Propulsion Systems Design

- Rocket engine basics
- Survey of the technologies
- Propellant feed systems
- Propulsion systems design
Overview of the Design Process

- **Program Objectives**
  - **System Requirements**

- **Vehicle-level Estimation**
  - (based on a few parameters from prior art)

- **System-level Estimation**
  - (system parameters based on prior experience)

- **System-level Design**
  - (based on discipline-oriented analysis)

Increasing complexity
- Increasing accuracy
- Decreasing ability to comprehend the "big picture"
Propulsion Taxonomy

- Mass Expulsion
  - Thermal
  - Non-Thermal
- Non-Mass Expulsion
**Thermal Rocket Exhaust Velocity**

- Exhaust velocity is

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

where

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

\[
R \equiv \text{universal gas const.} = 8314.3 \text{ Joules mole}^{-1} \text{ K}^{-1}
\]

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

- Ideal exhaust velocity is

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

- 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

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

- Expansion ratio

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

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

- $I_{sp,vacuum}=455$ sec  --> $I_{sp,sl}=333$ sec
Propulsion Taxonomy

- Mass Expulsion
  - Thermal
    - Chemical
      - Monopropellants
      - Bipopellants
        - Solids
        - Hybrids
        - Liquids
        - Air-Breathing
  - Non-Thermal
    - Non-Chemical

- Non-Mass Expulsion
Solid Rocket Motor

- Grain
- Case wall
- Nozzle
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
**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 $\text{N}_2\text{O}_4/\text{A50} = 1.8$ (by mass)

- $\text{N}_2\text{O}_4$ tank
  - Mass = 13,310 kg
  - Density = 1450 kg/m³
  - Volume = 9.177 m³ --> $r_{\text{sphere}} = 1.299$ m

- Aerozine 50 tank
  - Mass = 7390 kg
  - Density = 900 kg/m³
  - Volume = 8.214 m³ --> $r_{\text{sphere}} = 1.252$ 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: \( \rho_{He} = \frac{p_{g,0} M}{\mathcal{R} T_0} \) \( T=300^\circ\text{K} \) --> \( \rho=49.7 \text{ kg/m}^3 \) \( (300 \text{ psi} = 31.04 \text{ MPa}) \) \( M_{He}=185.1 \text{ kg} \)
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        - Solids
        - Hybrids
        - Liquids
        - Air-Breathing
          - Pressure-Fed
          - Pump-Fed
  - Non-Thermal
    - Non-Chemical
      - Nuclear
      - Electrical
        - Beamed
      - Solar
      - Cold Gas
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

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

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

- At 1 AU, \( P = 1394 \text{ W/m}^2 \)
- \( c = 3 \times 10^8 \text{ m/sec} \)
- \( T = 9 \times 10^{-6} \text{ N/m}^2 \)
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          - Pump-Fed
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      - Nuclear
        - Electrical
        - Solar
      - Non-Thermal
        - Ion
        - MPD
        - Beamed
          - Solar
          - Cold Gas
          - Microwave Sail
          - MagnetoPlasma
          - ED Tether
  - Non-Mass Expulsion
    - Non-Thermal