Professional Development
Short Course on
Tactical Missile Design

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Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Emphasis Is on Physics-Based, Analytical Sizing of Aerodynamic Configuration

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- Primary Emphasis
- Secondary Emphasis
- Tertiary Emphasis
- Not Addressed
### Tactical Missiles Are Different from Fighter Aircraft

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<th>Example of State-of-the-Art</th>
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**Symbols:**
- Superior
- Better
- Comparable
- Inferior

**Example of State-of-the-Art:**
- AGM-88
- AA-11
- SM-3
- PAC-3
- Javelin
- FIM-92
- GBU-31
- AGM-129
- AGM-86
- Storm Shadow
- LOCAAS

**2/24/2008 ELF**
Aero Configuration Sizing Parameters Emphasized in This Course

- Nose Fineness
- Diameter
- Propulsion Sizing / Propellant or Fuel
- Wing Geometry / Size
- Stabilizer Geometry / Size
- Flight Conditions (α, M, h)
- Thrust Profile
- Length
- Flight Control Geometry / Size
Conceptual Design Process Requires Evaluation of Alternatives and Iteration

- Mission / Scenario Definition
- Weapon Requirements, Trade Studies and Sensitivity Analysis
- Launch Platform Integration
- Weapon Concept Design Synthesis
- Technology Assessment and Dev Roadmap

Note: Typical conceptual design cycle is 3 to 9 months. House of Quality may be used to translate customer requirements to engineering characteristics. DOE may be used to efficiently evaluate the broad range of design solutions.
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

1. Define Mission Requirements
2. Establish Baseline
   - Aerodynamics
   - Propulsion
   - Weight
   - Trajectory
3. Meet Performance?
   - Yes: Measures of Merit and Constraints
     - Yes
     - No: Alt Mission
4. No: Resize / Alt Config / Subsystems / Tech
5. Alt Baseline
Examples of Air-Launched Missile Missions / Types

- **Air-to-air**
  - Short range ATA. AA-11. Maneuverability
  - Medium range ATA. AIM-120. Performance / weight
  - Long range ATA. Meteor. Range

- **Air-to-surface**
  - Short range ATS. AGM-114. Versatility
  - Medium range ATS. AGM-88. Speed
  - Long range ATS. Storm Shadow. Modularity
Examples of Surface-Launched Missile Missions / Types

- **Surface-to-surface**
  - Short range STS. Javelin.  
    - Size
  - Medium range STS. MGM-140.  
    - Modularity
    - Range

- **Surface-to-air**
  - Short range STA. FIM-92.  
    - Weight
  - Medium range STA. PAC-3.  
    - Accuracy
  - Long range STA. SM-3.  
    - High altitude

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### Aerodynamics Configuration Sizing Has High Impact on Mission Requirements

<table>
<thead>
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<th>Aero Configuration Sizing Parameter</th>
<th>Impact on Weapon Requirement</th>
<th>Aero Measures of Merit</th>
<th>Other Measures of Merit</th>
<th>Constraint</th>
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<td>Range / Maneuver</td>
<td>Time to Target</td>
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<tr>
<td>Nose Fineness</td>
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<td>〇</td>
<td>〇</td>
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<tr>
<td>Diameter</td>
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<td>Length</td>
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<tr>
<td>Wing Geometry / Size</td>
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<td>Stabilizer Geometry / Size</td>
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<td>Flight Control Geometry / Size</td>
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<td>Propellant / Fuel</td>
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<td>Thrust Profile</td>
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<tr>
<td>Flight Conditions ( $\alpha, M, h$ )</td>
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- **Very Strong**
- **Strong**
- **Moderate**
- **Relatively Low**
## Example of Assessment of Alternatives to Establish Future Mission Requirements

### Alternatives for Precision Strike

<table>
<thead>
<tr>
<th></th>
<th>Measures of Merit</th>
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<tr>
<td></td>
<td>Cost per Shot</td>
<td>Number of Launch Platforms Required</td>
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<tr>
<td><strong>Future Systems</strong></td>
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<tr>
<td>◆ Standoff platforms / hypersonic missiles</td>
<td>○</td>
<td>●</td>
</tr>
<tr>
<td>◆ Overhead loitering UCAVs / hypersonic missiles</td>
<td>○</td>
<td>○</td>
</tr>
<tr>
<td>◆ Overhead loitering UCAVs / light weight PGMs</td>
<td>●</td>
<td>○</td>
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<tr>
<td><strong>Current Systems</strong></td>
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<tr>
<td>◆ Penetrating aircraft / subsonic PGMs</td>
<td>●</td>
<td>–</td>
</tr>
<tr>
<td>◆ Standoff platforms / subsonic missiles</td>
<td>○</td>
<td>●</td>
</tr>
</tbody>
</table>

**Note:** Superior ☺ ☺ ☺ ☺ Good ☺ ☺ ☺ ☺ Average ○ ○ ○ ○ Poor – – – –

Note: C4ISR targeting state-of-the-art for year 2010 projected to provide sensor-to-shooter / weapon connectivity time of less than 2 m and target location error (TLE) of less than 1 m for motion suspended target.
C4ISR Tactical Satellites and UAVs Have High Impact on Mission Capability

Launch Platforms
- Fighter Aircraft
- Bomber
- Ship / Submarine
- UCAV

Precision Strike Weapons
- Hypersonic SOW
- Subsonic PGM
- Subsonic CM

Off-board Sensors
- Tactical Satellite
- UAV

Time Critical Targets
- TBM / TEL
- SAM
- C3
- Other Strategic

Note: C4ISR targeting state-of-the-art for year 2010 projected to provide sensor-to-shooter / weapon connectivity time of less than 2 m and target location error (TLE) of less than 1 m for motion suspended target.
Example of System-of-Systems Analysis to Develop Future Mission Requirements

1. Compare Targeting of Subsonic Cruise Missile Versus Hypersonic Missile

2. Time To Target

3. Alt / Speed / RCS Required For Survivability

4. Lethality

5. Campaign Model Weapons Mix (CM, Hypersonic Missile) Results (eg., Korean Scenario)

Selected For All:
- Value of Speed / Range
- Time Urgent Targets
- High Threat Environments
- Buried Targets
- Launch Platform Alternatives
- Operating and Attrition Cost in Campaign
- Weapon Cost in Campaign
- Mix of Weapons in Campaign
- Cost Per Target Kill
- C4ISR Interface
Example of Technological Surprise Driving Immediate Mission Requirements

Sidewinder AIM-9L (IOC 1977)

- Performance
  - +/- 25 deg off boresight
  - 6.5 nm range

Archer AA-11 / R-73 (IOC 1987)

- Performance
  - +/- 60 deg off boresight
  - 20 nm range

- New Technologies
  - TVC
  - Split canard
  - Near-neutral static margin
  - +/- 90 deg gimbal seeker
  - Helmet mounted sight

Note: AIM-9L maneuverability shortfall compared to Archer drove sudden development of AIM-9X.
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

1. Define Mission Requirements
2. Establish Baseline
   - Aerodynamics
   - Propulsion
   - Weight
   - Trajectory
3. Meet Performance?
   - Yes
     - Yes
     - No
   - No
     - Yes
     - No
4. Measures of Merit and Constraints
5. Alt Mission
6. Alt Baseline
7. Resize / Alt Config / Subsystems / Tech
Benefits of Baseline Design

- Allows simple, conceptual design methods to be used with good accuracy
- Well documented benchmark / configuration control / traceability between cause and effect
- Balanced subsystems
- Gives fast startup / default values for design effort
- Provides sensitivity trends for changing design
- Baselines can cover reasonable range of starting points
  - Baselines can normally be extrapolated to ±50% with good accuracy

Guidelines

- Don’t get locked-in by baseline
- Be creative
- Project state-of-the-art (SOTA) if baseline has obsolete subsystems
  - Sensors and electronics almost always need to be updated
Example of Missile Baseline Data

Configuration Drawing

Weight / Geometry

Flow Path Geometry

Aerodynamics

Ramjet Propulsion

Rocket Propulsion

Flight Performance

House of Quality

Pareto Sensitivity for DOE

2/24/2008
Baseline Design Data Allows Correction of Computed Parameters in Conceptual Design

\[ P_{CD, C} = \left( \frac{P_{B, C}}{P_{B, U}} \right) P_{CD, U} \]

- \( P_{CD, C} \): Parameter of conceptual design, corrected
- \( P_{B, C} \): Parameter of baseline, corrected (actual data)
- \( P_{B, U} \): Parameter of baseline, uncorrected (computed)
- \( P_{CD, U} \): Parameter of conceptual design, uncorrected (computed)

Example

- Ramjet Baseline with RJ-5 fuel (heating value = 11,300,000 ft-lbf/lbm)
- Advanced Concept with slurry fuel (40% JP-10 / 60% boron carbide = 18,500,000 ft-lbf/lbm)
- Flight conditions: Mach 3.5 cruise, 60k ft altitude, combustion temperature 4,000 R
- Calculate specific impulse \( (I_{SP})_{CD, C} \) for conceptual design, based on corrected baseline data

\[
\begin{align*}
(I_{SP})_{B, C} &= 1,120 \text{ s} \\
(I_{SP})_{B, U} &= 1387 \text{ s} \\
(I_{SP})_{CD, U} &= 2,271 \text{ s} \\
(I_{SP})_{CD, C} &= \left( \frac{(I_{SP})_{B, C}}{(I_{SP})_{B, U}} \right) (I_{SP})_{CD, U} = \left[ \frac{1120}{1387} \right] (2271) = 0.807 (2271) = 1,834 \text{ s}
\end{align*}
\]
Summary of This Chapter

♦ Overview of Missile Design Process
♦ Examples
  ♦ Tactical missile characteristics
  ♦ Conceptual design process
  ♦ SOTA of tactical missiles
  ♦ Aerodynamic configuration sizing parameters
  ♦ Processes that establish mission requirements
  ♦ Process for correcting design predictions
♦ Discussion / Questions?
♦ Classroom Exercise
Introduction Problems

1. The missile design team should address the areas of mission / scenario definition, weapon requirements, launch platform integration, design, and t______ a________.

2. The steps to evaluate missile flight performance require computing aerodynamics, propulsion, weight, and flight t________.

3. Air-to-air missile characteristics include light weight, high speed, and high m____________.

4. Air-to-surface missiles are often versatile and m______.

5. Four aeromechanics measures of merit are weight, range, maneuverability, and t___ to target.

6. The launch platform often constrains the missile span, length, and w______.

7. An enabling capability for hypersonic strike missiles is fast and accurate C____.

8. An enabling capability for large off boresight air-to-air missiles is a h_____ m______ sight.

9. A baseline design improves the accuracy and s____ of the design process.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
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- Measures of Merit and Launch Platform Integration
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- Establish Baseline
- Aerodynamics
- Propulsion
- Weight
- Trajectory
- Meet Performance?
  - Yes
    - Measures of Merit and Constraints
      - Yes
- Alt Mission
- Alt Baseline
- Resize / Alt Config / Subsystems / Tech
  - No
  - No
  - Yes
Drivers toward Small Diameter
- Decrease drag
- Launch platform diameter constraint

Drivers toward Large Diameter
- Increase seeker range and signal-to-noise, better resolution and tracking
- Increase blast frag and shaped charge warhead effectiveness (larger diameter ⇒ higher velocity fragments or higher velocity jet)
- Increase body bending frequency
- Subsystem diameter packaging
- Launch platform length constraint

Typical Range in Body Fineness Ratio 5 < l / d < 25
- Man-portable anti-armor missiles are low l / d (Javelin l / d = 8.5)
- Surface-to-air and air-to-air missiles are high l / d (AIM-120 l / d = 20.5)
Small Diameter Missiles Have Low Drag

\[ D = C_D q S_{\text{Ref}} = 0.785 C_D q d^2 \]

Note: \( D \) = drag in lb, \( C_D \) = drag coefficient, \( q \) = dynamic pressure in psf, \( d \) = diameter (reference length) in ft

Example for Rocket Baseline:
\[ d = 8 \text{ in} = 0.667 \text{ ft} \]
\[ \text{Mach 2, } h = 20k \text{ ft}, (C_D^0)_{\text{Powered}} = 0.95 \]
\[ q = \frac{1}{2} \rho V^2 = \frac{1}{2} \rho (Ma)^2 \]
\[ = \frac{1}{2} (0.001267) [(2)(1037)]^2 = 2,725 \text{ psf} \]
\[ D_0 / C_D^0 = 0.785 (2725)(0.667)^2 = 952 \]
\[ D_0 = 0.95 (952) = 900 \text{ lb} \]
Large Diameter Radar Seeker Provides Longer Detection Range and Better Resolution

\[ R_D = \left\{ \pi \sigma n^{3/4} / \left[ 64 \lambda^2 K T B F L \left( S / N \right) \right] \right\}^{1/4} P_t^{1/4} d \]

\[ \theta_{3dB} = 1.02 \lambda / d, \theta_{3dB} \text{ in rad} \]

Assumptions: Negligible clutter, interference, and atmospheric attenuation; non-coherent radar (only signal amplitude integrated); uniformly illuminated circular aperture; receiver sensitivity limited by thermal noise

Symbols:
- \( \sigma = \) Target radar cross section, m²
- \( n = \) Number of pulses integrated
- \( \lambda = \) Wavelength, m
- \( K = \) Boltzman's constant = 1.38 x 10⁻²³ J / K
- \( T = \) Receiver temperature, K
- \( B = \) Receiver bandwidth, Hz
- \( F = \) Receiver noise factor
- \( L = \) Transmitter loss factor
- \( S / N = \) Signal-to-noise ratio for target detection
- \( P_t = \) Transmitted power, W
- \( d = \) Antenna diameter

Example: Rocket Baseline
- \( d = 8 \text{ in} = 0.20 \text{ m}, P_t = 1000 \text{ W}, \lambda = 0.03 \text{ m (}\ f = 10 \text{ GHz)}\)
- \( R_D = \) Target detection range = \{ \pi (10) (100)^{3/4} / [64 (0.03)^2 (1.38 x 10⁻²³) (290) (10⁶) (5) (5)(10)]\}^{1/4} (1000)^{1/4} (0.203) = 13,073 \text{ m or 7.1 nm} \)
- \( \theta_{3dB} = 3\text{-dB beam width} = 1.02 (0.03) / 0.203 = 0.1507 \text{ rad or 8.6 deg} \)

Note for figure: \( \sigma = 10 \text{ m}^2, n = 100, \lambda = 0.03 \text{ m (}\ f = 10 \text{ GHz)}\), \( T = 290 \text{ K}, B = 10^6 \text{ Hz (} 10^{-6} \text{ s pulse}), F = 5, L = 5, S / N = 10 \)
Large Diameter IR Seeker Provides Longer Detection Range and Better Resolution

\[
R_D = \{ \frac{(l_T)_{\Delta \lambda} \eta_a A_o \{ D^* / [\Delta f_p^{1/2} (A_d)^{1/2}] \} (S/N)_D^{-1}}{1} \}^{1/2}
\]

\[
\text{IFOV} = \frac{d_p}{(\text{f-number} \cdot d_o)}
\]

Example: \(d_o = 5\text{ in} = 0.127\text{ m}\), exo-atmospheric
\[
L_\lambda = 3.74 \times 10^4 / \{ 4^5 \{ e^{1.44 \times 10^4 / (4 \cdot 300)} \} - 1 \} = 0.000224 \text{ W cm}^{-2} \text{ sr}^{-2} \mu\text{m}^{-1},
\]
\[
(l_T)_{\Delta \lambda} = 0.5 (0.000224) (4.2 - 3.8) 2896 = 0.1297 \text{ W / sr, A}_o = 256 \times 256 \times (20 \mu\text{m})^2 = 0.262 \text{ cm}^2, \text{ f-number} = 20 / [2.44 (4)] = 2.05
\]
\[
R_D = \{ 0.1297 (1) (0.01267) \{ 8 \times 10^{11} / [(250)^{1/2} (0.262)^{1/2}] \} (1)^{-1} \}^{1/2} = 12,740 \text{ m}
\]
\[
\text{IFOV} = 0.000020 / [2.05 (0.127)] = 0.0000769 \text{ rad}
\]

\begin{align*}
R_D &= \text{Target detection range, m} \\
(l_T)_{\Delta \lambda} &= \text{Target radiant intensity between } \lambda_1 \text{ and } \lambda_2 = \epsilon \\
L_\lambda &= \text{Spectral radiance (Planck’s Law)} = 3.74 \times 10^4 / \{ \lambda^5 \{ e^{1.44 \times 10^4 / (\lambda \cdot TT)} - 1 \} \}, \text{ W cm}^{-2} \mu\text{m}^{-1} \\
\Delta \lambda &= \text{Upper cutoff wavelength for detection, } \mu\text{m} \\
\lambda_1 &= \text{Lower cutoff wavelength for detection, } \mu\text{m} \\
A_T &= \text{Target planform area, cm}^2 \\
\lambda &= \text{Average wavelength, } \mu\text{m} \\
TT &= \text{Target temperature, K} \\
\text{IFOV} &= \text{Instantaneous field-of-view of pixel, rad} \\
f\text{-number} &= \frac{d_{\text{spot}}}{(2.44 \lambda)} \\
d_p &= \text{Pixel diameter, either } \mu\text{m} \\
d_{\text{spot}} &= \text{Spot resolution if diffraction limited} = d_p, \mu\text{m}
\end{align*}

Figure: \(d_T = 2\text{ ft (60.96 cm)}, TT = 300\text{ K}, \lambda_1 = 3.8 \mu\text{m}, \lambda_2 = 4.2 \mu\text{m}, \epsilon = 0.5, \lambda = 4 \mu\text{m}, \text{FPA (256 x 256, 20 } \mu\text{m})\), \(D^* = 8 \times 10^{11} \text{ cm Hz}^{1/2} / \text{W, (S/N)}_D = 1, \Delta f_p = 250 \text{ Hz.}\)
Missile Fineness Ratio May Be Limited by Impact of Body Bending on Flight Control

\[ \omega_{BB} = 18.75 \left( \frac{E \cdot t}{W \cdot (l/d)} \right)^{1/2} \]

Assumes body cylinder structure, thin skin, high fineness ratio, uniform weight distribution, free-free motion. Neglects bulkhead, wing / tail stiffness.

- \( \omega_{BB} \) = 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

Example for Rocket Baseline:

- \( l / d = 18 \)
- \( E_{AVG} = 19.5 \times 10^6 \) psi
- \( t_{AVG} = 0.12 \) in
- \( W = 500 \) lb
- \( E \cdot t / W = 19.5 \times 10^6 \cdot 0.12 / 500 = 4680 \) per in

\[ \omega_{BB} = 18.75 \left( 4680 / 18 \right)^{1/2} = 302 \text{ rad} / \text{sec} = 48 \text{ Hz} \]

\[ \omega_{Actuator} = 100 \text{ rad} / \text{sec} = 16 \text{ Hz} \]

\[ \frac{\omega_{BB}}{\omega_{Actuator}} = 302 / 100 = 3.02 > 2 \]

Nose Fineness Tradeoff

High Nose Fineness Superior Aerodynamically, Low Observables
Example: $l_N / d = 5$ tangent ogive

Low Nose Fineness Ideal Electromagnetically, High Propellant Length / Volume
Example: $l_N / d = 0.5$ (hemisphere)

Moderate Nose Fineness Compromise Dome
Examples: $l_N / d = 2$ tangent ogive, $l_N / d = 2$ faceted, $l_N / d = 2$ window, $l_N / d = 2$ multi-lens
Faceted and Flat Window Domes Can Have Low Dome Error Slope, Low Drag, and Low RCS

- **Firestreak**
- **Mistral**
- **SLAM-ER**
- **JASSM**
- **THAAD**

Faceted Dome (Mistral) Video
Supersonic Body Drag Driven by Nose Fineness while Subsonic Drag Driven by Wetted Area

\[
( C_{D_0} )_{\text{Body}} = ( C_{D_0} )_{\text{Body,Friction}} + ( C_{D_0} )_{\text{Base}} + ( C_{D_0} )_{\text{Body, Wave}}
\]

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

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

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

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

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

Example for Rocket Baseline:

\[
\begin{align*}
(C_{D_0})_{\text{Base, Wave}} & = 0.053 \left( \frac{18}{2725} \right)^{0.2} = 0.14 \\
(C_{D_0})_{\text{Base, Friction}} & = 0.25 / 2 = 0.13 \\
(C_{D_0})_{\text{Base, Powered}} & = (1 - 0.223) \left( \frac{0.25}{2} \right) = 0.10 \\
(C_{D_0})_{\text{Body, Wave}} & = 0.14 \\
(C_{D_0})_{\text{Base, Coast}} & = 0.14 + 0.13 + 0.14 = 0.41 \\
(C_{D_0})_{\text{Body, Powered}} & = 0.14 + 0.10 + 0.14 = 0.38
\end{align*}
\]
Moderate Nose Tip Bluntness Causes a Negligible Change in Supersonic Drag

Steps to Calculate Wave Drag of a Blunted Nose
1. Relate blunted nose tip geometry to pointed nose tip geometry

2. Compute \( (C_D_0)_{\text{Wave,SharpNose}} \) for sharp nose, based on the body reference area
   \[
   (C_D_0)_{\text{Wave,SharpNose}} = \left( 1.59 + \frac{1.83}{M^2} \right) \left\{ \tan^{-1} \left[ \frac{0.5}{l_N/d} \right] \right\}^{1.69}
   \]

3. Compute \( (C_D_0)_{\text{Wave,Hemi}} \) of the hemispherical nose tip \( (l_{\text{NoseTip}}/d_{\text{NoseTip}} = 0.5) \), based on the nose tip area
   \[
   (C_D_0)_{\text{Wave,Hemi}} = \left( 1.59 + \frac{1.83}{M^2} \right) \left\{ \tan^{-1} (0.5) \right\}^{1.69} = 0.665 \left( 1.59 + \frac{1.83}{M^2} \right)
   \]

4. Finally, compute \( (C_D_0)_{\text{Wave,BluntNose}} \) of the blunt nose, based on the body reference area
   \[
   (C_D_0)_{\text{Wave,BluntNose}} = (C_D_0)_{\text{Wave,SharpNose}} \left( \frac{S_{\text{Ref}} - S_{\text{NoseTip}}}{S_{\text{Ref}}} \right) + (C_D_0)_{\text{Wave,Hemi}} \frac{S_{\text{NoseTip}}}{S_{\text{Ref}}}
   \]

Example Rocket Baseline (\( d_{\text{Ref}} = 8 \text{ in} \)) with 10% Nose Tip Bluntness at Mach 2
- \( (C_D_0)_{\text{Wave,SharpNose}} = \left[ 1.59 + \frac{1.83}{(2)^2} \right] \left\{ \tan^{-1} (0.5/2.4) \right\}^{1.69} = 0.14 \)
- \( d_{\text{NoseTip}} = 0.10 \text{ (8)} = 0.8 \text{ in} \)
- \( S_{\text{NoseTip}} = \pi d_{\text{NoseTip}}^2 / 4 = 3.1416 \times 0.8^2 / 4 = 0.50349 \text{ in}^2 = 0.00349 \text{ ft}^2 \)
- \( (C_D_0)_{\text{Wave,Hemi}} = 0.665 \left[ 1.59 + \frac{1.83}{(2)^2} \right] = 1.36 \)
- \( (C_D_0)_{\text{Wave,BluntNose}} = 0.14 \left( 0.349 - 0.003 \right) / 0.349 + (1.36)(0.003) / (0.349) = 0.14 + 0.01 = 0.15 \)
Boattail Decreases Base Pressure Drag Area

Base Pressure Drag Area

Without Boattail

θ_BT

d_Ref

With Boattail

θ_BT

d_BT

During Motor Burn

After Motor Burnout

Note: Boattail angle θ_BT and boattail diameter d_BT limited by propulsion nozzle packaging, tail flight control packaging, and flow separation

Boattailing Reduces Drag for Subsonic Missiles

Note: Boattail half angle should be less than 10 deg, to avoid flow separation.

If \( \alpha \) negative, \( C_N \) negative

Based on slender body theory (Pitts, et al.) and cross flow theory (Jorgensen) references

Valid for \( l/d > 5 \)

Example \( l/d = \) length / diameter = 20

\[ C_N = |(a/b) \cos^2 \phi + (b/a) \sin^2 \phi| |\sin(2\alpha) \cos(\alpha/2)| + 2(l/d) \sin^2 \alpha \]
L / D Is Impacted by $C_{D0}$, Body Fineness, and Lifting Body Cross Section Geometry

\[
L / D = \frac{C_L}{C_D} = \frac{(C_N \cos \alpha - C_{D0} \sin \alpha)}{(C_N \sin \alpha + C_{D0} \cos \alpha)}
\]

For a lifting body,

\[
C_N = \left( a / b \right)^2 \cos^2(\phi) + \left( b / a \right)^2 \sin^2(\phi)
\]

\[
\left| \sin(2\alpha) \cos(\alpha/2) \right| + 2 \left( l / d \right) \sin^2 \alpha
\]

Note:
- If $\alpha$ negative, $L / D$ negative
- $d = 2 \left( a b \right)^{1/2}$
- Launch platform span constraints (e.g., VLS launcher) and length constraints (e.g., aircraft compatibility) may limit missile aero configuration enhancements
Lifting Body Requires Flight at Low Dynamic Pressure to Achieve High Aero Efficiency

L / D = C_L / C_D = ( C_N cos α - C_D0 sin α ) / ( C_N sin α + C_D0 cos α )

\[ | C_N | = \left[ \left( \frac{a}{b} \right) \cos^2(\phi) + \left( \frac{b}{a} \right) \sin^2(\phi) \right] \left[ \sin(2\alpha) \cos\left(\frac{\alpha}{2}\right) \right] + 2 \left( \frac{l}{d} \right) \sin^2\alpha \]

Example figure based on following assumptions:
Body lift only (no surfaces), cruise flight (lift = weight), W = L = 2,000 lb, d = 2 (a b)\(^{1/2}\), S = 2 ft\(^2\), l / d = 10, C_D0 = 0.2

Example:
- q = 500 psf
  - a / b = 1 \Rightarrow L / D = 2.40
  - a / b = 2 \Rightarrow L / D = 3.37
- q = 5,000 psf
  - a / b = 1 \Rightarrow L / D = 0.91
  - a / b = 2 \Rightarrow L / D = 0.96

Note. Example figure based on following assumptions:
- Body lift only (no surfaces)
- Cruise flight (lift = weight)
- W = L = 2,000 lb
- d = 2 (a b)\(^{1/2}\)
- S = 2 ft\(^2\)
- l / d = 10
- C_D0 = 0.2
Trade-off of Low Observables and $(L/D)_{\text{Max}}$ Versus Volumetric Efficiency

Advantages:
- $(L/D)_{\text{Max}}$
- Low RCS

Advantages:
- Payload
- Launch Platform Integration

Tailored Weapons

Conventional Weapons (circular cross section)

- Circular Cross Section
- Body Planform Area $(\text{Body Volume})^{2/3}$
$\Delta C_m / \Delta \alpha$ and Static Margin Define Static Stability

Statically Stable: $\Delta C_m / \Delta \alpha < 0$, with $x_{ac}$ behind $x_{cg}$

Statically Unstable: $\Delta C_m / \Delta \alpha > 0$, with $x_{ac}$ in front of $x_{cg}$

Note: Statically unstable missile requires high bandwidth autopilot.

Autopilot negative rate feedback provides stability augmentation.
Body Aerodynamic Center Is a Function of Angle of Attack, Nose Fineness, and Body Length

\[
\left( \frac{x_{AC}}{l_N} \right)_B = 0.63 \left( 1 - \sin^2 \alpha \right) + 0.5 \left( \frac{l_B}{l_N} \right) \sin^2 \alpha
\]

Note: Based on slender body theory (Pitts, et al.) and cross flow theory (Jorgensen) references. No flare.

\( (x_{AC})_B = \) Location of body aerodynamic center, \( l_N = \) length of nose, \( \alpha = \) angle of attack, \( l_B = \) total length of body.
Based on Slender Body Theory:

- \(( C_{N\alpha} )_F = 2 \left( \frac{d_F}{d} \right)^2 - 1 \)
- \(( x_{ac} )_F = x_F + 0.33 l_F \left( \frac{2 (d_F/d) + 1}{(d_F/d) + 1} \right)\)
- \(( C_{N\alpha} )_B = 2 \) per rad
- \(( x_{ac} )_B = 0.63 l_N \)

\(\Sigma M = 0\) at Aerodynamic Center. For a Body-Flare:

- \(( C_{N\alpha} )_B \left(\frac{x_{CG} - (x_{AC})_B}{d}\right) + (C_{N\alpha})_F \left(\frac{x_{CG} - (x_{AC})_F}{d}\right) = -\left((C_{N\alpha})_B + (C_{N\alpha})_F\right) \left(\frac{(x_{AC} - x_{CG})}{d}\right)\)

Static Margin for a Body-Flare

- \((x_{AC} - x_{CG})/d = -\left((C_{N\alpha})_B \left(\frac{x_{CG} - (x_{AC})_B}{d}\right) + (C_{N\alpha})_F \left(\frac{x_{CG} - (x_{AC})_F}{d}\right)\right) / \left((C_{N\alpha})_B + (C_{N\alpha})_F\right)\)
Example of Static Margin for THAAD (Statically Unstable Missile)

\[ \left( C_{N_\alpha} \right)_B = 57.7 \]
\[ \left( x_{ac} \right)_B = 57.7 \]
\[ \left( C_{N_\alpha} \right)_F = 230.9 + 0.33 \left( 12.0 \right) \left( \frac{2(18.7/14.6)+1}{(18.7/14.6)+1} \right) = 237.1 \text{ in} \]
\[ \left( x_{ac} \right)_F = 237.1 \]
\[ \left( x_{AC} \right) = 140.9 \]
\[ x_{CG_{Launch}} = 146.9 \text{ in} \]
\[ \left( x_{AC} - x_{CG} \right)_{Launch} / d = - \left\{ 2 \left[ \frac{146.9 - 57.7}{14.6} \right] + 1.28 \left[ \frac{146.9 - 237.1}{14.6} \right] \right\} / \left[ 2 + 1.28 \right] = -0.41 \]

\[ \left( C_{N_\alpha} \right)_B = 2 \left[ \frac{18.7}{14.6} \right]^2 - 1 \] = 1.28 per rad
\[ \left( x_{ac} \right)_F = 230.9 + 0.33 \left( 12.0 \right) \left( \frac{2(18.7/14.6)+1}{(18.7/14.6)+1} \right) = 237.1 \text{ in} \]
\[ \left( x_{ac} \right)_B = 0.63 \left( 91.5 \right) = 57.7 \text{ in} \]
\[ x_{CG_{Launch}} = 146.9 \text{ in} \]
\[ \left( x_{AC} - x_{CG} \right)_{Launch} / d = - \left\{ 2 \left[ \frac{146.9 - 57.7}{14.6} \right] + 1.28 \left[ \frac{146.9 - 237.1}{14.6} \right] \right\} / \left[ 2 + 1.28 \right] = -0.41 \]
**Tail Stabilizers Have Lower Drag While Flares Have Lower Aero Heating and Stability Changes**

<table>
<thead>
<tr>
<th>Type Stabilizer</th>
<th>Drag</th>
<th>Span</th>
<th>Heating</th>
<th>$\Delta C_{N_\alpha}$</th>
<th>Tail Control</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flare (e.g., THAAD)</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
</tr>
<tr>
<td>Tails (e.g., Standard Missile)</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
<td>–</td>
</tr>
</tbody>
</table>

Note: Superior, Good, Average, Poor
Wing Sizing Trades

◆ Advantages of Small Wing / Strake / No Wing
  • Range in high supersonic flight / high dynamic pressure
  • Max angle of attack
  • Launch platform compatibility
  • Lower radar cross section
  • Volume and weight for propellant / fuel

◆ Advantages of Larger Wing
  • Range in subsonic flight / low dynamic pressure
  • Lower guidance time constant*
  • Normal acceleration*
  • High altitude intercept*
  • Less body bending aeroelasticity (wing stiffens body)
  • Less seeker error due to dome error slope (lower angle of attack)
  • Less wipe velocity for warhead (lower angle of attack)
  • Lower gimbal requirement for seeker

*Based on assumption of aerodynamic control and angle of attack below wing stall
Most Supersonic Missiles Are Wingless

- Stinger FIM-92
- Grouse SA-18
- Grison SA-19 (two stage)
- Gopher SA-13
- Starburst
- Mistral
- Kegler AS-12
- Archer AA-11
- Gauntlet SA-15
- Magic R550
- Python 4
- U-Darter
- Python 5
- Derby / R-Darter
- Aphid AA-8
- Sidewinder AIM-9X
- ASRAAM AIM-132
- Grumble SA-10 / N-8
- Patriot MIM-104
- Starstreak
- Gladiator SA-12
- PAC-3
- Roland (two stage)
- Crotale
- Hellfire AGM-114
- ATACM MGM-140
- Standard Missile 3 (three stage)
- THAAD

Most Supersonic Missiles Are Wingless

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Wings, Tails, and Canards with Large Area and at High Angle of Attack Have High Normal Force

\[
\left( \frac{C_{N}}{S_{\text{REF}}/S_{W}} \right)_{\text{Wing}} = \begin{cases} 
4 \sin \alpha' \cos \alpha' / (M^2 - 1)^{1/2} + 2 \sin^2 \alpha' & \text{if } M > \left\{ 1 + \left[ \frac{8}{\pi A} \right]^2 \right\}^{1/2} \\
\left( \frac{\pi A}{2} \right) \sin \alpha' \cos \alpha' + 2 \sin^2 \alpha' & \text{if } M < \left\{ 1 + \left[ \frac{8}{\pi A} \right]^2 \right\}^{1/2} 
\end{cases}
\]

Note: Linear wing theory applicable if \( M > \left\{ 1 + \left[ \frac{8}{\pi A} \right]^2 \right\}^{1/2} \), slender wing theory applicable if \( M < \left\{ 1 + \left[ \frac{8}{\pi A} \right]^2 \right\}^{1/2} \), \( A = \text{Aspect Ratio} < 3 \), \( S_{\text{Surface}} = \text{Surface Planform Area} \), \( S_{\text{Ref}} = \text{Reference Area} \)

Example for Rocket Baseline Wing
\( A_{W} = 2.82 \)
\( S_{W} = 2.55 \text{ ft}^2 \)
\( S_{\text{Ref}} = 0.349 \text{ ft}^2 \)
\( \delta = 13 \text{ deg}, \alpha = 9 \text{ deg} \)
\( M = 2 \)
\( \left\{ 1 + \left[ \frac{8}{\pi A} \right]^2 \right\}^{1/2} = 1.35 \)
Since \( M > 1.35 \), use linear wing theory + Newtonian theory
\( \alpha' = \alpha_{W} = \alpha + \delta = 22^\circ \)
\( (C_{N})_{\text{Wing}} S_{\text{Ref}}/S_{W} = 4 \sin 22^\circ \cos 22^\circ / (2^2 - 1)^{1/2} + 2 \sin^2 22^\circ = 1.083 \)
\( (C_{N})_{\text{Wing}} = 1.083 \times (2.55) / 0.349 = 7.91 \)
Aerodynamic Center of a Thin Surface (e.g., Wing, Tail, Canard) Varies with Mach Number

\[
\left( \frac{x_{AC}}{c_{MAC}} \right)_{Surface} = \frac{[A(M^2 - 1)^{1/2} - 0.67]}{[2A(M^2 - 1)^{1/2} - 1]}, \text{ if } M > \sim 2
\]

\[
\left( \frac{x_{AC}}{c_{MAC}} \right)_{Surface} = 0.25, \text{ if } M < \sim 0.7
\]

Note: Based on linear wing theory

Thin wing \( \Rightarrow M (t/c) \ll 1 \)

\( (x_{AC})_{Surface} = \text{Surface aerodynamic center distance from leading edge of } (c_{MAC})_{Surface} \)

\( c_{MAC} = \text{Mean aerodynamic chord} \)

\( A = \text{Aspect ratio} = b^2 / S \)

Example: Rocket Baseline Wing

\( A = 2.82 \)

\( c_{MAC} = 13.3 \text{ in} \)

\( (x_{MAC})_{Wing} = 67.0 \text{ in} \)

\( M = 2 \)

\( (x_{AC} / c_{MAC})_{Wing} = 0.481 \)

\( (x_{AC})_{Wing} = 6.4 \text{ in from mac leading edge} = 73.4 \text{ in from nose tip} \)
Hinge Moment Increases with Dynamic Pressure and Effective Angle of Attack

\[ HM = N_{\text{Surface}} \left( x_{\text{AC}} - x_{\text{HL}} \right)_{\text{Surface}} \]

- **q = 436 psf (M = 0.8)**
- **q = 1242 psf (M = 1.35)**
- **q = 2725 psf (M = 2)**
- **q = 17031 psf (M = 5)**

**Note:** Based on linear wing theory, slender wing theory, and thin wing (M \( t / c \) \( \ll 1 \))

- \( N_{\text{Surface}} \) = Normal force on surface (two panels)
- \( (x_{\text{AC}} - x_{\text{HL}})_W \) = distance from surface aerodynamic center to hinge line of surface

**Example for Rocket Baseline Wing Control**

- \( c_{\text{mac}} = 13.3 \) in
- \( x_{\text{HL}} = 0.25 \) \( c_{\text{mac}} \)
- \( S_{\text{Ref}} = 0.349 \) ft\(^2\)
- \( S_W = 2.55 \) ft\(^2\)
- \( \delta = 13 \) deg, \( \alpha = 9 \) deg
- \( \alpha' = \alpha_W = \alpha + \delta = 22^\circ \)
- \( M = 2, h = 20k \) ft, \( q = 2725 \) psf
- \[ N_W = \left[ C_{N_W} \left( S_{\text{Ref}} / S_W \right) \right] qS_W = 1.083 \left( 2725 \right) \left( 2.55 \right) = 7525 \) lb
- \( x_{\text{AC}} / c_{\text{mac}} = 0.48 \)
- \( HM = 7525 \left( 0.48 - 0.25 \right) \left( 13.3 \right) = 23019 \) in – lb for two panels
Wings, Tails, and Canards Usually Have Greater Skin Friction Drag Than Shock Wave Drag

\[
(C_{D_0})_{\text{Surface, Friction}} = n_{\text{Surface}} \left\{ 0.0133 \left[ \frac{M}{(q \text{ cmac})} \right]^{0.2} \right\} (2 \frac{S_{\text{Surface}}}{S_{\text{Ref}}}), q \text{ in psf, } c_{\text{mac}} \text{ in ft}
\]

\[
(C_{D_0})_{\text{Surface, Wave}} = n_{\text{Surface}} \left( \frac{1.429}{M_{\text{ALE}}^2} \right) \left( 1.2 M_{\text{ALE}}^2 \right)^{3.5} \left[ \frac{2.4}{2.8 M_{\text{ALE}}^2 - 0.4} \right]^{2.5 - 1} \sin^2 \delta_{\text{LE}} \cos \Lambda_{\text{LE}}
\]

\[t_{\text{mac}} \frac{b}{S_{\text{Ref}}} \text{, based on Newtonian impact theory}\]

\[
(C_{D_0})_{\text{Surface}} = (C_{D_0})_{\text{Surface, Wave}} + (C_{D_0})_{\text{Surface, Friction}}
\]

\[n_{\text{Surfaces}} = \text{number of surface planforms ( cruciform = 2 )}\]
\[q = \text{dynamic pressure in psf}\]
\[c_{\text{mac}} = \text{length of mean aero chord in ft}\]
\[\gamma = \text{Specific heat ratio = 1.4}\]
\[M_{\text{ALE}} = M \cos \Lambda_{\text{LE}} = \text{Mach number \perp leading edge}\]
\[\delta_{\text{LE}} = \text{leading edge section total angle}\]
\[\Lambda_{\text{LE}} = \text{leading edge sweep angle}\]
\[t_{\text{mac}} = \text{max thickness of mac}\]
\[b = \text{span}\]

**Example for Rocket Baseline Wing:**

\[n_w = 2, M = 2, h = 20k \text{ ft (q = 2,725 psf)}, c_{\text{mac}} = 1.108 \text{ ft}, S_{\text{Ref}} = 50.26 \text{ in}^2, S_w = 367 \text{ in}^2, \delta_{\text{LE}} = 10.01 \text{ deg}, \Lambda_{\text{LE}} = 45 \text{ deg}, t_{\text{mac}} = 0.585 \text{ in}, b = 32.2 \text{ in}, M_{\text{ALE}} = 1.41 (M = 2)\]

\[\frac{M}{(q \text{ cmac})} = 2 / [2725 (1.108)] = 0.000662 \text{ ft / lb}\]

\[n S_{\text{Surface}} / S_{\text{Ref}} = 2 (367) / 50.26 = 14.60\]

\[(C_{D_0})_{\text{Wing, Friction}} = 0.090\]

\[(C_{D_0})_{\text{Wing, Wave}} = 0.024\]

\[(C_{D_0})_{\text{Wing}} = 0.024 + 0.090 = 0.11\]
Examples of Wing, Tail, and Canard Panel Geometry Alternatives

Parameter | Triangle (Delta) | Aft Swept LE Trapezoid | Bow Tie | Double Swept LE | Rectangle
--- | --- | --- | --- | --- | ---
Variation $x_{AC}$ | – | | | | |
Bending Moment / Friction | | | | | |
Supersonic Drag | | | | | |
RCS | | | | | |
Span Constraint | – | | | | |
Stability & Control | | | | | |
Aeroelastic Stab. | | | | | |

$\lambda = \text{Taper ratio} = c_T / c_R$

$A = \text{Aspect ratio} = b^2 / S = 2 b / [(1 + \lambda) c_R]$

$y_{CP} = \text{Outboard center-of-pressure} = (b / 6)(1 + 2\lambda) / (1 + \lambda)$

$c_{MAC} = \text{Mean aerodynamic chord} = (2 / 3)c_R (1 + \lambda + \lambda^2) / (1 + \lambda)$

Note: Superior | Good | Average | Poor
--- | --- | --- | ---
Based on equal surface area and equal span. Surface area often has more impact than geometry.
Examples of Surface Arrangement and Aerodynamic Control Alternatives

Two Panels (Mono-Wing)
Three (Tri-Tail)
Four (Cruciform)
Six*
Eight*

Folded
Wraparound
Extended
Balanced Actuation Control
Flap Control

Interdigitated
In-line

Note: More than four tails are usually free-to-roll pitch / yaw stabilizers, for low induced roll.
### Most Missiles Use Four Control Surfaces with Combined Pitch / Yaw / Roll Control Integration

<table>
<thead>
<tr>
<th>Control Integ</th>
<th>Control Surfaces</th>
<th>Example</th>
<th>Control Effect</th>
<th>Cost</th>
<th>Packaging</th>
</tr>
</thead>
<tbody>
<tr>
<td>✦ Pitch / Yaw</td>
<td>2</td>
<td>Stinger FIM-92</td>
<td>–</td>
<td>☺</td>
<td>☺</td>
</tr>
<tr>
<td>✦ Pitch / Roll</td>
<td>2</td>
<td>ALCM AGM-86</td>
<td>–</td>
<td>☺</td>
<td>☺</td>
</tr>
<tr>
<td>✦ Pitch / Yaw / Roll</td>
<td>3</td>
<td>SRAM</td>
<td>–</td>
<td>☺</td>
<td>☺</td>
</tr>
<tr>
<td>✦ Pitch / Yaw / Roll</td>
<td>4</td>
<td>Adder AA-12</td>
<td>☺</td>
<td>☺</td>
<td>☺</td>
</tr>
<tr>
<td>✦ Pitch + Yaw + Roll</td>
<td>5</td>
<td>Kitchen AS-4</td>
<td>☺</td>
<td>–</td>
<td>–</td>
</tr>
<tr>
<td>✦ Pitch / Yaw + Roll</td>
<td>6</td>
<td>Derby / R-Darter</td>
<td>☺</td>
<td>–</td>
<td>–</td>
</tr>
</tbody>
</table>

**Note:** Superior ☺ ☺ ☺ ☺ Good ☺ ☺ ☺ Average ☺ ☺ ☺ ☺ Poor ☺ ☺ ☺ ☺
There Are Many Flight Control Configuration Alternatives

Control Design Alternatives

- Tail
  - Cruciform (4)
  - Tri-tail (3)
  - Not Compressed Folded
  - Wraparound Switchblade

- Canard
  - Above
  - Rolling Airframe (2)

- Wing
  - Above

TVC or Reaction Jet Control
- Movable Nozzle
- Jet Tab
- Jet Vane
- Axial Plate
- Secondary Injection
- Normal Jet / JI
- Spanwise Jet / JI

Fixed Surface Alternatives

- Tail
  - Wingless
  - Wing
  - Strake / Canard
  - In Line with Controls
  - Interdigitated with Controls
  - Number (2, 3, 4)

- Tail + Wing
  - In Line with Controls
  - Interdigitated with Controls

- Tail (3, 4, 6, 8)
- Strake / Canard & Tail
- In Line with Controls
- Interdigitated with Controls

- Tail + Canard / Strake
- Tail + Wing

- Tail (3, 4, 6, 8)
Tail Control Is Efficient at High Angle of Attack

\[ C_{N \text{Trim}} \quad C_N \text{ at } \delta = 0 \]

Efficient Packaging

- Low Hinge Moment / Actuator Torque
- Low Induced Rolling Moment
- Efficient at High $\alpha$

Decreasced Lift at Low $\alpha$ if Statically Stable
Tail Control Is More Effective Than Conventional Canard Control at High Angle of Attack

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

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

Conventional Canard Control

Tail Control

$\alpha \sim \text{Angle of Attack (deg)}$

*Note: Additional forward fixed surfaces (such as Python 4) in front of movable canards alleviate stall at high $\alpha$. Free-to-roll tails (such as Python 4) alleviate induced roll from canard control at high $\alpha$. 

