THE MASS DRIVER PROJECT

**A PRELIMINARY DESIGN OF A MATERIAL
TRANSFER SYSTEM FOR SPACE CONSTRUCTION
PROJECTS USING OPEN SOURCE DESIGN
PRINCIPLES**

by

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ABSTRACT

The Mass Driver Project: A Preliminary Design Of a Material Transfer System For Space
Construction Projects Using Open Source Design Principles

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This project attempts to adapt open source software development principles to the evaluation and design of a mass driver system capable of transporting construction material from the Earth's surface to orbit as a stream of inline, discrete material packages following one after another. The design process does not identify any 'show stopper' physical effects. The final design of the system proposes using a novel launch rail concept to send a stream of 10 kg material packages to a 10 000 km apogee altitude, where a rocket motor fires to place the material package into a highly elliptical orbit. A satellite, in a similar orbit, collects the material packages and distributes them to a space construction project. Further work is required to evaluate the aerodynamic issues related to the material package stream. Open source collaboration on the design did not occur; however seven potential collaborators were identified.

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--Basic research is what I'm doing, when I don't know what I am doing -- Wernher von Braun.

DEDICATION

I would like to dedicate this work to the memory of my mother, Elona Gerl (nee Tyni).

I am sure she remains proud of my continued achievements.

--Halukas auttaja tulee käskemättä - A willing helper comes without asking – Finnish Proverb

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NOMENCLATURE

- a_{avg} – Average acceleration force (m/s^2)
 a_{ss} – Speed of Sound (m/s)
 $d_{spacing}$ – Material Package Spacing (m) - distance between material packages in stream
 h – Altitude (m) - height above Earth surface
 h_{max} – Maximum Altitude (m) - the highest altitude the material package obtains
 k_{air} – Specific Heat Ratio = 1.4 (for air)
 $m_{package}$ – Launch Mass (kg) - total mass of material package
 $m.f._{payload}$ – Mass fraction of the payload
 $m.f._{prop}$ – Propellant mass fraction for the rocket motor
 m_{rocket} – Mass of the rocket motor (kg)
 r_{apogee} – Apogee Radius (m) - highest altitude for the material package (apogee)
 $r_{Circ\_Launcher}$ – Radius of launcher (m)
 r_{Earth} – Radius of a spherical Earth (m) = 6371 km = 6 371 000 m
 r_{max} – Maximum Radius (m) - largest distance from the centre of the Earth to the material package at apogee (highest location)
 t_{step} – Time step (s) - the time duration between each calculation in the simulation
 $t_{step\ Multiplier}$ – Time step multiplier — (effects the length of the time steps)
 v_{circ} – Circularization Velocity (m/s) - velocity required for circular orbit (at apogee altitude)
 v_{Launch} – Launch Velocity (m/s) - the initial launch velocity
 $v_{package}$ – Velocity of the package (m/s)
 $v_{\theta(apogee)}$ – Tangential Velocity (m/s) - tangential velocity of material package at apogee
 A_F – Frontal Area (m^2) - frontal area of the material package
 A_{perp} – Area of launch package perpendicular to magnetic field (m^2)
 B – Magnetic Field Intensity (tesla)
 C_d – Drag coefficient for material package
 $C_{slip-stream}$ – Slip Stream Drag Reduction (%) - the drag reduction caused by the launch corridor
 $D_{package}$ – Material Package Diameter (m)
 E_{Launch} – Launch Energy (MJ) - launch Energy for material package
 F_D – Atmospheric drag force (N)
 F_G – Force of gravity (N)
 F_R – Resultant force of gravity (N)
 G – Gravitational Constant = $6.67 \times 10^{-11} m^3/kg s^2$
 I_{sp} – Specific Impulse (s) – performance parameter of rocket motor
 L/D – Length to Diameter Ratio of material package
 $L_{package}$ – Length (m) - length of material package
 M_{Earth} – Mass of Earth (kg) = $5.976 \times 10^{24} kg$
 MFR – Mass Flow Rate (kg/s) - the rate of mass transfer from the launcher mechanism
 P_{Launch} – Launch Power (MW) - the power used for the launch process
 R – Real Gas Constant ($J/kg K$) = $287 J/(kg K)$
 T_{atm} – atmospheric temperature (K)

V_{package} – Volume (m^3) - volume of material package

α_{Launch} – Launch Angle (degree) - the angle between the launch direction and the radial (vertical) direction

ρ_{atm} – atmospheric air density (kg/m^3)

ρ_{package} – Material Package Density (kg/m^3)

μ_0 – Magnetic permeability

ω_{launch} – Launch Frequency (Hz) - the launch frequency of material packages in launch corridor

ω_{Earth} – Rotation Speed of Earth (rad/s) = $-2\pi / (23 \text{ hr } 56 \text{ min } 4.09\text{s}) = 72.921159 \times 10^{-6} \text{ rad}/\text{s}$

Chapter 1 Introduction to the Mass Driver Project

1.1 Introduction – The Mass Driver Project

Large scale commercialization and development of space near Earth is hindered by the high cost of transporting material to space using current rocket technology; presently, the cost is approximately \$22 000 US/kg [1]. Much of this cost is due to the fact that rockets are not efficient, only approximately 5% of the takeoff mass of a large rocket is payload that reaches orbit. The rest of the rocket mass is 85% fuel and 10% propulsion machinery, which are discarded along the flight to orbit [2]. Smaller rockets have a payload mass that is less than 3% of the takeoff mass, with the remaining mass dedicated to fuel and propulsion [3]. It is unlikely that further substantial cost reductions in current rocket technology is possible, given that globally, at this time, approximately \$1 trillion dollars have been spent on rocket research, development and operations. Present rocket technology represents a mature technology [2]. Therefore, to get material from Earth's surface to orbit, at a substantially lower cost, a new launch technology must be developed.

Rocket research, development and operation programs have mainly been done by government agencies, government contracted organizations and government funded educational institutions. Using a traditional development method to create a new, lower cost space launch method would likely require a substantial political commitment and public funding. Therefore, the large scale development and commercialization of space is restricted because it is currently too expensive to rocket material up from Earth; yet, researching and developing a new, cheap launch technology is also too expensive to develop using traditional means.

A possible solution to this conflict is to reduce the cost of researching and developing the new launch technology. To reduce the development costs, this project investigates using the Open Source Software (OSS) development model to evaluate the use of mass driver technology to deliver a series of small material packages to Earth orbit. Section 1.4 introduces the mass driver material package stream transport system (MPSTS) that was proposed by Mr. Ed Panzeria, President of Axiomatic Product Development (APD) Ltd.

1.2 What is Open Source Software?

Open Source Software (OSS) is considered any software product that has the original source code for software available to the public for use and/or modification at no cost. Two factors created the OSS development process, firstly, the need for high quality web server and operating system software and secondly, a social backlash (in the developer community) against corporate monopolization of the software market. A group of programmers, who essentially shared the aforementioned views, formed a volunteer community to collaborate via the internet and create successful high quality software products like the Apache web server and Linux operating system [4].

OSS development is very cost effective because the volunteer community writes, tests and documents the software product for no monetary compensation. The compensation the volunteer community receives is a software product that likely better suits their needs than anything commercially available and the opportunity to meet and work with people with similar interests. Section 2.1 explains the organization of typical OSS projects.

1.3 Open Source + Engineering = Open Engineering

The application of OSS development principles to engineering design and development has been termed Open Engineering (OE) by Professor F. A. Salustri. A research goal, for this project, is to investigate the effectiveness of implementing the Open Engineering concept for a space technology project that requires the effort of a multidisciplinary team of experts. A space technology development project was chosen as an open engineering subject mainly for two reasons. Firstly, the potential for conflicts of interest for participants and their present employers is low, since, few commercial enterprises are researching alternative space launch technology. Secondly, developing an alternative space launch technology requires a multidisciplinary development team, comprised of professionals from science, engineering and business. An open source project has the potential to attract a broad range of people to form a collaborative multidisciplinary development team. The cost of employing a multidisciplinary development team for a space technology project would be very high; therefore, using an open source model could be

very cost effective. If the OE concept can be implemented for a complex space technology project, it is likely that the OE concept can be applied to many other fields of engineering.

When this project was conceived, it was believed that this was the first OE project. However, through the literature review performed, it was determined that at least three previous OE projects have been attempted, the open source car (OSCar) project, the OpenCores project and the ThinkCycle project. Section 2.2 in Chapter two provides the details of these three projects.

1.4 Mass Driver - Material Package Stream Transport Concept

Intuitively, one of the simplest space launch systems would have a launch mechanism on the ground that would impart the payload with enough momentum (velocity and energy) to reach orbit. This idea of using a single impulse or directly launching payload from Earth surface to orbit dates at least as far as Newton's thought experiment with a cannon on the side of a high mountain. Newton rationalized, that if a cannon mounted on the top of a very tall mountain, could fire a cannon ball at the appropriate velocity, the cannon ball would perpetually fall towards Earth or orbit Earth [3].

Research into a single impulse or direct method of launching material into space has been on going since Newton's thought experiment. The research into direct space launch technology has followed the research advancement of military gun technology. The original concepts simply used large gun powder cannons to fire material into space. As gun technology advanced, the space launch concepts envisioned using electromagnetic guns as a practical means of launching payloads to space. In the 1970's, Dr. G. O'Neill proposed the use of an electromagnetically driven launch tube named a mass driver to launch lunar soil from the Moon's surface; since then, the term mass driver is commonly used for electromagnetic launchers [3]. Sections 2.3 reviews some of the previously proposed direct launch concepts, Section 2.4 reviews two space plane concepts and Section 2.5 reviews the common launcher technology proposed for direct space launch.

Presently, all material arrives in Earth orbit either on a rocket or the Space Shuttle, therefore no direct space launch method has been successfully implemented. Many of the previously proposed space launch methods assumed the launch of large satellite sized payloads.

Mr. Ed Panzeria, of APD Ltd., proposed a space launch concept that is similar to the performance of a pitcher and catcher at a baseball game. The pitcher is responsible for accurately propelling the ball to the vicinity of the catcher and the catcher is responsible for positioning the glove to catch the ball. The ball is responsible for traveling the path from the pitcher to the catcher in a predictable manner. To allow the ball to be caught properly, the pitcher and catcher communicate with each other to ensure they both know the behavior of the ball.

For Mr. Panzeria's concept, the pitcher is replaced by a mass driver launch mechanism, the catcher is substituted by an orbiting 'catcher' satellite and the ball is substituted with a small payload of raw material, known as a material package (MP). Figure 1.1 depicts the 'catcher' satellite (CatSat) capturing a stream of material packages (shown in red). The performance of the mass driver launcher is nearly the same as a baseball pitcher, except a series of balls or material packages are rapidly launched one after another to form a material package stream. It is assumed that the sequential launch of MPs could create a tunnel or corridor of low-density air through

Figure 1.1: Material Stream Mass Driver Concept [5]

the atmosphere. The series of material packages could be visualized as a cable or string extending from the launcher up through the atmosphere to a 'catcher' satellite (CatSat). Once the material packages reach the vicinity of the CatSat, the satellite positions itself to capture the MP stream. Therefore, like the baseball, the MPs are merely responsible for traveling in a predictable manner through the atmosphere, the mass driver and CatSat have the 'brains' to control the flight and capture of the material package. Once the MPs have been captured by the CatSat, they are either utilized for an onboard manufacturing process or bundled together and launched to an appropriate space construction location.

Mr. E. Panzeria's concept will be referred to as the material package stream transport system (MPSTS) or the open mass driver system (OpenMDS) in subsequent Sections of this document. Section 2.5 and 2.6 will review the previously proposed launch mechanisms and 'catcher' satellite concepts that would be considered for use in the mass driver system design. Section 2.7 reviews tentative space construction projects that the mass driver system could be used for the supply of raw material.

The mass driver system is presumed to be more beneficial than other direct space launch systems primarily for two reasons:

1. The drag reduction experienced by the series of material packages might significantly reduce the launch requirements to reach orbit. Therefore, the design of the launch mechanism might be of lower performance than previously proposed direct launch systems.
2. The material package mass is intended to be small, in the order of grams to kilograms. Therefore the design of the launch mechanism should be easier since the performance requirements should be lower than previously proposed launch systems which tend to be designed for hundreds to thousands of kilogram launch masses.

The goal of this thesis project was to establish an open engineering development community to design and evaluate the physical feasibility and design of the aforementioned mass driver system. Chapters 3 and 4 will detail the preliminary design and feasibility review for the mass driver system performed solely by the author. Chapter 5 will review the efforts to establish a development community for the project and the design results of the development community. Chapter 6 summarizes the design process and has recommendations for further work.

Chapter 2 Literature Review

2.1 Open Source Software (OSS) Development

At this time, there is no standard method for organizing an Open Source Software (OSS) project; however, there is a general structure. Typically a founding person^a or group of dedicated experts^b initiates the project and the combined work of a community of volunteers advances the project. The fundamental tasks of the founding person or group are the same [4]:

1. State the requirements for the software product.
2. Divide the software product into work modules.
3. Present and promote the project on the Internet.
4. Select the 'best' efforts (code and documentation) of the developer community for compilation into a functional new version of the software product.

Note that item #2 requires the founder(s) to complete a significant amount of system design of the software product before dividing the work.

The developer community then is responsible for the following [4]:

1. Writing the programming code to complete the work modules.
2. Create documents explaining how code modules operate.
3. Create user manuals for the software product.
4. Test the functional versions of the software product for errors.
5. Report errors in the software product to the developer community.
6. Create and submit 'fixes' for the reported errors.

The development pace of an OSS project typically depends on the number of developers participating and their skill and enthusiasm. Many OS projects do not proceed or succeed because

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^a Linus Torvalds for the Linux operating system for example

^b An expert group for the Apache web server for example

the project fails to attract or keep a critical number of active developers. This critical number is unknown and variable [4].

A larger development group generally leads to a higher quality product because there are more eyes to find the errors and there is more competition amongst the developers to create superior work.

From case studies [4], it seems the successful OSS projects require a dedicated founding group to create a preliminary 'prototype' version of the software product to attract other developers. The Linux operating system began as a hobby of Linus Torvalds to create an operating system for his personal computer. He initially posted a message on Usenet as a survey of what people like and dislike about available operating systems. In few weeks, he then wrote Linux 0.01 himself and made it available to others on the internet. Enthusiasm for the Linux 0.01 was immediate; others downloaded, tested, tweaked and sent the modifications back to Torvalds. At the time of writing, this development cycle for Linux still continues to improve the program [6].

The purpose of the preliminary version of the software is to demonstrate the potential benefits and usefulness of the software product and the ability and dedication of the current project members. New members join feeling confident the project will continue and their works have the potential of becoming part of the product. The OSS development method can be considered more of an optimization process; the 'prototype' product created by the founder, is augmented by development community according to the 'wish' list created by the founding group. It is not known if an OSS project that requires discovering new knowledge or developing new technology can be successful [4].

2.1.1 Advantages of OSS Development

OSS development has created products that are of high quality and compete with commercial software products. OSS development process has the following advantages as compared to commercial software development [4]:

1. A Highly Parallel Development Process that Increases the Speed and Quality of Development

As mentioned in the previous Section, the project founder divides the project into a number of separate, independent work modules. Since each module can be worked on separately, the speed of the completion of the entire program can be greatly accelerated. Successful open source projects have a developer community that can be much larger than any commercial enterprise. Therefore, an open source project can be completed more quickly than commercial products because there is simply more people to work on the work modules.

Since there can be a large number of developers working on a project, it is common that many developers are working on the same work modules independently. This redundancy of labour in a commercial project could be viewed as a wasteful use of resources; however for an OSS project, since labour is essentially free, this redundancy is a powerful advantage because it creates competition among developers. Since, only the best work module will be selected to be included in the final version of the software product, developers will strive hard to ensure they submit the best work they can to ensure their work is included in the final product. Therefore, this redundancy of labour in OSS projects improves the quality of the product.

2. Involves a Community of Globally Distributed Developers

Large open source projects have the potential to attract developers from around the world. Internet technology allows users from around the world to participate and benefit from the project. The participation of people from different countries can identify special needs of people around the world, thus, the software product is more universally useable and more suitable for global markets.

3. Utilizes a Truly Independent Peer Review Process

Participants in an OSS project are not linked together by the traditional business hierarchy and professional or personal associations; therefore, the work submitted to the project receives an unbiased and honest evaluation of the quality of the work. Since the developer does not know who will be reviewing their code, it is suspected that the developer spends extra time to ensure that the work submitted is of the highest quality. By submitting his best work, the developer is attempting to ensure his work is included in the final version of the program.

4. Bugs Are Quickly Identified and Fixed

Typically, contacting a business to report a problem with their product is not a convenient nor rapid means of fixing your problem. The internet tools implemented by an OSS project allow the person using the software to easily and rapidly communicate with the developer community to report bugs or shortcomings of the software product. This rapid communication allows bugs to be reported and fixed and new versions to be released at a rapid rate. This rapid communication between the users and developers improves the quality and usefulness of the software product.

5. Products are Developed by Motivated and Dedicated People

Unlike traditional, commercial software organizations, which currently have a problem attracting and retaining appropriately skilled developers, open source projects typically attract a highly dedicated and motivated people due to the nature of the project. The reputation of some highly, skilled developers working on the OSS project may help to attract more people to the project for the opportunity to work with these individuals and view there work. Since all participants of OSS projects are volunteers, these participants must have a genuine interest in the product and a desire to produce a successful product. Typically, people are participating in the project because they want to use the software product or enjoy solving challenging problems.

6. Users are Highly Involved in the Development

User involvement is important for developing a software product that fulfills the needs of the user. OSS projects require that, like the developer, the user be highly involved in the development process, because the users typically are responsible for testing, reporting bugs, creating documentation and defining the program requirements. The Linus Law states that, "Given enough eyeballs, all bugs are found"; therefore, a high degree of user involvement should yield a high quality software product.

2.1.2 Software Tools Used for OSS Development

OSS projects lack formal business organization; therefore, it is very important that the software tools used by OSS projects reduce the amount of managing and co-ordination efforts that are required to be done by the project founder. One of the most common software tools used is Concurrent Versions Systems (CVS). CVS is itself an OSS product. CVS was developed

to manage revisions to program codes submitted by developers and provide a means of easily making the most current version of the program code available to the developers and users for downloading. CVS greatly simplifies the file management requirements of the project. Internet communication tools like e-mail, newsgroups, discussion boards and web sites are extensively used for communication and collaboration amongst the participants of the OSS projects.

2.1.3 Taboos and Norms in OSS Projects

Many OSS projects have well documented guidelines for participant conduct. Most of the guidelines emphasize that every participant must practice respect for one another.

