Overview of Liquid Propellant Rocket Engine Systems and the J-2X

Lecture for University of Stuttgart
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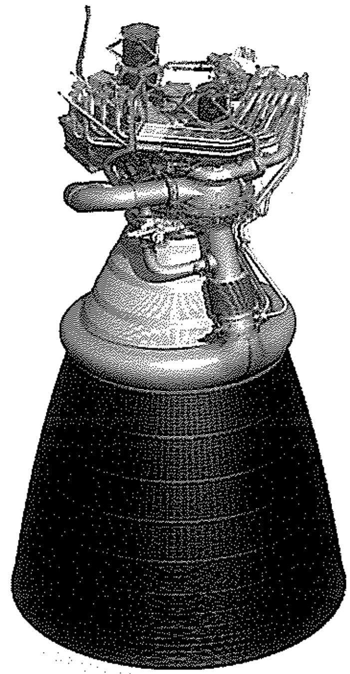

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Objective

- Conduct an introductory discussion to rocket engines using liquid propellants to serve as the foundation for a subsequent discussion on the J-2X upper stage engine under development by Pratt & Whitney Rocketdyne (PWR) for NASA.

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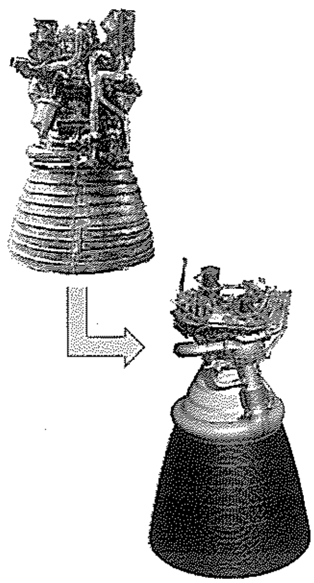
Outline

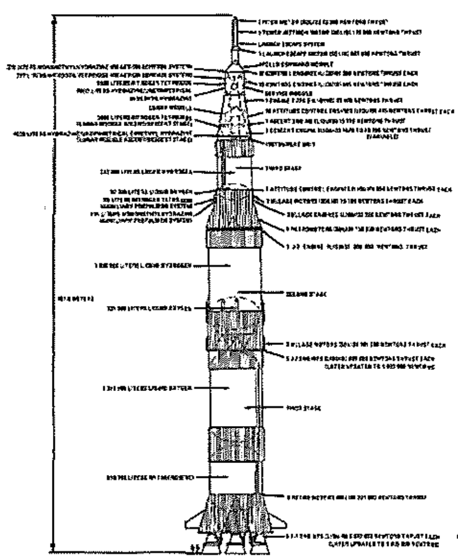


- Liquid Rocket Engine (LRE) Applications
- Liquid Propellants
- LRE Power Cycle Review
- The J-2X System

LRE Applications

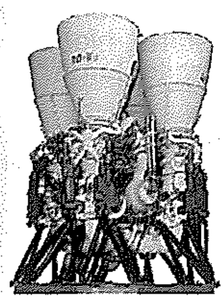
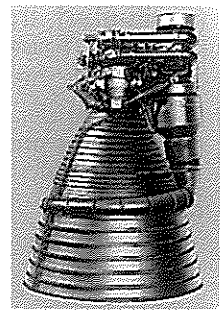
Integrated Launch Vehicle (Saturn-V)

- 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)



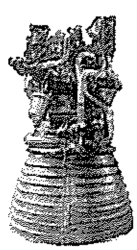
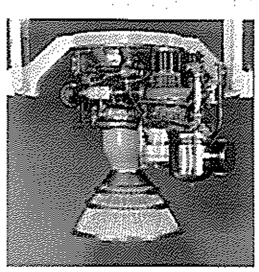
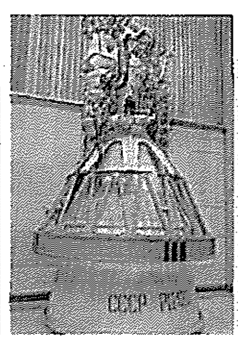
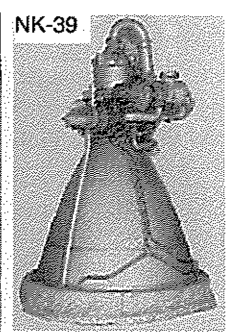
LRE Applications
Booster

- 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

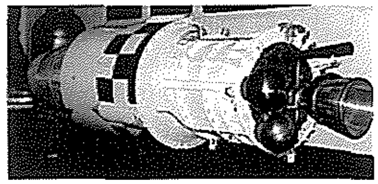
- Provide supplemental impulse for achieving orbit.
- High thrust, but less required than a booster engine
- Higher ε than booster, less than orbital LRE
- Examples
  - J-2
  - LR91 series
  - RD-120
  - NK-39

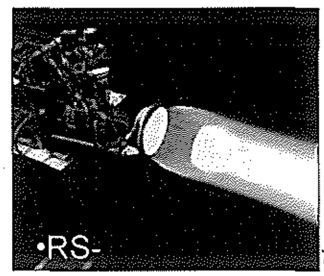





LRE Applications

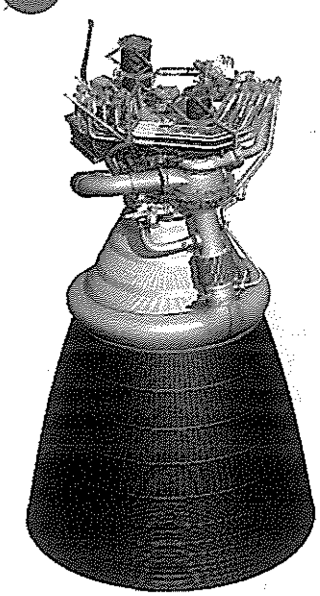
Upper Stage Engine (USE)

- Typically applied for final orbital insertion or modification of orbital parameters (similar to OME)
- Low to medium thrust (10 – 300 Klbf)
  - 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





J-2X Application



- The J-2X continues to be used in the same applications that the heritage J-2 was developed to support for the Saturn-V vehicle.
  - Sustainer / Upper Stage for Ares-1
  - Upper Stage (EDS) for Ares-V

Liquid Propellants

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Terms to Know

Liquid Propellants

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

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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

J-2X

* Also used as a monopropellant

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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

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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 starts).

