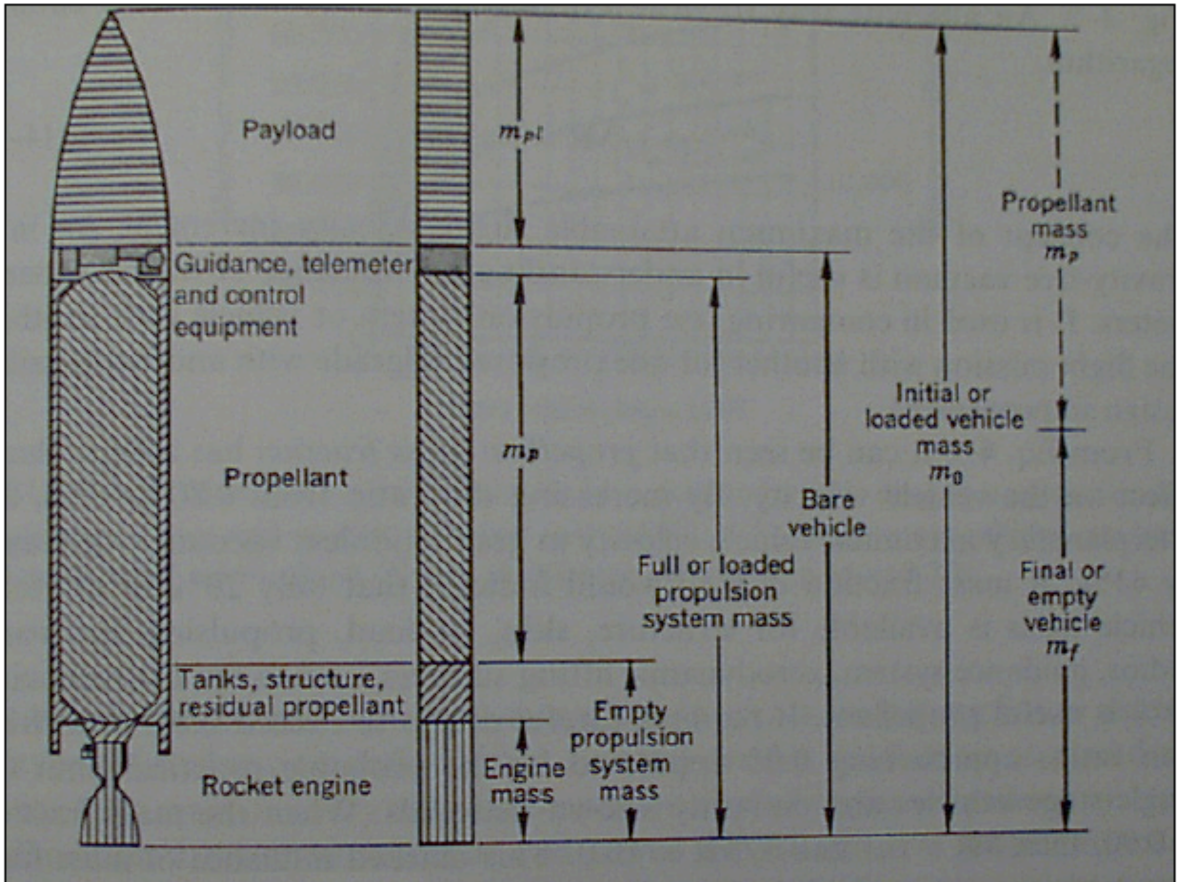
Choice of Launch System:



Definitions of various vehicle masses.

Taylor, Chapter 5.

Steps in the Selection Process

- Mission Needs and Objectives
  - dictate performance, trajectory, launch site
- Dedicated or shared launch
- Mission requirements
  - orbit altitude, inclination, right ascension
  - satellite weight and size
  - date
- Select candidate Launch systems (more than 1!)

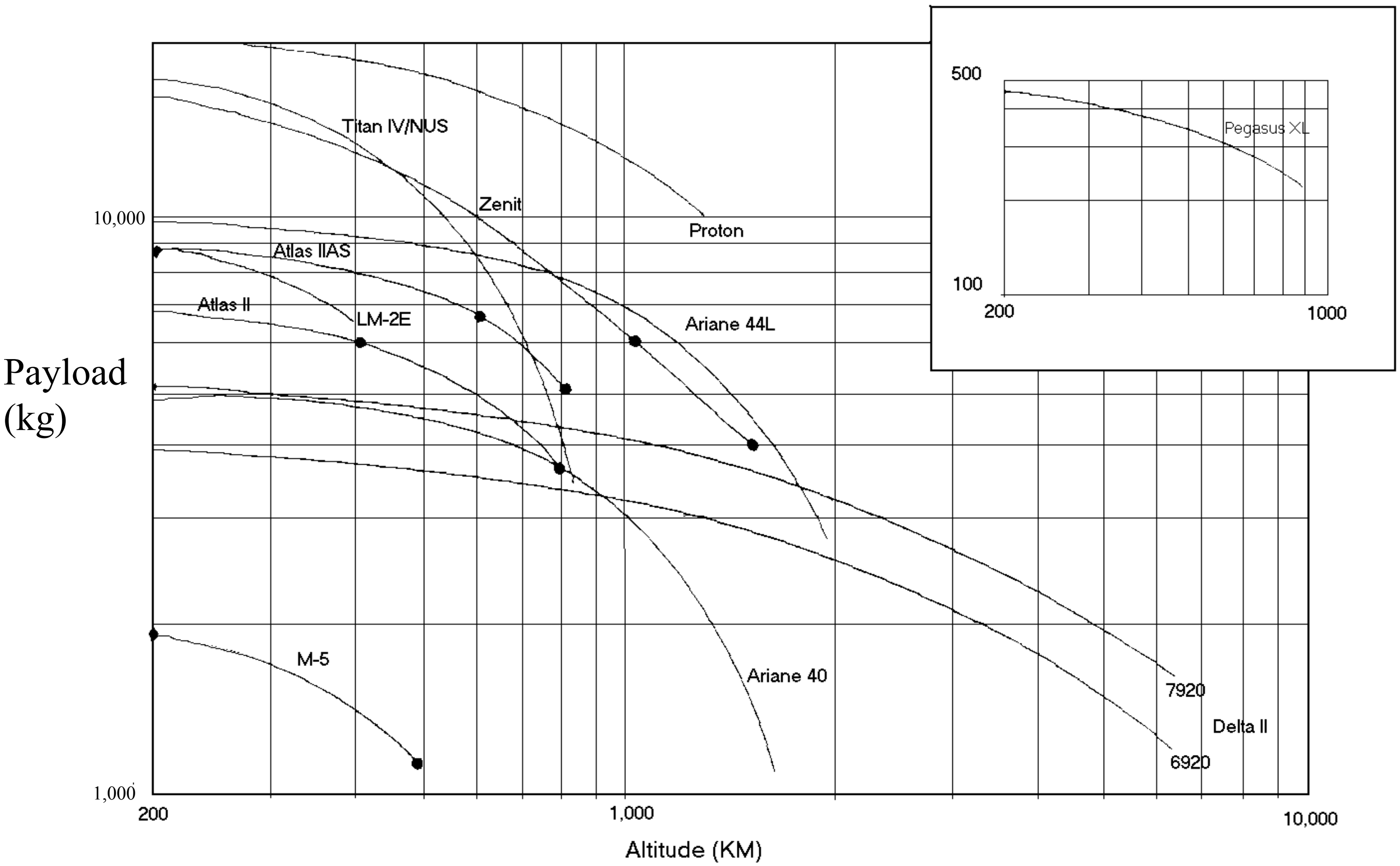
Selection Drivers

- Cost
- What Velocity (ΔV)?
- How Much Weight?
- Reliability
- Availability
- Secondary Issues
  - payload envelope
  - environments
  - interfaces

