Liquid Propulsion Systems – Evolution & Advancements

Launch Vehicle Propulsion & Systems

AIAA
American Institute of
Aeronautics and Astronautics

LPTC
*Liquid Propulsion
Technical Committee*

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Modules

1. Rocket Propulsion Fundamentals
2. LRE Applications
3. Liquid Propellants
4. Engine Power Cycles
5. Engine Components

Module 1:

Rocket Propulsion Fundamentals

TOPICS
- Thrust
- Specific Impulse
- Mixture Ratio
- I_{sp} vs. MR
- Density vs. I_{sp}
- Propellant Mass vs. Volume
- Area Ratio

Warning**: *Contents deal with math, physics and thermodynamics.***

Be afraid…be very afraid…

Terms

A	Area
a	Acceleration
F	Force (thrust)
g	Gravity constant (32.2 ft/sec2)
I	Impulse
m	Mass
P	Pressure
t	Time
T	Temperature
V	Velocity
Δ	Delta / Difference
ε	Area Ratio
γ	Ratio of specific heats

Subscripts

a	Ambient
c	Chamber
e	Exit
o	Initial state
r	Reaction
s	Stagnation
sp	Specific
t	Throat or Total

Rocket thrust can be explained using Newton's 2nd and 3rd laws of motion.

2nd Law: a force applied to a body is equal to the mass of the body and its acceleration in the direction of the force.

$$F = ma$$

3rd Law: For every action, there is an equal and opposite reaction.

$$F_a = -F_r$$

In rocket propulsion, a mass of propellant (m) is accelerated (via the combustion process) from initial velocity (V_o) to an exit velocity (V_e). The acceleration of this mass is written as:
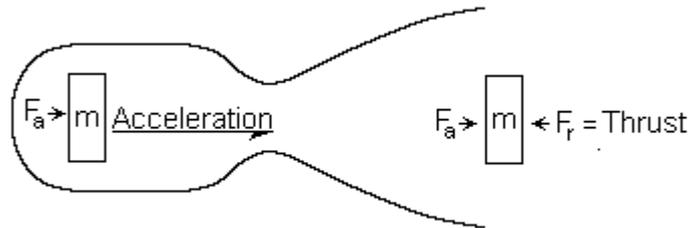
$$a = (V_e - V_o) / \Delta t$$

Combining terms, we get:

$$F = m(V_e - V_o)/t$$

Which can be rearranged:

$$F = \left(\frac{m}{t}\right)(V_e - V_o)$$

The term of mass of propellant divided by time is referred to as mass flow rate (m-dot), expressed as

$$F = \dot{m}(V_e - V_o)$$

Substituting weight flow rate for mass flow rate and assuming a zero initial velocity, the thrust produced by propellant flow (*momentum thrust*, F_1) is shown as

$$F_1 = \frac{\dot{m}}{g}(V_e)$$

Another component of thrust (*pressure thrust*, F_2) comes from the force exerted by external pressure differences on the system. This is described by the difference of the pressure of the flow leaving the engine (P_e) through the exit area (A_e) compared to the external (ambient) pressure (P_a).

$$F_2 = (P_e - P_a)A_e$$

In space, P_a is assumed to be zero (which explains why thrust rated at vacuum is higher than at sea level).

Combining the two thrust components gives

$$F = \frac{\dot{m}}{g}(V_e) + (P_e - P_a)A_e$$
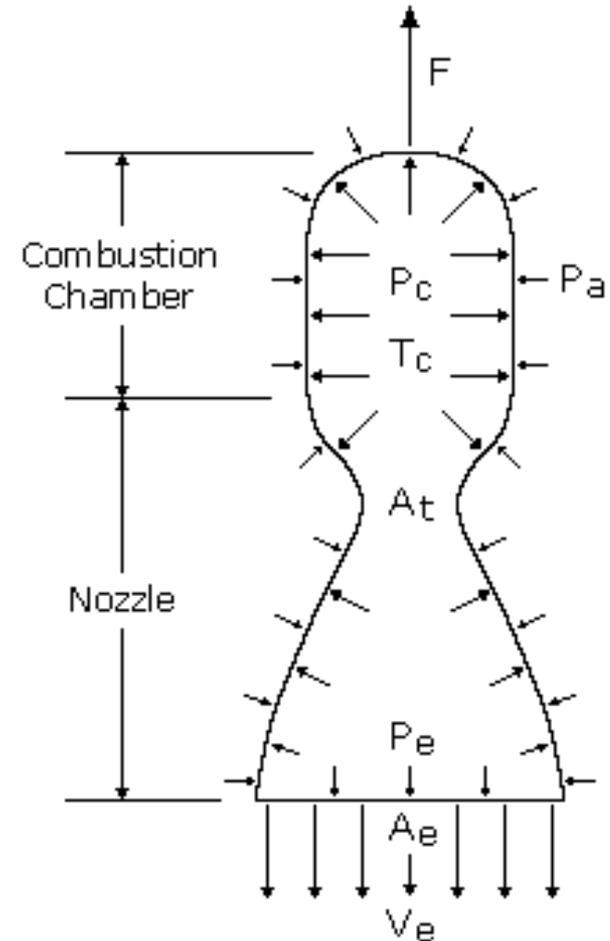
The *total impulse* (I_t) is the thrust integrated over the run duration (time, t)
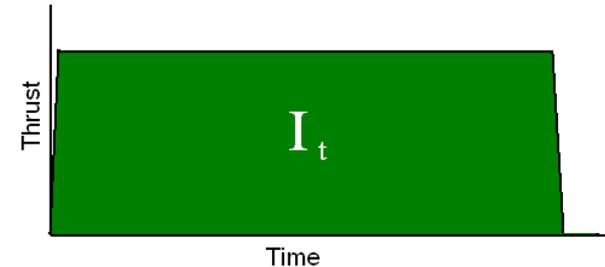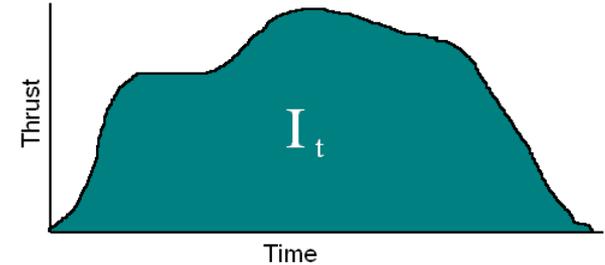
$$I_t = \int_0^t F \, dt$$

Assuming constant thrust and negligible transients (i.e., start and shutdown), this becomes

$$I_t = Ft$$

The *specific impulse*, (I_{sp}) is the total impulse generated per weight of propellant

$$I_{sp} = \frac{\int_0^t F \, dt}{g_o \int_0^t \dot{m} \, dt} = \frac{F}{\dot{m}}$$

This is really what you need to know about I_{sp}. If you want to learn some more, stay seated. The rest of you can go get some coffee for about 10 minutes…

Substituting the thrust equation for F gives

$$I_{sp} = \left(\frac{V_e}{g}\right) + (P_e - P_a)\left(\frac{A_e}{\dot{w}}\right)$$

Under optimal conditions the pressure of the flow exiting the nozzle is equal to that of the ambient pressure (i.e., optimally expanded). The Isp equation then becomes

$$I_{sp} = \frac{V_e}{g}$$

C_{pm}=molar specific heat of exhaust gas

M = molecular weight of exhaust gas

P_s= isentropic stagnation pressure of the flow at the throat (almost always equivalent to chamber pressure, P_c)

γ= ratio of specific heats of exhaust gas

Where the flow exit velocity is calculated by

$$V_e = \sqrt{\left(2gC_{pm}T_c / M\right)\left\{1 - \left(\frac{P_e}{P_s}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right\}}$$

Don't be intimidated by this – just understand that V_e (and hence I_{sp}) is enhanced by high release of energy from the combustion process (high C_{pm}) and low molecular weight (low M) of the exhaust

The equation for I_{sp} then becomes

$$I_{sp} = \sqrt{\left(\frac{2C_{pm}T_c}{gM}\right)\left\{1-\left(\frac{P_e}{P_s}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right\}}$$

From this, optimum I_{sp} becomes

$$I_{sp(opt)} = \sqrt{\left(\frac{2\gamma}{(\gamma-1)}\right)\left(\frac{RT_c}{gM}\right)\left\{1-\left(\frac{P_e}{P_s}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right\}}$$

By definition

$$\gamma = \frac{C_p}{C_v} = \frac{C_{pm}}{C_{vm}}$$

and

$$C_{pm} = C_{vm} + R$$

Where R is the universal gas constant, then

$$C_{pm} = \gamma/(\gamma-1)R$$

Primarily a function of the combustion process, which is influenced by the propellants, and the mixture ratio

Primarily a function of the expansion of the exhaust gases through the nozzle, which is influenced by back pressure and area ratio

From an empirical standpoint, determining I_{sp} from test results involves calculating the total impulse generated by the engine and dividing it by the weight of the propellant consumed.

$$I_{sp} = \frac{\int\limits_{0}^{t} F\,dt}{w_{consumed}}$$

The I_{sp} for flight engines is established (i.e., tagged) on the test stand under simulated flight conditions to ensure acceptance of engine operation and adequate propellant loading of the vehicle.
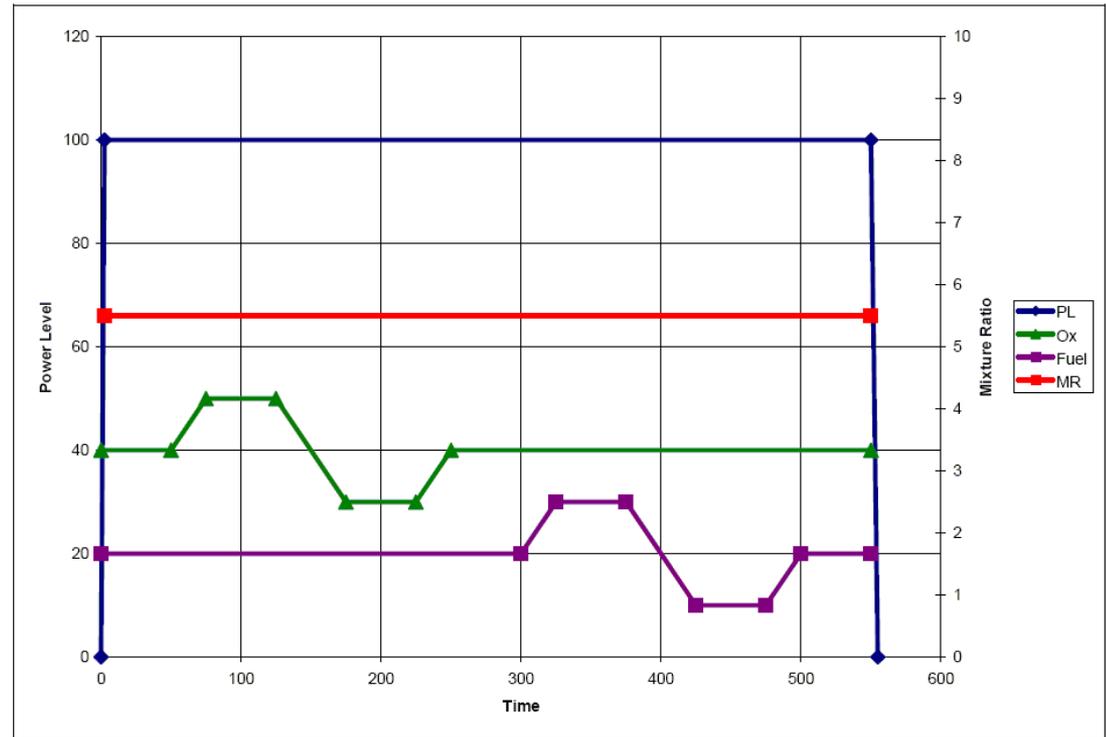
J-2X Acceptance Test Profile

Other primary performance parameters (i.e., Thrust, MR) are also tagged during acceptance testing.

Rocket propellants are mixed in relative quantities to produce the highest possible system I_{sp}. This ratio of propellant consumption is called mixture ratio, MR.

$$MR = \frac{\dot{w}_o}{\dot{w}_f}$$

In most cases, MR is selected for maximum energy release per weight of propellant. This can be achieved by mixing the propellants in a stoichiometric reaction in the combustion chamber, where all the propellants are thoroughly combusted. <u>However, a stoichiometric MR does not necessarily provide an optimized I_{sp}</u>.

☞ The SSME uses a MR of ~6 (stoichiometric for LO_2/ LH_2 combustion is 8) to reduce the internal and plume temperatures, but also to allow a small amount of H_2 to remain in the exhaust. The lighter molecule is able to accelerate to a higher velocity and generate higher kinetic energy ($KE = \frac{1}{2} mV^2$) than a H_2O steam exhaust.

I_{sp} vs. MR
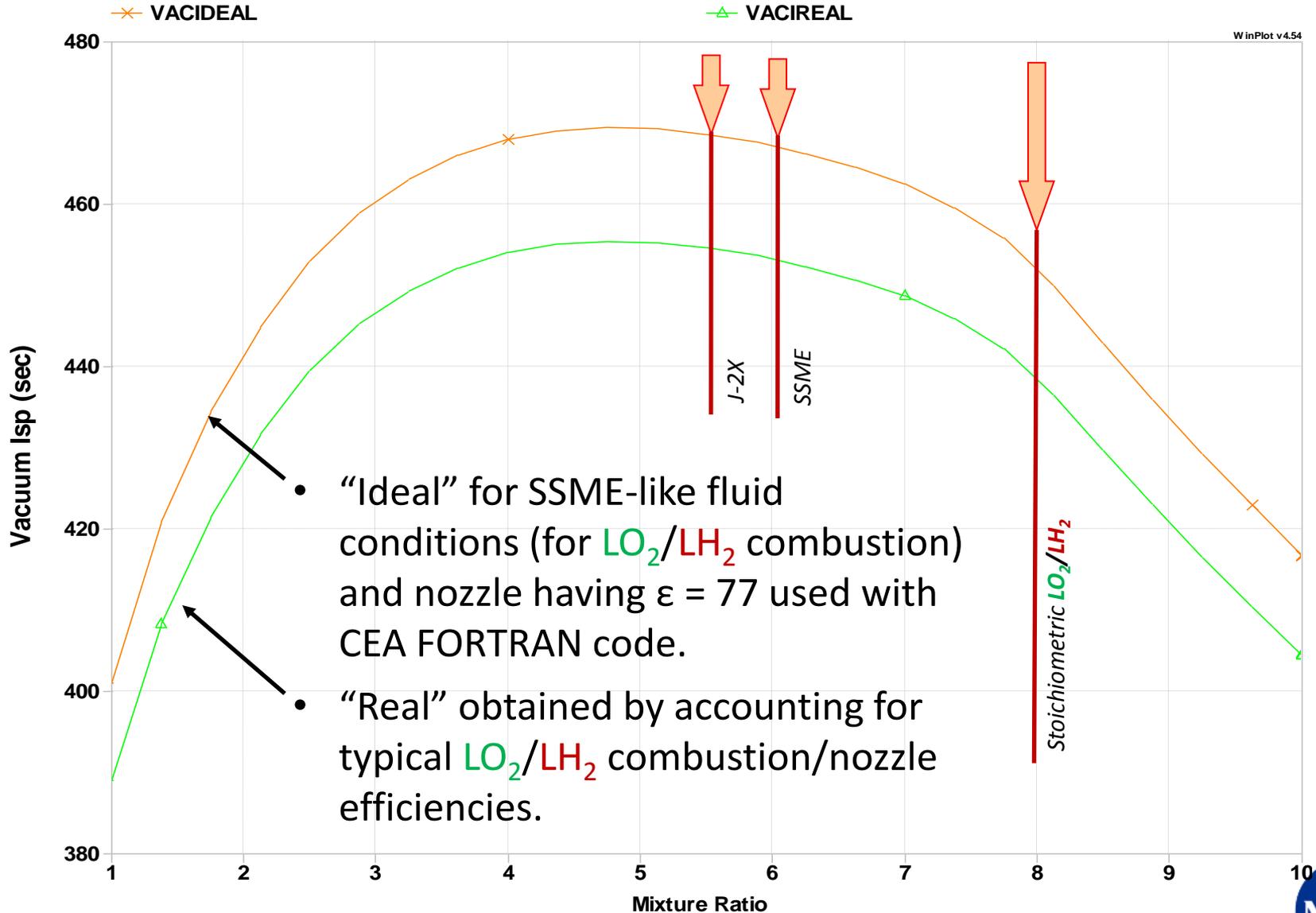
- Liquid bipropellant combinations offer a wide range of performance capabilities.

- Each combination has multiple factors that should be weighed when selecting one for a vehicle.
  - Performance (I_{sp})
  - Density (higher is better)
  - Storability (venting?)
  - Ground Ops (hazards?)
  - Etc.

- One of the more critical trades is that of performance versus density.

- LO_2/LH_2 offers the highest I_{sp} performance, but at the cost of poor density (thus increasing tank size).

- Trading I_{sp} versus density is sometimes referred to as comparing "bulk impulse" or "density impulse".

- As an example, the densities and I_{sp} performance of the following propellant combinations will be compared.

	Density (g/ml)	Density (lb/ft^3)
Hydrogen	0.07	4.4
Methane	0.42	26.4
RP-1	0.81	50.6
Oxygen	1.14	71.2

P_c = 300 psia expanded to 14.7 psia	MR (O/F)	I_{sp} (sec)
LO_2/LH_2	3.5	347[1]
LO_2/CH_4	2.33	263[2]
LO_2/RP-1	2.4	263[2]

(1) SC   (2) FC
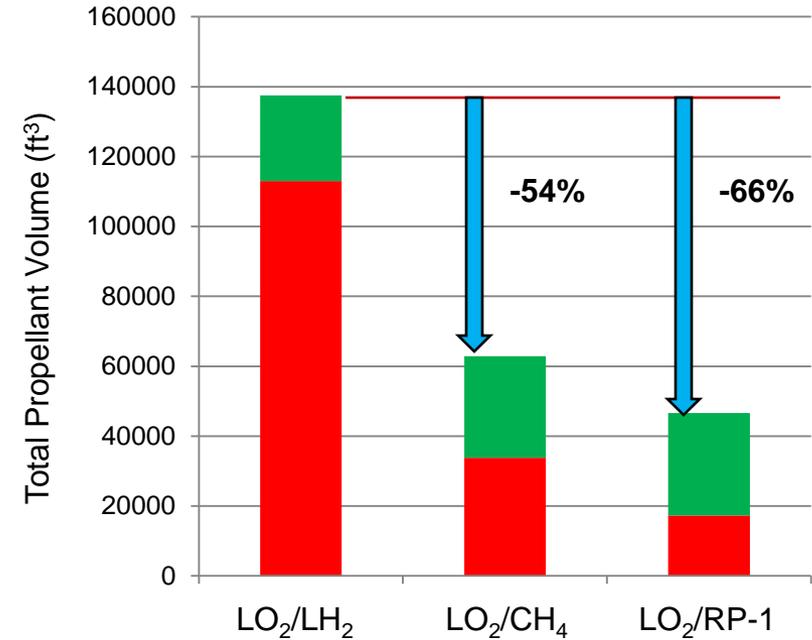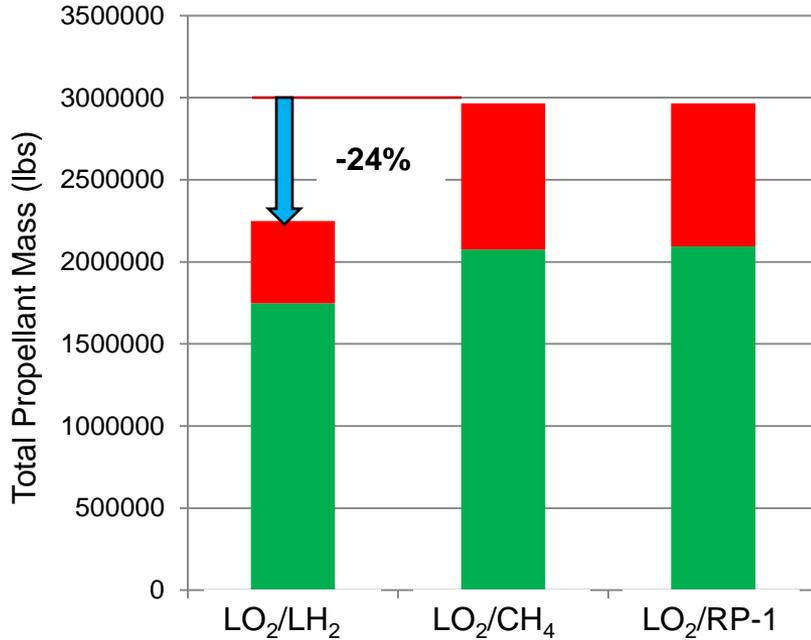
- For an impulse requirement similar to the 3 SSME's used on the Shuttle (~1.5 Mlbf for 520 seconds), the required propellant masses are calculated.

- LO$_2$/LH$_2$ requires 24% less propellant mass than the others.

- However…

- When the propellant mass is compared against the tank volume, there is a significant disparity from the low hydrogen density that can adversely impact the size (and total weight) of the vehicle.

- Lesson: I_{sp} isn't everything – especially with boost stages.

The parameter that determines exit velocity and pressure of the exhaust gases is *area ratio* or *nozzle expansion ratio*, MR

$$\varepsilon = \frac{A_e}{A_t}$$

- As ε increases, the exit velocity increases and the exit pressure decreases (higher I_{sp}).

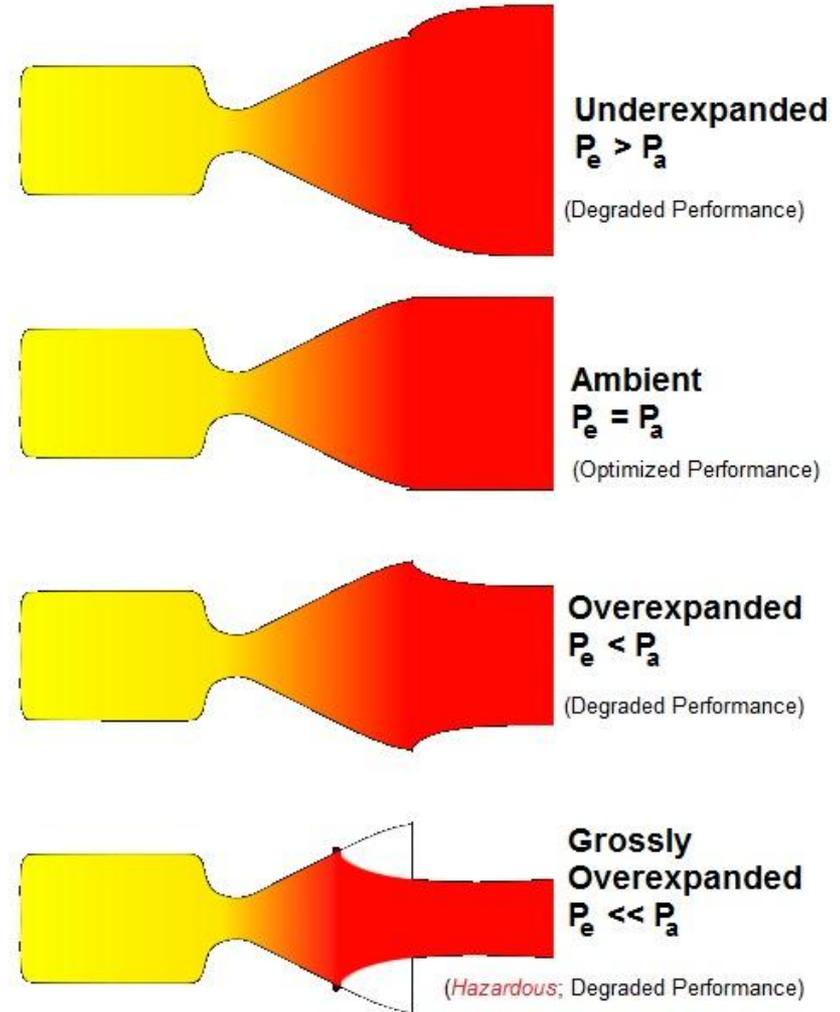- When possible, ε is selected so that $P_e = P_a$ and the engine operates at optimum thrust.

- For in-space propulsion (i.e., J-2X), the ε is made as large as the weight requirements or volume limitations permit.

- If $P_e > P_a$, then the nozzle is identified as underexpanded and will not provide optimal performance as the plume will continue to expand after exiting the nozzle.

