The Feasibility of Launching Small Satellites with a Light Gas Gun

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Abstract. This paper summarizes a study conducted for the Defense Advanced Research Projects Agency of the technical and economic feasibility of using a light gas gun to launch small satellites. The launcher concept is based upon a distributed-injection gun, which, in principle, can produce high muzzle velocities at relatively low acceleration levels. To establish initial system requirements for the launcher and spacecraft, the deployment of a large constellation of telecommunications satellites is chosen as a reference mission. This choice reflects the dominance of telecommunications in current commercial LEO market projections, but the results obtained for this mission are later generalized to encompass other applications. The spacecraft mass budget is most affected by large mass fraction allocations for structure and power subsystems. High acceleration loads are responsible for the increase in structural mass, and the increase in battery mass is tied to volume limitations that restrict the battery technology that can be used. The results of the financial analysis suggest that achieving a competitive specific launch cost requires a launch rate beyond current market projections. But a low-volume launch business could provide an attractive total mission cost relative to current systems.

Introduction

While rockets will certainly be used for transporting astronauts and very large payloads into space for years to come, their complexity and high cost inhibit access to space for many other purposes. The emerging requirement for maintaining large constellations of small satellites in low earth orbit (LEO) is just one of a number of reasons to consider cheaper launch methods. Several R&D programs are already underway to develop more economical rockets, but orders-of-magnitude reductions in cost will be difficult to achieve.

About a year ago, the Defense Advanced Research Projects Agency (DARPA) asked us to assess the economic and technical feasibility of launching payloads in the 10–1000 kilogram range using a gun. In principle, a gun is an attractive alternative to a rocket because it is simple, reusable, and can provide an order-of-magnitude increase in payload fraction. But its disadvantages are substantial, too. The launch vehicle must survive high g-loads, as well as the severe heating associated with transatmospheric flight at hypersonic speed. And if the gun is large, the orbits that can be reached may be limited to a single inclination. If these disadvantages can be mitigated, however, a gun launcher would be compatible with the "smaller/cheaper" trend in spacecraft design and would offer major improvements in operability.

A number of types of launcher have been proposed for gun launch to space, but they can generally be grouped into two categories, compressed gas and electromagnetic. The first serious efforts in this area were made in the early 1960's using conventional powder guns under the HARP project.[1]

The limitation on sound speed due to the high molecular weight of powder combustion products required launch vehicles incorporating multi-stage rockets, significantly restricting payload capacity. The project was terminated before payloads were successfully orbited, but this early work did demonstrate that payloads and rockets could survive the rigors of gun launch.

It was recognized early that launch velocity would have to be increased by a factor of three in order to build a system that was "more gun than rocket." A variety of electromagnetic launchers have been considered to this end, but despite substantial investment, progress in the hypervelocity regime has been disappointing since the pioneering work of the late 1970's.[2] In particular, electromagnetic launchers are relatively complex, and the lifetime of materials and components has proven problematic.

In contrast, light gas launchers have shown steady progress since they were first introduced shortly after the Second World War, and by the late 1960's muzzle velocity had exceeded escape velocity.[3] Like electromagnetic launchers, light gas launchers too suffer from barrel erosion and other problems, but do not become unattractive until much higher velocities are sought. Today, muzzle velocities in the 6-8 km s^{-1} range are routinely achieved in testing applications.

In the early 90's, the Strategic Defense Initiative Office (SDIO) considered a two-stage light-gas-gun launcher as a means for deploying the Brilliant Pebbles spacecraft. The requirement was to place up to four thousand 100 kg spacecraft into specified orbits at a rate of one launch every 30 minutes. Researchers at the Lawrence Livermore National Laboratory (LLNL) analyzed a three-tube, large-scale version of the SHARP light-gas-gun, which had been developed earlier by the last author (Hunter). They judged the system to be technically feasible.[4]

Using the experience gained in the SHARP project, one of the authors (Cartland) joined with Hunter in developing detailed conceptual designs for a proposed family of commercial gun launchers, known as the JVL (Jules Verne Launcher) series. These designs are based upon the distributed-injection launcher concept, and provide an important source of information for the present study.

The distributed-injection concept is a variation on the light-gas-gun theme. The launch package is accelerated by injecting working fluid at multiple points along the launch tube rather than having it expand over the entire length from a high-pressure reservoir located at the breech. The concept has been explored previously, both theoretically[5, 6] and experimentally[7], although the experiments were conducted at a scale much smaller than we are considering here. In its application to space launch, the distributed-injection technique is used to reduce the stresses on the launch vehicle (by flattening the acceleration profile) and to facilitate momentum management, rather than to achieve previously unattainable muzzle velocities. In fact, reaching orbit requires a muzzle velocity in the range of 40%-50% of the theoretical maximum, which is in keeping with the documented performance of light-gas-guns.

Objectives and Approach

Affordability was the dominant factor in the study. We were asked to consider practical limitations on the size of the launcher, to define recurring and non-recurring costs and achievable launch rates, and to compare the economics of gun launch to that of existing launch systems. We were also asked to identify launch-survivable spacecraft in the 10–1000 kg range that might be the basis for a viable commercial application. The general purpose was to help the government make informed decisions about the development of an operational launch capability based on light-gas-gun technology.

The approach we adopted is illustrated in Figure 1. The initial sizing of the system was based on preliminary construction cost estimates, rough estimates of potential market size, and judgements about technical risk. Because the relative ablation recession length increases rapidly as the size of the launch vehicle goes down, we decided that a system capable of launching spacecraft weighing only 10's of kilograms was too risky. On the other hand, with construction costs estimated to be over $2B, a system capable of launching spacecraft in the 1000 kg category was considered too expensive. Hence, we focussed the study on guns designed to launch spacecraft in the 100-kg range. We started with the JVL-200 (200-pound payload) launcher, which had been designed to limit peak launch loads to 2500 g's.

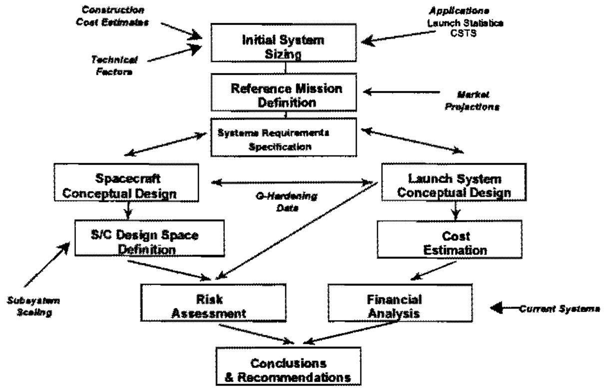

Figure 1. Overall Approach

In view of the requirement to assess commercial viability, the definition of a "Reference Mission" was guided by market projections.[8,9,10] At first we looked at a wide range of possible applications, but soon concentrated on three: 1) scientific research; 2) earth observation; and 3) telecommunications. These sectors appeared to have the highest potential for an active space market in the timeframe covered by the study; i.e., 2005-2030. The telecommunications sector, of course, is far and away the largest, accounting for more than 90% of the launches projected in the near-term and expected to grow into a one trillion-dollar annual business by 2001.[8,11]

We selected a telecommunications mission similar to the one pursued by "Big LEO" constellations, such as Iridium, which are aimed at providing real-time worldwide communications. Our purpose was not to propose an alternate system, but rather to use the choice to uncover the issues associated with launching a complex satellite with a gun. Our hope was that, once the effects of the launch environment on spacecraft subsystems were understood, the results could be generalized to other applications. As a starting point, we developed the crude set of system specifications shown in Table 1, which assumes that the mission can be carried out with a constellation having the same total mass as the Iridium constellation.

