JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 47, No. 2, March–April 2010

Overview of Advanced Concepts for Space Access

Andrew D. Ketsdever, Marcus P. Young, and Jason B. Mossman
U.S. Air Force Research Laboratory, Edwards Air Force Base, California 93524
and
Anthony P. Pancotti
ERC, Inc., Edwards Air Force Base, California 93524

DOI: 10.2514/1.46148

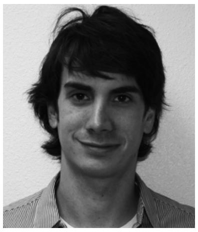
Nomenclature

A = area, m^2
c = speed of light, 2.99×10^8 m/s
F = thrust, N
I_{sp} = specific Impulse, s
$\dot{m}$ = mass flow, kg/s
m_{prod} = product mass, kg
P_{jet} = propulsive jet power, W
P_{prop} = stored propellant power available, W
$\Im$ = solar constant at Earth, 1358 W/m^2
t_b = burn time, s
v = velocity, m/s

Andrew D. Ketsdever is currently the Program Manager of Advanced Concepts at the U.S. Air Force Research Laboratory's (AFRL) Propulsion Directorate at Edwards Air Force Base. He has worked in the areas of advanced propulsion technology, micropropulsion, rarefied gas dynamics, and spacecraft–thruster interactions since starting at AFRL in 1992. Dr. Ketsdever received a Ph.D. in aerospace engineering from the University of Southern California in 1995. He is currently a member of the AIAA Nuclear and Future Flight Technical Committee, has authored or coauthored over 75 technical papers, and has coedited an AIAA Progress in Astronautics and Aeronautics series book entitled *Micropropulsion for Small Spacecraft*. He is an Associate Fellow of the AIAA.

Marcus P. Young received his B.S. in aerospace engineering (1998), M.S. in aerospace engineering (2000), and Ph.D. in aerospace engineering (2004) from the University of Southern California. Upon completion of his Ph.D. he joined the Advanced Concepts Group at the U.S. Air Force Research Laboratory's (AFRL) Propulsion Directorate at Edwards Air Force Base. He is currently the group lead of the Advanced Concepts Group which is investigating and developing revolutionary propulsion technologies. His primary fields of interest are spacecraft propulsion systems, advanced launch concepts, rarefied gas dynamics, and advanced power systems. He is a Member of the AIAA.

Jason B. Mossman is a Senior Aerospace Engineer at the U.S. Air Force Research Laboratory Propulsion Directorate at Edwards Air Force Base. He received his B.S. in aeronautical engineering from the University of California, Davis, and his M.S. in mechanical engineering from California State University, Fresno. His research interests include reusable launch vehicles, trajectory optimization, cost estimation, and ballistic missile design. He is a Senior Member of the AIAA.



Anthony P. Pancotti is a Senior Scientist with ERC, Inc., and a Member of the Advance Propulsion Group at the U.S. Air Force Research Laboratory (AFRL). His work entails the study of potential future propulsion concepts through area expertise and experimental research. He obtained his B.S. (2002) for the University of Buffalo and his M.S. (2004) and Ph.D. (2009) from the University of Southern California (USC). He currently runs the Collaborative High Altitude Flow Facility at USC, conducting experimental research for the AFRL. His research interests include plasma dynamics, space, high-altitude, and launch propulsion systems. He is a Student Member of the AIAA.

Presented as Paper 5121 at the 44th Joint Propulsion Conference, Hartford, CT, 20–23 July 2008; received 25 June 2009; revision received 5 January 2010; accepted for publication 18 January 2010.

ΔH = change in enthalpy, J
Δv = velocity increment, m/s
η = efficiency
ρ = reflectivity

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 and kerosene, to modern multistaged launch vehicles, the paradigm has been the same: combine fuel and oxidizer to extract chemical energy that 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, as defined by the ratio of jet power produced to chemical power available from Eq. (1) [1]:

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

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 liftoff weight into orbit. This seeming contradiction is a clear indication that revolutionary concepts are needed for space launch. These revolutionary concepts are not 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/s 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/s of total ΔV required to attain a low Earth orbit with a typical rocket trajectory, only about 1 km/s is involved in competing against the Earth's gravitational potential. About 0.5 to 0.75 km/s can be attributed to losses such as atmospheric drag, vehicle steering, and backpressure. Airbreathing-launch-assist trajectories would have higher losses. 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/s as an example, the energy required to reach a low Earth orbit is approximately 45 MJ/kg or 12.5 kWh/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 thousands to well over $10,000/kg [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 time frame. Two relevant U.S. Air Force 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, an 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 microsatellite, 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 that require propellant, and propellantless 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 The Boeing Company. The Delta IV was developed as part of the Evolved Expendable Launch Vehicle (EELV) program and had its first flight on 20 November 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; however, the full cost savings have yet to be realized.

