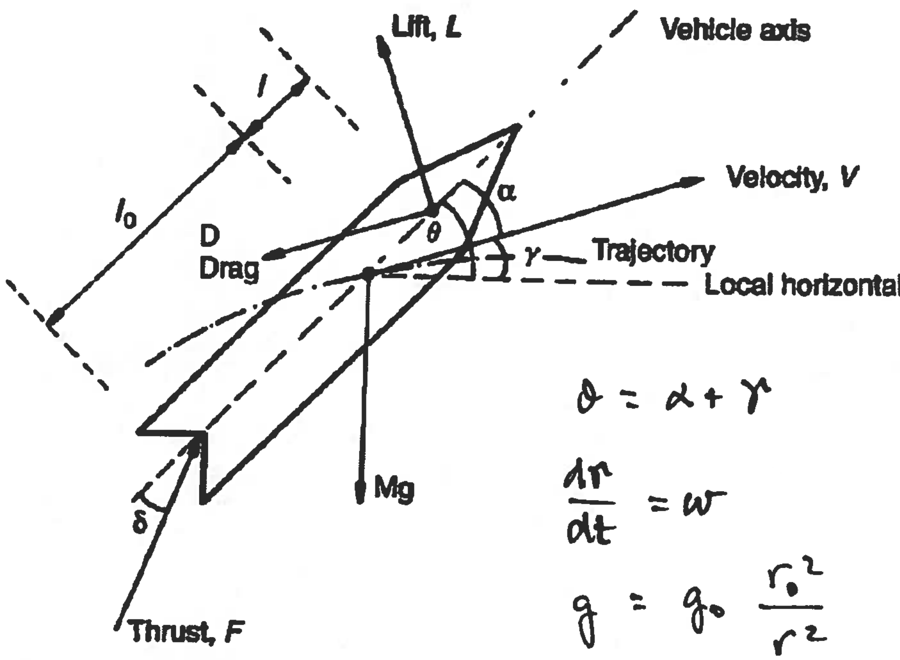
Launch Vehicles

Launch Vehicle Performance



$$\theta = \alpha + \gamma$$

$$\frac{dr}{dt} = w$$

$$g = g_0 \frac{r_0^2}{r^2}$$

Figure 7.1 Configuration and nomenclature for rocket motion in the vertical plane

- In flight direction (∥ to V): Forces on rocket

$$M \frac{dV}{dt} = F \cos(\alpha + \delta) - Mg \sin\gamma - D \quad 7.1$$

- Angular motion in pitch (θ) about C.M.:

$$I_p \frac{d^2\theta}{dt^2} = \underbrace{(L\cos\alpha + D\sin\alpha)\,l}_{\text{displacement of C.P. from C.M.}} - \underbrace{F l_0 \sin\delta}_{\text{thrust vectoring via nozzle gimbal}} \quad 7.3$$

- Small angles α and δ : force balance becomes

$$\frac{dV}{dt} = \underbrace{\frac{F}{M}}_{\substack{\text{thrust}\\ \text{(ideal)}}} - \underbrace{g\sin\gamma}_{\text{gravity}} - \underbrace{\frac{D}{M}}_{\text{drag}}$$

$$= \frac{1}{M}\left(-\frac{dM}{dt}\right) I_{sp}\, g_0 \quad - g\sin\gamma - \frac{D}{M}$$

$$= -I_{sp}\, g_0 \frac{d}{dt}(\ln M) \quad - g\sin\gamma - \frac{D}{M}$$

- Integrate to get velocity eqn:

$$V(t) = V_0 + \underbrace{I_{sp}\, g_0 \ln\left(\frac{M_0}{M}\right)}_{\Delta V_{ideal}} - \underbrace{\int_0^t g\sin\gamma\, dt}_{\Delta V_g} - \underbrace{\int_0^t \frac{D}{M}\, dt}_{\Delta V_D} \qquad 7.6$$

$$\boxed{\Delta V\Big|_t = V - V_0 = \Delta V_{ideal} \underbrace{- \Delta V_g - \Delta V_D}_{\substack{\text{propulsive losses}\\ \text{due to gravity + drag}}}} \qquad 7.7$$

- Competition: low ΔV_g → short burn time → high velocity

  low ΔV_D → low velocity in atmos $\left(D = \frac{1}{2}\rho V^2 S C_D\right)$

Earth Launch

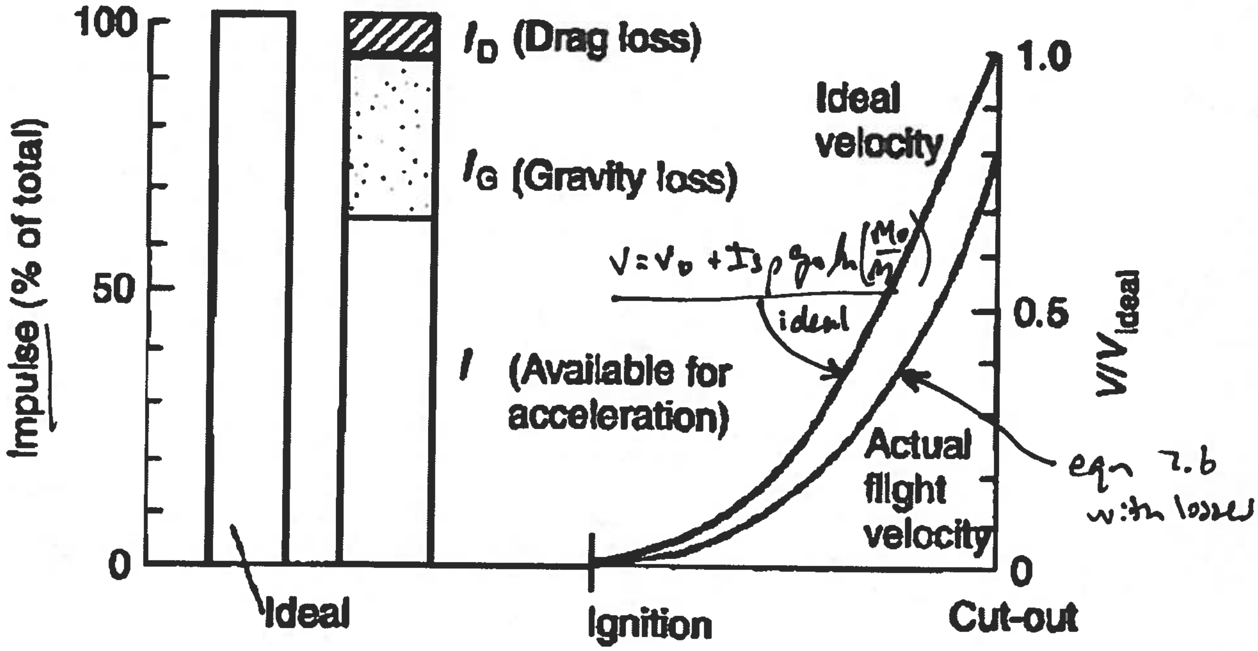


Figure 7.2 Illustration of launch losses due to drag and gravity

- Forces ⊥ to velocity vector (centripetal, thrust, weight, lift)

$$MV\frac{d\gamma}{dt} = F\sin(\alpha+\delta) - Mg\cos\gamma + L + \frac{MV^2}{r}\cos\gamma \qquad 7.2$$

- "Gravity turn": launch vertically, must turn in direction of orbit

Neglect all but gravitational terms in 7.2:

$$V\frac{d\gamma}{dt} = -g\cos\gamma,$$

γ = angle between velc. vector and local horizontal (decreasing $\rightarrow \cong 0$)

Integrate:

$$\sin\gamma(t) = \tanh\left[\tanh^{-1}(\sin\gamma_0) - \int_{t_0}^{t}\frac{g}{V}\,dt\right] \qquad 7.8$$

- Can modify this rate with:
  - thrust vectoring
  - attitude control thrusters
  - aerosurfaces

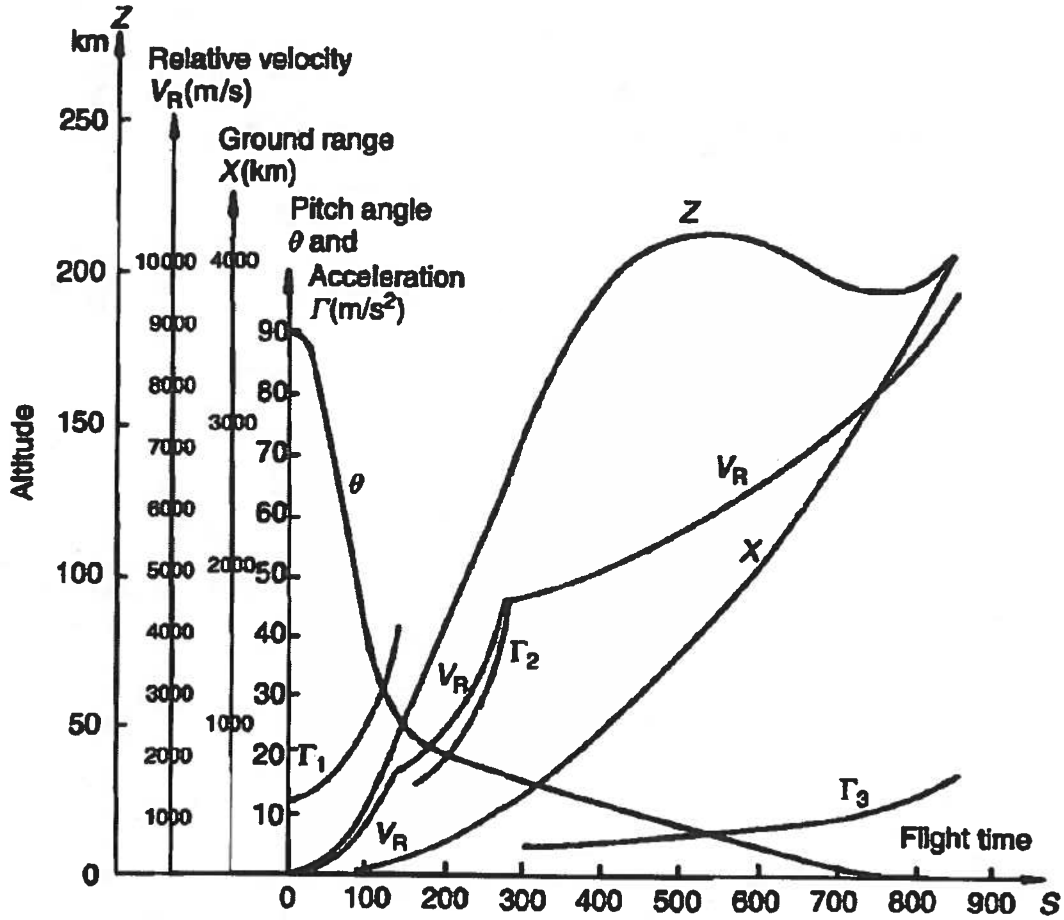


Figure 7.3 Typical Ariane flight profile

- note staging

$\theta \approx \gamma \quad (\alpha \simeq 0)$

up to 4.5 g's

- Trajectory prediction: must simplify for analytical results

ignore: ΔV_D

let $g = g_0$

$$\frac{\text{thrust}}{\text{weight}} = \text{constant} = \underbrace{\frac{F_0}{M_0 g_0}}_{\text{lift-off params}} = a$$

Now can form expression for single-stage mass ratio:

$$\frac{M_b}{M_0} = \exp\left[- \underbrace{\frac{V_b}{g_0 I_{sp}}}_{\text{ideal}} \; \underbrace{\frac{a^2}{(a^2-1)}}_{\text{reduction due to gravity}} \right]$$

b: burnout conditions

0 = launch conditions

Gravitational loss:

$$\Delta V_g = g_0 I_{sp} \ln\left(\frac{M_0}{M_b}\right) - V_b$$

$$= V_b / (a^2 - 1)$$

so want large $\frac{\text{thrust}}{\text{weight}}$ to minimize losses due to gravity large cost savings. One avenue of future work in this area is incorporating the effect of the correlation of the sine loads for harmonics of a single rotating shaft.

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R. B. Malla
Associate Editor

Optimum Thrust-to-Weight Ratio for Gravity-Turn Trajectories

G. E. Dorrington*
University of London,
London, England E1 4NS, United Kingdom

Nomenclature

a = vehicle thrust-to-weight ratio
g = gravitational acceleration, ms^{-2}
k = velocity constant, ms^{-1}
m = vehicle mass, kg
T = vehicle thrust, N
t = time, s
u = vehicle velocity, ms^{-1}
v_e = exhaust velocity, ms^{-1}
v_e^* = effective exhaust velocity, ms^{-1}
z = $\tan(\phi/2)$
ΔV_g = gravity loss, ms^{-1}
λ = payload ratio
μ_e = engine thrust-to-weight ratio
μ_{es} = design constant
σ_{es} = design constant
ϕ = vehicle flight path angle, rad

Subscripts

b = at burnout
opt = optimum (for maximum λ)
0 = initial

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*Lecturer in Aerospace Design, Department of Engineering, Queen Mary and Westfield College. Member AIAA.