- If $P_e < P_a$, then the nozzle is identified as overexpanded and will not provide optimal performance as the exit shock will migrate inside the nozzle. This can be hazardous from thrust imbalances and damage to the nozzle.
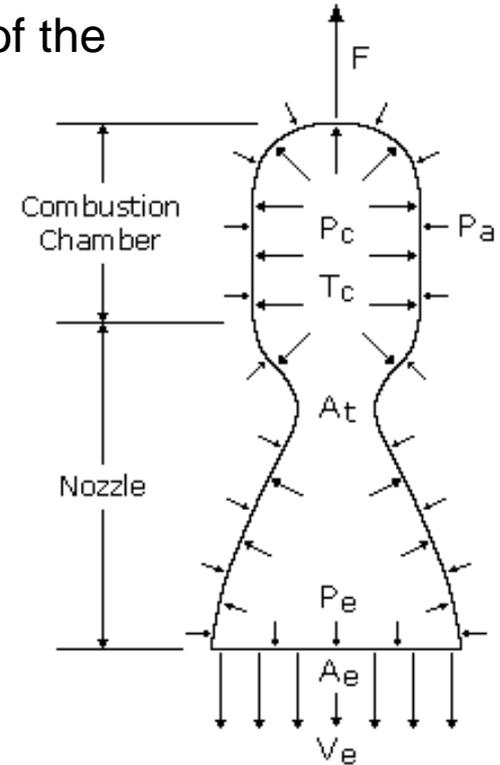
- Criteria to be considered for selecting a design ε can include the following:
  - ε provides the optimal integrated performance over the engine operating period. Trajectory analysis is used to determine the altitudes (and P_a) that the engine will operate, which can be used to provide an integrated I_{sp} based on ε.
  - ε is optimized to provide the maximum performance during a critical time of the engine operating period. Example: the ε for SSME is optimized at the altitude where the SRBs are staged to provide a needed performance boost at that critical time

Underexpanded
$P_e > P_a$
(Degraded Performance)

Ambient
$P_e = P_a$
(Optimized Performance)

Overexpanded
$P_e < P_a$
(Degraded Performance)

Grossly Overexpanded
$P_e \ll P_a$
(*Hazardous*; Degraded Performance)

Horsepower to Weight Comparison

High Power Density Comparison of Automobile Engines, Jet Engines, and Rocket Engines

HP/Wt

Shuttle Engine
7,000 lb
6,232,500 HP
HP/Wt = 907.4

Large Jet Engine
13,065 lb
1,950,300 HP
HP/Wt = 150

Small Jet Engine
2,890 lb
52,900 HP
HP/Wt = 18.3

Auto Engine
370 lb
200 HP
HP/Wt = 0.54

Indy Engine
275 lb
800 HP
HP/Wt = 2.91

1000
800
600
400
200
0
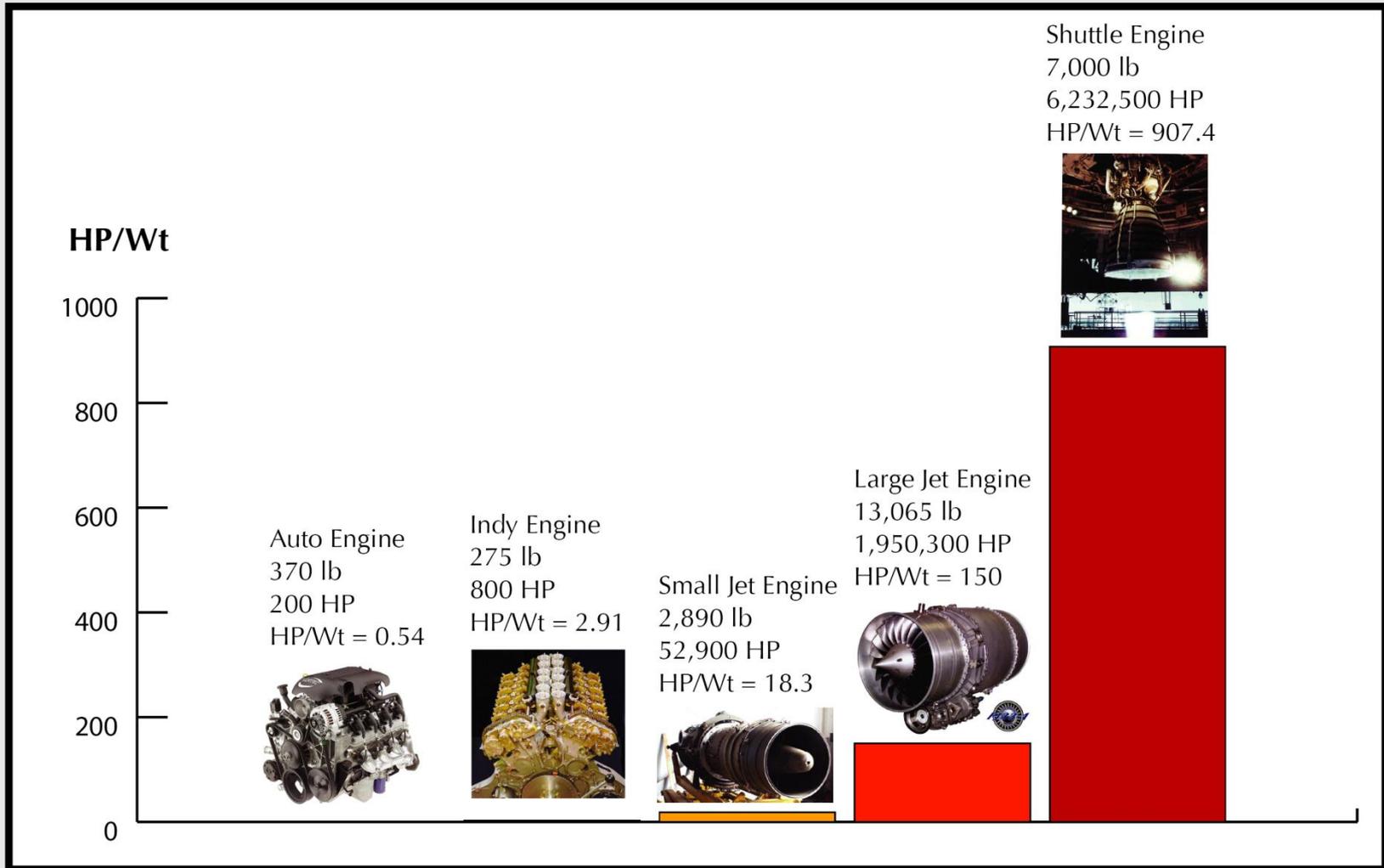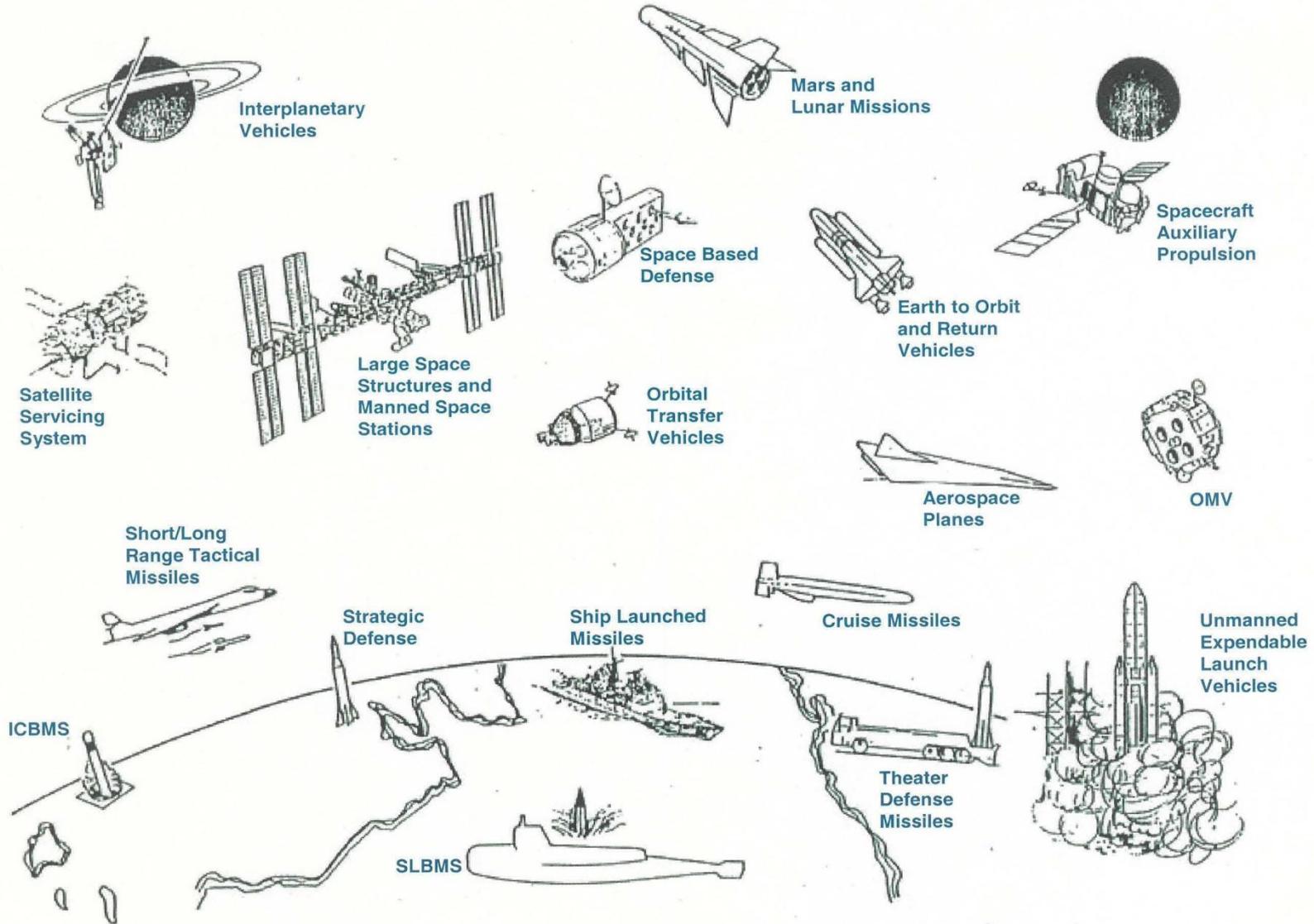
Module 2:

Rocket Propulsion Applications

Rocket Propulsion Applications

In-Space Propulsion Performance
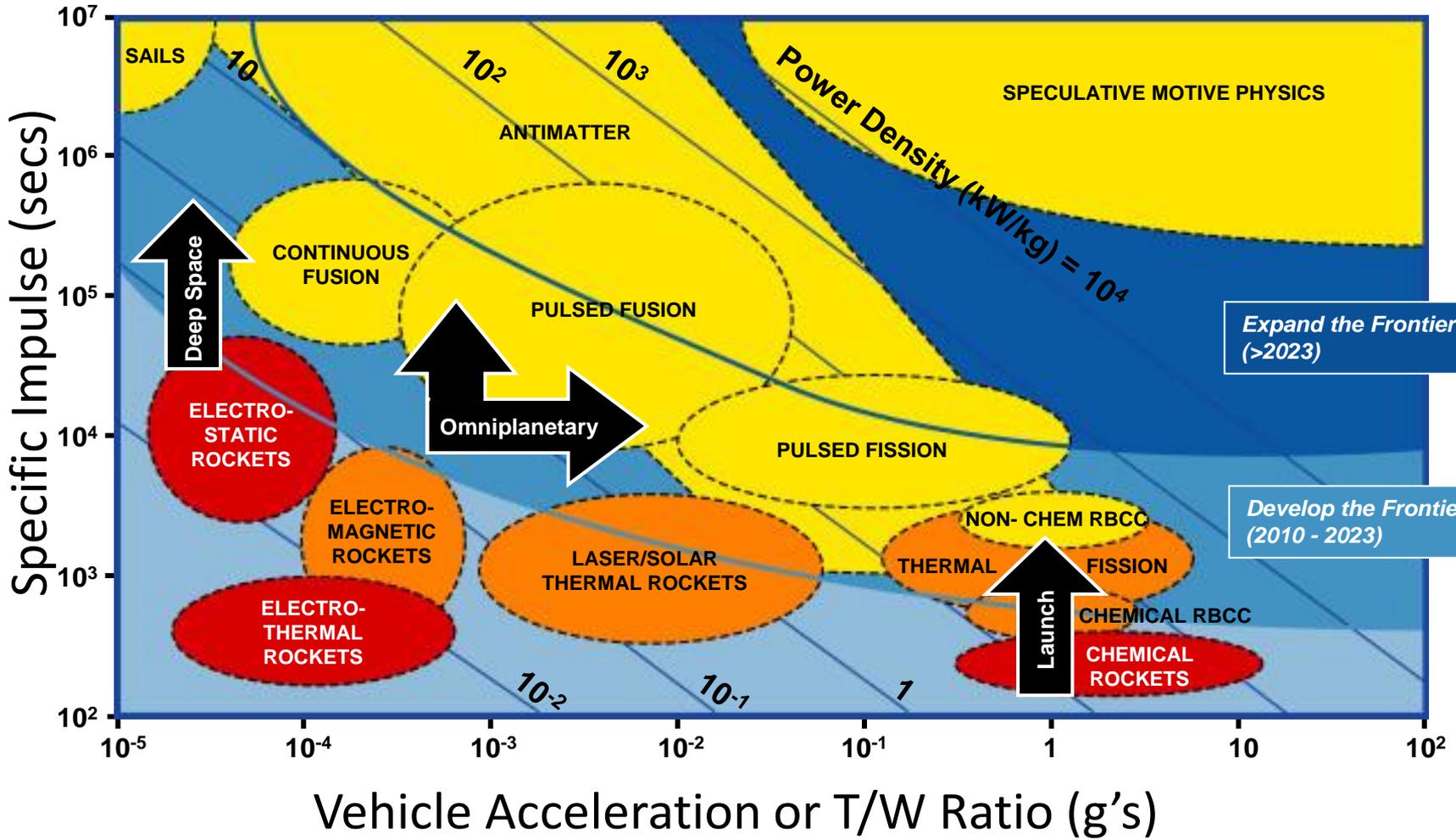
Jet Propulsion Options
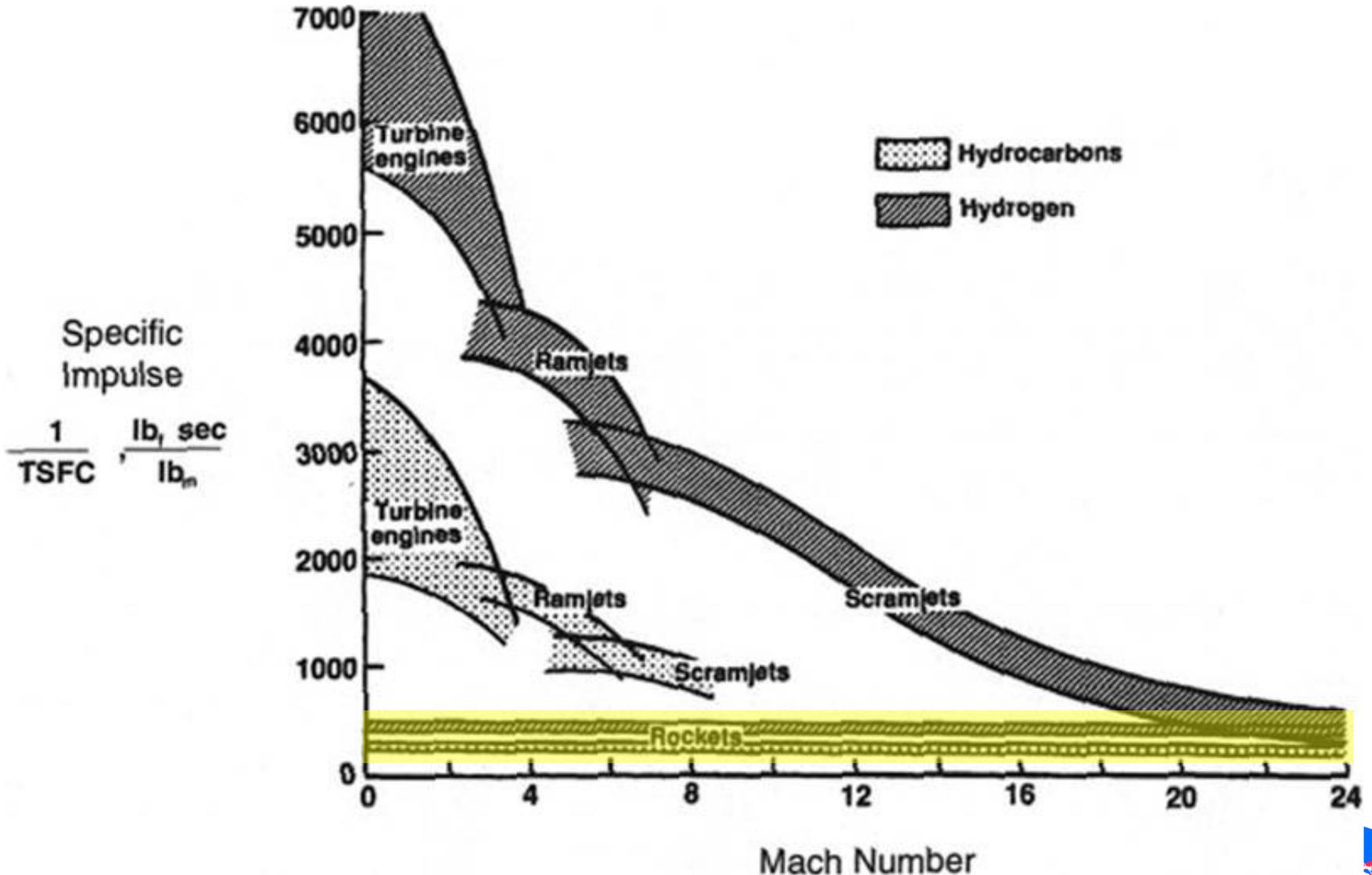
Propulsion Taxonomy

ETO Propulsion Systems

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                    ┌─────────────────────────────────┐
                    │   Space Propulsion Systems      │
                    │ For Space Access / Transportation│
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Earth to Orbit

In-Space

ETO Applications Range

Chemical Rocket

Chem/Thermal Combined Cycle

Beamed Energy

Launch Assist

Reusable

SSTO   TSTO

Shuttle Upgrades   Shuttle Derived

Expendable

for Heavy Lift Cargo
eg., HEDS

ELV Reliability

Upgrades and/or Safe
Return to Flight
Mods/Quality Adds

- Solids
  - Conventional
  - High Energy
- Liquids
  - LO_2/HC
  - Cryogenic
  - Storable
- Gels
- HEDM

- Air-Breathing
  - RBCC
- Combustion/Nuclear
  Combined Cycle
  - TBCC
- Scramjet

- Laser beams
- M-Wave

- Strap-Ons
- Mag-Lev
- E-M Accelerators
  - Rail Guns
- Gravity Assist
- Air Launch

In-Space Propulsion Systems

Space Propulsion Systems
For Space Access/Transportation

Earth to Orbit

In-Space

ISP Applications Range

Expendable
- Smart Independent
- Upper Stage (US) (eg., Centaur, IUS)
- Dumb US (eg., ASUS)
- S/C IPS

Microspacecraft
- Formation Flying

Reusable
- Space-Based
- Earth-Based

High-Value Manned Asset Control
- *ISS*
- Planetary Return/Ascent (w/people)
- Shuttle/RLV
- OMS and RCS

Spacecraft Control (Robotic / Unmanned)
- Drag Makeup
- Station Keeping
- Pointing and ACS
- Controlled Reentry/Disposal

Interstellar Flight

Chemical
- Solids
- Liquids
- HEDM
- Gels
- In situ Propellant Manufacturing
- Monopropellants

Solar
- Thermal
  - H_2
  - NH_3
- Solar Electric (SEP)
  - Electro Thermal (ET)
  - Electro Static (ES)
  - Electro Magnetic (EM)

Beamed Energy
- Laser
  - Thermal
  - Electric
- M-Wave

Nuclear
- Thermal
  - Fission
  - fusion
- Electric
  - ES
  - EM
- Antimatter

Propellantless Propulsion
- Tethers
- Sails
- Solar
- Plasma
- Gravity Assist
- Aeroassist

Breakthrough Physics
- Black Hole Assist
- Space Warps
- Gravo Magnetic
- Enhanced Weak Forces

2.7

LRE Applications

Space

- Booster (1st stage)
- Sustainer (2nd stage)
- USE (3rd+ stage)
- OME
- RCS / ACS
- Descent / Ascent
- Experimental Aircraft

RS-25

J-2X

Military

- SSM
  - ICBM
  - IRBM
  - SRBM
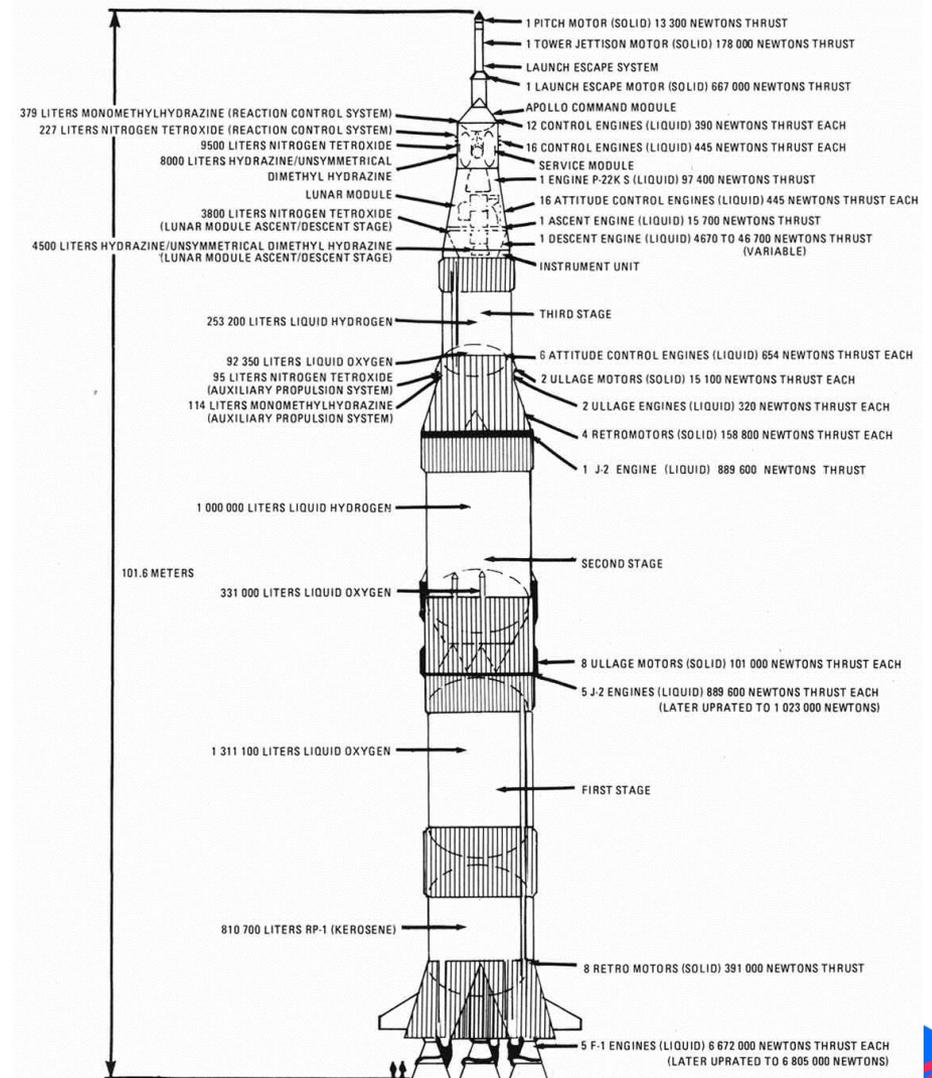 - SLBM
- SAM
- ASM
- AAM
- Aircraft Superperformance

LRE Applications

- Lift launch vehicle and payload from earth surface to LEO
  - Requires high thrust forces generated by direct thermal energy conversion
  - Two sources of thermal power available today
    - Chemical combustion
    - Nuclear - Not suitable because of radiation emitted

- Transfer payloads from LEO to into higher orbits or into planetary trajectories
  - Much lower thrust forces required
  - More options available

- Orbit maintenance, position control, station keeping and spacecraft attitude control
  - Small pulsing thrust forces required
  - Many options available

LRE Applications
## Integrated Launch Vehicle (Saturn-V)

- The Saturn-V launch vehicle used every rocket propulsion application
  - Booster (F-1 1st stage)
  - Sustainer (J-2 2nd stage)
  - Upper stage (J-2 3rd stage)
  - OMS (AJ10-137 SPS)
  - RCS (R-4D, SE-8)
  - Planetary Descent (VTR-10)
  - Planetary Ascent (RS-18)
- Propellant combinations used
  - LO_2/RP-1 (kerosene)
  - LO_2/LH_2
  - NTO/Aerozine-50
  - Solid (ullage motors)

