

Ram Accelerator as an Impulsive Space Launcher: Assessment of Technical Risks

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Abstract

The ram accelerator is a chemically powered hypervelocity mass driver that operates with in-tube propulsive cycles similar to airbreathing ramjets and scramjets. The launcher consists of a long tube filled with a pressurized gaseous fuel-oxidizer mixture in which a subcaliber projectile having the shape similar to that of a ramjet centerbody is accelerated. No propellants for this launch process are carried aboard the projectile; it effectively flies through its own propellant “tank”. The ram accelerator at the University of Washington has been operated at velocities up to nearly 3 km/s and in-tube Mach numbers greater than 7 in methane-based propellant mixtures. This Mach number capability corresponds to muzzle velocities greater than 7 km/s when using fuel-rich hydrogen-oxygen propellant. The combination of hypervelocity muzzle velocities and the ram accelerator’s inherent scalability to multi-ton payload sizes makes it suitable for direct space launch. Technical issues associated with the implementation of the ram accelerator technology for direct space launch applications are presented here.

Introduction

The ram accelerator is a projectile launcher conceived in 1983 at the University of Washington that uses chemical energy to accelerate projectiles to hypersonic speeds (Hertzberg *et al.* 1988). Although it resembles a conventional long-barreled cannon, the principle of operation of the ram accelerator is notably different, being closely related to that of a supersonic airbreathing ramjet engine. This device consists of a stationary tube, analogous to the cylindrical outer cowling of a ramjet engine (Fig. 1), filled with combustible gaseous mixtures (typically methane or hydrogen, oxygen, and diluents such as nitrogen, helium, or excess hydrogen) at fill pressures ranging from 5 to 200 atm. Frangible diaphragms close off each end of the tube to contain the propellant. The projectile is similar in shape to the centerbody of a ramjet and has a diameter smaller than the bore of the accelerator tube.

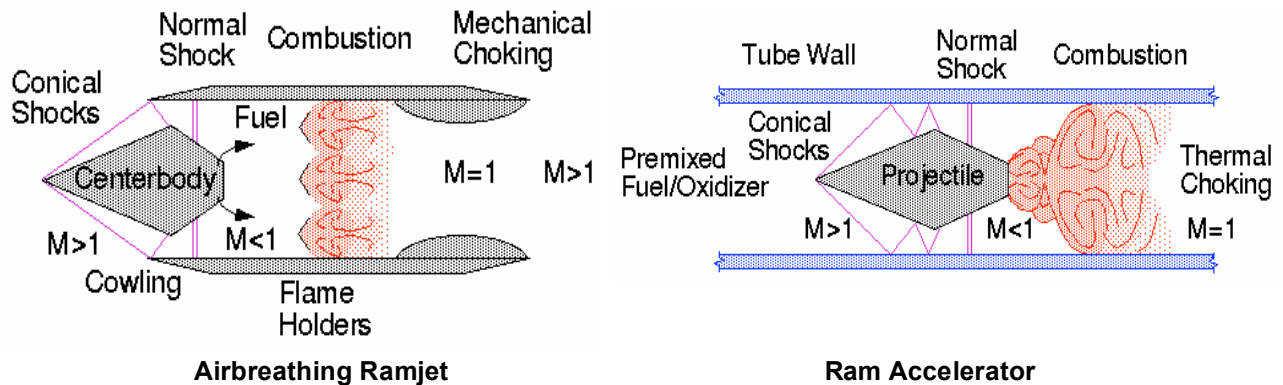


Fig. 1 Comparison of airbreathing ramjet with ram accelerator.

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The operational sequence of the ram accelerator (Fig. 2) is initiated by injecting the projectile into the ram accelerator tube at speeds greater than 500 m/sec by means of a conventional powder gun or light gas gun. A tube-sealing obturator is used with the subcaliber projectiles to prevent gun gas blowby. The acceleration of the projectile from rest compresses residual air in the gun's launch tube via a series of reflected shock waves (Stewart *et al.* 1998). When the projectile punctures the entrance diaphragm, the slug of shock-heated air ignites the propellant near the base of the projectile, and the obturator is rapidly decelerated and left behind. A stable combustion zone is thus formed which travels with the projectile, maintaining a high pressure on the projectile base that smoothly propels it forward, in a manner analogous to a surfer riding a breaking ocean wave (see Fig. 3). To keep the subcaliber projectile centered in the tube, the projectile is either fabricated with fins that span the bore or else the tube is equipped with several internal guide rails that guide the axisymmetric projectile (Fig. 4).

What distinguishes the ram accelerator from gun-like devices is that its source of energy (the combustible gas mixture) is uniformly distributed throughout the entire length of the launch tube, whereas in a gun the energy source is concentrated at the breech as either a charge of gunpowder or high pressure gas. During the ram acceleration process the highest pressure in the launch system is always at the base of projectile, rather than at the breech as in a gun (Fig. 3). Another key difference is that, unlike a gun, the bulk of the ram accelerator combustion products have relatively little velocity with respect to the launch tube, thus barrel erosion is significantly

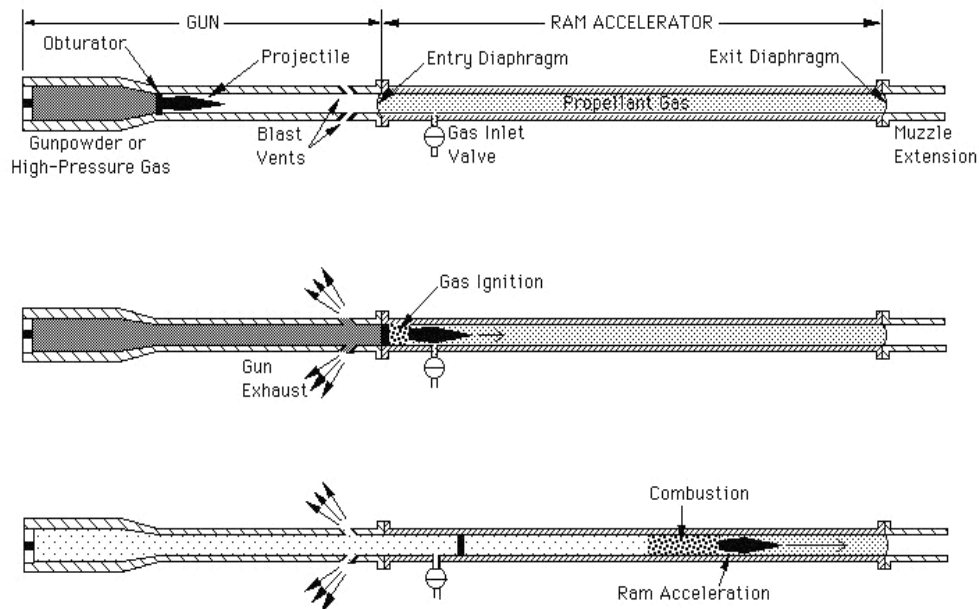


Fig. 2 Operational sequence of ram accelerator.

- (a) Gun is loaded with projectile and obturator, and charge of gunpowder or high pressure gas. Ram accelerator is loaded with combustible gas mixture at 5–200 atm. pressure.
- (b) Gun fires obturator/projectile combination into ram accelerator.
- (c) Combustion is initiated and propels with projectile, sustaining traveling pressure wave that accelerates projectile to high velocity.