Introduction

THE analysis of the ascent of a single-stage rocket from a planetary surface to orbit on a gravity-turn trajectory is well known.[1–4] This Note extends the analysis to quickly arrive at an expression for the burnout-to-initial mass ratio of a single-stage vehicle ascending in vacuo and thereafter derives some bounds on the optimum initial vehicle thrust-to-weight ratio. The analysis is not directly useful for detailed studies of vehicle performance because a gravity turn is not an optimal trajectory, but it may still be educationally instructive, or possibly useful for concept evaluation and preliminary design studies.

Analysis of a Single-Stage Rocket on a Gravity Turn

In a flat-Earth approximation,[1,2] ignoring atmospheric drag, the equations of motion of a rocket vehicle ascending on a gravity turn (with no thrust vectoring) are

$$m\frac{\mathrm{d}u}{\mathrm{d}t} = T - mg\cos\phi \tag{1a}$$

$$u\frac{\mathrm{d}\phi}{\mathrm{d}t} = g\sin\phi \tag{1b}$$

where ϕ is the angle of the flight path from the vertical, T is the vehicle thrust, m is the vehicle flight mass, u is the vehicle velocity, and g is the gravitational acceleration (assumed to be constant and equal to the surface value, g_0).

In the analysis that follows, it will be assumed that the vehicle thrust-to-weight ratio a is held constant at the initial (liftoff) value throughout the ascent:

$$a = T/mg = T_0/m_0g_0 = a_0 \tag{2}$$

In this particular case, the substitution $z = \tan(\phi/2)$ yields the solution[1]

$$u = kz^{a-1}(1+z^2) \tag{3a}$$

$$\frac{\mathrm{d}u}{\mathrm{d}z} = k\{(a-1)z^{a-1} + (a+1)z^a\} \tag{3b}$$

where the constant k is equal to half the burnout velocity, $k = u_b/2$, if and when $u = u_b$ and $m = m_b$ at $z = 1$ (a condition that is assumed hereafter).

If the thrust is approximated by

$$T = -v_e\frac{\mathrm{d}m}{\mathrm{d}t} \tag{4}$$

where the exhaust velocity v_e is assumed to be constant, then the rocket's acceleration is given by

$$\frac{\mathrm{d}u}{\mathrm{d}t} = -\frac{v_e^*}{m}\frac{\mathrm{d}m}{\mathrm{d}t} \tag{5}$$

where v_e^* is an effective exhaust velocity, $v_e^* = v_e(1 - a^{-1}\cos\phi)$. The burnout-to-initial mass ratio m_b/m_0 is found by integration with respect to velocity from $u = 0$ to the burnout velocity $u = u_b$:

$$\ell n\left\{\frac{m_b}{m_0}\right\} = -\int_0^{u_b}\frac{\mathrm{d}u}{v_e^*} \tag{6}$$

Noting $\cos\phi = (1-z^2)/(1+z^2)$, this integral becomes

$$\ell n\left\{\frac{m_b}{m_0}\right\} = -\int_0^{u_b}\left\{1 - \frac{(1-z^2)}{a(1+z^2)}\right\}^{-1}\frac{\mathrm{d}u}{v_e}$$

$$= -\frac{u_b}{v_e}\int_0^1 \frac{1}{2}a(1+z^2)z^{a-2}\,\mathrm{d}z \tag{7}$$

which can easily be solved to yield a simple expression:

$$\boxed{\frac{m_b}{m_0} = \exp\left\{-\frac{u_b}{v_e}\frac{a^2}{(a^2-1)}\right\}} \tag{8}$$

Hence, it can be seen that the burnout-to-initial mass ratio is lower than the ideal value, $\exp\{-u_b/v_e\}$, and the gravity loss incurred during the ascent is[2]

$$\Delta V_g = v_e \ln\{m_0/m_b\} - u_b = u_b/(a^2 - 1) \tag{9}$$

Optimum Thrust-to-Weight Ratio

Suppose that in a conceptual design study the initial mass m_0 of the vehicle considered is fixed, and the vehicle mass breakdown is written as

$$m_0 = m_b + m_{\text{propellant}} \tag{10a}$$

$$m_b = m_{\text{payload}} + m_{\text{misc}} + m_{\text{structure}} + m_{\text{engine}} \tag{10b}$$

where $m_{\text{propellant}}$ is the ascent propellant mass, m_{payload} is the payload, $m_{\text{structure}}$ is the structural mass, m_{engine} is the total engine mass, and m_{misc} is the mass of all other subsystems and miscellaneous items. Hence, for a prescribed value of v_e/v_b, the payload-to-initial mass ratio, $\lambda = m_{\text{payload}}/m_0$, is maximized when $m_b - m_{\text{misc}} - m_{\text{structure}} - m_{\text{engine}}$ is maximized.

Assume to begin with that m_{misc} and $m_{\text{structure}}$ are fixed. In this case λ is maximized when $m_b - m_{\text{engine}}$ is maximized. If the engine thrust-to-weight ratio $\mu_e = T_0/(g_0 m_{\text{engine}})$ is held constant, then [differentiating Eq. (8) with respect to a] it follows that λ is a maximum when $a = a_{\text{opt}}$ given by

$$\frac{2a_{\text{opt}}}{\left(a_{\text{opt}}^2 - 1\right)^2} = \mu_e^{-1}\frac{v_e}{u_b}\frac{m_0}{m_b} = \mu_e^{-1}\frac{v_e}{v_b}\exp\left\{\frac{u_b}{v_e}\frac{a_{\text{opt}}^2}{\left(a_{\text{opt}}^2 - 1\right)}\right\} \tag{11}$$

Now assume that m_{misc} and $m_{\text{structure}}$ vary as the vehicle design thrust-to-weight ratio is altered. An exact expression can no longer be defined (because the vehicle's structure and subsystem masses will depend on a variety of design factors[5]), but to first order it is not unreasonable to adopt an approximate linear relation:

$$(m_{\text{structure}} + m_{\text{engine}} + m_{\text{misc}})/m_0 = \sigma_{es} + \mu_{es}^{-1}a \tag{12}$$

where σ_{es} and μ_{es} are constants (and $\mu_{es} < \mu_e$). In this more general case, the condition for maximum λ is the same as Eq. (11) except the design constant μ_{es} replaces μ_e:

$$\frac{2a_{\text{opt}}}{\left(a_{\text{opt}}^2 - 1\right)^2} = \mu_{es}^{-1}\frac{v_e}{u_b}\frac{m_0}{m_b} \tag{13}$$

The optimum value of initial thrust-to-weight implied by Eq. (13) has a lower limit. The minimum feasible value of a_{opt} occurs as $m_s + m_e + m_{\text{misc}} \to m_b$ (or as $\lambda \to 0$) and also as $\sigma_{es} \to 0$, that is, when

$$\frac{2a_{\text{opt}}}{\left(a_{\text{opt}}^2 - 1\right)^2} \to \frac{v_e}{u_b}\frac{\mu_{es}^{-1}}{\left(\mu_{es}^{-1}a_{\text{opt}} + \sigma_{es}\right)} \to a_{\text{opt}}^{-1}\frac{v_e}{u_b} \tag{14}$$

or (rearranging this equation and solving the resulting quadratic) when

$$a_{\text{opt}} \to a_{\text{opt}_{\min}} = \left(\tfrac{1}{2}u_b/v_e\right)^{\frac{1}{2}} + \left(1 + \tfrac{1}{2}u_b/v_e\right)^{\frac{1}{2}} \tag{15}$$

Hence, irrespective of the actual values of σ_{es} and μ_{es}, the optimum vehicle thrust-to-weight ratio a_{opt} lies within certain bounds given by Eqs. (11) and (15), provided positive λ is achievable and provided λ needs to be maximized.

Limitations of the Analysis

Note that, in reality, constant in-flight vehicle thrust-to-weight ratio is unlikely. If a vehicle has multiple engines, then these engines might be shut down sequentially during the ascent to prevent acceleration limits from being exceeded; but it is unlikely that each engine would be throttled in such a way that the total thrust varies directly in proportion to vehicle flight mass. Moreover gravity-turn trajectories (with/without the condition assumed herein that burnout occurs at $z = 1$) are not optimal.[4] Nevertheless, despite these differences, the optimum conditions for vehicle thrust-to-weight found here appear to be useful for concept evaluation and preliminary estimates. To demonstrate this assertion, consider the following examples.

Single-Stage Rocket Ascent from Earth's Surface to Orbit

As a first example, consider a single-stage-to-orbit launch vehicle ascending to low Earth orbit, using liquid hydrogen and oxygen propellants, such that $v_e \cong 4500\ \text{ms}^{-1}$ and $v_e/u_b = 0.5$. If $\mu_e = 50$ (a typical value with current technology), then $m_b - m_{\text{engine}}$ is maximized when $m_b/m_0 \cong 0.1$ and $a_{\text{opt}} \cong 3$. If structural mass increases significantly with launch acceleration, then the optimum initial vehicle thrust-to-weight ratio (for maximum payload ratio) will be lower, but it will not be less than $a_{\text{opt}_{\min}} = 1 + \sqrt{2}$ at which point $m_b/m_0 \cong 0.089$. Hence, λ is maximized when a_{opt} lies between approximately 2.4 and 3. Note, however, that in reality this result ignores the effects of drag losses, etc. Hence, a lower initial vehicle thrust-to-weight ratio might be expected.

Apollo Lunar Module Ascent

As a second example, consider the ascent stage of the Apollo Lunar Module,[5] which had a near-constant ascent thrust of about $T = 15.57$ kN and an initial mass of about $m_0 = 4700$ kg on the lunar surface where $g_0 \cong 1.62\ \text{m/s}^2$, such that $a_0 \cong 2$ (depending more exactly on the mission number: Apollo 11–17). To attain lunar orbital velocity at $u_b \cong 1.63$ km/s, with an exhaust velocity of about $v_e = 3050\ \text{ms}^{-1}$, Eq. (8) gives $m_b/m_0 \cong 0.495$, which is slightly worse than the actual value of 0.52. Furthermore, assuming $m_e = 213$ kg (the mass of the ascent propulsion system[5]) such that $\mu_e \cong 45$, Eq. (11) gives $a_{\text{opt}} \cong 3.1$, and Eq. (15) gives $a_{\text{opt}_{\min}} \cong 1.64$. Hence, despite the differences in trajectory thrust profile, etc., the simple analysis presented herein does not give unreasonable results.

Acknowledgments

The work presented in this Note was initiated in 1988, while I was attending the International Space University (ISU) summer session at the Massachusetts Institute of Technology, Boston. It was first submitted to this journal on 20 July 1999, exactly 30 years after the Apollo 11 moon landing, as a deliberate tribute to that event. With that background in mind, I would like to thank the founders of ISU and the participants of the 1988 summer session, in particular Edwin E. ("Buzz") Aldrin, for his encouragement and for stressing the importance of basic trajectory/orbital analysis, even though this work is just a small contribution and somewhat overdue.

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J. A. Martin
Associate Editor

Mission Requirements

$$\Delta V_a = V_0 \left\{\frac{R_E + h_0}{R_E + h}\right\}^{1/2} \left\{1 - \left[1 + \left(\frac{2}{\frac{R_E + h}{R_E + h_0}}\right)\right]^{1/2}\right\},$$

$$V_b = V_0 \left\{\frac{2}{1 + \left(\frac{R_E + h_0}{R_E + h}\right)}\right\}^{1/2}, \quad V_0 = \left\{\frac{\mu}{R_E + h_0}\right\}^{1/2}$$

	Transfer orbit apogee height (km)	Burn-out velocity at 200 km, V_b (km s^{-1})	Circularization burn at apogee, ΔV_a (km s^{-1})
typical LEO:	200	$7.784 = V_0$	0
	500	7.870	0.085
	1 000	8.004	0.214
	10 000	9.299	1.199
GEO:	35 863	10.261	1.477
	∞	11.009	–

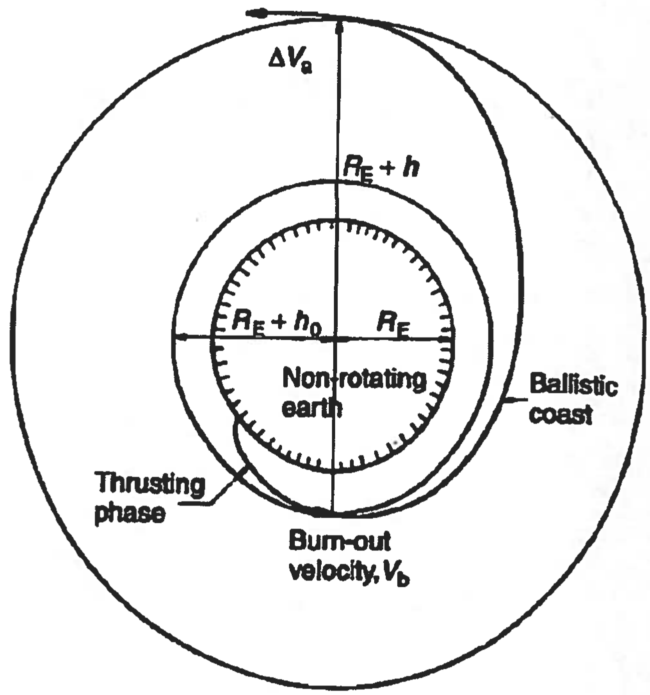


Figure 7.4 Launch vehicle burn-out velocities for spacecraft emplacement

$$\Delta V_{ideal} \doteq \Delta V + \underbrace{\Delta V_g + \Delta V_D}_{\text{1st stage of large rocket}}$$

ΔV_g: $\sim 1.1 \frac{km}{s}$ $\qquad$ ΔV_D: $\sim 0.2 \frac{km}{s}$

Multi-Stage Launch Vehicles (7.2.3)

- why not single-stage-to-orbit?

from 7.6, $V_{b_{ideal}} = I_{sp}\, g_0 \ln\Big(\underbrace{\frac{M_0}{M_b}}_{\equiv R}\Big)$

rocket mass ratio

- $M_0 = M_p + M_s + M_F$

M_0: initial rocket total mass
M_p: initial payload
M_s: initial structure
M_F: initial fuel (propellant)

let $\underbrace{p \equiv \frac{M_p}{M_0}}_{\text{fractional payload ratio}}$, and $\underbrace{\sigma = \frac{M_s}{M_F}}_{\text{propellant tankage struct. efficiency}}$

- What ISP will we need for $h_a = 200\,\text{km}$?

