SpaceX Propulsion

Tom Markusic
Space Exploration Technologies

46th AIAA/ASME/SAE/ASEE
Joint Propulsion Conference
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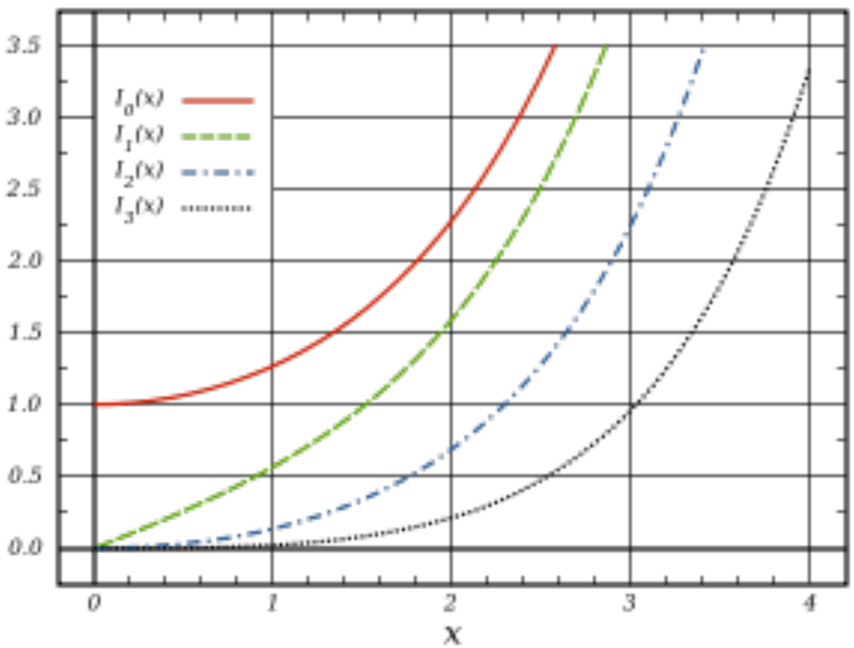
Inverse Hyperbolic Bessel Functions

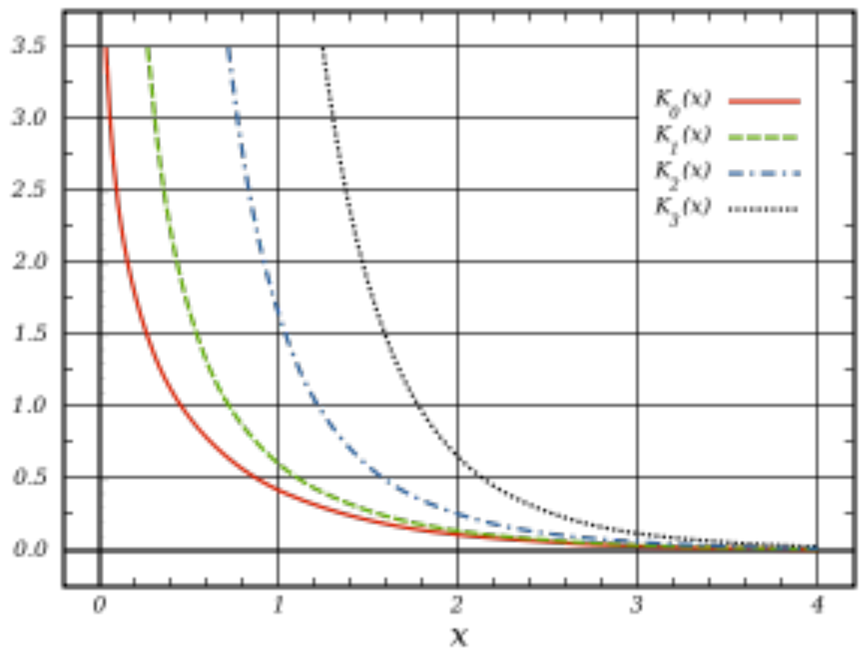
$$x^2 \frac{d^2 y}{dx^2} + x \frac{dy}{dx} - (x^2 + \alpha^2) y = 0.$$

$$I_\alpha(x) = i^{-\alpha} J_\alpha(ix) = \sum_{m=0}^{\infty} \frac{1}{m!\,\Gamma(m+\alpha+1)} \left(\frac{x}{2}\right)^{2m+\alpha}$$

$$K_\alpha(x) = \frac{1}{2} e^{-\frac{1}{2}\alpha\pi i} \int_{-\infty}^{+\infty} e^{-ix\sinh t - \alpha t} dt$$

$$K_\alpha(x) = \frac{\pi}{2} \frac{I_{-\alpha}(x) - I_\alpha(x)}{\sin(\alpha\pi)} = \frac{\pi}{2} i^{\alpha+1} H_\alpha^{(1)}(ix) = -\frac{\pi}{2} i^{\alpha+1} e^{-i\pi\alpha} H_\alpha^{(2)}(-ix).$$





