Why hypersonic propulsion?
What's different about it?
AirCycles4Hypersonics.xls spreadsheet
“Conventional” ramjet
Scramjets
Sidebar: Pulse detonation engines
Hypersonic propulsion - motivation

- Why use air even if you’re going to space?
  - Carry only fuel, not fuel + O\(_2\), while in atmosphere
    - 8x mass savings (H\(_2\)-O\(_2\)), 4x (hydrocarbons)
    - Actually more than this when ln( ) term in Brequet range equation is considered
  - Use aerodynamic lifting body rather than ballistic trajectory
    - Ballistic: need Thrust/weight > 1
    - Lifting body, steady flight: Lift (L) = weight (mg); Thrust (T) = Drag (D), Thrust/weight = L/D > 1 for any decent airfoil, even at hypersonic conditions

What’s different about hypersonic propulsion?

- Stagnation temperature T\(_t\) - measure of total energy (thermal + kinetic) of flow - is really large even before heat addition - materials problems
  \[ T_t = T \left( 1 + \frac{\gamma - 1}{2} M^2 \right) \]
  - T = static temperature - T measured by a thermometer moving with the flow
  - T\(_t\) = temperature of the gas if it is decelerated adiabatically to M = 0
  - \(\gamma\) = gas specific heat ratio = C\(_p\)/C\(_v\); M = Mach number = u/(\(\gamma RT\))\(^{1/2}\)
- Stagnation pressure - measure of usefulness of flow (ability to expand flow) is really large even before heat addition - structural problems
  \[ P_t = P \left( 1 + \frac{\gamma - 1}{2} M^2 \right)^{\gamma / (\gamma - 1)} \]
  - P = static pressure - P measured by a pressure gauge moving with the flow
  - P\(_t\) = pressure of the gas if it is decelerated reversibly and adiabatically to M = 0
  - Large P\(_t\) means no mechanical compressor needed at large M
What’s different about hypersonic propulsion?

- Why are $T_1$ and $P_1$ so important? Recall isentropic expansion to $P_e = P_a$ (optimal exit pressure yielding maximum thrust) yields

$$u = \frac{2\gamma \text{RT}}{\gamma - 1} \left[ \left( \frac{P_a}{P_1} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]$$

- … but it’s difficult to add heat at high $M$ without major loss of stagnation pressure.

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What’s different about hypersonic propulsion?

- High temperatures: $\gamma$ not constant, also molecular weight not constant - dissociation - use GASEQ (http://www.gaseq.co.uk) to compute stagnation conditions.

- Example calculation: standard atmosphere at 100,000 ft
  - $T_1 = 227K$, $P_1 = 0.0108$ atm, $c_1 = 302.7$ m/s, $h_1 = 70.79$ kJ/kg (atmospheric data from http://www.digitaldutch.com/atmoscalc/)
  - Pick $P_2 > P_1$, compress isentropically, note new $T_2$ and $h_2$
  - 1st Law: $h_1 + u_1^2/2 = h_2 + u_2^2/2$; since $u_2 = 0$, $h_2 = h_1 + (M_1 c_1)^2/2$ or $M_1 = \left[ 2(h_2 - h_1)/c_1^2 \right]^{1/2}$
  - Simple relations ok up to $M = 7$
  - Dissociation not as bad as might otherwise be expected at ultra high $T$, since $P$ increases faster than $T$

- Limitations of these estimates
  - Ionization not considered
  - Stagnation temperature relation valid even if shocks, friction, etc. (only depends on 1st law) but stagnation pressure assumes isentropic flow
  - Calculation assumed adiabatic deceleration - radiative loss (from surfaces and ions in gas) may be important.

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What's different about hypersonic propulsion?