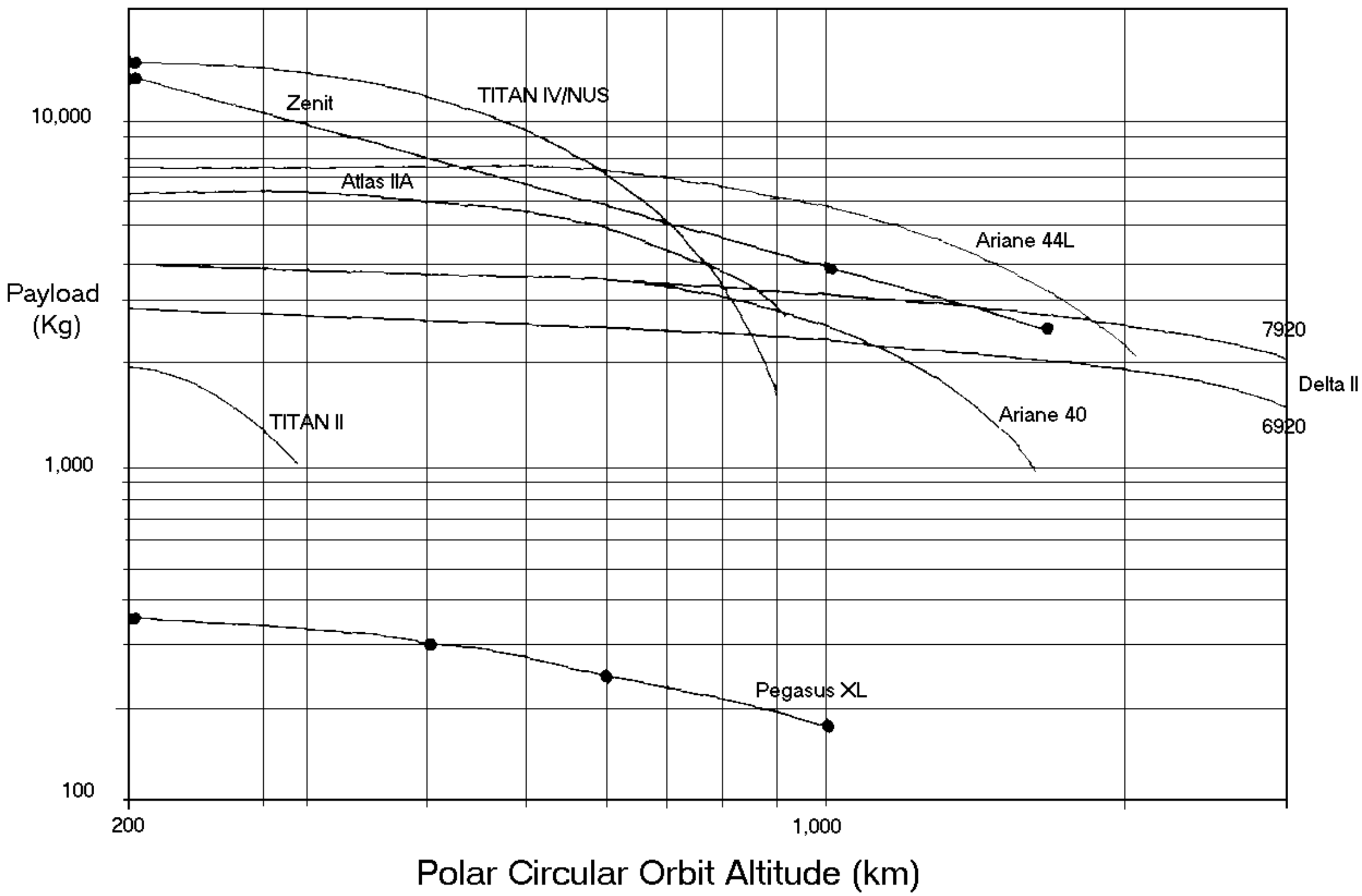
Launch System Issues

- Performance Capability - weight capacity to selected orbit.
- Vehicle availability - Is there a rocket available when you want to launch? How about a matching facility? Ground Stations (launch phase?)
- Spacecraft-to-launcher compatibility - Will your spacecraft survive the launch environments?
- Cost - can you afford it?
- Fairing Size - Will your satellite fit in the nose of the rocket?

Launch Performance I



Launch Performance II



Costs, US systems

Launch Vehicles	Maximum Payload-to-Orbit (kg) LEO	GTO	GEO	Unit Cost (FY00$M)	Cost per kg to LEO (FY00$K/kg)
USA					
Atlas II	6,580	2,810		80–90	12.2—13.7
Atlas II A	7,280	3,039		85–95	11.7–13.0
Atlas II AS	8,640	3,600		100–110	11.6–12.7
Athena 1	800			18	22.5
Athena 2	1,950			26	13.3
Athena 3	3,650			31	8.5
Delta II (7920, 7925)	5,089	1,840		50–55	9.8–10.8
Pegasus XL	460			13	28.3
Saturn V	127,000			820	6.5
Shuttle* (IUS or TOS)	24,400	5,900	2,360	400	16.4
Titan II	1,905			37	19.4
Titan IV	21,640	8,620	5,760 (Centaur)	214 (270)	9.9
Taurus	1,400	450		20–22	14.3–15.7

Costs, Foreign Systems

ESA					
Ariane 4 (AR40)	4,900	2,050		50–65	10.2–13.3
Ariane 4 (AR42P)	6,100	2,840		65–80	10.7–13.1
Ariane 4 (AR44L)	9,600	4,520		95–120	9.9–12.5
Ariane 5 (550 km)	18,000	6,800		130	7.2
CHINA					
Long March C23B	13,600	4,500	2,250	75	5.5
RUSSIA					
Proton SL-13	20,900			55–75	2.6–3.6
Kosmos C-1	1,400			11	7.9
Soyuz	7,000			13–27	1.9–3.9
Tsyklon	3,600			11–16	3.1–4.4
Zenit 2	13,740			38–50	2.8–3.6
JAPAN					
H-2	10,500	4,000	2,200	160–205	15.2–19.5
J-1	900			55–60	61.1–66.7
GTO = Geosynchronous Transfer Orbit; GEO = Geostationary Orbit; LEO = Low-Earth Orbit					

Multi-Stage Rockets

- Advantages:
  - Reduces total vehicle weight for the same payload and delta V
  - …or, increases payload from the same vehicle
  - Increases the max velocity for a given vehicle
  - Decreases required I_{sp}
- Disadvantages:
  - Increased Complexity
  - Decreased Reliability
  - Increased Cost
- Although additional stages improve performance – to a point – the greatest single improvement is with the second stage

Multi-stage Rockets (2)

- In general, the benefit of discarding the empty tanks and structures outweighs the additional cost and complexity.
- For a single stage rocket:

$$\Delta V = g_o I_{sp} \ln\left({m_i}/{m_f} \right) = g_o I_{sp} \ln\left({w_i}/{w_f} \right)$$

- For a multiple stage rocket:

$$\Delta V_t = \Delta V_1 + \Delta V_2 + \Delta V_3 + \ldots$$

- The improvement is because the final weight of stage 1 does not equal the initial weight of stage 2.

• Current state-of-the art-solution

Example

- Single stage Rocket, w_i=121000 lbs, 1000 lb. Payload, 12000 lb structure, I_{sp}=300 sec.

$$\Delta V = g_o I_{sp} \ln\left({}^{w_i}\!\big/_{w_f} \right)$$

$$\Delta V = 32.2 * 300 \ln\left({}^{121000}\!\big/_{13000} \right)$$

$$\Delta V = 21{,}550\ {}^{ft}\!\big/_{\sec}$$

Example (cont'd)

$$w_{i2} = 1{,}000 + 2{,}000 + 18{,}000 = 21{,}000$$

$$w_{f2} = 1{,}000 + 2{,}000 = 3{,}000$$

$$\Delta V_2 = 32.2 * 300 \ln(21000/3000) = 18{,}797\, ft/\sec$$

$$\Delta V_T = \Delta V_1 + \Delta V_2 = 18797 + 13155 = 31{,}952\, ft/\sec$$

Compare to 21,500 for the single stage rocket, same initial weight, structure weight, propellant weight and payload.

Example cont

- Two stage rocket, payload 1000 lbs., stage 1 weighs 10000 lbs. and has 90,000 lbs. propellant, stage 2 weighs 2000 lbs. and has 18000 lbs. propellant. ISP is 300 sec for both.

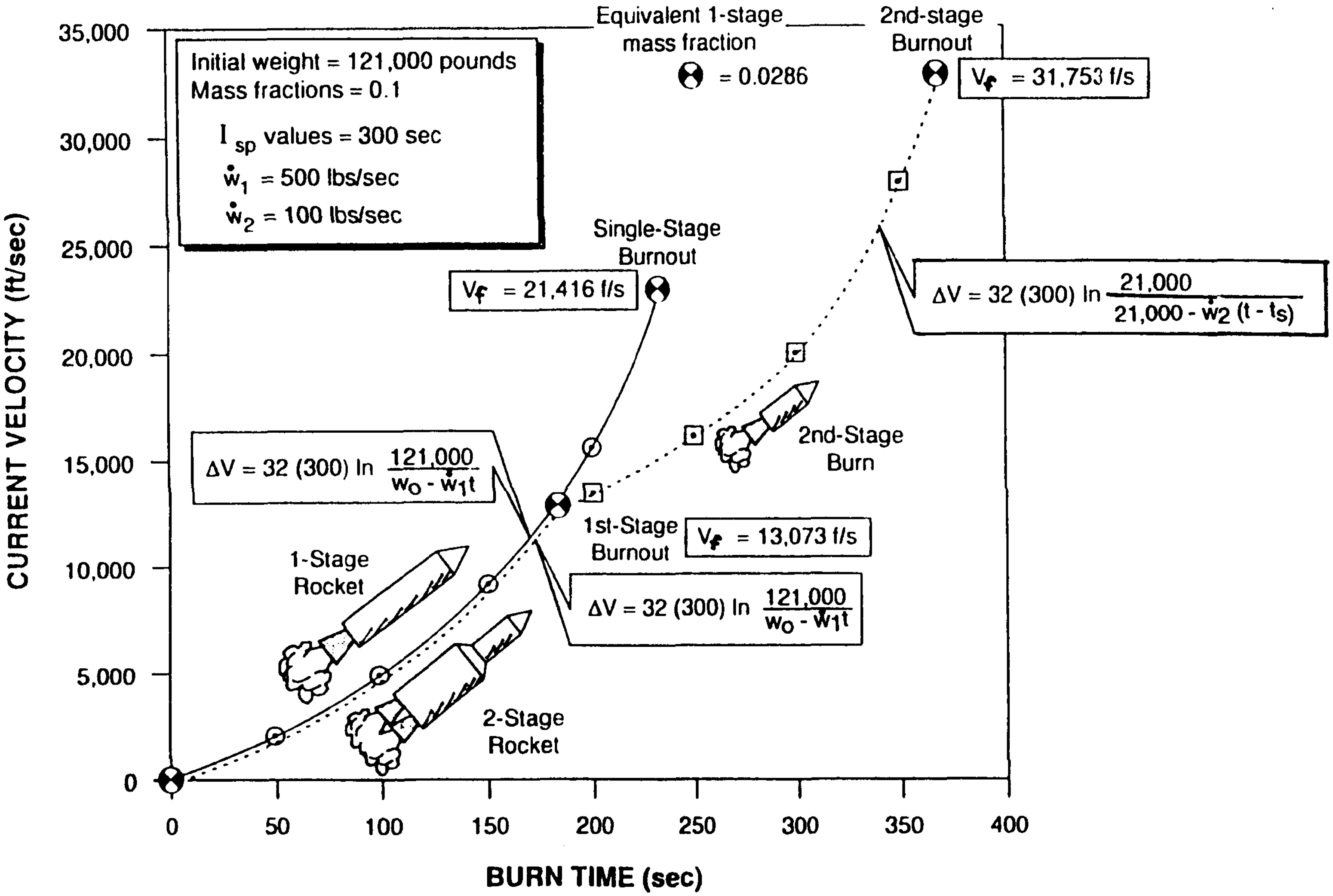
$$w_i = 1{,}000 + 10{,}000 + 90{,}000 + 2{,}000 + 18{,}000 = 121{,}000$$

