Heavy Lift Launch
How Boosters Work, Their History, And The Role Of Heavy Lift In Space Commercialization

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Booster Systems: Why Lift Systems and Their Planning are Critical to the Commercialization of Space

The current crisis in American human spaceflight has generated a great deal of concern and confusion in the general public, as well as in the aerospace sector, regarding booster systems, how they are developed, and what might be the best path forward for the United States and the rest of the world.

While there is little argument that the United States needs to have diverse and cost effective lift capability, there is debate regarding whether the approach to this necessary system is a national asset or an international asset with national components. This discussion has political, policy technical and commercial aspects that also have significant implications for the commercial space sector.

The human rated launch system, in particular, is facing a serious predicament. The resolution and resultant policy implications have serious long term consequences.

Simply put, the ability of the United States to play a significant role in the commercialization of space would be considerably limited if it did not have adequate launch capabilities, and further, the idea that commerce and trade are fundamental to the sustainability of a nation is accepted as a fact.

Therefore, in this chapter we begin with a discussion of the technical mechanics of booster systems, and then consider the policy implications and choices as they pertain to the development of commercial space.

The Physics: How a Launch Booster System Works

The intent of this section is to acquaint the reader with the terminology of launch systems. It is based on the work of Robert A. Braeunig, and of course a more detailed version may be found at his web site.[1] Most classical texts on rocket propulsion use similar nomenclature.

Providing a concise overview here requires an extensive degree of simplification, so the reader who is knowledgeable about the underlying science and technology is asked to forgive the omissions. The intent is to support an effective discussion of boosters, their history and their likely evolution in the future, particularly with respect to how commercialization, which heretofore was not a factor, makes its impact felt.

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1 http://www.braeunig.us/space/propuls.htm
"Rocket Propulsion", compiled and edited by Robert A. Braeunig, 1997, 2005, 2007, 2009. This web site is also the source of Figures 1 – 6.
For the interested reader, more complete information on rocket propulsion can be found in the classic reference, *Aircraft and Missile Propulsion, Volume I and II: Thermodynamics of Fluid Flow and Application to Propulsion Engines*, by M.J. Zucrow, Professor of Gas Turbines and Jet Propulsion, Purdue University, 1958, which was at one time a standard college course reference book. Although long out of print, it is sometimes available on-line for purchase.
The reader may also refer to *Elements of Propulsion, Gas Turbines and Rockets*, by J.D. Mattingly, 2006, AIAA Education Series, AIAA, and *Aerothermodynamics of Gas Turbine and Rocket Propulsion*, 3rd ed., by G.C. Oates, 1997, AIAA Education Series, AIAA.

Rocket Engines and Thrust

A typical rocket engine consists of the nozzle, the combustion chamber, and the injector, as shown in Figure 1. Thrust is the force that propels a rocket or spacecraft, and is measured in pounds, kilograms, or Newtons. Physically speaking, thrust is the result of pressure that is exerted on the wall of a combustion chamber.

Figure 1 shows a combustion chamber with an opening, the nozzle, through which gas can escape. The pressure distribution within the chamber is asymmetric; that is, inside the chamber the pressure varies little, but near the nozzle it decreases somewhat. The force due to gas pressure on the bottom of the chamber is not compensated for from the outside. The resultant force, F, is therefore due to the difference between internal and external pressure, and the resulting thrust occurs in the direction opposite to that of the gas jet, and pushes the chamber upwards.

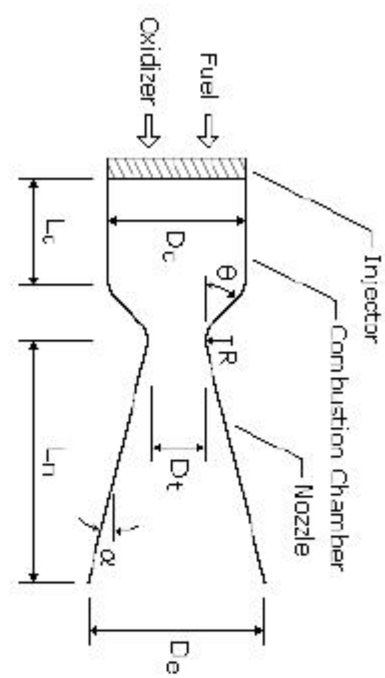

Figure 1
A Typical Rocket Engine

The burning of propellants takes place at high pressure in the combustion chamber. The chamber must be strong enough to contain the high pressure generated by, and the high temperature resulting from, the combustion process. Because of the high temperature, the chamber and nozzle are usually cooled. The chamber must also be of sufficient length (Lc) to ensure complete combustion before the gases enter the nozzle.

Nozzle

The function of the nozzle is to convert the chemical-thermal energy generated in the combustion chamber into kinetic energy. The nozzle converts the slow moving, high pressure, high temperature gas in the combustion chamber into high velocity gas of lower pressure and temperature. Since thrust is the product of mass and velocity, a very high gas velocity is desirable. Nozzles consist of a convergent and divergent section. The minimum flow area between the convergent and divergent section is called the nozzle throat. The flow area at the end of the divergent section is called the nozzle exit area. The nozzle is usually made long enough (or the exit area is great enough) such that the pressure in the combustion chamber is reduced at the nozzle exit to the pressure existing outside the nozzle. It is under this condition, Pe=Pa where Pe is the pressure at the nozzle exit and Pa is the outside ambient pressure, that thrust is maximum and the nozzle is said to be adapted. This is also called optimum or correct expansion. When Pe is greater than Pa, the nozzle is under-extended. When the opposite is true, it is over-extended.

Therefore, because atmospheric pressures vary by altitude, a nozzle is designed for the altitude at which it has to operate. At the Earth's surface, at the atmospheric pressure of sea level (0.1 MPa or 14.7 psi), the discharge of the exhaust gases is limited by the separation of the jet from the nozzle wall. In the cosmic vacuum, this physical limitation does not exist. Therefore, there have to be two different types of engines and nozzles, those that propel the first stage of the launch vehicle through the atmosphere, and those that propel subsequent stages or control the orientation of the spacecraft in the vacuum of space.

Specific Impulse

The specific impulse of a rocket, Isp, is the ratio of the thrust to the flow rate of the weight ejected, that is:

Isp = F / q g0

Where F is thrust, q is the rate of mass flow, and g0 is standard gravity (9.80665 m/s2).

Specific impulse is expressed in seconds. When the thrust and the flow rate remain constant throughout the burning of the propellant, the specific impulse is the time for which the rocket engine provides a thrust equal to the weight of the propellant consumed. Some typical values are given in Table 1.[2]

2 This is known as the Tsiolkovsky rocket equation.

Engine type	Scenario	SFC in lb/(lbf·h)	SFC in g/(kN·s)	Specific impulse (s)	Effective exhaust velocity (m/s)
NK-33 rocket engine	Vacuum	10.9	309	330	3,240
SSME rocket engine	Space shuttle vacuum	7.95	225	453	4,423
Ramjet	Mach 1	4.5	127	800	7,877
J-58 turbojet	SR-71 at Mach 3.2 (Wet)	1.9	53.8	1,900	18,587
Rolls-Royce/ Snecma Olympus 593	Concorde Mach 2 cruise (Dry)	1.195	33.8	3,012	29,553
CF6-80C2B1F turbofan	Boeing 747-400 cruise	0.605	17.1	5,950	58,400
General Electric CF6 turbofan	Sea level	0.307	8.696	11,700	115,000

Table 1
Specific Impulse of various rockets.
http://en.wikipedia.org/wiki/Specific_impulse

Note that for the different engine types, particularly engines that do not require or function on chemical reaction, the values and characteristics of the Isp are wildly different. Careful interpretation and examination of details is required to reach valid conclusions from this high level data.

Engine	Effective exhaust velocity (m/s, kg·m/s/kg)	Specific impulse (s)	Energy per kg of exhaust (MJ/kg)
Turbofan jet engine (actual V is ~300)	29,000	3,000	~0.05
Solid rocket	2,500	250	3
Bipropellant liquid rocket	4,400	450	9.7
Ion thruster	29,000	3,000	430
Dual Stage Four Grid Electrostatic Ion Thruster	210,000	21,400	22,500
VASIMR	290,000	30,000	43,000

Table 2
Effective exhaust velocities.
http://en.wikipedia.org/wiki/Specific_impulse

For a given engine, the specific impulse has different values on the ground and in the vacuum of space, because the ambient pressure is involved in the expression for the thrust. In evaluating engine performance it is therefore important to state whether specific impulse is the value at sea level, at high atmosphere, or in a vacuum.