379 LITERS MONOMETHYLHYDRAZINE (REACTION CONTROL SYSTEM)
227 LITERS NITROGEN TETROXIDE (REACTION CONTROL SYSTEM)
9500 LITERS NITROGEN TETROXIDE
8000 LITERS HYDRAZINE/UNSYMMETRICAL DIMETHYL HYDRAZINE
LUNAR MODULE
3800 LITERS NITROGEN TETROXIDE (LUNAR MODULE ASCENT/DESCENT STAGE)
4500 LITERS HYDRAZINE/UNSYMMETRICAL DIMETHYL HYDRAZINE (LUNAR MODULE ASCENT/DESCENT STAGE)

1 PITCH MOTOR (SOLID) 13 300 NEWTONS THRUST
1 TOWER JETTISON MOTOR (SOLID) 178 000 NEWTONS THRUST
LAUNCH ESCAPE SYSTEM
1 LAUNCH ESCAPE MOTOR (SOLID) 667 000 NEWTONS THRUST
APOLLO COMMAND MODULE
12 CONTROL ENGINES (LIQUID) 390 NEWTONS THRUST EACH
16 CONTROL ENGINES (LIQUID) 445 NEWTONS THRUST EACH
SERVICE MODULE
1 ENGINE P-22K S (LIQUID) 97 400 NEWTONS THRUST
16 ATTITUDE CONTROL ENGINES (LIQUID) 445 NEWTONS THRUST EACH
1 ASCENT ENGINE (LIQUID) 15 700 NEWTONS THRUST
1 DESCENT ENGINE (LIQUID) 4670 TO 46 700 NEWTONS THRUST (VARIABLE)
INSTRUMENT UNIT

253 200 LITERS LIQUID HYDROGEN
92 350 LITERS LIQUID OXYGEN
95 LITERS NITROGEN TETROXIDE (AUXILIARY PROPULSION SYSTEM)
114 LITERS MONOMETHYLHYDRAZINE (AUXILIARY PROPULSION SYSTEM)

THIRD STAGE
6 ATTITUDE CONTROL ENGINES (LIQUID) 654 NEWTONS THRUST EACH
2 ULLAGE MOTORS (SOLID) 15 100 NEWTONS THRUST EACH
2 ULLAGE ENGINES (LIQUID) 320 NEWTONS THRUST EACH
4 RETROMOTORS (SOLID) 158 800 NEWTONS THRUST EACH
1 J-2 ENGINE (LIQUID) 889 600 NEWTONS THRUST

1 000 000 LITERS LIQUID HYDROGEN

101.6 METERS

331 000 LITERS LIQUID OXYGEN

SECOND STAGE

8 ULLAGE MOTORS (SOLID) 101 000 NEWTONS THRUST EACH
5 J-2 ENGINES (LIQUID) 889 600 NEWTONS THRUST EACH (LATER UPRATED TO 1 023 000 NEWTONS)

1 311 100 LITERS LIQUID OXYGEN

FIRST STAGE

810 700 LITERS RP-1 (KEROSENE)

8 RETRO MOTORS (SOLID) 391 000 NEWTONS THRUST

5 F-1 ENGINES (LIQUID) 6 672 000 NEWTONS THRUST EACH (LATER UPRATED TO 6 805 000 NEWTONS)

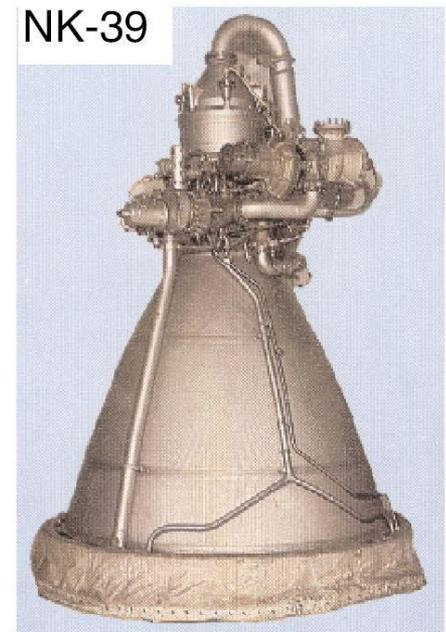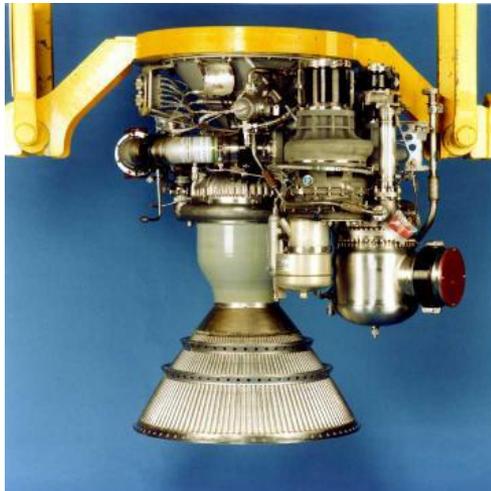# LRE Applications

Booster

- Provides initial propulsive thrust to launch vehicle
- Large thrust (T)
- High thrust-weight ratio (T/W)
- High specific impulse (I_{sp})
  - Area ratio (ε) limited by constraint from atmospheric pressure
- T, I_{sp}, and ε traded against propellants and power cycle
  - ε for SSME optimized at altitude for SRB staging
- Examples
  - F-1
  - RD-170
  - RD-180
 - RS-68
  - SSME
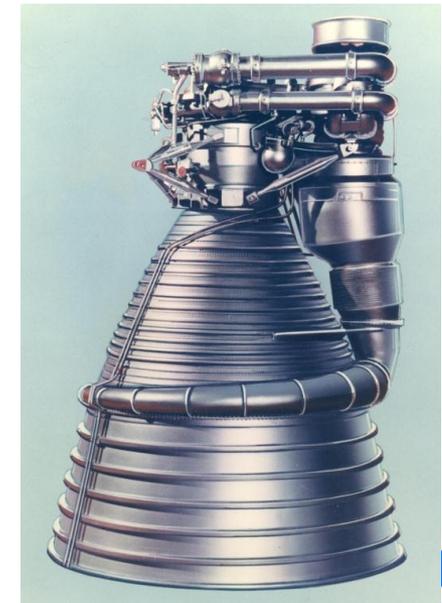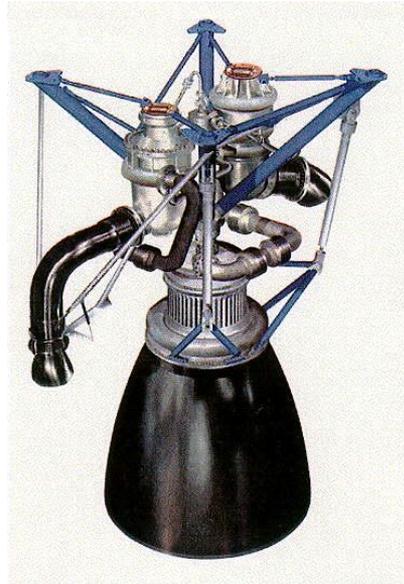
LRE Applications
Sustainer / 2ⁿᵈ Stage

- Provide supplemental impulse for achieving orbit.
- High thrust, but less required than a booster engine
- Higher ε than booster, less than orbital LRE
- Examples
  - J-2
  - LR91 series
  - RD-120
  - NK-39

NK-39

LRE Applications

Upper Stage Engine (USE)

- Typically applied for final orbital insertion or modification of orbital parameters

- Low to medium thrust (10 – 300 kN)
  - Dependent on upper stage / mission requirements

- Propellants typically hypergolic (multi-start, thrust-on-demand) or LO_2/LH_2 (fewer starts, higher energy, higher orbits, larger payloads)
  - Russians prefer LO_2/kerosene or NTO/UDMH for their upper stages

- Examples
  - RL10 family
  - Agena
  - Vinci

Agena

RS-72
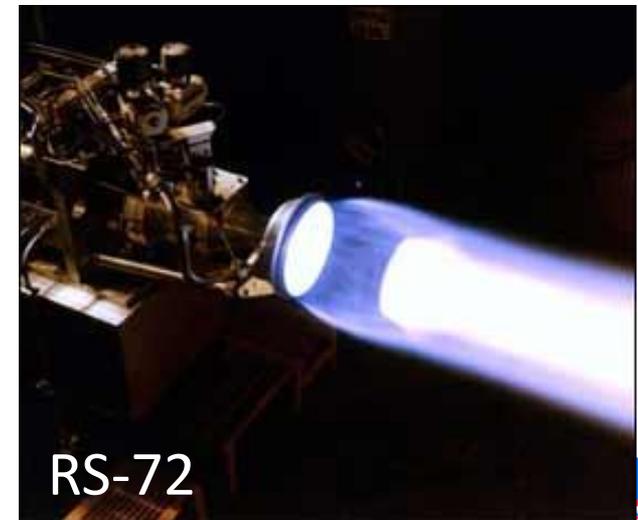
LRE Applications

Orbital Maneuvering Engine (OME)

- Moderate Thrust (0.1 – 20 Klbf thrust)
  - Dependent on vehicle mass and operational requirements
- Long life, multiple starts
- Typically pressure-fed hypergolic, radiatively cooled
  - Emerging interest in "green" propellants (i.e., LO_2/ethanol, HTP)
- Thrust on demand
  - Minimal preparatory procedures
- Examples
  - AJ10-137 SPS
  - AJ10-190 OME
  - TRW VTE

Aerojet AJ10-137 SPS Engine (Apollo)

NTO+AZ-50 / Pressure-fed / Ablative + Radiation + FFC

T: 20500 lbf I_{sp}: 310.5 sec P_c: 97 psia   ε: 62.5:1

LRE Applications
RCS/ACS

- Thrust-on-demand
- Throttleable
- Pulse-mode
- Low thrust (0.001 – 1000 lbf)
- Usually pressure-fed monopropellant or hypergolic bipropellants
  - Recent elevated interest in "green" propellants (i.e., LO_2/ethanol or HAN)
- Usually ablative or radiatively cooled with FFC
  - Regenerative cooling not used (thermal soakback)

LRE Applications
RCS Schematic
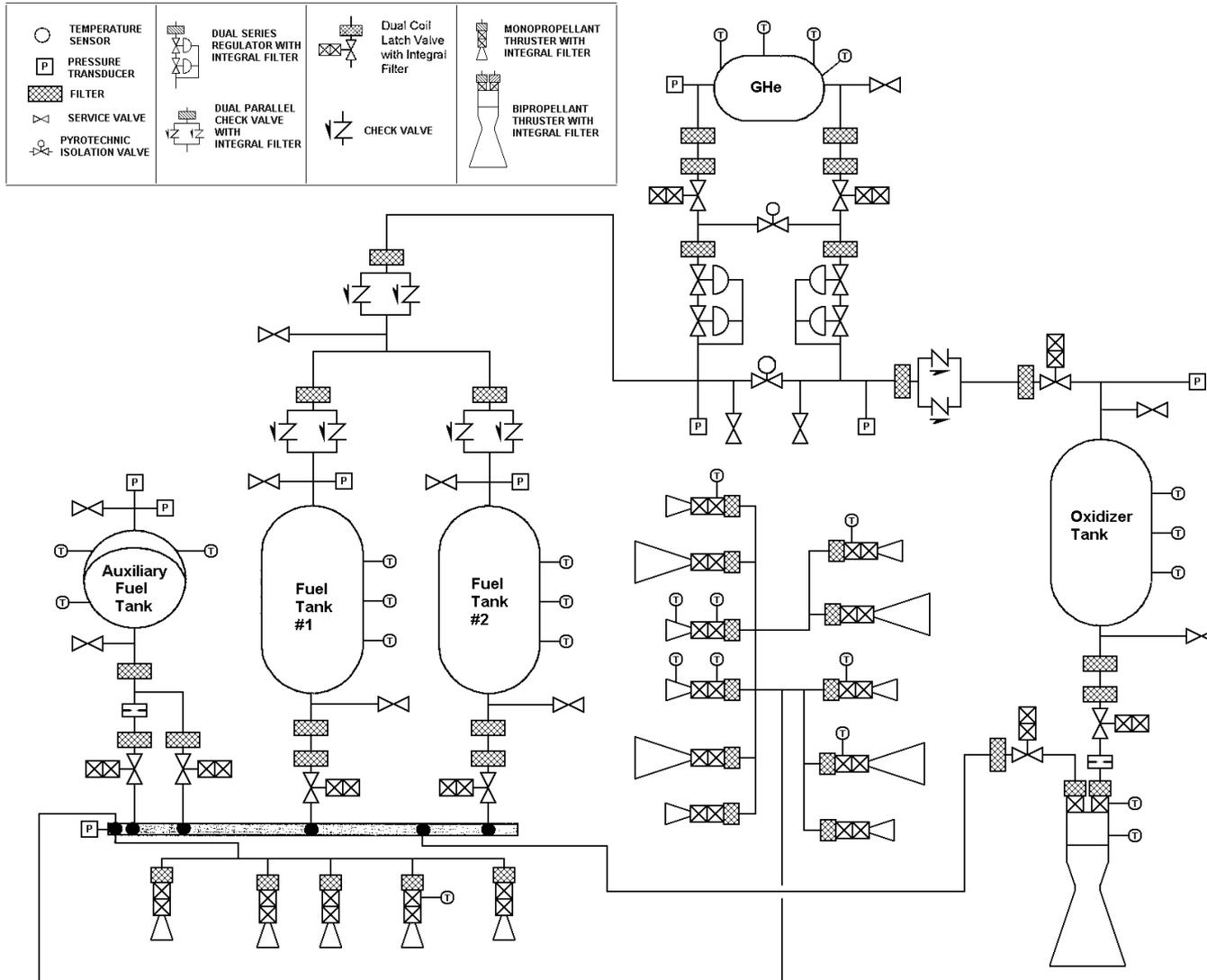
LRE Applications
RCS Thrusters
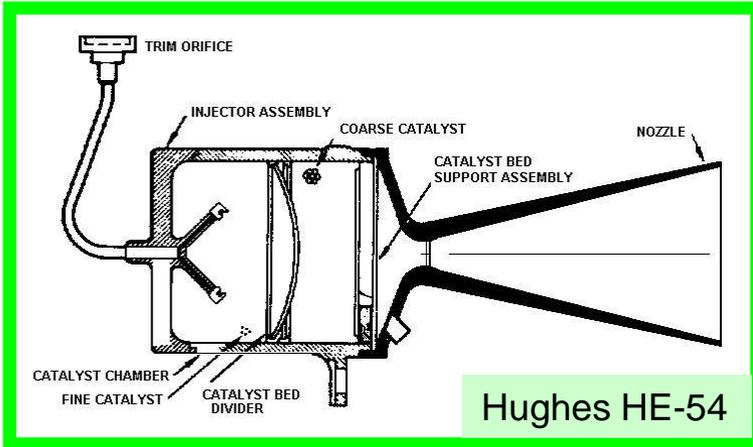
Hughes HE-54

- •Pressure-fed
- •Hypergolic biprop or Monoprop
- •Rad-cooled
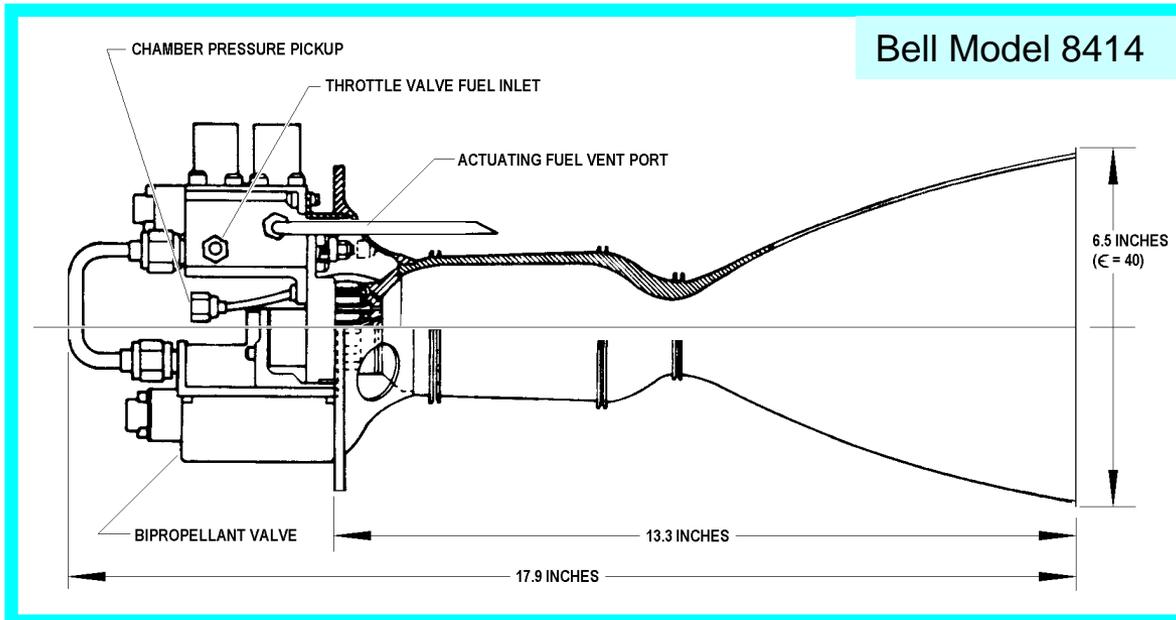
Bell Model 8414
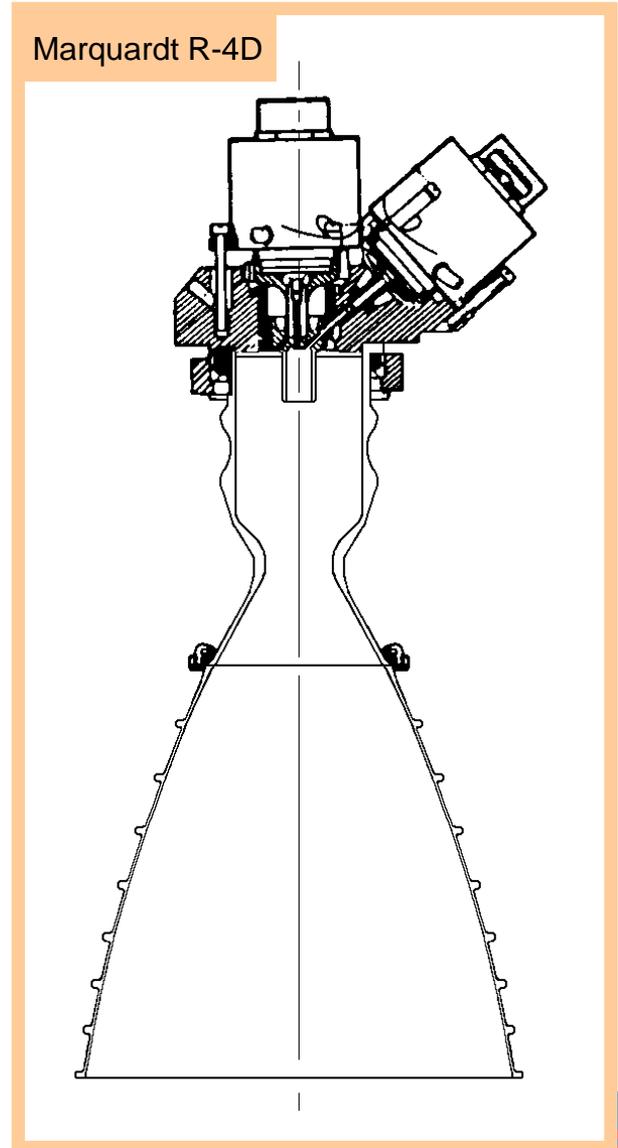
Marquardt R-4D

LRE Applications
Planetary Ascent / Descent

- Low/Moderate Thrust (0.6-10 Klbf)
- Throttleable (down to 5-10% power)
- Thrust-on-Demand
  - Fast response
  - High reliability
- Examples
  - Aerojet MR-80
  - TRW VTR-10 LMDE
  - Bell/Rkdn LMAE

MR-80 Viking Mars Descent Engine

N_2H_4 / Pressure-fed / Rad-cooled

T: 632 – 62 lbf (throttleable)

I_{sp}: 210 - 179 sec P_c: 250 – 27 psia

CATALYST BED CYLINDERS
RADIATION SHIELD
LOWER INJECTOR CLOSURE
INJECTOR PASSAGES
INJECTOR BOTTOM FLANGE
INJECTOR END PLATE
INJECTOR TOP FLANGE
AFT DOME CENTER SECTION
AFT DOME NOZZLE RING
INJECTOR BRANCHES
INTERIOR NOZZLES
INJECTOR WEBS
REACTOR HEATER
EXTERIOR NOZZLES
PROPELLANT VALVE MOUNT PLATE
REACTOR WALL
REACTOR WALL INSULATION
INJECTOR FEED TUBE
UPPER INJECTOR CLOSURE INSULATION
UPPER INJECTOR CLOSURE
VESPEL RADIATION SHIELD MOUNT BRACKET
THERMAL STANDOFF

LRE Applications
Planetary Ascent / Descent

Bell/Rocketdyne RS-18 LMAE (Apollo)

NTO+AZ-50 / Pressure-fed / Ablative + FFC

T: 3500 lbf I_{sp}: 307 sec P_c: 122 psia

MR: 1.6 ε: 45.6:1

TRW VTR-10 LMDE (Apollo)

NTO+AZ-50 / Pressure-fed / Ablative + FFC

T: 9850 – 1050 lbf (throttleable)

I_{sp}: 305 sec P_c: 104 psia

LRE Applications

Experimental Aircraft

- Moderate Thrust (6-50 Klbf)
- Throttleable / Restartable / High reliability
- Examples
  - XLR11-RM-#
  - XLR25-CW-1

FUEL TANK VENT LINE
FUEL TANKS
FUEL JETTISON LINE
XLR11 ENGINE
HYDROGEN PEROXIDE TANK
LOX JETTISON LINE
LOX TANK
AIR TANK - COCKPIT PRESSURE
HYDRAULIC PUMPS
DEFOG BLOWERS
PITOT STATIC FLOW XMTR
LOX PRIME LINE
TURBOPUMP EXHAUST LINE
TURBOPUMP
HYDRAULIC RESERVOIRS
DEFOG BATTERY
STATIC INVERTORS
UHF TRANCEIVERS
GASEOUS NITROGEN TANK
POWER SOURCE BATTERY
STATIC INVERTOR (BOTH SIDES)
EMERGENCY POWER SOURCE BATTERY
STABILITY AUGMENTOR (BOTH SIDES)
POWER SOURCE BATTERY

LRE Applications
Experimental Aircraft (X-15)

Reaction Motors XLR99-RM-1

- Throttleable (30%) / Restartable / High reliability / Monopropellant GG cycle
- LO_2 / NH_3
- 50 Klbf
- I_{sp}: 236 sec
- P_c: 600 psia
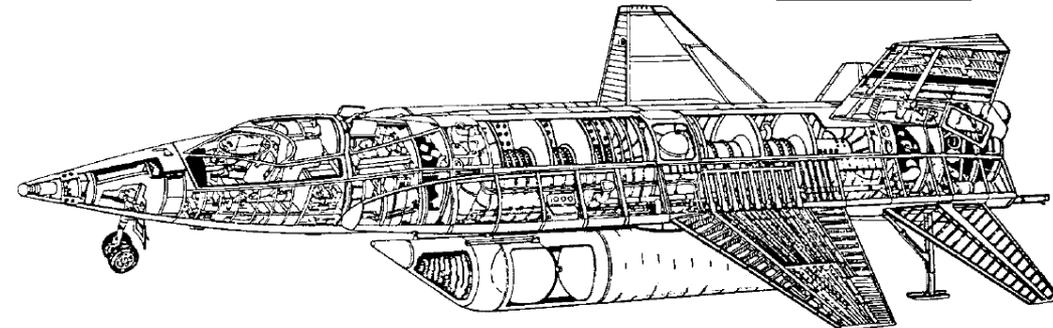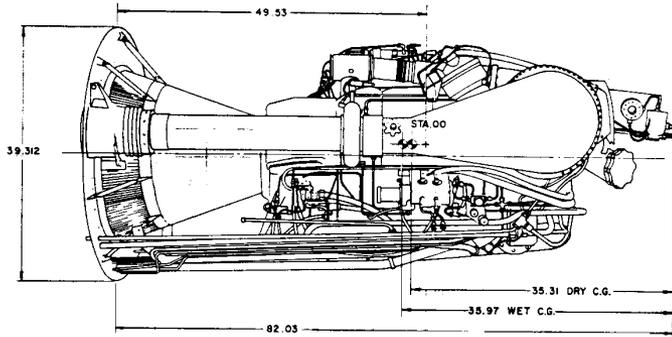
XLR99 Schematic
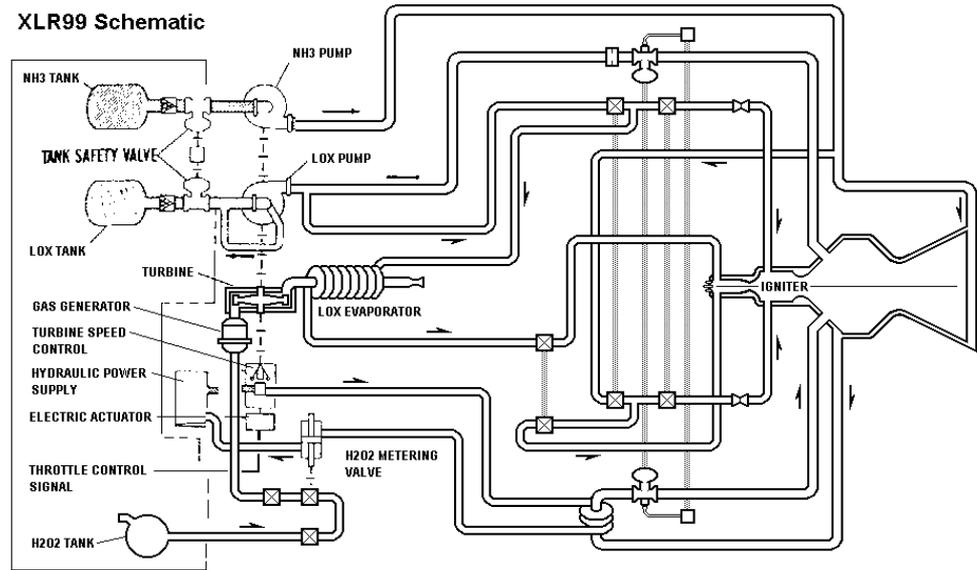
Backup / Supplemental
Military LRE Applications

(not included in short course material)

Module 3:

Liquid Propellants

Terms to Know

Liquid Propellants

- Fuel
- Oxidizer
- Monopropellant
- Catalyst
- Bipropellant
- Storable
- Space Storable
- Cryogenic
- Hypergolic

Some Liquid Propellants

STORABLE

- HAN – Hydroxylammonium Nitrate
- HTP – High Test Peroxide (H_2O_2)
- IRFNA – Inhibited Red Fuming Nitric Acid
- MMH – Monomethyl Hydrazine
- N_2H_4 - Hydrazine
- N_2O_4 - Nitrogen Tetroxide (NTO)
- RJ-1 – Ramjet Propellant 1
- RP-1 – Rocket Propellant 1
- TEA – Triethyl Aluminum
- TEB – Triethyl Boron
- UDMH – Unsymmetrical Dimethylhydrazine

SPACE STORABLE

- B_2H_6 - Diborane
- B_5H_9 - Pentaborane
- BrF_5 – Bromine Pentafluoride
- C_2H_6 – Ethane
- NH_3 – Ammonia
- N_2F_4 - Tetrafluorohydrazine

CRYOGENIC

- CH_4 – Methane
- FLOX – Mixture of LF_2 and LO_2
- LF_2 – Liquid Fluorine
- LH_2 – Liquid Hydrogen
- LO_2 – Liquid Oxygen
- OF_2 – Oxygen Difluoride

Propellants

- Propellants are the materials that are combusted by the engine to produce thrust.