$V_b = 7.8\ \text{km/s}$ (Fig 7.4)

$\sigma = 0.1$ typical

$M_p = 0$ best case performance $\Rightarrow R = 11$

$\therefore$ ISP $\cong$ 331 s for no payload, no drag, no gravity } clearly unrealistic $\rightarrow$ STAGE

Staging

- Velocity increment for i^{th} stage:

$$\Delta V_i = V_{b_i} - V_{b_{i-1}} = ISP_i \, g_0 \ln\left(\frac{1+\sigma_i}{p_i+\sigma_i}\right) \tag{7.13}$$

note that $\sigma_i = \left(\frac{M_p}{M_0}\right)_i$ and M_p is all hardware mass for above i^{th} stage

- Final burn-out velocity:

$$V_b = \sum_{i=1}^{n} ISP_i \, g_0 \ln\left(\frac{1+\sigma_i}{p_i+\sigma_i}\right) \tag{7.14}$$

- Many ways to optimize distr. of mass, but highly constrained by engineering and market realities.

- In practice, max # stages ~ 4

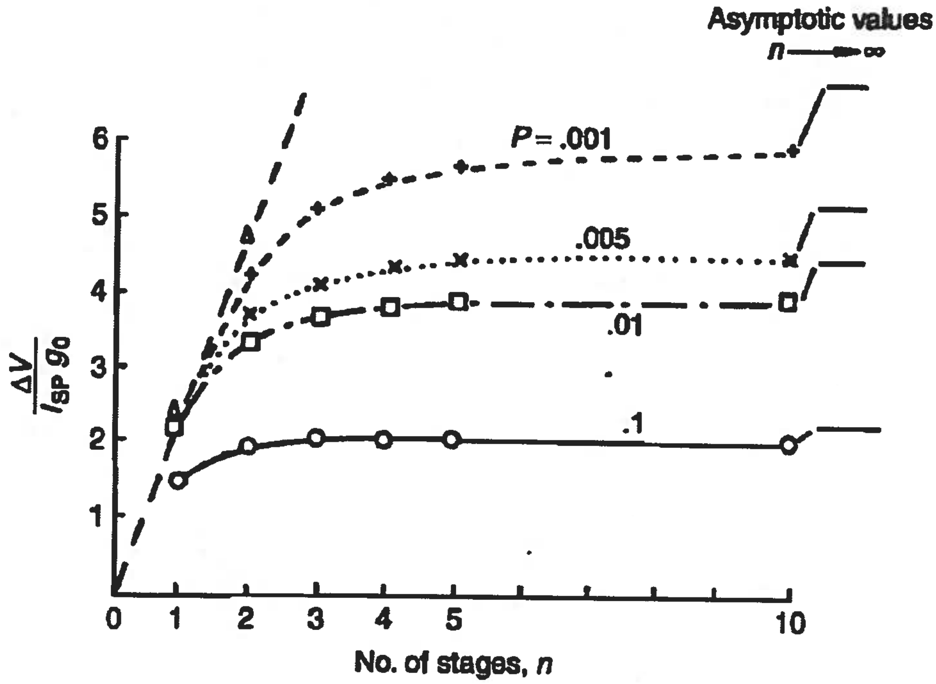


Figure 7.7 Variation of velocity increment with number of stages for fixed overall payload ratio P

Staging Example:

Same: overall mass M_0
payload ratio $p = \frac{M_P}{M_0}$
structure/fuel $\sigma = M_S/M_F$
ISP

- *Equivalent single-stage launcher ΔV*

Payload 455 kg

Structural mass 2447 kg ($\sigma = 0.15$)

Propellant mass 15957 kg

Mass ratio 6.5

□ $\boxed{\Delta V = 5.4 \text{ km s}^{-1}}$

for an average I_{sp} of 295s

- *Ideal three-stage launcher performance*

3rd stage payload 455 kg;

$$p_3 = \frac{455}{1439} = 0.32$$

2nd stage payload 1439 kg;

$$p_2 = \frac{1439}{4839} = 0.30$$

1st stage payload 4839 kg;

$$p_1 = \frac{4839}{18859} = 0.26$$

Overall payload ratio, $P = \prod_i p_i = 0.025$

□ $\Delta V = \left\{ \sum_i I_{SP,i}\, g_0 \ln\left[\frac{1+\sigma_i}{p_i + \sigma_i}\right]\right\} = \boxed{8.1 \text{ kms}^{-1}}$

Figure 7.8 Illustration of the benefits of multi-staging

Text 26.1: Launch Vehicle Selection

or: Choosing Your Rocket

"...The only natural predator of a spacecraft is a launch vehicle:

Major General Nathan Lindsay, USAF (retired)

"Most of the time, the launch vehicle will be a very expensive nuisance, and one of the greatest risks to the success of your mission."

Text 26.1: Launch Vehicle Selection

- Commercial launch vehicles have (mostly) evolved from NASA and DOD earlier designs.

  Continued development uses Government standards as a guide + framework, but not as rules unless required by a Government customer.

- Launch vehicle selection: balance
  - Performance (separated mass delivered to orbit)
  - Availability
  - Risk
  - Cost

- If you have several choices, go for most $\frac{\text{MASS}}{\$}$

- Examples: \$/Kg to LEO

Space Transportation Costs: Trends in Price Per Pound to Orbit 1990-2000

September 6, 2002

Futron Corporation
7315 Wisconsin Avenue
Suite 900W
Bethesda, Maryland 20814
301-913-9372
http://www.futron.com

Introduction

The cost of space transportation has been nothing less than an obsession for many people in the space industry. The perceived high cost of space transportation is generally viewed as one of the biggest obstacles, if not the biggest, to the growth of space commercialization and exploration. Civil space agencies, entrepreneurs, policy makers, legislators, satellite operators, and others have all focused on the cost of space transportation at one time or another. Even though each group has different reasons for being interested in launch costs, they all share the same fixation – lowering the cost of transporting payloads into space.

However, evaluating current and future launch vehicles on the basis of cost has been problematic. The costs of launch vehicles are usually compared by computing the cost to place one pound of payload into orbit. This "cost per pound to orbit" metric has become widespread in the industry, and has been used to establish goals for the cost performance of future launch vehicles. However, this seemingly simple and widely used value is fraught with standardization issues that, if not fully understood, make it a far less useful tool. This White Paper attempts to clarify the concept of price per pound to orbit and to identify some trends in the cost of space transportation using this metric.

What Is Price Per Pound to Orbit and How Is It Calculated?

The simplest way to study the cost of space transportation is to compare the prices of launch vehicles. Unfortunately, this is generally a case of comparing apples to oranges: all launch vehicles are not equal. A Pegasus XL, for example, costs far less than an Ariane 5, but is also a much smaller vehicle. Differences in vehicle size can mask more important cost differences caused by vehicle design, nation of manufacture, and other factors.

To compensate for this, the "price per pound to orbit" metric was developed to compare vehicles for their cost effectiveness, mostly in the comparison of vehicles in the design phase. Price per pound offers a simple way to normalize launch costs, permitting more meaningful comparisons among vehicles of different capabilities. This metric has gained wide acceptance, with almost every proposed new launch vehicle since the Space Shuttle using some type of price-per-pound target.

While price per pound is simple to compute arithmetically, it has many shortcomings. Determining the real price of a launch can be difficult, since the terms of many launch contracts are not made public. In many cases only generic prices for launch vehicles are available, reducing the accuracy of the calculation. The choice of orbital altitude and inclination affects the payload capacity of a vehicle, thus affecting the calculation. These and other factors, if not properly accounted for, reduce the effectiveness of the price-per-pound metric.

Purpose of This White Paper

As stated in the Executive Summary, the purpose of this White Paper is to clarify the concept of price per pound to orbit and to identify some trends in the cost of space transportation using this metric. This White Paper describes how various vehicles fare, generically, using the price per pound metric, and it describes the trends in the metric during the 1990s for both commercial non-geosynchronous orbit (NGSO) – which includes low Earth orbit (LEO), medium Earth orbit, and highly elliptical orbits – and geosynchronous orbit (GSO) launches.

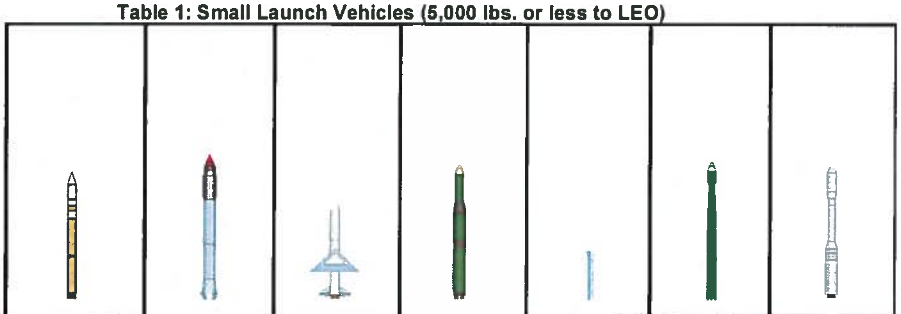
The analyses in this White Paper rely on a database of launch vehicle prices that Futron has maintained for almost a decade. The database is populated with publicly available information on launch prices and, every year, Futron performs an analysis to update the prices of all active launch vehicles in the database. This system worked well throughout the 1990s because there was sufficient publicly available information on launch prices that Futron could estimate price ranges with confidence.

Since the year 2000, the launch vehicle industry has become extremely competitive. Public sources on launch vehicle prices have all but dried up. Executives at Arianespace and Boeing Launch Services have been quoted as saying prices have dropped by 20-30% recently. Because of this uncertainty in estimating current prices of launch vehicles, Futron has restricted the analysis in this White Paper to the time period 1990-2000.

Generic Price Per Pound Calculation

Although the price-per-pound metric appears straightforward, there are a number of methods of computing it. One approach is to simply divide the estimated cost of a launch vehicle by its payload capacity. This approach permits a basic comparison of launch prices among various vehicles at a given point in time. Price-per-pound figures for a representative sample of commercial launch vehicles most commonly in use in the 1990s, as well as the Space Shuttle, are presented in Tables 1 through 3. The vehicles are divided into the four mass classes defined by the FAA Office of Commercial Space Transportation: small, medium, intermediate, and heavy, although for this discussion medium and intermediate vehicles will be grouped together. Separate price-per-pound figures are calculated for each vehicle's LEO and, where relevant, GTO (geosynchronous transfer orbit) capacity. (GTO is used here because most launch vehicles place GSO-bound payloads in an intermediate transfer orbit, from which the spacecraft maneuvers into GSO.) All prices are given in year 2000 dollars based on the latest price information provided during the decade, and do not include the costs of apogee kick motors or other payload injection means.

