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An Overview of Advanced Concepts for Space Access (Preprint)

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A wide range of advanced launch concepts have been proposed in an effort to revolutionize space access through either a significant reduction in launch costs or significant improvements in launch performance. This paper briefly summarizes commonly proposed advanced launch concepts, including both concepts that employ propellant and propellantless concepts. Each concept is briefly described along with its potential in two generic mission classes: small satellite launch to LEO and large satellite launch to GEO. It is shown theoretically that there is significant room for improvement in the cost and performance of current launch systems. It is also shown, however, that historical predictions of launch costs reductions and/or performance improvements for new technologies have been highly optimistic with realized costs and performance leading to only incremental improvements instead of revolutionary advancements. All of the reviewed technologies still have significant technical challenges to overcome before yielding fully operational systems. The associated risk makes it difficult to justify the large investments required to develop such systems, indicating that a development path with useful products at points in between the current state-of-the-art and final goal is necessary.

Nomenclature

A	=	Area (m^2)
c	=	Speed of light, 2.99×10^8 m/s
F	=	Thrust (N)
$\dot{m}$	=	Mass flow (kg/s)
m_{prod}	=	Product mass (kg)
P_{jet}	=	Propulsive jet power (W)
P_{prop}	=	Power available in stored propellant (W)
t_b	=	Burn time (s)
v	=	Velocity (m/s)
ΔH	=	Change in enthalpy (J)
ΔV	=	Velocity Increment (m/s)
η	=	Efficiency
ρ	=	Reflectivity
$\Im$	=	Solar constant at Earth (1358 W/m^2)

I. Introduction

The means of ferrying every man-made object taken from the ground to space has been through chemical combustion of one type or another. From the days of Sputnik, launched with a combination of liquid oxygen

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and kerosene, to modern, multi-staged launch vehicles, the paradigm has been the same – combine fuel and oxidizer to extract chemical energy which is then converted to kinetic energy. Current chemical propulsion technology is very efficient, on the order of 97-98%. For example, the Space Shuttle main engines (SSME) are approximately 97% efficient in the ratio of chemical power available to jet power produced from Eq (1).[1]

$$\eta = \frac{P_{jet}}{P_{prop}} = \frac{\dot{m}v^2}{2} \frac{m_{prod} t_b}{\Delta H^0_{rxn}} \tag{1}$$

This implies that there is little room to improve the performance of such systems, and the improvements that are made generally come at great expense. However, current launch vehicles only place a small percentage of their total lift-off weight into orbit. This seeming contradiction is a clear indication that revolutionary concepts are needed for space launch. These revolutionary concepts aren't necessarily required to be in the propulsion arena. For example, materials that can dramatically reduce the inert mass fraction will also aid in a launch vehicle's ability to place more payload into orbit with existing propulsion systems.

To achieve orbit, kinetic energy is supplied to the launch vehicle via the propulsion system. Some of this kinetic energy is converted into potential energy as the vehicle ascends. However, a significant fraction remains kinetic energy as the vehicle accelerates from a velocity of essentially zero (0.41 km/sec at Kennedy Space Center for prograde orbits) to a velocity approaching the orbital velocity required for the payload to remain in orbit. Of the approximately 9.5 km/sec of total ΔV required to attain a low-Earth orbit, only about 1 km/sec is involved in competing against the Earth's gravitational potential. About 0.5 to 0.75 km/sec can be attributed to losses such as atmospheric drag, vehicle steering, and back-pressure. The remaining ΔV is used to increase the velocity of the payload. From this perspective, advanced concepts that attempt to address reducing the amount of ΔV required by only reducing the potential energy change or reducing drag probably tend to complicate systems more than they enable them.[2]

Using a ΔV of 9.5 km/sec as an example, the energy required to reach a low-Earth orbit is approximately 45 MJ/kg or 12.5 kW-hr/kg. This equates to $1.75/kg at current peak-hour electricity rates and about $0.48/kg at off-hour rates. The energy requirement roughly doubles for placing payload mass into geosynchronous orbit. Current rates for access to space range from several thousand to well over 10,000 US dollars per kilogram.[3] Clearly there is also room for improvement in the area of cost for current chemical launch vehicles.

For decades, advanced propulsion concepts have sought alternative means to more easily and cost effectively access space; however, all of these concepts have yet to be truly realized. In an attempt to understand whether chemical systems can ever be replaced, an extensive array of advanced concepts has been investigated. The objective of this study was to assess the technology's potential of providing a cost-effective means of placing objects in orbit around the Earth within the next 15 to 50 year timeframe. Two Air Force relevant missions have been used in this study to assess potential launch concepts. The first mission involves placing a large communications satellite in geosynchronous Earth orbit (GEO) which is complicated by the requirement of a relatively large ΔV. For the GEO mission, an analysis based on the availability of a notional space tug, a LEO to GEO transfer vehicle, has been performed to assess the usefulness of the space tug concept from the perspective of launch vehicle design. The second mission involves placing a micro-satellite, with a mass of about 100 kg, in a low-Earth orbit (LEO) which is complicated by the requirements of low-cost and rapid response.

The study focused on two main categories of propulsion systems: those which require propellant and propellant-less systems. Many of these systems have also been analyzed for their potential in a launch assist role. For the purposes of this study, launch assist is defined as a family of technologies that can provide some fraction of the required orbital potential and/or kinetic energy using non-rocket based techniques in an attempt to greatly reduce launch costs.

Advanced concepts for launch or launch assist will need to have clear advantages over chemical systems to be of value. New launch concepts will have to improve performance, efficiency, cost, or the ability to rapidly respond to changing global situations. It is not clear if an improvement to any one of these areas is sufficient to justify a complete change in current launch infrastructure. However, it is clear that systems that improve several of these key areas will be viewed more favorably.

II. State of the Art Launch Technology

As stated above, advanced space launch systems must provide significant improvements in system performance or significant reductions in launch costs to be viable. It is, therefore, worthwhile to begin by providing a brief overview of the pertinent characteristics of state-of-the-art launch systems and describing the most important performance metrics for launch vehicle comparison. General state-of-the-art launch vehicle characteristics will be described along with specific characteristics for two representative launch systems, the Delta IV Heavy and the Minotaur IV. The Delta IV Heavy was chosen to represent the current state of the art in launch vehicles capable of launching large payloads to GEO. The Minotaur IV was chosen as a representative of the launch vehicle class capable of launching small payloads to LEO.

A. Delta IV Heavy: Large Satellite to GEO

The Delta IV Heavy is largest launch vehicle in the Delta IV family designed by Boeing. The Delta IV family of launch vehicles is shown in Fig. 1. The Delta IV was developed as part of the Evolved Expendable Launch Vehicle (EELV) program and had its first flight on November 20, 2002. The goal of the program was to reduce the launch costs by 25% while increasing the reliability by simplifying the design, manufacturing processes, and integration[4]. For example the RS-68 motor has 80% fewer parts than its predecessor.[5] It was hoped that the reduced cost and increased reliability would yield additional commercial customers, but in general, this didn't happen which greatly reduced the expected cost savings.

The Delta IV Heavy is a two stage launch system.[6] It uses three common booster cores that use LH_2/LO_2 for propellant and a Rocketdyne RS-68 engine for the first stage. Each RS-68 is capable of providing a sea level thrust of 2.891 MN (8.673MN total) at 410s Isp. The upper stage of the Delta IV Heavy is also a LH_2/LO_2 system with a RL-10B-2 motor. The upper stage provides a thrust at altitude of 110kN at a specific impulse of 462s. The Delta IV heavy is capable of placing 22,950 kg into LEO. The cost per unit mass of the Delta IV depends on the specifics of the launch, but is still around $10,000/kg.

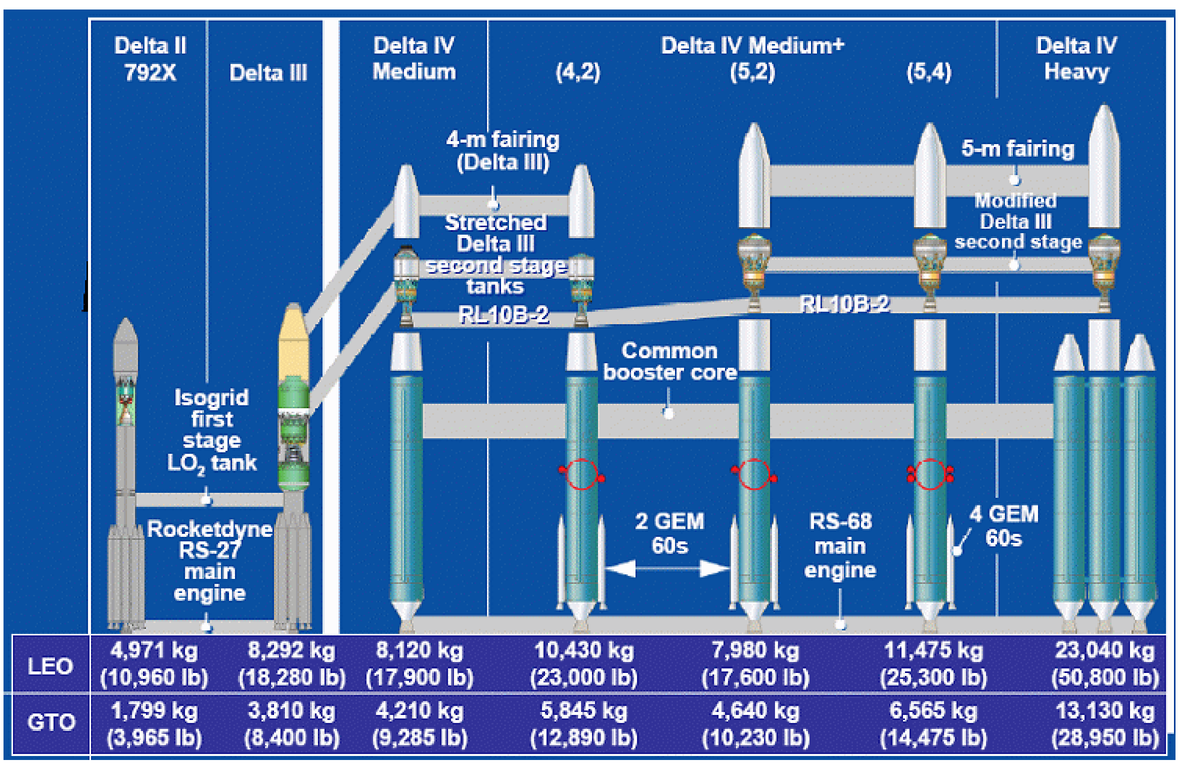

Figure 1. Delta IV Family of Launch Vehicles

B. Minotaur IV: Small Satellite to LEO

The Minotaur IV launch vehicle was developed by Orbital Sciences Corporation for the US Air Force as a cost effective means of launching small payloads into LEO.[7] The Minotaur IV reuses surplus Peacekeeper boosters as a means of achieving low launch costs. The Minotaur IV is a four stage all solid propellant rocket. The first three stages are government furnished Peacekeeper stages. The stages provide 2224kN, 1223kN, and 289kN of thrust. The fourth stage uses an Orion 38 design and provides orbit insertion.

