Rocket Fundamentals

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Contents

Rocket Thrust

- Newton's 3rd Law: Momentum (or *total impulse*) of rocket and payload is equal (and opposite) to that of the exhaust stream from rocket engine. In general unsteady flow,

$$p = \int V_e \, dm = \int V_e(t) \, (dm/dt) \, dt$$

where V_e = freestream exhaust velocity

dm = *mass increment*

(dm/dt) = *mass flow rate*

- Instantaneous force (thrust) produced by rocket engine in steady-state operation in vacuum, from Newton's 2nd Law:

$$T = dp/dt = (dm/dt) \, V_e$$

Specific Impulse

- In steady-state conditions (for simplicity) we have:

$$p = m\,V_e = I \quad (kg\ m/s)$$

- Increment of momentum (or impulse) *per unit mass of propellant* is simply the exhaust velocity:

$$V_e = I\,/\,m = I_{sp} = \textit{specific impulse (m/s)}$$

- Very customary to use weight flow (*mg*) instead of mass flow normalization, yielding:

$$I_{sp} = I_{sp}\,/\,g = V_e\ /\,g \qquad \textit{specific impulse}\ (s)$$

Ideal Rocket Equation

- Thrust equation now becomes

$$T = dp/dt = (dm/dt)\, V_e = (dm/dt)\, gI_{sp}$$

- Motion of rocket in vacuum in absence of gravity, or normal to gravity vector, from Newton's 2nd Law:

$$dV/\mathrm{d}t = a = T/m = (dm/dt)\, V_{\mathrm{e}} / m = (g\, I_{sp}/m)\, dm/dt$$

or

$$dV = g\, I_{sp}\, dm/m$$

Ideal Rocket Equation (cont.)

- Integrate to obtain:

$$\Delta V = gI_{sp} \ln (m_i/m_f) = gI_{sp} \ln MR$$

or

$$m_i/m_f = e^{\Delta V/gI\mathrm{sp}}$$

or

$$(m_i - m_f) / m_f = \Delta m / m_f = e^{\Delta V/gIsp} - 1$$

If all propellant is consumed, $\Delta m = m_p$ = propellant mass.

Ideal Rocket Equation (cont.)

- Let

  m_s = dry mass of structure

  m_p = propellant mass

  m_P = payload mass

  Then

  $$m_i / m_f = (m_p + m_s + m_P) / (m_s + m_P) = e^{\Delta V/gIsp}$$

  $$= (1 + \lambda) / (\varepsilon + \lambda)$$

  where

  $\lambda = m_P/(m_s + m_p)$ = payload ratio (*Saturn 5 = 0.05*)

  $\varepsilon = m_s/(m_s + m_p) = 1 - \eta$ = structural coefficient

  $\eta = m_p/(m_s + m_p)$ = propellant fraction (*STS ET = 0.964*)

Design Principles

- To maximize ΔV, both rocket effective exhaust speed, V_e, (or I_{sp}) and mass ratio, *MR*, must be maximized. The effective exhaust speed derives from propulsion system performance, while the mass ratio is a figure of merit of the structural design.
- The propulsion designer tries to maximize V_e (or I_{sp}) and the structural designer tries to maximize *MR*. A high *MR* is obtained through minimizing the dry mass that is taken to orbital velocity (light weight structure and staging or dropping the depleted propellant tanks during ascent.)
- Since the propulsion system is a significant element of mass of the rocket, the rocket designer must trade propulsion mass and I_{sp}. Typically the propulsion *T/W* (thrust to weight ratio) is more significant during early boost (1st stage) and I_{sp} is more important during late boost or for upper stages.

Rocket Engine Performance

- A rocket engine is a device for converting the potential energy of a hot, high pressure stagnant gas into directed kinetic energy.
- Actual engine performance depends on many factors
  - Throttling
    - Hard to do without degrading V_e.
    - Constant dm/dt can lead to excessive acceleration at burnout.
  - Ambient Pressure (e.g., altitude)

    $$T = (dm/dt)\,[V_e + (p_e - p_a)A_e/(dm/dt)\,] = (dm/dt)\,V_{eq}$$

    - $p_e > p_a$ implies underexpanded flow; i.e., not full nozzle efficiency
    - $p_e < p_a$ implies overexpanded flow and unwanted base drag

Engine Performance

- Exit area:

  $A_e / A^* = (1/M_e)\{2\ [l + (k - 1)\ M_e^2/2] / (k + 1)\}^{(k+1)/2(k-1)}$

  where

  $M_e = V_e/a_e$ = exit Mach number

  $a_e = kRT_e$ = exit speed of sound

  A^* = sonic throat area

  – Size does matter.