2/24/2008
About 70% of Tail Control Missiles Also Have Wings

- JASSM AGM-158  🇺🇸
- Maverick AGM-65  🇺🇸
- CALCM  🇺🇸
- JSOW AGM-154  🇺🇸
- Tomahawk BGM-109  🇺🇸
- Taurus KEPD 350  🇩🇪
- Storm Shadow / Scalp  🇬🇧
- Popeye AGM-142  🇺🇸
- Exocet MIM40  🇫🇷
- TOW2-BGM71D  🇺🇸
- AMRAAM AIM-120  🇺🇸
- Sunburn SS-N-22  🇳🇱
- Standard RIM-66 / 67  🇺🇸
- RBS-70 / 90  🇸🇪
- Shipwreck SS-N-19  🇧🇪
- Super 530  🇫🇷
- Sea Dart (two stage)  🇺🇸
- FSAS Aster  🇫🇷
- R-37 (AA-X-13)  🇷🇺
- Mica  🇫🇷
- Adder AA-12  🇷🇺
- Rapier 2000  🇬🇧
- SD-10 / PL-12  🇳🇱
- Seawolf  🇷🇺
### Tail Control Alternatives: Conventional Balanced Actuation Fin, Flap, and Lattice Fin

<table>
<thead>
<tr>
<th>Type of Tail Control</th>
<th>Control Effectiveness</th>
<th>Drag</th>
<th>Moment</th>
<th>RCS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Balanced Actuation Fin (Example: ASRAAM AIM-132)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Flap (Example: Hellfire AGM-114)</td>
<td></td>
<td></td>
<td>−</td>
<td></td>
</tr>
<tr>
<td>Lattice Fin (Example: Adder AA-12 / R-77)</td>
<td></td>
<td></td>
<td></td>
<td>−</td>
</tr>
</tbody>
</table>

Note: Superior ☄ ☄ ☄ ☄ Good ☄ ☄ Average − Poor − Poor
Lattice Fins Have 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 RCS (cavities, normal leading edges)

♦ High drag at transonic Mach number (choked flow)
Conventional Canard Control Is Efficient at Low Angle of Attack But Stalls at High Alpha

Efficient Packaging

Simplified Manufacturing

Increased Lift at Low $\alpha$ if Statically Stable

Stall at High $\alpha$ if Statically Stable

Induced Roll

Note: Additional forward fixed surface in front of movable canard alleviates stall at high $\alpha$. Free-to-roll tails alleviate induced roll at high $\alpha$. Dedicated roll control surfaces avoid roll control saturation and simplify autopilot design.
Canard Control Missiles Are Wingless and Most Are Supersonic

Stinger FIM-92
Grouse SA-18
Grison SA-19 (two-stage)
Gopher SA-13

Starburst
Gauntlet SA-15
Mistral
AIM-9L

Archer AA-11
Magic R 550
Python 4
U-Darter

Python 5
Derby / R-Darter
Aphid AA-8
Kegler AS-12

GBU-12
GBU-22
GBU-27
GBU-28

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Missiles with Split Canards Have Enhanced Maneuverability at High Angle of Attack

Note: Forward fixed surface reduces local angle-of-attack for movable canard, providing 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 tails and separate roll control ailerons.

Note: $\alpha'$ = Local angle of attack
Wing Control Requires Less Body Rotation But Has High Hinge Moment, Induced Roll and Stall

\[ \Delta C_N \sim C_{N\text{ trim}} \]

- Low Body \( \alpha \) / Dome Error Slope
- Fast Response (if skid-to-turn)
- Poor Actuator Packaging
- Large Hinge Moment
- Larger Wing Size
- Induced Roll
- Wing Stall

(\( \alpha \) small) cg
Wing Control Missile Susceptible to High Vortex Shedding

Strong vortices from wing interact with tail

Video of Vortices from Delta Wing at High Angle of Attack

Source: University of Notre Dame web site: http://www.nd.edu/~ame/facilities/SubsonicTunnels.html

Source: Nielsen Engineering & Research (NEAR) web site: http://www.nearinc.com/near/project/MISDL.htm
Wing Control Missiles 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
TVC and Reaction Jet Flight Control

Liquid Injection

Hot Gas Injection

Jet Vane

Axial Plate

Jet Tab

Movable Nozzle

Note: Jet vanes provide roll control and share actuators with aero control, but have reduced $I_{sp}$.

Note:
- TVC and reaction jet flight control provide high maneuverability at low dynamic pressure.
- TVC usually has lower time constant and miss distance than aero control.
- Reaction jets usually have lower time constant and miss distance than TVC.
- Reaction jets can be either impulse jets or controlled duration jets.

Reaction Jet

Jet flow

M$_\infty$

Jet inter.  Thrust  Jet interaction

M$_\infty$
Most Tactical Missiles with TVC or Reaction Jet Control Also Use Aero Control

♦ Jet Vane + Aero Control:
  - Mica
  - Sea Sparrow RIM-7
  - AIM-9X
  - Sea Wolf GWS 26
  - IRIS-T
  - A-Darter
  - Javelin

♦ Jet Tab + Aero Control:
  - Archer AA-11

♦ Reaction Jet + Aero Control:
  - PAC-3

♦ Movable Nozzle + Aero Control + Reaction Jet:
  - SM-3 Standard Missile
  - Aster FSAF 15

♦ Movable Nozzle + Reaction Jet:
  - THAAD

♦ Reaction Jet:
  - LOSAT

Example Video of TVC (FSAF-15 and Javelin)
Skid-to-Turn Is the Most Common Maneuver Law

- **Skid-To-Turn (STT)**
  - Advantage: Fast response
  - Features
    - Does not require roll commands from autopilot
    - Works best for axisymmetric cruciform missiles

- **Bank-To-Turn (BTT)**
  - Advantage: Provides higher maneuverability for planar wing, noncircular / lifting bodies, and airbreathers
  - Disadvantages
    - Time to roll
    - Requires fast roll rate
    - May have higher dome error slope
  - Features
    - Roll attitude commands from autopilot
    - Small sideslip

- **Rolling Airframe (RA)**
  - Advantage: Requires only two sets of gyros / accelerometers / actuators (packaging for small missile)
  - Disadvantages for aero control
    - Reduced maneuverability for aero control
    - Requires high rate gyros / actuators
    - Requires precision geometry and thrust alignment
  - Features
    - Bias roll rate and roll moment
    - Can use impulse steering (e.g., PAC-3, LOSAT)
    - Compensates for thrust offset
Asymmetric Inlets Require Bank-to-Turn Maneuvering

♦ Examples of Twin Inlet Missiles with Bank-to-Turn

♦ Twin Side Inlets Ramjet: ASMP  

♦ Twin Cheek Inlets Ducted Rocket: HSAD  

♦ Twin Cheek Inlets Ducted Rocket: Meteor  

♦ Examples of Single Inlet Missiles with Bank-to-Turn

♦ Chin Inlet Ramjet: ASALM  

♦ Bottom Inlet Turbojet: BGM-109 Tomahawk  

♦ Bottom Inlet Turbojet: Storm Shadow / Scalp  

♦ Top Inlet Turbofan: AGM-86 ALCM  

Note: Bank-to-turn maneuvering maintains low sideslip for better inlet efficiency.
X Roll Orientation Is Usually Better Than + Roll Orientation

+ Roll Orientation with Four Tail Surfaces Control of Pitch / Yaw / Roll, Looking Forward from Base

X Roll Orientation with Four Tail Surfaces Control of Pitch / Yaw / Roll, Looking Forward from Base

Note: + roll orientation usually has lower trim drag, less static stability and control effectiveness in pitch and yaw, and statically unstable roll moment derivative ( $C_{l\phi} > 0$ ).
X roll orientation has better launch platform compatibility, higher L / D, higher static stability and control effectiveness in pitch and yaw, and statically stable roll moment derivative ( $C_{l\phi} < 0$ ).
Trimmed Normal Force Is Defined at Zero

Pitching Moment

\[ \delta = \delta_{\text{Trim}} \text{ for either statically unstable tail control or statically stable canard control} \]

\[ \delta = 0 \]

\[ \delta = \delta_{\text{Trim}} \text{ for either statically stable tail control or statically unstable canard control} \]

\[ \delta = 0 \]

\[ \alpha_{\text{Trim}} @ C_m = 0 \]

\[ \delta = \delta_{\text{Max}} \]

\[ \text{Angle of Attack (Deg)} \]

\[ \text{Normal Force, } C_N \]

\[ \text{Pitching Moment, } C_m \]

\[ \text{Angle of Attack (Deg)} \]
Relaxed Static Margin Allows Higher Trim Angle of Attack and Higher Normal Force

Note: Rocket Baseline
\( X_{\text{CG}} = 75.7 \) in.
Mach 2

\[ (\alpha + \delta)_{\text{Max}} = 21.8 \text{ deg}, \ (C_{N_{\text{Trim}}})_{\text{Max}} \]

\[ \alpha / \delta = 0.75, \ (\text{Static Margin} = 0.88 \text{ Diam}) \]

\[ \alpha / \delta = 1.5, \ (\text{SM} = 0.43 \text{ Diam}) \]

\[ \alpha / \delta = \infty, \ (\text{SM} = 0) \]

\((C_{N_{\text{Trim}}})_{\text{max}}, \text{Max Trimmed Normal Force Coefficient of Rocket Baseline})\)
Tails Are Sized for Desired Static Margin

\[ \sum M = 0 \text{ at aerodynamic center} \]
\[ (C_{N\alpha})_B \left[ \frac{x_{CG} - (x_{AC})_B}{d} \right] + (C_{N\alpha})_W \left[ \frac{x_{CG} - (x_{AC})_W}{d} \right] S_W / S_{Ref} + (C_{N\alpha})_T \left[ \frac{x_{CG} - (x_{AC})_T}{d} \right] S_T / S_{Ref} \]
\[ = - \left[ (C_{N\alpha})_B + (C_{N\alpha})_W S_W / S_{Ref} + (C_{N\alpha})_T S_T / S_{Ref} \right] \left( \frac{x_{AC} - x_{CG}}{d} \right) \]

Static margin for a specified tail area is
\[ \frac{x_{AC} - x_{CG}}{d} = - \left[ (C_{N\alpha})_B \left[ \frac{x_{CG} - (x_{AC})_B}{d} \right] + (C_{N\alpha})_W \left[ \frac{x_{CG} - (x_{AC})_W}{d} \right] S_W / S_{Ref} + (C_{N\alpha})_T \left[ \frac{x_{CG} - (x_{AC})_T}{d} \right] \right] \left( \frac{S_T / S_{Ref}}{[C_{N\alpha}]_B + (C_{N\alpha})_W S_W / S_{Ref} + (C_{N\alpha})_T S_T / S_{Ref}} \right) \]

Required tail area for a specified static margin is
\[ \frac{S_T}{S_{Ref}} = \frac{(C_{N\alpha})_B \left[ \frac{x_{CG} - (x_{AC})_B}{d} \right] + (C_{N\alpha})_W \left[ \frac{x_{CG} - (x_{AC})_W}{d} \right] S_W / S_{Ref} + \left( (C_{N\alpha})_B + (C_{N\alpha})_W S_W / S_{Ref} \right) \left( \frac{x_{AC} - x_{CG}}{d} \right)}{[C_{N\alpha}]_T \left[ x_{AC} - x_{CG} \right]} \]
Larger Tail Area Is Required for Neutral Stability at High Mach Number

\[
\left( \frac{S_T}{S_{Ref}} \right)_{Neutral} = \left\{ \left( \frac{S_W}{S_{Ref}} \right) \left( \frac{x_{CG} - (x_{AC})_W}{d} \right) \right\} / \left\{ \left( \frac{(x_{AC})_T - x_{CG}}{d} \right) \left( C_{N\alpha} \right)_T \right\}
\]

Assumptions for figure:
• \( x_{CG} \approx l / 2, (x_{AC})_B \approx d, (x_{AC})_T \approx l - d \)
• \( \alpha < 6 \text{ deg}, \text{ turbulent boundary layer} \)
• \( (C_{N\alpha})_B = 2 \text{ per rad} \)
• \( (C_{N\alpha})_T = (C_{N\alpha})_W = \frac{4}{M^2 - 1} \), if \( M > \left\{ 1 + \left[ \frac{8}{(\pi A)} \right]^2 \right\}^{1/2} \)
• \( (C_{N\alpha})_T = (C_{N\alpha})_W = \frac{\pi A}{2}, \text{ if } M < \left\{ 1 + \left[ \frac{8}{(\pi A)} \right]^2 \right\}^{1/2} \)

Example Rocket Baseline:
\( l = 144 \text{ in}, d = 8 \text{ in}, S_w = 2.55 \text{ ft}^2, S_{Ref} = 0.349 \text{ ft}^2, A_w = 2.82, (c_{MAC})_W = 13.3 \text{ in}, x_{MAC} = 67.0 \text{ in from nose tip}, \text{ burnout} \)
\( x_{CG} = 76.2 \text{ in from tip} \), \( M_{max} = 3 \)

\( (x_{AC})_W = 0.49 \times 13.3 = 6.5 \text{ in from leading edge of MAC} \)

\( (x_{AC})_W = 6.5 + 67.0 = 73.5 \text{ in from nose} \)

\( \left( \frac{S_W}{S_{Ref}} \right)_{Neutral} = 0.14 \) (forward wing)

\( \left( \frac{S_T}{S_{Ref}} \right)_{Neutral} = 1.69 \) provides neutral stability

\( \left( \frac{S_T}{S_{Ref}} \right)_{Neutral} = 1.69 \times 0.349 = 0.59 \text{ ft}^2 \)
Stability and Control Derivatives Conceptual Design Criteria

| $\left| \frac{C_{l\delta r}}{C_{l\delta a}} \right| < 0.3$ (Roll Due to Rudder Deflection) | $\left| \frac{C_{l\phi}}{C_{l\delta a}} \right| < 0.5$ (Roll Due to Roll Angle) |

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{l\delta r}$ \hspace{1cm} $C_{l\delta a}$

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{l\phi}$ \hspace{1cm} $C_{l\delta a}$

| $\left| \frac{C_{n\delta a}}{C_{n\delta r}} \right| < 0.2$ (Yaw Due to Aileron Deflection) | $\left| \frac{C_{m\alpha}}{C_{m\delta}} \right| < 1$ (Pitch Due to $\alpha$) |

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{n\delta a}$ \hspace{1cm} $C_{n\delta r}$

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{n\delta a}$ \hspace{1cm} $C_{n\delta r}$

| $\left| \frac{C_{l\beta}}{C_{l\delta a}} \right| < 0.3$ (Roll Due to Sideslip) | $\left| \frac{C_{n\beta}}{C_{n\delta r}} \right| < 1$ (Yaw Due to Sideslip) |

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{l\beta}$ \hspace{1cm} $C_{l\delta a}$

$z$ \hspace{2cm} $y$ \hspace{2cm} $x$

$C_{n\beta}$ \hspace{1cm} $C_{n\delta r}$

Note: The primary control derivative (larger bold font) should be larger than the undesirable stability and control derivative.
Most of the Rocket Baseline Body Buildup Normal Force Is Provided by the Wing

\[
(C_N)_{\text{Total}} = (C_N)_{\text{Wing-Body-Tail}} \approx (C_N)_{\text{Body}} + (C_N)_{\text{Wing}} + (C_N)_{\text{Tail}}
\]

Note: \[(C_D)_{\text{Total}} = (C_D)_{\text{Wing-Body-Tail}} \approx (C_D)_{\text{Body}} + (C_D)_{\text{Wing}} + (C_D)_{\text{Tail}}\]

\[(C_m)_{\text{Total}} = (C_m)_{\text{Wing-Body-Tail}} \approx (C_m)_{\text{Body}} + (C_m)_{\text{Wing}} + (C_m)_{\text{Tail}}\]

Note for figure: \(M = 2, \delta = 0\)
Summary of Aerodynamics

- Conceptual Design Prediction Methods of Bodies and Surfaces
  - Normal force coefficient
  - Drag coefficient
  - Aerodynamic center / pitching moment coefficient / hinge moment
- Design Tradeoffs
  - Diameter
  - Nose fineness
  - Boattail
  - Lifting body versus axisymmetric body
  - Wings versus no wings
  - Tails versus flares
  - Surface planform geometry
  - Flight control alternatives
  - Maneuver alternatives
  - Roll orientation
  - Static margin / time to converge or diverge
  - Tail sizing
Summary of Aerodynamics (cont)

- Stability and Control Design Criteria
  - Static stability
  - Control effectiveness
  - Cross coupling
- Body Buildup
- New Aerodynamics Technologies
  - Faceted / window / multi-lens domes
  - Bank-to-turn maneuvering
  - Lifting body airframe
  - Forward swept surfaces
  - Neutral static margin
  - Lattice fins
  - Split canard control
  - Free-to-roll tails
- Discussion / Questions?
- Classroom Exercise
Aerodynamics Problems

1. Missile diameter tradeoffs include consideration of seeker range, warhead lethality, structural mode frequency, and d___.
2. Benefits of a high fineness nose include lower supersonic drag and lower r____ c____ s______.
3. Three contributors to drag are base drag, wave drag, and s___ f_______ drag.
4. To avoid flow separation, a boattail or flare angle should be less than __ deg.
5. A lifting body is most efficient at a d______ p_______ of about 700 psf.
6. At low angle of attack the aerodynamic center of the body is on the n___.
7. Subsonic missiles often have w____ for enhanced range.
8. The aerodynamic center of the wing is between 25% and 50% of the m___ a__________ c____.
9. Hinge moment increases with the local flow angle due to control surface deflection and the a____ o_ a______.
10. Increasing the surface area increases the s___ f_______ d___.

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11. Leading edge sweep reduces drag and r____ c____ s______.
12. A missile with six control surfaces, four surfaces providing combined pitch / yaw control plus two surfaces providing roll control, has an advantage of good c_____ e____________.
13. A missile with two control surfaces providing only combined pitch / yaw control has advantages of lower c____ and good p________.
14. A tail control missile has larger trim normal force if it is statically u_______.
15. Lattice fins have low h____ m______.
16. Split canards allow higher maximum angle of attack and higher m______________.
17. Two types of unconventional control are thrust vector control and r_______ j__ control.
18. The most common type of TVC for tactical missiles is j__ v___ control.
19. Three maneuver laws are skid to turn, bank to turn, and r______ a________.
20. Bank to turn maneuvering is usually required for missiles with a single wing or with a________ inlets.
Aerodynamics Problems (cont)

21. A missile is statically stable if the aero center is behind the c______ o___
g______.

22. Tail stabilizers have low drag while a f____ stabilizer has low aero heating 
and a relatively small shift in static stability.

23. If the moments on the missile are zero the missile is in t___.

24. Total normal force on the missile is approximately the sum of the normal 
forces on the surfaces (e.g., wing, tail, canard) plus normal force on the 
b___.

25. Increasing the tail area increases the s_____ m________.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- **Propulsion Considerations in Tactical Missile Design**
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

1. Define Mission Requirements
2. Establish Baseline
3. Aerodynamics
4. Propulsion
5. Weight
6. Trajectory
7. Meet Performance?
   - Yes
   - No
   - No
8. Measures of Merit and Constraints
9. Yes

Alternative Mission

Alternative Baseline

Resize / Alt Config / Subsystems / Tech
High Specific Impulse Is Indicative of Lower Fuel / Propellant Consumption

- Turbojet: $I_{sp}$ typically constrained by turbine temperature limit
- Ramjet: $I_{sp}$ typically constrained by combustor insulation temperature limit
- Scramjet: $I_{sp}$ typically constrained by thermal choking
- Solid Rocket: $I_{sp}$ typically constrained by safety
- Ducted Rocket
Cruise Range Is Driven by L/D, I<sub>sp</sub>, Velocity, and Propellant or Fuel Weight Fraction

R = ( L / D ) I<sub>sp</sub> V ln [ W<sub>L</sub> / ( W<sub>L</sub> – W<sub>P</sub> )], Breguet Range Equation

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Subsonic Turbojet Missile</th>
<th>Liquid Fuel Ramjet Missile</th>
<th>Hydrocarbon Fuel Scramjet Missile</th>
<th>Solid Rocket</th>
</tr>
</thead>
<tbody>
<tr>
<td>L / D, Lift / Drag</td>
<td>10</td>
<td>5</td>
<td>3</td>
<td>5</td>
</tr>
<tr>
<td>I&lt;sub&gt;sp&lt;/sub&gt;, Specific Impulse</td>
<td>3,000 s</td>
<td>1,300 s</td>
<td>1,000 s</td>
<td>250 s</td>
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<td>V&lt;sub&gt;AVG&lt;/sub&gt;, Average Velocity</td>
<td>1,000 ft / s</td>
<td>3,500 ft / s</td>
<td>6,000 ft / s</td>
<td>3,000 ft / s</td>
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<tr>
<td>W&lt;sub&gt;P&lt;/sub&gt; / W&lt;sub&gt;L&lt;/sub&gt;, Cruise Propellant or Fuel Weight / Launch Weight</td>
<td>0.3</td>
<td>0.2</td>
<td>0.1</td>
<td>0.4</td>
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<tr>
<td>R, Cruise Range</td>
<td>1,800 nm</td>
<td>830 nm</td>
<td>310 nm</td>
<td>250 nm</td>
</tr>
</tbody>
</table>

Note: Ramjet and Scramjet missiles booster propellant for Mach 2.5 to 4 take-over speed not included in W<sub>P</sub> for cruise. Rockets require thrust magnitude control (e.g., pintle, pulse, or gel motor) for effective cruise. Max range for a rocket is usually a semi-ballistic flight profile, instead of cruise flight. Multiple stages may be required for rocket range greater than 200 nm.
Solid Rockets Have High Acceleration Capability

Solid Rocket
\[ T_{\text{Max}} = 2 P_c A_t = m' V_e \]

Turbojet
\[ T_{\text{Max}} = \frac{\pi}{4} d^2 \rho_0 V_0^2 \left( \frac{V_e}{V_0} - 1 \right) \]

Ramjet
\[ T_{\text{Max}} = \frac{\pi}{4} d^2 \rho_0 V_0^2 \left( \frac{V_e}{V_0} - 1 \right) \]

Note:
- \( P_c \) = Chamber pressure, \( A_t \) = Nozzle throat area, \( m' \) = Mass flow rate
- \( d \) = Diameter, \( \rho_0 \) = Free stream density, \( V_0 \) = Free stream velocity,
- \( V_e \) = Nozzle exit velocity ( Turbojet: \( V_e \sim 2,000 \) ft/s, Ramjet: \( V_e \sim 4,500 \) ft/s, Rocket: \( V_e \sim 6,000 \) ft/s )
### Turbojet Nomenclature

<table>
<thead>
<tr>
<th>Inlet</th>
<th>Compressor</th>
<th>Combustor</th>
<th>Turbine</th>
<th>Nozzle</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>1</td>
<td>2</td>
<td>3</td>
<td>4</td>
</tr>
<tr>
<td></td>
<td>Inlet Entrance</td>
<td>Compressor Entrance</td>
<td>Compressor Exit</td>
<td>Turbine Entrance</td>
</tr>
<tr>
<td>0</td>
<td>Free Stream</td>
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<td></td>
</tr>
</tbody>
</table>

1. Inlet Entrance
2. Compressor Entrance
3. Compressor Exit
4. Turbine Entrance
5. Turbine Exit

```plaintext
Inlet                      Compressor                        Combustor       Turbine              Nozzle
```
High Temperature Compressors Are Required to Achieve High Pressure Ratio at High Speed

\[ T_3 \approx T_0 \left( 1 + \left[ \left( \frac{\gamma_0 - 1}{2} \right) M_0^2 \right] \left( \frac{p_3}{p_2} \right)^{\frac{\gamma_3 - 1}{\gamma_3}} \right) \]

\[ \gamma_0 = 1.4, \quad \gamma_3 \approx 1.29 + 0.16 e^{-0.0007 T_3} \]

Note: Ideal inlet; ideal compressor; low subsonic, isentropic flow

Example:
\[ M_0 = 2, \quad h = 60k \text{ ft (} T_0 = 398 \text{ R } \) \]
\[ \frac{p_3}{p_2} = 5 \Rightarrow T_3 = 1118 \text{ R, } \gamma_3 = 1.36 \]

\[ T_3 = \text{Compressor exit temperature in Rankine, } T_0 = \text{free stream temperature in Rankine, } \gamma = \text{specific heat ratio, } M_0 = \text{free stream Mach number, } p_3 = \text{compressor exit pressure, } p_2 = \text{compressor entrance pressure} \]
High Turbine Temperature Is Required for High Speed Turbojet Missiles

\[ T_4 \approx T_3 + \left( \frac{H_f}{c_p} \right) \frac{f}{a}, \text{T in R} \]

\[ c_{p4} \approx 0.122 \frac{T_4}{T_4}^{0.109}, c_p \text{ in BTU / lb / R} \]

**Example:**

\( M_0 = 2, h = 60K \text{ ft} \) (\( T_0 = 398 \text{ R}\)), \( p_3/p_2 = 5 \Rightarrow T_3 = 1118 \text{ R} \)

RJ-5 fuel (\( H_f = 14,525 \text{ BTU / lb}\)), \( c_p = 0.302 \text{ BTU / lb / R}, f/a = 0.067 \) (stochiometric) \( \Rightarrow T_4 = 1118 + \left( 14525 / 0.302 \right) 0.067 = 4,340 \text{ R} \)

\( T_4 = \) Turbojet turbine entrance temperature in Rankine, \( T_3 = \) compressor exit temperature in Rankine, \( H_f = \) heating value of fuel, \( c_p = \) specific heat at constant pressure, \( f/a = \) fuel-to-air ratio
### Turbine Material Temperature Limit Is a Constraint for a High Speed Turbojet Missile

<table>
<thead>
<tr>
<th>Max Short Duration Temp</th>
<th>Turbine Material</th>
<th>Temperature Constrained Turbines for Mach 4 Cruise</th>
<th>$I_{SP}$ for Mach 4 Cruise</th>
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<tbody>
<tr>
<td>$\approx 3,000R$</td>
<td>Nickel Super Alloys</td>
<td>Very Highly Constrained Turbojet, Air Turbo Rocket, Turbo Ramjet</td>
<td>$\approx 1,000$ s</td>
</tr>
<tr>
<td>$\approx 3,000R$</td>
<td>Titanium Aluminides (lighter weight than nickel super alloys)</td>
<td>Very Highly Constrained Turbojet, Air Turbo Rocket, Turbo Ramjet</td>
<td>$\approx 1,000$ s</td>
</tr>
<tr>
<td>$\approx 3,500R$</td>
<td>Single Crystal Nickel Aluminides</td>
<td>Highly Constrained Turbojet</td>
<td>$\approx 1,200$ s</td>
</tr>
<tr>
<td>$\approx 4,000R$</td>
<td>Ceramic Matrix Composites</td>
<td>Moderately Constrained Turbojet</td>
<td>$\approx 1,500$ s</td>
</tr>
<tr>
<td>$\approx 4,500R$</td>
<td>Rhenium Alloys</td>
<td>Moderately Constrained Turbojet</td>
<td>$\approx 2,000$ s</td>
</tr>
<tr>
<td>$\approx 5,000R$</td>
<td>Tungsten Alloys</td>
<td>Slightly Constrained Turbojet</td>
<td>$\approx 2,500$ s</td>
</tr>
</tbody>
</table>

Note: Constrained turbojet for Mach 4 cruise imposes a limit on turbine temperature that is less than ideal. Constraints could consist of a combination of:

- Constraint on compressor pressure ratio to limit turbine temperature
- Constraint on fuel-to-air ratio to limit turbine temperature
- Use of afterburner to limit turbine temperature
Turbine-Based Missiles Are Capable of Subsonic to Supersonic Cruise

- Turbojet
  - Firebee II
  - SS-N-19 Shipwreck

- Turbo Ramjet
  - Regulus II
  - SR-71

- Air Turbo Rocket
  - Diagram showing components such as hanger, solid fuel, engine, and thrusters.
Compressor Pressure Ratio for Maximum Thrust
Turbojet Is Limited by Turbine Temperature

\[
\left( \frac{p_3}{p_2} \right)_{T_{\text{max}}} \approx \left\{ \frac{T_4}{T_0} \right\}^{1/2} / \left\{ 1 + \left[ \frac{\gamma_0 - 1}{2} \right] M_0^2 \right\} \gamma_4 / (\gamma_4 - 1)
\]

Assumptions: Ideal turbojet (isentropic inlet, compressor, turbine, nozzle; low subsonic and constant pressure combustion; exit pressure = free stream pressure)

Example:
\[ M_0 = 2.0, \ h = 60k \ \text{ft} \ (T_0 = 390 \ \text{R} \ ) \ , \ T_4 = 3,000 \ \text{R}, \ \gamma_4 = 1.31 \]
\[ \left( \frac{p_3}{p_2} \right)_{T_{\text{max}}} = \left\{ \frac{(3000 / 390)^{1/2}}{1 + [(1.4 - 1)/2] 2.0^2} \right\}^{1.31/(1.31 - 1)} = 6.31
\]

Note:
\[ T_0 = \text{Free stream temperature} \]
\[ T_4 = \text{Turbine entrance temperature} \]
\[ \gamma = \text{Specific heat ratio} \]

Turbojet Thrust Is Limited by Turbine Maximum Allowable Temperature

\[ T_{\text{IdealMax}} / (p_0 A_0) = (\gamma_0 M_0 / a_0) (T / m^\prime)_{\text{IdealMax}} \]

Assumption: Ideal turbojet

\[
\begin{align*}
T_{\text{IdealMax}} &= \frac{\gamma_0 M_0}{a_0} T_{\text{m}^\prime} \\
T_{4} &= 2000 \text{ R} \\
T_{4} &= 3000 \text{ R} \\
T_{4} &= 4000 \text{ R} \\
T_{4} &= 5000 \text{ R}
\end{align*}
\]

Note:

\[
(\frac{T}{m^\prime})_{\text{IdealMax}} = V_e - V_0
\]

\[
V_e = \left\{ \frac{2 c_p T_{\text{si}} [1 - (p_0 / p_{\text{si}})]^{(\gamma_5 - 1)/\gamma_5]} \right\}^{1/2}
\]

\[
T_{\text{si}} \approx T_4 - T_3 + T_2
\]

\[
T_3 \approx T_2 \left(\frac{p_3}{p_2}\right)^{(\gamma_3 - 1)/\gamma_3}
\]

\[
T_2 \approx T_0 \left\{1 + \left[\left(\gamma_0 - 1\right)/2\right] M_0^2\right\}
\]

\[
p_{\text{si}} \approx p_4 \left(\frac{T_{\text{si}}}{T_4}\right)^{(\gamma_4 - 1)/\gamma_4}
\]

\[
p_4 = p_3
\]

\[
p_2 \approx p_0 \left\{1 + \left[\left(\gamma_0 - 1\right)/2\right] M_0^2\right\}^{\gamma_0/(\gamma_0 - 1)}
\]

\[
p_0 = \text{Free stream static pressure}
\]

\[
A_0 = \text{Free stream flow area into inlet}
\]

\[
T_4 = \text{Turbine entrance temperature}
\]

Example: \( M_0 = 2, h = 60 \text{ k ft} \) \( T_0 = 390 \text{ R}, p_0 = 1.047 \text{ psi} \), \( T_4 = 3,000 \text{ R}, \gamma_4 = 1.31, \left(\frac{p_3}{p_2}\right)_{T_{\text{max}}} = 6.31, p_2 = 8.19 \text{ psi}, p_3 = 51.7 \text{ psi}, A_0 = 114 \text{ in}^2, T_2 = 702 \text{ R}, T_3 = 1133 \text{ R}, \gamma_3 = 1.36
\]

\[
T_e = 2569 \text{ R}, \gamma_5 = 1.32, p_{\text{si}} = 23.0 \text{ psi}, V_e = 4524 \text{ ft} / \text{s}, \left(\frac{T}{m^\prime}\right)_{\text{IdealMax}} = 2588 \text{ ft} / \text{s},
\]

\[
T_{\text{IdealMax}} / p_0 A_0 = 7.49
\]

\[
T_{\text{IdealMax}} = 7.49 \left(\frac{1.047}{114}\right) = 894 \text{ lb}
\]
Turbojet Specific Impulse Decreases with Supersonic Mach Number

\[(I_{SP})_{\text{ideal@T_{max}}} \frac{g_c c_p T_0}{(a_0 H_f)} = \frac{T_{\text{IdealMax}}}{\left[\frac{p_0 A_0 \gamma_0 M_0}{(T_4 - T_3)}\right]}\]

Assumptions: Ideal turbojet (isentropic inlet, compressor, turbine, nozzle; flow, low subsonic, constant pressure combustion; exit pressure = free stream pressure), max thrust

Example:

\[M_0 = 2, h = 60\text{k ft (} T_0 = 390 \text{ R, } a_0 = 968 \text{ ft }/\text{s}), \text{ RJ-5 fuel (} H_f = 14,525 \text{ BTU }/\text{lbum }, T_4 = 3,000 \text{ R, } c_p = 0.293 \text{ BTU }/\text{lbum }/\text{R, } \gamma_0 = 1.4\]

Calculate \((I_{SP})_{\text{ideal@T_{max}}} \frac{g_c c_p T_0}{(a_0 H_f)} = 0.559\)

\[(I_{SP})_{\text{ideal@T_{max}}} = 0.559 \times \frac{968 \times 14525}{32.2 \times (0.293 \times 390) \times (968 - 968)} = 2136 \text{ s}\]

Note:

\(g_c = \text{Gravitational constant} = 32.2\)
\(c_p = \text{Specific heat at constant pressure}\)
\(T_0 = \text{Free stream temperature}\)
\(a_0 = \text{Free stream speed of sound}\)
\(H_f = \text{Heating value of fuel}\)
\(T_{\text{IdealMax}} = \text{Ideal maximum thrust}\)
\(\gamma = \text{Specific heat ratio}\)
\(T_4 = \text{Combustor exit temperature}\)
\(T_3 = \text{Compressor exit temperature}\)
Tactical Missile Ramjet Propulsion Alternatives

Liquid Fuel Ramjet

Ramjet Sustain Inboard Profile

Solid Fuel Ramjet

Boost

Sustain

Solid Ducted Rocket

Boost

Sustain

Note:
- Solid Ducted Rocket
  - Booster Propellant
  - Fuel

Rocket Boost Inboard Profile
High Specific Impulse for a Ramjet Occurs Using High Heating Value Fuel at Mach 3 to 4

\[
(I_{SP})_{\text{ideal}} \cdot \frac{g_c \cdot c_p}{a_0 \cdot H_f} = \left\{ M_0 \left( \frac{T_4}{T_0} \right) \left\{ 1 + \left[ \frac{\gamma_0 - 1}{2} \right] M_0^2 \right\}^{1/2} - 1 \right\} \left\{ 1 + \left[ \frac{\gamma_0 - 1}{2} \right] M_0^2 \right\} \left( \frac{T_4}{T_0} \right) / \left\{ 1 + \left[ \frac{\gamma_0 - 1}{2} \right] M_0^2 \right\} - 1 \right\}
\]

Assumptions: Ideal ramjet, isentropic inlet and nozzle, low subsonic and constant pressure combustion, exit pressure = free stream pressure, \( \phi \leq 1 \)

Example for Ramjet Baseline:

- \( M = 3.5 \), \( h = 60\,\text{k ft} \) (\( T_0 = 390 \,\text{R} \), \( a_0 = 968 \,\text{ft/s} \)),
- RJ-5 fuel (\( H_f = 14,525 \,\text{BTU/lbm} \), \( T_4 = 4,000 \,\text{R} \),
- \( c_p = 0.302 \,\text{BTU/lbm/R} \), \( \gamma_0 = 1.4 \)

Calculate

\[
(I_{SP})_{\text{ideal}} \cdot \frac{g_c \cdot c_p}{a_0 \cdot H_f} = 3.5 \left\{ \left( \frac{4000}{390} \right) / \left\{ 1 + \left[ \frac{(1.4 - 1)}{2} \right] 3.5^2 \right\}^{1/2} - 1 \right\} / \left\{ 1 + \left[ \frac{(1.4 - 1)}{2} \right] 3.5^2 \right\} \left( \frac{4000}{390} \right) / \left\{ 1 + \left[ \frac{(1.4 - 1)}{2} \right] 3.5^2 \right\} - 1 \right\} = 0.372
\]

\[
(I_{SP})_{\text{ideal}} = 0.372 \cdot \frac{(968)}{(14,525)} / \left[ 32.2 \cdot (0.302) \right] = 1387 \,\text{s}
\]

Note:

- \( g_c \) = Gravitational constant = 32.2
- \( c_p \) = Specific heat at constant pressure
- \( T_0 \) = Free stream temperature
- \( a_0 \) = Free stream speed of sound
- \( H_f \) = Heating value of fuel
- \( \gamma \) = Specific heat ratio
- \( T_4 \) = Combustor exit temperature

High Thrust for a Ramjet Occurs from Mach 3 to 5 with High Combustion Temperature

\[ T_{\text{ideal}} / \left( \phi p_0 A_0 \right) = \gamma_0 M_0^2 \left\{ \left[ \frac{T_4}{T_0} \right] / \{ 1 + [(\gamma_0 - 1) / 2] M_0^2 \} \right\}^{1/2} - 1 \]

Assumptions: Ideal ramjet, isentropic inlet and nozzle, low subsonic and constant pressure combustion, exit pressure = free stream pressure, \( \phi \leq 1 \)

Example for Ramjet Baseline:

\[ M_0 = 3.5, \alpha = 0 \text{ deg}, h = 60k \text{ ft} \left( T_0 = 390 \text{ R}, p_0 = 1.047 \text{ psi} \right), T_4 = 4,000 \text{ R}, \left( f / a \right) = 0.055, \phi = 0.82, A_0 = 114 \text{ in}^2, \gamma_0 = 1.4 \]

\[ T_{\text{ideal}} / \left( \phi p_0 A_0 \right) = 1.4 \left( 3.5 \right)^2 \left\{ \left[ \frac{4000}{390} \right] / \{ 1 + [(1.4 - 1) / 2] \left( 3.5 \right)^2 \} \right\}^{1/2} - 1 \] = 12.43

\[ T_{\text{ideal}} = 12.43 \left( 0.82 \right) \left( 1.047 \right) \left( 114 \right) = 1216 \text{ lb} \]

Note:

\( (T)_{\text{ideal}} = \text{Ideal thrust} \)

\( p_0 = \text{Free stream static pressure} \)

\( A_0 = \text{Free stream flow area into inlet} \)

\( \gamma_0 = \text{Free stream specific heat ratio} \)

\( M_0 = \text{Free stream Mach number} \)

\( T_4 = \text{Combustor exit temperature} \)

\( T_0 = \text{Free stream temperature} \)

\( \phi = \text{Equivalence ratio = fuel-to-air ratio / stochiometric fuel-to-air ratio} \)

Ramjet Combustor Temperature Increases with Mach Number and Fuel Flow

\[ T_4 \approx T_0 \{ 1 + [(\gamma_0 - 1) / 2] M_0^2 \} + (H_f / c_p) \left( \frac{f}{a} \right) \]

Assumptions: Low subsonic combustion. No heat transfer through inlet (isentropic flow). \( \phi \leq 1. \)

\( T_4 \) = combustor exit temperature in Rankine, \( T_0 \) = free stream temperature in Rankine, \( \gamma \) = specific heat ratio, \( M_0 \) = free stream Mach number, \( H_f \) = heating value of fuel, \( c_p \) = specific heat at constant pressure, \( f / a \) = fuel-to-air ratio.

Example:

- \( M_0 = 3.5 \)
- \( h = 60k \) ft (\( T_0 = 390 \) R)
- RJ-5 fuel (\( H_f = 14,525 \) BTU / lb / R)
- \( f / a = 0.055 \)
- \( \gamma_0 = 1.4 \)
- \( c_p = 0.122 T^{0.109} \) BTU / lbm / R.

Note: \( c_p \approx 0.302 +/- 5\% \) if 2500 R < T < 5000 R

- Then \( T_4 = 390 \{ 1 + [(1.4 - 1) / 2] (3.5)^2 \} + [(14525) / (0.302)] \cdot 0.055 = 3,991 \) R

Note: \( (f / a)_{\phi=1} \approx 0.067 \) for stochiometric combustion of liquid hydrocarbon fuel, e.g., RJ-5.
Ramjet Combustor Entrance Mach Number Should Be Low, to Avoid Thermal Choking

\[
(M_3)_{TC} = \frac{-b + \sqrt{b^2 - 4\gamma_3^2}}{2\gamma_3^2}^{1/2}
\]

\[
b = 2\gamma_3 + \frac{(T_{4t}/T_0)(1 + \gamma_4)^2}{(1 + 0.2M_0^2)[1 + (\gamma_4 - 1)/2]}
\]

Assumptions: Constant area combustion, \((\gamma_3 - 1)/2\) \(M_3^2 < 1\), isentropic inlet

Example:

\(M_0 = 2\), \(h = 60k\) ft \((T_0 = 390\) R\), \(T_{4t} = 4,000\) R, \(\gamma_0 = 1.4\)

\(\gamma_4 = 1.29 + 0.16e^{0.0007(4000)} = 1.300\)

\(T_{0t} = (1 + 0.2M_0^2)T_0 = 702\) R

\(\gamma_3 = 1.29 + 0.16e^{-0.0007(702)} = 1.388\)

\(b = 2(1.388) + (4000/390)(1 + 1.300)^2/(1 + 0.2(2^2)[1 + (1.300 - 1)/2]) = -24.211\)

\[(M_3)_{TC} = \frac{24.211 + \sqrt{(-24.211)^2 - 4(1.388^2)}}{2(1.388^2)}^{1/2} = 0.204\]

Note:

\((M_3)_{TC} = \) Combustor entrance Mach number with thermal choking \((M_4 = 1)\)

\(\gamma_3 = \) Specific heat ratio at combustor entrance

\(M_0 = \) Free stream Mach number

\(T_{4t} = \) Combustor exit total temperature

\(T_0 = \) Free stream static temperature

\(\gamma_4 = \) Specific heat ratio in combustion
A Ramjet Combustor with a Low Entrance Mach Number Requires a Small Inlet Throat Area

\[
\frac{A_{IT}}{A_3} = \left[\frac{\gamma + 1}{2}\right]^{\frac{\gamma + 1}{2(\gamma - 1)}} M_3 \left[\frac{1 + (\gamma - 1)}{2} M_3^2 \right]^{\frac{\gamma + 1}{2(\gamma - 1)}} = \left(\frac{216}{215}\right) M_3 \left(1 + \frac{M_3^2}{5}\right)^{-3}
\]

Assumptions: Isentropic inlet, \(M_{IT} = 1\), \(\gamma = 1.4\)

Note:
- \(A_{IT}\) = Inlet throat area
- \(A_3\) = Combustor entrance area
- \(M_3\) = Combustor entrance Mach number
- \(\gamma\) = Specific heat ratio

Example:
- Ramjet Baseline
  - \(A_{IT} = 41.9\) in\(^2\)
  - \(A_3 = 287\) in\(^2\)
  - \(\frac{A_{IT}}{A_3} = 41.9 / 287 = 0.1459\)
  - Assume sonic flow (\(M = 1\)) at \(A_{IT}\)
  - \(M_3 = 0.085\)
  - \(0.085 < (M_3)_{TC} = 0.204\)
Typical Ramjet Has Nearly Constant Pressure Combustion

- Assume Rayleigh Flow, with Heat Addition at
  - Constant Area
  - Negligible Friction
- Pressure Loss in Combustor is Given by
  - \[ \frac{p_4}{p_3} = \frac{1 + \gamma_3 M_3^2}{1 + \gamma_4 M_4^2} \]
- Mach Number Increase in Combustor Is Given by
  - \[ \frac{T_{4t}}{T_0} = \left[ \frac{1 + \gamma_3 M_3^2}{1 + \gamma_4 M_4^2} \right]^{2} \left( \frac{M_4}{M_3} \right)^{2} \left\{ 1 + \left[ \frac{\gamma_4 - 1}{2} \right] M_4^2 \right\} \left\{ 1 + \left[ \frac{\gamma_3 - 1}{2} \right] M_3^2 \right\} \]
- From Prior Example
  - \( M_0 = 2, h = 60k \text{ ft} \ (T_0 = 390 \text{ R}), T_{4t} = 4,000 \text{ R}, \gamma_0 = 1.4, \gamma_4 = 1.300, \text{ and } \gamma_3 = 1.388 \)
- Assume Ramjet Baseline with Sonic Inlet Throat
  - \( \frac{A_{IT}}{A_3} = 41.9 / 287 = 0.1459 \Rightarrow M_3 = 0.085 \)
- Solving Above Equations
  - \( M_4 = 0.304 \)
  - \( \frac{p_4}{p_3} = 0.902 \)
- Assumption of Nearly Constant Pressure Combustion Is Reasonably Accurate
  - 10% error
Minimum Length for the Combustor Is a Function of Combustion Velocity

\[(l_{\text{comb}})_{\text{min}} = t_{\text{comb}} V_{\text{comb}}\]

Example for \(t_{\text{comb}} = 0.002\) s and Subsonic Combustion Ramjet:
- \(V_{\text{comb}} = 200\) ft / s
- \((l_{\text{comb}})_{\text{min}} = 0.002 \times 200 = 0.4\) ft

Example for \(t_{\text{comb}} = 0.002\) s and Scramjet:
- \(V_{\text{comb}} = 3,000\) ft / s
- \((l_{\text{comb}})_{\text{min}} = 0.002 \times 3000 = 6.0\) ft
Ramjet Engine / Booster Integration Options

- **Low Cruise Drag (Modern Ramjets)**
  - Integral-Rocket Ramjet (IRR)
  - Forward Booster

- **High Cruise Drag**
  - Podded Ramjet
  - Podded Ramjet, Aft Drop-off Booster

Fuel
Boost Propellant

Aft Drop-off Booster
Podded Drop-off Booster
Podded IRR

## Ramjet Engine / Booster Integration Trades

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<tr>
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<th>Diameter</th>
<th>Weight</th>
<th>Ejectables</th>
<th>Cruise Drag</th>
<th>Carry Drag</th>
<th>Cost</th>
<th>Cycle Compatibility</th>
<th>Inlet Compatibility</th>
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<tr>
<td>Integral Rocket – Ramjet (IRR)</td>
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<tr>
<td>Aft Booster (Drop-off)</td>
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<td>Forward Booster</td>
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</tbody>
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Ramjets with Internal Boosters and No Wings Have Low Drag

- Poded Ramjet
- Poded IRR
- Poded Ramjet, Aft Drop Off Booster

- IRR
- Aft Drop Off Booster
- Forward Booster
- Poded Drop Off Booster

Note:
Nose Fineness Ratio ≥ 2.25
Nose Bluntness Ratio ≤ 0.20

## Ramjet Inlet Options

<table>
<thead>
<tr>
<th>Type Inlet</th>
<th>Sketch</th>
<th>Placement</th>
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</thead>
<tbody>
<tr>
<td>Nose</td>
<td><img src="Sketch.png" alt="Nose Sketch" /></td>
<td>Nose-full axisymmetric</td>
</tr>
<tr>
<td>Chin</td>
<td><img src="Sketch.png" alt="Chin Sketch" /></td>
<td>Forward upside in nose compression field-partial axisymmetric</td>
</tr>
<tr>
<td>Forward Cruciform Axisymmetric</td>
<td><img src="Sketch.png" alt="Forward Cruciform Sketch" /></td>
<td>Forward in nose compression field-cruciform (four) axisymmetric</td>
</tr>
<tr>
<td>Aft Cruciform Axisymmetric</td>
<td><img src="Sketch.png" alt="Aft Cruciform Sketch" /></td>
<td>Aft-cruciform (four) axisymmetric</td>
</tr>
<tr>
<td>Under Wing Axisymmetric</td>
<td><img src="Sketch.png" alt="Under Wing Sketch" /></td>
<td>In planar wing compression field-twin axisymmetric</td>
</tr>
<tr>
<td>Twin Two-dimensional</td>
<td><img src="Sketch.png" alt="Twin Sketch" /></td>
<td>Aft-twin cheek-mounted two dimensional</td>
</tr>
<tr>
<td>Underslung Axisymmetric</td>
<td><img src="Sketch.png" alt="Underslung Sketch" /></td>
<td>Aft underside-full axisymmetric</td>
</tr>
<tr>
<td>Underslung Two-dimensional</td>
<td><img src="Sketch.png" alt="Underslung Sketch" /></td>
<td>Aft underside-belly mounted two dimensional</td>
</tr>
<tr>
<td>Cruciform Two-dimensional</td>
<td><img src="Sketch.png" alt="Cruciform Sketch" /></td>
<td>Aft-cruciform (four) two dimensional</td>
</tr>
</tbody>
</table>

### Ramjet Inlet Concept Trades

<table>
<thead>
<tr>
<th>Type Inlet</th>
<th>Pressure Recovery</th>
<th>Carriage Envelope</th>
<th>Alpha Capability</th>
<th>Weight</th>
<th>Drag</th>
<th>Warhead Shrouding</th>
<th>Inlet Cost</th>
<th>Preferred Steering</th>
<th>Preferred Control</th>
<th>Prime Mission Suitability</th>
</tr>
</thead>
</table>

**Selection Factors**

- **Preferred Steering**
- **Preferred Control**
- **Prime Mission Suitability**

**Note:**
- BTT = Bank to Turn
- STT = Skid to Turn
- W = Wing
- C = Canard
- T = Tail

Current Supersonic Air-breathing Missiles Have Either a Nose Inlet or Axisymmetric Aft Inlets

- United Kingdom
  - Sea Dart GWS-30

- France
  - ASMP
  - ANS

- Russia
  - AS-17 / Kh-31
  - Kh-41
  - SS-N-22 / 3M80
  - SA-6
  - SS-N-19
  - SS-N-26

- China
  - C-101
  - C-301

- Taiwan
  - Hsiung Feng III

- India
  - BrahMos

Aft inlets have lower inlet volume and do not degrade lethality of forward located warhead.
Nose Inlet may have higher flow capture, pressure recovery, smaller carriage envelope, and lower drag.
**Shock on Inlet Cowl Lip Prevents Spillage**

- **Inlet w/o External Compression**
  - Inlet Swallows 100% of the Free Stream Flow
  - External Compression Required for Efficient Pressure Recovery if Mach Number > 2 and Inlet Start at Low Supersonic Mach number

- **External Compression Inlet (with Spillage)**
  - Shocks Converge Outside Inlet Lip (Results in Spillage Air)

- **External Compression Inlet (w/o Spillage)**
  - Inlet Swallows 100% of the Free Stream Flow
  - Shocks Converge at Inlet Lip (Inlet Captures Maximum Free Stream Flow)
Shock Wave Angle Increases with Deflection Angle and Decreases with Mach Number

\[
\tan (\alpha + \delta) = 2 \cot \theta_{2D} \left( M^2 \sin^2 \theta_{2D} - 1 \right) / \left[ 2 + M^2 \left( \gamma + 1 - 2 \sin^2 \theta_{2D} \right) \right], \text{ for 2D flow, perfect gas}
\]

Note: \( \theta_{2D} \) = 2D shock wave angle, \( M \) = Mach number, \( \alpha \) = angle of attack, \( \delta \) = body deflection angle, \( \gamma \) = specific heat ratio, \( \theta_{\text{conical}} \approx 0.81 \theta_{2D} \)

Example for Ramjet Baseline:

\( \delta = 17.7 \text{ deg}, \ M = 3.5, \ \alpha = 0 \text{ deg}, \ \gamma = 1.4 \)

\( \Rightarrow \theta_{2D} = 32 \text{ deg} \)

\( \theta_{\text{conical}} \approx 0.81 \theta_{2D} = 0.81 \left( 32 \right) = 26 \text{ deg} \)

Approximate estimate of \( \theta \):

\( \theta_{2D} \approx \mu + \alpha + \delta = \sin^{-1} \left( \frac{1}{M} \right) + \alpha + \delta \)

\( \theta_{\text{conical}} \approx 0.81 \theta_{2D} = 0.81 \left[ \sin^{-1} \left( \frac{1}{M} \right) + \alpha + \delta \right] \)
Capture Efficiency of an Inlet Increases with Mach Number

\[
\left( \frac{A_0}{A_c} \right)_{\text{conical}} = \frac{(h/l) \left(1 + \delta M + \alpha M\right)}{\left[\left(1 - 0.23\delta M + \alpha M\right)(\delta + h/l)\right]}, \text{ conical nose with forward inlet}
\]

\[
\left( \frac{A_0}{A_c} \right)_{2D} = \frac{(h/l) \left(1 + \delta M + \alpha M\right)}{\left[(1 + \alpha M)(\delta + h/l)\right]}, \text{ 2D nose with forward inlet}
\]

Note: \( A_0 / A_c \leq 1\), \( A_c \) = inlet capture area, \( A_0 \) = free stream flow area, \( \delta \) = deflection angle in rad, \( h \) = inlet height, \( l \) = distance from nose tip to inlet.