The Delta IV Heavy is a three-stage launch system [6]. It uses three common booster cores that use LH_2/LO_2 for propellant and a RS-68 engine for the first stage. Each RS-68 is capable of providing a sea level thrust of 2.891 MN (8.673 MN total) at an I_{sp} of 410 s. 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 110 kN at a specific impulse of 462 s. The Delta IV Heavy is capable of placing 23,040 kg into LEO or 13,130 kg into GEO transfer. The cost per unit mass of the Delta IV depends on the specifics of the launch, but is still around 10, 000/kg.

B. Minotaur IV: Small Satellite to LEO

The Minotaur IV launch vehicle was developed by Orbital Sciences Corporation for the U.S. 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 2224, 1223, and 289 kN of thrust. The fourth stage uses an Orion 38 design and provides orbit insertion.

The Minotaur IV can deliver 1750 kg to LEO. The first launch of the Minotaur IV launch vehicle, with the TacSat4 payload, is scheduled for early 2010. Its predecessor, the Minotaur I, had launch costs that 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 an advertised launch responsiveness of under six months [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 10,000 times 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. Figure 1 shows the distribution of current total launch costs for common U.S. launch systems (as of 2007) as a function of payload mass for the payload delivered into a 185 km (100 n mile) circular orbit with an inclination of 28.5 deg. 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 \$9000 and 50, 000/kg (4000 and 23, 000/lb). Total launch costs can be divided into two categories: recurring and nonrecurring. 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 toward more complicated higher-performance launch systems and toward simpler lower-performance 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 EELV program, for example, is an effort designed to maintain U.S. 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 toward simpler designs and to use standardized components to reduce the launch costs. Both launch systems are operational, but they have not yet achieved the projected cost savings. On the other end of the spectrum is the space shuttle. The space shuttle is arguably the most complicated machine ever 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 the original goals to be achieved [9]. The initially projected flight rate was 60 launches per year (40 from Kennedy Space Center and 20 from Vandenberg Air Force Base) with a two-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, they have not historically been achieved by either significantly more or less complicated systems.

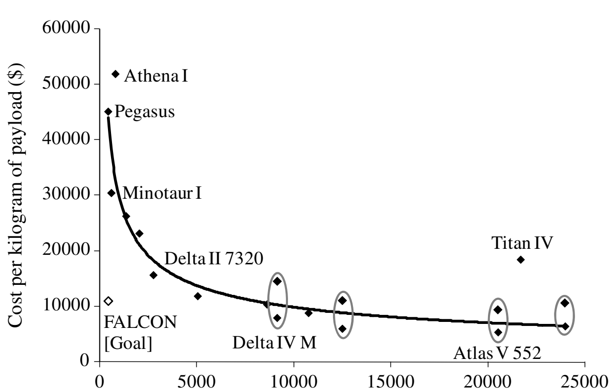


Fig. 1 Launch costs of common space launch systems.

D. Reliability

Launch reliability is another performance metric that is 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. Sauvageau and Allen [10] demonstrated that between 1964 and 2000 the reliability for U.S. launch vehicles was consistently between 0.91 and 0.95. 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 that would likely require significant improvement in all three areas. It was also shown that roughly two-thirds of the failures are due to propulsion elements, and the other one-third is due to a variety of nonpropulsion 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) [11]. 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 [12,13].

Different missions will obviously have different metrics that dominate the launch vehicle selection process. For example, some military missions may place a premium on responsiveness. Such missions would probably tolerate higher launch costs for quick access to space. Human missions will almost assuredly place a premium on reliability and will tolerate higher launch costs associated with redundancy, high-level component testing, and extensive systems testing. Low-cost microsatellite missions may be completely driven by the launch cost metric.