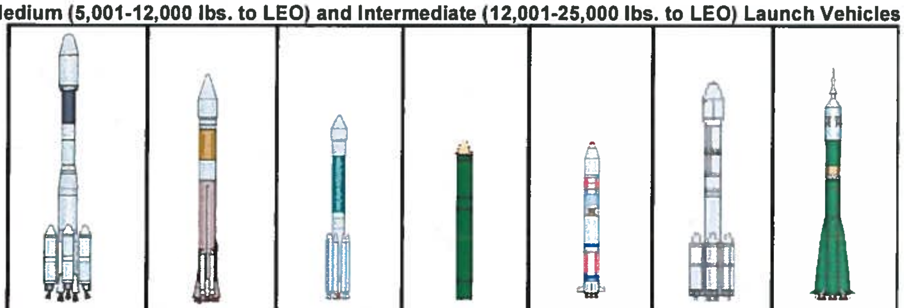
Table 1: Small Launch Vehicles (5,000 lbs. or less to LEO)



Vehicle name	Athena 2	Cosmos	Pegasus XL	Rockot	Shtil	START	Taurus
Country/Region of origin	USA	Russia	USA	Russia	Russia	Russia	USA
LEO capacity lb (kg)	4,520 (2,065)	3,300 (1,500)	976 (443)	4,075 (1,850)	947 (430)	1,392 (632)	3,036 (1,380)
Reference LEO altitude mi (km)	115 (185)	249 (400)	115 (185)	186 (300)	124 (200)	124 (200)	115 (185)
GTO capacity lb (kg)	1,301 (590)	0	0	0	0	0	988 (448)
Reference site and inclination	CCAFS 28.5 deg.	Plesetsk 62.7 deg.	CCAFS 28.5 deg.	Plesetsk 62.7 deg.	Barents Sea 77-88 deg.	Svobodny 51.8 deg.	CCAFS 28.5 deg.
Estimated launch price (2000 US$)	$24,000,000	$13,000,000	$13,500,000	$13,500,000	$200,000*	$7,500,000	$19,000,000
Estimated LEO payload cost per lb (kg)	$5,310 ($11,622)	$3,939 ($8,667)	$13,832 ($30,474)	$3,313 ($7,297)	$211 ($465)	$5,388 ($11,687)	$6,258 ($13,768)
Estimated GTO payload cost per lb (kg)	$18,448 ($40,678)	N/A	N/A	N/A	N/A	N/A	$19,234 ($42,411)

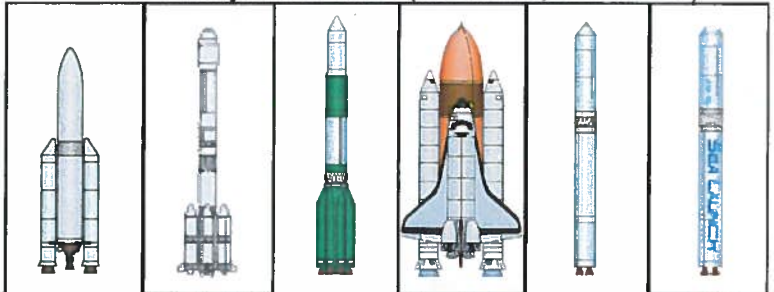
* Shtil launch costs partially subsidized by the Russian Navy as part of missile launch exercises

Table 2: Medium (5,001-12,000 lbs. to LEO) and Intermediate (12,001-25,000 lbs. to LEO) Launch Vehicles



Vehicle name	Ariane 44L	Atlas 2AS	Delta 2 (7920/5)	Dnepr	Long March 2C	Long March 2E	Soyuz
Country/Region of origin	Europe	USA	USA	Russia	China	China	Russia
LEO capacity lb (kg)	22,467 (10,200)	18,982 (8,618)	11,330 (5,144)	9,692 (4,400)	7,048 (3,200)	20,264 (9,200)	15,418 (7,000)
Reference LEO altitude mi (km)	124 (200)	115 (185)	115 (185)	124 (200)	124 (200)	124 (200)	124 (200)
GTO capacity lb (kg)	10,562 (4,790)	8,200 (3,719)	3,969 (1,800)	0	2,205 (1,000)	7,431 (3,370)	2,977 (1,350)
Reference site and inclination	Kourou 5.2 deg.	CCAFS 28.5 deg.	CCAFS 28.5 deg.	Baikonur 46.1 deg.	Taiyuan 37.8 deg.	Taiyuan 37.8 deg.	Baikonur 51.8 deg.
Estimated launch price (2000 US$)	$112,500,000	$97,500,000	$55,000,000	$15,000,000	$22,500,000	$50,000,000	$37,500,000
Estimated LEO payload cost per lb (kg)	$5,007 ($11,029)	$5,136 ($11,314)	$4,854 ($10,692)	$1,548 ($3,409)	$3,192 ($7,031)	$2,467 ($5,435)	$2,432 ($5,357)
Estimated GTO payload cost per lb (kg)	$10,651 ($23,486)	$11,890 ($26,217)	$13,857 ($30,556)	N/A	$10,204 ($22,500)	$6,729 ($14,837)	$12,598 ($27,778)

Table 3: Heavy Launch Vehicles (more than 25,000 lbs. to LEO)



Vehicle name	Ariane 5G	Long March 3B	Proton	Space Shuttle	Zenit 2	Zenit 3SL
Country/Region of origin	Europe	China	Russia	USA	Ukraine	Multinational
LEO capacity lb (kg)	39,648 (18,000)	29,956 (13,600)	43,524 (19,760)	63,443 (28,803)	30,264 (13,740)	34,969 (15,876)
Reference LEO altitude km (mi)	342 (550)	124 (200)	124 (200)	127 (204)	124 (200)	124 (200)
GTO capacity lb (kg)	14,994 (6,800)	11,466 (5,200)	10,209 (4,630)	13,010 (5,900)	0	11,576 (5,250)
Reference site and inclination	Kourou 5.2 deg.	Xichang 28.5 deg.	Baikonur 51.6 deg.	KSC 28.5 deg.	Baikonur 51.4 deg.	Odyssey Launch Platform 0 deg.
Estimated launch price (2000 US$)	$165,000,000	$60,000,000	$85,000,000	$300,000,000	$42,500,000	$85,000,000
Estimated LEO payload cost per lb (kg)	$4,162 ($9,167)	$2,003 ($4,412)	$1,953 ($4,302)	$4,729 ($10,416)	$1,404 ($3,093)	$2,431 ($5,354)
Estimated GTO payload cost per lb (kg)	$11,004 ($24,265)	$5,233 ($11,538)	$8,326 ($18,359)	$23,060 ($50,847)	N/A	$7,343 ($16,190)

Unlike the other vehicles listed in Tables 1-3, the Space Shuttle is not available commercially and thus does not have a launch *price*, per se, associated with it. Instead, the estimated *cost* (to NASA) to fly one shuttle mission is listed in Table 3. There are several ways to compute the cost of a shuttle mission, ranging from dividing the total NASA budget for the shuttle by the number of launches each year to estimating the marginal cost of one additional shuttle flight. The former method can produce per-launch costs of over $500 million, while the latter can lower the cost below $100 million. NASA's Space Transportation Architecture Study in the late 1990s estimated a shuttle launch cost of $300 million, based on an annual budget of $2.4 billion and eight flights a year, a rate NASA approached or achieved for most of the 1990s. We adopt the $300 million cost figure for this analysis, although we note that in the last few years the shuttle flight rate has dropped significantly without an appreciable decrease in the shuttle program budget, which would result in a sharp increase in per-launch costs.

The price-per-pound figures in Tables 1 through 3 span a wide range. There is a general trend of lower prices per pound for larger launch vehicles due to the economies of scale that a larger vehicle provides. The data also show that non-Western (Chinese, Russian, and Ukrainian) vehicles tend to have lower prices than their Western (American and European) counterparts, primarily because of lower labor and infrastructure costs. Table 4 shows that these differences in average price per pound can be significant.

Table 4: Average Price Per Pound for Western and Non-Western Launch Vehicles

Vehicle Class	LEO		GTO	
	Western	Non-Western*	Western	Non-Western*
Small	$8,445	$3,208	$18,841	N/A
Medium/Intermediate	$4,994	$2,407	$12,133	$9,843
Heavy	$4,440	$1,946	$17,032	$6,967

* The Zenit 3SL is considered a non-Western launch vehicle because of its Ukrainian and Russian heritage.

While this approach is simple, it has a key disadvantage: it treats launch vehicles as commodity items with a fixed price and capacity. In reality this is not the case. Negotiations for launch vehicles can result in widely varying prices, depending on customer requirements, the existing supply of and demand for launch services, and any special provisions like bulk buys of launch vehicles or the exchange of equity or services for launch services. In addition, each launch vehicle is used for one time only and can be uniquely tailored to some degree to meet the needs of each payload.

Specific Price Per Pound Calculation

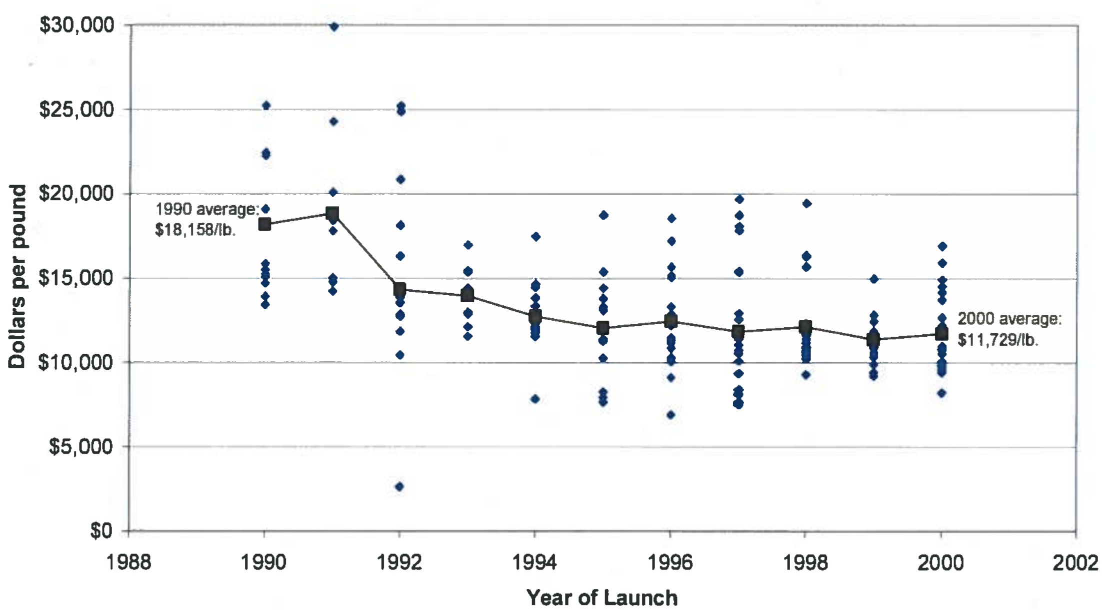
A more robust approach is to compute the price per pound of each launch individually. This increases the fidelity of the data and also provides a clearer view of pricing trends over time. The pricing information used in this study comes from Futron's database of launch vehicle prices. Wherever possible, the specific price for each launch is used; otherwise, the average price for that launch vehicle is used instead. All prices are normalized to constant year 2000 dollars to permit meaningful comparisons from year to year. For each launch the total mass of the payload, rather than the rated capacity of the vehicle, is used to compute the price per pound. Using payload mass instead of capacity makes only a modest difference in the price-per-pound results for GSO launches, as the mass of the payloads averaged 80-90% of the capacity of the vehicles (for some launches the payload mass exceeds the capacity of the vehicle – in those cases vehicles use reserve propellant designed to guarantee proper orbital insertion, trading away some accuracy for increased performance). For launches into NGSO the choice in approach does make a significant difference in the results, as noted later in this White Paper.

For this analysis we treat the GSO and NGSO launch markets separately. Other than the Delta 2, there is little overlap between these markets in the launch vehicles used. The technical requirements for reaching these orbits are also very different, affecting the payload capacity and thus price per pound. While the GSO market has a single, well-defined orbit, the NGSO market encompasses a wide array of orbital altitudes and inclinations. For this analysis, we will treat these as a group despite these differences, noting that most of these payloads are launched into low-Earth orbits several hundred kilometers high in non-polar inclinations. The results of this analysis are discussed in the following sections.

GSO Experience

The cost of launching commercial payloads into GSO, as measured using the price-per-pound metric, dropped significantly in the 1990s. The average price per pound fell from \$18,158 in 1990 to \$11,729 in 2000 (measured using constant year 2000 dollars), a drop of 35% during the decade. As illustrated in Figure 1, however, prices did not decrease steadily throughout the decade. Instead, there was a sharp decrease in launch prices in the first half of the decade, with prices then holding constant around \$12,000 per pound through the latter half of the 1990s.

