Propulsion Systems Design

- Lecture #18 - October 30, 2014
- Crew Systems/PPT Systems Project
- Rocket engine basics
- Survey of the technologies
- Propellant feed systems
- Propulsion systems design
Notes:

• I will be posting separately (later today) details on the third team project (Crew Systems/Power, Propulsion, and Thermal Systems) and the next problem set

• From here on out, a single problem set will be issued for each of the disciplinary lecture sets (CS, PPT, LSM, AVS)

• Today at 3:30 in AVW 2116, the guest speaker in ENAE 788X will be Robert Giglio, chief rover designer for Astrobotic
Propulsion Taxonomy

- Mass Expulsion
  - Thermal
    - Chemical
      - Monopropellants
      - Bipropellants
    - Non-Chemical
      - Nuclear
        - Electrical
      - Cold Gas
        - Solar
          - Solar Sail
          - Laser Sail
          - Microwave Sail
          - MagnetoPlasma
      - Beamed
        - MPD
          - Ion
- Non-Mass Expulsion
  - Nuclear
  - Electrical
  - Cold Gas
  - Solids
  - Hybrids
  - Liquids
  - Air-Breathing
  - Pressure-Fed
  - Pump-Fed

UNIVERSITY OF MARYLAND

ENAE 483/788D - Principles of Space Systems Design
Thermal Rocket Exhaust Velocity

• Exhaust velocity is

\[ V_e = \sqrt{\frac{2\gamma \, \mathcal{R} T_0}{\gamma - 1 \, \overline{M}}} \left[ 1 - \left( \frac{p_e}{p_0} \right)^{\frac{\gamma - 1}{\gamma}} \right] \]

where

\[ \overline{M} \equiv \text{average molecular weight of exhaust} \]

\[ \mathcal{R} \equiv \text{universal gas const.} = 8314.3 \, \frac{\text{Joules}}{\text{mole} \, ^\circ\text{K}} \]

\[ \gamma \equiv \text{ratio of specific heats} \approx 1.2 \]
Ideal Thermal Rocket Exhaust Velocity

• Ideal exhaust velocity is

\[ V_e = \sqrt{\frac{2\gamma \, R T_0}{\gamma - 1 \, M}} \]

• This corresponds to an ideally expanded nozzle
• All thermal energy converted to kinetic energy of exhaust
• Only a function of temperature and molecular weight!
Thermal Rocket Performance

- Thrust is
  \[ T = \dot{m}V_e + (p_e - p_{amb})A_e \]

- Effective exhaust velocity
  \[ T = \dot{m}c \Rightarrow c = V_e + (p_e - p_{amb}) \frac{A_e}{\dot{m}} \]

- Expansion ratio
  \[
  \frac{A_t}{A_e} = \left( \frac{\gamma + 1}{2} \right)^{\frac{1}{\gamma - 1}} \left( \frac{p_e}{p_0} \right)^{\frac{1}{\gamma}} \sqrt{\frac{\gamma + 1}{\gamma - 1}} \left[ 1 - \left( \frac{p_e}{p_0} \right)^{\frac{\gamma - 1}{\gamma}} \right]
  \]

\[ I_{sp} = \frac{c}{g_0} \]
A Word About Specific Impulse

• Defined as “thrust/propellant used”
  – English units: lbs thrust/(lbs prop/sec)=sec
  – Metric units: N thrust/(kg prop/sec)=m/sec

• Two ways to regard discrepancy -
  – “lbs” is not mass in English units - should be slugs
  – Isp = “thrust/weight flow rate of propellant”

• If the real intent of specific impulse is

\[ I_{sp} = \frac{T}{\dot{m}} \text{ and } T = \dot{m}V_e \text{ then } I_{sp} = V_e !!! \]
Nozzle Design