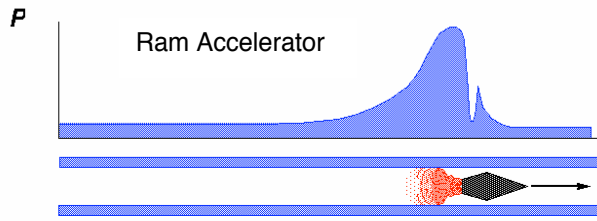
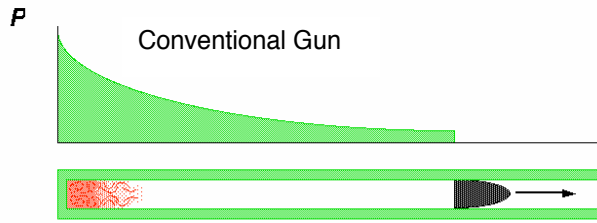


Fig. 3 Pressure profile of conventional gun and ram accelerator launch cycles.

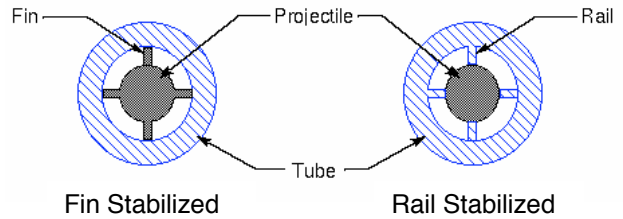


Fig. 4 Methods for centering subcaliber projectiles in launch tube.

reduced. Only a small volume of high pressure gas exits the tube with the projectile, resulting in minimal muzzle blast and recoil. Since the launch characteristics of the ram accelerator are independent of the initial conditions within the breech of a gun, the projectile acceleration and muzzle velocity can be readily tailored to specific needs by adjusting the propellant composition and fill pressure in the launch tube.

Ram accelerator propulsive cycles are distinguished by the velocity regime in which they operate (Hertzberg *et al.* 1991). In the subdetonative velocity regime; i.e., below the propellant's Chapman-Jouguet (CJ) detonation speed, the propulsive cycle behaves as if the flow were thermally choked behind the projectile, as shown in Fig. 5a. In this case the volume expansion of the rapidly combusting propellant maintains a shock train on the aft-body of the projectile and establishes a region of thermal choking at full tube area behind projectile. The thrust coefficient (i.e., ratio of thrust to the product of fill pressure and tube area) for this propulsive cycle is relatively insensitive to projectile geometry (Bruckner *et al.* 1991) and is plotted versus in-tube Mach number in Fig. 6. Maximum thrust typically occurs near Mach 3 and decreases to zero at the propellant CJ detonation speed. Gaseous propellants exist that have CJ detonation speeds of up to ~ 4 km/s, thus this represents the upper velocity limit of thermally choked ram accelerator operation.

In the superdetonative velocity regime, i.e., above the propellant CJ detonation speed, the combustion process occurs at supersonic velocity completely within the confines of the annular region between the projectile and tube wall, as shown in Fig. 5b. The thrust coefficient versus Mach number characteristics of this propulsive mode for the ideal case of no drag and when viscous drag is included are shown in Fig. 6 (Higgins 2006). The net thrust decreases with increasing Mach number until the thrust equals drag, at which point the projectile continues to cruise at constant velocity. The Mach number at which the thrust-equals-drag condition is reached depends strongly on the projectile geometry and propellant composition. Conservative

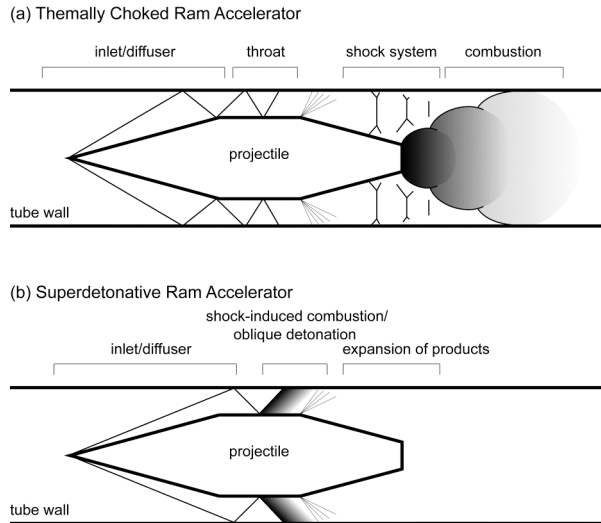


Fig. 5 Thermally choked and superdetonative ram accelerator propulsive modes.

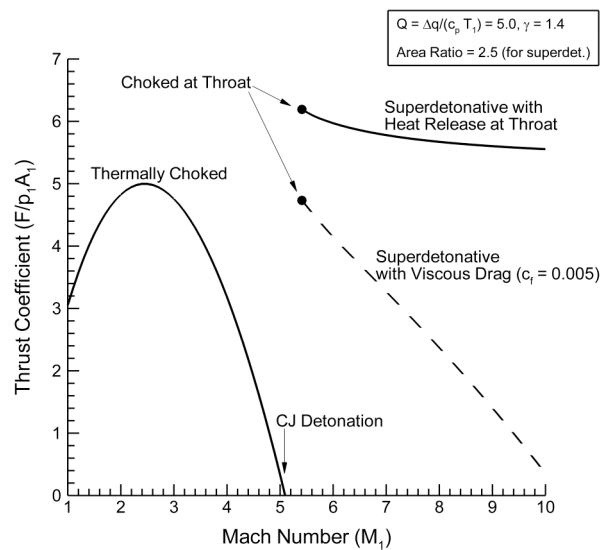


Fig. 6 Theoretical thrust-Mach profiles of ram accelerator propulsive modes.

estimates of performance limits of this propulsion mode indicate that operation at greater than Mach 8 is feasible in hydrogen-oxygen based propellants. This would readily lead to muzzle velocities in excess of 7 km/s.

Experimental Results

Thermally choked ram accelerator operation has been experimentally demonstrated in the velocity range of 0.7 to 2.7 km/s with launcher bores ranging from 25 to 120 mm. Ram accelerator operation over this velocity range is achieved by partitioning the launch tube into separate sections, or stages, each filled with a different propellant mixture, as shown in Fig. 7. By selecting the sequence of propellants in such a manner that their speed of sound and detonation speed increase towards the exit of the ram accelerator, the projectile Mach number can be kept within limits that maximize thrust and efficiency, resulting in high average acceleration and a higher final velocity than is achievable with a single propellant stage.

Experiments in the 16-m-long, 38-mm-bore ram accelerator facility at the University of Washington have demonstrated that this multi-stage approach is practical and effective for accelerating projectiles to hypersonic velocities, as indicated by the velocity-distance data in Fig. 8. Note that the experimental data correlate very well with the velocity profiles predicted by the theoretical model of the thermally choked propulsive mode (Bruckner *et al.* 1991). In the single-stage experiment shown in Fig. 8, the projectile actually accelerates up to and beyond the propellant CJ detonation speed. Positive thrust at the CJ speed arises from the combustion moving up onto the projectile body and the cessation of thermal choking. Ultimately the combustion moves completely onto the body and the projectile is ram-accelerated in the superdetonative propulsive mode until it reaches the thrust-equals-drag state. Having the projectile make a transition to another propellant before this limit is reached enables multi-stage superdetonative ram accelerator operation to accelerate payloads to velocities suitable for direct space launch applications. Experimental investigations carried out by Hertzberg *et al.* (1991),

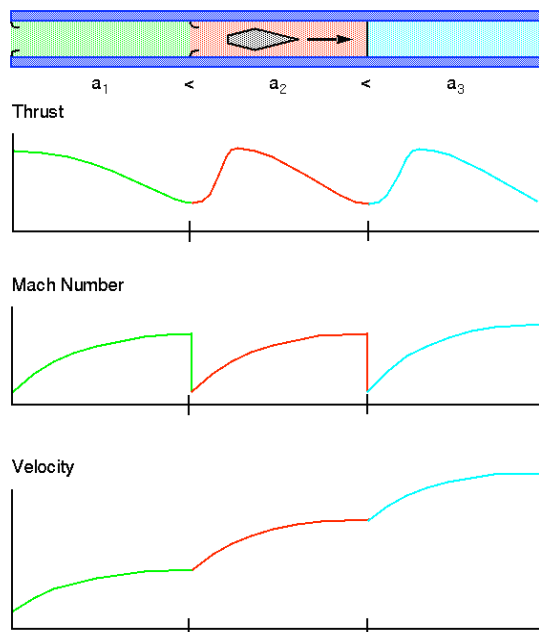


Fig. 7 Multi-staging ram accelerator to maintain high average acceleration and reach hypervelocity.