Example for baseline ramjet (conical nose):
- \( h = 3 \) in
- \( l = 23.5 \) in
- \( h / l = 0.1277 \)
- \( A_c = 114 \) in\(^2\)
- \( \delta = 17.7 \) deg (0.3089 rad)
- \( M = 3.5, \alpha = 0 \) deg
- \( A_0 / A_c = 0.81 \Rightarrow A_0 = 92 \) in\(^2\)
- Spillage = \( A_c - A_0 = 114 - 92 = 22 \) in\(^2\)
Isentropic Compression Allows Inlet Start at Lower Mach Number

\[
\frac{A_{IT}}{A_0} = 1.728 \left( M_{IE} \right)_{start} \left[ 1 + 0.2 \left( M_{IE} \right)_{start}^2 \right]^3, \text{ Assumptions: } 2-D \text{ inlet, Isentropic flow through inlet (} n = \infty), \gamma = 1.4
\]

\[
\frac{A_{IT}}{A_0} = \left( M_{IE} \right)_{start} \left[ \frac{0.4 \left( M_{IE} \right)_{start}^2 + 2}{2.4 \left( M_{IE} \right)_{start}^2} \right]^{3.5} \left[ \frac{2.8 \left( M_{IE} \right)_{start}^2 - 0.4}{2.4} \right]^{2.5} \left[ \frac{1.2 / (1 + 0.2 \left( M_{IE} \right)_{start}^2)}{2.4} \right]^{2.5}, \text{ Assumptions: } 2-D \text{ inlet, single normal shock (} n = 1), \gamma = 1.4
\]

Note: \( A_{IT} = \) inlet throat area, \( A_0 = \) free stream flow area, \( (M_{IE})_{start} = \) inlet entrance start Mach number, \( \gamma = \) specific heat ratio, \( n = \) number of shocks

Example for ramjet baseline
\[
A_{IT} = 0.29 \text{ ft}^2
\]
\[
A_c = 114 \text{ in}^2 = 0.79 \text{ ft}^2 \Rightarrow \frac{A_{IT}}{A_c} = 0.367
\]

Process:
1. Assume \((M_{IE})_{start}\)
2. Compute capture efficiency \(\frac{A_{IT}}{A_0}\)
3. Compute \((M_{IE})_{start}\) and compare with assumed \((M_{IE})_{start}\)
4. Iterate until convergence

Limit for isentropic compression
- From Prior Figure, \(\frac{A_0}{A_c} = 0.53\)
- Compute \(A_{IT} / A_c = (A_{IT} / A_0) / (A_0 / A_c) = 0.367 / 0.53 = 0.69 \Rightarrow (M_{IE})_{start} = 1.8\)

Ramjet baseline has mixed compression with \(n = 5\). Actual inlet start Mach number is \((M_{IE})_{start} > 1.8\)
Forebody Shock Compression Reduces the Inlet Entrance Mach Number

\[
(M_{IE})_{2D} = \left\{ \frac{36 M_0^4 \sin^2 \theta_{2D} - 5 [ M_0^2 \sin^2 \theta_{2D} - 1 ] [ 7 M_0^2 \sin^2 \theta_{2D} + 5 ]}{[ 7 M_0^2 \sin^2 \theta_{2D} - 1 ][ M_0^2 \sin^2 \theta_{2D} + 5 ]} \right\}^{1/2}
\]

\[
\tan (\alpha + \delta) = 2 \cot \theta_{2D} \left( M_0^2 \sin^2 \theta_{2D} - 1 \right) / \left[ 2 + M_0^2 \left( 2.4 - 2 \sin^2 \theta_{2D} \right) \right]
\]

Assumptions: 2D flow, perfect gas, \( \gamma \) = specific heat ratio = 1.4

Note: \( M_{IE} \) = inlet entrance Mach number, \( M_0 \) = free stream Mach number, \( \theta \) = oblique shock angle, \( \alpha \) = angle of attack, \( \delta \) = body deflection angle

Example for ramjet baseline

\( \delta = 17.7 \) deg

\( (M_{IE})_{\text{start}} = 1.8 \) (from prior example)

Compute \( M_0 = 2.55 \)

Note: Ramjet baseline forebody is conical, not 2D
Optimum Forebody Deflection Angle(s) for Best Pressure Recovery Increases with Mach Number

Note: $\delta_{\text{Total}}$ = Total deflection angle, $\delta_1$ = 1st deflection angle, $\delta_2$ = 2nd deflection, $\delta_3$ = 3rd deflection.

Optimum deflection angle provides equal loss in total pressure across each shock wave.

Optimum deflection angles are nearly equal for $M > 4$.

Example: Optimum forebody deflection angles for double wedge ($n = 3$) at Mach 2: $\delta_1 = 10.4$ deg, $\delta_2 = 11.2$ deg $\Rightarrow \delta_{\text{Total}} = 10.4 + 11.2 = 21.6$ deg

**Oblique Shocks Prior to the Inlet Normal Shock Are Required to Satisfy MIL-E-5008B**

MIL-E-5008B Requirement: \( \frac{p_{t_{\text{inlet}}}}{p_{t_0}} = 1 - 0.075 ( M - 1 )^{1.35} \)

- **n = 1** (Normal Shock)
- **n = 2** (1 Optimum Oblique Shock + Normal Shock)
- **n = 3** (2 Opt Oblique Shocks + Normal Shock)
- **n = 4** (3 Opt Oblique Shocks + Normal Shock)
- **Ideal Isentropic Inlet**
- **MIL-E-5008B**

**Note:**
- 2D flow assumed
- \( p_{t_{\text{inlet}}} \) = Inlet total pressure
- \( p_{t_0} \) = Free stream total pressure

Example: MIL-E-5008B requirement for Mach 3.5 (\( \frac{p_{t_{\text{inlet}}}}{p_{t_0}} = \eta_{\text{inlet}} = 0.74 \)) can be satisfied only if there are more than three oblique shocks prior to inlet normal shock.

High Density Fuels Provide Higher Volumetric Performance but Have Higher Observables

<table>
<thead>
<tr>
<th>Type Fuel</th>
<th>Density, lb / in³</th>
<th>Volumetric Performance, BTU / in³</th>
<th>Low Observables</th>
</tr>
</thead>
<tbody>
<tr>
<td>Liquid Ramjet ( RJ-4, RJ-5, RJ-6, RJ-7 )</td>
<td>~ 0.040</td>
<td>○ 581</td>
<td>○</td>
</tr>
<tr>
<td>HTPB</td>
<td>~ 0.034</td>
<td>○ 606</td>
<td>○</td>
</tr>
<tr>
<td>Slurry ( 40% JP-10 / 60% carbon )</td>
<td>~ 0.049</td>
<td>○ 801</td>
<td>○</td>
</tr>
<tr>
<td>Solid Carbon ( graphite )</td>
<td>~ 0.075</td>
<td>○ 1132</td>
<td>–</td>
</tr>
<tr>
<td>Slurry ( 40% JP-10 / 60% aluminum )</td>
<td>~ 0.072</td>
<td>○ 866</td>
<td>–</td>
</tr>
<tr>
<td>Slurry ( 40% JP-10 / 60% boron carbide )</td>
<td>~ 0.050</td>
<td>○ 1191</td>
<td>–</td>
</tr>
<tr>
<td>Solid Mg</td>
<td>~ 0.068</td>
<td>○ 1200</td>
<td>–</td>
</tr>
<tr>
<td>Solid Al</td>
<td>~ 0.101</td>
<td>○ 1300</td>
<td>–</td>
</tr>
<tr>
<td>Solid Boron</td>
<td>~ 0.082</td>
<td>○ 2040</td>
<td>–</td>
</tr>
</tbody>
</table>

Superior ☺ Above average ☻ Average ○ Below average –

2/24/2008
Ducted Rocket Design Implications

Excess Fuel from Gas Generator
- ~ 30% ⇒ Behaves more like a rocket (higher burn rate, higher burn temperature, lower I<sub>SP</sub>)
- ~ 70% ⇒ Behaves more like a ramjet (higher I<sub>SP</sub>, lower burn rate, lower burn temperature)

Choice of Fuel
- Metal (e.g., B, Al, Mg) ⇒ Higher I<sub>SP</sub>, higher density, deposits, higher observables
- Carbon based (e.g., C, HTPB) ⇒ Lower observables, higher reliability, lower I<sub>SP</sub>

Choice of Oxidizer
- AP ⇒ Higher burn rate, lower hazard, HCl contrail
- Min Smoke (e.g., HMX, RDX) ⇒ Lower Observables, lower heating value, lower burn rate, hazardous

Thrust Magnitude Control Approaches
- Pintle or valve in gas generator throat
- Retractable wires in grain
High Propellant Fraction Increases Burnout Velocity

\[ \Delta V = -g_c I_{sp} \ln \left( 1 - \frac{W_p}{W_i} \right) \]

Assumption: \( T >> D, T >> W \sin \gamma, \gamma = \text{const} \)

Example: Rocket Baseline
- \( W_{i,\text{boost}} = W_L = 500 \text{ lb}, W_{p,\text{boost}} = 84.8 \text{ lb} \)
- \( I_{sp,\text{boost}} = 250 \text{ s} \)
- \( \frac{W_{p,\text{boost}}}{W_i} = 84.8 / 500 = 0.1696 \)
- \( \Delta V = -32.2 \times (250) \ln (1 - 0.1696) = 1496 \text{ ft/s} \)
High Specific Impulse Requires High Chamber Pressure and Optimum Nozzle Expansion

\[ I_{SP} = c_d \left\{ \left[ \frac{2 \gamma^2}{(\gamma - 1)} \right] \left[ \frac{2}{(\gamma + 1)} \right] \left( \frac{\gamma + 1}{\gamma - 1} \right) \left[ 1 - \left( \frac{p_e}{p_c} \right) \left( \frac{\gamma - 1}{\gamma} \right) \right] \right\}^{1/2} + \left( \frac{p_e}{p_c} \right) \varepsilon - \left( \frac{p_0}{p_c} \right) \varepsilon \right\} \frac{c^*}{g_c} \]

\[ T = w'_p I_{SP} = (\frac{g_c}{c^*}) p_c A_t I_{SP} \]

\[ \varepsilon = \left\{ \left[ \frac{2}{(\gamma + 1)} \right] \left[ \frac{1}{(\gamma - 1)} \right] \left[ \frac{1}{(\gamma + 1)} \right] \right\}^{1/2} \left\{ \left( \frac{p_e}{p_c} \right) \left[ 1 - \left( \frac{p_e}{p_c} \right) \left( \frac{\gamma - 1}{\gamma} \right) \right] \right\}^{1/2} \]

Note:
- \( \varepsilon \) = nozzle expansion ratio
- \( p_e \) = exit pressure
- \( p_c \) = chamber pressure
- \( p_0 \) = atmospheric pressure
- \( w'_p \) = propellant weight flow rate
- \( A_t \) = nozzle throat area (minimum, sonic, choked)
- \( \gamma \) = specific heat ratio = 1.18 in figure
- \( c_d \) = discharge coefficient = 0.96 in figure
- \( c^* \) = characteristic velocity = 5,200 ft/s in figure
- \( h = 20k \) ft, \( p_0 = 6.75 \) psi in figure

Example for Rocket Baseline:
- \( \varepsilon = A_e / A_t = 6.2 \Rightarrow p_e / p_c = 0.02488, A_t = 1.81 \) in\(^2\)
- \( (p_c)_{boost} = 1769 \) psi, \( p_e = 44 \) psi, \( (I_{SP})_{boost} = 257 \) s
- \( (I_{SP})_{\varepsilon=6.2} / (I_{SP})_{\varepsilon=1} = 257 / 200 = 1.29 \)
- \( (T)_{boost} = (32.2 / 5200) (1769) (1.81) (257) = 5096 \) lb
- \( (p_c)_{sustain} = 301 \) psi, \( p_e = 7.49 \) psi, \( (I_{SP})_{sustain} = 239 \) s
- \( (I_{SP})_{\varepsilon=6.2} / (I_{SP})_{\varepsilon=1} = 240 / 200 = 1.20 \)
- \( (T)_{sustain} = (32.2 / 5200) (301) (1.81) (240) = 810 \) lb
High Propellant Weight Flow Rate Requires High Chamber Pressure and Large Nozzle Throat

\[ w_p = g_c \cdot p_c \cdot A_t / c^* \]

Rocket Baseline \( A_t \) for Boost:
\( c^* = 5200 \text{ ft} / \text{s} \)
\( (p_c)_{\text{boost}} = 1,769 \text{ psi} \)
\[ w_p = W_p / t_b = 84.8 / 3.69 = 23.0 \text{ lb} / \text{s} \]
\[ p_c \cdot A_t = c^* \cdot w_p / g_c = 5200 \cdot 23.0 / 32.2 = 3,714 \text{ lb} \]
\[ A_t = 3714 / 1769 = 2.10 \text{ in}^2 \]

Note: \( A_t \) = nozzle throat area, \( c^* \) = characteristic velocity, \( w_p \) = propellant weight flow rate, \( g_c \) = gravitational constant, \( p_c \) = chamber pressure
High Chamber Pressure Requires Large Propellant Burn Area and Small Nozzle Throat

\[ Ab = g_c p_c A_t / ( \rho c^* r ) \]

\[ r = r_{1000 \text{ psi}} ( p_c / 1000 )^n \]

Example \( Ab \) for Rocket Baseline:

\[ A_t = 1.81 \text{ in}^2 \]
\[ \rho = 0.065 \text{ lb} / \text{in}^3 \]
\[ n = 0.3 \]
\[ r_{1000 \text{ psi}} = 0.5 \text{ in} / \text{s} \]
\[ c^* = 5,200 \text{ ft} / \text{s} \]
\[ T_{\text{atmosphere}} = 70 \degree \text{F} \]

For sustain ( \( p_c = 301 \text{ psi} \)):

• \( r = 0.5 ( 301 / 1000 )^{0.3} = 0.35 \text{ in} / \text{s} \)
• \( Ab = 149 \text{ in}^2 \)

For boost ( \( p_c = 1,769 \text{ psi} \) )

• \( r = 0.59 \text{ in} / \text{s} \)
• \( Ab = 514 \text{ in}^2 \)

Note: \( A_b \) = propellant burn area, \( g_c \) = gravitation constant, \( A_t \) = nozzle throat area, \( \rho \) = density of propellant, \( c^* \) = characteristic velocity, \( r \) = propellant burn rate, \( r_{1000 \text{ psi}} \) = propellant burn rate at \( p_c = 1,000 \text{ psi} \), \( p_c \) = chamber pressure, \( n \) = burn rate exponent
Conceptual Design Sizing Process for a Rocket Motor

1. Define Altitude and Required Thrust-time

2. Assume Propellant (Characteristic Velocity, Nominal Burn Rate, Burn Rate Exponent), Chamber Pressure, Burn Area, and Nozzle Geometry (Expansion Ratio, Throat Area)

3. Compute $I_{sp}$ and Thrust

   **OK?**
   - Yes
   - No

4. Compute Propellant Weight Flow Rate and Propellant Used

   **OK?**
   - Yes
   - No

5. Determine Diameter and Length to Satisfy $w_p$ and $A_e$

   **OK?**
   - Yes
   - No
Conventional Solid Rocket Thrust-Time Design Alternatives with Propellant Cross-Section

Example Mission

• Cruise

• Dive at approximate constant dynamic pressure

• Climb at approximate constant dynamic pressure

• Fast launch – approximate cruise

• Fast launch – approximate cruise – high speed terminal

Thrust Profile

Example Web Cross Section / Volumetric Loading

Note: High thrust and chamber pressure require large surface burn area.

End Burner Radial Slotted Tube

Medium Burn Rate Propellant

High Burn Rate Propellant

Photo Courtesy of BAE
Conventional Rocket Has Fixed Burn while Thrust Magnitude Control Can Vary Burn Interval

Conventional Fixed Burn Interval (Boost)

- End Burning

Conventional Fixed Burn Interval (Boost – Sustain)

- Radial Boost
- End Burning Sustain
- Simultaneous Burning

Concentric Radial Burning

- High Burn Rate Boost
- Low Burn Rate Sustain

Pulse Motor TMC Variable Burn Interval (Boost – Coast – Boost / Sustain - Coast)

1st Pulse: Radial Boost
2nd Pulse: End Burning Sustain
Separate Burning (Pulsed Motor)

1st Pulse: Radial Boost
2nd Pulse: Radial Sustain / Boost
Separate Burning (Pulsed Motor)

Note: Each pulse increases motor cost approximately 40%.
Tactical Rocket Motor Thrust Magnitude Control Alternatives

Solid Pulse Motor
- ☑ High $I_{sp}$
- ☹ Limited Pulses

Solid Pintle Motor
- ☑ Continuously Select Up to 40:1 Variation in Thrust
- ☑ Reduce MEOP on Hot Day
- ☹ Good $I_{sp}$ Only If Burn Rate Exponent $n \rightarrow 1$

Bi-propellant Gel Motor
- ☑ High $I_{sp}$
- ☑ Duty Cycle Thrust
- ☑ Insensitive Munition
- ☹ Lower Max Thrust
- ☹ Toxicity
### Solid Rocket Propellant Alternatives

<table>
<thead>
<tr>
<th>Type</th>
<th>$I_{SP}$, Specific Impulse, s</th>
<th>$\rho$, Density, lb / in³</th>
<th>Burn Rate @ 1,000 psi, in / s</th>
<th>Safety</th>
<th>Observables</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Min Smoke</strong></td>
<td>220 - 255</td>
<td>0.055 - 0.062</td>
<td>0.25 - 2.0</td>
<td>☺</td>
<td>–</td>
</tr>
<tr>
<td><strong>Reduced Smoke</strong></td>
<td>250 - 260</td>
<td>0.062</td>
<td>0.1 - 1.5</td>
<td>☺</td>
<td>–</td>
</tr>
<tr>
<td><strong>High Smoke</strong></td>
<td>260 - 265</td>
<td>0.065</td>
<td>0.1 - 3.0</td>
<td>☺</td>
<td>–</td>
</tr>
</tbody>
</table>

- **Min Smoke**: No Al fuel or AP oxidizer. Either Composite with Nitramine Oxidizer (CL-20, ADN, HMX, RDX) or Double Base. Very low contrail ($H_2O$).
- **Reduced Smoke**: No Al (binder fuel). AP oxidizer. Low contrail ($HCl$).
- **High Smoke**: Al fuel. AP oxidizer. High smoke ($Al_2O_3$).

- Superior (☺)
- Above Average (☻)
- Average (○)
- Below Average (–)
# Steel is the Most Common Motor Case Material

<table>
<thead>
<tr>
<th>Type</th>
<th>Temperature</th>
<th>Volumetric Efficiency</th>
<th>Weight</th>
<th>IM</th>
<th>Airframe / Launcher Attachment</th>
<th>Cost</th>
</tr>
</thead>
<tbody>
<tr>
<td>♦ Steel</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>♦ Aluminum</td>
<td></td>
<td></td>
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<td></td>
<td></td>
</tr>
<tr>
<td>♦ Strip Steel / Epoxy Laminate</td>
<td></td>
<td></td>
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<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>♦ Composite</td>
<td></td>
<td></td>
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<td></td>
<td></td>
</tr>
<tr>
<td>♦ Titanium</td>
<td></td>
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<td></td>
</tr>
</tbody>
</table>

- Superior
- Above Average
- Average
- Below Average
**Heat Transfer Drives Rocket Nozzle Materials, Weight, and Cost**

<table>
<thead>
<tr>
<th>Rocket Nozzle Element</th>
<th>High Heating (High Chamber Pressure or Long Burn) ⇒ High Cost / Heavy Nozzle</th>
<th>Low Heating (Low Chamber Pressure or Short Burn) ⇒ Low Cost / Light Weight Nozzle</th>
</tr>
</thead>
<tbody>
<tr>
<td>Housing Material Alternatives</td>
<td>♦ Steel</td>
<td>♦ Cellulose / Phenolic</td>
</tr>
<tr>
<td></td>
<td>♦ Tungsten Insert</td>
<td>♦ Aluminum</td>
</tr>
<tr>
<td></td>
<td>♦ Rhenium Insert</td>
<td></td>
</tr>
<tr>
<td></td>
<td>♦ Molybdenum Insert</td>
<td></td>
</tr>
<tr>
<td>Throat Material Alternatives</td>
<td>♦ Silica / Phenolic Insert</td>
<td>♦ Cellulose / Phenolic Insert</td>
</tr>
<tr>
<td></td>
<td>♦ Graphite / Phenolic Insert</td>
<td>♦ Silica / Phenolic Insert</td>
</tr>
<tr>
<td></td>
<td>♦ Silicone Elastomer Insert</td>
<td>♦ Graphite Insert</td>
</tr>
<tr>
<td>Exit Cone, Dome Closeout, and Blast Tube Material Alternatives</td>
<td>♦ No Insert</td>
<td>♦ Carbon – Carbon Insert</td>
</tr>
<tr>
<td></td>
<td>♦ Glass / Phenolic Insert</td>
<td></td>
</tr>
</tbody>
</table>
Summary of Propulsion

- **Emphasis**
  - Turbojet propulsion
  - Ramjet propulsion
  - Rocket propulsion

- **Conceptual Design Prediction Methods**
  - Thrust
  - Specific impulse

- **Design Trades**
  - Turbojet turbine material, compressor ratio, and cycle
  - Ramjet engine / booster / inlet integration
  - Ramjet fuel
  - Propellant burn area requirement
  - Nozzle throat area
  - Nozzle expansion ratio
  - Rocket motor grain
  - Thrust magnitude control
Summary of Propulsion (cont)

- Design Trades (cont)
  - Solid propellant alternatives
  - Motor case material alternatives
  - Nozzle materials

- New Propulsion Technologies
  - Hypersonic turbojet
  - Ramjet / ducted rocket
  - Scramjet
  - Combined cycle propulsion
  - High temperature turbine materials
  - High temperature combustor
  - Oblique shock airframe compression
  - Mixed compression inlet
  - Low drag inlet
  - High density fuel / propellant
  - Endothermic fuel
Summary of Propulsion (cont)

♦ New Propulsion Technologies (cont)
  ♦ Solid rocket thrust magnitude control
  ♦ High burn exponent propellant
  ♦ Low observable fuel / propellant

♦ Discussion / Questions?

♦ Classroom Exercise (Appendix A)
1. An advantage of turbojets compared to ramjets is s_____ thrust.
2. The specific impulse of a turbojet is often limited by the maximum allowable temperature of the t______.
3. The specific impulse of a ramjet is often limited by the maximum allowable temperature of the c________.
4. Ducted rockets are based on a fuel-rich g__ g________.
5. A safety advantage of solid rocket propulsion over liquid propulsion is less t______.
6. A rocket boost to a take-over Mach number is required by ramjets and s______.
7. Parameters that enable the long range of subsonic cruise turbojet missiles are high lift, low drag, available fuel volume, and high s______ i______.
8. High thrust and high acceleration are achievable with s____ r_____ propulsion.
9. In a turbojet the power to drive the compressor is provided by the t______.
10. The compressor exit temperature is a function of the flight Mach number and the compressor pressure ratio.

11. Compressor exit temperature, fuel heating value, and fuel-to-air ratio determine the turbojet temperature.

12. Three types of turbine based propulsion are turbojet, turbo ramjet, and augmentor.

13. Mach number and fuel-to-air ratio determine the ramjet temperature.

14. An example of a ramjet with low drag and light weight is an inlet ramjet.

15. Russia, France, China, United Kingdom, Taiwan, and India are the only countries with currently operational ramjet missiles.

16. 100% inlet capture efficiency occurs when the forebody shock waves intercept the inlet.

17. Excess air that does not flow into the inlet is called supersonic air.
18. Starting a ramjet inlet at lower supersonic Mach number requires a larger area of the inlet t_____.

19. Optimum pressure recovery across shock waves is achieved when the total pressure loss across each shock wave is e____.

20. The specific impulse and thrust of a ramjet are a function of the efficiency of the combustor, nozzle, and i____.

21. High density fuels have high payoff for v_____ limited missiles.

22. The specific impulse of a ducted rocket with large excess fuel from the gas generator can approach that of a r_____.

23. High speed rockets require large p________ weight.

24. At the throat, the flow area is minimum, sonic, and c_____.

25. For an optimum nozzle expansion the nozzle exit pressure is equal to the a________ pressure.

26. High thrust and chamber pressure are achievable through a large propellant b___ area.
Propulsion Problems (cont)

27. Three approaches to solid rocket thrust magnitude control are pulse motor, pintle motor, and g__ motor.

28. A high burn exponent propellant allows a large change in thrust with only a small change in chamber p_______.

29. Three tradeoffs in selecting a solid propellant are safety, observables, and s_______ i_______.

30. A low cost motor case is usually based on steel or aluminum material while a light weight motor case is usually based on c_______ material.

31. Rockets with high chamber pressure or long burn time may require a t_______ throat insert.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- **Weight Considerations in Tactical Missile Design**
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

- Define Mission Requirements
- Establish Baseline
- Aerodynamics
- Propulsion
- Weight
- Trajectory
- Meet Performance?
- Measures of Merit and Constraints

- Yes
- Alt Mission
- Alt Baseline
- Resize / Alt Config / Subsystems / Tech
- No
Designing Light Weight Missile Has High Payoff

- Production cost
- Logistics cost
- Size
- Firepower
- Observables
- Mission flexibility
- Expeditionary warfare
Flight Performance (Range, Speed, Maneuverability) Sensitive to Subsystem Weight

- Dome - Seeker - Guidance and Control - Propulsion - Wings - Stabilizers
- Structure - Power Supply - Warhead - Insulation - Data Link - Flight Control

- High Sensitivity
- Low Sensitivity
- Minor Sensitivity
Missile Range is a Function of Launch Weight, Propellant Weight, and Specific Impulse

\[
\Delta V \approx -g_c I_{sp} \ln \left( 1 - \frac{W_{\text{Propellant}}}{W_i} \right)
\]

\[
R \approx \frac{V^2 \sin (2\theta_i)}{g_c}
\]

Assumptions:

- \(\theta_i\) = Launch Incidence Angle = 45 deg for max range
- Thrust Greater Than Drag and Weight
- Flat, Non-rotating Earth

For Two-Stage Missile with \(W_i\)\textsubscript{Min}:

\[\Delta V_1 = \Delta V_2\]

Example: Two-Stage Missile with Minimum Weight and \(R_{\text{max}} = 200\) nm = \(1.216 \times 10^6\) ft

Assume \(I_{sp} = 250\) sec, \(W_{\text{Payload}} = 500\) lb, \(W_{\text{inert}} = 0.2 W_{\text{Propellant}}\)

\[V = \left[\frac{32.2 \times 1.216 \times 10^6}{2}\right]^{1/2} = 6251\] ft / s

\[\Delta V_1 = \Delta V_2 = V / 2 = 3125\] ft / s

\[W_{i,\text{Second Stage}} = W_{\text{Payload}} + W_{\text{inert}} + W_{\text{Propellant}} = 814\] lb

\[W_{i,\text{First Stage}} = W_{\text{inert}} + W_{\text{Propellant}} = 85 + 427 = 512\] lb

\[W_i = W_{i,\text{First Stage}} + W_{i,\text{Second Stage}} = 1326\] lb

Compare: Single-Stage Missile, \(R = 200\) nm

\[\Delta V = 6251 = -32.2 \times (250) \ln \left[ 1 - \frac{W_{\text{Propellant}}}{W_{\text{Payload}} + 0.2 W_{\text{Propellant}} + 500} \right] \Rightarrow W_p = 767 \Rightarrow W_i = 1420\] lb
Missile Weight Is a Function of Diameter and Length

\[ W_L = 0.04 \, l \, d^2 \]

Units: \( W_L \) (lb), \( l \) (in), \( d \) (in)
Most Subsystems for Tactical Missiles Have a Density of about 0.05 lb / in$^3$
Modeling Weight, Balance, and Moment-of-Inertia
Is Based on a Build-up of Subsystems

Example Missile Configuration

Structure and Subsystems
Warhead and Structure
Inlet Structure and Subsystems
Aero Surfaces
Engine Structure and Subsystems
Fuel

Legend

Assume Uniform Weight Distribution For a Given Segment

\[ x_{CG} = \sum \left( x_{subsystem1} \frac{W_{subsystem1}}{W_{total}} + x_{subsystem2} \frac{W_{subsystem2}}{W_{total}} + \ldots \right) \]

\[ I_y = \sum \left[ \left( I_{y,subsystem1} \right)_{local} + W_{subsystem1} \frac{\left( x_{subsystem1} - x_{CG} \right)^2}{g_c} + \left( I_{y,subsystem2} \right)_{local} + W_{subsystem2} \frac{\left( x_{subsystem2} - x_{CG} \right)^2}{g_c} + \ldots \right] \]
Moment-of-Inertia Is Higher for High Fineness Ratio Body

\[
\begin{align*}
(l_y, \text{cylinder})_{\text{local}} &= \left[ \frac{W}{d^2} \cdot \frac{g_c}{g_c} \right] \left[ \frac{1}{16} + \left( \frac{1}{12} \right) \left( \frac{l}{d} \right)^2 \right] \\
(l_y, \text{cone})_{\text{local}} &= \left[ \frac{W}{d^2} \cdot \frac{g_c}{g_c} \right] \left[ \frac{3}{80} + \left( \frac{3}{80} \right) \left( \frac{l}{d} \right)^2 \right]
\end{align*}
\]

Example for Ramjet Baseline at Launch \( (x_{cg} = 8.04 \text{ ft}) \)

Assume missile can be approximated as a conical nose-cylinder

For the cone, \( d = 1.25 \text{ ft}, \frac{l}{d} = 1.57, W_{\text{cone}} = 15.9 \text{ lb}, x_{cg,\text{cone}} = 1.308 \text{ ft} \)

For the cylinder, \( \frac{l}{d} = 7.22, d = 1.698 \text{ ft}, W_{\text{cylinder}} = 2214 \text{ lb}, x_{cg,\text{cylinder}} = 8.09 \text{ ft} \)

\[
l_y = (l_y, \text{cone})_{\text{local}} + W_{\text{cone}} \left( x_{cg,\text{cone}} - x_{CG} \right)^2 / g_c + (l_y, \text{cylinder})_{\text{local}} + W_{\text{cylinder}} \left( x_{cg,\text{cylinder}} - x_{CG} \right)^2 / g_c
\]

\[
(l_y, \text{cone})_{\text{local}} = \left[ 15.9 \left( \frac{1.25}{16} \right)^2 / 32.2 \right] \left[ 0.0375 + 0.0375 \left( \frac{1.57}{1.698} \right)^2 \right] = 0.10 \text{ slug-ft}^2
\]

\[
(l_y, \text{cylinder})_{\text{local}} = \left[ 2214 \left( \frac{1.698}{32.2} \right)^2 / 32.2 \right] \left[ 0.0625 + 0.0833 \left( \frac{7.22}{8.09} \right)^2 \right] = 872 \text{ slug-ft}^2
\]

\[
l_y = 0.10 + 22.4 + 872 + 0.16 = 895 \text{ slug-ft}^2
\]
Structure Design Factor of Safety Is Greater for Hazardous Subsystems / Flight Conditions

- Pressure Bottle (2.50 / 1.50)
- Ground Handling Loads (1.50 / 1.15)
- Captive Carriage and Separation Flight Loads (1.50 / 1.15)
- Motor Case (MEOP) (1.50 / 1.10)
- Free Flight Loads (1.25 / 1.10)
- Castings (1.25 / 1.25)
- Fittings (1.15 / 1.15)
- Thermal Loads (1.00 / 1.00)

Note:
- MIL STDs include environmental (HDBK-310, NATO STANAG 4370, 810F, 1670A), strength and rigidity (8856), and captive carriage (8591).
- The entire environment (e.g., manufacturing, transportation, storage, ground handling, captive carriage, launch separation, post-launch maneuvering, terminal maneuvering) must be examined for driving conditions in structure design.
- FOS $\Delta$ for castings is expected to be reduced in future as casting technology matures.
- Reduction in required factor of safety is expected as analysis accuracy improves will result in reduced missile weight / cost.
### Structure Concepts and Manufacturing Processes for Low Parts Count

#### Geometry Alternatives

<table>
<thead>
<tr>
<th>Geometry Alternatives</th>
<th>Structure Concept Alternatives</th>
<th>Composites</th>
<th>Metals</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lifting Body Airframe</td>
<td>Monocoque</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td></td>
<td>Integrally Hoop Stiffened</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td></td>
<td>Integrally Longitudinal Stiffened</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td>Axisymmetric Airframe</td>
<td>Monocoque</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td></td>
<td>Integrally Hoop Stiffened</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td></td>
<td>Integrally Longitudinal Stiffened</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td>Surface</td>
<td>Solid</td>
<td>●</td>
<td>●</td>
</tr>
<tr>
<td></td>
<td>Sandwich</td>
<td>●</td>
<td>●</td>
</tr>
</tbody>
</table>

#### Structure Manufacturing Process Alternatives

- Vacuum Assist RTM
- Compression Mold
- Filament Wind
- Pultrusion
- Thermal Form
- Vacuum Bag / Autoclave
- Cast
- High Speed Machine
- Forming
- Strip Laminate

Note: Manufacturing process cost is a function of recurring cost (unit material, unit labor) and non-recurring cost (tooling).

Note: Blue circles indicate very low parts count, lighter blue circles indicate low parts count, white circles indicate moderate parts count, and grey circles indicate high parts count.
Low Parts Count Manufacturing Processes for Complex Airframes

- Vacuum Assisted RTM

- Filament Wind

- Pultrusion

- Metal Casting
### Tactical Missile Airframe Material Alternatives

<table>
<thead>
<tr>
<th>Type</th>
<th>Material</th>
<th>Tension ((\sigma_{TU}/\rho))</th>
<th>Buckling Stability ((\sigma_{Buckling}/\rho))</th>
<th>Max Short – Life Temp</th>
<th>Thermal Stress</th>
<th>Joining</th>
<th>Cost</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Metallic</td>
<td>Aluminum 2219</td>
<td>○</td>
<td>○</td>
<td>−</td>
<td>−</td>
<td>○</td>
<td>○</td>
<td>○</td>
</tr>
<tr>
<td>Increasing Cost</td>
<td>Steel PH 15-7Mo</td>
<td>○</td>
<td>−</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>−</td>
</tr>
<tr>
<td></td>
<td>Titanium 6Al-4V</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>−</td>
</tr>
<tr>
<td>Composite</td>
<td>S994 Glass / Epoxy and S994 Glass / Polyimide</td>
<td>○</td>
<td>○</td>
<td>○</td>
<td>−</td>
<td>○</td>
<td>○</td>
<td>○</td>
</tr>
<tr>
<td>Increasing Cost</td>
<td>Glass or Graphite Reinforce Molding</td>
<td>−</td>
<td>○</td>
<td>−</td>
<td>−</td>
<td>−</td>
<td>−</td>
<td>−</td>
</tr>
<tr>
<td></td>
<td>Graphite / Epoxy and Graphite Polyimide</td>
<td>○</td>
<td>−</td>
<td>−</td>
<td>−</td>
<td>−</td>
<td>−</td>
<td>−</td>
</tr>
</tbody>
</table>

**Note:**
- **Superior**
- **Above Average**
- **Average**
- **Below Average**

2/24/2008

ELF
Strength – Elasticity of Airframe Material Alternatives

\[ \sigma_t = \frac{P}{A} = E \varepsilon \]

Note:
- High strength fibers are:
  - Very small diameter
  - Unidirectional
  - High modulus of elasticity
  - Very elastic
  - No yield before failure
  - Non forgiving failure
- Metals:
  - Ductile,
  - Yield before failure
  - Allow adjacent structure to absorb load
  - Resist crack formation
  - Resist impact loads
  - More forgiving failure

E, Young’s modulus of elasticity, psi
P, Load, lb
\( \varepsilon \), Strain, in / in
A, Area, in\(^2\)
Room temperature

Graphite Fiber w/o Matrix
(400 – 800 Kpsi)

Kevlar Fiber w/o Matrix

Titanium Alloy (Ti-6Al-4V)

Very High Strength Stainless Steel
(Ph 15-7 Mo, TH 900)

High Strength Stainless Steel
(Ph 15-7 Mo, TH 1050)

Aluminum Alloy (2219-T81)

Glass Fiber w/o Matrix

Strength – Elasticity of Airframe Material Alternatives
Structural Efficiency at High Temperature of Short Duration Airframe Material Alternatives

Graphite / Epoxy (\(\rho = 0.065\) lb/in\(^3\)), 0±45-90 Laminate

Graphite / Polyimide (\(\rho = 0.057\) lb/in\(^3\)), 0±45-90 Laminate

Ti\(_3\)Al (\(\rho = 0.15\) lb/in\(^3\))

Ti-6Al-4V Annealed Titanium (\(\rho = 0.160\) lb/in\(^3\))

PH15-7 Mo Stainless Steel (\(\rho = 0.277\) lb/in\(^3\)). Note: Thin wall steel susceptible to buckling.

Graphite / Polyimide (\(\rho = 0.057\) lb/in\(^3\)), 0±45-90 Laminate

Graphite / Epoxy (\(\rho = 0.065\) lb/in\(^3\)), 0±45-90 Laminate
Hypersonic Missiles without External Insulation Require High Temperature Structure

\[ T_r = T_0 \left( 1 + 0.2 \; r \; M^2 \right) \]

Note:
- \( \gamma = 1.4 \)
- \( T_r = \) Recovery Temperature, R
- \( T_0 = \) Free stream temperature, R
- \( T_{\text{max}} = \) Max temperature capability
- No external insulation assumed
- \( r \) is recovery factor
- \( h = 40k \) ft ( \( T_{\text{Free Stream}} = 390 \) R )
- Stagnation \( r = 1 \)
- Turbulent boundary layer \( r = 0.9 \)
- Laminar boundary layer \( r = 0.8 \)
- Short-duration flight (less than 30 m), but with thermal soak

Graphite Polyimide

Titanium Aluminide \( \approx 2,500 \) °F

Single Crystal Nickel Aluminides \( \approx 3,000 \) °F

Ceramic Matrix Composite \( \approx 3,500 \) °F

Nickel Alloys (e.g., Inconel, Rene, Hastelloy, Haynes)

Graphite Epoxy

Titanium Aluminide

Ceramic Matrix Composite

Nickel Alloys

Titanium Aluminide

Single Crystal Nickel Aluminides

Ceramic Matrix Composite

Nickel Alloys

Titanium Aluminide

Single Crystal Nickel Aluminides

Ceramic Matrix Composite

Nickel Alloys

Titanium Aluminide

Single Crystal Nickel Aluminides

Ceramic Matrix Composite

Nickel Alloys

Titanium Aluminide

Single Crystal Nickel Aluminides

Ceramic Matrix Composite

Nickel Alloys
## Structure / Insulation Trades for Short Duration Flight

### Example Structure / Insulation Concepts

<table>
<thead>
<tr>
<th>Mach</th>
<th>Tmax</th>
<th>k</th>
<th>c</th>
<th>ρ</th>
<th>α</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Hot Metal Structure (e.g., Al) without Insulation</td>
<td>600</td>
<td>0.027</td>
<td>0.22</td>
<td>0.101</td>
<td>0.000722</td>
</tr>
<tr>
<td>Hot Metal Structure (e.g., Al)</td>
<td>600</td>
<td>0.027</td>
<td>0.22</td>
<td>0.101</td>
<td>0.000722</td>
</tr>
<tr>
<td>Internal Insulation (e.g., Min-K)</td>
<td>2000</td>
<td>0.0000051</td>
<td>0.24</td>
<td>0.012</td>
<td>0.00000106</td>
</tr>
<tr>
<td>Self-insulating Composite Structure (e.g., Graphite Polyimide)</td>
<td>1100</td>
<td>0.000109</td>
<td>0.27</td>
<td>0.057</td>
<td>0.00000410</td>
</tr>
<tr>
<td>Ext Insulation (e.g., Micro-Quartz Paint)</td>
<td>1200</td>
<td>0.0000131</td>
<td>0.28</td>
<td>0.012</td>
<td>0.00000226</td>
</tr>
<tr>
<td>Cold Metal Structure (e.g., Al)</td>
<td>600</td>
<td>0.027</td>
<td>0.22</td>
<td>0.101</td>
<td>0.000722</td>
</tr>
<tr>
<td>Internal Insulation (e.g., Min-K)</td>
<td>2000</td>
<td>0.0000051</td>
<td>0.24</td>
<td>0.012</td>
<td>0.00000106</td>
</tr>
</tbody>
</table>

**Note:**
- Tactical missiles use passive thermal protection (no active cooling).
- Small thickness allows more propellant/fuel for diameter constrained missiles (e.g., VLS launcher).
- Weight and cost are application specific.
- \( T_{\text{max}} \) = max temp capability, °F; \( k \) = thermal conductivity, BTU/s/ft²/°F/ft; \( c \) = specific heat or thermal capacity, BTU/lbm/°F; \( \rho \) = density, lbm/in³; \( \alpha \) = thermal diffusivity = \( k / (\rho \cdot c) \), ft²/s.
External Insulation Has High Payoff for Short Duration Flight

Example Airframe Temperature with No External Insulator – Steel Airframe Selected.