- Pressure ratio $p_0/p_e=100$ (1470 psi-->14.7 psi)
  $A_e/A_t=11.9$

- Pressure ratio $p_0/p_e=1000$ (1470 psi-->1.47 psi)
  $A_e/A_t=71.6$

- Difference between sea level and ideal vacuum $V_e$

\[
\frac{V_e}{V_{e,\text{ideal}}} = \sqrt{1 - \left(\frac{p_e}{p_0}\right)^{\gamma-1}}
\]

- $I_{sp,\text{vacuum}}=455$ sec  --> $I_{sp,\text{sl}}=333$ sec
Solid Rocket Motor

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Solid Propellant Combustion Characteristics

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Solid Grain Configurations

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Short-Grain Solid Configurations

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Advanced Grain Configurations

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
A completed J-2 rocket engine (left), with its pumps and lines installed. The basic engine structure is built up from a series of hollow tubes (right).
Liquid Propellant Feed Systems

(a) Pump-fed rocket

(b) Pressure-fed rocket
Space Shuttle OMS Engine

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 2001
Turbopump Fed Liquid Rocket Engine

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Sample Pump-fed Engine Cycles

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Gas Generator Cycle Engine

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
SSME Engine Cycle

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 2001
Liquid Rocket Engine Cutaway

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 2001
H-1 Engine Injector Plate
Injector Concepts

From G. P. Sutton, Rocket Propulsion Elements (5th ed.)
John Wiley and Sons, 1986
TR-201 Engine (LM Descent/Delta)
Solid Rocket Nozzle (Heat-Sink)

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Ablative Nozzle Schematic

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Active Chamber Cooling Schematic

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Boundary Layer Cooling Approaches

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Hybrid Rocket Schematic

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Hybrid Rocket Combustion

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
## Thrust Vector Control Approaches

<table>
<thead>
<tr>
<th>Gimbal or hinge</th>
<th>Jet vanes</th>
<th>Small control thrust chambers</th>
<th>Turbine exhaust gas control</th>
<th>Side injection</th>
</tr>
</thead>
<tbody>
<tr>
<td>Universal joint suspension</td>
<td>Four rotating heat resistant aerodynamic vanes in jet</td>
<td>Two or more gimbaled auxiliary thrust chambers</td>
<td>Gimbal on turbine exhaust nozzle</td>
<td>Secondary fluid injection on one side only</td>
</tr>
</tbody>
</table>

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Reaction Control Systems

- Thruster control of vehicle attitude and translation
- “Bang-bang” control algorithms
- Design goals:
  - Minimize coupling (pure forces for translation; pure moments for rotation) except for pure entry vehicles
  - Minimize duty cycle (use propellant as sparingly as possible)
  - Meet requirements for maximum rotational and linear accelerations
Single-Axis Equations of Motion

\[ \tau = I \ddot{\theta} \]

\[ \frac{\tau}{I} t = \dot{\theta} + C_1 \]

at \( t = 0 \), \( \dot{\theta} = \dot{\theta}_o \) \( \implies \frac{\tau}{I} t = \dot{\theta} - \dot{\theta}_o \)

\[ \frac{1}{2} \frac{\tau}{I} t^2 + \dot{\theta}_o t = \theta + C_2 \]

at \( t = 0 \), \( \theta = \theta_o \) \( \implies \frac{1}{2} \frac{\tau}{I} t^2 + \dot{\theta}_o t = \theta - \theta_o \)

\[ \frac{1}{2} \left( \dot{\theta}^2 - \dot{\theta}_o^2 \right) = \frac{\tau}{I} (\theta - \theta_o) \]
Attitude Trajectories in the Phase Plane
Gemini Entry Reaction Control System

RCS FUNCTION

RE-ENTRY MODULE

ROLL

YAW

PITCH
Apollo Reaction Control System
Apollo CSM RCS Assembly
Lunar Module Reaction Control System
LM RCS Quad
Viking Aeroshell RCS Thruster
Space Shuttle Primary RCS Engine

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Monopropellant Engine Design