$$w_{f1} = 1{,}000 + 10{,}000 + 2{,}000 + 18{,}000 = 31{,}000$$

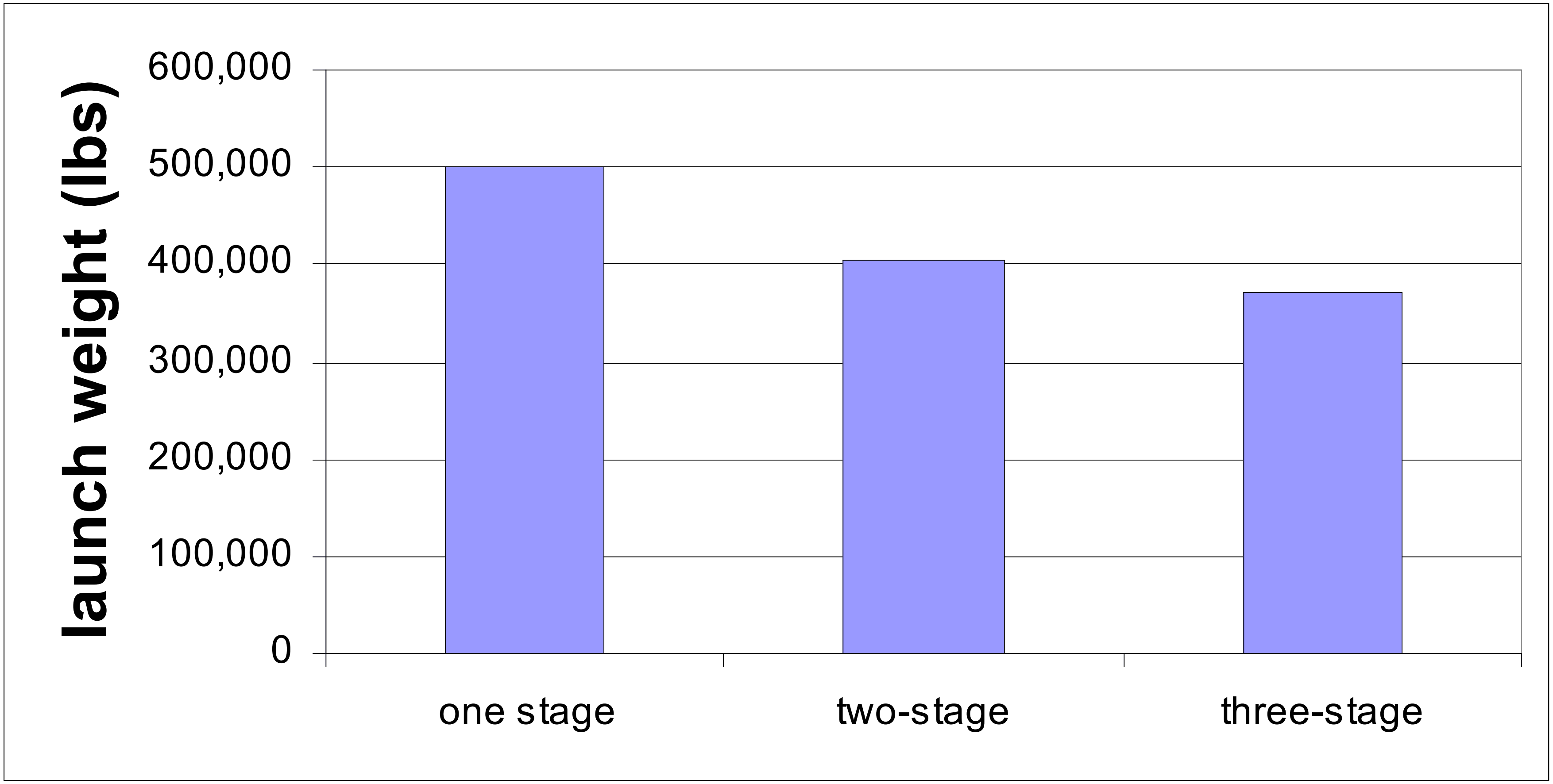
$$\Delta V_1 = 32.2 * 300 \ln\left(\frac{121000}{31000}\right)$$

