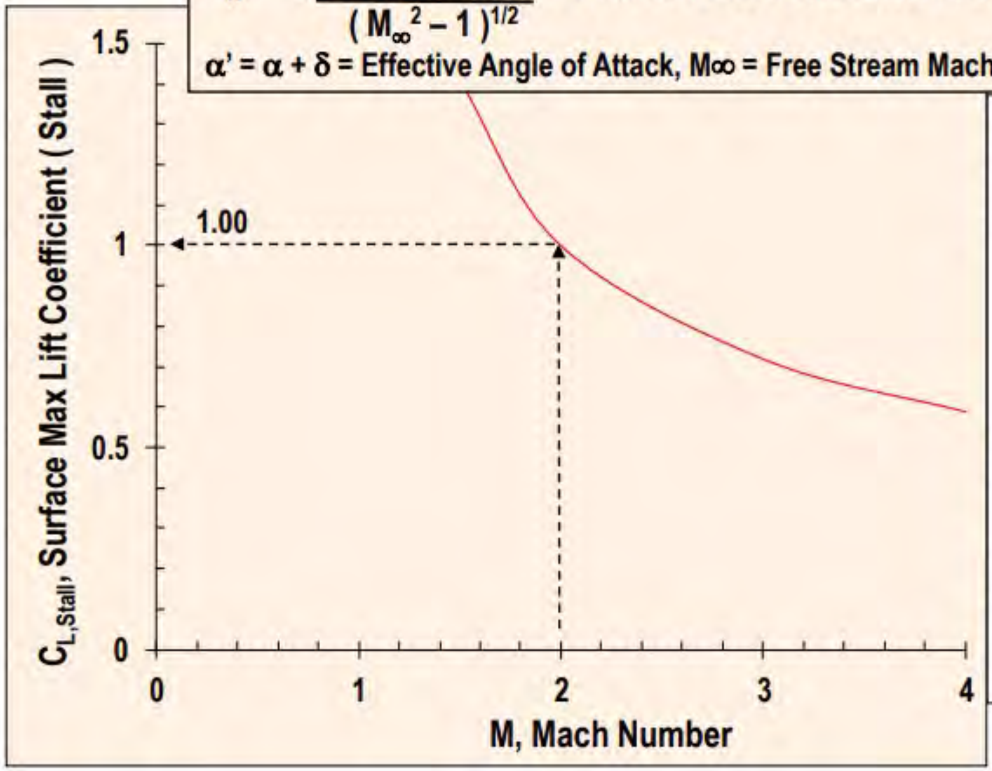
C_N = Normal Force Coefficient of Surface
C_L = Lift Coefficient of Surface
C_D = Drag Coefficient of Surface
δ = Deflection Angle of Surface
α = Angle of Attack of Missile
α' = Local Angle of Attack of Surface
Λ_{LE} = Leading Edge Sweep of Surface

Surface Maximum Lift (Stall) Decreases with Supersonic Mach Number

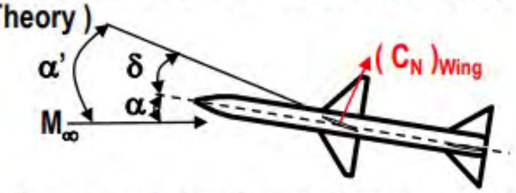
$$C_{L,Stall} \approx C_N \cos \alpha'_{stall}$$

$C_N = 4 \frac{|\sin \alpha' \cos \alpha'|}{(M_\infty^2 - 1)^{1/2}} + 2 \sin^2\alpha'$, Based on Surface Planform Area and Linear Wing Theory

$\alpha' = \alpha + \delta$ = Effective Angle of Attack, M_∞ = Free Stream Mach Number



Example: Rocket Baseline Wing Control Surface (Λ_{LE} = 45 deg $\Rightarrow \alpha'_{stall} \approx (\alpha + \delta)_{stall} \approx$ 22 deg), A = 2.82 & M = 2 $\Rightarrow$ Linear Wing Theory)



Using Linear Wing Theory Plus Newtonian Impact Theory, Calculate:

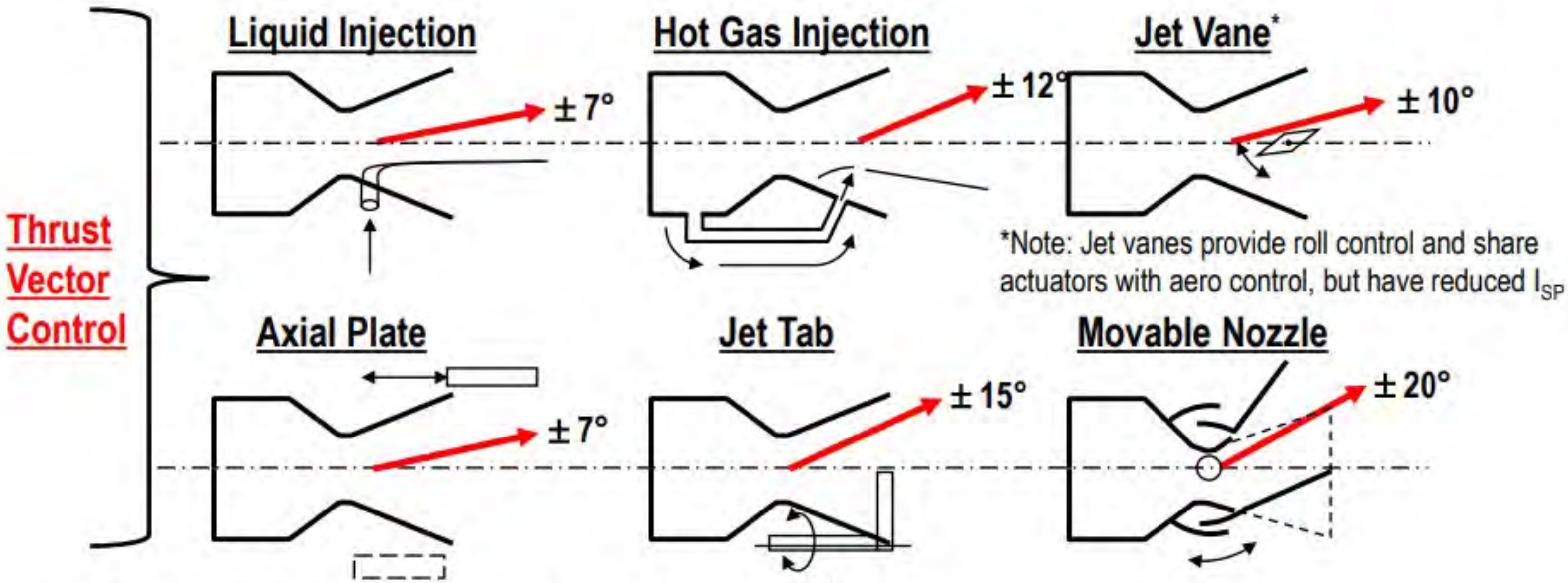
$C_N = 4 \sin \alpha' \cos \alpha' / (M_\infty^2 - 1)^{1/2} + 2 \sin^2\alpha'$
$= 4 \sin 22 \text{ deg} \cos 22 \text{ deg} / (2^2 - 1)^{1/2} + 2 \sin^2 22 \text{ deg} = 0.80 + 0.28 = 1.08$