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
Cold Gas Thruster Exhaust Velocity

Assume nitrogen gas thrusters

\[ V_e = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{RT_0}{\bar{M}}} \left[ 1 - \left( \frac{p_e}{p_0} \right)^{\frac{\gamma - 1}{\gamma}} \right] \]

\[ \bar{M} = 28 \]
\[ T_0 = 300 \text{ K} \]
\[ \gamma = 1.4 \]
\[ R = 8314.3 \]
\[ p_0 = 300 \text{ psi} \]
\[ p_e = 2 \text{ psi} \]

\[ V_e = \sqrt{\frac{2(1.4)}{1.4 - 1} \frac{8314.3(300)}{28}} \left[ 1 - \left( \frac{2}{300} \right)^{\frac{1.4 - 1}{1.4}} \right] = 689 \text{ m/sec} \]
# Cold-gas Propellant Performance

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Molecular Mass</th>
<th>Density[^a] (lb/ft³)</th>
<th>Theoretical Specific Impulse (sec)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hydrogen</td>
<td>2.0</td>
<td>1.21</td>
<td>296</td>
</tr>
<tr>
<td>Helium</td>
<td>4.0</td>
<td>2.37</td>
<td>179</td>
</tr>
<tr>
<td>Methane</td>
<td>16.0</td>
<td>12.10</td>
<td>114</td>
</tr>
<tr>
<td>Nitrogen</td>
<td>28.0</td>
<td>17.37</td>
<td>80</td>
</tr>
<tr>
<td>Air</td>
<td>28.9</td>
<td>19.3</td>
<td>74</td>
</tr>
<tr>
<td>Argon</td>
<td>39.9</td>
<td>27.60</td>
<td>57</td>
</tr>
<tr>
<td>Krypton</td>
<td>83.8</td>
<td>67.20</td>
<td>39</td>
</tr>
<tr>
<td>Freon 14</td>
<td>88.0</td>
<td>60.01</td>
<td>55</td>
</tr>
<tr>
<td>Carbon dioxide</td>
<td>44.0</td>
<td>Liquid</td>
<td>67</td>
</tr>
</tbody>
</table>

[^a]: At 3500 psia and 0°C.
Total Impulse

- Total impulse \( I_t \) is the total thrust-time product for the propulsion system, with units \(<\text{N-sec}>\)

\[
I_t = Tt = \dot{m}v_e t
\]

\[
t = \frac{\rho V}{\dot{m}}
\]

\[
I_t = \rho V v_e
\]

- To assess cold-gas systems, we can examine total impulse per unit volume of propellant storage

\[
\frac{I_t}{V} = \rho v_e
\]
Performance of Cold-Gas Systems

![Graph showing performance of different gases for cold-gas systems.](image)

- **Exhaust Velocity (m/sec)**: The horizontal axis represents the exhaust velocity of different gases. The exhaust velocity is measured in meters per second (m/sec).

- **Total Impulse (Ns/cm^2)**: The vertical axis represents the total impulse delivered by the gases. The total impulse is calculated as the product of the exhaust velocity and the mass flow rate, all integrated over time.

Different gases are represented by different markers on the graph:
- **Red squares** indicate exhaust velocity data points.
- **Blue diamonds** show total impulse data points.

The gases listed on the x-axis include:
- Hydrogen
- Helium
- Methane
- Nitrogen
- Air
- Oxygen
- Argon
- Krypton
- Freon 14
- Carbon Dioxide
- Nitrous Oxide

This graph illustrates how different gases perform in cold-gas systems, considering their exhaust velocity and total impulse deliverability.
Self-Pressurizing Propellants (CO$_2$)
Self-Pressurizing Propellants ($N_2O$)