Near-term Propulsion Needs

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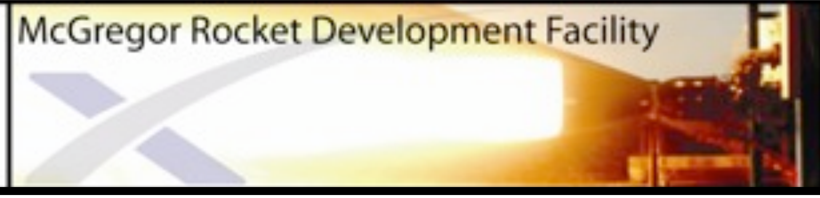

J-2X

HLLV Propulsion

- Merlin 2 uses scaled-up, flight proven Merlin 1 design
- SpaceX can develop and flight qualify the Merlin 2 engine in ~3 years at a cost of ~$1B. Production: ~$50M/engine
- J-2X development already in progress under Constellation program

Merlin 2

	Merlin 2	J-2X
Propellant	LOX/RP	LOX/LH_2
Thrust (vac) [klbf]	1,700	292
Isp (vac) [sec]	322	448
T/W [lbf/lbm]	150	55

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Solar Electric Propulsion for Cargo Tug

NEXT Ion Thruster

usek BHT-20K Hall Thruster

NASA 457M Hall Thruster

- Cluster of ~5 high TRL thrusters process 100 kWe solar power
- Next generation tug uses single high power thruster, such as NASA 457M
- Third generation tug uses nuclear electric propulsion at megawatt levels

	NEXT	BHT-20	457M
Propellant	Xenon	Xenon	Xenon
Power [kWe]	7	20	96
Thrust [mN]	236	1080	3300
Isp [sec]	4100	2750	3500
Efficiency [%]	70	72	58

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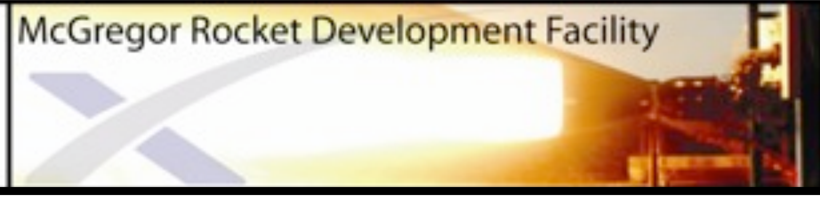

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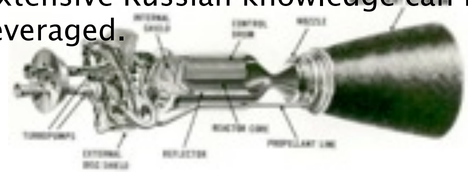
Nuclear Thermal Propulsion for Mars Stage

- NERVA derived technology
  - Total thrust ~ 60 klbf, using 2 to 6 NDR
  - Propellant: hydrogen, Isp ~ 930 sec
- ISRU or pre-deployed propellant for return mission
- Technology has been verified with >17 Hours of hot-fire tests, including restarts. No additional developmental, terrestrial tests (with nuclear) fuel are required.
- Extensive Russian knowledge can be leveraged.



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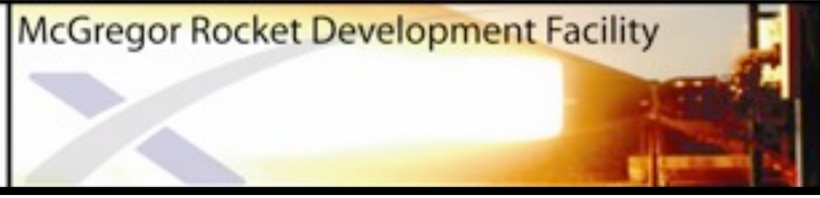