Liquid Fueled Rocket Engines

Liquid fueled rocket engines and the associated booster systems require tanks for two commodities: fuel and oxidizer. The fuels are liquid phase as the name implies, but can be also cryogenic. Oxidizers may be oxygen, always cryogenically tanked, but may not involve oxygen at all if the propellant and oxidizer are what is known as hypergolic, or upon mixing, spontaneously decompose to 'combustion' products. Table 3 shows some of the more common fuel / oxidizer pairs.

The table also shows monopropellant entries. These are essentially decomposed over a catalyst bed or can be heated to cause 'combustion' or decomposition. All of these reactions are exothermic, giving off heat and energy to be turned mechanically into thrust.

Nitric acid / hydrazine-base fuel
Nitrogen tetroxide / hydrazine-base fuel
Hydrogen peroxide / RP-1 (including catalyst bed)
Liquid oxygen / RP-1
Liquid oxygen / ammonia
Liquid oxygen / liquid hydrogen (GH2 injection)
Liquid oxygen / liquid hydrogen (LH2 injection)
Liquid fluorine / liquid hydrogen (GH2 injection)
Liquid fluorine / liquid hydrogen (LH2 injection)
Liquid fluorine / hydrazine
Chlorine trifluoride / hydrazine-base fuel

Table 3
Typical Fuels For Liquid Rocket Engines.
Shown as Fuel/Oxidizer.

Liquid bi-propellant rocket engines can be categorized according to their power cycles, that is, how power is derived to feed propellants to the main combustion chamber. Described below are some of the more common types.

Gas-generator cycle: The gas-generator cycle, also called open cycle, taps off a small amount of fuel and oxidizer from the main flow (typically 3 to 7 percent) to feed a burner called a gas generator. The hot gas from this generator passes through a turbine to generate power for the pumps that sends propellants to the combustion chamber. The hot gas is then either dumped overboard, or sent into the main nozzle downstream.

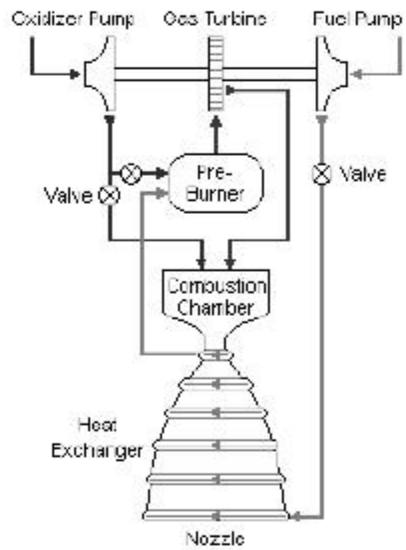

Figure 2
Staged Combustion

Increasing the flow of propellants into the gas generator increases the speed of the turbine, which increases the flow of propellants into the main combustion chamber, and hence, the amount of thrust produced.

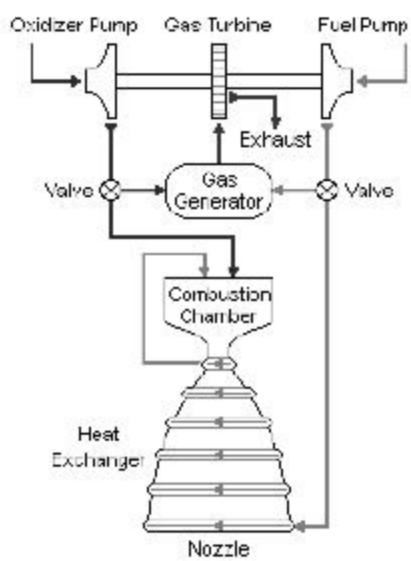

Figure 3
Gas Generator

The gas generator must burn propellants at a less-than-optimal mixture ratio to keep the temperature low for the turbine blades. Thus, the cycle is appropriate for moderate power requirements but not high-power systems, which would have to divert a large portion of the main flow to the less efficient gas-generator flow.

As in most rocket engines, some of the propellant in a gas generator cycle is used to cool the nozzle and combustion chamber, increasing efficiency and allowing higher engine temperature.

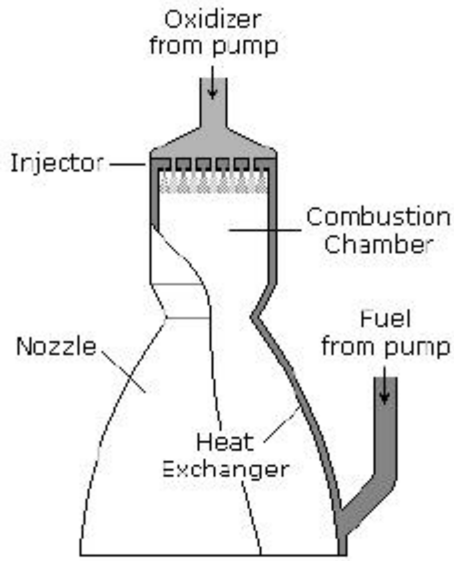

Figure 4
Regenerative Cooling

How The Engine Is Cooled

Regenerative cooling is the most widely used method of cooling a thrust chamber, and is accomplished by flowing high-velocity coolant over the back side of the chamber's hot gas wall to convectively cool the hot gas liner. The coolant with the heat input from cooling the liner is then discharged into the injector and utilized as a propellant. This is shown in Figure 4.

Earlier thrust chamber designs, such as the V-2 and Redstone, had low chamber pressure, low heat flux and low coolant pressure requirements, which could be satisfied by a simplified 'double wall chamber' design with regenerative and film cooling. For subsequent rocket engine applications, however, chamber pressures were increased and the cooling requirements became more difficult to satisfy. It was necessary to design new coolant configurations that were more efficient structurally and had improved heat transfer characteristics.

This led to the design of 'tubular wall' thrust chambers, by far the most widely used design approach for the vast majority of large rocket engine applications. These chamber designs have been successfully used for the Thor, Jupiter, Atlas, H-1, J-2, F-1, RS-27 and several other Air Force and NASA rocket engines. The primary advantage of the design is that it's lightweight and accrues a large experience base. As chamber pressures and hot gas wall heat fluxes have continued to increase (>100 atm), still more effective methods have been needed.

Solid Rocket Motors

Solid rockets motors store propellants in solid form. The fuel is typically powdered aluminum, and the oxidizer is ammonium perchlorate. A synthetic rubber binder such as polybutadiene holds the fuel and oxidizer powders together. Though lower performing than liquid propellant rockets, the operational simplicity of a solid rocket motor often makes it the propulsion system of choice. There is also an emerging technology that has the oxidizer injected into the solid propellant chamber and therefore the solid rocket motor can, unlike its oxidizer / fuel blended cousin, be turned on and off by use of the oxidizer control.

Solid Fuel Geometry

A solid fuel's geometry determines the area and contours of its exposed surfaces, and thus its burn pattern. There are two main types of solid fuel blocks used in the space industry, cylindrical blocks with combustion at a front, or surface, and cylindrical blocks with internal combustion. In the first case, the front of the flame travels in layers from the nozzle end of the block towards the top of the casing. This so-called ‘end burner’ produces constant thrust throughout the burn. In the second, more usual case, the combustion surface develops along the length of a central channel. Sometimes the channel has a star shaped, or other, geometry to moderate the growth of this surface.

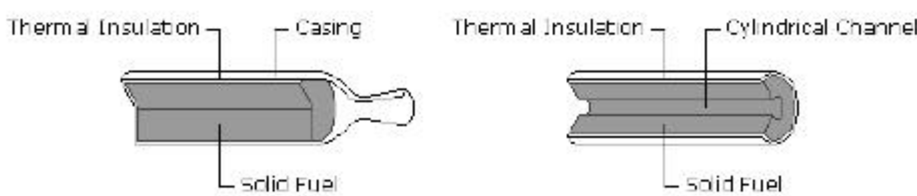

Figure 5
Solid Fuel Geometry

The shape of the fuel block for a rocket is chosen for the particular type of mission it will perform. Since the combustion of the block progresses from its free surface, as this surface grows, geometrical considerations determine whether the thrust increases, decreases or stays constant.

In Figure 6, we see fuel blocks with a cylindrical channel (1) develop their thrust progressively. Those with a channel and also a central cylinder of fuel (2) produce a relatively constant thrust, which reduces to zero very quickly when the fuel is used up. The five-pointed star profile (3) develops a relatively constant thrust that decreases slowly to zero as the last of the fuel is consumed. The ‘cruciform’ profile (4) produces progressively less thrust. Fuel in a block with a ‘double anchor’ profile (5) produces a decreasing thrust that drops off quickly near the end of the burn. The ‘cog’ profile (6) produces a strong initial thrust, followed by an almost constant lower thrust.