- Mass flow rate:

  $dm/dt = p_e A^*\{(k/RT_e)[2/(k+1)]^{(k+1)/(k-1)}\}^{1/2}$

Engine Performance (cont.)

- Chamber pressure and temperature:

$$V_e^{\,2} = kR_{\text{gas}}T_c[1 - (p_e/p_c)^{(k-1)/k}]/(k-1)$$

where

k = ratio of specific heats = c_p/c_v

p_e = nozzle exit pressure

p_c = combustion chamber pressure

T_c = combustion chamber temperature

R_{gas} = exhaust flow specific gas constant

- Higher pressure and higher temperature increase V_e
- Flame temperature of chemical propellants determines T_c
- Turbopump performance determines p_c.

Engine Performance (cont.)

- Chamber pressure and temperature related isentropically to exit conditions:

$$T_e / T_c = (p_e / p_c)^{(k-1)/k}$$

Turbine vs. Rocket Engines

Turbine Engines

- Internal operating pressure ~300 psi
- Turbine temperatures ~ 3300 F
- T/W ~ 6 @ T~40,000 lbf
- Room temperature propellants
- Mission time at max thrust ~ 20%
- Idle to max thrust < 5 sec

Rocket Engines

- Internal operating pressures ~6000 psi
- Turbine temperatures ~ 1300 F
- T/W~60 @ T~400,000 lbf
- Cryogenic propellants (-290F to -423F)
- Mission time at max thrust ~95%
- Idle to max thrust ~ 1 sec