Typical rules are:

- Attach your name to all work that you submit.
- Plagiarism of code is forbidden; it is never acceptable to remove the name of the original developer from any portion of code and submit it as your own code.
- The existing maintainer of a section of code must be respected; if a change is to be made it, the maintainer should be advised.
- Every participant in the project is treated as an equal.
- All participants should be modest about their work.
- Forking of the project into two similar but separate projects must be avoided.

2.2 Open Source Hardware Development

When the Mass Driver project was proposed it was believed that it would be the first attempt at adapting open source principles to the design of a non-software related product. However, three open source hardware projects have been identified, the OSCar project, OpenCores and ThinkCycle. The following three Sections discuss each of the projects.

2.2.1 Open Source Car (OSCAR) Project

In Germany in 1999, the OSCar project was founded by Markus Merz, a former BMW manager turned e-commerce consultant, as a project to design a “world” car using open source software principles. The project had the following goals:

- Create a superior automobile design, which utilizes a global input of perhaps millions of automotive engineers and enthusiasts rather than input of a few senior executives as per conventional design processes.
- The automobile design should be modular to allow different design versions to be interchangeable.

- The automobile design should be innovative, incorporating features that not currently available on automobiles.
- Environmentally friendly powertrain options, like fuel cells, hybrid-electric or pure electric should be considered as a possible design option.

Merz and his consultant company created a web site, www.theOSCarproject.org, which provided participants with tools such as newsgroups, on-line chat, electronic white boards, forums and databases to aid the design process. The design of the project was separated into work modules based on BMW's structure, such as a powertrain work module, body work module, electronics work module etc. Merz also created a manifesto, stating the project's goals and principles. This manifesto proclaimed that like OSS projects, all design information will be public knowledge; however, unlike traditional OSS projects, all major design decisions will be determined by a vote involving all participants [7].

The project attracted a large community of mostly German automotive design engineers who generally had an enthusiasm for cars but experienced creative oppression in their current jobs at large automotive firms. The community decided that the Catia CAD program and its open source clone OpenCascade^{*} would be used as the universal 'language' for sharing design information. Catia and OpenCascade were chosen because it was familiar to many professionals and both Catia and OpenCascade are capable of operating on home PCs. After approximately 36 months, a final design of the car was achieved and a prototype was built [8]. The prototype was built to entice an automotive manufacture to produce the design; however, at the time of writing, it seems the project failed to attract a manufacture. Additionally, the project was slated to restart in November 2003; however, at the time of writing, the project appears to be at a standstill.

2.2.2 OpenCores Project

The OpenCores project was founded by Damjan Lampret, in Slovenia, in 2000, as an attempt to apply OSS development principles to design of digital circuits, commonly known as 'cores'. The cores are a collection of millions of digital gates that form the integrated circuits that are imprinted on to microchips. The complexity of the cores has increased with the improved performance of microchips; this complexity has caused an increase in the development time and

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^{*} Open Cascade Web Site - <http://www.opencascade.org>

cost of cores. To mitigate the increased development time and cost, the OpenCores project proposes creating a library of core designs that are free for all to use. It is assumed that OpenCore designs would be of high quality since a large user/developer community would exist to create, document and improve the design of OpenCores. Prior to OpenCores, all core designs were purchased from commercial vendors; however, these designs were typically poorly documented and difficult to integrate into the design of the complete microchip.

The main objective of the OpenCores project is to develop open source designs for cores that are freely available, usable and reusable. Other objectives for this project are to provide complete documentation for OpenCore designs, develop standards for OpenCore designs and create tools and methods for open source hardware design [9].

At the time of writing, the OpenCores site hosted 247 core projects, with about 51% of these design projects complete and ready for manufacturing. The project organization is similar to open source software projects; each project is started and supported by an individual known as a maintainer. The OpenCores site has a 'classified' section to advertise for help on the project. An interested individual can view the work modules requiring completion and submit their own work. Like many OSS projects, OpenCores web site uses the CVS program to record the revision history and provide the most current version of the design to the community. The web site also offers, design and testing software to ensure all contributors are using compatible software for the design process.

Given that relatively large numbers of OpenCore projects have been complete, the OpenCore project has been successful at attracting a community of developers and users. According the web site, the developer/user community is a dynamic group, meaning that individuals regularly join and leave projects. This group represent a range of people from students to professionals who are primary motivation to participate is entertainment and personal interest [9].

With more than half of the OpenCore projects completed at the time of writing, it suggests that the open source design of computer hardware is viable. At the time of writing, at

least one new OpenCore project was created suggesting the growth of the OpenCore site will continue.

2.2.3 ThinkCycle: Open Collaborative Design

The ThinkCycle project is an initiative started by MIT graduate students to attempt to use open source software development principles to solve humanitarian problems of developing countries. These problems include topics such as increasing food production, providing clean water and providing better healthcare. The idea was to create a web accessible database, at www.thinkcycle.org, that could receive real-world problems for developing countries and provide a space to store all the materials and documents related to the solution of the problem. The name ThinkCycle was chosen because the term summarizes the notion that designing a solution to a problem requires the people to *think*, propose a preliminary design and then repeat this design *cycle* until a satisfactory final design is created [10].

The ThinkCycle concept subsidiary goal was to change the way engineering design is taught at university. Traditional engineering design courses typically give students a problem that already has a solution, therefore, the students can either optimize the existing solution or reinvent the wheel. Engineering designs courses using problems from ThinkCycle challenge the students to solve real-world problems with no present viable solution.

The typical ThinkCycle project proceeds according to the following steps:

1. A non-government agency (NGO) or other stakeholder submits a problem to the ThinkCycle web site.
2. Domain experts review the problem and decide if the problem is appropriate to release to the ThinkCycle database.
3. Design faculty select problems from the ThinkCycle database to issue to their student design teams.
4. Student teams work the problem by brainstorming, setting design restraints, investigating possible technology, proposing preliminary concepts, developing the final design and building prototypes. The information created during each phase of design is posted on the web site in the "ThinkSpace" for the project.
5. During step 4, a peer review, by domain experts and other collaborators, occurs to provide advice and resources to the student teams.

6. The students end the project with a final report and presentation that is made available to the NGO, other stakeholders and to the public via the web site.

Unlike OSS projects, where the developer community could be comprised of nearly anybody, students are mainly responsible for the design work for ThinkCycle. The roles of the typical non-student participants for a ThinkCycle project are to provide problems, advice and resources to support the student design teams [10].

At the time of writing, the project web site, www.thinkcycle.org, had 149 posted problems and 55 design projects working on a solution to a posted problem. Three of these design projects have resulted in the construction of a successful prototype; these projects are an inexpensive device for cholera treatment, a passive incubator for premature infants and bio-sand filters for water treatment. The most successful project was the design of the device for cholera treatment. The treatment of cholera typically requires saline to be pumped into the patient. Developing countries require inexpensive and easily operated medical equipment to administer saline. The typical devices used for treatment are either a computer controlled IV pump which is easy to use but costly or an IV drip valve that is cheap but not easily used by unskilled people. A group of MIT students, lead by Professor Timothy Prestero, collaborated with physicians and various engineers to explore various possible design concepts for a cheap treatment device. The student team used the suggestions from the collaborators to propose a concept design that is based on rotameters used by chemical engineers. The proposed device cost about \$1.25 to manufacture and is intuitive to use, so even untrained people can use the device. The design is so successful, negotiations with a medical equipment manufacture occurred and MIT applied for a patent on the device. The other two projects are not as successful and are in the prototype stages.

The application for a patent raises a legal issue for open source hardware projects. Should anybody own patent rights to a device whose final design is the result of contributed work of many people and all the design work is publicly available on the internet?

2.3 Proposed Direct Space Launch Systems

Directly launching a payload from Earth surface to space has several advantages over using a conventional rocket to place material into space. Firstly, a direct launch system potentially can offer substantially lower launch costs because most of the infrastructure for the system is reusable

and remains on Earth; unlike rockets which are costly, complex, single use machines. Also the recurring launch costs of a direct space launch are low, since ideally only the launch energy input and the launch vehicle are consumed during each launch. Launching bodies at hypersonic velocities is advantageous since the aerodynamic drag approaches a minimum value at high hypersonic velocities (Mach 7 and greater see Figure 3.7 for example). Therefore, launching bodies at high hypersonic velocities is energy efficient since the drag is reduced. The total mass of these launch bodies is also reduced since it does not require a propulsion system for atmospheric flight.

A body launched directly from Earth has a few limitations. A body directly launched from Earth will travel a ballistic trajectory that will return to Earth unless either a second impulse (propulsive force) occurs to place the body in Earth orbit or the body escapes Earth's gravity field. Launching projectiles at high hypersonic speeds from the Earth surface is technically difficult to achieve. The design of these launch bodies is complicated by the high mechanical and thermal loads. The design of the launcher mechanism is difficult because the launch process requires the launch vehicle to achieve high velocity and energy in a short time period. Many concepts have been proposed; however, the research into the technology for hypersonic launch of large projectiles is not sufficiently mature.

The following sub-Sections of 2.3 review a few of the previously proposed direct launch concepts. These concepts range from systems that are technically feasible with present technology to concepts that will be feasible in the future when the required technology and materials mature. The sub-Section of 2.3 is intended to introduce the complete launch system concepts. Section 2.4 will review two space launch concepts that can be considered the alternative to expendable rockets and direct space launch, the space plane. Section 2.5 will review the hypervelocity launcher devices proposed for direct space launches. Section 2.5 is intended to explain basic operating principles of most of the launcher technologies mentioned in direct space launch concepts of Section 2.3.

2.3.1 Bishop South Pole Accelerator

When the Mass Driver project was initiated, it was assumed that launching a series of bodies one after another from Earth surface to orbit was an original idea. However, through the literature review, it was determined that Mr. Forrest Bishop proposed a similar concept in his

unpublished work on the concept entitled the *South Pole Accelerator* in 1981 [11]. F. Bishop proposed using an electromagnetic coil launcher situated at the South Pole to launch a series of slender bodies, one after another. Bishop proposed that this rapid launch of in-line bodies would create a 'launch corridor' of rarified air (plasma) through the atmosphere. To maintain the 'launch corridor', Bishop estimated a launch frequency of at least 10 Hz was necessary. He suggested that the launch bodies could be coated with solid hydrogen or ice to help reduce the density of the air in the launch corridor and thermally protect the launch body. The launch bodies are to travel to a very high altitude where Earth's gravity force nearly matches the gravity force of the Sun. At this altitude, a miniature attitude control system could be used to position the launch body in Earth orbit with minimal additional propulsive impulse [11].

Mr. Bishop was contacted regarding providing further details of his concept, particularly the launch corridor. He responded that he would find his notes on the concept and post the information on his web site. At the time of writing, the details of the launch corridor did not appear on Mr. Bishop's web site.

2.3.2 ESA Nano-Satellite Rail Launcher

The European Space Agency (ESA) considered the feasibility of augmenting their current Pegasus railgun technology to be capable of launching small payloads to sub-orbital and orbital destinations. The development plan for this launcher is proposed to occur in two stages [12].

First, the sub-orbital launcher would be developed to launch 4 kg payloads to an altitude of 115 km. This sub-orbital launcher is intended to be a more cost-effective means of launching experimental payloads that are currently launched using sounding rockets. It was determined that a slender, cylindrical vehicle could be launched with a velocity of 2.15 km/s, at angle of 85° from the horizontal, would reach an altitude of 115 km. This vehicle would be 4 kg total mass,

Figure 2.1: Proposed ESA Sub-Orbital Rail Launched Vehicle [12]

including a 1 kg payload, and have a diameter of 50 mm and a length of 540 mm. Figure 1.2 depicts the proposed sub-orbital launch vehicle.

A CFD analysis, for the sub-orbital launch, determined that the vehicle would experience the maximum temperature of 1034K at the nose cone, approximately 0.1 seconds after launch. The duration of significant atmospheric heating would be approximately 40 seconds. Given these thermal conditions, it was concluded that conventional metals or high temperature composites would be suitable materials for the thermal loads.

The railgun launcher required to launch this sub-orbital projectile was determined to be approximately four times larger than the current Pegasus research facility. The length of the launcher would be about 22 meters and the energy bank would be 30 MJ. The efficiency of the railgun launcher was estimated to be about 30%.

The second stage of the development proposes a railgun launcher capable of placing 5 kg nano-satellite payloads into Low Earth Orbit (LEO). The cost of launching these nano-satellites is intended to be cheaper than the current cost range of about \$950, 000 to \$1 500 000 for piggy-back rides on large conventional rocket missions. The launch velocity, to reach LEO, was determined to be approximately 6 km/s at a launch angle of about 30° from the horizontal. The launch vehicle that would carry the 5 kg nano-satellite payload, would be a long slender cylinder with a diameter of 120 mm, a length of 2900 mm and a total mass of about 41 kg. The payload mass fraction for this concept would be about 12.2%. The vehicle total mass includes the mass of a hybrid rocket motor for orbital insertion of the satellite. Solid and liquid rocket motors were also considered, however, neither was selected since the solid fuel has a potential to be damaged during launch and liquid fuel propulsion was 5 kg heavier than the hybrid propulsion. Figure 2.2 depicts the proposed orbital launch vehicle.

A CFD analysis for the orbital vehicle launch, determined the thermal load on the vehicle to be significantly greater than the sub-orbital case. At 0.1 seconds after launch, the vehicle experiences a maximum temperature at the tip of the nose cone of 2900 K and 1700 K at the leading edges of the fins. The duration of the atmospheric heating was determined to be

approximately 20 seconds. It is likely that conventional refractory materials could be used as thermal protection.

Figure 2.2: Proposed ESA Orbital Rail Launched Vehicle [12]

The railgun required to launch the orbital projectile is about 180 m in length, about 30 times longer than the Pegasus facility, and the energy bank is 3.5 GJ, which is about 350 times larger than the Pegasus facility.

The study concluded the feasibility of both launchers is possible, however technical challenges exist. A staged approach to development would be beneficial since it allows the experience and technology improvements to be incorporated into the subsequent launcher design.

2.3.3 McNab's Rail Launcher Concept

Mr. Ian McNab proposed a direct launch concept similar to the ESA orbital rail launcher. McNab's goal was to create a direct launcher capable of directly launching a 1000 kg vehicle from the Earth surface to LEO at a frequency of a few launches per day [13].

McNab and Dr. M. Erengil first performed an analysis to determine a satisfactory launch velocity, launch angle, payload mass and orbital altitude. McNab and Erengil assumed the launch vehicle was a Rodman cone, with a base diameter of 1 meters and a length of 6 meters, and the vehicle was launched from an equatorial location, at a 2000 meter elevation. Figure 2.3 depicts the concept for the launch vehicle. From this analysis, they determined that a launch velocity of 7.5 km/s at an angle of 17.5° from the vertical would be capable of placing a 254 kg payload in a 544 km orbit. The payload mass fraction of this concept would be about 25.4%. At 100% launcher

efficiency, the launch would require 35 GJ of energy. This launch scenario represents a compromise between maximizing the mass of payload and minimizing the required launch velocity. The analysis did not consider the mass required for thermal protection, attitude control or vehicle docking/retrieval.

Figure 2.3: McNab Launch Vehicle Concept [13]

McNab considered eight different launch technologies, namely railguns, coilguns, electrothermal chemical guns, light gas guns, RAM accelerator, blast wave accelerator, slingatron and laser launchers. Railgun technology was selected since experimental results have demonstrated it can achieve a launch velocity of 7.5 km/s for small masses and further military railgun research was expected to occur. Railgun research is important for this concept since current railgun technology can not supply the required 35 GJ of energy to the launch body.

McNab concluded that the design of a railgun direct launcher was not impeded by any physical barriers and the cost of launching material to space could be as low as \$528/kg. Before a detailed design of the system occurs, McNab recommended a number of issues must be investigated. Most significantly, the performance of railguns operating with large launch masses must be tested and verified. The useful lifetime of present railgun design must be increased to ensure a launch system is capable of frequent launches. Additionally, the thermal protection and aerodynamic conditions affecting the projectile must be analyzed in detail.

2.3.4 DARPA Satellite Light Gas Gun Launch Concept

The Defense Advanced Research Project Agency (DARPA) funded a study to investigate the technical and economic feasibility of directly launching a constellation of small communication satellites, each with a mass ranging from 10 kg to 1000 kg, to a 700 km circular LEO [14].

The study proposed utilizing light gas gun technology. Unlike other launch technologies, light gas guns have a proven capability of routinely achieving the launch of small masses to high velocities. Hypervelocity experiments routinely use light gas guns to achieve velocities of 8 km/s.

A dominant goal for the proposed light gas gun launch system is it must be affordable. It was decided that launching a satellite payload with a mass of 113 kg (250 lb) would represent a good compromise between the estimated \$2 billion for a system for launching 1000 kg satellite payload and the limited capability of a 10 kg satellite launcher. The total launch mass was estimated to be about 682 kg which includes the 113 kg satellite payload, 223 kg carbon composite thermal protection system, 14 kg hydrazine fuelled attitude control system and a 250 kg solid rocket motor. The payload mass fraction was estimated to be about 16.5%. The solid rocket motor is capable of supplying a velocity change of 2.1 km/s to circularize the satellite in a 700 km orbit. The launch vehicle is a slender cone with a length to diameter ratio of 11. The structural mass of the vehicle is reduced by situating the satellite payload at the base of the cone and the delicate rocket motor at the top of the cone. Figure 2.4 depicts the conceptual design of the launch vehicle. The launch velocity for the vehicle to reach a 700 km orbit is estimated to be 7 km/s at a 20 degree launch angle.

Figure 2.4: DARPA Launch Vehicle Concept [14]

The proposed light gas gun required to launch the vehicle has a length of 1.52 km, a bore of 0.635 m (25 in) and would require about 2 700 tonnes of high quality steel. Instead of just injecting hydrogen gas at the breech of the launch tube, the study proposed that fifteen injectors distributed along the side of the launch tube could be used to achieve a nearly constant acceleration of the launch vehicle. Each gas injector requires a fast acting valve capable handling high pressure, about 70 MPa or 10 000 psi, and have an opening time of about 1 ms. The cost of the hydrogen gas required for each launch represents about \$700/kg of the launch cost; to reduce the launch costs it was recommended that the gas should be recycled at the end of the launch. To recycle the gas, the end of the launch tube requires a fast acting valve to allow the vehicle to exit without allowing the hydrogen to escape.