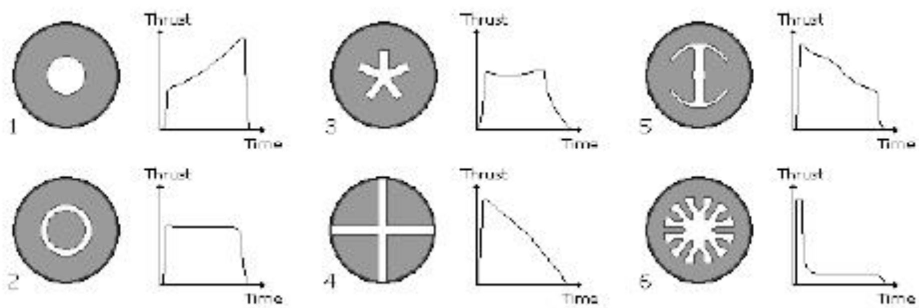

Figure 6
Solid Fuel Blocks

Burn Rate of Solid Rocket Boosters

Regression, typically measured in millimeters per second (or inches per second), is termed burn rate. This rate can differ significantly for different propellants, or for one particular propellant, depending on various operating conditions as well as formulation. Knowing quantitatively the burning rate of a propellant, and how it changes under various conditions, is of fundamental importance in the successful design of a solid rocket motor.

Propellant burning rate is influenced by many factors, the most significant being: combustion chamber pressure, initial temperature of the propellant grain, velocity of the combustion gases flowing parallel to the burning surface, local static pressure, and motor acceleration and spin. These factors are discussed below.

Burn rate is profoundly affected by combustion chamber pressure. The usual representation of the pressure dependence on burn rate is the Saint-Robert's Law,

$$r = a P_c^n$$

Where ‘r’ is the burn rate, ‘a’ is the burn rate coefficient, ‘n’ is the pressure exponent, and ‘Pc’ is the combustion chamber pressure. The values of ‘a’ and ‘n’ are determined empirically for a particular propellant formulation and cannot be theoretically predicted. It is important to realize that a single set of ‘a’ and ‘n’ values are typically valid over a distinct pressure range. More than one set may be necessary to accurately represent the full pressure regime of interest.

Example ‘a’ and ‘n’ values are 5.6059* (pressure in MPa, burn rate in mm/s) and 0.35 respectively for the Space Shuttle SRBs, which gives a burn rate of 9.34 mm/s at the average chamber pressure of 4.3 MPa or roughly 3000 lbs / in2

* NASA publications give a burn rate coefficient of 0.0386625 (pressure in PSI, burn rate in inch/s).

Monopropellant Engines

By far the most widely used type of propulsion for spacecraft attitude and velocity control is monopropellant hydrazine. Its excellent handling characteristics, relative stability under normal storage conditions, and clean decomposition products have made it the standard. The general sequence of operations in a hydrazine thruster is:

When the attitude control system signals for thruster operation, an electric solenoid valve opens, allowing hydrazine to flow. The action may be pulsed (as short as 5 ms) or long duration (steady state).

The pressure in the propellant tank forces liquid hydrazine into the injector, it then enters as a spray into the thrust chamber and contacts the catalyst beds.

The catalyst bed consists of alumina pellets impregnated with iridium. Incoming hydrazine heats to its vaporizing point by contact with the catalyst bed and with the hot gases leaving the catalyst particles. The temperature of the hydrazine rises to a point where the rate of its decomposition becomes so high that the chemical reactions are self-sustaining.

By controlling the flow variables and the geometry of the catalyst chamber, a designer can tailor the proportion of chemical products, the exhaust temperature, the molecular weight, and thus the enthalpy (heat content) for a given application. For a thruster application where specific impulse is paramount, the designer attempts to provide 30-40% ammonia dissociation, which is about the lowest percentage that can be maintained reliably. For gas-generator application, where lower temperature gases are usually desired, the designer provides for higher levels of ammonia dissociation.

Finally, in a space thruster, the hydrazine decomposition products leave the catalyst bed and exit from the chamber through a high expansion ratio exhaust nozzle to produce thrust.

Monopropellant hydrazine thrusters typically produce a specific impulse of about 230 to 240 seconds.

Other suitable propellants for catalytic decomposition engines are hydrogen peroxide and nitrous oxide. However, their performance is considerably lower than that obtained with hydrazine, a specific impulse of about 150 s with H_2O_2 and about 170 s with N_2O.

Monopropellant systems have successfully provided orbit maintenance and attitude control functions, but lack the performance to provide a large, weight-efficient change in velocity (ΔV) maneuvers required for orbit insertion.

Bipropellant systems are attractive because they can provide all three functions with one higher performance system, but they are more complex than the common solid rocket and monopropellant combined systems.

A third alternative is dual mode systems. These systems are hybrid designs that use hydrazine both as a fuel for high performance bipropellant engines and as a monopropellant with conventional low-thrust catalytic thrusters. The hydrazine is fed to both the bipropellant engines and the monopropellant thrusters from a common fuel tank.

Cold gas propulsion is just a controlled, pressurized gas source and a nozzle. It represents the simplest form of rocket engine. Cold gas has many applications where simplicity and/or the need to avoid hot gases are more important than high performance. The Manned Maneuvering Unit used by astronauts is an example of such a system.[3]

Staging of Multi-staged Boosters

Multistage rockets allow improved payload capability for vehicles with a high ΔV requirement, such as launch vehicles or interplanetary spacecraft.

In a multistage rocket, propellant is stored in smaller, separate tanks rather than a larger single tank as in a single-stage rocket. Since each tank is discarded when empty, energy is not expended to accelerate the empty tanks, so a higher total ΔV is obtained. Alternatively, a larger payload mass can be accelerated to the same total ΔV. For convenience, the separate tanks are usually bundled with their own engines, with each discardable unit called a stage.

Multistage rocket performance is described by the same rocket equation as single-stage rockets, but must be determined on a stage-by-stage basis. The velocity increment, ΔV_i, for each stage is calculated as before,

$$\Delta V_i = C_i \, LN\left(\frac{m_{oi}}{m_{fi}}\right)$$

Where 'm_{oi}' represents the total vehicle mass when stage 'i' is ignited, and 'm_{fi}' is the total vehicle mass when stage 'i' is burned out but not yet discarded.

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[3] Suggested reading on the simplicity of solid rockets coupled with some technological innovations, including a detailed review of the military interest, development and employment of the solid rocket launcher is found in *Journal Of Propulsion And Power*, Vol. 19, No. 6, November–December 2003; "Solid Rocket Enabling Technologies and Milestones in the United States," Leonard H. Caveny, Robert L. Geisler, Russell A. Ellis, and Thomas L. Moore, Chemical Propulsion Information Agency, Columbia, Maryland 21044.

It is important to realize that the payload mass for any stage consists of the mass of all subsequent stages plus the ultimate payload itself. The velocity increment for the vehicle is then the sum of those for the individual stages where n is the total number of stages.

$$\Delta V_{total} = \sum_{i=1}^{n} \Delta V_i$$

We define the payload fraction as the ratio of payload mass to initial mass, or mpl / mo.

For a multistage vehicle with dissimilar stages, the overall vehicle payload fraction depends on how the ΔV requirement is partitioned among stages. Payload fractions will be reduced if the ΔV is partitioned sub optimally. The optimal distribution may be determined by trial and error. A ΔV distribution is postulated and the resulting payload fraction calculated. The ΔV distribution is varied until the payload fraction is maximized.

After the selection of the ΔV distribution, vehicle sizing is accomplished by starting with the uppermost or final stage (whose payload is the actual deliverable payload) then calculating the initial mass of this assembly. This assembly then forms the payload for the previous stage and the process repeats until all stages are sized. Results reveal that to maximize payload fraction for a given ΔV requirement:

1. Stages with higher Isp should be above stages with lower Isp.
2. More ΔV should be provided by the stages with the higher Isp.
3. Each succeeding stage should be smaller than its predecessor, and
4. Similar stages should provide the same ΔV.

These design 'rules of thumb' are uniformly reflected in the discussion to follow, but with the caveat that the extent of application is modified by the optimization that was done on the stages and total system as a routine part of its design. Said another way, these rules of thumb will not provide sufficient guidance to build a customized launch booster. Mission profiles and side issues such as whether or not a system is 'human rated,' a term that is discussed later, are important drivers as well. Table 4 shows the relative performance of some well known launch systems.