Table 1: System Specifications

Orbit Altitude	700 km
Orbit Inclination	90 degrees
Max Deployment Rate	300 per year
Max Launch Rate	2 per day
Mission Life	5 years
LEO constellation	512 in 32 planes
On Orbit Spares	32
S/C Mass (Wet)	113 kg
S/C Volume	0.17 m^3
S/C Density	665 kg/m^3
Average axial acceleration	1,640 g's
Peak axial acceleration	2,500 g's
Average lateral acceleration	15 g's
Vibration loads	1700 g's @ 25 - 250 Hz
Launch Thermal	< 50 °C

Recalling Figure 1, the general approach called for the system requirements, the launch system conceptual design, and the spacecraft system conceptual design to be established through iteration. In practice, we had to be content with a single pass. In the following two sections, we will describe the launcher and spacecraft systems and note the major technical risks associated with them. These descriptions will be followed by a discussion of the financial analysis results.

Launch System

Launcher

Simply stated, the requirement here is for a survivable launch of a 113 kg (250-lb) spacecraft to a 700 kilometer polar orbit. Vehicle design considerations, to be discussed below, and ballistic/orbital mechanics lead to a distributed injection system capable of launching a 682 kg (1500-lb) package at an initial elevation of 22 degrees with a muzzle velocity of 7 km s^{-1}. Somewhat arbitrarily, we limit maximum acceleration to 2500 g's. This is a load that can easily be sustained by modern electronics with little or no hardening, and is low enough to make survivable designs for more g-sensitive components plausible. For purely practical reasons, gas temperature is limited to 1500 K and peak pressure to 70 MPa (10 ksi), with a target average launch tube pressure of 35 MPa (5 ksi). With these restrictions, the launcher has a bore diameter of 63.5 cm (25 in) and a length of 1.52 km (5000 ft). Fabrication of the launch tube will require about 2.7 million kg of highquality gun steel (*e.g.* A723) to provide a safety factor of 3 on yield at peak system pressure. Figure 2 shows an artist's conception of the launcher.[12]

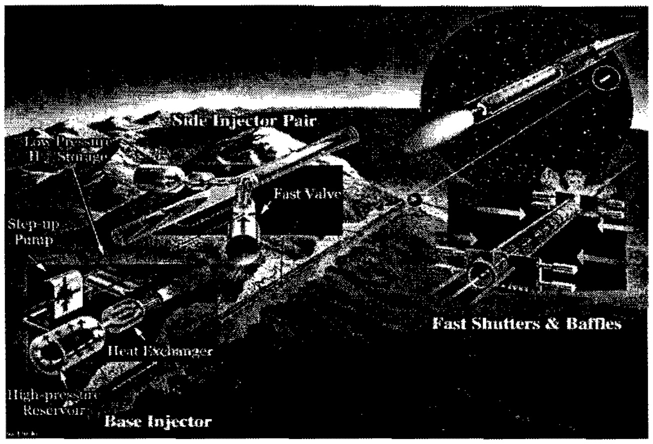

Figure 2. Distributed injection light gas launcher.

The distributed injection system consists of a base injector and 15 side injector pairs that are separated by 150 diameters to mitigate drag . Each injector comprises a high-pressure hydrogen reservoir, a heat exchanger, and a high-speed valve. A pumping system requires 1 hr to charge the high-pressure reservoirs to 70 MPa (10 ksi) with hydrogen from a 14 MPa (2 ksi) storage reservoir. The high-pressure hydrogen passes through a heat exchanger, reaching 1500 K, before entering the launch tube at an angle of 20 deg by way of a high-speed valve. Simulations show that performance begins to suffer if the working fluid is injected more than 10 diameters behind the projectile, and falls off rapidly after 50 diameters, requiring precision timing and valve opening times on the order of 1 ms near the muzzle.

As described here, the launcher will operate with approximately 10 million SCF of hydrogen. The hydrogen working fluid could be sacrificed on every launch, but its cost (~ $80K) is a significant fraction of the total launch cost. Thus the hydrogen is captured with a series of baffles (a "silencer") and fast shutters at the muzzle, and returned through a scrubber system to the low-pressure storage reservoir for reuse. The pumping system is sized to complete hydrogen recovery in 2 hrs. Evacuation of the launch tube takes one hour and is essential since the otherwise enclosed air has a mass comparable to that of the launch package.

Figures 3a and 3b show results from a scaled simulation of the distributed injection gas dynamics, using a base injector and two side injectors. The simulation is used to verify performance, and aids in sizing system components. Both the launch mass and tube length (*i.e.* energy or number of injectors) have been scaled by 3/16, thereby preserving the 7 km s^{-1} muzzle velocity. The code employed here, SIDEHEAT, includes a real gas equation of state, working fluid wall friction, and heat loss to the walls, and handles shocks with the Godonuv method.[13] Figure 3a shows details of the projectile base pressure, which are integrated in Figure 3b to give velocity. As an academic note, Fig. 3b illustrates the sequential effects on velocity of viscosity, chambrage, and distributed injection, assuming a fixed total reservoir volume and pressure. In the simplest case, inviscid flow and no chambrage, the code reproduces the well known analytic relation between the pressure and velocity ratios.[5,6]

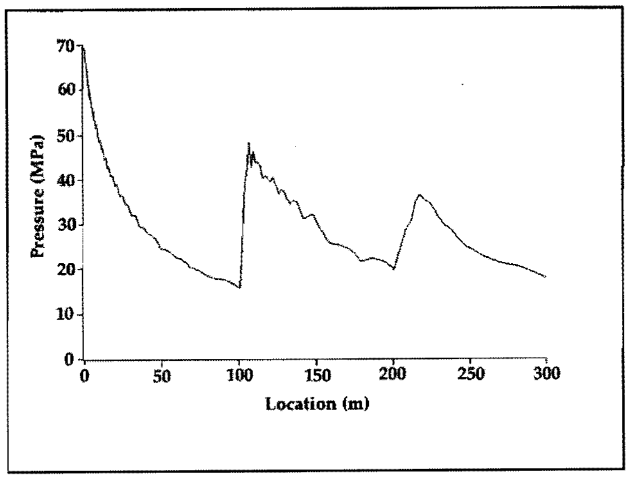

Figure 3a. Simulation of launch package base pressure.