III. Advanced Launch Concepts Employing Propellant

A. Nuclear

The concept of nuclear fission rockets was first proposed in the late 1940s. Since then, the development of launch vehicles based on Nuclear Engine for Rocket Vehicle Application (NERVA), particle bed, and cermet reactors have been studied [14]. Fission represents a specific energy density of approximately 7×10^{13} J/kg at 100% efficiency, which is nearly seven orders of magnitude greater than chemical reactions can provide. Concepts range from 1000 s to upward of 5000 s of specific impulse. Most of these concepts use a working gas (typically hydrogen) as the 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 sociopolitical one. Nuclear fission systems may hold the most promise of fulfilling the need for improved rocket-only performance in the short term, based on the substantial amount of previous work performed. Nuclear fission represents the highest technology readiness level (TRL) of any purely rocket concept presented in this study [14].

A nuclear-fission-based space tug, an LEO-to-GEO transfer vehicle, has also been investigated [15,16]. 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 when 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 break-even 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. 2 as a function of the space-tug specific impulse. The analysis presented in Fig. 2 assumes a total $\Delta V = 4.178$ km/s for a transfer from LEO (185 km altitude, $i = 28.7$ deg) 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 kilograms per kilowatt. 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 does nuclear fission, again at 100% efficiencies. Kammash and Lee [17] 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 use 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 used [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.

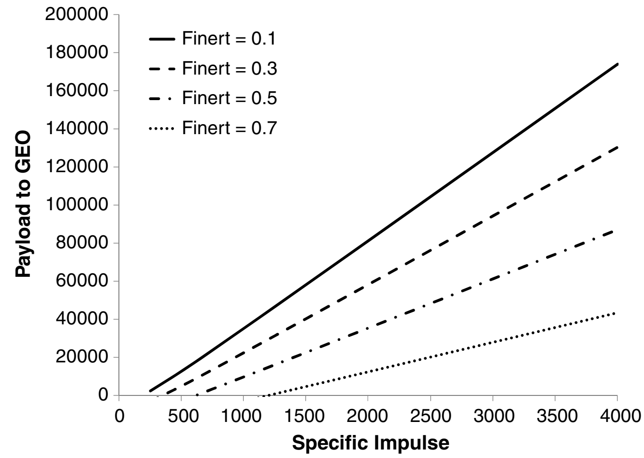


Fig. 2 Payload mass versus specific impulse for a notional nuclear space tug as a function of inert mass fraction (F_{inert}).

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. One advantage to either microwave or laser beamed-energy propulsion concepts is that the power source, and thus a large mass, remains on the ground and is not integrated onto the launch vehicle. 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 nonspace applications. The development of these systems for launch can certainly be leveraged from programs such as the airborne laser 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 in which they are attenuated by atmospheric species. Although atmospheric windows exist in which the transmission of certain wavelengths is highly efficient, beam attenuation through kilometers of atmosphere is always a concern. Beam scattering from atmospheric constituents and free-space losses will 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 [18] in 1965. Laser propulsion was first introduced by Kantrowitz [19] in 1972, although the concept of photon propulsion was first put forth in 1953 [20]. Since these pioneering papers, an extensive amount of research has been performed that has led to a wide variety of beamed-energy propulsion concepts (see, for

Table 1 Trade space for a nuclear space tug

System	Inert mass fraction of GTO stage	Specific impulse, s	Mass to GEO, kg	Notes
Delta IV Heavy, RL-10B upper stage		425 (RL-10B)	12,552	Assumes $\Delta V = 4.178$ km/sec for transfer from 185-km-alt parking orbit at $i = 28.7$ deg to GEO orbit.
Thermal nuclear tug	0.7	1000 (NTP)	<0	$F_{\mathrm{inert}} = 0.7$ from [1], hydrogen propellant
Thermal nuclear tug	0.455	1000 (NTP)	12,552	Break even with two Delta IV Heavy launches, hydrogen propellant
Electric nuclear tug	0.699	2000 (NEP)	12,552	Break even with two Delta IV Heavy launches
Electric nuclear tug	0.754	2500 (NEP)	12,552	Break even with two Delta IV Heavy launches
Thermal nuclear tug	0.3	1000 (NTP)	22,393	Possible F_{inert}, hydrogen propellant
Electric nuclear tug	0.45	2500 (NEP)	55,226	Possible F_{inert}

example, [21]). In the past several decades, advances in laser and microwave power generation have been significant, 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 that 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 [22,23] 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. The concept by Bae [22,23] improves the efficiency by increasing the photon flux by additionally amplifying the laser pulse in an intervening optical cavity. In the photonic laser thruster concept, a thrust amplification to Eq. (2) of up to 3000 times has been demonstrated [23] 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.