$$\Delta V_1 = 13{,}155\, ft / \sec$$

Velocity profiles, 1 vs 2 stage Rocket



Comparison of staging launch weights

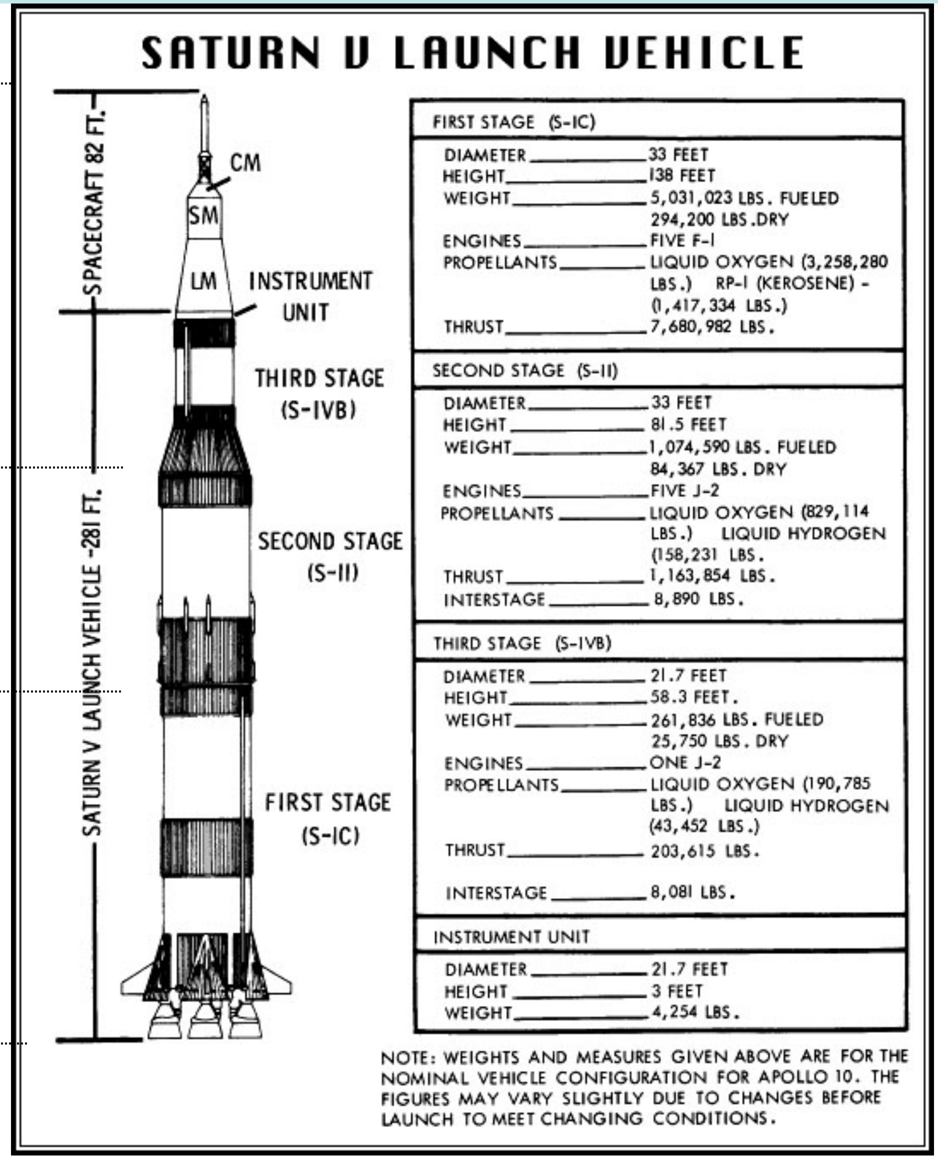


The launch stack

1st Launch stack

2nd launch stack

3rd launch stack

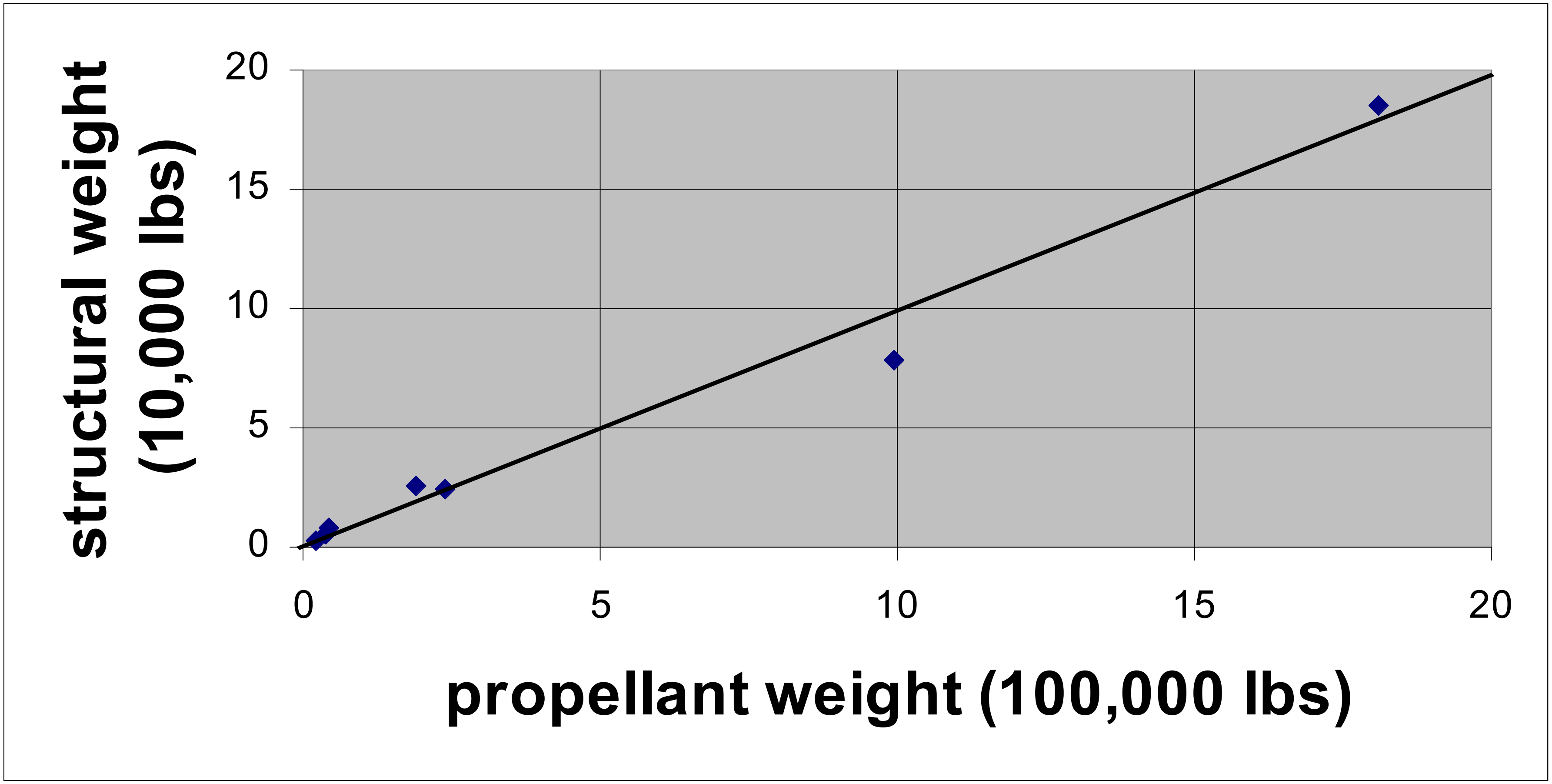


Secondary Considerations

Mass Fractions

- What percentage of each vehicle is devoted to each of the functions … e.g.
  - Gross Propellant mass fraction: 0.85
  - Gross Structure mass fraction: 0.14
  - Gross Payload mass fraction: 0.01
    - spacecraft bus
    - upper stages
    - payload

Structural weight correlation



Margins

- Budget resources!
- Power, Weight, Propellant, Dollars, computer memory space,…...
- Develop an allocation for each component or subsystem, and keep a reserve.
- Weight is the resource that most affects launch systems.
- Weight and Power Budgets Always Grow!

Availability

- Reliability - How likely is it that this one will blow up?
- Production capacity - How many are there, and how fast can the supplier deliver another?
- Operations support - Range issues - How many compatible launch facilities are there, and what is their turnaround time?
- Stand-down after failure.

$$A = 1 - \left[L(1 - R)T_d \Big/ \left(1 - \frac{1}{S}\right) \right]$$

A=availability
L=launch rate
R=reliability
T_d=stand down
S=surge capacity

Launch Environments

A Whole Lot of Shaking Going On

Payload Integration

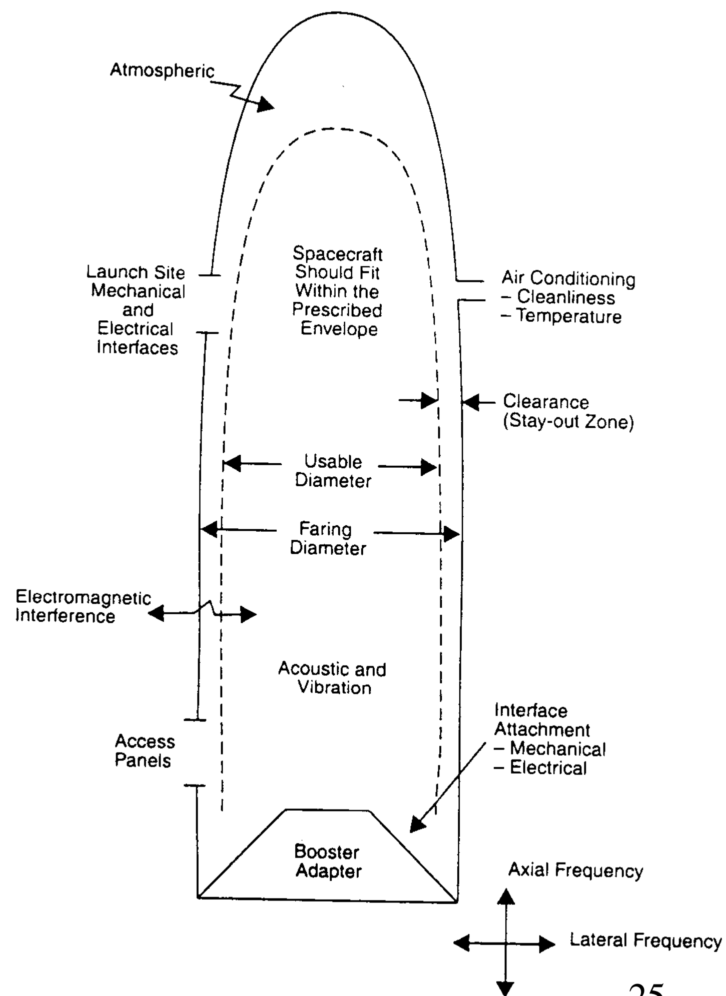
- Match the environments and interfaces of your satellite to several launch vehicles. - design for the worst case.
  - Fairing size and shape
  - Maximum Accelerations
  - Vibration Frequencies and magnitudes
  - Acoustic frequencies and magnitudes
  - Temperature extremes
  - air Cleanliness
  - Orbital Insertion Accuracy
  - Interfaces to launch site and vehicle

Environments and Constraints

Parameter	Typical Value/Comment	Reference
Payload Fairing Envelope	Consult user guide	Fig. 18-8, Table18-4
Payload to Launch Vehicle Interface	Specified bolt pattern	Launch vehicle user guides
Environments		
Thermal	10–35 °C	Launch vehicle user guides
Pad	188 BTU · ft^2/ hr	
Ascent fairing radiant	100–150 BTU · ft^2/ hr	
Aeroheating		
Electromagnetic	Consult range and launch vehicle user guides	
Contamination	Satisfy class 10,000 air	Sec. 18.3
Venting	Maximum of 1 psi differential	Fig. 18-9
Acceleration	5–7 g	Table 18-8
Vibration	0.1 g^2/Hz	Table 18-9, Fig. 18-10
Acoustics	140 dB	Fig. 18-12
Shock	4,000 g	Fig. 18-11

Fairing issues

- Size
- Margins (clearances)
- Protection from aerodynamic loads
  – Heat
  – Buffeting
- Protection from contamination



Structural and Electrical I/F

- Bolt patterns and adapter Rings - part of the payload weight budget.
- Electrical I/F - matching plugs, voltage sense.
- Optical and R/F I/F - depending on the payload, it may need to be tested, examined, or stimulated before launch, but after mating to the launch vehicle.
- Separation devices and separation control circuits
- Communications architecture for the launch and insertion phase.

Payload Environments

- Contamination - conditioned and filtered air post-mate and pre-launch.
- Thermal environment - keep the satellite within the design range (or design the range to match what the vehicle can support.)
- Pressure - flight environment can increase pressure. Satellite and fairing must vent excess pressure as the vehicle approaches vacuum

