Chemical/Nuclear Propulsion
Space System Design, MAE 342, Princeton University
Robert Stengel

- Thermal rockets
- Performance parameters
- Propellants and propellant storage

Chemical (Thermal) Rockets

- Liquid/Gas Propellant
  - Monopropellant
  - Cold gas
  - Catalytic decomposition
- Bipropellant
  - Separate oxidizer and fuel
  - Hypergolic (spontaneous) ignition
  - External ignition
  - Storage
    - Ambient temperature and pressure
    - Cryogenic
    - Pressurized tank
  - Throttleable
  - Start/stop cycling

- Solid Propellant
  - Mixed oxidizer and fuel
  - External ignition
  - Burn to completion
- Hybrid Propellant
  - Liquid oxidizer, solid fuel
  - Throttleable
  - Start/stop cycling

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http://www.princeton.edu/~stengel/MAE342.html
Cold Gas Thruster
(*used with inert gas*)

Moog Divert/Attitude Thruster and Valve

Monopropellant Hydrazine Thruster

- Catalytic decomposition produces thrust
- Reliable
- Low performance
- Toxic
Bi-Propellant Rocket Motor

Thrust / Motor Weight ~ 70:1

- Liquid oxygen: 94 K, 4710 kPa, 11.5 kgs
- Liquid hydrogen: 90 K, 4540 kPa, 2.41 kgs

- Combustion chamber: 3215 K, 2500 kPa
- Neck: 3083 K, 2035 kPa

- exit cross section: 930 m, 2.6 kPa
- Hydrogen for cooling: 850 K
- Hydrogen for cooling: 0.15 kPa
- nozzle diverging
- converging
- Inlet size: 4440 in, 2.7 m

Hypergolic, Storable Liquid-Propellant Thruster

- Spontaneous combustion
- Reliable
- Corrosive, toxic
Pressure-Fed and Turbopump Engine Cycles

**Pressure-Fed Rocket Cycle**

- Fuel Tank
- Oxidizer Tank
- Control Valves
- Pressurized gas
- Combustion Chamber
- Heat exchanger
- Nozzle

**Gas-Generator Rocket Cycle, with Nozzle Cooling**

- Fuel Pump
- Turbine
- Oxidizer Pump
- Exhaust
- Heat exchanger
- Nozzle

Staged Combustion Engine Cycles

**Staged Combustion Rocket Cycle**

- Fuel Pump
- Turbine
- Oxidizer Pump
- Fuel-rich Gas
- Control Valves
- Combustion Chamber
- Heat Exchanger
- Nozzle

**Full-Flow Staged Combustion Rocket Cycle**

- Fuel Pump
- Turbine
- Oxidizer Pump
- Fuel-rich Gas
- Oxidizer-rich Gas
- Control Valves
- Combustion Chamber
- Heat Exchanger
- Nozzle
German V-2 Rocket Motor, Fuel Injectors, and Turbopump