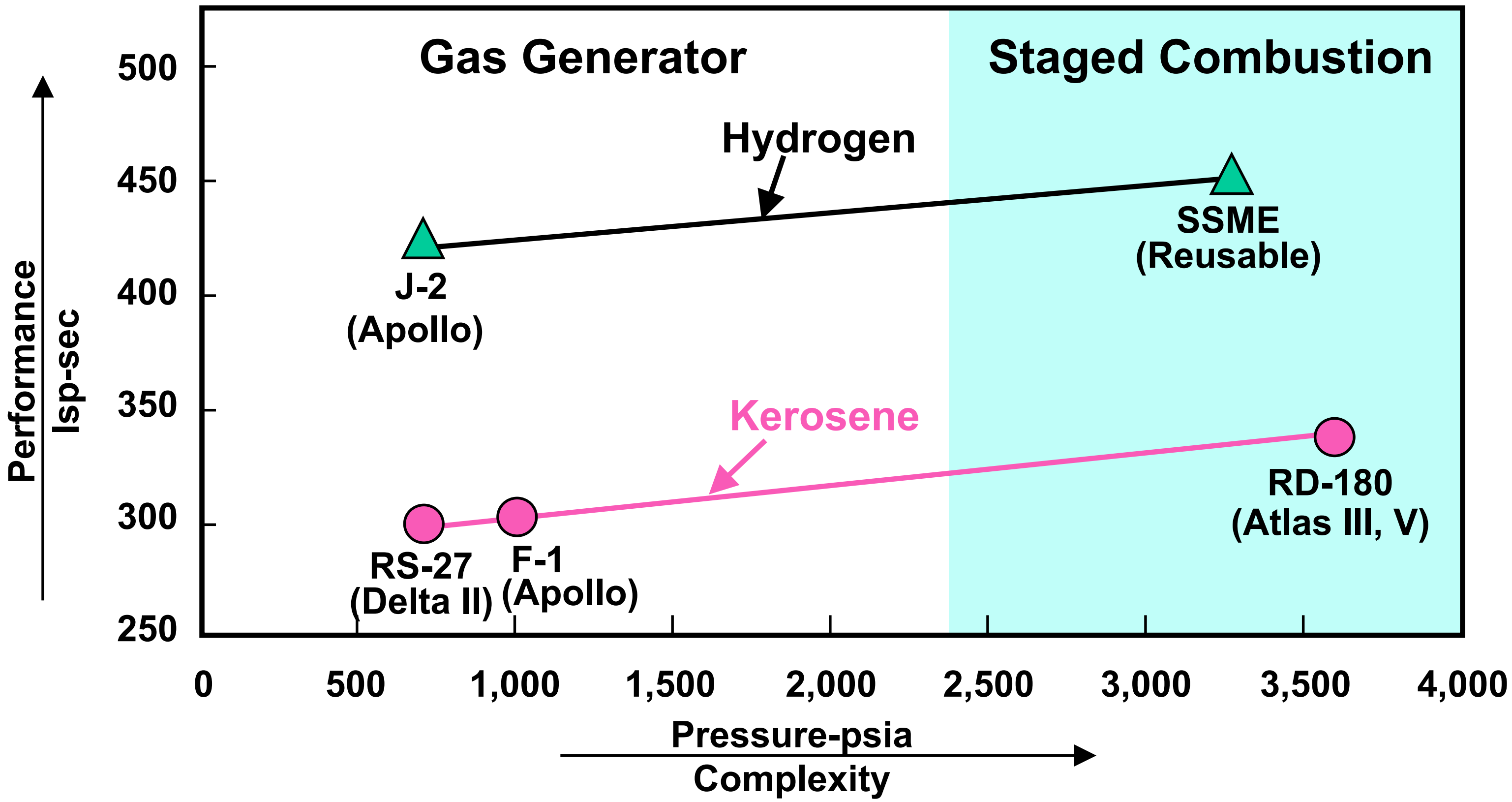
Rocket Engine Design Considerations

Typical I_{sp} and T/W for Representative Propulsion Systems

Propellant	**I_{sp} (s)**	**T/W**
Cold Gas	60-250	
Liquid Rocket		
Monopropellant	140-235	
Lox/Kerosene	300-340	70-100
Lox/Hydrogen	400-460	40-80
Solid Rocket	260-310	
Hybrid Rocket	290-350	

- Hydrogen propellants offer high I_{sp} and excellent upper stage performance
- Hydrocarbon systems yield lower I_{sp}, but higher T/W and MR and offer excellent 1st stage performance.
- For a given propellant combination, closed-cycle engines offer higher I_{sp} than open-cycle engines. Closed cycle engines expand all propellant through the main combustion chamber and nozzle, while open-cycle engines dump turbine drive propellant overboard.
- Performance parameters for some existing propulsion systems are shown on the following page. Staged combustion engines are closed cycle engine and gas generator engines are open cycle.

I_{sp} for Existing Propulsion Systems



Representative Existing Rocket Engines

Engine	**Vacuum Thrust (lbf)**	**Fuel**	**Oxidizer**	**I_{sp} (sec)**	**Expansion Ratio**
RL10B-2	25,000	LH_2	LO_2	464	285:1
SSME	470,000	LH_2	LO_2	459	77.5:1
RS-27A	237,000	RP-1	LO_2	302	12:1
RS-68	745,000	LH_2	LO_2	410	21.5:1
RD-170	1,777,000	RP-1	LO_2	331	36.4:1
Viking 4B	177,000	UH25	N_2O_4	293.5	30.8:1
Vulcain-2	304,000	LH_2	LO_2	433	58.5:1

Circular Orbit Velocity

- For a circular orbit, the velocity is constant throughout the orbit:

$$V_{circ} = (\mu / r_{circ})^{1/2}$$

where μ is the gravitational parameter and r_{circ} is the radius of the circular orbit to the planetary center. μ_E = 398,600 km^3/s^2 and Earth equatorial radius R_E = 6378 km, thus a 185 km (100 n. mi.) altitude circular LEO orbit has a circular velocity of approximately 7.8 km/s.

- This is the velocity to be supplied by the ascent vehicle.

Ascent Losses and Flight Performance

- Launch vehicle must actually supply 25-30% higher ideal ΔV than V_{circ}; i.e., 9.8 – 10.2 km/s – due to various losses:
  - Thrust Loss
    - Over- or under-expansion of nozzle relative to ambient pressure.
  - Gravity
    - Minimize by going as fast as possible, as soon as possible.
    - High T/W helps, but very high g-loads are incompatible with most cargo and with the desire to limit launch vehicle mass.
  - Drag ($D = \frac{1}{2} \rho V^2 S C_D$)
    - Thin vehicles reduce drag at expense of volumetric efficiency.
    - Reduce drag by avoiding high speed at low altitude
  - Steering Loss
    - Final velocity vector must be horizontal, while liftoff is vertical.
    - Energy spent going in the wrong direction is wasted; must pitch to horizontal as soon as possible – which increases drag.