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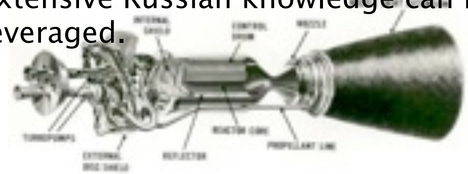
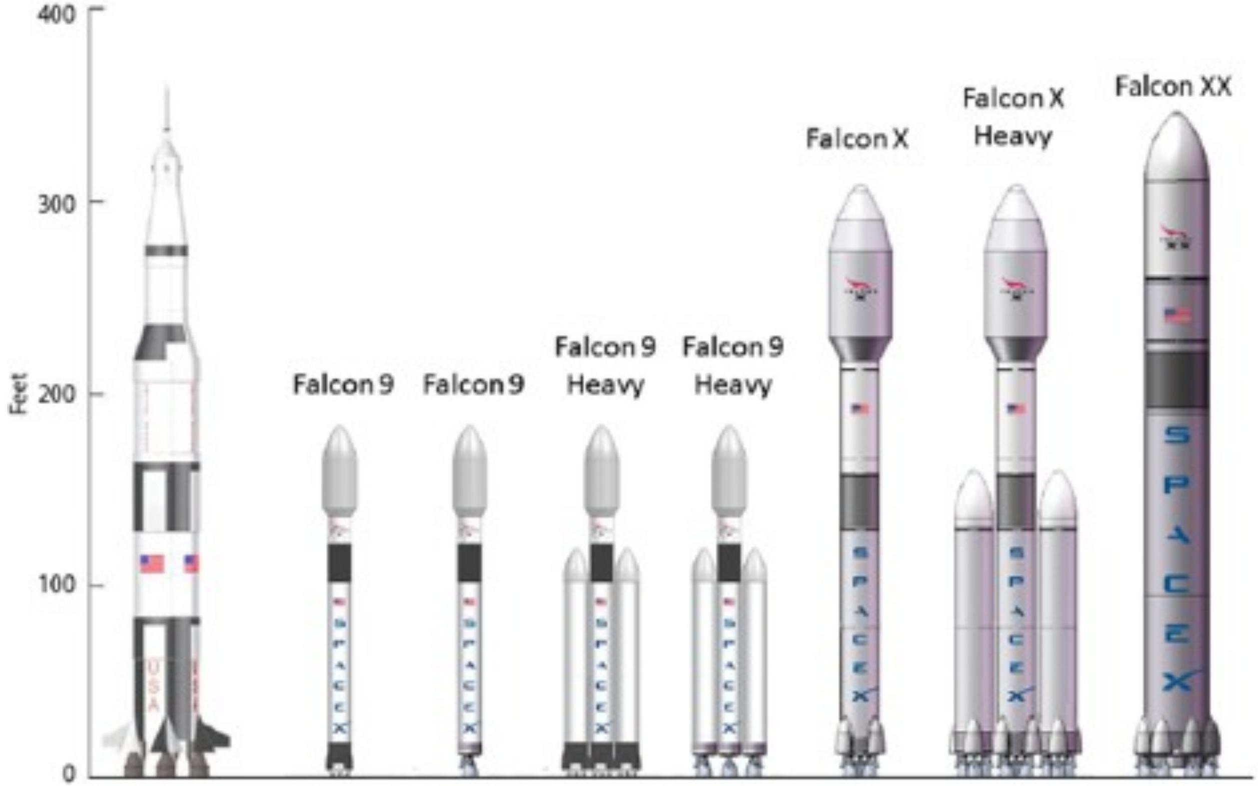


LOX/Methane Propulsion for Ascent/Desc

- ISRU-derived methane will be used for ascent/descent propulsion
- Strong developmental programs currently underway at Aerojet, ATK/XCOR
- SpaceX Merlin 1 engine may be reconfigurable to for LOX/methane, providing a large (~100 klbf) GG cycle engine for ascent/descent

Aerojet, T = 5.5 k-lbf, Isp = 350 sec

ATK/XCOR, T =7.5 k-lbf, Isp = ?

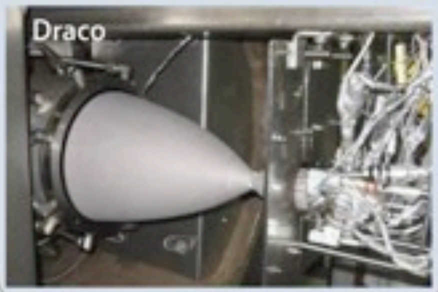


VEHICLE	Falcon 9	Falcon 9	Falcon 9 Heavy	Falcon 9 Heavy	Falcon X	Falcon X Heavy	Falcon XX
1st Stage Engines	Merlin 1D	Merlin 2	Merlin 1D	Merlin 2	Merlin 2	Merlin 2	Merlin 2
Core Diameter (meters)	3.6	3.6	3.6	3.6	6	6	10
Number of Cores	1	1	3	3	1	3	1
Engines per Core	9	1	9	1	3	3	6
Engine Thrust (sea level, lbf)	120k	1.2M	120k	1.2M	1.2M	1.2M	1.7M
Total Lift-off Thrust (lbf)	1.08M	1.2M	3.24M	3.6M	3.6M	10.8M	10.2M
Engine Out Capability?	Yes	No	Yes	No	Yes	Yes	Partial
Mass to LEO (kg)	10.5k	11.5k	32k	34k	38k	125k	140k