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, mostly for satellites and space probes
  - 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.

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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)
    - Disadvantage – Not selective about what it uses as fuel. It prefers hydrogen or hydrocarbons, but will consume almost anything with an oxidation potential. For example...

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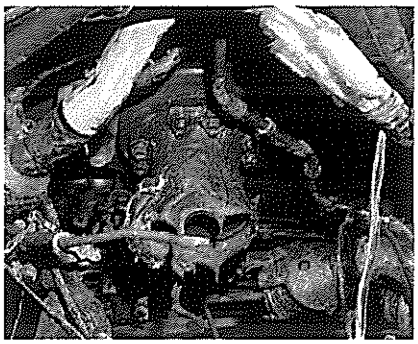
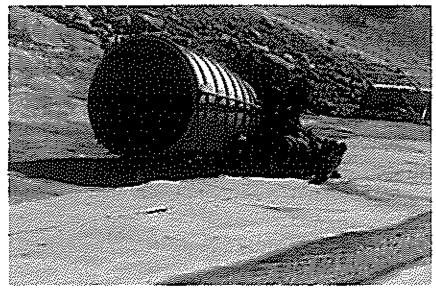
A Bad Day...

On 27 Mar 1985, SSME 2308 suffered a catastrophic failure when a weld failed on a critical fuel line, causing the engine to operate at a LO_2-rich state. Deprived of LH_2 fuel, the LO_2 effectively used the metal (Inconel) of the engine itself as fuel (aka "hardware-rich").





The damage was such that the engine melted off the test stand and the lower half fell down in the flame bucket.

LO_2 is a great oxidizer, but demands respect to what it can do.

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Mixture Ratio

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 mean 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 = ½ mV^2) than a H_2O steam exhaust.

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Density vs. I_{sp}

- 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

Pc = 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

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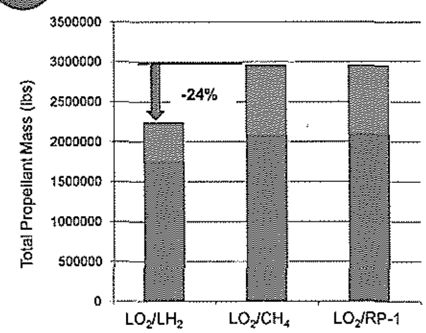
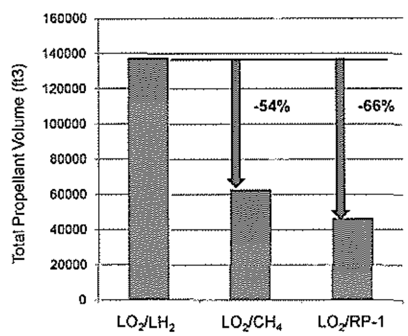
Propellant Mass vs. Volume





- For an impulse requirement similar to the 3 SSME's used on the Shuttle (1.5M lbf 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 of the vehicle.
- Lesson: I_{sp} isn't everything – especially with boost stages.

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LRE Power Cycle Review

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

J-2X

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LRE Cycle Uses & Trades

	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. Easier MR and thrust level throttling characteristics.	High performance (closed cycle). Simplier than mult prebumer 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 diluent 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

- The power "cycle" refers to how energy is generated to power the turbopump(s)
- A number of thermodynamic cycle options exist
- Which one used depends on application or mission requirements
- One cycle is not right for every application

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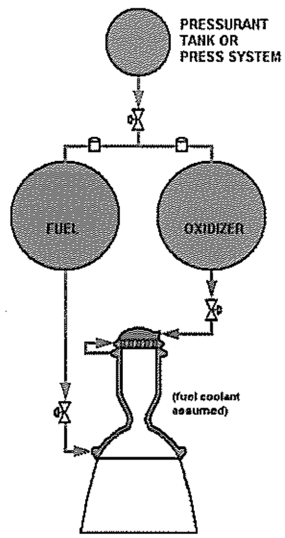
Terms & Acronyms

- FTP (Fuel Turbopump)
- GG (Gas Generator)
- GGFV (Gas Generator Fuel Valve)
- GGOV (Gas Generator Oxidizer Valve)
- HEX (Heat Exchanger)
- MCC (Main Combustion Chamber)
- 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.
- The selection of power cycle can be driven by many factors:
  - Propellants
  - Performance (thrust, specific impulse)
  - Safety / Reliability
  - Reusability
  - Technical Risk
  - Cost / Schedule
  - Etc.

Pressure-Fed "Cycle"



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

* Assumes use of hypergolic propellants

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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

- <u>GG Cycle Variations</u>
- Monopropellant GG
- Bipropellant GG
  - Common-shaft main TPA
  - Separate fuel and oxidizer TPAs
    - Series turbines
    - Parallel turbines

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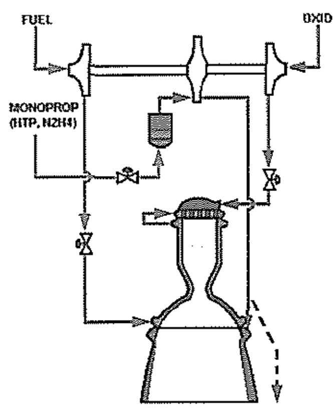
Gas Generator (GG) Cycle

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)

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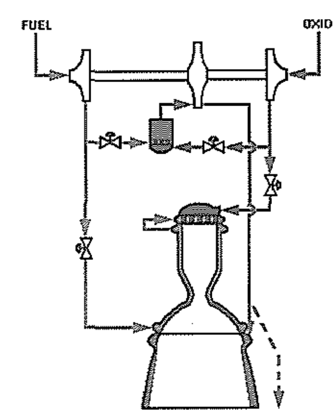
Gas Generator (GG) Cycle