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 [24]. The power density required to ionize a typical working gas is in the range of 5×10^{14} and 10^{15} W/m^2, once again emphasizing the need for high-power laser systems [25].

The laser lightcraft concept, experimentally demonstrated in [26], is envisioned to be a multistage system with the first stage driven by an airbreathing aerospike, using 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 concepts. In either configuration, the main idea is to focus the laser beam within the lightcraft geometry to break down the ambient air, thus forming an air plasma. As the vehicle ascends, the air density decreases to the point at which stored propellant is necessary. 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 onboard, a kind of laser-heated resistojet is envisioned. Kare [27] described a vehicle that 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. To minimize the cost of high-power laser development, a module-laser approach has been investigated. A noncoherent array of laser diodes could potentially be used instead of a single high-power ($\sim$100 MW) coherent laser. The array of diode lasers has the advantage of being relatively compact, efficient, and scalable. In the study by Kare [27], 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 tens 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 placed into two categories that include heat-exchanger or propellant-heating and plasma-formation options. Since the 1930s, microwave source development has seen exponential increases in Pf^2 (power times frequency^2) [28]. Current estimates suggest that an array of 300 gyrotron sources operating at 140 GHz and 1 MW power levels is sufficient to place a 1000 kg satellite into orbit [29]. 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 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. [30] 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. Atmospheric windows exist between 1–40, 130–160, and 200–300 GHz. A optimization of frequency is required for these systems that 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. [31] 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 upward of 1000 MW of output power would be necessary to produce the thrust required for a launch vehicle.

Parkin [32] described 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 [27] 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 s. 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 [29] 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. 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 [33]. Using the notional laser and microwave heat-exchanger concepts, Kare and Parkin [33] 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, 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, in which 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 [31].

C. Advanced Chemical (High-Energy-Density Materials)

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 with minimal molecular weight of the products. 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 s.

However, the maximum ΔV 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 are also 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 [34] has chronicled propellant development efforts in the United States. If all else is held constant, in order to achieve higher specific impulse with higher propellant density, the chamber temperature must be increased. The relationship between specific impulse, temperature, and exhaust species molecular weight is shown in Fig. 3. Currently, materials considerations limit chamber temperature to about 3900 K 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 [35]. These hydrocarbon fuels are isomers of existing hydrocarbons with novel arrangements of the atoms and increased bond energies as a result. Another class of propellants that has been considered for high-energy-density applications is polynitrogen [36,37]. 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.

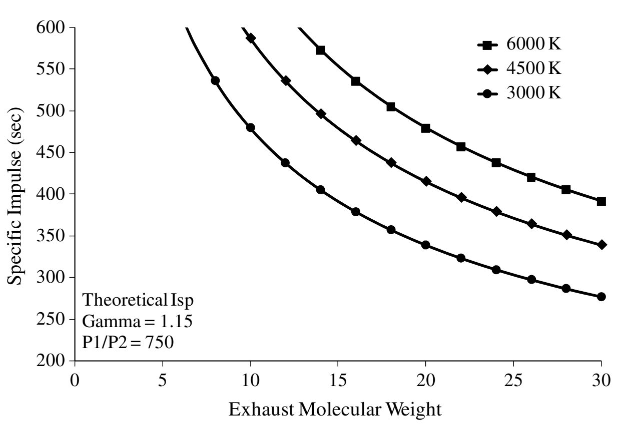


Fig. 3 Specific impulse as a function of temperature and exhaust molecular weight.

A conceptual study by Cole et al. [38] 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 s 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 s. 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 that 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. Although 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. Alternative Airbreathing Engines

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 liftoff weight of approximately 2×10^6 kg. Approximately 6.2×10^5 kg (30%) of that is liquid-oxygen oxidizer, whereas only 24,400 kg (1.2%) is considered useful payload [39]. Significant potential, therefore, exists for airbreathing 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. Airbreathing launch concepts are typically designed to be reusable to offset the added financial cost of the additional complexity of the airbreathing engine over a traditional rocket engine.

