Section 4.4: Supersonic Combustion Ramjets (SCRAMjets) and Combined Cycle Engines
ScramJet Applications

Space Access

Weapons

Hypersonic Missile (Time-critical targets)

Mid-Term

Near-Term

Hypersonic Cruiser (Global Reach/Attack)

Far-Term

RLV (Affordable, timely access to space)

Pursue Stepping-Stone Approach
POTENTIAL AIR-BREATHING HYPersonic VEHICLE APPLICATIONS AND THEIR FLIGHT ENVELOPES

- **Cruise Aircraft**
  - Mach 4-8 (HC/H₂)
  - Theater aircraft and weapons
  - Missiles (tactical and strategic)
  - Transport aircraft
  - Mach 8-18 (H₂)
  - Global aircraft and weapons
  - Missiles (tactical and strategic)

- **Space Access**
  - Mach 4-8 (HC/H₂)
  - 2STO 1st stage
  - 3STO 2nd stage
  - Mach 8-18 (H₂)
  - 2STO 1st stage
  - 3STO 2nd stage
  - Mach 25 (H₂)
  - SSTO

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**Altitude (kilofeet)**

- 0
- 50
- 100
- 150
- 200

**Speed (kilofeet per sec)**

- 0
- 5
- 10
- 15
- 20
- 25

**HC = Hydrocarbon**

**H₂ = Liquid Hydrogen**

Most Technologies Are Common to Both Systems
Operations Payoffs for Airbreathing Launch

- Decreased gross lift-off weight, resulting in smaller facilities and easier handling
- Wider range of emergency landing sites for intact abort
- Powered flyback/go-around & more margin at reduced power
- Self-ferry & taxi capabilities
- Greatly expanded launch windows (double or triple)
- Rapid orbital rendezvous (up to three times faster than rockets)
- Wider array of landing sites from orbit, with 2,000-mile cross range and increased range
- Reduced sensitivity to weight growth
Fuel Efficiencies of Various High Speed Propulsion Systems

**Turbojet**
- Uses atmospheric air
- Complex turbomachinery
- Subsonic combustion
- Mach range: 0 to 3

**Ramjet**
- Uses atmospheric air
- "Ram" compression
- No turbomachinery
- Subsonic combustion
- Mach range: 2 to 6

**Rocket**
- Carries oxidizer
- No turbomachinery
- Subsonic combustion
- Mach range: 0 to 25

**Scramjet**
- Uses atmospheric air
- "Ram" compression
- No turbomachinery
- Supersonic combustion
- Mach range: 5 to ~25

*Scramjets have the highest efficiency above Mach 6*

*Minimal turbo-machinery*

*Propulsion systems comparison*

*Hydrogen fuel*
Ideal Ramjet Cycle Analysis Revisited

T-s Diagram
Ideal Ramjet: *Inlet and Diffuser with normal shock*

- Mechanical Energy is Dissipated into Heat
- Huge Loss in Momentum
Ideal Ramjet: *Inlet and Diffuser* with Oblique shock

- So ... we put a spike in front of the inlet

- How does this spike Help?

- By forming an Oblique Shock wave ahead of the inlet
Thermodynamic Efficiency of Ideal Ramjet, revisited

\[ \eta = 1 - \left( \frac{P_A}{P_B} \right)^{\frac{\gamma-1}{\gamma}} \frac{T_C - \left( \frac{P_{0_B}}{P_{0_A}} \right)^{\frac{1}{\gamma}} T_B}{T_C - T_B} \]

i) As engine pressure ratio, \( P_B/P_A \), goes up \( \ldots \) \( \eta \) goes up

ii) As combustor temperature difference \( T_C-T_B \) goes up \( \ldots \) \( \eta \) goes up

iii) As inlet total pressure ratio (\( P_{0_B}/P_{0_A} \)) goes down \( \ldots \) (stagnation pressure loss goes up) \( \ldots \) \( \eta \) goes down
Scramjet Design Issues, I

- What if we keep engine flow path supersonic to minimize stagnation pressure loss?

- How do we keep the Inflow supersonic?

- Series of very weak (highly oblique) shockwaves and expansion shocks keep the flow supersonic throughout the engine
Scramjet Design Issues, I
Inlet “Point Design”

\[ M_\infty = M_{\text{design}} \]

On-design scramjet inlet operation

- SCRAMjets are VERY … 
  *Sensitive* to inlet mach number
Inlet “Point Design” (cont’d)

Turbulent Mixing causes large momentum loss

\[ M_\infty < M_{\text{design}} \]

Off-design scramjet inlet operation

Turbulent Mixing causes large momentum loss

\[ M_\infty > M_{\text{design}} \]

Off-design scramjet inlet operation
X-43A Side by Side Comparison

- Subtle but important shape differences Mach 10 Inlet likely would not start at Mach 7

Mach 7 Vehicle

Mach 10 Vehicle
ScrRamjet Design Issues, I

- What if we keep engine flow path supersonic to minimize stagnation pressure loss?
- How do we keep the Inflow supersonic?

- Series of very weak (highly oblique) shockwaves and expansion shocks keep the flow supersonic throughout the engine.
Background on Supersonic Inlet Design

- Anderson, Chapter 4 pp. 152-164
Rules for Shock Wave Reflections from Solid and Free Boundaries

1. Waves Incident on a Solid Boundary reflect in a Like manner;
   Compression wave reflects as compression wave, expansion wave reflects as expansion wave

2. Waves Incident on a Free Boundary reflect in an Opposite manner;
   Compression wave reflects as expansion wave, expansion wave reflects as compression wave

Schematic of the diamond wave pattern in the exhaust from a supersonic nozzle.
Geometry of a Shock Wave: Reflected From a Solid Boundary

\[ \beta_1 \neq \beta_2 \quad \text{• Not Billiards … Why?} \]
SCRAMJet Inlet Design Example

- Example Calculation

\[ M_1 = 3.6 \]

\[ \theta = 20^\circ \]

\[ \gamma = 1.4 \]

\[ \tan(\theta) = \frac{2 \left( M_1^2 \sin^2(\beta) - 1 \right)}{\tan(\beta) \left[ 2 + M_1^2 \left[ \gamma + \cos(2\beta) \right] \right]} \rightarrow \]

\[ = 20^\circ \]
SCRAMJet Inlet Design Example (cont’d)

\[ \tan(\theta) = \frac{2\left(M_1^2 \sin^2(\beta) - 1\right)}{\tan(\beta)\left[2 + M_1^2\left[\gamma + \cos(2\beta)\right]\right]} \]

\[ \frac{180}{\pi} \arctan \left( \frac{2 \left(3.6^2 \sin^2\left(\frac{\pi}{180} \cdot 34.1102\right) - 1\right)}{\left(\tan\left(\frac{\pi}{180} \cdot 34.1102\right)\right) \left(2 + 3.6^2 \left(1.4 + \cos\left(\frac{\pi}{180} \cdot 2 \cdot 34.1102\right)\right)\right)} \right) = 20^\circ \]

\[ \beta_1 = 34.1102^\circ \]
SCRAMJet Inlet Design Example (cont’d)