Ascent Vehicle Design Considerations

- Engines can be clustered to provide vehicle steering (roll/pitch/yaw) as well as potential engine-out capability. Engine-out capability provides significant improvement in ascent reliability. (Two Saturn 5 engine-out experiences; no loss of mission.)
- Size and envelope (length, diameter, angular motion) become critical design parameters for rocket engines, especially in multiple engine clusters and upper stages.
  - High combustion chamber pressure enables high gas expansion (thus high V_e or I_{sp}) in a smaller envelope.
  - High combustion chamber pressure combined with a closed cycle provides the highest performance in the smallest envelope, but greatly increases the turbopump horsepower requirements. A comparison of the fuel turbopump power density (power/mass) is illustrative.

Engine	Fuel Pump Power Density
J-2 (open cycle, 700 psi chamber pressure) -	20
SSME (closed cycle, 3000 psi chamber pressure) -	100

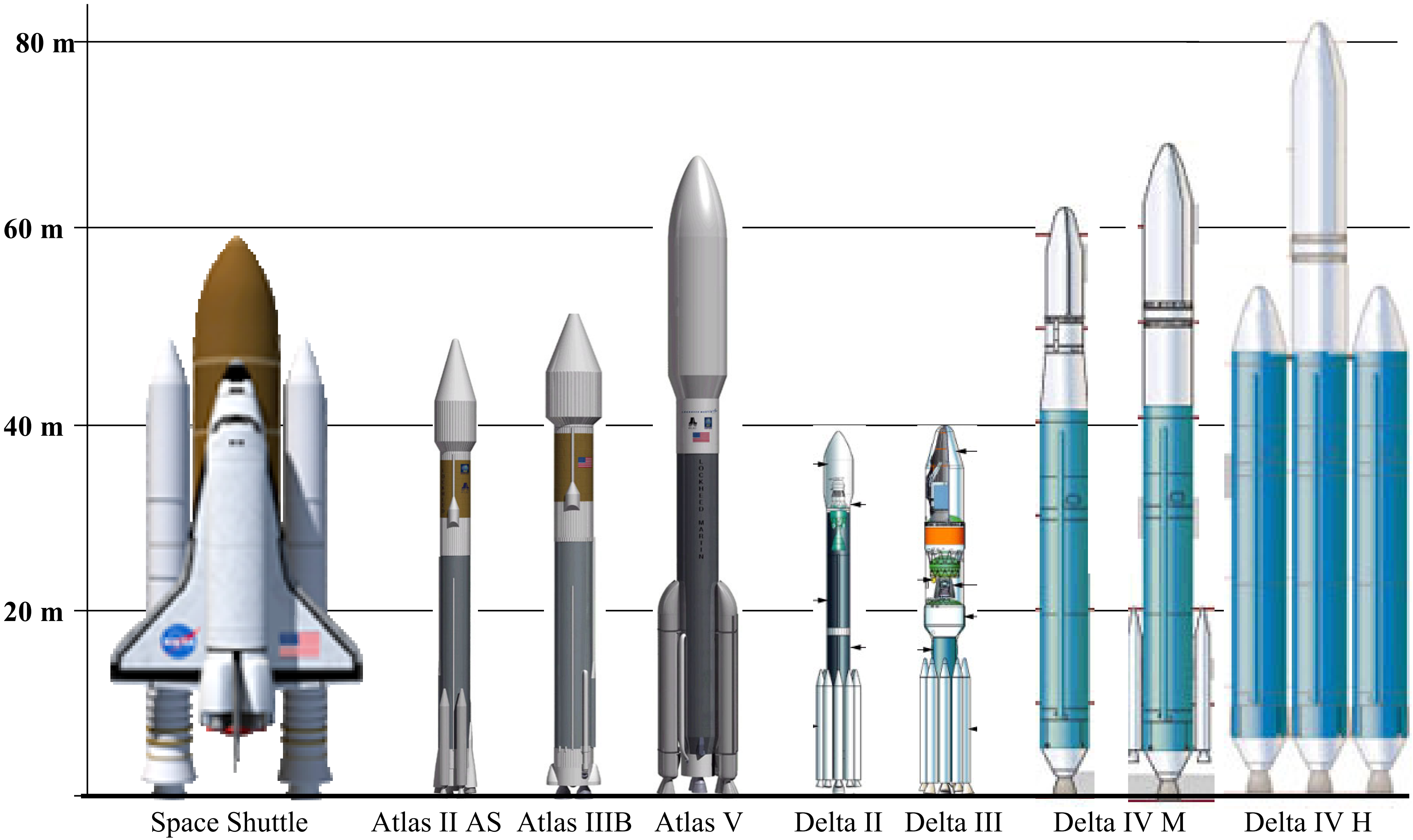
- Parallel burn (engine operation from ground to orbit) provides the opportunity to monitor the health of the engine prior to liftoff, whereas in a serial burn configuration the engine is started at altitude after staging.
- I_{sp} can be enhanced for the serial burn engine at operating altitude by optimizing the nozzle expansion ratio. The I_{sp} of the parallel burn engine is compromised because it must operate at sea level as a booster engine and at altitude as an upper stage engine.