Airbreathing engines show significant specific-impulse advantages over traditional rocket engines over specific ranges of vehicle Mach number. Figure 4 illustrates the potential specific impulse available from both airbreathing technologies and rockets. Rockets consistently provide a significantly lower specific impulse over the entire range in Mach numbers than the optimal airbreathing technology with the same fuel, but no single airbreathing technology can operate over the entire range of Mach numbers required to reach orbit (Mach = 0–25). This naturally leads to proposed airbreathing-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 [40]. The

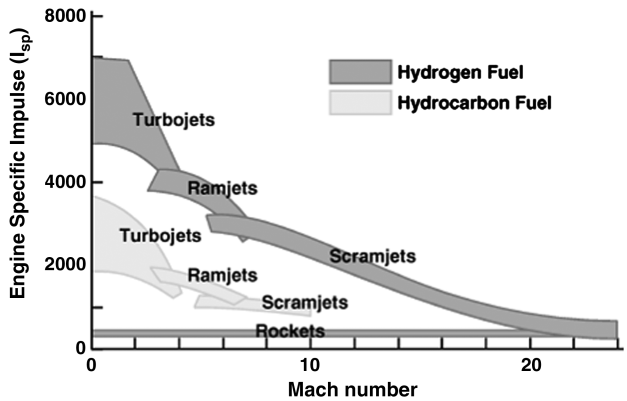


Fig. 4 Airbreathing and rocket performance verses Mach number.

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.

1. Pulsed-Detonation Engines

In addition to ramjets and scramjets, there are alternative technologies that may be incorporated into airbreathing 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 and deflagrations operate as a constant pressure cycle, so, theoretically at least, detonations can more efficiently convert the stored chemical energy to propulsive energy [41]. Practically, however, the PDEs that have been experimentally tested have not yet yielded the predicted performance improvements over similar ramjets [42]. Other potential advantages for PDEs include the ability to produce static thrust, they can be made with very few moving parts, and they can potentially operate on multiple fuel/oxidizer combinations. With the ability to operate with either ambient air or stored oxygen, PDEs can theoretically operate from the ground to hypersonic speeds with a single stage. PDEs are not commercially available, but the first flight of an aircraft powered by a PDE occurred on 31 January 2008 [43].

2. Rocket-Based Combined Cycle

RBCC propulsion systems are one class of combined-cycle propulsion systems 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 four 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 investigated. Daines and Segal [44] gave a review of the technology and critical technological issues associated with RBCCs. Tang and Chase [45] gave a more recent review of the history and current status of relevant airbreathing hypersonic flight. It is oftentimes 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 [46]. Analysis has also shown that direct-ascent trajectories are not feasible with RBCCs because of their low thrust-to-weight ratio [47]. The scramjet portion of the RBCC is the one cycle that is not currently technologically available, but it is the focus of demonstration programs like the X-51A [48] and the HyCAUSE program [49]. One major challenge of an RBCC is transitioning between the different engine cycles in a predictable and controlled way and without any flameouts or other faults. Dependable hypersonic airbreathing flight is still the critical limiting component that must be demonstrated before RBCCs can be truly evaluated.

3. Turbine-Based Combined Cycle

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 airbreathing 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. Partially counteracting this potential performance benefit is the additional mass of the turbine that must be carried even when the vehicle is operating in the scramjet mode. 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 [50]. Accurate cost estimates will not be available, however, until the required scramjet technology is available.

IV. Propellantless Advanced Launch Concepts

A. Electromagnetic (Rail and Coil)

Over the last couple of decades, there has been interest in electromagnetic rail systems for various applications. The U.S. Army has investigated weaponized systems for future combat vehicles with greater lethality than the 70 t Abrams but with a weight less than 20 t. The U.S. Navy has also shown interest in this technology for long-range shore bombardment. Current 5 in. guns have muzzle energy of 11 MJ. It is estimated that similar-sized electromagnetic (EM) railguns would be capable of operating at 20 MJ muzzle energy and could achieve 300 to 800 km range based on a 2500 m/s velocity [51]. The U.S. 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 with 20,000/kg for the space shuttle, if the required launch rate can be achieved [52].