Example Airframe Temperature with 0.012 in Insulator – Aluminum Airframe Acceptable for Short Duration.

Note: Short Range Air-to-Air Missile
Launch ~ 0.9 Mach at 10k ft Altitude
Atmosphere ~ Hot Day (1% Risk) Mil-HDBK-310
Phenolic Composites Are Good Insulators for High Temperature Structure and Propulsion

<table>
<thead>
<tr>
<th>Material Type</th>
<th>Characteristics</th>
<th>Density (lb/in³)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bulk Ceramics</td>
<td>Melt, ρ ~ 0.20 lb/in³</td>
<td>0.20</td>
</tr>
<tr>
<td></td>
<td>Zirconium Ceramic, Hafnium Ceramic</td>
<td></td>
</tr>
<tr>
<td>Graphites</td>
<td>Burn, ρ ~ 0.08 lb/in³</td>
<td>0.08</td>
</tr>
<tr>
<td></td>
<td>Carbon / Carbon</td>
<td></td>
</tr>
<tr>
<td>Porous Ceramics</td>
<td>Melt, ρ ~ 0.12 lb/in³</td>
<td>0.12</td>
</tr>
<tr>
<td></td>
<td>Resin Impregnated</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Carbon-Silicon Carbide</td>
<td></td>
</tr>
<tr>
<td>Medium Density Phenolic Composites</td>
<td>Char, ρ ~ 0.06 lb/in³</td>
<td>0.06</td>
</tr>
<tr>
<td></td>
<td>Nylon Phenolic, Silica</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Phenolic, Glass</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Phenolic, Carbon</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Phenolic, Carbon</td>
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<tr>
<td></td>
<td>Phenolic, Graphite</td>
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<td></td>
<td>Phenolic, Phenolic</td>
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<tr>
<td>Low Density Composites</td>
<td>Char, ρ ~ 0.03 lb/in³</td>
<td>0.03</td>
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<tr>
<td></td>
<td>Micro-Quartz Paint, Glass-Cork-Epoxy, Silicone Rubber</td>
<td></td>
</tr>
<tr>
<td>Plastics</td>
<td>Sublime, Depolymerizing</td>
<td>0.06</td>
</tr>
<tr>
<td></td>
<td>Teflon</td>
<td></td>
</tr>
</tbody>
</table>

**Note:** Assumed Weight Per Unit Area of Insulator / Ablator = 1 lb/ft²

**Graph:**
- **T_{max}, Max Temperature Capability, R**
- **Insulation Efficiency, Minutes To Reach 300° F at Back Wall**

**Phenolic Composites Are Good Insulators for High Temperature Structure and Propulsion.**
A “Thermally Thin” Surface (e.g., Metal Airframe) Has Uniform Internal Temperature

Thermally thin surface (e.g., metal airframe) has uniform internal temperature.

\[
\frac{dT}{dt}\bigg|_{t=0} = \left( T_r - T_{\text{initial}} \right) \frac{h}{(c \rho z)}
\]

\[
T = T_r - \left( T_r - T_{\text{initial}} \right) e^{-\frac{ht}{(c \rho z)}}
\]

\[
h = k \frac{N_{\text{NU}}}{x}
\]

Thermally thin \( \Rightarrow h \left( \frac{z}{k} \right)_{\text{surface}} < 0.1 \)

Example for Rocket Baseline Airframe:

- Aluminum skin w/o external insulation
- \( c = 0.215 \) BTU/lb/R, \( \rho = 0.10 \) lb/in\(^3\), \( k = 0.027 \) BTU/s/ft\(^2\)/R/ft
- \( z = 0.16 \) in = 0.0133 ft
- \( h = k \frac{N_{\text{NU}}}{x} \)

Assume Mach 2 sustain flight, 20k ft altitude (\( T_0 = 447 \) R, \( k = 3.31 \times 10^{-6} \) BTU/s/ft\(^2\)/R/ft), Turbulent boundary layer, \( x = 1.6 \) ft

\[
\text{Re}_x = \rho_0 M a_0 x / \mu_0 = 12.56 \times 10^6
\]

\[
N_{\text{NU}} = 0.0271 \text{ Re}^{0.8} = 12947
\]

\[
h = k \frac{N_{\text{NU}}}{x} = 0.0268 \text{ BTU/s/ft}^2/R
\]

Test: \( h \left( \frac{z}{k} \right)_{\text{surface}} = 0.0132 < 0.1 \Rightarrow \text{thermally thin} \)

Calculate \( T_r = T_0 [1 + 0.2 r M^2] = 447 [1 + 0.2 (0.9) (2)^2] = 769 \) R

At \( t = 0 \), Assume \( T_{\text{initial}} = 460 \) R, or 0° F

\[
\left( \frac{dT}{dt} \right)_{t=0} = \left( 769 - 460 \right) \frac{0.0268}{\left( 0.215 \right) \left( 172.8 \right) \left( 0.01333 \right)} = 17 \degree F/s
\]

At a sustain time \( t = 10 \) s, \( T = 769 - \left( 769 - 460 \right) e^{-0.0268 \left( 10 \right) / \left( 0.215 \right) \left( 172.8 \right) \left( 0.01333 \right)} = 589 \) R, or 129° F

Note: No external insulation; thermally thin structure (uniform internal temperature); “Perfect” insulation behind airframe; 1-D heat transfer; Turbulent boundary layer; Radiation neglected; \( \frac{dT}{dt} = \text{Temperature rate, R} / \text{s}; T = \text{Recovery (max) temperature, R}; h = \text{Convection heat transfer coefficient, BTU/s/ft}^2/R; c = \text{Specific heat, BTU/lb/R}; \rho = \text{Density, lb/ft}^3; z = \text{Thickness, ft}; k = \text{Conductivity, BTU/s/ft}^2/R/ft; \text{Re} = \text{Reynolds number}; N_{\text{NU}} = \text{Nusselt number}

A “Thermally Thick” Surface (e.g., Radome) Has a Large Internal Temperature Gradient

\[
\frac{[T(z,t) - T_{\text{initial}}]}{[T_r - T_{\text{initial}}]} = \text{erfc} \left\{ \frac{z}{\left[ 2 (\alpha t)^{1/2} \right]} \right\} - e^{(\frac{h z}{k})} + h^2 \frac{\alpha t}{k} \text{erfc} \left\{ \frac{z}{\left[ 2 (\alpha t)^{1/2} \right]} + h (\frac{\alpha t}{k})^{1/2} \right\}
\]

\[
\frac{[T(0,t) - T_{\text{initial}}]}{[T_r - T_{\text{initial}}]} = 1 - e^{h^2 \frac{\alpha t}{k}} \text{erfc} \left[ h (\frac{\alpha t}{k})^{1/2} \right]
\]

 applicable for thermally thick surface: \( z / \left[ 2 (\alpha t)^{1/2} \right] > 1 \)

Example: Rocket Baseline Radome
\( z = 0.25 \text{ in} = 0.0208 \text{ ft}, k = 5.96 \times 10^{-4} \text{ BTU} / \text{ft}^2 / \text{s} / \text{R}, \)
\( \alpha = 1.499 \times 10^{-5} \text{ ft}^2 / \text{s} \)
Mach 2, 20k ft alt \( (T_0 = 447 \text{ R}) \), Turbulent boundary layer,
\( x = 19.2 \text{ in} = 1.6 \text{ ft}, t = 10 \text{ s}, T_r = 769 \text{ R}, T_{\text{initial}} = 460 \text{ R} \)
\( \Rightarrow h = 0.0268 \text{ BTU} / \text{s} / \text{ft} \Rightarrow (\frac{h}{k})(\frac{\alpha t}{k})^{1/2} = 0.491 \)
Test: \( z / \left[ 2 (\alpha t)^{1/2} \right] = 0.0208 / \left\{ 2 \left[ 1.499 \times 10^{-5} (10) \right]^{1/2} \right\} = 0.849 < 1 \Rightarrow \text{not quite thermally thick} \)
Inner wall \( \Rightarrow h z / k = 0.935 \)
\( \frac{[T(0.0208,10) - T_{\text{initial}}]}{[T_r - T_{\text{initial}}]} = 0.0608 \)
\( T(0.0208,10) = 479 \text{ R} \) (Note: \( T_{\text{inner}} \approx T_{\text{initial}} \))
Surface \( \Rightarrow h z / k = 0 \)
\( \frac{[T(0,10) - T_{\text{initial}}]}{[T_r - T_{\text{initial}}]} = 0.372 \)
\( T(0,10) = 575 \text{ R} \)


2/24/2008
Internal Insulation Temperature Can Be Predicted Assuming Constant Flux Conduction

\[
\frac{T(z, t) - T_{\text{initial}}}{T(0, t) - T_{\text{initial}}} = e^{-z^2/(4 \alpha t)} - (\pi / \alpha t)^{1/2} (z / 2) \text{erfc}\left\{\frac{z}{2 (\alpha t)^{1/2}}\right\}
\]

Applicable for thermally thick surface: \(z / [2 (\alpha t)^{1/2}] > 1\)

Example for Rocket Baseline Airframe Insulation:

0.10 in Min-K Internal Insulation behind 0.16 in aluminum Skin

Assume \(M = 2\), 20k ft alt, \(x = 1.6\) ft, \(T_{\text{initial}} = 460\) R, \(t = 10\) s, \(z_{\text{Min-K}} = 0.10\) in = 0.00833 ft, \(\alpha_{\text{Min-K}} = 0.00000106\) ft\(^2 / s\), \(k = 5.96 \times 10^{-4}\) BTU / s / ft, \(h = 0.0268\) BTU / s / ft

Test: \(z / [2 (\alpha t)^{1/2}] = 0.00833 / [2 \times 0.00000106 (10)^{1/2}] = 1.279 > 1 \Rightarrow\) thermally thick
\((\alpha t)^{1/2} / z = [0.00000106 (10)]^{1/2} / 0.00833 = 0.3907\)

\[
\frac{T_{\text{Min-K}}(\ 0.0217, 10\ ) - 460\ }{[T_{\text{Min-K}}(\ 0, 10\ ) - 460\ ]} = 0.0359
\]

Assume \((T_{\text{inner}})_{\text{aluminum}} = (T_{\text{outer}})_{\text{Min-K}}\)

From prior example, \((T_{\text{inner}})_{\text{aluminum}} = 569\) R at \(t = 10\) s

Then, \((T_{\text{outer}})_{\text{Min-K}} = 569\) R at \(t = 10\) s

Compute, \((T_{\text{inner}})_{\text{Min-K}} = 460 + (569 - 460 \times 0.0338 = 460 + 4 = 464\) R

Note: 1-D conduction heat transfer, Radiation neglected, Constant heat flux input, \(T(z, t) =\) Inner temperature of insulation at time \(t\), \(T_{\text{initial}} =\) Initial temperature, \(T(0, t) =\) Outer temperature of insulation at time \(t\), \(\alpha =\) Diffusivity of insulation material, \(ft^2 / s\); \(z_{\text{max}} =\) Thickness of insulation material, \(ft\); erfc = Complementary error function

A Sharp Nose Tip / Leading Edge Has High Aerodynamic Heating

\[ h_r = \frac{N_{NUr} \ k_r}{d_{NoseTip}} \]
\[ N_{NUr} = 1.321 \ Re_{dNoseTip}^{0.5} \ Pr^{0.4} \]

Example for Rocket Baseline Nose Tip:
Assume \( M = 2 \), 20k ft alt, stagnation (\( T_r = 805 \) R) for a sharp nose tip (e.g., 1% blunt)

\[ d_{NoseTip} / d_{Ref} = 0.01 \Rightarrow d_{NoseTip} = 0.01 \ (8 \text{ in}) = 0.08 \text{ in} = 0.00557 \text{ ft} \]

\[ Re_{dNoseTip} = \rho_0 \ V_0 \ d_{NoseTip} / \mu_r = 3.39 \times 10^4 \]

\[ N_{NUr} = 223 \]

\[ h_r = 0.1745 \text{ BTU} / \text{ ft}^2 / \text{s} / \text{ R} \]

Outer surface temperature after 10 s heating in sustain flight (\( M = \text{const}, \ T_r = \text{const} \)):

\[ \left[ T (0, t) - T_{initial} \right] / \left[ T_r - T_{initial} \right] = 1 - e^{h^2 \alpha t / k^2} \]

\[ \text{erfc} \left\{ h (\alpha t)^{1/2} / k \right\} \]

\[ \left[ T (0, 10) - 460 \right] / \left[ 805 - 460 \right] = 1 - e^{ \left[ (0.1745)^2 \left( 1.499 \times 10^{-5} \right) / \left( 5.96 \times 10^{-4} \right)^2 \right] \text{erfc} \left\{ (0.1745) \left[ 1.499 \times 10^{-5} \ (10)^{1/2} / (5.96 \times 10^{-4}) \right] \right\}} = 0.845 \]

\[ T (0, 10) = 460 + 345 (0.845) = 752 \text{ R} \]

Tactical Missile Radiation Heat Loss Is Usually Small Compared to Convective Heat Input

Radiation Heat Flux at Equilibrium Temperature, BTU / ft² / s

\[ Q_{Rad} = 4.76 \times 10^{-13} \varepsilon T^4 \]

\( Q_{Rad} \) in BTU / ft² / s, \( T \) in R

Example: Ramjet Baseline
Assume:
- Titanium skin with emissivity \( \varepsilon = 0.3 \)
- Long duration (equilibrium) heating at Mach 4
- \( h = 80 \text{ ft}, T_0 = 398 \text{ R} \)
- Turbulent boundary layer (\( r = 0.9 \)) \( \Rightarrow T = T_r = T_0 (1 + 0.2 r M^2) = 1513 \text{ R} \)

Calculate:
\[ Q_{Rad} = 4.76 \times 10^{-13} (0.3) (1513)^4 \]
\[ = 0.748 \text{ BTU / ft}^2 / \text{s} \]
Design Concerns for Localized Aerodynamic Heating and Thermal Stress

Body Joints
- Hot missile shell
- Cold frames or bulkheads
- Causes premature buckling

IR Domes / RF Radomes
- Large temp gradients due to low thermal conduction
- Thermal stress at attachment
- Low tensile strength
- Dome fails in tension

Flare / Wedge Corner Flow
- Shock wave – boundary layer interaction
- Separated Flow
- High heating at reattachment

Leading Edges
- Hot stagnation temperature on leading edge
- Small radius prevents use of external insulation
- Cold heat sink material as chord increases in thickness leads to leading edge warp
- Shock wave interaction with adjacent body structure

Note: $\sigma_{TS} = \text{Thermal stress from restraint in compression or tension} = \alpha E \Delta T$
$\alpha = \text{coefficient of thermal expansion}, E = \text{modulus of elasticity}, \Delta T = T_2 - T_1 = \text{temperature difference}$.

Example: Thermal Stress for Rocket Baseline Pyroceram Dome, $\alpha = 3 \times 10^{-6}, E = 13.3 \times 10^6 \text{ psi}$
Assume $M = 2, h = 20k \text{ ft alt}, t = 10 \text{ s}$. Based on prior figure, $\Delta T = T_{\text{OuterWall}} - T_{\text{InnerWall}} = 102 R$
Then $\sigma_{TS} = 3 \times 10^{-6} (13.3 \times 10^6) (102) = 4,070 \text{ psi}$
Examples of Aerodynamic Hot Spots

- Nose Tip
- Leading Edge
- Flare
Tactical Missile Body Structure Weight Is about 22% of the Launch Weight

\[ \frac{W_{BS}}{W_L} \approx 0.22 \]

Example for 500 lb missile

- \( W_L = 500 \text{ lb} \)
- \( W_{BS} = 0.22 \times 500 = 110 \text{ lb} \)

Note: \( W_{BS} \) includes all load carrying body structure. If motor case, engine, or warhead case carry external loads then they are included in \( W_{BS} \). \( W_{BS} \) does not include tail, wing, or other surface weight.
Body Structure Thickness Is Based on Considering Many Design Conditions

- Structure Design Conditions That May Drive Airframe Thickness
  - Manufacturing
  - Transportation
  - Carriage
  - Launch
  - Fly-out
  - Maneuvering

- Contributors to Required Thickness for Cylindrical Body Structure
  - Minimum Gage for Manufacturing: \( t = 0.7 \frac{d}{\left( \frac{p_{\text{ext}}}{E} \right)^{0.4}} \) \( \approx 0.06 \text{ in} \) if \( p_{\text{ext}} \approx 10 \text{ psi} \)
  - Localized Buckling in Bending: \( t = 2.9 \frac{r}{\sigma} \frac{1}{E} \)
  - Localized Buckling in Axial Compression: \( t = 4.0 \frac{r}{\sigma} \frac{1}{E} \)
  - Thrust Force: \( t = T / (2 \pi r) \)
  - Bending Moment: \( t = M / (\pi \sigma r^2) \)
  - Internal Pressure: \( t = p r / \sigma \)

- High Risk (1), Moderate Risk (2), and Low Risk (3) Estimates of Required Thickness
  1. \( t = \text{FOS} \times \text{Max} \left( t_{\text{MinGage}}, t_{\text{Buckling,Bending}}, t_{\text{Buckling,AxialCompression}}, t_{\text{AxialLoad}}, t_{\text{Bending}}, t_{\text{InternalPressure}} \right) \)
  2. \( t = \text{FOS} \times \left( t_{\text{MinGage}}^2 + t_{\text{Buckling,Bending}}^2 + t_{\text{Buckling,AxialCompression}}^2 + t_{\text{AxialLoad}}^2 + t_{\text{Bending}}^2 + t_{\text{InternalPressure}}^2 \right)^{1/2} \)
  3. \( t = \text{FOS} \times \left( t_{\text{MinGage}} + t_{\text{Buckling,Bending}} + t_{\text{Buckling,AxialCompression}} + t_{\text{AxialLoad}} + t_{\text{Bending}} + t_{\text{InternalPressure}} \right) \)
Localized Buckling May Be A Concern for Thin Wall Structure

\[ \sigma_{\text{Buckling, Bending}} / E \approx 0.35 \left( \frac{t}{r} \right) \]
\[ \sigma_{\text{Buckling, Axial Compression}} / E \approx 0.25 \left( \frac{t}{r} \right) \]

Note: Thin wall cylinder with local buckling

\[ \sigma_{\text{Buckling}} / E = \text{Nondimensional buckling stress} \]
\[ \sigma_{\text{Buckling, Bending}} = \text{Buckling stress in bending} \]
\[ \sigma_{\text{Buckling, Axial Compression}} = \text{Buckling stress in axial compression} \]

\[ E = \text{Young’s modulus of elasticity} \]
\[ t = \text{Airframe thickness} \]
\[ r = \text{Airframe radius} \]

Min thickness for fab and handling \( \approx 0.06 \text{ in} \)

Example for Rocket Baseline in Bending:
4130 steel motor case, \( E = 29.5 \times 10^6 \text{ psi} \)
\[ \sigma_{\text{yield}} = 170,000 \text{ psi} \]
\[ t = 0.074 \text{ in}, \ r = 4 \text{ in} \]
\[ t / r = 0.0185 \]
\[ \sigma_{\text{Buckling, Bending}} / E \approx 0.35 \left( \frac{0.0185}{0.0185} \right) = 0.006475 \]
\[ \sigma_{\text{Buckling, Bending}} \approx 191,000 \text{ psi} \]
\[ \sigma_{\text{Buckling, Bending}} \approx \sigma_{\text{yield}} \]

Note: Actual buckling stress can vary +/- 50%, depending upon typical imperfections in geometry and the loading.
Process for Captive and Free Flight Loads Calculation

Free Flight

Maneuver Per Design Requirements

Weight load of bulkhead section

Air Load Obtained By Wind Tunnel

Air Load

Max Aircraft Maneuver Per MIL-A-8591

Carriage Load

Weight load of bulkhead section

Air Load

Note: MIL-A-8591 Procedure A assumes max air loads combine with max g forces regardless of angle of attack.

Example of $\alpha_{\text{max}}$ Calculated by MIL-A-8591 Using Procedure A for F-18 Aircraft Carriage:

$$\alpha_{\text{max}} = 1.5 \frac{n_z,\text{max}}{W_{\text{max}} / (C_{l_{\text{z}}}, q,S_{\text{Ref}}/_{\text{aircraft}})}$$

$$\alpha_{\text{max}} = 1.5 \left( 5 \right) \left( 49200 \right) / \left[ 0.05 \left( 1481 \right) \left( 400 \right) \right] = 12.5 \text{ deg}$$
Maximum Bending Moment Depends Upon Load Distribution

Example for Rocket Baseline:
1. $c = 4$, ejection load
2. $l = 144$ in

$\Rightarrow$

3. $C = 8$ for uniform loading
4. $N = 10,000$ lb (20 g)

$\Rightarrow$

$M_B = 360,000$ in-lb

$M_B = N l / c$

$C = 8$ for uniform loading

$w = \text{load per unit length}$

$C = 7.8$ for linear loading

$C = 6$ for linear loading to center

$C = 4$ for load at center (e.g., ejection load)

$N = \text{Normal Force}$

$C = 1$ for load at end (e.g., control force)

$N = \text{Normal Force}$
Bending Moment May Drive Body Structure Weight

Example for Rocket Baseline:
• Body has circular cross section
• 2219-T81 aluminum skin ( \( \sigma_{\text{ult}} = 65,000 \) psi )
• \( r = 4 \) in
• Ejection load = 10,000 lb ( 20 g )
• \( M_B = 360,000 \) in \( \cdot \) lb
• FOS = 1.5
• \( t = 360,000 \cdot (1.5) / [\pi (4)^2 (65,000)] = 0.16 \) in

\[
t = \frac{M_B \cdot \text{FOS}}{\pi r^2 \sigma_{\text{Max}}}
\]

Note / Assumptions:
Thin cylinder
Circular cross section
Solid skin
Longitudinal strength
Axial load stress and thermal stress assumed small compared to bending moment stress
\[
\sigma = \frac{M_B}{I_z} = \rac{M_B \cdot r}{\pi r^3 t} = \rac{M_B}{\pi r^2 t}
\]
Tactical Missile Propellant Weight Is about 72% of Rocket Motor Weight

\[
\frac{W_P}{W_M} \approx 0.72
\]

Example for Rocket Baseline

\[
\begin{align*}
W_M &= 209 \text{ lb} \\
W_P &= 0.72 \times 209 = 150 \text{ lb}
\end{align*}
\]

Note: \(W_M\) includes propellant, motor case, nozzle, and insulation.

Increased propellant fraction if:
- High volumetric loading
- Composite case
- Low chamber pressure
- Low flight loads
- Short burn time
Motor Case Weight Is Usually Driven By Stress from Internal Pressure

Assume motor case is axisymmetric, with a front ellipsoid dome and an aft cylinder body.

\[
\sigma_t = -\int_0^{\pi/2} p r \sin \theta \, d\theta
\]

\[
\sigma_t = \frac{p r}{2 t}
\]

(\sigma_t)_{Hoop Stress} = \frac{p r}{t}

(\sigma_t)_{Longitudinal Stress} = \left[ 2 + \left(\frac{a}{b}\right)^2 \right] \frac{p (a b)^{1/2}}{6 t}

If \( a = b \) (hemi dome of radius \( r \)), then

(\sigma_t)_{Longitudinal Stress} = \frac{p r}{2 t}

With metals – the material also reacts body bending.

In composite motor designs, extra (longitudinal) fibers must usually be added to accommodate body bending.
A Composite Motor Case Is Usually Lighter Weight

- **Calculate Maximum Effective Operating Pressure (Burst Pressure)**
  
  \[ p_{\text{burst}} = p_{\text{Boost, Room Temp}} \times e^{\pi k \Delta T} \times (\text{Design Margin for Ignition Spikes, Welds, Other Design Uncertainty}) \]

  Assume Rocket Motor Baseline: diameter = 8 in., length = 55 in, ellipsoid dome \( a/b = 2 \), \( p_{\text{Boost, Room Temp}} = 1769 \text{ psi} \), \( \pi k = (\Delta p / \Delta T) / p_c = 0.14\% / ^\circ\text{F} \)

  Assume Hot day \( T = 160^\circ \text{F} \) \( \Rightarrow e^{\pi k \Delta T} = e^{0.0014 (160 - 70)} = 1.134 \). Uncertainty factor is 1.134, \( 1\sigma \)

  Assume a \( 3\sigma \) uncertainty design margin is provided by, \( p_{\text{burst}} \approx 1769 x (1.134)^3 = 2,582 \text{ psi} \)

- **Assume Ultimate Factor of Safety FOS = 1.5**

- **Rocket Baseline Steel Case \( (\sigma_t)_{\text{ult}} = 190,000 \text{ psi} \)**
  
  \[
  t_{\text{Hoop}} = (\text{FOS}) \times p_{\text{burst}} \times r / \sigma_t = 1.5 \times 2582 \times 4.0 / 190,000 = 0.082 \text{ in} \\
  t_{\text{Dome}} = (\text{FOS}) \times p_{\text{burst}} \times (a/b)^{1/2} \times [2 + (a/b)^2] / (6 \sigma_t) = 1.5 \times 2582 \times [4 \times 2]^{1/2} \times [2 + (2)^2] / [6 \times 190,000] = 0.058 \text{ in} \\
  \]

  Weight = \( W_{\text{Cylinder}} + W_{\text{Dome}} = \rho \pi d t_{\text{Hoop}} l + \rho (2 \pi a b) t_{\text{Dome}} = 30.8 + 0.8 = 31.6 \text{ lb for steel case} \)

- **Try Graphite Fiber at \( \sigma_t = 450,000 \text{ psi Ultimate, Assume 60\% Fiber / 40\% Epoxy Composite}**
  
  \[
  t_{\text{Hoop}} = 1.5 \times 2582 \times 4.0 / [450,000 (0.60)] = 0.057 \text{ in radial fibers for internal pressure load} \\
  t_{\text{Dome}} = 0.041 \text{ in, for internal pressure load} \\
  \]

  Weight = 11.1 lb for composite case (w/o insulation, attachment, aft dome, and body bending fiber)

  Must also add about 0.015 in of either longitudinal fibers or helical wind to counteract body bending load
A Low Aspect Ratio Delta Wing Allows Lighter Weight Structure

\[ W_{\text{Surface}} = \frac{\sigma_{\text{max}}^{1/2}}{\sqrt[2]{\rho S N_{\text{max}}}} = \left[ A \left( 1 + 2 \lambda \right) \right]^{1/2} \]

\[ W_{\text{Surface}} = \rho S t_{\text{mac}} \]

\[ t_{\text{root}} = \left[ \frac{\text{FOS} N_{\text{max}} A \left( 1 + 2 \lambda \right)}{\sigma_{\text{max}}} \right]^{1/2} \]

Assumption: Uniform loading

**Note:**
Surface is 2 panels (Cruciform wing has 4 panels)

\[ W_{\text{Surface}} = \text{Surface weight sized by bending moment} \]

\[ \rho = \text{Density} \]

\[ \sigma_{\text{max}} = \text{Maximum allowable (ultimate) stress} \]

\[ t_{\text{mac}} = \text{Thickness of mean aero chord} \]

\[ t_{\text{root}} = \text{Thickness of root chord} \]

\[ N_{\text{max}} = \text{Maximum load} \]

\[ A = \text{Aspect ratio} \]

\[ \lambda = \text{Taper ratio} \]

**Example for Rocket Baseline Wing (2219-T81 Aluminum):**

\[ A = 2.82, \lambda = 0.175, c_r = 19.4 \text{ in}, \sigma_{\text{max}} = \sigma_{\text{ult}} = 65k \text{ psi} \]

Assume \( M = 2, h = 20k \text{ ft}, \alpha + \delta = 22 \text{ deg} \)

From prior example, \( N_{\text{max}} = 7525 \text{ lb} \)

Calculate \( t_{\text{root}} = \left[ \frac{1.5 \left( 7525 \right) \left( 2.82 \right) \left[ 1 + 2 \left( 0.175 \right) \right]}{65000} \right]^{1/2} = 0.813 \text{ in} \)

\[ t_{\text{root}} / c_{\text{root}} = 0.813 / 19.4 = 0.0419 = t_{\text{mac}} / c_{\text{mac}} \]

\[ t_{\text{mac}} = 0.0419 \left( 13.3 \right) = 0.557 \text{ in} \]

\[ W_{\text{wing}} \sigma_{\text{max}}^{1/2} = \left[ \rho S N_{\text{max}} \right]^{1/2} = \left[ A \left( 1 + 2 \lambda \right) \right]^{1/2} = 1.95 \]

\[ W_{\text{wing}} = 20.6 \text{ lb for 1 wing (2 panels)} \]
Dome Material Is Driven by the Type of Seeker and Flight Environment

<table>
<thead>
<tr>
<th>Seeker Dome Material</th>
<th>Density (g/cm³)</th>
<th>Dielectric Constant</th>
<th>MWIR / LWIR Bandpass</th>
<th>Transverse Strength (10^3 psi)</th>
<th>Thermal Expansion (10⁻⁶/°F)</th>
<th>Erosion, Knoop (kg/mm²)</th>
<th>Max Short-Duration Temp (°F)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>RF-IR Seeker</strong></td>
<td></td>
<td></td>
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<td>Zinc Sulfide (ZnS)</td>
<td>4.05</td>
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<td>☀️</td>
<td>☀️ 18</td>
<td>☀️ 4</td>
<td>☀️ 350</td>
<td>☀️ 700</td>
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<td>Zinc Selenide (ZnSe)</td>
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<td>☀️</td>
<td>☀️ 8</td>
<td>☀️ 4</td>
<td>☀️ 150</td>
<td>☀️ 600</td>
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<tr>
<td>Sapphire / Spinel</td>
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<td>☀️ 8.5</td>
<td>☀️  -</td>
<td>☀️ 28</td>
<td>☀️ 3</td>
<td>☀️ 1650</td>
<td>☀️ 1800</td>
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<td>☀️ 3.7</td>
<td>☀️  -</td>
<td>☀️ 8</td>
<td>☀️ 0.3</td>
<td>☀️ 600</td>
<td>☀️ 2000</td>
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<tr>
<td>Silicon Nitride (Si₃N₄)</td>
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<td>☀️  -</td>
<td>☀️ 90</td>
<td>☀️ 2</td>
<td>☀️ 2200</td>
<td>☀️ 2700</td>
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<td>Diamond (C)</td>
<td>3.52</td>
<td>☀️ 5.6</td>
<td>☀️  -</td>
<td>☀️ 400</td>
<td>☀️ 1</td>
<td>☀️ 8800</td>
<td>☀️ 3500</td>
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<tr>
<td><strong>RF Seeker</strong></td>
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<tr>
<td>Pyroceram</td>
<td>2.55</td>
<td>☀️ 5.8</td>
<td>☀️  -</td>
<td>☀️ 25</td>
<td>☀️ 3</td>
<td>☀️ 700</td>
<td>☀️ 2200</td>
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<tr>
<td>Polyimide</td>
<td>1.54</td>
<td>☀️ 3.2</td>
<td>☀️  -</td>
<td>☀️ 17</td>
<td>☀️ 40</td>
<td>☀️ 70</td>
<td>☀️ 400</td>
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<td><strong>IR Seeker</strong></td>
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<td>Mag. Fluoride (MgF₂)</td>
<td>3.18</td>
<td>☀️ 5.5</td>
<td>☀️</td>
<td>☀️ 7</td>
<td>☀️ 6</td>
<td>☀️ 420</td>
<td>☀️ 1000</td>
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<tr>
<td>Alon (Al₂O₃N₅)</td>
<td>3.67</td>
<td>☀️ 9.3</td>
<td>☀️  -</td>
<td>☀️ 44</td>
<td>☀️ 3</td>
<td>☀️ 1900</td>
<td>☀️ 1800</td>
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<td>Germanium (Ge)</td>
<td>5.33</td>
<td>☀️ 16.2</td>
<td>☀️</td>
<td>☀️ 15</td>
<td>☀️ 4</td>
<td>☀️ 780</td>
<td>☀️ 200</td>
</tr>
</tbody>
</table>

- Superior
- Above Average
- Average
- Below Average
- Poor

Dome Material is driven by the type of seeker and flight environment.
Radome Weight May Be Driven by Optimum Thickness Required for Efficient Transmission

\[ W_{\text{OptTrans}} = \rho S_{\text{wet}} t_{\text{OptTrans}} \]

\[ t_{\text{OptTrans}} = 0.5 n \frac{\lambda_0}{(\varepsilon - \sin^2 \theta_i)^{1/2}} \]

Note:
- \( W_{\text{OptTrans}} \) = Weight at Optimum Transmission
- \( \rho \) = Density
- \( S_{\text{wet}} \) = Surface wetted area
- \( t_{\text{OptTrans}} \) = Optimum thickness for 100% transmission
- \( n \) = Integer (1, 2, …)
- \( \lambda_0 \) = Wavelength in air
- \( \varepsilon \) = Dielectric constant
- \( \theta_i \) = Radar signal incidence angle = 90 deg - \( \delta - \theta \)
- \( \theta \) = Surface local angle
- \( \delta \) = Seeker look angle

Example for Rocket Baseline Pyroceram Radome:
- \( \varepsilon = 5.8, \rho = 0.092 \text{ lb/in}^3, \lambda_0 = 1.1 \text{ in}, n = 1, \text{ tangent ogive, } l = 19.2 \text{ in}, d = 8 \text{ in, } S_{\text{wet}} = 326 \text{ in}^2 \)
- \( \delta = 0 \text{ deg } \Rightarrow (\theta_i)_{\text{avg}} \approx 90 - 0 - 11.8 = 78.2 \text{ deg} \)
- \( t_{\text{OptTrans}} = 0.5 (1)(1.1) / (5.8 - 0.96)^{1/2} = 0.25 \text{ in} \)
- \( W_{\text{OptTrans}} = 0.092 (326)(0.25) = 7.5 \text{ lb} \)
# Missile Electrical Power Supply Alternatives

\[ W = W_E E + W_P P \]

<table>
<thead>
<tr>
<th>Measure of Merit</th>
<th>Power Supply</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Generator</td>
</tr>
<tr>
<td>Storage Life</td>
<td>![Superior]</td>
</tr>
<tr>
<td>Cost</td>
<td>![Superior]</td>
</tr>
<tr>
<td>( W_E ), Weight / Energy</td>
<td>0.0007</td>
</tr>
<tr>
<td>( kg / kW-s )</td>
<td>![Superior]</td>
</tr>
<tr>
<td>( W_P ), Weight / Power</td>
<td>1.4</td>
</tr>
<tr>
<td>( kg / kW )</td>
<td>![Superior]</td>
</tr>
</tbody>
</table>

- Superior
- Above Average
- Average
- Below Average

Example for Thermal battery: If \( E = 900 \text{ kW-s} \), \( P = 3 \text{ kW} \) \( \Rightarrow W = W_E E + W_P P = 0.0125 \times 900 + 0.3 	imes 3 = 12.2 \text{ kg} \)

Note: Generator provides highest energy with light weight for long time of flight (e.g., cruise missile).

Lithium battery provides nearly constant voltage suitable for electronics. Relatively high energy with light weight.

Thermal battery provides highest power with light weight (may be required for actuators).
Electromechanical Actuators Are Light Weight and Reliable

\[ W = W_T T_S \]

<table>
<thead>
<tr>
<th>Measure of Merit</th>
<th>EM</th>
<th>Cold Gas Pneumatic</th>
<th>Hydraulic</th>
</tr>
</thead>
<tbody>
<tr>
<td>( W_T ), Weight / Stall Torque ( lb / in-lb )</td>
<td>⭕️ 0.0025</td>
<td>⭑ 0.0050</td>
<td>⭑ 0.0034</td>
</tr>
<tr>
<td>Rate ( deg / s )</td>
<td>⭒ Up to 800</td>
<td>⭑ Up to 600</td>
<td>⭒ Up to 1000</td>
</tr>
<tr>
<td>Bandwidth ( Hz )</td>
<td>⭒ Up to 40</td>
<td>⭖ Up to 20</td>
<td>⭒ Up to 60</td>
</tr>
<tr>
<td>Reliability</td>
<td>⭒</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Cost</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Superior ☝️ Above Average ☝️ Average ☝️ Below Average

Note:
- Actuation system weight based on four actuators.
- Cold gas pneumatic actuation weight includes actuators, gas bottle, valves, regulator, and supply lines.
- Hydraulic actuation weight includes actuators, gas generator or gas bottle, hydraulic reservoir, valves, and supply lines.
- Stall torque \( \approx 1.5 \) maximum hinge moment of single panel.

\[
W = W_T T_S
\]

Example weight for rocket baseline hydraulic actuation at Mach 2, 20k ft alt, \( \alpha = 9 \) deg, with max control deflection of wing (\( \delta = 13 \) deg ) \( \Rightarrow \) Hinge moment of one panel = 11,500 in-lb. \( T_S = 1.5 \ (11500) = 17,250 \) in-lb \( \Rightarrow W = W_T T_S = 0.0034 \ (17250) = 59 \) lb

Schematic of Cold Gas Pneumatic Actuation
Examples of Electromechanical Actuator Packaging

Canard (Stinger)  

Tail (AMRAAM)  

Jet Vane / Tail (Javelin)  

Movable Nozzle (THAAD)
Summary of Weight

- Conceptual Design Weight Prediction Methods and Weight Considerations
  - Missile system weight, cg, moment of inertia
  - Factors of safety
  - Aerodynamic heating
  - Structure
  - Dome
  - Propulsion
  - Insulation
  - Power supply
  - Actuator

- Manufacturing Processes for Low Parts Count, Low Cost
  - Precision castings
  - Vacuum assisted RTM
  - Pultrusion / Extrusion
  - Filament winding
Summary of Weight (cont)

- Design Considerations
  - Airframe materials
  - Insulation materials
  - Seeker dome materials
  - Thermal stress
  - Aerodynamic heating

- Technologies
  - MEMS
  - Composites
  - Titanium alloys
  - High density insulation
  - High energy and power density power supply
  - High torque density actuators

- Discussion / Questions?
- Classroom Exercise (Appendix A)
1. Propulsion system and structure weight are driven by f____ o_s_____ requirements.
2. For a ballistic range greater than about 200 nautical miles, a t__ s_____ missile is lighter weight.
3. Tactical missile weight is proportional to v_____.
4. Subsystem d______ for tactical missiles is about 0.05 lb / in^3
5. Modeling weight, balance, and moment-of-inertia is based on a build-up of s_________.
6. Missile structure factor of safety for free flight is usually about 1.25 for ultimate loads and about 1.10 for y____ loads.
7. Manufacturing processes that can allow low parts count include vacuum assisted resin transfer molding of composites and c_______ of metals.
8. Low cost airframe materials are usually based on aluminum and steel while light weight airframe materials are usually based c_________ materials.
9. Graphite fiber has high strength and high m______ o_ e_________.

Weight Problems
10. The recovery factors of stagnation, turbulent boundary layer, and laminar boundary layer are 1.0, 0.9, and ___ respectively.

11. The most popular types of insulation for temperatures greater than 4,000 R are charring insulators based on p_______ composites.

12. Tactical missiles experience transient heating, and with increasing time the temperature approaches the r_______ temperature.