The Minotaur IV can deliver 1750kg to LEO. The first scheduled launch of the Minotaur IV launch vehicle is December 2008. Its predecessor, the Minotaur I, had launch costs were around $30,000/kg which is still below launch systems with similar capabilities (Pegasus and Athena) because it reused the solid boosters. The Minotaur family of rockets have not yet experienced any failures and have demonstrated some level of responsive capability with a responsive launch solutions of under 6 months advertised.[8]

C. Launch Costs

One common focus of launch system research is an attempt to significantly reduce the launch cost per unit kilogram of payload mass. If a satellite launch system is viewed as simply a method of transferring kinetic and potential energy to a payload then the absolute limit to the minimum cost of the launch system is simply the market cost of the added energy. As described in the introduction the cost of existing launch systems is approximately 10,000x higher than the direct cost of the energy added to the payload indicating, theoretically at least, that there is room for significantly reducing the launch costs. A significant reduction of launch costs could lead to a correspondingly significant reduction in satellite costs by enabling cheaper, less reliable components to be used in constructing the satellite. This would, in turn, lead to a reduction in the total cost of a spacecraft which would be enabling for missions such as large platoons of satellites, space based solar power systems, and a continual presence at lower altitudes.

Figure 2 shows the distribution of current launch costs for common US launch systems (as of 2007) as a function of mass for the case of launching a satellite into a 185 km (100nmi) circular orbit with an inclination of 28.5 degrees. In general the cost per unit mass decreases as the delivered payload mass increases, but achieved launch system costs (at current launch rates) are between $9,000/kg and $50,000/kg ($4,000/lb and $23,000/lb). Total launch costs can be divided into two categories: recurring and nonrecurring. In order to significantly reduce launch costs both categories must be addressed. Nonrecurring launch costs such as research and development costs will not be discussed in this paper for brevity, but they must not be neglected in selecting between alternative launch vehicles for development. Efforts are being made to reduce the recurring launch costs in all directions including towards more complicated, higher performance launch systems and towards simpler, lower performance launch systems.

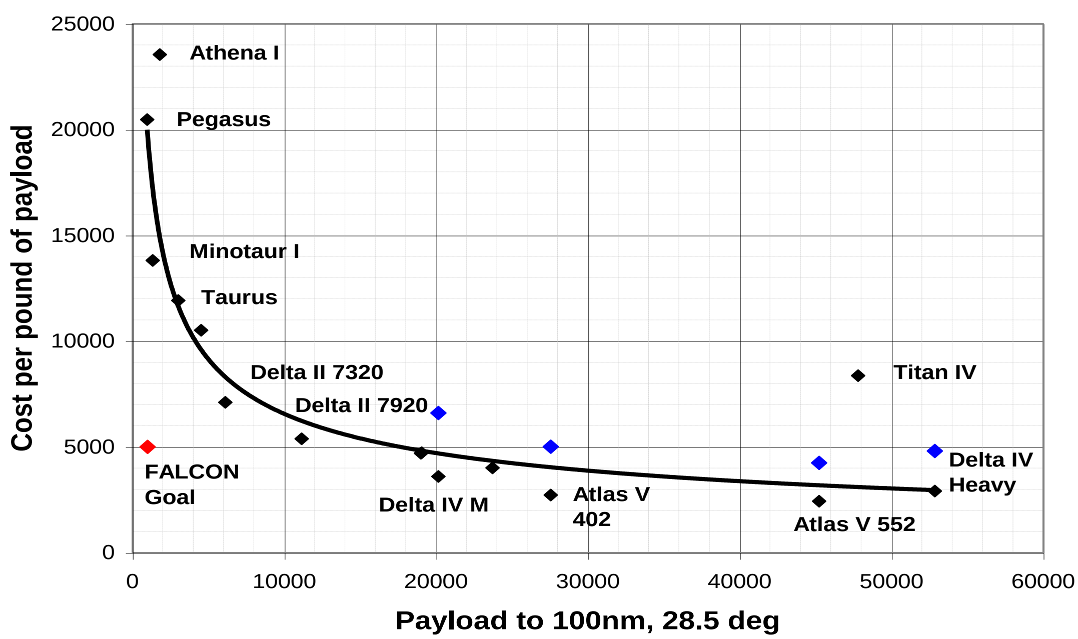

Figure 2. Launch Costs of Common Space Launch Systems

Historically, a number of efforts have attempted to reduce the cost per unit mass of space launch systems while maintaining their reliability. To date, these efforts have not yielded revolutionary reductions in launch costs. The

Evolved Expendable Launch Vehicle (EELV) program, for example, is an effort designed to maintain US launch capability, but at reduced cost and increased reliability. Two families of rockets (the Atlas V and the Delta IV) were developed and are in operation. The primary development philosophy was to evolve existing launch systems towards simpler designs and to use standardized components to reduce the launch costs. Both launch systems are operational, but they have not achieved the projected cost savings due to the lack of commercial interest.[9] On the other end of the spectrum is the Space Shuttle. The Space Shuttle is potentially the most complicated machine every created and was designed to be the first partially reusable launch system in an effort to greatly reduce the launch costs and increase the flight rate. In the case of the shuttle, the design philosophy of partial reusability (with increased complexity) has not enabled achievement of the original goals.[10] The initially projected flight rate was 60 launches per year (40 from KSC and 20 from Vandenberg) with a 2 week turnaround time. The achieved flight rate has been approximately one order of magnitude less. The initial estimated launch costs for the Space Shuttle were \$200/kg, but the achieved costs are closer to \$20,000/kg. It is apparent, therefore, that although significant cost reductions are possible, historically significant reductions haven't been achieved by either significantly more complicated systems or significantly less complicated systems.

D. Reliability

Launch reliability is another performance metric useful in evaluating and comparing launch vehicles. Reliability is defined as the likelihood that the launch vehicle will perform as expected and deliver the payload into the required final orbit. Allen demonstrated that between 1964 and 2000 the reliability for US launch vehicles was consistently between 0.91 and 0.95.[11] In general, reliability depends on the total parts count, the reliability of individual parts, and the redundancy built into the design through the arrangement of the parts. Significantly increasing the reliability of a launch vehicle is a daunting task which would likely require significant improvement in all three areas. It was also shown that roughly 2/3 of the failures are due to propulsion elements, while the other third is due to non-propulsion elements. In general, upper stages fail more often than lower stages for both solid and liquid systems, and guidance is the most common nonpropulsion failure. Solid propellant components typically have slightly better reliability, and monolithic solid boosters have higher reliability than segmented motors, which introduce additional failure modes. In general, simpler, lower performing systems appear to have increased reliability. It should also be noted that proposed reusable systems, which require high flight rates to be competitive, require increased reliability while using more complicated, higher performing systems.

E. Other Considerations

There are many other potential system metrics that can be considered when evaluating launch vehicles. Only a few will be mentioned in this section. Increasing the payload mass fraction is often listed as a motivation for developing new advanced launch systems. Payload mass fractions are commonly below 1% (with about 14% structural mass fraction and about 85% propellant mass fraction).[12] Large improvements in payload mass fraction are theoretically possible, but if they do not result in a corresponding reduction in launch cost per unit mass or increased reliability or responsiveness then the net effect would not be as appealing. The responsiveness or delay time it takes to get a payload launched is another important performance metric. Responsiveness is currently measured in months, but there is hope that it can be reduced to hours.[13,14] It is also worth keeping in mind some of the extreme physical parameters that are associated with current space launch systems. Additional parameters such as the thrust per unit weight and peak power flux must also be considered. The space shuttle main engine (SSME), for example, has a throat area of 600 cm^2 and produces > 6GW of power yielding an energy flux of greater than 10MW/cm^2.

IV. Launch Concepts Employing Propellant

A. Nuclear

The concept of nuclear fission rockets was first proposed in the late 1940's. Since then, the development of launch vehicles based on NERVA, Particle Bed and CERMET reactors have been studied.[15] Fission represents a specific energy density of approximately 7 x 10^{13} J/kg at 100% efficiency, which is nearly 7 orders of magnitude greater than chemical reactions can provide. Concepts range from 1000 sec to upwards of 5000 sec of specific impulse. Most of these concepts utilize a working gas, typically hydrogen, as propellant which is heated by the fission reactor. Although the specific mass of these systems remains relatively high, the increased specific impulse realized more than makes up for the increase in inert mass fraction. Although nuclear fission has never been used to launch a payload into orbit, there remain only a few technical issues in doing so. The main hindrance to developing an actual fission-based launch vehicle remains a socio-political one.

Nuclear fission-based space tugs, a LEO to GEO transfer vehicle, have also been investigated.[16,17] Although the space tug concept is not necessarily related to launch vehicles, it would have a direct impact on the size and capability of a launch system. For example, a launch vehicle may only need to be capable of placing a large payload in LEO in essence delivering it to the space tug system already in orbit. This requires a much lower ΔV for the launch vehicle than placing the payload into GEO directly. The benefits of a nuclear space tug is shown in Table 1 as a function of the space tug's inert mass fraction using either nuclear thermal propulsion (NTP) or nuclear electric propulsion (NEP). The values in Table 1 assume a total space tug mass (without payload) of 22,000 kg, the payload mass that a typical Delta-IV heavy can place into LEO. The breakeven scenario involves the mass to GEO of two Delta-IV heavy launches. For the tug scenario, the first Delta-IV heavy launch gets the nuclear space tug (22,000 kg) into LEO. The subsequent Delta-IV gets the desired GEO payload to the LEO tug for transfer. If low inert mass fractions are possible through advanced materials and engineering, then relatively large payloads can be delivered to GEO as shown in Fig. 3 as a function of the space tug specific impulse. The analysis presented in Fig. 3 assumes a total ΔV = 4.178 km/sec for a transfer from LEO (185 km altitude, i=28.7°) to GEO. The usefulness of a nuclear space tug relies on the inert mass fraction, F_{inert}, that is achievable by the system. Current estimates suggest that the specific mass of a fission propulsion system lies in the tens of kg/kW. A significant investment in future research needs to be devoted to specific mass reductions in fission systems for both space tug and direct launch applications.

Nuclear fusion is important for advanced propulsion concepts from the standpoint that it promises an order of magnitude greater specific energy density than nuclear fission, again at 100% efficiencies. Kammash and Lee[18] described a system wherein a high-density plasma is confined and heated to thermonuclear temperatures. Typically, electrical power is the limiting factor in many designs for launch vehicles that do not utilize chemical reactions. However, this issue is overcome by the high power density of nuclear fusion reactors in this particular design. Typical fusion systems are most likely going to be massive and complicated devices. Fusion reactors may also have excellent specific mass specifications ranging from 0.5 to 0.05 kg/kW depending on the fuel utilized.[18] Future research and development is obviously needed in this area to reach reasonable efficiencies (i.e. gains greater than unity). A major benefit to this development for space applications is the large terrestrial application potential of fusion reactors. Space efforts can leverage the already robust research in this area as the technology matures.

System	Inert Mass Fraction of GTO Stage	Specific Impulse (s)	Mass to GEO (kg)	Notes
Delta IV Heavy , RL-10 upper stage (x2)		425 (RL-10)	12,552	Assumes ΔV = 4.178 km/sec for transfer from 185 km altitude parking orbit at i=28.7° to GEO orbit.
Thermal Nuclear Tug	0.7	1000 (NTP)	< 0	Finert=0.7 from Ref. 1, Hydrogen propellant
Thermal Nuclear Tug	0.455	1000 (NTP)	12,552	Breakeven with 2 Delta IV Heavy launches, Hydrogen propellant
Electric Nuclear Tug	0.699	2000 (NEP)	12,552	Breakeven with 2 Delta IV Heavy launches
Electric Nuclear Tug	0.754	2500 (NEP)	12,552	Breakeven with 2 Delta IV Heavy launches
Thermal Nuclear Tug	0.3	1000 (NTP)	22,393	Possible Finert, Hydrogen propellant
Electric Nuclear Tug	0.45	2500 (NEP)	55,226	Possible Finert

Table 1. Trade space for a nuclear space tug.