Figure 1: Estimated Launch Price Per Pound for Commercial GSO Payloads (constant 2000$)



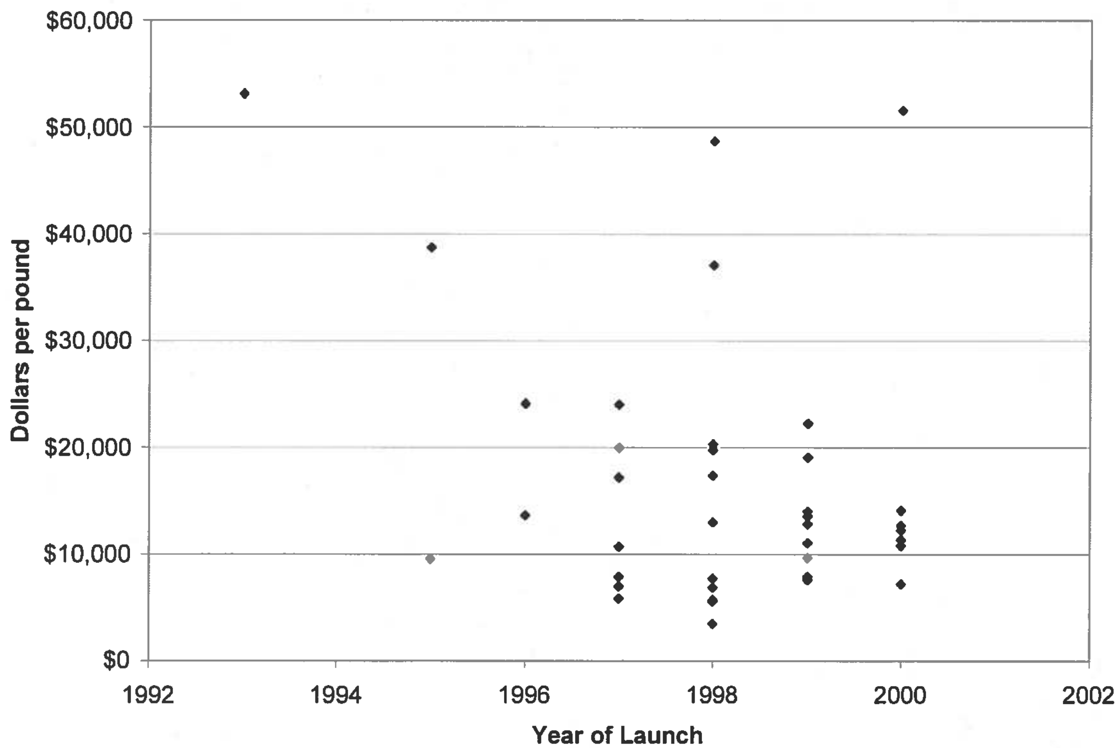
This pricing trend can be explained in large part by increased competition in the commercial launch industry, notably the introduction of the Chinese Long March and the Russian Proton launch vehicles to the market in the early and mid 1990s, respectively. These vehicles were aggressively priced compared to their Western counterparts, creating downward pressure on prices for the overall launch market. While three families of

Western launch vehicles – Ariane, Atlas, and Delta – account for 81% of the commercial GSO launches in this time frame, the average price per pound for these launches in a given year was as much as $1,000 higher than the average for all vehicles. By the end of the decade, however, Western vehicles had reached rough parity on a price-per-pound basis with their non-Western counterparts; this can be seen in Figure 1 by noticing the reduced scatter in the data points in the late 1990s versus the early 1990s. The departure of Chinese boosters from the commercial GSO market in the late 1990s because of reliability and export control issues likely also played a role in stabilizing prices by removing one of the sources of downward pressure on prices.

NGSO Experience

There is considerably less data for commercial NGSO launch pricing, simply because there have been far fewer commercial launches to NGSO than to GSO: there were 64 commercial NGSO launches in the 1990-2000 period, compared to nearly 200 commercial GSO launches. The price-per-pound data for those commercial NGSO launches, normalized to constant year 2000 dollars, are plotted in Figure 2.

Figure 2: Estimated Launch Price Per Pound for Commercial NGSO Payloads (constant 2000$)



Unlike the commercial GSO data, there are no clear trends in the price per pound of commercial NGSO launches. Most launches since 1995 cluster near $10,000 per pound, although there are some significantly higher outliers. These results can be explained in large part by the fact that commercial NGSO payloads have generally utilized a far smaller portion of the vehicle's stated capacity than GSO launches. While GSO payloads used 80-90% of a vehicle's capacity on average during the 1990-2000 period, NGSO payloads have used less than half of a vehicle's stated capacity during this time, even though many NGSO launches are multi-manifested with several spacecraft on each launch vehicle. Technical limitations of putting spacecraft in multiple orbital planes on one launch, as well as a desire to limit the exposure of a satellite constellation to a launch failure, put limits on multi-manifested launches that fall short of the vehicle's payload capacity. Many of the data points in Figure 2 with values greater than $20,000 a pound can be explained by launches that use a quarter or less of the capacity of the vehicles.

Conclusions

The price per pound of launch vehicles can be computed in two ways: a generic estimate using estimated launch vehicle costs and published payload capacities, and a more specific computation for each launch event using the cost of the vehicle for that launch and the mass of its payload. The generic metric offers a qualitative measure of launch industry costs, with lower prices per pound for larger as well as non-Western vehicles. The specific metric avoids a number of disadvantages of the generic metric and permits a more detailed, quantitative study of launch costs over the period.

The commercial GSO launch market has shown, on a specific price-per-pound basis, significant reductions in launch costs between 1990 and 2000. This can be attributed in large part to increased competition, particularly the introduction in the first half of the decade of lower-priced Chinese and Russian launch vehicles. The overall commercial launch market adjusted by lowering prices, stabilizing around $12,000 per pound in the latter half of the 1990s, nearly one-third less than launch costs at the beginning of the decade.

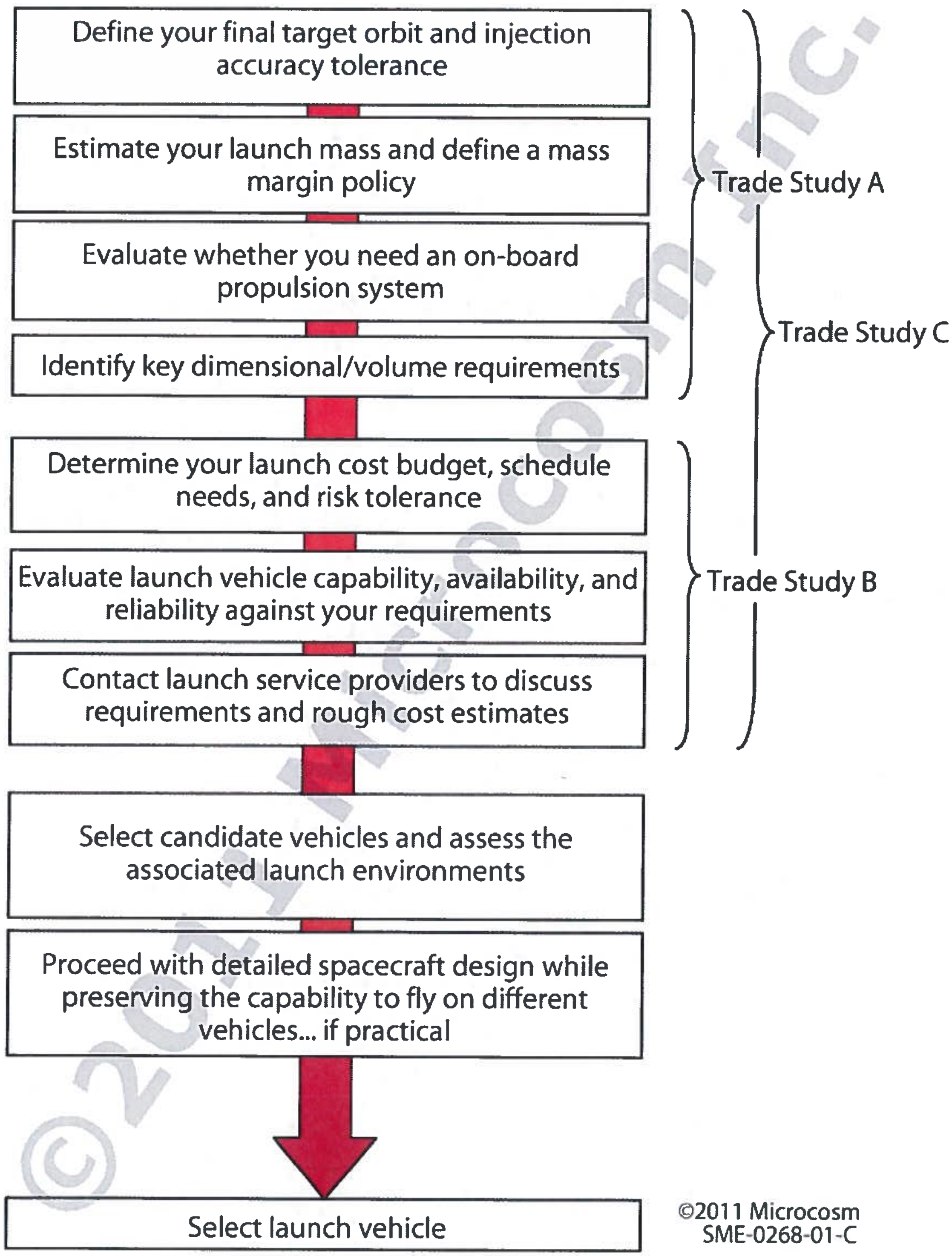
While this decrease is significant, it has failed to stimulate increased commercial activity. Since the late 1990s demand for commercial launches has dropped dramatically, from over 35 launches a year in the late 1990s to only 16 in 2001. Most industry forecasts show that commercial launch activity is unlikely to return to the peak levels of the late 1990s for the next ten years, despite the reduction in launch costs. This demonstrates the price inelasticity of the current launch market.

Regarding NGSO launch prices, there is no clear trend in the price-per-pound metric, other than a clustering around $10,000 per pound in the late 1990s. While this is lower than GSO launches, it is not as low as one might expect, because NGSO payloads have generally used a lower fraction of a vehicle's capacity than GSO payloads.

Although any decrease in the cost of space access is heartening, the decreases in price per pound seen in the last decade, as significant as they may be, appear to fall well short of what is needed to trigger major increases in commercial space activity. Far deeper cuts in the price per pound to orbit, such as the $1,000/pound goal of NASA's Space Launch Initiative, may be necessary to promote growth of the commercial space sector.

Text 26.1: Launch Vehicle Selection (for a given payload)

Process Flow: (Vastly simplified)



Text 26.1: Launch Vehicle Selection

Target Orbit and Injection Accuracy

Vehicle	LEO Apse Altitude (km)	Inclination (deg)
Minotaur IV (HAPS)*	< ±18.5	±0.05
Dnepr	±4–10	±0.04–0.05
Rockot	±1.5%**	±0.05
Delta II (two-stage)	±9.3	±0.05

* Hydrazine Auxiliary Propulsion System
**e.g., ±6 km for 300 km altitude

good example of modern US vehicle accuracy

- Types of mission for spacecraft in orbit:
  - communications
  - exploration – surface sensors
  - surveillance
  - education
  - astronomy
  - etc

Each defines spacecraft w/ different design specs, and different orbits → these define the launch vehicle requirements

- Orbit injection error → require propulsive ΔV to correct

Text 26.1: Launch Vehicle Selection

- Orbit injection accuracy :
    - liquid prop rockets : highest accuracy
      can command shut down
    - solid prop rockets : lower accuracy
      shutdown when fuel is gone
- Final stage solid rocket → "kick motor"
- Orbit adjustment by auxiliary propulsion → "trim"

Example : Minotaur IV all-solid stages with
$\underbrace{\text{liquid hypergolic trim propulsion system}}$

but... add'l mass & cost + complexity

Text 26.1: Launch Vehicle Selection

Launch Vehicle Capability, Availability, Reliability

- Capability: launch vehicle's ability to meet spacecraft requirements
  (launchers may be modified, but $$)

- Availability:
  - in stock or planned manufacture
  - launch site influence: can't launch every rocket at every launch site.
    Also recall $i \geq$ latitude of launch site
    changing i is prop-expensive
    Some launch sites (Guiana) are ~ equatorial

- Reliability: Based on past performance
  (table 26-2)