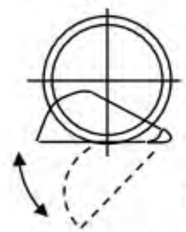
$C_{L,\,stall} \approx 1.08 \cos 22$ deg = 1.00, Based on Surface Planform Reference Area

Note: Figure Based on

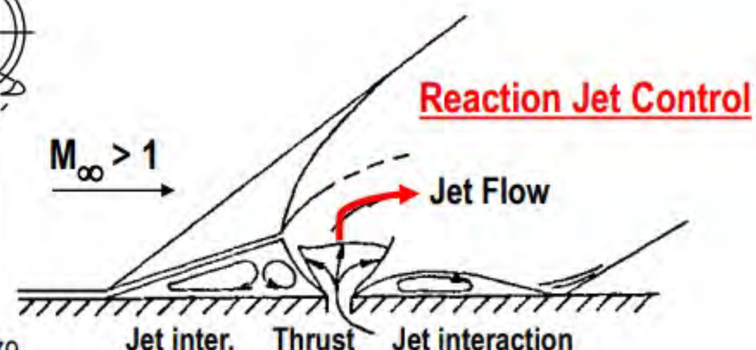
- Leading Edge Sweep $\Lambda_{LE} \leq$ 45 deg $\Rightarrow \alpha'_{stall} \approx (\alpha + \delta)_{stall} \approx$ 22 deg
- Linear Wing Theory Plus Newtonian Impact Theory
- Surface Planform Reference Area.

TVC and Reaction Jet Flight Control Provide High Maneuverability at Low Dynamic Pressure



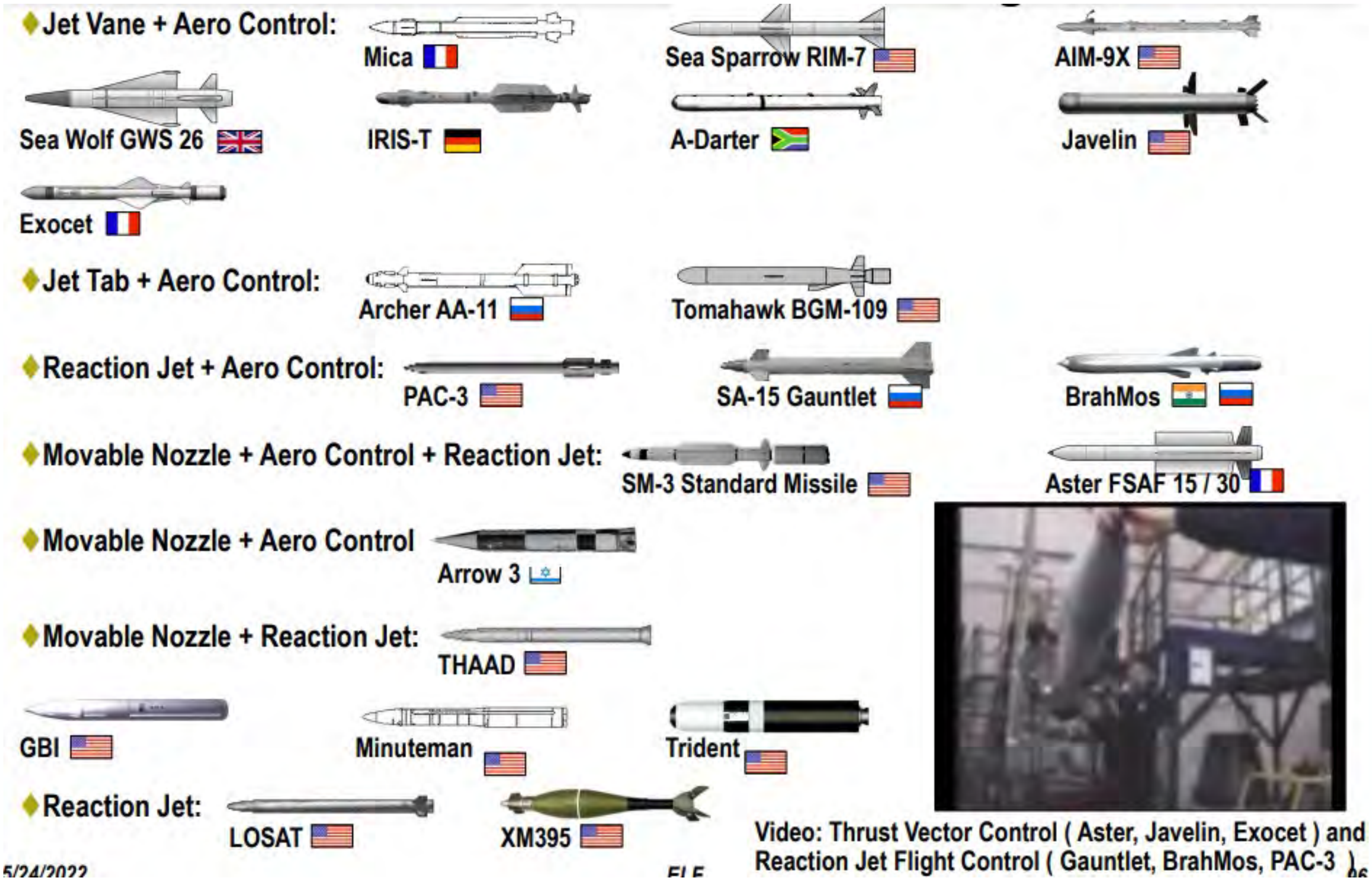


- •TVC and reaction jet flight control provide high maneuverability at low dynamic pressure
- •TVC usually has lower time constant and smaller miss distance than aero control
- •Reaction jets used for divert and attitude control
- •Reaction jets usually have lower time constant and smaller miss distance than TVC
- •Reaction jets can be either impulse jets or controlled duration jets