The study concluded that using a light gas gun is technically and economically feasible. Further work is required to evaluate the design of the thermal protection system and the aerodynamic stability of the launch vehicle; however, a successful design is likely with current

materials and methods. The development of the fast acting valves for the hydrogen injectors was identified as critical issue for the design of the light gas gun launcher. The study considered the tentative design of a 113 kg communication satellite payload and concluded that this mass is likely too small to be useful for communication satellites. About 75% of the 113 kg satellite mass would be structural material and a solar power energy generation system. Only about 3.2 kg would be available for the components for the communication satellite. Therefore, the study concluded that the 113 kg satellite payload would be useful for satellites that require smaller solar power systems. Further investigation would be required to determine the optimum mass of a communication satellite. The cost to launch the 113 kg payload was estimated to be about \$5500/kg, assuming 300 satellites are launched per year. The cost is comparable to low cost Russian rocket launches but much lower than other rockets systems.

2.3.5 Palmer and Dabiri Electromagnetic Launcher Concept

Miles Palmer and Ali Dabiri proposed creating an electromagnetic launch system capable of launching a 1 kg nano-satellite payload into LEO. The authors proposed that these small payloads can be small independent satellites or a part of a constellation of satellites in LEO as a replacement for a geosynchronous communications satellite. Alternatively, the payload launched can be used to send a resupply or refueling vehicle to satellites or space stations [15].

Palmer and Dabiri performed a detailed analysis to determine the launch parameters capable of placing a 1 kg satellite in LEO with minimal launch energy. The analysis included the variation of rocket motor mass needed for different launch parameter and the variation of the coolant mass need for cooling the projectile during atmospheric transit for different launch parameters. The analysis determined that the optimum launch velocity was about 5.5 km/s and the optimum launch angle was in the range of 22 -26°, these parameters minimized the launch energy to 1400 MJ (assuming 100% efficiency). The total mass of the launch projectile was estimated to be 18.6 kg, including 0.3 kg of coolant mass. The payload mass fraction for the concept would be about 5.4%.

The proposed launch vehicle was a slender cone; this geometry minimizes aerodynamic drag force and moves the center of gravity rearward to allow a statically stable flight body. It was

determined that the nose tip of the slender cone would require active cooling during atmospheric transit. Lithium was assumed as the coolant material for the nose tip, since it has a high specific heat. The authors postulated that the total mass of the projectile may be reduced by using a simple, pyrotechnically-timed fuse to initiate the insertion motor at ballistic apogee, instead, of utilizing an electronic trigger mechanism for the orbital insertion motor.

Palmer and Dabiri selected the launch mechanism to utilize railgun technology, since this technology has demonstrated launch velocities in the range of 6 – 10 km/s for small masses. The railgun required, is a scaled up version of military designs, with a bore of 0.6 meters and a length of 250 meters. The relatively short length of the launcher allows the railgun launcher to be capable of adjusting the launch direction and launch angle to allow missions to a greater number of Earth orbits. The shorter launch length subjects the launch vehicle to acceleration forces of approximately 64 000 Gs. Typically, undeveloped and costly pulsed power energy storage systems have been proposed for space railgun concepts. Palmer and Dabiri proposed that a bank of lead acid batteries charging a large inductor would be an inexpensive means of energy storage for the railgun instead of implementing pulsed power technology. The electricity for the railgun is to be ‘turned on’ using explosive switches; since an explosive switch minimizes electric arcing through actuation by a pyrotechnic charge.

Palmer and Dabiri’s study suggests that an electromagnetic space launch system is a near term possibility without any obvious physical barriers. The railgun for launching 1 kg payloads could serve as a technology demonstrator, a scaled up version could be implemented to launch conventional large satellites. The cost of launching payloads could be as low as \$20 to \$80/kg range. Commencing research into the use of a battery bank and inductor energy supply system may lead to a successful alternative space launch system. However further work is needed to develop the explosive switches required for the proposed railgun design.

2.3.6 NASA Nuclear Waste Disposal Rail Launch Concept

In 1982, NASA contracted Columbus Laboratories to study the feasibility and viability of an Earth to space electromagnetic (EM) launch system capable of launching to Earth orbit and beyond. This study considered using EM launchers for applications ranging from deep space probes to a booster for chemical rockets to atmospheric experiments; however, only two of these

applications were selected to be studied in detail. The first application was the launch of small satellite payloads to Earth orbit and second application was disposing of high-level nuclear waste by launching it out of the solar system [16].

The final concept proposed a launch facility that is capable of launching both orbital payloads and nuclear waste disposal missions. It was originally considered that a single railgun launcher capable of adjusting its launch angle would be used, however this design option became technically demanding since the length of the required railgun launcher would be approximately 2040 meters. It was decided that each application would have a dedicated railgun launcher. The launch rail for the orbital application would be inclined at a 20 degree angle from the horizontal, to minimize the size of the orbital insertion motor required. The railgun launcher for nuclear waste disposal would be oriented vertically, to minimize atmospheric drag. The launch velocity for the orbital application was estimated to be between 5 and 10 km/s for sending a launch vehicle, with a high ballistic co-efficient, to LEO. It was estimated that a launch velocity of about 20 km/s was required to launch nuclear waste out of the solar system. The analysis estimated that atmospheric drag accounted for about 5% of the launch velocity with the pressure drag being dominate and skin drag only accounting for about 1 to 2 % of the total drag. The maximum acceleration experienced by the payloads was estimated to be 2500 gee for an orbital payload and 10 000 gee for a nuclear waste payload.

The railgun design for both applications would be similar. The air would be evacuated from each railgun launch tube to reduce drag on the launch vehicle. The vacuum at the end of the launch tube would be maintained by a diaphragm that would be ruptured by a laser just prior to the vehicle exit. The electrical input to each railgun would be supplied by a nuclear reactor. Since the launch event requires a large amount of electricity over a short period of time, a means of storing and then quickly delivering this energy is needed. Therefore, each railgun would use homopolar generators (HGP) for energy storage since HPGs represent a technology that might require less cost and development than other technologies like battery systems, capacitors and explosive flux generators. Each railgun would utilize a distributed energy supply system, that is, current would be fed into the rails at a number of points along the rail rather than just at the base. A distributed energy supply increases the efficiency of the railgun but increases the complexity of

the railgun design. The supply current, for each segment of the railgun, would be turned on by a high current, solid state switch that is actuated by the magnetic field intensity of the previous segment of rail. The study noted that technology for these field actuated switches, capable of handling high current loads, require significant research and development.

Both railguns were required to be capable of frequent launches; it was proposed that an automated milling and repair machinery could be designed to recondition the rails at certain time intervals. The lifetime of these space railguns could not be accurately estimated at the time of the study, since experimentation with high current railguns has not been performed.

The launch vehicle for each application was assumed to be a slender cylinder with fins for stabilization. The study recommended that further research should be done to investigate the effectiveness of fins and study spin stabilization as an alternative. The study noted that spin stabilization would require the design of a railgun launcher capable of spinning the launch vehicle during launch. The launch vehicle for the orbital application requires a rocket motor for orbital insertion of the satellite payload. A liquid rocket motor was sized to supply the 2.1 km/s velocity increase to place a 650 kg satellite into a circular 500 km orbit. It was postulated that the rocket systems could later be retrieved by the space shuttle and returned to Earth for reuse. The total mass of this launch vehicle was estimated to be 6500 kg including the satellite payload mass of 650 kg. The payload mass fraction for the orbital launch vehicle would be about 10%. The launch vehicle for the nuclear waste disposal application is simplified since no rocket motor is needed. This launch vehicle was estimated to have a total mass of 2055 kg including 250 kg of high-level nuclear waste payload. The payload mass fraction for the nuclear waste launch would be about 12.2%. Figure 2.5 depicts a schematic of the proposed launch vehicles and the launch facility.

The thermal analysis for the launch vehicles indicated that conventional materials like refractory metals could be used for the nose cone to handle the thermal and mechanical loads. Carbon-carbon composites could be used for the sides to reduce the overall mass of the launch vehicle yet provide adequate thermal protection. The total heat input into the launch vehicle would be less than the heat input experienced by typical reentry vehicles because the duration of the heating event is short. A hypersonic launch creates significant heating loads that last a few

Figure 2.5: Schematic of NASA Rail Launch Facility and Vehicles [16]

seconds versus the typical reentry heating loads that lasts a few minutes. However, the launch vehicle would experience higher temperatures than typical reentry vehicles.

The study also considered the environmental effects that the proposed railgun launch system would have. Like present rocket launches, the weather conditions for launches are important. Winds would have little effect on the launch operations; however, cloud cover would inhibit launch. It was estimated that the presence of clouds would increase the ablation, or wear rate, of the nose cone causing the destruction of the vehicle. Launch during precipitation would result in destruction of the launch vehicle. The launch would generate significant noise issues. The sonic boom generated at the exit of the launch rail was estimated to be lethal at a radius of 15 meters and 30 meters from the exit for an orbital launch and nuclear waste disposal launch respectively. The orbital and nuclear waste disposal launch would rupture eardrums at 40 and 75 meters from the exit respectively. Windows would be broken by an orbital and nuclear waste disposal launch at 120 and 240 meters from the exit respectively.

The study recommended the launch facility be located on a remote island near the equator because it offers many advantages. High-level nuclear waste is a security risk; a remote island

location offers an easier location to guard the material. The equatorial location with a launch due east maximizes the use of earth's rotation for placing payloads in equatorial orbits. The remote island offers a location that is isolated, so the sonic booms generated would pose a risk to fewer people. Finally, a remote island offers a 'safe', low risk location for aborted launch vehicles to crash since the island is surrounded by an ocean.

The study recommended continued research into military railgun technology may lead to a space launcher application. No technical barriers to the development of a space railgun launcher were identified. The progress of railgun technology up to 1982, suggested that a practical system would likely be available in the 2020 time period.

2.3.7 Molder Distributed Propellant Launcher Rail

Professor S. Molder proposed a unique method of utilizing a solid or gel propellant to accelerate a launch body on a rail. The concept can be considered an externally fuelled single-stage rocket that is constrained to travel on a rail-road rail [17]. Figure 2.6 depicts a schematic of Professor Molder's Distributed Propellant Launch Rail (DPLR) concept.

Figure 2.6: Schematic of Molder's Distributed Propellant Launch Rail [17]

The rail-road rail has a slot milled along the top edge of the rail for the full length of the rail. The slot is filled with a solid or gel mono-propellant. The launch vehicle that rides the rail is essentially a diverging rocket nozzle that is fitted over the slot of mono-propellant. An igniter system is fitted to the launch vehicle just prior to the beginning of the rocket nozzle. The operation of this launch vehicle is simple; the igniter system initiates the combustion of the mono-propellant in the rail just ahead of the launch vehicle. The combustion gases then flow into the inlet of the rocket nozzle and out through the outlet at the rear of the launch body. The expansion of the combustion gases through an increasing area rocket nozzle creates a propulsive force that accelerates the launch vehicle forward. Therefore, the launch vehicle does not carry propellant.

The performance and efficiency of the launch system is improved because the propellant expended is used only to accelerate the igniter, rocket nozzle structure and payload mass. Unlike conventional rockets where the propellant is expended to accelerate the total mass of the storage tanks, rocket motor, remaining propellant and payload.

For a base case, S. Molder assumed the mass of the launch vehicle, length, nozzle half-angle and nozzle exit diameter to be 700 kg, 2.32 m, 10 degrees and 0.82 m respectively. The rail for the base case was assumed to be 1.5 km in length and have a 3 cm by 5 cm slot along the total length. The mass of the propellant loaded into the rail was assumed to be 2864 kg. The resulting final velocity of the launch vehicle traveling with drag equivalent to sea-level (1 atmosphere) was calculated to be 8.12 km/s with a maximum acceleration of 2 382 Gs.

The base case values were varied individually to determine the effects different launch parameters would cause. Molder determined that operating the launch vehicle in a vacuum would have a small effect on the final velocity; final velocity for operation in a vacuum would increase 2.2% from 8.12 to 8.30 km/s. Changing the total mass of fuel in the rail via altering the dimensions of the slot caused a significant change in performance. Changing the propellant mass from 1909 to 5728 kg, increased the acceleration and velocity by a factor of 2 and 1.43 respectively. Changing the total mass of the launch vehicle also has a significant effect on performance. By increasing the mass of the launch vehicle from 700 kg to 1000 kg, the final velocity decreased by 17.6% from 8.12 to 6.69 km/s.

Molder concluded that the calculated performance of the concept is capable of accelerating launch vehicles to velocities required for direct launch and recommended that further work be done to investigate the interaction of the fuel combustion and launch vehicle dynamics. For this concept to operate properly, the combustion of the propellant must be matched to the speed of the vehicle. If the combustion process is too slow, the gases will not be properly expanded through the rocket nozzle. Likewise, if the combustion process is too rapid, then the combustion gases will not be utilized properly by the rocket nozzle.

2.3.8 High Altitude Research Project (HARP)

In the 1960's Gerald Bull lead a research program to utilize a conventional naval powder cannon to propel large payloads to high altitudes. The cannon used was a surplus 16 inch diameter naval gun, originally mounted in a battleship; however, this naval cannon was modified to increase the total length of the barrel. Special gun powder charges were used in the cannon to maintain the base pressure on the payload to accelerate it to high velocities. These payloads experienced accelerations of 5000 to 20 000 gee and exited the cannon at velocities up to 3.6 km/s [2]. Figure 2.7 is a photo of the HARP cannon launching a payload [18].

Figure 2.7: HARP [18]

It was demonstrated in 1968, that the payload could reach sub-orbital altitudes of 180 km. At this altitude, a stable satellite orbit can be established since it is above the sensible atmosphere limit of about 150 km. The payloads launched ranged from 68 kg to 909 kg in mass and demonstrated that complex components like electronics and solid rocket motors could survive the launch acceleration from a cannon. The payloads were typically used to obtain high altitude meteorological data that previously was accessible by only by sounding rockets. However, near the end of the program in 1968, launch vehicles known as the Martlet 2G-1 and Martlet 4, were being developed to reach LEO. The Martlets launch vehicles were staged rockets launched from the cannon and capable of inserting a small satellite in to LEO. The HARP program ended due to financial difficulties before either of the Martlets launch vehicles could be demonstrated.

HARP demonstrated that directly launching a payload from the Earth surface to space was a near term possibility. However, a gun powder cannon has a maximum launch velocity of about 4 km/s. This velocity limit results from the maximum velocity of the gun powder combustion gases. It is therefore unlikely that future gun powder cannons would be capable of launching payloads much higher than the 180 km achieved by HARP, since the launch velocity for HARP was near the maximum velocity. To date the results of HARP represents the highest launch energy achieved by any direct space launch system at 440 MJ.

2.3.9 StarTram - Maglev Space Launch Tunnel

In 2001, Plus Ultra Technologies Inc. proposed an electromagnetic space launch concept capable of launching reusable manned and unmanned vehicles, with a mass of 200 tonne to Earth orbit at an estimated cost of \$30/kg. The vehicle is capable of carrying a payload of 70 tonnes to orbit. The proposed system therefore has a payload mass fraction of 35% [19].

The StarTram concept proposes accelerating manned launch vehicles at 2.5 Gs down a 1280 km long launch tunnel to reach a final velocity of 8.023 km/s at the end of the launch tunnel. The launch vehicle travels down the launch tunnel located on Earth surface similar to a Maglev train. A superconducting magnet onboard the launch vehicle interacts with an aluminum guide track located in the launch tunnel to levitate the vehicle. A linear synchronous motor (LSM) is incorporated into the guide track to propel the vehicle at nearly 100% efficiency. If the launch tunnel is evacuated of air, it is estimated that about 20 GW of electricity should be capable of accelerating the vehicle to 8 km/s. To further minimize the aerodynamic drag experience by the launch vehicle, the concept proposes that the vehicle enter a second evacuated tunnel angled at 5 degrees from the horizontal to guides the vehicle up to an altitude of 22 km. At 22 km altitude, the vehicle exits the tunnel at 8 km/s and travels through the remaining atmosphere experiencing significantly reduced drag and aerothermal effects (air density is 5% of sea-level values). The end of the tunnel utilizes a high velocity stream of air or steam to maintain the vacuum in the tunnel yet allows the vehicle to exit.

Constructing a structure to support the guide tunnel to a height of 22 km using conventional materials would be a challenging task. This concept proposes that superconducting cables located on the guide tube and ground below the tube could be used to generate a magnetic repulsion force capable of supporting the weight of the guide tunnel. It has been estimated that a total current of 14 mega-amperes and 280 mega-amperes in the superconducting cables located on the guide tunnel and ground respectively, would generate a repulsive force of 4 tonnes/m at an altitude of 22 km. The guide tunnel would be stabilized against disturbing forces like winds using a network of high strength polymer cables moored to the tunnel and the ground. Figure 2.8 is a sketch of the launch vehicle exiting the end of the magnetically levitated launch tube.

Figure 2.8: Sketch of the StarTram Concept [19]

The authors of the concept acknowledge that the concept proposed is advanced. However, the authors insist the design requirements can be fulfilled with existing materials and superconducting technology and appears to be economically and technically feasible. Future advancement in high temperature superconductors would make the design less complicated and more cost effective.

2.3.10 Lofstrom Launch Loop

Mr. Keith Lofstrom proposed using a very long ribbon of iron traveling at high velocity in a loop as a means to store and deliver energy to launch a five tonne payload to geosynchronous or near lunar orbits [20]. Lofstrom likened the operation of this ribbon of iron to a stream of water from a hose. If the stream of water were angled upward to the sky and traveling in a vacuum, it would follow a parabolic ballistic trajectory back to Earth. If the water stream encounters a flat plate with a slight downward angle, the stream of water will be deflected downward and the plate would experience an upward force. Suppose the entire stream is encased in a frictionless hose, then the downward deflection of the stream by the hose could create an upward force large enough to support the weight of the hose. The water stream travels back to Earth and can be recycled and pumped back into the beginning of the hose to create a closed loop. Instead of a stream of water, a ribbon of flexible iron traveling at high velocity could create an upward force capable of supporting a stationary mass without the need for structural support materials.

Lofstrom proposed a loop of iron with a cross-section of 50 mm wide by 7.6 mm thick traveling at 14 km/s could be used to support a launch track at an altitude of 80 km above Earth's surface. At an 80 km altitude, conventional construction materials cannot be used to create the massive support towers required;

Figure 2.9: Schematic of the Launch Loop [20]

therefore, the momentum of the iron ribbon is used to support the launch track. The launch loop begins at the Earth's surface near the equator, where the iron ribbon is launched at 14 m/s at angle of 20 degrees from the horizontal. Figure 2.9 depicts a schematic for the Launch Loop. The ribbon travels up to the 80 km altitude where it encounters an electromagnetic deflector station that redirects the ribbon in nearly a horizontal direction along the launch track. At this first electromagnetic deflector station, 5 tonne launch vehicles are loaded on to the launch track. The launch vehicles have a magnetic base that interacts with both the high velocity ribbon and permanent magnets located on the launch track to accelerate and levitate the vehicle. The vehicle is accelerated at about 3 Gs, down the 2000 km launch track to a final velocity of 10.5 km/s and exits the launch track. At the end of the launch track, the ribbon encounters a second electromagnetic deflector to redirect the ribbon at a 20 degree angle back towards Earth. Once the ribbon reaches Earth, it is accelerated using linear induction motors back toward the initial launch site to complete the loop.