The column labeled 'mass ratio' is the ratio of lift off to delivered mass. This might be considered in the future to be the metric for commercial launch systems as the ratio has a 1:1 relationship to the cost per pound of delivered mass to any orbit class, be it LEO, GEO, or lunar. Note that even 40+ years after successful missions in Apollo, the Saturn V (which was human-rated) is still nearly the performance of the Ariane V, (which was not human rated). That masterpiece of design, created essentially by the Peenemunde staff with support from the American industrial base, remains unrivaled, and will be a benchmark for a long time to come.

Vehicle	Takeoff Mass	Final Mass	Mass ratio	Payload fraction
Ariane 5 (vehicle + payload)	746,000 kg (~1,645,000 lb)	2,700 kg + 16,000 kg (~6,000 lb + ~35,300 lb)	39.9	0.975
Titan 23G first stage	117,020 kg (258,000 lb)	4,760 kg (10,500 lb)	24.6	0.959
Saturn V	3,038,500 kg (~6,700,000 lb)	13,300 kg + 118,000 kg (~29,320 lb + ~260,150 lb)	23.1	0.957
Space Shuttle (vehicle + payload)	2,040,000 kg (~4,500,000 lb)	104,000 kg + 28,800 kg (~230,000 lb + ~63,500 lb)	15.4	0.935
Saturn 1B (stage only)	448,648 kg (989,100 lb)	41,594 kg[3] (91,700 lb)	10.7	0.907
Virgin Atlantic GlobalFlyer	~181,000 kg (400,000 lb)	1,678.3 kg (3,700 lb)	6	0.83
V2	13,000 kg (~28,660 lb) (12.8 ton)	not available	3.85	0.74
X-15	15,420 kg (34,000 lb)	6,620 kg (14,600 lb)	2.3	0.57
Concorde	~181,000 kg (400,000 lb)	not available	2	0.5
Boeing 747	~363,000 kg (800,000 lb)	not available	2	0.5

Table 4

The relative performance of some well known launch systems.

The History of Booster Systems in the United States, and their Missions

Since their initial operational capability in the 1950s, space access launch vehicles have undergone nearly as many capability classification definitions as the vehicle series have had vehicle variants. Evolution of the vehicles has resulted in what today is generally accepted as 'rules of thumb' rather than clear distinctions with hard metrics. Moreover, orbital altitude, eccentricity, and inclination plane are independent variables that greatly effect on-orbit mass delivery for any launch system.

This is often perplexing, so the intent here is to develop an understanding of how a booster system, or space launch vehicle operates, linking this to the history of vehicle evolution.

Launch vehicle systems are tailored by the vendor to accommodate the greatest market share of their customer's target operational payloads and orbits. This accounts for much of the confusion when it comes to comparative vehicle capability based on top-level nomenclature.

There are three primary lift categories to consider as a baseline: low, medium, and heavy. However, there are only two lift categories to consider when dealing with access to space for human beings, namely medium and heavy lift.

For the sake of brevity, the discussion here centers upon payload delivery mass only, principally to circular Low Earth Orbit (LEO) at inclinations between 28 and 52 degrees. It is important to note that LEO metrics have direct bearing on accepted mission profiles for exploration beyond LEO, principally because payload mass is parked in LEO before transfer insertion to points beyond LEO, including the Moon and other Solar System destinations.

There are currently three American rocket systems capable of medium and heavy lift, Delta, Atlas, and Titan. (In addition, the Russian Proton and the ESA Ariane are also heavy lift systems.)

Atlas

Atlas V is an active expendable launch system. Atlas V was formerly operated by Lockheed Martin, and is now operated by the Lockheed Martin-Boeing joint venture United Launch Alliance.

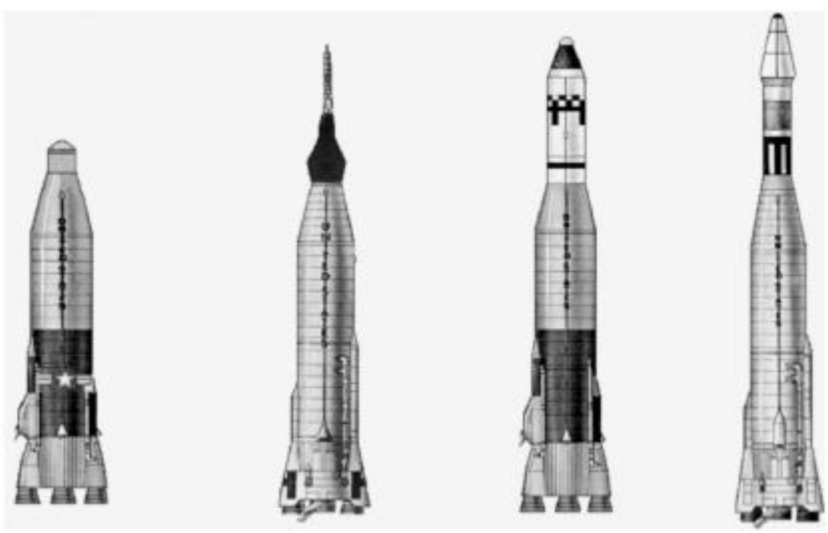

Figure 7
Early Atlas Family
Referred to as Atlas 1 or Atlas A-F Series

Each Atlas V rocket uses a Russian-built RD-180 engine burning kerosene and liquid oxygen to power its first stage and an American-built RL10 engine burning liquid hydrogen and liquid oxygen to power its upper stage.

Figure 8
Atlas 2 – 5 Family

Delta

The original Delta rockets used a modified version of the Thor, the first ballistic missile deployed by the United States, as their first stage. The Thor was designed in the mid-1950s. Subsequent satellite and space probe flights soon followed, using a Thor first stage with several different upper stages. The fourth upper stage used on the Thor was the Thor 'Delta,' delta being the fourth letter of the Greek alphabet. Eventually the entire Thor-Delta launch vehicle came to be called *"Delta."*

NASA intended Delta as *"an interim general purpose vehicle"* for communication, meteorological, and scientific satellites and lunar probes during the early 1960s, and planned to replace Delta with other rocket designs when they came on-line. Due to its reliability, however, Delta remains very much in use. Delta rockets are currently manufactured and launched, as with Atlas, by the United Launch Alliance.

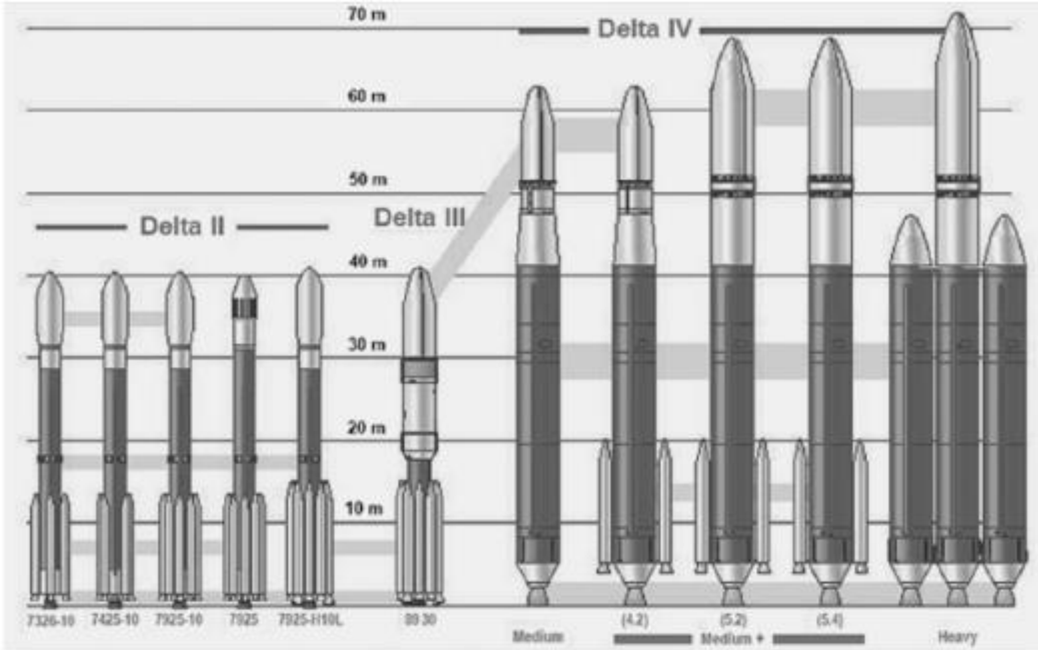

Figure 9
Delta Series (post IRBM)