13. The inner wall temperature is nearly the same as the surface temperature for a t_______ t____ structure.

14. A thermally thick surface is a good i_______.

15. A low conductivity structure is susceptible to thermal s_____.

16. The minimum gauge thickness is often set by the m__________ process.

17. A very thin wall structure is susceptible to localized b______.

18. Ejection loads and flight control loads often result in large b______ moment.

19. An approach to increase the tactical missile propellant / motor weight fraction over the typical value of 72% would be c_______ motor case.
20. The required rocket motor case thickness is often driven by the combustion chamber p_______.
21. A low aspect ratio delta wing has reduced w______.
22. For low speed missiles, a popular infrared dome material is z___ s_______.
23. A thermal battery provides high p____.
24. The most popular type of actuator for tactical missiles is an e________________ actuator.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

1. Define Mission Requirements
2. Establish Baseline
   - Alt Mission
   - Alt Baseline
3. Aerodynamics
4. Propulsion
5. Weight
6. Trajectory
7. Meet Performance?
   - Yes
   - No
   - Yes
   - No
   - Yes
   - No
   - Yes
8. Measures of Merit and Constraints
9. Resize / Alt Config / Subsystems / Tech
Flight Envelope Should Have Large Max Range, Small Min Range, and Large Off Boresight

Examples of Max / Min Range Limitations:
- Fire Control System Range and Off Boresight
- Seeker Range, Gimbal Angle, and Tracking Rate
- Maneuver Capability
- Time of Flight
- Closing Velocity
Conceptual Design Modeling Versus Preliminary Design Modeling

Conceptual Design Modeling

1 DOF [ Axial force ( $C_{DO}$ ), thrust, weight ]

2 DOF [ Normal force ( $C_N$ ), axial force, thrust, weight ]

3 DOF point mass [ 3 aero forces ( normal, axial, side ), thrust, weight ]

3 DOF pitching [ 2 aero forces ( normal, axial ), 1 aero moment ( pitching ), thrust, weight ]

4 DOF [ 2 aero forces ( normal, axial ), 2 aero moments ( pitching, rolling ), thrust, weight ]

Preliminary Design Modeling

6 DOF [ 3 aero forces ( normal, axial, side ), 3 aero moments ( pitching, rolling, yawing ), thrust, weight ]
1-DOF Coast Equation May Have Good Accuracy Near Zero Angle of Attack

\[
\frac{\dot{V}_{2-DOF}}{\dot{V}_{1-DOF}}, \quad \text{Predicted Deceleration Comparison for Rocket Baseline}
\]

Note:
- \( \dot{V}_{2-DOF} \) = Two-degrees-of-freedom deceleration
- \( \dot{V}_{1-DOF} \) = One-degree-of-freedom deceleration
- Rocket baseline during coast
- Mach 2, \( h = 20,000 \) ft
- \( \alpha_{\text{Trim}} \approx 0.3 \) deg for 1-g flyout
3-DOF Simplified Equations of Motion Show Drivers for Configuration Sizing

\[ I_y \theta'' \approx I_y \alpha'' \approx q S_{Ref} \ d C_{m\alpha} \ \alpha + q S_{Ref} \ d C_{m\delta} \ \delta \]

\[ \left( \frac{W}{g_c} \right) \ V \gamma' \approx q S_{Ref} \ C_{N\alpha} \ \alpha + q S_{Ref} \ C_{N\delta} \ \delta - W \ \cos \gamma \]

\[ \left( \frac{W}{g_c} \right) \ V'' \approx T - C_A \ S_{Ref} \ q - C_{N\alpha} \ \alpha^2 S_{Ref} \ q - W \ \sin \gamma \]

Configuration Sizing Implication

High Control Effectiveness \( \Rightarrow C_{m\delta} > C_{m\alpha}, I_y \text{ small} \ (W \text{ small}), q \text{ large} \)

Large / Fast Heading Change \( \Rightarrow C_N \text{ large}, W \text{ small}, q \text{ large} \)

High Speed / Long Range \( \Rightarrow \text{Total Impulse large}, C_A \text{ small}, q \text{ small} \)

Note: Based on aerodynamic control
For Long Range Cruise, Maximize $V_{Isp}$, $L / D$, and Weight Fraction of Fuel / Propellant

\[ R = ( V_{Isp} )( L / D ) \ln \left[ \frac{W_{BC}}{W_{BC} - W_P} \right], \text{Breguet Range Equation} \]

Example: Ramjet Baseline at Mach 3 / 60k ft alt
\[
R = 2901 \left( 1040 \right) \left( 3.15 \right) \ln \left[ \frac{1739}{1739 - 476} \right] = (9,503,676) \ln \left[ \frac{1}{1 - 0.2737} \right] = 3,039,469 \text{ ft} = 500 \text{ nm}
\]

Note: $R$ = cruise range, $V$ = cruise velocity, $I_{sp}$ = specific impulse, $L$ = lift, $D$ = drag, $W_{BC}$ = weight at begin of cruise, $W_p$ = weight of propellant or fuel
Efficient Steady Flight Is Enhanced by High L / D and Light Weight

Steady Level Flight

\[ T = D \]
\[ L = W \]

\[ T = W / (L / D) \]

Steady Climb

\[ \sin \gamma_C = (T - D) / W = \frac{V_C}{V_\infty} \]

\[ V_C = (T - D) V_\infty / W \]

\[ R_C = \Delta h / \tan \gamma_C = \Delta h (L / D) \]

Steady Descent (Glide)

\[ \sin \gamma_D = (D - T) / W = \frac{V_D}{V_\infty} \]

\[ V_D = (D - T) V_\infty / W \]

\[ R_D = \Delta h / \tan \gamma_D = \Delta h (L / D) \]

Note:
- Small Angle of Attack
- Equilibrium Flight
- \( V_C \) = Velocity of Climb
- \( V_D \) = Velocity of Descent
- \( \gamma_C \) = Flight Path Angle During Climb
- \( \gamma_D \) = Flight Path Angle During Descent
- \( V_\infty \) = Total Velocity
- \( \Delta h \) = Incremental Altitude
- \( R_C \) = Horizontal Range in Steady Climb
- \( R_D \) = Horizontal Range in Steady Dive (Glide)

**Flight Trajectory Lofting / Shaping Provides Extended Range**

Lofted Trajectory Design Guidelines for Horizontal Launch:
- High thrust-to-weight $\approx 10$ for safe separation
- Rapid pitch up minimizes time / propellant to reach efficient altitude
- Climb at $\alpha \approx 0$ deg with thrust-to-weight $T / W \approx 2$ and $q \approx 700$ psf to minimize drag / propellant
- Apogee at $q \approx 700$ psf, followed by either $(L / D)_{MAX}$ cruise or $(L / D)_{MAX}$ glide
Small Turn Radius Using Aero Control Requires High Angle of Attack and Low Altitude Flight

\[ R_T = \frac{V}{\gamma} \approx 2 \frac{W}{(g_c C_N S_{Ref} \rho)} \]

Assumption: Horizontal Turn

Note for Example Figure:
- \( W = \text{Weight} = 2,000 \text{ lb} \)
- \( a / b = 1 \) (circular cross section), No wings
- \( C_N = \sin 2 \alpha \cos (\alpha / 2) + 2 (l / d) \sin^2 \alpha \)
- \( l / d = \text{Length} / \text{Diameter} = 10 \)
- \( S_{Ref} = 2 \text{ ft}^2 \)
- \( C_D = 0.2 \)
- \( (L / D)_{Max} = 2.5 \)
- \( q(L / D)_{Max} \approx 700 \text{ psf} \)
- \( \alpha(L / D)_{Max} = 15 \text{ deg} \)
- \( T(L / D)_{Max} = 740 \text{ lb} \)

Example:
- \( \Delta \alpha = 10 \text{ deg} \)
- \( C_N = 0.94 \)
- \( h = 40k \text{ ft (} \rho = 0.000585 \text{ slug / ft}^3) \)
- \( R_T = 2 \frac{2,000}{[(32.2)(0.94)(2)(0.000585)]} = 112,000 \text{ ft} \)

Note: Require \( R_T \text{Missile} \leq R_T \text{Target} \) for small miss distance
High Turn Rate Using Aero Control Requires High Angle of Attack and High Velocity

\[ \gamma' = g_c C_N \rho V S_{Ref} / (2 W), \text{ rad / s} \]

Assumption: Horizontal Turn

Example for Lifting Body at Altitude \( h = 20,000 \text{ ft} \):

Assume:
- \( W = \text{Weight} = 2,000 \text{ lb} \)
- \( a / b = 1 \) (circular cross section)
- No wings
- Negligible tail lift
- Neutral static stability

\[ C_N = \sin 2 \alpha \cos (\alpha/2) + 2(\text{l/d})\sin^2 \alpha \]

- \( S_{Ref} = 2 \text{ ft}^2 \)
- \( \text{l/d} = \text{Length} / \text{Diameter} = 10 \)
- \( \alpha = 15 \text{ deg} \)
- \( V = 2000 \text{ ft / s} \)

Then:
- \( N = \text{Normal Force} = C_N \rho V S_{Ref} \)

\[ C_N = \sin [2(15)] \cos (15/2) + 2(10)\sin^2 (15) = 0.50 + 1.34 = 1.835 \]

\[ q = \text{Dynamic Pressure} = 0.5 \rho V^2 = 0.5 \left( 0.001267 \right) \left( 2000 \right)^2 = 2534 \text{ psf} \]

\[ N = 1.835 \times 2534 \times (2) = 9,300 \text{ lb} \]

\[ N / W = 9300 / 2000 = 4.65 \text{ g} \]

\[ \gamma' = 32.2 \times 1.835 \times 0.001267 \times 2 \times 2 / [2(2000)] \]

\[ = 0.0749 \text{ rad / s} = 4.29 \text{ deg / s} \]
For Long Range Coast, Maximize Initial Velocity and Altitude and Minimize Drag Coefficient

For Long Range Coast, Maximize Initial Velocity and Altitude and Minimize Drag Coefficient

\[
\frac{V}{V_i} = \left\{ 1 - \left[ \frac{g_c \sin \gamma}{V_i} \right] t \right\} / \left\{ 1 + \left[ \frac{g_c \rho_{AVG} S_{Ref} (C_{D_0})_{AVG} V_i}{(2W)} \right] t \right\}
\]

\[
\left[ \frac{g_c \rho_{AVG} S_{Ref} (C_{D_0})_{AVG}}{(2W)} \right] R = \ln \left\{ 1 - \left[ \frac{g_c^2 \rho_{AVG} S_{Ref} (C_{D_0})_{AVG}}{(2W)} \right] \frac{1}{2} \right\} + \left[ \frac{g_c \rho_{AVG} S_{Ref} (C_{D_0})_{AVG} V_i}{(2W)} \right] t
\]

Example for Rocket Baseline:

- \( W = W_{BO} = 367 \text{ lb}, S_{Ref} = 0.349 \text{ ft}^2, V_i = 2,151 \text{ ft/s}, \gamma = 0 \text{ deg}, (C_{D_0})_{AVG} = 0.9, h = 20,000 \text{ ft}, (\rho = 0.00127 \text{ slug/ft}^3), t = 10 \text{ s}\)
- \( [(g_c \rho S_{Ref} (C_{D_0})_{AVG} V_i)/(2W)] t = [(32.2 (0.00127) (0.349) (0.9) (2151)) / [2 (367)]] 10 = 0.376\)
- \( V/V_i = 0.727 \Rightarrow V = 0.727 \times 2151 = 1564 \text{ ft/s}, [[(g_c \rho S_{Ref} (C_{D_0})_{AVG})]/(2W)] R = 0.319 \Rightarrow R_{coast} = 18,300 \text{ ft or } 3.0 \text{ nm}\)

Note: Based on 1 DOF coast
\[ dV/dt = -g_c C_{D_0} S_{Ref} q/W - g_c \sin \gamma \]

Assumptions:
- \( \gamma = \text{constant} \)
- \( \alpha \approx 0 \text{ deg} \)
- \( D > W \sin \gamma \)

\( V = \) velocity during coast
\( V_i = \) initial velocity ( begin coast )
\( R = \) coast range
\( V_x = V \cos \gamma, V_y = V \sin \gamma \)
\( R_x = R \cos \gamma, R_y = R \sin \gamma \)
For Ballistic Range, Maximize Initial Velocity, Optimize Launch Angle, and Minimize Drag

\[
\frac{V_x}{V_i} = \cos \gamma_i / \left( 1 + \left[ g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W) \right] t \right)
\]

\[
\frac{V_y + g_c t}{V_i} = \sin \gamma_i / \left( 1 + \left[ g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W) \right] t \right)
\]

\[
\frac{V_x}{V_i} = \cos \gamma_i / \left( 1 + \left[ g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W) \right] t \right)
\]

\[
\frac{V_y + g_c t}{V_i} = \sin \gamma_i / \left( 1 + \left[ g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W) \right] t \right)
\]

Assumptions: Thrust = 0, \( \alpha = 0 \) deg, \( D > W \sin \gamma \), flat earth

Nomenclature: \( V \) = velocity during ballistic flight, \( V_i \) = initial velocity, \( R_x \) = horizontal range, \( h \) = altitude, \( h_i \) = initial altitude, \( V_x \) = horizontal velocity, \( V_y \) = vertical velocity

Example for Rocket Baseline:

- \( W = 367 \) lb, \( S_{Ref} = 0.349 \) ft\(^2\), \( V_i = V_{BO} = 2,151 \) ft / s, \( \gamma_i = 0 \) deg, \( (C_{D0})_{AVG} = 0.9 \), \( h_i = 20,000 \) ft, \( \rho_{AVG} = 0.00182 \) slug / ft\(^3\), \( t = 35 \) s

- \( [g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W)] t = \{ 32.2 (0.00182) (0.349) (0.9) (2151) / [2 (367)] \} = 35 \) ft

- \( V_x / V_i = 0.354 \Rightarrow V_x = 762 \) ft / s, \( (V_y + 32.2 t) / V_i = 0.354 \Rightarrow V_y = -1127 \) ft / s, \( [g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W \cos \gamma_i)] R_x = 1.037 \Rightarrow R_x = 42,900 \) ft or 7.06 nm, \( [g_c \rho_{AVG} S_{Ref} (C_{D0})_{AVG} V_i / (2W \sin \gamma_i)] (h - h_i + 16.1 t^2) = 1.037 \Rightarrow h = 0 \) ft
High Propellant Weight, High Thrust, and Low Drag Provide High Burnout Velocity

\[ \Delta V / (g_c I_{SP}) = - [1 - (D_{AVG} / T) - (W_{AVG} \sin \gamma / T)] \ln (1 - W_p / W_i) \]

Example for Rocket Baseline:
Assume \( \gamma = 0 \) deg
Assume \( M_i = 0.8, h = 20k \) ft
\( W_i = W_L = 500 \) lb
For boost, \( W_p = 84.8 \) lb
\( W_p / W_L = 0.1696 \)
\( I_{SP} = 250 \) s
\( T_B = 5750 \) lb
Assume \( D = D_{AVG} = 635 \) lb
\( D_{AVG} / T = 0.110 \)
\( \Delta V / [(32.2)(250)] = -(1 - 0.110) \ln (1 - 0.1696) = 0.1654 \)
\( \Delta V = (0.1654)(32.2)(250) = 1331 \) ft / s

Note: 1 DOF Equation of Motion with \( \alpha \approx 0 \) deg, \( \gamma = \) constant, \( W_i = \) initial weight, \( W_{AVG} = \) average weight, \( W_p = \) propellant weight, \( I_{SP} = \) specific impulse, \( T = \) thrust, \( M_i = \) initial Mach number, \( h = \) altitude, \( D_{AVG} = \) average drag, \( \Delta V = \) incremental velocity, \( g_c = \) gravitation constant, \( V_x = V \cos \gamma, V_y = V \sin \gamma, R_x = R \cos \gamma, R_y = R \sin \gamma \)

Note: \( R = (V_i + \Delta V / 2) t_B, \) where \( R = \) boost range, \( V_i = \) initial velocity, \( t_B = \) boost time
High Missile Velocity and Lead Are Required to Intercept High Speed Crossing Targets

VM sin L = VT sin A, Proportional Guidance Trajectory

Note:
Proportional Guidance
VM = Missile Velocity
VT = Target Velocity
A = Target Aspect
L = Missile Lead Angle
≈ Seeker Gimbal

Example:
L = 30 deg
A = 45 deg
VM / VT = sin (45°) / sin (30°) = 1.42
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

Define Mission Requirements

- Establish Baseline
  - Aerodynamics
    - Propulsion
      - Weight
      - Trajectory
        - Meet Performance?
          - Yes
            - Measures of Merit and Constraints
          - No
            - No

- Alt Mission
  - Alt Baseline
    - Resize / Alt Config / Subsystems / Tech
      - No
Summary of Flight Performance

◊ Flight Performance Activity in Missile Design
  ◦ Compute range, velocity, time-to-target, off boresight
  ◦ Compare with requirements

◊ Discussed in This Chapter
  ◦ Equations of motion
  ◦ Flight performance drivers
  ◦ Propulsion alternatives range comparison
  ◦ Steady level flight required thrust
  ◦ Steady climb and steady dive range prediction
  ◦ Cruise prediction
  ◦ Boost prediction
  ◦ Coast prediction
  ◦ Ballistic flight prediction
  ◦ Turn radius and turn rate prediction
  ◦ Target lead for proportional homing guidance
Flight Performance Strongly Impacted by
- Aerodynamics
- Propulsion
- Weight

Discussion / Questions?
Classroom Exercise ( Appendix A )
Flight Performance Problems

1. Flight trajectory calculation requires input from aero, propulsion, and w_____.
2. Missile flight envelope can be characterized by the maximum effective range, minimum effective range, and o__ b________.
3. Limitations to the missile effective range include the fire control system, seeker, time of flight, closing velocity, and m_______ capability.
4. 1 DOF simulation requires modeling only the thrust, weight, and a____ f____.
5. A 3 DOF simulation that models 3 aero forces is called p____ m____ simulation.
6. A simulation that includes 3 aero forces ( normal, axial, side ), 3 aero moments ( pitch, roll, yaw ), thrust, and weight is called a _ DOF simulation.
7. The pitch angular acceleration $\dot{\theta}$ is approximately equal to the second time derivative of the a____ o_ a_____.
8. Cruise range is a function of velocity, specific impulse, L / D, and f___ fraction.
9. If thrust is equal to drag and lift is equal to weight, the missile is in s_____ I_____ flight.
10. Turn rate is a function of normal force, weight, and v_______.

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11. Coast range is a function of initial velocity, weight, drag, and the t___ of flight.
12. Incremental velocity due to boost is a function of $l_{sp}$, drag, and p________ weight fraction.
13. To intercept a high speed crossing target requires a high speed missile with a high g_____ angle seeker.
14. An analytical model of a rocket in co-altitude, non-maneuvering flight can be developed by patching together the flight phases of boost and c____.
15. An analytical model of a rocket in a short range, off-boresight intercept can be developed by patching the flight phases of boost and t___.
16. An analytical model of a guided bomb in non-maneuvering flight can be developed from the flight phase of a steady d____.
17. An analytical model of an unguided weapon can be developed from the b_______ flight phase.
18. An analytical model of a ramjet in co-altitude, non-maneuvering flight can be developed by patching the flight phases of boost and c_____.

Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Missile Concept Synthesis Requires Evaluation of Alternatives and Iteration

1. Define Mission Requirements
2. Establish Baseline
   - Aerodynamics
   - Propulsion
   - Weight
3. Trajectory
4. Meet Performance?
   - Yes
     - Measures of Merit and Constraints
   - No
     - Alt Mission
     - Alt Baseline
     - Resize / Alt Config / Subsystems / Tech

Note: The diagram shows a decision-making process where the outcome of each step influences the next, leading to the final evaluation of measures of merit and constraints.
Measures of Merit and Launch Platform Integration Should Be Harmonized

- Robustness
- Lethality
- Miss Distance
- Carriage and Launch Observables
- Other Survivability Considerations
- Reliability
- Cost
- Balanced Design
- Launch Platform Integration / Firepower
Tactical Missiles Must Be Robust

- Tactical Missiles Must Have Robust Capability to Handle
  - Adverse Weather
  - Clutter
  - Local Climate
  - Flight Environment Variation
  - Uncertainty
  - Countermeasures
  - EMI / EMP

- This Section Provides Examples of Requirements for Robustness
Adverse Weather and Cloud Cover Are Pervasive

North Pole Region

North Atlantic

Deserts (Sahara, Gobi, Mojave)

Argentina, Southern Africa and Australia

South Pole Region

Annual Average Fraction of Cloud Cover

Cloud Cover Over Ocean

Cloud Cover Over Land

Rising Air

Descending Air

Latitude Zone

North Pole Region

North Atlantic

Deserts (Sahara, Gobi, Mojave)

Argentina, Southern Africa and Australia

South Pole Region

Note: Annual Average Cloud Cover

Global Average = 61%

Global Average Over Land = 52%

Global Average Over Ocean = 65%

Radar Seekers Are Robust in Adverse Weather

Note:

EO attenuation through cloud at 0.1 g / m³ and 100 m visibility
EO attenuation through rain at 4 mm / h
Humidity at 7.5 g / m³
Millimeter wave and microwave attenuation through cloud at 0.1 gm / m³ or rain at 4 mm / h

- EO sensors are ineffective through cloud cover
- Radar sensors have good to superior performance through cloud cover and rain

Radar Seekers Are Desirable for Robust Operation within the Troposphere Cloud Cover

Note:
- IR seeker may be able to operate “Under the Weather” at elevations less than 2,000 ft using GPS / INS midcourse guidance.
- IR attenuation through cloud cover greater than 100 dB per km. Cloud droplet size (0.1 to 50 μm) causes resonance.
- mmW has ~2 dB/km attenuation through rain. Typical rain drop size (~4 mm) is comparable to mmW wavelength.
### Precision Strike Missile Target Sensors Are Complemented by GPS / INS / Data Link Sensors

<table>
<thead>
<tr>
<th>Sensor</th>
<th>Adverse Weather Impact</th>
<th>ATR / ATA in Clutter</th>
<th>Range</th>
<th>Moving Target</th>
<th>Volume Search Time</th>
<th>Hypersonic Dome Comp.</th>
<th>Diameter Required</th>
<th>Weight and Cost</th>
<th>Maturity</th>
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</table>

**Note:**  
- Superior  
- Good  
- Below Average  
- Poor
Imaging Sensors Enhance Target Acquisition / Discrimination

Imaging LADAR

Imaging Infrared

SAR

Passive Imaging mmW

Video of Imaging Infrared

Video of SAR Physics
Assume Exo-atmospheric Intercept with

- Target diameter $D_T = 2$ ft ($A_T = 2919$ cm$^2$), temperature $T_T = 300$ K, emissivity $\varepsilon = 0.5$
- Diameter of seeker aperture $d_o = 5$ in ($A_o = 0.01267$ m$^2$)
- Diameter of pixel detector $d_p = 40$ $\mu$m
- Spot resolution if diffraction limited $= d_{\text{spot}} = d_p = 40$ $\mu$m
- Temperature of pixel detector $T_d = 77$ K
- Focal plane array size 256x256 FPA ($A_d = 1.049$ cm$^2$)
- Pixel detector bandwidth $\Delta f_p = 50$ Hz ($t_{\text{integ}} = 0.00318$ s)
- Required signal-to-noise ratio for detection $(S/N)_D = 5$

First Calculate MWIR Seeker Detection Range

- Radiant intensity of target within seeker bandwidth $(I_T)_{\Delta\lambda} = \varepsilon L_{\lambda} (\lambda_2 - \lambda_1) A_T$, W sr$^{-1}$
- Spectral radiance of target $L_{\lambda} = 3.74 \times 10^4 / \{ \lambda^5 \{ e^{\{1.44 \times 10^4 / (\lambda T_T)\}} - 1 \} \}$, W cm$^{-2}$ sr$^{-2}$ $\mu$m$^{-1}$
- Assume $\lambda = 4$ $\mu$m, $\lambda_2 = 5$ $\mu$m, $\lambda_1 = 3$ $\mu$m, then
- $L_{\lambda} = 0.000224$ W cm$^{-2}$ sr$^{-2}$ $\mu$m$^{-1}$, $(I_T)_{\Delta\lambda} = 0.643$ W sr$^{-1}$
- Assume Hg$_{0.67}$Cd$_{0.33}$Te detector at $\lambda = 4$ $\mu$m and 77 K $\Rightarrow D^* = 8 \times 10^{11}$ cm Hz$^{1/2}$ W$^{-1}$
- $R_D = \{ \{ 0.643 \} \{ 1 \} \{ 0.01267 \} \{ 5 \}^{1/2} \{ 1.049 \}^{1/2} \}^{1/2} = 13,480$ m
Next, Calculate LWIR Seeker Detection Range

\[
R_D = \left\{ \left( I_T \right)_{\Delta \lambda} \eta_a A_0 \left\{ D^* / \left[ \left( \Delta f_p \right)^{1/2} \left( A_d \right)^{1/2} \right] \right\} \left( S / N \right)_{D^{-1}} \right\}^{1/2}, \text{m}
\]

\[
\left( I_T \right)_{\Delta \lambda} = \varepsilon L_\lambda \left( \lambda_2 - \lambda_1 \right) A_T, \text{W sr}^{-1}
\]

\[
L_\lambda = 3.74 \times 10^4 / \{ \lambda^5 \left\{ e^{1.44 \times 10^4 / \left( \lambda \cdot T_T \right)} - 1 \right\} \}, \text{W cm}^{-2} \text{sr}^{-2} \mu\text{m}^{-1}
\]

Assume \( \lambda = 10 \mu\text{m} \), \( \lambda_2 = 13 \mu\text{m} \), \( \lambda_1 = 7 \mu\text{m} \), then

\[
L_\lambda = 0.00310 \text{ W cm}^{-2} \text{ sr}^{-2} \mu\text{m}^{-1}, \left( I_T \right)_{\Delta \lambda} = 26.7 \text{ W sr}^{-1}
\]

Assume Hg_{0.80}Cd_{0.20}Te detector at \( \lambda = 10 \mu\text{m} \) and 77 K \( \Rightarrow D^* = 5 \times 10^{10} \text{ cm}^{1/2} \text{ Hz}^{1/2} \text{ W}^{-1} \)

\[
R_D = \left\{ \left( 26.7 \right) \left( 1 \right) \left( 0.01267 \right) \left\{ \left( 5 \times 10^{10} \right) / \left[ \left( 50 \right)^{1/2} \left( 1.049 \right)^{1/2} \right] \right\} \left( 5 \right)^{-1} \right\}^{1/2} = 21,600 \text{ m}
\]

MWIR Seeker Versus LWIR Seeker Selection Depends Upon Target Temperature

\[
(\lambda)_{\text{max}} = 2898 / T_T, \text{ Wein’s Displacement Law, } T_T \text{ in K}
\]

Example: \( T_T = 300 \text{ K} \)

\[
(\lambda)_{\text{max}} = 2898 / 300 = 10.0 \mu\text{m}
\]
GPS / INS Provides Robust Seeker Lock-on in Adverse Weather and Clutter

Note: = Target Aim Point and Seeker Tracking Gate, GPS / INS Accuracy = 3 m, Seeker 640 x 480 Image, Seeker FOV = 20 deg, Proportional Guidance Navigation Ratio = 4, Velocity = 300 m / s, G&C Time Constant = 0.2 s.

Seeker Lock-on @ 250 m to go (1 pixel = 0.14 m)
3 m GPS / INS error ⇒ n_{Mreq} = 1.76 g, σ = 0.9 m

Seeker Lock-on @ 850 m to go (1 pixel = 0.47 m)
3 m GPS / INS error ⇒ n_{Mreq} = 0.15 g, σ < 0.1 m

Seeker Lock-on @ 500 m to go (1 pixel = 0.27 m)
3 m GPS / INS error ⇒ n_{Mreq} = 0.44 g, σ < 0.1 m

Seeker Lock-on @ 125 m to go (1 pixel = 0.07 m)
3 m GPS / INS error ⇒ n_{Mreq} = 7.04 g, σ = 2.2 m
Data Link Update at Seeker Lock-on Reduces Moving Target Error

\[ \text{TE}_{\text{Seeker Lock-on}} = \left[ \text{TLE}^2 + (V_T \cdot t_{\text{Latency}})^2 \right]^{1/2} \]

Example:
\begin{align*}
\text{TLE} &= 10 \text{ m} \\
\text{t}_{\text{Seeker Lock-on}} &= 100 \text{ s} \\
\text{t}_{\text{Update}} &= 90 \text{ s} \\
V_T &= 10 \text{ m/s} \\
\text{t}_{\text{Latency}} &= \text{t}_{\text{Seeker Lock-on}} - \text{t}_{\text{Update}} = 100 - 90 = 10 \text{ s} \\
\text{TE}_{\text{Seeker Lock-on}} &= \left[ \text{TLE}^2 + (V_T \cdot \text{t}_{\text{Latency}})^2 \right]^{1/2} \\
&= \left[ 10^2 + (10 \cdot 10)^2 \right]^{1/2} \\
&= 100.5 \text{ m}
\end{align*}

Note: \( t_{\text{Seeker Lock-on}} \) = Seeker Lock-on time, \( t_{\text{Update}} \) = Data Link Update Time, \( V_T \) = Target Velocity, TLE = Target Location Error at Update = 10 m
Optimum Cruise Is a Function of Mach Number, Altitude, and Planform Geometry

Note:
- U.S. 1976 Standard Atmosphere
- For Efficient Cruise, \((L/D)_{\text{Max}}\) for Cruising Lifting Body Typically Occurs for \(500 < q < 1,000\) psf
- \((L/D)_{\text{Max}}\) for Cruise Missile with Low Aspect Ratio Wing Typically Occurs for \(200 < q < 500\) psf
- \(q \approx 200\) psf lower limit for aero control
Missile Guidance and Control Must Be Robust for Changing Events and Flight Environment

Example High Performance Missile Has
- Low-to-High Dynamic Pressure
- Negative-to-Positive Static Margin
- Thrust / Weight / cg Transients
- High Temperature
- High Thermal Load
- High Vibration
- High Acoustics

- Air Launch at Low Mach (high $\alpha$) / Deploy Compressed Carriage Surfaces
- Booster Ignition
- Engine Start Transient
- Booster Shutdown Transient at High Mach
- Pitch-Up at High Alpha
- Climb
- Engine Shutdown Transient
- Cruise
- Level Out
- Pitch-Over at High Alpha
- Vertical Launch in Cross Wind (high $\alpha$) / Deploy Compressed Carriage Surfaces
- Dive
- Terminal at High Dynamic Pressure
- Precision Impact at $\alpha \approx 0$ Deg
- Pitch-Over at High Alpha
- Precision Impact at $\alpha \approx 0$ Deg
Design Robustness Requires Consideration of Flight Altitude

Troposphere (h < 36,089 ft)
- $T / T_{SL} = 1 - 6.875 \times 10^{-6} h$, h in ft
- $p / p_{SL} = (T / T_{SL})^{5.2561}$
- $\rho / \rho_{SL} = (T / T_{SL})^{4.2561}$

Stratosphere (h > 36089 ft)
- $T = \text{constant} = 390 R$
- $p / p_{SL} = 0.2234 e^{-(h - 36089) / 20807}$
- $\rho / \rho_{SL} = 0.2971 e^{-(h - 36089) / 20807}$

Note: $T_{SL} = \text{Temperature at sea level}$, $p_{SL} = \text{pressure at sea level}$, $\rho_{SL} = \text{density at sea level}$, $c_{SL} = \text{speed of sound at sea level}$, $h = \text{altitude in ft}$.
Design Robustness Requires Consideration of Cold and Hot Atmospheres

Note:

- Based on properties at sea level

U. S. 1976 Standard Atmosphere: Temperature = 519 R, Density = 0.002377 slug/ft³, Speed of sound = 1116 ft/s

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ELF
Design Robustness Is Required to Handle Uncertainty

Example Normalized PDF

Typical % Error from Forecast Value

Note for normal distribution: $PDF = \left\{ \frac{1}{\sigma \sqrt{2\pi}} \right\} e^{\left( \frac{(x - \mu)^2}{2\sigma^2} \right)}$
### Counter-Countermeasures by Missile Enhance

**Design Robustness**

#### Examples of CM (Threat)

- **EOCM**
  - directed laser
  - flare
  - smoke
- **RFCM**
  - active radar
  - jammer
  - chaff
- **Decoy**
- **Low Observables**
- **Speed**
- **Altitude**
- **Maneuverability**
- **Lethal Defense**

#### Examples of CCM (Missile)

- **Imaging Seeker**
- **Multi-spectral / Multi-mode Seeker**
- **Temporal Processing**
- **Hardened GPS / INS**
  - standoff acquisition
  - Integrated GPS / INS
  - directional antenna
  - pseudolite / differential GPS
- **ATR / ATA**
- **Speed**
- **Altitude**
- **Maneuverability**
- **Low Observables**
- **Saturation**
Examples of Countermeasure-Resistant Seekers

IIR (I^2R AGM-130) .................................................................

Two Color IIR (Python 5)

Acoustic - IIR (BAT) .............................................................

IIR – LADAR (LOCAAS) ........................................................

ARH – mmW (AARGM)

ARH - IIR (Armiger) .............................................................
Examples of Targets where Size and Hardness Drive Warhead Design / Technology

- Small Size, Hard Target: Tank ⇒ Small Shaped Charge, EFP, or KE Warhead
- Deeply Buried Hard Target: Bunker ⇒ KE / Blast Frag Warhead
- Large Size Target: Building ⇒ Large Blast Frag Warhead
76% of Baghdad Targets Struck First Night of Desert Storm Were C³ Time Critical Targets


Source: AIR FORCE Magazine, 1 April 1998
Type of Target Drives Precision Strike Missile
Size, Speed, Cost, Seeker, and Warhead

- **Anti-Fixed Surface Target Missiles** (large size, wings, subsonic, blast frag warhead)
  - AGM-154
  - Storm Shadow / Scalp
  - KEPD-350
  - BGM-109
  - AGM-142

- **Anti-Radar Site Missiles** (ARH seeker, high speed or duration, blast frag warhead)
  - AGM-88
  - AS-11 / Kh-58
  - ARMAT
  - Armiger
  - ALARM

- **Anti-Ship Missiles** (large size, KE / blast frag warhead, and high speed or low altitude)
  - MM40
  - AS-34 Kormoran
  - AS-17 / Kh-31
  - BrahMos
  - SS-N-22 / 3M80

- **Anti-Armor Missiles** (small size, hit-to-kill, low cost, shape charge, EFP, or KE warhead)
  - Hellfire
  - LOCAAS
  - MGM140
  - AGM-65
  - LOSAT

- **Anti-Buried Target Missiles** (large size, high fineness, KE / blast frag warhead)
  - CALCM
  - GBU-28
  - GBU-31
  - Storm Shadow
  - MGM-140
# Examples of Light Weight Air Launched Multi-Purpose Precision Strike Weapons

<table>
<thead>
<tr>
<th>Weapon</th>
<th>Fixed Surface Targets&lt;sup&gt;(1)&lt;/sup&gt;</th>
<th>Moving Targets&lt;sup&gt;(2)&lt;/sup&gt;</th>
<th>Time Critical Targets&lt;sup&gt;(3)&lt;/sup&gt;</th>
<th>Buried Targets&lt;sup&gt;(4)&lt;/sup&gt;</th>
<th>Adverse Weather&lt;sup&gt;(5)&lt;/sup&gt;</th>
<th>Firepower&lt;sup&gt;(6)&lt;/sup&gt;</th>
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<tbody>
<tr>
<td>Example New Missile</td>
<td>Superior</td>
<td>Good</td>
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<td>AGM-65</td>
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<td>Hellfire / Brimstone / Longbow</td>
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<td>Average</td>
<td>Superior</td>
</tr>
</tbody>
</table>

Note:
- Superior
- Good
- Average
- Poor

---

(1) - Multi-mode warhead desired. GPS / INS provides precision (3 m) accuracy.
(2) - Seeker or high bandwidth data link required for terminal homing.
(3) - High speed with duration required ⇒ High payoff of high speed / loiter and powered submunition.
(4) - KE penetration warhead required ⇒ High impact speed, low drag, high density, long length.
(5) - GPS / INS, SAR seeker, imaging mmW seeker, and data link have high payoff.
(6) - Light weight required. Light weight also provides low cost.
Blast Is Effective at Small Miss Distance

\[ \Delta p / p_0 = \frac{37.95}{z p_0^{1/3}} + \frac{154.9}{(z p_0^{1/3})^2} + \frac{203.4}{(z p_0^{1/3})^3} + \frac{403.9}{(z p_0^{1/3})^4} \]

\[ z = \frac{r}{c^{1/3}} \]

Note:
Based on bare sphere of pentolite (\( E_c^{1/2} = 8,500 \text{ ft/s} \))
\( \Delta p \) = overpressure at distance \( r \) from explosion
\( p_0 \) = undisturbed atmospheric pressure, psi
\( z \) = scaling parameter = \( r / c^{1/3} \)
\( r \) = distance from center of explosion, ft
\( c \) = explosive weight, lb

Example for Rocket Baseline Warhead:
\( c = 38.8 \text{ lb} \)
\( h = 20k \text{ feet}, p_0 = 6.8 \text{ psi} \)
\( r = 10 \text{ ft} \)
\( z = \frac{10}{(38.8)^{1/3}} = 2.95 \)
\( z p_0^{1/3} = 5.58 \)
\( \Delta p / p_0 = 13.36 \)
\( \Delta p = 90 \text{ psi} \)

Guidance Accuracy Enhances Lethality

Rocket Baseline Warhead Against Typical Aircraft Target

\[
P_K > 0.5 \text{ if } \sigma < 5 \text{ ft ( } \Delta p > 330 \text{ psi, fragments impact energy } > 130k \text{ ft-lb/ft}^2 )
\]

\[
P_K > 0.1 \text{ if } \sigma < 25 \text{ ft ( } \Delta p > 24 \text{ psi, fragments impact energy } > 5k \text{ ft-lb/ft}^2 )
\]

Note: Rocket Baseline 77.7 lb warhead
\[C / M = 1, \text{ spherical blast, } h = 20k \text{ ft.}\]

Video of AIM-9X Flight Test Missile Impact on Target
( No Warhead )
Warhead Blast and Fragments Are Effective at Small Miss Distance

Hellfire 24 lb shaped charge warhead fragments are from natural fragmenting case

Roland 9 kg explosively formed warhead multi-projectiles are from preformed case

Video of Guided MLRS 180 lb blast fragmentation warhead
Maximum Total Fragment Kinetic Energy
Requires High Charge-to-Metal Ratio

\[ KE = \left( \frac{1}{2} \right) M_m V_f^2 = E_c M_c / \left( 1 + 0.5 \frac{M_c}{M_m} \right) \]

Note:
Based on Gurney Equation
Cylindrical Warhead
KE = Total Kinetic Energy
\( M_m \) = Total Mass Metal Fragments
\( V_f \) = Fragment Velocity
\( E_c \) = Energy Per Unit Mass Charge
\( M_c \) = Mass of Charge
\( M_{wh} \) = Mass of Warhead = \( M_m + M_c \)

Example:
Rocket Baseline Warhead
\( M_c = 1.207 \text{ slug} \)
\( M_m = 1.207 \text{ slug} \)
\( \frac{M_c}{M_m} = 1 \)

\[ E_c M_c = 52,300,000 \times 1.207 = 63,100,000 \text{ ft-lb} \]
\[ KE = \frac{63100000}{1 + 0.5 \times 1} \]
\[ = 42,100,000 \text{ ft-lb} \]

Multiple Impacts Are Effective Against Threat Vulnerable Area Subsystems

\[ P_K = 1 - \left(1 - \frac{A_v}{A_{tp}}\right)^{n_{hits}} \]

Note:
- \(A_v\) = Target vulnerable area
- \(A_{tp}\) = Target presented area

Example:
- If \(A_v / A_p = 0.1\), \(n_{hits} = 22\) gives \(P_K = 0.9\)
- If \(A_v / A_p = 0.9\), \(n_{hits} = 1\) gives \(P_K = 0.9\)
Small Miss Distance Improves Number of Warhead Fragment Hits

\[ n_{\text{hits}} = n_{\text{fragments}} \left[ \frac{A_p}{4\pi\sigma^2} \right] \]

- **Wwh = 5 lb**
- **Wwh = 50 lb**
- **Wwh = 500 lb**

Example for Rocket Baseline:

\[ W_{WH} = 77.7 \text{ lb} \]
\[ M_c / M_m = 1, \ W_m = 38.8 \text{ lb} = 17,615 \text{ g} \]
Average fragment weight = 3.2 g
\[ n_{\text{fragments}} = 17615 / 3.2 = 5505 \]
\[ A_p = \text{Target presented area} = 20 \text{ ft}^2 \]
\[ \sigma = \text{Miss distance} = 25 \text{ ft} \]
\[ n_{\text{hits}} = 5505 \left\{ \frac{20}{(4\pi)(25)^2} \right\} = 14 \]

Kinetic energy per square foot. = KE
\[ / \left( 4\pi\sigma^2 \right) = 42100000 / \left[ 4\pi(25)^2 \right] = 5,360 \text{ ft-lb / ft}^2 \]

Note:
- Spherical blast with uniformly distributed fragments
- \[ n_{\text{hits}} = n_{\text{fragments}} \left[ \frac{A_p}{4\pi\sigma^2} \right] \]
- Warhead charge / metal weight = 1
- Average fragment weight = 50 grains ( 3.2 g )
- \[ A_p = \text{Target presented area} = 20 \text{ ft}^2 \]
High Fragment Velocity Requires High Charge-to-Metal Ratio

\[ V_f = \left( 2E_c \right)^{1/2} \left[ \left( \frac{M_c}{M_m} \right) / \left( 1 + 0.5 \frac{M_c}{M_m} \right) \right]^{1/2} \]

Note: Based on the Gurney equation for a cylindrical warhead

Example:
Baseline Rocket Warhead
HMX Explosive
\( \frac{M_c}{M_m} = 1 \)
\( V_f = 8,353 \text{ ft/s} \)

HMX Explosive (\( 2E_c \))^{1/2} = 10,230 ft/s
TNT Explosive (\( 2E_c \))^{1/2} = 7,600 ft/s
\( V_f = \text{Fragment initial velocity, ft/s} \)
\( E_c = \text{Energy per unit mass of charge, ft}^2 / \text{s}^2 \)
\( M_c = \text{Mass of charge} \)
\( M_m = \text{Total mass of all metal fragments} \)
\( M_{wh} = \text{Mass of warhead} = M_m + M_c \)
Note: Typical air-to-air missile warhead
• Fragments initial velocity 5,000 ft/s
• Sea level
• Average fragment weight 3.2 g
• Fewer than 0.3% of the fragments weigh more than 9.7 g for nominal 3.2 g preformed warhead fragments
• Small miss distance gives less reduction in fragment velocity, enhancing penetration
Hypersonic Hit-to-Kill Enhances Energy on Target for Missiles with Small Warheads

\[ \frac{E_T}{E_C} = \left[ \frac{1}{2} \left( \frac{W_{\text{Missile}}}{g_c} \right) V^2 + E_C \left( \frac{W_C}{g_c} \right) \right] / \left[ E_C \left( \frac{W_C}{g_c} \right) \right] \]

Example for Rocket Baseline:
- \( W_{\text{Missile}} = 367 \text{ lb} \)
- \( W_C = 38.8 \text{ lb} \)
- \( W_{\text{Missile}} / W_C = 9.46 \)
- \( V = 2,000 \text{ ft} / \text{s} \)
- \( (1/2) \left( \frac{W_{\text{Missile}}}{g_c} \right) V^2 = 22.8 \times 10^6 \text{ ft-lb} \)
- \( E_C \left( \frac{W_C}{g_c} \right) = 63.1 \times 10^6 \text{ ft-lb} \)
- \( E_T / E_C = 1.36 \)

Note: Warhead explosive charge energy based on HMX, \( (2E_C)^{1/2} = 10,230 \text{ ft} / \text{s} \).

1 kg weight at Mach 3 closing velocity has kinetic energy of 391,000 J \( \Rightarrow \) equivalent chemical energy of 0.4 lb TNT.
Kinetic Energy Warhead Density, Length, and Velocity Provide Enhanced Penetration

\[
P / d = [(l / d) - 1] \left( \frac{\rho_p}{\rho_T} \right)^{1/2} + 3.67 \left( \frac{\rho_p}{\rho_T} \right)^{2/3} \left[ \frac{\rho_T V^2}{\sigma_T} \right]^{1/3}
\]

Note:
- \( V > 1,000 \text{ ft/s} \)
- \( l / d > 2 \)
- Non-deforming (high strength, sharp nose) penetrator
- \( l = \) Penetrator length
- \( d = \) Penetrator diameter
- \( V = \) Impact velocity
- \( \rho_p = \) Penetrator density
- \( \rho_T = \) Target density
- \( \sigma_T = \) Target ultimate stress

Example for 250 lb Steel Penetrator
- \( l / d = 10 \)
- \( l = 48 \text{ in (4 ft)} \)
- \( d = 4.8 \text{ in (0.4 ft)} \)
- Concrete target
- \( \rho_p = 0.283 \text{ lb/in}^3 (15.19 \text{ slug/ft}^3) \)
- \( \rho_T = 0.075 \text{ lb/in}^3 (4.02 \text{ slug/ft}^3) \)
- \( V = 4,000 \text{ ft/s} \)
- \( \sigma_T = 5,000 \text{ psi (720,000 psf)} \)
- \( P / d = [10 - 1] \left( \frac{15.19}{4.02} \right)^{1/2} + 3.67 \left( \frac{15.19}{4.02} \right)^{2/3} \left[ \frac{4.02 (4000)^2}{720000} \right]^{1/3} = 57.3 \)
- \( P = (57.3) (0.4) = 22.9 \text{ ft} \)

Examples of Kinetic-Kill Missiles

- Standard Missile 3 (NTW)
- PAC-3
- THAAD
- LOSAT
- LOSAT Video

Source: U.S. Army
**CEP Approximately Equal to $1 \sigma$ Miss Distance**

For a normal distribution of error:
- Probability $< 1\sigma$ miss distance = 0.68
- Probability $< 2\sigma$ miss distance = 0.95
- Probability $< 3\sigma$ miss distance = 0.997

A Collision Intercept Has Constant Bearing for a Constant Velocity, Non-maneuvering Target

Example of Miss
( Line-of-Sight Angle Diverging )
( Line-of-Sight Angle Rate \( L' \neq 0 \) )

Example of Collision Intercept
( Line-of-Sight Angle Constant )
( Line-of-Sight Angle Rate \( L' = 0 \) )

Note: \( L = \) Missile Lead
\( A = \) Target Aspect
A Maneuvering Target and Initial Heading Error Cause Miss

Maneuvering Target:

\[ \tau \frac{d^2 Z}{dt^2} + \frac{dZ}{dt} + N' \frac{Z}{t_o - t} = -N' \frac{\cos A}{\cos L} \frac{1}{2} \alpha_T t^2 \]

Initial Heading Error:

\[ \tau \frac{d^2 Z}{dt^2} + \frac{dZ}{dt} + N' \frac{Z}{t_o - t} = -V_M \gamma_{M_0} \]

Note: \( t_o - t = 0 \) at intercept, causing discontinuity in above equations.