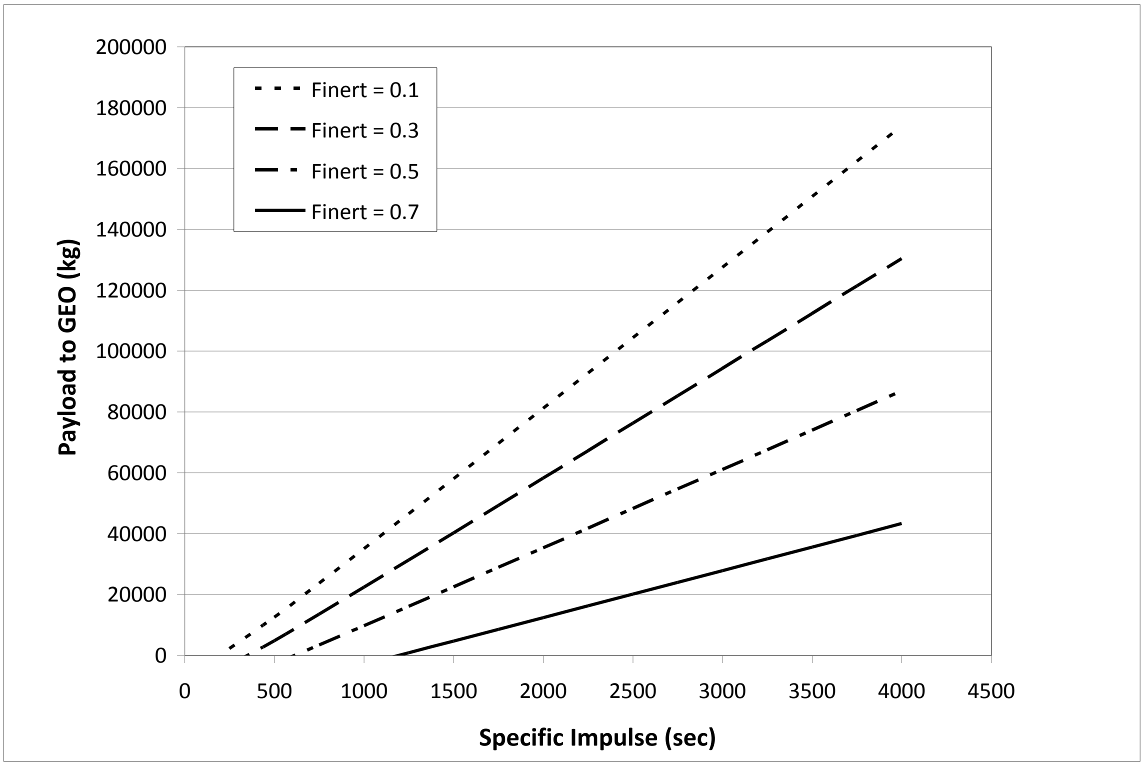

Figure 3. Payload mass versus specific impulse for a notional nuclear space tug as a function of inert mass fraction (Finert).

B. Beamed Energy

For the purposes of this study, beamed energy propulsion is defined as energy addition to a propellant through either laser or microwave energy directed from the ground or space to a launch vehicle. The overriding advantage to either microwave or laser propulsion concepts is that the power source, and thus a large mass, is not integrated on the launch vehicle, remaining on the ground. This can provide some level of mass savings over conventional chemical systems, although the magnitude of the mass savings is obviously concept dependent. The second advantage is the possibility for higher specific impulse since the limitation on chemical energy production is removed. However, there are some disadvantages for propulsion applications that need to be addressed. First, laser and microwave sources with enough energy for launch applications do not currently exist, although development continues in both of these areas for non-space applications. The development of these systems for launch can certainly be leveraged from programs such as the Air-Borne Laser (ABL) and fusion research; however, specific development for propulsive applications will still be costly. Another disadvantage is that the electromagnetic beams must propagate through the atmosphere where they are attenuated by atmospheric species. Although atmospheric windows exist where the transmission of certain wavelengths is highly efficient, beam attenuation through kilometers of atmosphere is always a concern. Beam scattering from atmospheric constituents may also pose problems for certain wavelengths. In addition, highly accurate pointing, tracking, and beam focusing will be required by the ground-based beamed power system throughout the launch profile to orbit. For example, the turning of the vehicle for final orbit insertion may be problematic for a fixed, ground-based beamed power facility.

A beamed microwave rocket concept was first introduced by Shad and Moriarty[19] in 1965. Laser propulsion was first introduced by Kantrowitz[20] in 1972 although the concept of photon propulsion was first put forth in 1955.[21] Since these pioneering papers, an extensive amount of research has been performed which has led to a wide variety of beamed energy propulsion concepts.[22] In the past several decades, advances in laser and microwave power generation have been significant, potentially warranting another look at these systems for a launch vehicle configuration.

1. Laser Propulsion Concepts

Laser propulsion concepts can be broken down into four main categories based on the thrust mechanism employed: heat exchange, plasma formation (gas breakdown), laser ablation, and photon pressure. From the standpoint of photon pressure, the laser beam is used to provide photons which "push" on the vehicle. In general the momentum transfer possible from photons to a surface is extremely small precluding it from a launch vehicle application. However, recent work by Bae[23,24] has brought the possibility of launching small payloads to orbit using photon pressure. Equation (2) gives the force produced by a flux, $\Im$, of photons as it interacts normal to a surface.

$$F = \frac{\Im}{c}(1+\rho)A \tag{2}$$

As can be seen in Eq. (2), several factors can act to increase the force due to photon pressure including highly reflective surfaces and a large photon flux. Concepts have been developed that attempt to increase the number of photon reflections from a surface through the use of optical cavities (both resonant and non-resonant). The concept by Bae improves the efficiency by increasing the photon flux by additionally amplifying the laser pulse in an intervening optical cavity. In the Photonic Laser Thruster (PLT) concept, a thrust amplification to Eq. (2) of up to 3,000 times has been demonstrated[24] by forming an optical cavity between two planar surfaces. From initial results, thrust-to-power ratios (F/P) approaching 20 μN/W have been demonstrated using a pulsed Nd:YAG laser. Assuming no losses, approximately 5 MW of laser power would be required for a thrust-to-weight (F/W) ratio of one for a 10 kg payload. Although unlikely to provide enough thrust at reasonable power levels for launch with current laser systems, the trend in high power laser development could make this a viable system for launching small payloads in the future.

In an effort to increase the thrust produced by a laser interacting with a material, laser ablation thrusters have been developed. Laser ablation involves the removal and subsequent acceleration of atoms or molecules from a solid surface through laser irradiation. Although mostly for spacecraft propulsion applications, some concepts have extended into the launch vehicle domain.[25,26] In two notional designs by Phipps, et al. the payload mass fraction to low Earth orbit ranged between 4% and 27.5% depending on the size of the vehicle and the number and configuration of laser launch stations utilized. These payload mass fractions show tremendous promise in the range of total launch vehicle mass from 10 to 20 kg. Scaling this technology to higher liftoff masses will require large amounts of laser power and presumably equally large-scale development programs. In general, laser ablation propulsion is capable of providing much higher thrust levels than the PLT concept; however, this benefit comes at the price of carrying propellant (laser ablatant) on the vehicle.

In order to achieve a relatively high thrust laser propulsion system, high power lasers producing high temperature gas flows are necessary. Plasma formation in a nozzle can create temperatures as high as 10^4 - 10^5 K. However, sustaining a plasma in a high mass flow environment requires power levels of 100 to 1000 MW for a typical launch system. Pulsed laser systems have been proposed to ionize the propellant inside a nozzle increasing the thrust generated by creating a high temperature plasma jet.[27] The power density required to ionize a typical working gas is in the range of 5 x10^{14} and 10^{15} W/m^2 once again emphasizing the need for high power laser systems.[28] These concepts generally suffer from the requirement for highly accurate focusing optics on the launch vehicle.

The "laser lightcraft" concept demonstrated in ref. 29 is envisioned to be a multi-stage system with the first stage driven by an air-breathing aerospike, utilizing a beamed, ground-based laser to form air detonations that propel the vehicle. Two types of lightcraft engines have been examined using either simple laser-thermal or more complex magnetohydrodynamic (MHD) concepts. In either configuration, the main idea is to focus the laser beam within the lightcraft geometry to breakdown the ambient air, thus forming an air plasma. The second stage would use the same ground-based laser in combination with a working fluid stored on the vehicle. The second stage would operate when the atmospheric density decreased below a critical value. The breakdown of air for sustaining pulsed detonation waves to propel the vehicle has the advantage of not having to store fuel on-board the vehicle. However as the vehicle ascends, the air density decreases to the point where stored propellant is necessary. Concepts which used stored liquid propellants and solid ablative material were investigated. The initial lightcraft design was a reconnaissance or telecommunications vehicle weighing 100 kg and envisioned to be boosted by a 100 MW-class, ground-based laser.

The final category of laser propelled vehicles involves beaming a laser to a heat exchanger located on the launch vehicle. Through heat exchange with a working propellant, also stored on-board, a kind of laser-heated resistojet is envisioned. Kare[30] describes a vehicle which uses a lightweight, flat plate heat exchanger to couple laser energy to a hydrogen propellant. The hot gas produced in the heat exchanger is then expelled through a rather conventional nozzle. Although heat exchangers are traditionally inefficient, microchannels are used to maintain laminar flow and provide a large surface area. By keeping the operating temperature at approximately 1000 °C, re-radiation losses are minimized and the heat exchanger can be made of simple materials. The heat exchanger efficiency is also relatively independent of laser wavelength so that effectively any laser can be used. To minimize the cost of high power laser development, a module laser approach has been investigated. A non-coherent array of laser diodes could potentially be used instead of a single high power (~100 MW) coherent laser. The array of diode lasers has the advantage of being relatively compact, efficient, and scalable. However, the incoherent nature of the laser sources requires a much larger aperture, possibly on the order of several thousand square meters. In the study by Kare[26], a notional vehicle could place from 50 to 200 kg into low Earth orbit using the heat exchanger method. However, the total payload mass fraction is only slightly greater than 2%.

Although laser propulsion concepts have the potential to place a large payload mass fraction into low-Earth orbit, available laser power in the foreseeable future will limit the total payload mass to 10's of kilograms. These systems are also liable to be complex and expensive and ultimately may not drive down the cost of space access. They can be envisioned as fulfilling a rapid response role once a laser-based launch facility is constructed; however, weather and safety issues may be significant.

2. Microwave Propulsion Concepts

Microwave propulsion concepts can be put into two categories that include heat exchanger or propellant heating and plasma formation options. Since the 1930's, microwave source development has seen exponential increases in Pf^2 (power x frequency^2).[31] Current estimates suggest that an array of 300 gyrotron sources operating at 140 GHz and 1MW power levels is sufficient to place a 1000 kg satellite into orbit.[32] Gyrotrons appear to be one of the most versatile vacuum electronic devices capable of producing high average power in the 30-300 GHz range. The maximum average power range for gyrotrons is approximately 2 MW; however, some types of gyrotrons can produce upwards of 30 MW peak power. Conversion efficiencies of approximately 50% are expected from current gyrotron sources.[33] The microwave frequency being used for launch vehicle concepts depends on several factors such as atmospheric propagation, air breakdown, coupling efficiency, and the overall size of the ground-based microwave station. For example, a relatively low microwave frequency can result in transmitter diameters that are several hundred meters assuming a transmission length of 100 km. Higher frequencies will act to reduce the required transmission diameter; however, atmospheric attenuation and plasma formation (breakdown) need to be considered.