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

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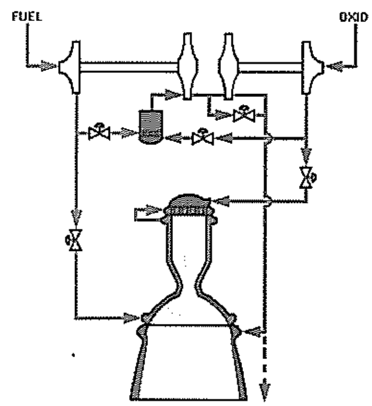
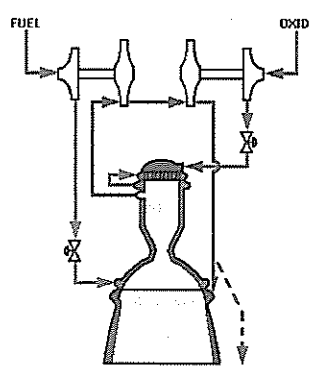
Gas Generator (GG) Cycle

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 → J-2X

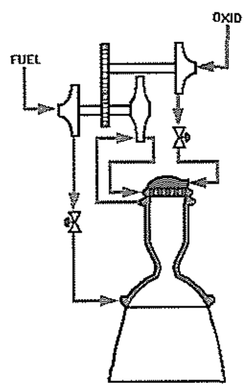
Tapoff Cycle



- 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

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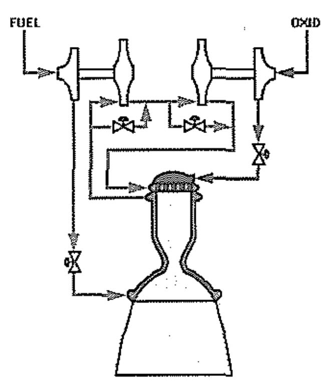
Expander Cycle

Dual TPA



- 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

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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
- Usually requires the use of boost pumps to increase propellant pressure entering main pumps to prevent cavitation

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

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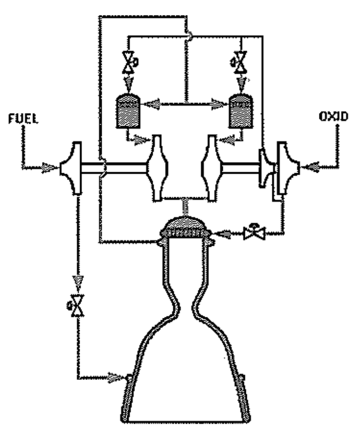
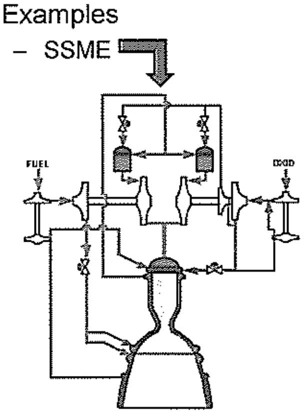
Fuel-Rich SC Cycle

Dual Preburners



- Permits independent MR and thrust level throttleability
- Examples
  - SSME



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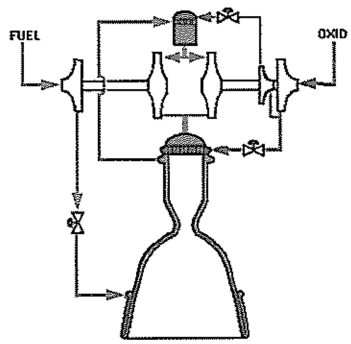
Fuel-Rich SC Cycle

Single Preburner, Dual TPAs



- 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)

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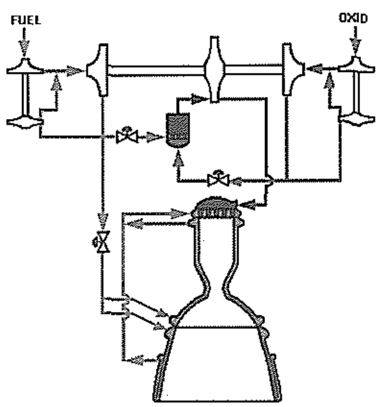
Oxidizer-Rich SC Cycle

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)

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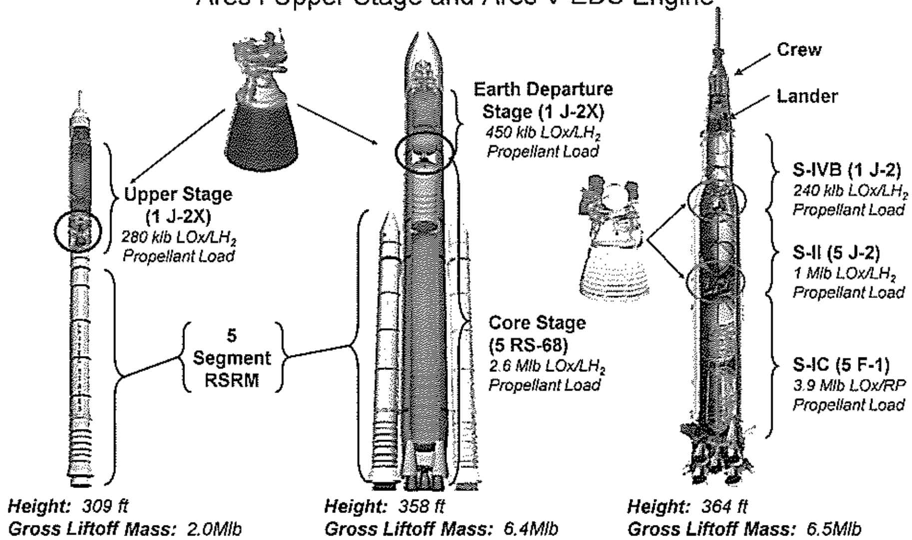
J-2X: Apollo-era Derivative Engine