SPACEX

Testing Survey

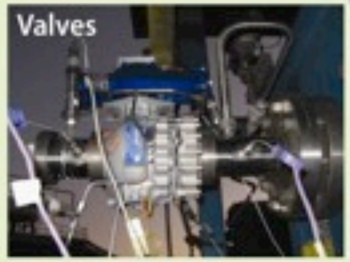
Engine Testing

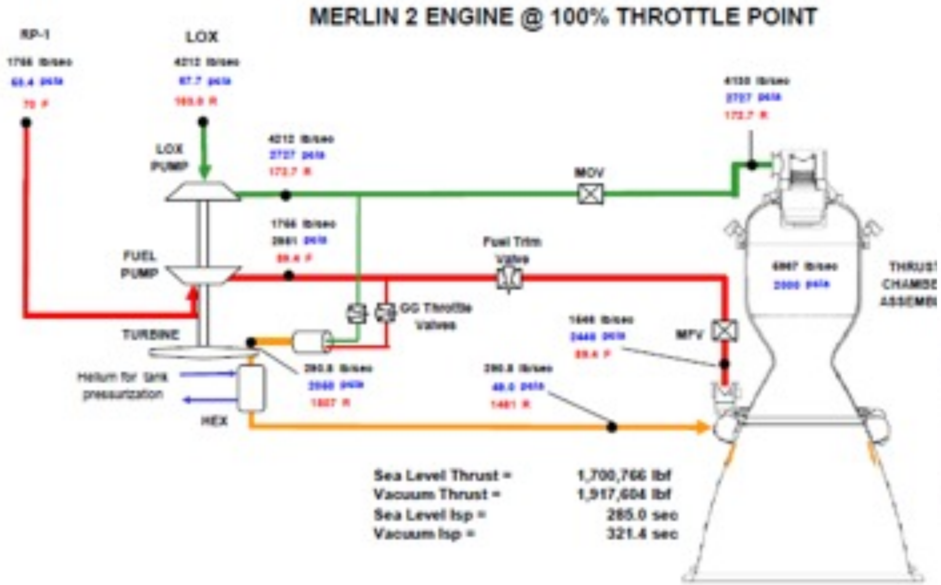


Stage Testing

Structural Testing

Component Testing



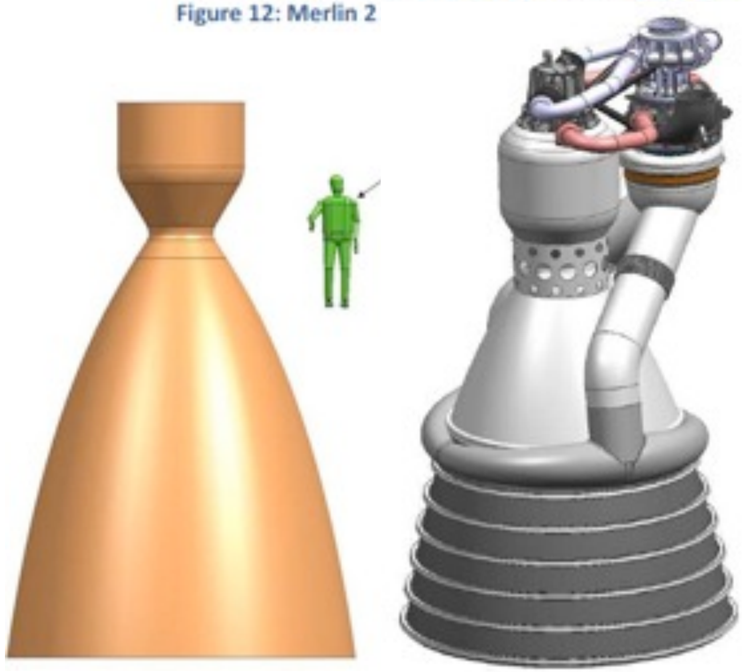


100% RPL		
RP1 Flowrate	16,700	GPM
LOX Flowrate	26,500	GPM

Table 1: Merlin 2 Engine Summary

Parameter	70% Throttle Point	100% Throttle Point
Propellants	LOX/RP-1	LOX/RP-1
Sea Level Thrust	1.20M lbf	1.70M lbf
Vacuum Thrust	1.42M lbf	1.92M lbf
Sea Level Isp	271 seconds	285 seconds
Vacuum Isp	320 seconds	321 seconds
Chamber Pressure	1472 psia	2000 psia
Nozzle Area Ratio	25:1	25:1
Mixture Ratio (Ox/fuel)	2.4	2.4
Throttle Setting	70 %	100 %