The launch loop has demanding design requirements. To reduce atmospheric heating, the iron ribbon is to be enclosed in high vacuum sheath structure that is also must be supported by the momentum of the iron ribbon. The launch loop is stabilized in the position against disturbing forces like wind using long cables extending from 80 km altitude to Earth. The total length of the required iron ribbon loop is about 4900 km and its mass is about 15 600 tonnes. The kinetic energy stored in the ribbon rotating at 14 km/s is 1.5×10^6 GJ, and would require a typical 500 MW

power generation station about 60 days output to accelerate the ribbon to full speed; the power loss to the deflectors and other losses accounts for about 200 MW at full speed operation. A 500 MW power supply, would capable of launching 35 vehicles per day.

Lofstrom acknowledges the design requirements are demanding, and the system may ultimately be impractical or too expensive if research into the concept continues. However, through proposing this concept, it is hoped that it may inspire others to propose other low cost and high volume launch systems based on ideas from his concept.

2.4 Other Novel Space Launch System Proposals

This Section will introduce two concepts that propose using a reusable space plane to transfer material to orbit. The concepts have been proposed as a more cost-effective method of transport than the current expendable rockets and Space Shuttle vehicles. The both concepts, the ESA Hopper and HASTOL system, are reusable space plane systems capable of taking off horizontally, delivering the payload to orbit and landing horizontally, much like a conventional aircraft. However ESA design focuses on providing a near term system by utilizing existing rocket motor designs, the HASTOL design focuses on providing a system for future use because some critical technology must be developed to make the HASTOL system feasible.

2.4.1 ESA Hopper Space Plane

In 2003, ESA funded a study of a reusable launch vehicle, named the Hopper which was intended to be a unmanned, fully automated, reusable rocket powered launch vehicle. The Hopper vehicle is intended to achieve lower launch costs than conventional rocket technology. Lower launch costs are possible because each mission uses propellant more effectively to place the payload in orbit, the vehicle is reusable and the vehicle is less risky; therefore, a safer means of transporting payloads [21]. Figure 2.10 depicts a sketch of the Hopper concept.

Figure 2.10: ESA Hopper Concept [21]

The operation sequence of the Hopper is relatively straightforward. The Hopper vehicle complete with an internal payload ranging from 7.2 to 13.1 tonnes^c, takes off horizontally from a launch rail located near the equator in French Guyana. The vehicle is accelerated on the launch rail to velocity of about 185 m/s. The Hopper vehicle leaves the launch rail and burns the three main liquid rocket engines until it reaches about 100 km altitude. The vehicle coasts upwards on a ballistic trajectory to 130 km altitude where a blast of pressurized helium is used to discharge from the vehicle the payload and an upper stage rocket motor. The liquid rocket motor of the upper stage ignites and propels the payload to either a circular LEO or a Geosynchronous transfer orbit (GTO). The Hopper vehicle continues its ballistic trajectory and returns to Earth near Europe about 4500 km down range from the launch site. The Hopper vehicle is returned to the launch site by boat and readied for the next mission. It is intended that one Hopper vehicle could complete 10 to 20 missions per year.

The Hopper launch concept has a number of advantages. The horizontal launch rail enhances safety by mechanically guiding the vehicle during launch and reduces the mass of propellant and wing structure because the launch sled assists the vehicle to a high take-off velocity. Horizontal launch is desirable since it requires less thrust during takeoff than vertical launch; therefore, the main engines can be smaller or reduced throttle can be used to maximize engine life. The Hopper system is almost entirely reusable; the only expendable component for each mission is the upper stage. Unlike the Space Shuttle, the Hopper vehicle does not enter orbit; therefore, the total mass of the Hopper is reduced since less propellant, thermal protection and structural mass is required. The payload mass fraction of the vehicle is 1.5 and 2.5% for a GTO and a LEO mission respectively. Since the Hopper vehicle does not reach orbit, the design of the vehicle is less demanding. The design goal of the Hopper is to use existing technology, like the main engines from the Araine rocket, to maximize safety and reliability of the system while minimizing development and manufacturing costs.

It is intended that the Hopper System could be developed as early as 2015. However, at the time of writing this manuscript, the concept was not selected for further development.

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^c 7.2 tonne payloads can be placed in GEO and 13.1 tonne payloads can be placed in LEO.

2.4.2 HASTOL System

The Hypersonic Airplane Assisted Space Tether Orbital Launch (HASTOL) system is a joint proposal made, in 1999, by Boeing/MDC Phantom Work, Tethers Unlimited Inc. and the University of Maryland for a space launch system. The HASTOL system is intended to be a fully reusable system that is capable of placing payloads in orbit at cost less than expendable rockets. The HASTOL concept is novel because it eliminates the need to use conventional rocket engines to place material into orbit [22].

As the name implies, a hypersonic aircraft is used for the first stage of the payload transport. The hypersonic airplane takes off horizontally

Figure 2.11: Boeing DF-9 Hypersonic Aircraft Concept [22]

from a conventional runway carrying a 14 tonne payload. The study proposes using an existing concept design for the hypersonic airplane, the Boeing DF-9, depicted in Figure 2.11, to reduce the development cost of the system. The DF-9 aircraft is fitted with three distinct propulsion systems for use at different velocities. An air breathing turbo ramjet is used for velocities below Mach 4.5. An air breathing ram/scram jet is used for velocities above Mach 4.5 and a linear rocket system is used at altitudes above 100 km where air is rare. The DF-9 is intended to carry the payload to an altitude of 100 km, traveling at 3.6 km/s. At 100 km altitude, the DF-9 and the payload rendezvous with a grappling mechanism attached to the tip of a rotating space tether. The grappling mechanism^d captures the payload, removes it from the DF-9 and carries the payload into orbit. Figure 2.12 depicts the rendezvous of the payload and tether. The DF-9 returns empty to a conventional runway. The space tether is a 640 km long rotating cable that is located in an

^d See Section 2.6.2 for grapple mechanism details

elliptical orbit. The space tether carries the payload for one complete orbit and releases the payload into a stable elliptical orbit. The payload mass fraction for this system is estimated to be about 5%.

Unlike the conventional means of orbiting a payload by supplying energy from a

Figure 2.12: Rotating Tether During Payload Transfer [22]

chemical rocket, the space tether achieves orbiting of the payload by transferring a portion of its own orbital energy to the payload. Orbiting the payload reduces the energy of the space tether; therefore, the maximum altitude of the tether orbit, the apogee, is reduced. To return the space tether to its original orbit, the space tether uses an electrodynamic propulsion system to boost the apogee of the orbit without using propellants. An electrodynamic propulsion system boosts the altitude of the tether by supplying electrical current from solar panels to a 10 km long conducting wire. The current in the wire interacts with the Earth's magnetic field to create a force increases the apogee of the tether.

The HASTOL system is based on near term technologies; however significant advancement in the design of both the hypersonic aircraft and the tether is required. The DF-9 requires the development of the propulsion systems to make the design practical. According to the study, the design of the tether system is practical with commercially available material. However, since no orbiting tethers exist at the time of writing, significant research is required in two areas, the development of an electrodynamic propulsion system and the evaluation of the lifetime of the tether material in the space environment.

2.5 Mass Driver Launcher Technology

Section 2.3 introduced entire alternative systems for launching material to orbit. This Section details operation and capabilities of the proposed launcher technologies used for directly launching material from Earth surface to orbit. These launch technologies are grouped into three categories: chemical, electromagnetic and mechanical launch systems. Section 2.3 did not include a launch concept for a mechanical launcher because no detailed complete launch system could be found at the time of writing; however, a discussion of the operation of a mechanical launcher for hypersonic payloads can be found in Section 2.5.3.1. The launch technologies discussed in the following Sections will be considered as possible design solutions for the launcher for the MPSTS mentioned in Section 1.4.

2.5.1 Chemical Launch Systems

Chemical launch systems utilize the energy resulting from a chemical reaction to accelerate the payload. Primary benefit for a chemical launch system is the storage of the launch energy is generally easy and safely accomplished by storing the chemical reagents in tanks. Generally, however chemical launch systems are an inefficient means of accelerating payloads. The following Sections will briefly outline the technology and operation of a conventional cannon, a light gas gun, a RAM tube and an electrothermal launcher.

2.5.1.1 Conventional Cannon Launcher

Conventional cannon launchers use the chemical reaction of gunpowder to generate high pressure gas at the base of a projectile to accelerate the projectile down a barrel or launch tube. Figure 2.13 depicts a schematic for a conventional cannon

Figure 2.13: Conventional Cannon Launcher [23]

launcher. The molecular weight and temperature of the gas created by the ignition of gunpowder limits the maximum attainable velocity of the projectile to a range of 3 to 4 km/s [2]. The most successful attempt to use a conventional cannon for space launch was mentioned in Section 2.3.8

the HARP (High Altitude Research Project); however, a Canadian company, Columbaid^c is continuing the research into convention cannon technology for space launch.

The benefits for using conventional cannon technology for space launch is it is a simple, well understood technology that is capable of launching relatively large payloads multiple times per day. The HARP gun launched projectiles up to 900 kg and was capable of being reloaded and repositioned in about one hour; therefore, nearly 24 launches per day are possible. However, as demonstrated in the HARP project, a pure cannon launch with a maximum projectile velocity of 4 km/s cannot place payloads much higher than 180 km altitude. The velocity limitation of conventional cannons is the primary disadvantage for this technology. To increase the velocity of the projectile, methods such as using liquid propellants and distributing the gunpowder charges have been proposed.

2.5.1.2 Light Gas Gun Launcher

Light Gas gun launchers utilize a compressed gas injected at the base of the gun to accelerate a payload down a barrel or launch tube [14]. Figure 2.14 depicts a schematic for a light gas

Figure 2.14: Light Gas Gun Launcher [23]

gun launcher. Light gas guns represent a mature technology, since research into light gas guns began in 1948 as a means of performing hypervelocity experiments. In these hypersonic experiments, light gas guns operated with compressed hydrogen have accelerated small payloads to velocities of 10 km/s. A light gas gun, capable of firing large payloads at 10 km/s, would be more than capable of sending the payloads to LEO altitudes. However, light gas gun technology has two problems. Firstly, at launch velocities greater than about 8 km/s, the erosion rate of launch tube bore becomes significant. Fortunately, a launch velocity of 7 to 8 km/s is sufficient for a payload to reach LEO and reduces the erosion rate of the bore to acceptable levels. Secondly, since the high pressure gas is injected into the base, the payload experiences a non-uniform acceleration with the initial acceleration force being high at the base or breech then gradually

^c See Columbaid web site for launcher details. www.columbaid.com

reducing as the payload moves away from the breech. To rectify this problem, the light gas gun design in Section 2.3.4 has proposed the compressed gas would be injected at multiple points along the gun to create a more uniform acceleration. Figure 2.15 depicts a schematic of the distributed injection light gas gun launcher.

Figure 2.15: Distributed Injection Light Gas Gun Launcher [23]

A uniform acceleration reduces the peak acceleration force; therefore, the structural mass and gee hardening requirements of the payload would be reduced.

2.5.1.3 RAM Launcher

A RAM launcher is composed of two parts, first a launch tube filled with a gaseous propellant mixture and secondly a payload with a special geometry. To accelerate the payload, the payload is propelled into the launch tube with enough velocity to cause shock compression and

Figure 2.16: RAM Launcher [23]

combustion of the gaseous propellant in the wake of the payload [13]. Figure 2.16 depicts a schematic for the RAM launcher. Just like a conventional cannon, the combustion creates gas pressure to accelerate the payload down the launch tube. The movement of the payload causes the combustion, gas generation and acceleration to occur down the entire length of the launch tube.

Experiments with RAM launch tubes have obtained velocities of up to 2.7 km/s [24]. However, to achieve the velocities required for orbital launch, approximately 7 km/s, it has been estimated that the launch tube must be divided into 15 regions of gaseous propellant with different mixtures. Each of these regions must be separated from each other by a fast acting valve, door or barrier mechanism, which can quickly open when the payload approaches.

2.5.1.1 Electrothermal Launcher

An electrothermal launcher uses electrical current to improve the function of a conventional cannon [13]. Figure 2.17 depicts the schematic for an electrothermal launcher. Just like a conventional cannon, an electrothermal launcher uses the combustion of a gunpowder-like

material to generate a high pressure gas to accelerate a payload. However, to increase the pressure of the gas and the acceleration of the payload, electrical energy is used to excite the combustion gases into a plasma state. Electrothermal technology can be considered

Figure 2.17: Electrothermal Launcher [23]

an extension of conventional cannon technology. However, just like conventional cannon technology, electrothermal technology cannot achieve the velocity required for orbital launch.

2.5.2 Electromagnetic Launch Systems

Electromagnetic launch systems utilize electrical energy to accelerate payloads. Electromagnetic launch technology is potentially a means of launching payloads to velocities required for orbital launch with high overall efficiency. Electromagnetic technology allows for the acceleration force on the payload to be constant and accurately controlled; however, the requirements for the electrical energy storage and electrical switching (turning on and off the current) have limited the success of electromagnetic launch technology. The following two Section introduce two common electromagnetic launchers, the railgun launcher and the coilgun launcher.

2.5.2.1 Railgun Launchers

A railgun launcher is comprised of two components. The main component is a pair of parallel conductive rails connected to a source of direct current; one rail is connected to the positive polarity and the other connected to the negative polarity.

Figure 2.18: Railgun Launcher [23]

The second component of a railgun launcher is known as an armature. The armature slides between the two parallel rails and carries the payload. Figure 2.18 depicts a schematic of the railgun launcher. To launch the payload, the armature is positioned between the two rails to complete the electrical circuit. The flow of current through the armature generates a propulsive force, known as the Lorentz force, to accelerate the armature and payload. The Lorentz force

occurs because the magnetic field formed around the armature is perpendicular to the field formed around the rails; therefore the fields repel each other [12]. At the end of the rails, the armature and payload depart from the rails at a high velocity. Shortly after the armature and payload exit the railgun launcher, the armature and payload separate; the payload follows a trajectory to orbit while the armature falls back to Earth.

Experiments with railgun technology have had encouraging results for space launch applications. Laboratory experiments have demonstrated that small payloads, in the order of one gram, can be accelerated to velocities up to 12 km/s [2]. The findings suggest that there are no fundamental barriers to accelerating larger payloads to these velocities.

A railgun launcher has the potential to be a low cost space launch system for a number of reasons. The railgun launcher is a highly reusable system; generally, the only reoccurring cost for each launch is the cost of the electricity consumed and the cost of the armature. The cost of electricity to launch payloads to high velocities is negligible. For example the cost of electricity for launching a 1250 kg payload to a 260 km altitude would be about \$2200 or \$1.76 /kg [13]. This cost assumes a 30% launcher efficiency and a cost of electricity of 8¢/kWh. The railgun launcher is a relatively simple machine to construct, operate and maintain; therefore, the cost associated with each should be low.

Railgun launcher technology does have some significant technical issues to solve before a practical system can launch large payload, in the order of 10's to 1000's of kilograms. A fundamental problem with railgun technology is developing the equipment needed to handle the high current loads, millions of amps [12], for launching large payloads. Launching large payloads requires an energy storage system capable of storing gigajoules of electricity. Various technologies for energy storage such as batteries, capacitors and rotating momentum machines have been suggested and investigated, but further work is needed for a practical energy storage system. Launching large payloads also requires heavy-duty switches to turn on the power to the rails at the appropriate time. Developing switch technology capable of turning on and off millions of amps requires further development. The design of an armature and rails capable of conducting millions of amps without significantly damaging the rails is another design area that must be developed

before a practical railgun launcher could be constructed. Many of the problems with railgun technology may be mitigated in the future because the technology receives extensive military research funding.

2.5.2.2 Coil-gun Launchers

The coil gun launcher, like the railgun launcher, utilize a magnetic field to accelerate a payload down a launch tube. Figure 2.19 depicts a schematic of a coil gun launcher.

Figure 2.19: Coil-gun Launcher [23]

The coil gun is similar to an electric induction motor, except it can be considered as unrolled and known as a linear induction motor. The stator coils or drive coils are stacked co-axially to form the launch tube. Like the railgun, the payload for the coil-gun requires a conducting armature to be attached or built into the payload^f. When the drive coil directly behind the payload armature energizes, a magnetic field is created; the expansion of the magnetic field induces a current in the armature. The interaction of the drive coil magnetic field and the induced armature current creates a force accelerating the payload forward [13]. A controller precisely energizes the drive coils sequentially according to the position of the payload. The result is the drive coils create a traveling magnetic field wave that is synchronized with the motion of the payload. The traveling magnetic field exerts a constant and controllable acceleration force on the payload. The magnetic field wave created by the drive coils center the payload in the launch tube; therefore, the operation does not suffer from the problem of launch tube erosion that plagues other technologies.

Major research and experiments with coil-guns were performed by Dr. G. O'Neill in the 1970's for his Mass Driver concept for transporting material from the lunar surface [25]. The Mass Driver concept enhanced the performance of the coil-gun by utilizing superconducting materials for the armature on the payload. The Mass Driver concept improved the overall efficiency of the launch system by recycling the payload armature and its kinetic energy back into the launch system. After the payload was launched, an additional section of coil on the launch

^f A payload composed entirely of a conducting material can serve as the armature but the performance would be poor due to Eddy current losses.

tube is used to slow the armature and generate electricity to feed back into the energy storage system. The theoretical efficiency of the Mass Driver system was estimated to be about 90% [2].

A practical coil-gun launcher for space launch purposes is not likely in the near term. The best coil-gun experiments have accelerated 180 g masses to 1.3 km/s [2]. Accelerating payloads to high velocities with coil-guns have been hindered by the capability of the switching technology to provide high voltages to quickly energize the drive coil. Maintaining a magnetic traveling wave synchronized with a high velocity payload is a challenging requirement for the switching technology. Like railguns, the energy storage requirements exceed present technology for launching large payloads.

2.5.3 Mechanical Launch System

Mechanical launch technology utilizes the momentum of a large moving object to accelerate a relatively small payload. The following Section details the operation of the Slingatron concept, a relatively new concept, proposed in 1998, for a rotating machine capable of launching payloads to orbital velocities.