Source: Brebner, G. G., "The Control of Missiles," AGARD-LS-98, Feb 1979

Most Missiles with TVC or Reaction Jet Flight Control Also Use Aerodynamic Flight Control



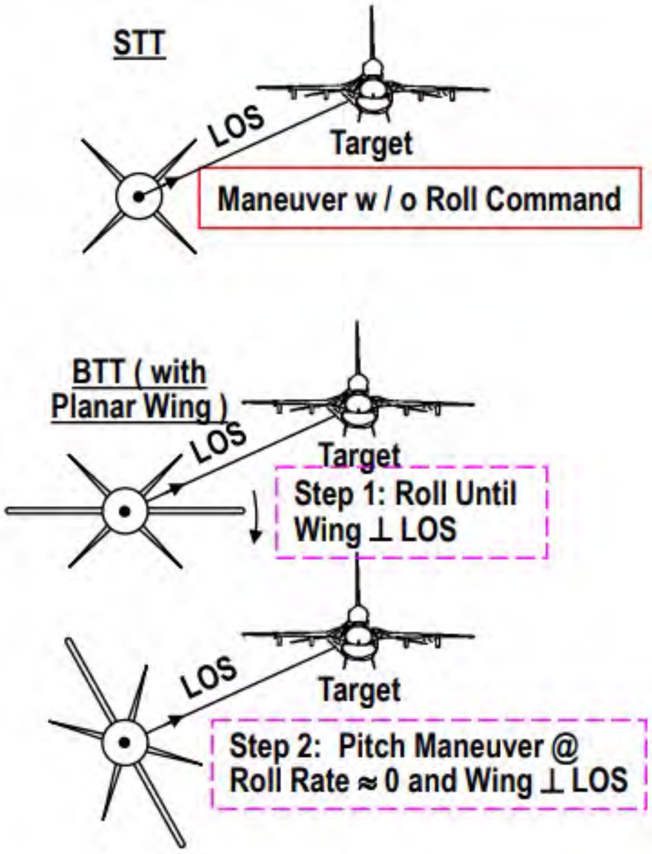
Skid-to-Turn is the Most Common Maneuver Law for Missiles

Skid-To-Turn (STT)

- Advantage ☺: Fast response
- Disadvantage ☹: Usually limited to axisymmetric cruciform missiles with low aspect ratio
- Feature:
  - Usually small roll attitude / rate commands from autopilot

Bank-To-Turn (BTT)

- Advantage ☺: Higher maneuverability for mono-wing, noncircular / lifting bodies, and airbreathers
- Disadvantages ☹:
  - Time to roll
  - Roll rate limited by gain for radome error slope stability
- Features
  - Large roll attitude commands from autopilot
  - Small sideslip

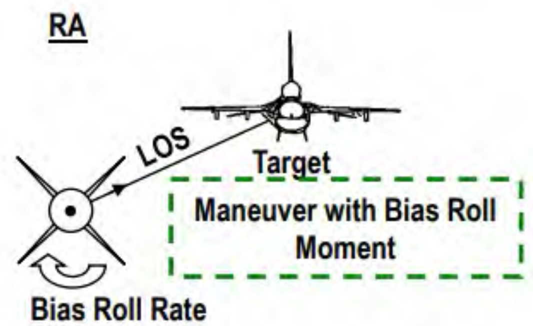


Note: LOS is line-of-sight

Skid-to-Turn is the Most Common Maneuver Law for Missiles (cont)

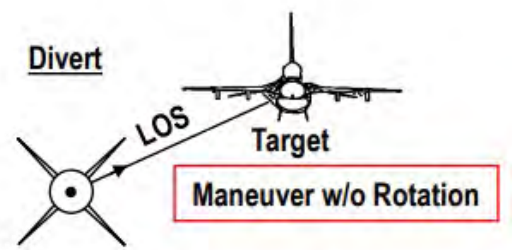
Rolling Airframe (RA)

- Advantages ☺:
  - Requires fewer gyros / accelerometers / actuators
  - Compatible with rosette scan / pseudo Image seeker
- Disadvantages ☹:
  - Reduced maneuverability for aero control
  - Requires higher rate gyros / actuators / seeker tracking
  - Higher drag with coning flight trajectory
  - Requires precision geometry and thrust alignment
  - Induces radial stress
  - Thrust varies with roll rate
- Features
  - Bias roll rate (~ 10 Hz) from bias roll moment
  - Can use "bang-bang" / impulse steering
  - Compensates for thrust offset



Divert

- Advantages ☺:
  - Lower time constant
  - Less effect of radome error slope
  - Often has smaller miss distance
- Disadvantages ☹:
  - Usually higher cost
  - May not provide sufficient maneuverability
- Features
  - Direct lift / side force w/o rotation
  - Either wing, blended canard – tail, or divert reaction jet control