Figure 12: Merlin 2



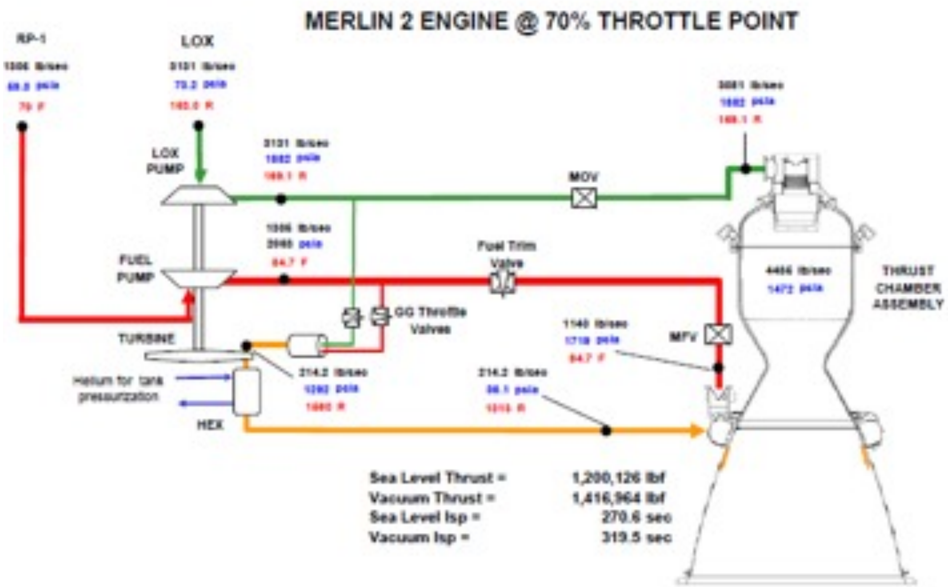


Figure 11: Merlin 2 State Point Diagram at 70% Throttle

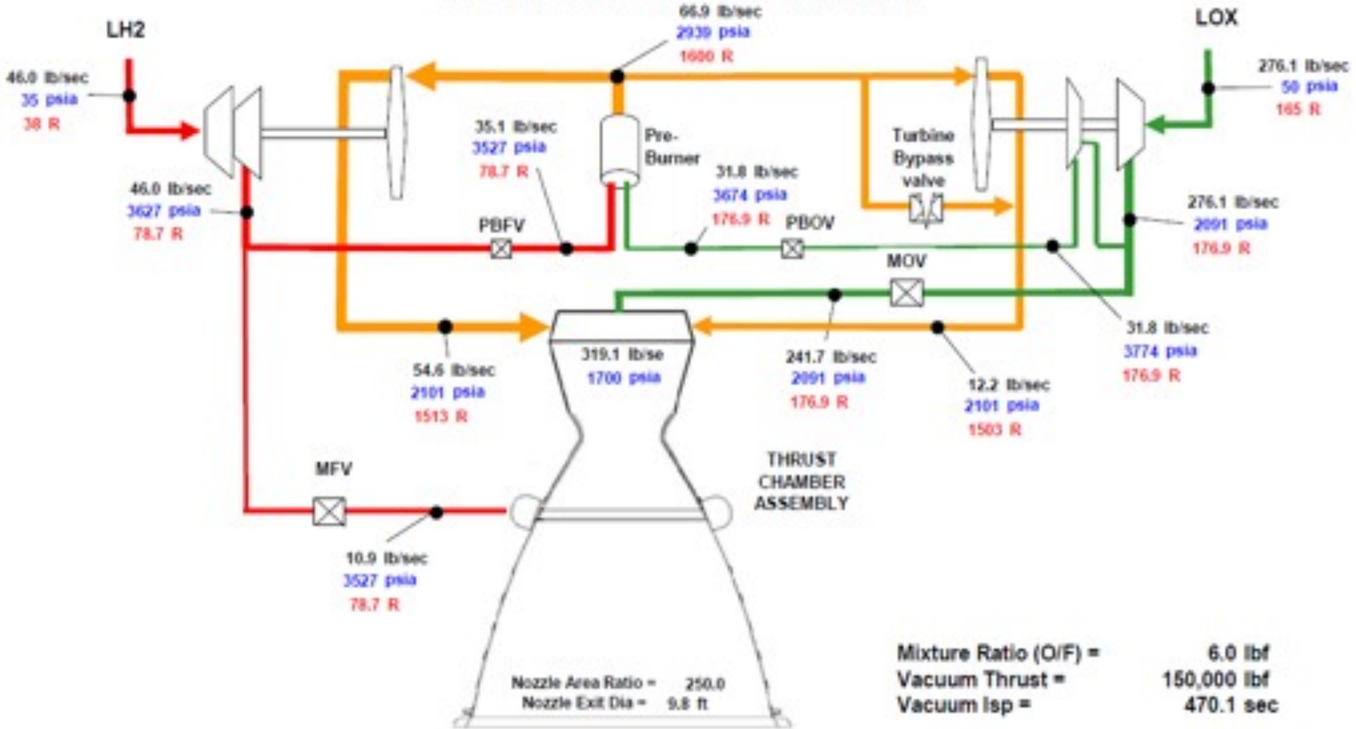
SPACEX

Raptor

McGregor Rocket Development Facility

Parameter	Value
Propellants	LOX/LH2
Vacuum Thrust	150,000 lbf
Vacuum Isp	470.1 seconds
Chamber Pressure	1700 psia
Nozzle Area Ratio	250:1
Mixture Ratio (Ox/fuel)	6.0
Throttle Range	50 - 100 %