2.5.3.1 Slingatron Launcher

The operation of the Slingatron launcher is similar to swirling coffee in a coffee cup by gyrating the mug. The mass of the coffee is constrained to travel in a circle by the coffee mug, however, a gyrating circular motion exerts a tangential force on the coffee that causes the acceleration. To increase the velocity of the coffee, the gyration frequency must also increase. Another everyday example for the Slingatron operation is the whirling of a mass at the end of a string. If the mass is whirled at a constant frequency and the string is lengthened, the mass at the end of the string will be pulled to a higher velocity. If the string is lengthened at a constant rate, the mass will travel in a spiral [26].

The Slingatron concept proposes that a payload could be accelerated by constraining it to travel inside a spiral shaped launch tube subjected to constant circular gyration. The circular gyration of the spiral shaped launch tube is performed by the rotation of a number of swing arm drive units distributed along the spiral and operated by electric motors or turbines. Figure 2.20

depicts a schematic for the Slingatron launcher. One of the main advantages, of the Slingatron concept, is the supply of power required for launch is distributed among multiple drive units and the supply can be spread over a period of time to slowly increase the gyration frequency. Unlike electromagnetic launchers, storing launch energy is simple, the rotational momentum of the spiral launch tube and swing arms act as the storage device.

To launch a payload, the spiral launch tube is accelerated to the required constant gyration frequency and the payload is injected into the launch tube near the center of the spiral. Given a sufficient gyration frequency, minimal friction forces and a long spiral length, the payload would exit the spiral at orbital launch velocities of about 8 km/s. This launcher is capable of multiple payloads launches at launch frequency equal to the gyration frequency; however, the drive units must be capable of maintaining a constant gyration frequency.

The most important requirement for the proper operation of the Slingatron is to ensure friction forces on the payload are minimized. The payload experiences two sources of friction, aerodynamic drag in the launch tube if air is present and sliding friction between the payload and the launch tube due to centrifugal forces. Research by D. Tidman indicates that if the launch tube was evacuated of air and payload was supported by a gas bearing, the friction forces would be sufficiently low.

A preliminary design for launching a 1000 kg cylindrical payload with a diameter of 0.64 m at 8 km/s has been proposed. The proposed launcher would require a gyration frequency of 4.75 Hz and the mass of the spiral launch tube and swing arms would be 9525 tonnes. The length of the spiral track traversed by the payload is 28 miles. For comparison a typical passenger car weighs

Figure 2.20: Slingatron Launcher and Swing Arm [26]

1.4 tonnes, therefore the rotating mass for this launcher is equivalent to rotating nearly 7000 cars at a frequency of 4.75 rotations per second. Practically, it does not seem realistic to construct a machine with such a large rotating mass moving at these frequencies.

2.6 Proposed Catcher Mechanisms

If a payload without an orbital insertion motor is launched to space directly from Earth, at a velocity less than Earth escape velocity, a means of capturing the payload in space is necessary to prevent it from returning to Earth. The following Sections review capture or catcher mechanisms that have been previously proposed. Section 2.6.1 reviews the catcher concepts developed for the Moon Mass Driver system proposed by G. O'Neill. Section 2.6.2 reviews the grappling mechanism for the rotating tether implemented by HASTOL concept described in Section 2.4.2. Section 2.6.3 reviews the proposed grappling mechanisms for the Momentum eXchange Electrodynamic Reboost (MXER) rotating tether system proposed by NASA and Tennessee Technological University.

2.6.1 NASA Summer Design Series Catcher Concepts

In the summer of 1977, NASA hosted a design symposium to propose a design for a system capable of transporting soil from the moon surface to space construction projects located at Lagrange points⁸. The proposed transport system was to use an electromagnetic coil launcher, a Mass Driver, to launch sintered packets or bags of lunar material from the Moon. Various catcher concepts were proposed to capture these material packages in space. The catcher concepts were classified in two categories, passive catcher mechanisms with no moving components and active catcher mechanisms with moving components.

2.6.1.1 Passive Catcher Mechanism

Passive catcher mechanisms rely on the deformation of materials to absorb the kinetic energy of incoming material packages. The first passive concept proposed was a concept that utilized a large disc of crushable material such as expanded foam or bonded glass fibers. The diameter of this disc depends on the launch accuracy and dispersion of the material packages; it

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⁸ Lagrange points are five locations in space where the Earth and Moon gravity forces are equal and opposite in direction, therefore a mass located at a Lagrange point is stationary relative to the Earth and Moon.

was assumed a 100 m diameter would be suitable. The kinetic energy of the incoming material packages would be dissipated by the deformation of the material. It was estimated that the material package traveling at 200 m/s would penetrate 1.3 m into foam material. After a period of time, the material with the lunar material imbedded is retrieved and processed to extract the lunar soil. This concept was not considered feasible since initially about 454 tonnes of material from Earth s required [27].

The second proposed passive concept was composed of two major components, a stationary rim with a grid of cables and a spinning Kevlar bag in the shape of a cone. Figure 2.21 depicts this concept. Like the previous passive catcher concept, the diameter of the rim was estimated to be 100 m to allow for dispersion of the material packages. The incoming material packages are broken up into fine particles when they strike the grid of cables attached to the rim.

The spray of fine particles hits the Kevlar bag and comes to a rest. The centrifugal force, created by the rotation of the bag, prevents the fine particles from escaping the bag. The concept was not considered feasible due to the uncertainty about the material packages breaking up and being retained as planned [27].

Figure 4-28.— *Passive bag-catcher concept.*

Figure 2.21: NASA Passive Catcher Concept [27]

2.6.1.2 Active Catcher Mechanism

The active catcher mechanism relies on moveable components to absorb and transfer the kinetic energy of the incoming material packages into another form of energy. The proposed active catcher mechanism operates similar to the arresting cable for the deceleration of aircraft on an aircraft carrier. Figure 2.22 depicts a schematic of this cable catcher concept. The proposed concept utilizes radar to detect the position of incoming material packages. The catcher system orients itself to intercept the material package in a cable netting similar to a volleyball net. Each

end of the cable netting is attached and wound around a reel. When the material package contacts the netting, the material package's momentum creates a force that causes the reels to rotate, letting out more cable. The motion of the rotating reels can be used to turn generators or flywheels to convert the kinetic energy of the package into a useful form of energy [27].

Figure 2.22: NASA Active Catcher Concept [27]

2.6.2 HASTOL Grapple Concept

The HASTOL concept, described in Section 2.4.2, required the end of the rotating tether to intercept a payload carried by a hypersonic aircraft. The grapple mechanism design is challenging because the rendezvous point only occurs at a narrow region of space due to the velocity difference between the aircraft and the end of the tether. In addition, the grapple mechanism must be capable of sustaining the acceleration force of the payload after it is captured. The preliminary concept for the grapple mechanism was proposed as a hook extended from the payload and an 'attaching' ring mounted on the end of the tether. The operation is simple; the aircraft travels a straight flight path with the payload hook extended above the aircraft and the 'attaching' ring at the end of the tether approaches the payload hook from behind. The hook latches on to the ring upon contact [22].

Figure 2.23 depicts the schematic of the 'attaching' ring assembly attached to the tether. The 'attaching' ring is connected to a shock absorbing suspension cradle that allows six degrees of movement for minor misalignment of the payload hook. The top cylindrical section could include a winch assembly for adjusting the position of the 'attaching' ring. The 'catching' operation occurs very quickly so it is expected that the entire process would be automated with

Figure 2.23: HASTOL Grapple Mechanism [22]

the use of differential GPS systems similar to those used for landing commercial aircraft. The investigators suggest further work is needed to refine the grapple design for issues such as atmospheric heating, control and power supply.

2.6.3 MXER Grapple Concept

NASA's In-space Propulsion Technology Program initiated the Momentum eXchange Electrodynamic Reboost (MXER) Tether Program to design a rotating tether for transporting satellites from LEO to geosynchronous transfer orbit (GTO) without the use of rocket technology. The program includes participants from Tennessee Technological University, Lockheed Martin Astronautics and Tethers Unlimited [28]. A web site was setup, at for collaborating and proposing catcher mechanisms for the end of the MXER rotating tether [29].

The design challenge was to create a payload capture mechanism that would allow the rotating end of the MXER tether in an elliptical orbit to securely connect with a payload in circular LEO. The time period for capture, that is time period when the payload and the end of the tether both share the same position and have zero relative velocity, is very brief. Therefore, the 'catcher' mechanism must operate very quickly. The ideal case is the payload and tether end share the same position and have zero velocity at time of capture, however practically, the 'catcher' mechanism must accommodate some degree of error in position and relative velocity [28].

At the time of writing three concepts for the catcher mechanism were proposed. However, the only information available about the concepts was animations of the mechanisms operations available on the collaboration web site. Therefore, the following details about the concepts were inferred from the animations [29].

The first concept, known as the Iris mechanism, is the preferred design. Like the HASTOL payload, a hook extends from the payload with an extendable boom. Figure 2.24 depicts a payload with the hook and boom extended ready to be captured by the tether. The catcher mechanism attached to the tether is called an iris,

Figure 2.24: Payload Boom and Hook [29]

presumably because the mechanism opens and closes. Initially the mechanism forms a large square, see Figure 2.25a. For capture to occur, the hook of the payload must be inside the perimeter of this square. If the hook is in the perimeter of the square, the four 'posts' at each corner move together very quickly to reduce the size of the square to snare the hook. Figures 2.25 a, b and c depict the process.

Figure 2.25: Iris 'Catcher' Mechanism (a) before capture, (b) during capture and (c) at capture [29]

It appears that each of the four 'posts' contain two active motors (probably electric) to create the linear motion required to reduce the size of the square to achieve capture. The 'posts' probably contain a sensor to detect the presence of the hook inside the perimeter of the square to trigger the capture event.

The second proposed catcher mechanism utilizes the same hook and boom for the payload; however the capture mechanism on the tether differs. On the tether, a structure that is similar to a fishing net is used to snare the payload hook. Figure 2.26 depicts the net capture mechanism.

Figure 2.26: Net Capture Mechanism [29]

Figure 2.27: Bear Trap Capture Mechanism [29]

It is assumed that this mechanism was not selected because it would be difficult to ensure that the payload is securely captured by the net and the mechanism would make release of the payload difficult.

The third capture method operates like a bear trap. It is presumed that the payload utilizes a hook similar to the previous examples. A bear trap like mechanism located on the end of the tether closes around the payload hook at the appropriate time. Figure 2.27 depicts the bear trap. It is presumed this concept was a preliminary concept since the level of work on the concept is limited.

2.7 Space Construction Projects

Five space construction projects of different magnitude have been identified from the literature review. These five projects will be considered as the possible range of space construction projects that the material transfer capabilities of the Mass Driver Transport System, mentioned in Section 1.4, could be used to achieve. The five construction projects are listed below in table 2.1 from smallest to largest project [27&30]. Figure 2.28 depicts the projects. The specifications for an Ohio class submarine and Nimitz class aircraft carrier are included in table 2.1 and Figure 2.28 for comparison purposes.

Table 2.1: Specifications for Five Space Construction Projects

Space Construction. Project	Total Mass (tonne)	Length (m)	Rocket Launches*	Population (people)	Project Date
MIR Space Station	67.5	46	6	2 to 6	1990's
International Space Station (ISS)	228	80	45	2 to 7	2000's
Island One Habitat	10 620 000	830	2 000 000	10 000	2080's**
Island Two Habitat	53 000 000	895	10 000 000	75 000	2100's**
Island Three Habitat	124 000 000	8950	24 000 000	820 000	2120's**
<u>Comparison</u>					
Submarine	19 051	171	1693	155	1980's
Aircraft Carrier	98 557	333	8760	5680	1980's

* Number of rocket launches to supply total mass based on 11.25 tonne per rocket.

** Based on the development time between MIR and ISS.

Figure 2.28: Tentative Timeline for Five Space Construction Projects

(Comparison)

Nimitz Class
Aircraft Carrier

Mass: 98 557 tonne
Length: 333 m
Width: 41 m
Crew: 5680

(Comparison)

Ohio Class
Submarine

Mass: 19 051 tonne
Length: 171 m
Diameter: 10.1 m
Crew: 155

Tentative Space Structure Development Timeline

MIR Space Station

Mass: 67.5 tonne
Length: 46 m
Diameter: 4.1 m
Rocket Launches: 6
Crew: 2 - 6

Status: De-orbited 1997

International Space
Station (ISS)

Mass: 228 tonne
Length: 80 m
Rocket Launches: 45
Crew: 2 - 7

Status: Under Construction

Island "One"
Torus Habitat

Mass: 10.62 M tonne
Major Radius: 830 m
Minor Radius: 65 m
Rocket Launches: 2 M
Population: 10 000

Status: Proposed NASA
Design

Island "Two"
Spherical Habitat

Mass: 53 M tonne
Radius: 895 m
Rocket Launches: 10 M
Population: 75 000

Status: Proposed NASA
Design

Island "Three"
Cylindrical Habitat

Mass: 124 M tonne
Length: 8950 m
Radius: 895 m
Rocket Launches: 24 M
Population: 820 000

Status: Proposed NASA
Design

From 1975 NASA Space Habitat Design Series

Chapter 3 Evaluation and Design of the Preliminary Open Mass Driver System

3.1 Design Problem and Approach

The goal of this project is to establish and aid an open engineering development community for the Open Mass Driver System (OpenMDS) described in Section 1.4. This design process for the OpenMDS is intended to identify any technical limitations that would make the design impossible. Therefore, this study is concerned only with the physical feasibility of the OpenMDS and is not concerned at this time with the economic feasibility of the mass driver system. The design of the mass driver will be first limited to a systems level design; a successful feasible systems design will then proceed towards a more detailed design also with the aid of the development community. Before an open engineering development community can be established, it is assumed that some form of ‘prototype’ design of the mass driver system will be required to attract a development community, since successful open source software projects tend to have an initial prototype version of the software to attract a development community. At the commencement of this project it was assumed that a detailed explanation of the operation of the concept would be sufficient to attract developers; however, through attempts^h at attracting developers, it became evident the ‘prototype’ design required details to prove the concept was feasible and proves the commitment of the project founders.

Section 1.4 introduced Mr. Panzeria’s mass driver material package transport system as an Earth based launcher capable of sending a stream of small construction material packages to Earth orbit. An orbiting ‘catcher’ satellite (CatSat) situates itself to collect the construction material packages in orbit. The collected MPs are then distributed as construction material to space construction sites. The proposed concept describes how the mass driver system would operate; however, to evaluate the physical feasibility of the mass driver system, a number of design parameters must be defined and evaluated. As a starting point, Section 3.2 describes the preliminary design parameters that were chosen as the baseline mass driver system. The goal of the design process for this Chapter is to develop a systems level design for the ‘prototype’ OpenMDS. This Chapter will describe the evaluation of the first version of OpenMDS. The following Chapter, Chapter four, will discuss the ‘evolution’ of the OpenMDS design towards a

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^h see Chapter 5 for these attempts and results

feasible prototype design. Chapter five will discuss the attempts to establish an open source development community and the results from the open source community.

3.2 Proposed Open Mass Driver System Design

The mission for the OpenMDS will be considered to be the supply of raw material to one of the five space construction projects introduced in Section 2.7. These projects range from a small MIR space station sized project to a large 820 000 person habitat. The range of the construction will demonstrate the scale of space construction project that the proposed mass driver system design could be used to support. The baseline mass driver system design was considered to be comprised of three sub-systems, the launcher, material package (or pellets) and 'catcher' satellite sub-systems. Figure 3.1 depicts a concept map for the OpenMDS. The concept map visually represents the sub-systems, initial design parameters and their relationships to each other. This concept map is not a complete diagram, but it records the initial design parameters identified for the OpenMDS. The following three Sections provide a description of each sub-system and the assumed design parameters for the OpenMDS.

3.2.1 The Launcher Sub-system

The launcher (or mass driver) sub-system is responsible for launching a series of high velocity material packages one after another to form a material package stream (MPS). For the preliminary OpenMDS, it is assumed the launcher would be located on Earth's surface at the equator and the launch direction would be in an easterly direction to take advantage of Earth's rotation. The launch site was assumed to be at sea-level, to account for the majority of elevations that are located on the equator.

The concept map, Figure 3.1, indicates the important design parameters for establishing a material package stream is the launch velocity, launch frequency and launch energy. These three launch parameters are affected by the design parameters of the material packages and the 'catcher' satellite (CatSat). Once the launch velocity, frequency and energy are determined, then the appropriate launcher technology, from Section 2.5, can be selected. Therefore, to design the launcher sub-system all the important design parameters for the OpenMDS depicted in Figure 3.1 will be used to calculate the launch velocity, frequency and energy requirements. Details of the calculation method and results are discussed in Section 3.3, 3.4 and 3.5.

Figure 3.1: Concept Map for Open Mass Driver System

3.2.2 Material Package Sub-system

The material packages (or pellets) are ideally comprised of only the raw material needed for the space construction project. These material packages are assumed to have a total mass of less than 10 kg with each being composed of a solid construction material, like steel. It is assumed that having a small material package mass will be beneficial because it will simplify the design of the

launcher sub-system. The preliminary OpenMDS will consider material packages masses ranging from 1 gram, or pebble size, to 10 kg, or bowling ball size.

The concept map identifies four major design parameters, in addition to the mass, for the material packages, namely the geometry, stability, drag and thermal protection. To address the geometry, it is initially assumed the material package would be a simple cylinder. It was initially assumed that flight stability of the material package would be provided by fins. The drag on each material package is a complex situation because each material package is traveling in the wake, or low pressure region of the preceding material package; section 3.3 and 3.4 discuss the details of how the drag on the material package was approximated. A single launch body would require a thermal protection system to reduce the erosion or ablation of the body during hypersonic atmospheric flight. In Section 2.3, three studies [12, 14 & 16] suggested a passive thermal protection system utilizing conventional refractory materials would be sufficient for thermal protection; however two studies [13 & 15] concluded a high performance active cooling system is required for adequate thermal protection of the launch vehicles. Determining the thermal effects experienced by an individual material package in a material package stream is a challenging problem, since the thermal effects are highly coupled to the hypersonic fluid flow about the material package stream. Early in this project, Dr. Salustri suggested that evaluating the actual fluid flow conditions about the material package stream would be beyond the scope of this project. Since the thermal effects are highly associated with the actual fluid flow conditions, determining the thermal effects experience by the material packages must be deferred until flow effects about the material package stream are known. It will be assumed that a thermal protection system is possible for a material package stream since previous studies [12, 13, 14, 15 & 16] have determined a thermal protection system is possible for a single launch body.