- \( M_{1n} = M_1 \sin \beta_1 = 3.6 \sin \left( \frac{\pi}{180} \cdot 34.1102 \right) = 2.0188 \)

- Normal Shock Solver -->

\[
M_2n = 0.574168 \rightarrow M_2 = \frac{M_2n}{\sin(\beta_1-\theta)} = \frac{0.574168}{\sin \left( \frac{\pi}{180} \cdot (34.1102 - 20) \right)} = 2.3552
\]
SCRAMJet Inlet Design Example (cont’d)

\[
\theta - \beta - M \text{ solver, for } M_2 = 2.3552 \text{ and } \theta_2 = 20^\circ,
\]

\[
\beta_2 = 45.0534^\circ \quad \rightarrow \quad \Phi = 45.0534^\circ - 20^\circ = 25.0534^\circ
\]

\[
(M_2 n)_{\beta_2} = 2.3552 \sin(45.0534) = 0.649303
\]

\[
M_3 = \frac{(M_2 n)_{\beta_2}}{\sin(\beta_2 - \theta)} = \frac{0.649303}{\sin \left( \frac{\pi}{180} \left( 45.0534 - 20 \right) \right)} = 1.5333
\]

So we have
Gotten inside
The duct and
Still are supersonic!
Off Design Operation

- What if we do our ramp geometry incorrectly for inlet mach?

θ - β - M solver, for M₂ = 2.3552 and θ₂ = 20°,

- M = 1.533

• 20° > θ max

Detached Shockwave!

MAE 6530 - Propulsion Systems II
Off Design Operation
(cont’d)

Mach reflection ... localized strong shockwave ... starts bad train of events leading to flow separation and possible unstart.
Geometry of a Shock Wave: Reflected
From another Shock wave

• Shock waves not only reflect from solid boundaries, they also reflect from each other

• Need “more tools” to analyze problem completely
Pressure-Deflection Diagrams

"Right Running" Oblique Shockwave

\[ \theta' \]

Locus of all possible static pressure values behind oblique shock wave for given upstream conditions

"Left Running" Oblique Shockwave

\[ \theta_2 \]

\[ \theta_{\text{max}} \]

\[ \theta_1 \]

\[ \theta_2 \]

\[ \theta_{\text{max}} \]
• Look at example 1

\[ M_1 = 3.6 \]

\[ \theta = 20^\circ \]

\[ \gamma = 1.4 \]

\[ \beta_2 = \beta_1 = 34.1102^\circ \]

• Left running wave

\[ M_{1n} = M_1 \sin \beta_1 = 3.6 \sin \left( \frac{\pi}{180} \times 34.1102 \right) = 2.0188 \]

• Normal Shock Solver -->

\[ M_2n = 0.574168 \rightarrow M_2 = \frac{M_{2n}}{\sin(\beta_1 - \theta)} = \frac{0.574168}{\sin \left( \frac{\pi}{180} (34.1102 - 20) \right)} = 2.3552 \]

\[ p_2/p_1 = 4.588283 \]
Pressure-Deflection Diagrams (cont’d)

- Look at example 1

\[ M_1 = 3.6 \]
\[ \theta = 20^\circ \]
\[ \gamma = 1.4 \]

- Right running wave

\[
\left( M_2 n \right)_{\beta_2} = 2.3552 \sin(45.0534) = 0.649303
\]

\[
M_3 = \frac{\left( M_2 n \right)_{\beta_2}}{\sin(\beta_2 - \theta)} = \frac{0.649303}{\sin \left( \frac{\pi}{180} (45.0534 - 20) \right)} = 1.5333
\]

\[
p_3/p_2 = 3.075094 \quad \rightarrow \quad p_3/p_1 = (p_3/p_2)(p_2/p_1) = (3.075094)(4.588283) = 14.109
\]
Intersection of Shocks of Opposite Families

Intersection of “left-running” and “right running” Shock waves
• Shock emanating from \{A,B\} intersect and continue as refracted shocks \{C,D\} .. At intersection point E, \(p_4 = p'_4\)
Intersection of Shocks of Opposite Families, Example

- Example: $M_1 = 3$, $p_1 = 1 \text{ atm}$, $\theta_2 = 20^\circ$, $\theta_3 = 15^\circ$
- Plot $p, \theta$ diagram, $p_4$, and flow direction $\Phi_4$
Numerical Calculations for Example

From the $\theta$-$\beta$-$M$ solver: For $\theta_2 = 20^\circ$ and $M_1 = 3$, $\beta = 37.8^\circ$.

$$M_{n_1} = M_1 = \sin \beta = (3 \times \sin (37.8) = 1.839; \frac{p_2}{p_1} = 3.783$$

$$M_{n_2} = 0.6078; M_2 = \frac{M_{n_2}}{\sin(\beta - \theta)} = \frac{0.6078}{\sin(37.8 - 20)} = 1.99$$

For $\theta_3 = 15^\circ$ and $M_1 = 3$, $\beta = 32.2$

$$M_{n_1} = M_1 \sin \beta = (3 \times \sin (32.2) = 1.60; \frac{p_3}{p_1} = 2.82$$

$$M_{n_3} = 0.6684; M_3 = \frac{M_{n_3}}{\sin(\beta - \theta)} = \frac{0.6684}{\sin(32.2 - 15)} = 2.26$$
### Numerical Calculations for Example

(cont’d)

For the upstream flow represented by region 2, plot a pressure deflection diagram from the following calculations:

\[
\begin{array}{ccccccc}
\beta_2 & \theta_4' & M_{n_2} = M_2 \sin \beta & \frac{p_4'}{p_2} & \frac{p_4'}{p_4} = \frac{p_4}{p_2} & \Delta \theta = \theta_2 - \theta_4' \\
30 & 0 & 1 & 1 & & 3.783 & 20 \\
33.3 & 4 & 1.09 & 1.219 & & 4.61 & 16 \\
37.2 & 8 & 1.20 & 1.513 & & 5.72 & 12 \\
41.6 & 12 & 1.32 & 1.866 & & 7.06 & 8 \\
46.7 & 16 & 1.45 & 2.286 & & 8.65 & 4 \\
53.5 & 20 & 1.60 & 2.820 & & 10.67 & 0 \\
\end{array}
\]