Acceleration Loads

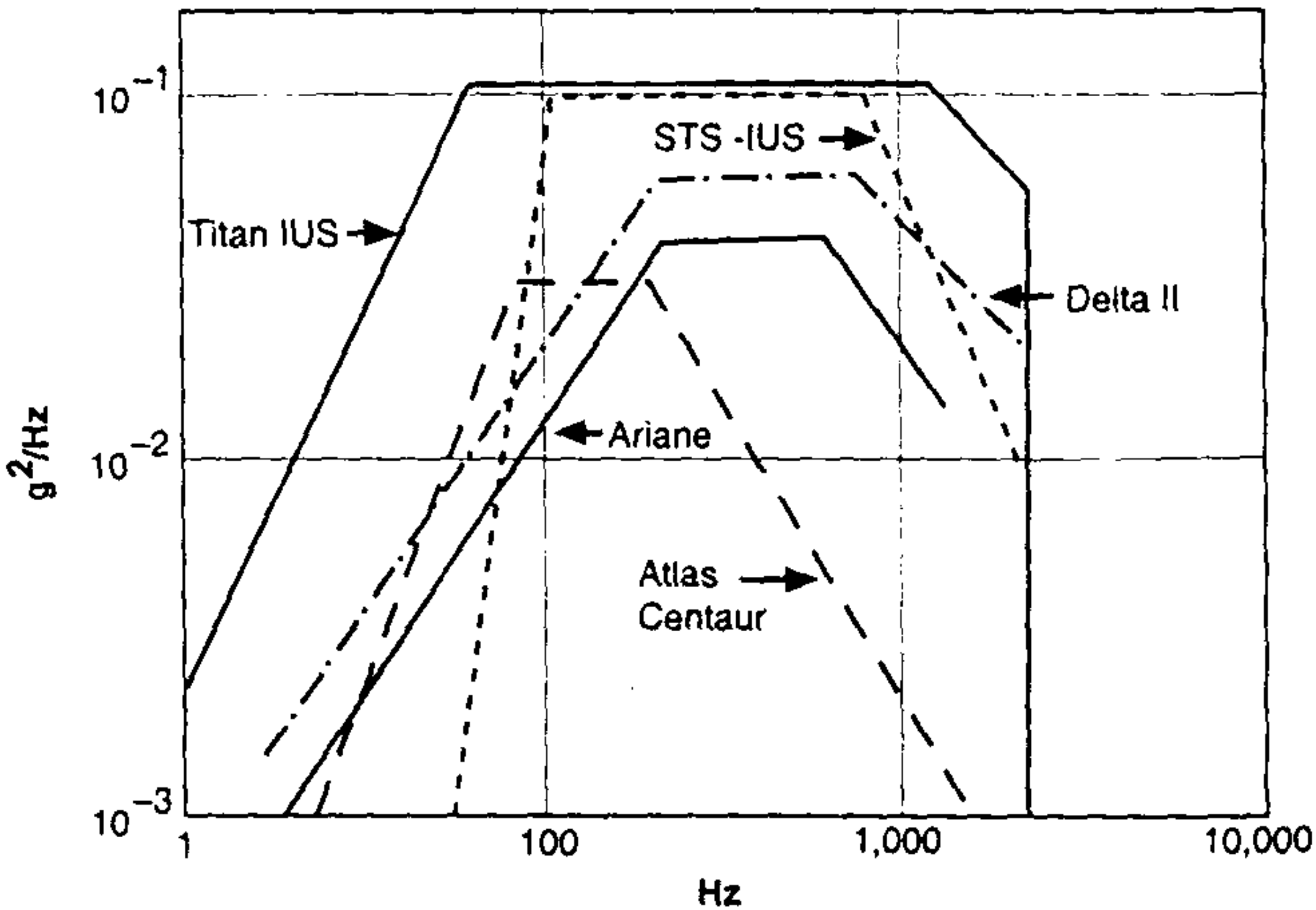
- Static (steady state) and Dynamic (vibration) loads on the vehicle.
- Design for the worst case sum, with margin.
- Causes
  - vehicle acceleration
  - variable engine thrust
  - aerodynamic drag
  - acoustic pressure from the engine
  - response of the vehicle (frequency response)

Acceleration table

	Lift-Off		Max Airloads		Stage 1 Shutdown (Booster)		Stage 2 Shutdown (Booster)	
Vehicle	Axial	Lateral	Axial	Lateral	Axial	Lateral	Axial	Lateral
T34D/IUS								
Steady State	+1.5	—	+2.0	—	0 to +4.5	—	0 to +2.5	—
Dynamic	±1.5	±5.0	±1.0	±2.5	±4.0	±2.0	±4.0	±2.0
Atlas-II								
Steady State	+1.3	—	+2.2	+0.4	+5.5	—	+4.0	—
Dynamic	±1.5	±1.0	±0.3	±1.2	±0.5	±0.5	±2.0	±0.5
Delta (max* all series)								
Steady State	+2.4	—	—	—	—	—	—	—
Dynamic	±1.0	+2.0 to +3.0	—	—	—	—	+6.0	—
H-II								
Steady State	—	—	—	—	—	—	—	—
Dynamic	±3.2	±2.0	—	—	—	—	±5.0	±1.0
Shuttle								
Steady State with IUS	+3.2	+2.5	+1.1 to 3.2	+0.25 to − 0.59	—	—	+3.2	+0.59
Dynamic	+3.5	+3.4	—	—	—	—	—	—

* 2σ Values

Vibration Environment

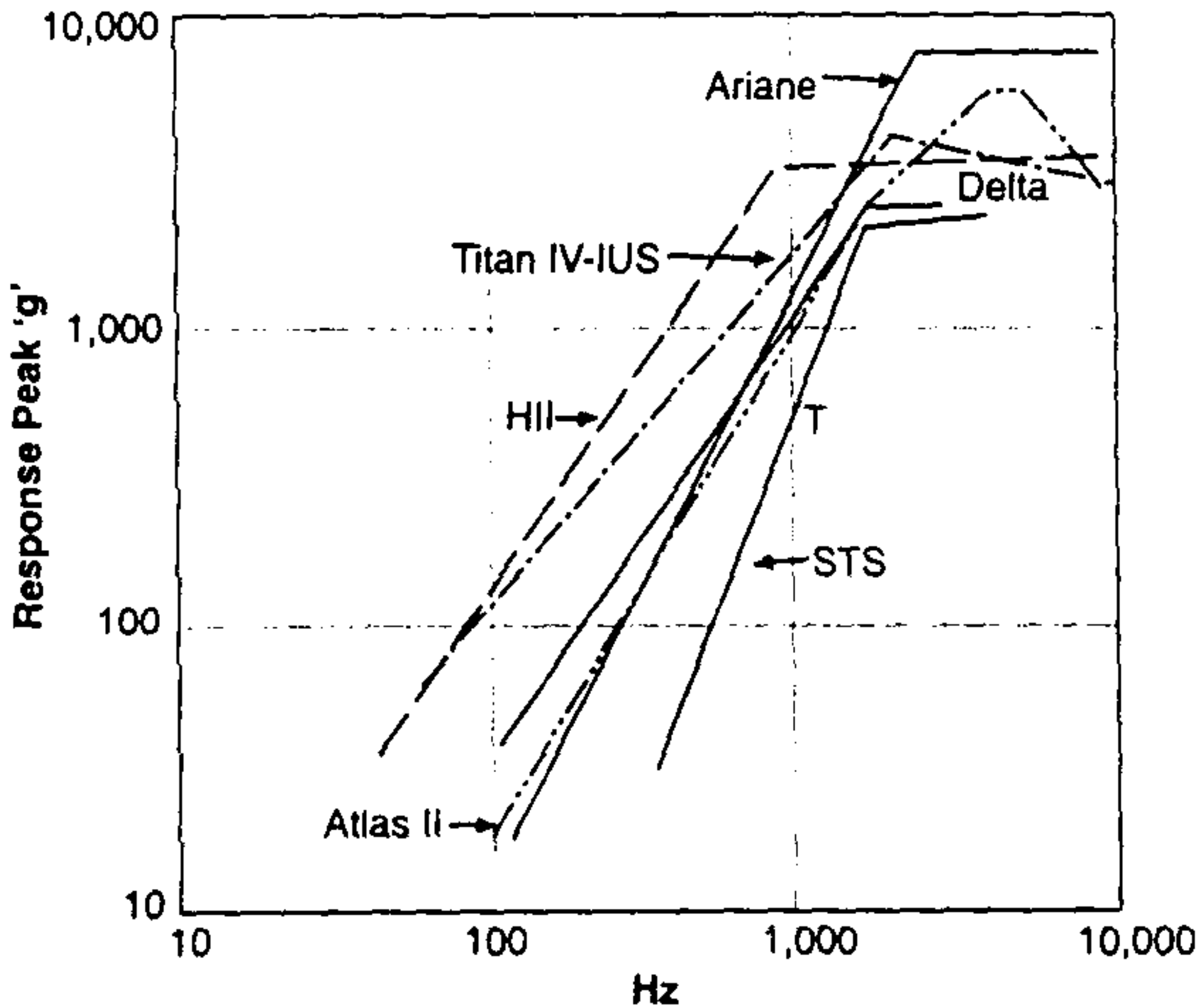


Fundamental Frequencies

Launch System	Fundamental Frequency (Hz)	
	Axial	Lateral
Atlas II, IIA, IIAS	15	10
Ariane 4	*	10
Delta 6925/7925	35	15
Long March 2E	26	10
Pegasus, XL	18	18
Proton	30	15
Space Shuttle	13	13
Titan II	24	10

* 31 Hz for dual payloads, 18 Hz for single payloads.

Shock Environments



Expendable vs. Reusable Launch Systems

Why Develop Re-Usable Launch Systems?

• The surface of Earth lies at the bottom of a deep gravity well and a vast ocean of air

...... the sheer speed required to attain orbit demands a very high order of launch vehicle performance.

• Although US acquired capability to place payloads and people to orbit several decades ago

.... space travel is still an enormously complex, expensive, and dangerous undertaking

• Extremely high cost of space access presents tremendous limitation to large-scale space commercialization

.... to achieve a profit, value of current commercial payloads must literally exceed their weights in gold

Why Develop Re-Usable Launch Systems? (concluded)

• **A NASA Study Conducted in 1992 concluded that in to achieve large-scale space commercialization and/or militarization, then we must**

-- 1) Reduce payload cost to low Earth orbit (LEO) from $20,000 /pound to $1000 /pound within 10-20 years (possible)

-- 2) to $100 /pound within 25-30 years (very unlikely)

-- 3) and finallly, to tens of dollars /pound within 40-50 years. (very, very unlikely)

Why Develop RLV's?

• All space launches to date (except Space Shuttle launches) are based on launch technologies identical to technologies used for warhead delivery.

-- Most or all of the launch-stack is thrown-away each time.

• In 1980's Space Shuttle became the first large-scale launch vehicle in which a substantial portion was reusable.

-- Reasoning "if we don't have to throw the vehicle away each time, launches should be cheaper."

-- Mostly a platitude of "faith" little initial analysis performed to support this conclusion

-- But is this reasoning too simple to account for the real-world factors that are involved in a launch process?

Why Develop RLV's?