Launch Vehicle

Figure 4 depicts the major components of the launch vehicle. The 113 kg (250-lb) spacecraft is contained in a 0.17 m^3 (6 ft^3) compartment aft. The low drag configuration aeroshell (length/diameter ~11) provides thermal protection and structural support, and is jettisoned after atmospheric egress. A single-stage solid rocket motor fires prior to apogee and injects the spacecraft into a circular orbit. An integral attitude control system orients the projectile after the aeroshell is discarded, corrects for thrust misalignment during motor firing, and may be used for orbital trim.

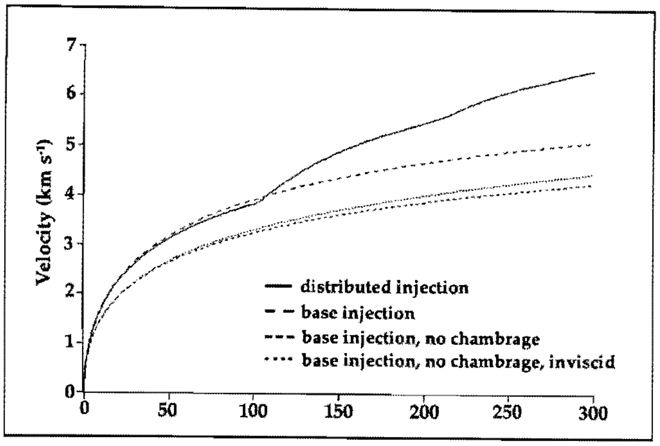

Figure 3b. Simulation of velocity for various constant pV reservoir configurations

About 14 kilograms are reserved for the hydrazine-fueled attitude control system. The system includes 6 thrusters (2 pitch and 4 yaw/roll) with I_{sp} = 230 s. An additional 82 kg (180 pounds) is allocated for the carbon composite sabot (not shown) that supports and protects the launch vehicle while it is in-bore.

A typical mission for a 700-kilometer polar launch might unfold as follows. At t-1 hr the step-up pumps begin to charge the high-pressure reservoirs with hydrogen from the storage reservoir. As the countdown proceeds, the temperature is raised to operational level in the heat exchangers, and launch is initiated by switching the high-speed valve in the base injector. The position of the launch package is sensed in-bore, and the side injector pairs are sequentially triggered. At t+0.44 s, the launch vehicle exits the muzzle and sheds its sabot. The aeroshell is jettisoned at t+300 s, at which point the launch vehicle is at an altitude of well over 500 km, and the attitude control system orients the launch vehicle in preparation for a motor firing at t+545 s. After a 91 second burn, orbit is nominally achieved at t+636 s. Figure 5 is a simulation of the launch vehicle's velocity profile during this mission.[17]

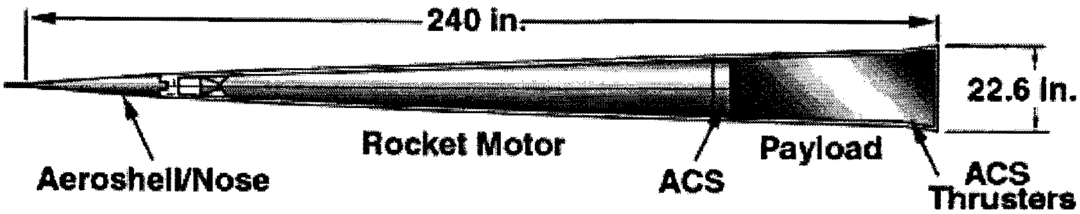

Figure 4. Schematic of the launch vehicle

The launch vehicle can be described as an "inverse" re-entry vehicle, and employs similar methods for thermal protection. The aeroshell is primarily of carbon composite construction and weighs 223 kg (490 lb.). Analytic[14] and computational[15] analysis of ablation predicts approximately 7.6 cm (3 in) of nose cone recession, though incorporation of an aerospike might reduce both drag and deformation during atmospheric egress. The power law body ($r=Ax^{0.65}$) with 7.5 deg base flare ensures both low drag (C_d = 0.016) and passive stability, although the margin of stability was estimated to be very small.[16]

As currently envisioned the solid rocket motor weighs 250 kg (550 pounds). It has a steel case (*e.g.* D6AC) and an NH_4ClO_4/Al propellant with mass fraction of 0.84. The geometry allows for an expansion ratio greater than 20, giving an I_{sp} of at least 270 s. With a 6080 N thrust and a burn time of 91 s, the motor supplies the required $\Delta v = 2.1$ km s^{-1} to orbit the spacecraft.

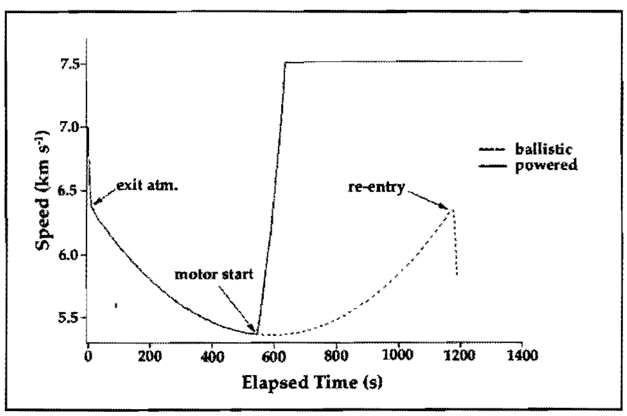

Figure 5. Simulation of launch vehicle velocity profile for a LEO mission.

Scaling and Performance Considerations

Many trade-offs can be made between launcher design, vehicle design, and the manner in which the mission is executed. Practical considerations and the current state of the requisite technologies can help to bound the design space. Given the unconventional nature of the gun launch concept, prudence suggests a conservative approach. Consider, as one example, the launch tube. Our conventional solution is steel construction, even though the melting point of steel limits the working fluid temperature to less than 1700 K. The 1500 K working temperature assumed here further limits sound speed, and hence muzzle velocity, shifting more of the velocity burden to the injection motor. However, in spite of a conservative selection of material and operating margin, the overall system performance remains impressive.

Examination of launch vehicle scaling reveals an important point, and serves to illustrate some of these trade-offs. Take as fixed a number of parameters such as muzzle velocity, orbital altitude, drag coefficient, and average base pressure. Under these conditions, the in-bore stresses are invariant as the system is scaled photographically, so structural mass fraction can remain constant. However, a higher launch mass means a higher ballistic coefficient and better penetration of the atmosphere. A shallower launch angle can then be tolerated, and is in fact necessary to reach a fixed apogee. The inherently higher angular momentum of the shallower trajectory reduces the Δv requirement for the injection motor.

