Launch Vehicles

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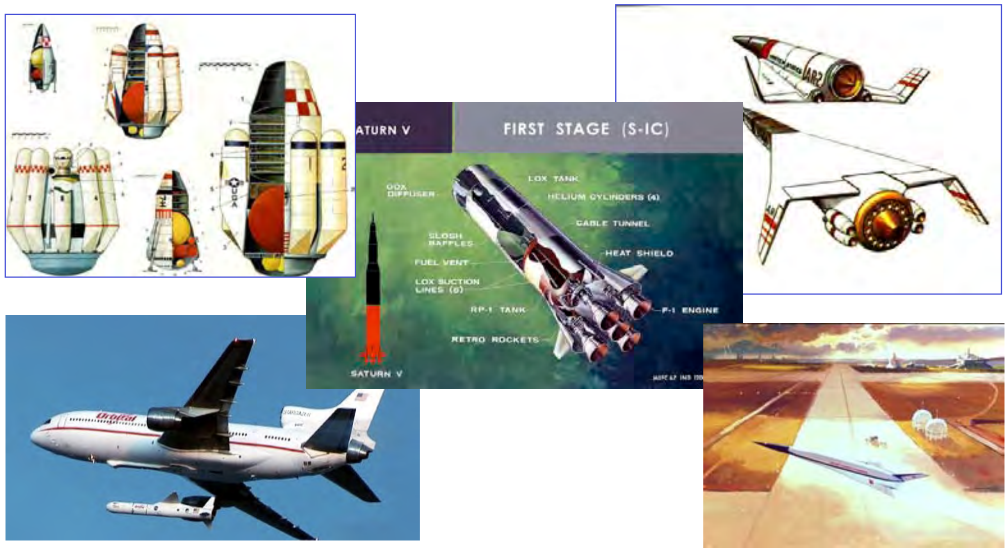
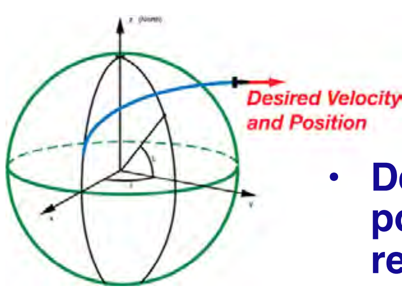
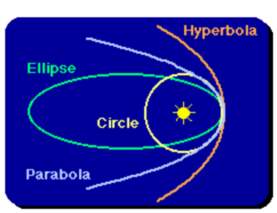


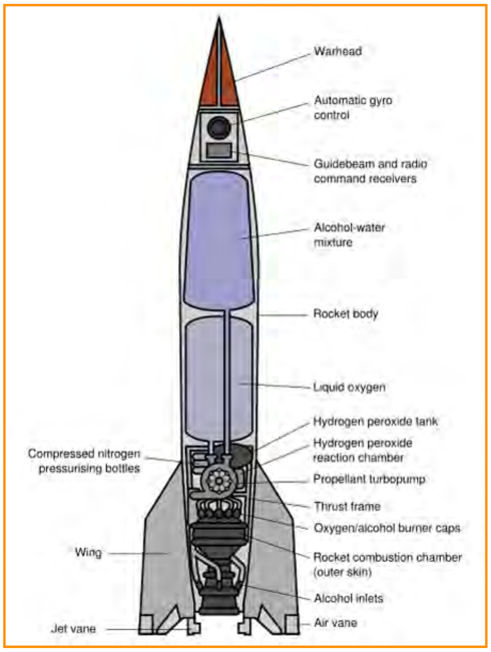
Launch Goals



- **Deliver payload to desired position and velocity, safely, reliably, and on time**
  - **Low earth orbit: ~24,000 ft/s = ~7.3 km/s**
  - **Escape: >36,000 ft/s = ~11 km/s**
- **Minimize launch cost and propellant use**
- **Minimize hazard to infrastructure and damage to environment**

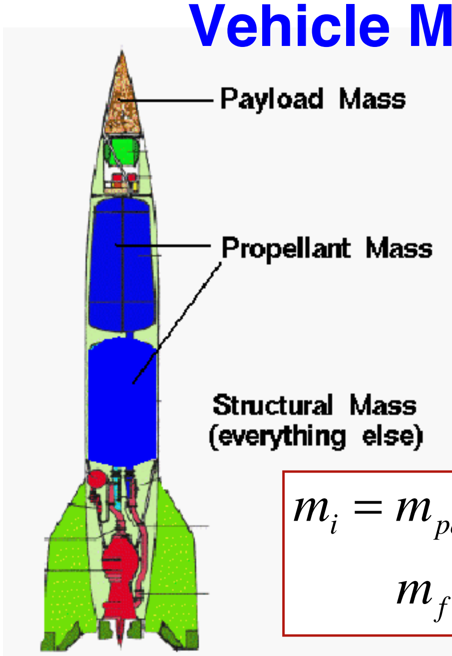


Launch Vehicle Systems



- **Propulsion and Power**
  - Main engines
  - Attitude-control thrusters
  - Retro-rockets
  - Ullage rockets
  - Turbo-pumps
  - Batteries, fuel cells
  - Pressurizing bottles
  - Escape/destruct systems
- **Structure**
  - Skin, frames, ribs, stringers, bulkheads
  - Propellant tanks
  - Fins, control surfaces
  - Inter-stage adapters, fairings
  - Heat shields, insulation
- **Electronics**
  - Guidance and control computers
  - Sensors and actuators
  - Radio transmitters and receivers
  - Radar transponders
  - Antennas
- **Reusable launchers/ orbiters**
  - Wings, parachutes
  - Landing gear
  - Orbital maneuvering units
  - Robot arms
  - Life support systems (manned vehicle)

Vehicle Mass Components



Initial and final masses of a single-stage rocket

$$m_i = m_{payload} + m_{structure/engine} + m_{propellant}$$

$$m_f = m_{payload} + m_{structure/engine}$$

Launch Vehicle Configuration Design Goals

- Minimum weight -> sphere
- Minimum drag -> slender body
- Minimum axial load -> low thrust
- Minimum lateral load -> sphere
- Minimum gravity loss -> high thrust

Configuration Design Goals

- Maximum payload -> lightweight structure, high mass ratio, multiple stages, high specific impulse
- Perceived simplicity, improved range safety -> single stage
- Minimum cost -> low-cost materials, economies of scale
- Minimum environmental impact -> non-toxic propellant

https://en.wikipedia.org/wiki/Comparison_of_orbital_launch_systems

The Rocket Equation

Ideal velocity increment of a rocket stage, ΔV_I (gravity and aerodynamic effects neglected)

$$\left(V_f - V_i\right) \equiv \Delta V_I = I_{sp} g_o \ln\left(\frac{m_i}{m_f}\right) \equiv I_{sp} g_o \ln \mu$$

$$m_i = m_{payload} + m_{structure/engine} + m_{propellant}$$
$$m_f = m_{payload} + m_{structure/engine}$$

$$\mu \triangleq \frac{m_i}{m_f} : \quad \text{Mass Ratio}$$

$$\Delta V_I = I_{sp} g_o \ln\left(\frac{m_i}{m_f}\right) = I_{sp} g_o \ln \frac{m_f + m_{propellant}}{m_f}$$

Ideal Velocity Increment for Single Stage with Various Specific Impulses

Ideal Velocity Increment, km/s

Mass Ratio	Isp = 220 s	= 275 s	= 400 s	= 500 s	= 850 s
2	1.50	1.90	2.70	3.40	5.78
3	2.40	3.00	4.30	5.30	9.16
4	3.00	3.80	5.40	6.80	11.56
5	3.50	4.30	6.30	7.90	13.42

Single stage to orbit with payload (ΔV_I~ 7.3 km/s)?