- Bipropellant liquid rocket systems consist of a *fuel* and an *oxidizer*. They are the most common due to their high performance, but are more complex.

- Several propellants can be used singularly as *monopropellants* (i.e. HTP, N_2H_4, UDMH), which release energy when they decompose either when heated or catalyzed.

- The mission / requirements of the vehicle will directly effect the selection of propellants and configuration (power cycle) of the propulsion system(s).

- The primary propellant types to be discussed are:

  - Storable

  - Space Storable

  - Cryogenic

Propellant Types - Storable

- *Storable* propellants are liquid at sea level conditions of temperature and pressure and can be stored indefinitely in sealed tanks.

- One drawback of storable propellants is that, with the exception of kerosene-based fuels (RP-1, RJ-1) they are invariably toxic, reactive, corrosive, and difficult to handle.

- Most storable propellant combinations are *hypergolic*, meaning that they ignite spontaneously when in contact with each other.

  - Hypergolic propellant combinations are primarily used for small thruster applications.

  - Elimination of the ignition system reduces engine complexity and enables thrust-on-demand capability (quick start with minimal prep) and pulse mode (multiple rapid

- Historical note: The ability to cast large solid rocket motors did not evolve until the late 1950's, so initial large-thrust rocket propulsion systems used liquid propellants, mostly storables.

  - The evolution of RFNA > IRFNA > NTO had strategic impacts (U.S.A. vs. Soviet Union) in terms of performance and storability.

Propellant Types – Storable
Monopropellants

- *Monopropellants* are storable liquid propellants that can be induced to decompose to a gaseous state in the presence of a catalyst (or contamination) and release heat that can be converted to thrust.

  - Catalysts – Shell 405, silver/cobalt plated wire gauze, sodium or potassium permanganate, etc.

  - Some monopropellants can be used in bipropellant systems as either a fuel (N_2H_4, UDMH) or an oxidizer (HTP), which can enable more operational flexibility

- The performance (i.e., I_{sp}) is lower than that of bipropellant systems, but the systems are more simple (higher reliability).

- One drawback of monopropellant systems is that the reactive nature of the propellant requires high standards of cleanliness to prevent uncontrolled decomposition from contaminants.

- Examples
  - Hydrogen Peroxide (H_2O_2, HTP) up to 90-98% concentration
  - Hydrazine (N_2H_4) most commonly used
  - UDMH (used in GG in RD-119)
  - HAN (experimental)

Propellant Types – Space Storable

- *Space storable* propellants are liquid in the temperatures of space and generally have a net boiling point greater than 230°R.

- They can be stored for longer periods of time than cryogenic propellants when in space and depending on the storage tank design, thermal environment, and tank pressure.

- They are generally more energetic than most storable propellant combinations, but are rarely used due to their extreme toxicity, reactivity and handling difficulties.

- Actual application of space storable propellants in an operational propulsion system is rare due to the toxicity hazards.

  - Beginning in the late 1950's, the USAF studied the use of space storable propellants in upper stages. The findings indicated that the operational hazards did not justify the performance gains.

  - The XLR99-RM-1 rocket engine for the X-15 experimental hypersonic aircraft used a LO_2/NH_3 propellant combination.

Propellant Types – Cryogenic

- *Cryogenic* propellants are liquefied gases at extremely low temperatures (approx. 30°R to 230°R) and are typically the most energetic types of propellants.

- However, they are more difficult to store for any length of time (vaporization losses) and require provisions for venting the propellant tank.

- LO_2 and LH_2 are the most commonly used liquid cryogenic propellants, and will be used in the J-2X.

  - LH_2
    - Advantages – High performance, excellent coolant
    - Disadvantage – Low density (~4.5 lb/ft^3 vs. 72 lb/ft^3 for LO_2, resulting in a disproportionate size in propellant tanks)
  - LO_2
    - Advantages – Non-toxic, high reactivity to fuel (high performance). Only fluorine is a better oxidizer.
    - Disadvantage – Not selective about what it uses as fuel. It prefers hydrogen or hydrocarbons, but will consume almost anything with an oxidation potential.

Backup / Supplemental

Propellant Types – Storable
Evolution of Domestic Kerosene for Rocket Propulsion

Ether

Gasoline

Goddard

(Performance)

Alcohol

Alcohol + water

Project Hermes

(Combustion stability)

JP-4 Kerosene

Navaho / Jupiter

(Performance)

RP-1 Kerosene

Project REAP / Atlas

(Performance repeatability)

RP-2 Kerosene

(Ultra-Low Sulfur)

SLI / 2GRLV

(System Reusability)
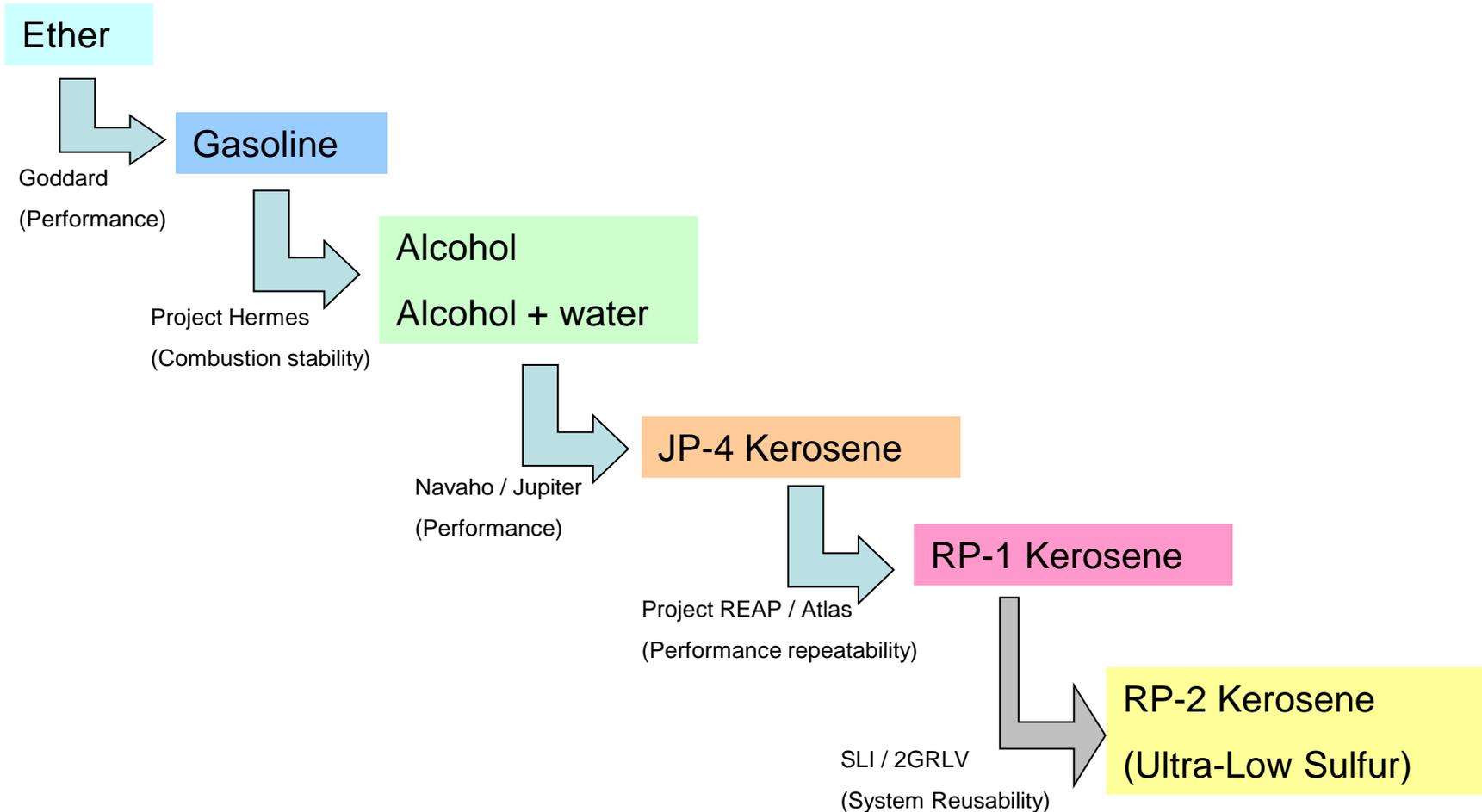
Propellant Types – Storable
Kerosene Forms - Domestic

Fuel	Formula	MW	HF (cal/mol)	Density (lb/ft3)
JP-4	$C_{10}H_{20}$	140.27	-63756	48.5
JP-5	$C_{10}H_{19}$	139.26	-57177	50.97
JP-7	$C_{10}H_{21}$	141.28	-70187	49.5
JP-9	$C_{652}H_{999}$	8835.2	-335528	58.12
JP-10	$C_{10}H_{16}$	136.24	-25219	58.06
RJ-4	$C_{12}H_{20}$	164.29	-44930	57.66
RJ-5	$C_{140}H_{184}$	1867.0	+199776	66.17
RJ-6	$C_{700}H_{999}$	9414.7	-92195	64.3
RP-1	$C_{10}H_{19.5}$	139.5	-57600	50.3
Quadricyclane	C_7H_8	92.14	+72200	57.36

Propellant Types – Storable
Nitric Acid Blends - Domestic

Blend	Description
FNA	Fuming Nitric Acid
SFNA	Stable Fuming Nitric Acid (obsolete)
HDA	High Density Acid (see description for IRFNA IV)
MHDA	Modified High Density Acid
RFNA	Red Fuming Nitric Acid, 83.5-84.5% HNO_3 + 14% NO_2 + 1.5-2.5% H_2O, ref. MIL-P-7254
IRFNA	Inhibited RFNA, 82.3% HNO_3 + 15% NO_2 + 2% H_2O + 0.7% HF, ref. MIL-P-7254 and AGC-77002
IRFNA III-A	Type III-A IRFNA, 83.4% HNO_3 + 14% NO_2 + 2% H_2O + 0.6% HF, ref. MIL-P-7254 and AGC-77002
IRFNA III-B	Type III-B IRFNA, ref. MIL-P-7254E-1
IRFNA IV	Type IV IRFNA, aka HDA, 54.3% HNO_3 + 44% NO2 + 1% H2O + 0.7% HF, ref. MIL-P-7254
WFNA	White Fuming Nitric Acid, 97.5% HNO_3 + 2% H_2O + 0.5% NO_2, ref. MIL-P-7254
IWFNA	Inhibited WFNA, a blend of WFNA and 0.6% HF as a passivant
MWFNA	Modified White Fuming Nitric Acid
MDIFNA	Maximum Density Inhibited Fuming Nitric Acid, 54.8% HNO_3 + 44% N_2O_4 + 0.7% HF + 0.5% H_2O

Module 4:

Power Cycles

Objective

Continue the overview discussion of liquid propellant rocket engines to focus on the various power cycles that are used to power the turbomachinery.

LRE Power Cycle Review

- Pressure-Fed
- GG, Monopropellant
- GG, Bipropellant, Single TPA
- GG, Bipropellant, Dual TPA, Series Turbines
- GG, Bipropellant, Dual TPA, Parallel Turbines
- Tap-Off
- Expander, Single TPA
- Expander, Dual TPA
- Fuel-Rich Staged Combustion, Dual Preburners
- Fuel-Rich Staged Combustion, Single Preburner
- Oxidizer-Rich Staged Combustion
- Full-Flow Staged Combustion

Terms & Acronyms

- FTP (Fuel Turbopump)
- GG (Gas Generator)
- GGFV (Gas Generator Fuel Valve)
- GGOV (Gas Generator Oxidizer Valve)
- HEX (Heat Exchanger)
- MCC (Main Combustion Assembly)
- MFV (Main Fuel Valve)
- MOV (Main Oxidizer Valve)
- Nozzle
- NE (Nozzle Extension)
- OTBV (Oxidizer Turbopump Bypass Valve)
- OTP (Oxidizer Turbopump)
- TCA (Thrust Chamber Assembly)
- TPA (Turbopump Assembly)

Rocket Engine Cycles

- A rocket engine "cycle" refers to the power cycle that the engine system uses to power the turbopumps to pressurize the propellants.
- A number of thermodynamic cycle options exist
- Which one used depends on application or mission requirements
- One cycle is not right for every application
- The selection of power cycle can be driven by many factors:
    - Propellants
    - Performance (thrust, specific impulse)
    - Safety / Reliability
    - Reusability
    - Technical Risk
    - Cost / Schedule
    - Etc.

LRE Cycle Uses

	Staged Combustion, Single Chamber Tripropellant	Staged Combustion, Dual Preburner	Staged Combustion, Single Preburner	Gas Generator	Expander	Tap-off
Advantages	Highest integrated performance available (closed cycle). Maximizes propellant bulk density and Isp.	High performance (closed cycle). Very attractive for reusable applications. Easler MR and thrust level throttling characteristics.	High performance (closed cycle). Simplier than multi preburner options to left. Very attractive for reusable applications	Simple cycle, low production costs, easlier to develop	High reliability, benign failure modes (containted), simple cycle	Simple cycle with fewer parts, lower production costs, easier maintainability
Disadvantages	Most difficult to develop. Will be very expensive. Production cost makes reusable applications mandatory. Vehicle must be very performance driven such as SSTO.	More difficult to develop than single PB. Tends to be very expensive. Failure modes tend to be more involved. Production cost makes reusable applications almost mandatory.	More difficult to develop. Tends to be more expensive. Failure modes tend to be more involved.	Lower performance because of open cycle. Performance level makes this unattactive for most reusable applications.	Limited to LOX/LH2 propellants only. Limited performance because of heat transfer limitations.	Hot gas duct that taps off from the MCC and mixes dilluent fuel to regulate gas temperature. Lower performance (Open cycle).
Potential Applications	Reusable SSTO.	Booster or upperstage, reusable rockets	Booster or upperstage, reusable or expendible rockets (May depend on propellant choices)	Booster or upper stage, expendible rockets	Booster or upperstage, reusable or expendible rockets	Booster or upper stage, expendible rockets

Pressure-Fed "Cycle"
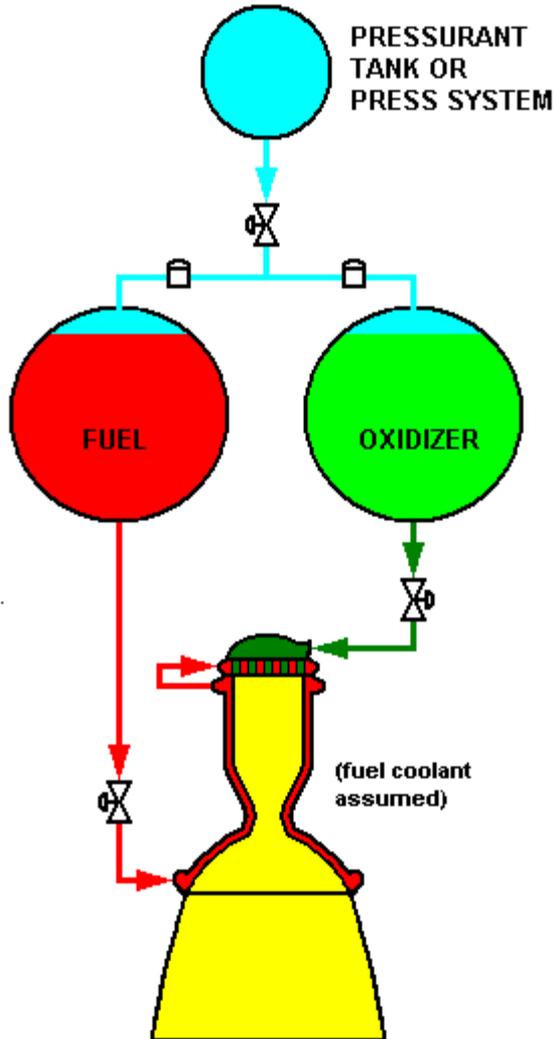
Not really a cycle, since it does not power turbomachinery, but serves as a basis of comparison. While this is the simplest system, it is also the most performance-limited in comparison to pump-fed systems.

- Excellent reliability
- Robust start/shutdown*
  - "Thrust on demand"
- Good storability*
- Good throttleability
- Acceptable performance
  - Trade thrust vs. tank weight
- Examples
  - Aerojet AJ10-190 (STS)
  - Aerojet AJ10-118 (Delta II)
  - Most RCS/ACS systems

* Assumes use of hypergolic propellants

4.7

Gas Generator (GG) Cycle

- One of the first power cycles developed for rocket propulsion
- Uses either dedicated or common propellants in gas generator (GG) to produce turbine drive gas
- Turbine exhaust dumped, resulting in degraded I_{sp} performance
- Good reliability
- Robust start/shutdown
- Lower operating pressures mitigate the need for boost pumps
- Can utilize almost any viable bipropellant combination

GG Cycle Variations

- Monopropellant GG
- Bipropellant GG
  - Common-shaft main TPA
  - Separate fuel and oxidizer TPAs
    - Series turbines
    - Parallel turbines

Gas Generator (GG) Cycle
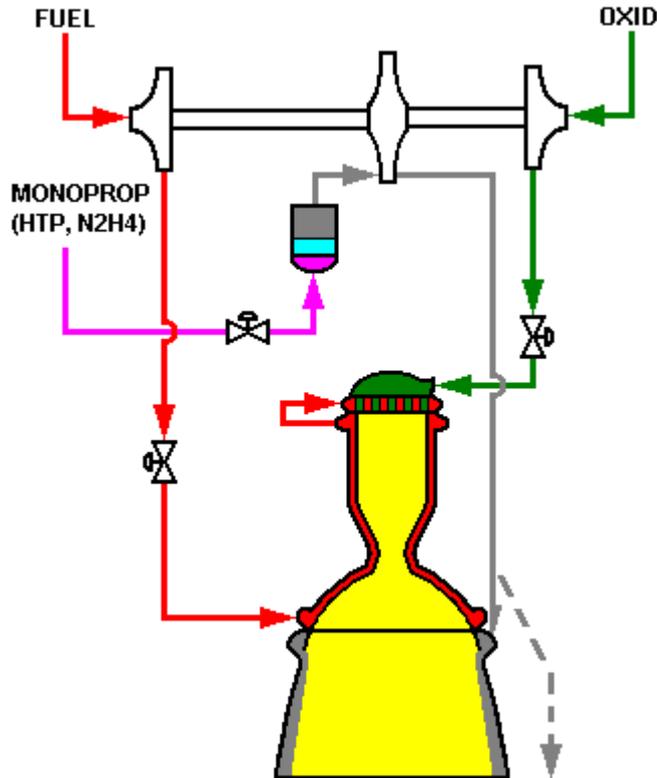
Monopropellant GG

- Early/original power cycle
- Acceptable performance
    - Independent monopropellant control provides more reliable system, but at the cost of increased weight of third propellant
- Examples
    - A-4 (V-2)
    - A-6 (Navaho-I)
    - A-7 (Redstone)
    - RD-107/108 (R-7 family)
    - XLR99-RM-1 (X-15)
    - AR2-3A (F-104)

Gas Generator (GG) Cycle

Bipropellant, Single TPA
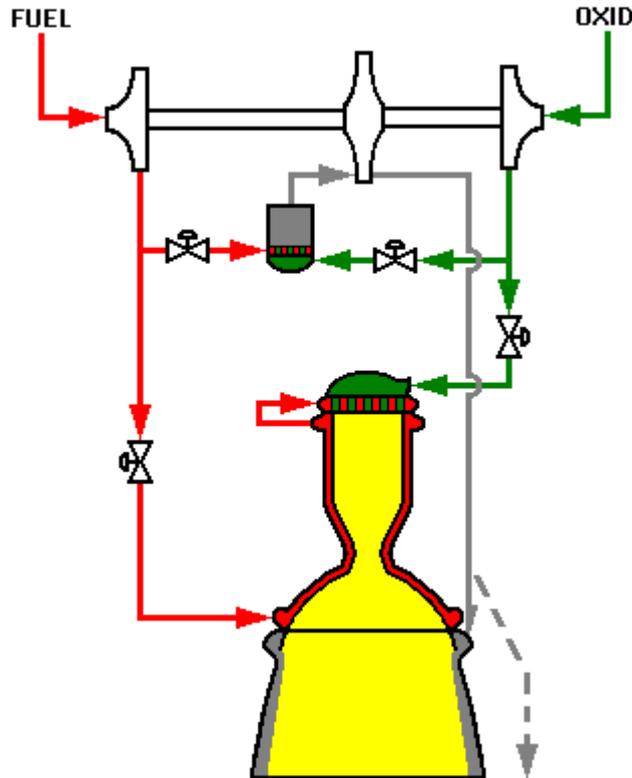
- Improved performance over monopropellant GG
  - Bootstrap start
  - T/W improved by elimination of 3rd propellant
- This cycle works well for propellants with similar fluid properties (i.e., density, viscosity = LO$_2$/RP-1) to allow a common shaft RPM.
- Examples
  - F-1
  - Atlas MA-2, -3, -5, -5A
  - Navaho-II, -III
  - MC-1 / Fastrac
  - S-3D → H-1 → RS-27