- \( N' = \) Effective navigation ratio = \( N \left[ \frac{V_M}{V_M - V_T \cos A} \right] \)
- \( N = \) Navigation ratio = \( \frac{d\gamma}{dt} / \frac{dL}{dt} \)
- \( \tau = \) Missile time constant, \( V_M = \) Velocity of missile, \( \gamma_{M_0} = \) Initial flight path angle error of missile, \( t_o = \) Total time of flight, \( \alpha_T = \) Acceleration of target, \( V_T = \) Velocity of target

Reference: Jerger, J.J., Systems Preliminary Design
Missile Time Constant Causes Miss Distance

- $\tau$ is a measure of missile ability to respond to target condition changes
- $\tau$ equals elapsed time from input of target return until missile has completed 63% or $(1 - e^{-1})$ of corrective maneuver ($t = \tau$)
- $\tau$ also called “rise time”
- Contributions to time constant $\tau$:
  - Control effectiveness ($\tau_\delta$)
  - Control dynamics (e.g., actuator rate) ($\tau_\delta^*$)
  - Dome error slope ($\tau_{\text{Dome}}$)
  - Guidance and control filters ($\tau_{\text{Filter}}$)
  - Other G&C dynamics (gyro dynamics, accelerometer, processor latency, etc)
  - Seeker errors (resolution, latency, blind range, tracking, noise, glint, amplitude)
- Approach to estimate $\tau$:
  - $\tau = \tau_\delta + \tau_\delta^* + \tau_{\text{Dome}}$

Example for Rocket Baseline:
$M = 2$, $h = 20k$ ft, coast
$\tau = \tau_\delta + \tau_\delta^* + \tau_{\text{Dome}}$
$= 0.096 + 0.070 + 0.043 = 0.209$ s
Assumptions for $\tau_\delta$
- Control surface deflection limited
- Near neutral stability

Equation of motion is
$\alpha = \left[ \rho \frac{V^2 S d C_{m_\delta}}{2 I_y} \right] \delta_{\text{Max}}$

Integrate to solve for $\alpha_{\text{Max}}$
$\alpha_{\text{Max}} = \left[ \rho \frac{V^2 S_{\text{Ref}} d C_{m_\delta}}{8 I_y} \right] \delta_{\text{Max}}^2$

$\tau_\delta$ is given by
$\tau_\delta = \left[ 8 I_y \left( \frac{\alpha_{\text{Max}}}{\delta_{\text{Max}}} \right) / \left( \rho \frac{V^2 S_{\text{Ref}} d C_{m_\delta}}{} \right) \right]^{1/2}$

Contributors to small $\tau_\delta$
- Low fineness (small $I_y / (S_{\text{Ref}} d)$)
- High dynamic pressure (low altitude / high speed)
- Large $C_{m_\delta}$

Example for Rocket Baseline:
$W = 367 \text{ lb}$, $d = 0.667 \text{ ft}$, $S_{\text{Ref}} = 0.349 \text{ ft}^2$, $I_y = 94.0 \text{ slug-ft}^2$,
$M = 2$, $h = 20k \text{ ft}$ ($\rho = 0.001267 \text{ slug/ft}^3$),
$\alpha_{\text{Max}} = 9.4 \text{ deg}$, $\delta_{\text{Max}} = 12.6 \text{ deg}$, $C_{m_\delta} = 51.6 \text{ per rad}$,
$\tau_\delta = \left\{ 8 \left( 94.0 \right) \left( 9.4 / 12.6 \right) / \left[ 0.001267 \left( 2074 \right)^2 \left( 0.349 \right) \right. \right.$
$\left. \left( 0.667 \right) \left( 51.6 \right) \right]^{1/2} = 0.096 \text{ s}$
Assumptions for $\tau_{\delta}$:
- Control surface rate limited ($\dot{\delta} = \dot{\delta}_{\text{Max}}$)
- Near neutral stability

Equation of motion for $\dot{\delta} = \pm \dot{\delta}_{\text{Max}}$
- $\alpha^\ddot{} = [\rho V^2 S d C_{m\delta} / (2 I_y)] \dot{\delta}_{\text{Max}}$

Equation of motion for "perfect" response $\dot{\delta} = \infty$, $\delta = \delta_{\text{Max}}$
- $\alpha^\ddot{} = [\rho V^2 S d C_{m\delta} / (2 I_y)] \delta_{\text{Max}}$

$\tau_{\delta}$ is difference between actual response to $\alpha_{\text{Max}}$ and "perfect" ($\tau_{\delta}$) response

Then
- $\tau_{\delta} = 2 \delta_{\text{Max}} / \dot{\delta}_{\text{Max}}$

Example for Rocket Baseline
- $\delta_{\text{Max}} = 360 \text{ deg/s}$, $\delta_{\text{Max}} = 12.6 \text{ deg}$
- $\tau_{\delta} = 2 (12.6 / 360) = 0.070 \text{ s}$

Note:
- $\tau_{\delta}$
  - Response for control rate limit
  - Response for no control rate limit
**Time Constant \( \tau_{Dome} \) for Radome Is Driven by Dome Error Slope**

- \(| R | = 0.05 \left( \frac{L_N}{d} - 0.5 \right) \left[ 1 + 15 \left( \frac{\Delta f}{f} \right) \right] / \left( \frac{d}{\lambda} \right)\)
- \(\tau_{Dome} = N' \left( \frac{V_C}{V_M} \right) \frac{\alpha}{\gamma'} \)
- \(\alpha / \gamma' = \alpha \left( \frac{W}{g_c} \right) \frac{V_M}{\left\{ q S_{Ref} \left[ C_{N\alpha} + C_{N\delta} / \left( \frac{\alpha}{\delta} \right) \right] \right\}}\)
- Substituting gives \(\tau_{Dome} = N' W V_C \frac{\alpha}{\gamma'} / \left\{ g_c q S_{Ref} \left[ C_{N\alpha} + C_{N\delta} / \left( \frac{\alpha}{\delta} \right) \right] \right\}\)

**Example for Rocket Baseline at M = 2, h = 20k ft, q = 2725 psf**

Assume \(V_T = 1,000 \text{ ft/s}, \) giving \(V_C = 3,074 \text{ ft/s}\)

Assume \(N' = 4, f = 10 \text{ GHz}, \) or \(\lambda = 1.18 \text{ in}, \Delta f / f = 0.02\)

Configuration data are \(l_N/ d = 2.4, d = 8 \text{ in}, S_{Ref} = 0.349 \text{ ft}^2, W = 367 \text{ lb}, \) \(C_{N\alpha} = 40 \text{ per rad}, \) \(C_{N\delta} = 15.5 \text{ per rad}, \) \(\alpha / \delta = 0.75\)

Compute \(| R | = 0.05 \left( 2.4 - 0.5 \right) \left[ 1 + 15 \left( 0.02 \right) \right] / \left( 8 / 1.18 \right) = 0.0182 \text{ deg/deg}\)

\(\tau_{Dome} = 4 \left( 367 \right) \left( 3074 \right) \left( 0.0182 \right) / \left[ 32.2 \left( 2725 \right) \left( 0.349 \right) \left( 40 + 15.5 / 0.75 \right) \right] = 0.043 \text{ s}\)
High Initial Acceleration Is Required to Eliminate a Heading Error

\[ \frac{a_M t_0}{V_M \gamma_M} = N' \left( 1 - \frac{t}{t_0} \right)^{N' - 2} \]

Example: Exoatmospheric Head-on Intercept, \( N' = 4 \)

Midcourse lateral error at \( t = 0 \) (seeker lock-on) = 200 m, 1σ

\[ R_{\text{lock-on}} = 20000 \text{ m} \Rightarrow \gamma_M = 200 / 20000 = 0.0100 \text{ rad} \]

\[ V_M = 5000 \text{ m/s}, V_r = 5000 \Rightarrow t_0 = R_{\text{lock-on}} / (V_M + V_r) = 20000 / (5000 + 5000) = 2.00 \text{ s} \]

\[ \frac{a_M t_0}{V_M \gamma_M} = 4 \]

\[ a_M = 4 \times (5000 \times 0.0100 / 2.00) = 100 \text{ m/s}^2 \]

\[ n_M = 100 / 9.81 = 10.2 \text{ g} \]
Missile Minimum Range May Be Driven By 4 to 8 Time Constants to Correct Initial Heading Error

\[ \sigma_{HE} = V_M \gamma_M t_0 e^{t_0/\tau} \sum_{j=1}^{N'-1} \left\{ \left( \frac{N' - 1}{j} \right)! \left( \frac{N' - 2}{j - 1} \right)! \right\} \]

If \( N' = 3 \), \( \sigma_{HE} = V_M \gamma_M t_0 e^{t_0/\tau} \left[ 2 \left( t_0 / \tau \right) - ( t_0 / \tau )^2 / 2 \right] \)

If \( N' = 4 \), \( \sigma_{HE} = V_M \gamma_M t_0 e^{t_0/\tau} \left[ 3 \left( t_0 / \tau \right) - 2 \left( t_0 / \tau \right)^2 + \left( t_0 / \tau \right)^3 / 6 \right] \)

If \( N' = 5 \), \( \sigma_{HE} = V_M \gamma_M t_0 e^{t_0/\tau} \left[ \left( t_0 / \tau \right) - ( 3 / 2 ) \left( t_0 / \tau \right)^2 + \left( t_0 / \tau \right)^3 / 2 - \left( t_0 / \tau \right)^4 / 24 \right] \)

If \( N' = 6 \), \( \sigma_{HE} = V_M \gamma_M t_0 e^{t_0/\tau} \left[ \left( t_0 / \tau \right) - 2 \left( t_0 / \tau \right)^2 + \left( t_0 / \tau \right)^3 - \left( t_0 / \tau \right)^4 / 6 + \left( t_0 / \tau \right)^5 / 120 \right] \)

Note: Proportional Guidance

\( (\sigma_{HE})_{\text{Max}} \) shown in figure is the envelope of adjoint solution

\( (\sigma_{HE})_{\text{Max}} = \text{Max miss distance} (1 \sigma) \) from heading error, ft

\( V_M = \text{Velocity of missile}, \text{ft} / \text{s} \)

\( \gamma_M = \text{Initial heading error}, \text{rad} \)

\( t_0 = \text{Total time to correct heading error}, \text{s} \)

\( \tau = \text{Missile time constant}, \text{s} \)

\( N' = \text{Effective navigation ratio} \)

Example: Ground Target, \( N' = 4, \tau = 0.2, \) GPS / INS error = 3 m, \( R_{\text{lock-off}} = 125 \text{ m}, \gamma_M = 3 / 125 = 0.024 \text{ rad}, V_M = 300 \text{ m} / \text{s}, t_0 = 125 / 300 = 0.42 \text{ s} \)

\[ t_0 / \tau = 0.42 / 0.2 = 2.1, (\sigma_{HE})_{\text{Max}} / (V_M \gamma_M t_0) = 0.12 \]

\[ (\sigma_{HE})_{\text{Max}} = 0.12 (300)(0.024)(0.42) = 2.2 \text{ m} \]

References:


Required Maneuverability Is about 3x the Target Maneuverability for an Ideal (τ = 0) Missile

\[
\frac{n_M}{n_T} = \left[ \frac{N'}{N' - 2} \right] \left[ 1 - (1 - t/t_0)^{N' - 2} \right]
\]

Where
- \(t\) = Elapsed Time
- \(t_0\) = Time to Target
- \(N'\) = Effective Navigation Ratio

Assumptions:
- \(\tau = 0\)
- \(V_M > V_T\)

Example:
- \(\tau = 0, N' = 3, t/t_0 = 1\)
- \(\Rightarrow n_M/n_T = 3\)
Target Maneuvers Require 6 to 10 Time Constants to Settle Out Miss Distance

\[ \sigma_{\text{MAN}} = g_c n_T \tau^2 e^{(t_0 / \tau)} \sum_{j=2}^{N'-1} \left\{ (N' - 3)! \left[ - \frac{(t_0 / \tau)^j}{j!} \right] / \left( (j - 2)! (N' - j - 1)! j! \right) \right\} \]

If \( N' = 3 \), \( \sigma_{\text{MAN}} = g_c n_T \tau^2 e^{(t_0 / \tau)} \left[ (t_0 / \tau)^2 / 2 \right] \)

If \( N' = 4 \), \( \sigma_{\text{MAN}} = g_c n_T \tau^2 e^{(t_0 / \tau)} \left[ (t_0 / \tau)^2 / 2 - (t_0 / \tau)^3 / 6 \right] \)

If \( N' = 5 \), \( \sigma_{\text{MAN}} = g_c n_T \tau^2 e^{(t_0 / \tau)} \left[ (t_0 / \tau)^2 / 2 - (t_0 / \tau)^3 / 3 + (t_0 / \tau)^4 / 24 \right] \)

If \( N' = 6 \), \( \sigma_{\text{MAN}} = g_c n_T \tau^2 e^{(t_0 / \tau)} \left[ (t_0 / \tau)^2 / 2 - (t_0 / \tau)^3 / 2 + (t_0 / \tau)^4 / 8 - (t_0 / \tau)^5 / 120 \right] \)

Note: Proportional Guidance

\( \left( \frac{\sigma_{\text{MAN}}}{g_c n_T \tau^2} \right)_{\text{Max}} \) is the envelope of adjoint solution

\( \left( \frac{\sigma_{\text{MAN}}}{g_c n_T \tau^2} \right)_{\text{Max}} = \text{Max miss (1} \sigma) \) from target accel, ft

\( n_T = \text{Target acceleration, g} \)

\( g_c = \text{Gravitation constant, 32.2} \)

\( \tau = \text{Missile time constant, s} \)

\( N' = \text{Effective navigation ratio} \)

\( \tau_0 = \text{Time of flight, s} \)

References:


2/24/2008

ELF
An Aero Control Missile Has Reduced Miss Distance at Low Altitude / High Dynamic Pressure

(\sigma_{\text{Man}})_{\text{Max}} = 0.13 g_c n_T \tau^2 @ N' = 4, t_0 / \tau = 2

Note: Proportional guidance
Target maneuver initiated for max miss (t_0 / \tau = 2)
(\sigma_{\text{Man}})_{\text{Max}} in figure = Envelope of adjoint miss distance
\tau = Missile time constant, s
N' = Effective navigation ratio = 4
n_T = Target acceleration, g
g_c = Gravitation constant = 32.2

Example for Rocket Baseline at Mach 2, coasting
Assume:
• n_T = 5g, V_T = 1,000 ft / s, head-on intercept
• h = 20k ft \Rightarrow \tau = 0.209 s
(\sigma_{\text{Man}})_{\text{Max}} = 0.13 (32.2)(5)(0.209)^2 = 0.9 ft
• h = 80k ft \Rightarrow \tau = 1.17 s
(\sigma_{\text{Man}})_{\text{Max}} = 0.13 (32.2)(5)(1.17)^2 = 28.7 ft
Glint Miss Distance Driven by Seeker Resolution, Missile Time Constant, and Navigation Ratio

\[ \sigma_{\text{Glint}} = K_{N'} \left( \frac{W}{\tau} \right)^{1/2} \]
\[ K_{N'} = 0.5 \left( 2 K_{N'} = 4 \right)^{N'/4} \]
\[ K_{N'} = 4 = 1.206 \]
\[ W = \left( b_T \right)_{\text{Res}}^2 / \left( 3 \pi^2 B \right) \]

Note:
- Proportional guidance
- Adjoint miss distance
- \( \sigma_{\text{Glint}} \) = Miss distance due to glint noise, ft
- \( W \) = Glint noise spectral density, ft^2 / Hz
- \( \tau \) = Missile time constant, s
- \( N' \) = Effective navigation ratio
- \( (b_T)_{\text{Res}} \) = Target span resolution at seeker blind range, ft
- \( B \) = Noise bandwidth, Hz (1 < B < 5 Hz)

Example: Rocket Baseline at Mach 2, \( h = 20k \) ft altitude \( \Rightarrow \tau = 0.209 \) s

Assume:
- \( N' = 4 \)
- \( B = 2 \) Hz
- \( (b_T)_{\text{Res}} = b_R = 40 \) ft (radar seeker beam width resolution of target wing span)

Calculate:
\[ W = \left( 40 \right)^2 / \left[ 3 \pi^2 (2) \right] = 27.0 \text{ ft}^2 / \text{Hz} \]
\[ \sigma_{\text{Glint}} / (b_T)_{\text{Res}} = K_{N'} \left( W / \tau \right)^{1/2} / (b_T)_{\text{Res}} \]
\[ = 1.206 \left( 27.0 / 0.209 \right)^{1/2} / 40 = 0.343 \]
\[ \sigma_{\text{Glint}} = 0.343 \left( 40 \right) = 13.7 \text{ ft} \]

Reference:

2/24/2008
Minimizing Miss Distance with Glint Requires Optimum Time Constant and Navigation Ratio

\[ \sigma = \left[ \left( \sigma_{\text{MAN}} \right)_{\text{Max}}^2 + \left( \sigma_{\text{Glint}} \right)^2 \right]^{1/2} \]

Note:
- Proportional guidance
- Adjoint miss distance
- \( (\sigma_{\text{MAN}})_{\text{Max}} \) = Max miss distance from target maneuver, ft
- \( \sigma_{\text{Glint}} \) = Miss distance from glint noise, ft
- \( \tau \) = Missile time constant, s
- \( N' \) = Effective navigation ratio

Example for Rocket Baseline at Mach 2, \( h = 20k \text{ ft altitude} \Rightarrow \tau = 0.209 \text{ s} \)

Assume:
- \( N' = 4 \)
- \( B = 2 \text{ Hz} \)
- \( (b_T)_{\text{Res}} = 40 \text{ ft} \)
- \( n_T = 5g, V_T = 1,000 \text{ ft} / \text{s}, \text{ Head-on} \)

From prior figures:
- \( (\sigma_{\text{MAN}})_{\text{Max}} = 0.9 \text{ ft}, \sigma_{\text{Glint}} = 13.7 \text{ ft} \)

Calculate:
- \( \sigma = \left[ \left( \sigma_{\text{MAN}} \right)_{\text{Max}}^2 + \left( \sigma_{\text{Glint}} \right)^2 \right]^{1/2} = 13.7 \text{ ft} \)

Reference:

2/24/2008

ELF
Missile Carriage RCS and Launch Plume Are Considerations in Launch Platform Observables

- **Missile Carriage Alternatives**
  - Internal Carriage: Lowest Carriage RCS
  - Conformal Carriage: Low Carriage RCS
  - Conventional External Pylon or External Rail Carriage: High Carriage RCS

- **Plume Alternatives**
  - Min Smoke: Lowest Launch Observables (H₂O Contrail)
  - Reduced Smoke: Reduced Launch Observables (e.g., HCl Contrail from AP Oxidizer)
  - High Smoke: High Launch Observables (e.g., Al₂O₃ Smoke from Al Fuel)
Examples of Weapon Bay Internal Carriage and Load-out

Center Weapon Bay Best for Ejection Launchers

F-22 Bay Loadout: 3 AIM-120C, 1 GBU-32
F-117 Bay Loadout: 1 GBU-27, 1 GBU-10
B-1 Bay Loadout: 8 AGM-69

Side Weapon Bay Best for Rail Launchers

AMRAAM Loading in F-22 Bay
F-22 Side Bay: 1 AIM-9 Each Side Bay
RAH-66 Side Bay: 1 AGM-114, 2 FIM-92, 4 Hydra 70 Each Side Bay

Video
Minimum Smoke Propellant Has Low Observables

High Smoke Example: AIM-7
Particles (e.g., metal fuel) at all atmospheric temperature.

Reduced Smoke Example: AIM-120
Contrail (HCl from AP oxidizer) at T < -10° F atmospheric temperature.

Minimum Smoke Example: Javelin
Contrail (H₂O) at T < -35° F atmospheric temperature.
High Altitude Flight and Low RCS Enhance Survivability

\[ P_t = \left( \frac{4 \pi}{3} \right) P_r R^4 \left( \frac{G_t G_r \sigma}{\lambda^2} \right) \]

Note:
- Range Slant Angle = 20 deg, G_t = Transmitter Gain = 40 dB, G_r = Receiver Gain = 40 dB, \( \lambda = \) Wavelength = 0.03 m, \( P_r = \) Receiver Sensitivity = 10^{-14} W, \( \sigma = \) radar cross section (RCS)
- Based on Radar Range Equation with \( (S/N)_{\text{Detect}} = 1 \) and Unobstructed Line-of-Sight

Example for \( P_t = 50,000 \) W:
- Not detected if:
  - \( h > 25k \) ft for \( \sigma = 0.001 \) m^2
  - \( h > 77k \) ft for \( \sigma = 0.1 \) m^2

\[ Pt, \text{ Radar Transmitted Power Required for Detection, W} \]
\[ h, \text{ Geometric Altitude, kft} \]
Mission Planning and High Speed Enhance Survivability

\[ t_{\text{exp}} = 2 \left( \frac{R_{\text{max}}}{V} \right) \cos \left[ \sin^{-1} \left( \frac{y_{\text{offset}}}{R_{\text{max}}} \right) \right] - t_{\text{react}} \]

Note: Based on assumption of constant altitude, constant heading flyby of threat SAM site with an unobstructed line-of-sight. \( t_{\text{exp}} \) = exposure time to SAM threat, \( R_{\text{max}} \) = max detection range by SAM threat, \( V \) = flyby velocity, \( y_{\text{offset}} \) = flyby offset, \( t_{\text{react}} \) = SAM site reaction time from detection to launch.

**Example:**

- \( y_{\text{offset}} = 7 \text{ nm}, R_{\text{max}} = 10 \text{ nm} = 60750 \text{ ft}, \) \( y_{\text{offset}} / R_{\text{max}} = 0.7, t_{\text{react}} = 15 \text{ s} \)
  - If \( V = 1000 \text{ ft} / \text{s}, t_{\text{react}} \left( \frac{V}{R_{\text{max}}} \right) = 0.247 \)
  - \( t_{\text{exp}} \left( \frac{V}{R_{\text{max}}} \right) = 2 \cos \left[ \sin^{-1} \left( \frac{7}{10} \right) \right] - 15 \left( \frac{1000}{60746} \right) = 1.428 - 0.247 = 1.181 \)
  - \( t_{\text{exp}} = 1.181 \left( \frac{60746}{1000} \right) = 71.7 \text{ s} \)

- If \( V = 4000 \text{ ft} / \text{s}, t_{\text{react}} \left( \frac{V}{R_{\text{max}}} \right) = 0.988 \)
  - \( t_{\text{exp}} \left( \frac{V}{R_{\text{max}}} \right) = 0.440 \)
  - \( t_{\text{exp}} = 0.440 \left( \frac{60746}{4000} \right) = 6.7 \text{ s} \)
Low Altitude Flight and Terrain Obstacles
Provide Masking from Threat

\[ h_{\text{mask}} = (h_{\text{mask}})_{\text{obstacle}} + (h_{\text{mask}})_{\text{earth}} = h_{\text{obstacle}} \left( \frac{R_{\text{los}}}{R_{\text{obstacle}}} \right) + \left( \frac{R_{\text{los}}}{7113} \right)^2 \]

**Example:**

- \( h_{\text{obstacle}} = 200 \) ft
- \( R_{\text{obstacle}} = 5.0 \) nm = 30395 ft
- \( R_{\text{los}} = 10.0 \) nm = 60790 ft
- \( h_{\text{mask}} = 200 \left( \frac{60790}{30395} \right) + \left( \frac{60790}{7113} \right)^2 = 400 + 73 = 473 \) ft above terrain

- Height of low hill or tall tree ≈ 100 ft
- Height of moderate hill ≈ 200 ft
- Height of high hill ≈ 500 ft
- Height of low mountain ≈ 1000 ft
Insensitive Munitions Improve Launch Platform Survivability

- Critical Subsystems
  - Rocket motor or fuel tank
  - Warhead

- Severity Concerns Ranking of Power Output - Type
  1. Detonation (≈ 0.000002 s rise time)
  2. Partial detonation (≈ 0.0001 s rise time)
  3. Explosion (≈ 0.001 s rise time)
  4. Deflagration or propulsion rise time (≈ 0.1 s rise time)
  5. Burning (> 1 s)

- Design and test considerations (MIL STD 2105C)
  - Fragment / bullet impact or blast
  - Sympathetic detonation
  - Fast / slow cook-off fire
  - Drop
  - Temperature
  - Vibration
  - Carrier landing (18 ft/s sink rate)
High System Reliability Is Provided by High Subsystem Reliability and Low Parts Count

Typical System Reliability

\[ R_{\text{system}} \approx 0.94 = R_{\text{Arm}} \times R_{\text{Launch}} \times R_{\text{Struct}} \times R_{\text{Auto}} \times R_{\text{Act}} \times R_{\text{Seeker}} \times R_{\text{In Guid}} \times R_{\text{PS}} \times R_{\text{Prop}} \times R_{\text{Fuze}} \times R_{\text{W/H}} \]

Typical Event / Subsystem

- Arm (0.995 – 0.999)
- Launch (0.990 – 0.995)
- Structure (0.997 – 0.999)
- Autopilot (0.993 – 0.995)
- Actuators (0.990 – 0.995)
- Seeker (0.985 – 0.990)
- Inertial Guidance (0.995 – 0.999)
- Power Supply (0.995 – 0.999)
- Propulsion (0.995 – 0.999)
- Fuze (0.987 – 0.995)
- Warhead (0.995 – 0.999)
Sensors, Electronics and Propulsion Drive
Missile Production Cost

Structure
• Rocket • Airbreather

Seeker

Power Supply

Dome

Guidance and Control

Propulsion
• Rocket • Airbreather

Warhead and Fuzing

Aerothermal Insulation

Wings

Data Link

Flight Control

Stabilizers

Cost

Note:
System assembly and test ~ 10% production cost
Propulsion and structure parts count / cost of airbreathing missiles are higher than that of rockets

- Very High ( > 25% Production Cost )
- High ( > 10% )
- Moderate ( > 5% )
- Relatively Low ( < 5% )
Sensors and Electronics Occupy a Large Portion of a High Performance / High Cost Missile.

Example: Derby / R-Darter Missile

Cost Considerations

- Life Cycle
  - System Development and Demonstration (SDD)
  - Production
  - Logistics

- Culture / processes
  - Relative Emphasis of Cost, Performance, Reliability, Organization Structure
  - Relaxed Mil STDs
  - IPPD
  - Profit

- Competition
SDD Cost Is Driven by Schedule Duration and Risk

\[ C_{SDD} = 20,000,000 \times t_{SDD}^{1.90} \] (for \( t_{SDD} \) in years)

Example:
5 year (medium risk) SDD program
\[ C_{SDD} = 20,000,000 \times (5)^{1.90} \]
\[ = 20,000,000 \times 121.55 \]
\[ = 2,431,000,000 \] mill.

Note: SDD required schedule duration depends upon risk. Should not ignore risk in shorter schedule.

-- SDD cost based on 1999 US$
Light Weight Missiles Have Low Unit Production Cost

\[ C_{1000th} \approx 6100 \times W_L^{0.758}, \ (W_L \text{ in lb}) \]

Example:
2,000 unit buy of 100 lb missile:
\[ C_{1000th} \approx 6100 \times W_L^{0.758} = 6100 \times (100)^{0.758} = \$200,000 \]

Cost of 2,000 missiles = 2000 \times (\$200,000) = \$400,000,000

Note:
- Unit production cost based on 1999 US$
Learning Curve and Large Production Reduce Unit Production Cost

\[ C_x = C_{1st} L^{\log_2 x}, \quad C_{2x} = L C_x, \text{ where } C \text{ in U.S. } 99\$ \]

Example:
For a learning curve coefficient of \( L = 80\% \), cost of unit \#1000 is 11\% the cost of the first unit

Contributors to the learning curve include:
- More efficient labor
- Reduced scrap
- Improved processes


Labor intensive learning curve: \( L < 0.8 \)
Machine intensive learning curve: \( L > 0.8 \)
Low Parts Count Reduces Missile Unit Production Cost

Note: Tactical Tomahawk has superior flexibility (e.g., shorter mission planning, in-flight retargeting, BDI / BDA, modular payload) at lower parts count / cost and higher reliability. Enabling technologies for low parts count include: casting, pultrusion / extrusion, centralized electronics, and COTS.
Tactical Missile Culture Is Driven by Rate
Production of Sensors and Electronics

Copperhead Seeker and Electronics Production

Patriot Control Section Production

Video of Hellfire Seeker and Electronics Production
## Logistics Cost Considerations

### Peacetime Logistics Activity
- **Contractor Post-production Engineering**
  - Training Manuals / Tech Data Package
  - Simulation and Software Maintenance
  - Configuration Management
  - Engineering Support
  - System Analysis
  - Launch Platform Integration
  - Requirements Documents
  - Coordinate Suppliers
- **Storage Alternatives**
  - Wooden Round (Protected)
  - Open Round (Humidity, Temp, Corrosion, Shock)
- **Reliability Maintenance**
  - Surveillance
  - Testing
- **Maintenance Alternatives**
  - First level (depot)
  - Two level (depot, field)
- **Disposal**

### Wartime Logistics Activity
- **Deployment Alternatives**
  - Airlift
  - Sealift
- **Combat Logistics**
  - Launch Platform Integration
  - Mission Planning
  - Field Tests
  - Reliability Data
  - Maintainability Data
  - Effectiveness Data
  - Safety Data
Logistics Cost Lower for Simple Missile Systems

Simple: Stinger
More Sophisticated: Hawk and SLAMRAAM
Complex: PAC-3

Very Complex: THAAD

Video of Logistics Alternatives
Logistics Is Simpler for Light Weight Missiles

Support personnel for installation with 50 lb lift limit per person
Support personnel for installation with 100 lb lift limit per person
Machine lift for installation

Support Personnel required for Installation vs. Missile Weight, lb

Predator (21 lb)  Sidewinder (190 lb)  Sparrow (500 lb)  Laser Guided Bomb (2,500 lb)

Video of Simple Logistics for a Light weight Missile
Small MEMS Sensors Can Provide Health Monitoring, Reducing Cost and Weight

- Micro-machined Electro-Mechanical Systems (MEMS)
  - Small size / low cost semiconductor manufacturing process
  - 2,000 to 5,000 sensors on a 5 in silicon wafer

- Wireless (RF) Data Collection and Health Monitoring

- Distributed Sensors Over Missile
  - Stress / strain
  - Vibration
  - Acoustics
  - Temperature
  - Pressure

- Reduced Logistics Cost and Improved Reliability
  - Health monitoring

- Reduced Weight and Production Cost
  - More Efficient Design
Missile Carriage Size, Shape, and Weight Are Driven by Launch Platform Compatibility

<table>
<thead>
<tr>
<th>US Launch Platform</th>
<th>Launcher</th>
<th>Carriage Span / Shape</th>
<th>Length</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Surface Ships</td>
<td>VLS</td>
<td>22” x 22”</td>
<td>263”</td>
<td>3400 lb</td>
</tr>
<tr>
<td>Submarines</td>
<td>CLS</td>
<td>22”</td>
<td>263”</td>
<td>3400 lb</td>
</tr>
<tr>
<td>Fighters / Bombers / UCAVs</td>
<td>Rail / Ejection</td>
<td>~24” x 24”</td>
<td>~168”</td>
<td>~500 lb to 3000 lb</td>
</tr>
<tr>
<td>Ground Vehicles</td>
<td>Launch Pods</td>
<td>~28” x 28”</td>
<td>158”</td>
<td>3700 lb</td>
</tr>
<tr>
<td>Helos</td>
<td>Rail</td>
<td>~13” x 13”</td>
<td>70”</td>
<td>120 lb</td>
</tr>
</tbody>
</table>
Light Weight Missiles Enhance Firepower

Max Strike Weapon Weight

Inboard Asymmetric Bring-Back Load Limit

Outboard Asymmetric Bring-Back Load Limit

Configuration for Day Operation with Bring-Back Load

Configuration for Night Operation with Bring-Back Load
Launch Envelope Limitations in Missile / Launch Platform Physical Integration

- **Off Boresight**
  - Seeker field of regard $\Rightarrow$ potential obscuring from launch platform

- **Minimum Range**
  - Launcher rail clearance and aeroelasticity $\Rightarrow$ miss at min range
  - Helo rotor downwash $\Rightarrow$ miss at min range

- **Safety**
  - Launcher retention $\Rightarrow$ potential inadvertent release, potential hang-fire
  - Launch platform local flow field $\alpha$, $\beta$ $\Rightarrow$ potential unsafe separation
  - Launch platform maneuvering $\Rightarrow$ potential unsafe separation
  - Handling qualities with stores $\Rightarrow$ potential unsafe handling qualities
  - Launch platform bay / canister acoustics $\Rightarrow$ missile factor of safety
  - Launch platform bay / canister vibration $\Rightarrow$ missile factor of safety
Store Separation Wind Tunnel Tests Are Required for Missile / Aircraft Compatibility

F-18 Store Compatibility Test in AEDC 16T

AV-8 Store Compatibility Test in AEDC 4T

Types of Wind Tunnel Testing for Store Compatibility
- Flow field mapping with probe
- Flow field mapping with store
- Captive trajectory simulation
- Drop testing

Example Stores with Flow Field Interaction: Kh-41 + AA-10
Examples of Rail Launched and Ejection Launched Missiles

Example Rail Launcher: Hellfire / Brimstone

Example Ejection Launcher: AGM-86 ALCM

Video of Hellfire / Brimstone Carriage / Launch

Video of AGM-86 Carriage / Launch
Examples of Safe Store Separation

- AMRAAM Rail Launch from F-16
- Laser Guided Bombs Drop from F-117
- Video of Rapid Drop (16 Bombs) from B-2
Examples of Store Compatibility Problems

Unsafe Separation
Hang-Fire
Store Aeroelastic Instability
# MIL-STD-8591 Aircraft Store Suspension and Ejection Launcher Requirements

<table>
<thead>
<tr>
<th>Store Weight / Parameter</th>
<th>30 Inch Suspension</th>
<th>14 Inch Suspension</th>
</tr>
</thead>
<tbody>
<tr>
<td>♦ Weight Up to 100 lb</td>
<td>Not Applicable</td>
<td>Yes</td>
</tr>
<tr>
<td>• Lug height (in)</td>
<td></td>
<td>0.75</td>
</tr>
<tr>
<td>• Min ejector area (in x in)</td>
<td></td>
<td>4.0 x 26.0</td>
</tr>
<tr>
<td>♦ Weight 101 to 1,450 lb</td>
<td>Yes</td>
<td>Yes</td>
</tr>
<tr>
<td>• Lug height (in)</td>
<td>1.35</td>
<td>1.00</td>
</tr>
<tr>
<td>• Min lug well (in)</td>
<td>0.515</td>
<td>0.515</td>
</tr>
<tr>
<td>• Min ejector area (in x in)</td>
<td>4.0 x 36.0</td>
<td>4.0 x 26.0</td>
</tr>
<tr>
<td>♦ Weight Over 1,451 lb</td>
<td>Yes</td>
<td>Not Applicable</td>
</tr>
<tr>
<td>• Lug height (in)</td>
<td>1.35</td>
<td></td>
</tr>
<tr>
<td>• Min lug well (in)</td>
<td>1.080</td>
<td></td>
</tr>
<tr>
<td>• Min ejector area (in x in)</td>
<td>4.0 x 36.0</td>
<td></td>
</tr>
</tbody>
</table>

## Ejection Stroke

![Ejection Stroke Diagram]
<table>
<thead>
<tr>
<th>Rail Launcher</th>
<th>Forward Hanger</th>
<th>Aft Hanger</th>
</tr>
</thead>
<tbody>
<tr>
<td>LAU-7 Sidewinder Launcher</td>
<td>2.260</td>
<td>2.260</td>
</tr>
<tr>
<td>LAU 117 Maverick Launcher</td>
<td>1.14</td>
<td>7.23</td>
</tr>
</tbody>
</table>

Note: Dimensions in inches.

- LAU 7 rail launched store weight and diameter limits are ≤ 300 lb, ≤ 7 in
- LAU 117 rail launched store weight and diameter limits are ≤ 600 lb, ≤ 10 in
Compressed Carriage Missiles Provide Higher Firepower

Baseline AIM-120B AMRAAM

Compressed Carriage AIM-120C AMRAAM (Reduced Span Wing / Tail)

Baseline AMRAAM: Loadout of 2 AMRAAM per F-22 Semi-Bay

Compressed Carriage AMRAAM: Loadout of 3 AMRAAM per F-22 Semi-Bay

17.5 in 17.5 in

12.5 in 12.5 in 12.5 in

Video of Longshot Kit on CBU-97

Note: Alternative approaches to compressed carriage include surfaces with small span, folded surfaces, wrap around surfaces, and planar surfaces that extend (e.g., switch blade, Diamond Back, Longshot).
Example of Aircraft Carriage and Fire Control Interfaces

Example: ADM-141 TALD (Tactical Air-Launched Decoy) Carriage and Fire Control Interfaces
Example of Ship Weapon Carriage and Launcher, Mk41 VLS

- 8 Modules / Magazine
- Tomahawk Launch
- 8 Canister Cells / Module
- Standard Missile Launch

Cell Before Firing
Cell After Firing

Canister Cell Hatch
Ship Deck
Missile Cover

Exhaust Hatch
Plenum
Module Gas Management
**Robustness Is Required for Carriage, Shipping, and Storage Environment**

<table>
<thead>
<tr>
<th>Environmental Parameter</th>
<th>Typical Requirement</th>
<th>Video: Ground / Sea Environment</th>
</tr>
</thead>
<tbody>
<tr>
<td>✤ Surface Temperature</td>
<td>-60° F* to 160° F</td>
<td></td>
</tr>
<tr>
<td>✤ Surface Humidity</td>
<td>5% to 100%</td>
<td></td>
</tr>
<tr>
<td>✤ Rain Rate</td>
<td>120 mm / h**</td>
<td></td>
</tr>
<tr>
<td>✤ Surface Wind</td>
<td>100 km / h steady***</td>
<td></td>
</tr>
<tr>
<td></td>
<td>150 km / h gusts****</td>
<td></td>
</tr>
<tr>
<td>✤ Salt fog</td>
<td>3 g / mm² deposited per year</td>
<td></td>
</tr>
<tr>
<td>✤ Vibration</td>
<td>10 g rms at 1,000 Hz: MIL STD 810, 648, 1670A</td>
<td></td>
</tr>
<tr>
<td>✤ Shock</td>
<td>Drop height 0.5 m, half sine wave 100 g / 10 ms: MIL STD 810, 1670A</td>
<td></td>
</tr>
<tr>
<td>✤ Acoustic</td>
<td>160 dB</td>
<td></td>
</tr>
</tbody>
</table>

Note: MIL-HDBK-310 and earlier MIL-STD-210B suggest 1% world-wide climatic extreme typical requirement.

* Lowest recorded temperature = -90° F. 20% probability temperature lower than -60° F during worst month of worst location.

** Highest recorded rain rate = 436 mm / h. 0.5% probability greater than 120 mm / h during worst month of worst location.

*** Highest recorded steady wind = 342 km / h. 1% probability greater than 100 km / h during worst month of worst location.

**** Highest recorded gust = 378 km / h. 1% probability greater than 150 km / h during worst month of worst location.
Summary of Measures of Merit and Launch Platform Integration

- Measures of Merit
  - Robustness
  - Warhead lethality
  - Miss distance
  - Carriage and launch observables
  - Other survivability considerations
  - Reliability
  - Cost

- Launch Platform Integration
  - Firepower, weight, fitment
  - Store separation
  - Launch platform handling qualities, aeroelasticity
  - Hang-fire
  - Vibration
  - Standard launchers
  - Carriage and storage environment

- Discussion / Questions?
- Classroom Exercise ( Appendix A )
Measures of Merit and Launch Platform Integration Problems

1. IR signal attenuation is greater than 100 dB per km through a c_____.
2. GPS / INS enhances seeker lock-on in adverse weather and ground c______.
3. A data link can enhance missile seeker lock-on against a m_____ target.
4. An example of a missile counter-counter measure to flares is an i______ i_____ seeker.
5. Compared to a mid-wave IR seeker, a long wave IR seeker receives more energy from a c___ target.
6. High fineness kinetic energy penetrators are required to defeat b_____ targets.
7. For the same lethality with a blast fragmentation warhead, a small decrease in miss distance allows a large decrease in the required weight of the w______.
8. For a blast / frag warhead, a charge-to-metal ratio of about one is required to achieve a high total fragment k______ e______.
9. A blast fragmentation warhead tradeoff is the number of fragments versus the individual fragment w______.
10. Kinetic energy penetration is a function of the penetrator diameter, length, density, and v______.

11. In proportional homing guidance, the objective is to make the line-of-sight angle rate equal to z____.

12. Aeromechanics contributors to missile time constant are flight control effectiveness, flight control system dynamics, and dome e_____ s____.

13. Miss distance due to heading error is a function of missile navigation ratio, velocity, time to correct the heading error, and the missile t___ c______.

14. A missile must have about t_____ times the maneuverability of the target.

15. Minimizing the miss distance due to radar glint requires a high resolution seeker, an optimum missile time constant and an optimum n________ r____.

16. Weapons on low observable launch platforms use i_______ carriage.

17. Weapons on low observable launch platforms use m______ smoke propellant.

18. For an insensitive munition, burning is preferable to detonation because it releases less p____.
19. Missile system reliability is enhanced by subsystem reliability and low p____ count.

20. High cost subsystems of missiles are sensors, electronics, and p______.

21. Missile SDD cost is driven by the program duration and r___.

22. Missile unit production cost is driven by the number of units produced, learning curve, and w____.

23. First level maintenance is conducted at a d____.

24. A standard launch system for U.S. Navy ships is the V____ L____ S____.

25. Most light weight missiles use rail launchers while most heavy weight missiles use e______ launchers.

26. Higher firepower is provided by c______ carriage.

27. The typical environmental requirement from MIL-HDBK-310 is the _% world-wide climatic extreme.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Sizing Examples

- **Rocket Baseline Missile**
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- **Ramjet Baseline Missile**
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- **Computer Aided Conceptual Design Sizing Tools**

- **Soda Straw Rocket Design, Build, and Fly**
Air-to-Air Engagement Analysis Process and Assumptions

- F-pole range provides kill of head-on threat outside of threat weapon launch range
- Aircraft contrast for typical engagement
  - \( C = 0.18 \)
- Typical visual detection range by target (Required F-pole range)
  - \( R_D = 3.3 \) nm
- Typical altitude and speed of launch aircraft, target aircraft, and missile
  - \( h = 20k \) ft altitude
  - \( V_L = \text{Mach} 0.8 = 820 \) ft / s
  - \( V_T = \text{Mach} 0.8 = 820 \) ft / s
  - \( V_M = 2 \ V_T = 1,640 \) ft / s
**Assumed Air-to-Air Engagement Scenario for Head-on Intercept**

- **t = 0 s (Launch Missile)**
  - Blue Aircraft (820 ft/s)
  - Blue Missile (1640 ft/s)

- **RL = Launch Range = 10.0 nm**

- **t = tf = 24.4 s (Missile Impacts Target)**
  - Blue Aircraft (820 ft/s)
  - Red Aircraft Destroyed

- **RF = Missile Flight Range = 6.7 nm**

- **RF-pole = 3.3 nm**

- Red Aircraft (820 ft/s)

---

**Equations:**

- \( RL = V_M t_f + V_T t_f \)
- \( RF-pole = V_M t_f - V_L t_f \)
Target Contrast and Size Drive Visual Detection and Recognition Range

\[ R_D = 1.15 \left[ \frac{A_p}{C - C_T} \right]^{1/2}, \quad R_D \text{ in nm, } A_p \text{ in ft}^2 \]

\[ R_R = 0.29 R_D \]

Example:
If \( C = 0.18 \)

\( R_D = 3.3 \text{ nm} \)

\( R_R = 1.0 \text{ nm} \)

Note:
- \( R_D \) = Visual detection range for probability of detection \( P_D = 0.5 \)
- \( C \) = Contrast
- \( C_T \) = Visual threshold contrast = 0.02
- \( A_p \) = Target presented area = 50 ft\(^2\)
- \( R_R \) = Visual recognition range
- \( \theta_F \) = Pilot visual foveal angle = 0.8 deg
- Clear weather
- Pilot search glimpse time = 1 / 3 s
High Missile Velocity Improves Standoff Range

\[ \frac{R_{F-Pole}}{R_L} = 1 - \frac{(V_T + V_L)}{(V_M + V_T)} \]

Note: Head-on intercept

- \( R_{F-Pole} = \) Standoff range at intercept
- \( R_L = \) Launch range
- \( V_M = \) Missile average velocity
- \( V_T = \) Target velocity
- \( V_L = \) Launch velocity

Example:
- \( V_L = V_T \)
- \( V_M = 2V_T \)
- Then \( \frac{V_T}{V_M} = \frac{V_L}{V_M} = 0.5 \)
- \( R_{F-Pole} / R_L = 0.33 \)
- \( R_{F-Pole} = R_D = 3.3 \text{ nm} \)
- \( R_L = 3.3 / 0.33 = 10.0 \text{ nm} \)
Missile Flight Range Requirement Is Greatest for a Tail Chase Intercept

\[
\left( \frac{R_F}{R_L} \right)_{\text{Head-on}} = \left( \frac{V_M}{V_T} \right) / \left[ \left( \frac{V_M}{V_T} \right) + 1 \right]
\]

\[
\left( \frac{R_F}{R_L} \right)_{\text{Tail Chase}} = \left( \frac{V_M}{V_T} \right) / \left[ \left( \frac{V_M}{V_T} \right) - 1 \right]
\]

Examples:
- **Head-on Intercept**
  - \( V_M = 1,640 \text{ ft/s}, V_T = 820 \text{ ft/s} \)
  - \( \frac{V_M}{V_T} = 1640 / 820 = 2 \)
  - \( \frac{R_F}{R_L} = 2 / (2 + 1) = 0.667 \)
  - \( R_L = 10.0 \text{ nm} \)
  - \( R_F = 0.667 (10.0) = 6.67 \text{ nm} \)
- **Tail Intercept at same conditions**
  - \( \frac{R_F}{R_L} = 2 / (2 - 1) = 2.0 \)
  - \( R_F = 2.0 (10.0) = 20.0 \text{ nm} \)
Note: Dimensions in inches

## Mass Properties of Rocket Baseline Missile

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight, lb.</th>
<th>C.G. STA, in.</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 Nose (Radome)</td>
<td>4.1</td>
<td>12.0</td>
</tr>
<tr>
<td>3 Forebody structure</td>
<td>12.4</td>
<td>30.5</td>
</tr>
<tr>
<td>Guidance</td>
<td>46.6</td>
<td>32.6</td>
</tr>
<tr>
<td>2 Payload Bay Structure</td>
<td>7.6</td>
<td>54.3</td>
</tr>
<tr>
<td>Warhead</td>
<td>77.7</td>
<td>54.3</td>
</tr>
<tr>
<td>4 Midbody Structure</td>
<td>10.2</td>
<td>73.5</td>
</tr>
<tr>
<td>Control Actuation System</td>
<td>61.0</td>
<td>75.5</td>
</tr>
<tr>
<td>5 Aftbody Structure</td>
<td>0.0</td>
<td>–</td>
</tr>
<tr>
<td>Rocket Motor Case</td>
<td>47.3</td>
<td>107.5</td>
</tr>
<tr>
<td>Insulation (EDPM – Silica)</td>
<td>23.0</td>
<td>117.2</td>
</tr>
<tr>
<td>6 Tailcone Structure</td>
<td>6.5</td>
<td>141.2</td>
</tr>
<tr>
<td>Nozzle</td>
<td>5.8</td>
<td>141.2</td>
</tr>
<tr>
<td>Fixed Surfaces</td>
<td>26.2</td>
<td>137.8</td>
</tr>
<tr>
<td>Movable Surfaces</td>
<td>38.6</td>
<td>75.5</td>
</tr>
<tr>
<td>Burnout Total</td>
<td>367.0</td>
<td>76.2</td>
</tr>
<tr>
<td>Propellant</td>
<td>133.0</td>
<td>107.8</td>
</tr>
<tr>
<td>Launch Total</td>
<td>500.0</td>
<td>84.6</td>
</tr>
</tbody>
</table>
### Rocket Baseline Missile Definition

#### Body

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dome Material</td>
<td>Pyroceram</td>
</tr>
<tr>
<td>Airframe Material</td>
<td>Aluminum 2219-T81</td>
</tr>
<tr>
<td>Length, in</td>
<td>143.9</td>
</tr>
<tr>
<td>Diameter, in</td>
<td>8.0</td>
</tr>
<tr>
<td>Airframe thickness, in</td>
<td>0.16</td>
</tr>
<tr>
<td>Fineness ratio</td>
<td>17.99</td>
</tr>
<tr>
<td>Volume, ft³</td>
<td>3.82</td>
</tr>
<tr>
<td>Wetted area, ft²</td>
<td>24.06</td>
</tr>
<tr>
<td>Nozzle exit area, ft²</td>
<td>0.078</td>
</tr>
<tr>
<td>Boattail fineness ratio</td>
<td>0.38</td>
</tr>
<tr>
<td>Nose fineness ratio</td>
<td>2.40</td>
</tr>
<tr>
<td>Nose bluntness</td>
<td>0.0</td>
</tr>
<tr>
<td>Boattail angle, deg</td>
<td>7.5</td>
</tr>
</tbody>
</table>

**Movable surfaces (forward)**

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Aluminum 2219-T81</td>
</tr>
<tr>
<td>Planform area, ft² (2 panels exposed)</td>
<td>2.55</td>
</tr>
<tr>
<td>Wetted area, ft² (4 panels)</td>
<td>10.20</td>
</tr>
<tr>
<td>Aspect ratio (2 panels exposed)</td>
<td>2.82</td>
</tr>
<tr>
<td>Taper ratio</td>
<td>0.175</td>
</tr>
<tr>
<td>Root chord, in</td>
<td>19.4</td>
</tr>
<tr>
<td>Tip chord, in</td>
<td>3.4</td>
</tr>
<tr>
<td>Span, in (2 panels exposed)</td>
<td>32.2</td>
</tr>
<tr>
<td>Leading edge sweep, deg</td>
<td>45.0</td>
</tr>
</tbody>
</table>
### Movable surfaces (continued)

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mean aerodynamic chord, in</td>
<td>13.3</td>
</tr>
<tr>
<td>Thickness ratio</td>
<td>0.044</td>
</tr>
<tr>
<td>Section type</td>
<td>Modified double wedge</td>
</tr>
<tr>
<td>Section leading edge total angle, deg</td>
<td>10.01</td>
</tr>
<tr>
<td>$x_{mac}$, in</td>
<td>67.0</td>
</tr>
<tr>
<td>$y_{mac}$, in (from root chord)</td>
<td>6.2</td>
</tr>
<tr>
<td>Actuator rate limit, deg / s</td>
<td>360.0</td>
</tr>
</tbody>
</table>