WOW! HOT WARM COLD

<table>
<thead>
<tr>
<th>Temperature</th>
<th>Species</th>
</tr>
</thead>
<tbody>
<tr>
<td>5000K</td>
<td>N+O+e⁻</td>
</tr>
<tr>
<td>3000K</td>
<td>N₂+O</td>
</tr>
<tr>
<td>1000K</td>
<td>N₂+O₂</td>
</tr>
<tr>
<td>200K</td>
<td>N₂+O₂</td>
</tr>
</tbody>
</table>

Stagnation pressure (atm)

Stagnation temperature (K)

Mach number

Conventional” ramjet

- Incoming air decelerated isentropically to M = 0 - high T, P
- No compressor needed
- Heat addition at M = 0 - no loss of P₁ - to max. allowable T (called T₁)
- Expand to Pₙ = P₁
- Doesn't work well at low M - P/P₁ & T/T₁, low - Carnot efficiency low
- As M increases, P/P₁ and T/T₁ increases, cycle efficiency increases, but if M too high, limited ability to add heat (T₁ close to Tₘₐₓ) - good efficiency but less thrust
“Conventional” ramjet - effect of $M_1$

- "Banana" shaped cycles for low $M_1$, tall skinny cycles for high $M_1$, "fat" cycles for intermediate $M_1$

Basic ramjet $T_2/T_1 = 7$

“Conventional” ramjet example

- Example: $M_1 = 5$, $T_2/T_a = 12$, $\gamma = 1.4$
- Initial state (1): $M_1 = 5$, $T_1 = T_a$, $P_1 = P_a$
- State 2: decelerate to $M_2 = 0$
  - $T_2 = T_a\left(1 + \frac{\gamma - 1}{2} M_1^2\right) = 6T_a$
  - $P_2 = P_a\left(1 + \frac{\gamma - 1}{2} M_1^2\right)^{\gamma-1} = 529.1P_a$
- State 4: add at heat const. $P$; $M_4 = 0$, $P_4 = 529.1P_a$, $T_4 = T_a = 12T_a$
- State 9: expand to $P_9 = P_1 = P_a$
  - $P_4 = 529.1P_a = P_9\left(1 + \frac{\gamma - 1}{2} M_9^2\right)^{\gamma-1} = P_a\left(1 + \frac{1.4 - 1}{2} M_9^2\right)^{1/\gamma-1} \Rightarrow M_9 = 5.00$
  - $T_4 = 12T_a = T_9\left(1 + \frac{\gamma - 1}{2} M_9^2\right) = T_a\left(1 + \frac{1.4 - 1}{2} 5^2\right) \Rightarrow T_9 = 2T_a$

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“Conventional” ramjet example

- Specific thrust (ST) (assume FAR $\ll 1$)

\[ \text{Thrust} = \dot{m} \left[ (1 + \text{FAR}) u_0 - u_1 \right] + (P_0 - P_1) A_0; \text{FAR} \ll 1, P_0 = P_1 \]

\[ \Rightarrow ST = \frac{\text{Thrust}}{\dot{m} a_1} = \frac{u_0}{a_1} - \frac{u_1}{a_1} = \frac{\dot{m} a_0}{a_0 a_1} - M_1 = M \sqrt{\frac{T_0}{T_1}} - M_1 = 5 \sqrt{2} - 5 = 2.07 \]

- TSFC and overall efficiency

\[ \text{TSFC} \equiv \frac{\text{Heat input}}{\text{Thrust} \cdot a_1} = \frac{\dot{m} C_p (T_4 - T_2)}{\text{Thrust} \cdot a_1} = \frac{\dot{m} a_1}{\text{Thrust}} \frac{C_p}{a_1} (T_4 - T_2) \]

\[ = \frac{1}{ST} \frac{\gamma}{\gamma - 1} R (12 T_1 - 6 T_1) = \frac{1}{2.07 1.4 - 1} (12 - 6) = 7.24; \]

\[ \eta_o = \frac{M_1}{\text{TSFC}} = \frac{5}{7.24} = 0.691 \]

“Conventional” ramjet - effect of $M_1$

- Basic ramjet $\tau_\lambda = 7$

- Plots showing thrust and fuel consumption, specific thrust, TSFC, and overall efficiency vs. flight mach number $M_1$. 