**Pick these values**

**Calculate these values**

\[
\begin{align*}
M_1 &= 3 \\
M_3 &= 1.99 \\
\theta_2 &= 20^\circ \\
\frac{p_2}{p_1} &= 3.783 \\
\tan(\theta_4') &= \frac{2\left(M_2^2 \sin^2(\beta_2) - 1\right)}{\tan(\beta_2) \left[ 2 + M_2^2 [\gamma + \cos(2\beta_2)] \right]}
\end{align*}
\]
Numerical Calculations for Example

(cont’d)

For the upstream flow represented by region 3, plot a pressure deflection diagram from the following calculations:

\[
\begin{align*}
\beta_3 & \quad \theta_4 \quad M_{n_3} = M_3 \sin \beta \quad \frac{p_4}{p_2} \quad \frac{p_4}{p_1} = \frac{p_4}{p_3} \frac{p_3}{p_1} \quad \Delta \theta = \theta_4 \theta_3
\end{align*}
\]

<table>
<thead>
<tr>
<th>(\beta_3)</th>
<th>(\theta_4)</th>
<th>(M_{n_3})</th>
<th>(\frac{p_4}{p_2})</th>
<th>(\frac{p_4}{p_1})</th>
<th>(\Delta \theta = \theta_4 \theta_3)</th>
</tr>
</thead>
<tbody>
<tr>
<td>26</td>
<td>0</td>
<td>1</td>
<td>1</td>
<td>2.82</td>
<td>-15</td>
</tr>
<tr>
<td>29</td>
<td>4</td>
<td>1.096</td>
<td>1.23</td>
<td>3.47</td>
<td>-11</td>
</tr>
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<td>33</td>
<td>8</td>
<td>1.23</td>
<td>1.598</td>
<td>4.51</td>
<td>-7</td>
</tr>
<tr>
<td>36.8</td>
<td>12</td>
<td>1.35</td>
<td>1.96</td>
<td>5.53</td>
<td>-3</td>
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<tr>
<td>41.5</td>
<td>16</td>
<td>1.50</td>
<td>2.458</td>
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<td>1</td>
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<tr>
<td>46.8</td>
<td>20</td>
<td>1.65</td>
<td>3.01</td>
<td>8.49</td>
<td>5</td>
</tr>
<tr>
<td>53.7</td>
<td>24</td>
<td>1.82</td>
<td>3.698</td>
<td>10.43</td>
<td>9</td>
</tr>
</tbody>
</table>

\(\theta_3 = 15^\circ\)

\(M_1 = 3\)

\(M_3 = 2.26\)

\[
\frac{p_3}{p_1} = 2.82
\]

\[
\tan(\theta_4) = \frac{2 \left[M_3^2 \sin^2(\beta_3) - 1 \right]}{\tan(\beta_3) \left[ 2 + M_3^2 \left[ \gamma + \cos(2\beta_3) \right] \right]}
\]
Pressure-deflection-diagram for Example

\[ M = 2.32 \quad (M_3) \]
\[ M = 1.99 \quad (M_2) \]

\[ \theta_4 = \Phi - \theta_2 \]
\[ \theta_4' = \theta_2 - \Phi \]

\[ \Phi = \text{“physical flow angle Relative to inlet direction”} \]
From the graph on previous page

\[ p_4 = p_4' = \frac{p_4}{p_1} \]

\[ p_1 = (8.3)(1) = 8.3 \text{ atm} \]

\[ \Phi = 4.5^\circ \]

\[ \Theta_3 = 15^\circ \]

\[ \Theta_2 = 20^\circ \]

\[ M_1 = 3 \]

\[ \Phi_4 \]

• Entropy change across \{BE, and ED\} = entropy change across Shocks \{AE, and EC\} …. Creates “slip line” emanating From E between regions 4 and 4’
Intersection of Shocks of The Same Family

• Will Mach waves intersect shock wave?

Left running mach line will
Always intersect left running shock

1. \( u_1 > a_1 \rightarrow \mu_1 < \beta \)
   • \( u_2 < a_2 \rightarrow \mu_2 > \beta - \theta \)

\[
\begin{align*}
\sin(\beta) &= \frac{u_1}{V_1} \rightarrow \sin(\mu_1) = \frac{a_1}{V_1} \\
\sin(\beta - \theta) &= \frac{u_2}{V_2} \rightarrow \sin(\mu_2) = \frac{a_2}{V_2}
\end{align*}
\]

Yes!
Not a big stretch to generalize to

- Conditions in region 3 driven by region 1 conditions and processed by shocks \( \{AC, \text{ and } BC\} \)

- Conditions in region 5 driven by region 1 conditions and processed only by one shock \( \{CD\} \)

"Two Left Running Shock waves"
Intersection of Shocks of The Same Family (cont’d)

- Conditions in region 3 driven by Upstream conditions and processed by shocks \{AC, and BC\}

- Conditions in region 5 driven by Upstream conditions and processed Only by shock \{CD\}

- Entropy change across \{CD\} ≠ entropy change across Shocks \{AC, and BC\} …

  Creates “slip line” emanating From C between regions 4 and 5

\[ P_4 = P_5 \]

MAE 5420 - Compressible Fluid Flow
Intersection of Shocks of The Same Family (cont’d)

• within slip zone $P_4 = P_5$
  and $\theta_4 = \theta_5$,

• in general for arbitrary
  Shock waves … $P_3 \neq P_5$

But .. $P_4 = P_5$

… How?…Reflected wave E…

… weak shock or
  Expansion wave depending
On relationship of $P_3$ to $P_5$

“Two Left Running Shock waves”
Intersection of Shocks of The Same Family

(cont’d)

• within slip zone $P_4 = P_5$
  and $\theta_4 = \theta_5$, also $\theta_3 = \theta_5$

$P_3 > P_5$

… expansion fan drops

$P_3$ to $P_3$ ( = $P_5$)

“Two Left Running Shock waves”
Intersection of Shocks of The Same Family
(cont’d)

• within slip zone $P_4 = P_5$
  and $\theta_4 = \theta_5$, also $\theta_3 = \theta_5$

$P_3 < P_5$

… weak shock wave compresses $P_3$ to $P_4 \ (= P_5)$

• Model must be smart enough
  To accommodate this difference

Must iteratively adjust
  Strength of waves CD and CE
  Until $P_4 = P_5$ on output side

“Two Left Running Shock waves”
ScrRamjet Design Issues, II

- Supersonic flow makes flow control within the combustion chamber more difficult.

- Massflow entering combustion chamber must mix with fuel and have sufficient time for initiation and reaction, while traveling supersonically through combustion chamber, before the burned gas is expanded through the thrust nozzle.