Combustion ChamberInjectors
### H-1 ENGINE

<table>
<thead>
<tr>
<th>Vehicle Effectivity</th>
<th>SA-200</th>
<th>155 SEC</th>
<th>200,000 LB</th>
<th>260.5 MIN</th>
<th>261.0 MIN</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (sea level)</td>
<td>155 SEC</td>
<td>155 SEC</td>
<td>205,000 LB</td>
<td>260.5 MIN</td>
<td>261.0 MIN</td>
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<tr>
<td>Thrust duration</td>
<td>155 SEC</td>
<td>155 SEC</td>
<td>205,000 LB</td>
<td>260.5 MIN</td>
<td>261.0 MIN</td>
</tr>
<tr>
<td>Specific Impulse</td>
<td>155 SEC</td>
<td>155 SEC</td>
<td>205,000 LB</td>
<td>260.5 MIN</td>
<td>261.0 MIN</td>
</tr>
<tr>
<td>LB-sec/lb</td>
<td>155 SEC</td>
<td>155 SEC</td>
<td>205,000 LB</td>
<td>260.5 MIN</td>
<td>261.0 MIN</td>
</tr>
<tr>
<td>Engine wt dry</td>
<td>1,820 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
</tr>
<tr>
<td>(INBD)</td>
<td>1,820 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
</tr>
<tr>
<td>(OUTBD)</td>
<td>1,820 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
<td>2,100 LB</td>
</tr>
<tr>
<td>Engine wt burnout</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
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<tr>
<td>(INBD)</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
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<tr>
<td>(OUTBD)</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
<td>2,200 LB</td>
</tr>
<tr>
<td>Exit-to-throat area ratio</td>
<td>8 TO 1</td>
<td>8 TO 1</td>
<td>8 TO 1</td>
<td>8 TO 1</td>
<td>8 TO 1</td>
</tr>
<tr>
<td>Propellants</td>
<td>LOX &amp; RP-1</td>
<td>LOX &amp; RP-1</td>
<td>LOX &amp; RP-1</td>
<td>LOX &amp; RP-1</td>
<td>LOX &amp; RP-1</td>
</tr>
<tr>
<td>Mixture ratio</td>
<td>2.23+2%</td>
<td>2.23+2%</td>
<td>2.23+2%</td>
<td>2.23+2%</td>
<td>2.23+2%</td>
</tr>
<tr>
<td>Contractor: NAA/Rocketdyne</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Vehicle Application: Saturn IB/SA-IB Stage (Eight Engines)</td>
<td></td>
<td></td>
<td></td>
<td></td>
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</table>

### F-1 ENGINE

<table>
<thead>
<tr>
<th>Vehicle Effectivity</th>
<th>SA-501 Thru SA-505</th>
<th>150 SEC</th>
<th>165 SEC</th>
<th>260 SEC MIN</th>
<th>263 MIN</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust (sea level)</td>
<td>150 SEC</td>
<td>165 SEC</td>
<td>263 MIN</td>
<td>260 SEC MIN</td>
<td>263 MIN</td>
</tr>
<tr>
<td>Thrust duration</td>
<td>150 SEC</td>
<td>165 SEC</td>
<td>263 MIN</td>
<td>260 SEC MIN</td>
<td>263 MIN</td>
</tr>
<tr>
<td>Specific Impulse</td>
<td>150 SEC</td>
<td>165 SEC</td>
<td>263 MIN</td>
<td>260 SEC MIN</td>
<td>263 MIN</td>
</tr>
<tr>
<td>LB-sec/lb</td>
<td>150 SEC</td>
<td>165 SEC</td>
<td>263 MIN</td>
<td>260 SEC MIN</td>
<td>263 MIN</td>
</tr>
<tr>
<td>Engine weight</td>
<td>18,416 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
</tr>
<tr>
<td>Dry</td>
<td>18,416 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
<td>18,500 LB</td>
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<tr>
<td>Engine weight</td>
<td>20,096 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
</tr>
<tr>
<td>Burnout</td>
<td>20,096 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
<td>20,180 LB</td>
</tr>
<tr>
<td>Exit-to-throat area ratio</td>
<td>16 TO 1</td>
<td>16 TO 1</td>
<td>16 TO 1</td>
<td>16 TO 1</td>
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</tr>
<tr>
<td>Propellants</td>
<td>LOX &amp; RP-1</td>
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<td>2.27+2%</td>
</tr>
<tr>
<td>Contractor: NAA/Rocketdyne</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Vehicle Application: Saturn V/S-IC Stage (Five Engines)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Origins of the F-1

• Air Force legacy (1955)
  • Design undertaken before vehicle or mission were identified
• Big engine, big problems
  • 16:1 nozzle expansion
  • 6.67 MN thrust
• F-1 turbopumps
  • Oxygen: 24,811 gal/min
  • RP-1: 15,741 gal/min
• F-1 injector
• Combustion instability
  • Significant theoretical work by Luigi Crocco and David Harrje, Princeton
USSR RD-107/8 Rocket Motors

**RD-107**
4 combustion chambers, 2 verniers

**RD-108**
4 combustion chambers, 4 verniers

**R-7 Base**
4-RD-107, 1-RD-108

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**RD-180 Engine Schematic**
(used on Atlas V)
Special Shuttle Main Engine (RS-25)

SpaceX Merlin Family

Merlin 1A (ablative nozzle)  Merlin 1C (vacuum nozzle)  Merlin 1D (throtttable)