$$\mu_{required} = e^{\Delta V_I / I_{sp} g_0}$$

Required Mass Ratio for Various Velocity Increments

$$\mu_{required} = e^{\Delta V_I / I_{sp} g_0}$$

Ideal Velocity Increment, km/s	Required Mass Ratio Isp = 240 s	= 400 s
7	19.6	6.0
8	29.9	7.7
9	45.7	9.9
10	69.9	12.8
11	106.9	16.5
12	163.5	21.3

... and there are velocity losses due to gravity and aerodynamic drag

Ratios Characterizing a Rocket Stage

Mass ratio

$$\mu = m_{initial} / m_{final}$$

Payload ratio (as large as possible)

$$\lambda = m_{payload} / m_{initial}$$

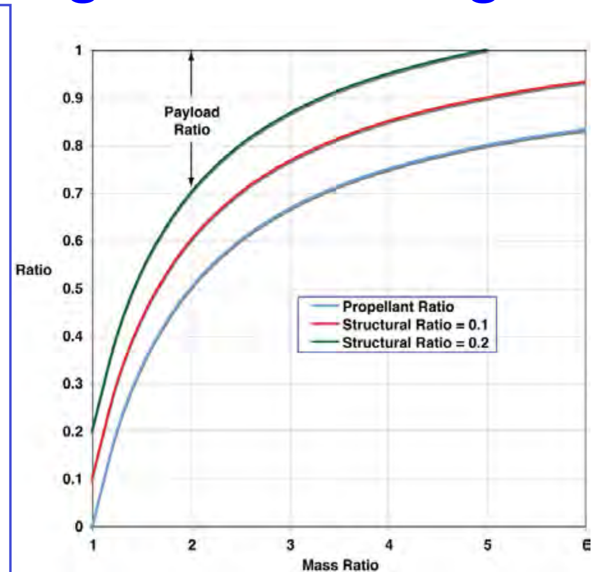
Structural ratio (typically 0.1 - 0.2)

$$\eta = m_{structure/engine} / m_{initial}$$

Propellant ratio

$$\varepsilon = m_{propellant} / m_{initial} = \frac{\mu - 1}{\mu}$$

$$\lambda + \eta + \varepsilon = 1$$



Payload is what's left after propellant and structure are subtracted

Ideal Velocity Increment for a Two-Stage Rocket

For each stage

$$\Delta V_{I_j} = I_{sp_j} g_o \ln \frac{m_{pay_j} + m_{s/e_j} + m_{prop_j}}{m_{pay_j} + m_{s/e_j}}, \quad j = 1,2$$

For both stages

$$\Delta V_I = \Delta V_{I_1} + \Delta V_{I_2}$$

$$= I_{sp_1} g_o \ln \frac{m_{pay_1} + \left(m_{s/e_1} + m_{prop_1}\right)}{m_{pay_1} + m_{s/e_1}} + I_{sp_2} g_o \ln \frac{m_{pay_2} + m_{s/e_2} + m_{prop_2}}{m_{pay_2} + m_{s/e_2}}$$

$$= I_{sp_1} g_o \ln \frac{m_{init_2} + \left(m_{s/e_1} + m_{prop_1}\right)}{m_{init_2} + m_{s/e_1}} + I_{sp_2} g_o \ln \frac{\left(m_{pay_2} + m_{s/e_2} + m_{prop_2}\right)}{m_{pay_2} + m_{s/e_2}}$$

$$= g_o \left[I_{sp_1} \ln \mu_1 + I_{sp_2} \ln \mu_2 \right]$$

Ideal Velocity Increment for a Multiple-Stage Rocket

- **Ideal velocity increment of an *n*-stage rocket**

$$\Delta V_I = g_o \sum_{j=1}^{n} I_{sp_j} \ln \mu_j$$

- **With equal specific impulses**

$$\Delta V_I = I_{sp} g_o \ln\left(\mu_1 \bullet \mu_2 \bullet \ldots \mu_n\right) \equiv I_{sp} g_o \ln\left(\mu_{overall}\right)$$

$$= I_{sp} g_o \ln \mu^n$$

Required Mass Ratios for Multiple-Stage Rockets

- Staging reduces ***individual mass ratios*** to achievable values
- With equal specific impulses for each stage

Required Individual Mass Ratio

Ideal Velocity Increment, km/s	Single Stage Isp = 240 s	= 400 s	= 850 s	Two Stages Isp = 240 s	= 400 s	= 850 s	Three Stages Isp = 240 s	= 400 s	= 850 s
7	19.55	5.95	2.32	4.42	2.44	1.52	2.69	1.81	1.32
8	29.90	7.68	2.61	5.47	2.77	1.62	3.10	1.97	1.38
9	45.72	9.91	2.94	6.76	3.15	1.72	3.58	2.15	1.43
10	69.92	12.79	3.32	8.36	3.58	1.82	4.12	2.34	1.49
11	106.92	16.50	3.74	10.34	4.06	1.93	4.75	2.55	1.55
12	163.50	21.29	4.22	12.79	4.61	2.05	5.47	2.77	1.62

https://en.wikipedia.org/wiki/Multistage_rocket

Mass-Ratio Effect on Final Load Factor

- Thrust-to-weight ratio = load factor

$$\frac{Thrust}{Weight} = n\,(load\ \ factor) = \frac{Thrust}{mg_o}$$

$$n_{initial} = \frac{Thrust}{m_{initial}g_o};\quad n_{final} = \frac{Thrust}{m_{final}g_o}$$

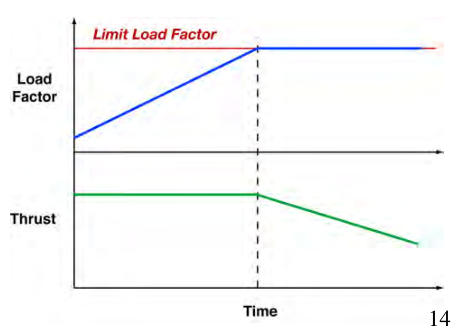
- If thrust is constant

$$\frac{n_{final}}{n_{inital}} = \frac{m_{initial}}{m_{final}} = \mu$$

- If thrust is reduced, limit load factor can be enforced

Final Load Factor, g

Initial Load Factor	Mass Ratio = 2	Mass Ratio = 5
1.3	2.6	6.5
2	4	10
3	6	15



Overall Payload Ratio of a Multiple-Stage Rocket

$$\lambda_{overall} = \frac{\left(m_{payload}\right)_n}{\left(m_{initial}\right)_1} = \frac{\left(m_{payload}\right)_n}{\left(m_{initial}\right)_n} \bullet \frac{\left(m_{payload}\right)_{n-1}}{\left(m_{initial}\right)_{n-1}} \bullet \ldots \frac{\left(m_{payload}\right)_1}{\left(m_{initial}\right)_1}$$
$$= \lambda_1 \bullet \lambda_2 \bullet \ldots \lambda_n$$

Feasible design goal: Choose stage mass ratios to maximize overall payload ratio

Payload Ratios of a Two-Stage Rocket

- **For equal specific impulses**

$$\Delta V_I = I_{sp} g_o \left[\ln \mu_1 + \ln \mu_2\right]$$
$$= I_{sp} g_o \left[\ln \mu_1 \mu_2\right] = I_{sp} g_o \left[\ln \mu_{overall}\right]$$

- **Payload ratios for different structural ratios**

$$\lambda_1 = \frac{1}{\mu_1} - \eta_1 = \frac{1 - \mu_1 \eta_1}{\mu_1} \qquad \lambda_2 = \frac{1 - \mu_2 \eta_2}{\mu_2}$$

Maximum Payload Ratio of a Two-Stage Rocket

Overall payload ratio

$$\boxed{\lambda_{overall} = \lambda_1 \lambda_2 = \frac{(1-\mu_1\eta_1)(1-\mu_2\eta_2)}{\mu_{overall}}}$$