**SCRAM … M>>1 in burner**

**Rocket … M<<1 in burner**

\[ aA + bB \xrightarrow{k_f} cC + dD \]

Reaction times are Critically tuned to Flow path mach number
Review of 1-D Combustion Model

Consider generic combustor model

i) **Enthalpy Balance**

\[ \dot{Q} = \dot{m}_{fuel} \cdot h_{fuel} \rightarrow \dot{Q} + \dot{m}_{air} \left( h_3 + \frac{V_3^2}{2} \right) = \left( \dot{m}_{air} + \dot{m}_{fuel} \right) \left( h_4 + \frac{V_4^2}{2} \right) \]
Review of 1-D Combustion Model \((2)\)

Consider generic combustor model

\[
\dot{Q} = \dot{m}_{\text{fuel}} \cdot h_{\text{fuel}} \rightarrow \dot{Q} + \dot{m}_{\text{air}} \left( h_3 + \frac{V_3^2}{2} \right) = \left( \dot{m}_{\text{air}} + \dot{m}_{\text{fuel}} \right) \left( h_4 + \frac{V_4^2}{2} \right)
\]

\[
\dot{m}_{\text{fuel}} \cdot h_{\text{fuel}} + \dot{m}_{\text{air}} \left( h_3 + \frac{V_i^2}{2} \right) = \left( \dot{m}_{\text{air}} + \dot{m}_{\text{fuel}} \right) \left( h_e + \frac{V_e^2}{2} \right)
\]

\[
\frac{\dot{m}_{\text{fuel}}}{\dot{m}_{\text{air}}} \cdot h_{\text{fuel}} + \left( h_3 + \frac{V_3^2}{2} \right) = \left( 1 + \frac{\dot{m}_{\text{fuel}}}{\dot{m}_{\text{air}}} \right) \left( h_4 + \frac{V_4^2}{2} \right) \rightarrow
\]

\[
\frac{1}{f} \cdot h_{\text{fuel}} + \left( h_3 + \frac{V_3^2}{2} \right) = \left( \frac{f + 1}{f} \right) \left( h_4 + \frac{V_4^2}{2} \right) \rightarrow f = \frac{\dot{m}_{\text{air}}}{\dot{m}_{\text{fuel}}}
\]
Review of 1-D Combustion Model (3)

- Stagnation Pressure Ratio Across Combustor is Proportional to the ratio of the fuel enthalpy and the incoming air stagnation enthalpy

\[
\frac{T_{04}}{T_{03}} = \left( \frac{f}{f+1} \right) \left( \frac{1}{f} \cdot \frac{h_{\text{fuel}}}{c_{p4} T_{03}} + \frac{c_{p3}}{c_{p4}} \right) = \left( \frac{f}{f+1} \right) \left( \frac{c_{p3}}{c_{p4}} \right) \left( 1 + \frac{1}{f} \cdot \frac{h_{\text{fuel}}}{c_{p3} T_{03}} \right)
\]

- Resulting Mach Number Change Across Combustor

\[
M_4^2 \left[ 1 + \frac{\gamma_4 - 1}{2} M_4^2 \right] = \left( \frac{f+1}{f} \right)^2 \left[ \frac{T_{04}}{T_{03}} \right] \left[ \frac{\gamma_3 R_{g4}}{\gamma_4 R_{g3}} \right] M_3^2 \left[ 1 + \frac{\gamma_3 - 1}{2} M_3^2 \right]
\]
Review of 1-D Combustion Model (4)

\[ F(M_3) \equiv \left( \frac{f + 1}{f} \right)^2 \left[ \frac{T_{04}}{T_{03}} \right]\left[ \frac{\gamma_3 R_{g_4}}{\gamma_4 R_{g_3}} \right] M_{3}^2 \left[ 1 + \frac{\gamma_3 - 1}{2} M_{3}^2 \right] \]

\[ \left[ \frac{\gamma - 1}{2} - \gamma^2 F(M_{inlet}) \right] M_{exit}^4 + \left[ 1 - F(M_{inlet})2\gamma \right] M_{exit}^2 - F(M_{inlet}) = 0 \]

→ use quadratic formula

\[ M_{exit} = \sqrt{-\left[ 1 - F(M_{inlet})2\gamma \right] \pm \sqrt{\left[ 1 - F(M_{inlet})2\gamma \right]^2 + 4 \left[ \frac{\gamma - 1}{2} - \gamma^2 F(M_{inlet}) \right] F(M_{inlet})}} \]
On the Difference Between Subsonic and Supersonic Combustion

\[ M_{\text{exit}} = \sqrt{\frac{-[1 - F(M_{\text{inlet}})2\gamma] \pm \sqrt{[1 - F(M_{\text{inlet}})2\gamma]^2 + 4\left[\frac{\gamma - 1}{2} - \gamma^2 F(M_{\text{inlet}})\right] F(M_{\text{inlet}})}}{2\left[\frac{\gamma - 1}{2} - \gamma^2 F(M_{\text{inlet}})\right]}} \]

two solutions...pick subsonic solution if \((M_{\text{inlet}} < 1, \Delta q > 0)\)

...pick supersonic solution if \((M_{\text{inlet}} > 1, \Delta q > 0)\)

Cannot cross over Mach = 1
Otherwise second law of thermodynamics is violated
On the Difference Between Subsonic and Supersonic Combustion (2)

- MAE 5320 Lecture 5.4 (Anderson Chapter 3), Rayleigh Equations

\[
H_0^{(*)} = \frac{c_p \cdot T_0}{c_p \cdot T_0^{(*)}} = 2(1+\gamma) \cdot \frac{M^2 \left[ 1 + \frac{\gamma - 1}{2} M^2 \right]}{\left[ 1 + \gamma M^2 \right]^2}
\]

\[
H^{(*)} \equiv \left[ \frac{c_p \cdot T}{c_p \cdot T^{(*)}} \right] = \left[ \frac{(1+\gamma) \cdot M}{1 + \gamma \cdot M^2} \right]^2
\]

\[
\frac{s - s^{(*)}}{c_p} = \ln \left[ M^2 \cdot \left( \frac{1 + \gamma}{1 + \gamma \cdot M^2} \right)^{\frac{\gamma+1}{\gamma}} \right]
\]

\[
\frac{\Delta q^{(*)}}{c_p \cdot T_o} = \frac{T_0^{*}}{T_o} - 1 = \frac{1 - H_0^{*}}{H_0^{*}}
\]

- Set of Parametric curves with property ratios as a function of Mach number

- Relates current conditions to those that occur for thermal choke point

- Entirely at function of Current Mach number

- Prescribed heat addition Necessary to thermally choke flow
Rayleigh Flow Curves

Rayleigh Curve, Stagnation Temperature Ratio

\( \frac{T_0}{T_0^*} \)