Oda, et al.[33,34] has described a system that uses a gas discharge to produce thrust. The gas discharge is formed near the focal point of a high power, pulsed microwave beam. The system uses the ambient air as propellant. The beam is delivered from a ground-based microwave system at 170 GHz. The induced plasma absorbs the remaining microwave pulse and expands through their device rapidly, causing a shock wave driven impulse. The breakdown intensity for a gas such as air has been shown to be frequency dependent[35] with larger frequencies requiring a higher breakdown intensity. Therefore, air breakdown schemes would benefit from lower frequency; however breakdown in the atmosphere before reaching the launch vehicle may be an issue. Atmospheric windows exist between 1-40 GHz, 130-160 GHz, and 200-300 GHz. A optimization of frequency is required for these systems which yields a plasma at the launch vehicle through focusing of the microwave beam, but does not lead to breakdown in the intervening atmosphere. A similar concept is also described by Nakagawa, et al.[36] using 110 GHz frequency with an output power less than 1 MW. In this study, the maximum coupling coefficient was found to be 395 N/MW indicating that upwards of 1000 MW of output power would be necessary to produce the thrust required for a launch vehicle. Obviously, improvements to the coupling coefficient are envisioned for this concept.

Parkin[32,37] describes a microwave thermal thruster that incorporates a heat exchanger to absorb beamed microwave energy from the ground in a similar fashion to that described by Kare[30] for laser beams. This concept uses hydrogen propellant heated in a heat exchanger to a temperature of 2800 K yielding a specific impulse of over 1000 sec. The heat exchanger was designed to operate at power levels exceeding 1 GW. A limiting factor in the performance of this device is the maximum operating temperature which is limited by the materials used in the heat exchanger. In a notional design, Parkin and Culick[32] suggest a payload mass fraction approaching 5-15% after system optimization. Since air or other propellant breakdown is not desired in this configuration, higher microwave frequencies (140 GHz) are used to additionally avoid atmospheric breakdown within an atmospheric transmission window.

3. A Brief Comparison of Beamed Propulsion Concepts

The benefit of microwave systems is a wider range of operating frequencies and a lower cost per unit power than laser-based systems. Both systems can leverage extensive industrial and military development. Both systems can also use an array of sources instead of a single larger source to perform their intended missions which should lead to reduced cost overall. Transmission through the atmosphere plagues both system as cloud cover, rain, and launch attempts for low altitude (i.e. sea level) can cause high attenuation levels. The efficiency of both laser and microwave transmitters is expected to be in the 35% to 60% range with laser system efficiency generally higher.[38] Using the notional laser and microwave heat exchanger concepts, Kare and Parkin[38] estimated the cost of a beam source capable of launching 100 kg. Their conclusion was that both systems would cost in excess of $2 billion USD with the microwave system costing slightly less (but probably within the error estimates of the comparison). Even though the microwave system exhibits about 30% less cost per Watt of power generated, it will require almost 2.5 times the power to be generated at the source. The increased power generation presumably comes from the coupling coefficients with the heat exchanger where laser energy in visible wavelengths is expected to couple with higher efficiency. The coupling between beamed laser energy and microwave energy for air plasma concepts appears to be comparable.[36]

4. Thrust Augmentation Using Beamed Power

Microwave and laser beamed energy could also be used to augment the thrust and specific impulse of more traditional launch vehicles. Beamed energy into the nozzle of a typical liquid hydrogen/liquid oxygen system could be used to further heat the expanding gas to temperatures beyond what chemistry can provide. Coupling to the water vapor in the nozzle (gaseous or liquid droplets) could either be done directly to the neutral gas or could be done through plasma formation in the nozzle. Coupling to the expanding gas in the supersonic region of the nozzle may be difficult, and energy addition on the diverging side of the nozzle throat will be inefficient from a thermal energy conversion standpoint. However, energy addition to the diverging section of the nozzle may provide the most cost effective means of energy addition. Increasing the temperature in the combustion region (subsonic) of the thruster could lead to higher mass and cost since the use of expensive materials to survive the increased heat flow could be necessary. A detailed study looking at the cost per Newton of thrust increase or the cost per second of Isp increase needs to be performed to investigate the usefulness of thrust augmentation. Concepts have also been envisioned where thrust augmentation can be provided, in effect, through beamed power to operate magnetohydrodynamic (MHD) devices on a launch vehicle.[39]

C. Advanced Chemical (HEDM)

The performance of a chemical rocket is heavily influenced by the properties of the propellant used. Specific impulse is proportional to the square root of the chamber temperature divided by the mean molecular weight of the exhaust species. To maximize specific impulse, the propellant combination must release a large amount of energy to obtain a high chamber temperature, and must have minimal molecular weight. The combination of oxygen and hydrogen has proven to be among the most effective, with engines such as the SSME achieving a specific impulse in excess of 450 sec.

However, the maximum delta-V that a rocket stage can achieve is proportional to specific impulse and the natural logarithm of the vehicle mass ratio (the ratio of vehicle initial mass to final mass). So although a low molecular weight propellant helps increase specific impulse, such propellants also are low in density and thus require large propellant tanks, reducing the vehicle mass ratio.

Historically, launch vehicle designers have had to choose between low density but high specific impulse propellant combinations such as oxygen / hydrogen, and higher density but lower specific impulse propellant combinations such as oxygen / kerosene. Chemists have sought after propellants that might provide both high specific impulse and high density for decades. Clark[40] has chronicled propellant development efforts in the United States.

As described above, specific impulse is proportional to the square root of chamber temperature divided by the molecular weight of the exhaust, so if the molecular weight of the exhaust increases due to an increase in propellant density, chamber temperature must increase accordingly to maintain a given specific impulse. The relationship between specific impulse, temperature, and exhaust species molecular weight is shown in Fig. 4.

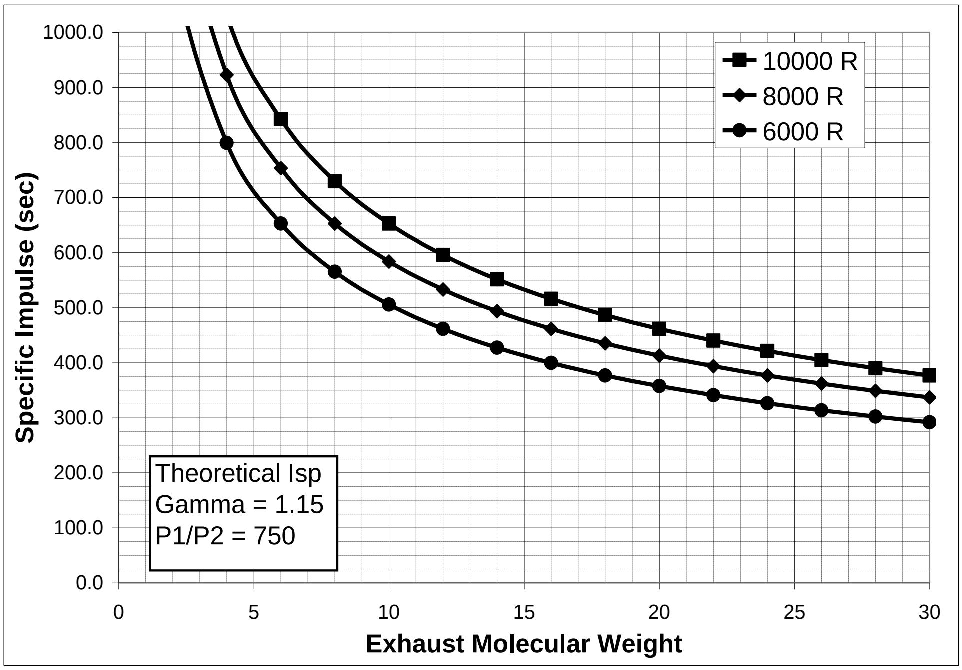

Figure 4. Specific Impulse as a Function of Temperature and Exhaust Molecular Weight

Currently, materials considerations limit chamber temperature to about 7000ºR if the chamber is actively cooled. Any new propellants with higher energy densities would likely require higher combustion temperatures than current materials allow.

For a chemical rocket, the energy that raises the working fluid up to the chamber temperature comes from the chemical bonds in the propellant. High energy density propellants are usually novel chemicals with many high energy chemical bonds to supply the necessary energy during the reaction. One class of high energy density fuels that has been studied are the strained ring hydrocarbons.[41,42,43] These hydrocarbon fuels are isomers of existing hydrocarbons with novel arrangements of the atoms and increased bond energies as a result.

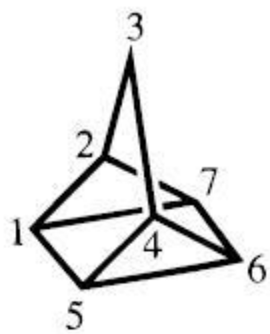

Figure 5. Structure of Quadricyclane molecule C_7H_8, a strained ring hydrocarbon

Another class of propellants that has been considered for high energy density applications is polynitrogen.[44,45,46,47] The decomposition reaction of theoretical compounds such as N_4 or N_8 into N_2 molecules releases a great deal of energy. So far, research that has produced compounds containing the N_5^+ ion but has not yet produced a polynitrogen compound that would be a useful propellant.

A conceptual study by Cole, et al.[48] looks at metallic hydrogen as a potential propellant. Above 4.5 Mbar, solid molecular hydrogen is hypothesized to become an atomic solid with metallic properties. Recombination of the hydrogen atoms could then release 216 MJ/kg of specific energy, far exceeding the approximately 10 MJ/kg from the SSME. Initial calculations indicate that specific impulses as high as 1700 sec could be achieved if an adequate chamber material can be found to withstand the high temperatures. The engine concept developed in their study suggests that a diluted mixture of metallic hydrogen with cryogenic liquid hydrogen could potentially produce a specific impulse in the range of 900-1100 sec. In this concept the liquid hydrogen is used as a coolant. Unfortunately, metallic hydrogen has yet to be produced on Earth.

These efforts have highlighted challenges that are common to all high energy density propellant research. One key challenge is that molecules which can release large amounts of energy also tend to be less stable than desired. Another challenge is that synthesis of these molecules often requires many steps, which can dramatically increase the cost of the propellant. While propellants such as hydrogen, RP-1 (kerosene) or methane are readily made in large quantities with existing processes and infrastructure, there would not be many applications for high energy density materials except for use in rockets and explosives, which puts these materials at an additional economic disadvantage. Toxicity and material compatibility have not yet been evaluated for many high energy density materials, which would be required before they could be adopted.

D. Hypersonic Air Breathing Vehicles

Launch vehicles employing atmospheric oxygen as the oxidizer can potentially yield a revolutionary increase in payload mass fraction by exchanging the oxidizer mass that a traditional rocket carries for useful payload mass (at least partially). NASA's Space Shuttle, for example, has a gross lift-off weight of approximately $2x10^6$ kg. Approximately $6.2x10^5$ kg (30%) of that is LOX oxidizer while only 24,400kg (1.2%) is considered useful payload.[49] Significant potential, therefore, exists for air breathing launch systems if the potential improvement in payload mass fraction is not counteracted by other factors such as increases in engine and structural mass, increases in gravitational losses, and increases in drag losses. The payload mass fraction may not be the only (or even most important) launch vehicle performance metric so other metrics such as the launch cost per unit mass must also be included in comparisons. Air breathing launch concepts are typically designed to be reusable to offset the added financial cost of the additional complexity of the air breathing engine over a traditional rocket engine.

Air breathing engines show significant specific impulse advantages over traditional rocket engines over specific ranges of vehicle Mach number. Figure 6 illustrates the potential specific impulse available from both air breathing technologies and rockets. Rockets consistently provide a significantly lower specific impulse over the entire range in Mach numbers than the optimal air breathing technology with the same fuel, but no single air breathing technology can operate over the entire range of Mach numbers required to reach orbit (Mach = 0-25). This naturally leads to proposed air breathing launch vehicle concepts that employ multiple integrated propulsion technologies. A wide variety of ramjet, scramjet, and combined cycle devices have been proposed and tested and a good overview of both the historical development and current state is given by Fry.[50] The two most common categories of solutions for launch applications are commonly referred to as the rocket based combined cycle (RBCC) and turbine based combined cycle (TBCC). Although a wide variety of combined cycle concepts have been proposed, the technology does not currently exist to build any of them so much of this discussion will center on the individual cycles.