- **Example …. Space Shuttle**

  -- Originally envisioned as a measure that would dramatically cut launch cost. .. One size fits all launch & delivery system

  -- However, the current average cost of a medium-lift expendable launch is Approximately $80–$120 million dollars.

  -- Current estimates of Shuttle launch costs run as high as $400 million.

  -- Clearly "man rating" is a factor in cost

- **Becomes obvious that reusability is not the dominant economic factor involved in launch costs.**

  -- What factors are important?

Launch Cost Model

• Groundbreaking paper presented by Dr. James S. Wertz (SMAD) at the International Aerospace Federation Congress in October 2000 addressed this misconception.

-- This paper presented an analytical launch cost model that considered a wide range of cost elements and allowed an objective assessment of launch costs to be performed.

■ Key factors
  - 1) cost of development,
  - 2) cost of recovery,
  - 3) cost of refurbishment,
  - 4) cost of insurance.

-- For a reusable launch vehicle these factors are significantly larger than for an expendable launch stack.

The only cost not incurred by the RLV is the cost of the ELV hardware and assembly.

For a minimal number of flights, the RLV costs far exceed the costs of the ELV hardware and assembly.

Launch Cost Model

$$C_{launch} = C_{development} + C_{vehicle} + C_{flightops} + C_{recovery} + C_{refurb} + C_{insurance}$$

Where:

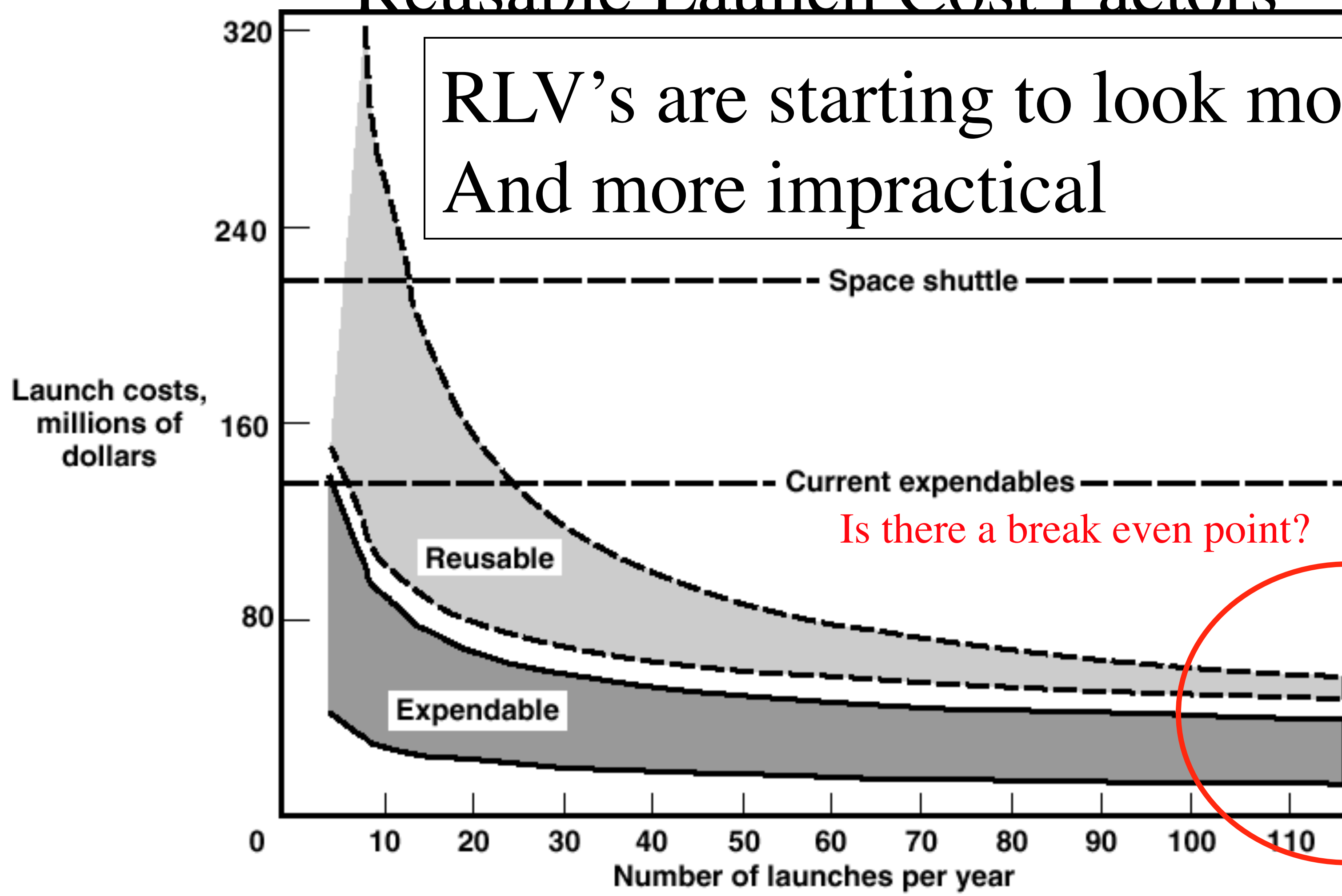
C_{launch}	$\equiv$	Total cost of launch in FY00 dollars (excludes inflation)
$C_{development}$	$\equiv$	Amortization of nonrecurring development cost
$C_{vehicle}$	$\equiv$	Reusable: Amortization of vehicle production cost Expendable: Recurring production cost (Theoretical First Unit cost reduced by learning curve)
$C_{flightops}$	$\equiv$	Total cost of flight operations per flight
$C_{recovery}$	$\equiv$	Recurring cost of recovery (reusable only)
C_{refurb}	$\equiv$	Refurbishment cost (reusable only)
$C_{insurance}$	$\equiv$	Cost of launch insurance

Comparison of Expendable vs. Reusable Launch Cost Factors

ELV	RLV	FACTOR	DISCUSSION
X	X	Amortization of Non-recurring development production cost	Higher for RLV due to larger nonrecurring cost
X	X	**ELV** Recurring production cost **RLV** Amortization of production cost	ELV uses learning curve: RLV is more complex and expensive to produce Amortization rather than recurring production is the major RLV cost savings
	X	Recovery cost	$0 for ELV
	X	Refurbishment cost	May be substantial for RLV; $0 for ELV
X	X	Flight Operations	RLV has more complex systems; more expensive check-out and recovery
X	X	Vehicle insurance	Depends on both replacement cost and reliability; ELV or RLV could be cheaper

Comparison of Expendable vs. Reusable Launch Cost Factors

RLV's are starting to look more
And more impractical



Is there a break even point?

Cost per launch as compared with average launch rate, 2001–2015.

SSTO

- What would the shuttle I_{sp} have to be in order to get to orbit in a single stage (SSTO)?
- Assume same propellant mass fraction

$$I_{sp\ SSTO} = \frac{\Delta V_{SSTO}}{g_0 \ln\left[1 + P_{mf}\right]} = \frac{7608_{m/\sec}}{9.806_{m/\sec^2} \times \ln\left[1 + 5.33\right]} = 420.5_{\sec}$$

- In terms of efficiency we are already there .. If we could just figure a way for the SSME's to produce … 1.5 millions of thrust each!

Single Stage to ORBIT Example

- **Is there a break even cost point for RLV's?**

-- The "faithful" believed so, How?

SSTO!

-- Single Stage to orbit,

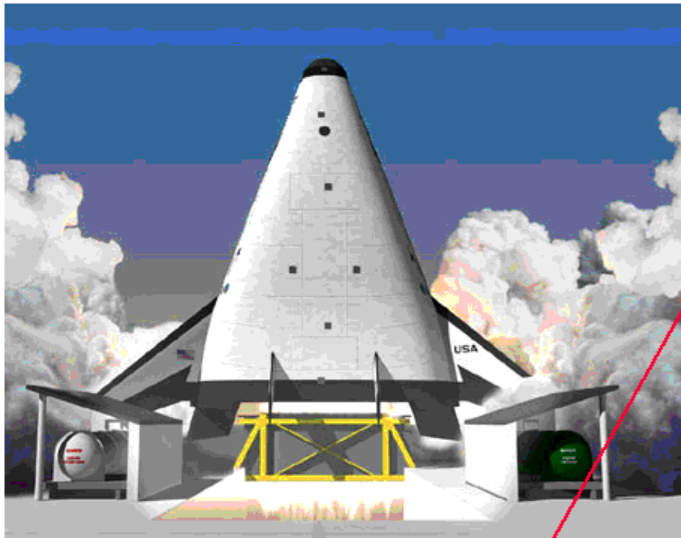
"Holy Grail"of the Economic Space program

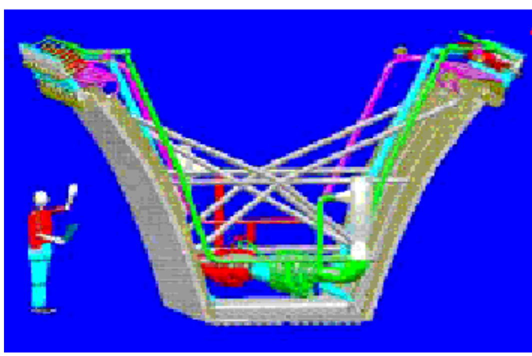
- We'll show

In the next section that SSTO is Impractical with Current state of technology

SSTO: A Real World Example

Lockheed-Martin "Venture-Star" TO LEO





RS-2200 Engine : (Venture-Star)