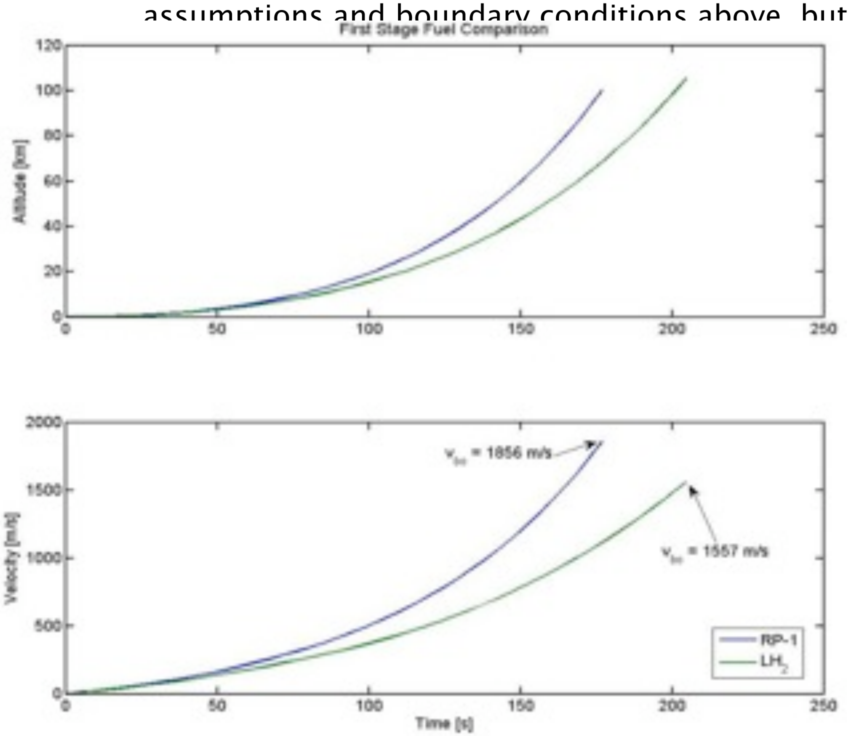
RAPTOR ENGINE STATEPOINT DIAGRAM



LOX/RP versus LOX/LH2 Booster

Fundamentals

- Simple 1-D dynamic model used to compare LOX/RP and LOX/LH2 first stage performance for a HLLV
  - First, for both propellants, propellant mass was chosen to yield the same ΔV (3.6 km/s) for a given payload (750 MT), consistent with Saturn V, but with no external forces.
  - Typical engine performance and tank mass fractions assumed.
  - Initial T/W fixed at 1.2 for both cases. Ballistic trajectory.
  - Equations of motion again integrated using assumptions and boundary conditions above, but



HLLV 1st Stage Propulsion

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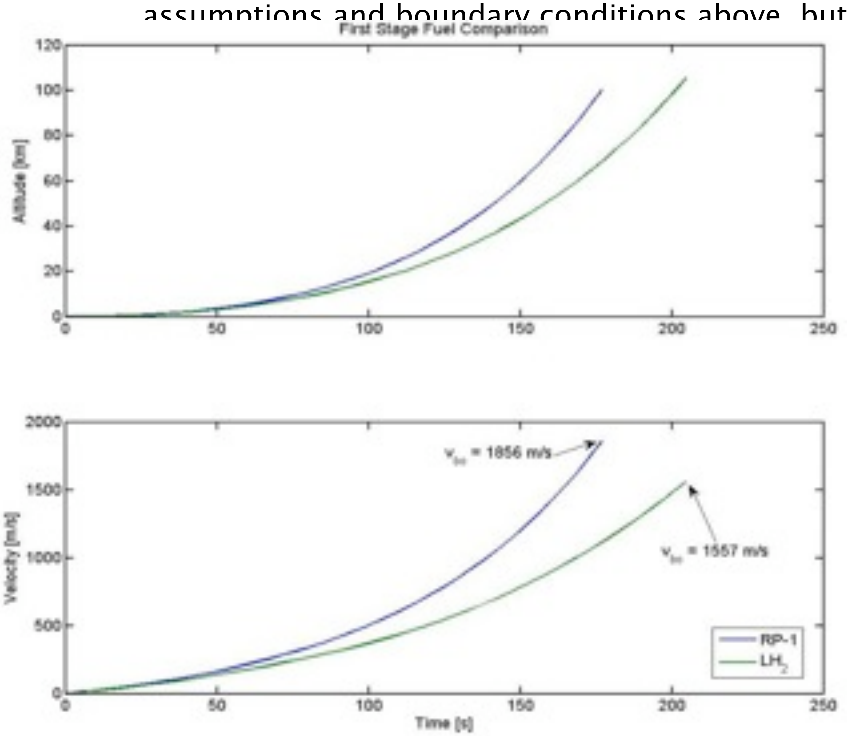