There have been several EM rail systems conceived for launch to LEO, each with their inherent technical problems that would need to be overcome. Because of the fact that the density (which dominates drag losses) at 50,000 ft is reduced to about 87% of the value at sea level, it has been conceived to mount an EM rail system in a large aircraft. This would reduce the aerothermal loading; however, it places limits on payload size and would require even larger acceleration because of the inherently short track length. In addition, issues arise such as the placement of the gun and pulsed-power equipment within the airframe. Research would also need to be done to minimize launch effects (torque and recoil) on the aircraft [53]. Another option is a ground launch facility that would use longer tracks to accelerate larger payloads. Although the acceleration experienced during launch would still be large (greater than 1000 g), 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. Because of 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 8 MJ [54]. Flight-test firing of a 2 kg projectile to an altitude of 120 km was performed [55]. Laboratory rail systems have achieved very high exit velocities, accelerating a small 7 g launch package to 7 m/s [53], showing that there is no fundamental barrier to achieve the require muzzle velocity.

Currently, there are four critical issues that need to be addressed to see the success of electromagnetic rail launch. First, velocities greater than the currently demonstrated 7 km/s have to be achieved with acceptable acceleration limits and payload size. Second, the development of a pulsed-power system that can deliver the necessary megaamperes of current and appropriate power to the track. Third, aerothermal loads on the projectile during its transatmospheric flight need to be addressed. Finally, nanosatellite technology must advance in order to withstand the high levels of acceleration [53].

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 [56]. This ablated plasma and gas leads to drag and unwanted secondary arcing. Another major loss mechanism in EM railguns is friction-related stresses and strains on the rail and projectile, causing both a negative acceleration force and fatigue on components. High aerothermal 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 mechanisms is plasma venting [57].

The amount of energy needed to be stored for an EM rail launch system is tremendous. To launch a modest 1250 kg package would require muzzle energy of 35 GJ. Assuming an 80% energy conversation, an input energy of 44 GJ would be required. It has been suggested that these energy levels could be supported using high-speed rotating electrical generators [58]. Even a very light payload of 10 kg would require 250 MJ, which is comparable to one of the largest energy-storage facilities at Sandia National Laboratories. To date, laboratory rail guns have only demonstrated muzzle energies of about 9 MJ.

Launching a payload to LEO using a railgun requires velocities of approximately 10.6 km/s, including losses [59]. Because of 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 needed, adding to the overall mass of the launch vehicle. Studies have shown [60] 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, accelerations of several thousand times that of gravity are required [57]. Aircraft launch concepts that use much shorter tracks can experience 50,000 g 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. To accomplish a mass deposition to LEO equivalent to that of current rocket systems, a large number of launches would be necessary. To achieve the 500 t a year to orbit, conceived electrometric systems would require an average of five launches a day [52]. This brings up many issues of launch infrastructure and logistics, space trafficking, and launch waste materials.

Coil guns use electromagnetic coils to accelerate a magnetic payload to high velocities. Coil guns do not require sliding contacts, indicating that they may have longer lifetimes than railguns. The current technology limitations of coil guns include high-voltage fast-acting switches and parasitic resistance/energy dissipation. Coil guns have higher predicted performance, but have slightly lower achieved performance due to limitations in available technology. Fundamentally, electromagnetic systems 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 5 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 significantly more than 1 GW of power.

As an example, assuming that 1 GW of power is available, it would be possible to launch roughly 600 kg at 3.4 km/s and a 30 deg incline, assuming suitable terrain could be found for the 6-km-long launcher. Although most payloads and rocket motors are not designed to withstand the resulting 100 g acceleration, projects (see the Gas Guns section) 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 of several kilometers/second 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.

B. Gas Guns

The idea of launching payloads using gun launch has been around since Verne [61] wrote about it in 1865. Morgan [62] compiled a good history of the development of gun launch systems. 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 gas-dynamic 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 3 km/s, which is sufficient for launch-assist, but is insufficient for direct-launch applications [62]. A representative of the state of the art in large gas-dynamic guns is the High Altitude Research Project (HARP) gun effort from the 1960s. The guns were funded to study hypersonic reentry, but were also an incremental step in the development of a gun launch system. The HARP 16 in. gun was assembled from surplus artillery tubes. The final version fired a 180 kg projectile at 3.6 km/s (5500 g) and it reached an altitude of 180 km. The next generation was to be a full launch system, with the cannon launching 1300 kg at 1.8 km/s, delivering a payload of 90 kg 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).