Condition for a maximum with respect to first stage mass ratio

$$\boxed{\frac{\partial \lambda_{overall}}{\partial \mu_1} = \frac{\left(-\eta_1 + \frac{\mu_{overall}\eta_2}{\mu_1^{\ 2}}\right)}{\mu_{overall}} = 0}$$

Maximum Payload Ratio of a Two-Stage Rocket

Optimal stage mass ratios

$$\boxed{\mu_1 = \sqrt{\mu_{overall} \, \eta_2 / \eta_1}; \quad \mu_2 = \sqrt{\mu_{overall} \, \eta_1 / \eta_2}}$$

Optimal payload ratio

$$\boxed{\lambda_{overall} = \frac{1}{\mu_{overall}} - 2\sqrt{\eta_1\eta_2 / \mu_{overall}} + \eta_1\eta_2}$$

Also see

http://www.princeton.edu/~stengel/Prop.pdf

Scout Launch Vehicle
(1961-1994)

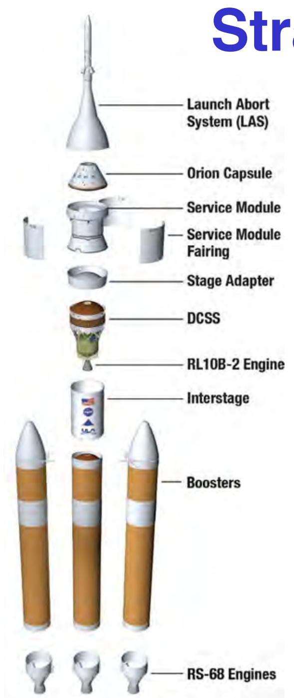
- Liftoff mass = 16,450 kg
- 4 solid-rocket stages

- Overall mass ratio = 34
- Overall payload ratio = 0.00425 = 0.425% (67-kg payload)

Typical Figures for Scout

Stage	Isp, s, vac (SL)	Mass Ratio	Payload Ratio	Structural Ratio	Impact Range, km
1 (Algol)	284 (238)	2.08	0.358	0.123	~60
2 (Castor)	262 (232)	2.33	0.277	0.152	~250
3 (Antares)	295	2.53	0.207	0.189	~2500
4 (Altair)	280	2.77	0.207	0.154	Orbit

Strap-On Boosters



- High volumetric specific impulse is desirable for first stage of multi-stage rocket
- Strap-on solid rocket boosters are a cost-effective way to increase mass and payload ratios

Expendable vs. Reusable Launch Vehicles

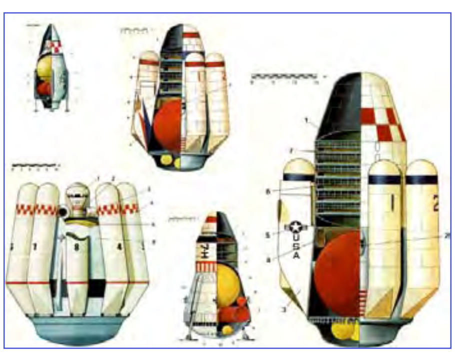
- **Expendable Vehicle**
  - Low cost per vehicle
  - New vehicle for each launch
  - Low structural ratio
  - Continued production
  - Launch preparation
  - Upgrade in production

- **Reusable Vehicle**
  - High initial cost
  - High structural ratio
  - Maintenance and repair
  - Non-reusable parts and supplies
  - Launch preparation
  - Return to launch site
  - Upgrade
  - Replacement cost

Heavy-Lift "Big Dumb Boosters" *c. 1963*

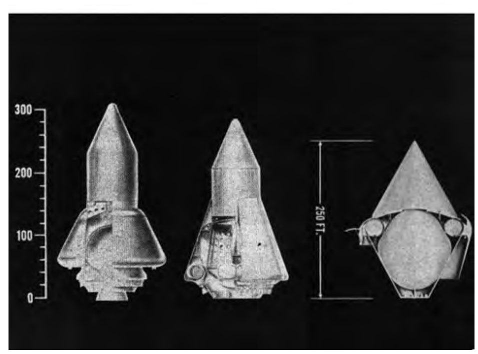
Objective: 450,000 kg to low earth orbit

Douglas Single-Stage-to-Orbit



- Plug nozzle
- Nozzle = Reentry Heat Shield
- Fully recoverable

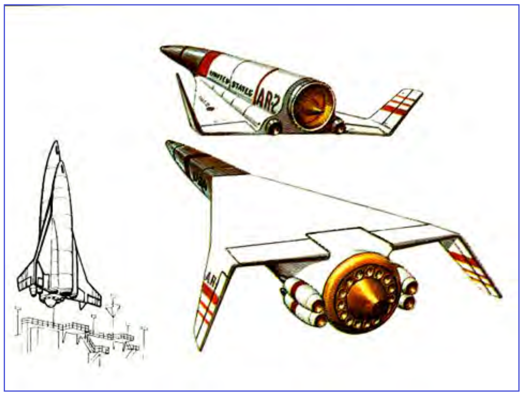
General Dynamics, Martin, and Douglas Concepts



- 1-1/2 stage, fully recoverable
- Recovery at sea
- Ducted rocket

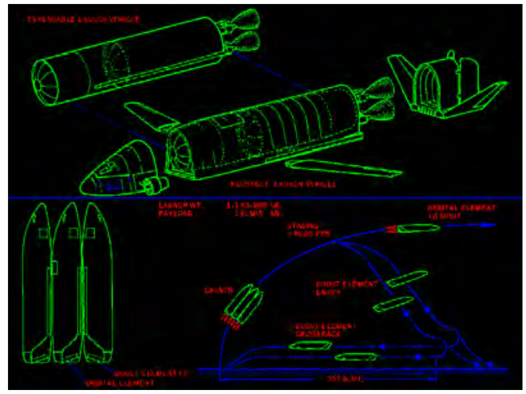
Vertical Takeoff, Horizontal Landing Vehicles

Martin Astro-Rocket



Heat shield-to-heat shield

General Dynamics Triamese



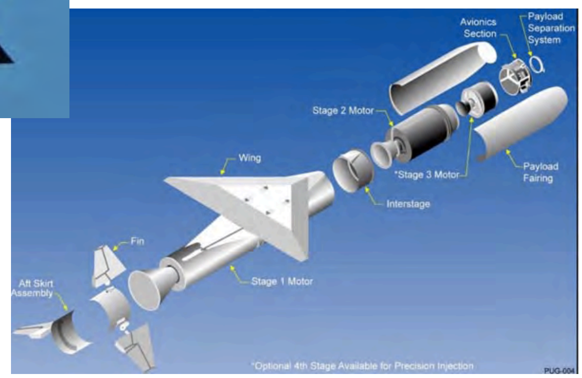
Three "identical" parallel stages

Horizontal vs. Vertical Launch

- **Feasibility of "airline-like" operations?**
- **Use of high I_{sp} air-breathing engines**
- **Rocket stages lifted above the sensible atmosphere**
- **Flexible launch parameters**

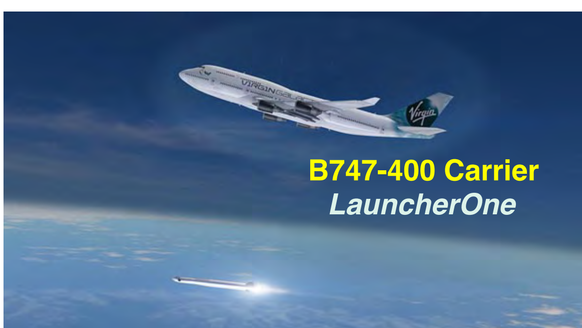
Pegasus Air-Launched Rocket

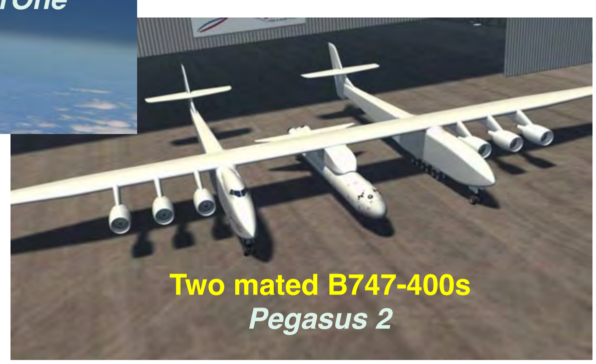
- Initial mass: 18,000 to 23,000 kg
- Payload mass: 440 kg