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Scramjet (Supersonic Combustion RAMjet)

- What if $T_t > T_{\text{max}}$ allowed by materials or $P_t > P_{\text{max}}$ allowed by structure?
  - Can't decelerate to $M = 0$!
- Need to mix fuel & burn supersonically, never allowing air to decelerate to $M = 0$
- Many laboratory studies, very few successful test flights (e.g. X-43 below)

Australian project:

US project (X-43): http://www1.nasa.gov/missions/research/x43-main.html
- Steady flight (thrust $\approx$ drag) achieved at $M_1 \approx 9.65$ at 110,000 ft altitude ($u_1 \approx 2934$ m/s $= 6562$ mi/hr)
- 3.8 lbs. $H_2$ burned during 10 - 12 second test
- Rich $H_2$-air mixtures ($\phi \approx 1.2 - 1.3$), ignition with silane ($SiH_4$, ignites spontaneously in air)
- …but no information about the conditions at the combustor inlet, or the conditions during combustion (constant $P$, $T$, area, …?)
- Real-gas stagnation temperatures 3300K (my model, slide 36: 3500K), surface temperatures up to 2250K (!)

- Acceleration to steady flight achieved at $M_1 \approx 5$ at 70,000 ft for 140 seconds using hydrocarbon fuel
Diffuser
- Mach number decrements from flight Mach number ($M_1$) to the specified value after the diffuser in 25 equal steps
- Stagnation pressure decrements from its value at $M_1$ ($P_{1t}$) to $\pi_d P_{1t} = \eta_d^{\gamma/(\gamma-1)}P_{1t}$ in 25 equal steps
- Static $P$ and $T$ are calculated from $M$ and $P_{1t}$, $T_{1t}$
- Sound speed $c$ is calculated from $T$, then $u$ is calculated from $c$ and $M$
- No heat input or work output in diffuser, but may have wall heat transfer
- Shocks not implemented (usually one would have a series of oblique shocks in an inlet, not a single normal shock)

Combustor
- Heat addition may be at constant area (Rayleigh flow), $P$ or $T$
- Mach number after diffuser is a specified quantity (not necessarily zero) - Mach number after diffuser sets compression ratio since there is no mechanical compressor
- Rayleigh curves starting at states 1 and 2 included to show constant area / no friction on T-s
- Nozzle
  - Static (not stagnation) pressure decrements from value after afterburner to specified exhaust pressure in 25 equal steps
  - Stagnation pressure decrements from its value after afterburner ($P_{7t}$) to $\pi n P_{7t} = \eta n^{\gamma/(\gamma-1)} P_{7t}$ in 25 equal steps
  - Static $P$ and $T$ are calculated from $M$ and $P_{t}$, $T_{t}$
  - Sound speed $c$ is calculated from $T$, then $u$ is calculated from $c$ and $M$
  - No heat input or work output in diffuser, but may have wall heat transfer
  - Heat transfer occurs according to usual law

- Combustion parameter $\tau_\lambda = T_{4t}/T_1$ (specifies stagnation temperature, not static temperature, after combustion)
- Caution on choosing $\tau_\lambda$
  - If $\tau_\lambda T_1 < \tau_1 T_1$ ($\tau_1 = 1 + (\gamma-1)/2 M_1^2$) (maximum allowable temperature after heat addition > temperature after deceleration) then no heat can be added (actually, spreadsheet will try to refrigerate the gas…)
  - For constant-area heat addition, if $\tau_\lambda T_1$ is too large, you can’t add that much heat (beyond thermal choking point) & spreadsheet “chokes”
  - For constant-T heat addition, if $\tau_\lambda T_1$ is too large, pressure after heat addition < ambient pressure - overexpanded jet - still works but performance suffers
  - For constant-P heat addition, no limits! 😊 But temperatures go sky-high 😊
  - All cases: $f$ (fuel mass fraction) needed to obtain specified $\tau_\lambda$ is calculated - make sure this doesn’t exceed $f_{\text{stoichiometric}}$!
Hypersonic propulsion (const. T) - T-s diagrams

- With Const. T combustion, maximum temperature within sane limits, but as more heat is added, P decreases, eventually \( P_4 < P_9 \).
- Also, latter part of cycle has low Carnot-strip efficiency since constant T and P lines will converge.