Supersonic
Subsonic

Rayleigh Curve 3, Non-Dimensional Heat Addition

\( \frac{\Delta q}{c_p \cdot T_0} \)

Initial Mach

Rayleigh Curve, Static Temperature Ratio

\( \frac{s - s^*}{c_p} \)

Cooling
Heating
Subsonic
Supersonic

Rayleigh Curve 4, Non-Dimensional Entropy

\( \frac{s - s^*}{c_p} \)

Initial Mach

Fig 3-13
Anderson pp. 108, 109
On the Difference Between Subsonic and Supersonic Combustion (4)

\[ H^* = \frac{T}{T^*} \]

Sonic Point
\((M = 1)\)

**Figure 3-13 From Anderson**

- Maximum entropy change occurs at sonic point

- Heating to supersonic Mach numbers starting from subsonic flow violates second law of thermodynamics

\[ S^* = \frac{S - S^*}{c_p} \]

*Figure 3.13 | The Rayleigh curve.*
On the Difference Between Subsonic and Supersonic Combustion

Rayleigh Curve (Anderson pp. 108, 109)

1. For supersonic flow in region 1, i.e., $M_1 > 1$, when heat is added
   a. Mach number decreases, $M_2 < M_1$
   b. Pressure increases, $p_2 > p_1$
   c. Temperature increases, $T_2 > T_1$
   d. Total temperature increases, $T_{o2} > T_{o1}$
   e. Total pressure decreases, $p_{o2} < p_{o1}$
   f. Velocity decreases, $u_2 < u_1$

2. For subsonic flow in region 1, i.e., $M_1 < 1$, when heat is added
   a. Mach number increases, $M_2 > M_1$
   b. Pressure decreases, $p_2 < p_1$
   c. Temperature increases for $M_1 < \gamma^{-1/2}$ and decreases for $M_1 > \gamma^{-1/2}$
   d. Total temperature increases, $T_{o2} > T_{o1}$
   e. Total pressure decreases, $p_{o2} < p_{o1}$
   f. Velocity increases, $u_2 > u_1$
\[
\begin{align*}
M_{\text{inlet}} < 1 & \iff +\Delta q \text{ increases Mach} \implies M_{\text{exit}} > M_{\text{inlet}} \\
M_{\text{inlet}} > 1 & \iff +\Delta q \text{ decreases Mach} \implies M_{\text{exit}} < M_{\text{inlet}} \\
M_{\text{inlet}} < 1 & \iff -\Delta q \text{ decreases Mach} \implies M_{\text{exit}} < M_{\text{inlet}} \\
M_{\text{inlet}} > 1 & \iff -\Delta q \text{ increases Mach} \implies M_{\text{exit}} > M_{\text{inlet}}
\end{align*}
\]
Thermal Choking in Ramjet

Non-adiabatic flow is accelerated to mach 1 without divergent nozzle by adding heating

\[ M_{\text{exit}} > M_{\text{inlet}} \]
Sufficient Heat Must be to Drive Combustor Exit Flow to Mach 1

\[ M_{exit} < M_{inlet} \]
Thermal Choking Comparisons

• Looking at slide 30 … There are several features shown in these plots that have important implications for the ramjet flow.

• First is that much more heat can be added to a subsonic flow than to a supersonic flow before thermal choking occurs.

• Second stagnation pressure losses due to heat addition in subsonic flow are relatively small and cannot exceed about 20% of the stagnation pressure of the flow

• ScramJets are inherently Thermally inefficient!
ScrRamjet design issues, II (cont’d)

- Supersonic flow places stringent requirements on the pressure and temperature of the flow, and requires that the fuel injection and mixing be extremely efficient.

- Propulsion controller designed to maintain stable operation while achieving necessary performance
  - Flameout - low fuel flow condition where hydrogen-only combustion is not sustained
  - Unstart - high fuel flow condition where shock train moves through isolator causing normal shock to occur. Choked nozzle result … with flow spill out … likely that shockwave forced back up to inlet. Very bad
ScrRamjet design issues, II (cont’d)

- Typical Equivalence Ratio Schedule for Scramjet Burn

![Graph showing the equivalence ratio schedule for a scramjet burn. The graph includes three lines labeled $\Phi_{\text{total}}$, $\Phi_{\text{fuel}}$, and $\Phi_{\text{igniter}}$. The equivalence ratio, $\Phi$, is shown on the y-axis, and time, in seconds, is shown on the x-axis. The graph indicates a transition from rich to lean conditions over time.](image_url)
ScrRamjet design issues, II (cont’d)

• In typical ramjet inflow is decelerated to subsonic speeds and then reaccelerated via nozzle to supersonic speeds to produce thrust. This deceleration, which is produced by a normal shock, creates a total enthalpy loss which limits the upper operating point of a ramjet engine.

• In supersonic combustion, enthalpy of freestream air entering the scramjet engine is large compared to the heat energy released by the combustion reaction.

• Depending on fuel, combustion equivalence ratio, and freestream altitude, potential combustion heat release is equal to freestream flow enthalpy between Mach 8 and Mach 10.

• Heat released from combustion at Mach 25 is only 10% of total enthalpy of working fluid.

• Design of a scramjet engine is as much about minimizing drag as maximizing thrust.

• Integration of engine into airframe is Key.
ScrRamjet design issues, II (cont’d)

- Integration of engine into airframe is Key

Mach 20+ design

Mach 7-10 design
Example Enthalpy Calculation

• Air enters Mach 10 SCRAM inlet at 30 km altitude

\[ p_\infty = 1.17180 \text{ kPa}, \ 226.65 \degree K, \ \theta_{ramp} = 4 \degree \]

Conditions at 2:

\[
\tan(\theta) = \frac{2 \left( M_1^2 \sin^2(\beta) - 1 \right)}{\tan(\beta) \left[ 2 + M_1^2 \left( \gamma + \cos(2\beta) \right) \right]} \quad \Rightarrow \quad \begin{align*}
\beta_1 &= 8.6531 \degree \\
M_2 &= 8.622746 \\
p_\infty / p_\infty &= 2.474152 \\
T_\infty / T_\infty &= 1.323222 \\
P_0 / P_{0\infty} &= 0.928350
\end{align*}
\]
Example Enthalpy Calculation (cont’d)

\[ \beta_1 = 8.6531^\circ \]
\[ M_2 = 8.622746 \]
\[ \frac{p_3}{p_\infty} = 2.474152 \]
\[ \frac{T_3}{T_\infty} = 1.323222 \]
\[ \frac{P_{03}}{P_{0\infty}} = 0.928350 \]

\[ \beta_2 = 9.521^\circ \]
\[ M_3 = 7.576 \]
\[ \frac{p_3}{p_2} = 2.20665 \]
\[ \frac{T_3}{T_2} = 1.271723 \]
\[ \frac{P_03}{P_{02}} = 0.951393 \]

\[ P_3 = 5.496 \cdot 1.1718 = 6.550 \text{ kPa} \]

\[ T_3 = 1.6831 \cdot 226.65 = 381.47 \text{ °K} \]
Example Enthalpy Calculation (cont’d)