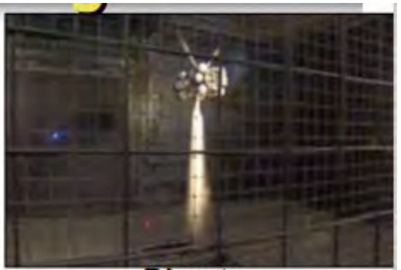
Note: LOS is line-of-sight

Examples of Skid-to-Turn, Bank-to-Turn, Rolling Airframe, and Divert Maneuvering

Skid-To-Turn (STT): Sea Sparrow

Bank-to-Turn (BTT): JASSM

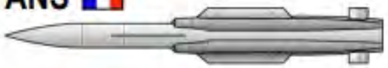
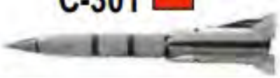
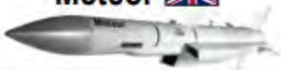
Rolling Airframe (RA): SeaRAM



Divert: MKV

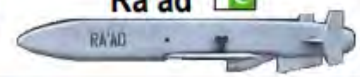
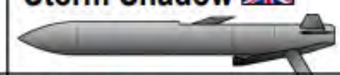
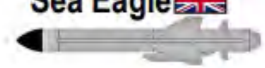
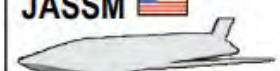
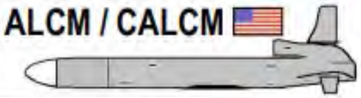
Video of Sea Sparrow, JASSM, SeaRAM, and MKV Flight Trajectories

Non-Cruciform Inlets Require Bank-to-Turn Maneuvering

Type Inlet	Location	Propulsion	Example Missile
Twin	Side	Ramjet	ASMP  ANS 
"	"	"	C-101  C-301 
"	"	Turbojet	Taurus KEPD-350 
"	Cheek	Ducted Rocket	HSAD  Meteor 
Single	Bottom Scoop	Scramjet	X-51 SED 
"	"	Ramjet	ASALM 
"	"	Turbojet	Tomahawk  RBS-15  SOM 

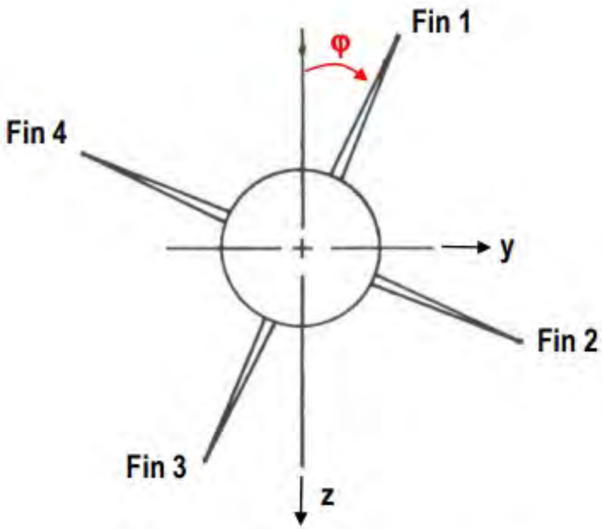
Note: Bank-to-turn maneuvering maintains low sideslip ($\beta \approx 0$ deg) with better inlet efficiency

Non-Cruciform Inlets Require Bank-to-Turn Maneuvering (cont)

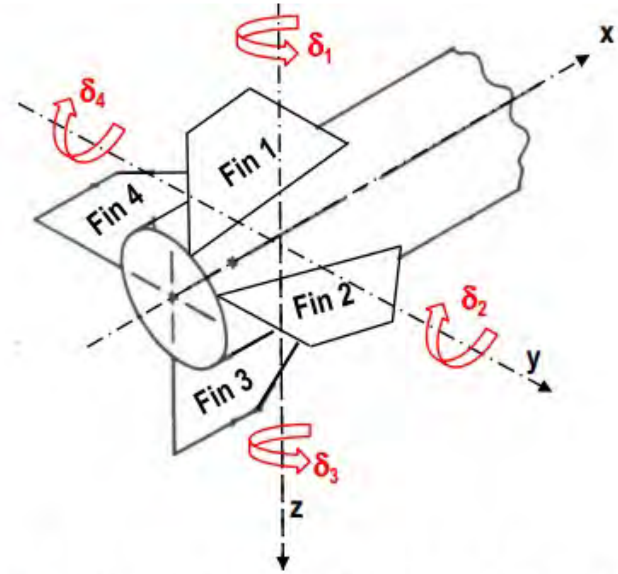
Type Inlet	Location	Propulsion	Example Missile
Single	Bottom Scoop (cont)	Turbojet	NSM  TORGOS  Ra'ad 
"	"	"	Storm Shadow  Sizzler 
			Delilah  Kh-35 
"	"	"	Hyunmoo III 
"	"	"	Babur  Sea Eagle 
"	Bottom Flush	"	JASSM  Harpoon 
"	"	"	Gabriel 
"	Top	Turbofan	ALCM / CALCM 

Note: Bank-to-turn maneuvering maintains low sideslip ($\beta \approx 0$ deg) with better inlet efficiency

Typical Sign Convention for Cruciform Missile Roll Angle and Flight Control Surface Deflection



+ φ looking forward from base



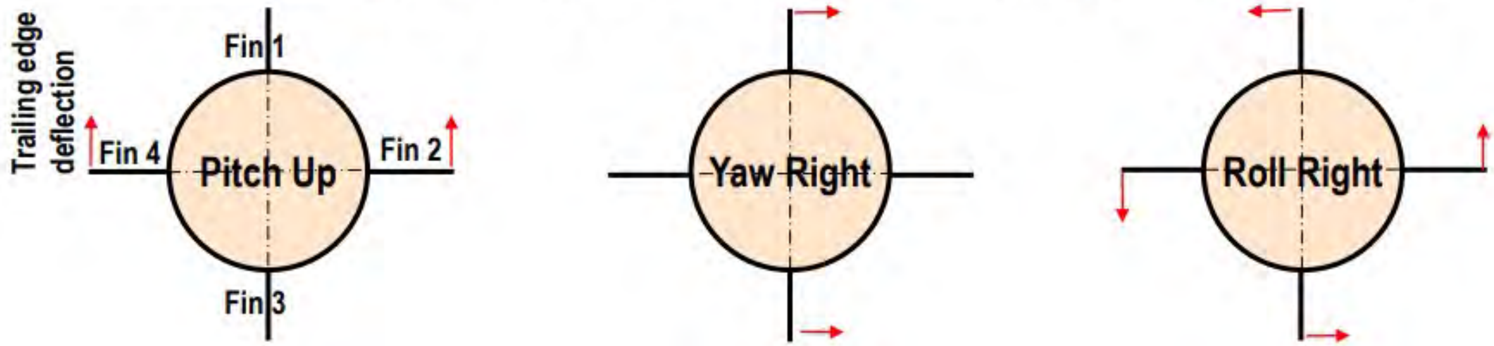
Fins 1 and 3 have trailing edge right for + deflection
Fins 2 and 4 have trailing edge up for + deflection
In above figure Fins 1 and 2 have + deflection