Const. T combustion, \( M_1 = 10; M_2 = 2.61 \) (\( T_2 = 2000K \))
Stoich. \( \text{H}_2\)-air (f = 0.0283, \( Q_R = 1.2 \times 10^8 \text{ J/kg} \) ⇒ \( \tau_\lambda = 35.6 \))

Hypersonic propulsion (const. T) - effect of \( M_1 \)

- Minimum \( M_1 = 6.28 \) - below that \( T_2 < 2000 \) even if \( M_2 = 0 \)
- No maximum \( M_2 \)
- \( \eta_{\text{overall}} \) improves slightly at high \( M_1 \) due to higher \( \eta_{\text{thermal}} \) (lower \( T_9 \))

Const. T combustion, \( M_1 = \text{varies}; M_2 \) adjusted so that \( T_2 = 2000K \);
\( \text{H}_2\)-air (\( Q_R = 1.2 \times 10^8 \text{ J/kg} \)), \( \tau_\lambda \) adjusted so that f = f_{stoichiometric}

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Hypersonic propulsion (const. T) - effect of $\tau_\lambda$

- $M_1 = 10$, $T_2 = 2000K$ specified $\Rightarrow M_2 = 2.61$
- At $\tau_\lambda = 21.1$ no heat can be added
- At $\tau_\lambda = 35.6$, $f = 0.0283$ (stoichiometric $H_2$-air)
- At $\tau_\lambda = 40.3$ (assuming one had a fuel with higher heating value than $H_2$), $P_4 = P_9$
- $f$ & Specific Thrust increase as more fuel is added ($\tau_\lambda$ increasing), $\eta_{\text{overall}}$ & $I_{\text{SP}}$ decrease due to low $\eta_{\text{thermal}}$ at high heat addition (see T-s diagram)

Hypersonic propulsion (const. T) - effect of $M_2$

- Maximum $M_2 = 3.01$ - above that $P_4 < P_9$ after combustion (you could go have higher $M_2$ but why would you want to - heat addition past $P_4 = P_9$ would reduce thrust!)
- No minimum $M_2$, but lower $M_2$ means higher $T_2$ - maybe beyond materials limits (after all, high $T_1$ is the whole reason we’re looking at alternative ways to burn at hypersonic Mach numbers)
- $\eta_{\text{overall}}$ decreases at higher $M_2$ due to lower $\eta_{\text{thermal}}$ (lower $T_2$)
Hypersonic propulsion (const. T) - effect of $\eta_d$

- Obviously as $\eta_d$ decreases, all performance parameters decrease
- If $\eta_d$ too low, pressure after stoichiometric heat addition < $P_1$, so need to decrease heat addition (thus $\tau_\lambda$)
- Diffuser can be pretty bad ($\eta_d \approx 0.25$) before no thrust

Hypersonic propulsion (const. T) - effect of $\eta_n$

- Obviously as $\eta_n$ decreases, all performance parameters decrease
- Nozzle can be pretty bad ($\eta_n \approx 0.32$) before no thrust, but not as bad as diffuser
Hypersonic propulsion (const. P) - T-s diagrams

- With Const. P combustion, no limitations on heat input, but maximum temperature becomes insane (actually dissociation & heat losses would decrease this T substantially)
- Carnot-strip (thermal) efficiency independent of heat input; same as conventional Brayton cycle (s-P-s-P cycle)

Const. P combustion, \( M_1 = 10; M_2 = 2.61 \) \( (T_2 = 2000K) \)
Stoich. \( H_2 \)-air \( (f = 0.0283, Q_R = 1.2 \times 10^8 \text{ J/kg} \Rightarrow \tau_\lambda = 35.6) \)

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Hypersonic propulsion (const. P) - performance

- \( M_1 = 10, T_2 = 2000K \) specified \( \Rightarrow M_2 = 2.61 \)
- Still, at \( \tau_\lambda = 21.1 \) no heat can be added
- At \( \tau_\lambda = 35.6, f = 0.0283 \) (stoichiometric \( H_2 \)-air)
- No upper limit on \( \tau_\lambda \) (assuming one has a fuel with high enough \( Q_R \))
- \( f \) & Specific Thrust increase as more fuel is added \( (\tau_\lambda \text{ increasing}), \eta_{\text{overall}} \text{ & } I_{SP} \)
decrease only slightly at high heat addition due to lower \( \eta_{\text{propulsive}} \)

Const. P combustion, \( M_1 = 10; M_2 = 2.61 \) \( (T_2 = 2000K) \)
\( H_2 \)-air \( (Q_R = 1.2 \times 10^8 \text{ J/kg}) \), \( \tau_\lambda \) (thus \( f \)) varies

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Hypersonic propulsion (const. A) - T-s diagrams

- With Const. A combustion, heat input limited by thermal choking, maximum temperature even more insane than constant P
- … but Carnot-strip efficiency is awesome!