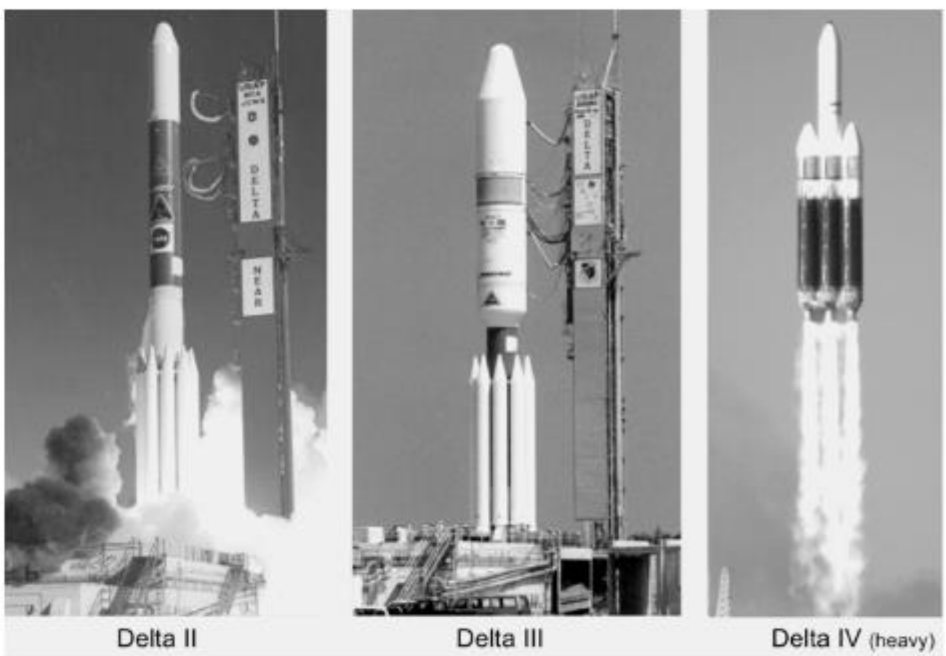

Figure 10
Delta Launch Vehicles Currently Operational

Figure 11
Delta Launch Vehicles Currently Operational

Titan

As of 2006, the Titan family of rockets is no longer in use. The high cost of using hydrazine and nitrogen tetroxide fuels, along with the special care that was needed due to their toxicity, proved too expensive compared to the higher-performance liquid hydrogen or RP-1-fueled vehicles (kerosene), with a liquid oxygen oxidizer. Titan is owned by the Lockheed Martin company, which decided to extend its Atlas family of rockets instead of its more expensive Titans. The final Titan was launched from Vandenberg Air Force Base on 19 October 2005, carrying a secret payload for the National Reconnaissance Office (NRO). There are about twenty Titan II rockets at the Aerospace Maintenance and Regeneration Center near Tucson, Arizona, that are set to either be scrapped or used as monuments

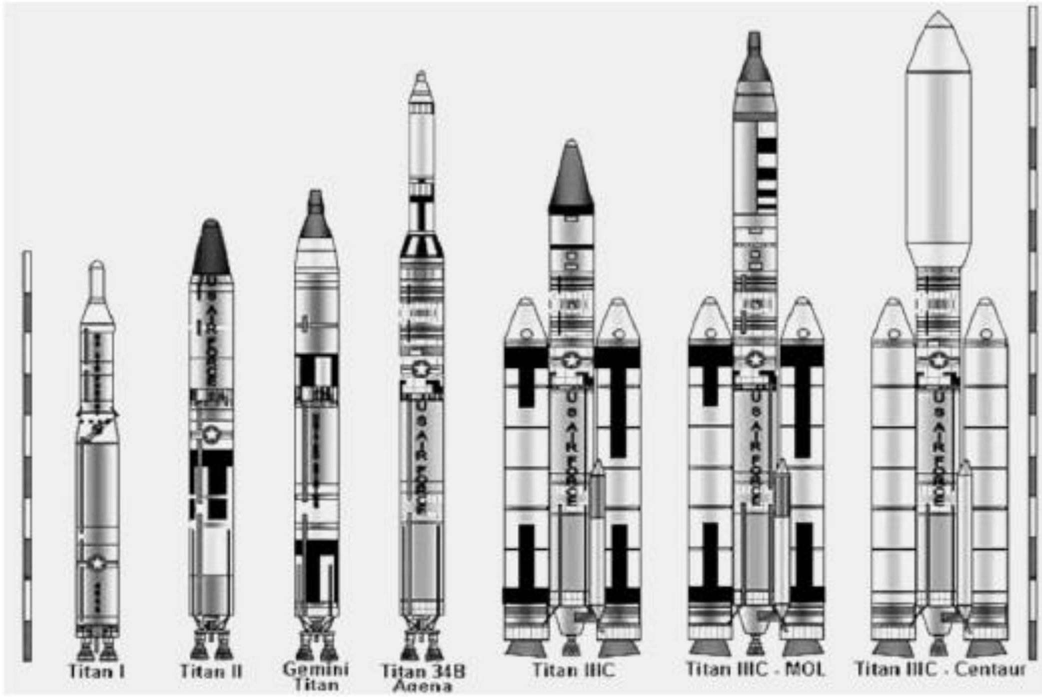

Figure 12
Titan Family of Missile and Launch Vehicles

Categories of Booster Rocket Systems

In and across the booster 'Lift Categories,' there are three Rules of Thumb that guided the definition of the booster systems.

The First Rule of Thumb: Medium and Heavy Lift

Traditional medium lift to LEO is roughly 10,000 – 40,000 lbs; plus or minus 5,000 lbs. 50,000 lbs is considered the absolute maximum in the 'traditional' medium lift category systems.

The medium lift category covers the most ubiquitous launch systems, which have collectively conducted a tremendous variety of diverse missions, including human spaceflight. They span nearly half a century of payloads from every conceivable customer base to virtually all points above Earth's atmosphere, including unmanned probes that have explored nearly all of the Sun's planets. Some have even ventured beyond the solar system itself.

Today, medium lift rocket propulsion systems are common, with several countries possessing domestically produced systems of similar capability.

However, some terminology pertaining to the first rule of thumb has morphed. Yesterday's 'medium lift' isn't always applied today, and the upper bracket of traditional medium lift is often referred to as 'heavy' lift.

The reason for this is that 'heavy' lift was a term that initially applied only to human exploration, principally beyond LEO, which is to say

Apollo. The first and arguably only true heavy lift rockets were the Saturn series, which were developed for the sole purpose of sending humans beyond LEO.

The originator of the early NASA booster systems was the military, which meant that the Saturn series of rockets set the heavy lift standard in both civilian and military space. For nearly half a century and still counting, the Saturn V remains unchallenged as the record for throw weight to LEO. It could place a whopping 250,000 lbs into a 28 degree inclination LEO orbit per launch.

In the decades since the Saturn first flew, 200,000 lbs to LEO has remained a traditional lunar reference mission minimum requirement for human exploration. The planned Ares V booster (estimated 300,000 lbs to LEO) substantially exceeds that. If funded for development, the Ares V would be, by far, the most powerful launch system ever developed, and would open up the entire solar system to human exploration. It would also provide a catalyst of unquestionable value to the prospects of space commercialization.

Looking at the Saturn as compared with other launch systems tells a story in itself, as seen in Figure 11.

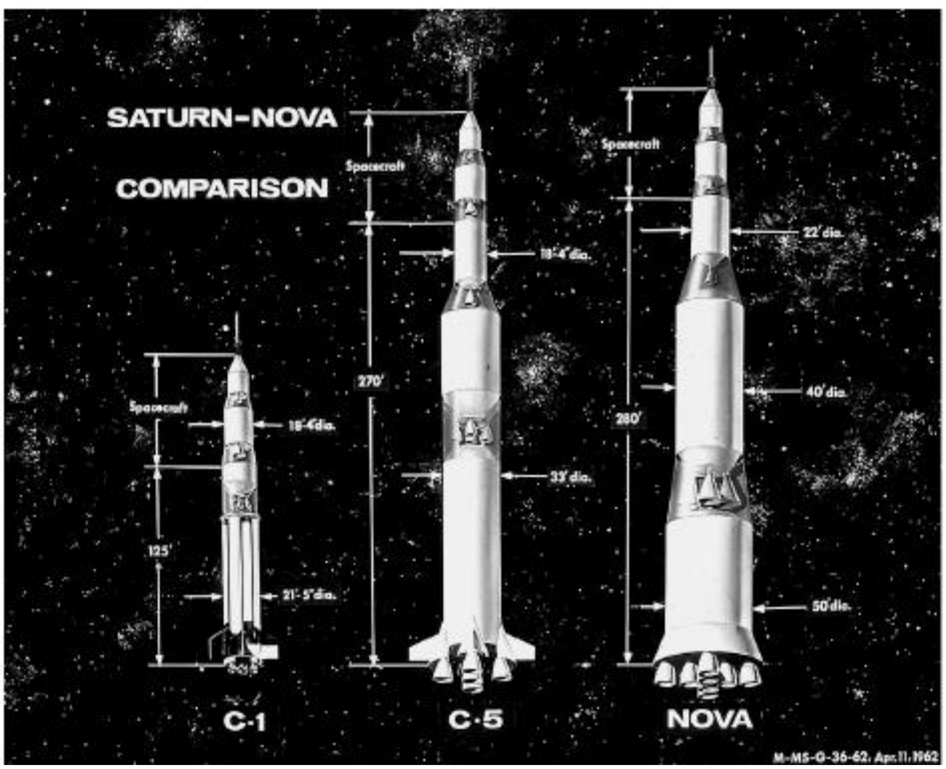

Figure 13
1962 graphic comparison of Saturn C-1, C-5 and proposed NOVA launch vehicles. NASA – Marshall Space Flight Center
http://mix.msfc.nasa.gov/IMAGES/HIGH/9902050.jpg