δ_e = Equivalent elevator deflection (+ δ_e produces + (up) pitching moment) = $[(\delta_2 + \delta_4)/2] \cos\varphi - [(\delta_1 + \delta_3)/2] \sin\varphi$
δ_r = Equivalent rudder deflection (+ δ_r produces + (right) yawing moment) = $[(\delta_2 + \delta_4)/2] \sin\varphi + [(\delta_1 + \delta_3)/2] \cos\varphi$
δ_a = Equivalent aileron deflection (+ δ_a produces + (clockwise) rolling moment) = $(\delta_2 + \delta_3 - \delta_1 - \delta_4)/4$

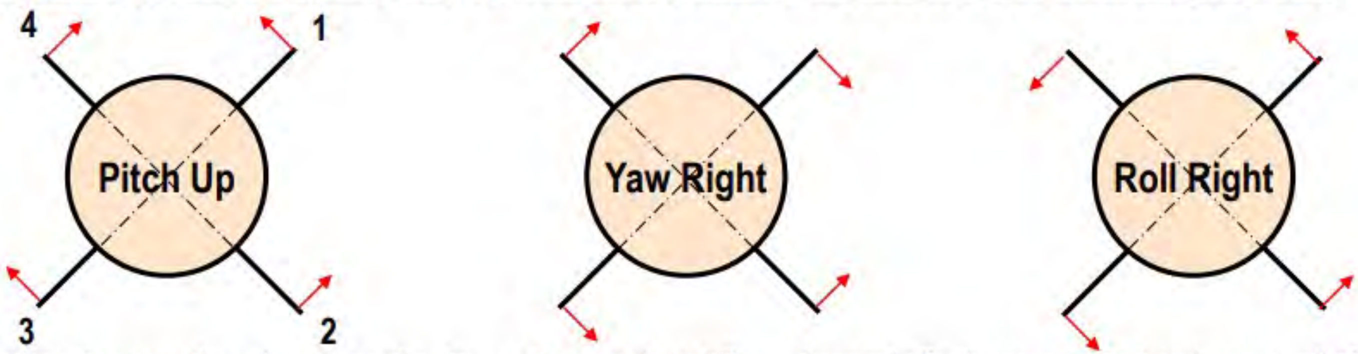
Note: For minimum total fin deflection (⇨ lowest total hinge moment, lowest drag, highest control effectiveness):
$\delta_1 = \delta_r - \delta_a$, $\delta_2 = \delta_e + \delta_a$, $\delta_3 = \delta_r + \delta_a$, $\delta_4 = \delta_e - \delta_a$

X Roll Orientation Flight is Usually Better Than + Roll Orientation Flight

+ Roll Orientation (φ = 0 deg), Cruciform Tail Control, Looking Forward from Base



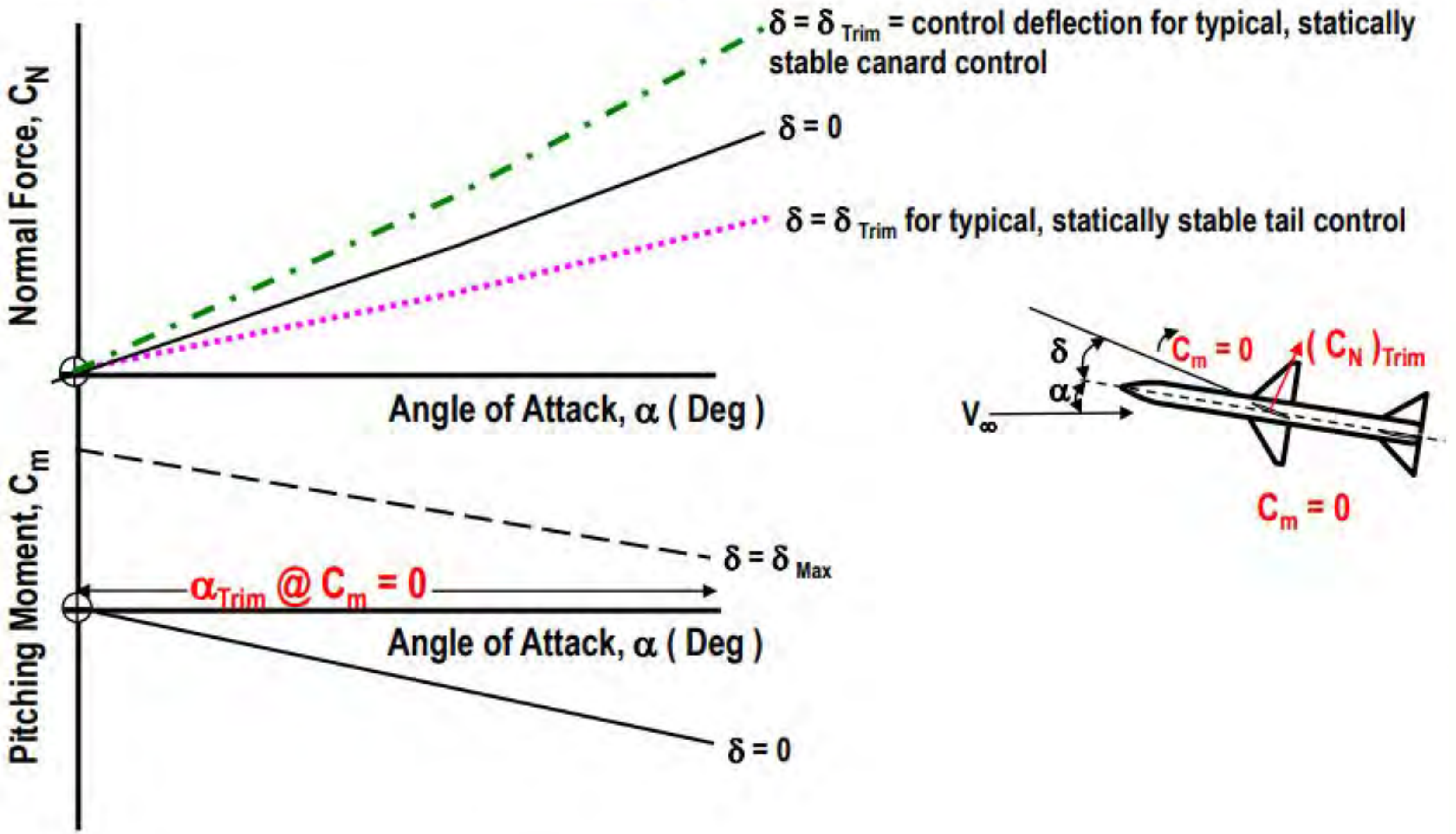
X Roll Orientation (φ = 45 deg), Cruciform Tail Control, Looking Forward from Base