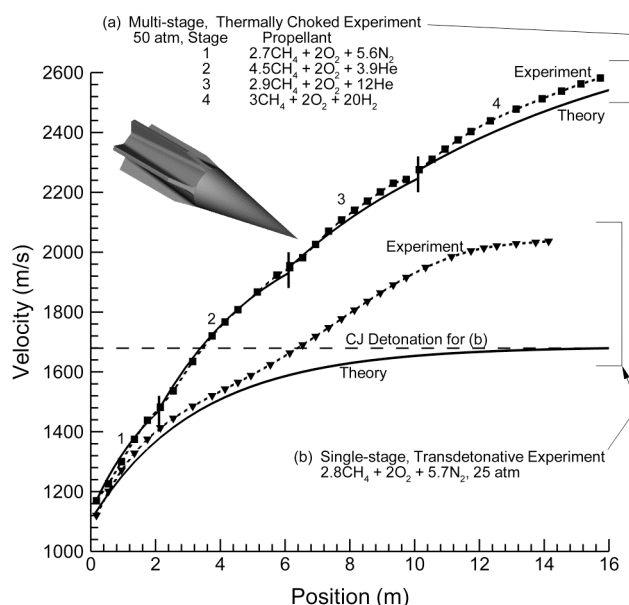


Fig. 8 Multi-stage and single-stage results of ram accelerator experiments.

Seiler, *et al.* (1998), and Knowlen *et al.* (1996, 2007) have demonstrated superdetonative ram accelerator operation in the velocity and Mach number ranges of 1.6 to 2.5 km/s and 6 to 7, respectively.

The most effective and efficient ram accelerator propulsive cycle in the velocity range from 0.7 to 3 km/s is the thermally choked mode (Hertzberg *et al.* 1988). This propulsive cycle operates at the lowest Mach numbers ($2.5 < M < 4.5$) and has a low peak cycle pressure. Subdetonative ram accelerator operation at velocities greater than 3 km/s, however, has not yet been demonstrated. Above 3 km/s, superdetonative ram accelerator performance becomes competitive and is certainly essential above 4 km/s. In order to reach Earth orbit velocities, ram accelerator launchers must operate with a superdetonative propulsive cycle at in-tube Mach numbers ranging from around 6 to 10. It is essential for more laboratory experiments to be carried out in this Mach range and, ultimately, at velocities of 6–8 km/s to realize the full potential of the ram accelerator for direct space launch applications.

Ram Accelerator Space Launch System Parameters

The main components of a ram accelerator space launcher system are an initial launch gun, thermally choked ram accelerator section, and a superdetonative ram accelerator section (Bruckner and Hertzberg 1987, Knowlen and Bruckner 2001). An initial launcher is required to accelerate the projectile from rest up to the minimum entrance velocity needed to enable thermally choked ram accelerator operation, which can be as low as 0.5 km/s for $\text{CH}_4/\text{O}_2/\text{CO}_2$ propellant. Combustion-driven gas guns such as that used for the pump-tube driver stage of the SHARP facility could be employed for this purpose (Scott 1996). The basic design considerations for a full-scale direct space launch ram accelerator facility capable of launching a

2000 kg projectile to 6–8 km/s with a peak acceleration of 1500 g are presented. The tube bore diameter varies from 1.13 m in the subdetonative stages to 1.0 m in the superdetonative stages, as discussed below. The tube walls are assumed to be fabricated from steel alloy having a yield stress of 1200 MPa and are thick enough to provide a safety factor of at least 2 based on the theoretical peak cycle pressure. The parameters for this launch tube are then used as a basis to scale smaller launch facilities and evaluate costs.

An ideal ram accelerator would have a propellant composition gradient that enables operation at nearly constant Mach number so that the projectile would experience minimal variations in acceleration. Although it is possible to reliably introduce mixture gradients in tubes, it will be more practical to use adjacent stages of various lengths having different propellants to control the in-tube Mach number and acceleration history of the projectile. For the system proposed here, the propellants (all at the same pressure) are separated from each other with isolation valves (i.e., ball or gate valves) that can be opened just prior to firing. This approach eliminates the need for the projectile to penetrate diaphragms between stages. A diaphragm at the entrance to the thermally choked section is still required; however, it has negligible impact on the projectile when it is penetrated at relatively low entrance velocity. Alternatively, the entrance diaphragm can be burst before the projectile reaches it to effect “low velocity starting”; i.e., initiate ram accelerator operation at velocities around 300–500 m/s (Knowlen *et al.* 2000). At the exit, a pre-scored diaphragm that can be explosively burst just prior to the arrival of the projectile is preferred because it is unlikely that a large-scale valve could be opened quickly enough to keep from losing a substantial fraction of the final propellant.

A large bore ram accelerator space launch system, such as the example presented here, would likely use internal rails to support axisymmetric projectiles, as shown in Fig. 4. This allows the tube cross sectional area to be different in the thermally choked and superdetonative ram accelerator sections. Theoretical modeling of the acceleration performance of both ram accelerator propulsive modes has proven to be quite good for experiments carried out to date. Thus these theoretical performance models were used to estimate the size-scale and wall thickness of the direct space launch facility. Detailed discussions of the numerical procedures and theoretical underpinnings of the ram accelerator performance modeling can be found in Bruckner *et al.* (1991), Knowlen *et al.* (1996, 1998) and Bundy *et al.* (2004).

The projectile for the 1-m-bore launcher is a cone-cylinder-cone configuration, as shown in Fig. 5, with nose and rear cone half angles of 15° and 20°, respectively. The cylindrical midbody has a diameter of 0.85 m and length 2.70 m. To enhance the flame holding capability of the projectile when operating in the thermally choked mode, the rear cone is truncated to give a 0.5 m-diameter base. For a net mass of 2000 kg, the density of this projectile is 1000 kg/m³ and its geometry is compatible with ram accelerator operation in all of its propulsive modes. The outer structure is used only as an aeroshell during the launch process and then discarded at the muzzle exit to release an aerodynamically stable atmospheric transit projectile.

The velocity-distance and acceleration-distance profiles of a projectile being accelerated from 0.7 to 6 km/s in the ram accelerator section of the proposed direct space launch system are shown in Fig. 9 (Knowlen *et al.* 2001). The first seven ram accelerator stages are designed for thermally choked operation in the velocity range of 0.7 to 3 km/s and have a bore of 1.13 m to provide a projectile-to-tube inner diameter ratio of 0.75. The propellant, stage length, entrance

and exit velocities (V_{in} and V_{out}), and average acceleration for each stage are listed in Table 1. The entrance and exit Mach numbers of each stage are typically 3 and 3.7, respectively, which results in accelerations of 1200 ± 300 g. The peak cycle pressure under these conditions is 100 MPa, which dictates that the tube wall thickness be at least 10 cm to maintain the desired safety factor. The total length of the thermally choked ram accelerator section is 362 m, which results in a tube mass of 1120 metric Tons (rail mass not included).