An interesting detail concerning the Saturn program is that the Saturn V was not the end of the evolutionary cycle for this behemoth. There was a planned Mars rocket that would have also been a very heavy lift capability for the lunar missions, called the Saturn VIII or *"Nova."* Few details remain as to its projected performance, but a visual comparison shown in Figure 13 hints that this is well over 500,000 lbs to LEO.

The second Rule of Thumb: Lift to Lunar

As noted, traditional heavy lift systems are based on the lunar exploration reference mission profiles established for Apollo, which is 200,000 lbs or more to LEO.

While the Space Shuttle stack is technically a heavy lift system, the Orbiter returns to earth, which means that the current Shuttle system can deliver a maximum on-orbit payload of less than 50,000 lbs. This is one of the driving reasons why cargo versions of the Shuttle stack (SSME's bolted to a payload can instead of to the Orbiter) were much studied in early ISS evaluations as well as in preliminary exploration systems architectures.

Traditional heavy lift terminology has also morphed since the retirement of the Saturn series in 1975. Based principally on military lift requirements, the upper limit of traditional medium lift category is increasingly referred to as 'heavy' lift and is often confused with exploration 'heavy' lift. This is an important distinction and serves to clarify Rule of Thumb 3.

The third Rule of Thumb: Lift to LEO

Human access to ISS – or comparable adventures in the commercial arena - from U.S. facilities requires 30,000 – 50,000 lbs of lift to LEO.

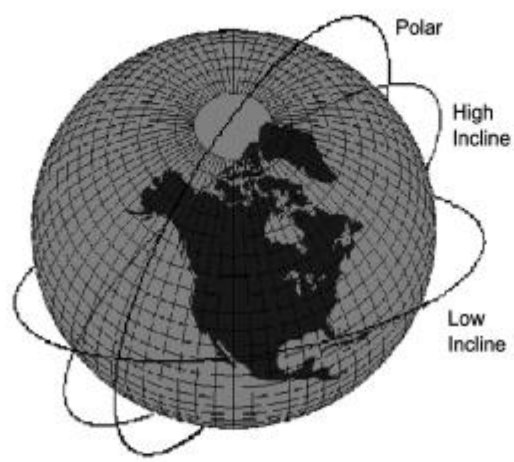

Figure 14
Earth Orbits
High inclination orbits require more energy to achieve compared with low incline orbits, because the Earth's spin helps a spacecraft achieve orbit at low incline.

Lift requirements for human access to ISS complicates the LEO scenario because of the space station's relatively high orbital inclination (51.6 degrees) and the consequential additional energy required to get there when compared to low inclination LEO orbital planes (U.S. traditional 28 degrees).

This is the flashpoint of discussion for U.S. based human launch systems because with the cancellation of Ares I, only two American launch systems outside of further Shuttle service could conceivably transport crews to ISS – Delta IV Heavy and Atlas V Heavy.

Neither of these vehicles are Saturn class at this time, although there is much discussion over maximum developmental limits that, if realized, could make both Delta and possibly Atlas true Saturn class launch vehicles. Figure 15, taken from Boeing literature on Delta IV, shows what could be done to 'grow the family' to very high useable payload size.

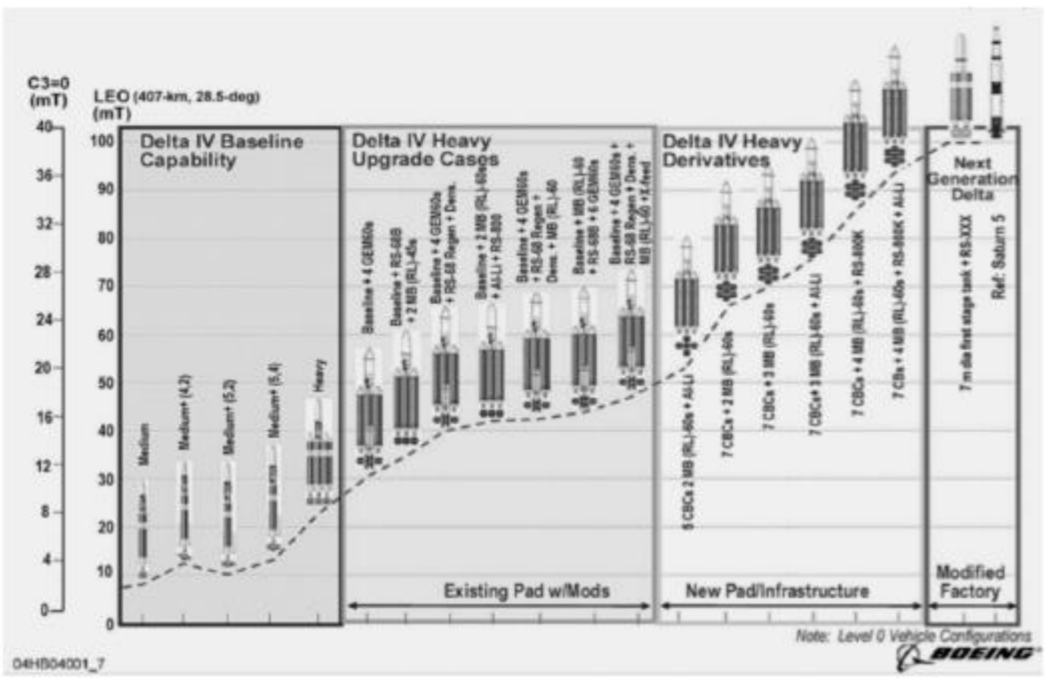

Figure 15
Delta IV Baseline and Upgrades

For the time being, it is sufficient to say that any vehicle capable of delivering 50,000 lbs to the ISS should satisfy current Orion command and service module lift requirements. The challenge is to connect launches and accumulated operational experience with a growth plan for the launcher that tracks and slightly leads the forecast need for lift capability.

As suggested by Figures 16 and 18, the difficulty is not confined to the launch system but spills over into the launch pad and the ground infrastructure as well. Accumulated experience in all aspects of the system is essential to developing long term reliability, and therefore confidence on the part of commercial customers. These points are illustrative of the nature of this business, and the difficulties of operating from a position of policy and pragmatism; the results may be economically and technically disastrous.

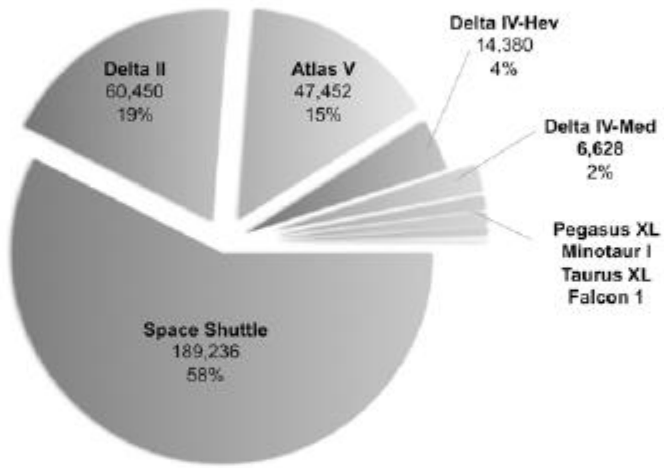

Figure 16
US Space Deliveries, 2005 – 2009
Payload Mass Launched (kg)

Heavy Lift Launch for Space Commerce

In the short term, which is defined here as the next five to seven years, there are only three American solutions that could satisfy the above launch requirements for NASA, military, or commercial missions:

1. Extending the Space Shuttle Program
2. 'Human Rating' for the Delta IV Heavy.
3. 'Human Rating' for the Atlas V heavy.