It is postulated that the ideal case is, the material package stream would create slip-stream fluid flow effects that would create a low density tube of air to form around the material package stream. Therefore ideally, the reduced air density around the material package stream would reduce the thermal effects on individual material such that a thermal protection material is not required; therefore, the material package could be completely raw material. A detailed analysis of the fluid flow and thermal characteristics is needed to determine if the physical effects about the

material package stream conform to the proposed ideal case. Potentially, a member of the open engineering development community of this project would perform this analysis. Without a detailed analysis of the fluid flow and thermal effects, the design of the OpenMDS will precede according to assumptions for the material package stream fluid flow characteristics.

The concept map, Figure 3.1, depicts that the design parameters for the material packages are affected by the design parameters of the launcher and ‘catcher’ satellite sub-systems. Therefore, like the design of the launcher sub-system, the design parameters of the OpenMDS will be used to calculate the design parameters for the material packages. Details of the calculation method and results are discussed in Section 3.3, 3.4 and 3.5.

3.2.3 ‘Catcher’ Satellite Sub-system

The ‘catcher’ satellite (CatSat) sub-system is responsible for capturing the material package stream and either utilizing the material for onboard manufacturing processes or distributing the material to a space construction project. The concept map, Figure 3.1, identifies the altitude of the CatSat as the main design parameter. The altitude of the CatSat affects the design parameters of the launcher and material packages. To simplify the design of the launcher sub-system, it is initially assumed the altitude of the CatSat would correspond to a geosynchronous (GEO-SYNC) Earth orbit.

A GEO-SYNC altitude simplifies the design of the launcher sub-system because the CatSat would be stationary with respect to a launcher. At GEO-SYNC, the satellite is orbiting at the same rate as Earth’s rotation; therefore, the launcher can operate continuously without having a launch window. The required launcher mechanism could be constructed as a stationary device pointing at the same region of the sky, since it would not be required to slew the launch direction.

3.3 Evaluation of Baseline Mass Driver System

Since this is to be an ‘open engineering’ project, it is very important that all the information regarding the project be accessible to all participants of the project. To facilitate the openness of the project, ideally, all software used for the design of the mass driver should be either an OSS product or at least a common, low-cost commercial product. An extensive search was conducted to find the aforementioned software products to calculate the launch of hypervelocity masses from

the Earth's surface. A software product for evaluating the OpenMDS was sought because it would speed up the development time of the 'prototype' design; therefore, more effort could be allotted to attracting a volunteer development team.

The closest software candidate was NASA's Optimal Trajectories by Implicit Simulationⁱ (OTIS) code, distributed by NASA Glenn Research Center's Commercial Technology Office. The program is capable of calculating the flight of a hypervelocity body from Earth's surface to orbit. The program is a quasi-freeware product. The program is available for free download to United States citizens only; however, the program relies on a non-linear optimization program, SNOPT, which must be purchased separately from Stanford Business Software Inc. Since the SNOPT program must be purchased and the author and many potential participants are not American citizens, the OTIS program was not further considered for the evaluation of the mass driver system.

Since no software product for evaluating the mass driver system was freely available, a calculation method had to be created. A goal of this study is to evaluate the physical feasibility of the OpenMDS concept; therefore, an order of magnitude estimate for design parameters would be sufficient to assess the physical feasibility of the preliminary design. At this stage of development, a highly accurate calculation method is not required. A more accurate calculation method will be required for the detailed design of the OpenMDS. The launch of a series of hypersonic masses from the Earth surface is a complicated situation to calculate; therefore, the calculation method evolved from first a simple mathematical model of the situation to progressively more detailed models. The OpenMDS was evaluated using two calculation methods, first a 2-dimensional gravity-only mathematical model and then a 1-dimensional simulation model which accounts for gravity, aerodynamic drag and approximate launch corridor effects. The following two Sections introduce the calculation methods used.

3.3.1 Introduction to the 2-D Gravity-Only Mathematical Model

The mathematical model considers the 2-dimensional launch of a body from Earth's surface with only Earth's gravity acting on the body. A system of equations for the launch body

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ⁱ Optimal Trajectory by Implicit Simulation (OTIS) website - <http://otis.grc.nasa.gov/request.shtml>

was defined and solved using the Maple mathematical solver program. Using the Maple program hinders the ‘openness’ of the solution since the program is not freely available. However, potential participants to the project could use a numerical method or another mathematical solving program to obtain design parameter values. The user enters the launch mass, launch angle and launch velocity into the Maple work sheet. Then Maple is used to calculate the maximum altitude or apogee of the launch mass, the velocity at apogee and the launch energy. Figure 3.2 depicts a schematic for the first calculation method. The launch velocity was manually varied to determine the velocity required to reach an apogee at GEO-SYNC for a range of launch angles and masses. Section 3.4.1 details the calculation method and Section 3.5.1 details the results for the 2-dimensional launch in Earth’s gravity field.

Figure 3.2: Schematic for 2D Gravity-only Model

3.3.2 Introduction to the 1-D Gravity and Aerodynamic Drag Simulation Model with Approximate Launch Corridor Effects

The second calculation method considered a 1-dimensional launch from Earth’s surface (vertical launch) with Earth’s gravity, aerodynamic drag and an approximate drag reducing launch corridor acting on a single launch body. This model improves the functionality of the first model by considering the aerodynamic drag on a hypersonic body traveling through the various regions of Earth’s atmosphere and also approximates the launch corridor created by the slip-stream effects.

Evaluating the slip-stream aerodynamic drag effects about a series of hypersonic inline bodies and the air properties of the launch corridor created by a steady state hypersonic material package stream is a fluid flow problem that has not been extensively studied. Section 4.2.4 details what is known about the aerodynamic effects of hypersonic slip-stream flows. Therefore, instead

of attempting to evaluate the actual slip-stream effects, this simulation model was programmed to establish a range of possible design values by evaluating ideal case and worst case situations. The ideal case is the operation of a steady state material package stream which causes vacuum conditions in the launch corridor or a zero density tunnel through the atmosphere. The calculation method approximates this ideal case by eliminating the aerodynamic drag on the material package (100% drag reduction case). The worst case for the launch is the operation of the steady state material package stream that does not change the air density conditions of the launch corridor (0% drag reduction). The calculation method approximates the worst case by imposing the full aerodynamic drag on the material package; the same conditions the material package would experience if it traveled alone through the atmosphere.

Before the simulation model was attempted, an analytical solution similar to mathematical model #1 was initially attempted but proved to be erroneous. Since the analytical solution was erroneous, a simulation model was created by programming an Excel spread sheet. A simulation solution was a better solution because it was easier to calculate the drag effects and atmospheric properties than the analytic approach.

The user is required to input the launch velocity, launch frequency, material package diameter, material package density, launch mass and percent drag reduction caused by the launch corridor. The ideal and worst case launch corridor conditions are specified by the user; the user can specify the ideal case of a perfect vacuum launch corridor by specifying 100% drag reduction or any value up to no drag reduction corresponding to 0% drag reduction. The spreadsheet then calculates the launch energy, launch apogee, velocity at apogee, spacing between material packages and length to diameter ratio of the material packages. The launch velocity was manually varied to obtain a GEO-SYNC apogee for the range of launch mass from 1 gram to 10 kg. The material package diameter and density are used by the spreadsheet to calculate the length of the material package. The drag calculation used by the model was based on experimental data for a hypersonic finned missile with a length to diameter ratio of 24 [31]. The material package diameter was manually varied to until a length to diameter ratio of 24 was achieved. Figure 3.3 depicts a schematic for the second mathematical model. Section 3.4.2 details the calculation method and

Section 3.5.2 details the results for the 1-dimensional launch with Earth's gravity and aerodynamic drag.

The Excel program improves the 'openness' of the information related to the project. Excel is a common software product installed on many home PC; therefore, it is likely that many potential users would have access to the program and be familiar with operating the program. Additionally, the Excel program was chosen because the author is more familiar with programming in Excel than other programming languages. This familiarity expedited the evaluation of the mass driver system since little learning was necessary to program the simulation.

Figure 3.3: Schematic for the 1D Gravity and Atmospheric Drag Simulation Model

3.4 Calculation Method and Details

The following three Sections review the details of how the two calculations methods perform the calculations to evaluate the OpenMDS.

3.4.1 Calculations for 2-D Gravity Only Mathematical Model

Figure 3.4a is a diagram of the 2-dimensional, gravity-only mathematical model viewed from the top of Earth looking down at the North Pole. The diagram depicts an eastward launch from an arbitrary location on Earth's equator. A circular co-ordinate system is depicted with an origin at the centre of the Earth. Figure 3.4b depicts the free body diagram of the launch mass. The only force acting on the launch mass is the force of gravity in the radial direction.

3.4.1.2 Calculation Method

The mathematical solution began by considering the gravity force acting on the launch mass in the radial direction (e_r),

$$F_G = \frac{G M m}{r^2} e_r = m a_r \quad [3]$$

Divide the above equation by m ,

$$a_r = \frac{G M}{r^2} e_r \quad (\text{eq. 3.1})$$

From a dynamics textbook, [32], the equation for acceleration in circular co-ordinates is,

$$a(t) = [a_r(t) - r(t) \times \omega^2(t)] e_r + [r(t) \times a_\theta(t) + 2 \times v_r(t) \times \omega(t)] e_\theta \quad (\text{eq. 3.2})$$

Breaking Equation 3.2 into its components, e_r and e_θ , and equating the gravity force,

$$a_r(t) - [r(t) \times \omega^2(t)] = -\frac{G M}{r^2} \quad (\text{eq. 3.3})$$

$$[r(t) \times a_\theta(t)] + [2 \times v_r(t) \times \omega(t)] = 0 \quad (\text{eq. 3.4})$$

Isolating $a_r(t)$ in equation 3.3,

$$\boxed{a_r(t) = [r(t) \times \omega^2(t)] - \frac{G M}{r^2}} \quad (\text{eq. 3.5})$$

$$\boxed{\frac{dr^2}{dt} = r \left(\frac{d\theta}{dt} \right)^2 - \frac{GM}{r^2}}$$

Isolating $a_\theta(t)$ in equation 3.4,

$$\boxed{a_\theta(t) = \frac{-2 \times v_r(t) \times \omega(t)}{r}} \quad (\text{eq. 3.6})$$

$$\boxed{\frac{d\theta^2}{dt} = \frac{-2}{r} \frac{dr}{dt} \frac{d\theta}{dt}}$$

To solve the system of equations the initial launch conditions must be specified. The initial position of the launch site (at time zero) was specified as the radius of the Earth's equator as,

$$r(0) = 6\,371\,000 \text{ m} \quad (\text{eq. 3.7})$$

and the longitude (at time zero) was specified at an arbitrary longitude location of 30 degrees as,

$$\theta(0) = 0.5236 \text{ rad} \quad (\text{eq. 3.8})$$

The initial launch velocity (at time zero) in the radial direction was calculated as follows,

$$V_r(0) = V_{\text{Initial}} \cos \alpha_{\text{Launch}} \quad (\text{eq. 3.9})$$

The initial launch velocity (at time 0) in the tangential direction was calculated as follows,

$$V_\theta(0) = V_{\text{Initial}} \sin \alpha_{\text{Launch}} + \omega_{\text{Earth}} \times r_{\text{Earth}} \quad (\text{eq. 3.10})$$

The Maple program was used to solve the system of differential equations formed by equations 3.5 and 3.6 and given the initial conditions, equations 3.7, 3.8, 3.9 and 3.10 specified by the user. Maple could not find a closed form solution to the differential equations, so the Runge Kutta numerical method was used to determine the maximum radius, r_{max} , or apogee of the ballistic trajectory and the angular rotation of the material package at apogee. As mentioned in the Section 3.3.1, the user varied the input parameters manually until an apogee at GEO-SYNC was determined.

After suitable launch parameters were determined, the tangential velocity of the material package at apogee was calculated as follows,

$$V_\theta(t_{\text{apogee}}) = r_{\text{max}} \times \omega(t_{\text{apogee}}) \quad (\text{eq. 3.11})$$

where t_{apogee} is the time corresponding to the apogee location

The tangential velocity for a satellite in a circular orbit at the apogee was calculated as follows,

$$V_{\text{circ}} = \left(\frac{GM}{r_{\text{max}}} \right)^{1/2} \quad (\text{eq. 3.12})$$

The circular tangential velocity represents the velocity of the catcher satellite. The velocity difference between the material package at apogee and an orbiting catcher satellite was calculated as follows,

$$\Delta V = V_{\text{circ}} - V_\theta(t_{\text{apogee}}) \quad (\text{eq. 3.13})$$

This velocity difference represents the capture velocity of the material package that the capture mechanism must be design to accommodate.

The apogee altitude was calculated as follows,

$$r_{\text{apogee}} = r_{\text{max}} - r_{\text{Earth}} \quad (\text{eq. 3.14})$$

The launch energy, E_{Launch} , was calculated as follows,

$$E_{\text{Launch}} = \frac{1}{2} m (V_{\text{initial}})^2 \quad (\text{eq. 3.15})$$

Section 3.5.1 details the range of results obtained by this mathematical model.

3.4.2 Calculations for 1-D Gravity, Aerodynamic Drag Mathematical Model with Approximate Launch Corridor Effects

Like the first mathematical model, the simulation model is used to determine the launch parameters required for a material package to reach GEO-SYNC. This second calculation method improves the launch results by considering aerodynamic drag on the material package and approximating the effects of the launch corridor. The spreadsheet for the simulation model was developed under the following assumptions.

3.4.2.1 Assumptions

1. The launch angle is vertical and the launched mass will follow a straight line trajectory since Earth rotation is neglected for this model.
2. The lower atmospheric density was assumed to be defined by a density function defined by data from the ICAO (International Civil Aviation Organisation) Standard Atmosphere [33].
3. The upper atmosphere was assumed to follow a density function defined by the 1976 US Standard Atmosphere [34].
4. The atmospheric temperature was defined by functions for each temperature region depicted in Figure 3.5 [33].
5. All temperature functions between temperature regions were assumed to be linear.
6. Experimental total drag coefficient as a function of Mach number for a hypervelocity missile was used [31]. The experimental drag co-efficient assumed a fin stabilised missile with an L/D of 24.
7. The package material was assumed to be steel and the geometry was assumed to be a simple cylinder.
8. The simulation model assumed constant deceleration during each time step.

3.4.2.2 Input, Output and Constant Parameters

Input Parameters Specified by User

- time step (t_{step}) – the time duration between each calculation in the simulation (s)
Launch Velocity (v_{Launch}) – the initial velocity of the material package at exit of launcher (m/s)
Material Package Mass (m_{package}) – total mass of material package (kg)
Material Package Diameter (D_{package}) – the diameter of the material package (m)
Material Package Density (ρ_{package}) – the density of the material package (kg/m³)
Launch Frequency (ω_{launch}) – the launch frequency of material packages in launch corridor (Hz)
Slip Stream Drag Reduction ($C_{\text{slip-stream}}$) – the percent drag reduction caused by the launch corridor (%)

Output Parameter of Simulation Spreadsheet

- Frontal Area (A_F) – frontal area of the material package (m²)
Volume (V_{package}) – volume of material package (m³)
Length (L_{package}) – length of material package (m)
Length to Diameter (L/D) Ratio – Length to Diameter ratio of material package
Launch Energy (E_{Launch}) – Launch energy of material package (MW)
Material Package Spacing (d_{spacing}) - Distance between material packages in stream
Mass Flow Rate (MFR) – the rate of mass transfer from the launcher mechanism (kg/s)
Launch Power (P_{Launch}) – the power used for the launch process
Maximum Altitude (h_{max}) – the highest altitude the material package obtains (m) (manually found in simulation table)

Constant Required

- Mass of Earth (M_{Earth}) = 5.976×10^{24} kg
Radius of a spherical Earth (r_{Earth}) = 6371 km = 6 371 000 m
Gravitational Constant (G) = 6.67×10^{-11} m³/kg s² [3]
Specific Heat Ratio (k) = 1.4 (for air)
Real Gas Constant (R) = 287 J/(kg K) [35]

Figure 3.5: Overview of 1D and Atmospheric Drag Simulation Model

3.4.2.3 Calculation Method

The Figure 3.5, above, depicts a vertical launch from an arbitrary location on Earth's equator to a GEO-SYNC orbit. The density and temperature regions of the atmosphere are depicted. The distances shown in the Figure are not to scale.

The simulation spreadsheet performs the following calculation steps:

1. The user inputs the desired time step (calculation interval), launch velocity, launch altitude (specified as sea-level), launch frequency, material package mass, material package diameter, material package density and percent slip-stream drag reduction. The user may also change the constant properties for the Earth and the air.
2. The frontal area, volume, length, L/D (length to diameter) ratio, launch energy and spacing of the material package is calculated. The mass transfer rate and the launch power of the system is calculated. See Part A for the calculations.

NOTE: It is important that the L/D ratio close to 24, since the drag coefficient data used by the spreadsheet is based on an L/D ratio of 24 (see Part F). L/D ratio is adjusted by the user by manually changing either the diameter of the package or the density.

3. At time = 0s, the initial launch altitude (sea-level) and velocity inputted by the user are entered into the simulation table.
4. The altitude for the time step is used to calculate the **density of air, ρ_{air}** , according to the density functions given in Part B.
5. The altitude for the time step is used to calculate the **temperature of air, T_{air}** , according to the temperature functions given in Part C.
6. The **speed of sound, a_{ss}** , at the altitude of the time step is calculated according to the equation in Part D. The speed of sound is required to calculate the Mach number of the material package.
7. The **Mach number, Ma** , of the package for the time step is calculated according to the equation in Part E. The Mach number is required to determine the drag co-efficient, C_d .
8. The **drag coefficient, C_d** , for the material package, with a given Mach number, is determined from experimental data detailed in Part F.
9. The **gravity (F_G), drag (F_D) and resultant (F_R) force** for the material package is calculated according to equations in Part G.
10. The **average acceleration, a_{avg}** , (deceleration) experienced by the material package during the time step, is calculated according to Part H.
11. The **altitude and velocity** for the next time step is calculated using the average acceleration according to equations in Part I.
12. Steps 4 through 11 are repeated for the following time steps.

NOTE: The simulation ends when the velocity of the package becomes zero or negative. The user may have to copy and paste the final row of the simulation table to extend the simulation time to achieve zero or negative velocity.