Hypersonic propulsion (const. A) - performance

- $M_1 = 10, T_2 = 2000K$ specified $\Rightarrow M_2 = 2.61$
- Still, at $\tau_\lambda = 21.1$ no heat can be added
- At $\tau_\lambda = 30.5$, thermal choking at $f = 0.0193 < 0.0283$
- $f$ & Specific Thrust increase as more fuel is added ($\tau_\lambda$ increasing), $\eta_{\text{overall}}$ & $I_{\text{SP}}$ decrease slightly at high heat addition due to lower $\eta_{\text{propulsive}}$

**Const. A combustion, $M_1 = 10$; $M_2 = 2.61$ ($T_2 = 2000K$)**

$H_2$-air ($Q_R = 1.2 \times 10^8$ J/kg) $\Rightarrow \tau_\lambda = 30.1$; (can’t add stoichiometric amount of fuel at constant area for this $M_1$ and $M_2$)

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Consider a very simple propulsion system in a standard atmosphere at 100,000 feet (227K and 0.0107 atm, with $\gamma = 1.4$) in which

1. Incoming air is decelerated isentropically from $M = 15$ until $T = 3000K$
2. Heat is added at constant $T$ until ambient pressure is reached (not a good way to operate, but this represents a sort of maximum heat addition)

(a) To what Mach number could the air be decelerated if the maximum allowable gas temperature is 3000K? What is the corresponding pressure?

\[
T_i \left(1 + \frac{\gamma - 1}{2} M^2_i\right) = T_f \left(1 + \frac{\gamma - 1}{2} M^2_f\right)
\]

\[
(227K) \left(1 + \frac{1.4 - 1}{2} 15^2\right) = (3000K) \left(1 + \frac{1.4 - 1}{2} M^2_f\right)
\]

\[
M^2_f = \frac{2}{1.4 - 1} \left[\frac{227K}{3000K} \left(1 + \frac{1.4 - 1}{2} 15^2\right) - 1\right] = 12.403 \Rightarrow M_f = 3.522
\]

\[
P_f = \left(\frac{T_f}{T_i}\right)^{\frac{\gamma - 1}{\gamma}} = \left(\frac{3000K}{227K}\right)^{\frac{1.4 - 1}{1.4}} = 8391; P_f = 8391P_i = 8391(0.0107) = 89.79 atm
\]

(b) What is the exit Mach ($M_e$) number? What is the area ratio?

\[
P_e / P_f = \exp \left[\frac{\gamma}{2} (M_e^2 - M^2_i)\right]; \ln (P_e / P_f) = \frac{\gamma}{2} (M_e^2 - M^2_i)
\]

\[
M_e^2 = M^2_i - \frac{2}{\gamma} \ln (P_e / P_f) = 3.522^2 - \frac{2}{1.4} \ln (1/8391) = 25.31 \Rightarrow M_e = 5.031
\]

\[
A_e / A_i = \frac{M_i}{M_e} \exp \left[\frac{\gamma}{2} (M_e^2 - M^2_i)\right] = \frac{3.522}{5.031} \exp \left[\frac{1.4}{2} (5.031^2 - 3.522^2)\right] = 5869
\]

(c) What is the specific thrust?

\[
ST = \text{Thrust}/m_e c_1 = \dot{m}_e (u_e - u_i)/m_e c_1 = (M_e c_e - M_i c_i)/c_1 = \frac{M_e (T_e / T_i)^{1/2} - M_i}{T_e / T_i}
\]

\[
ST = M_e (T_e / T_i)^{1/2} - M_i = 5.031(3000K/227K)^{1/2} - 15 = 3.289
\]
Example - continued

(d) What is the thrust specific fuel consumption?