### Fixed surfaces (aft)

<table>
<thead>
<tr>
<th>Property</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Aluminum 2219-T81</td>
</tr>
<tr>
<td>Modulus of elasticity, $10^6$ psi</td>
<td>10.5</td>
</tr>
<tr>
<td>Planform area, ft$^2$ (2 panels exposed)</td>
<td>1.54</td>
</tr>
<tr>
<td>Wetted area, ft$^2$ (4 panels)</td>
<td>6.17</td>
</tr>
<tr>
<td>Aspect ratio (2 panels exposed)</td>
<td>2.59</td>
</tr>
<tr>
<td>Taper ratio</td>
<td>0.0</td>
</tr>
<tr>
<td>Root chord, in</td>
<td>18.5</td>
</tr>
<tr>
<td>Tip chord, in</td>
<td>0.0</td>
</tr>
<tr>
<td>Span, in (2 panels exposed)</td>
<td>24.0</td>
</tr>
<tr>
<td>Leading edge sweep, deg</td>
<td>57.0</td>
</tr>
<tr>
<td>Mean aerodynamic chord, in</td>
<td>12.3</td>
</tr>
<tr>
<td>Thickness ratio</td>
<td>0.027</td>
</tr>
<tr>
<td>Section type</td>
<td>Modified double wedge</td>
</tr>
<tr>
<td>Section leading edge total angle, deg</td>
<td>6.17</td>
</tr>
<tr>
<td>$x_{mac}$, in</td>
<td>131.6</td>
</tr>
<tr>
<td>$y_{mac}$, in (from root chord)</td>
<td>4.0</td>
</tr>
</tbody>
</table>
### Rocket Baseline Missile Definition (cont)

#### References values
- Reference area, ft²: 0.349
- Reference length, ft: 0.667
- Pitch / Yaw Moment of inertia at launch, slug-ft²: 117.0
- Pitch / Yaw Moment of inertia at burnout, slug-ft²: 94.0

#### Rocket Motor Performance (altitude = 20k ft, temp = 70° F)
- Burning time, sec (boost / sustain): 3.69 / 10.86
- Maximum pressure, psi: 2042
- Average pressure, psi (boost / sustain): 1769 / 301
- Average thrust, lbf (boost / sustain): 5750 / 1018
- Total impulse, lbf-s (boost / sustain): 21217 / 11055
- Specific impulse, lbf-s / lbm (boost / sustain): 250 / 230.4

#### Propellant
- Weight, lbm (boost / sustain): 84.8 / 48.2
- Flame temperature @ 1,000 psi, °F: 5282 / 5228
- Propellant density, lbm / in³: 0.065
- Characteristic velocity, ft / s: 5200
- Burn rate @ 1000 psi, in / s: 0.5
- Burn rate pressure exponent: 0.3
### Rocket Baseline Missile Definition (cont)

#### Propellant (continued)
- **Burn rate sensitivity with temperature, % / °F**: 0.10
- **Pressure sensitivity with temperature, % / °F**: 0.14

#### Rocket Motor Case
- **Yield / ultimate strength, psi**: 170,000 / 190,000
- **Material**: 4130 Steel
- **Modulus of elasticity, psi**: $29.5 \times 10^6$ psi
- **Length, in**: 59.4
- **Outside diameter, in**: 8.00
- **Thickness, in (minimum)**: 0.074
- **Burst pressure, psi**: 3140
- **Volumetric efficiency**: 0.76
- **Grain configuration**: Three slots + web
- **Dome ellipse ratio**: 2.0

#### Nozzle
- **Housing material**: 4130 Steel
- **Exit geometry**: Contoured (equiv. 15°)
- **Throat area, in²**: 1.81
- **Expansion ratio**: 6.2
- **Length, in**: 4.9
- **Exit diameter, in**: 3.78
Rocket Baseline Missile Has Boost-Sustain Thrust - Time History

Thrust – 1,000 lb

Boost Total Impulse = \( \int T \, dt = 5750 \times 3.69 = 21217 \text{ lb-s} \)

Sustain Total Impulse = \( \int T \, dt = 1018 \times 10.86 = 11055 \text{ lb-s} \)

Note: Altitude = 20k ft, Temperature = 70° F
Total impulse drives velocity change
Rocket Baseline Missile Aerodynamic Characteristics

\[ S_{Ref} = 0.349 \text{ ft}^2, \quad l_{Ref} = d = 0.667 \text{ ft}, \quad \text{CG at STA 75.7}, \delta = 0 \text{ deg} \]

- Pitching Moment \( C_m \)
  - \( M = 1.2 \) and \( 1.5 \)
  - \( S_{Ref} \) and \( l_{Ref} \)
  - \( \delta = 0 \text{ deg} \)

- Normal Force \( C_N \)
  - \( M = 1.2 \) and \( 0.6 \)
  - \( 1.5 \) and \( 0.6 \)
  - \( 1.2 \) and \( 4.60 \)
  - \( 1.5 \) and \( 3.95 \)
  - \( 2.0 \) and \( 2.35 \)
  - \( 2.0 \) and \( 2.87 \)
  - \( 2.35 \) and \( 4.60 \)

\( \alpha, \text{Angle of Attack – Deg} \)
Rocket Baseline Missile Aerodynamic Characteristics (cont)


graphs showing aerodynamic characteristics:

- \( C_{A0} = C_{A\alpha} = 0 + K_1 \delta^2 + K_2 \alpha \delta \)
- \( K_1, K_2 \sim \text{Per Deg}^2 \)
- \( C_{A\alpha} \), \( C_{m\delta} \), \( C_{n\delta} \) at \( \alpha = 0 \text{ deg} \), Per Deg

Mach Number range: 0 to 5
High Altitude Launch Enhances Rocket Baseline Range

- **Burnout**: $V_{\text{max}} = 2524$ ft/s
- **Termination Mach = 1.5**: $V_{\text{max}} = 2147$ ft/s
- **Coast**: $V_{\text{max}} = 1916$ ft/s

- **ML = 0.7**
- **$C_{\text{D AVG}} = 0.65$**
- **Constant Altitude Flight**

Altitude ~ $10^3$ ft

Range ~ nm
Low Altitude Launch and High Alpha Maneuvers
Enhance Rocket Baseline Turn Performance

Note: Off boresight envelope that is shown does not include the rocket baseline seeker field-of-regard limit (30 deg).
Paredo Shows Range of Rocket Baseline Driven by $I_{sp}$, Propellant Weight, Drag, and Static Margin

Note: Rocket baseline:
- $h_L = 20k$ ft, $M_L = 0.7$, $M_{EC} = 1.5$
- $R@M_L = 0.7, h_L = 20k = 9.5$ nm

Example: 10% increase in propellant weight $\Rightarrow$ 8.8% increase in flight range
Assumptions

1 degree of freedom
Constant altitude

Simplified equation for axial acceleration based on thrust, drag, and weight
\[ n_x = \frac{T - D}{W} \]

Missile weight varies with burn rate and time
\[ W = W_L - \frac{W_P}{t_B} \cdot t \]

Drag is approximated by
\[ D = C_{D0} q S \]
Example of Rocket Baseline Axial Acceleration

\[ n_x = \frac{(T - D)}{W} \]

Note:
- \( t_f = 24.4 \, \text{s} \)
- \( M_L = 0.8 \)
- \( h_L = 20,000 \, \text{ft} \)
- \( T_B = 5750 \, \text{lb} \)
- \( t_B = 3.69 \, \text{s} \)
- \( T_S = 1018 \, \text{lb} \)
- \( t_S = 10.86 \, \text{s} \)
- \( D = 99 \, \text{lb at Mach 0.8} \)
- \( D = 1020 \, \text{lb at Mach 2.1} \)
- \( W_L = 500 \, \text{lb} \)
- \( W_P = 133 \, \text{lb} \)
Example of Rocket Baseline Missile Velocity vs Time

\[ \Delta V / (g_c \ I_{SP}) = - (1 - D_{AVG} / T) \ln (1 - W_p / W_i), \text{During Boost-Sustain} \]

\[ V / V_{BO} = 1 / \left\{ 1 + t / \left[ 2 W_{BO} / \left[ g_c \rho_{AVG} S_{Ref} \left( C_{D0} \right)_{AVG} V_{BO} \right] \right] \right\}, \text{During Coast} \]

Note:

- \( M_L = 0.8 \)
- \( h_L = 20k \text{ feet} \)
Range and Time-to-Target of Rocket Baseline Missile Meet Requirements

\[ R = \Delta R_{\text{boost}} + \Delta R_{\text{sustain}} + \Delta R_{\text{coast}} \]

Note:
- \( M_L = 0.8 \)
- \( h_L = 20k \text{ ft} \)

\((R_F)_{\text{req}} = 6.7 \text{ nm @ } t = 24.4 \text{ s}\)
Sizing Examples

- **Rocket Baseline Missile**
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- **Ramjet Baseline Missile**
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- **Computer Aided Conceptual Design Sizing Tools**

- **Soda Straw Rocket Design, Build, and Fly**
Example of Wing Sizing to Satisfy Required Maneuver Acceleration

- Size Wing for the Assumptions
  - \( (n_z)_{\text{Required}} = 30 \text{ g to counter 9 g maneuvering target} \)
  - \( (n_z) = \Delta (n_z)_{\text{Wing}} + \Delta (n_z)_{\text{Body}} + \Delta (n_z)_{\text{Tail}} \)

- Rocket Baseline @
  - Mach 2
  - 20,000 ft altitude
  - 367 lb weight (burnout)

- From Prior Example, Compute
  - \( \alpha_{\text{Wing}} = \alpha'_{\text{Max}} = (\alpha + \delta)_{\text{Max}} = 22 \text{ deg for rocket baseline} \)
  - \( \alpha = 0.75\delta \), \( \alpha_{\text{Body}} = \alpha_{\text{Tail}} = 9.4 \text{ deg} \)
  - \( \Delta (n_z)_{\text{Body}} = q S_{\text{Ref}} (C_N)_{\text{Body}} / W = 2725 (0.349) (1.28) / 367 = 3.3 \text{ g (from body)} \)
  - \( \Delta (n_z)_{\text{Tail}} = q S_{\text{Tail}} [(C_N)_{\text{Tail}} (S_{\text{Ref}} / S_{\text{Tail}})] / W = 2725 (1.54) (0.425) / 367 = 4.9 \text{ g (from tail)} \)
  - \( \Delta (n_z)_{\text{Wing}} = (n_z)_{\text{Required}} - \Delta (n_z)_{\text{Body}} - \Delta (n_z)_{\text{Tail}} = 30 - 3.3 - 4.9 = 21.8 \text{ g} \)
  - \( (S_W)_{\text{Required}} = \Delta (n_z)_{\text{Wing}} W / \{q [(C_N)_{\text{Wing}} (S_{\text{Ref}} / S_{\text{Wing}})]\} = 21.8 (367) / \{(2725) (1.08)\} = 2.72 \text{ ft}^2 \)

Note: \( (S_W)_{\text{Rocket Baseline}} = 2.55 \text{ ft}^2 \)

2/24/2008

Video of Intercept of Maneuvering Target

Note: (SW)RocketBaseline = 2.55 ft²
Wing Sizing to Satisfy Required Turn Rate

Assume

(γ' )\text{Required} > 18 \ deg / s \ to \ counter \ 18 \ deg / s \ maneuvering \ aircraft

Rocket Baseline @

- Mach 2
- 20,000 ft altitude
- 367 lb weight (burnout)
- γ_i = 0 deg

Compute

\[ \gamma' = \frac{g_c \ \mathbf{n} / \mathbf{V} = \left[ q \ S_{\text{Ref}} \ \mathbf{C}_{N\alpha} \ \alpha + q \ S_{\text{Ref}} \ \mathbf{C}_{N\delta} \ \delta - W \cos(\gamma) \right] / \left[ \left( W / g_c \right) \mathbf{V} \right] }{\alpha / \delta = 0.75} \]

\[ \alpha' = \alpha + \delta = 22 \ deg \ \Rightarrow \ \delta = 12.6 \ deg, \ \alpha = 9.4 \ deg \]

\[ \gamma' = \left[ 2725 \ (0.349)(0.60)(9.4) +2725 \ (0.349)(0.19)(12.6) - 367 \ (1) \right] / \left( 367 / 32.2 \right)(2074) = 0.31 \ rad / s \ or \ 18 \ deg / s \]

Note: (S_W)\text{RocketBaseline} \Rightarrow 18 \ deg / s \ Turn \ Rate
Wing Sizing to Satisfy Required Turn Radius

- Assume Maneuvering Aircraft Target with
  - $\gamma' = 18 \text{ deg/s} = 0.314 \text{ rad/s}$
  - $V = 1000 \text{ ft/s}$
  - $(R_T)_{\text{Target}} = \frac{V}{\gamma'} = \frac{1000}{0.314} = 3183 \text{ ft}$

- Assume Rocket Baseline @
  - Mach 2
  - 20,000 ft altitude
  - 367 lb weight (burnout)

- Compute
  - $\gamma' = 18 \text{ deg/s}$ (prior figure)
  - $(R_T)_{\text{RocketBaseline}} = \frac{V}{\gamma'} = \frac{2074}{0.314} = 6602 \text{ ft}$

Note: $(R_T)_{\text{RocketBaseline}} > (R_T)_{\text{Target}} \Rightarrow$ Rocket Baseline Can Be Counter-measured by Target in a Tight Turn

- Counter-Countermeasure Alternatives
  - Larger Wing
  - Higher Angle of Attack
  - Longer Burn Motor with TVC
Sizing Examples

- **Rocket Baseline Missile**
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- **Ramjet Baseline Missile**
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- **Computer Aided Conceptual Design Sizing Tools**

- **Soda Straw Rocket Design, Build, and Fly**
## Combined Weight / Miss Distance Drivers: Nozzle Expansion and Motor Volumetric Efficiency

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline</th>
<th>Sensitivity Variation</th>
<th>W*</th>
<th>σ*</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fixed surface number of panels</td>
<td>4</td>
<td>3</td>
<td>+0.054</td>
<td>+0.100</td>
</tr>
<tr>
<td>Movable surface number of panels</td>
<td>4</td>
<td>2</td>
<td>+0.071</td>
<td>+0.106</td>
</tr>
<tr>
<td>Design static margin at launch</td>
<td>0.40</td>
<td>0.30</td>
<td>+0.095</td>
<td>+0.167</td>
</tr>
<tr>
<td>Wing movable surface sweep (deg)</td>
<td>45.0</td>
<td>49.5</td>
<td>-0.205</td>
<td>+0.015</td>
</tr>
<tr>
<td>Tail fixed surface sweep (deg)</td>
<td>57.0</td>
<td>60.0</td>
<td>+0.027</td>
<td>+0.039</td>
</tr>
<tr>
<td>Wing movable surface thickness ratio</td>
<td>0.044</td>
<td>0.034</td>
<td>+0.041</td>
<td>+0.005</td>
</tr>
<tr>
<td>Nose fineness ratio</td>
<td>2.4</td>
<td>2.6</td>
<td>-0.016</td>
<td>-0.745</td>
</tr>
<tr>
<td>Rocket chamber sustain pressure (psi)</td>
<td>301</td>
<td>330</td>
<td>-0.076</td>
<td>-0.045</td>
</tr>
<tr>
<td>Boattail fineness ratio (length / diameter)</td>
<td>0.38</td>
<td>0.342</td>
<td>+0.096</td>
<td>+0.140</td>
</tr>
<tr>
<td>Nozzle expansion ratio</td>
<td>6.2</td>
<td>6.82</td>
<td>-0.114</td>
<td>-0.181</td>
</tr>
<tr>
<td>Motor volumetric efficiency</td>
<td>0.76</td>
<td>0.84</td>
<td>-0.136</td>
<td>-0.453</td>
</tr>
<tr>
<td>Propellant density (lb/in³)</td>
<td>0.065</td>
<td>0.084</td>
<td>-0.062</td>
<td>+0.012</td>
</tr>
<tr>
<td>Boost thrust (lb)</td>
<td>5,750</td>
<td>6,325</td>
<td>+0.014</td>
<td>-0.018</td>
</tr>
<tr>
<td>Sustain thrust (lb)</td>
<td>1,018</td>
<td>1,119</td>
<td>+0.088</td>
<td>+0.246</td>
</tr>
<tr>
<td>Characteristic velocity (ft/s)</td>
<td>5,200</td>
<td>5,720</td>
<td>-0.063</td>
<td>-0.077</td>
</tr>
<tr>
<td>Wing location (percent total length)</td>
<td>47.5</td>
<td>42.75</td>
<td>+0.181</td>
<td>-0.036</td>
</tr>
</tbody>
</table>

Baseline: Weight = 500 lb, Miss distance = 62.3 ft

W* = weight sensitivity for parameter variation = ΔW / W

σ* = miss distance sensitivity for parameter variation = Δσ / σ

Note: Strong impact with synergy
Strong impact
Moderate impact with synergy
Moderate impact
## A Harmonized Missile Can Have Smaller Miss Distance and Lighter Weight

### Judicious changes

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline Value</th>
<th>Missile Configured for Minimum:</th>
<th>Harmonized</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boost thrust (lb)</td>
<td>5,750</td>
<td>3,382</td>
<td>3,382</td>
</tr>
<tr>
<td>Wing location (percent missile length to 1/4 mac)</td>
<td>47.5</td>
<td>47</td>
<td>44</td>
</tr>
<tr>
<td>Wing taper ratio</td>
<td>0.18</td>
<td>0.2</td>
<td>0.2</td>
</tr>
<tr>
<td>Nose fineness ratio</td>
<td>2.4</td>
<td>3.2</td>
<td>2.55</td>
</tr>
<tr>
<td>Nose blunting ratio</td>
<td>0.0</td>
<td>0.05</td>
<td>0.05</td>
</tr>
<tr>
<td>Nozzle expansion ratio</td>
<td>6.2</td>
<td>15</td>
<td>15</td>
</tr>
<tr>
<td>Sustain chamber pressure (psi)</td>
<td>301</td>
<td>1,000</td>
<td>1,000</td>
</tr>
<tr>
<td>Boattail fineness ratio</td>
<td>0.38</td>
<td>0.21</td>
<td>0.21</td>
</tr>
<tr>
<td>Tail leading edge sweep (deg)</td>
<td>57</td>
<td>50</td>
<td>50</td>
</tr>
</tbody>
</table>

### Technology limited changes

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline Value</th>
<th>Missile Configured for Minimum:</th>
<th>Harmonized</th>
</tr>
</thead>
<tbody>
<tr>
<td>No. wing panels</td>
<td>4</td>
<td>2</td>
<td>2</td>
</tr>
<tr>
<td>No. tail panels</td>
<td>4</td>
<td>3</td>
<td>3</td>
</tr>
<tr>
<td>Wing thickness ratio</td>
<td>0.044</td>
<td>0.030</td>
<td>0.030</td>
</tr>
<tr>
<td>Wing leading edge sweep (deg)</td>
<td>45</td>
<td>55</td>
<td>55</td>
</tr>
<tr>
<td>Static margin at launch (diam)</td>
<td>0.4</td>
<td>0.0</td>
<td>0.0</td>
</tr>
<tr>
<td>Propellant density (lb/ in³)</td>
<td>0.065</td>
<td>0.084</td>
<td>0.084</td>
</tr>
<tr>
<td>Motor volumetric efficiency</td>
<td>0.76</td>
<td>0.84</td>
<td>0.84</td>
</tr>
</tbody>
</table>

### Measures of Merit

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline Value</th>
<th>Missile Configured for Minimum:</th>
<th>Harmonized</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total weight (lb)</td>
<td>500</td>
<td>385.9</td>
<td>390.1</td>
</tr>
<tr>
<td>Miss distance (ft)</td>
<td>62.3</td>
<td>63.1</td>
<td>16.2</td>
</tr>
<tr>
<td>Time to target (s)</td>
<td>21.6</td>
<td>23.8</td>
<td>23.6</td>
</tr>
<tr>
<td>Length (in)</td>
<td>144</td>
<td>112.7</td>
<td>114.9</td>
</tr>
<tr>
<td>Mach No. at burnout</td>
<td>2.20</td>
<td>2.08</td>
<td>2.07</td>
</tr>
<tr>
<td>Weight of propellant (lb)</td>
<td>133</td>
<td>78.3</td>
<td>85.4</td>
</tr>
<tr>
<td>Wing area (in²)</td>
<td>368.6</td>
<td>175.5</td>
<td>150.7</td>
</tr>
<tr>
<td>Tail area (in²)</td>
<td>221.8</td>
<td>109.1</td>
<td>134.5</td>
</tr>
</tbody>
</table>

*Note: □ Value of driving parameter*
Baseline Missile vs Harmonized Missile

Nose Fineness: \{ 2.4, 2.6 \}

Weight (lb): \{ 500, 390 \}

Propellant Density (lb/in³): \{ 0.065, 0.084 \}

Surfaces: \{ 4 wings / 4 tails, 2 wings / 3 tails \}

144"
Sizing Examples

- **Rocket Baseline Missile**
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- **Ramjet Baseline Missile**
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- **Computer Aided Conceptual Design Sizing Tools**

- **Soda Straw Rocket Design, Build, and Fly**
Lofted Glide Trajectory Provides Extended Range

- Using Rocket Baseline, Compare
  - Lofted Launch-Coast-Glide Trajectory
  - Lofted Launch-Ballistic Trajectory
  - Constant Altitude Trajectory

- Assume for Lofted Launch-Coast-Glide Trajectory:
  - $\gamma_i = 45\,\text{deg}$
  - $\gamma = 45\,\text{deg}$ during boost and sustain
  - $\gamma = 45\,\text{deg}$ coast
  - Switch to $(L/D)_{\max}$ glide at optimum altitude
  - $(L/D)_{\max}$ glide trajectory after apogee
  - $h_i = h_f = 0\,\text{ft}$

- Velocity, Horizontal Range, and Altitude During Initial Boost @ $\gamma = 45\,\text{deg}$
  \[ \Delta V = -g_c I_{sp} \left[ 1 - \frac{D_{AVG}}{T} - \frac{W_{AVG} \sin \gamma}{T} \right] \ln \left( 1 - \frac{W_p}{W_i} \right) = -32.2 \left( \frac{250}{5750} \right) \left[ 1 - \frac{419}{5750} - \frac{458 \times 0.707}{5750} \right] \ln \left( 1 - \frac{84.8}{500} \right) = 1,303\,\text{ft/s} \]
  \[ \Delta R = \left( V_i + \Delta V / 2 \right) t_B = \left( 0 + 1303 / 2 \right) 3.69 = 2,404\,\text{ft} \]
  \[ \Delta R_x = \Delta R \cos \gamma_i = 2404 \times 0.707 = 1,700\,\text{ft} \]
  \[ \Delta R_y = \Delta R \sin \gamma_i = 2404 \times 0.707 = 1,700\,\text{ft} \]
  \[ h = h_i + \Delta R_y = 0 + 1700 = 1,700\,\text{ft} \]
Lofted Glide Trajectory Provides Extended Range (cont)

- **Velocity, Horizontal Range, and Altitude During Sustain @ γ = 45 deg**
  \[ \Delta V = -g \ c \ \text{I}_{sp}\left[1 - \frac{(D_{AVG} / T)}{(W_{AVG}\sin\gamma / T)}\right]\ln\left(1 - \frac{W_p}{W_i}\right) = -32.2 \ (230.4) \left[1 - (650 / 1018) - 391 \ (0.707) / 1018\right]\ln\left(1 - 48.2 / 415.2\right) = 81 \text{ ft/sec}\]
  
  \[V_{BO} = 1303 + 81 = 1,384 \text{ ft/s}\]
  \[\Delta R = (V_i + \Delta V / 2) \ t_B = (1303 + 81 / 2) \ 10.86 = 14,590 \text{ ft}\]
  \[\Delta R_x = \Delta R \cos \gamma_i = 14590 \ (0.707) = 10,315 \text{ ft}\]
  \[\Delta R_y = \Delta R \sin \gamma_i = 14,590 \ (0.707) = 10,315 \text{ ft}\]
  \[h = h_i + \Delta R_y = 1700 + 10315 = 12,015 \text{ ft}\]

- **Velocity, Horizontal Range, and Altitude During Coast @ γ = 45 deg to h@((L / D)_{max})**
  \[V_{coast} = V_i \left\{1 - \left[\frac{(g \ c \sin\gamma)}{V_i}\right] t\right\} / \left\{1 + \left[\frac{g \ c \rho_{AVG} \ S_{Ref} \ (C_{D_0})_{AVG} \ V_i}{(2 \ W)} t\right]\right\} = 1384 \left\{1 - \left[\frac{(32.2 \ (0.707))}{1384}\right] t\right\} / \left\{1 + \left[\frac{32.2 \ (0.001338) \ (0.349) \ (0.7) \ (1384)}{(2 \ (367))} t\right]\right\} = 674 \text{ ft/s}\]
  \[R_{coast} = \left\{\frac{2 \ W}{g \ c \rho_{AVG} \ S_{Ref} \ (C_{D_0})_{AVG}}\right\} \ln\left\{1 - \left[\frac{g \ c^2 \rho_{AVG} \ S_{Ref} \ (C_{D_0})_{AVG}}{(2 \ W)}\right] \sin\gamma t^2 + \left[\frac{g \ c \rho_{AVG} \ S_{Ref} \ (C_{D_0})_{AVG} \ V_i}{(2 \ W)} t\right]\right\} = \left\{\frac{2 \ (367)}{32.2 \ (0.001338) \ (0.349) \ (0.7)} \ln\left\{1 - \left[(32.2)^2 \ (0.001338) \ (0.349) \ (0.7) / (2 \ (367))\right] t^2 + \left[32.2 \ (0.001338) \ (0.349) \ (0.7) \ (1384) / (2 \ (367))\right] t\right\} = 17148 \text{ ft}\]
  \[(R_x)_{coast} = (R_y)_{coast} = R_{coast} \sin\gamma = 17148 \ (0.707) = 12124 \text{ ft}\]
Lofted Glide Trajectory Provides Extended Range (cont)

Flight Conditions At End-of-Coast Are:
- \( t = 35 \text{ s} \)
- \( V = 674 \text{ ft} / \text{s} \)
- \( h = 24,189 \text{ ft} \)
- \( q = 251 \text{ psf} \)
- \( M = 0.66 \)
- \( (L / D)_{\text{max}} = 5.22 \)
- \( \alpha (L / D)_{\text{max}} = 5.5 \text{ deg} \)

Initiate \( \alpha = \alpha (L / D)_{\text{max}} = 5.5 \text{ deg} \) at \( h = 24,189 \text{ ft} \)

Incremental Horizontal Range During \((L / D)_{\text{max}}\) Glide Is
- \( \Delta R_x = (L / D) \Delta h = 5.22 (24189) = 126,267 \text{ ft} \)

Total Horizontal Range for Elevated Launch-Coast-Glide Trajectory Is
- \( R_x = \sum \Delta R_x = \Delta R_{x,\text{Boost}} + \Delta R_{x,\text{Sustain}} + \Delta R_{x,\text{Coast}} + \Delta R_{x,\text{Glide}} = 1700 + 10315 + 12124 + 126267 = 150406 \text{ ft} = 24.8 \text{ nm} \)
Lofted Glide Trajectory Provides Extended Range (cont)

Note: Rocket Baseline

- End of boost
- End of sustain
- Lofted ballistic apogee, t = 35 s, V = 667 ft/s, h = 21,590 ft
- Lofted coast apogee, t = 35 s, V = 674 ft/s, h = 24,189 ft
- Lofted ballistic impact, t = 68 s, γ = -71 deg, V = 1368 ft/s
- Lofted glide impact, t = 298 s, γ = -10.8 deg, V = 459 ft/s
- Co-altitude flight impact, t = 115 s, V = 500 ft/s

Co-altitude flight impact, t = 115 s, V = 500 ft/s
Sizing Examples

- Rocket Baseline Missile
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- Ramjet Baseline Missile
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- Computer Aided Conceptual Design Sizing Tools

- Soda Straw Rocket Design, Build, and Fly
Ramjet Baseline Is a Chin Inlet Integral Rocket

Ramjet (IRR)

## Mass Properties of Ramjet Baseline Missile

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight, lb</th>
<th>CG Sta, in</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nose</td>
<td>15.9</td>
<td>15.7</td>
</tr>
<tr>
<td>Forebody Structure</td>
<td>42.4</td>
<td>33.5</td>
</tr>
<tr>
<td>Guidance</td>
<td>129.0</td>
<td>33.5</td>
</tr>
<tr>
<td>Payload Bay Structure</td>
<td>64.5</td>
<td>60.0</td>
</tr>
<tr>
<td>Warhead</td>
<td>510.0</td>
<td>60.0</td>
</tr>
<tr>
<td>Midbody Structure</td>
<td>95.2</td>
<td>101.2</td>
</tr>
<tr>
<td>Inlet</td>
<td>103.0</td>
<td>80.0</td>
</tr>
<tr>
<td>Electrical</td>
<td>30.0</td>
<td>112.0</td>
</tr>
<tr>
<td>Hydraulic System for Control Actuation</td>
<td>20.0</td>
<td>121.0</td>
</tr>
<tr>
<td>Fuel Distribution</td>
<td>5.0</td>
<td>121.0</td>
</tr>
<tr>
<td>Aftbody Structure</td>
<td>44.5</td>
<td>142.5</td>
</tr>
<tr>
<td>Engine</td>
<td>33.5</td>
<td>142.5</td>
</tr>
<tr>
<td>Tailcone Structure</td>
<td>31.6</td>
<td>165.0</td>
</tr>
<tr>
<td>Ramjet Nozzle</td>
<td>31.0</td>
<td>165.0</td>
</tr>
<tr>
<td>Flight Control Actuators</td>
<td>37.0</td>
<td>164.0</td>
</tr>
<tr>
<td>Fins (4)</td>
<td>70.0</td>
<td>157.2</td>
</tr>
<tr>
<td>End of Cruise</td>
<td>1,262.6</td>
<td>81.8</td>
</tr>
<tr>
<td>Ramjet Fuel (11900 in³)</td>
<td>476.0</td>
<td>87.0</td>
</tr>
<tr>
<td>Start of Cruise</td>
<td>1,738.6</td>
<td>83.2</td>
</tr>
<tr>
<td>Boost Nozzle (Ejected)</td>
<td>31.0</td>
<td>164.0</td>
</tr>
<tr>
<td>Frangible Port</td>
<td>11.5</td>
<td>126.0</td>
</tr>
<tr>
<td>End of Boost</td>
<td>1,781.1</td>
<td>84.9</td>
</tr>
<tr>
<td>Boost Propellant</td>
<td>449.0</td>
<td>142.5</td>
</tr>
<tr>
<td>Booster Ignition</td>
<td>2,230.1</td>
<td>96.5</td>
</tr>
</tbody>
</table>
# Ramjet Baseline Missile Definition

## Inlet

<table>
<thead>
<tr>
<th>Type</th>
<th>Mixed compression</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Titanium</td>
</tr>
<tr>
<td>Conical forebody half angle, deg</td>
<td>17.7</td>
</tr>
<tr>
<td>Ramp wedge angle, deg</td>
<td>8.36</td>
</tr>
<tr>
<td>Cowl angle, deg</td>
<td>8.24</td>
</tr>
<tr>
<td>Internal contraction ratio</td>
<td>12.2 Percent</td>
</tr>
<tr>
<td>Capture area, ft²</td>
<td>0.79</td>
</tr>
<tr>
<td>Throat area, ft²</td>
<td>0.29</td>
</tr>
</tbody>
</table>

## Body

<table>
<thead>
<tr>
<th>Dome Material</th>
<th>Silicon nitride</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airframe Material</td>
<td>Titanium</td>
</tr>
<tr>
<td>Combustor Material</td>
<td>Insulated Inconel</td>
</tr>
<tr>
<td>Length, in</td>
<td>171.0</td>
</tr>
<tr>
<td>Diameter, in</td>
<td>20.375</td>
</tr>
<tr>
<td>Fineness ratio</td>
<td>8.39</td>
</tr>
<tr>
<td>Volume, ft³</td>
<td>28.33</td>
</tr>
<tr>
<td>Wetted area, ft²</td>
<td>68.81</td>
</tr>
<tr>
<td>Base area, ft² (cruise)</td>
<td>0.58</td>
</tr>
<tr>
<td>Boattail fineness ratio</td>
<td>N/A</td>
</tr>
<tr>
<td>Nose half angle, deg</td>
<td>17.7</td>
</tr>
<tr>
<td>Nose length, in</td>
<td>23.5</td>
</tr>
</tbody>
</table>
### Ramjet Baseline Missile Definition (cont)

#### Tail (Exposed)

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Titanium</td>
</tr>
<tr>
<td>Planform area (2 panels), ft²</td>
<td>2.24</td>
</tr>
<tr>
<td>Wetted area (4 panels), ft²</td>
<td>8.96</td>
</tr>
<tr>
<td>Aspect ratio (2 panels exposed)</td>
<td>1.64</td>
</tr>
<tr>
<td>Taper ratio</td>
<td>0.70</td>
</tr>
<tr>
<td>Root chord, in</td>
<td>16.5</td>
</tr>
<tr>
<td>Span, in. (2 panels exposed)</td>
<td>23.0</td>
</tr>
<tr>
<td>Leading edge sweep, deg</td>
<td>37.0</td>
</tr>
<tr>
<td>Mean aerodynamic chord, in</td>
<td>14.2</td>
</tr>
<tr>
<td>Thickness ratio</td>
<td>0.04</td>
</tr>
<tr>
<td>Section type</td>
<td>Modified double wedge</td>
</tr>
<tr>
<td>Section leading edge total angle, deg</td>
<td>9.1</td>
</tr>
<tr>
<td>$x_{mac}$, in</td>
<td>150.3</td>
</tr>
<tr>
<td>$y_{mac}$, in (from root chord)</td>
<td>5.4</td>
</tr>
</tbody>
</table>

#### Reference values

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reference area, ft²</td>
<td>2.264</td>
</tr>
<tr>
<td>Reference length, ft</td>
<td>1.698</td>
</tr>
</tbody>
</table>
Engine Nomenclature and Flowpath Geometry for Ramjet Baseline

Subscripts
0 Free stream flow into inlet (Example, Ramjet Baseline at Mach 4, $\alpha = 0$ deg $\Rightarrow A_0 = 104$ in². Note: $A_c = 114$ in²)
1 Inlet throat (Ramjet Baseline $A_1 = A_{IT} = 41.9$ in²)
2 Diffuser exit (Ramjet Baseline $A_2 = 77.3$ in²)
3 Flame holder plane (Ramjet Baseline $A_3 = 287.1$ in²)
4 Combustor exit (Ramjet Baseline $A_4 = 287.1$ in²)
5 Nozzle throat (Ramjet Baseline $A_5 = 103.1$ in²)
6 Nozzle exit (Ramjet Baseline $A_6 = 233.6$ in²)
Ref Reference Area (Ramjet Baseline Body Cross-sectional Area, $S_{Ref} = 326$ in²)

\[
(C_{D_0})_{\text{Nose Corrected}} = (C_{D_0})_{\text{Nose Uncorrected}} \times \left( 1 - \frac{A_c}{S_{Ref}} \right)
\]

$A_c$ = Inlet capture area

$S_{Ref}$ = Reference area

Engine Station Identification

Nose Corrected Area = $A_c = 114$ in²

20.375 in

120°

$S_{Ref}$ = Reference Area

Ramjet Engine Station Identification

2/24/2008 ELF 327
Aerodynamic Characteristics of Ramjet Baseline

Source: Reference 27, based on year 1974 computer program from Reference 32.

S_{Ref} = 2.264 \text{ ft}^2
I_{Ref} = d_{Ref} = 1.698 \text{ ft}
X_{cg} @ Sta 82.5 \text{ in}
\delta = 0 \text{ deg}
Aerodynamic Characteristics of Ramjet Baseline (cont)

Pitching Moment Coefficient, $C_m$

Source: Reference 27, based on year 1974 computer program from Reference 32.
Aerodynamic Characteristics of Ramjet Baseline (cont)

\[ C_{m_{\alpha}} \sim \text{per deg} \]

\[ C_{N_\delta} \sim \text{per deg} \]

\[ C_D_0 \sim \text{per deg} \]

\[ M, \text{ Mach Number} \]

Source: Reference 27, based on year 1974 computer program from Reference 32.
Thrust Modeling of Ramjet Baseline

Note:
- Standard atmosphere
- $T = T_{\text{max}} \phi$
- $\phi = 1$ if stochiometric $(f/a = 0.0667)$
- $\alpha = 0 \, \text{deg}$

Example: $M = 3.5$, $h = 60k \, \text{ft}$, $\phi = 1$ \implies Max Thrust = 1,750 lb

Figure based on Reference 27 prediction
Specific Impulse Modeling of Ramjet Baseline

- $M$, Mach Number
- $ISP$, Specific Impulse, s

Note:
- Standard atmosphere
- $\varphi \leq 1$
- $I_{SP}$ based on Reference 27 computer prediction.

Example: $M = 3.5 \Rightarrow I_{SP} = 1,120$ s
Rocket Booster Acceleration / Performance of Ramjet Baseline

Boost Thrust ~ 1000 lb

Time ~ s

Boost Range ~ nm

h, Altitude 1,000 ft

Burnout Mach Number

h, Altitude 1,000 ft

ISP_{Booster} = 250 s

Standard atmosphere

M_L = 0.80

Constant altitude flyout
Ramjet Baseline Has Best Performance at High Altitude

Example, Mach 3 / 60k ft flyout ⇒ 445 nm. Breguet Range Prediction is $R = V I_{sp} ( L / D ) \ln \left[ \frac{W_{BC}}{W_{BC} - W_f} \right] = 2901 \left( \frac{1040}{3.15} \right) \ln \left( \frac{1739}{1739 - 476} \right) = 3039,469$ ft or 500 nm. Predicted range is 10% greater than baseline missile data.

Note: $M_L = 0.8$, Constant Altitude Fly-out
From Paredo Sensitivity, Ramjet Baseline Range Driven by $I_{SP}$, Fuel Weight, Thrust, and $C_{D0}$

Example: At Mach 3.0 / 60k ft altitude cruise, 10% increase in fuel weight $\Rightarrow$ 9.6% increase in flight range
Ramjet Baseline Flight Range Uncertainty Is +/- 7%, 1 $\sigma$

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline Value at Mach 3.0 / 60k ft</th>
<th>Uncertainty in Parameter</th>
<th>$\Delta R / R$ from Uncertainty</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Specific Impulse</td>
<td>1040 s</td>
<td>+/- 5%, 1$\sigma$</td>
<td>+/- 5%, 1$\sigma$</td>
</tr>
<tr>
<td>2. Ramjet Fuel Weight</td>
<td>476 lb</td>
<td>+/- 1%, 1$\sigma$</td>
<td>+/- 0.9%, 1$\sigma$</td>
</tr>
<tr>
<td>3. Cruise Thrust ($\phi = 0.39$)</td>
<td>458 lb</td>
<td>+/- 5%, 1$\sigma$</td>
<td>+/- 2%, 1$\sigma$</td>
</tr>
<tr>
<td>4. Zero-Lift Drag Coefficient</td>
<td>0.17</td>
<td>+/- 5%, 1$\sigma$</td>
<td>+/- 4%, 1$\sigma$</td>
</tr>
<tr>
<td>5. Lift Curve Slope Coefficient</td>
<td>0.13 / deg</td>
<td>+/- 3%, 1$\sigma$</td>
<td>+/- 1%, 1$\sigma$</td>
</tr>
<tr>
<td>6. inert Weight</td>
<td>1205 lb</td>
<td>+/- 2%, 1$\sigma$</td>
<td>+/- 0.8%, 1$\sigma$</td>
</tr>
</tbody>
</table>

Level of Maturity Based on Flight Demo of Prototype, Subsystem Tests, and Integration

- Wind tunnel tests
- Direct connect, freejet, and booster firing propulsion tests
- Structure test
- Mock-up
- Hardware-in-loop simulation
- Flight Test

Total Flight Range Uncertainty at Mach 3.0 / 60k ft Flyout

$\Delta R / R = \left[ (\Delta R / R)_1^2 + (\Delta R / R)_2^2 + (\Delta R / R)_3^2 + (\Delta R / R)_4^2 + (\Delta R / R)_5^2 + (\Delta R / R)_6^2 \right]^{1/2} = +/- 6.9\%, 1\sigma$

$R = 445$ nm +/- 31 nm, 1$\sigma$
Sizing Examples

- Rocket Baseline Missile
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- Ramjet Baseline Missile
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- Computer Aided Conceptual Design Sizing Tools

- Soda Straw Rocket Design, Build, and Fly
**Slurry Fuel and Efficient Packaging Provide Extended Range Ramjet**

<table>
<thead>
<tr>
<th>propulsive / Configuration</th>
<th>fuel type / volumetric performance (btu / in3) / density (lb / in3)</th>
<th>fuel volume (in3) / fuel weight (lb)</th>
<th>ISP (s) / cruise range at Mach 3.5, 60k ft (nm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Liquid Fuel Ramjet</td>
<td>RJ-5 / 581 / 0.040</td>
<td>11900 / 476</td>
<td>1120 / 390</td>
</tr>
<tr>
<td>Ducted Rocket (Low Smoke)</td>
<td>Solid Hydrocarbon / 1132 / 0.075</td>
<td>7922 / 594</td>
<td>677 / 294</td>
</tr>
<tr>
<td>Ducted Rocket (High Performance)</td>
<td>Boron / 2040 / 0.082</td>
<td>7922 / 649</td>
<td>769 / 366</td>
</tr>
<tr>
<td>Solid Fuel Ramjet</td>
<td>Boron / 2040 / 0.082</td>
<td>7056 / 579</td>
<td>1170 / 496</td>
</tr>
<tr>
<td>Slurry Fuel Ramjet</td>
<td>40% JP-10, 60% boron carbide / 1191 / 0.050</td>
<td>11900 / 595</td>
<td>1835 / 770</td>
</tr>
</tbody>
</table>

**Note:**

Flow Path

Available Fuel

\[ R_{cruise} = V I_{SP} ( L / D ) \ln \left[ \frac{W_{BC}}{W_{BC} - W_f} \right] \]
Sizing Examples

- Rocket Baseline Missile
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- Ramjet Baseline Missile
  - Range robustness
  - Propulsion and fuel alternatives
  - Velocity control

- Computer Aided Conceptual Design Sizing Tools

- Soda Straw Rocket, Design, Build, and Fly
Example of Ramjet Velocity Control Through Fuel Control

Note: Ramjet baseline, vertical impact at sea level, steady state velocity at impact, T = thrust, W = weight, D = drag, \( W_{BO} \) = burnout weight, \( C_{D0} \) = zero-lift drag coefficient, \( M_i \) = impact Mach number, \( T_{required} \) = required thrust for steady state flight, \( w_f \) = fuel flow rate, \( I_{SP} \) = specific impulse, \( \phi \) = equivalence ratio (\( \phi = 1 \) stochiometric)
Sizing Examples

- Rocket Baseline Missile
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- Ramjet Baseline Missile
  - Range robustness
  - Propulsion and fuel alternatives
  - Surface impact velocity

- Computer Aided Conceptual Design Sizing Tools

- Soda Straw Rocket Design, Build, and Fly
Objective of Conceptual Design

- Search Broad Solution Space
- Iterate to Design Convergence

Characteristics of Good Conceptual Design Sizing Code

- Fast Turnaround Time
- Easy to Use
- Directly Connect Predictions of Aeromechanics and Physical Parameters to Trajectory Code
- Simple, Physics Based Methods
- Includes Most Important, Driving Parameters
- Provides Insight into Relationships of Design Parameters
- Stable Computation
- Imbedded Baseline Missile Data
- Human Designer Makes the Creative Decisions
Example of DOS-Based Conceptual Sizing Computer Code – ADAM

- Conceptual Sizing Computer Program
  - Advanced Design of Aerodynamic Missiles (ADAM)
  - PC compatible
  - Written in DOS

- Aerodynamic Module Based on NACA 1307 Calculates
  - Static and dynamic stability derivatives
  - Control effectiveness and trim conditions

- 3, 4, 5, and 6-DOF Simulation Modules
  - Proportional guidance
  - Input provided automatically by aerodynamic module

- Configurations Benchmarked with Wind Tunnel Data

- Greater than 50 Input Parameters Available
  - Defaults to benchmark configuration(s)
Example of Spreadsheet Based Conceptual Sizing Computer Code - TMD Spreadsheet

◆ Conceptual Sizing Computer Code
  ◆ Tactical Missile Design (TMD) Spreadsheet
  ◆ PC compatible
  ◆ Windows Excel spreadsheet

◆ Based on Tactical Missile Design Short Course and Textbook
  ◆ Aerodynamics
  ◆ Propulsion
  ◆ Weight
  ◆ Flight trajectory
  ◆ Measures of merit
Example of Spreadsheet Based Conceptual Sizing Computer Code, TMD Spreadsheet

Define Mission Requirements [Flight Performance (R_{Max}, R_{Min}, V_{AVG}), MOM, Constraints]

Establish Baseline (Rocket, Ramjet)

Aerodynamics Input (d, l, I_n, A, c, t, x_{cg})
Aerodynamics Output [C_{D_0}, C_N, x_{ac}, C_{m_k}, L/D, S_T]

Propulsion Input (p_c, \epsilon, c^*, A_b, A_p, A_0, H_t, \phi, T_4, Inlet Type)
Propulsion Output [I_{sp}, T_{cruise}, p_{l_2}/p_{l_0}, w, T_{boost}, T_{sustain}, \Delta V_{Boost}]

Weight Input (W_L, W_P, \rho, \sigma_{max})
Weight Output [W_L, W_P, h, dT/dt, T, t, \sigma_{buckling}, M_B, \sigma, W_{subsystems}, x_{cg}, I_y]

Trajectory Input (h_i, V_i, Type (cruise, boost, coast, ballistic, turn, glide))
Trajectory Output (R, h, V, and \gamma versus time)

Meet Performance?

