Propulsion Systems Design

• Lecture #18 - October 29, 2020
• Rocket engine basics
• Survey of the technologies
• Propellant feed systems
• Propulsion systems design
Grading Rubric for Team Project #2

- Overall design (interior/exterior views, dimensioned 3-view, functionality in mission application) – 10 pts
- Habitability design (crew accommodations, usage allocation, galley, waste management, windows) - 10 pts
- Life support systems design (trade studies, installation in CAD design, provision of resupply) – 10 pts
- Accommodation of visiting vehicles (docking ports, hatches, cargo interfaces) – 10 pts
- EVA accommodations (airlock(s)/suitports, suit donning/doffing/recharging stations, dust remediation) – 10 pts
Grading Rubric for Term Project #2

- Logistics management (volume allocation, crew access, ability to restock) – 10 pts
- Concept of operations (How will the habitat be used? “day in the life”… how well does it fit the mission?) – 10 pts
- Presentation quality (“telling the story”, engaging the reader, adhering to principles from class) - 10 pts
- CAD quality (details, fidelity, complexity) - 20 pts
- “Above and beyond” (extra effort outside of nominal expectations) - 10 pts extra credit
Thermal Rocket Exhaust Velocity

• Exhaust velocity is

\[ v_e = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{\mathcal{R} T_o}{\bar{M}}} \left[ 1 - \left( \frac{p_e}{p_o} \right)^{\frac{\gamma - 1}{\gamma}} \right] \]

where

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

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

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

- Ideal exhaust velocity is
  \[ v_{e,\text{ideal}} = \sqrt{\frac{2\gamma \, R \, T_0}{\gamma - 1 \, \bar{M}}} \]

- 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
  \[ T = \dot{m}c \implies c = v_e + (p_e - p_{amb}) \frac{A_e}{\dot{m}} \]
  \[ I_{sp} = \frac{c}{g_o} \]
- Expansion ratio
  \[ \frac{A_t}{A_e} = \left( \frac{\gamma + 1}{2} \right)^{\frac{1}{\gamma-1}} \left( \frac{p_e}{p_o} \right)^{\frac{1}{\gamma}} \sqrt{\frac{\gamma + 1}{\gamma - 1} \left[ 1 - \left( \frac{p_e}{p_o} \right)^{\frac{\gamma - 1}{\gamma}} \right]} \]
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 vacuum $V_e$

$$
\frac{v_{e1}}{v_{e2}} = \sqrt{\frac{\frac{\gamma - 1}{p_o^\gamma} - \frac{\gamma - 1}{p_{e1}^\gamma}}{\frac{\gamma - 1}{p_o^\gamma} - \frac{\gamma - 1}{p_{e2}^\gamma}}}
$$

- $I_{sp,\text{vacuum}}=455$ sec  -->  $I_{sp,\text{sl}}=397$ 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
Liquid Rocket Engine

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

Oxidizer tank
Fuel tank
Pump
Turbine
Pump
Gas generator
Valve
Thrust chamber

High-pressure gas

Oxidizer tank
Fuel tank
Valve
Thrust chamber
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 Engine Schematic
SpaceX Merlin 1D Engines
Falcon 9 Octoweb Engine Mount
RD-180 Engine(s) (Atlas V)
SSME Powerhead Configuration
SSME Engine Cycle

SSME FLOW DIAGRAM

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986
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 gimballed auxiliary thrust chambers</td>
<td>Gimbal on turbine exhaust nozzle</td>
<td>Secondary fluid injection on one side only</td>
</tr>
</tbody>
</table>

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 \quad \implies \quad \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 \quad \implies \quad \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
Apollo Reaction Control System Thrusters
RCS Quad
Apollo CSM RCS Assembly
Lunar Module Reaction Control System
LM RCS Quad
Viking Aeroshell RCS Thruster
Viking RCS Thruster Schematic
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

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

\[ \bar{M} = 28 \]
\[ T_0 = 300 \, K \]
\[ \mathcal{R} = 8314.3 \]

\[ p_0 = 300 \, psi \]
\[ p_e = 2 \, psi \]
\[ \gamma = 1.4 \]

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

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Molecular Mass</th>
<th>Density (^a) (lb/ft(^3))</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\(^\circ\)C.
Total Impulse