Gas Generator (GG) Cycle

Bipropellant GG, Dual TPA, Series Turbines
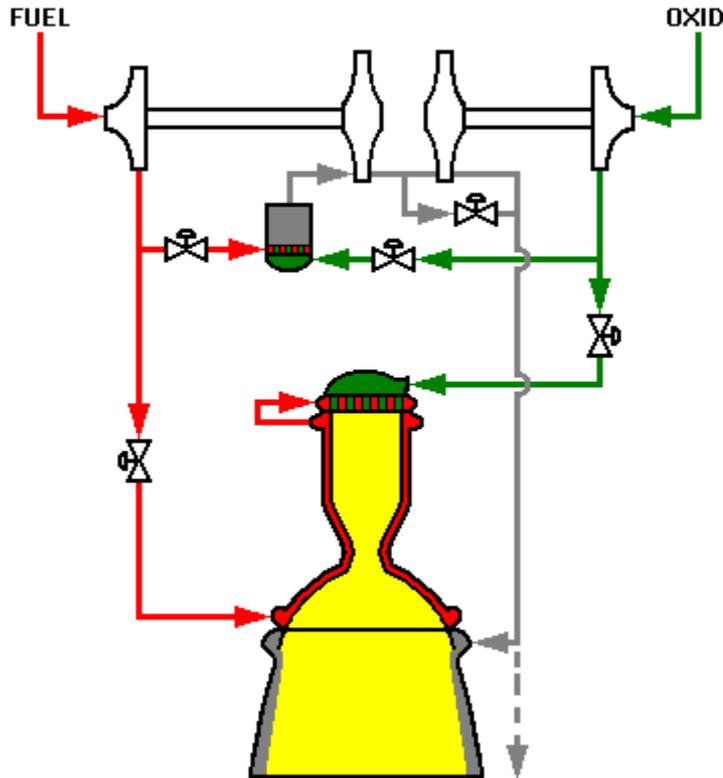
- Allows independent thrust and MR control
- This cycle works well for propellants with different fluid properties (LO_2/LH_2) that require different pump speeds.
- Examples
  - J-2 $\rightarrow$ J-2X

Gas Generator (GG) Cycle
Bipropellant GG, Dual TPA, Parallel Turbines
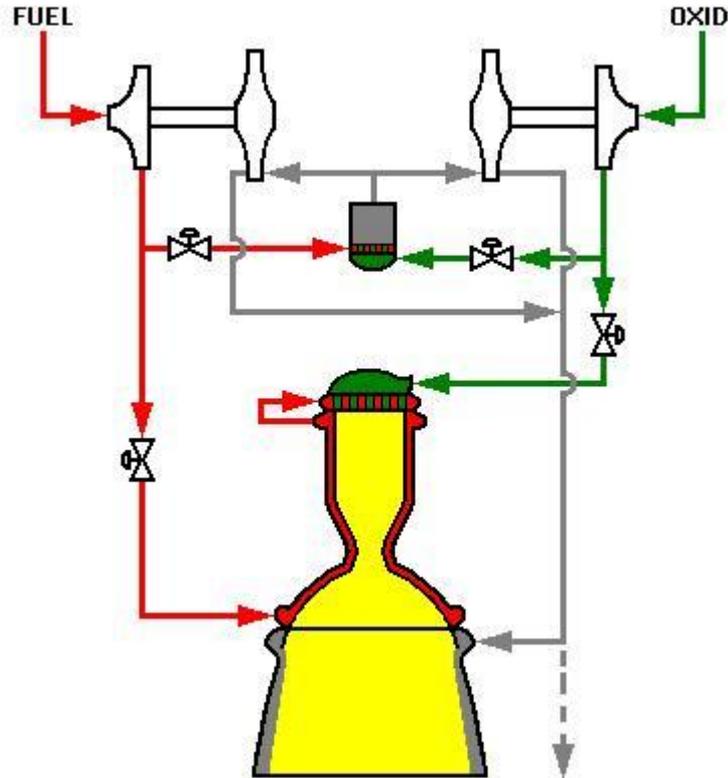
- Allows independent thrust and MR control
- This cycle works well for propellants with different fluid properties (LO$_2$/LH$_2$) that require different pump speeds.
- Examples
  - Vulcain
  - RS-68

Tapoff Cycle
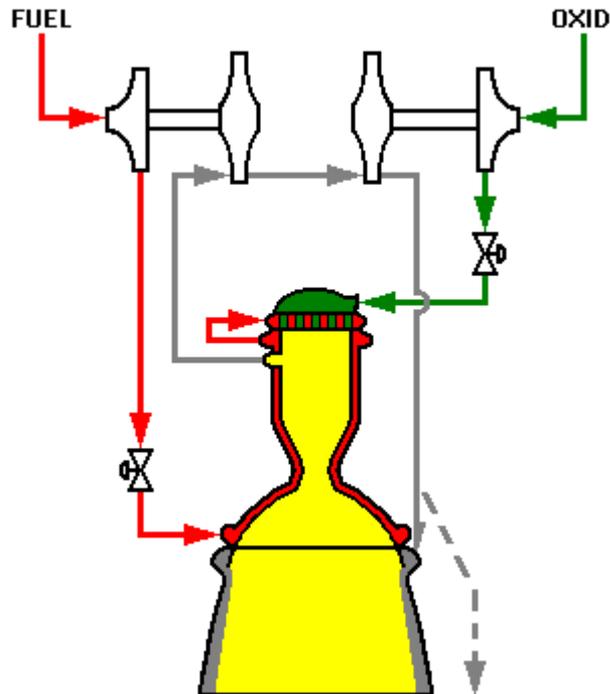
- Turbopumps driven by hot gas tapped from main chamber
- Good throttleability
- Low operational experience
- Simplicity offers potential high reliability
- Examples
  - J-2S

Expander Cycle
Single TPA
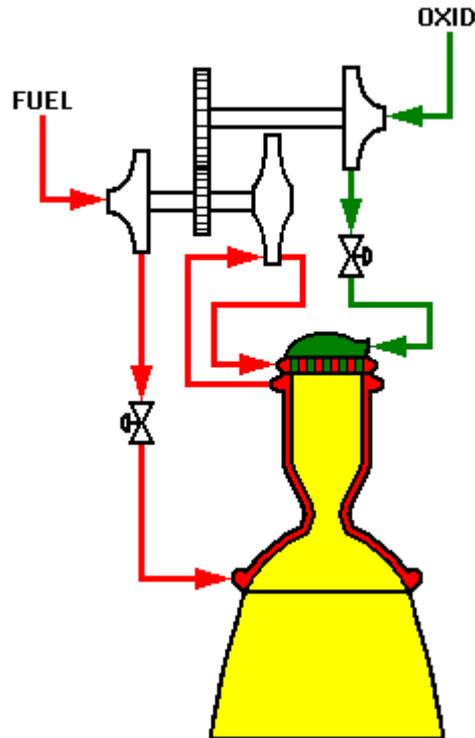
- Good throttleability
- Thrust limited by ability to utilize heated fuel
  - Requires high heat transfer efficiency and/or multi-stage turbine to extract work
  - T/W impact
- High reliability
- Benign failure modes
- Examples
  - RL10 family

Expander Cycle
Dual TPA
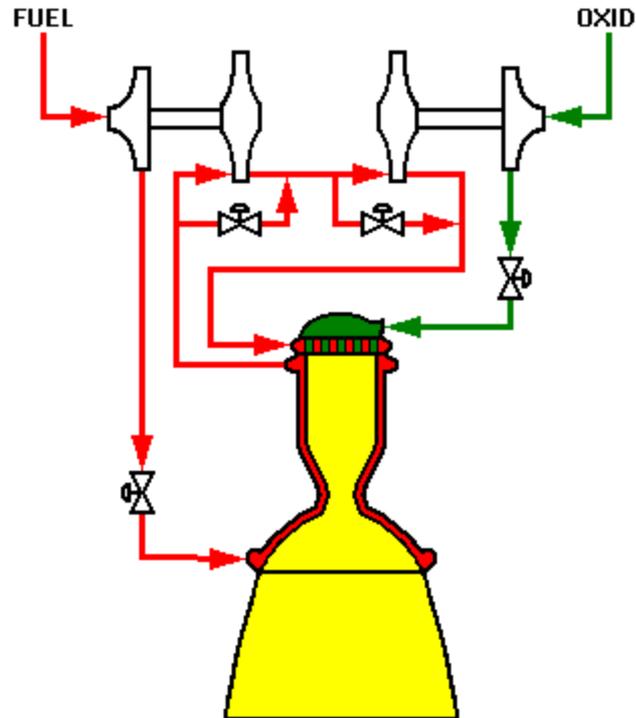
FUEL
OXID

- Good throttleability
- Turbine bypasses permit independent thrust and MR control
- Thrust limited by ability to utilize heated fuel
- High reliability
- Benign failure modes
- Examples
  - RL60
  - MB-xx

Staged Combustion (SC) Cycle

- Utilizes all propellants to generate thrust

- High performance (thrust, I_{sp}, T/W)

- High I_{sp} requires high operating pressures

- Good reliability, but high operating conditions demand vigilance

- The high operating pressures usually requires the use of boost pumps to increase propellant pressure entering main pumps.

SC Cycle Variations

- Fuel-Rich (FRSC)
  - Often used with LO_2/LH_2 propellants

- Oxidizer-Rich (ORSC)
  - Often used with LO_2/Kerosene propellants
  - NTO/UDMH also used

- Full-Flow (FFSC)
  - One experimental system developed (IPD) using LO_2/LH_2 propellants
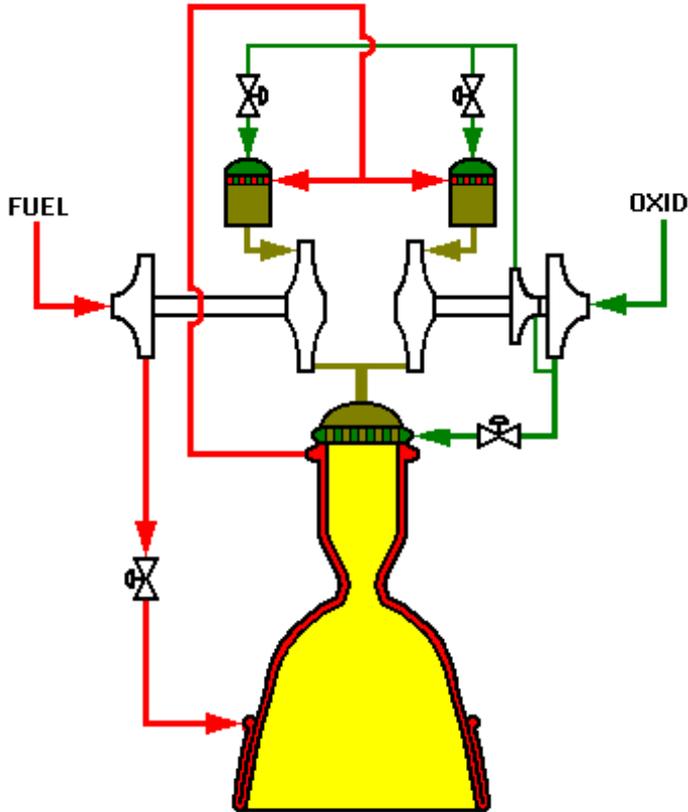
Dual Preburners

- Permits independent MR and thrust level throttleability
- Examples
  – SSME (with boost pumps)

4.17
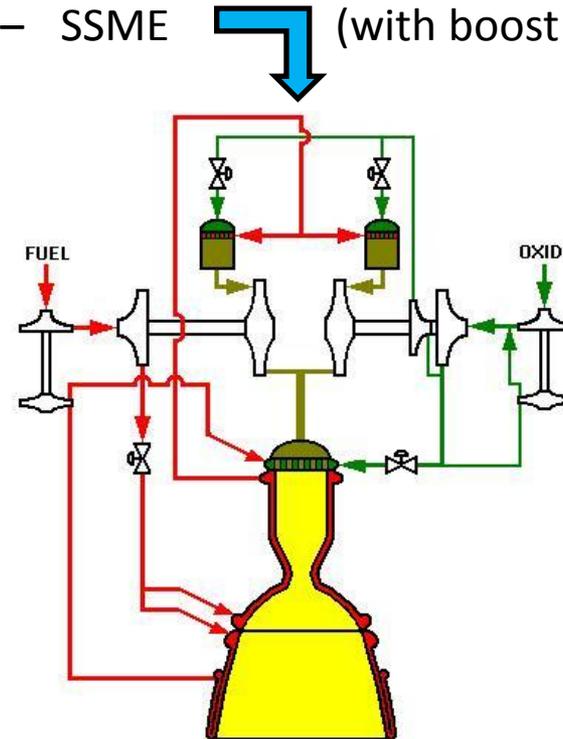
Fuel-Rich SC Cycle
Single Preburner, Dual TPAs

FUEL
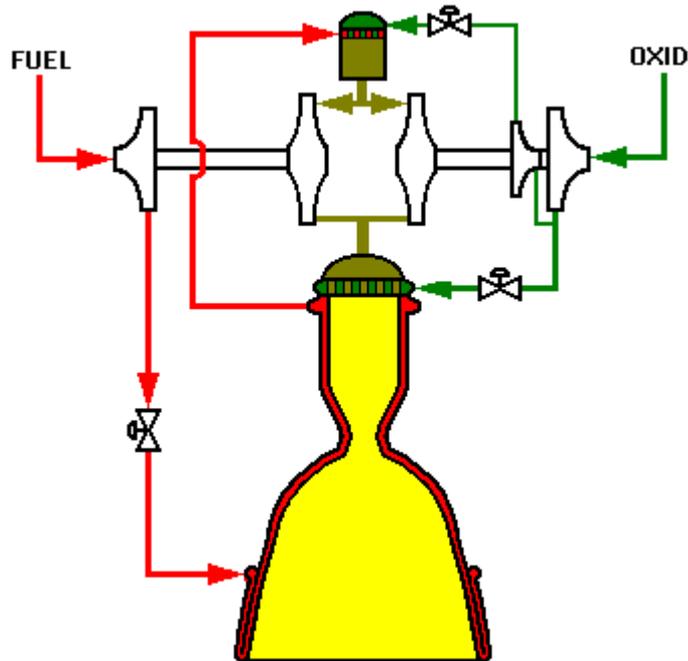
OXID

- Permits some MR and thrust level throttleability
- Better system simplicity offers better reliability than DPFRSC system
- Examples
  - RD-0120
  - LE-7
  - RS-30 ASE (DDT&E incomplete)
  - COBRA (DDT&E incomplete)

Oxidizer-Rich SC Cycle
Single TPA with Boost Pumps
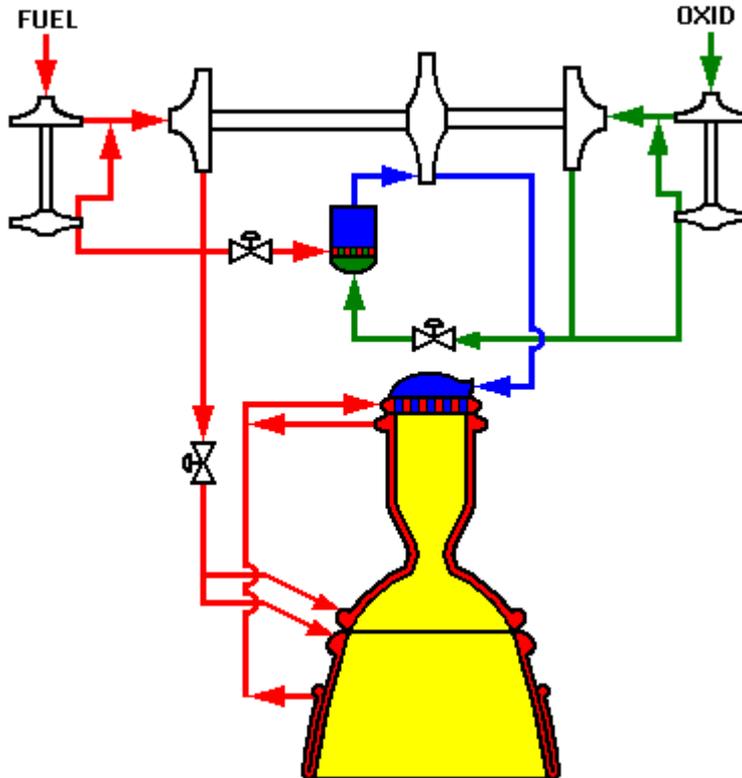
- Good Reliability
- Requires use of materials resistant to ignition in an oxidizer-rich environment
  - Requires exotic coatings
- Used exclusively in Russia
- Examples
  - RD-253
  - RD-170 Family
    - RD-170, -171, -172
    - RD-180
    - RD-191
  - RS-84 (DDT&E incomplete)

Full-Flow SC Cycle
Two TPAs with Two PBs

FUEL

OXID
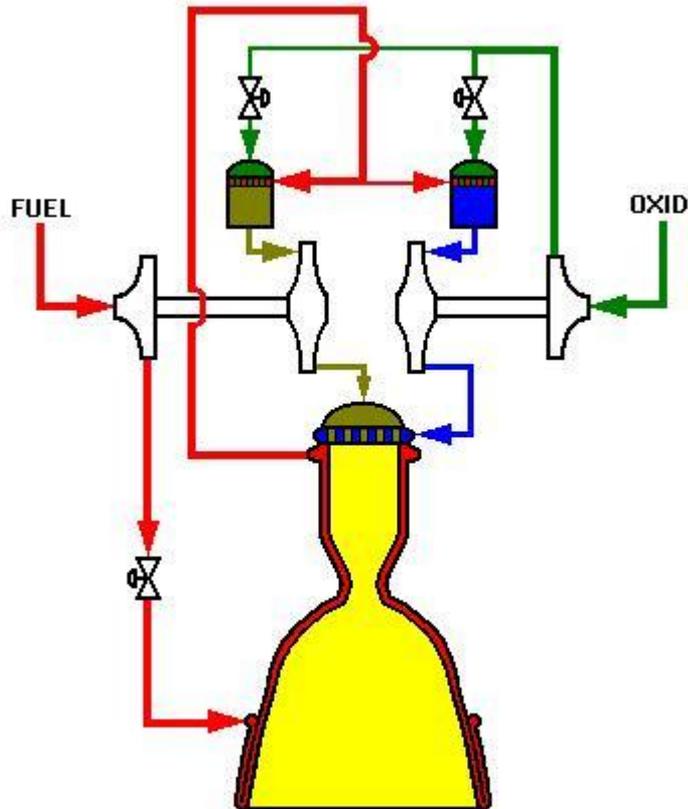
- System complexity degrades reliability and increases cost
- Complicated flow management requires complex transient and mainstage control
- Examples
  - IPD (Integrated Powerhead Demonstrator)

Other Cycles

- In addition to the primary power cycles already discussed, there are numerous variations of these that can be investigated to pursue higher performance.

- These variations are limited by the different permutations of components and system enhancements that can be integrated.

- Also, occasionally a "wildcat" power cycle will be proposed. Often they will promise high performance, but are hampered by high technical risk and low validated substantiation.

  - Dual-expander cycle

  - Augmented expander cycle

  - Cycles using tripropellants

  - Combined cycle system

  - …and then there are the real crazy ones…

Backup Slides

Example Engines

Gas Generator (GG) Cycle
Monopropellant GG
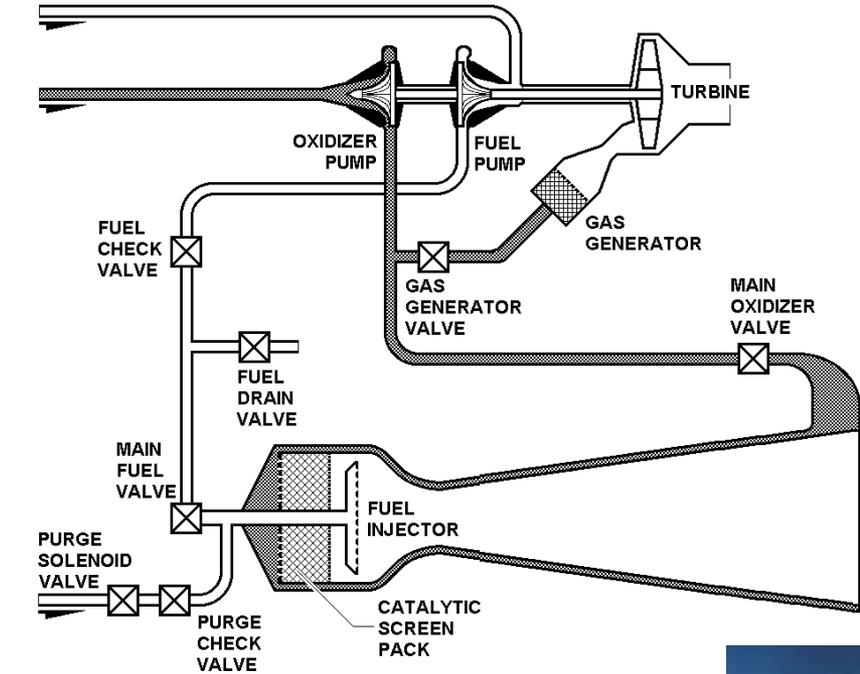
Rocketdyne AR2-3

- F-104 superperformance
- 90% HTP / JP-4
- 3000-6600 lbf thrust
- I_{sp}: 234 sec (alt)
- P_c: 560 psia
- Initially used in 1957, later in 2000 considered for X-37 propulsion
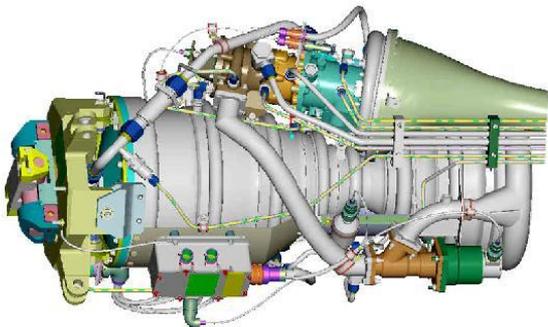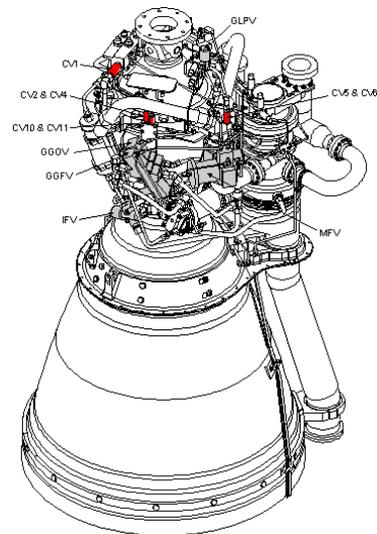
Gas Generator (GG) Cycle
Bipropellant GG, Single TPA
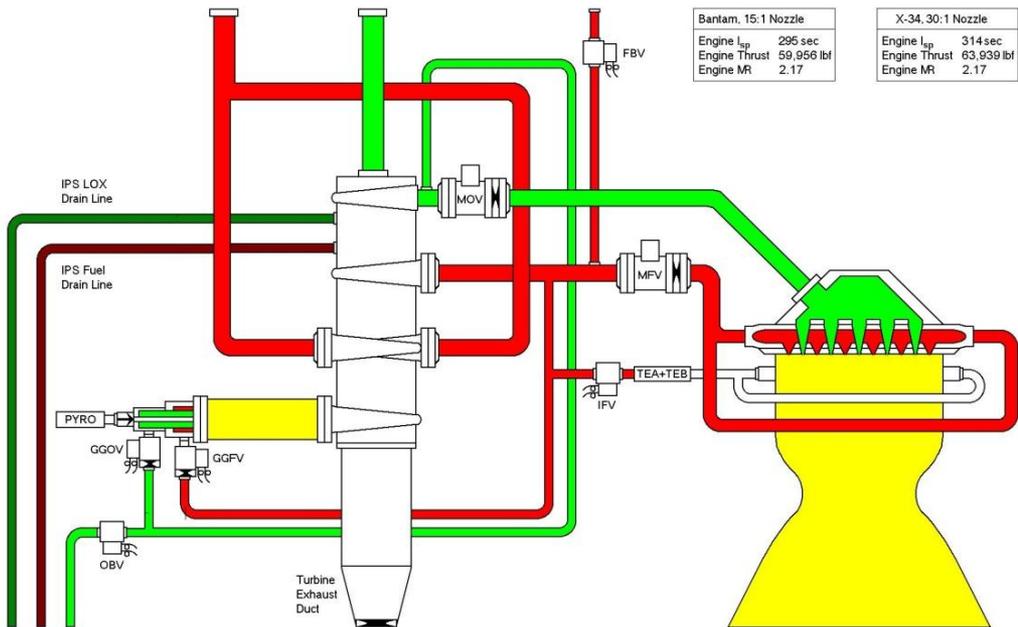
Bantam, 15:1 Nozzle	
Engine I_{sp}	295 sec
Engine Thrust	59,956 lbf
Engine MR	2.17