Roll control from turbine exhaust
Blue Origin BE-4

- LOX/Liquefied natural gas
- United Launch Alliance has chosen as motor for the Vulcan launch vehicle
- Thrust = 2.5 MN (550,000 lb)

RD-181 and RD-191

- RD-181 to be used on Orbital-ATK Antares
- RD-191 to be used on NPO Energomash Angara
Solid-Fuel Rocket Motor

- Thrust is proportional to burning area
- Rocket grain patterns affect thrust profile
- Propellant chamber must sustain high pressure and temperature
- Environmentally unfriendly exhaust gas

Hybrid-Fuel Rocket Motor

- SpaceShipOne motor
  - Nitrous oxide
  - Hydroxy-terminated polybutadiene (HTPB)
- Issues
  - Hard start
  - Blow back
  - Complete mixing of oxidizer and fuel toward completion of burn
Rocket Thrust

\[
\text{Thrust} = \dot{m}_{\text{propellant}} V_{\text{exhaust}} + A_{\text{exit}} (p_{\text{exit}} - p_{\text{ambient}}) \equiv \dot{m} c_{\text{eff}}
\]

\[
c_{\text{eff}} = \frac{\text{Thrust}}{\dot{m}} = \text{Effective exhaust velocity}
\]

\[
\dot{m} \equiv \text{Mass flow rate of on-board propellant}
\]

Specific Impulse

\[
I_{sp} = \frac{\text{Thrust}}{\dot{m} g_o} = \frac{c_{\text{eff}}}{g_o}, \quad \text{Units} = \frac{m}{s} = \text{seconds}
\]

\[
g_o \equiv \text{Gravitational acceleration at earth's surface}
\]

- \(g_o\) is a normalizing factor for the definition
- Chemical rocket specific impulse (vacuum)
  - Solid propellants: < 295 s
  - Liquid propellants: < 510 s

- Space Shuttle Specific Impulses
  - Solid boosters: 242-269 s
  - Main engines: 455 s
  - OMS: 313 s
  - RCS: 260-280 s
Specific Impulse

Specific impulse is a product of characteristic velocity, $c^*$, and rocket thrust coefficient, $C_F$.

\[
I_{sp} = \frac{\text{Thrust}}{m g_o} = \frac{c_{\text{eff}}}{g_o} = C_F \frac{c^*}{g_o} = \frac{V_{\text{exhaust}}}{g_o} \quad \text{when} \quad C_F = 1, \quad p_e = p_{\text{ambient}}
\]

- Characteristic velocity is related to
  - combustion chamber performance
  - propellant characteristics

- Thrust coefficient is related to
  - nozzle shape
  - exit/ambient pressure differential

The Rocket Equation

Ideal velocity increment of a rocket stage, \(\Delta V_I\) (gravity and aerodynamic effects neglected)

\[
\frac{dV}{dt} = \frac{\text{Thrust}}{m} = \frac{\dot{m} c_{\text{eff}}}{m} = -\frac{d m}{dt} \frac{I_{sp} g_o}{m}
\]

\[
\int_{V_i}^{V_f} dV = -I_{sp} g_o \int_{m_i}^{m_f} \frac{d m}{m} = -I_{sp} g_o \ln \frac{m_f}{m_i}
\]

\[
(V_f - V_i) \equiv \Delta V_I = I_{sp} g_o \ln \left( \frac{m_i}{m_f} \right) \equiv I_{sp} g_o \ln \mu
\]
Volumetric Specific Impulse

Specific impulse

\[
\Delta V_i = I_{sp\,g_0} \ln \mu = I_{sp\,g_0} \ln \left( \frac{m_{\text{final}} + m_{\text{propellant}}}{m_{\text{final}}} \right) = I_{sp\,g_0} \ln \left( 1 + \frac{m_{\text{propellant}}}{m_{\text{final}}} \right)
\]

\[
= I_{sp\,g_0} \ln \left( 1 + \frac{\text{Density}_{\text{propellant}} \cdot \text{Volume}_{\text{propellant}}}{m_{\text{final}}} \right)
\]

\[
\approx g_o I_{sp} \left( \frac{\rho_{\text{propellant}} \cdot Vol_{\text{propellant}}}{m_{\text{final}}} \right) = g_o \left( I_{sp\rho_{\text{propellant}}} \right) \frac{Vol_{\text{propellant}}}{m_{\text{final}}}
\]