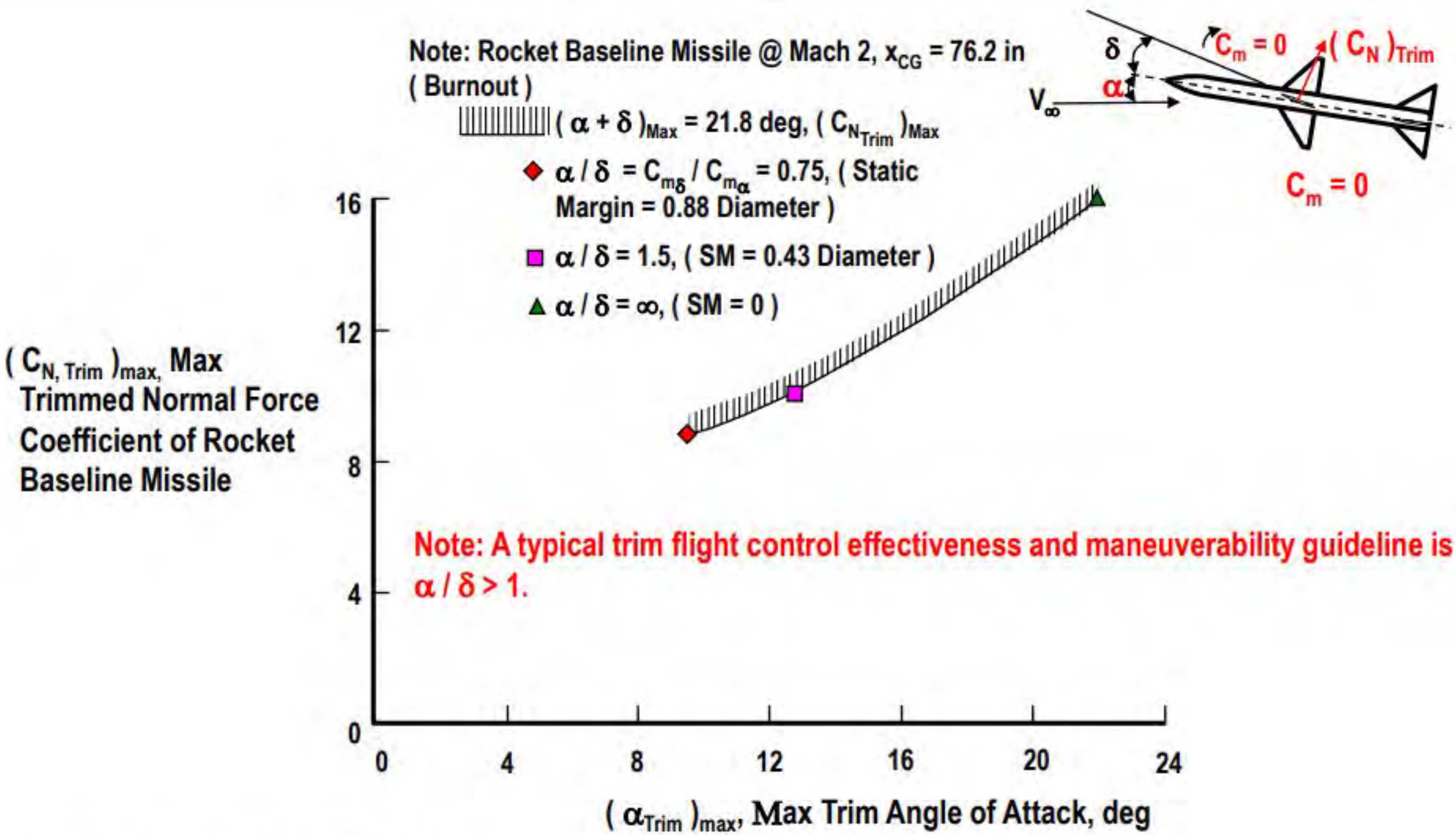
Note: + roll orientation sometimes has lower trim drag and less static stability and control effectiveness in pitch and yaw. + roll often has statically unstable roll moment derivative ($C_{l_\phi} > 0$) in supersonic flight.

X roll orientation usually has better launch platform compatibility, higher lift-to-drag ratio, higher static stability and control effectiveness in pitch and yaw. X roll often has statically unstable roll moment derivative ($C_{l_\phi} > 0$) in subsonic flight.