Text 26.1: Launch Vehicle Selection

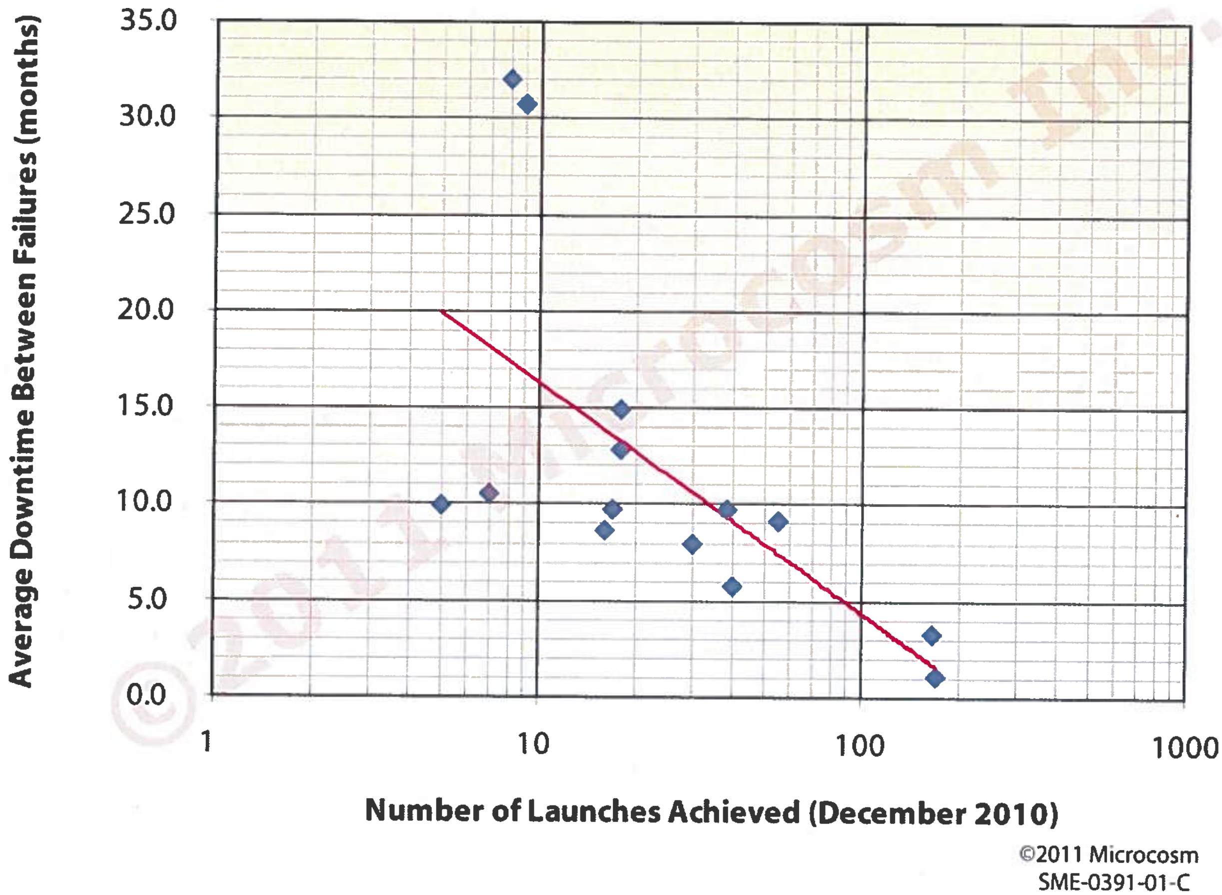
Launch Vehicle Reliability

Launch Vehicle	No. of Successful Launches	Total No. of Launches	R
Atlas V	23	23	1.000
Delta II	163	165	0.988
Delta IV	15	15	1.000
Falcon 1	2	5	0.400
Falcon 9	2	2	1.000
Minotaur I	9	9	1.000
Minotaur IV	2	2	1.000
Pegasus XI	37	40	0.925
Space Shuttle	130	132	0.985
Taurus	6	8	0.750
Long March 2C/D	46	46	1.000
Long March 3A/B/C	36	38	0.947
Long March 4	22	22	1.000
Ariane 5	52	55	0.945
PSLV	17	18	0.944
GSLV	4	7	0.571
Shavit	6	9	0.667
H-IIA	17	18	0.944
H-IIB	2	2	1.000
Dnepr	15	16	0.938
Proton (since 1970)	321	348	0.922
Rockot	16	17	0.941
Soyuz	1654	1753	0.944
Zenit	28	30	0.933

Table 26-2. Reliability Experience of Launch Vehicles as of December 31, 2010.

- Every vehicle with > 50 launches has had a failure
- Cost and time (and reputation Δ) to recover

Text 26.1: Launch Vehicle Selection



∴ more experience in launching → faster recovery from failure

- Failure at the launch ahead of you will affect your schedule.

also Wikipedia: "Comparison of Orbital Launch Systems"

Text 26.1: Launch Vehicle Selection

Nation	Vehicle	Orbits Serviced	Performance (mT) LEO	GTO	Active Launch Sites*	Comments
USA	Atlas V	LEO, GTO, GSO, Escape	9.4–18.5	4.8–8.9	CCAFS, VAFB	Two payload fairing sizes, one to five strap-on solid motors
	Delta II	LEO, GTO, Escape	2.5–5.5	0.8–2.0	CCAFS, VAFB	Three, four or nine strap-on solid motors
	Delta IV	LEO, GTO, Escape	9.2–22.6	4.3–13	CCAFS, VAFB	Two upper stage sizes, two or four strap-on solid motors, or three-core "Heavy"
	Falcon 1	LEO	1	n/a	RTS	
	Falcon 9	LEO, GTO	10.4	4.5	CCAFS	
	Minotaur I	LEO	0.6	n/a	WFF, VAFB	Uses decommissioned ICBM motors for the first two stages
	Minotaur IV	LEO	1.7	n/a	WFF, VAFB	Uses decommissioned ICBM motors for the first three stages
	Pegasus XL	LEO	0.45	n/a	Air-mobile	Prior launches from CCAFS, RTS, VAFB, WFF, and the Canary Islands
	Taurus	LEO	1.4	n/a	VAFB	Vehicle and launch equipment designed to be road-, rail-, and sea-portable to austere launch sites
China	Long March 2C/D	LEO, GTO	3.8	1.2	JSLC, TSLC, XSLC	
	Long March 3A/B/C	GTO	n/a	2–5.5	XSLC	Primarily marketed for GTO missions
	Long March 4	LEO	4.2	n/a	JSLC, TSLC	Mostly sun-synchronous use
European Union	Ariane 5	LEO, GTO	20	10	CSG	Two upper stage choices, ECA (cryogenic) and ES (storable) for LEO and GTO missions, respectively
India	PSLV	LEO, GTO	1.6	1	SDSC	The LEO performance figure quoted by ISRO is for sun-synchronous
	GSLV	GTO	n/a	2.5	SDSC	No LEO performance quoted by ISRO
Israel	Shavit	LEO	0.3*	n/a	Palamchim AFB	Launches to date have been westward, however a polar capability is being offered and the launch system design is intended to operate from austere sites
Japan	H-IIA	LEO, GTO	n/a	4–6	TSC	Two to four strap-on solid rocket motors. This vehicle is primarily marketed for GTO missions
	H-IIB	LEO	16.5	8	TSC	Used for International Space Station servicing missions to date
Russia	Dnepr	LEO	3.7	n/a	Baikonur, Yasny	Converted RS-20 (NATO SS-18) ICBM, with commercial services marketed by ISC Kosmotras
	Proton	LEO, GTO, Escape	23	6.9	Baikonur	Commercial services marketed by International Launch Services (ILS)
	Rockot	LEO	2	n/a	Plesetsk	Converted UR100N (NATO SS-19) ICBM, with commercial services marketed by Eurockot, GmBH
	Soyuz	LEO, GTO, Escape	5.0–8.2	1.5–3.0	Baikonur, Plesetsk, CSG	Commercial services marketed by Starsem (Baikonur) and Arianespace (CSG)
Ukraine	Zenit	LEO, GTO	n/a	n/a	Baikonur, Sea-mobile	Commercial services marketed by Sea Launch, LLC

*** Site Names/Acronyms**

Baikonur—Baikonur Cosmodrome, Kazakhstan
CCAFS—Cape Canaveral Air Force Station, Florida
CSG—Guiana Space Center, Guiana, France
JSLC—Jiuquan Satellite Launch Center, China
RTS—Reagan Test Site, Kwajalein Atoll, Marshall Islands
SDSC—Satish Dhawan Space Center, Sriharikota, India
TSC—Tanegashima Space Center, Japan
TSLC—Taiyuan Satellite Launch Center, China
VAFB—Vandenberg Air Force Base, California, United States
XSLC—Xichang Satellite Launch Center, China
WFF—NASA Wallops Flight Facility, Virginia, United States
Yasny—Yasny Launch Base, Orenburg Region, Russia

Nation	Vehicle	Orbits Serviced	Performance (mT) LEO	Performance (mT) GTO	Active Launch Sites*	Comments
USA	Atlas V	LEO, GTO, GSO, Escape	9.4–18.5	4.8–8.9	CCAFS, VAFB	Two payload fairing sizes, one to five strap-on solid motors
	Delta II	LEO, GTO, Escape	2.5–5.5	0.8–2.0	CCAFS, VAFB	Three, four or nine strap-on solid motors
	Delta IV	LEO, GTO, Escape	9.2–22.6	4.3–13	CCAFS, VAFB	Two upper stage sizes, two or four strap-on solid motors, or three-core "Heavy"
	Falcon 1	LEO	1	n/a	RTS	
	Falcon 9	LEO, GTO	10.4	4.5	CCAFS	
	Minotaur I	LEO	0.6	n/a	WFF, VAFB	Uses decommissioned ICBM motors for the first two stages
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	Pegasus XL	LEO	0.45	n/a	Air-mobile	Prior launches from CCAFS, RTS, VAFB, WFF, and the Canary Islands
	Taurus	LEO	1.4	n/a	VAFB	Vehicle and launch equipment designed to be road-, rail-, and sea-portable to austere launch sites
China	Long March 2C/D	LEO, GTO	3.8	1.2	JSLC, TSLC, XSLC	
	Long March 3A/B/C	GTO	n/a	2–5.5	XSLC	Primarily marketed for GTO missions
	Long March 4	LEO	4.2	n/a	JSLC, TSLC	Mostly sun-synchronous use

Nation	Vehicle	Orbits Serviced	Performance (mT) LEO	Performance (mT) GTO	Active Launch Sites*	Comments
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India	PSLV	LEO, GTO	1.6	1	SDSC	The LEO performance figure quoted by ISRO is for sun-synchronous
	GSLV	GTO	n/a	2.5	SDSC	No LEO performance quoted by ISRO
Israel	Shavit	LEO	0.3*	n/a	Palamchim AFB	Launches to date have been westward, however a polar capability is being offered and the launch system design is intended to operate from austere sites
Japan	H-IIA	LEO, GTO	n/a	4–6	TSC	Two to four strap-on solid rocket motors. This vehicle is primarily marketed for GTO missions
	H-IIB	LEO	16.5	8	TSC	Used for International Space Station servicing missions to date
Russia	Dnepr	LEO	3.7	n/a	Baikonur, Yasny	Converted RS-20 (NATO SS-18) ICBM, with commercial services marketed by ISC Kosmotras
	Proton	LEO, GTO, Escape	23	6.9	Baikonur	Commercial services marketed by International Launch Services (ILS)
	Rockot	LEO	2	n/a	Plesetsk	Converted UR100N (NATO SS-19) ICBM, with commercial services marketed by Eurockot, GmBH
	Soyuz	LEO, GTO, Escape	5.0–8.2	1.5–3.0	Baikonur, Plesetsk, CSG	Commercial services marketed by Starsem (Baikonur) and Arianespace (CSG)
Ukraine	Zenit	LEO, GTO	n/a	n/a	Baikonur, Sea-mobile	Commercial services marketed by Sea Launch, LLC

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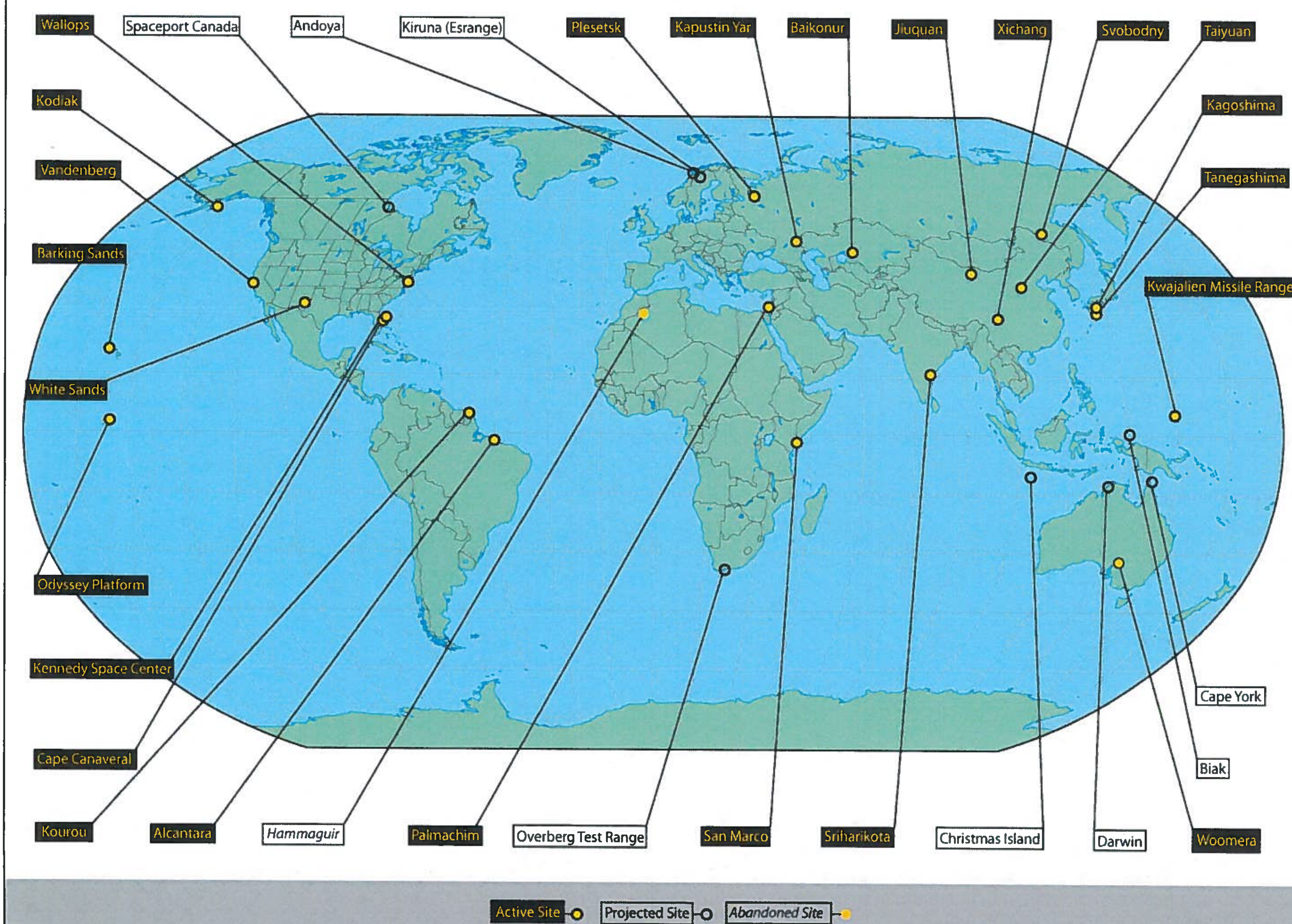
RTS—Reagan Test Site, Kwajalein Atoll, Marshall Islands
SDSC—Satish Dhawan Space Center, Sriharikota, India
TSC—Tanegashima Space Center, Japan
TSLC—Taiyuan Satellite Launch Center, China
VAFB—Vandenberg Air Force Base, California, United States
XSLC—Xichang Satellite Launch Center, China
WFF—NASA Wallops Flight Facility, Virginia, United States
Yasny—Yasny Launch Base, Orenburg Region, Russia