Manufacturer: Boeing Rocketdyne

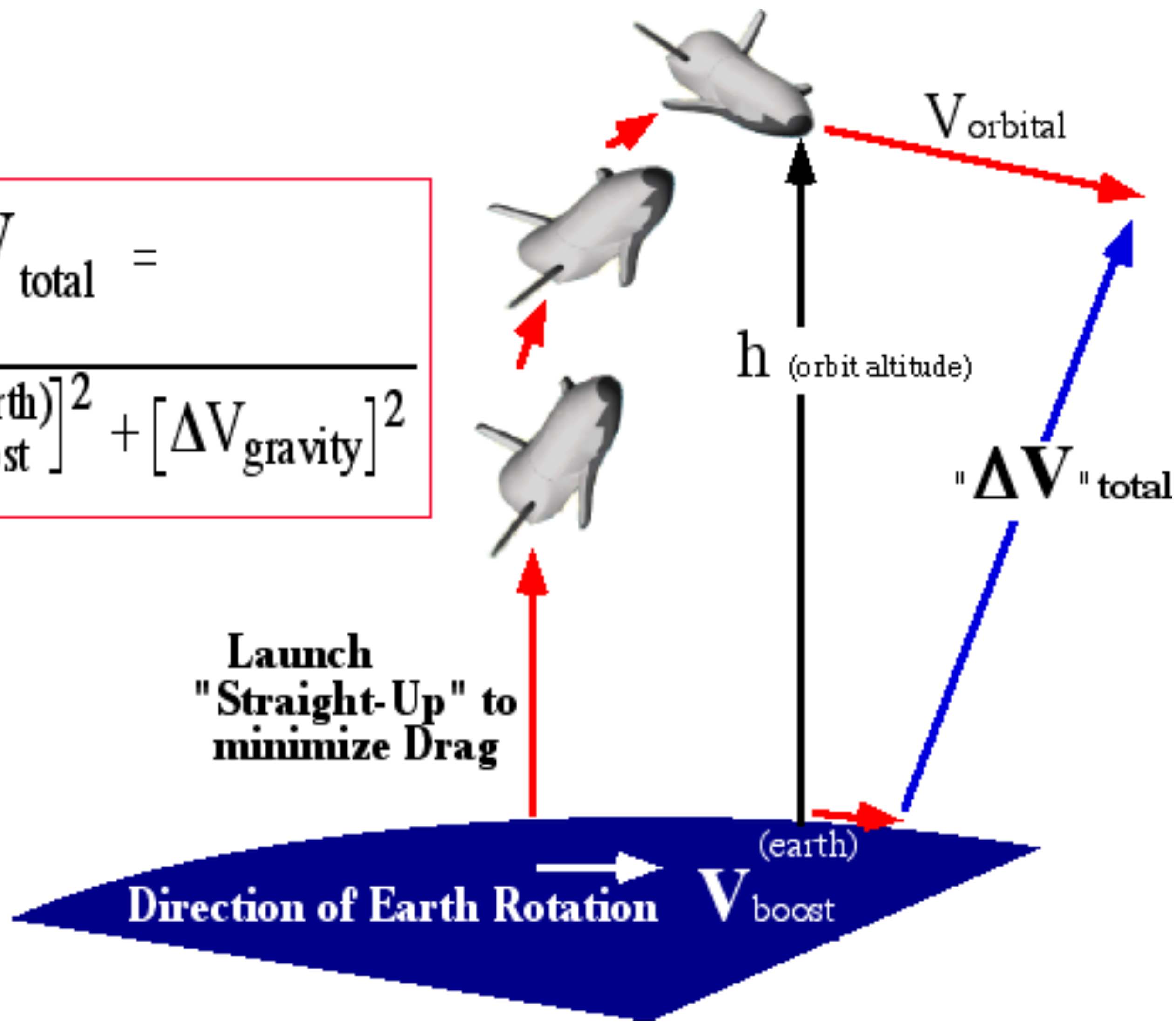
Weight:	**8000 lbs.**
Max Thrust:	**520,000 lbf (Liftoff)**
	564,000 lbf (Space)
Liftoff I_{sp}:	**420 sec**

Mean I_{sp} 453.3

- Use Liftoff I_{sp} --- "Conservative Scenario"
- Calculate the required delta-v to launch into 160 km AGL LEO orbit inclined at 35 deg to equator
- Calculate the propellant mass fraction required to reach this orbit

SSTO: A Real World Example (cont'd)

$$\Delta V_{total} = \sqrt{\left[V_{orbital} - V_{boost}^{(earth)}\right]^2 + \left[\Delta V_{gravity}\right]^2}$$



SSTO: A Real World Example (cont'd)

Could Venture Star Actually Have Achieved SSTO?

- Compute Earth Rotational velocity at 35° (Edwards AFB) latitude

$$V_{rot_{Earth}} = \omega_{Earth} \times r_{Earth} \times \cos[Lat] =$$

$$[0.000072722 \text{ radians}] \times \left[6371 \text{ km} \times 1000 \frac{m}{km}\right] \times \cos\left[\frac{35\pi}{180} \text{ radians}\right] = 379.5 \frac{m}{sec}$$

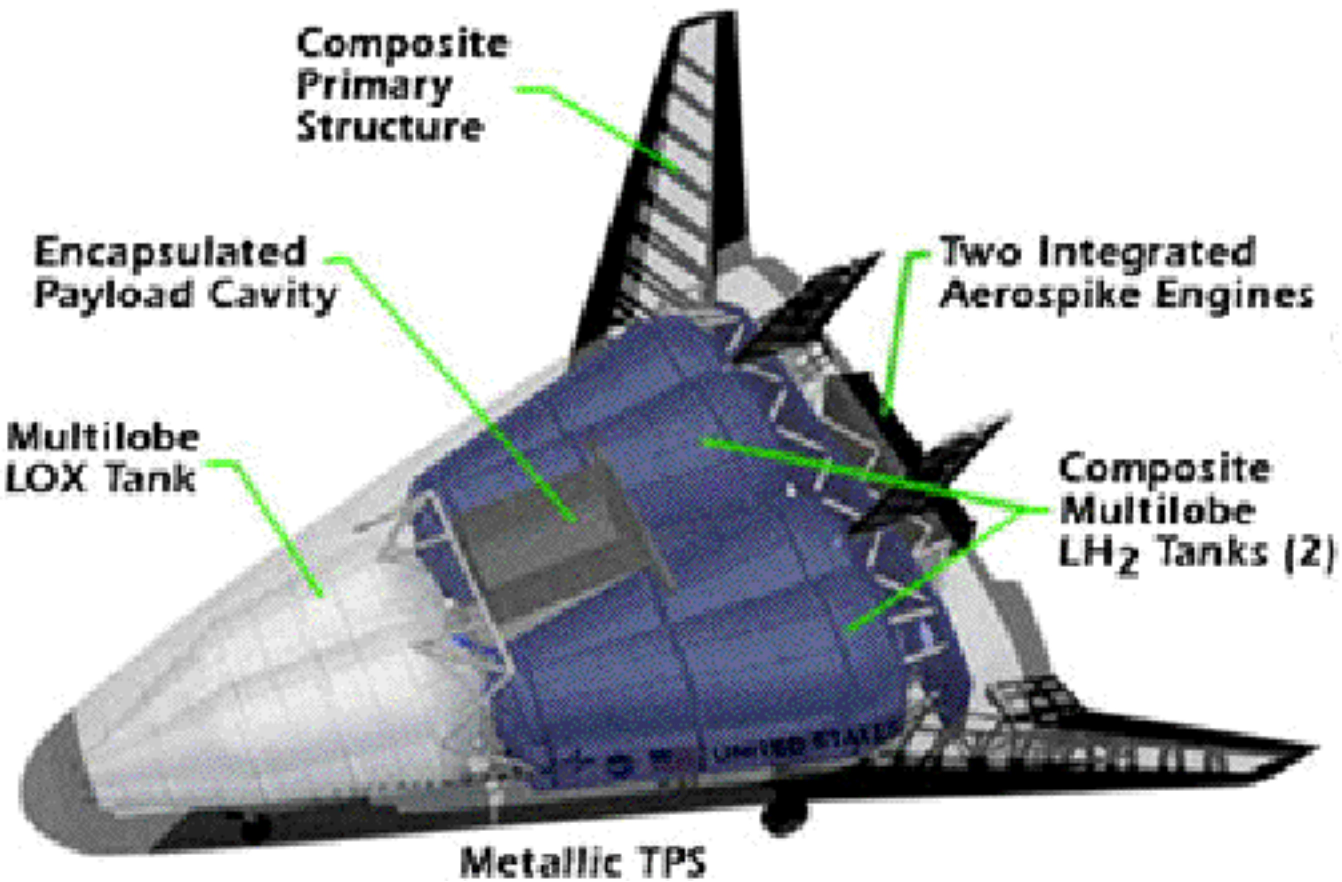
$$\Delta V_{total}^{(EAFB)} = \sqrt{\left[[7812.2 - 379.5] \frac{m}{sec}\right]^2 + \left[1750.8 \frac{m}{sec}\right]^2} =$$

$$7636.1 \frac{m}{sec}$$

ΔV required for Orbit

SSTO: A Real World Example (cont'd)

Venture-Star Fuel Capacities



SSTO: A Real World Example (cont'd)

Venture-Star Fuel Capacities

$$\left[\begin{array}{c}\text{LOX Tank Capacity: 635,000 liters} \\ \\ \text{LH}_2 \text{ Tank Capacity: } 2 \times 900{,}000 \text{ liters}\end{array}\right] \Rightarrow \boxed{\begin{array}{c}\text{TOTAL CAPACITY:} \\ \\ \text{2,435,000 LITERS}\end{array}}$$

$$\begin{array}{l}\text{LOX Mass: } 635{,}000 \text{ liters} \times 1.14 \dfrac{\text{kg}}{\text{liter}} = 723{,}900 \text{ kg} \\ \text{LH}_2 \text{ Mass: } 2 \times 900{,}000 \text{ liters} \times 0.07 \dfrac{\text{kg}}{\text{liter}} = 126{,}000 \text{ kg}\end{array} \Rightarrow \boxed{\begin{array}{c}\text{TOTAL CAPACITY:} \\ \text{849,900 kg}\end{array}}$$