Trimmed Normal Force is Defined at Zero Pitching Moment



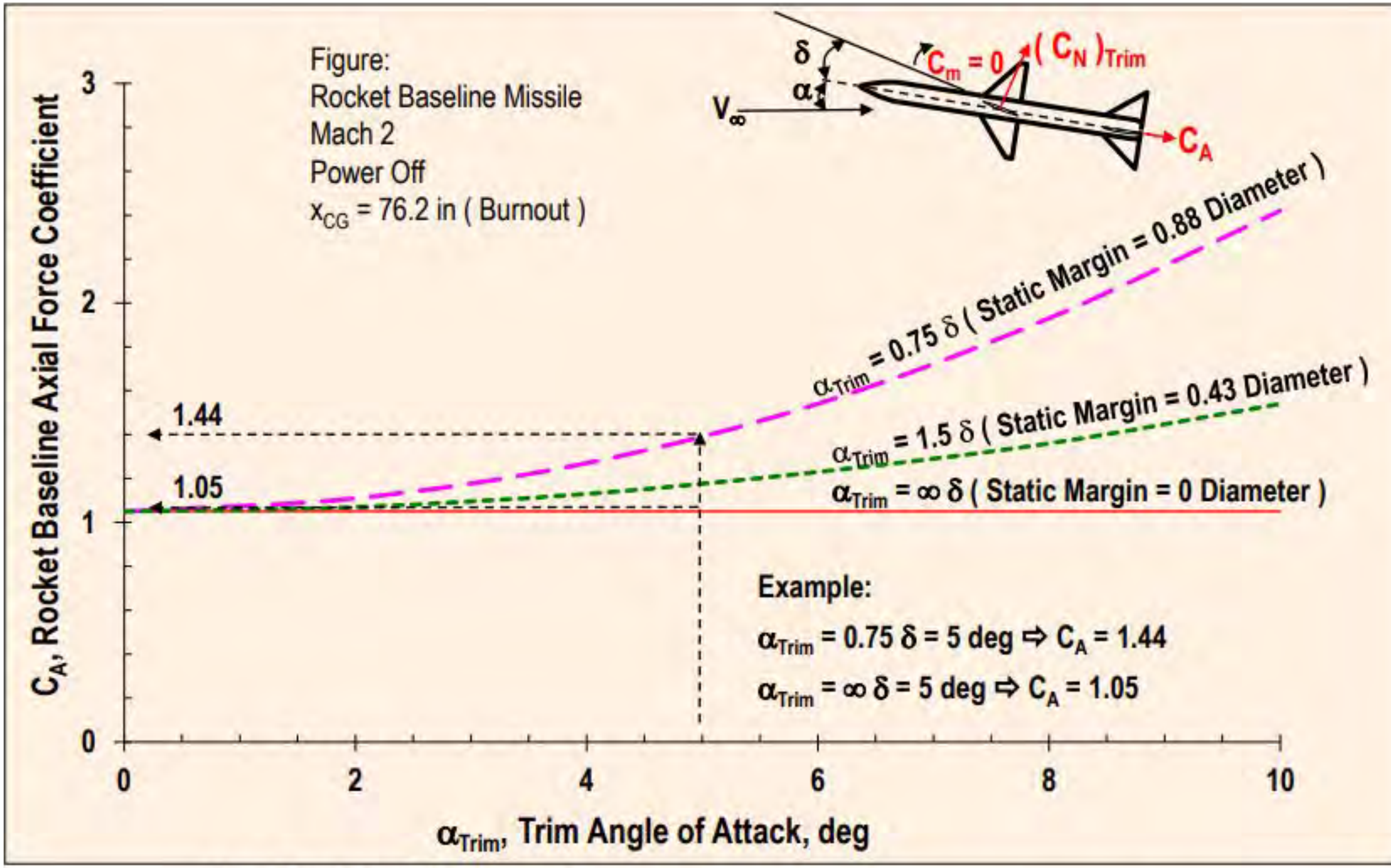
Relaxed Static Stability Margin Allows Higher Trim Angle of Attack and Higher Normal Force



Note: Based on Data of Chapter 7

SM = Static Margin = Distance Between Aerodynamic Center and Center of Gravity ($x_{AC} - x_{CG}$)

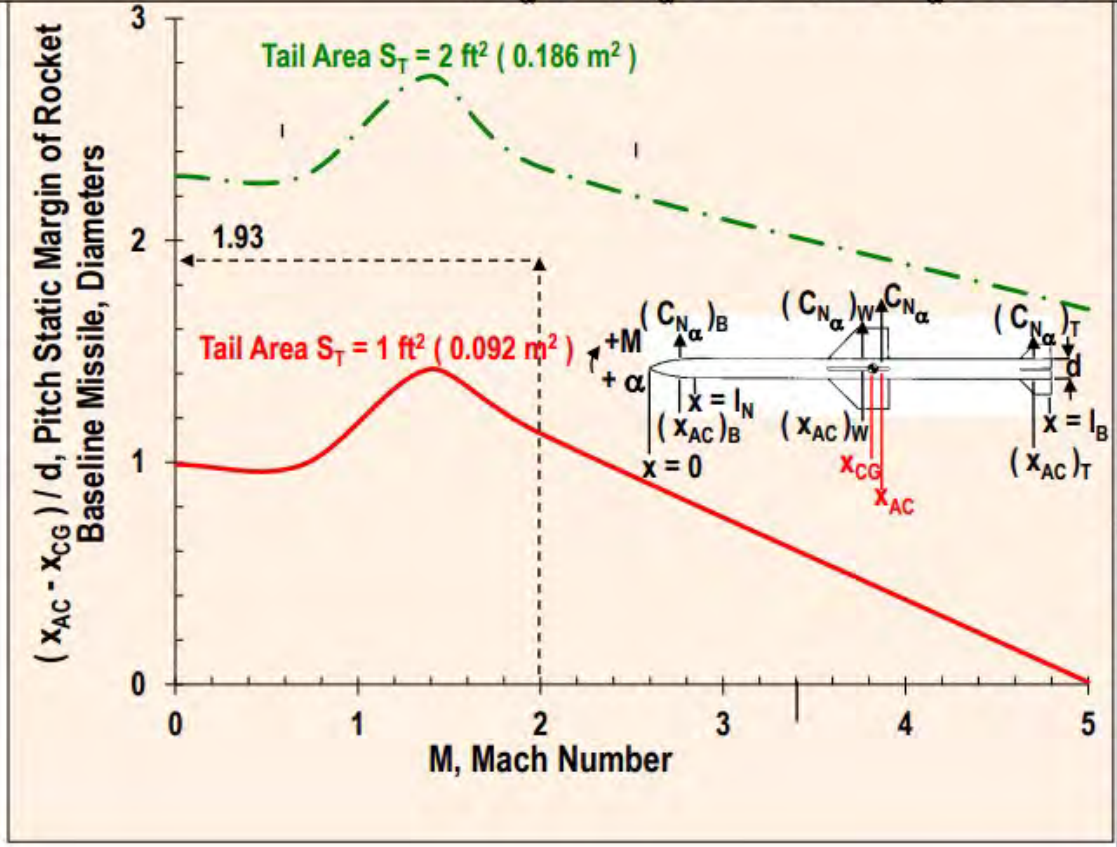
Relaxed Static Stability Margin Reduces Drag