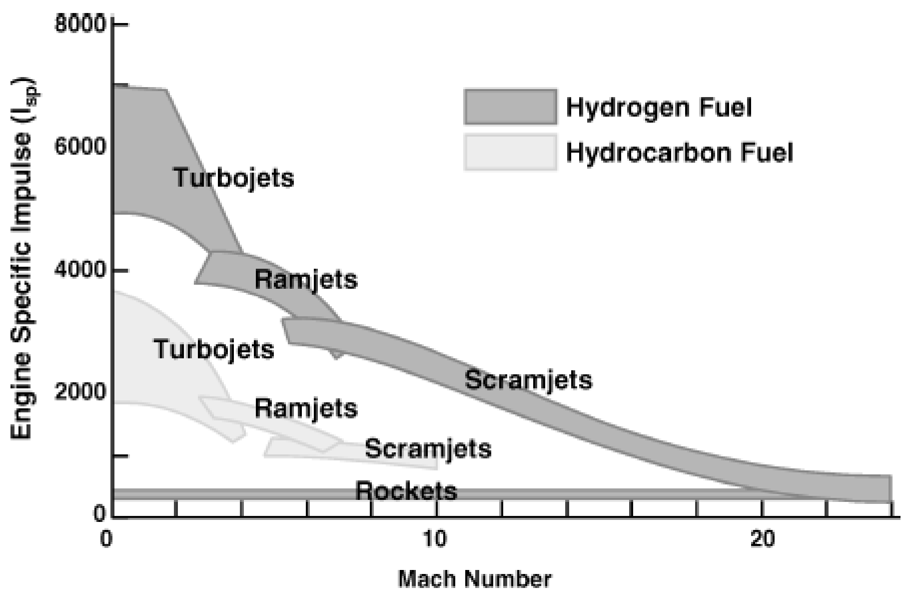

Figure 6. Air Breathing and Rocket Performance Verses Mach Number

1. Ramjets

Ramjets are relatively simple jet engines that have no moving parts. Unlike the workhorse turbofan engine which uses rotating machinery to provide the necessary compression for efficient combustion, the ramjet simply uses the dynamic pressure due to the forward motion of the jet as shown in Fig. 7. The ramjet nozzle intakes ambient gas and efficiently slows it to subsonic speeds for combustion. The fuel is then injected into the flow and combusted with oxygen from the local atmosphere in the combustion chamber. The hot products are then exhausted out of a nozzle to produce thrust. Ramjets can't operate under stationary conditions and require another technology (turbojet in TBCCs or rocket RBCCs) to boost it to sufficient speeds for operation. Ramjets can operate between roughly Mach 0.5 and Mach 5. The low Mach number limit is due to low compression produced by the slow speeds and the high Mach number limit is due to dissociation.

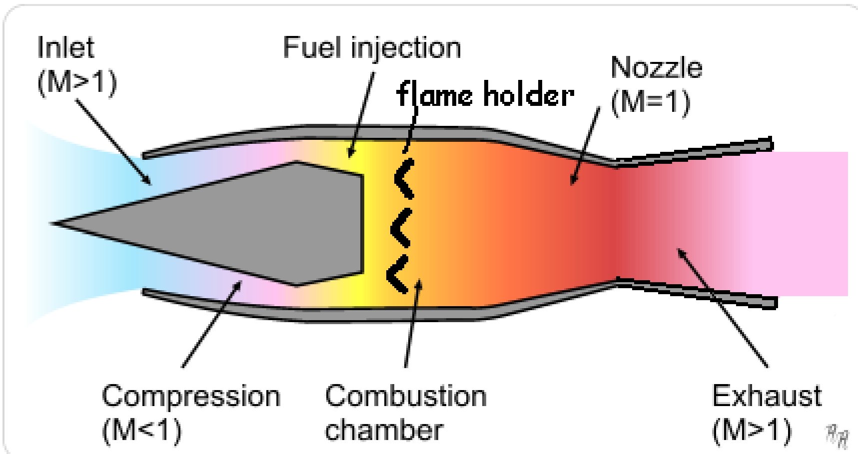

Figure 7. Typical Ramjet

The first known ramjet publication is from 1913.[50] In the century since then ramjet propulsion systems have become relatively mature technology with successful applications in missiles[51], aircraft[52], and even an experimental helicopter[53]. The RIM-8 Talos missile successfully integrated rocket and ramjet technology into a single vehicle.[54] The J58 engine on the SR-71 Blackbird was a successful integration of a turbojet and ramjet engine.[52] Ramjet technology, ramjet/rocket integration technology, and ramjet/turbojet integration technology, therefore, can all be considered technologically available.

2. Scramjets

A scramjet (supersonic combustion ramjet) is a variation of the ramjet where the flow in the combustor remains supersonic as shown in Fig. 8. Scramjets, like ramjets, have few or no moving parts and compress the incoming gas for combustion. Unlike ramjets, however, scramjets maintain a supersonic flow throughout the engine so supersonic combustion takes place. Scramjets require a minimum of at least Mach 5 to operate. The upper limit on the operational Mach number is not accurately know, but is estimated to be between Mach 8-10 for hydrocarbon fuels and could be up to perhaps Mach 25 for hydrogen fuel.[50] Three of the major technological hurdles still requiring solutions are the difficulties in designing for range of conditions, minimizing weight, and surviving and controlling the thermal load.

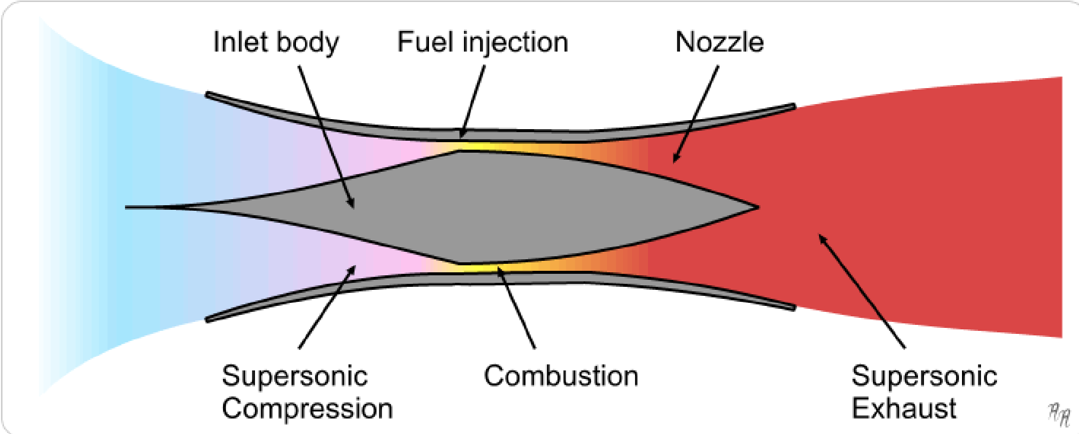

Figure 8. Typical Scramjet

Scramjets are being investigated around the world.[55] No scramjet powered complete vehicles have ever flown. Model scramjets (X-43A) have successfully flown up to Mach numbers of 10,[56] but no flown scramjet has ever been designed to survive for multiple flight tests. Both ramjets and scramjets have significantly lower thrust to weight ratios (~2 versus ~100) than rockets indicating that they will spend a significantly longer time (15-30 minutes) at high speed in the atmosphere and that the gravitational losses and aerodynamic losses will be significant. Significant scramjet development is required to advance the technology to the level that it can be integrated into combined cycle launch systems.

3. Pulsed Detonation Engines

In addition to ramjets and scramjets there are alternative technologies that may be incorporated into air breathing combined cycle technology in the future. One such technology is the pulsed detonation engine (PDE). PDEs are conceptually simple devices. Fuel and air are mixed in the closed end of a tube, ignited into a supersonic detonation, and then exhausted through a nozzle at the opposite end of the tube. Generally, detonations are treated as a constant volume cycle while deflagrations operate as a constant pressure cycle so, theoretically at least, detonations can more efficiently convert the stored chemical energy to propulsive energy.[57] Practically, however, the PDEs that have been experimentally tested have not yet yielded the predicted performance improvements over similar ramjets.[58] PDEs are not commercially available, but the first flight of an aircraft powered by a PDE occurred on January 31, 2008.[59]

4. Rocket Based Combined Cycle

Rocket-based combined cycle propulsion systems (RBCCs) are one class of combined cycle propulsion systems (CCPs) that attempts to achieve a system integrated specific impulse significantly higher than tradition chemical propulsion systems in order to lead to a significant reduction in the launch costs. A traditional RBCC consists of 4 heavily integrated cycles (rocket-ejector, ramjet, scramjet, rocket) that operate individually at various ranges of Mach number. More recently RBCCs with a rocket cycle and any number of other cycles have been looked at. Daines and Segal give a review of the technology and critical technological issues associated with RBCCs.[60] Tang and Chase give a more recent review of the history and current status of relevant air breathing hypersonic flight.[61] It is often times intuitively assumed that RBCC based RLVs will greatly reduce the cost of launch, but recent analysis indicates that the launch costs are likely to be in line with EELV costs.[62] Analysis has also shown that direct ascent trajectories are not feasible with RBCCs because of their low thrust to weight ratio.[63] As indicated earlier, the scramjet portion of the RBCC is not currently technologically available, but it is the focus of demonstration programs like the X-51A[64] and the HyCAUSE program[65]. Dependable hypersonic airbreathing flight is still the critical limiting component that must be demonstrated before RBCCs can be truly evaluated.

5. Turbine Based Combined Cycle (TBCC)

Turbine-based combined cycle propulsion systems (TBCCs) are similar to RBCCs, but they use a turbine based propulsion system for the first (lowest speed) mode of operation. As shown earlier a turbine based air breathing propulsion system has a significant (order of magnitude) performance advantage over rockets at low Mach numbers. It is expected, therefore, that replacing the rocket with a turbine based propulsion system should yield a higher performance system. A similar analysis for the cost of a specific TBCC system design showed that the launch costs were slightly above that for a similar RBCC and similar to EELV costs.[66] Accurate cost estimates will not be available, however, until the required scramjet technology is available.

III. Propellantless Launch Concepts

A. Electromagnetic (Rail)

Over the last couple of decades there has been interest in electromagnetic rail systems for various applications. The Army has investigated weaponized systems for future combat vehicles with greater lethality than the 70 ton Abrams but with a weight less than 20 tons. The Navy has also shown interest in this technology for long range shore bombardment. Current 5 inch guns have muzzle energy of 11MJ. It is estimated that similar sized EM railguns would have the capable of operating at 20 MJ muzzle energy and could achieve 300 to 800 km range based on a 2,500 m/s velocity[67] The Air Force interest in EM rails lies in the feasibility of a low cost, small payload to LEO launch system. The cost per unit mass could be as low as $600/kg compared to $20,000/kg of the space shuttle if the required launch rate can be achieved.[68]

There have been several EM rail systems conceived for launch to LEO, each with their inherent technical problems that would need to be overcome. Due to the fact that at 50,000 ft the density, dominated drag losses, is reduced to about 87% of the value at sea level, it has been conceived to mount a EM rail system in a large aircraft. This would reduce the aero-thermal loading; however it places limits on payload size and would require even larger gee-forces because of the inherently short track length. In addition, issues arise such as the placement of the gun and pulsed power equipment within airframe. Research would also need to be done to minimize launch effects (torque and recoil) on the aircraft.[67] Another option is a ground launch facility that would use longer tracks to accelerate larger payloads. While the gee forces experienced during launch would still be large (1000's of gee) they would be much more manageable. These facilities could be built near the equator on the side of a mountain. Major work still needs to be done with large bore rails which have not achieved as high muzzle velocities as their smaller bore counterparts. Due to the lower velocities of this concept (relative to required launch velocities), EM systems in a launch-assist role seem the most feasible way to reach orbit.