- Orbital-ATK
- Three solid-rocket stages launched from an aircraft
- Aerodynamic lift used to rotate vehicle for climb

Virgin Galactic LauncherOne and Vulcan Aerospace Stratolauncher





Specific Energy Contributed in Boost Phase

Total Energy = Kinetic plus Potential Energy (relative to flat earth)

$$\boxed{\mathbb{E} = \frac{mV^2}{2} + mgh}$$

Specific Total Energy = Energy per unit weight = Energy Height (km)

$$\boxed{\mathbb{E}' = \frac{V^2}{2g} + h}$$

Specific Energy Contributed in Boost Phase

Specific Energy contributed by first stage of launch vehicle

Less remaining drag loss (typical)

Plus Earth's rotation speed (typical)

	Altitude, km	Mach Number	Earth-RelativeVelocity, km/s	Remaining Drag Loss, km/s	Earth Rotation Speed, km/s	Specific Kinetic Energy, km	Total Specific Energy, km	Percent of Goal
Scout 1st-Stage Burnout	22	4	1.2	0.05	0.4	123.42	145.42	3.93%
Subsonic Horizontal Launch	12	0.8	0.235	0.15	0.4	12.05	24.05	0.65%
Supersonic Horizontal Launch	25	3	0.93	0.04	0.4	85.57	110.57	2.99%
Scramjet Horizontal Launch	50	12	3.6	0	0.4	829.19	879.19	23.74%
Target Orbit	300	25	7.4		0.4	3403.34	3703.34	

Trans-Atmospheric Vehicles
(Aerospace Planes)

- **Power for takeoff**
  - Turbojet/fans
  - Multi-cycle air-breathing engines/scramjets
  - Rockets
- **Single-stage-to-orbit**
  - Carrying dead weight into orbit
  - High structural ratio for wings, powerplants, and reusability
  - SSTO has very low payload ratio

Trans-Atmospheric Vehicle Concepts

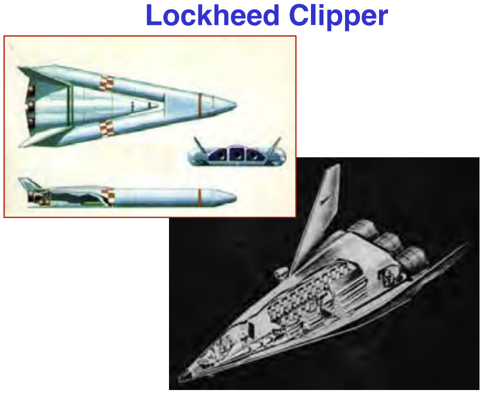
Various approaches to staging

Boeing TAV

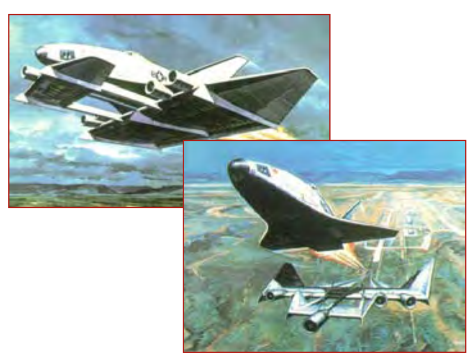
Rockwell TAV

Rockwell StarRaker

Lockheed Clipper

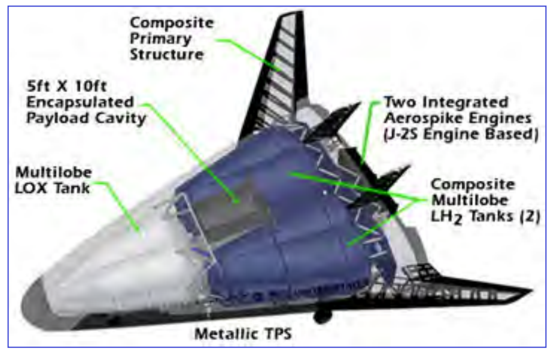


Lockheed TAV

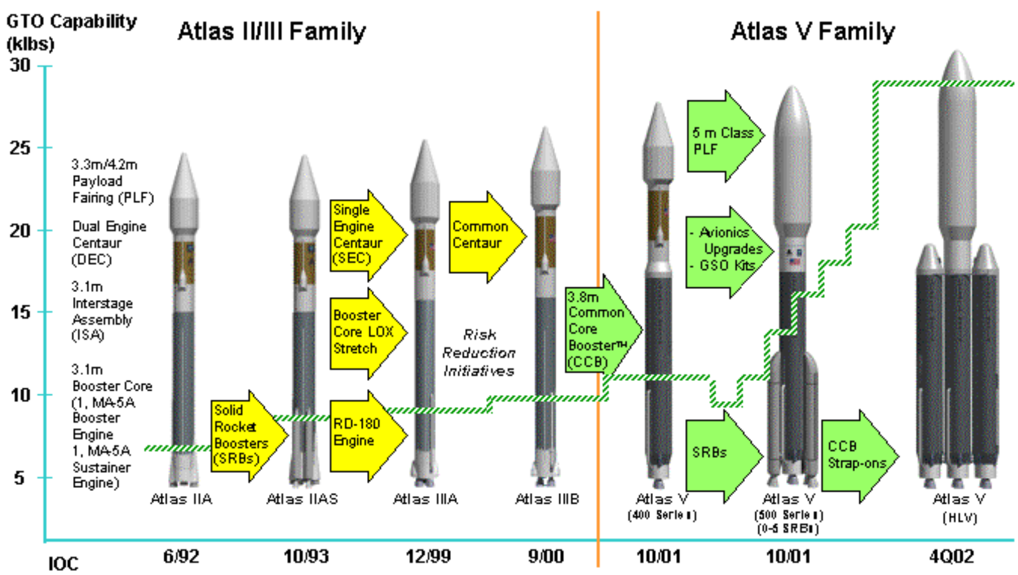


Venture Star/X-33

- Reusable, single-stage-to-orbit, proposed Space Shuttle replacement
- Advertised payload ratio = 2% (dubious)
- X-33: Sub-orbital test vehicle
- Improved thermal protection
- Linear spike nozzle rocket
- Program cancelled following tank failure in X-33 testing