Trade Studies

- Recent NASA-led "Heavy Lift Launch Vehicle Study" compared many configurations of LOX/LH2, LOX/RP, SRB propulsion for a HLLV.
  - Configuration with 6 Lox/RP engine first stage competitive with all concepts in performance and mission capture metrics
  - Configuration with 6 Lox/RP engine first stage shown to provide benefits in safety and annual recurring cost metrics above all LOX/LH2 and SRB configurations

Operations

Handling.

- Deep cryogenic (-432 F) vs room temperature for RP
- LH_2 has high infrastructure investment for test and launch

Safety.

- LH2 leaks lead to detonation risk—extensive monitoring required
- RP leaks are easily (visually) detectable, low explosion risk

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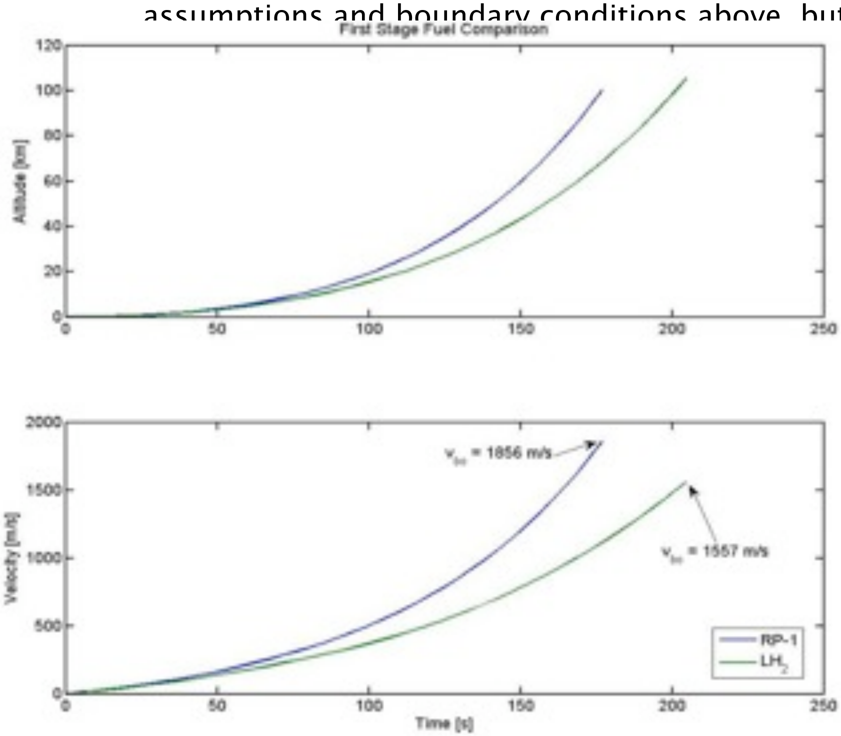