Volumetric specific impulse

\[I_{sp\,vol} \triangleq VI_{sp} = I_{sp\rho_{\text{propellant}}}
\]

Volumetric Specific Impulse

- For fixed volume and final mass, increasing volumetric specific impulse increases ideal velocity increment

<table>
<thead>
<tr>
<th>Density, g/cc</th>
<th>Isp, s, SL</th>
<th>Vlsp, s (g/cc), Isp, s, SL, vac</th>
<th>Vlsp, s (g/cc), vac</th>
</tr>
</thead>
<tbody>
<tr>
<td>LOX/Kerosene</td>
<td>1.3</td>
<td>265</td>
<td>345</td>
</tr>
<tr>
<td>LOX/LH2 (Saturn V)</td>
<td>0.28</td>
<td>360</td>
<td>101</td>
</tr>
<tr>
<td>LOX/LH2 (Shuttle)</td>
<td>0.28</td>
<td>390</td>
<td>109</td>
</tr>
<tr>
<td>Shuttle Solid Booster</td>
<td>1.35</td>
<td>242</td>
<td>327</td>
</tr>
</tbody>
</table>

- Saturn V Specific Impulses, vacuum (sea level)
  - 1st Stage, 5 F-1 LOX-Kerosene Engines: 304 s (265 s)
  - 2nd Stage, 5 J-2 LOX-LH2 Engines: 424 s (~360 s)
  - 3rd Stage, 1 J-2 LOX-LH2 Engine: 424 s (~360 s)
Typical Values of Chemical Rocket Specific Impulse

- Chamber pressure = 7 MPa (low by modern standards)
- Expansion to exit pressure = 0.1 MPa

<table>
<thead>
<tr>
<th>Solid-Propellant Rockets</th>
<th>Visp, kg/s(m^3) x 10^3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Double-Base</td>
<td>Isp, s</td>
</tr>
<tr>
<td>AFU</td>
<td>196</td>
</tr>
<tr>
<td>ATN</td>
<td>235</td>
</tr>
<tr>
<td>JPN</td>
<td>250</td>
</tr>
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</table>

<table>
<thead>
<tr>
<th>Composite</th>
<th>Isp, s</th>
</tr>
</thead>
<tbody>
<tr>
<td>JPL 540A</td>
<td>231</td>
</tr>
<tr>
<td>TRX-H609</td>
<td>245</td>
</tr>
<tr>
<td>PBAN (SSV)</td>
<td>260</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Hybrid-Fuel Rocket</th>
<th>Isp, s</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuel</td>
<td>Oxidizer</td>
</tr>
<tr>
<td>HTPB</td>
<td>N2O</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Liquid-Fuel Rockets</th>
<th>Isp, s</th>
</tr>
</thead>
<tbody>
<tr>
<td>Monopropellant</td>
<td></td>
</tr>
<tr>
<td>Hydrogen Peroxide</td>
<td>165</td>
</tr>
<tr>
<td>Hydrazine</td>
<td>199</td>
</tr>
<tr>
<td>Nitromethane</td>
<td>255</td>
</tr>
<tr>
<td>Bopropellant</td>
<td></td>
</tr>
<tr>
<td>Fuel</td>
<td>Isp, s</td>
</tr>
<tr>
<td>Kerosene</td>
<td>Oxygen</td>
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<td></td>
<td>Flourine</td>
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<tr>
<td></td>
<td>Red Fuming</td>
</tr>
<tr>
<td>Hydrogen</td>
<td>Nitric Acid</td>
</tr>
<tr>
<td></td>
<td>Oxygen</td>
</tr>
<tr>
<td></td>
<td>Flourine</td>
</tr>
<tr>
<td></td>
<td>Nitrogen</td>
</tr>
<tr>
<td>UDMH</td>
<td>Tetroxide</td>
</tr>
</tbody>
</table>