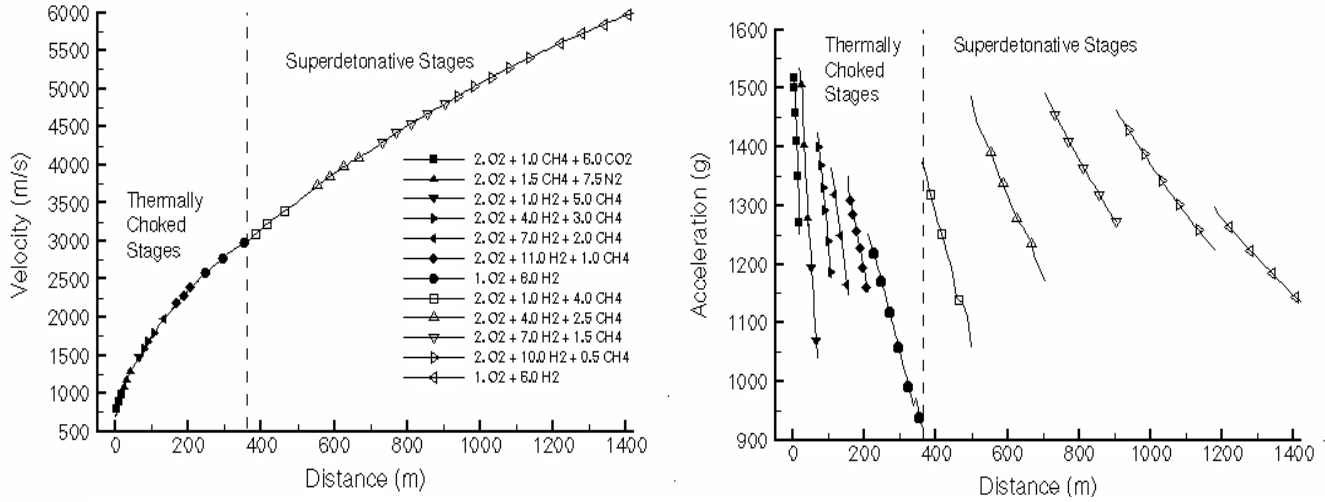


Fig. 9 Velocity-distance and acceleration-distance profiles of ram accelerator sized to launch 2000 kg, 0.85-m-diameter projectiles at 6 km/s.

TABLE 1: Parameters for Ram Accelerator System For Launching an 85 cm diameter, 2000 kg Projectile.

Launch Section Stages	I.D. / O.D. (m)	Length (m)	V_{in} (km/s)	V_{out} (km/s)	Acceleration Average (g)
Thermally Choked					
1.0CH ₄ + 2O ₂ + 6.0CO ₂	1.13 / 1.33	18.3	0.70	1.00	1420
1.5CH ₄ + 2O ₂ + 7.5N ₂	(all)	24.9	1.00	1.30	1410
5.0CH ₄ + 2O ₂ + 1.0H ₂		24.6	1.30	1.50	1160
3.0CH ₄ + 2O ₂ + 4.0H ₂		38.6	1.50	1.80	1310
2.0CH ₄ + 2O ₂ + 7.0H ₂		47.4	1.80	2.10	1260
1.0CH ₄ + 2O ₂ + 11 H ₂		55.3	2.10	2.40	1240
8.0 H ₂ + 1O ₂		152	2.40	3.00	1080
Superdetonative					
4.0CH ₄ + 2O ₂ + 1.0H ₂	1.00 / 1.56	135	3.00	3.50	1220
2.5CH ₄ + 2O ₂ + 4.0H ₂	(all)	206	3.50	4.20	1330
5.0CH ₄ + 2O ₂ + 7.0H ₂		200	4.20	4.80	1380
0.5CH ₄ + 2O ₂ + 10 H ₂		275	4.80	5.50	1340
6.0 H ₄ + 1O ₂		242	5.50	6.00	1210

6.0 H ₄ + 1O ₂		173	6.00	6.30	1090
7.0 H ₄ + 1O ₂		551	6.30	7.00	862
8.0 H ₄ + 1O ₂		1240	7.00	8.00	616

The remaining stages of the ram accelerator section are designed for operation in the superdetonative velocity regime. The bore of this section is 1.00 m, which results in a projectile-to-tube inner diameter ratio of 0.85. Five superdetonative stages are used to accelerate the projectile from 3 to 6 km/s (Fig. 9) in a distance of 1058 m (dashed line in Table 1). The propellants were chosen to limit the acceleration to 1500 g while operating in the Mach number range of $7 \leq M \leq 8$ for velocities up to 6 km/s. This resulted in an average acceleration of ~ 1300 g. To reach 7 or 8 km/s, extra stages are required (Table 1) in which the projectile operates up to Mach 9 or 10, respectively. The performance of the superdetonative propulsive mode decreases at velocities greater than 6 km/s because the benefits of reducing Mach number by adding more hydrogen are offset by the reduction in heat release due to the increased dilution of the fuel-rich propellant. Even though the average thrust decreases as the projectile accelerates beyond 6 km/s, reasonable length accelerator tubes are feasible for muzzle velocities up to 8 km/s. The peak cycle pressure is 250 MPa for the superdetonative propulsive mode, which results in a wall thickness of 0.28 m and launch tube masses of 9530, 16,000, and 27,200 metric Tons for muzzle velocities of 6, 7, and 8 km/s, respectively.

The total propellant mass used per ram accelerator launch to 8 km/s is ~ 20 times the mass of the projectile; e.g., ~ 40 metric Tons for a 2000 kg projectile. This is comparable to the propellant mass fraction of conventional rocket boosters for LEO payloads. Except for the first two low speed thermally choked stages in Table 1 (0.7 to 1.3 km/s), the only propellants used in the baseline ram accelerator system are methane, hydrogen and oxygen. All of these are readily available and have well established distribution systems. Remote launch sites may have LNG and LOX trucked in whereas the hydrogen may have to be produced on site by either steam-reforming methane or electrolyzing water. Cryogenic storage of the propellants offers several logistic advantages in addition to reduced storage volume requirements. Loading the launch tube with propellant can be carried out with conventional cryogen handling systems in a very rapid manner; i.e., in the time span of minutes, if necessary, by using liquid rocket turbo-machinery. The cryogens are loaded at sub-ambient temperature, which reduces the actual pressure during the filling process. Convective heating from the tube walls will ultimately bring the propellants up to ambient temperature and the desired fill pressure; however, this could take hours. Nevertheless, operation of the ram accelerator at sub-ambient temperature does not incur any penalties (thrust is actually proportional to propellant density, not fill pressure), and may even have some beneficial effects in reducing heat transfer at high Mach number. Thus, if necessary, launch rates of once per hour are certainly feasible using conventional cryogen transfer equipment. Operational issues of the ram accelerator under sub-ambient temperature conditions, however, have yet to be thoroughly investigated.

The baseline ram accelerator system parameters determined for the 2000 kg projectile are used as a basis for scaling down to smaller launchers with higher acceleration. Of particular interest is a system that has approximately twice the average acceleration, half the barrel length, and a reduced bore to enable the launch of 300 kg projectiles at 6 km/s. This is about the smallest scale that a direct space launch ram accelerator could be built that would still be able to deliver ~ 80 kg of payload to LEO. There are several potential markets of interest for payloads of this size, thus the infrastructure costs of such a system are indicative of the minimum required investment to build a direct space launch ram accelerator.

Launch Facility Cost Estimates

No one can say for sure how much money is required to build a ram accelerator facility of sufficient size for space launch; however, there are some comparable systems from which to draw estimates in a first-order effort to determine the upper bound for facility cost. Here, we will attempt to determine the cost of a baseline ram accelerator launcher system having a 500-mm-bore and length of 800 m that is capable of launching a 300 kg projectile at 6 km/s.