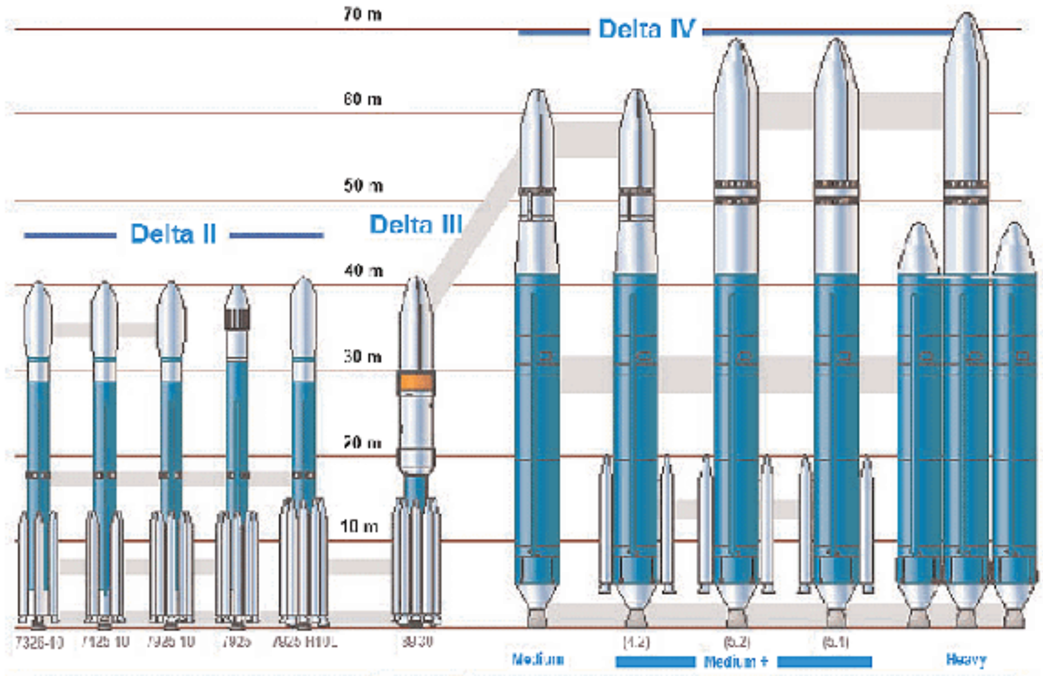


ULA Atlas Evolution



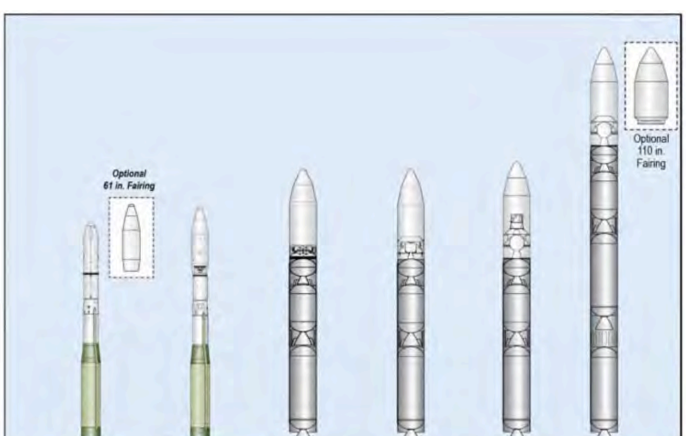
Implementing a Low Risk Evolution Process

ULA Delta Evolution



Orbital-ATK Minotaur and Antares

Minuteman ICBM Derivative



	Minotaur I	Minotaur I Lite	Minotaur IV Lite	Minotaur IV	Minotaur V	Minotaur VI
S1	M55A1 (GFE)	M55A1 (GFE)	SR-118 S1 (GFE)	SR-118 S1 (GFE)	SR-118 S1 (GFE)	SR-118 S1 (GFE)
S2	SR 19 (GFE)	SR 19 (GFE)	SR-119 S2 (GFE)	SR-119 S2 (GFE)	SR-119 S2 (GFE)	SR-118 S2 (GFE)
S3	Orion 50XL	Orion 50XL	SR-120 S3 (GFE)	SR-120 S3 (GFE)	SR-120 S3 (GFE)	SR-119 S3 (GFE)
S4	Orion 38	(N/A)	Super HAPS	Orion 38 (STAR 48BV Optional)	STAR 48BV	SR-120 S4 (GFE)
S5	HAPS (Optional)	(N/A)	(N/A)	HAPS (Optional)	STAR 37	STAR 48BV
S6	(N/A)	(N/A)	(N/A)	(N/A)	(N/A)	STAR 37 (Optional)*
Application	*Spacelift*	*Suborbital/Target*	*Suborbital/Target*	*Spacelift*	*MTO/GTO/Lunar*	*LEO/MTO/GTO/Lunar*
Performance:						
	584 kg to LEO	680 kg Ballistic	3000 kg Ballistic	1591 kg to LEO (1837 kg w/Star 48BV)	650 kg to MTO 532 kg to GTO 342 kg to TLI	3144 kg to LEO 1025 kg to MTO* 866 kg to GTO* 560 kg to TLI*

LEO = 200 km, 28.5°
Ballistic = VAFB to RTS (6,600 km, 20° Reentry Angle)

RD-181 Motor

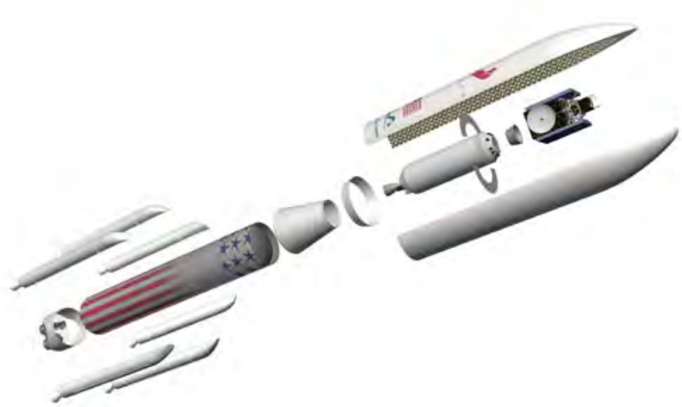
10/28/2014 Failure ***https://www.youtube.com/watch?v=9V1_BiTkHJ4***

Blue Origin New Shepard and OTS

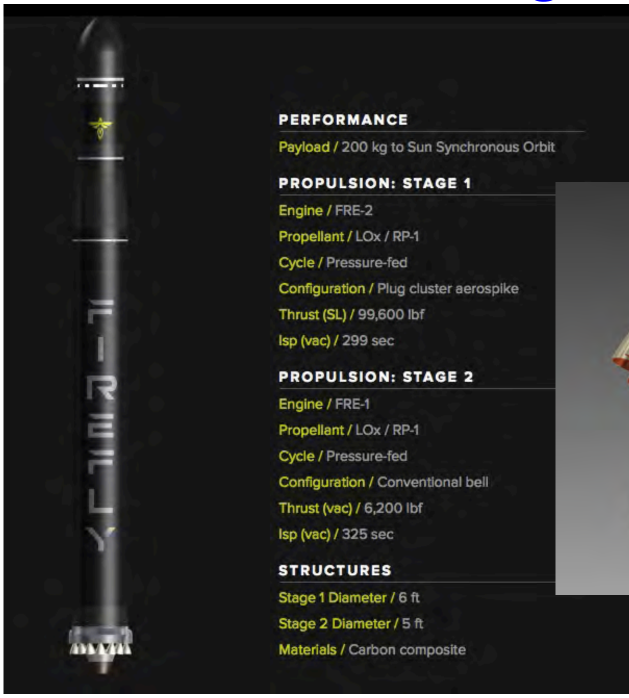
ULA Vulcan Launcher

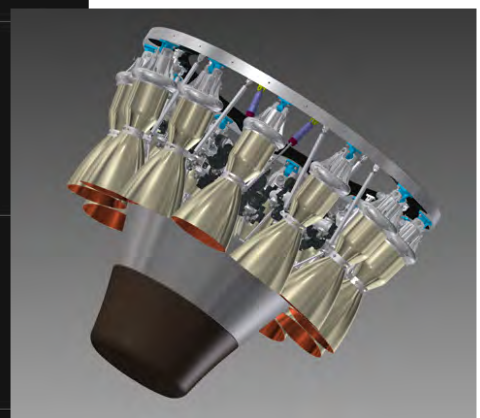
Successor to Delta 4 Heavy and Atlas Heavy

- 0 – 6 Orbital-ATK solid-rocket boosters
- LOX/Methane 1st stage, derived from Delta 4 (2 Blue Origin BE-4)
- LOX/LH2, Centaur or ACES 2nd stage (1 or 4 Aerojet Rocketdyne RL-10C)