\[ TSFC = \frac{\text{(Heat input)}}{\text{Thrust} \times c_i} = \frac{[\dot{m}_i(C_p(T_{3i} - T_{2i})c_i)]}{[\text{Thrust} \times c_i^2]} \]

\[ = \frac{[\dot{m}_i c_i]}{\text{Thrust}} \times \left[ \frac{\gamma}{(\gamma - 1)} \right] R(T_{3i} - T_{2i}) = \frac{[\gamma / (\gamma - 1)]}{[\text{ST}]} \times [1 / (\gamma - 1)] \times \left[ (T_{3i} - T_{2i}) / T_i \right] \]

\[ T_o = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) \]

\[ T_a = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) \]

\[ T_2 = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) - \frac{T_a - T_o}{T_i} \]

\[ T_3 = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) - \frac{T_a - T_o}{T_i} \]

\[ T_0 = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) - \frac{T_a - T_o}{T_i} \]

\[ T_s = T_i \left( 1 + \frac{\gamma - 1}{2} M_i^2 \right) - \frac{T_a - T_o}{T_i} \]

\[ TSFC = \frac{\gamma}{\gamma - 1} \left( \frac{T_o - T_a}{T_i} \right) \frac{T_i - T_s}{T_i} = 3.279 \frac{1}{1.4 - 1} (34.11) = 26.01 \text{ (lousy!)} \]

(c) Can stoichiometric hydrogen-air mixtures generate enough heat to accomplish this?

Determine if the heat release per unit mass \( f_{\text{stoich}} Q_b \) is equal to or greater than the heat input needed \( C_P(T_{3i} - T_{2i}) \).

\[ C_P(T_{3i} - T_{2i}) = \frac{\gamma R T_o}{\gamma - 1} \frac{T_i - T_s}{T_i} = 1.4 \left( \left[ \frac{8.314 \text{ J/moleK}}{0.02897 \text{ kg/mole}} \right] / 1.4 - 1 \right) (34.11) (227 \text{ K}) \]

\[ f_{\text{stoich}} Q_b = (0.0283) (1.20 \times 10^4 \text{ J/kg}) = 3.396 \times 10^4 \text{ J/kg} \]

Heat input \( C_P(T_{3i} - T_{2i}) \) is 7.777 \times 10^4 \text{ J/kg}.

 Requirement is higher by a factor of \( \approx 2.3 \), so H₂-air cannot provide this much heat release.

Summary

- Propulsion at high Mach numbers is very different from conventional propulsion because
  - The optimal thermodynamic cycle (decelerate to \( M = 0 \)) yields impractically high \( T \) & \( P \)
  - Deceleration from high \( M \) to low \( M \) without major \( P \) losses is difficult
  - Propulsive efficiency \( \approx 2u_i / (u_i + u_o) \) is always high
- 3 ways of adding heat discussed
  - Constant \( T \)
    - Probably most practical case
    - Low efficiency with large heat addition
    - Large area ratios
  - Constant \( P \) - best performance but very high \( T \)
  - Constant \( A \) - thermal choking limits heat input
Sidebar topic: pulse detonation engine

- Discussed in more detail in AME 514
- Simple system - fill tube with detonable mixture, ignite, expand exhaust
- Something like German WWII “buzz bombs” that were “Pulse Deflagration Engines”
- Advantages over conventional propulsion systems
  - Nearly constant-volume cycle vs. constant pressure - higher ideal thermodynamic efficiency
  - No mechanical compressor needed
  - (In principle) can operate from zero to hypersonic Mach numbers

Fill Tube → Detonate Mixture → Exhaust

Refill Tube, Repeat

Courtesy Fred Schauer

Pulse detonation engine concept

- Challenges (i.e. problems...)
  - Detonation initiation in small tube lengths
  - Deceleration of gas to low M at high flight M
  - Fuel-air mixing
  - Noise

Kailasanath (2000)
PDE Research Engine - Wright-Patterson Air Force Base

- Pontiac Grand Am engine driven by electric motor used as air pump to supply PDE
- Allows study of high frequency operation, multi-tube effects

Photos courtesy F. Schauer

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Videos of $H_2$-air PDE in operation

- 1 Tube @ 16Hz
- 4 Tubes @ 4Hz each
- 2 Tubes @ High Frequency

Videos courtesy F. Schauer

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Performance of “laboratory” PDE

- Performance (Schauer et al., 2001) using H₂-air similar to predictions unless too lean (finite-rate chemistry, not included in calculations)

![Graph showing performance vs. PHI]

Effect of tube fill fraction

- Better performance with lower tube fill fraction - better propulsive efficiency (accelerate large mass by small Δu, just like turbofan vs. turbojet), but this is of little importance at high M₁ where Δu << u₁

![Graph showing effect of tube fill fraction]