Of larger impact is the thermal protection scaling. Increasing the launch mass decreases the relative amount of surface area requiring protection, but the higher ballistic coefficient and more shallow launch angle yield a higher velocity and longer path length in the atmosphere. For the range of interest here, surface area effects dominate aerothermal considerations, and relatively less shielding is required at higher launch mass.[18] The net effect is that rocket motor and heat shield mass can be traded for spacecraft mass as the total launch mass increases. Figure 6 illustrates this behavior.[19] Note that the spacecraft mass fraction exceeds that of conventional launch vehicles by an order of magnitude or more. Note also that the slight improvement in launcher performance with size due to reduced drag and heat loss to the walls has been ignored.

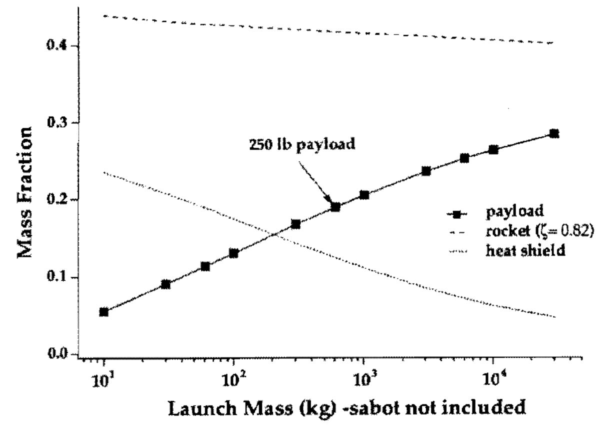

Figure 6. Spacecraft mass fraction scaling.

The launcher is optimized for the mission it is designed to perform. The question then arises as to how performance is affected by off optimum operation given that reorientation of a large launcher is difficult. Small inclination changes are accomplished with the rocket motor and, as is well known, impose a significant penalty on spacecraft mass. The same is not necessarily true for launch to different altitudes. As Figure 7 shows, altitudes below ballistic apogee can be reached with little change in the total required Δv by entering a Hohmann transfer ellipse. Spacecraft mass will likely still suffer to some extent to accommodate a more complex two-pulse motor.

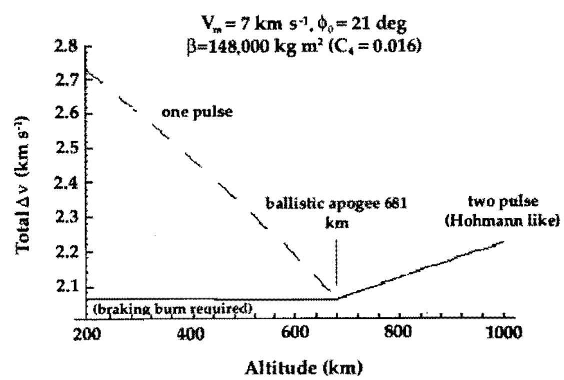

Figure 7. Injection motor Δv requirements versus orbital altitude.

Technical Risks

With respect to the launcher, a number of technologies require testing and integration. The critical technology is the injection process and the engineering of the injector. Valves with throats of tens of centimeter diameter must open within a few tens of bore diameters of the projectile's passing in a carefully timed sequence. In short, the valves must open at "bullet" type velocities both precisely and repeatedly. Pre-accelerated valves for hydrogen capture, such as would be required at the muzzle, have been demonstrated[20], but reliability, maintainability, and synchronization remain issues for all of the high speed valves. The principles of operation of the heat exchangers are understood, but further analysis and some experimentation is necessary to ensure proper throughput, function, and robustness. Finally, simulation of distributed injection launcher performance is a valuable design tool, but code predictions must be validated against actual performance data. All of the above could be adequately tested with a heavily exercised, scaled prototype.

Several launch vehicle issues bear further investigation. Thermal loads appear manageable using standard re-entry vehicle materials and techniques, although the aerothermal environment for egress is more severe. Of particular concern is hypersonic stability, especially with respect to how it is affected by ablation. Analysis, simulation, and experimentation are essential. Also, launch acceleration loads are two orders of magnitude higher than those encountered in conventional space launch. This is a large step for the space launch community, but the loads in question are survivable for many payload components using standard industry design practices.[21] More g-sensitive components, such as large optics and deployable structures, need closer study.

Spacecraft System

Subsystem Analysis

Traditionally, a spacecraft is designed to meet fixed launch vehicle parameters; but in this study we had the luxury of optimizing the gun, launch vehicle, and spacecraft to support a specific mission, and exploring departures from that mission as a result of launcher constraints. This circumstance allowed an iterative loop to exist between the launcher and spacecraft designs that normally is not there.

The starting point was the set of system requirements associated with the reference mission, which was shown earlier. From this set we derived the subsystem requirements shown in Table 2. We then considered various subsystem configurations and evaluated them against launch system constraints; e.g., volume, g-load level, etc. If a configuration failed to meet the mission requirements, or could not stay within the launch system constraints, we pursued other solutions at the system or subsystem level. If a solution still could not be found, changes in launch system requirements were considered.

Table 2: Subsystem Requirements

Subsystem	
Power	
Bus Voltage	22 – 34 V
Average Load	251 Watts
Battery	16.0 AH
ADACs	
Attitude Knowledge	
Roll	+/- 0.6°
Pitch	+/- 0.9°
Yaw	+/- 1.2°
Propulsion	
Position Maintenance	
In Track	+/- 2.0 km
Cross Track	+/- 1.7 km
Orbit Adjustment	
Fix Inclination Error	+/- 0.057°
Fix Apogee Error	+/- 14 km
Shift in Orbit	+/- 6°
TT&C	
Availability	
Cmd Up link Data Rate	1 kb/sec
Cmd Down link Data Rate	1 kb/sec
Structures	
Quasi-Static Loads	2,500 g's

Table 3 shows the subsystem breakout in terms of mass, volume and associated mass fraction that was obtained after one iteration. Assuming that the power system is sized adequately, we note that there is almost no mass available for the RF payload, implying that the reference mission cannot be carried out with the initial gun design. Given the limitations of time and funding, we were not able to converge on a spacecraft design that satisfied the original mission requirements; but we obtained enough information during the analysis to uncover some of the major influences of high g-loads and packaging constraints on subsystem design.

Table 3: Subsystem Allocation

Subsystem	Kg	Vol. (ft^3)	Mass Fraction
Attitude Determination & Control	10.6	0.34	9%
Propulsion (wet)	18.3	0.72	16%
Power	29.7	2.55	26%
Structures	44.2	0.91	39%
Thermal	3.1	0.10	3%
Command and Data Handling	Part of Payload		
GNS Unit	1.3	0.10	1%
TT&C	0.9	0.05	1%
Harnessing	1.7	0.10	2%
Available for Payload	*3.2*	*1.1*	*3%*
Total Spacecraft Mass	113.0	6.0	100%

Major G-Load and Packaging Influences

As noted earlier, conceptual designs for each of the spacecraft subsystems were developed with a system level requirement to meet a peak load of 2,500g. No structural amplification or attenuation effects were considered. As expected, due their small mass and volume, electronic components were relatively insensitive to g-loads. The subsystems most affected by the high acceleration loads, and other launcher limitations, were the structural, power, and attitude determination and control (ADAC) subsystems.