Ares I Upper Stage and Ares V EDS Engine



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Engine Lineage

J-2X: Adding a New Member to the Family

	1960-1970	*1965-1971*	*1996-2001*	*2006-*
Configuration	J-2	J-2S	X-33	J-2X
Thrust	230 klb	265 klb	261 klb	294 klb
Isp	425 sec	436 sec	419 sec	448 sec
Mass	3,492 lb	3,800 lb	7,500 lb	5,450 lb
Length	116 in	116 in	79 in	185 in

Requirements

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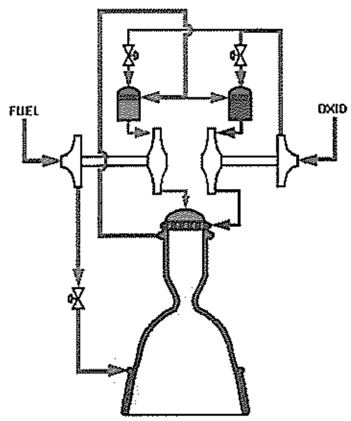
Full-Flow SC Cycle

Two TPAs with Two PBs



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

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The J-2X System

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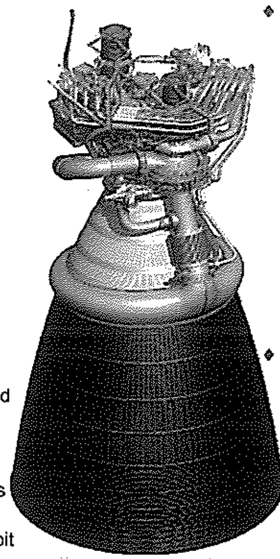
Design Overview

- **Mission**: Common upper stage engine for Ares I and Ares V
- **Development Philosophy**: Evolved hardware and mature technology where possible, aggressive schedule, early risk reduction testing, requirements-driven
- **Key Features**:
  - LOX/LH_2 GG cycle
  - Series turbines with throttle capability through Lox turbine bypass
  - Open loop, pneumatically actuated valves
  - On-board engine controller and health monitoring
  - Tube-wall regen nozzle/large passively-cooled nozzle extension, turbine exhaust gas boost/cooling
  - Helium spin start – with on-orbit restart capability



- **USE Key Requirements**
  - Vacuum Thrust: 294,000 lbf (1307 kN)
  - Specific impulse: 448 sec (min)
  - Mixture ratio: 5.5
  - Run duration: 500 seconds
  - Weight: 5,535 (2,516 kg)
  - Size: 120" dia x 185" long
  - Life: 8 starts / 2600 sec
  - Ares V specific: on-orbit restart, 82% thrust (4.5 mixture ratio)
- **Major Hardware Flow**
  - Production – Pratt & Whitney Rocketdyne, Canoga Park, CA
  - Engine assembly – SSC, MS, Bldg 9101
  - Test – SSC, MS, Stands A1, A2, A3
  - Stage integration – MAF, LA

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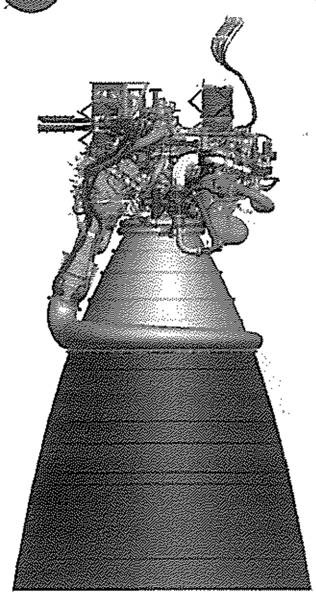
J-2X Basic Statistics



	•English	•Metric
• Cycle	GG	GG
• Thrust, vac	294 klbs	1308 kN
• Isp, vac (min)	448 s	448 s
• Pc	1,337 psia	9.218 MPa
• MR	5.5	5.5
• AR (geometric)	94.4	94.4
• Weight (max)	5,450 lbs	2472 kg
• Secondary Mode MR	4.5	4.5
• Secondary Mode PL	82%	82%
• Restart	1	1
• Service Life Starts	8	8
• Service Life Seconds	2,600 s	2,600 s
• Length (max)	185 in	4.699 m
• Exit Dia. (max)	120 in	3.048 m

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J-2X Major Elements

Gimbal Block
- Based on J-2 & J-2S design
- Potential upgrade to more modern, demonstrated materials

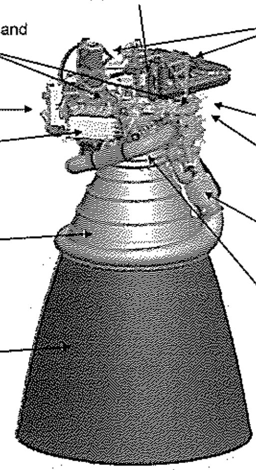
Turbomachinery
- Based on J-2S MK-29 design
- Modified to meet J-2X performance and current design standards

Gas Generator
- Leveraged from RS-68 design

Engine Controller
- Based on directly on RS-68 design and software architecture

Tube-Wall Regeneratively-Cooled Nozzle Section
- Based on long history of RS-27 success (Delta II/III)

Nozzle Extension
- Refractory metallic radiatively-cooled with emissivity coating



Flexible Inlet Ducts
- Based on J-2 & J-2S ducts
- Adjusted to meet J-2X performance
- Altered as necessary to meet current design standards

Open-Loop Pneumatic Control
- Similar to J-2 & J-2S design

Valves
- Ball-sector traceable to XRS-2200 and RS-68

Heat Exchanger
- Based on J-2 experience on as used on S-IVB

Channel wall MCC
- Based on RS-68 demonstrated technology

The J-2X engine is based on heritage systems with directly traceable flight experience.