To date there have been a number of milestones in EM railguns. The integrated launch package has been tested with a 4 kg projectile and an exit velocity of 2 km/s, demonstrating a total muzzle energy of 8MJ[69]. Flight test firing of a 2 kg projectile to an altitude of 120 km was preformed.[70] It has been suggested that the 32MJ facility at the German-French Research Institute of Saint-Louis be used to further these flight tests.[71] Laboratory rail systems have achieved very high exit velocities, accelerating a small 7g launch packages to 7 m/s,[72] showing that there is no fundamental barrier of achieve the require muzzle velocity.

Currently there have been four critical issues that need to be address to see the success of electromagnetic launch. First, velocities greater then 7 km/s have to be achieved with acceptable acceleration limits and payload size. Second, the devolvement of pulse power system that can deliver the necessary mega-Amperes of current and appropriate power to the track. Third, aero-thermal loads on the projectile during its trans-atmospheric flight need to be addressed. And finally, nano-satellite technology must advance in order to withstand the high levels of acceleration.[72]

The principle mechanism limiting the velocity was identified as viscous drag on the plasma and neutral gas ablated from the bore wall by plasma radiation.[73] This ablated plasma and gas leads to drag and unwanted secondary arcing. Another major loss mechanism in EM railguns is friction related stresses and strain on the rail and projectile, causing both a negative acceleration force as well as fatigue on components. High aero-thermal loads and projectile ablation require an overall launch package that can survive the tremendous loads on the way to orbit. It has also been argued that one of largest loss mechanism is plasma venting.[74]

The amount of energy needed to be stored for an EM rail launch system is tremendous. To launch a modest 1,250kg package would require muzzle energy or 35 GJ. Assuming an 80% energy conversation, an input energy of 44GJ would be required. It has been suggested that these energy levels could be supported using high-speed rotating electrical generators.[75] Even a very light payload of 10 kg would require 250MJ which is comparable to one of the largest energy storage facilities at Sandia National Laboratory. To date laboratory rail guns have only demonstrated muzzle energies of about 9 MJ.

Launch velocities for a railgun launch to LEO require velocities of approximately 10.6 km/s including losses.[76] Due to the large velocities relatively low in the atmosphere, the projectile is subject to high drag and thermal loads, which can lead to severe material ablation. Therefore advanced materials or complicated cooling systems are need, adding to the overall mass of the launch vehicle. Studies have shown[77] that nose tip passive or active cooling methods are possible.

Electromagnetic launch will almost always involve very large acceleration. Even for tracks of moderate length, extremely high accelerations are needed in order to achieve the greater than 7 km/s velocities needed to reach orbit. For a 1 km track, several thousand gees are required.[74] Aircraft launch concepts that uses much shorter tracks can experience 50,000 gees or more. Because of these extreme forces, no manned launch would be possible; even electronics, sensors, classic propellants, and other delicate equipment could not be launched in this manner. Only rugged payloads would be possible, such as supplies, material, advanced fuels and water. In addition to the rugged nature of the payload, payload size is also a consideration. In order to accomplish an equivalent mass deposition to LEO as current rocket systems, a large number of launches would be necessary. To achieve the 500 tons a year to orbit, conceived electrometric systems would require an average of 5 launches a day.[68] This brings up many issues of launch infrastructure and logistics, space trafficking, and launch waste materials.

B. Elevator

The idea of a space platform that reached from the ground to geosynchronous orbit (GEO) was first put forward by Konstantin Tsiolkovsky in 1895. This concept would allow a payload to be placed into orbit without the need for traditional rockets, although rockets were certainly not "traditional" in the late 19th century. It was only later in 1903 that Tsiolkovsky's theory on overcoming Earth's gravity using rockets (reaction vehicles) was published. Modern versions of the space elevator concept include a cable which traverses from the ground to GEO, a massive counterweight above GEO, "climbers" which deliver payload from the ground to GEO, and power beaming (laser or microwave) concepts to propel the climbers up the cable. For these concepts, a cable is lowered from GEO in order to deploy a second cable upwards thus achieving a desired orbital altitude. [78]

The major advantage of the space elevator concept is that there is no need to produce or store any energy to access space. Once the space elevator is in place, subsequent launches can be performed quickly and relatively inexpensively. The major disadvantages of the space elevator concept are the need for extremely high tensile strength materials, high power requirements for most climber concepts, and cost to name a few. To combat the high tensile strength requirement (65-120 GPa), carbon nanotubes (CNTs) have been proposed for the cable material with a theorized tensile strength of 130 GPa.[78] The density of CNTs is also low (1300 kg/m^3) which allows for a relatively low cable mass. Currently, carbon nanotubes have achieved lengths of several centimeters.[79] By tapering the cable or ribbon, sufficient support strength can be achieved. For CNTs, the cable's cross sectional area at GEO would need to be 2 to 10 times larger than the cross sectional area on the ground to support itself.[80] Further development at potentially high cost is required to produce CNTs applicable to the space elevator concept.

It has been proposed that the counterweight at an altitude beyond GEO could be a captured nearby asteroid.[78] Since a counterweight on the order of several metric tonnes is required for the most concepts, the idea of capturing an asteroid for the counterweight is not out of the question. Launching this amount of mass beyond GEO from the Earth would be prohibitively expensive using today's technology.

Several problems with the space elevator concept have been identified including micrometeoroid and orbital debris impact, tropospheric weather, atomic oxygen interactions in low-Earth orbit, Van Allen radiation interactions in medium-Earth orbit, vibration, safety, security, and economic issues. Besides the cost of the cable material, another major economic limitation appears to be that potentially thousands of kilograms would need to be lifted to space via conventional rockets.

There are many technical, economic, and political challenges to the implementation of the space elevator concept, rendering its implementation in the near future infeasible. However, what was nearly impossible to consider in the early part of the 20th century, has gained momentum since the discovery of CNTs in 1991, and the cost of producing CNTs has dramatically decreased in the past several years. Future advances in materials and materials processing may further enable space elevator applicable technologies.

C. Space Platforms and Towers

Whereas the space elevator seeks to build a structure that reaches beyond GEO, space platforms and towers are being considered that would only reach up to 100 km. These towers would then be used to launch rockets carrying payloads to higher orbits or interplanetary space. Currently, the world's tallest structure is a television transmission tower which is 629m high.[78] Building structures significantly taller than this is not only technically challenging but also expensive. Bolonkin[81] suggests that inflatable towers could be constructed using lighter-than-air gases for factors of 100 less cost than traditional structures. The act of inflating the tower would also lift the intended payload to a desired altitude; however, other propulsive means would be required to lift the payload beyond the inflatable altitude with significant ΔV requirements depending on the desired final orbit. Although the tower may have several other uses such as a communications tower, space weather platform, astronomical observations, or space tourism, the direct benefit to satellite launch versus the cost of the structure in a launch assist role is not immediately clear. Additionally, only about 10% of the energy needed to attain low-Earth-orbit is due to the Earth's gravitational potential. By far, the majority of the energy is kinetic to reach the required orbital velocities. From this perspective, launching from a space platform or tower at an altitude of 100 km does not necessarily lead to a large advantage to accessing space.

D. Gravity Modification and other Breakthrough ideas

Many advanced concepts for launch involve the modification or complete removal of gravity as a means for accessing space. Studies involving transient mass fluctuations[82], gravity shielding[83], and even warp drives[84] have been conducted which look at altering gravity. These concepts, when applied to the problem of launch, act to reduce the amount of potential energy required to attain orbit. As previously discussed, the gravitational potential accounts for only 15-20% of the total ΔV required to reach LEO. The remaining ΔV is required to change the vehicle's velocity from the Earth's rotational velocity (0.46 km/sec at the equator) to orbital velocities (~7.8 km/sec in LEO). Thus, these concepts do not generally address the major issue of launching payloads to LEO. A study by Tajmar and Bertolami[2] suggests that the "gains in terms of propulsion would be modest (from these concepts) and lead to no breakthrough." Although there would be advantages to launch vehicles using the concepts investigated in Ref. 2, they were not deemed to be sufficiently beneficial to justify the cost of development and implementation. The study looked at inertial mass modification, gravitational mass modification, and gravitometric field utilization.

Inertial mass modification involves changing the initial mass of a launch vehicle and/or its propellant. Tajmar and Bertolami[2] suggest (and correctly so) that the modification of the inertial mass of both the vehicle and the propellant leads to no influence at all on the overall system. However, Millis[85] suggests that inertial mass modification of just the propellant could lead to breakthrough-type propulsion applications. In this case, the inertial mass of the propellant is increased as it is expelled from the vehicle. It must be pointed out, however, that there are no current schemes for achieving such a modification in any case.

Reducing the gravitational mass of a launch vehicle would lead to a direct ΔV reduction, meaning lower propellant mass would be required to reach orbit. For a 100 km LEO satellite, the ΔV reduction would be approximately 1.4 km/sec if the gravitational mass of the vehicle were reduced to zero. However, a ΔV of 7.5 km/sec would still be required to attain orbit requiring a subsequent launch strategy. This would be akin to using the gravitational mass modification concept in a launch-assist role. For a GEO satellite the required ΔV could be reduced from 13 km/sec to about 3 km/sec which would dramatically reduce the cost of a traditional launcher to this altitude. Obviously, this concept is a long way from fruition and would have to compete with more traditional concepts such as high efficiency electric propulsion or nuclear orbital transfer vehicles.

Gravitomagnetic fields involve generating a Lorentz force analog of gravity. The generated electrodynamic fields would interact with the Earth's magnetic field to generate thrust. For most configurations and reasonable power levels, only a very small force could actually be generated. Since the minimum requirement for a launch vehicle is to have a thrust-to-weight ratio greater than one, this approach is not likely to lead to a breakthrough in launch vehicle applications. Tajmar and Bertolami[2] suggest that this concept offers to additional benefit over current tether technology even if proposed superconductor concepts could be employed.

Other advanced concepts that could be employed for space access include antimatter and quantum vacuum energy (or Casimir) forces. The Casimir force[86] has been suggested to exist due to the energy state of vacuum or empty space. It has been demonstrated that the vacuum energy can force flat plates together when the plate distance is very small (on the order of microns). Although a typical launch site would not have a vacuum condition, this condition can be approached as the plate separation distance gets extremely small even at atmospheric conditions. Nevertheless, the force generated by quantum vacuum energy is extraordinarily small and would not be applicable to launch vehicles.

Antimatter converts all of its mass to energy during its annihilation with normal matter. Antiproton annihilation has been suggested by Forward[87] as a means of propulsion. During annihilation, antiprotons convert nearly two-thirds of their energy into charged particles which can be harnessed to produce thrust. Antimatter is a highly concentrated means of energy storage with a specific energy density of 9 x 10^{16} J/kg compared with about 10^{7} J/kg for chemical reactions. With a high specific energy density, high specific impulse, high thrust systems can be envisioned that could one day be applicable to launch vehicles. The current limitation on this technology is the production rate and subsequent storage of antimatter. Capture of antiprotons produced in current facilities remains a difficult task, and although long-term storage has been demonstrated, storage is limited to very small quantities on the order of 10^{11} particles/cm^{3}.[88] Another significant drawback to antimatter propulsion systems is the current production level which is not anywhere near capable of supporting a launch effort. Additionally, the efficiency of current production process and trapping is in the range of 10^{-8} indicating that significant power is required to produce a relatively small amount of contained antimatter.[88]

E. Launch Assist

Many of the technologies listed above may first be viable for providing only a limited fraction of the total velocity increment required to reach orbit. Although in theory the velocity increment could be added at any part of the launch trajectory, for this discussion only the first stage launch assist technologies will be discussed. In general first stage launch assist technologies can provide a reduction in the velocity increment delivered by the chemical rocket by increasing the initial kinetic or potential energy or by reducing the losses. Traditionally this has been envisioned by either launching from higher altitudes or launching with an increased initial velocity. First a brief description of the potential of launch assist will be described and then individual launch assist technologies will be briefly discussed.