To handle the higher-than-normal launch loads, we assumed the spacecraft's primary support structure to be a simple ribbed cylinder. We examined four materials: titanium (Ti 6AL-4V), aluminum (AL 6061-T6 and AL 7075-T6) and a metal matrix composite (AL SiCp/6061-T6). Their characteristics are compared in Table 4.

Table 4: Structural Material Trades

Material	Strength/ Density (psi/pci)	Thermal Conductivity (Btu/hr•ft•°F)	Relative Cost
Al-6061.T6	370,000	96	Lowest
Al-7075.T6	*770,000*	*75*	*Low*
Ti-6AL.4V	812,000	4	Middle
Al-SiCp/6061.T6	868,000	100	Highest

AL 7075 T6 was chosen as our primary structural material because of its relatively high strength, low cost and good thermal conductivity. Even with the use of this alloy, the structural mass fraction for the primary structure turned out to be 39%. (The total mass comprised 35.8 kg of primary structure and an estimated 8.5 kg of secondary structure.) This mass fraction is considerably higher than the 8% - 15% fraction that is typical of spacecraft designed for conventional launchers.

Designing a power subsystem to meet both the demands of a communications payload and the launcher constraints was a challenge. Since the communications payload was poorly defined, we decided to look at the power subsystem parametrically to see how much power would be available from various solar array and battery configurations. Both Si and GaAs solar arrays were assessed in two body-fixed and two deployed configurations. The arrays were not designed to articulate, although in an operational system that would probably be a requirement. The most powerful configuration was a GaAs array having a length about equal to that of the launch vehicle and a lateral dimension defined by its inner circumference, which could be stowed internally during transatmospheric flight. Such an array would be capable of producing slightly more than 250 Watts.

We studied four types of batteries: NiH_2, Li-Ion, NaS and NiCd. We rejected NaS batteries because of their experimental nature and thermal requirements. Li-Ion technology, while promising, was also rejected because of the battery's inability to meet the high number of discharge cycles associated with the orbit. We would have preferred NiH_2 batteries, given their strong space legacy and high power density (approximately 1.6 x NiCd), but they require about twice the packaging volume of NiCd batteries. The volume constraint makes them incompatible with the initial vehicle design. We also believe they would be sensitive to high acceleration loads. In contrast, NiCd batteries have good acceleration immunity. If we choose NiCd batteries, the total power subsystem, including the solar array and associated electronics, has a mass of approximately 30 kg, or about 26% of the total spacecraft mass.

Items such as star cameras, reaction wheels and spinning earth horizon sensors were shown to have sensitivities to high acceleration loads, so some technology investments would be necessary to develop a survivable ADAC subsystem. We do not believe the design problems to be insurmountable, however.

Generalized Results

The small mass fraction available for the RF payload does not imply that a complex telecommunications satellite cannot be launched with a gun, but rather that the initial sizing of the gun was too restrictive for the chosen reference mission. An RF payload places high demands on power and volume. In this section, we use the results of the subsystem analysis to generalize the results to other possible payloads. The basic question is: what payload power-mass combinations are compatible with a 113 kg spacecraft launched at 2500 g's? In addressing this question, we also take into account the effects of a changed mission by allowing for higher power density batteries and more advanced structural materials.

Figure 8 shows a series of curves that are broken into three sets. The first set is associated with a structural mass fraction of 39%. The second set assumes a composite structure with a mass fraction of 20%. The bottom curve shows a "realistic" design, where the packaging density is limited to 450 kg/m^3 and (because of their volume efficiency and their legacy of high acceleration applications) NiCd batteries are used. The curves in each set correspond to different battery technologies: NiCd, NiH_2 and "advanced technology," by which we mean either Li-Ion or NaS batteries.

Any point in the area underneath any of these seven lines is a possible design point within the class of spacecraft addressed in this study. For example, if one had a 100 Watt payload that required 20 kg, a 39% mass fraction would be allowable, but an advanced technology battery would be required. If 40 kg of payload mass were required then structural mass fraction would have to drop.

Figure 9 shows the same set of curves for a spacecraft that does not require a propulsion system, but still must meet the other subsystem requirements, such as navigation and pointing. As one can see, when the propulsion system (18.3 kg) is removed from the spacecraft a different set of curves are generated. Points on these curves indicate accessible payloads, as long as the mission does not require propulsion. Clearly in this case there is more power and mass available to the payload without employing advanced technologies.