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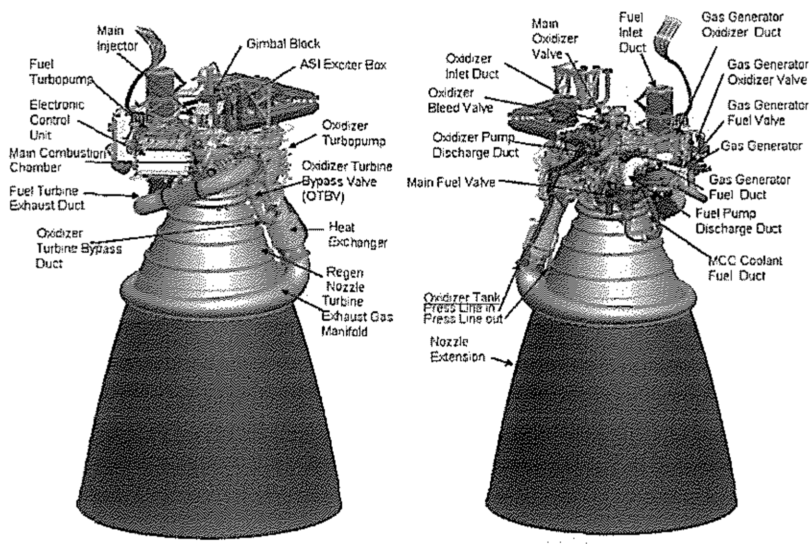
J-2X System Configuration



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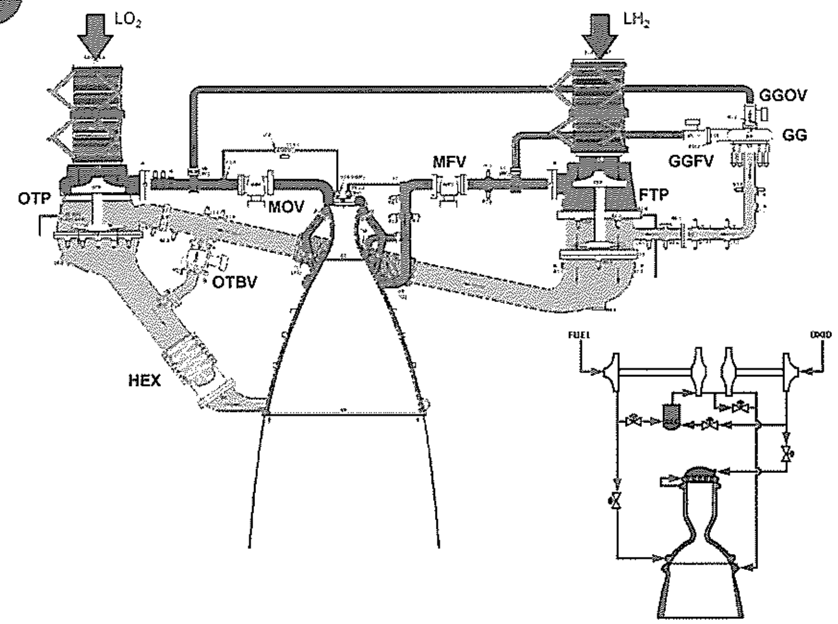
J-2X Schematic



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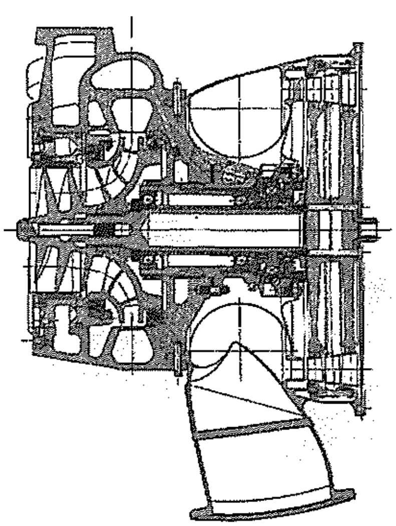
J-2 Mk 15-O Oxidizer Turbopump



- Originally used on J-2, modified for J-2X
- Heritage basis for the J-2X turbopump (Mk 72-O)
- Modified to meet J-2X requirements (i.e., performance, structural margins) and utilize technological improvements

TPA Characteristics (230-Klbf thrust)	Mark 15-0	
	Pump	Turbine
Speed (RPM)	8753	8753
Inlet pressure (psia)	39	89.5
Discharge pressure (psia)	1114	34.4
Inlet temperature (°R)	165°	1220°
Discharge temperature (°R)		1080°
Flow rate (lbm/sec)	450.7	7.2
Efficiency, η (%)	80.0	48.4
Horsepower (hp)	---	2358
Turbopump weight (lbm)	305	

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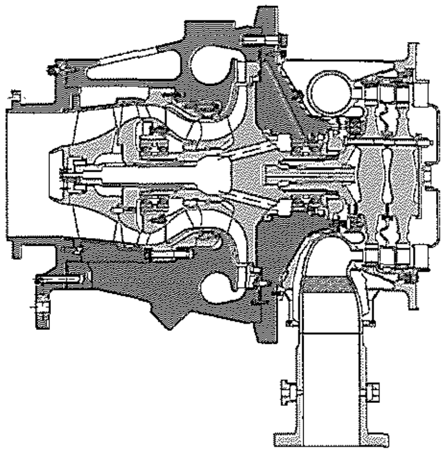
J-2S Mk 29-F Fuel Turbopump



- Not used on original J-2 system, developed for J-2S engine
- Heritage basis for the J-2X turbopump (Mk 72-F)
- Modified to meet J-2X requirements (i.e., performance, structural margins) and utilize technological improvements
  - Hydrostatic bearings

TPA Characteristics (265-Klbf thrust)	Mark 29-F	
	Pump	Turbine
Speed (RPM)	29935	29935
Inlet pressure (psia)	30	992
Discharge pressure (psia)	1908	109
Inlet temperature (°R)	40°	1660°
Discharge temperature (°R)		1210°
Efficiency (%)	76.0	46.0
Horsepower (hp)		13939