Launchers

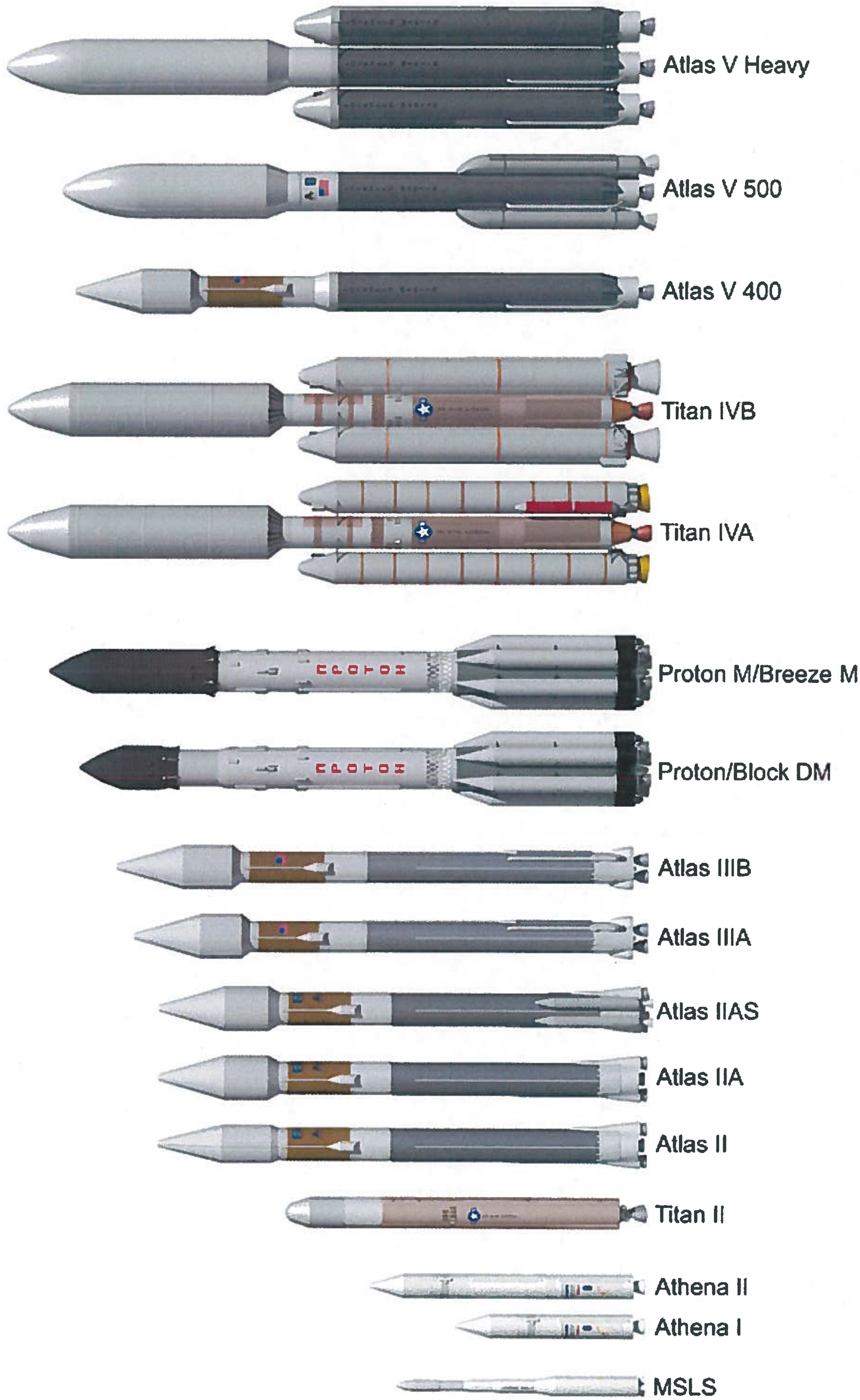
Countries with Civilian Launch Capability

- Brazil (2)
- China (9)
- European Space Agency (5)
- Japan (3)
- India (3)
- Israel (1)
- Russia (17)
- Ukraine (6)
- USA (21) excl Shuttle

Worldwide Launch Sites



Rocket Comparison

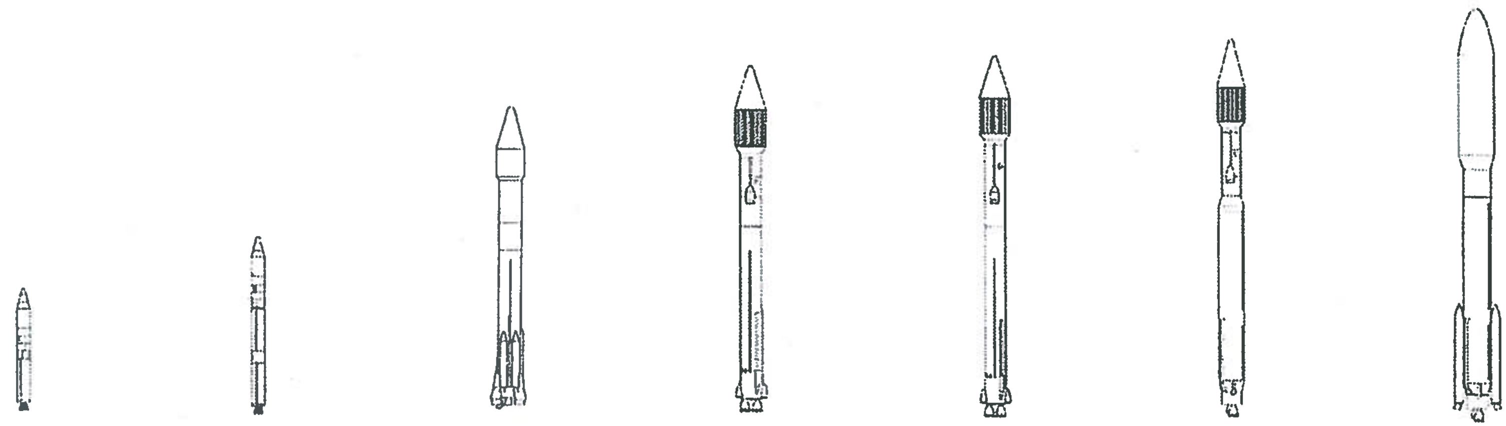


Heavy-Lift Rockets

	Payload to LEO (lbm)
Past:	
• Saturn 5	260,000
• Space Shuttle	28,800
Current:	
• Soyuz FG	7,000
• Falcon 9	10,454
• Atlas V 500	20,520
• Delta IV Heavy	23,260
• Titan IVB	21,680
• Long March 2F	?
• Ariane 5ECB	~21,600
• Proton M	21,000
Future:	
• SLS	308,000

US Launchers

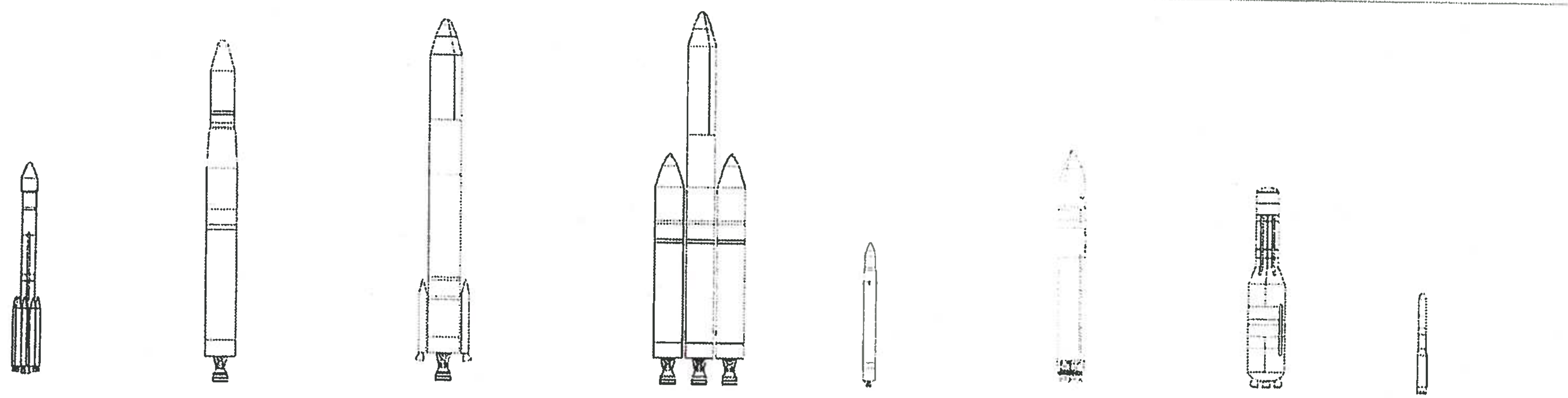
United States



Vehicle	**Athena I**	**Athena II**	**Atlas IIAS**	**Atlas IIIA**	**Atlas IIIB**	**Atlas V 400**	**Atlas V 500**
Performance							
LEO Maximum	820 kg (1805 lbm)	2065 kg (4520 lbm)	8618 kg (19,000 lbm)	8640 kg (19,050 lbm)	10,759 kg (23,720 lbm)	12,500 kg (27,558 lbm)	Up to 20,520 kg (45,239 lbm)
SSO	360 kg (790 lbm)	1165 kg (1565 lbm)	?	—	—	?	?
GTO	—	590 kg (1290 lbm)[1]	3719 kg (8200 lbm)	4037kg (8900 lbm)	4119 kg (9081 lbm)	Up to 4950 kg (10,913 lbm)	8670 kg (19,114 lbm)
Cost	$40–45 million	$45–50 million	Price negotiable	Price negotiable	Price negotiable	Price negotiable	Price negotiable
First Flight	1995	1998	1993	2000	2002	2002	2003
Launch Site(s)	Cape Canaveral Kodiak	Cape Canaveral Kodiak	Cape Canaveral Vandenberg	Cape Canaveral	Cape Canaveral	Cape Canaveral	Cape Canaveral

US Launchers

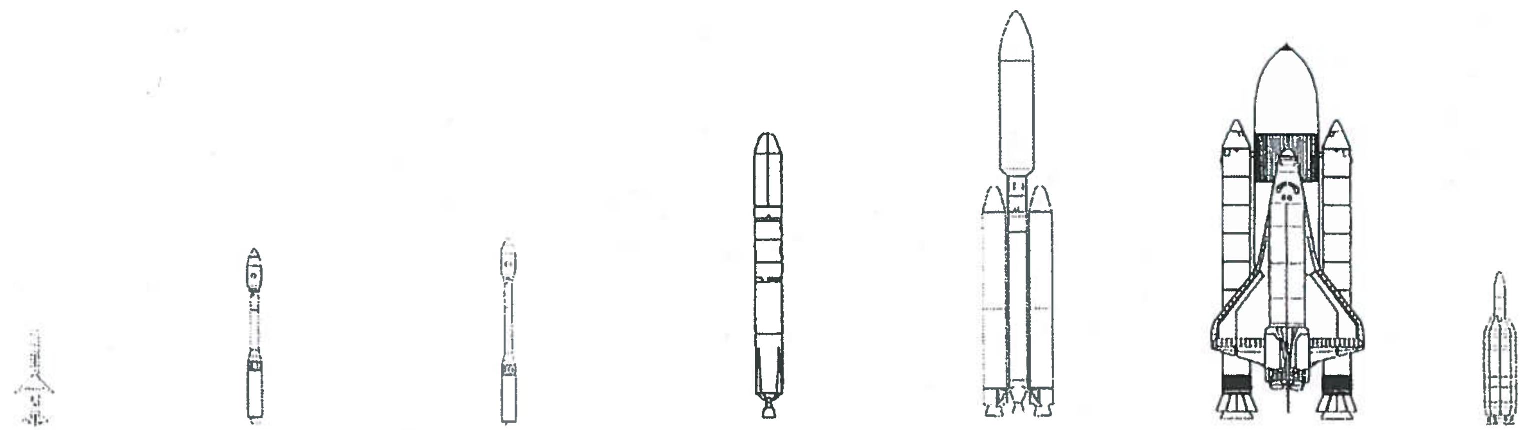
United States (continued)