Expendable vs. Reusable Design

- The highest performance rockets are expendable, multi-stage designs, with lox/hydrocarbon 1st stage and LH_2 upper stage(s). Operations cost is driven by vehicle stage integration and replacement of expended hardware.
  - The Saturn V was the rocket with the highest payload mass fraction.

- Reusable stages could have lower operations cost and higher reliability than expendable stages if the launch rate is high and the reusable stages can be maintained efficiently. Performance suffers because of the higher structural weight of reusable systems (structural margins, thermal protection systems, wings, landing gear, etc.). To approach the performance of an expendable system, material/structures technology must be matured to reduce weight, resulting in increasing development cost.

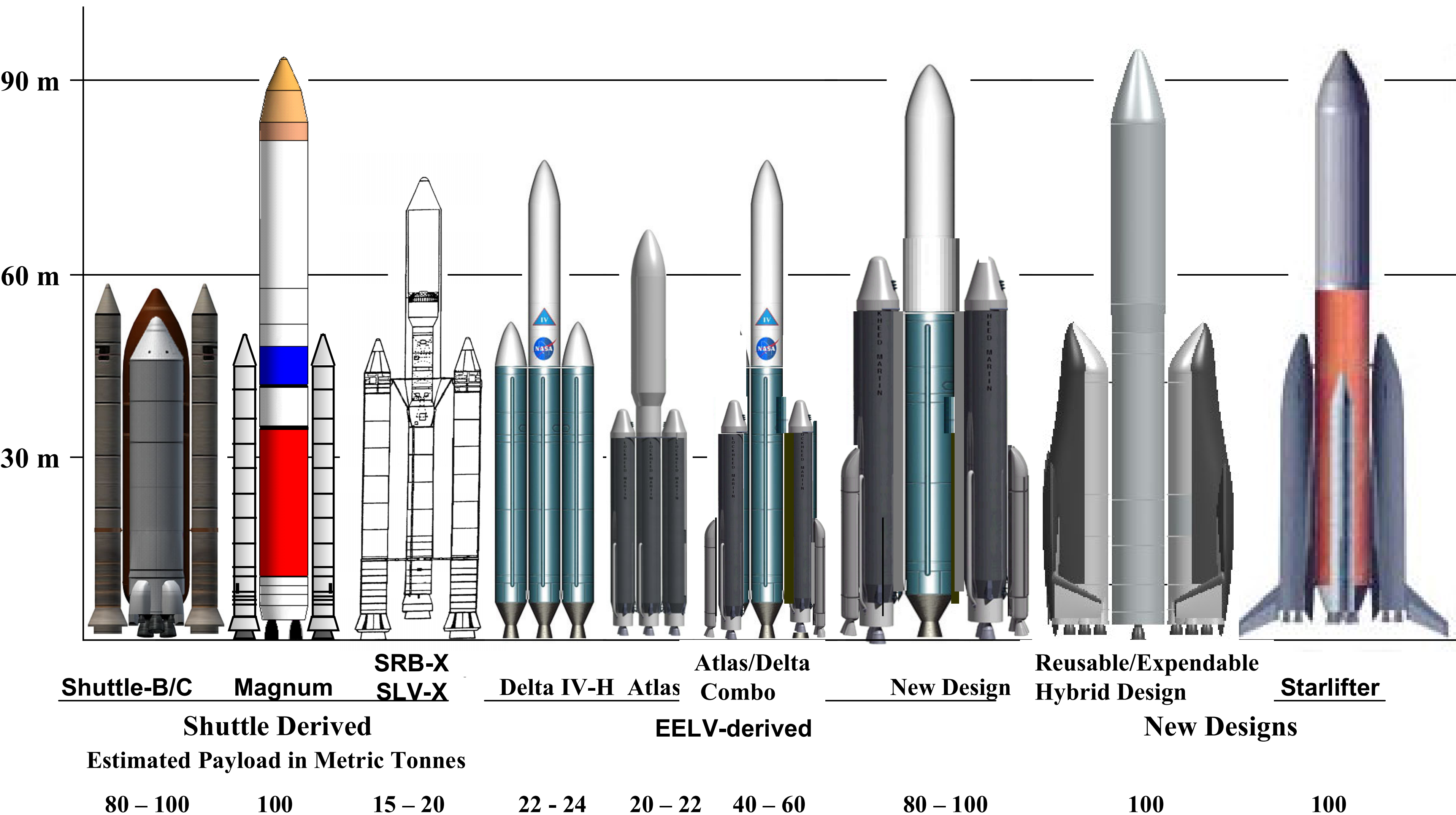
Existing Launch Vehicles



Payload to LEO @ 28.5° in Metric Tonnes

22(landing constraint) 8.6 10.7 20 5 8 8 – 13 25

Launch Vehicle Trade Space

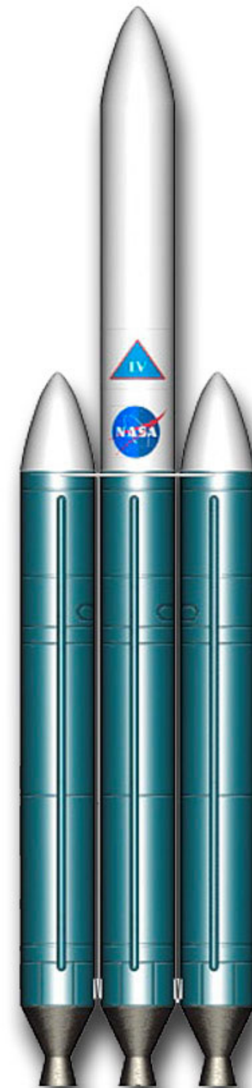


Heavy Lift Launch Vehicle (HLLV) Options

- *Shuttle-Derived HLLV Class* — *EELV Heavy Class*

Shuttle-Derived HLLV

Atlas V 552



Delta IV-H

Atlas V Heavy Launch Vehicle Configuration

Atlas V 552

Vehicle Configuration:

- Common Core Booster in Production
- Common Core Booster :
  - **Lox RP-1 (Kerosene)**
  - **RD-180 engine**
  - **933,370 lbf thrust (Vac) (1 engine)**
  - **I_{SP} 338 sec (Vac)**
- Payload: 20.6 mt (45k lbm) to 185 km Circ @ 28.5°
  17.0 mt (37k lbm) to 460 km Circ @ 51.6°