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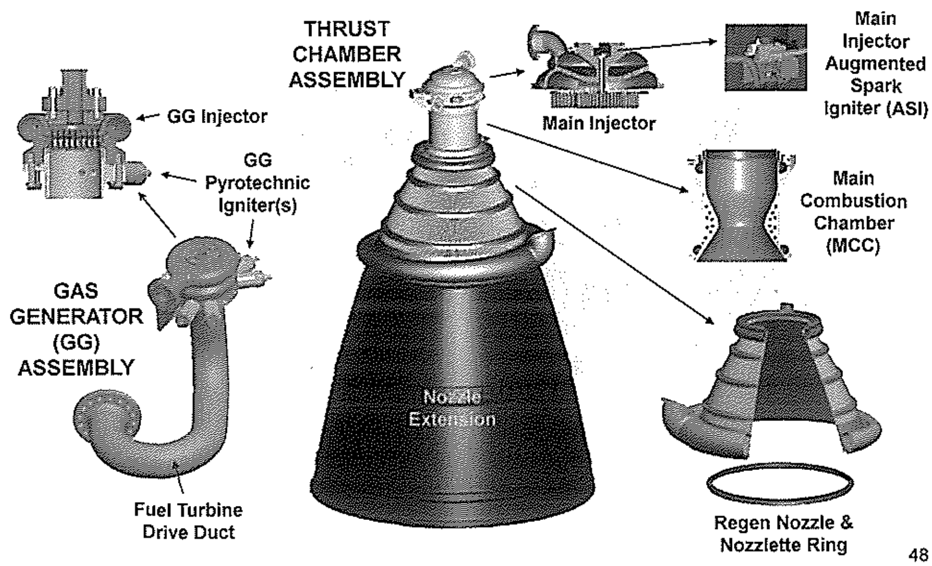
Combustion Devices

Component Preliminary Designs



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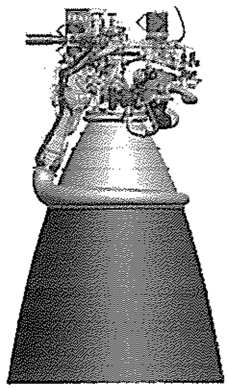
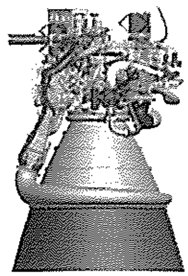
J-2X w Stub Nozzle



Full-Scale Nozzle Extension
- 92:1 Exit Area Ratio
- Overall Length 95.16 in.



Stub Nozzle Extension
- 59:1 Exit Area Ratio
- Overall Length 37.57 in.

- Going to a stub nozzle, consistent with what we would test in A-2, we would have a 59:1 area ratio (vs. an optimized 92:1) engine with the following performance deltas from our baseline:
  - Vacuum specific impulse: -13 seconds (435s guaranteed minimum vs. 448s guaranteed minimum baseline)
  - Vacuum thrust: -9,000 lbf (285,000 lbf nominal thrust vs. 294,000 lbf nominal thrust baseline)
  - Engine mass: -30 lbm (5505 lbm NTE vs. 5535 lbm NTE baseline)

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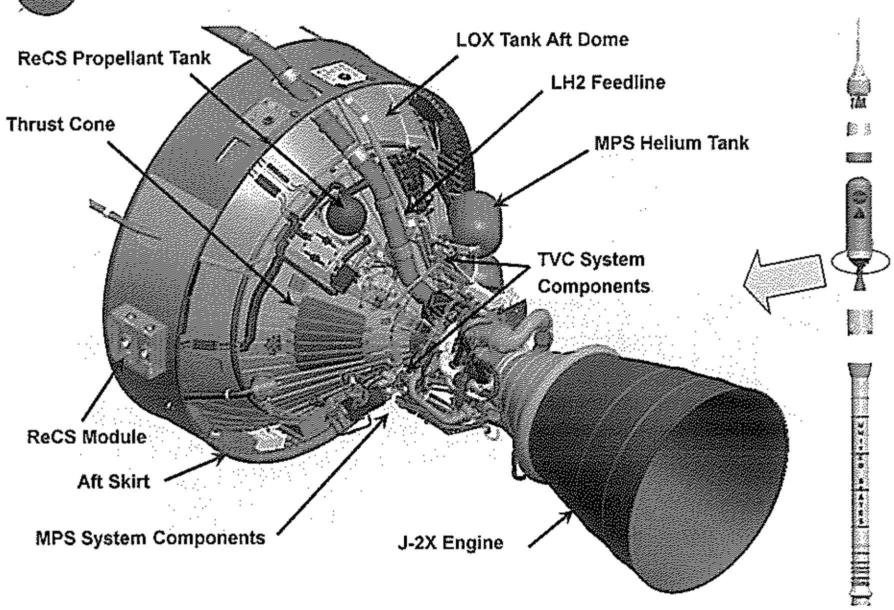
J-2X and Ares I Upper Stage Integration



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J-2X Component & Subscale Testing

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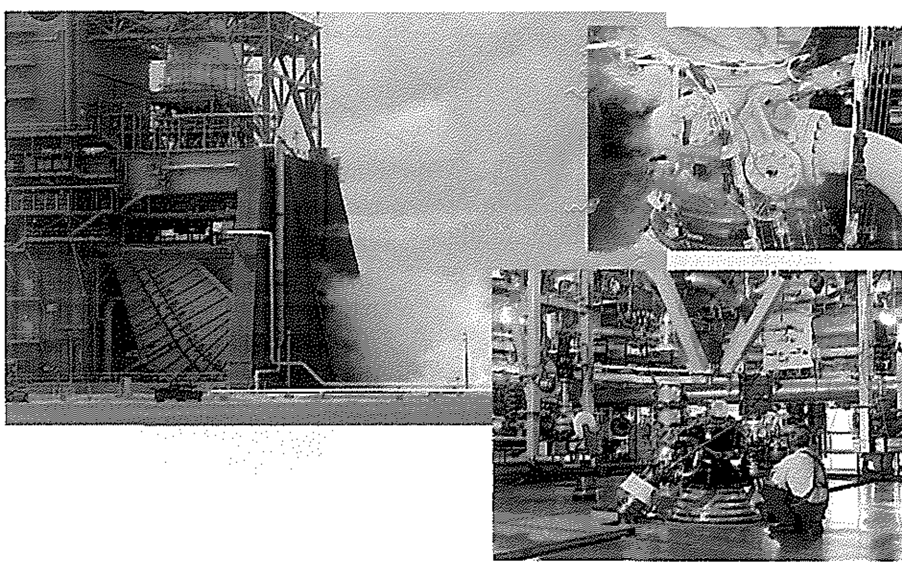
Powerpack 1 (PPA-1) Testing, SSC