The upper cost boundary for a ram accelerator launch facility is represented by the SHARP/JVL light gas gun effort. Public data on SHARP is scant; nonetheless, a published estimate for a proposed 1520-meter-long JVL light gas gun cited a cost of \$298M (Gilreath *et al.* 1998, 1999). The basis for this number comes from an estimate provided by Morrison-Knudsen, the company that built the Alaskan Pipeline. Proportioning this cost to an 800-meter-long ram accelerator launch tube results in an estimate of \$157M. It can be argued, however, that the ram accelerator facility cost will be significantly lower than that of a light gas gun because it is inherently a much simpler device, as discussed below.

The cost estimate for the proposed 1520-meter light gas gun includes a total of 15 high pressure propellant storage tanks and heat exchangers. Each storage tank holds 18,900 cubic meters of hydrogen gas at 70 MPa, and a temperature of 1500 K. If spherical and made of high-strength titanium, each storage tank would be 33 meters in diameter with a wall thickness of 57 cm (almost 2 ft). The net mass of such a tank would be roughly 8000 metric Tons, which contains enough material to construct the entire length of a ram accelerator launcher capable of 6 km/s muzzle velocity. In addition to 15 of these high pressure tanks, the SHARP/JVL facility includes other complexities such as fast-opening/high-capacity valves at each high pressure storage tank and a hydrogen gas reclamation baffle at the muzzle of the light gas gun. Furthermore, the design and fabrication of the large-scale pump tube and acceleration reservoir for this system is a significant engineering challenge. For these and other reasons, we expect an 800-m-long by 500-mm-bore ram accelerator launcher to cost much less than \$157M.

Perhaps a better comparable cost estimate is the Alaskan pipeline itself, because it has many of the same building requirements as a ram accelerator—both are essentially large pipes. Each half mile (equivalent to 800 m) of Alaskan pipeline cost roughly \$16M in 2006 dollars, as compared to the \$157M suggested by the light gas gun. The ram accelerator will have more complexities associated with it than the Alaskan pipeline, such as high pressure gas feed systems (operating at 5 MPa or 725 psi, with the total launch tube volume just 1/5th of one light gas gun high pressure storage tank), the need for closure mechanisms to separate sections of different gas composition, potentially an active control system for tube sections to maintain precision alignment, as well as the fact the pipe sections will be thicker than the Alaskan pipeline. Because of this, the \$16M per 800 m of Alaskan Pipeline is considered the lower bound of scaling the ram accelerator launch system cost. The only firm conclusion that we can reach is that the true system cost is somewhere between \$16M and \$157M; however, this number is likely closer to the lower bound due to the vastly lower system complexity of ram accelerator compared to the SHARP/JVL light gas gun. The authors propose that a system cost in the range of \$40 to \$50M dollars is not unreasonable.

Launch Site Altitude Considerations

Integral to the performance characterizations of the ram accelerator as a direct space launch system is the location at which it is to be situated. There has been some discussion about the benefits of a high altitude launch site. In general, the higher the altitude of the launch site, the less velocity loss that occurs while traversing the lower atmosphere, as well as allowing for a thermal protection system of lower mass. Nevertheless, a high altitude site implies a site that is remote from civilization and all of its infrastructure benefits. Thus, the advantages of higher launch altitudes must be quantified in order to analyze optimal site location. To this end, a numerical model was constructed to analyze the ballistic flight path of a ram-accelerator-launched projectile. This model assumes a constant drag coefficient, a 22° launch angle, a standard 3-layer atmosphere, a spherical Earth, and an impulsive maneuver for the apogee kick. The numerical integration procedure was implemented in Octave, an open source Matlab clone. The influences of drag coefficient and muzzle altitude on the necessary orbit insertion ΔV determined from this model are shown in Fig. 10.

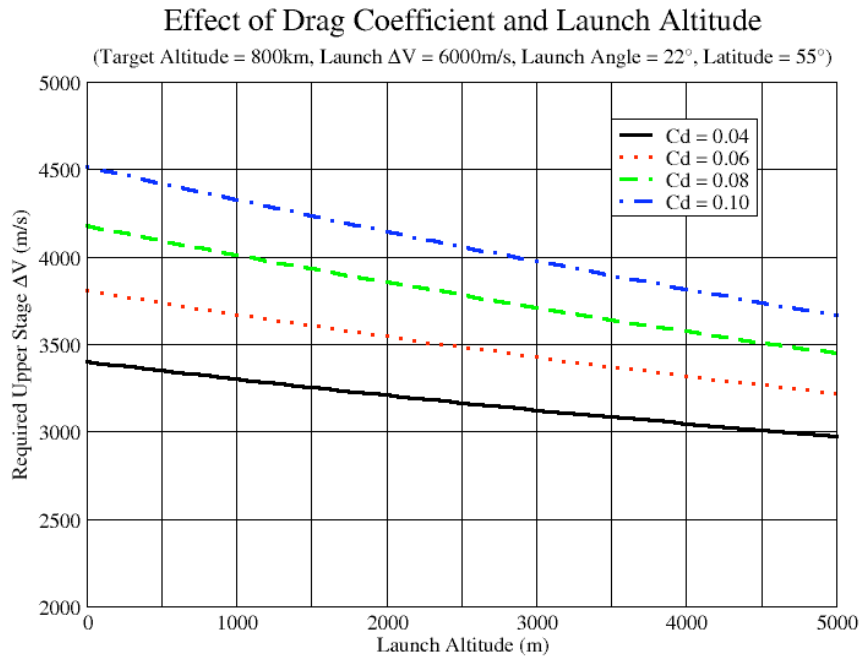


Fig. 10 Upper Stage Energy Requirements for Various Drag Coefficients

The main metric of performance that is used is the upper stage ΔV required to attain the target orbit because it determines the payload fraction of the total vehicle mass delivered to orbit. It is evident in Fig. 10 that the upper stage ΔV is heavily dependent on the drag coefficient. Realistic values of drag coefficient are difficult to determine *a priori* since no one has ever launched a projectile at Mach 18+ at sea level. McNab (2003) suggests a Rodman cone shape for a rail-gun launched projectile. Some empirical data exist for these shapes, but only up to Mach 4. The study carried out by Zielinski and Garner (1991) did, however, yield an empirical equation from Rodman cone flight data:

$$C_D = 0.67M^{-0.82}$$

At Mach 18 (~6 km/s launch velocity), this equation gives a drag coefficient of ~0.06. Using this drag coefficient as a starting point, one can determine how the upper stage ΔV varies with altitude. It was found that for every 500 m of elevation, approximately 100 m/s in upper stage ΔV was saved. For an upper stage solid rocket having a specific impulse (Isp) of 325 sec, this translates into roughly an extra 1% payload mass fraction increase per 500 m in altitude (i.e., from roughly 30% at sea level, to 43% at 5000 m altitude). The conclusion drawn from this is that while higher site altitude is prudent, it is not critical. Most likely, other factors such as proximity to pre-existing infrastructure, downrange requirements, and environmental impact will dominate over launch site altitude concerns.

Launch Noise

Under *The National Environmental Policy Act of 1970*, successfully obtaining a launch license (or any other activity that could affect the environment) in the United States requires the acceptance of an environmental impact statement. A ram accelerator launch facility will most likely have its greatest impact on the surroundings due to the noise generated by the projectile passing through the lower atmosphere at hypersonic speeds. In order to estimate the sonic boom strength, two methods of analysis were employed. One was the Whitcomb far field equation for a truncated cone, and the other a normalized weak shock theory for hypersonic booms (Pan and Sotomayer, 1972). Both methods yielded similar results, thus only the sound overpressure amplitudes predicted by normalized weak shock theory are shown in Fig. 11. These plots were determined for a constant velocity, 2000 kg, 85-cm-diameter projectile launched at 20° inclination angle in the standard 3-layer atmosphere model. The decay in sound intensity along the ground track of flight path is primarily the result of the projectile's rapid ascent.