For a gun launch system to directly launch a payload, it must achieve muzzle velocities significantly above the roughly 3 km/s that a gas-dynamic 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) [62]. 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 4000 atm it bursts a disk and propels the projectile out

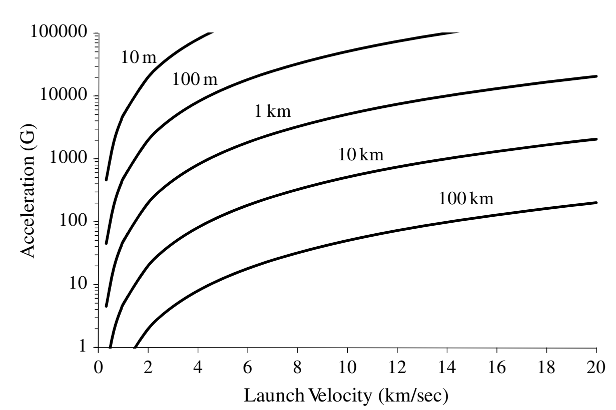


Fig. 5 Acceleration as a function of launch velocity for various track lengths.

the end of the gun. The SHARP gun demonstrated launching a 5 kg projectile at 3 km/s, but the next phase of the project to increase the velocity and angle the gun upward was not funded. Both gas-dynamic guns and light-gas guns have demonstrated impressive performance and are candidates for launch-assist systems.

C. Elevator

The idea of a space platform that reached from the ground to geosynchronous orbit (GEO) was first put forward in 1895. As currently envisioned, this concept would allow a payload to be placed into orbit without the need for traditional rockets. Modern versions of the space-elevator concept include a cable that traverses from the ground to GEO, a massive counterweight above GEO, climbers that 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 upward, thus achieving a desired orbital altitude [63].

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 [63]. 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 [64]. 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 [65]. Further development at potentially high cost is required to produce CNTs applicable to the space-elevator concept.

Several problems with the space-elevator concept have been identified, including the need for a massive counterweight, 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. In addition to 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.

D. 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 that is 629 m high [63]. Building structures significantly taller than this is not only technically challenging but also expensive. Bolonkin [66] suggested that inflatable towers could be constructed using lighter-than-air gases for factor-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 required to reach orbital velocities in LEO is kinetic. 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.

E. Balloon Launch

Envisioned balloon launch systems are simple low-performance systems [67–69]. The launch systems typically have no control during balloon ascent. Balloon-assisted sounding rocket launch of 18 kg payloads to altitude of up to 100 km was demonstrated by the Rockoon flights in the 1950s. 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 from NASA [70]. The balloons have envelope volumes up to and exceeding 500, 000 m^3 and are capable of delivering a payload of 2721 kg to an altitude of 33.5 km 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.

F. 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 [71], gravity shielding [72], and even warp drives [73] 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/s at the equator) to orbital velocities ($\sim$7.8 km/s 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 [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 use.

Reducing the gravitational mass of a launch vehicle would lead to a direct ΔV reduction, meaning that lower propellant mass would be required to reach orbit. For a 100 km LEO satellite, the ΔV reduction would be approximately 1.4 km/s if the gravitational mass of the vehicle were reduced to zero. However, a ΔV of 7.5 km/s 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/s to about 3 km/s, 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.

Another advanced concept that could be employed for space access is antimatter. Antimatter converts all of its mass to energy during its annihilation with normal matter. Antiproton annihilation has been suggested by Forward [74] as a means of propulsion. During annihilation, antiprotons convert nearly two-thirds of their energy into charged particles that can be harnessed to produce thrust. Antimatter is a highly concentrated means of energy storage with a specific energy density of 9×10^{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 [75]. In addition, 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 [75].

V. 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 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.

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):

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

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/s of the velocity budget [11]. The burnout velocity increment is dependent on the launch-site properties and the final orbit, but is typically between 7.5 km/s (LEO) and 11 km/s (GEO).

Figure 6 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 30 MJ/kg for LEO orbits and 60 MJ/kg for GEO orbits (neglecting losses). In LEO orbits the total mechanical energy is predominantly kinetic energy. At an altitude of approximately 3200 km 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.

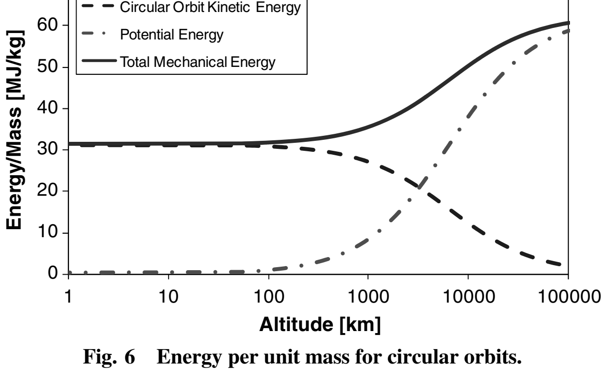


Fig. 6 Energy per unit mass for circular orbits.