Part A) Calculated Design Parameters

The following design parameters are calculated using the material package mass (m_{package} in kg), material package diameter (D_{package} in meters), material density (ρ_{package} in kg/m^3), launch velocity (v_{launch} in m/s) and launch frequency (ω_{launch} in Hz) entered by the user. Figure 3.6 is a schematic that depicts most of the calculated values.

$$\text{Frontal Area} - A_F = \pi \times (D_{\text{package}}/2)^2 \quad (\text{m}^2) \quad (\text{eq. 3.16})$$

$$\text{Volume} - V_{\text{package}} = m_{\text{package}} / \rho_{\text{package}} \quad (\text{m}^3) \quad (\text{eq. 3.17})$$

$$\text{Length} - L_{\text{package}} = V_{\text{package}} / A_F \quad (\text{m}) \quad (\text{simple cylinder assumption}) \quad (\text{eq. 3.18})$$

$$\text{L/D Ratio} - L/D = L_{\text{package}} / D_{\text{package}} \quad (\text{eq. 3.19})$$

$$\text{Launch Energy} - E_{\text{Launch}} = 0.5 \times m_{\text{package}} \times v_{\text{Launch}}^2 \quad (\text{MJ}) \quad (\text{eq. 3.20})$$

$$\text{Spacing} - d_{\text{spacing}} = v_{\text{Launch}} \times 1/\omega_{\text{launch}} - L_{\text{package}} \quad (\text{m}) \quad (\text{eq. 3.21})$$

$$\text{Mass Flow Rate} - \text{MFR} = m_{\text{package}} \times \omega_{\text{launch}} \quad (\text{kg/s}) \quad (\text{eq. 3.22})$$

converted to (tonnes/hr) multiplying by 3.6

$$\text{Launch Power} - P_{\text{Launch}} = E_{\text{Launch}} \times \omega_{\text{launch}} \quad (\text{MW}) \quad (\text{eq. 3.23})$$

Figure 3.6: Schematic for Launch Parameters

Part B) Air Density Functions

Lower Atmosphere

The ICAO data for the lower atmosphere density as a function of altitude was charted in Excel and a 2nd degree polynomial function was fitted to the data [33]. The atmospheric density function was determined to be:

$$\rho_{\text{atm}} = 2 \times 10^{-9} h^2 - 1 \times 10^{-4} h + 1.1904 \quad (\text{eq. 3.24})$$

valid for $0 \text{ m} < h < 19\,000 \text{ m}$ ($\approx 62\,000 \text{ ft}$)

where:

ρ_{atm} – atmospheric air density in kg/m^3 .

h – altitude (height above Earth surface) in meters (m).

Upper Atmosphere

The upper atmospheric density function used is obtained from the 1976 US Standard Atmosphere and defined as:

$$\rho_{\text{atm}} = 11 \times \exp(-h/6000) \quad (\text{eq. 3.25})$$

valid for $70\,000 \text{ m}$ ($\approx 229\,658 \text{ ft}$) $< h < 118\,000 \text{ m}$ ($\approx 387\,139 \text{ ft}$)

and

$$\rho_{\text{atm}} = [(h/6000) - 95]^{-3} / 2600 \quad (\text{eq. 3.26})$$

valid for $118\,000 \text{ m}$ ($\approx 387\,139 \text{ ft}$) $< h < 200\,000 \text{ m}$ ($\approx 656\,168 \text{ ft}$)

and its assumed that,

$$\rho_{\text{atm}} = 0 \text{ for } h > 200\,000 \text{ m} (\approx 656\,168 \text{ ft}) \quad (\text{eq. 3.27})$$

'Middle' Atmosphere

For $19\,000\text{ m} < h < 70\,000\text{ m}$ the atmospheric air density is linearly interpolated between 0.0124 kg/m^3 (@ $19\,000\text{ m}$) and $9.4324 \times 10^{-5}\text{ kg/m}^3$ (@ $70\,000\text{ m}$), since a density function for this region could not be found at this time. The interpolation function was determined to be:

$$\rho_{\text{atm}} = -2.4129 \times 10^{-7} (h - 19\,000) + 0.0124 \quad (\text{eq. 3.28})$$

valid for $19\,000\text{ m} (\approx 62\,000\text{ ft}) < h < 70\,000\text{ m} (\approx 229\,658\text{ ft})$.

Part C) Air Temperature Functions

Troposphere Region ($0 < h < 10\text{ km}$) ($288.15\text{ K} < T_{\text{atm}} < 220.0\text{ K}$)

The temperature of this region decreases linearly. Therefore, the temperature function is defined by:

$$T_{\text{atm}} = -6.815 \times 10^{-3} h + 288.15 \quad \text{valid for } 0 \leq h < 10\,000\text{ m} (\approx 32\,808\text{ ft}) \quad (\text{eq. 3.29})$$

where:

T_{atm} – atmospheric temperature in Kelvin (K).

h – altitude (height above Earth surface) in meters (m).

Stratosphere Region ($10\text{ km} \leq h < 50\text{ km}$) ($220.0 \leq T_{\text{atm}} < 270\text{ K}$)

The temperature of this region increases exponentially, however it will be assumed that the variation is linear since only the end point temperatures are known at this time. The temperature function is defined by:

$$T_{\text{atm}} = 1.25 \times 10^{-3} (h - 10\,000) + 220.0 \quad (\text{eq. 3.30})$$

valid for $10\,000\text{ m} \leq h < 50\,000\text{ m} (\approx 164\,042\text{ ft})$

Mesosphere Region ($50\text{ km} \leq h < 90\text{ km}$) ($270\text{ K} \leq T_{\text{atm}} < 180\text{ K}$)

The temperature of this region decreases linearly. Therefore, the temperature function is defined by:

$$T_{\text{atm}} = -2.25 \times 10^{-3} (h - 50\,000) + 270.0 \quad (\text{eq. 3.31})$$

valid for $50\,000\text{ m} \leq h < 90\,000\text{ m} (\approx 295\,275\text{ ft})$

Thermosphere Region ($90\text{ km} \leq h < 300\text{ km}$) ($180\text{ K} \leq T_{\text{atm}} < 1500\text{ K}$)

The temperature of this region increases exponentially, however it will be assumed that the variation is linear since only the end point temperatures are known at this time. The temperature function is defined by:

$$T_{\text{atm}} = 6.2857 \times 10^{-3} (h - 90\,000) + 180.0 \quad (\text{eq. 3.32})$$

valid for $90\,000\text{ m} \leq h < 300\,000\text{ m} (\approx 984\,252\text{ ft})$

Exosphere Region ($h \geq 300\text{ km}$) ($T_{\text{atm}} = 1500\text{ K}$)

The temperature of this region is assumed to be constant.

$$T_{\text{atm}} = 1500 \quad \text{valid for } h \geq 300\,000\text{ m} (\approx 984\,252\text{ ft}) \quad (\text{eq. 3.33})$$

Part D) Speed of Sound

The speed of sound at each time step is calculated using the gas constants and air temperature as follows [35]:

$$a_{ss} = [k_{air} \times R \times T_{atm}]^{0.5} \quad (\text{eq. 3.34})$$

where:

a_{ss} – Speed of Sound (m/s)

k_{air} – Specific Heat Ratio - c_p/c_v (1.4 for air)

R – Ideal Gas Constant (J/kg K)

T_{atm} – Temperature of air (K)

Part E) Mach Number

The Mach number, Ma , of the projectile for each time step is calculated as follows [35]:

$$Ma = v_{package} / a_{ss} \quad (\text{eq. 3.35})$$

where:

$v_{package}$ – velocity of the package (m/s)

a_{ss} – speed of sound (m/s)

The Mach number is used in the next calculation to determine the drag coefficient.

Part F – Drag Coefficient

Experimental data for the drag coefficient, C_d , vs. Mach number for a finned missile is depicted in Figure 3.7 [31]. The experimental data is valid for a body with an L/D of 24. This data is used for the drag co-efficient of the package.

The spreadsheet contains a table of C_d vs. Mach number values obtained from Figure 3.7. The spreadsheet uses the Mach number of the material package to 'look up' the appropriate drag coefficient, C_d . The advantage of hypersonic travel can be observed in Figure 3.7, which is the drag co-efficient decreases as the velocity surpasses about Mach 1.8. At high hypersonic speeds (> Mach 8) the drag-coefficient approaches an asymptotic minimum value of about 0.1.

Figure 3.7: Drag Coefficient (C_d) vs. Mach Number for a Finned Missile [31]

Part G - Forces Acting on the Launch Body

Figure 3.8 is a free body diagram (FBD) that depicts two forces acting on a 1-D launch body, the gravity force and the drag force.

The force of gravity, F_G (N), was calculated as [3]:

$$F_G = [G M m] / r^2 = [G M_E m_{\text{package}}] / (R_E + h)^2 \quad (\text{eq. 3.36})$$

where:

$$G = 6.67 \times 10^{-11} \text{ m}^3 / \text{kg s}^2$$

$$M_E = 5.976 \times 10^{24} \text{ kg and } R_E = 6\,371\,000 \text{ m}$$

The total drag force, F_D (N), on the launched body can be calculated as [35]:

$$F_D = (0.5 \times C_d \times \rho_{\text{atm}} \times v^2 \times A_F) \times (1 - C_{\text{slip-stream}} / 100) \quad (\text{eq. 3.37})$$

where:

ρ_{atm} – density of local atmospheric air the body is traveling through – an air density is a function of altitude.

v – velocity of launched body.

$C_{\text{slip-stream}}$ – Factor inputted by user as the percent slip-stream reduction

This calculation method is a simplistic means of evaluating a complex gas dynamic process occurring in the launch corridor. A slip-stream reduction factor of one hundred percent drag reduction is equivalent to the ideal case where the launch corridor conditions are a perfect vacuum. A slip-stream reduction factor of zero percent is equivalent to the case where the launch corridor is the full density of the atmosphere at each region. These two cases represent the range of the expected drag effect, therefore the actual drag reduction effect exists at some point in this range.

Figure 3.8: Free Body Diagram for Material Package

The resultant force, F_R , on the launch body can be calculated by summing:

$$F_R = F_G + F_D \quad (\text{eq. 3.38})$$

Part H) Average Acceleration on the Material Package

The average acceleration force, a_{avg} (m/s^2) on the material package was calculated as:

$$a_{avg} = F_R / m_{package} \quad (\text{eq. 3.39})$$

where:

F_R – Resultant Force acting on the launch body (N)

$m_{package}$ – Mass of package (kg)

Part I) Altitude and Velocity for Subsequent Time Steps

The average acceleration, a_{avg} , of the time step was used to calculate the altitude and velocity for the subsequent time step as follows [32] :

$$h_{(t+1)} = h_{(t)} + [v_{(t)} \times t_{step}] + [0.5 \times a_{avg} \times (t_{step}^2)] \quad (\text{eq. 3.40})$$

where:

$h_{(t+1)}$ – altitude of the package for the next time step (m)

$h_{(t)}$ – altitude of the package in the present time step (m)

$v_{(t)}$ – velocity of the package in the present time step (m/s)

a_{avg} – average acceleration during the present time step (m/s^2)

t_{step} – length of time of the time step (s)

$$v_{(t+1)} = v_{(t)} + [a_{avg} \times t_{step}] \quad (\text{eq. 3.41})$$

where:

$v_{(t+1)}$ – velocity of the package in the next time step (m/s)

$v_{(t)}$ – velocity of the package in the present time step (m/s)

a_{avg} – average acceleration during the present time step (m/s²)

t_{step} – length of time of the time step (s)

The results for the launch of material packages using the second mathematical model are detailed in Section 3.5.2.

3.5 Results from Calculation Methods for the Open Mass Driver System

The calculation methods were used to create a range of possible design parameters for the OpenMDS. From performance estimates obtained from the calculation methods, a set of design parameters is selected and presented as the preliminary OpenMDS in Section 3.6. This preliminary design represents the first design cycle for the mass driver system. The next two Sections will present the performance results and the observed trends obtained from the two mathematical models used.

3.5.1 Results for 2-D Gravity-Only Mathematical Model

The results parameters obtained from the first calculation method were the maximum altitude (or apogee) of the material package and the angular velocity of the material packages at apogee. These two resulting parameters were obtained over a range of launch angles and launch velocities. The results were obtained by first, setting the launch angle to 0 degrees (vertical launch) and then incrementing the launch velocity up from 1 km/s until the launch velocity corresponding to a GEO-SYNC apogee was achieved. This process was repeated for a launch angle of 25, 35, 45, and 55 degrees.

The results of the apogee altitude vs. launch velocity are presented in Figure 3.9. It can be observed from Figure 3.9, that the lowest launch velocity to achieve any given altitude corresponds to a 0 degree or a vertical launch. This result is expected, since a vertical launch represents the shortest distance to any altitude. Therefore, the minimum launch velocity to reach a GEO-SYNC apogee was determined to be 10.306 km/s, launched at a zero degree angle. The largest angle, 55 degrees, consequently requires the largest launch velocity, 17.97 km/s, to reach a GEO-SYNC apogee. The required launch velocities for the other launch angles are listed in Table 3.1.

Figure 3.9: Results from 2D Gravity Only Model – Apogee Altitude vs. Launch Velocity of Material Packages

Table 3.1: Launch Results for 2D Gravity Only Model

Launch Angle	Launch Velocity (km/s)	r_{max} (m)	$r_{Apogee} = r_{max} - r_{Earth}$ (km)	V_{Theta} (m/s)	$V_{Circular Orbit}$ (m/s)	Delta V (m/s)	Specific Launch Energy (MJ/kg)
0	10.306	4.2145E+07	3.5774E+04	70.3	3075.4	3005.0	53.11
25	11.37	4.2086E+07	3.5715E+04	797.8	3077.5	2279.7	64.64
35	12.58	4.2096E+07	3.5725E+04	1162.4	3077.1	1914.7	79.13
45	14.575	4.2150E+07	3.5779E+04	1628.1	3075.2	1447.1	106.22
55	17.97	4.2201E+07	3.5830E+04	2292.5	3073.3	780.8	161.46

The general trend observed in Figure 3.9 and table 3.1 is as the launch angle increases, the required launch velocity to reach a particular altitude also increases.

TREND 1: $\uparrow$ Launch Angle $\Rightarrow \uparrow$ Launch Velocity

At the maximum altitude (apogee), the angular velocity (v_0 or v_{Theta}), of the material package was recorded for the same range of launch angles and launch velocities. Table 3.1 lists the angular velocities for the material packages at a GEO-SYNC apogee. It can be observed in Table 3.1, that the general trend is as the launch angle increases, the angular velocity of the material

package also increases. This trend reflects the fact that launching at angles other than vertical, adds a tangential velocity component that increases with increasing angle.

TREND 2: $\uparrow$ Launch Angle $\Rightarrow$ $\uparrow$ Angular Velocity of Material Package

Since the material package is to be captured by the CatSat, the difference between the material package at apogee and the angular velocity of a satellite in a circular orbit at the apogee altitude is represented as the Δv quantity in Table 3.1. The general trend observed in Table 3.1 is as the launch angle increases, the velocity difference between the material package and the satellite decreases.

TREND 3: $\uparrow$ Launch Angle $\Rightarrow$ $\downarrow$ Velocity Difference between Satellite and Package

Figure 3.10 is a graph of the velocity difference vs. launch velocity results. From Figure 3.10, it can be observed that for low launch velocities, thus low apogee altitudes, the effect of launch angle on launch velocity is relatively small since the lines in Figure 3.10 are close together. However, at higher launch velocities, thus higher altitudes, a larger launch angle significantly decreases the velocity difference since the lines in Figure 3.10 diverge. The 55 degree launch angle has two velocities, thus altitudes that correspond to satellite locations where the velocity difference between the satellite and material package is zero. These zero velocity locations correspond to satellite orbits at about 3000 km and 11 000 km altitude. Note these altitudes are much lower than the 35 793 km altitude required for a GEO-SYNC orbit.

Figure 3.10: Results from 2D Gravity Only Model – Velocity Difference vs. Launch Velocity of Material Packages

3.5.2 Results for 1-D Gravity and Aerodynamic Drag Simulation Model

This model improves the results of the previous by including aerodynamic drag and an approximate launch corridor effect. However, this model essentially assumes a vertical launch; therefore, the effects of launch angle are excluded from these results. The results for this 1-dimensional model are concerned with evaluating how the size of the material package, traveling through the atmosphere, effects the launch requirements and the performance of the mass driver system.

The results for this model considered the launch of material packages with masses ranging from 1 gram to 10 kg to GEO-SYNC altitudes. The launch corridor conditions were specified by the author to be either 0% slip-stream drag reduction, thus full atmosphere, or 100% drag reduction, thus perfect vacuum. The 100% drag reduction case corresponds to the 0 degree launch angle in the first mathematical model. The material density of the material package was specified as steel and diameter of the material package was adjusted until a length to diameter ratio of 24 was achieved (the drag calculations are valid only for bodies with an L/D ratio of 24). The time step or calculation interval for the simulation was specified to be 1 second. Therefore, the

forces on the material package were calculated for each second of the flight. The launch frequency was adjusted until the spacing between the material packages in the stream was 10 times the material package diameter. At this stage of evaluation, the value of 10 diameters separation was chosen as an estimate suggested by Professor S. Molder for the required spacing for slip-stream drag reduction to occur. Section 4.2.4 discusses the results of previous studies investigating slip-stream or interference drag reduction at subsonic and hypersonic regimes.

Table 3.2 lists the results obtained for the vertical launch of material packages ranging in mass from 1 gram to 10 kg to a GEO-SYNC apogee altitude. The smallest material package, 1 g total mass, has dimensions similar to a needle with a diameter of 1.09 mm and a length of 45.8 mm. The largest material package, 10 kg total mass, has dimensions similar to a steel bar, with a diameter of 40.9 mm and a length of 990 mm. The dimensions for the material packages between these two extremes are listed in table 3.2.

Table 3.2: Results from 1D Simulation Model for Vertical Launch of 1 Gram to 10 Kg Material Package Streams

Parameter	1g Material Package Stream	10g Material Package Stream	100g Material Package Stream	1 kg Material Package Stream	10 kg Material Package Stream
Package Diameter	1.09 mm	4.09 mm	8.80 mm	19.00 mm	40.90 mm
Package Length	45.8 mm	100 mm	210 mm	460 mm	990 mm
Package Separation (10 Dia)	10.9 mm	40.90 mm	88.00 mm	190.00 mm	409.00 mm
Launch Velocity					
0 % Drag Reduction	49.79 km/s	21.18 km/s	14.38 km/s	12.04 km/s	11.07 km/s
100% Drag Reduction	10.31 km/s	10.31 km/s	10.31 km/s	10.31 km/s	10.31 km/s
Launch Energy					
0 % Drag Reduction	1.24 MJ	2.24 MJ	10.34 MJ	72.47 MJ	612.95 MJ
100% Drag Reduction	0.05 MJ	0.53 MJ	5.31 MJ	53.1 MJ	530.97 MJ
Launch Frequency**	160 000 Hz (pkg/s)	73 720 Hz (pkg/s)	34 200 Hz (pkg/s)	15 900 Hz (pkg/s)	7 374 Hz (pkg/s)
Muzzle Power**	8.50×10^3 MW	3.91×10^4 MW	1.82×10^5 MW	8.44×10^6 MW	3.92×10^8 MW
Number 500 MW Reactors Required for Supply Power**	17 Reactors	78.29 Reactors	363.2 Reactors	1688.5 Reactors	7830.7 Reactors
Mass Transfer Rate**	576 tonnes/hr	2654 tonnes/hr	12 312 tonnes/hr	57 240 tonnes/hr	265 464 tonnes/hr
Percent 2003 World Steel Production**	0.48%	2.23%	10.35%	48.11%	223.12%
Material Supply Time ** For:					
Mir Space Station	7.03 minutes	1.53 minutes	19.74 seconds	4.25 seconds	0.91 seconds
International Space Station	23.75 minutes	5.15 minutes	1.11 minutes	14.33 seconds	3.09 seconds
Habitat for 10 000	25.62 months	5.56 months	1.20 months	0.26 months	0.05 months
Habitat for 70 000	127.76 months	27.73 months	5.98 months	1.29 months	0.28 months
Habitat for 820 000	298.20 months	64.72 months	13.95 months	3.00 months	0.65 months

** Quantities assume 100% drag reduction - a perfect vacuum launch corridor

The obvious trend for the geometry of the material package is, as the material package mass increases, the dimensions of the package increases since the density is assumed constant.