- Payload Faring: 17.7’ x 76.8’ (5.4m x 23.4m)

Upperstage:

- Pressure stabilized tanks
- **Cryogenic (Hydrogen) RL-10A-4-2**
  - **22,300 lbf thrust (2 engines)**
  - **I_{SP} 450 (Vac)**
- .021k mt (45,826 lbm) propellant loading
- Engine restart capability

Performance data limited to 6g’s, minor performance loss for 3g’s

Delta IV Heavy Launch Vehicle Configuration

Delta IV - H

Vehicle Configuration:

- Common Core Booster In Production
- Common Core Booster :
  - **Lox/Hydrogen**
  - **RS-68**
  - **745,000 lbf thrust (Vac) (1 engine per booster)**
  - **ISP 410 sec (Vac)**
- Payload: 22.5 mt (50k lbm) to 185 km Circ @ 28.5°
  22.5 mt (50k lbm) to 460 km Circ @ 51.6°
- Payload Faring:16.4’ x 65.0’ (5.0m x 19.8m)

Upper Stage:

- RL-10B-2
  - **Lox/Hydrogen**
  - **24,750 lbf thrust (2 engines)**
  - **ISP 466 sec (Vac)**
- .027k mt (60,000 lbm) propellant loading
- Engine restart capability

Performance data limited to 6g’s, minor performance loss for 3g’s

Shuttle-Derived HLLV Configuration

Representative Shuttle-Derived HLLV

Vehicle Characteristics

Cargo Only	93.5 mt (.206 mlbs)
	85.0 mt 9.187 mlb
Payload (56 x 278 km @ 28.5°)	
Payload Faring	25’ X 90’ (7.62m x 27.43m)

Booster (5-segment): not in production

Propellants	**HTPB (Solid Propellant)**
Sea Level Thrust	**3.33 mlb each**
Sea Level Isp	**265 sec**

External Tank (SLWT w/ 5 ft stretch):

Propellants	**LO2/LH2**
Engines	**3 SSME Engines (104%)**
Vacuum thrust	**492 klb ea Vac Isp= 453 sec**
Sea Level thrust	**397 klb ea SL Isp = 365 sec**