Density 1300 kg/m³

Density 625 kg/m³
N$_2$O Performance Augmentation

- Nominal cold-gas exhaust velocity $\sim$600 m/sec
- N$_2$O dissociates in the presence of a heated catalyst $2N_2O \rightarrow 2N_2 + O_2$
  
  engine temperature $\sim$1300°C
  
  exhaust velocity $\sim$1800 m/sec
- NOFB (Nitrous Oxide Fuel Blend) - store premixed N$_2$O/hydrocarbon mixture
  
  exhaust velocity $>$3000 m/sec
Pressurization System Analysis

Adiabatic Expansion of Pressurizing Gas

\[ p_{g,0} V_g^\gamma = p_{g,f} V_g^\gamma + p_L V_L^\gamma \]

Known quantities:
- \( P_{g,0} = \) Initial gas pressure
- \( P_{g,f} = \) Final gas pressure
- \( P_L = \) Operating pressure of propellant tank(s)
- \( V_L = \) Volume of propellant tank(s)

Solve for gas volume \( V_g \)
Boost Module Propellant Tanks

- Gross mass 23,000 kg
  - Inert mass 2300 kg
  - Propellant mass 20,700 kg
  - Mixture ratio $N_2O_4/A50 = 1.8$ (by mass)

- $N_2O_4$ tank
  - Mass = 13,310 kg
  - Density = 1450 kg/m$^3$
  - Volume = 9.177 m$^3$ --> $r_{sphere} = 1.299$ m

- Aerozine 50 tank
  - Mass = 7390 kg
  - Density = 900 kg/m$^3$
  - Volume = 8.214 m$^3$ --> $r_{sphere} = 1.252$ m
Boost Module Main Propulsion

- Total propellant volume \( V_L = 17.39 \text{ m}^3 \)
- Assume engine pressure \( p_0 = 250 \text{ psi} \)
- Tank pressure \( p_L = 1.25 \times p_0 = 312 \text{ psi} \)
- Final GHe pressure \( p_{g,f} = 75 \text{ psi} + p_L = 388 \text{ psi} \)
- Initial GHe pressure \( p_{g,0} = 4500 \text{ psi} \)
- Conversion factor 1 psi = 6892 Pa
- Ratio of specific heats for He = 1.67
  \[
  (4500 \text{ psi}) V_g^{1.67} = (388 \text{ psi}) V_g^{1.67} + (312 \text{ psi})(17.39 \text{ m}^3)^{1.67}
  \]
- \( V_g = 3.713 \text{ m}^3 \)
- Ideal gas:
  \[
  \rho = 49.7 \text{ kg/m}^3 \quad (4500 \text{ psi} = 31.04 \text{ MPa})
  \]
  \[
  T = 300^\circ \text{K} \quad \Rightarrow \quad \rho_{He} = \frac{p_{g,0} \bar{M}}{\mathcal{R} T_0}
  \]
  \[
  M_{He} = 185.1 \text{ kg}
  \]
Nuclear Thermal Rockets

- Heat propellants by passing through nuclear reactor
- Isp limited by temperature limits on reactor elements (~900 sec for H2 propellant)
- Mass impacts of reactor, shielding
- High thrust system
VASIMR Engine Concept

1st stage: helicon plasma generator
2nd stage: ion cyclotron resonance power amplifier
3rd stage: magnetic nozzle
Ion Propulsion

- Uses electrostatic forces to accelerate ions
- Injects electrons to keep beam neutral
- High Isp (~3000 sec) at low thrust (~10 N)
- Substantial mass penalty for electrical power generation
Solar Sails

• Sunlight reflecting off sail produces momentum transfer

\[ T = 2mV = 2mc \]

\[ E = mc^2 \Rightarrow m = \frac{E}{c^2} \Rightarrow \dot{m} = \frac{E}{t} \frac{1}{c^2} = \frac{P}{c^2} \]

• At 1 AU, \( P = 1394 \) W/m\(^2\)
• \( c = 3 \times 10^8 \) m/sec
• \( T = 9 \times 10^{-6} \) N/m\(^2\)