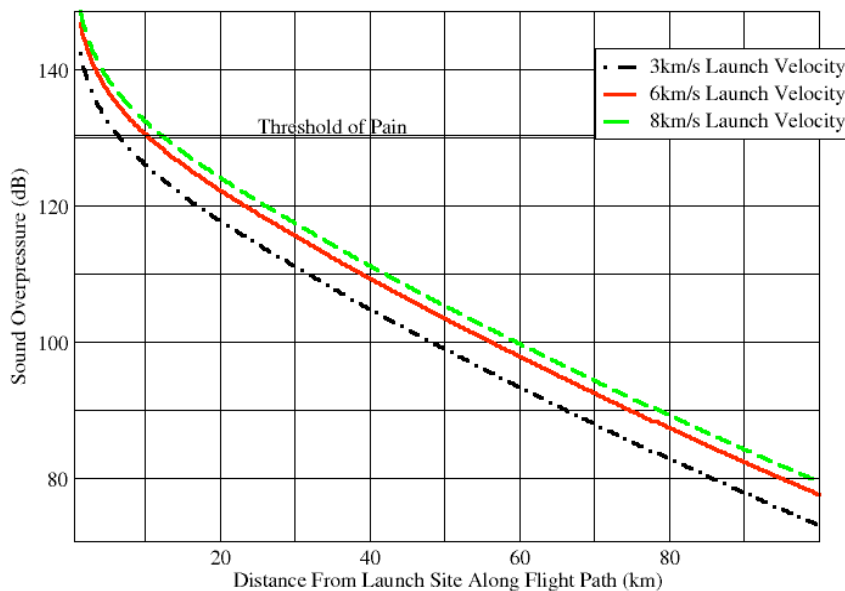


Fig. 11 Hypersonic projectile sound intensity along ground track.

It comes as no surprise that a hypersonic launch is quite noisy. Based on the noise estimates here, a 6 km/s ram accelerator launch site should be placed well over 10 km away from human habitation. Environmental concerns encompass wildlife too. A full environmental impact analysis is beyond the scope of this paper, but it should be obvious that the site location should be placed well away from both aquatic life and land animals (Cheng and Lee, 2004). A high-mountain desert location, for example, might be a good choice.

Payload Considerations

At a general systems level, a ram accelerator-launched satellite is identical to a regular communications satellite and must perform the same functions, thus similar systems must be present. The primary difference between a standard satellite and one launched with a ram accelerator is that the latter must withstand a much more severe launch environment than what is typically experienced with a rocket. Two issues in particular, the structural loads and thermal environment, dominate the technical risks to be considered.

On the structural side, the projectile is subjected to maximum acceleration (depending on the number of stages) of 2000 g for a 1.4-km-long ram accelerator launcher for a period of less than one second. At the same time, it experiences a significant low-frequency vibration from the ram accelerator combustion, which is discussed later. On the thermal side, it must then traverse the lower atmosphere while traveling at Mach 18 or greater. Nevertheless, it will be shown that these issues have already been well studied and can be solved with present-day engineering.

Impulsive-Launched Projectile Structural Loading

Previously Tested High-G Systems

There have been two significant earlier efforts to develop direct to space gun-launch technologies, the first started in 1959 (Project HARP) and carried on through the early '60s by Gerald Bull (Bull and Murphy, 1991), and the latest in 1996 by John Hunter and Harry Cartland with the Jules Verne Launch company (SHARP/JVL) that was funded under the DARPA TTO.

Many consider Gerald Bull the pioneer of direct space launch development, even though the technology he employed, powder guns, could not place a very sizable payload into orbit. Nevertheless, project HARP launched many hundreds of projectiles into suborbital trajectories. The maximum altitude achieved was 180 kilometers, and complex projectiles were routinely subjected to 10,000 g of acceleration. Since HARP's ultimate goal was to deliver a working satellite to LEO, much development effort was spent on constructing satellite system components that could survive the initial launch accelerations of the 16-inch-bore naval cannon (Murphy and Bull, 1966).

Components that were tested successfully by Bull and his group included sun sensors, horizon scanning sensors, accelerometers, NiCd batteries, on-board computer, cold-gas ACS system, and solid rocket motors. These systems were fired both in suborbital trajectories and horizontally at a backstop with a smaller launcher, as reported by Marks *et al.* (1966). Peak accelerations exceeded 10,000 g for each case. After recovery, the sensors were then tested inside a laboratory against their corresponding pre-launch reference. Considering this was

accomplished in the early '60s, more reliable components can undoubtedly be developed with the micro-electronic technology of today (Davis, 2002).

Modern Day COTS High-G Components

It is currently possible to purchase an accelerometer/laser ring gyro commercial off-the-shelf (COTS) from BAE systems that will survive 20,000 g peak acceleration and still function properly afterwards (BAE Systems website, 2006). Among other things, this technology has most recently been incorporated into a class of smart munitions, the *XM982 Precision Guided Extended Range Artillery Projectile*, which has capabilities of target identification, short-term loitering, and of course guided flight. This projectile is intended for use with the Army's Future Combat System (Global Security's website, 2006), and undoubtedly many will be produced.

Under the SHARP research contract, fairly typical COTS electronics were tested in a similar fashion to HARP's methods. According to Dr. Harold Gilreath (2006), the lead engineer of the SHARP effort, *"The issue of g-hardening was studied extensively, including its cost. Hardening the structure and electronics seems straightforward and relatively inexpensive. We went so far as to test commercial electronic packages, such as cell phones, in an air gun at several thousand g's (with just a little hardening) and had no failure. In the end, we decided that none of the loading issues were show stoppers."* Neither HARP nor SHARP had decided that the g-loading problem was insurmountable for structures and all electronics-based subsystems. Deployables and other delicate items will still require some creative engineering for stowage but, as shown below, there are readily available solutions to these problems too.

Static Load Mitigation

The static loading problem can be analyzed using typical structural equations of bending moments, yield stress, and so on. For the accelerating body problem, loads arise much like a bridge or large concrete structure; i.e., the loading at any given point is due to the weight of the structure above. The stresses incurred from this type of loading for a continuous column are determined in the same manner used for hydrostatic pressure in a liquid column. Straightforward calculations show that a constant cross-section column of titanium can be over 9 meters in length at 2000 g before the stress at its base reaches compressive yield limits (neglecting buckling). Since the length of a full-scale 2000 kg projectile is about 3.4 meters (Bruckner and Hertzberg, 1987), a titanium structure will readily support the acceleration loading.

Depending on orientation, payload components are also subjected to bending moments and stress concentrations from launch-acceleration loading. For example, a typical electronics board is supported by a number of posts, thus bending moments can develop between the posts that lead to stress concentrations at their attachment points. These stresses will always be higher than the simple compressive case. If the board is encased in an epoxy or plastic; however, the load can be more evenly distributed over the bottom of the board, which would significantly relieve the stress concentrations and bending moments. For a typical circuit board with surface-mount components (e.g., chips, transformers, etc.), the compressive stresses developed are two orders of magnitude lower than the typical yield stress of the board, which is approximately 70 MPa (Jeng *et al.* 2003-2004). Unfortunately, there is very little literature on this subject, ostensibly because

applications for hardening protection against prolonged (as opposed to shock loading) loads over 1000 g are rare.

Dynamic Loading Considerations

The dynamic problem is primarily concerned with vibrations, acoustics, and combustion instabilities. The major source of vibrations for both conventional rockets, as well as the ram accelerator, is generated from combustion. The ram accelerator environment, however, is several orders of magnitude “noisier” than a typical rocket launch environment. This is understandable when one looks at the rate and confinement of energy release during launch. At any given moment of operation, the combustion intensity calculations in the Appendix indicate that full-scale ram accelerator may generate many tens of gigawatts of acoustic power within just centimeters of the projectile body. As a comparison, the Saturn V rocket lower stage generated 58 GW of power, although for a much longer period of time than that for the launch process of a full-scale ram accelerator launcher.