Figure 7 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 8.85 km. 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 0.087 MJ/kg, or much less than 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 12.2 km, which yields an increased potential energy of 0.12 MJ/kg over launch at sea level, or only about 0.4% of the total required mechanical energy. Similarly, even near-space dirigibles at 20 km altitude would only increase the potential energy by 0.2 MJ/kg (0.7%). 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.

The situation is similar when considering kinetic energy. Using the rotation of the Earth's surface can yield approximately 0.1 MJ/kg (less than 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 are a small component of the total energy required to reach space.

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. It is a challenge for launch-assist technologies to provide a sufficient fraction of the total required ΔV to warrant their added complexity.

VI. Discussion

A brief overview of the concepts considered in this study is shown in Table 2. The last column of Table 2 indicates whether a proposed system (small-scale or full mockup) has been demonstrated. Essentially, this would indicate a TRL of 5 based on the NASA TRL definitions. By its very 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,

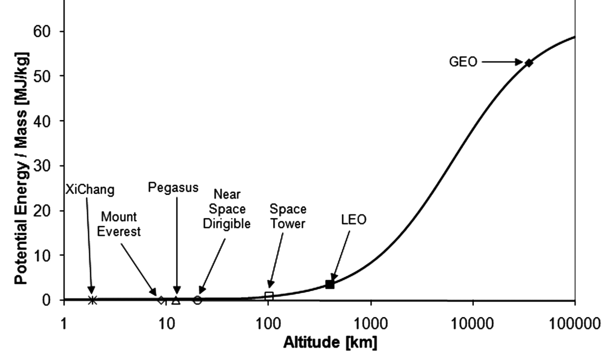


Fig. 7 Potential energy per unit mass for various launch-assist technologies.

Table 2 Concept summary

Concept	Propellant used (Y/N)	Working fluid	Energy source	Specific impulse, s	Demonstrated at system-level TRL 5 (Y/N)
Nuclear	Y	Hydrogen	Onboard fission reactor	1000–5000 [14]	Y [14]
Beamed energy (laser)	Y	Hydrogen/ablated material/air	Beamed	600 (heat exchanger, Kare [27])	Y (some concepts) [26]
Beamed energy (microwave)	Y	Hydrogen	Beamed	1000 (heat exchanger, Parkin [32])	N
Advanced chemical	Y	Polynitrogen, metallic hydrogen	Chemical	900–1100 (Cole et al. [38])	N
Airbreathing engines	Y	Air/hydrocarbon or hydrogen	Chemical	850–1550	Y [43,48]
Electromagnetic rail	N	——	Electrical	N/A	Y, but not to orbital velocities [55]
Space elevator	N	——	Solar	N/A	N
Breakthrough physics ideas	Y and N	Various	Various	Various	N

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. Only 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 that 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 nanostructured 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.

VII. Conclusions

Of the concepts presented, nuclear fission systems may hold the most promise of fulfilling the need for improved rocket performance in the short term, assuming that considerable mass reductions can be achieved. This is based on the substantial amount of previous work performed on this concept, making it the highest TRL, purely rocket concept presented in this study. That is not to say that there are not major challenges ahead for this concept, but the past development in this concept and relatively high TRL make it a strong candidate. 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. Airbreathing combined-cycle systems also appear to hold more immediate promise for launch system applications, since the basic technology underlying the concepts are relatively mature (high TRL). 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 onboard propellant. Currently, 85–90% of a typical launch vehicle is propellant, indicating the major benefit of these types of concepts. However, like nearly all advanced concepts, beamed-energy propulsion has yet to be successfully demonstrated. 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 propulsive efficiency of chemical-launch vehicles is extremely high. There is limited room for improvement, and further improvement will come at greater 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.

Acknowledgments

This work was supported by the Propulsion Directorate of the U.S. Air Force Research Laboratory. The authors wish to thank Roy Hilton and Ingrid Wysong for their support of this effort. The authors also acknowledge the support of Kassandra Merritt on the analysis of electromagnetic systems.

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R. Braun
Associate Editor