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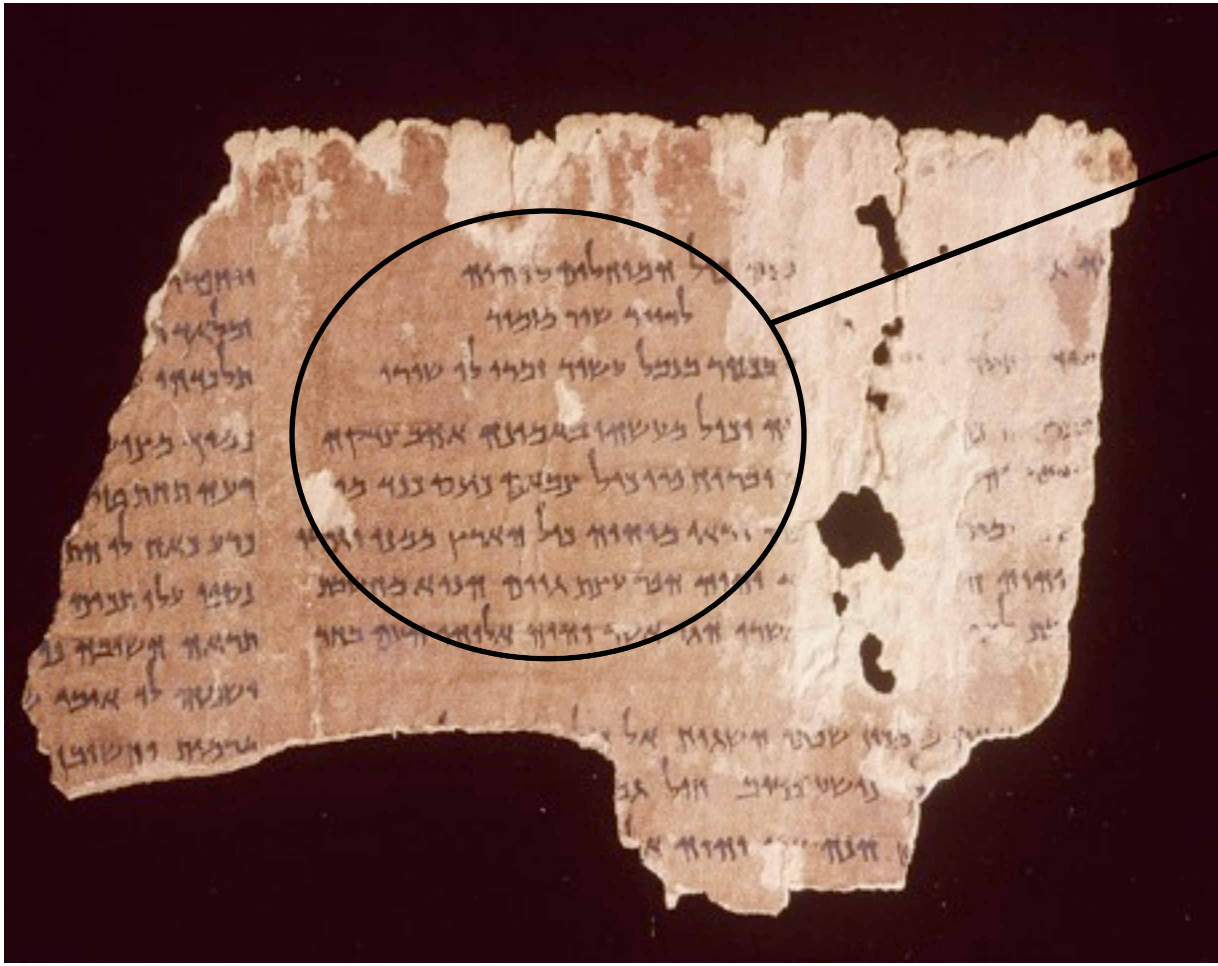
RP staged combustion versus GG cycle

- First Stage ΔV simplified model compared Merlin 2 gas generator cycle engine with scaled up RS-84 derived staged combustion engine.
  - Mass of Merlin 2 based on current design (sea level thrust = 1.7 Mlbf). Mass of RS-84 derived engine estimated by linearly scaling thrust and assuming T/W is constant.
  - Merlin 2 vac Isp = 322.1 sec, RS-84 derived vac Isp = 334.6 sec.
  - Modeled Falcon X with F9 flight trajectory (250 km x 34.5 deg).
  - Found burnout velocity for Merlin 2 stage and RS-84 derived stages to be 3526 m/sec and 3527 m/sec,

Dead Sea Scrolls

McGregor Rocket Development Facility



"Black water shall elevate thy children to the heavens. Purify it. But thou shalt not combine it in a ratio greater than one kikkar to twenty shekkels, nor shalt thou burn rocks. Thus saith the lord."

- # Assumptions for Mission and Vehicle Sizing

HLLV T/W	1.2
1st Stage Payload	750 MT
RP-1 inert mass fraction	0.06
LH2 inert mass fraction	0.08
RP-1 Isp	300 s
LH2 Isp	420 s
RP O/F ratio	2.27
LH2 O/F ratio	5.5
Stage height, excluding	36 m
RP-1 GLOM	3040 MT
LH2 GLOM	2060 MT
RP-1 Burnout time	177 s
LH2 Burnout time	205 s
RP-1 Stage diameter	8.7 m
LH2 Stage diameter	11.3 m

SEP Isp	2750 s
SEP thrust per engine	1.08 N
Xenon tank mass fraction	0.1
SEP structural and margin mass	0.1
Solar Arrays and PPU mass fraction	3.5 kg/kW
Low-thrust Delta V LEO to Phobos	11.2 km/s
NTR Isp	930 s
Delta V LEO to TMI	4.2 km/s
Delta V TMI to MOC	2.5 km/s
Delta V MOC to Phobos Capture	0.4 km/s
NTR 15k lbf-thrust engine mass	2600 kg
NTR tank mass fraction	0.1
Earth Aerocapture Delta V savings	3.2 km/s