The discussion of dynamic loading is complicated because the ram accelerator has two primary modes of operation; i.e., thermally choked (in-tube Mach 3-5, subsonic combustion) and superdetonative (greater than Mach 6 in-tube, supersonic combustion). The thermally choked regime is better understood than the superdetonative regime because typical combustion processes, such as are found in a car engine or jet engine, are subsonic. As for supersonic combustion, the projectile is actually moving faster than the bulk of the pressure oscillations, as indicated by the experimental data presented by Hinkey *et al.* (1992). This results in a less severe dynamic environment than that of the thermally choked propulsive mode; hence the thermally choked velocity regime is considered the limiting case.

Acoustic Energy Transfer

In the thermally choked propulsive mode, there are two distinct phenomena at work. The first is the ordinary “noise” of combustion. Using the procedure outlined in the Appendix, it is possible to estimate the worst-case dynamic environment due to this source. For a 6 km/s ram accelerator with a 2000 kg projectile, 1-m-bore, and a launch tube length of 1.4 km a peak load of 31.5 g combustion “noise” is predicted. For a ram accelerator with a 300 kg projectile, 500-mm-bore by 800-m-long, and the same 6 km/s exit velocity a peak load of 57 g is predicted. These dynamic load values are within the range of what off-the-shelf test equipment can withstand in vibration, as well as within the range of what can be successfully hardened against with present-day engineering.

Combustion Instabilities

The second source of pressure oscillations is due to combustion instabilities within the ram accelerator tube. Combustion instabilities are very complex phenomena, but in the thermally choked regime the amplitude of these oscillations can be bound with some certainty with empirical data. Experimental tube wall data (Hinkey *et al.* 1992) indicate that pressure oscillations due to combustion instabilities in the ram accelerator occur in a narrow frequency band varying between 2,000 and 10,000 Hz with amplitude of ± 25 MPa for combustion pressures of ~ 200 MPa, which arise from operation at 5 MPa fill pressure. The net result is that this

oscillation will exert an acceleration variation of 10 to 20% of the nominal acceleration. Stresses due to dynamic loads, however, fall off as the inverse square of the frequency, so while the amplitude of these pressure oscillations are two orders of magnitude greater than those of the combustion noise, their net effect is 2 to 3 orders of magnitude less than the stresses incurred from combustion noise.

Dynamic Load Mitigation

Dynamic loads cause catastrophic failure either by exceeding yield stress or through fatigue failure. Since the vibration environment is of short duration, we are concerned more with the former. Catastrophic failure typically happens through resonance. Turbulent subsonic combustion has a characteristic noise power spectrum density (Calvin, 1994) that is constant up to about 200 Hz, above which the intensity decreases with frequency as $\omega^{-1.5}$ (Rajaram and Lieuwen, 2003). Unfortunately, most electronics boards have resonant frequencies lower than 100 Hz, which inconveniently is also where a large portion of the dynamic energy is.

Typical solutions for this problem include the use of vibration isolators (which requires swing space), or installing stiffeners on the boards, thus raising the resonant frequency (Steinberg, 2000). A better solution has already been alluded to; i.e., encasing the electronic components in a block of plastic or potting epoxy. Criticisms of similar methods stem from maintenance concerns since it is not feasible to diagnose and repair problems on a board that has been fully potted. Being that maintenance is not yet economical for a satellite in orbit, this is not considered to be a major concern.

Coping with Fragile Components: Inflatable Solar Arrays & Communication Dishes

Since no one has orbited a complete and functioning impulsive-launched communications satellite, certain components have obviously never been tested at high accelerations and are in need of some development work. As was seen with project HARP, compact packages of integrated electronics and simple mechanical systems can be designed to survive acceleration loading in excess of many thousands of g 's. Deployables such as solar arrays and communication dishes, however, are relatively fragile structures. To provide deployable structures that can withstand the high- g launch environment, using inflatable components may provide the best solution. There is some precedent for using inflatables in high- g systems, as exemplified by a gun-launched UAV with inflatable wings (Brown *et al.* 2001).

A column of Mylar material (as found in inflatable space structures) will self-load well below its yield stress if its length is less than a 4.8 meters when packed (density = 1400 kg/m^3 , yield stress at 100 MPa). A 7-m-diameter inflatable antenna such as the one tested aboard STS-77 had packed dimensions of a disk 36-cm-diameter and less than 10-cm-thick, minus the struts that can be packed in a similar fashion. Folded properly, stress concentrations are kept small and the Mylar material does not experience any loading approaching its yield stress, making inflatables excellent candidates for high- g launch load survival.

As for solar arrays, new photovoltaic films are commercially available that are bondable to a variety of materials, including polymers such as Mylar (Fairly, 2004). Depending on the photovoltaic material used, efficiencies between 8% and 18% are possible (EEEC energy

website, 2006). The net result is a multi-kilowatt solar array attached to the back of the inflatable communications dish, yielding a respectable power/mass ratio (ranges from average to exceptional, depending on assumptions of thin-film thickness).

The primary reason inflatable deployables, such as the ones described above, have not yet seen much use in space applications is their relatively short lifetimes. A typical spacecraft mission lifetime is usually greater than 10 years, but Mylar and thin film photovoltaics (especially amorphous silicon) will decompose when exposed to UV light or free radicals in the upper atmosphere before the completion of a decade-long mission. The anticipated satellite design lifetime for some immediate applications, however, is between 1 and 2 years, which is well within the design lifetimes of typical inflatables (Cassapakis, 2005).

Aerodynamic Heating for Impulsively-Launched Space Vehicles

The heat transfer issues for a vehicle launched directly from the surface of the Earth at 6-8 km/s with gun-like devices are similar, yet substantially different from those arising during re-entry. The primary goal for manned-vehicle re-entry is to dissipate sufficient kinetic energy to enable a “soft” landing without overheating the payload. Thus these vehicles generally have low ballistic coefficients (low weight to drag ratio) and spend a relatively long time (several minutes) in the upper reaches of the atmosphere at very high velocity. Ablative shields (Mercury, Gemini, Apollo, Corona film canisters) and/or refractory thermal protection systems (Space Shuttle) are routinely used to withstand the intense aerodynamic heating during re-entry. Strategic payloads launched by ICBMs, however, are designed for high-speed re-entry with low drag profiles in order to minimize defensive response times. Thermal protection systems involving heat sinks and insulation barriers have been developed for these applications. In both cases, the worst of the aerodynamic heating during re-entry typically occurs at high altitude and low density on a time scale of minutes.

Impulsive space launch from the surface of the Earth requires the vehicle to pass through the densest part of the atmosphere while traveling at its highest velocity. This significantly enhances the peak heat transfer rate compared to re-entry. The exposure time to the maximum heat flux conditions, however, is much shorter (on the order of seconds); thus, the overall aerodynamic heating is not necessarily any worse. For example, to attain 100 km of altitude from a 20 degree launch angle, the ascent vehicle will travel ~300 km through the atmosphere. At an initial velocity of 8 km/s the vehicle will be above 60% of the atmosphere (~10 km) within 4 sec and, assuming its average velocity to be 6 km/s thereafter, it will reach the upper limits of the atmosphere, ~50 km, in less than 20 sec and 100 km within 1 min of launch, at which point its thermal protection aeroshell can be discarded in preparation for orbit insertion maneuvers.