X-34, 30:1 Nozzle	
Engine I_{sp}	314 sec
Engine Thrust	63,939 lbf
Engine MR	2.17

NASA/MSFC MC-1 Fastrac

- X-34 hypersonic aircraft
- LO_2 / RP-1
- 64 Klbf thrust (vac)
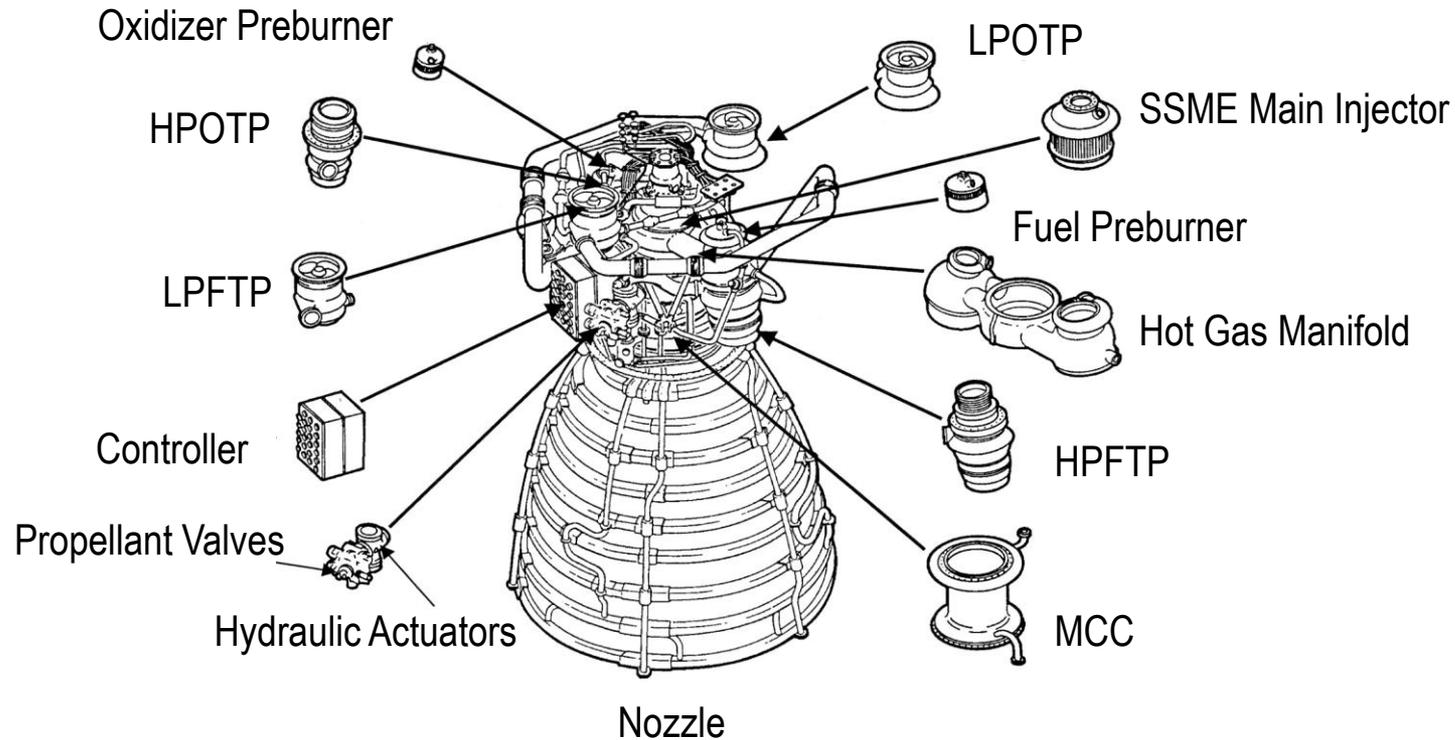- I_{sp}: 314 sec (vac)
- P_c: 650 psia

4.24

Gas Generator (GG) Cycle
Bipropellant GG, Dual TPA, Parallel Turbines

Nota: The engine is shown in operation.

COLOR KEY

- OXYGEN
- HYDROGEN
- HOT GAS
- HELIUM CONTROL
- HELIUM PRESSURIZATION

Note: The engine is shown in operation.

Ref.: Snecma brochure

HM60 Vulcain Mk-1

- Ariane-V booster
- 252 Klbf thrust
- I_{sp}: 430 sec (vac)
- P_c: 1624 psia
- MR: 5.3
- ε: 45:1

Large RP-1 Engine Comparisons
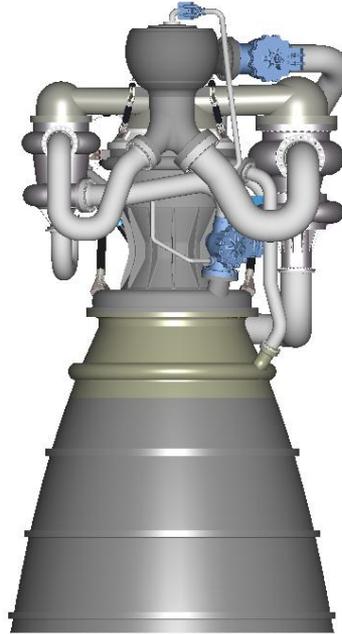
	RD-180	RS-84 (*)	F-1
Cycle	ORSC	ORSC	GG
T (sl/vac, Klbf)	860 / 934	1050 / 1168	1522 / 1748
I_{sp} (sl/vac, secs)	311.9 / 338.4	301.5 / 335.4	265.4 / 304.1
P_c (psia)	3722	2800	982
ε	36.4	36	16
W (lbm)	12225	15985	18616
T/W (sl/vac)	70.3 / 76.4	65.7 / 73.1	82 / 94
MR	2.72	2.70	2.27
Envelope (L/OD, inches)	146 / 124	181 / 108	220 / 143
Life (starts)	10 (20 w refurb at 10)	40 (100 goal)	20
Reliability	0.998 @ 95% confidence	>0.99990	0.990 @ 96.6 confidence

Module 5:

Subsystems & Components

Oxidizer Preburner

LPOTP

HPOTP

SSME Main Injector

Fuel Preburner

LPFTP

Hot Gas Manifold

Controller

HPFTP

Propellant Valves

Hydraulic Actuators

MCC

Nozzle

Objective

Continue the overview discussion of liquid propellant rocket engines to focus on the different components used to form the engine system.

Subsystems and Components

- Turbomachinery

- Combustion Devices

- Valves & Actuators

- Engine Control System

- Ancillary Systems

- Engine Integration Hardware

Turbomachinery

J-2 Mk-15F FTP (LH_2)

RD-0120 OBP (LO_2)

RD-107 TPA (LO_2/RG-1)

Oxidizer Inlet

Fuel Inlet

GN$_2$

LN$_2$ Heat Exchanger

Steam Inlet

Turbine Exhaust

LN$_2$

Thermal Standoff / Indexing Key

H$_2$O$_2$ Inlet

H$_2$O$_2$ Discharge

Fuel Discharge

Oxidizer Discharge
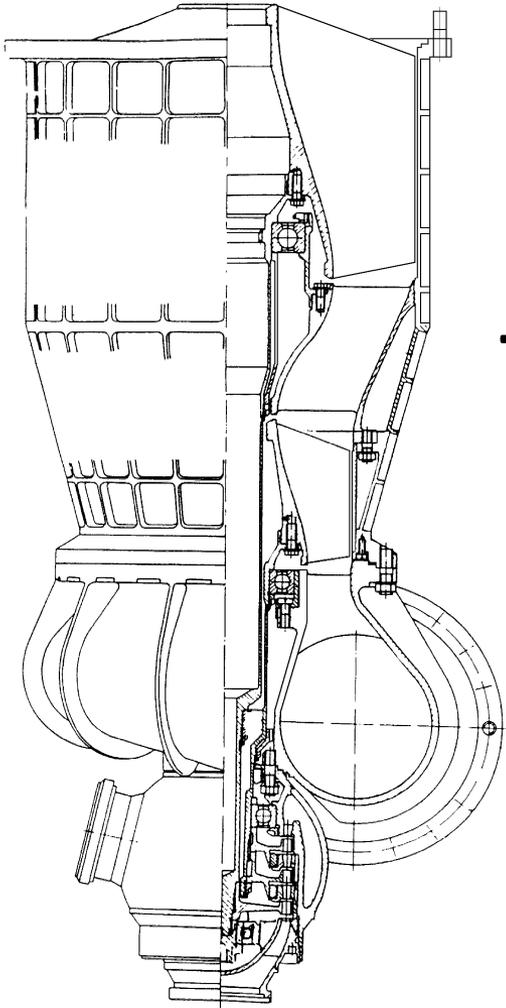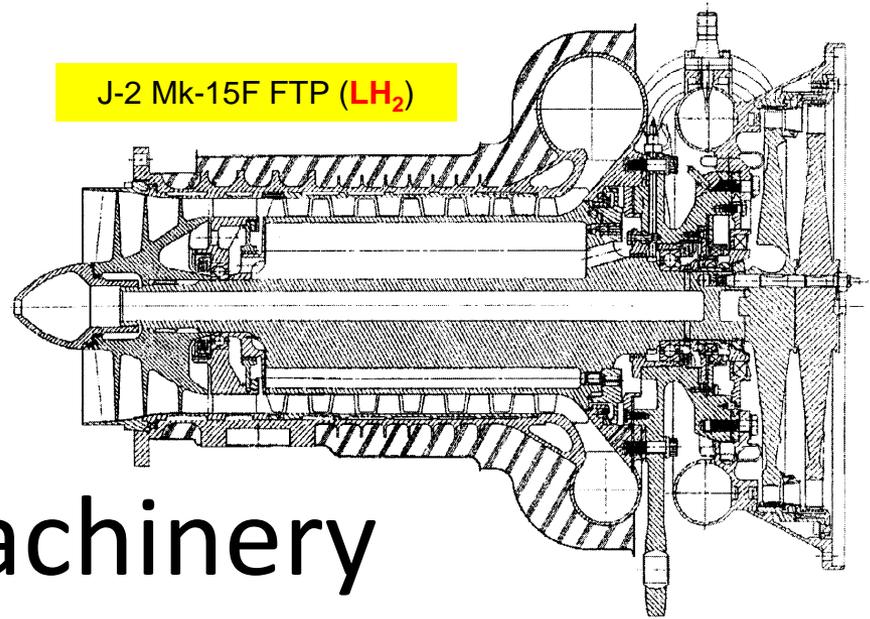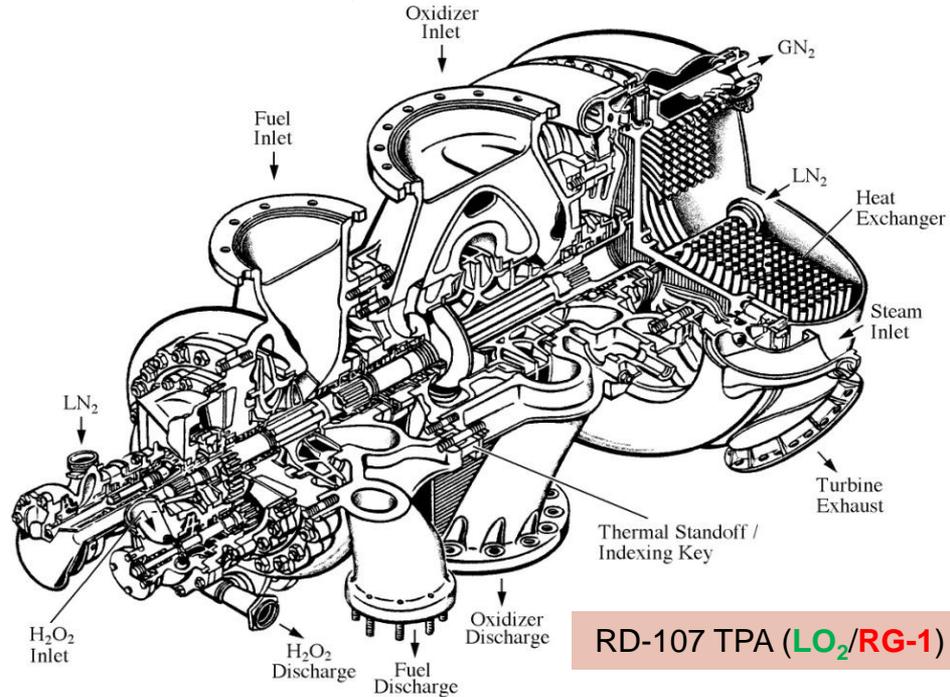
Turbomachinery

- The function of the rocket engine turbopump is to receive the liquid propellants from the vehicle tanks at low pressure and supply them to the combustion chamber at the required flow rate and injection pressure.

- The energy to power the turbine itself is provided by the expansion of high pressure gases, which are usually mixtures of the propellants being pumped.

- Turbopumps are primarily separated into the pump section and the turbine section, separated by a seal package and with the rotor supported by bearings.

- Depending on the system, the turbomachinery can be
  - in separate units for the fuel and oxidizer, or
  - together in an integrated housing, with the pumps on a single shaft with a turbine, or with the pumps on separate shafts connected by a gearbox.
  - Have boost pumps upstream of the main pumps.

- Examples
  - SSME / RS-25
    - LPFTP
    - LPOTP
    - HPFTP
    - HPOTP
  - J-2X
    - FTP
    - OTP
  - RD-0120
    - FBP
    - OBP
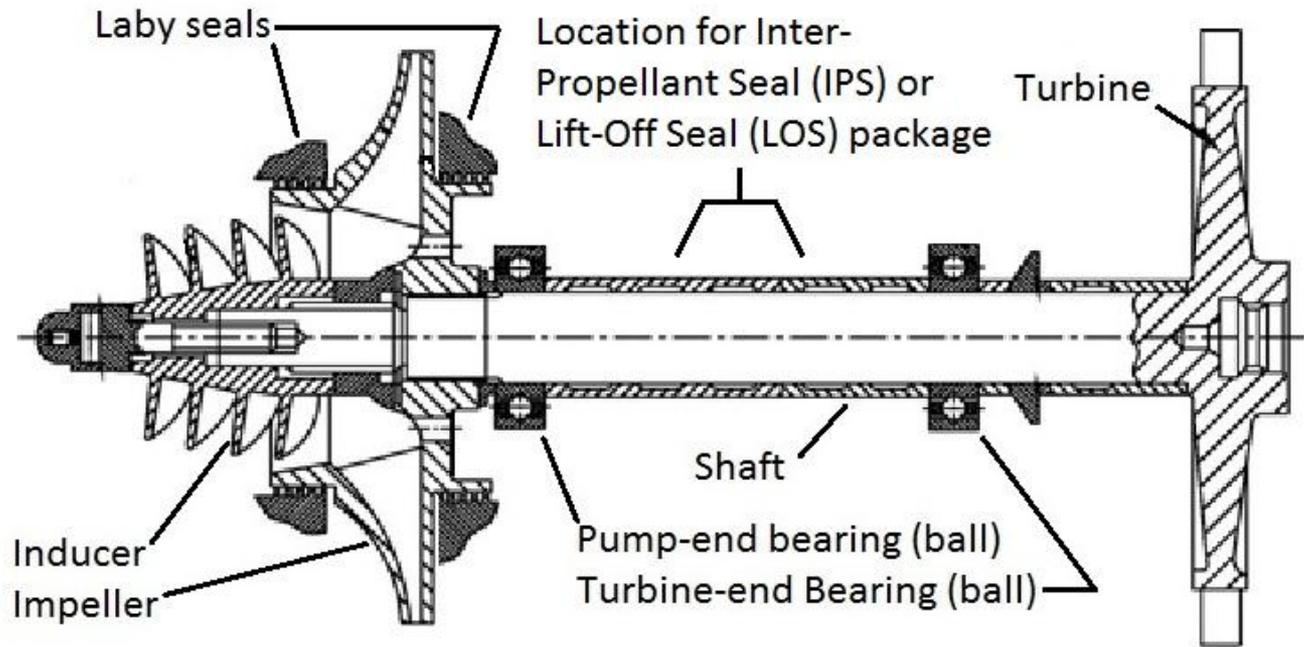   - MTU

Power Density

Horsepower to Weight Comparison

High Power Density Comparison of Automobile Engines, Jet Engine, and SSME High Pressure Pumps

SSME Fuel Pump
775 lb
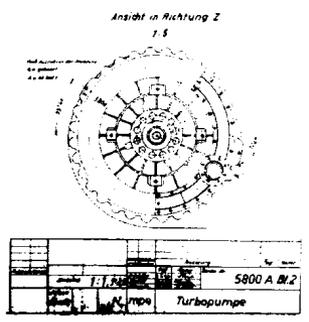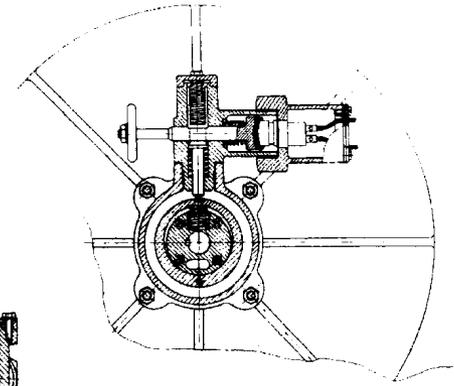72,000 HP
HP/Wt = 93

SSME LOX Pump
575 lb
29,000 HP
HP/Wt = 50

Jet Engine
2,800 lb
52,900 HP
HP/Wt = 18.9

Indy Engine
275 lb
800 HP
HP/Wt = 2.91

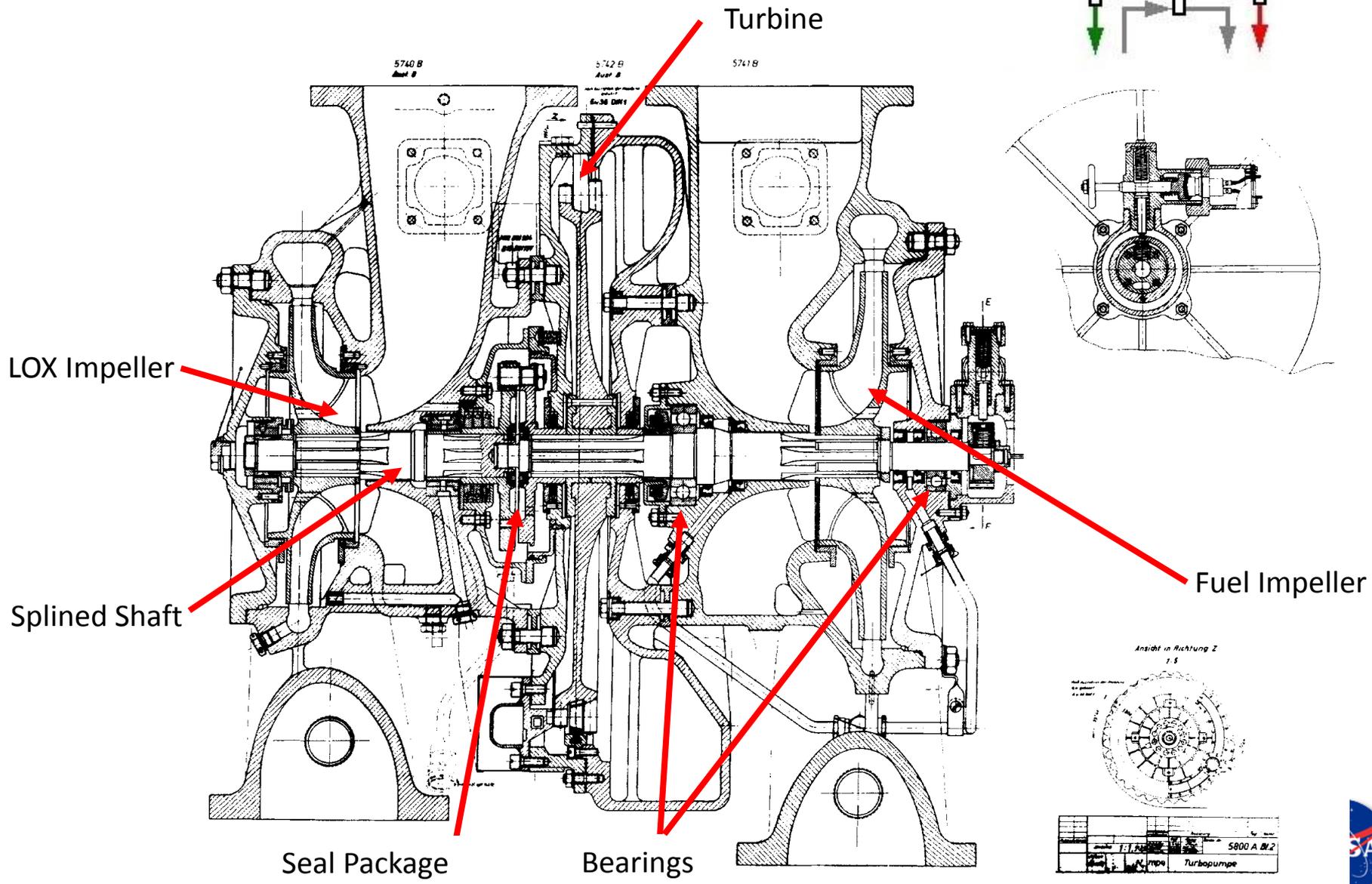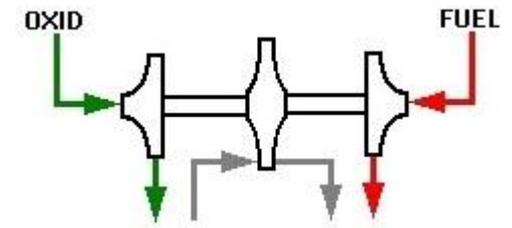Auto Engine
370 lb
200 HP
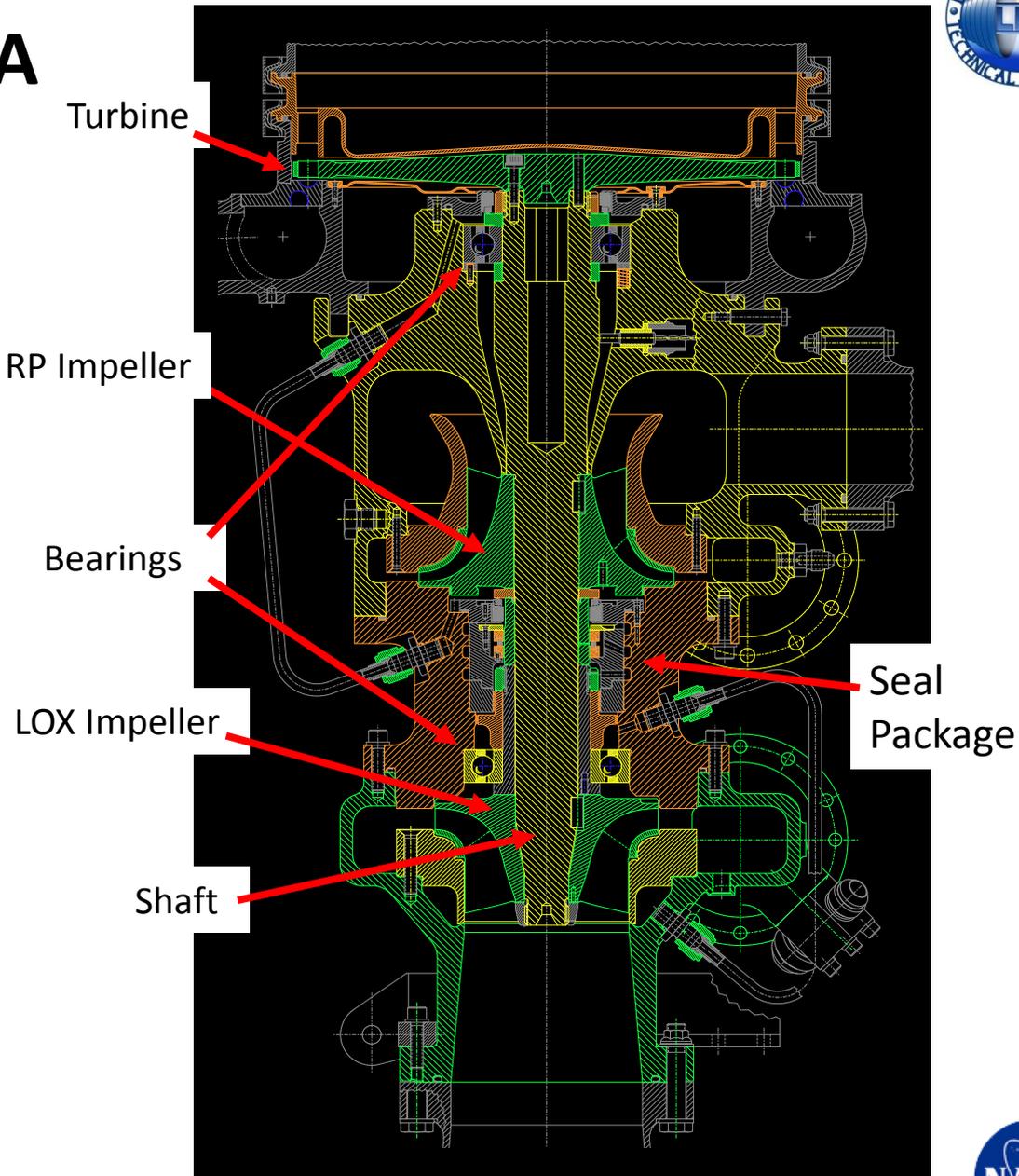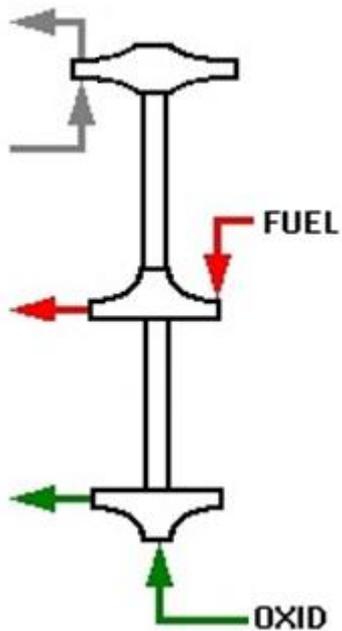HP/Wt = .54