Note: Based on Data of Chapter 7
.SM = Static Margin = Distance Between Aerodynamic Center and Center of Gravity ($x_{AC} - x_{CG}$)

Missile Static Margin is Driven by Tail Area and Static Margin Predicition Has Large Uncertainty

$$x_{AC} - x_{CG} \approx \frac{(C_{N_\alpha})_B [(x_{AC})_B - (x_{CG})] + (C_{N_\alpha})_W [(x_{AC})_W - (x_{CG})](S_W / S_{Ref}) + (C_{N_\alpha})_T [(x_{AC})_T - (x_{CG})](S_T / S_{Ref})}{(C_{N_\alpha})_B + (C_{N_\alpha})_W S_W / S_{Ref} + (C_{N_\alpha})_T S_T / S_{Ref}}$$

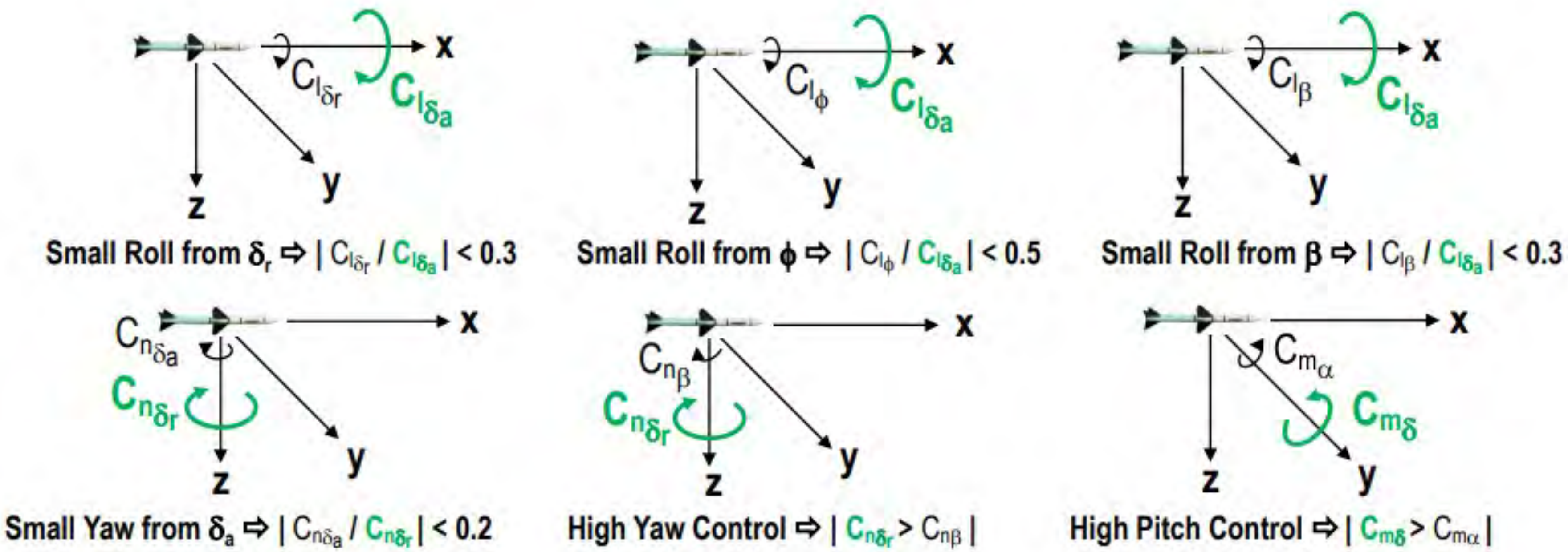


Assumptions for Figure:
Rocket Baseline Missile
Body length l = 144 in, Nose length l_N = 19.2 in, d = 8 in, Wing area S_W = 2.55 ft^2, Reference area S_{Ref} = 0.349 ft^2, A_W = 2.82, $(c_{MAC})_W$ = 13.3 in, Burnout x_{CG} = 76.2 in from tip, $\alpha < 6$ deg, turbulent boundary layer.

Using Simplified Prediction Methods:
$(C_{N_\alpha})_B$ = 2 per rad, $(x_{AC})_B / l_N = 0.63$, $(C_{N_\alpha})_T = (C_{N_\alpha})_W = 4 / [M^2 - 1]^{1/2}$, if $M > \{1 + [8 / (\pi A)]^2\}^{1/2}$, $(C_{N_\alpha})_T = (C_{N_\alpha})_W = \pi A / 2$, if $A < 3$ and $M < \{1 + [8 / (\pi A)]^2\}^{1/2}$, $(x_{AC})_B = 0.63\, l_N$,

Example Rocket Baseline Tail (S_T = 1.54 ft^2), M = 2 :
$(C_{N_\alpha})_T = (C_{N_\alpha})_W = 4 / [M^2 - 1]^{1/2} = 2.31$, $(x_{AC})_B / l = 0.084$, $(x_{AC})_W / l = 0.510$, $(x_{AC})_T / l = 0.955$
$(x_{AC} - x_{CG}) / d = 1.93$
From Chapter 7: $(x_{AC} - x_{CG}) / d = 0.88$
There is Typically Large Uncertainty in Conceptual Design Static Margin

Stability & Control Requires High Flight Control Effectiveness



Stabilty & Control Cross Coupling is a Concern for Lifting Bodies (S&C Cross Coupling Often > 30%)

M2-F2 Lifting Body

X-24B Lifting Body

Video of M2-F2 Lifting Body Flight Test