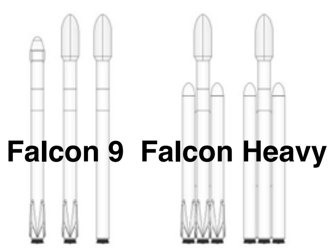


Firefly Launch Vehicle and Aero-Plug Motor



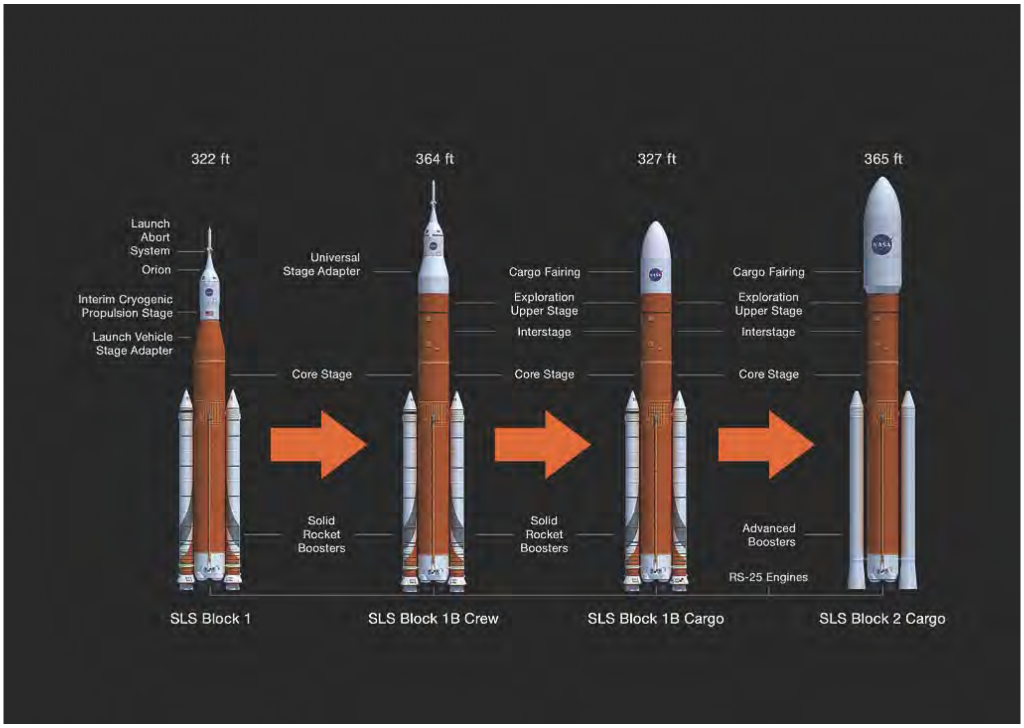


SpaceX Falcon 9, Heavy



- **Falcon Heavy**
  - **Payload to Low Earth Orbit – 53,000 kg**
  - **Payload to Geosynchronous Orbit = 21,200 kg**
  - **Liftoff Mass = 1.462 x 10^6**
  - **2 stages, plus 2 boosters, all LOX/RP-1**
  - **27 Merlin 1D (SL) 1^{st} stage rockets**
  - **1 Merlin 1D (vac) 2^{nd} stage rocket**

Space Launch System



Launch Vehicle Structural Loads

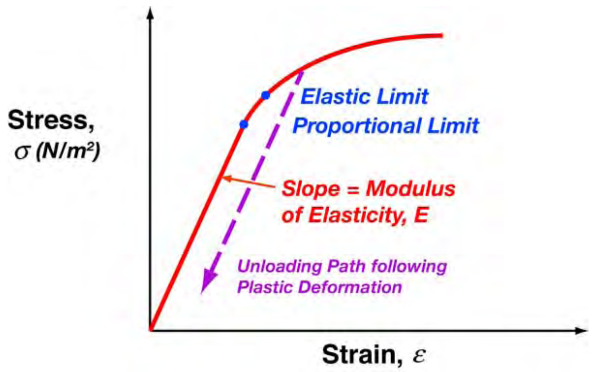
- Static/quasi-static loads
  - Gravity and thrust
  - Propellant tank internal pressure
  - Thermal effects
    - Rocket
    - Cryogenic propellant
    - Aerodynamic heating
- Dynamic loads
  - Bending and torsion
  - “Pogo” oscillations
  - Fuel sloshing
  - Aerodynamics and thrust vectoring
- Acoustic and mechanical vibration loads
  - Rocket engine
  - Aerodynamic noise

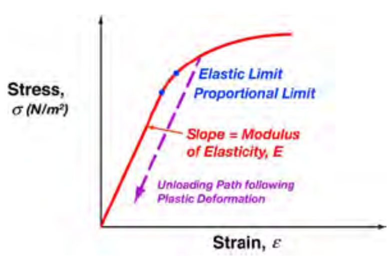
Structural Material Properties

- **Stress, σ: Force per unit area**
- **Strain, ε: Elongation per unit length**

$$\sigma = E\,\varepsilon$$

Proportionality factor, *E*: Modulus of elasticity, or Young's Modulus
Strain deformation is reversible below the elastic limit
Elastic limit = yield strength
Proportional limit ill-defined for many materials

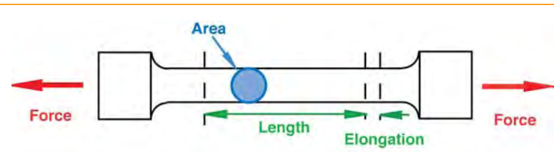




Structural Material Properties

Material Properties (Wikipedia)

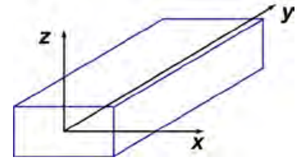
	Young's Modulus, GPa	Elastic Limit, MPa	Density, g/cm^3
Aluminum Alloy	69	400	2.7
Carbon-Fiber Composite	530	-	1.8
Fiber-Glass Composite	125-150	-	2.5
Magnesium	45	100	1.7
Steel	200	250-700	7.8
Titanium	105-120	830	4.5



Structural Stiffness

- **Geometric stiffness of a structure that bends about its *x* axis is portrayed by its area moment of inertia**

$$I_x = \int_{z_{\min}}^{z_{\max}} x(z) z^2 \, dz$$

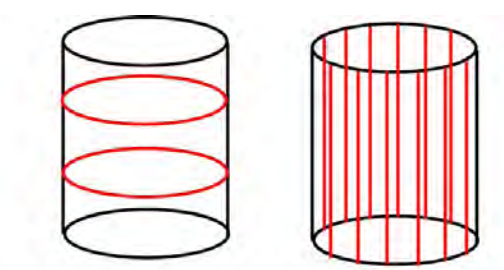


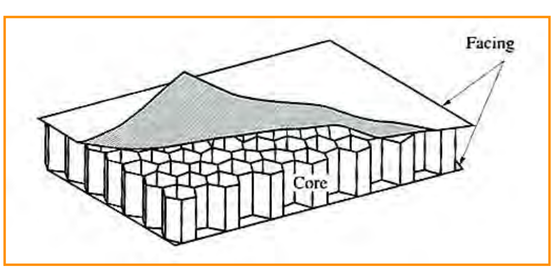
Area moment of inertia for simple cross-sectional shapes

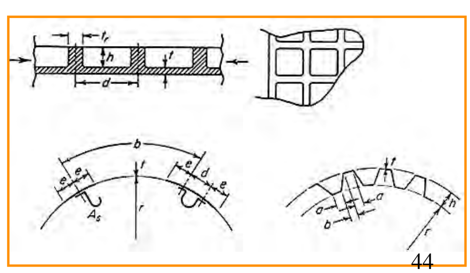
- **Solid rectangle of height, *h*, and width, *w*:** $I_x = wh^3/12$
- **Solid circle of radius, *r*:** $I_x = \pi r^4/4$
- **Circular cylindrical tube with inner radius, r_i, and outer radius, r_o:** $I_x = \pi\left(r_o^4 - r_i^4\right)/4$