HP/Wt

100
80
60
40
20
0
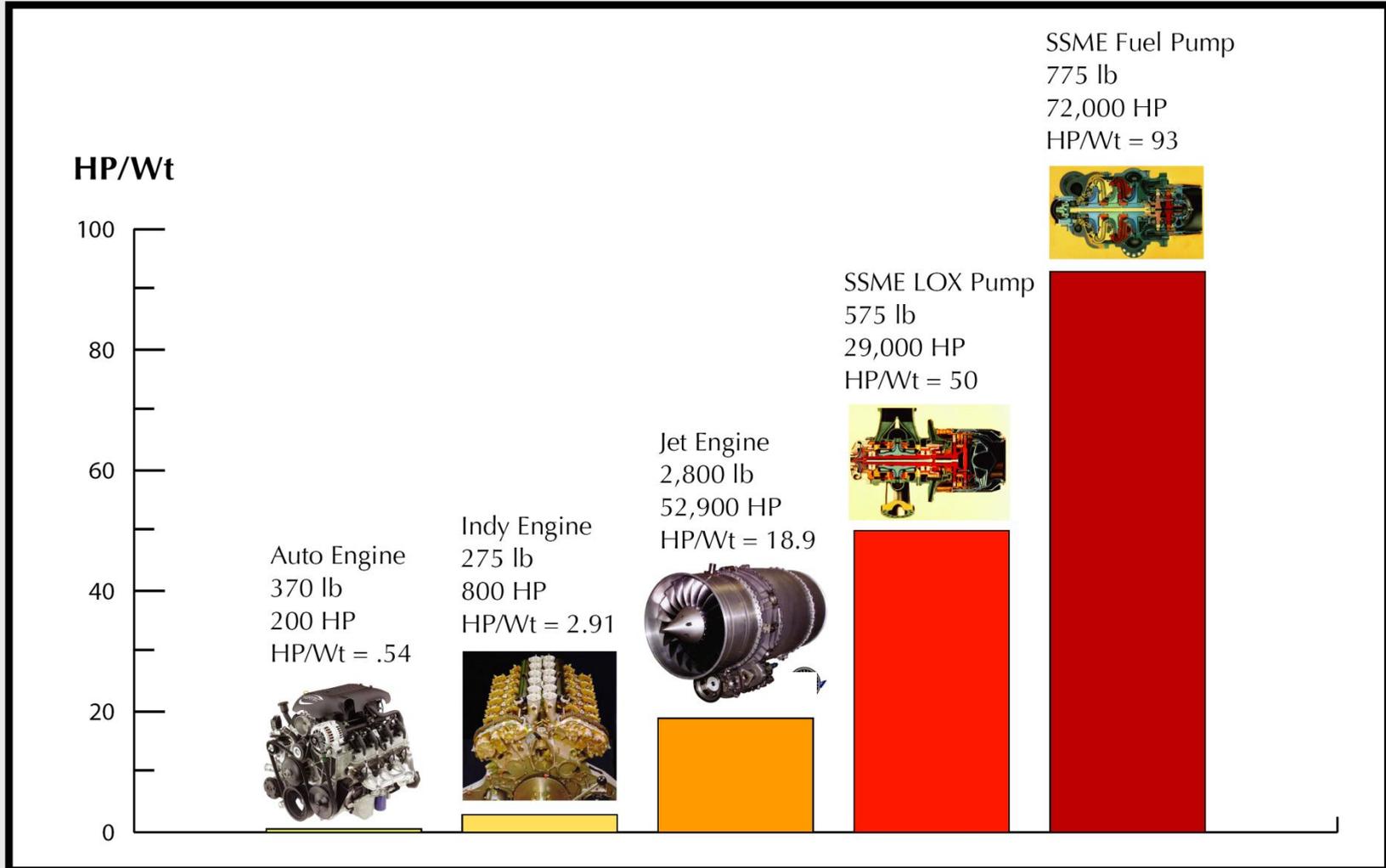
Turbopump Anatomy

- Turbopumps have most of the same basic types of parts. Almost all turbopumps currently in use rely on centrifugal impellers
  - Housing / volute (not shown below)
  - Impeller
  - Inducer
  - Shaft
  - Turbine
 - Bearings
  - Seals

A-4 TPA (ca. 1942)

OXID

FUEL

Turbine

LOX Impeller

Splined Shaft

Fuel Impeller

Seal Package

Bearings

MC-1 TPA

Turbine

RP Impeller

Bearings

LOX Impeller

Shaft

Seal Package

FUEL

OXID

Bearings

- Bearing packages are used to securely support the turbopump rotor as it rotates at operational speeds. They are designed to tolerate radial and axial loads.
  - Ball – most commonly used
  - Roller – fewer, but with higher radial load capacity
 - Hydrostatic – useful with fuel turbopumps
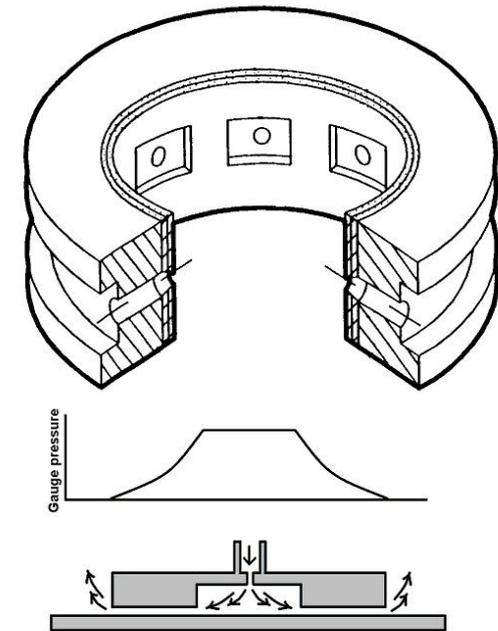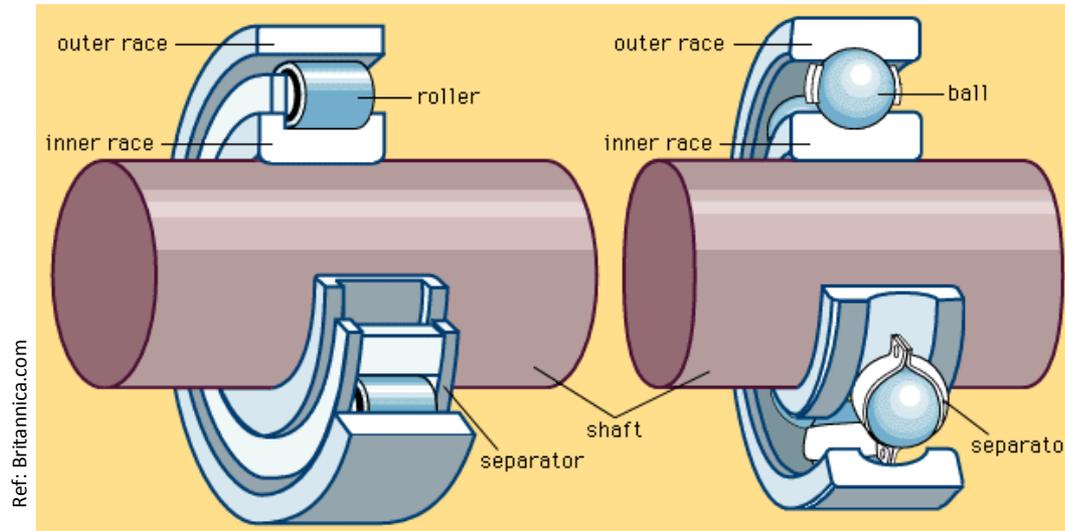
Ref: Britannica.com

Ref: Wikipedia.com & U.S. Patent Office

Seals

- Seals are internal assemblies that are essential to prevent uncontrolled mixing of fluids (propellants, hot turbine gas) in the turbopump.

- Depending on the fluids being separated, there are multiple options available. More common examples include:

  - *Labyrinth "laby" seal* - fluorocarbon seals that contact ridges around the impeller face and control the amount of propellant flow around the impeller shrouds.

  - *Lift-off seal* – used in LH2 TPA and acts as a check valve permitting flow in only one direction. It prevents LH2 leakage into the turbine end of the turbopump prior to engine start and after engine cutoff. During engine start, a pressure unbalance develops across the seal to offset the spring load and retract the seal, allowing liquid hydrogen to enter and cool the turbine end.

  - *Intermediate seal* – uses a barrier purge in oxidizer TPAs to prevent internal mixing of the oxidizer being pumped with the hot fuel-rich turbine drive gas.

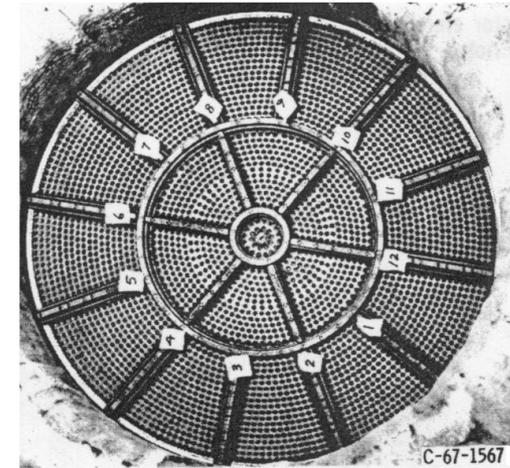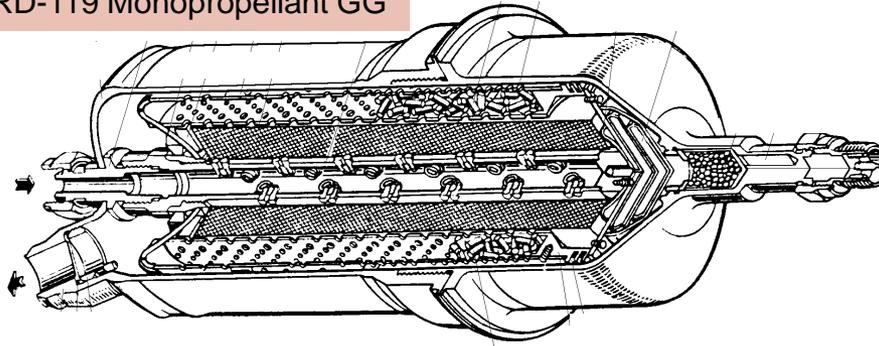Ref: Answers.com

Labyrinth "Laby" seal
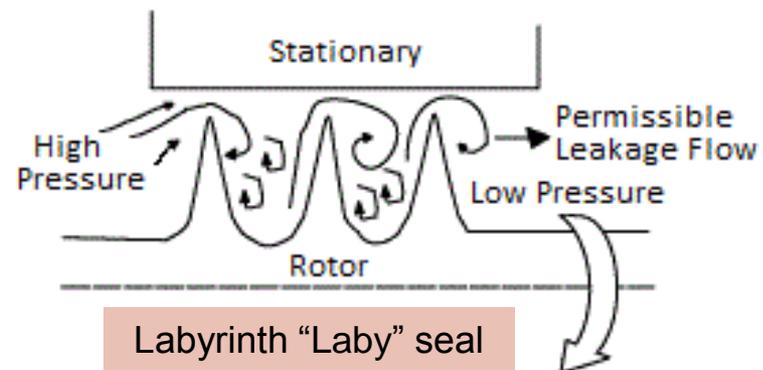
Turbo Concerns - Cavitation

- What is cavitation?
  - The formation of bubbles in the propellant flow when the static pressure in the fluid flow becomes less than the fluid's vapor pressure.
  - Cavitation can occur when the pump inlet pressure is incompatible (lower) with the rotational speed and suction performance of the inducer.

- Why is cavitation bad? What can happen?
  - *Immediate drop-off of exit flow and pressure* – can cause LOX-rich excursion (high MR) if fuel pump excessively cavitates. LOX pump cavitation is less severe and will cause the engine system to power down.
  - *Elimination of hydraulic brake to turbopump rotor* – can cause overspeed and possible rotor burst.
  - *Material erosion of inducer impeller* – hardware damage and progressive degradation of pump performance.

- How is cavitation prevented?
  - Design using experience and validated analytical codes.
  - Extensive testing and data analysis of the inducer/impeller at the part, component, subsystem and system level to validate design and identify the "no-fly" zones.
  - Post-test inspections of inducer/impeller surfaces to check for material degradation.
  - Redlines on the propellant inlet pressure to shutdown the engine.
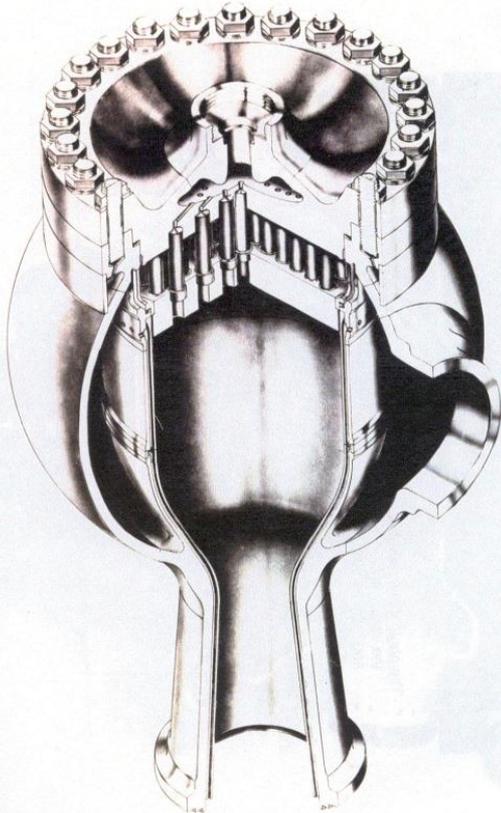
Turbo Concerns – Rotordynamic Instability

- What is rotordynamic instability?
  - Can occur when the rotor exhibits an excessive rotational imbalance that exceeds the ability of the bearings to control. Influenced by the stiffness of the bearings, the operational speed of the turbopump and the balance of the rotor.

- Why is rotordynamic instability bad? What can happen?
  - *Rapid degradation of seals and bearings* – accelerated increase in instability. Can liberate bearing and seal material into propellant flow, causing additional damage.
  - *Rubbing between stationary and rotating parts* – can cause friction ignition.

- How is rotordynamic instability prevented?
  - Design using experience and validated analytical codes with emphasis on rotor design and bearing selection.
  - Progressive balance testing and adjustment during assembly
  - Redlines on the turbopump vibration measured by accelerometers to shutdown the engine during testing. Accel redlines typically not used on flight to prevent false cut.

Combustion Devices
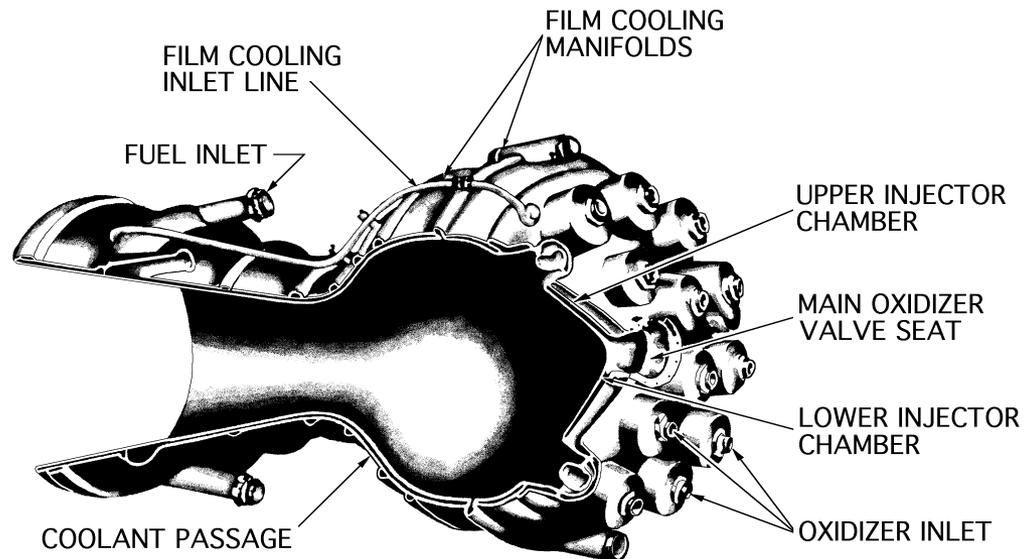

RD-119 Monopropellant GG

M-1 Main Injector

RD-170 PB

FILM COOLING INLET LINE

FILM COOLING MANIFOLDS

FUEL INLET

UPPER INJECTOR CHAMBER

MAIN OXIDIZER VALVE SEAT

LOWER INJECTOR CHAMBER

OXIDIZER INLET

COOLANT PASSAGE

A-4 TCA

5.14

Combustion Devices

- Combustion Devices are components responsible for the production and control of the combustion process.
  - GG (Gas Generator)
  - MCC (Main Combustion Assembly)
  - Nozzle
  - NE (Nozzle Extension)
  - Preburner
  - TCA (Thrust Chamber Assembly)
- Start & Ignition System

- Examples
  - SSME / RS-25
    - FPB
    - OPB
    - MCC
    - Nozzle
  - J-2X
    - GG
    - MCE
    - Nozzle
    - Nozzle Ext.

TCA's and MCC's

The difference between a TCA and MCC is fairly subjective, but the TCA can be regarded as the integrated assembly of the MCC, MINJ and nozzle.
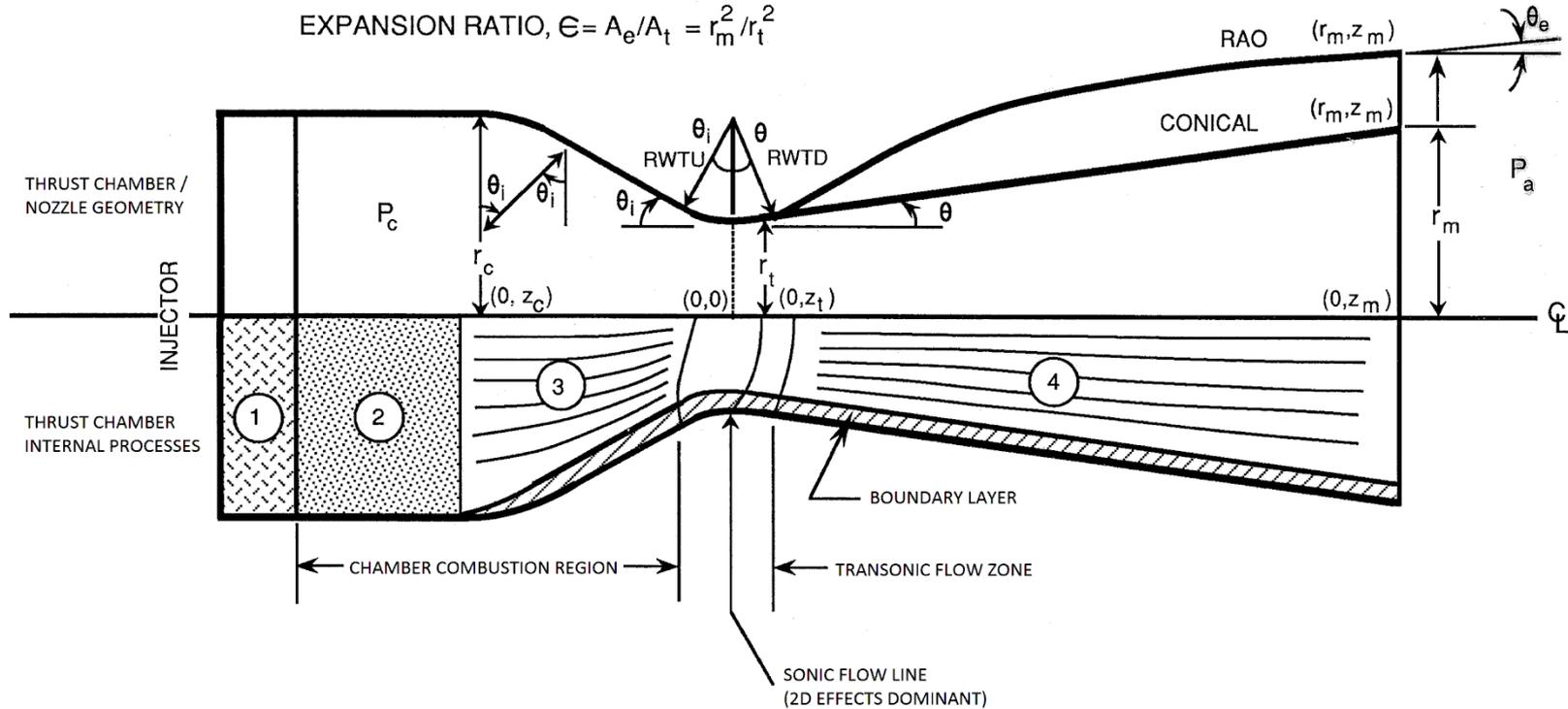

A-4 and Redstone TCAs
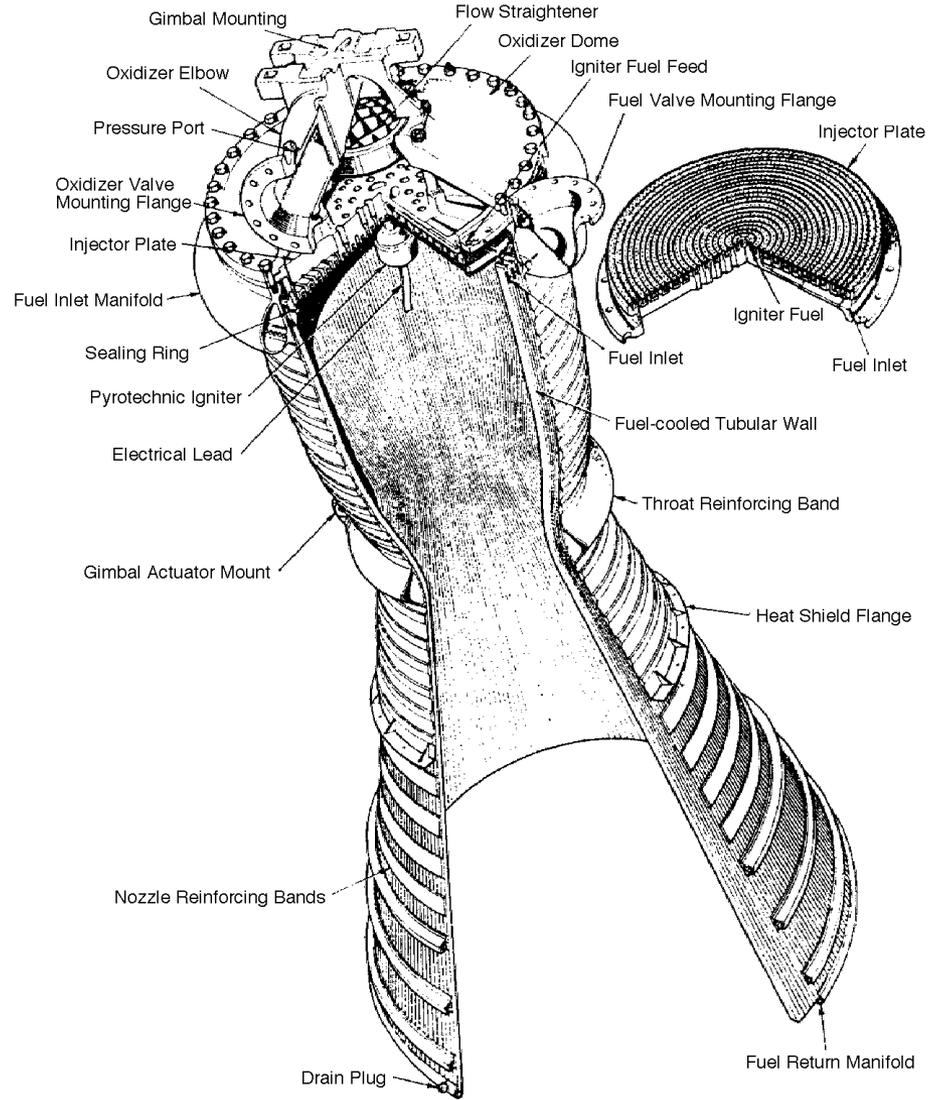
Rolls Royce RZ2 TCA