High ballistic coefficient is desirable for an impulsive-launch ascent vehicle to minimize velocity loss to aerodynamic drag; however, the corresponding slender geometry results in severe heating of the nose tip and forebody. Many studies have addressed this issue for direct space launch applications over the past 30 years. The consensus of these studies is that modern ablative and/or transpiration cooling techniques will protect sea-level-launched projectiles at 8 km/s during atmospheric transit with a tolerable mass loss. In addition, significant thermal protection system mass reduction would be realized if the muzzle altitude were elevated to 5000 m. The main advantage of an ablative protection system is its passive nature and relatively

low complexity; however, blunting of the nose tip increases drag and thus velocity loss during atmospheric transit. Conversely, the transpiration cooling schemes maintain the shape of the nose tip; however, they are more complicated in that on-board equipment is required to control the injection timing and flow rate of the coolant. Brief reviews of the heat transfer analyses for impulsive-launch space vehicles that have been carried out in some pertinent technical publications are provided below.

Previous Work on Thermal protection

During the 1980s there was significant interest in using light gas guns and electromagnetic railguns as direct space launchers. Hawke *et al.* (1982) proposed the use of an EM railgun for launch velocities of up to 20 km/s for projectiles in the mass range of 1-200 kg. They reviewed prior work on kinetic energy loss for ballistic launch and noted that the necessary muzzle velocity increases as the projectile mass decreases due to ablation effects. On this basis they proposed the use of tungsten aeroshells to enable low-mass projectiles (i.e., several kilograms) to survive launch to orbit. Hunter and Hyde (1989) suggested the use of a light gas gun in the velocity range of 5-7 km/s for 1000-4000 kg projectiles having carbon-carbon thermal protection systems. They estimated the mass loss due to ablation to be less than 20 kg for a 6 km/s launch based on simulations carried out by Sandia National Laboratory, and raised concerns about the lack of experimental ablation rate data at stagnation pressures greater than 120 atm (stagnation pressure maximum is ~300 atm and rapidly drops with altitude). Fair *et al.* (1989) compared the status of solid-propellant rockets with the EM railgun technology potential at the time. They refer to a hypervelocity projectile design incorporating transpiration cooling by expelling combustion product gases through the nose tip. In addition, the concept of injecting a liquid combustible jet ahead of the projectile that mixes with the air at the shock front and burns to reduce the atmosphere density is mentioned. No detailed heat transfer analyses were actually presented in any of the papers cited above.

Palmer and Dabiri (1989) considered transpiration cooling for EM railgun-launched projectiles with lithium coolant. Launch conditions optimized to deliver 1 kg of payload to LEO resulted in coolant mass ranging from 1% to 30% total projectile mass for muzzle velocities ranging from 4.5 to 12 km/s, respectively. Conversely, the coolant mass fraction was less than 1% in all cases considered when the projectile mass was increased to 500 kg. Bruckner and Hertzberg (1987) and Kaloupis and Bruckner (1988) carried out ablation and shape change calculations during atmospheric transit for a 2000 kg projectile launched at 7 to 10 km/s at various inclination angles and muzzle altitudes. They assumed the mass loss resulted in symmetrical blunting of carbon-carbon nose tips and accounted for the corresponding increase in drag. Results of these calculations indicate that the projectile would lose ~1% of initial mass and 20-50% of initial velocity by the time it leaves the atmosphere (velocity loss increases with increasing muzzle velocity).

Bogdanoff (1992) carried out an in-depth analysis of aerodynamic heating for ablative and transpiration cooled nose cones of 2000 kg projectiles being launched with muzzle velocities of 7 and 10 km/s. In the case of carbon-carbon ablation for the 10 km/s mission, the projectile mass loss was less than 3%. For the same mission using NH₃ for transpiration cooling, the net mass loss was 2-3%. Other transpiration coolants such as CH₄, H₂, and H₂O were considered; however, NH₃ was chosen to eliminate potential plugging of fine transpiration passages due to

carbon deposits and prevent oxidation damage from O and O₂ arising from water dissociation at high temperatures. This particular paper of Bogdanoff presents the most detailed engineering analysis of the aerodynamic heating problem found in the literature cited herein.

Morgan (1997) provides an overview of gun technologies for space launch applications and suggests that the thermal protection systems for high Mach re-entry vehicles can be used to meliorate the aerodynamic heating problem. Gilreath et al. (1998, 1999) present a detailed design for a 682 kg launch vehicle that employs an aeroshell of primarily carbon composite construction with an overall mass fraction of ~10%. They refer to analytical and computational modeling that predicts the nose cone recession to be approximately 1% the length of the vehicle at a muzzle velocity of 7 km/s.

Aerodynamic Heating Consensus

Others who have recently considered the impulsive launch aerodynamic heating problem have summarily dismissed it as being tractable (e.g., McNab 2003, Cocks *et al.* 2005). In general, as previously stated, modern carbon-carbon ablation and transpiration thermal protection systems are deemed adequate for Earth atmosphere transit at velocities up to 8 km/s. Nevertheless, there are intriguing possibilities in using the new generation of light-weight ceramic ablators; e.g., phenolic impregnated carbon ablators (PICA) for the Stardust (12.6 km/s re-entry velocity) and the Genesis sample return capsules (comet dust and solar wind particles, respectively) (Olyncik *et al.* 1999). The implementation of these thermal protection technologies on impulsive-launched space vehicles will certainly enhance the robustness of this means for LEO access while increasing the payload mass fraction.

Conclusion

The ram accelerator is a promising technology for direct space launch applications. Practical launchers for 300 kg projectiles have been estimated to cost around \$40-50M. The environmental impact of muzzle blast may be a more significant factor in launch site location than altitude, although proximity to technical support infrastructure is another pertinent factor. There do not appear to be any technical hurdles that preclude the immediate implementation of acceleration-hardened satellites for communications and Earth observation missions. Upon reaching key performance milestones in laboratory experiments, the ram accelerator can be readily scaled up to sizes appropriate for satellite launches to LEO.

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Appendix

Acoustic Energy Transfer

Using the following characteristics, it is possible to estimate the worst-case dynamic environment due to the ordinary “roar” of combustion in the thermally choked ram accelerator propulsive mode.

1. A practical upper bound of the energy of acoustic noise generation is 1/100,000 of combustion energy output.¹
2. Typical in-tube conditions at $M = 4$ and a fill pressure of 5 MPa will have a propellant sound speed of 1,000 m/s, and a density of 80 kg/m³.
3. Acoustic impedance of a ram accelerator launch tube is $Z = \frac{\rho c}{A}$ where ρ is steel density, c is sound speed in steel, and A is tube wall cross-sectional area.
4. Similar to the case of an electrical circuit, the RMS pressure (voltage) in terms of a power input is given by $\frac{P_{RMS}^2}{Z} = P_{avg}$
5. For a constant acceleration profile with a fixed-length ram accelerator, the average power output can be determined by the following equation: $P_{avg} = \frac{mV_{exit}^3}{4d} 10^{-5}$
6. The RMS pressure acting on the rear of the projectile is given by the resulting equation:

$$P_{RMS} = \sqrt{\frac{B_c V_{exit}^3 \rho c \cdot 10^{-5}}{4d}}$$

where B_c = ballistic coefficient, V_{exit} = ram accelerator exit velocity, ρ = average in-tube gas density, c = average in-tube gas sound speed, and d = length of ram accelerator

7. A good rule of thumb is that the peak acceleration experienced due to a dynamic load is three times the RMS acceleration.
8. For a 6 km/s ram accelerator with a 2000 kg projectile, 1.0-m-bore, and a 1600-m length the RMS acceleration is 10.5 g, which gives a peak acceleration of 31.5 g.
9. For a ram accelerator with a 300 kg projectile, 500-mm-bore, 800 m length, and the same 6 km/s exit velocity the RMS acceleration is 19 g, which gives a peak acceleration of 57 g.

These dynamic load values are well within the range of what off-the-shelf test equipment can be successfully hardened against with present-day engineering.