Structural Stiffeners

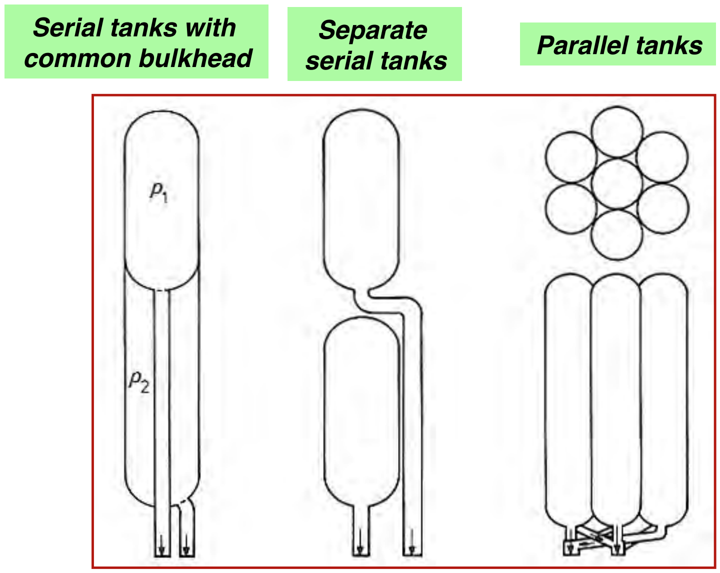
- **Axial stiffeners provide high I_x per unit of cross-sectional area**
- **Circular stiffeners increase resistance to buckling**
- **Honeycomb and waffled surfaces remove weight while retaining I_x**



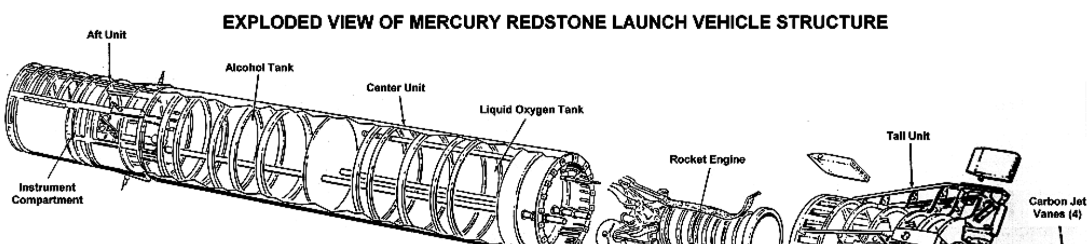




Propellant Tank Configurations for Launch Vehicles



Mercury-Redstone Structure

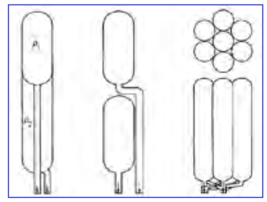


Semi-monocoque structure (load-bearing skin stiffened by internal components)

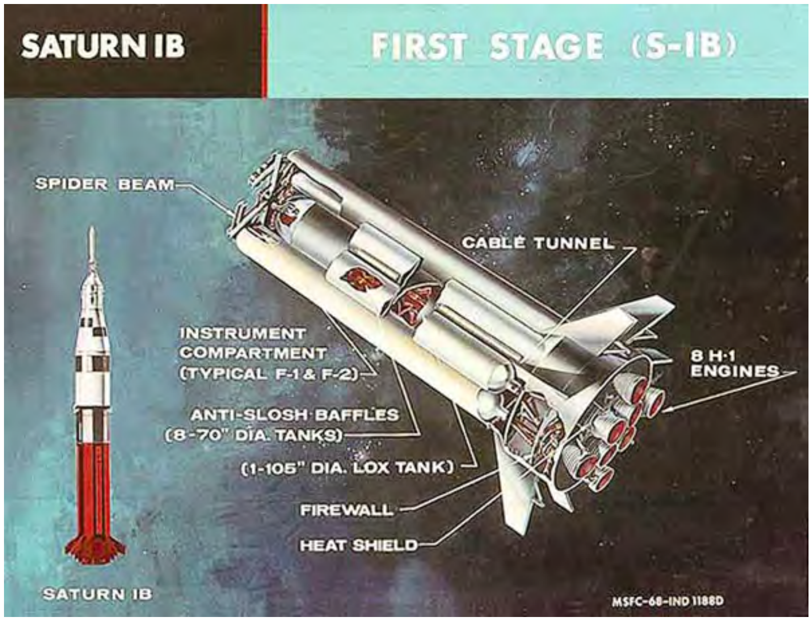
External skin, internal tanks separated by longerons and circular stiffeners

Aerodynamic and exhaust vanes for thrust vectoring

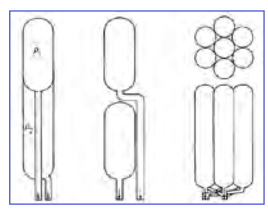
Saturn IB First Stage



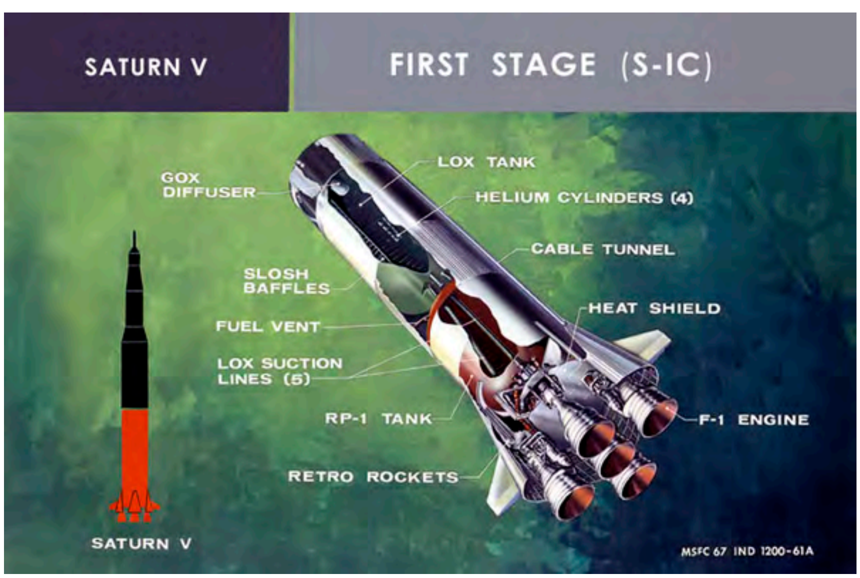
Parallel tanks, external bracing



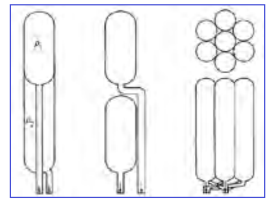
Saturn V First Stage



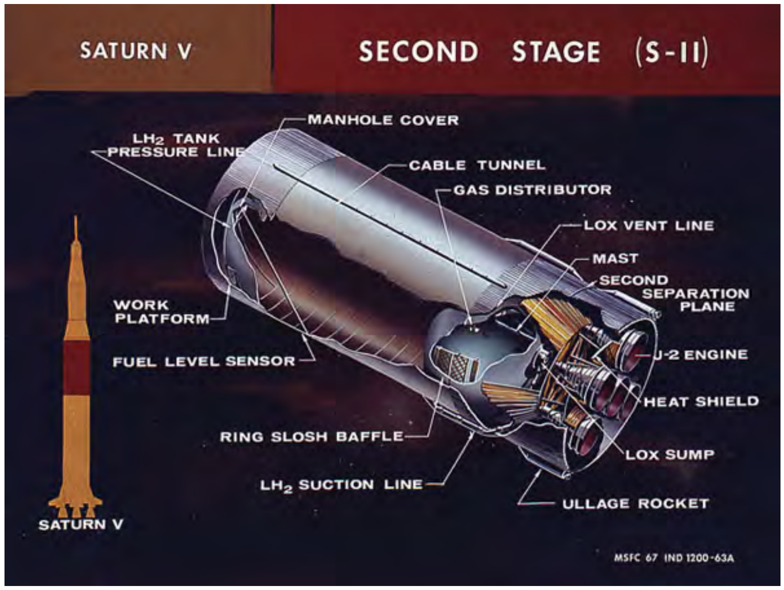
Separate serial tanks within semi-monocoque structure