- Specific Enthalpy of flow entering combustor:

\[
h_3 = C_p T_3 + \frac{\left[\sqrt{\gamma R_g T_3 M_3}\right]^2}{2} = \frac{(1.4 \cdot 287.056 \cdot 381.47)^{0.5} \cdot 7.576}{2} + 1004.7 \cdot 381.47
\]

\[
= 4.7828 \text{ MJ/kg}
\]

\[
p_3 = 5.496 \cdot 1.1718 = 6.550 \text{ kPa}
\]

\[
T_3 = 1.6831 \cdot 226.65 = 381.47 \text{ °K}
\]
Example Enthalpy Calculation (cont’d)

- CEA Calculation: GH₂ fuel, Φ=1

\[ p_3 = 6.550 \text{ kPa} \]

\[ T_3 = 381.47 \text{ °K} \]

\[ \Delta h_{\text{combustion}} = C_p \left[ T_{\text{combustion}} - T_3 \right] = \frac{R_u}{M_w} \frac{\gamma}{\gamma - 1} \left[ T_{\text{combustion}} - T_3 \right] = \]

\[ \left( \frac{8314.4126}{24.519} \right) \left( \frac{1.1565}{1.1565 - 1} \right) (2238.92 - 381.47) = 4.655 \text{ MJ/kg} \]

Combustion Enthalpy is Less than Freestream Enthalpy!
Example Enthalpy Calculation (cont’d)

- Calculate thermodynamic efficiency of engine

\[
\eta = 1 - \left( \frac{p_\infty}{p_3} \right)^{\frac{\gamma-1}{\gamma}} \left[ \frac{T_C - \left( \frac{P_0_3}{P_0_\infty} \right)^{\frac{\gamma-1}{\gamma}} T_3}{T_C - T_3} \right]
\]

\[
\begin{align*}
T_B &= 381.47 \, ^\circ\text{K} \\
T_C &= 2238.92 \, ^\circ\text{K} \\
p_3/p_\infty &= 5.4960 \\
T_3/T_\infty &= 1.6831 \\
P_0_3/P_0_\infty &= 0.8832
\end{align*}
\]

\[
\gamma \sim (1.1565+1.4)/2 = 1.27825
\]

\[
1 - \left( \frac{1}{5.4960} \right)^{\frac{1.27825-1}{1.27825}} \left[ 2239.92 - \left( \frac{0.8832}{1.27825} \right) \right]^{381.47} \\
\frac{381.47}{(2239.92 - 381.47)} = 0.3061
\]

About 50% as efficient as our previous ramjet design (section 9.1) Operating at Mach 4 and 10km altitude (\(\eta=0.6\))
ScrRamjet Design Issues, III

• look at Mach 4 Ramjet problem (Section 9.1) …

let $P_{0B}/P_{0A} = 1$

$$
\eta = 1 - \left(\frac{P_A}{P_B}\right)^{\frac{\gamma - 1}{\gamma}} \frac{T_C - \left(\frac{P_{0B}}{P_{0A}}\right)^{\frac{\gamma - 1}{\gamma}} T_B}{(T_C - T_B)} =
$$

8% increase in Efficiency … Compared to Ramjet … but can we Do this? … no!

$$
1 - \frac{38.422^{1.4}}{(2662.82 - 1.856.61)} (2662.82 - 856.61) = 0.6475
$$
ScrRamjet design issues, III (cont’d)

- Supersonic flow cannot maintain stable combustion below ~ Mach 6

- Scramjets are feasible only for sustaining hypersonic speeds, not for achieving them from zero velocity

Scramjets Require Additional Cycles For An Operational Vehicle … and TRL for these technologies are low
ScrRamjet design issues, III (cont’d)

Combined cycle rocket engines

... Engines that are part jet engine
And part rocket motor!

... or part turbojet and part
SCRAMjet

Rocket + Scramjet = 
\textit{Rocket Based Combined Cycle (RBCC)}

OR

Turbine + Scramjet =
\textit{Turbine Based Combined Cycle (TBCC)}
Rocket-Based
Combined Cycle (RBCC) Mission Profile

Velocity as a Function of Altitude, HTOHL Dual-Fuel Strutjet, 65% Vc
LOX/Propane Rocket, LH2 RamScramjet, LOX/LH2 Rocket
RBCC Mission Profile (2)

The shaded areas on the velocity bars represent a range of uncertainty regarding the optimum transition velocity between the various engine modes or propellant types.

RBCC
RBCC Mission Profile (3)
Proposed RBCC Demo Program

Integrated System Test of an Air-breathing Rocket (ISTAR)
Turbine Based Combined Cycle (TBCC), I

Turbine-Scramjet Combination Engine

“Saddle Bag” Concept

- Separate flow paths
- Cleaner design

Low-Speed
Supersonic RTA (Mach 0-4+)
Revolutionary Turbine Accelerator

High-Speed
Hypersonic Scramjet (Mach 4-15)

Credit: Chuck McClinton NASA
TBCC Mission Profile

The shaded areas on the velocity bars represent a range of uncertainty regarding the optimum transition velocity between the various engine modes or propellant types.

TBCC
Turbine Based Combined Cycle (TBCC), II
(cont’d)

• Share parts of same flow path

• Weight savings
Rocket Based Combined Cycle (RBCC), I

- Ducted Rocket Approach
Rocket Based Combined Cycle (RBCC), II

- Dual Combustor Approach

- There is a precedent for this design
Pratt and Whitney J58 Turbojet/Ramjet Combined Cycle Engine

Part Turbo-jet

Part Ram-jet

Dual Burner … Same flow path

https://www.youtube.com/watch?v=F3ao5SCedIk
Pratt and Whitney J58 Turbojet/Ramjet Combined Cycle Engine (cont’d)

• Above mach 3 a portion of the flow bypasses the turbine and burns directly in afterburner providing about 80% or thrust …

• At lower speeds the engine operates as a normal supersonic Turbojet … same nozzle used by both operational modes
Thermal management

- The nature of the inlet design and need to minimize wave drag
  Mandate very sharp leading edges

- Leading edges generate extreme hypersonic heating rates
  In excess of 100 watts/cm$^2$
Heating is Minimized by Blunt Body

- Detached Normal Shockwave On Blunt Leading Edge Produces High level of Drag and Dissipates significant Portion of heat into flow