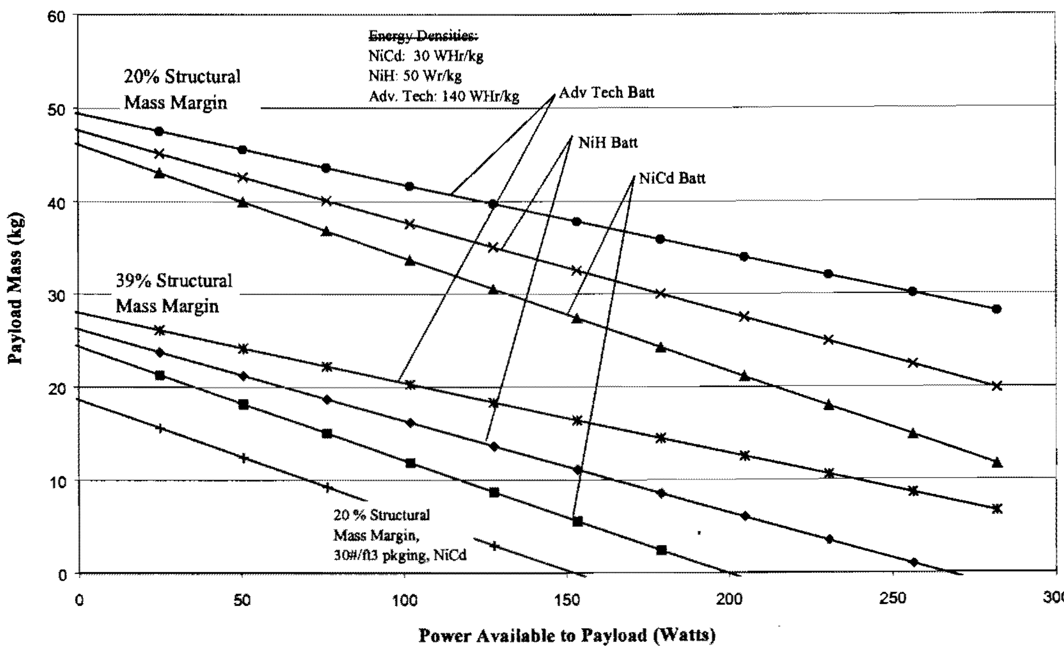

Figure 8: Acceptable Spacecraft Design Curves

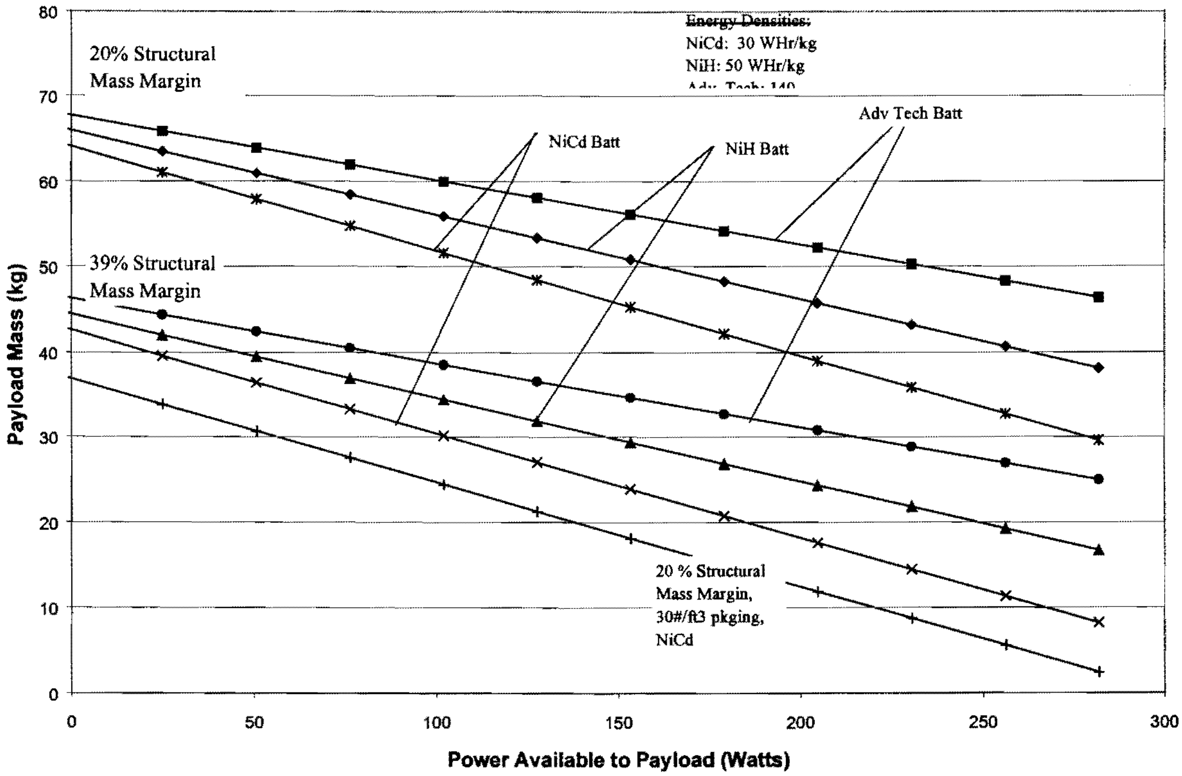

Figure 9: Acceptable Spacecraft Design Curves (No Propulsion)

Major Technical Risks

The major technical risks appear at both the system and subsystem levels and are associated with both high acceleration loads and the small packaging volume. Conventional packaging densities, coupled with the increased mass devoted to support structure, reduces the payload mass substantially.

High acceleration loads affect all subsystems, but components such as the large optics in a star camera, reaction wheels, or spinning earth horizon sensors will need special attention. Issues associated with packaging and deploying solar arrays and antennas are of paramount importance. Gun-launched spacecraft will require high packaging densities, but with the industry looking towards micro- and nano-spacecraft, high density designs may be possible in the future.

Financial Analysis

Assumptions & Methodology Overview

The major premise shaping the financial analysis was that the gun launch system would be a commercial entity, operating as a business that offers competitive returns to investors. While the current analysis does not fully address all of the issues inherent in the term "business operation," it provides significant insight and a solid foundation for future study.

To accomplish the assessment, we used a financial tradeoff analysis. Just as the launcher system can be described by a "physical operating parameter space" of interrelated parameters and characteristics, such as maximum pressure, barrel length, and payload, it can also be described by a "financial operating parameter space" of interrelated parameters, such as construction cost, operating cost, rate of return, and cost of launch vehicle. The physical characteristics of the launcher system link the two parameter spaces together, so that tradeoffs in one area lead to changes in the parameters of the other. For example, if the payload mass increases then either the internal rate of return increases or the cost per kg decreases. In the following sections, we describe the parametric analysis we conducted to explore the feasible "financial operating region" for the launch system, and then compare the results to conventional systems.

Affordability Metrics and Concepts

We used several standard financial metrics as our FOM (figure of merit) for affordability. This section provides a quick review of those measures and concepts discussed in this paper. A more complete review of Internal Rate of Return and Net Present Value can be found in Blanchard[22]. Stewart[23] contains an excellent discussion of learning curves.

The Net Present Value (NPV) is a metric that quantifies the value of an income stream (which can contain either positive or negative cash flows in each period) over a period of time, taking the interest rate into account. In mathematical terms:

$$NPV = \sum_{t=0}^{n} F_t (1+i)^{-t}$$

where
NPV = Net Present Value
t = time period in years
n = number of years
F_t = net cash flow in year t

The Internal Rate of Return (IRR) is a measure of profitability. It is the interest rate that, when applied to a stream of cash flows causes the NPV to be zero. It is analogous to the return on a mutual fund or certificate of deposit. Mathematically, IRR is defined as the interest rate i^* such that:

$$0 = \sum_{t=0}^{n} F_t (1+i^*)^{-t}$$

All other terms are as defined previously for NPV. NPV and IRR were both used as parameters and as FOM's in the analysis.

The Learning Curve, or progress function, is a means of quantifying how familiarity and experience with the completion of a product lead to greater efficiency and cost reduction in production. Learning curves are frequently expressed as percentages. An 85% learning curve implies that a cost reduction of 15% occurs when the number of articles is doubled. The fourth unit produced would cost 85% of the cost of the second unit, the eighth unit would cost 85% of the fourth unit, and so forth. Mathematically, the learning curve relationship is defined by the following equation:

$$Y_t = tY_1^{\,b},$$

where
t = unit number
Y_t = cost of unit number t

The exponent b is defined by

$$b = \frac{\ln(m)}{\ln(2)},$$

where m is the learning curve rate expressed as a decimal. The learning curve rate and initial unit cost of the launch vehicle were used as parameters in the analysis.

Financial Model

The financial model provided a year by year representation of cash inflows and outflows for the launch complex based on selected physical and financial parameters. Several key assumptions were built into the model, and not subject to variation during the course of the analysis. In general, these assumptions were determined by the ground rules of considering the launcher as a commercial system that had to recoup its operating and construction costs. The model has a fifteen-year planning horizon, with a time resolution of one year. We assumed that the first three years were devoted to the construction of the system. The maximum launch rate during the fourth year was either 50 per year or the prevailing yearly launch rate, whichever was less. This assumption allowed for possible construction delays and a "shakedown" period for launcher system operation. There was no allowance for "down time" for major maintenance or refurbishment. We assumed that any major maintenance could be completed without affecting the yearly launch rate.