Thrust Chamber Geometry & Internal Processes

CONTRACTION RATIO, $\epsilon = A_c/A_t = r_c^2/r_t^2$

EXPANSION RATIO, $\epsilon = A_e/A_t = r_m^2/r_t^2$

THRUST CHAMBER / NOZZLE GEOMETRY

THRUST CHAMBER INTERNAL PROCESSES

INJECTOR

RAO (r_m, z_m)

CONICAL (r_m, z_m)

P_a

P_c

θ_i θ RWTU RWTD

$(0, z_c)$ $(0,0)$ $(0,z_t)$ $(0,z_m)$ C_L

BOUNDARY LAYER

CHAMBER COMBUSTION REGION TRANSONIC FLOW ZONE

SONIC FLOW LINE (2D EFFECTS DOMINANT)

1. **INJECTION ATOMIZATION ZONE** (MIXING EFFECTS DOMINANT)

2. **RAPID COMBUSTION ZONE** (MIXING & VAPORIZATION EFFECTS DOMINANT)

3. **STREAM TUBE COMBUSTION ZONE** (VAPORIZATION EFFECTS DOMINANT; BOUNDARY LAYER ASSUMPTION BECOMES VALID)

4. **SUPERSONIC EXPANSION ZONE** (TWO-DIMENSIONAL, REACTION KINETIC, AND BOUNDARY LAYER ASSUMPTIONS BECOME VALID)
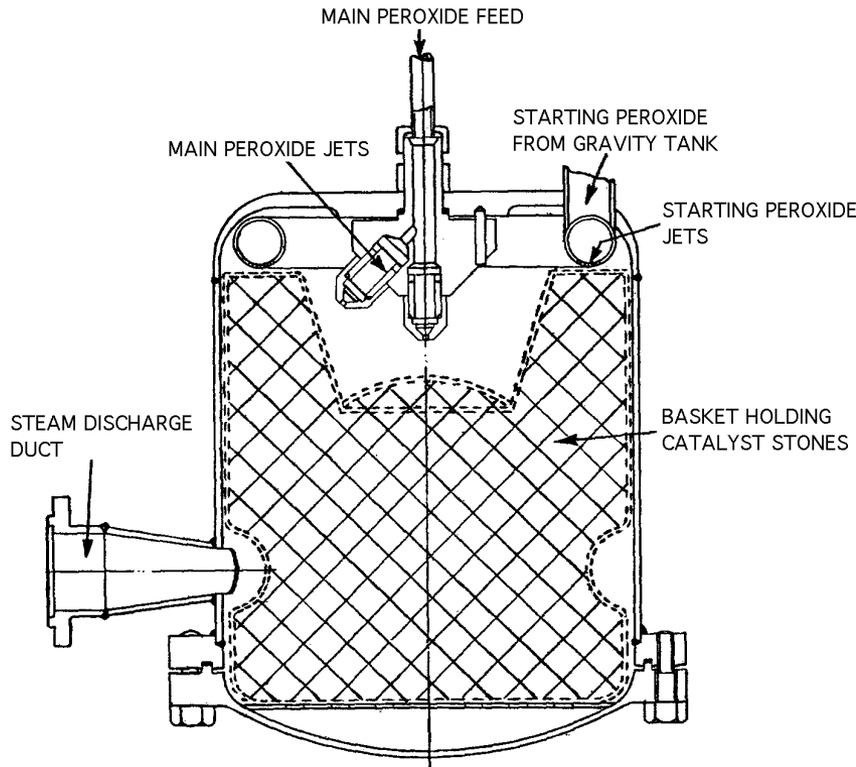
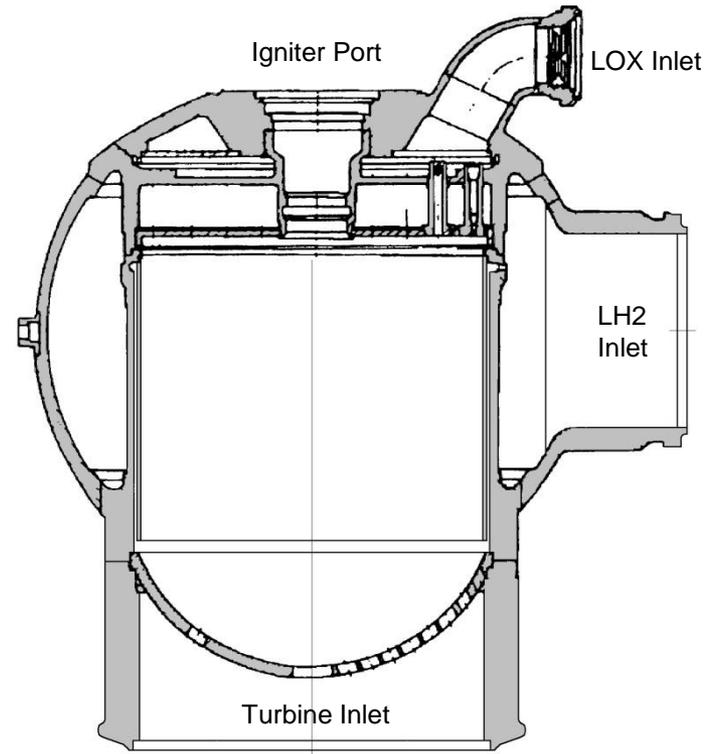η_{ER} η_{ER} $\eta_{ER}, \Delta F_{BL}$ $\eta_{2D}, \eta_{KIN}, \Delta F_{BL}$

5.17

GG's and PB's

Gas Generators or Preburners generate hot gas to drive the turbines, either by tapping off and combusting the main propellants or decomposing a third monopropellant. They are similar in many respects to MCCs or TCAs, only on a smaller scale

Typical Monopropellant GG Component Layout Shown with 109-509A2 GG Cross Section

Typical PB Component Layout Shown with RD-0120 PB Cross Section
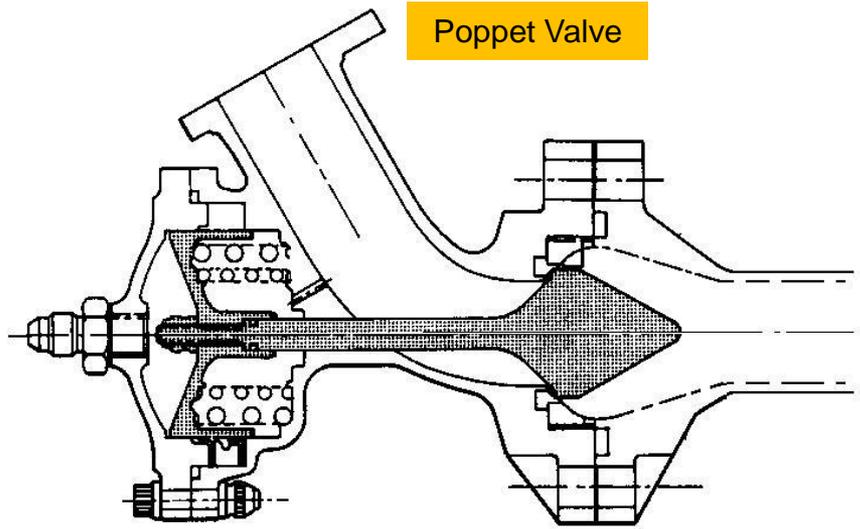
5.18

CD Concerns – Combustion Instability

- What is combustion instability?
  - Occurs when different factors in the vehicle and engine systems trigger pressure oscillations of various frequencies and amplitudes.
  - Depending on the frequency of pressure oscillations, the instability is classified into three categories
    - *Chugging* (10-400 Hz) – Linked to interactions with the propellant feed system or vehicle and the combustion chamber.
    - *Buzzing* (400-1000 Hz) – Linked to interactions with the engine structure, mixture ratio, and propellant feed system resonances
    - *Screaming* (>1000 Hz) – Linked to the combustion process and acoustic resonance properties in the combustion chamber.

- Why is combustion instability bad? What can happen?
  - *Excessive pressure vibration* – Accelerated LCF/HCF damage, especially if the oscillations couple with the engine or vehicle structure.
  - *Excessive heat transfer* – Material degradation and hot-wall erosion.

- How is combustion instability prevented?
  - Baffles
  - Acoustic cavities
  - Bomb testing

A thorough explanation of combustion instability can be found on the 1 December 2011 J-2X blog written by W.D. Greene at "http://blogs.nasa.gov/cm/newui/blog/viewpostlist.jsp?blogname=J2X"
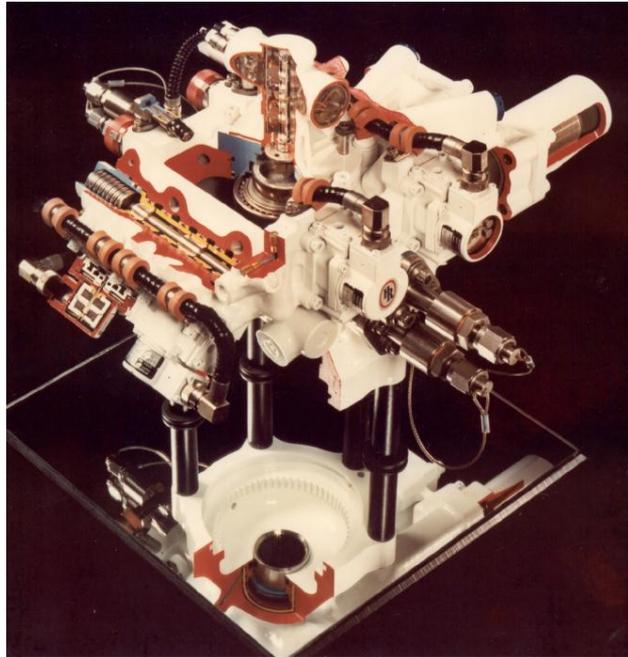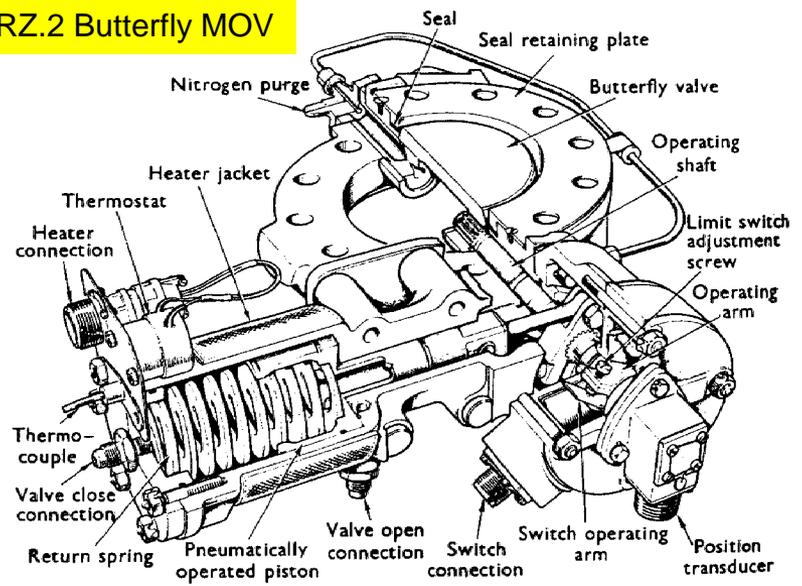
Start & Ignition

- The *start system* provides the initial pressure impulse to the system to bootstrap the power cycle.
  - Tank head – RS-25, RL10
  - GHe spin – J-2X
  - Solid propellant GG (SPGG) – J-2S

- The *ignition system* provides the energy pulse required to initiate combustion. Each combustion device in the system generally requires an independent ignition system as the timing is usually different within the start transient. The required pulse level depends on the propellants.
  - Spark –most LO_2/LH_2 engines
  - Pyrotechnic – used largely in GGs where an energy pulse is adequate
  - Hypergolic (TEA/TEB) – most LO_2/RP-1 engines
  - Laser – relatively low TRL, but shows promise in terms of flexibility and control.
  - Fluorine injection – hypergolic with hydrogen

TEA/TEB – Triethyl Aluminum/Triethyl Boron (a storable fuel blend that is hypergolic with oxygen)

Poppet Valve

Valves & Actuators

RZ.2 Butterfly MOV

Seal
Seal retaining plate
Nitrogen purge
Butterfly valve
Heater jacket
Operating shaft
Thermostat
Limit switch adjustment screw
Heater connection
Operating arm
Thermo-couple
Valve close connection
Return spring
Pneumatically operated piston
Valve open connection
Switch connection
Switch operating arm
Position transducer
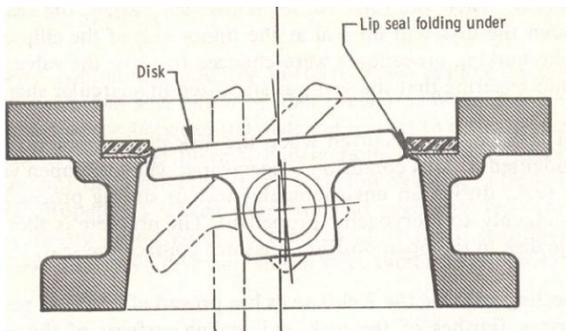
Valves & Actuators

- Valves
  - Main Propellant Valves (MOV, MFV)
  - Secondary Propellant Valves
    - GG Valves (GGFV, GGOV)
    - Preburner Valves (FPOV, OPOV)
    - Chamber Coolant Valve (CCV)
  - Hot Gas Valves (OTBV)
  - Ancillary Valves
    - Purge Valves
    - Check Valves
    - Solenoid Valves
- Actuators
  - Pneumatic/Hydraulic
  - Electro-mechanical

- Examples
  - SSME / RS-25
    - MFV
    - MOV
    - FPOV
    - OPOV
    - CCV
  - J-2X
    - MFV
    - MOV
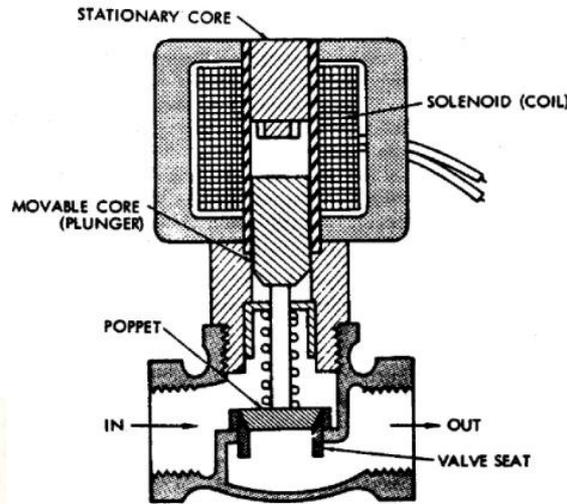   - GGFV
    - GGOV
    - OTBV

Valves

- Valves are used throughout all engine systems to control and direct propellants, pressurants, and other working fluids to the various engine elements and are considered an integral part of the engine system.

- As part of engine cycle selection trade studies, the complexity and number of valves are considered.

- Extremes in pressure, temperature, vibration, and other system / cycle induced conditions are considered and included in the engine trade studies.
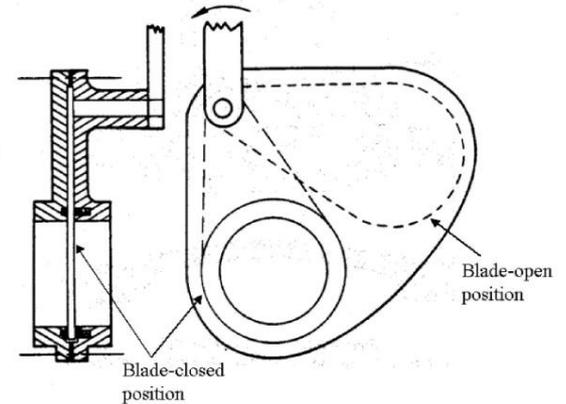
- Valve Types
    - Ball Valve
    - Butterfly Valve
    - Blade Valve
    - Sleeve Valve
    - Poppet Valve
    - Check Valve
    - Pressure Actuated Valve

Poppet valve with solenoid actuator

Blade valve - consists of a flat disc that is fixed perpendicular to the flow stream or duct axis.

Butterfly valve showing the lip seal and disc rotation

Actuators

- Most valves require an operator or actuator.

- Types of actuators
  - Solenoid
  - Hydraulic
  - Pneumatic
  - Electromechanical

- Design Considerations
  - Fluid Medium
  - Fluid Temperature
  - Fluid Pressure
  - Flow Capacity
  - Flow Characteristics
  - Leakage
  - Operating Life
  - Control Requirements
  - Loads and Environments

Main propellant valve hydraulic actuator for SSME

MOTOR
3 PHASE
BRUSHLESS DC

RESOLVER
SINGLE SPEED
BRUSHLESS

OUTPUT SPLINE

HARMONIC DRIVE
120 : 1
DOUBLE ECCENTRIC
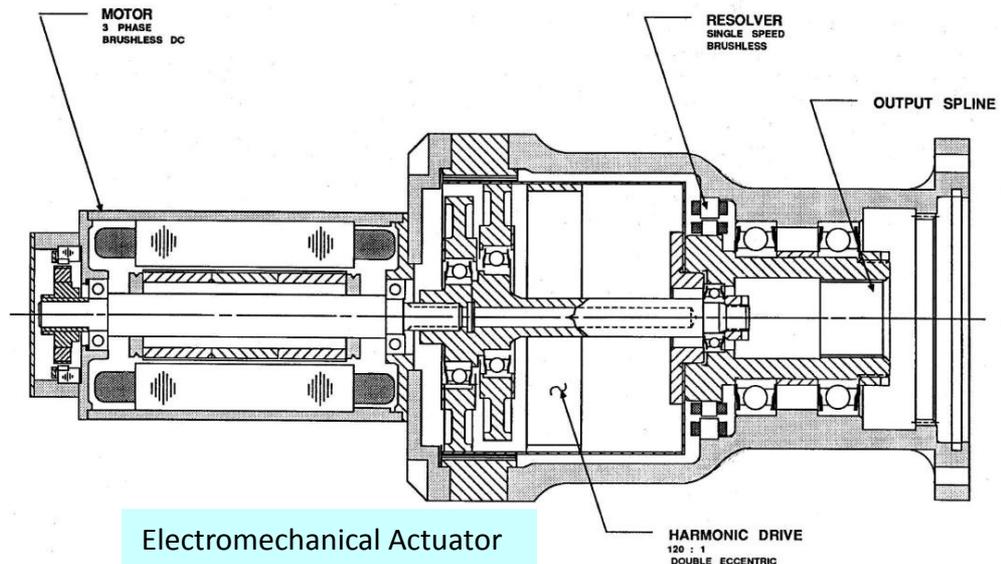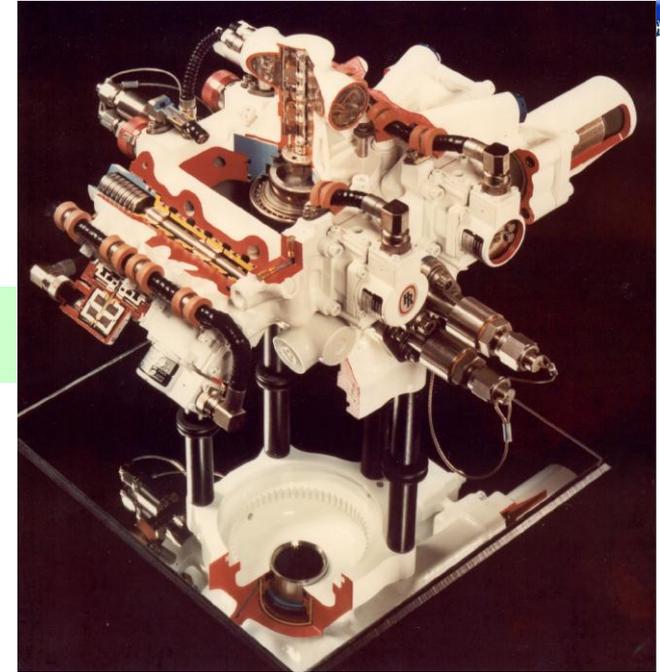
Electromechanical Actuator

Control System

Engine Control System (J-2X)

- Control of the J-2X engine is accomplished physically by opening valves that control the flow of propellants to the combustion devices along with properly timed ignition and pressurized helium spin to initiate engine start, and by closing the valves to stop propellant flow to shutdown the engine.

- The control system consists of components required to perform the functions of starting and shutting down the engine. Four subsystems work together to accomplish these functions:
  - engine control subsystem
  - propellant valves subsystem
  - ancillary valves subsystem
  - pneumatic subsystem

- The engine control system has <u>open</u> loop control authority over the engine. In other words, the J-2X control system does not have a feedback (i.e., <u>closed</u>) loop like RS-25 to control and self correct the positions of the propellant valves or engine parameters, like chamber pressure or engine thrust. The engine operating level (thrust, mixture ratio) is controlled by orifices located throughout the engine.

Engine Control System (J-2X)

- The J-2X engine control system is comprised of the following components:
  - Dual Channel Engine Control Unit (ECU)
  - Main Injector Exciter Unit (MIEU)
  - Pneumatic Actuator Control Assembly (PACA)
  - Pneumatic Purge Control Assembly (PPCA)
  - Propellant Valves
  - Ancillary System Valves
  - Health Monitoring Instrumentation
  - Critical Instrumentation
- The engine control system performs engine _control_ functions:
  - open loop control of the five main propellant valves via pneumatic solenoids
  - sequence control of the ancillary valves for purge, bleed, and engine ignition
  - sequence control of two GG pyrotechnic devices and main injector spark exciter for engine ignition
  - provide engine initiated shutdown status to the vehicle

Engine Control System (J-2X)

- The engine control system performs engine _health monitoring_ functions:
  - monitor operational flight sensors for engine status and health conditions
  - perform commanded engine checkout
  - provide operational flight sensor data to vehicle

- The Engine Controller Unit (ECU) receives a command to initiate engine start from the vehicle. The ECU actuates propellant valves using a solenoid driver, which subsequently opens a solenoid to pressurize the pneumatic actuator on the propellant valve. The ECU also activates a spark exciter which is used in the ASI system to start ignition in the main chamber and pyrotechnic igniters to start ignition in the GG. The ancillary valves participate by controlling functions such as purge and bleed.

Ancillary Systems

Ancillary Systems

- Major system, subsystem and components have been shown to accomplish the primary engine functions. In addition, LREs also include ancillary functionality in support of the major engine functions. These functions are accomplished by the complex plumbing of minor fluid lines, which perform the following:
    - Pre-start conditioning of the engine, facilitated by the <u>bleed system</u>
    - Supplying pressurants to the vehicle for tank pressurization, facilitated by the <u>tank pressurization system</u>
    - Engine start, facilitated by the <u>helium spin start system</u> and the <u>augmented spark ignition system</u>
    - Valve actuation, facilitated by the <u>pneumatic actuation system</u>
    - Engine purging, facilitated by the <u>purge system</u>
    - Routing of propellant leakage, facilitated by the <u>drain system</u>

Engine Integration Hardware

Engine Integration Hardware

- HEX (Heat Exchanger)
- Pogo Suppressor
- Gimbal
- Brackets
- Rigid Ducts
- Flex Ducts
- Small lines (bleeds, purges)

Backup

Additional References

- General
  - NASA-SP 8000 series: ~20 publically-available documents developed in the 1970's that provide detailed insight into most LRE components. For example:
    - NASA SP-8109, "Liquid Rocket Engine Centrifugal Flow Turbopumps," 1973 http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19740020848_1974020848.pdf

- Turbomachinery
  - Stangeland, M.L., "Turbopumps for Liquid Rocket Engines," PWR Threshold, 1988 http://www.pwrengineering.com/articles/turbopump.htm