- High Drag Profiles Have Lower Levels of Total Hypersonic Heating
• Sharp Leading Edge … Much Higher Hypersonic Lift-to-Drag, but also significantly higher heating

\[
q_{\text{max}} \approx \frac{1}{\sqrt{R_{LE}}}
\]

Oblique Shockwave

• Flow attached at leading edge
  Heating impinges directly

• More Exotic Thermal Protection Systems Required
SCRAMJET DESIGN ISSUES, IV (cont’d)

- Sharp leading Edge has very high heating because of small radius

- Mach number and flow density are also Key players

\[
q_{LE} = H_f \left[ h_0 - h_{wall} \right] \frac{1}{R_{LE}} \sqrt{\frac{2}{\rho_{0_2}} \left( \frac{P_{0_2} - P_\infty}{\rho_{0_2}} \right)} \left[ C_{p_{\infty}} T_\infty + \frac{V_\infty^2}{2} - C_{p_{wall}} T_{wall} \right]^{1/2}
\]

Stagnation heating Rate
SCRAMJET DESIGN ISSUES, IV (cont’d)

• Small LE radius also has lower thermal capacity and the problem is compounded

\[ T_{\text{wall}} = \left( \Phi(\theta) H_{tf} \right) \left[ C_p T_\infty + \frac{V_\infty^2}{2} - C_{p\text{wall}} T_{\text{wall}} \right] + \left[ \frac{\alpha}{2} \sigma T_2^4 - \varepsilon \sigma T_{\text{wall}}^4 \right] \left[ \rho_{\text{LE}} C_{p\text{LE}} \tau_{\text{LE}} \right] \]

• Shuttle tile manages heat by having high emissivity and very high heat capacity .. But it limited to \(< 2000^\circ\text{K}\)
**SCRAMJET DESIGN ISSUES, IV**  
(cont’d)

- Technology readiness level (TRL) for UTHC TPS systems very low < 3/10

Mach 12+ TPS for sharp leading edge

“Ultra-High-Temperature Ceramics” (UHTC)
Even with matured UTHC TPS heating will have to be actively managed for long duration hypersonic flight …

\[
\dot{T}_{\text{wall}} = \left( \Phi(\theta)H_{tf} \right) \left[ C_p T_\infty + \frac{V_\infty^2}{2} - C_{p_{wall}} T_{\text{wall}} \right] + \left[ \frac{\alpha}{2} \sigma T_2^4 - \varepsilon \sigma T_{\text{wall}}^4 \right] - \left( \frac{\dot{q}}{\text{removed}} \right)
\]

Where do you put the heat you remove?
SCRAMJET DESIGN ISSUES, IV (cont’d)

• How About Ablative leading edges

  uneven recession
  causes detached
  leading edge shockwave

• Detached shockwave effects inlet flow path increases drag

• Emitted gases can effect mixture ratio of engine

• Non-receding charring ablative (NRCA)
SCRAMjet flight tests

• The high cost of flight testing and the unavailability of full enthalpy ground facilities have hindered scramjet development.

• A large amount of the experimental work on scramjets has been undertaken in cryogenic facilities, direct-connect tests, or burners, each of which simulates one aspect of the engine operation.

• Further, vitiated facilities, storage heated facilities, arc facilities and the various types of shock tunnels each have limitations which have prevented perfect simulation of scramjet operation.

• Full Enthalpy, full dynamic pressure data is a REAL RARITY
SCRAMjet flight tests (cont’d)

WHY FLIGHT TESTS?

"...to separate the real from the imagined and to make known the overlooked and the unexpected problems..." Hugh L. Dryden
SCRAMjet flight tests, CIAM

- Russian CIAM … mid 1990’s
- Supersonic Combustion
  Never verified by peer review … debate rages
SCRAMjet flight tests, HyShot

- U. Queensland (Australian) Hyshot Flight tests

- Flight 1 failed, Flights 2-4 successful
SCRAMjet flight tests, HyShot (cont’d)

- U. Queensland (Australian) Hyshot Flight tests
- Hyshot II First verified SCRAM flight operation July 30, 2002
- Engine only tests, not an integrated vehicle .. Hyshot I, II
  Flowpath tests …Never intended to produce more thrust than drag
HyShot II Mission Profile

- Terrier-Orion Mk 70 rocket
- Max liftoff spd: Mach 8+
- Liftoff accl: 22 g (60 g for 0.5 s)
- Apogee: 330 km
  - Nose is pushed over, cone
  - ejected (Bang-Bang maneuver)
- Max descent spd: Mach 7.6
  - Scramjet stage
  - Hydrogen Fueled
SCRAMjet flight tests, HyShot (cont’d)

• HyShot III Flight, March 25, 2006

• More Sophisticated 4-chamber axi-symmetric inlet design

• Teaming with British company Qinetiq

• Positive thrust accelerated vehicle from Mach 6.8 to Mach 8.0

• Hyshot IV data still being analyzed
SCRAMjet flight tests, HyShot (cont’d)
SCRAMjet flight tests, X-43A

- NASA X-43A, three flights

Flight 1, June 2 2001 … booster failure, terminated flight

Not a good sign

Here Comes the MIB.

Mishap Investigation Board
SCRAMjet flight tests, X-43A (cont’d)

- Took three years to get Problem fixed … flight 2, *March 27, 2004* successful Mach 7 max engine operation, *Flight 3, November 16, 2004*, Mach 10 successful operation
X-43A Vehicle

- Fuel: Hydrogen
- Igniter: Silane
- Thermal Barrier: Shuttle tiles
- Engine Coolant: Glycol/Water
- Nitrogen Purge
- Electric Actuators
X-43A firsts (cont’d)

• First flight of Integrated Scramjet Vehicle
  – Successful high dynamic pressure, high Mach, non-symmetrical stage separation (required for TSTO)

• Verified performance, operability and controllability
  – Airframe-integrated Scramjet
  – Integrated, powered, hypersonic airbreathing Vehicle

• Verified engineering application of NASA-Industry-University hypersonic vehicle design tools

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Credit: Chuck McClinton NASA
X-43A Lessons learned

• X-43 airframe drag (and lift) was slightly higher than nominal predicted, but within uncertainty prediction
• Scramjet engine performance was very close to preflight predictions (positive acceleration for M 7, Cruise for M 10)
• Control deflections to trim engine induced moments were very close to preflight predictions
• Other hypersonic vehicle technologies were as predicted
  — Aerodynamic stability and control
  — Natural and Tripped boundary layer transition
  — Airframe and wing structure
  — Thermal loads/Gap heating
  — TPS
  — Internal environment
  — Launch vehicle stiffness

Credit: Chuck McClinton NASA
USAF X-51A

X-51A FIRST FLIGHT
May 2010

Pratt & Whitney Rocketdyne

PTE = Performance Test Engine
GDE = Ground Demonstrator Engine
X = Experimental

Pratt & Whitney
A United Technologies Company
USAF X-51A (2)

- First Hydrocarbon Fueled ScramJet
USAF X-51A (3)

X-51A Cruiser Assembly

- Control System and Flight Test Instrumentation
- Fuel Tank
- Batteries
- Guidance Control Unit
- Fuel Pump
- Scramjet Engine Mounted on Underside of the Vehicle
Small Scale SCRAM experiments

- Simplified Approach to Scramjet Testing (SAST)
- Propulsion & Performance Branch (RP), NASA Dryden
- Small directionally-symmetric Mach 6 Scramjet design
- Configuration is aerodynamically stable
Small Scale SCRAM experiments
(cont’d)

SAST Design / Test Team

• **NASA Dryden, Edwards, CA**
  – Vehicle and experiment design, fabrication, and analysis.
  – Data acquisition / telemetry (Code FT).