Exhaust Velocity vs. Thrust Acceleration
Rocket Characteristic Velocity, $c^*$

$$c^* = \frac{1}{\Gamma} \sqrt{\frac{R_o T_c}{M}}$$, where
$$\Gamma = \sqrt{\gamma \left( \frac{2}{\gamma + 1} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}}}$$

$R_o$ = universal gas constant = $8.3 \times 10^3$ kg m$^2$/s$^2$ °K
$T_c$ = chamber temperature, °K
$M$ = exhaust gas mean molecular weight
$\gamma$ = ratio of specific heats (~1.2-1.4)

Rocket Characteristic Velocity, $c^*$

$$c^* = \frac{P_c A_t}{m} = \text{exhaust velocity if } C_F = 1$$
Rocket Thrust Coefficient, $C_F$

$$C_F = \frac{\text{Thrust}}{p_c A_t} = \lambda \gamma \sqrt{\left( \frac{2\gamma}{\gamma - 1} \right) \left[ 1 - \left( \frac{p_e}{p_c} \right)^{(\gamma - 1)/\gamma} \right] + \left( \frac{p_e - p_{ambient}}{p_c} \right) A_c}$$

**Thrust** = $\lambda \dot{m} v_e + A_c (p_e - p_{ambient})$

$\lambda$ : reduction ratio (function of nozzle shape)

$C_F$ typically 0.5 - 2

---

Thrust Coefficient, $C_F$, vs. Nozzle Expansion Ratio

![Graph showing $C_F$ vs. Expansion Ratio](image)
Mixture Ratio, $r$

$$r = \frac{\dot{m}_{\text{oxidizer}}}{\dot{m}_{\text{fuel}}}$$
$$\dot{m}_{\text{fuel}} = \frac{\dot{m}_{\text{total}}}{1 + r}; \quad \text{"leaner"} < r < \text{"richer"}$$

- **Stoichiometric mixture:**
  complete chemical reaction of propellants
- **Specific impulse maximized**
  with lean mixture ratio, $r$ (i.e.,
  below stoichiometric maximum)

**Effect of Pressure Ratio on Mass Flow**

In choked flow, mass flow rate is maximized

$$\dot{m} = \frac{\Gamma p_c A_t}{\sqrt{R_o T_c}}$$

Choked flow occurs when

$$\frac{p_e}{p_c} \leq \left(\frac{2}{\gamma + 1}\right)^{\gamma/\gamma-1} \approx 0.53$$
Combustion Instability

- Complex mix of species, phases, pressures, temperatures, and flows
- Cavity resonance

Harrje, NASA SP-194, 1972
Shock Diamonds

When $p_e \neq p_m$, exhaust flow is over- or underexpanded.
Effective exhaust velocity < maximum value.

Rocket Nozzles

1. Conventional conical nozzle
2. Conventional bell-type nozzle
3. Spike-type nozzle
4. Plug-type nozzle
5. Expansion-deflection type nozzle
Rocket Nozzles

• Expansion ratio, $A_e/A_t$, chosen to match exhaust pressure to average ambient pressure
  – Ariane rockets: Viking V for sea level, Viking IV for high altitude

• Rocket nozzle types
  – DeLaval nozzle
  – Isentropic expansion nozzle
  – Spike/plug nozzles
  – Expansion-deflection nozzle
Linear Spike/Plug Nozzles

Throttling, Start/Stop Cycling
Reaction Control Thrusters

- Direct control of angular rate
- Unloading momentum wheels or control-moment gyros
- Reaction control thrusters are typically on-off devices using
  - Cold gas
  - Hypergolic propellants
  - Catalytic propellant
  - Ion/plasma rockets
- Thrusters commanded in pairs to cancel velocity change

Apollo Lunar Module RCS  Space Shuttle RCS  RCS Thruster

Divert and Attitude Control Thrusters

https://www.youtube.com/watch?v=W8efpDBvTDE
https://www.youtube.com/watch?v=71qgl6bdddM8
https://www.youtube.com/watch?v=KBMU6l6GsdM
https://www.youtube.com/watch?v=JURQYH669_g
Nuclear Propulsion