Venture-Star Empty Weight

- **Original Specs were set at 100,000 kg**

  ... but by 2000 that had grown to ~135,000 kg

- **Target payload to LEO 25,000 kg, "dry weight"**

  .... Original Specs ---- 125,000 kg

  .... 2000 --- 125,000 kg

$GTO_{WT} = 974{,}900$ kg

only 3.6%

$GTO_{WT} = 1{,}009{,}900$ kg

SSTO: A Real World Example (cont'd)

- Based on original Dry mass, 100,000 kg

Circa: 1995

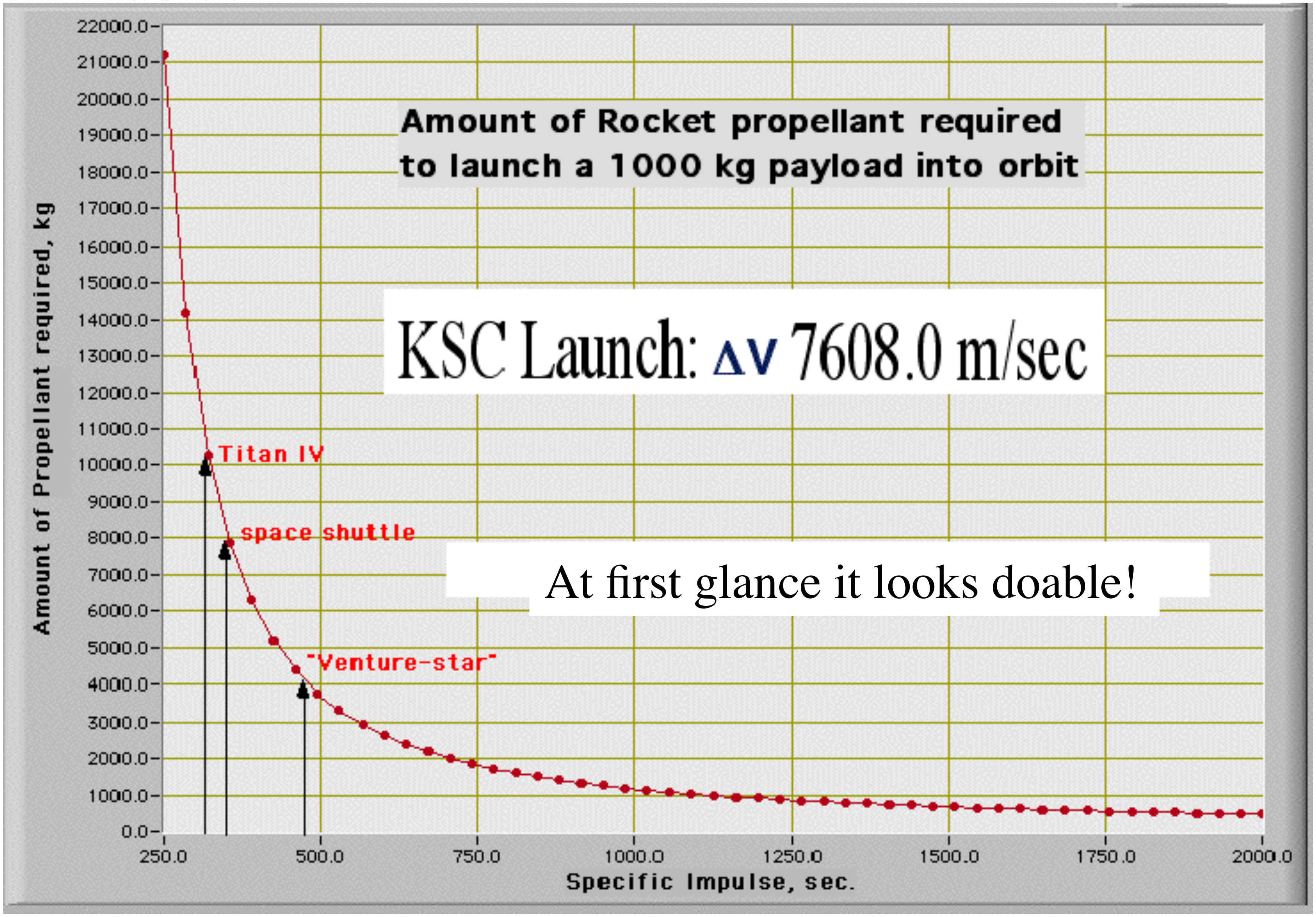
Propellant Mass Fraction:

$$\frac{849{,}900 \text{ kg}}{125{,}000 \text{ kg}} = 6.799$$

Circa: 2000

Propellant Mass Fraction:

$$\frac{849{,}900 \text{ kg}}{160{,}000 \text{ kg}} = 5.312$$



KSC Launch: ΔV 7608.0 m/sec

At first glance it looks doable!

SSTO: A Real World Example (cont'd)

Venture Star: Max ΔV Achievable:

Circa: 1995

$$\Delta V_{max} = g_0 I_{sp} \ln[1 + P_{mf}] =$$

$$9.81 \times 453.3 \ln[1+6.799] = 9133.9 \frac{m}{sec}$$

Required ΔV: $7636.1 \frac{m}{sec}$

Circa: 2000

$$\Delta V_{max} = g_0 I_{sp} \ln[1 + P_{mf}] =$$

$$9.81 \times 453.3 \ln[1+5.312] = 8183.3 \frac{m}{sec}$$

SSTO: A Real World Example (cont'd)

Venture Star/ X-33 : Postscript

• With 7% Drag loss You can't even reach orbit

When aerodynamic drag is factored in (~ 5% for SSTO trajectory) is factored in... achievable

+ optimized drag losses

$$\mathrm{Max}\left[\Delta V_{\mathrm{total}}^{(\mathrm{EAFB})}\right] \approx 8183.3 \times (1-.05) = 7772.2\ \frac{\mathrm{m}}{\mathrm{sec}}$$

Required ΔV: = 7636.1 m/sec

A Real World Example (concluded)

Venture Star/ X-33 : Postscript

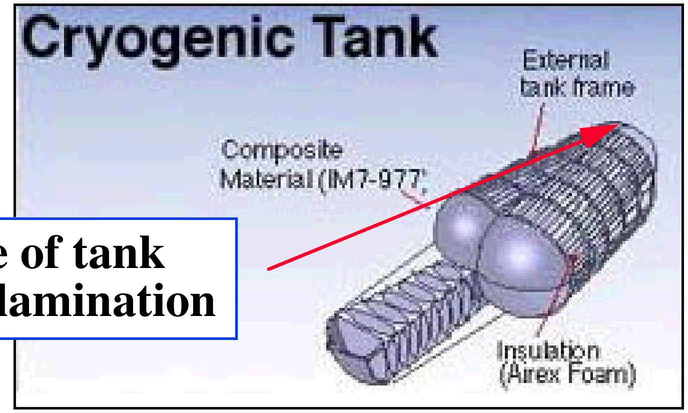
$$\mathrm{Max}\left[\Delta V_{\mathrm{total}}^{(\mathrm{EAFB})}\right] \approx 8183.3 \times (1-.05) = 7772.2 \ \frac{\mathrm{m}}{\mathrm{sec}}$$

Required ΔV: = 7636.1 m/sec

- This is a "Razor Thin margin"

- **Failure of the Lightweight weight Composite fuel tanks put them over the top in dry weight and killed the program**

X-33 Tank: What Went Wrong?



Site of tank de-lamination

- **LH$_2$ Fuel Tanks**

  Graphite/epoxy composite design intended to reduce structural weight, and withstand load of fuel and forces exerted by other X-33 structures.

- **Tank failed *after* qualification testing**

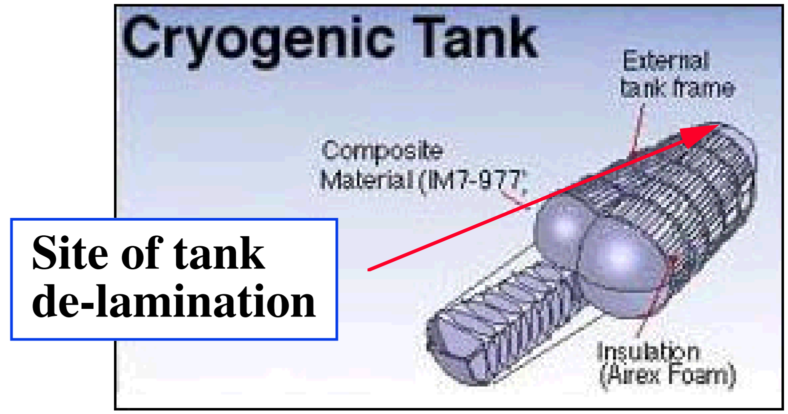
  While tank was filled with LH2 during testing air in composite structure was liquified

  Resulting vacuum in tank honeycomb cells caused external GN$_2$ purge gas to be drawn in from outside, and some gaseous H$_2$ was drawn in from inside

  After testing, when tank was purged of cryogenics, structured heated up, entrapped liquified air returned to gaseous state, and large pressures within the internal cells of the structure were created

  Unanticipated large internal pressures caused catastrophic de-lamination of the tank along the front lobe seam

X-33 Tank: What Went Wrong? (concluded)



- **So for Now ... it apears the human race will have to settle for a TSTO (Two-stage-to-Orbit) RLV at best**

Falcon 9 Commercial Launch Vehicle (http://www.spacex.com/falcon9)

Reusable two-stage to orbit vehicle