1. The Potential of Launch Assist

Replacing the first stage of a chemical rocket based launch system by an alternative technique has the potential to reduce the total weight, complexity, and cost of the chemical rocket portion of the space launch system. It must be remembered, however, that system level performance metrics such as cost, reliability, and availability have not yet been proven for the discussed launch assist systems. As discussed in the introduction, historical predictions for the cost savings potential of advanced space launch systems have systematically been overly optimistic and for this reason the discussion below focuses simply on whether the launch assist system can have an effect on the total required velocity increment either by providing initial potential or kinetic energy or through a reduction in the traditional velocity increment losses. The total required velocity increment is vehicle and mission dependent, but some general statements can be made. As simple expression for the total design velocity increment is obtained by treating it as simply the sum of the burnout, gravity, and drag velocity increments as shown in Eq (3).

$$\Delta V_{design} = \Delta V_{burnout} + \Delta V_{gravity} + \Delta V_{drag} \tag{3}$$

Gravity and drag losses are dependent on the specific launch vehicle and trajectory, but together these losses account for 1.0 to 1.5 km/sec of the velocity budget.[12] While the burnout velocity increment is dependent on the launch site properties and the final orbit, but is typically between 7.5km/s (LEO) and 11km/s (GEO).

Figure 9 shows the total mechanical energy per unit mass breakdown of circular orbits as a function of altitude. The total required energy per unit mass varies between approximately 30MJ/kg for LEO orbits to 60MJ/kg for GEO orbits (neglecting losses). In LEO orbits the total mechanical energy is predominantly kinetic energy. At an altitude of approximately 3,200km the total mechanical energy is composed of equal parts kinetic and potential energy and in GEO the total mechanical energy is mostly (92%) potential energy. A launch assist system should provide a measurable fraction of the total required energy (velocity increment) to be viable.

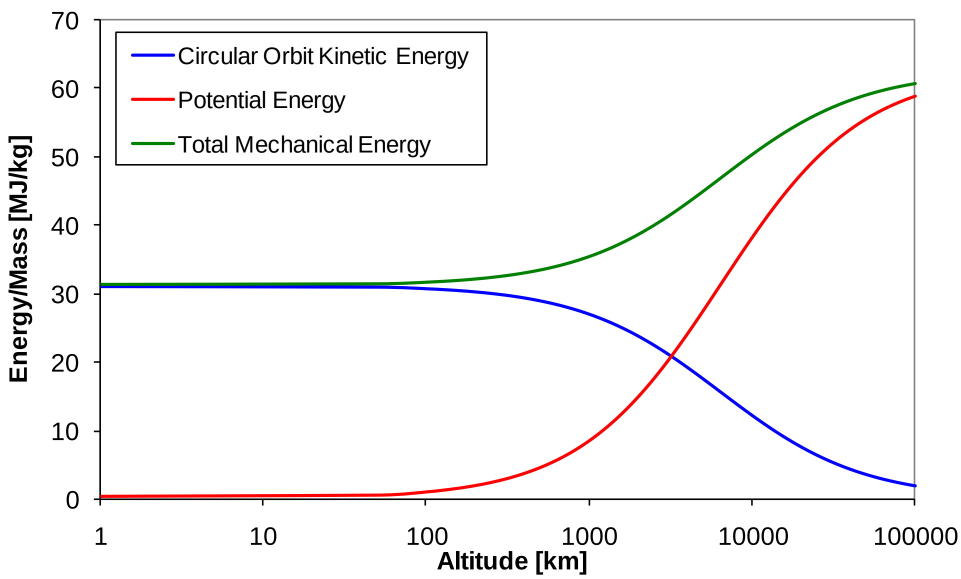

Figure 9. Energy Per Unit Mass for Circular Orbits

Figure 10 shows the same energy per unit mass breakdown, but for only potential energy and compared to some potential energies relevant for launch assist evaluations. The summit of Mount Everest is the highest point above sea level on the Earth's surface at 8.85km. Although it is not a practical launch site, it does indicate that placing a launch site at higher altitudes can yield no greater than an increase in energy per unit mass of 0.087MJ/kg or << 1% of the total mechanical energy. Clearly increasing the launch site altitude does not yield significant increases in potential energy. The Pegasus launch system typically releases the launch rocket at an altitude of 12.2km which yields an increased potential energy of 0.12MJ/kg over launch at sea level or only about 0.4% of the total required mechanical energy. Similarly, even near space dirigibles at 20km altitude would only increase the potential energy by 0.2 MJ/kg (0.7%). In order for the increased potential energy from launch vehicle altitude to make a useful impact the launch must take place at an altitude that is a significant fraction of the desired altitude which is not currently technologically feasible. Additional benefits of launching at high altitude such as reducing the drag losses or improving the launch availability by launching from above weather systems may be relatively more important.

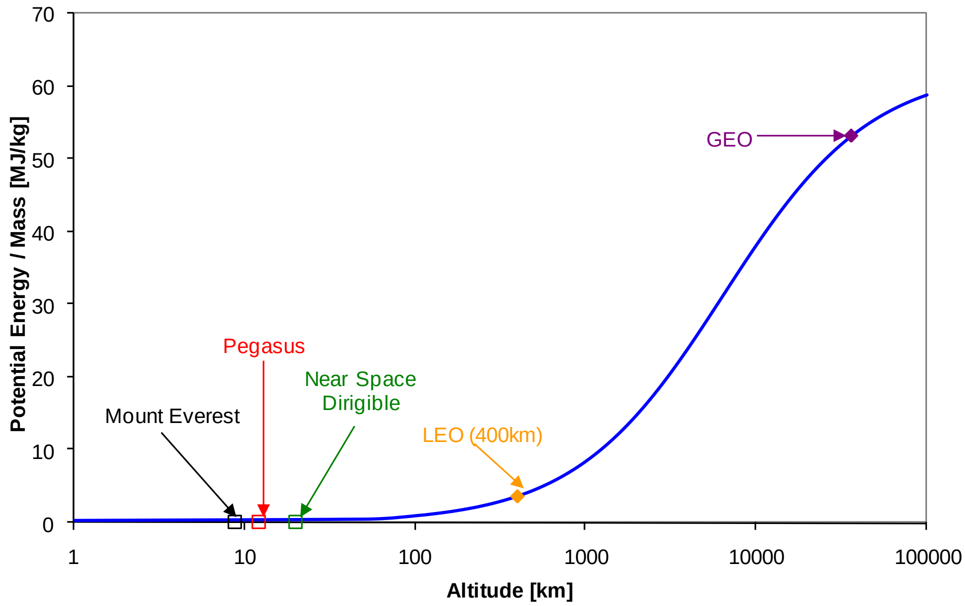

Figure 10. Potential Energy per Unit Mass for Various Launch Assist Technologies

Figure 11 shows the same breakdown for kinetic energy per unit mass as a function of altitude. Using the rotation of the Earth's surface can yield approximately 0.1MJ/kg (<0.5%). The Pegasus launch vehicle releases the rocket at approximately Mach 0.8 which corresponds to less than 0.1% of the total LEO mechanical energy. It appears, therefore, that the effects of the initial launch velocity from aircraft or the Earth's rotation is a small component of the total energy required to reach space.

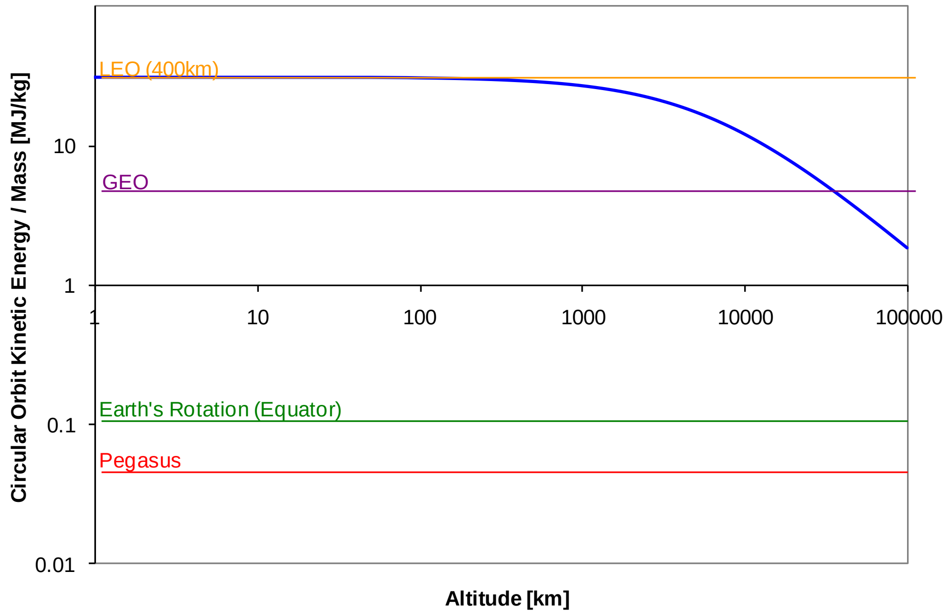

Figure 11. Potential Energy per Unit Mass for Various Kinetic Launch Assist Technologies

The last potential for launch assist technologies is to reduce the velocity increment losses. Drag and gravity losses can account for up to 10% of the total design velocity increment. Significantly reducing the required velocity increment due to losses is difficult, however, because reducing gravity losses typically requires operating at higher thrust to weight ratios which would tend to increase the drag losses unless the initial launch takes place at high altitudes with greatly reduced atmospheric pressure.

2. Air Launch

Air launch systems employ either fixed wing aircraft or balloons/dirigibles to launch payloads at altitudes above 10km. Sarigul-Klijn published a summary of the potential of fixed wing air launch systems.[89] A wide variety of air launch systems have been proposed[90], but only one system is currently being used, the Pegasus rocket[91] from Orbital Sciences Corporation. The Pegasus rocket uses a L-1011 aircraft to carry the rocket/payload to an altitude of approximately 12km where it is released at approximately Mach 0.8. As discussed above this doesn't represent a significant fraction of the total mechanical energy required for LEO or GEO, but it does have several advertised advantages. Air launch typically occurs above weather systems. Assuming that the launch site isn't also experiencing bad weather the aircraft can take-off and rise above the weather to launch the rocket/payload. Air launch systems can also yield any launch azimuth without costly plane change maneuvers. Air launching allows the use of more efficient first stage nozzles. This has not resulted in a significant cost advantage for the Pegasus launch vehicle. The Pegasus launch vehicle is capable of placing 443kg into orbit at a price of just over $40,000/kg. As shown in Fig. 2 this is consistent with trends of other launch vehicles as a function of the payload mass. One additional limitation for air launch systems is that they are size limited because they require very large aircraft and balloons to launch even small payloads.