(A definition of 'human rating' is provided below.)

Extending Space Shuttle operations is not impossible at this late juncture, but because such an undertaking would require substantial logistical contract re-instatement and may quickly prove to be cost prohibitive when compared to human rating heavy versions of either Delta or Atlas. Shuttle continuation remains a very contentious issue on those grounds.

The shutdown of the Shuttle Program is significant for many reasons, including operations work force stability, cost of payload per pound, loss of a human rated vehicle availability, and others. However the principle concern is the gap left in the launch market, as shown in Figure 11.

Clearly the supposition that transition from government to commercial launch systems can be smooth, painless, and swift would not be an accurate conclusion based upon this Figure. The path may be long, arduous, and difficult, especially in the current fiscal situation and business climate, projected to last well beyond 2012.

Human Rating

Human Rating refers to the suitability of a launch system for transporting humans. As stated in the latest NASA led forum on Human Rating, the definition is:

> *"A human-rated system accommodates human needs, effectively utilizes human capabilities, controls hazards, and manages safety risk associated with human spaceflight, and provides, to the maximum extent practical, the capability to safely recover the crew from hazardous situations.*
>
> *"The overall objective is to provide the safest possible design that can accomplish the mission, given the constraints on the program, mass, volume, schedule, and cost."*[4]
>
> NASA NPR 8705.2

This definition leads to several connected, deliberate, and nearly irrevocable decisions. First, engineers and managers must always concern themselves with these questions:

- What is the failure mode set that would threaten significantly human life on-board?
- What are the limits that must be designed in, such as g force limits, maneuvers rates, accessibility, operability, and the like?
- What is the over-system-design capable of in terms of reliability, in on and off nominal situations?

In the un-manned arena, the primary criterion is 'mission success.' If one translates that into booster design, the ability to not destroy the launch pad, have adequate range safety, and effective abort scenarios (destruction of vehicle) at any point causes only the loss of the booster system and the payload, but there is no loss of life. There are no repercussions except an upset customer.

In human spaceflight, the metric is 'safety of the crew, the vehicle, and mission,' in that very specific and non-negotiable order. Once we

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4 *Aerospace America*, AIAA monthly publication, August 2010, Page 26 – 41.

inject humans into the equation, we see that the design has radically different requirements and attributes. Escape from a malfunctioning booster is one such requirement; the ability to retrieve a crew from abort scenarios is mandatory. G forces also become an issue, something that is difficult (as previously discussed) in solid rockets, since the solid rocket burn rate is not controllable once ignited.

And the complications are additive. A theoretical example is a booster is doing 3 g's and the requirement to have an escape system for the crew that puts separation distance adequate for booster destruction g force requirements at 11 g's (not uncommon for escape systems). The additive effect is 14 g's - very significant, and lethal.

A practical example is that John Glenn rode an Atlas in the Mercury program. At the time the success rate for Atlas was under 50%, and the destruct rate for boosters was around 20%. This launch system was so unstable that the launch silo crews received hazardous duty pay. It is not conceivable that this situation would repeat itself in the quest for the commercialization of space, the return to the moon or a mission to Mars today, because the social tolerance for this risk level and the sense of immediacy from the Russian Sputnik that promulgated the decision to use this unreliable booster in 1963 could not be repeated today.

Therefore, despite a family evolution much like the Atlas, Titan, and Delta families the Ariane Program, shown in Figure 17, having no history of human cargo, was not designed to do so, and therefore would require yet another variant to approach the human rating issue.

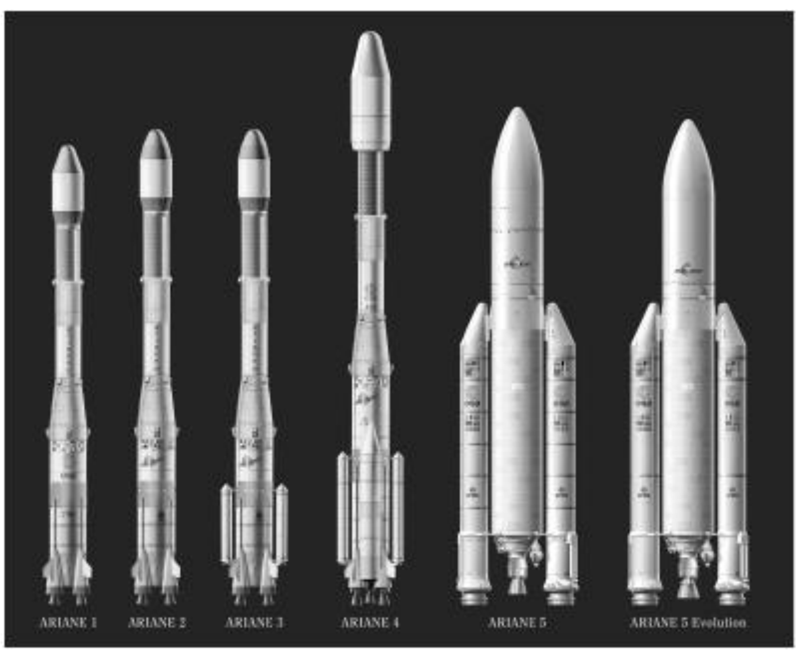

Figure 17
Ariane Launch Vehicles

Given the maturity of the design, this is largely impractical. Why? Because if there is an integrated optimized design that has mass-to-orbit as the metric, everything from choice of toxicity of propellants to upper limits of g forces are required to be rethought. Looking back to 1963, John Glenn, in true test pilot fashion, took on a mission that had a very good chance of taking his life. While test pilots of this ilk are still out there, the society and the organizations like NASA have lost the stomach for this kind of risk. History, then, must be viewed carefully so as not to mislead as to what is currently feasible from a social, cultural and technology perspective.

It is interesting that the NASA Forum on Human Rating did not come up with anything new that conflicted with or added to the NPR 8705 requirements. What was decided amounted to recognition that this is a very real design factor that will influence the cost of the total system, and certainly will be an issue when routine commercial space access is being considered. Human rating is going to be a seminal issue in the development of systems to serve as the transportation system in the space commercialization adventure.

This is an additive complexity to the evolution of commercial launchers and the commercialization of space. We sit today on the cusp of a direction change in space faring. It is inevitable that a transition to commercial service providers will occur, but the question is, When will the probability of success for commercial launch systems even come close to the success factors that the current Russian and American human rated space launch systems enjoy today?

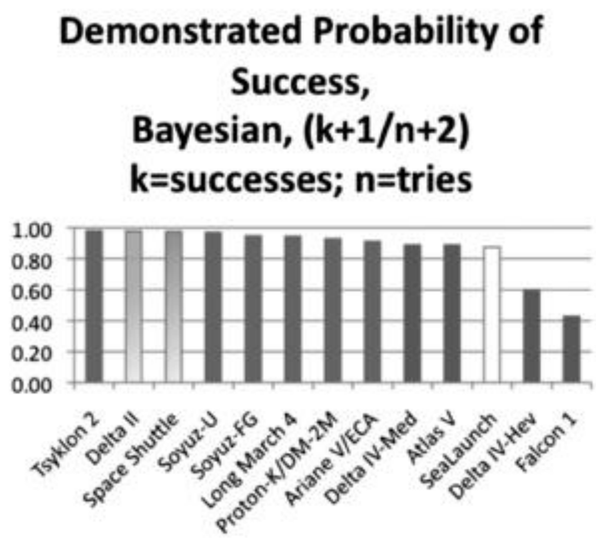

Figure 18
Demonstrated Probability of Success

Moreover, looking at how long it took to get these systems reliable to the current degree, at a commercially viable cost, what does this mean for the commercial arena? These are unanswered and critical questions that policy change alone cannot force in reality.

The simple Demonstrated Probability of Success analysis shown in Figure 18 indicates that the likely gap is closer to 10 years than 5 years, as that's how long it is likely to take to accumulate enough launches on any new system to assure reliability to any sufficient degree, irrespective of what capital, technology and initiative is expended by the commercial sector. In the unforgiving realms of space, it is not appropriate or supportable to 'push the limits' with reckless abandon.