Saturn V Second Stage

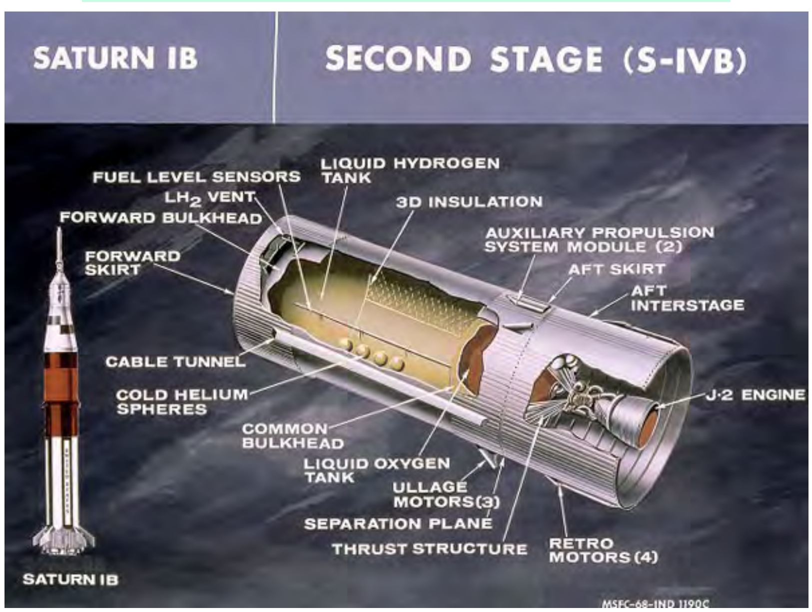


Integral serial tanks, with common bulkhead



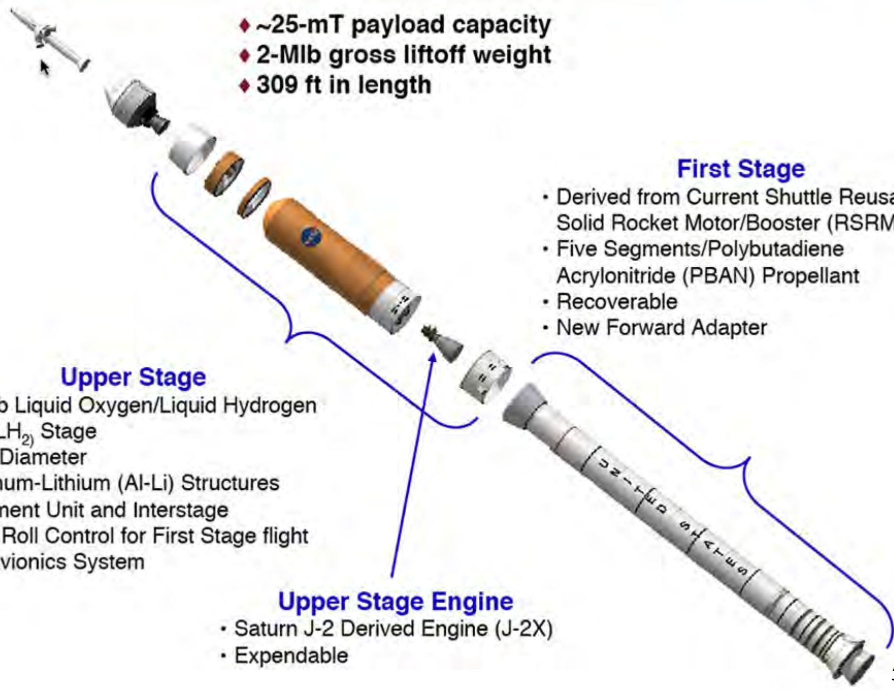
Saturn IB Second Stage/ Saturn V Third Stage

Serial tanks, with common bulkhead



Ares I Crew Launch Vehicle

- ~25-mT payload capacity
- 2-Mlb gross liftoff weight
- 309 ft in length



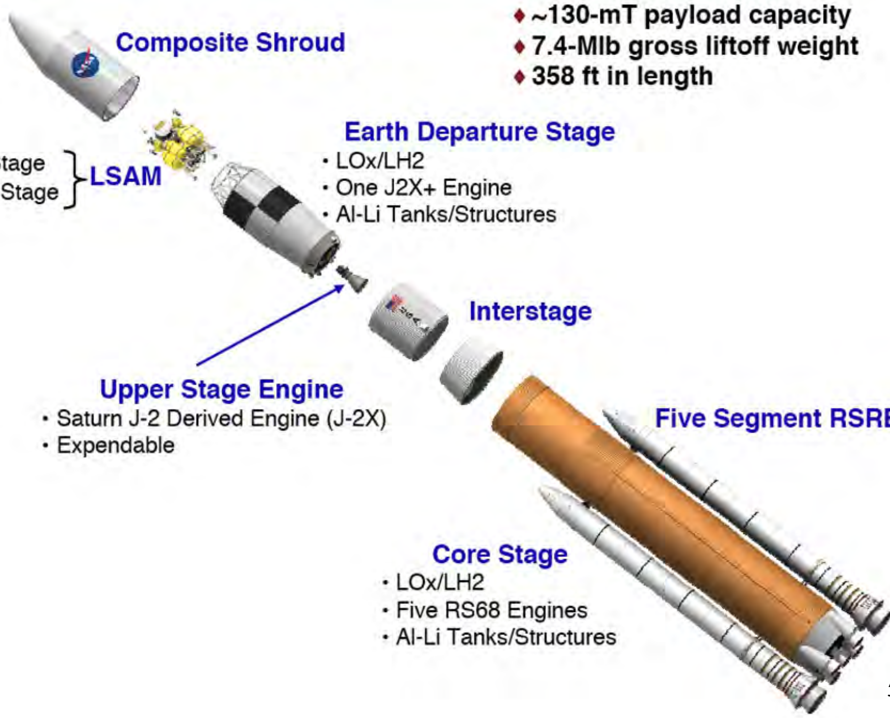
First Stage
- Derived from Current Shuttle Reusable Solid Rocket Motor/Booster (RSRM/B)
- Five Segments/Polybutadiene Acrylonitride (PBAN) Propellant
- Recoverable
- New Forward Adapter

Upper Stage
- 280-klb Liquid Oxygen/Liquid Hydrogen (LOX/LH_2) Stage
- 5.5-m Diameter
- Aluminum-Lithium (Al-Li) Structures
- Instrument Unit and Interstage
- RCS / Roll Control for First Stage flight
- CLV Avionics System

Upper Stage Engine
- Saturn J-2 Derived Engine (J-2X)
- Expendable

Ares V Cargo Launch Vehicle

- ~130-mT payload capacity
- 7.4-Mlb gross liftoff weight
- 358 ft in length



Composite Shroud

Ascent Stage
Descent Stage
} **LSAM**

Earth Departure Stage
- LOx/LH2
- One J2X+ Engine
- Al-Li Tanks/Structures

Interstage

Upper Stage Engine
- Saturn J-2 Derived Engine (J-2X)
- Expendable

Five Segment RSRBs

Core Stage
- LOx/LH2
- Five RS68 Engines
- Al-Li Tanks/Structures

Next Time:
Spacecraft Structures