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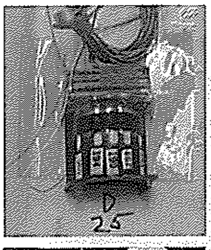
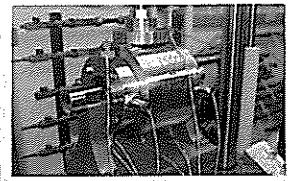
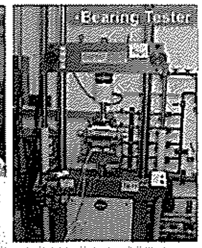
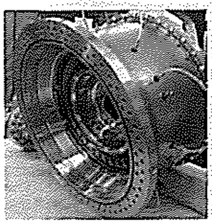
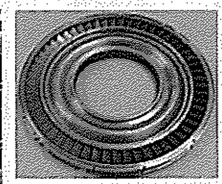
Turbomachinery Testing

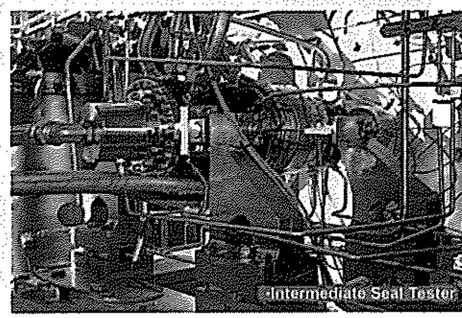












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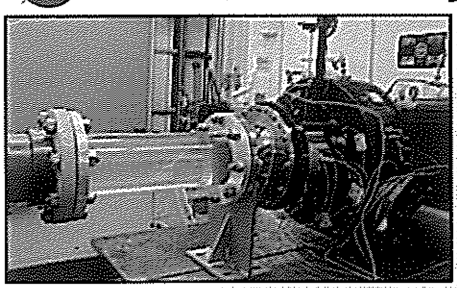
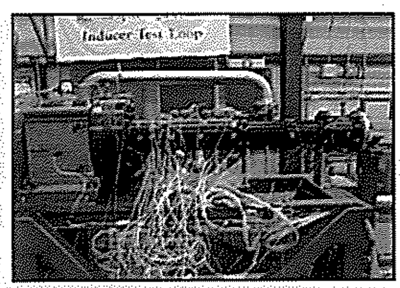
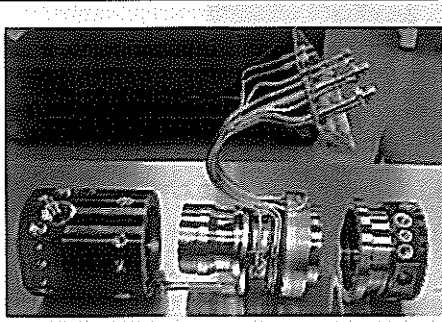
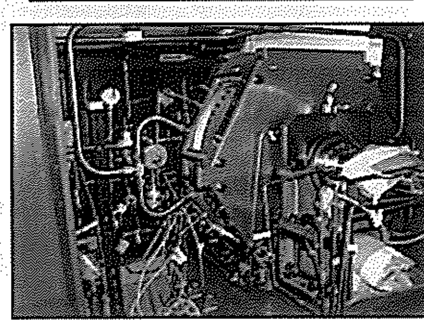
Turbomachinery Testing (cont'd)









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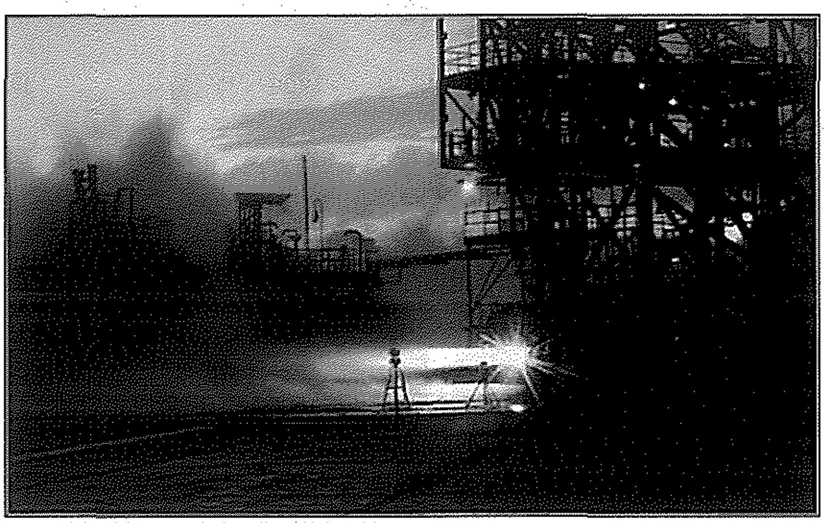
Subscale Main Injector Testing, MSFC