Construction costs were fully amortized by the end of the fifteenth year. We assumed that a quantity of funds was borrowed at the start of the program. The funds not utilized for construction in a year were invested at the prevailing interest rate. All funds borrowed for construction were expended by the end of the third year. Using an initial sensitivity analysis on the fund expenditure profile (the percentage of total construction funds expended each year of the construction period) we determined that the impact of varying the profile within reasonable limits was marginal. The analysis was performed with a profile that assumed 50% expended in the first year, 25% in the second, and 25% in the third. The model automatically calculated the required funds based on the construction costs, the expenditure profile, and the interest rate. This calculation established the yearly construction loan payment. Eight percent was used as the prevailing interest for all calculations. The actual cost figures were derived from estimates based on a mix of Cost Estimating Relationships (CER's) and analogous component estimates that were prepared for the construction of the JVL-200 launcher in Adak, Alaska.

Due to the inherent uncertainty in cost estimates based on such a preliminary design, our sensitivity analysis (discussed later) was designed to ensure we were able to bound the construction costs. The construction costs were broken down to a Level II Work Breakdown Structure (WBS). Some selected components included the launch tube, injectors, hydrogen storage, hydrogen heaters, valves, handling equipment and site work.

Operating costs were divided into fixed and variable costs. Fixed costs represented "housekeeping" costs that were independent of the number of launches per year. The fixed costs (payroll, supplies and site maintenance) were taken to be $4,000,000 per year, based on the JVL-200 launch cost estimates. Variable launch costs represented a "per launch" cost including labor, hydrogen, etc., but did not include the cost of the launch vehicle. These variable launch costs were also based on the JVL-200 launch costs. The data provided variable costs for 75, 150, 300 and 600 launches per year. Variable costs were approximately $5,200 per launch at a launch rate of 300 per year. We performed a log transformation on the data and then based the costs on a linear regression model of the transformed data. This regression was used to estimate the "per launch" costs in the model for the varying launch rates under analysis.

Three cost elements were tied to the launch vehicle: 1) the initial vehicle R&D costs; 2) the cost of the first vehicle (Y_1); and 3) the learning curve rate for launch vehicle production. The inclusion of these elements reflected the assumption that the business entity operating the launch complex would be responsible for developing and building the launch vehicle. The model also provided the option to use a fixed cost per vehicle, reflecting the possibility that the vehicles would be purchased from some other commercial source instead of being developed internally. These two approaches yielded slightly different results, due to the effects of the discount rate.

We considered several methods for estimating the cost of the first launch vehicle. One method made use of the NASA Advanced Mission Cost Model[24], which provided a ROM (rough order of magnitude) cost estimate for a missile having characteristics similar to the launch vehicle. More detailed estimates were based on subsystem-level pricing of the conceptual vehicle shown in Figure 4, and on inflation-adjusted cost breakdowns for the Brilliant Pebbles[4] and HARP[1] vehicles. The latter estimating methods produced similar vehicle costs, ranging between $280,000 and $320,000 per vehicle. Costs associated with the launch vehicle assumed great importance in certain launch parameter scenarios, and so we chose to use the higher fidelity approach in our work.

Several cost elements were not modeled due to the large amount of time that would have been required to analyze them. These elements included taxes, insurance, depreciation, range clearance costs, and the cost of a major refurbishment of the launch complex. There were no land acquisition costs allocated. We assumed that any environmental impact approvals were obtained with reasonable processing costs that were included in the construction costs. We further assumed that there were no additional costs or delays to the project due to litigation, strikes, or other causes.

Representative values for several key financial parameters are contained in Table 5.

Table 5: Financial Parameters Representative Values

Quantity	Value
Interest Rate (per year)	8.0%
Amortization Period (years)	15
Spacecraft Weight (kg)	113
Price per kg	$3,486
Initial Launch Vehicle Cost	$320,000
Launch Vehicle Learning Curve Rate	90.0%
Average Launch Vehicle Cost	$109,589
Launcher Construction costs	$298,000,000
Yearly Construction Loan Amortization	$32,928,784
Fixed Launcher Operating Costs	$4,000,000
Variable Launcher Operating Costs	$15,551,000
Average Yearly Launch Vehicle Costs	$32,876,584
Launches per year	300
Yearly Gross Revenue	$118,597,576
Internal Rate of Return	8.0%
Net Present Value	$0

Analysis Methodology

In our parametric analysis, the launch system was characterized by the following parameters: gross income per launch (either $/kg to orbit or $/launch), spacecraft weight, launch rate, fixed launch costs, variable launch costs, initial launch vehicle cost, launch vehicle learning curve rate, IRR, construction cost and interest rate. Interest rate remained fixed throughout the period. Construction cost and payload weight were determined by the launcher variant under evaluation. Fixed and variable launch costs were based on the JVL data as discussed previously.

In order to ensure that we captured the true range of launch costs, we conducted excursions with increases of 50% in variable launch costs and fixed launch costs. We also performed excursions with a 50% decrease in these values. We carried out a similar analysis on the construction costs: a 50% increase, a 50% and a 100% decrease. The cost decreases have also served to provide data points to include the effect of construction cost subsidies.

Cost Analysis

We will illustrate the results with a few of the graphs and tables we developed in the course of the analysis. All the graphs presented in this section will be for the values indicated in the Table 5 unless otherwise indicated.

flows, allowing us to predict when yearly revenue would recover cost, and when the net cumulative cash flow became positive.

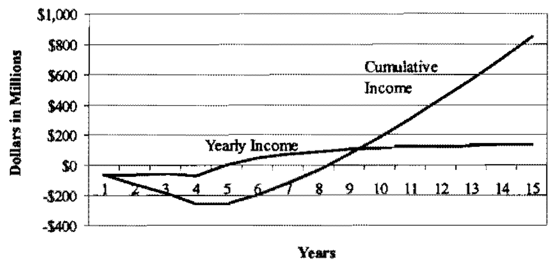

Figure 10: Income Stream Graphs

We used IRR as a figure of merit as well as a parameter. We gained useful information by holding all parameters constant except for the cost of the first launch vehicle and the learning curve rate for the production of launch vehicles. Analyses of this kind allowed us to determine the range of launch vehicle parameters that would allow rates of return comparable with other competing projects of similar payoff and risk under specified market assumptions (launch price and launch rate). This information was then used to help explore a financially feasible region for launch vehicle parameters.