Delta II	Delta IV Medium	Delta IV Medium+	Delta IV Heavy	Falcon I	Falcon V	K-1	Minotaur
Up to 5102 kg (11,249 lbm)	8870 kg (19,554 lbm)	Up to 13,327 kg (29,381 lbm)	23,260 kg (51,280 lbm)	668 kg (1472 lbm)	5040 kg (11,088 lbm)	4600 kg (10,150 lbm)	607 kg (1339 lbm)
Up to 3186 kg (7025 lbm)	6832 kg (15,062 lbm)	10,863 kg (23,949 lbm)	19,665 kg (43,354 lbm)	408 kg (898 lbm)	3173 kg (6980 lbm)	1250 kg (2750 lbm)	317 kg (700 lbm)
Up to 1841 kg (4058 lbm)	3934 kg (8672 lbm)	6411 kg (14,135 lbm)	12,369 kg (27,269 lbm)	—	1500 kg (3300 lbm)	1570 kg (3460 lbm)	—
Price negotiable	Price negotiable	Price negotiable	Price negotiable	$5.9 million	$12 million	$17 million	$15–20 million
1990	2002	2002	2004	2004	2005	?	2000
Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Woomera Nevada Test Site	Vandenberg Others

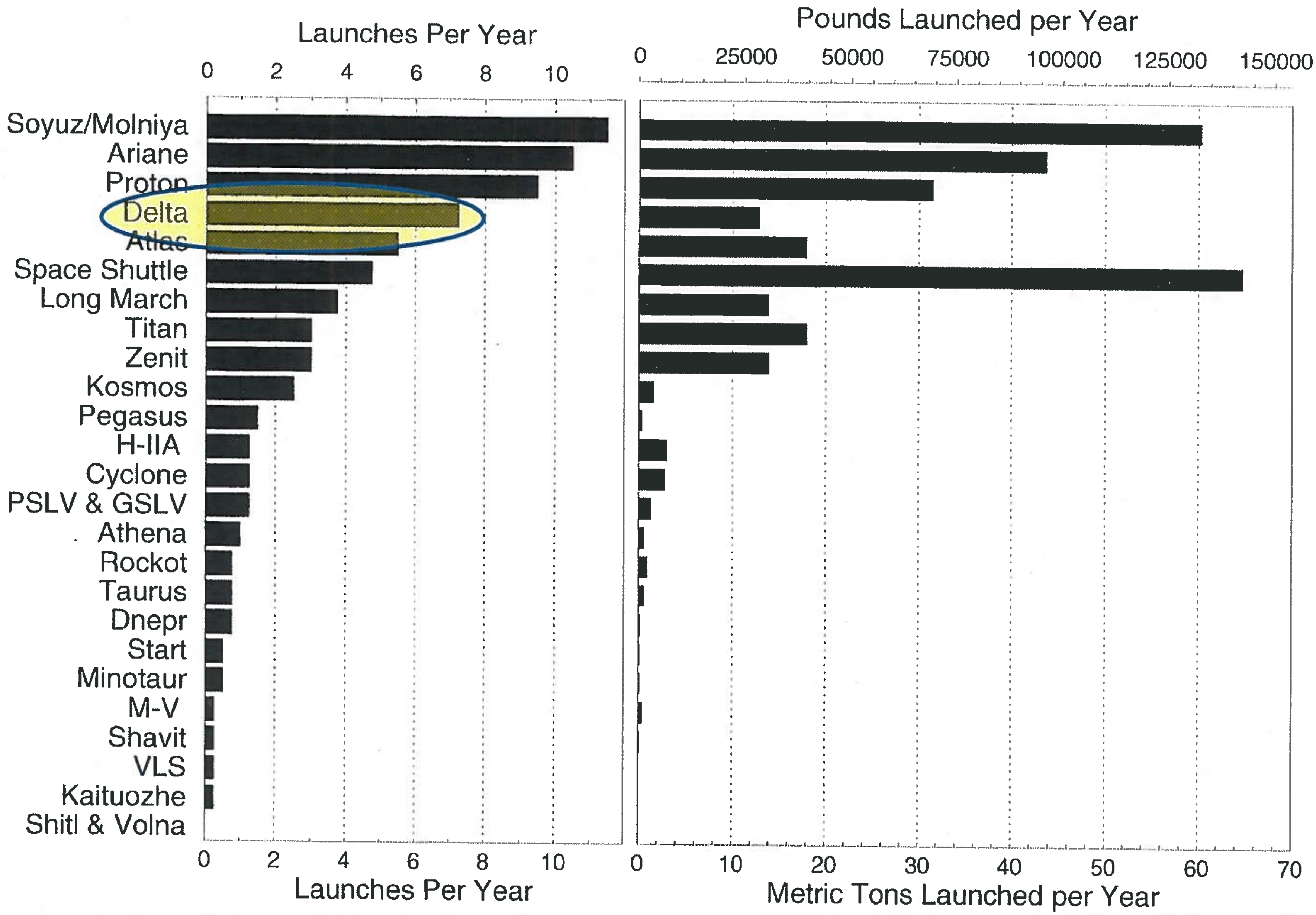
US Launchers

United States (continued)



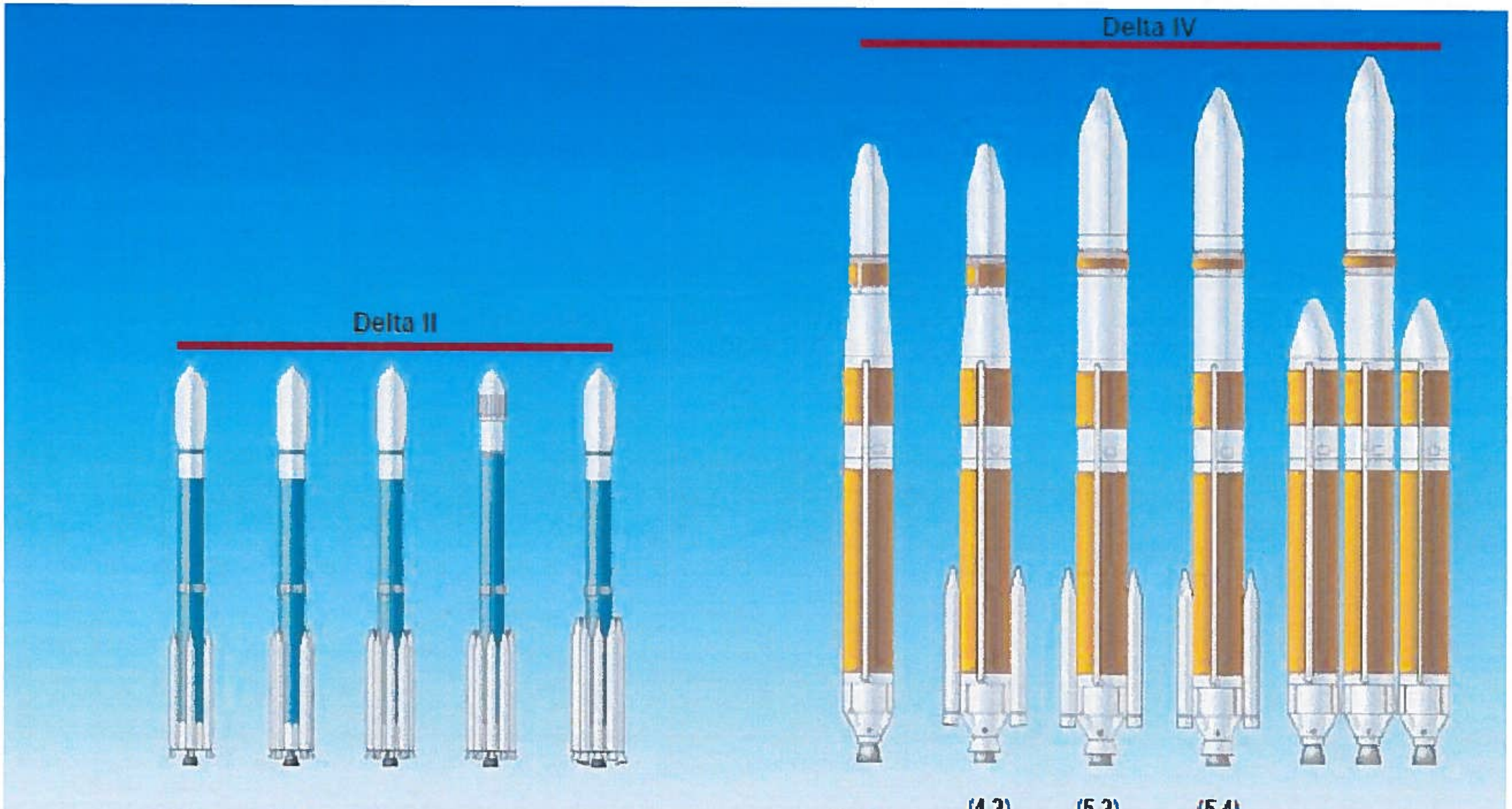
Vehicle	**Pegasus XL**	**Commercial Taurus**	**Taurus XL**	**Titan II**	**Titan IVB**	**Space Shuttle**	**Scorpius**
Performance							
LEO Maximum	443 kg (997 lbm)	1370 kg (3020 lbm)	1590 kg (3505 lbm)	1900 kg (4200 lbm)	21,680 kg (47,800 lbm)	28,800 kg (63,500 lbm)	314 kg (700 lbm)
SSO	190 kg (420 lbm)	720 kg (1590 lbm)	860 kg (2000 lbm)	1100 kg (2425 lbm)	?	—	125 kg (276 lbm)
GTO	—	495 kg (1090 lbm)	557 kg (1228 lbm)	—	?	—	—
Cost	$15–25 million	$25–47 million	$25–47 million	$30–40 million	$350–450 million	$450–750 million	$2.9 million
First Flight	1994	1998	2004	1988	1997	1981	2006
Launch Site(s)	Vandenberg Wallops Cape Canaveral Others	Vandenberg Others	Vandenberg Others	Vandenberg	Cape Canaveral Vandenberg	Kennedy Space Center	Vandenberg Cape Canaveral Wallops

Launch Vehicle Utilization, 1999–2002 Average



One–quarter of the launch vehicle families perform two-thirds of all launches and carry 80% of all cargo.

Example: US Delta IV Med/Heavy

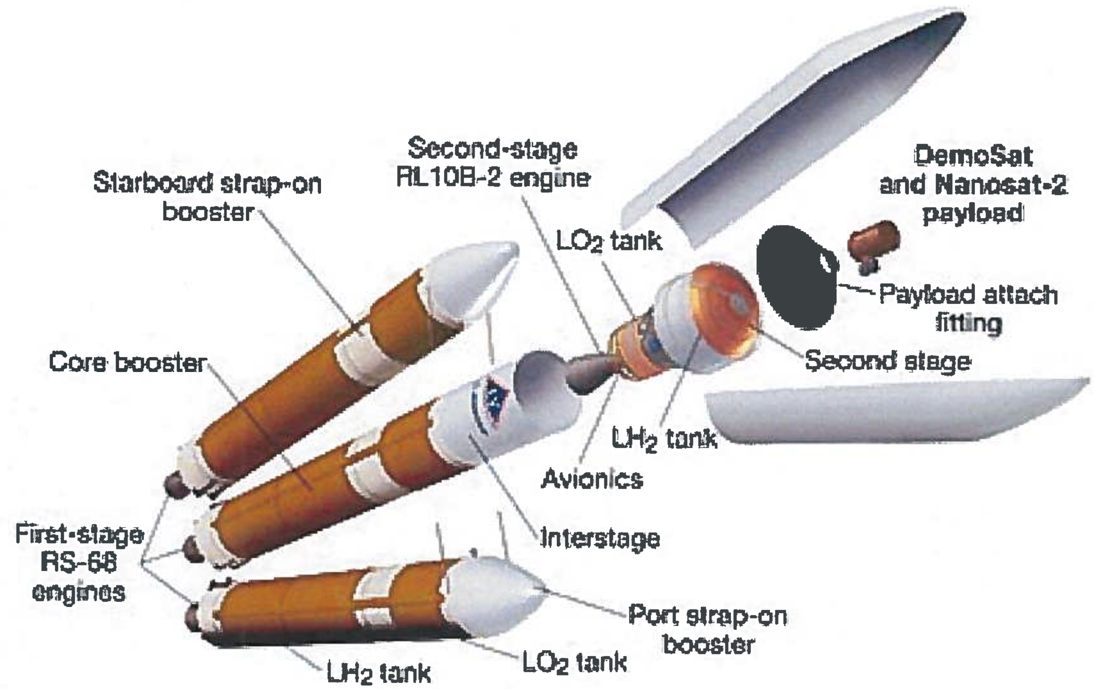


		7326-10	742X-10	7925-10	7925-9.5	7925-H10	Medium	(4,2)	(5,2) Medium+	(5,4)	Heavy
GTO	(kg)	894	1,032	1,759	1,832	2,133	4,231	5,941	4,869	6,822	12,757
LEO	(kg)	2,472	2,880	4,598	4,783	5,648	9,106	12,305	10,616	13,864	21,892
GEO	(kg)						1,138	2,036	1,686	2,786	6,276

GTO = 185 x 35,786 km at 28.7 deg Delta II and 27.0 deg Delta IV. LEO = 407 km circular at 28.7 deg.
GEO = 35,786 km circular at 0 deg

Figure 1. Family of Delta Launch Vehicles

Example: US Delta IV Med/Heavy



Total weight at launch approx 1/3 shuttle launch weight