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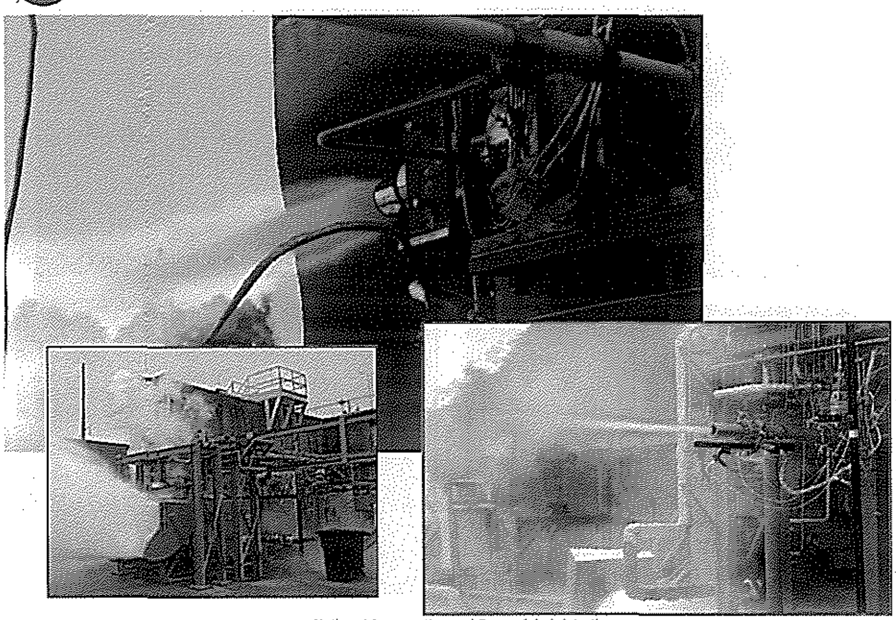
Workhorse Gas Generator Testing, MSFC



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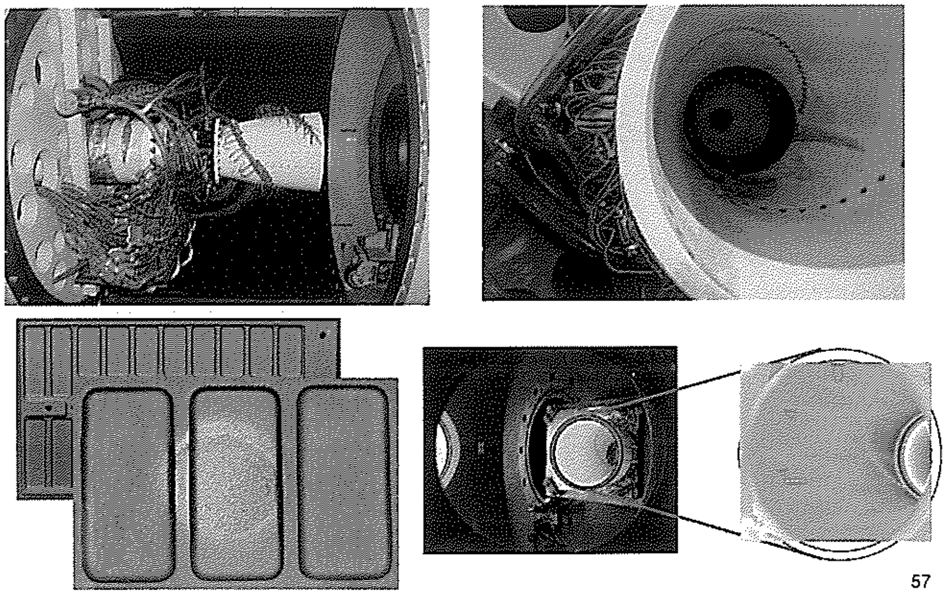
Nozzle Extension Testing, MSFC



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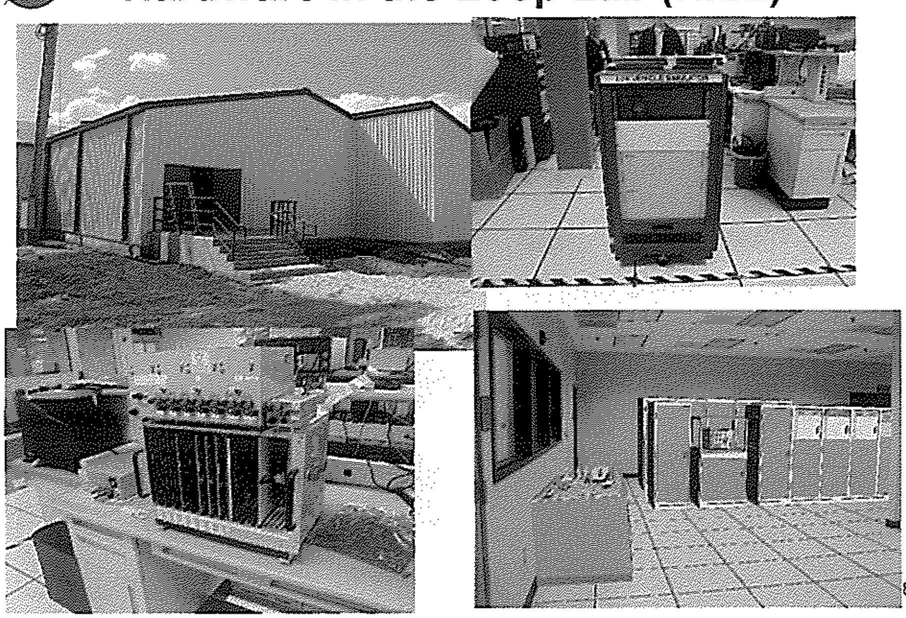
Hardware In the Loop Lab (HILL)



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Current Status

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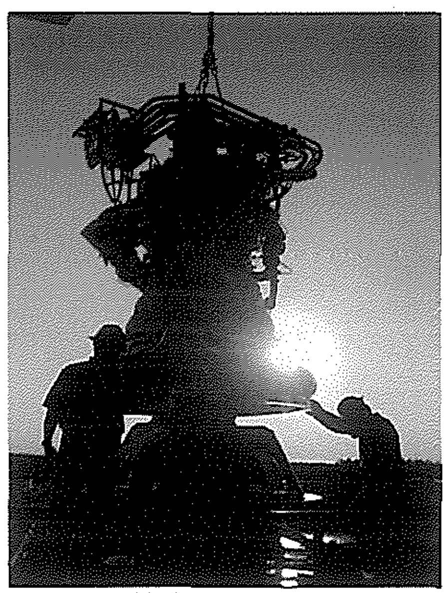
J-2X E10001 Testing, SSC

- Recently, the first J-2X development engine (E10001) completed assembly and installation into the A-2 test facility at NASA-SSC.
- Full system testing has begun and is expected to continue to support system certification for use on the NASA Space Launch System (SLS).



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Q&A

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