- Nuclear reaction produces thermal energy to heat inert working fluid
  - Solid core
  - Liquid core
  - Gaseous core
- High propellant temperature leads to high specific impulse
- Working fluid chosen for low molecular weight and storability

\[ c^* = \frac{1}{\Gamma} \sqrt{\frac{R_T c}{M}} \]

Solid-Core Nuclear Rocket

- Operating temperature limited by
  - melting point of reactor materials
  - cracking of core coating
  - matching coefficients of expansion
- Possible propellants: hydrogen, helium, liquid oxygen, water, ammonia
- \( I_{sp} = 850 - 1,000 \) sec
- \( T / W \sim 7:1 \)
Project Rover, 1955-1972

NERVA Rocket, $I_{sp} \approx 900$ sec

NERVA-Powered Mars Mission

Kiwi-B4-A Reactor/Rocket
Liquid/Particle-Core Nuclear Rocket

- Nuclear fuel mixed with working fluid
- In principle, could operate above melting point of nuclear fuel
- $I_{sp} \approx 1,300 – 1,500$ sec
- Conceptual
- Massive radioactive waste

Open-Cycle Gas Core Nuclear Rocket

- Toroidal circulation of working fluid confines nuclear fuel to center
- Fuel does not touch the wall
- Conceptual
- Massive radioactive waste
- $I_{sp} \approx 3,000 – 5,000$ sec
Closed-Cycle Gas Core Nuclear Rocket

- “Nuclear light bulb”
- Nuclear fuel contained in quartz container
- $I_{sp} \approx 1,500 – 2,000$ sec
- Conceptual

Nuclear-Pulse (“Explosion”) Rocket - Project Orion

“Physics packages” ejected behind the pusher plate

[https://en.wikipedia.org/wiki/Project_Orion_(nuclear_propulsion)]
Next Time:
Launch Vehicles

SUPPLEMENTAL MATERIAL
Propellant Tanks

Propellant must be kept near the exit duct without bubbles during thrusting

Ion/Plasma Thrusters

<table>
<thead>
<tr>
<th>Engine</th>
<th>Propellant</th>
<th>Required power kW</th>
<th>Specific impulse s</th>
<th>Thrust mN</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSTAR</td>
<td>Xenon</td>
<td>2.3</td>
<td>3,300 to 1,700</td>
<td>92 max</td>
</tr>
<tr>
<td>NEXT</td>
<td>Xenon</td>
<td>6.9</td>
<td>4,300</td>
<td>236 max</td>
</tr>
<tr>
<td>HiPEP</td>
<td>Xenon</td>
<td>20–50</td>
<td>6,000–9,000</td>
<td>460–670</td>
</tr>
<tr>
<td>Hall effect</td>
<td>Xenon</td>
<td>25</td>
<td>3,290</td>
<td>950</td>
</tr>
<tr>
<td>FEEP</td>
<td>Liquid Cesium</td>
<td>$6 \times 10^{-5}$–0.06</td>
<td>6,000–10,000</td>
<td>0.001–1</td>
</tr>
<tr>
<td>VASIMR</td>
<td>Argon</td>
<td>200</td>
<td>3,000–12,000</td>
<td>~5,000</td>
</tr>
<tr>
<td>DS4G</td>
<td>Xenon</td>
<td>250</td>
<td>19,300</td>
<td>2,500 max</td>
</tr>
</tbody>
</table>
Variable Specific Impulse Magnetoplasma Rocket (VASIMR)

<table>
<thead>
<tr>
<th>Propellant</th>
<th>Required power kW</th>
<th>Specific impulse s</th>
<th>Thrust mN</th>
</tr>
</thead>
<tbody>
<tr>
<td>Argon</td>
<td>200</td>
<td>3,000–12,000</td>
<td>~5,000</td>
</tr>
</tbody>
</table>

DAWN Spacecraft

<table>
<thead>
<tr>
<th>Engine</th>
<th>Propellant</th>
<th>Required power kW</th>
<th>Specific impulse s</th>
<th>Thrust mN</th>
</tr>
</thead>
<tbody>
<tr>
<td>NSTAR</td>
<td>Xenon</td>
<td>2.3</td>
<td>3,300 to 1,700</td>
<td>92 max</td>
</tr>
</tbody>
</table>