• **NAWC-Weapons Division, Research Rockets Branch, White Sands Missile Range, NM**
  – Ground, range, and flight operations and safety.

• **NAWC-Weapons Division, Pt. Mugu, CA**
  – Flush mount antenna design, fabrication, and test.

• **NAWC, China Lake, CA**
  – Payload welding.

• **Industrial Solid Propulsion (ISP), Las Vegas, NV**
  – Rocket motors.
SAST Objectives

- Build experience with hypersonic flight test techniques and instrumentation at NASA Dryden.

- Evaluate the feasibility and value of a simple, low cost hypersonic flight testbed.

- Obtain hypersonic propulsion flight data for a simplified scramjet engine.

- Get operational expertise through high flight rate
Simplified Approach to SCRAMjet Testing (SAST)

**SAST Payload Internal Arrangement**

- Rate Gyros (2) (on cone base bulkhead)
- Linear Accelerometers (3)
- S and C-band Transmitters
- Instrumentation Boards
- Flush Antennas
- Fuel Bottle (inside cone)
- Scramjet Flow Paths
- Pyrotechnic Valve
- Fuel Injector Location (same for other flow path)
- Batteries (3)
- ESP (Pressure) Unit
- Pressure Switch (on aft bulkhead)
Simplified Approach to SCRAMjet Testing (SAST)

Scramjet Engine Design

- **Simple cone (9.5°) forebody**
  - High dynamic pressure flight delivers high combustor entrance pressure using simple conical shock-on-lip, shock on shoulder configuration.

- **Self-starting scoop inlet**
  - Swept sidewall, spilling design allows fixed geometry inlet to start at about Mach 3 and obtain full capture at design Mach of 6.

- **Diverging isolator (1°)**
  - Diverging isolator prevents combustor-inlet interactions.

- **Single orifice, normal fuel injection**
  - Fuel injection scheme chosen for simplicity but is easily changed to more optimum designs.

- **High pressure, gaseous hydrogen-silane fuel**
  - Proven hydrogen-silane pyrophoric fuel used for auto-ignition.

- **Diverging combustor / nozzle (4°)**
  - Conservative expansion ratio provides measurable engine thrust with low external aerodynamic drag.
Simplified Approach to SCRAMjet Testing (SAST)

SAST Trajectory

- Launch at elevation angle, $Q_E$, of 79 deg.
- Ballistic trajectory to impact. As directed by WSMR, no destruct system will be used.
- Test point (at rocket motor burnout) of Mach 5.2, 17,000 ft. approx. 6 seconds after launch.
- Maximum altitude of approx. 150,000 ft.
- Impact point approx. 18 nmi. downrange.
- Total flight time of approx. 220 seconds.
- Dispersion footprint within WSMR range.
Rocket Motor Description

• Viper-V Block-II solid rocket motor
• Manufacturer -- Industrial Solid Propulsion, Inc.
• Propellant -- 87% solids HTPB/AP/AL
• Dimensions -- 131 in. length and 7 in. dia.
• Weight -- 225 lbs.
• Thrust -- approx. 6,000 lbs for 5.5 seconds
• Motor case -- carbon/epoxy composite
• Launch lugs -- fixed T-rail aft lug, ejectable T-rail forward lug
Scramjet Fuel System

- High pressure, gaseous blow down system.
- Gaseous hydrogen-silane fuel stored in fuel tank at 1800 psi.
- Pyrotechnic valve used to release fuel to fuel injectors.
- Pressure switch sensing booster burn-out opens pyrotechnic valve.
- Fuel injectors sized for initial ER=0.2.
- Scramjet burn time of about 2 sec. (with decreasing ER).
- Predicted peak combustion pressure of about 300 psi and change in force of about 150 lbs.
Viper V Block II Motor Cross Section

Booster Burn through
First Flight March 2000

- Guess What? Booster Failure

- Dust yourself off, try it again!!!
Hypersonic Physics - Propulsion

- Natural and forced boundary layer transition
- Turbulence
- Separation caused by shock-boundary layer interaction
- Shock-shock interaction heating (Type 3 and 4)
- Isolator shock trains
- Cold-wall heat transfer
- Fuel injection, penetration and mixing
- Finite rate chemical kinetics
- Turbulence-chemistry interaction
- Boundary layer relaminarization
- Recombination chemistry
- Catalytic wall effects

- Lot of Promise but Long way to go

- Most of these phenomena were modeled in the design tools. Some were avoided by application of a uncertainty factors.

- X-43 success demonstrates an engineering level understanding of “the physics”. A better understanding of these issues will be beneficial for optimization of vehicle performance, but not “enabling”

- All designs share the same physics
Homework 4.3, Part 1

A ramjet operates at an altitude of 10,000 m ($T_a = 223\, K$, $P_a = 0.26\, atm$, $\gamma = 1.4$) at a Mach number of 1.7. The external diffusion is based on an oblique shock and on a normal shock, as described in the shown figure.

![Diagram of shock waves and pressure recovery](image_url)

Calculate
- Stagnation pressure recovery, $\frac{P_{02}}{P_{0a}}$?
- At what Mach number does the oblique shock become detached?
- What is the distance $x$, from the cone tip to the outer inlet lip, for the condition described in the figure?
- What is the best turning angle $\theta$ in terms of highest pressure ratio, $\frac{P_{02}}{P_{0a}}$?

Assume $\infty = a$
Homework 4.3, Part 2

\[ \beta_1 = 30^\circ \]
\[ M_1 = 2.8 \]
\[ p_1 = 1 \text{ atm} \]
\[ T_1 = 300^\circ \text{K} \]

- Calculate
  \[ \beta_2 \]
  \[ M_3 \]
  \[ p_3 \]
  \[ T_3 \]

- Assume \( \gamma = 1.4 \)
Questions??