No \[ R_{Max}, R_{Min}, V_{AVG} \]

No \[ p_{Blast}, P_K, n_{Hit}, V_{fragments}, P_{KE}, KE_{Warhead}, \tau_{Total}, \sigma_{HE}, \sigma_{MAN}, R_{detect}, C_{SDD}, C_{1000th}, C_{unit} \]

Yes

Measures of Merit and Constraints

Yes
Example of TMD Spreadsheet Sizing Code
Verification: Air-to-Air Range Requirement

♦ Example Launch Conditions
  ♦ h_L = 20k ft
  ♦ M_L = 0.8

♦ Example Requirement
  ♦ R_F = 6.7 nm with t_f < 24.4 s

♦ Solutions for Rocket Baseline
  ♦ ADAM: R_F = 6.7 nm at t_f = 18 s
  ♦ TMD Spreadsheet: R_F = 6.7 nm at t_f = 19 s
  ♦ 3 DOF using wind tunnel aero data: R_F = 6.7 nm at t_f = 21 s

♦ Differences in Flight Time to 6.7 nm Mostly Due to Zero-Lift Drag Coefficient. For Example:
  ♦ ADAM prediction at Mach 2.0: (C_D0)_coast = 0.53
  ♦ TMD Spreadsheet prediction at Mach 2.0: (C_D0)_coast = 0.57
  ♦ Wind tunnel aero data at Mach 2.0: (C_D0)_coast = 1.05

♦ Wind Tunnel Data / Baseline Missile Data Correction Required to Reduce Uncertainty in C_D0
Sizing Examples

- **Rocket Baseline Missile**
  - Standoff range requirement
  - Wing sizing requirement
  - Multi-parameter harmonization
  - Lofted range comparison

- **Ramjet Baseline Missile**
  - Range robustness
  - Propulsion and fuel alternatives
  - Surface impact velocity

- **Computer Aided Conceptual Design Sizing Tools**

- **Soda Straw Rocket Design, Build, and Fly**
Example of Design, Build, and Fly Customer Requirements

- **Objective** – Design, Build, and Fly Soda Straw Rocket with:
  - Flight Range Greater Than 90 ft
  - Weight Less Than 2 g

- **Furnished Property**
  - Launch System
  - Distance Measuring Wheel
  - Weight Scale
  - Micrometer Scale
  - Engineer’s Scale
  - Scissors

- **Furnished Material**
  - 1 “Giant” Soda Straw: 0.28 in Diameter by 7.75 in Length, Weight = 0.6 g
  - 1 Strip Tabbing: ½ in by 6 in, Weight = 1.4 g
  - 1 Ear Plug: 0.33 – 0.45 in Diameter by 0.90 in Length, Weight = 0.6 g
  - 1 “Super Jumbo” Soda Straw: 0.25 in Diameter by 7.75 in Length
Example of Design, Build, and Fly Customer Requirements (cont)

- Furnished Property Launch System with Specified Launch Conditions
  - Launch Tube Diameter: 0.25 in
  - Launch Tube Length (e.g., 6 in)
  - Launch Pressure (e.g., 30 psi)
  - Launch Elevation Angle (e.g., 40 deg)

- Predict Flight Trajectory Range and Compare with Test
Soda Straw Rocket Launcher and Targeting

1. Pump
2. Pressure Tank
3. Air Hose
4. Pressure Gauge
5. Launch Switch
6a. Solenoid Valve Launcher (0.025 s average response)
6b. Manual Valve Launcher (0.1 s average response)
7. Inclinometer
8. Launch Tube
9. Rocket on Launcher
10. Laser Pointer Targeting Device
11. Rockets with Various Length, Tail Geometry, Nose Geometry, and Other Surfaces
It Is Easy to Make a Soda Straw Rocket

1. Cut Large Diameter “Giant” Soda Straw to Desired Length

2. Twist and Squeeze Ear Plug to Fit Inside Soda Straw

3. Slide Ear Plug Inside Soda Straw

4. Cut Adhesive Tabs to Desired Height and Width of Surfaces

5. Apply Adhesive Tabs to Soda Straw

6. Wrap Front of Ear Plug and Straw with Tape

7. Slide Giant Soda Straw Rocket Over Smaller Diameter “Super Jumbo” Soda Straw Launch Tube
Soda Straw Rocket Baseline Configuration

Ear Plug     Soda Straw                              Strip Tabbing

\[ l = 7.0 \text{ in} \]
\[ l_c = 6.0 \text{ in} \]

0.28 in
0.25 in
0.5 in
0.28 in
0.5 in
## Soda Straw Rocket Baseline Weight and Balance

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight, g</th>
<th>cg Station, in</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nose (Plug)</td>
<td>0.6</td>
<td>0.5</td>
</tr>
<tr>
<td>Body (Soda Straw)</td>
<td>0.5</td>
<td>3.5</td>
</tr>
<tr>
<td>Fins (Four)</td>
<td>0.5</td>
<td>6.75</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>1.6</strong></td>
<td><strong>3.39</strong></td>
</tr>
</tbody>
</table>
## Soda Straw Rocket Baseline Definition

### Body

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material Type</td>
<td>HDPE Plastic</td>
</tr>
<tr>
<td>Material density, lb / in³</td>
<td>0.043</td>
</tr>
<tr>
<td>Material strength, psi</td>
<td>4,600</td>
</tr>
<tr>
<td>Thickness, in</td>
<td>0.004</td>
</tr>
<tr>
<td>Length, in</td>
<td>7.0</td>
</tr>
<tr>
<td>Diameter, in</td>
<td>0.28</td>
</tr>
<tr>
<td>Fineness ratio</td>
<td>25.0</td>
</tr>
<tr>
<td>Nose fineness ratio</td>
<td>0.5</td>
</tr>
</tbody>
</table>

### Fins

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Material</td>
<td>Plastic</td>
</tr>
<tr>
<td>Planform area, in² (2 panels exposed)</td>
<td>0.25</td>
</tr>
<tr>
<td>Wetted area, in² (4 panels)</td>
<td>1.00</td>
</tr>
<tr>
<td>Aspect ratio (2 panels exposed)</td>
<td>1.00</td>
</tr>
<tr>
<td>Taper ratio</td>
<td>1.0</td>
</tr>
<tr>
<td>Chord, in</td>
<td>0.5</td>
</tr>
<tr>
<td>Span (exposed), in</td>
<td>0.5</td>
</tr>
<tr>
<td>Span (total including body), in</td>
<td>0.78</td>
</tr>
<tr>
<td>Leading edge sweep, deg</td>
<td>0</td>
</tr>
<tr>
<td>Xmac, in</td>
<td>6.625</td>
</tr>
<tr>
<td>Nose</td>
<td>Foam</td>
</tr>
<tr>
<td>------</td>
<td>------</td>
</tr>
<tr>
<td>Material Type</td>
<td>Foam</td>
</tr>
<tr>
<td>Material density, lb / in³</td>
<td>0.012</td>
</tr>
<tr>
<td>Average diameter</td>
<td>0.39 in</td>
</tr>
<tr>
<td>Length</td>
<td>0.90 in</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Reference Values</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Reference area, in²</td>
<td>0.0616</td>
</tr>
<tr>
<td>Reference length, in</td>
<td>0.28</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Thrust Performance</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Inside cavity length, in</td>
<td>6.0</td>
</tr>
<tr>
<td>Typical Pressure, psi</td>
<td>30</td>
</tr>
<tr>
<td>Maximum thrust @ 30 psi pressure, lb</td>
<td>1.47</td>
</tr>
<tr>
<td>Time constant, s (standard temperature)</td>
<td>0.025</td>
</tr>
</tbody>
</table>
For body-tail geometry, static margin given by

\[
\frac{(x_{AC} - x_{CG})}{d} = - \left\{ \left( \frac{C_{N_{\alpha}}}{d} \right)_B \left[ \frac{x_{CG} - (x_{AC})_B}{d} \right] + \left( \frac{C_{N_{\alpha}}}{d} \right)_T \left[ \frac{x_{CG} - (x_{AC})_T}{d} \right] \right\} \frac{S_T}{S_{Ref}} \frac{S_{Ref}}{S_{Ref}}
\]

For baseline soda straw configuration

- \(x_{CG} = 3.39\) in, \(d = 0.28\) in, \(C_{N_{\alpha}})_B = 2\) per rad, \(S_T = 0.25\) in\(^2\), \(S_{Ref} = 0.0616\) in\(^2\)
- \((x_{AC})_B = \left( \frac{x_{AC}}{l_N} \right) l_N = 0.63(0.14) = 0.09\) in
- \((C_{N_{\alpha}})_T = \pi A_T / 2 = \pi (1) / 2 = 1.57\)
- \((x_{AC})_T = 6.5 + 0.25(\text{cmac})_T = 6.63\)

Substituting

\[
\frac{(x_{AC} - x_{CG})}{d} = - \left\{ 2 (3.39 - 0.09) / 0.28 + [1.57 (3.39 - 6.63) / 0.28] [(0.25) / 0.0616] \right\} / [2 + 1.57 (0.25) / 0.0616] = 6.00\) (statically stable)
\]

\(x_{AC} = 6.00 (0.28) + 3.39 = 5.07\) in from nose
Soda Straw Rocket Has High Acceleration Boost Performance

Thrust (T) from Pressurized Tube of Area A

\[ T = (p - p_0)A = p_{\text{gauge}} \left(1 - e^{-t/\tau}\right)A \]

\[ a \approx \frac{32.2T}{W}, \ V = \int a \ dt, \ s = \int V \ dt \]

Example:
Assume \( p_{\text{gauge}} = 30 \text{ psi} \), \( l_t = 6 \text{ in} \), \( \tau = 0.025 \text{ s} \) (Average for Solenoid Valve), \( s = l_c = 6 \text{ in} \)

Thrust Equation Is:

\[ T = 30 \left(1 - e^{-t/0.025}\right) \times 0.0491 = 1.4726 \times \left(1 - e^{-40.00t}\right) \]

Note: Actual Boost Thrust Lower (Pressure Loss, Boundary Layer, Launch Tube Leakage, Launch Tube Friction)

Equations for Acceleration (a), Velocity (V), and Distance (s) During Boost Are:

\[ a \approx \frac{32.2T}{W} = 32.2 \left(1 - e^{-40.00t}\right) / 0.00352 = 13471.1 \left(1 - e^{-40.00t}\right) \]

\[ V = \int a \ dt = 13471.1t + 336.78e^{-40.00t} - 336.78 \]

\[ s = \int V \ dt = 6735.57t^2 - 8.419e^{-40.00t} - 336.78t + 8.419 \]

End of Boost Conditions Are:

\[ s = l_c = 6 \text{ in} = 0.500 \text{ ft} \Rightarrow t = 0.0188 \text{ s} \]

\[ a = 7123 \text{ ft/s}^2 = 221 \text{ g} \]

\[ V = 75.2 \text{ ft/s} \]

\[ q = \frac{1}{2} \rho V^2 = \frac{1}{2} \left(0.002378\right) (75.2)^2 = 6.72 \text{ psf} \]

\[ M = \frac{V}{c} = 75.2 / 1116 = 0.0674 \]
Most of the Soda Straw Rocket Drag Coefficient Is from Body Skin Friction

\[
C_{D_0} = (C_{D_0})_{\text{Body,Friction}} + (C_{D_0})_{\text{Base,Coast}} + (C_{D_0})_{\text{Tail,Friction}} \\
= 0.053 \left( \frac{1}{d} \right) \left( \frac{M}{(q \, I)} \right)^{0.2} + 0.12 + n_T \left( \frac{0.0133}{(M/(q \, c_{mac})^{0.2}} \right) \left( 2 \, \frac{S_T}{S_{Ref}} \right)
\]

Example: \( V = 75.2 \text{ fps}, S_T = 0.00174 \text{ ft}^2, S_{Ref} = 0.000428 \text{ ft}^2 \Rightarrow \frac{S_T}{S_{Ref}} = 4.07 \)

Compute:

\[
\begin{align*}
C_{D_0} & = 0.053 \left( 25.0 \right) \left( \frac{0.0674}{(6.72)(0.583)} \right)^{0.2} + \\
& \quad + 0.12 + 2 \left( \frac{0.0133}{(6.72)(0.0417)} \right)^{0.2} \left( 4.07 \right) = 0.58 + 0.12 + 0.16 = 0.86
\end{align*}
\]

Note:

- Above Drag Coefficient Not Exact
- Based on Assumption of Turbulent Boundary Layer
- Soda Straw Rocket Small Size and Low Velocity \( \Rightarrow \) Laminar Boundary Layer \( \Rightarrow \) Large Boundary Layer Thickness on Aft Body at Tails

Compute Drag Force:

\[
D_{\text{max}} = C_D \, q_{\text{max}} \, S_{\text{Ref}} = 0.86 \left( 6.72 \right) \left( 0.000428 \right) = 0.00247 \text{ lb}
\]

Compare Drag Force to Weight:

\[
\frac{D_{\text{max}}}{W} = \frac{0.00247}{0.00352} = 0.70
\]

Note: Drag Force Smaller Than Weight
Soda Straw Rocket Baseline Has a Ballistic Flight Range Greater Than 90 Feet

\[ R_x = \left\{ 2 \frac{W \cos \gamma_i}{\left[ \frac{g_c \rho S_{Ref} C_{D0}}{V_i} \right]} \right\} \ln \left\{ 1 + \frac{t}{2 \frac{W}{\left[ \frac{g_c \rho S_{Ref} C_{D0}}{V_i} \right]}} \right\} \]

\[ h = \left\{ 2 \frac{W \sin \gamma_i}{\left[ \frac{g_c \rho S_{Ref} C_{D0}}{V_i} \right]} \right\} \ln \left\{ 1 + \frac{t}{2 \frac{W}{\left[ \frac{g_c \rho S_{Ref} C_{D0}}{V_i} \right]}} \right\} + h_i - \frac{gt^2}{2} \]

Example, Assume \( l_t = 6 \text{ in}, \ p_{gauge} = 30 \text{ psi}, \ \gamma_i = 30 \text{ deg}, \ \tau = 0.025 \text{ sec}, \ \text{Soda Straw Baseline}, \ t = t_{impact} = 1.8 \text{ s} \)

\[ \text{Horizontal Range At Impact} = R_x = \left\{ 2 \left( \frac{0.00352}{32.2 \left( 0.002378 \right) \left( 0.000428 \right) \left( 0.86 \right)} \right) \cos \gamma_i \right\} \ln \left\{ 1 + \frac{t}{2 \frac{0.00352}{32.2 \left( 0.002378 \right) \left( 0.000428 \right) \left( 0.86 \right) \left( 75.2 \right)}} \right\} \]

\[ = 249.8 \cos \gamma_i \ln \left( 1 + 0.301 t \right) \]

\[ = 249.8 \left( 0.866 \right) \ln \left[ 1 + 0.301 \left( 1.8 \right) \right] = 93.7 \text{ ft} \]

\[ \text{Height At Impact} = h = \left\{ 2 \left( \frac{0.00352}{32.2 \left( 0.002378 \right) \left( 0.000428 \right) \left( 0.86 \right)} \right) \sin \gamma_i \right\} \ln \left\{ 1 + \frac{t}{2 \left( \frac{0.00352}{32.2 \left( 0.002378 \right) \left( 0.000428 \right) \left( 0.86 \right) \left( 75.2 \right)}} \right\} + h_i - 32.2 t^2 / 2 \]

\[ = 249.8 \sin \gamma_i \ln \left( 1 + 0.301 t \right) + h_i - 32.2 t^2 / 2 = 249.8 \left( 0.5 \right) \ln \left[ 1 + 0.301 \left( 1.8 \right) \right] + h_i - 32.2 \left( 1.2 \right)^2 / 2 \]

\[ = h_i + 1.9 \text{ ft} \]
Soda Straw Rocket Range Driven by Inside Chamber Length and Launch Angle

Note: Decreased chamber length \( \Rightarrow \) shorter duration thrust (decreased total impulse) \( \Rightarrow \) decreased end-of-boost velocity

Soda Straw Rocket Baseline:

\[ W = \text{Weight} = 0.00423 \text{ lb} \]

\[ l_c = \text{inside chamber length} = 6 \text{ in} \]

\[ \tau = \text{Time constant to open solenoid valve} = 0.025 \text{ s} \]

\[ p_{\text{gauge}} = \text{gauge pressure} = 30 \text{ psi} \]

\[ \gamma_l = \text{Initial / launch angle angle} = 30 \text{ deg} \]

\[ l_t = 7 \text{ in} \]

\[ V = \text{Launch velocity} = 75.2 \text{ fps} \]

\[ C_{D0} = \text{Zero-lift drag coefficient} = 0.86 \]

\[ t_{\text{impact}} = \text{Time from launch to impact} = 1.8 \text{ s} \]

\[ R_x = \text{Horizontal range} = 94 \text{ ft} \]
Soda Straw Rocket Baseline Flight Range
Uncertainty is +/- 2.4%, 1σ

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Baseline Value</th>
<th>Uncertainty in Parameter</th>
<th>ΔR / R Due to Uncertainty</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Inside Chamber Length</td>
<td>6 in</td>
<td>+/- 2%, 1σ</td>
<td>+/- 1.5%, 1σ</td>
</tr>
<tr>
<td>2. Launch Angle</td>
<td>30 deg</td>
<td>+/- 3%, 1σ</td>
<td>+/- 1.7%, 1σ</td>
</tr>
<tr>
<td>3. Gauge Pressure</td>
<td>30 psi</td>
<td>+/- 3%, 1σ</td>
<td>+/- 0.5%, 1σ</td>
</tr>
<tr>
<td>4. Weight</td>
<td>1.6 g</td>
<td>+/- 6%, 1σ</td>
<td>+/- 0.4%, 1σ</td>
</tr>
<tr>
<td>5. Solenoid Time Constant</td>
<td>0.025 s</td>
<td>+/- 20%, 1σ</td>
<td>+/- 0.2%, 1σ</td>
</tr>
<tr>
<td>6. Zero-Lift Drag Coefficient</td>
<td>0.86</td>
<td>+/- 20%, 1σ</td>
<td>+/- 0.2%, 1σ</td>
</tr>
</tbody>
</table>

♦ Estimate of Level of Maturity / Uncertainty of Soda Straw Rocket Baseline Parameters Based on
  ♦ Wind tunnel test
  ♦ Thrust static test
  ♦ Weight measurement
  ♦ Prediction methods

♦ Total Flight Range Uncertainty for 30 psi launch at 30 deg
  ♦ ΔR / R = [ (ΔR / R)₁² + (ΔR / R)₂² + (ΔR / R)₃² + (ΔR / R)₄² + (ΔR / R)₅² + (ΔR / R)₆² ]¹/² = +/- 2.4%, 1σ
  ♦ R = 94 ft +/- 2.3 ft, 1σ
### House of Quality Translates Customer Requirements into Engineering Emphasis

#### Table: Design Characteristics Sensitivity Matrix

<table>
<thead>
<tr>
<th></th>
<th>Flight Range</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chamber Length</td>
<td>7</td>
<td>3</td>
</tr>
<tr>
<td>Tail Planform Area</td>
<td>8</td>
<td>2</td>
</tr>
<tr>
<td>Weight</td>
<td>6</td>
<td>4</td>
</tr>
</tbody>
</table>

\[
\begin{align*}
74 &= (7 \times 8 + 3 \times 6) \\
26 &= (7 \times 2 + 3 \times 4)
\end{align*}
\]

#### Note on Design Characteristics Sensitivity Matrix: (Room 5):
- **++ Strong Synergy**
- **+ Synergy**
- **0 Near Neutral Synergy**
- **- Anti-Synergy**
- **- - Strong Anti-Synergy**

*Note: Based on House of Quality, inside chamber length most important design parameter.*
DOE Explores the Broad Possible Design Space with a Reasonably Small Set of Alternatives

Design Space for Design of Experiments (DOE)

<table>
<thead>
<tr>
<th>Engineering Characteristics Range</th>
<th>( l_c ), Inside Chamber Length, in</th>
<th>( S_T ), Tail Planform Area, in²</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lower Value</td>
<td>4</td>
<td>0.125</td>
</tr>
<tr>
<td>Upper Value</td>
<td>6</td>
<td>0.25</td>
</tr>
</tbody>
</table>

Full Factorial DOE Based on Upper / Lower Values of \( k = 2 \) Parameters:
Number of Combinations = \( 2^k = 2^2 = 4 \)

<table>
<thead>
<tr>
<th>Concept</th>
<th>Sketch</th>
<th>( l_c ), Inside Chamber Length, in</th>
<th>( S_T ), Tail Planform Area, in²</th>
</tr>
</thead>
<tbody>
<tr>
<td>“Big Kahuna”</td>
<td>![Sketch]</td>
<td>6</td>
<td>0.25</td>
</tr>
<tr>
<td>“Shorty”</td>
<td>![Sketch]</td>
<td>4</td>
<td>0.25</td>
</tr>
<tr>
<td>“Stiletto”</td>
<td>![Sketch]</td>
<td>6</td>
<td>0.125</td>
</tr>
<tr>
<td>“Petite”</td>
<td>![Sketch]</td>
<td>4</td>
<td>0.125</td>
</tr>
</tbody>
</table>

Note: DOE concepts should emphasize customer driving requirements and the driving engineering characteristics.
Engineering Experience Should Guide the DOE Set of Alternatives and the Preferred Design

As an Example, for the Soda Straw Rocket, from Experience We Know That

- Soda Straw Rocket Must Fit on Launcher
- Maximum Boost Velocity Occurs When Chamber Length = Launch Tube Length
- Three or Four Tails Best for Stability
- Tails That Are Too Small May Result in an Unstable Flight
- Tails That Are Too Large Add Weight and Cause Trajectory Dispersal
- Canards Require Larger Tails for Stability, Add Weight, and Cause Trajectory Dispersal
- Wings Add Weight, Add Drag, and Cause Trajectory Dispersal
Engineering Experience Should Guide the DOE Set of Alternatives and Preferred Design (cont)

As an Example, Soda Straw Rocket Geometry Should Be Comparable to an Operational Rocket with Near-Neutral Static Stability (e.g., Hydra70)

<table>
<thead>
<tr>
<th>Concept</th>
<th>Sketch</th>
<th>I / d, Total Length / Diameter</th>
<th>b / d, Total Tail Span / Diameter</th>
<th>c / d, Tail Chord / Diameter</th>
</tr>
</thead>
<tbody>
<tr>
<td>“Big Kahuna”</td>
<td></td>
<td>25</td>
<td>2.79</td>
<td>2</td>
</tr>
<tr>
<td>“Shorty”</td>
<td></td>
<td>17.9</td>
<td>2.79</td>
<td>2</td>
</tr>
<tr>
<td>“Stiletto”</td>
<td></td>
<td>25</td>
<td>1.89</td>
<td>2</td>
</tr>
<tr>
<td>“Petite”</td>
<td></td>
<td>17.9</td>
<td>1.89</td>
<td>2</td>
</tr>
<tr>
<td>Hydra 70</td>
<td></td>
<td>15.1</td>
<td>2.66</td>
<td>1</td>
</tr>
</tbody>
</table>

Note: For a subsonic rocket with the center-of-gravity in the center of the rocket, slender body theory and slender surface theory give total tail span and chord for neutral stability of $b_{\text{Neutral Stability}} \approx 2 \ d$ and $c_{\text{Neutral Stability}} > \approx d$ respectively.
As an Example, for the Soda Straw Rocket Design We Should

- Reflect Customer Emphasis of Requirements for
  - Range
  - Weight
- Provide Balanced Emphasis of Most Important Engineering Characteristics
  - Chamber Length
  - Tail Size / Span
Summary of Sizing Examples

♦ Rocket Powered Missile (Sparrow Derived Baseline)
  ♦ Standoff range requirement
  ♦ Wing area sizing requirements for maneuverability, turn rate, and turn radius
  ♦ Multi-parameter harmonization
  ♦ Ballistic versus lofted glide flight range

♦ Ramjet Powered Missile (ASALM Derived Baseline)
  ♦ Robustness in range uncertainty
  ♦ Propulsion and fuel alternatives
  ♦ Surface target impact velocity

♦ Computer Aided Sizing Tools for Conceptual Design
  ♦ ADAM
    ♦ Analytical prediction of aerodynamics
    ♦ Numerical solution of equations of motion
Summary of Sizing Examples (cont)

- Computer Aided Sizing Tools for Conceptual Design (cont)
  - TMD analytical sizing spreadsheet (based on this text)
    - Analytical prediction of aero, propulsion, and weight
    - Closed form analytical solution of simplified equations of motion

- Soda Straw Rocket Design, Build, and Fly
  - Static margin
  - Drag
  - Performance
  - Sensitivity study
  - House of Quality
  - Design of Experiment (DOE)

- Discussion / Questions?
- Classroom Exercise (Appendix A)
Sizing Examples Problems

1. Required flight range is shorter for a head-on intercept and it is longer for a t____ c____ intercept.
2. The rocket baseline center-of-gravity moves f______ with motor burn.
3. The rocket baseline is an a_______ airframe.
4. The rocket baseline thrust profile is b____ s______.
5. The rocket baseline motor case and nozzle are made of s____.
6. The rocket baseline flight range is driven by $I_{sp}$, propellant weight fraction, drag, and s_____ m_____.
7. Contributors to the maneuverability of the rocket baseline are its body, tail, and w____.
8. Although the rocket baseline has sufficient g’s and turn rate to intercept a maneuvering aircraft, it needs a smaller turn r_____.
9. Compared to a co-altitude trajectory, the rocket baseline has extended range with a l_____ glide trajectory.
10. The ramjet baseline has a c____ inlet.
11. The Mach 4 ramjet baseline has a t_______ airframe.
Sizing Examples Problems (cont)

12. Although the ramjet baseline combustor is a nickel-based super alloy, it requires insulation, due high temperature. The super alloy is insulating.

13. The flight range of the ramjet baseline is driven by $I_{sp}$, weight, thrust, zero-lift coefficient, and the weight fraction of fuel.

14. Extended range for the ramjet baseline would be provided by more efficient packaging of subsystems and the use of special fuels.

15. A conceptual design sizing code should be based on the simplicity, speed, and robustness of parametric based methods.

16. The House of Quality room for design characteristics weighted importance indicates which engineering design characteristics are most important in meeting the customer requirements.

17. Paredo sensitivity identifies the design parameters that are most influential.

18. DOE concepts should emphasize the customer driving requirements and the driving engineering characteristics.

19. If the total tail span (including body diameter) is twice the body diameter, the missile is approximately non-spherical.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Relationship of Technology Assessment / Roadmap to the Development Process

- Technology Roadmap Establishes Time-phased Interrelationships for
  - Technology development and validation tasks
  - Technology options
  - Technology goals
  - Technology transition (ATD, ACTD, DemVal, PDRR, SDD)

- Technology Roadmap Identifies
  - Key, enabling, high payoff technologies
  - Technology drivers
  - Key decision points
  - Critical paths
  - Facility requirements
  - Resource needs
Relationship of Design Maturity to the US Research, Technology, and Acquisition Process

- **Basic Research**
  - 6.1: ~ $0.1B
  - 6.2: ~ $0.3B

- **Exploratory Development**
  - 6.3: ~ $0.9B
  - 6.4: ~ $0.5B

- **Advanced Development**
  - 6.5: ~ $1.0B
  - Production: ~ $6.1B
  - System Upgrades: ~ $1.2B

- **Demonstration & Validation**
  - System Development and Demonstration

- **Technology Development**
  - ~ 10 Years

- **Prototype Demonstration**
  - ~ 4 Years

- **Full Scale Development**
  - ~ 5 Years

- **Limited**
  - 2 Years

- **First Block**
  - ~ 5 Years

- **Production Upgrades**
  - ~ 5-15 Years

**Maturity Level**
- Conceptual Design
  - Drawings (type): < 10 (subsystems)
- Preliminary Design
  - Drawings (type): < 100 (components)
- Detail Design
  - Drawings (type): > 100 (parts)
- Production Design
  - Drawings (type): > 1000 (parts)

**Note:**
- Total US DoD Research and Technology for Tactical Missiles ≈ $1.8 Billion per year
- Total US DoD Acquisition (SDD + Production + Upgrades) for Tactical Missiles ≈ $8.3 Billion per year
- Tactical Missiles ≈ 11% of U.S. DoD RT&A budget
- US Industry IR&D typically similar to US DoD 6.2 and 6.3A
Technology Readiness Level (TRL) Indicates the Maturity of Technology

- **TRL 1-3**
  - Category 6.1
  - Basic research

- **TRL 4**
  - Category 6.2A
  - Exploratory development of a component, conceptual design studies, and prediction methods

- **TRL 5**
  - Category 6.2B
  - Exploratory development of a subsystem

- **TRL 6**
  - Category 6.3
  - Advanced tech demo of a subsystem

- **TRL 7**
  - Category 6.4
  - Prototype demonstration

Initial assessment → component test → subsystem test → integrated subsystems → integrated missile
Conceptual Design Has Broad Alternatives While Detail Design Has High Definition

![Graph showing the typical number of alternative concepts or number of design drawings over time.](image)

- **Number of Concepts**
- **Number of Drawings**

- **Time (Years)**: 0, 5, 10, 15
- **Typical Number of Alternative Concepts or Number of Design Drawings**: 1, 10, 100, 1000

Timeline:
- **Conceptual Design**
- **Preliminary Design**
- **Detail Design**
- **Production Design**
US Tactical Missile Follow-On Programs Occur about Every 24 Years

- **Short Range ATA, AIM-9, 1949 - Raytheon**
- **Medium Range ATA, AIM-7, 1951 - Raytheon**
- **Anti-radar ATS, AGM-45, 1961 - TI**
- **Long Range STA, MIM-104, 1966 - Raytheon**
- **Man-portable STS, M-47, 1970 - McDonnell Douglas**
- **Long Range STS, BGM-109, 1972 - General Dynamics**
- **Long Range ATS, AGM-86, 1973 - Boeing**
- **Medium Range ATS, AGM-130, 1983 - Rockwell**
- **Javelin (gunner survivability, lethality, weight), 1989 - TI**
- **AIM-9X (maneuverability), 1996 - Hughes**
- **AIM-120 (autonomous, speed, range, weight), 1981 - Hughes**
- **AGM-88 (speed, range), 1983 - TI**
- **PAC-3 (accuracy), 1992 - Lockheed Martin**
- **Long Range ATS, AGM-129 (RCS), 1983 - General Dynamics**
- **AGM-130 (RCS), 1983 - General Dynamics**
- **Medium Range ATS, AGM-130, 1983 - Rockwell**
- **JASSM (cost, range, observables), 1999 - LM**
- **Hypersonic Missile, > 2007**
- **Hypersonic Missile > 2007**

### Timeline

- **1950 - 1965:** Short Range ATA, AIM-9, 1949 - Raytheon
- **1965 - 1970:** Medium Range ATA, AIM-7, 1951 - Raytheon
- **1970 - 1975:** Anti-radar ATS, AGM-45, 1961 - TI
- **1975 - 1980:** Long Range STA, MIM-104, 1966 - Raytheon
- **1985 - 1990:** Long Range STS, BGM-109, 1972 - General Dynamics
- **1990 - 1995:** Long Range ATS, AGM-86, 1973 - Boeing
- **1995 - > 2000:** Medium Range ATS, AGM-130, 1983 - Rockwell

**Year Entering SDD**

- 2/24/2008
Missile Design Validation / Technology Development Is an Integrated Process

- Rocket Static
- Turbojet Static
- Ramjet Tests
- Direct Connect-Freejet

- Structure Test

- Wind Tunnel Tests

- Hardwaste In-Loop Simulation

- Propulsion Model

- Model Digital Simulation

- IM Tests

- Lab Tests

- Flight Test Progression
  - Captive Carry, Jettison, Separation, Unpowered Guided Flights, Powered Guided Flights, Live Warhead Flights

- Lab Tests

- Tower Tests

- Environment Tests
  - Vibration
  - Temperature

- Autopilot / Electronics

- Witness / Arena Tests

- Environment Tests

- IM Tests

- Sled Tests

- Sensors

- Seeker

- Actuators / Initiators

- Warhead

- Ballistic Tests

- IM Tests

- Sled Tests

Propulsion

Airframe

Guidance and Control

Power Supply

Warhead

Aero Model

Wind Tunnel Tests

Model Digital Simulation

Lab Tests

Tower Tests

Environment Tests

IM Tests

Sled Tests
Examples of Missile Development Tests and Facilities

- Airframe Wind Tunnel Test
- Propulsion Static Firing with TVC
- Propulsion Direct Connect Test
- Propulsion Freejet Test
Examples of Missile Development Tests and Facilities (cont)

- Warhead Arena Test ..............................................................
- Warhead Sled Test ...............................................................
- Insensitive Munition Test ......................................................
- Structure Test .....................................................................
Examples of Missile Development Tests and Facilities (cont)

- Seeker Test
- Hardware-In-Loop
- Environmental Test
- Submunition Dispenser Sled Test
Examples of Missile Development Tests and Facilities (cont)

♦ RCS Test

♦ Store / Avionics Test

♦ Flight Test

♦ Video of Facilities and Tests
Missile Flight Test Should Cover Extremes of Flight Envelope

Example: Ramjet Baseline Propulsion Test Validation (PTV)

Note: Seven Flights from Oct 1979 to May 1980.

Flight 1 failure of fuel control. As a result of the high thrust, the flight Mach number exceeded the design Mach number.
Example of Aero Technology Development

- Conceptual Design (5 to 50 input parameters) Prediction
- Preliminary Design (50 to 200 input parameters) Prediction
  - Missile DATCOM. Contact: AFRL. Attributes include: Low cost
  - MISL3. Contact: NEAR. Attributes include: Modeling vortex shedding
  - SUPL. Contact: NEAR. Attributes include: Paneling complex geometry
  - AP02. Contact: NSWC. Attributes include: Periodic updates
  - CFD. Contact: Georgia Tech. Attributes include: Model runs on Parallel Processing PCs
- Preliminary Design Optimization
  - Response Surface Model: Contact: Georgia Tech. Attributes include 10x more rapid computation
  - Probabilistic Analysis: Contact: Georgia Tech. Attributes include an evaluation of design robustness
Example of Aero Technology Development (cont)

- Wind Tunnel Test Verification
  - Body buildup force and moment
  - Control effectiveness and hinge moment
  - Store carriage and separation
  - Flow field (may be required)
  - Pressure distribution (may be required)
  - Plume, heat transfer, and dynamic stability (usually not required)
  - Inlet (if applicable)

- 3 to 6-DOF Digital Simulation
- Hardware-in-loop Simulation
- Detail Design (over 200 input parameters)
- Flight Test Validation
Example of Missile Technology State-of-the-Art Advancement: Air-to-Air Missile Maneuverability

![Graph showing the advancement of air-to-air missile maneuverability over time, with specific missiles and their introduction dates highlighted.]
Example of Missile Technology State-of-the-Art Advancement: Ramjet Propulsion

![Graph showing advancements in ramjet propulsion technology over time with various missile models and their corresponding flight demonstration years and cruise Mach numbers. The graph includes symbols for different missile models such as Cobra, X-7, Vandal/Talos, St-450, SE 4400, RARE, Bloodhound, BOMARC, Typhon, STATEX, D-21, CROW, SA-6, Sea Dart, LASRM, ALVRJ, 3M80, ASALM, AS-17/Kh-31, ASMP, ANS, Kh-41, SLAT, BrahMos, and Meteor. The x-axis represents the year of flight demonstration, ranging from 1950 to 2010, and the y-axis represents the cruise Mach number.]
Summary of Development Process

- Development Process
  - Technology roadmap
  - Development activities
  - Time frame
- Level of Design Maturity Related to Stage of Development
- Missile Follow-on Programs
- Subsystems Development Activities
- Subsystems Integration and Missile System Development
- Flight Test Activities
- Missile Development Tests and Facilities
- State-of-the-Art Advancement in Tactical Missiles
- New Technologies for Tactical Missiles
- Discussion / Questions?
- Classroom Exercise (Appendix A)
Development Process Problems

1. A technology roadmap establishes the high payoff technologies g____.
2. The levels of design maturity from the most mature to least mature are production, detail, preliminary, and c_________ design.
3. Technology transitions occur from basic research to exploratory development, to advanced development, to d___________ and v__________.
4. Approximately 11% of the U.S. RT&A budget is allocated to t_______ m_______.
5. In the U.S., a tactical missile has a follow-on program about every __ years.
6. Compared to the AIM-9L, the AIM-9X has enhanced m____________.
7. Compared to the AIM-7, the AIM-120 has autonomous guidance, lighter weight, higher speed, and longer r____.
8. Compared to the PAC-2, the PAC-3 has h__ t_ k___ accuracy.
9. Guidance & control is verified in the h_______ in l___ simulation.
10. Airbreathing propulsion ground tests include direct connect tests and f______ tests.
11. Aerodynamic force and moment data are acquired in w___ t_____ tests.
Outline

- Introduction / Key Drivers in the Design Process
- Aerodynamic Considerations in Tactical Missile Design
- Propulsion Considerations in Tactical Missile Design
- Weight Considerations in Tactical Missile Design
- Flight Performance Considerations in Tactical Missile Design
- Measures of Merit and Launch Platform Integration
- Sizing Examples
- Development Process
- Summary and Lessons Learned
- References and Communication
- Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Evaluate Alternatives and Iterate the System-of-Systems Design

- Mission / Scenario Definition
- Weapon Requirements, Trade Studies and Sensitivity Analysis
- Launch Platform Integration
- Weapon Concept Design Synthesis
- Technology Assessment and Dev Roadmap

Update

Initial
Revision

Trades / Eval

Effectiveness / Eval

Initial Reqs

Alt Concepts

Baseline Selected

Initial Carriage / Launch

Iteration

Refine Weapons Req

Alternate Concepts ⇒ Select Preferred Design ⇒ Eval / Refine

Initial Tech Trades Initial Revised Roadmap Roadmap

Note: Typical design cycle for conceptual design is usually 3 to 9 months
Exploit Diverse Skills for a Balanced Design

Customer (requirements pull)
⇒ mission / MIR weighting

Operations analysts
⇒ system-of-systems analysis

System integration engineers
⇒ launch platform integration

Missile design engineers
⇒ missile concept synthesis

Technical specialists (technology push)
⇒ technology assessment / roadmap
Utilize Creative Skills

♦ Use Creative Skills to Consider Broad Range of Alternatives
♦ Ask Why? of Requirements / Constraints
♦ Project into Future (e.g., 5 – 15 years)
  ◆ State-of-the-art (SOTA)
  ◆ Threat
  ◆ Scenario / Tactics / Doctrine
  ◆ Concepts
  ◆ Technology Impact Forecast
♦ Recognize and Distill the Most Important, Key Drivers
♦ Develop Missile Concept that is Synergistic within a System-of-Systems
♦ Develop Synergistic / Balanced Combination of High Leverage Subsystems / Technologies
Identify and Quantify the High Payoff Measures of Merit

- Max / Min Range
- Time to Target
- Robustness
- Lethality
- Reliability
- Miss Distance
- Survivability
- Observable
- Weight
Start with a Good Baseline

I would have used the wheel as a baseline.
Conduct Balanced, Unbiased Tradeoffs

- Aerodynamics
- Propulsion
- Structures
- Production
- Seeker
- Guidance and Control
- Warhead – Fuze
Evaluate Many Alternatives

Note: Although all of the above are supersonic air-to-air missiles, they have different configuration geometry
Search a Broad Design Solution Space (Global Optimization vs Local Optimization)

Local Optimum (e.g., Lowest Cost Only in Local Solution Space)

Global Optimum (e.g., Lowest Cost in Global Solution Space) within Constraints
Evaluate and Refine as Often as Possible

What God hath made, man shall not changeth.
Provide Balanced Emphasis of Analytical vs Experimental

Albert Einstein: "The only real valuable thing is intuition."

Thomas Edison: "Genius is 1% inspiration and 99% perspiration."
Use Design, Build, and Fly Process, for Feedback That Leads to Broader Knowledge

1. Design
   - Prediction Satisfies Customer Requirements?
     - No
     - Build
     - Is it Producible?
       - No
       - Fly (Test)
       - Test Results Satisfy Customer Requirements and Consistent with Prediction?
         - No
         - Design
         - Yes

2. Climb Ladder of Knowledge
   - Data
   - Failure / Success
   - Information
   - Knowledge
   - Understanding
   - Wisdom

Where is the wisdom we have lost in knowledge? Where is the knowledge we have lost in information?--T. S. Eliot (The Rock)

Knowledge comes by taking things apart: analysis. But wisdom comes by putting things together.--John A. Morrison

We are drowning in information but starved for knowledge.--John Naisbitt (Megatrends: Ten New Directions Transforming Our Lives)

We learn wisdom from failure much more than from success. We often discover what will do, by finding out what will not do; and probably he who never made a mistake never made a discovery.--Samuel Smiles (Self Help)
Evaluate Technology Risk
Keep Track of Assumptions and Develop Real-Time Documentation

It’s finally finished! . . .
Develop Good Documentation

Technology Roadmap

Unit production cost and development cost

Weight and balance

Three-view drawing of preferred concept(s)

Traceability of system driving MIRs

Sensitivity of system / subsystem parameters

Mission flight profiles of preferred concept(s)

Aero and propulsion characteristics

Justification of recommended concept(s)

Sketches of alternative concepts

MIRs Weighting
Utilize Group Skills

Balance the Tradeoff of Importance vs Priority

- Production Programs / Detail Design
- SDD Programs / Preliminary Design
- Advanced Programs / Conceptual Design
Evaluate Alternatives and Iterate the Configuration Design

Define Mission Requirements

Establish Baseline

Aerodynamics

Propulsion

Weight

Trajectory

Meet Performance?

Yes

Measures of Merit and Constraints

No

Resize / Alt Config / Subsystems / Tech

Alt Mission

Alt Baseline
## Configuration Sizing Conceptual Design Guidelines: Aeromechanics

<table>
<thead>
<tr>
<th>Configuration Sizing Parameter</th>
<th>Aeromechanics Design Guideline</th>
</tr>
</thead>
<tbody>
<tr>
<td>Body fineness ratio</td>
<td>$5 &lt; \frac{l}{d} &lt; 25$</td>
</tr>
<tr>
<td>Nose fineness ratio</td>
<td>$l_N / d \approx 2$ if $M &gt; 1$</td>
</tr>
<tr>
<td>Boattail or flare angle</td>
<td>$&lt; 10\text{ deg}$</td>
</tr>
<tr>
<td>Efficient cruise dynamic pressure</td>
<td>$q &lt; 1,000\text{ psf}$</td>
</tr>
<tr>
<td>Missile homing velocity</td>
<td>$V_M / V_T &gt; 1.5$</td>
</tr>
<tr>
<td>Ramjet combustion temperature</td>
<td>$&gt; 4,000^\circ\text{ F}$</td>
</tr>
<tr>
<td>Oblique shocks prior to inlet normal shock to satisfy MIL-E-5008B</td>
<td>$&gt; 2$ oblique shocks / compressions if $M &gt; 3.0$, $&gt; 3$ shocks / compressions if $M &gt; 3.5$</td>
</tr>
<tr>
<td>Inlet flow capture</td>
<td>Shock on cowl lip at $M_{\text{max,cruise}}$</td>
</tr>
<tr>
<td>Ramjet Minimum cruise Mach number</td>
<td>$M &gt; 1.2 \times M_{\text{Inlet,Start}}$, $M &gt; 1.2 M_{\text{Max,Thrust}} = \text{Drag}$</td>
</tr>
<tr>
<td>Subsystems packaging</td>
<td>Maximize available volume for fuel / propellant</td>
</tr>
</tbody>
</table>
## Configuration Sizing Conceptual Design Guidelines: Guidance & Control

<table>
<thead>
<tr>
<th>Configuration Sizing Parameter</th>
<th>G&amp;C Design Guideline</th>
</tr>
</thead>
<tbody>
<tr>
<td>Body bending frequency</td>
<td>$\omega_{BB} &gt; 2 \omega_{ACT}$</td>
</tr>
<tr>
<td>Trim control power</td>
<td>$\alpha / \delta &gt; 1$</td>
</tr>
<tr>
<td>Neutral stability tail-body</td>
<td>If low aspect ratio, $b / d \approx 2$, $c / d &gt; \approx 1$</td>
</tr>
<tr>
<td>Stability &amp; control cross coupling</td>
<td>$&lt; 30%$</td>
</tr>
<tr>
<td>Airframe time constant</td>
<td>$\tau &lt; 0.2 \text{ s}$</td>
</tr>
<tr>
<td>Missile maneuverability</td>
<td>$n_M / n_T &gt; 3$</td>
</tr>
<tr>
<td>Proportional guidance ratio</td>
<td>$3 &lt; N' &lt; 5$</td>
</tr>
<tr>
<td>Target span resolution by seeker</td>
<td>$&lt; b_{\text{target}}$</td>
</tr>
<tr>
<td>Missile heading rate</td>
<td>$\gamma_M &gt; \gamma_T$</td>
</tr>
<tr>
<td>Missile turn radius</td>
<td>$R_{TM} &lt; R_{TT}$</td>
</tr>
</tbody>
</table>
Wrap Up (Part 1 of 2)

- Missile design is a creative and iterative process that includes:
  - System considerations
  - Missile concepts and sizing
  - Flight trajectory evaluation

- Cost / performance drivers may be “locked in” during conceptual design.

- Missile design is an opportunity for a diverse group to work together for a better product:
  - Military customer ⇒ mission / scenario definition
  - Operations analysts ⇒ system-of-systems modeling
  - System integration engineers ⇒ launch platform integration
  - Missile design engineers ⇒ missile concept synthesis
  - Technical specialists ⇒ technology assessment / technology roadmap
Wrap Up (Part 2)

- The missile conceptual design philosophy requires:
  - Iteration, iteration, iteration
  - Evaluation of a broad range of alternatives
  - Traceable flow-down allocation of requirements
  - Starting with a good baseline
  - Paredo sensitivity analysis to determine the most important, driving parameters
  - Synergistic compromise / balanced subsystems and technologies that are high leverage
  - Awareness of technology SOTA / technology assessment
  - Technology impact forecast
  - Robust design
  - Creative design decisions made by the designer (not the computer)
  - Fast, simple, robust, physics-based prediction methods
Introduction / Key Drivers in the Design Process
Aerodynamic Considerations in Tactical Missile Design
Propulsion Considerations in Tactical Missile Design
Weight Considerations in Tactical Missile Design
Flight Performance Considerations in Tactical Missile Design
Measures of Merit and Launch Platform Integration
Sizing Examples
Development Process
Summary and Lessons Learned
References and Communication
Appendices (Homework Problems / Classroom Exercises, Example of Request for Proposal, Nomenclature, Acronyms, Conversion Factors, Syllabus)
Bibliography of Other Reports and Web Sites

- System Design
  - “Periscope,” http://www.periscope1.com/
Bibliography of Other Reports and Web Sites (cont)

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- **Aerodynamics**
  - “Missile Aerodynamics,” NATO AGARD LS-98, February 1979
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- Materials and Heat Transfer-
- “NASA Ames Research Center Thermal Protection Systems Expert (TPSX) and Material Properties Database”, [http://tpsx.arc.nasa.gov/tpsxhome.shtml](http://tpsx.arc.nasa.gov/tpsxhome.shtml)
Bibliography of Other Reports and Web Sites (cont)

- Materials and Heat Transfer (continued)

- Guidance, Navigation, Control, and Sensors
Follow-up Communication

I would appreciate receiving your comments and corrections on this text, as well as any data, examples, or references that you may offer.

Thank you,

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Outline

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