- Total impulse $I_t$ is the total thrust-time product for the propulsion system, with units $<$N-sec$>$

$$ I_t = Tt = \dot{m}v_et $$

$$ t = \frac{\rho V}{\dot{m}} $$

$$ I_t = \rho Vv_e $$

- 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

Exhaust Velocity (m/sec) vs. Total Impulse (Nsec/m^3)

- Hydrogen
- Helium
- Methane
- Nitrogen
- Air
- Oxygen
- Argon
- Krypton
- Freon 14
- Carbon Dioxide
- Nitrous Oxide

**Legend:**
- Exhaust Velocity
- Total Impulse
Self-Pressurizing Propellants (CO₂)
Self-Pressurizing Propellants ($N_2O$)

![Graph showing vapor pressure vs. temperature for $N_2O$.](image)

- Density $1300 \text{ kg/m}^3$
- Density $625 \text{ kg/m}^3$
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 $\sim 1300^\circ C$
  - 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 $\text{N}_2\text{O}_4/A50 = 1.8$ (by mass)

- $\text{N}_2\text{O}_4$ tank
  - Mass = 13,310 kg
  - Density = 1450 kg/m$^3$
  - Volume = 9.177 m$^3$ $\Rightarrow r_{\text{sphere}} = 1.299$ m

- Aerozine 50 tank
  - Mass = 7390 kg
  - Density = 900 kg/m$^3$
  - Volume = 8.214 m$^3$ $\Rightarrow r_{\text{sphere}} = 1.252$ 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:
  \[ \rho = 49.7 \text{ kg/m}^3 \] (4500 psi = 31.04 MPa)
  $M_{He} = 185.1 \text{ kg}$
Autogenous Pressurization

- Use gaseous propellants to pressurize tanks with liquid propellants
- Heat exchanger to gasify and warm propellants, then route back into ullage volume
- Eliminates need for pressurized gases for ullage and high-pressure storage bottles (e.g., Falcon 9 failures)
- Issue: start-up transient
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
Electrostatic Ion Thruster
Hall-Effect Thruster

boron nitride walls

anode / gas distributor

inner magnetic coil

magnetic circuit

outer magnetic coil

cathode neutralizer

axial electric field

radial magnetic field

thruster exhaust
Ion Engine Schematic

- **Propellant Injection**
- **Electrons Emitted by Hollow Cathode Traverse Discharge and are Collected by Anode**
- **Electrons Impact Atoms to Create Ions**
- **Magnetic Field Enhances Ionization Efficiency**
- **Ions Electrostatically Accelerated**
- **Magnet Rings**
- **Anode**
- **Positive Grid (+1090V)**
- **Negative Grid (-225V)**
- **Electrons Injected into Beam for Neutralization**
- **Hollow Cathode Plasma Bridge Neutralizer**
- **Discharge Plasma**
- **Ion Beam**
Ion Engines Existing/In Development

- NSTAR (NASA Solar Technology Application Readiness)
  - DS1 and Dawn
  - 30 cm, 2.3 kW, 92 mN, 3120 sec
- NEXT (NASA Evolutionary Xenon Thruster)
  - Available 2019
  - 6.9 kW power, 236 mN thrust, Isp 4190 sec
- HiPEP (High Power Electric Propulsion)
  - TRL 3
  - 670 mN, 39.3 kW, 7 mg/sec prop, Isp 9620 sec
Variable Specific Impulse Magnetoplasma

Step 1. Injector feeds neutral gas

Step 2. Helicon coupler ionizes propellant

Step 3. Superconductor generates magnetic field that confines plasma

Step 4. ICH coupler heats plasma to ~ 1 million degrees

Step 5. Thrust generated as plasma escapes magnetic confinement

VASIMR Engine Concept

1st stage: helicon plasma generator
2nd stage: ion cyclotron resonance power amplifier
3rd stage: magnetic nozzle
VASIMR Operating Specifications

- Optimum operating point
  - Isp 5000 sec
  - Thrust 5.7 N
  - Power 200 kW

- Can be derated for higher thrust at lower Isp

- Compare to ion engines at equivalent power
  - 87 NSTAR thrusters: 8 N, 3120 sec Isp
  - 29 NEXT thrusters: 6.8 N, 4190 sec Isp
  - 5 HiPEP thrusters: 3.4 N, 9620 sec Isp
Solar Sails

- Sunlight reflecting off sail produces momentum transfer

\[ T = 2\dot{m}V = 2\dot{mc} \]

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

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

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