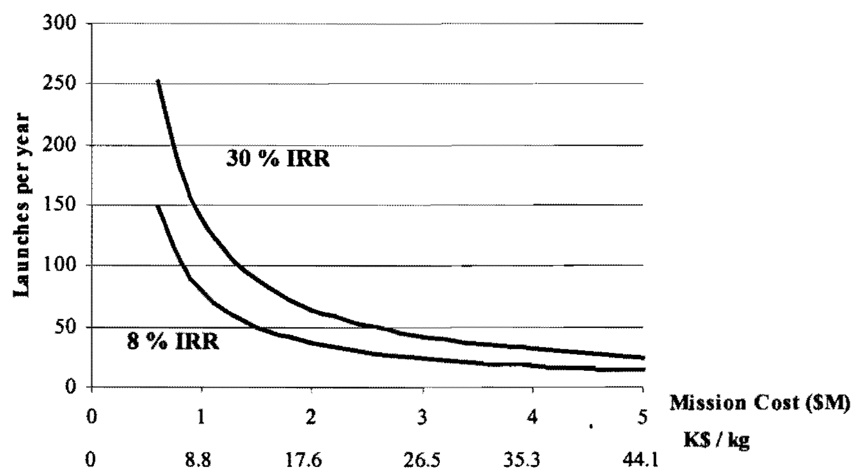

Figure 11: Launch Costs

The income stream graph shown in Figure 10 corresponds to an IRR of 20%. The first year that the operation turns a yearly profit is in year 5. However, cumulative cash flow is not positive until year 9. Analyses of this type provided time data for income

We also examined the effect of varying the launch rate. Our original concept was that the launcher would probably be most cost effective in a "mass market," with a launch rate in the vicinity of 300 launches per year. When we varied the launch rate and looked at the required revenue stream expressed in terms of dollars per launch rather than dollars per kg, we found possible alternate operating points, as shown by Figure 11. The lower trace on this graph shows those combinations of total mission cost and launch rate which result in a breakeven situation (NPV = 0). The upper trace shows those combinations that resulted in an IRR of 30%. (This value was chosen as representative of the minimum IRR needed to make the launch system as attractive as alternate technology investments.) Analysis of graphs such as these led us to consider launch rates on the order of one launch per week. Although the cost per kg in this regime is much higher, the cost of approximately $2.5M per launch is within the typical budget of small satellite researchers.

The study also allowed us to determine the major cost drivers for the operation of the launch system. We had reasonable estimates of most of the costs involved in calculating the Total Ownership Cost (TOC), with the exception of major maintenance and disposal costs. We found that variations of 50% increases or decreases in construction or operating costs had little effect on the overall cost element distribution. Table 6 illustrates two typical potential operating points, and highlights a key result: reductions in launch vehicle cost offer the greatest opportunity for driving the launch costs down further.

Table 6: Cost Element Breakdown

	Yearly Launch Rate	
Cost Area	**300 Launches**	**52 Launches**
Construction	9%	30%
Fixed Operating	1%	4%
Variable Operating	4%	4%
Launch Vehicle	86%	63%
Total	100%	100%

Comparison with Other Systems

We will conclude the financial discussion with a brief comparison of the gun launch system to current launch systems. All cost data for competitive systems is taken from Isakowitz[25]. A selected portion of this data is contained in Table 7. We should be careful to point out that these comparisons are crude and indirect. They do not account for differences in launch vehicle scale, orbital parameters, or economic factors.

Table 7: Comparative LEO Payloads & Costs

Vehicle	kg LEO	$M/Launch	$/kg
Pegasus XL	460	$12	$26,087
LLV-1	800	$16	$20,000
Conestoga 1620	889	$19	$21,372
Taurus	1400	$19	$13,571
LLV-2	1990	$21	$10,553
LLV-3	3655	$25	$6,840
Delta 7925	3990	$48	$11,905
AR40	4900	$53	$10,714
Atlas II	5510	$80	$14,519
AR44P	6900	$88	$12,681
SL-4	7000	$19	$2,643
Zenit-2	13740	$40	$2,911
AR5	18000	$120	$6,667
Proton D1	20900	$60	$2,871
Titan IV/SRM U	21640	$222	$10,259

As we noted above, the estimated mission cost for the gun launch system would be approximately $2.5M to place a 113 kg satellite into LEO at a launch rate of about one per week. This cost compares favorably to that of the nearest current system (Pegasus), even allowing for multiple satellites to be carried on Pegasus. The most recent estimate puts the Pegasus launch cost at $16M to place 250 kg into a 600 km sun-synchronous orbit[26].

To achieve a favorable specific cost ($ per kg), the launch rate must be near the limitations of the launch system, which is 300 launches per year. This launch rate is probably sustainable from a mechanical point of view, but we are unable to justify such launch rates based on even the most optimistic market projections and assumptions about market share. While it is certainly possible that the existence of the capability to perform nearly daily launches of small satellites might be the enabling technology for a new era of the exploitation of near earth space we cannot quantify such a belief. There are also issues regarding the limitations inherent in the fixed inclination of the launching system that have not been addressed in this analysis.

Conclusions and Observations

We conclude that placing small satellites into orbit using a distributed-injection light gas gun is technically feasible, provided that certain critical developments are made. For the launcher, the key components are fast acting, high flow rate valves and an endurable high-temperature, high-pressure hydrogen heater. Thermal protection, aerodynamic stability, and packaging efficiency represent significant problems for the vehicle.

Because of the large power levels required for high-throughput global telecommunications, our initial design iteration indicated that spacecraft in the 100-kg class cannot meet the requirements of the reference mission, given the limits on payload mass and volume. In broadening the mission set, however, we found that a large range of useful, less power-intensive payloads and missions were possible. The actual range of possible payload weight and power are dependent on the level of technology incorporated in the spacecraft structure and batteries, as well as the level of the requirement for on-orbit propulsion.

Our economic analysis showed that a payload-to-orbit cost of approximately $5500 per kg would be required to yield an internal rate of return (IRR) of 30%, if the launch rate could be pushed to 300 per year. At the same specific launch cost, the operation would break-even with 150 launches per year. While this cost is comparable to present rates, the cost of conventional launches is likely to come down in the future as new and upgraded launch vehicles enter the competition. As a result, we have some concern about the market attractiveness of the light-gas gun launcher for applications in which specific launch cost is important. Complexities are likely to arise when designing a system for a gun environment, and even 150 launches per year goes beyond current projections for small satellites.

On the other hand, when approached from the perspective of total mission cost, the gun launch system looks very interesting. With a launch rate as low as one per week, a total mission cost of approximately $2.5M would yield an internal rate of return of 30% and a total mission cost as low as $1.5M would permit break-even operation. These mission costs are considerably less than current systems and provide for some interesting possibilities for small satellite operations. The fundamental question relates to market elasticity. If a launch system were available that provided such a low cost per mission, would we see a dramatic increase in the demand for launches? (If we build it, will they come?)

It appears to us that further analysis of the viability of a light gas gun for affordable access to space is warranted. Our future work is likely to focus on total system optimization based on overall cost and utility, and on the evaluation of more advanced missions enabled by affordable on-demand access to space; e.g.; on-orbit satellite refurbishment or upgrade. Developing a higher fidelity design of the launch vehicle to better uncover potential challenges, and increasing the fidelity of the system level cost estimates are also part of our plans.

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19. This plot may be suspect at the extremes of the range. No account is taken of any penalties that may be associated with fabrication at very large or small scale.

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