Envisioned balloon launch systems are simple, low performance systems.[92,93,94] The launch systems typically have no control during balloon ascent. Balloon assisted sounding rocket launch of 18kg payloads to altitude of up to 100km was demonstrated by the Rockoon flights by Van Allen in the 1950's.[95] Launching larger satellites into permanent orbit is very difficult with balloon assisted launch. The state of the art in large balloons is the ultra long duration balloon (ULDB) from NASA.[96] The balloon has an envelope volume of 631,500m^3 and is capable of delivering a payload of 2,721kg to an altitude of 33.5km and maintaining the altitude for 100 days. Balloons with volumes significantly larger than that become increasingly more difficult. The delivered mass (payload and launch vehicle) is almost an order of magnitude lower than the Pegasus rocket, but it is also launched at a much higher altitude. Balloon launch assist appears likely to occur for only very small satellites.

3. Electromagnetic Launch

There are two electromagnetic technologies that have been proposed for use as launch assist systems, coilguns and railguns. Both systems have been discussed in detail elsewhere so only their launch assist abilities will be discussed here. Coilguns use electromagnetic coils to accelerate a magnetic payload to high velocities. Coilguns do not require sliding contacts indicating that they may have longer lifetimes than railguns. The current technology limitations of coilguns include high-voltage, fast-acting switches and parasitic resistance/energy dissipation. Coilguns have higher predicted performance, but have slightly lower achieved performance due to limitations in available technology. Railguns have successfully launched gram size projectiles at up to 10km/s, but kilogram size projectiles have only reached several km/s. Proposed systems often require > 100s of kg at several km/s indicating that the technology is still orders of magnitude from being able to provide the basic performance.

Fundamentally, this type of system will be constrained by the energy storage and instantaneous power available, the real estate (and funds) available to construct the launcher, and the acceleration the payload can withstand. Figure 12 illustrates the basic kinematic relationship between desired velocity, track length, and acceleration. It is not possible to achieve all of the desirable traits of high velocity, a short track, and low acceleration; some compromise must be made. The total energy and instantaneous power available to drive the apparatus must also be considered; useful missions could easily require >> 1 GW of power.

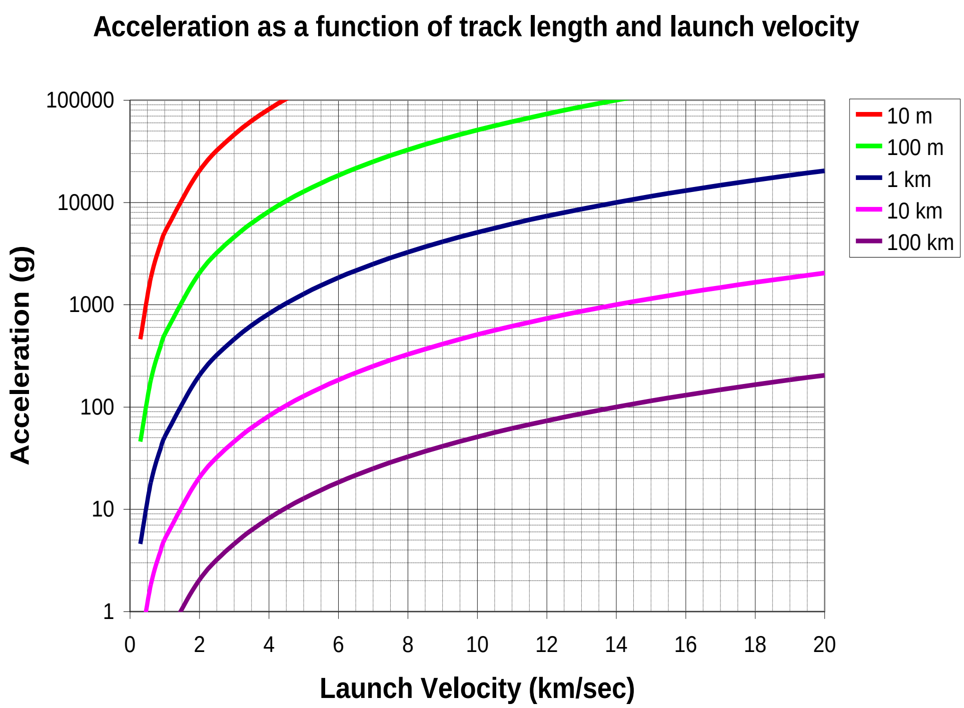

Figure 12. Acceleration as a Function of Launch Velocity for Various Track Lengths

As an example, assuming that 1 GW of power is available, it would be possible to launch roughly 600 kg at 3.4 km/sec and a 30 degree incline, assuming suitable terrain could be found for the 6 km long launcher. While most payloads and rocket motors are not designed to withstand the resulting 100g acceleration, projects led by Bull (see gun-launched section below) have developed preliminary designs of rocket motors that could survive such an environment. If most of the 600 kg launch mass were such a rocket motor, it is reasonable to suggest that the payload to orbit could be as high as 35 kg for such an arrangement.

Using the electromagnetic system to deliver a payload directly to orbit appears to be more challenging. A payload would need to be launched with a velocity several km/sec faster than orbital velocity as it will quickly lose this energy to drag while transiting the lower atmosphere. The aeroheating on the payload as well as the power and acceleration needed to reach such high speeds with a track of practical length casts some doubt on the feasibility of such a concept.

4. Gun Launch

The idea of launching payloads using gun launch has been around since Jules Verne wrote about it in his story *Voyage from Earth to the Moon* in 1865. Morgan has compiled a good history of the development of gun launch systems.[97] This review will not, therefore, cover the historical development of gun launch, but will briefly detail the current state of the art. Guns are commonly divided into the classes of gasdynamic guns and light gas guns. Gas dynamic guns consist of a long high-strength tube with only one end open. The other end is packed with explosive charge and the projectile. Ignition of the propellant fills the chamber behind the projectile with high-temperature high-pressure gas accelerating the projectile. Nitrocellulose powered guns are limited to exhaust velocities of approximately 3km/s which is sufficient for launch assist, but is insufficient for direct launch applications.[97] A representative of the state of the art in large gas dynamic guns is the HARP gun effort led by Gerald Bull in the 1960s. The guns were funded to study hypersonic reentry, but were also an incremental step in the development of a gun launch system. Figure 13 shows a firing from a 16 inch gun. The gun was assembled from surplus artillery tubes. The final version fired a 180kg projectile at 3.6 km/s (5,500 gees) and it reached an altitude of 180km. The next generation was to be a full launch system with the cannon launching 1,300kg at 1.8km/s delivering a payload of 90kg to LEO. In addition to the basic launch demonstration the HARP effort also demonstrated hardened payloads that could withstand the launch conditions (including solid rocket fuel).

Figure 13. HARP 16 Inch Gun Firing

In order for a gun launch system to directly launch a payload it must achieve muzzle velocities significantly above the roughly 3km/s that a gasdynamic gun can achieve. Launch assist systems could also benefit from higher muzzle velocities. This requires either a higher temperature or lighter gas molecule which is the design philosophy behind the light gas gun. The current state of the art in demonstrated light gas guns is the Super High Altitude Research Project (SHARP).[97] The SHARP gun was developed by Lawrence Livermore National Laboratory with the hopes of providing a significant fraction of the required orbital velocity to a payload. Light gas guns work by using an explosion of typically gunpowder to drive a piston and compress hydrogen gas. Once the gas reaches approximately 4,000atm it bursts a disk and propels the projectile out the end of the gun. The SHARP gun demonstrated launching a 5kg projectile at 3km/s, but the next phase of the project to increase the velocity and angle the gun upward was not funded. Both gasdynamic guns and light gas guns have demonstrated impressive performance and are candidates for launch assist systems. Both technologies suffer, however, from not being able to obtain the required funding (billions of dollars) for the final phase of the technology development. Gilreath et. al. have also shown that even if a light gas gun launch system is technologically feasible it appears to not make economic sense unless the number of required launches grows significantly.[98]

IV. Discussion

By definition, an advanced launch concept would offer vast improvements over current launch capabilities thereby forever changing the way in which payloads reached orbit. Although improvements to chemical rockets can certainly be envisioned, "game-changing" increases over present capabilities are not likely to emerge from derivatives of current designs. As an analogy, there were relatively few advances in transportation for millennia, from the invention of the wheel up until the industrial revolution. On the great increase in power available from advanced technologies of steam and internal combustion engines could bring about a revolution in the way people traveled. The invention of the airplane brought further revolution, and perhaps one day advances in space travel will provide yet another mass transit breakthrough. However, finding solutions which can offer a similar dramatic improvement over today's chemical rockets will be extremely challenging.

The key will lie in the safe and efficient production, storage, and transfer of large amounts of energy. Tens of megaJoules of energy are required to move the mass of a single kilogram from the surface of the Earth to LEO. For chemical rockets, this energy is delivered in a matter of minutes with gigaWatts of power produced at liftoff. As the rocket equation indicates there are several ways to improve on the amount of ΔV a launch vehicle can provide. First, higher specific impulse would be extremely beneficial. For chemical systems, this means higher chamber temperatures which would easily surpass the melting point limits of most known materials. Second, a reduction in liftoff (or initial) mass would also aid in increasing the ΔV of the launch vehicle. Composite materials and advanced nano-structured materials may lead to lighter weight vehicles which could increase payload mass fractions. Removing components, such as power sources, on traditional vehicles and placing them on the ground could lead to significant reductions in liftoff mass. Launch assist concepts which essentially act as a first stage propulsion system can also lead to reduced mass of the overall vehicle. However as with all new technologies, there are challenges to the implementation of each of these ideas. In some cases, advanced concepts will only act to complicate launch vehicles potentially increasing the cost per unit mass to orbit while decreasing reliability.

Of the concepts presented in this manuscript, nuclear fission rockets hold the most promise of fulfilling the need for improved rocket performance in the short-term. That is not to say that there aren't major challenges ahead for this concept, but the past development in this concept and relatively high technology readiness level (TRL) puts it ahead in the game. Obviously safety, reliability and cost are going to be major factors in the use of fission reactors for launch vehicles. Higher specific impulse implies that for the same ΔV, more payload mass can be taken to orbit if the inert mass can be controlled. Combined cycle systems also seem to hold more "immediate" promise for launch system applications since the basic technology underlying the concepts are relatively mature. In these systems, high specific impulse comes from the fact that the oxidizer is readily available in the atmosphere for much of the flight trajectory.

Beamed energy concepts offer the benefit of separating the power source from the launch vehicle. If the energy required to achieve orbit were generated at a remote (ground) site and then transmitted to the vehicle, there could potentially be no need for on-board propellant. Currently, 85-90% of a typical launch vehicle is propellant indicating the major benefit of these types of concepts. If propellant were necessary, only fuel would be required saving mass from not having to store oxidizer as well. However, beamed energy propulsion has yet to be successful due to the poor efficiency during the conversion of the supplied energy to propulsive force. Hypervelocity launchers using guns or magnets to accelerate payloads also separate the power source from the vehicle. However, the high velocity needed to reach orbit combined with relatively short launch "tracks" can lead to extraordinarily high launch loads.

Propellantless concepts may be the key to revolutionizing the launch vehicle paradigm. However, all of these systems are only at the conceptual phase. The technical challenges for a concept like the space elevator may be several decades away from being solved. To complicate matters even more, the initial investment cost of such a system may be forever prohibitive, regardless of the promised returns.

Large research efforts have been funded to improve chemical rockets. Many of these efforts have met with limited success. The current efficiency of chemical launch vehicles is extremely high. There is limited room for improvement, and further improvement will come at great cost. To deliver a payload to orbit, current launch vehicles require the rapid conversion of staggering amounts of energy. This will continue to be the case for most of the advanced concepts envisioned for launch. Basic research into the subsystems used for energy production, storage, and transmission will be required. It is clear that significant advancement in launch vehicle technology is necessary to usher in a new era in space access. It is also clear that when programs are put into place with appropriate levels of commitment, major advances can be envisioned in the near term.

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