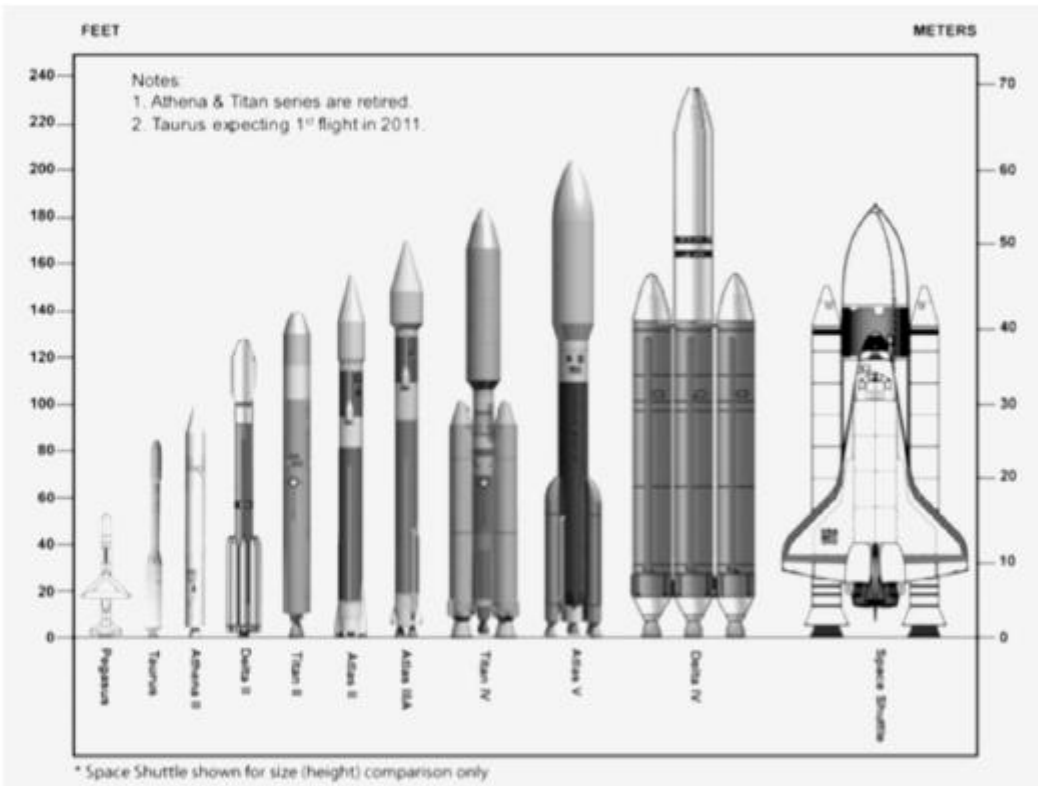

Figure 19
Current U.S. Launch Vehicle Family

A brief glance at the current suite of vehicles, as shown in Figure 19, indicates clearly the satellite market has a rich set of choices. The human spaceflight area, however, has very few. In fact, the human space segment, post Shuttle, is in need of not only a new very heavy lift launch system, but an interim upgrade and certification of some existing non-human rated 'cargo' vehicles. Why does this dichotomy exists at all?

A Brief History of Launch Vehicle Development

All U.S. and Soviet / Russian launch vehicles originated as military systems whose primary mission was to deliver nuclear weapons. Both the

Russians and Americans built upon knowledge and systems captured from German rocket experts immediately following World War II.

The first U.S. and Soviet systems were long range artillery rockets (like the Redstone), and were all largely based on improvements to the WWII V-2 series. Although its range was only 200 miles, the V2 was groundbreaking and unmatched in capability for its time, and it is widely regarded as the first 'strategic' ballistic missile. It of course was not human rated and intended for military uses only.

Increasing the range and payload capacity of these rocket delivery systems for nuclear weapons was the next step for launch systems, and they had advantages over aircraft delivery in three primary areas: 1) enormously faster response, 2) no flight crew risk, and 3) are virtually impossible to intercept in flight. Increasing the range and payload capacity of these launch vehicles was a central factor in the United States achieving strategic advantage its adversaries.

In this case, a human rated system (aircraft) was completely replaced by a non-human rated system (rocket), with attendant stark differences in the infrastructure, the risk profiles, and the delivery focus.

It is important to note that, while much effort was made to miniaturize tactical fission weapons, because strategic nuclear weapons of the early period (1950s and 1960s) were generally large and heavy, the 'bigger is better' rule applied to ICBM launch vehicles. The trend continued with the development of the first thermonuclear (fusion) weapons because they were huge by comparison to fission weapons.

Human rated systems must accommodate a lot of life support gear, a necessary and unavoidable launch mass. The best design case is miniaturization and outstanding design that minimizes booster lift requirements. However, if a brutally large launch weight capacity system is available, such as the Saturn V, Aries V, or the C-8 Nova, the design optimization, complexity, and even reliability of that life support gear can be relaxed in favor of redundancy, and its associated weight can easily increase.

This is yet another difference between the historical launch system evolution and the future commercialization arena. The driving forces are radically different, and if history is used in an uninformed way it could lead to disastrous programmatic decisions.

Range distinctions, which is related to booster capability, evolved quickly, and were classified into three main groups: 1)artillery rockets, range 50-200 statute miles, 2) Intermediate Range Ballistic Missile (IRBM), range 200-3,000 statute miles, and 3) Intercontinental Ballistic Missile (ICBM), range 3,000-10,000 statute miles.

Only categories 2), and 3) had influence in the space launch arena. However, the technology of all three groups is to a degree cross-cutting and there was significant influence shared by emerging technologies in areas such as computing hardware and the emergence of digital flight systems.

Summary: The Next Generation of Heavy Lift Systems

A short summary of the continuously serving and therefore successful launch system vehicles reveals the following:

- Space Shuttle: The world's only human rated, reusable, winged spacecraft. About to be retired.
- Delta: Began IRBM from NATO land bases.
- Delta II: The series marked Delta's departure from weapons delivery service. Used for military and civilian satellite launch.
- Delta III: Added lift capability to missions & customers accrued during series II service.
- Delta IV: Follow-on to series II with the added prospect for use as a human launch system.
- Atlas: Started as the USA's principal ICBM of the late 1950s and early 1960s. It evolved away from weapons delivery service in the mid 1960s. Series I-V evolved similarly to and as a prime competitor of Delta.

In this chapter we have discussed the history and the technology that not only explains how the national and international community has arrived today with the arsenal of launch systems available as we approach the departure point for the commercialization of space.

We have also discussed technical and socio-cultural challenges, the lexicon of rocketry and what is meant by such key terms as 'heavy lift,' and the significance of human rated, heavy lift system for space commerce.

Looking forward, the longer term view 25 years out will certainly see the development of new heavy lift systems, but in the short and medium terms the issues remain quite challenging, as cheap, reliable access to space is not yet a reality.

Our view is that the emergence of space commercialization has loftier motivation and more constructive intent than the adapted weapons systems that are the mainstays of our current heavy lift capabilities, but it will take demonstrated markets of significant size in order for the technical development costs to be borne by the commercial sector, as well as significant innovation to develop new systems that bring the costs down. An order of magnitude cost reduction is of course too much to ask, and yet from a commercial perspective this is exactly what the emerging generation of entrepreneurs is asking for. And as generation of bright rocket scientists are well aware of these goals, they are working at them, in both public and private employment, in the US and in other nations.

Thomas E. Diegelman

Tom Diegelman has been in the aerospace community for over 35 years, involved in research, development and operation of training simulators, ground based flight control installations and facility operations. Tom started his career with Cornell Aeronautical Laboratory as a research engineer, working on early versions of shuttle handling quality study simulations and shuttle shock tunnel testing.

Tom moved to Houston in the late 70's to join Singer / Link Flight Simulation and worked in the Shuttle Mission Training Facility (SMTF) as a model developer, and later a manager of simulation projects. In 1988, Tom joined NASA to lead the $170M redesign of the SMTF. Assignments at NASA / JSC include projects in advanced mission control technology, technology development, and facility operations control. He served as Facility Manager for Mission Control for 3 years before accepting an account manager position in the Technology Transfer Office, developing partnerships and Space Act Agreements.

The design of the training facility for the Constellation Program culminated his nearly 30 years of experience at JSC in Jake Garn astronaut training facility. Tom was elected to Seabrook City Council in 2006, and served two terms, during which he worked closely with the Port of Houston Authority on the Seabrook / Bayport Terminal Facility issues. Tom is a member of Baytran, a non-profit organization promoting inter-modal transportation solutions in the Houston / Bay Area, and continues to be involved in local, state and federal government on behalf of the space technology community.

He is also coauthor Of Chapter 16, *A Space Commercialization Model: Ocean Ports and Inter-Modal Transportation.*

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