TREND 4: Material Package Mass $\uparrow \Rightarrow$ Dimensions of Material Package $\uparrow$

The spacing between the material packages in the stream ranges from a distance of 10.9 mm for the 1 gram material package to 409 mm for the 10 kg material package. Table 3.2 details the spacing for the other material packages. Since the spacing between the material packages is assumed to be ten times the diameter of the material package, the distance between the material package increases as the mass of the material package increases.

TREND 5: Material Package Mass $\uparrow \Rightarrow$ Spacing (distance) between Material Package $\uparrow$

The required launch velocity to reach GEO-SYNC for each material package is presented in Table 3.2 as two separate values. One velocity value assumes 0% drag reduction occurs in the launch corridor; therefore, each material package travels through the complete atmosphere. The second velocity value assumes the opposite condition, that is, the material package experience 100% drag reduction in the launch corridor or no aerodynamic drag. If no aerodynamic drag (100% drag reduction) acts on the material package, then the required launch velocity to reach GEO-SYNC was determined to be 10.31 km/s, shown in Table 3.2 for all material packages; the required launch velocity is independent of mass for the no drag situation. The no drag, vertical launch condition also serves a means of validating the calculation of the simulation spreadsheet. As stated in Section 3.5.1, the first calculation method determined the launch velocity for a no drag, vertical launch (0 degree launch angle) to GEO-SYNC altitude to be 10.306 km/s. Therefore, the analytical solution from the first model corresponds closely to the results obtained from the 1-D simulation spreadsheet. The similarity in results validates the calculation method of the simulation spreadsheet with regards to Earth's gravity field.

When full aerodynamic drag is acting on the material package, the launch velocity to reach GEO-SYNC altitude depends on the mass of the material package. From table 3.2, the required launch velocity ranges from 11.07 km/s for a 10 kg material package (a 7.4% increase over the no

drag case) to 49.79 km/s for a 1 gram material package (a 383% increase over the no drag case). Figure 3.11 is a graph of the results for the launch velocities of the material packages in Table 3.2.

Figure 3.11: Results from 1D Simulation Model – Launch Velocity and Energy for 1 Gram to 10 Kg Material Packages to GEO-SYNC

Material packages with a mass of less than about 1 kg have launch velocity requirements that exponentially increase. Therefore, material packages less than 1 kg potentially can greatly benefit from slip-stream drag reduction (i.e. 383% for 1 gram package) because reducing the drag would bring the drag line (blue line) in Figure 3.11, down near to the no drag line (red line) in Figure 3.11. Material packages greater than about 1 kg, would not benefit as much from slip-stream drag reduction since the difference between the drag and no drag lines is small, (i.e. 7.4% for 10 kg material packages). Figure 3.12 extends the size of material package to 1000 kg. Notice that the drag and no drag lines approach each other near 1000 kg.

Figure 3.12: Results from 1D Simulation Model – Launch Velocity and Energy for 1 Gram to 1000 Kg Material Packages to GEO-SYNC

Therefore, for large material packages, the effect of atmospheric drag is small. The physical reasoning behind this result is, larger material packages have more mass and less frontal area than smaller material packages; therefore, larger material packages have greater momentum at lower velocities and are not decelerated by the atmosphere as much as smaller material packages.

Therefore, the general trends observed for launch velocity is, launch velocity is constant for all mass not experiencing aerodynamic drag; however, for material packages subjected to drag forces, increasing mass causes the required launch velocity to decrease.

**TREND 6: Material Package Mass $\uparrow \Rightarrow$ Launch Velocity Constant (for 100% drag reduction)
 $\Rightarrow$ Launch Velocity $\downarrow$ (0% drag reduction)**

Figures 3.12 also depict the launch energy associated with each material package. The launch energy increases linearly with the size of the material package. The general trend is an increase in the material package mass causes the launch energy to increase.

TREND 7: Material Package Mass $\uparrow \Rightarrow$ Launch Energy $\uparrow$

The required launch frequency to achieve a stream of material packages, with 10 diameter spacing between them, is effected by the launch velocity of the material package and the size of the material package. The launch frequency ranged from 7 374 Hz for 10 kg material packages to 160 000 Hz for 1 gram material packages. For comparison, the frequency of the fastest conventional gun (GE Vulcan rotary cannon)ⁱ is about 120 shots/s (Hz) and the oscillation frequency of a quartz crystal in a wrist watch is about 32 000 Hz^k. The general trend is an increase in the material mass causes a decrease in the launch frequency.

TREND 8: Material Package Mass $\uparrow \Rightarrow$ Launch Frequency $\downarrow$

The ideal, 100% efficient, supply power needed to operate the mass driver material stream is listed in Table 3.2. The launch mechanism for launch a stream of 1 gram material packages was determined to require about 8500 MW, which is equivalent to the output of 17 nuclear reactor power generation units (500 MW each). The 10 kg material package stream launcher requires a considerably larger supply power of 3 920 000 MW, which is equivalent to about 7830 nuclear reactor units. The general trend is an increase in material package mass causes an increase in the supply power required.

TREND 9: Material Package Mass $\uparrow \Rightarrow$ Supply Power $\uparrow$

The productivity of the mass driver launch system is defined by the mass transfer rate. The calculated mass transfer rate for the material package stream is depicted in Table 3.2. The mass transfer rate for the 1 gram material package stream was determined to be 576 tonnes/hr, which is equivalent to 0.48% of the 2003 world steel production [36]. The 10 kg material package stream has a mass transfer rate of 262 464 tonnes/hr, which is equivalent to about 223% of the 2003 world steel production. The general trend is an increase in material package mass causes an increase in the mass transfer rate.

TREND 10: Material Package Mass $\uparrow \Rightarrow$ Mass Transfer Rate $\uparrow$

ⁱ Obtained from Wikipedia.org - http://en.wikipedia.org/wiki/M61_Vulcan

^k Obtained from How Stuff Works website - <http://electronics.howstuffworks.com/quartz-watch2.htm>

The usefulness of the mass driver system is defined by the time required to supply material to space construction projects. The time required to supply material continuously was determined using the mass transfer rate for each material package mass and the size of the project from Section 2.7. The 1 gram material package stream could supply construction material for a Mir station in 7 minutes, an international station in about 24 minutes, a 10 000 person habitat in about 26 months, a 70 000 person habitat in 128 months and an 820 000 person habitat in about 298 months. The 10 kg material package stream could supply construction material for a Mir station in 0.9 seconds, an international station in about 3 seconds, a 10 000 person habitat in about 0.05 months, a 70 000 person habitat in 0.28 months and an 820 000 person habitat in about 0.65 months.

3.6 Implications of Results on Open Mass Driver System Design

Below are the important trends from the results of the mathematical model.

TREND 1: $\uparrow$ Launch Angle $\Rightarrow$ $\uparrow$ Launch Velocity

TREND 2: $\uparrow$ Launch Angle $\Rightarrow$ $\uparrow$ Angular Velocity of Material Package

TREND 3: $\uparrow$ Launch Angle $\Rightarrow$ $\downarrow$ Velocity Difference between Satellite and Package

To simplify the design of the launcher, it is desired that the launch velocity requirement for the launcher to be minimum, this requires the launch angle to be minimized. To simplify the design of the catcher satellite, it is desired that the velocity difference between the material package and satellite be minimized, this requires the launch angle to be maximized. Therefore, a single launch angle value can not be beneficial both for the launcher and the catcher satellite. The selected launch angle will represent a compromise between reducing the launch velocity for the launcher and reducing the velocity difference for the catcher satellite. The preliminary design of the mass driver assumes the launch angle is near vertical, since the results of the second mathematical model are for a vertical launch. (Selecting an appropriate launch angle is deferred to the next Chapter, Chapter four, where further analysis is detailed.)

The following are the important trends from the simulation spread sheet.

**TREND 6: Material Package Mass $\uparrow \Rightarrow$ Launch Velocity Constant (100% drag reduction)
 $\Rightarrow$ Launch Velocity $\downarrow$ (0% drag reduction)**

TREND 7: Material Package Mass $\uparrow \Rightarrow$ Launch Energy $\uparrow$

TREND 8: Material Package Mass $\uparrow \Rightarrow$ Launch Frequency $\downarrow$

TREND 9: Material Package Mass $\uparrow \Rightarrow$ Supply Power $\uparrow$

TREND 10: Material Package Mass $\uparrow \Rightarrow$ Mass Transfer Rate $\uparrow$

To simplify the design of the launcher, it is desired that the launch velocity, launch energy, launch frequency and supply power be minimized. By observing the trends listed above it is obvious that simultaneously minimizing all four aforementioned values is not possible. To minimize the launch velocity and launch frequency the mass of the material package should be increased but increasing the mass of the material package increases the required launch energy and power. Therefore, just like the launch angle, the selected mass of the material package represents a trade off between the benefits of increasing the mass to reduce the launch velocity and energy and the benefits of reducing the mass to reduce the launch energy and supply power requirements.

The material transfer rate is also linked to the mass of the material package. However, it cannot be simply stated that maximizing or minimizing this parameter has a clear benefit. The next Section will use this parameter as the starting point for the preliminary design for the OpenMDS.

3.6.1 Preliminary Design for the Open Mass Driver System

The logical place to start the design is to consider the mission that the mass driver system will be used to supply construction material. The important parameter determining the usefulness of the mass driver system is the rate that it can transfer construction material to orbit. If the material transfer rate is too small, the system will be not be capable of keeping up with demand for the construction material. Conversely, if the rate is too high, the material will be supplied at a rate

greater than that could be utilized, so the material would have to be stored or discarded and the mass driver system could potentially sit idle waiting for the next job.

The desired material transfer rate is more easily determined if the time to supply material to a space construction project is considered. Consider a space construction project to construct a habitat for 10 000 people. This population of people is equivalent to the size of many small towns. From the results in table 3.2, a 10 kg material package stream operating continuously could provide the required material to this project in about 1.5 days (0.05 months). The energy supply for the 10 kg material package stream is the highest for the material package masses considered at the equivalent power of about 7830 nuclear reactor units (3 920 000 MW¹) or about 0.02% of 2002 world electrical power output [37]. However, the 10 kg material package stream offers the lowest launch velocity, in table 3.2, at 11.07 km/s for the 0% drag reduction case and the lowest launch frequency requirements (of the masses considered) at 7374 Hz (packages/s) for a 10 diameter separation.

The 1 gram material package stream provides material at a more reasonable rate for future space construction projects. The required material to build a 10 000 person space habitat, equivalent to a small town, could be supplied in a little more than two years. The supply power for the 1 gram material package launcher is the smallest, of the material package masses considered, at 8500 MW, which is equivalent to 17 nuclear reactor units or about 0.0005% of the 2002 world electrical power output [37]. However, the 1 gram material package launcher has the highest launch velocity requirements for the 0% drag reduction case at 49.79 km/s and the highest launch frequency requirements at 160 000 Hz (packages/s) to maintain a 10 diameter separation.

To better establish what an acceptable launch velocity for the baseline mass driver design should be, the launch velocity for the range of material packages masses was compared to the values published for launcher technologies in the studies mentioned in Section 2.3 and other direct launch studies. Figure 3.12 compares the material package launch velocity both for 0% and 100% drag reduction to the actual launcher systems (solid data points) and proposed launcher systems

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¹ Power requirements assume 100% launcher efficiency.

(hollow data points). Table A-1 in appendix A, lists the launch velocities and references were the data points in Figure 3.13 were obtained from.

Figure 3.13: Comparison of Launch Velocity for Present and Proposed Launcher Technology to the Material Package Stream Mass Driver

The minimum launch velocity that the launch technology must achieve for the baseline design is the 10.31 km/s for the ideal case that the material packages travel with 100% drag reduction. The potential launch technology would preferably exceed this minimum launch velocity because it is possible that the launch corridor conditions will not cause a 100% reduction in aerodynamic drag. The maximum launch velocity requirement for the launcher technology corresponds to the velocity required to launch the material packages with 0% drag reduction. This possible range of velocity values is depicted in Figure 3.13 as a red shaded region.

Therefore in Figure 3.13, it can be seen that two actual railgun designs (two solid triangles) have accelerated mass in the order of grams to velocities that exceed the 10.31 km/s minimum requirement. However, when the maximum launch velocity requirement for the material packages is considered, one of the actual railgun designs meet the velocity requirement for all material package masses greater than about 100 grams. It must be pointed out that, these two actual railgun designs are based on accelerating small masses, in the order of grams; scaling the railgun design up to accelerate masses greater than 100 grams might be technically difficult. Two proposed railgun designs, (one being the NASA Nuclear waste concept of Section 2.3.6), propose utilizing railgun technology capable of exceeding the minimum launch velocity and meet the maximum launch velocity for material package masses greater than nearly 10 grams. The Lofstrom launch loop of Section 2.3.10, is depicted in Figure 3.13 as the two proposed novel launchers (blue star data points) capable of achieving and slightly exceeding the minimum launch velocity requirements. The majority of the actual and proposed launcher technologies depicted in Figure 3.13 are below the minimum launch velocity because most of the investigation into direct launch to space is focussed on delivering satellites to LEO.

To better establish the launch energy requirements for the range of material package masses, the launch energy for the 0% and 100% drag reduction were compared to the launch energy of actual and proposed launcher technologies in Figure 3.14.

From Figure 3.14, the three conventional cannon launchers (solid blue diamonds) meet the required launch energy for the entire range of material packages considered. The conventional cannon launchers however, do not meet the minimum velocity requirement (seen in Figure 3.13); therefore, this technology is not a suitable for the mass driver launcher technology. However, four actual railgun designs (solid red triangles centered around the 1 kg launch package) exceed the energy requirements for the material packages with mass less than about 100 grams. An actual light gas gun design is also capable of meeting the energy requirements for material packages with mass less than about 1 kg. Proposed railgun, coilgun, and the novel launchers (Molder launch rail of Section 2.3.7 and Lofstrom launch loop of Section 2.3.10) have launch energies that exceed the requirements of the material packages.

Figure 3.14: Comparison of Launch Energy for Present and Proposed Launcher Technology to the Material Package Stream Mass Driver

Therefore, for the preliminary OpenMDS, when considering the material transfer rate, launch velocity and launch energy effects, it seems that the extreme material package masses, the 1 gram and 10 kg material package streams, have drawbacks that outweigh the benefits. Even though the 1 gram material package stream has very desirable material transfer rate the 1 gram stream has the risk of requiring launch velocity that is nearly five times the minimum launch velocity if the launch corridor drag reduction effects are small. Additionally, the 160 000 Hz (packages/second) launch frequency is unlikely to be achieved in the near term since the launch frequency is three orders of magnitude greater than the fastest rotary cannon and more than four times greater than the oscillation frequency of the quartz crystal in a wrist watch.

On the other end of the mass range, as previously mentioned, the 10 kg material package has a material transfer rate that is capable of supplying enough material to construct a 10 000

person habitat in about 1.5 days. This material transfer rate is obviously too high, since it is unlikely that any near term construction process could utilize material at this rate since it is equivalent to building an entire town in a day and a half. From table 3.2 the rate of material transferred, associated with the 10 kg material package, is equivalent to more than double (223%) the 2003 world production of steel. Also the power requirements of 7830 nuclear reactor units for the 10 kg launcher seem unpractical for any near term launch facility.

Therefore, a trade-off exists between the potential high launch velocity and launch frequency of the 1 gram material package stream and the high material transfer rate of the 10 kg material package stream, it was decided that the baseline mass driver design will utilize a 10 gram material package stream. The 10 gram material package stream has the potential of requiring a launch velocity of only about twice the minimum launch velocity or 21.18 km/s and the launch frequency is lowered to less than half the 1 gram frequency at 73 720 Hz (packages/s). The number of nuclear reactor units to supply the 10 gram material package stream is about four times greater than the 1 gram material package stream at 79 reactor units but much less than the 10 kg material package stream. The material transfer rate for the 10 gram material package stream is capable of supply all the construction material to a 10 000 person habitat in about 5.56 months.

Considering the launch velocity and energy comparison of figures 3.13 and 3.14, it seems that the railgun technology is the most capable technology since actual systems have been demonstrated that achieve the velocity and energy requirements for the minimum requirements and proposed railgun designs nearly meet the maximum launch velocity requirements for the no drag reduction case. None of the potential launch technologies reviewed in Section 2.5 are capable of rapid successive launches. The railgun and coilgun seem to be the easiest technologies to be enhanced to provide rapid launch capability, since conceptually it would be 'merely' a matter of providing sufficient electrical energy, an armature, and a material package. The coilgun was not chosen since actual and proposed designs do not meet the minimum velocity requirements. The light gas gun, Molder's launch rail and the conventional cannon were not chosen because the designs do not easily allow rapid successive launch capabilities. The Lofstrom launch loop seems to be capable of the minimum launch velocity and energy requirements however this technology was not chosen because the concept requires further research and demonstration.

Figure 3.15 summarizes the preliminary design of the mass driver system. This preliminary design represents the first design cycle for the mass driver system. The resulting performance of the 10 gram material package stream is not ideal. In fact, the material transfer rate for continuous operation seems too high to practical for any near term space project and even the minimum velocity requirement are challenging to achieve with present and proposed launch technologies. Therefore, instead of continuing the design work on this preliminary concept, the next Chapter, Chapter four, describes the next iteration of the design cycle. The goal of the design work describe in Chapter four, is to define a mass driver system with lower launch velocity requirements, lower mass transfer rate and generally a system design that could be useful in the near term for the supply of space construction projects.

Figure 3.15: Preliminary Design for the Mass Driver Material Package System

Chapter 4 Evaluation and Design of an Improved Open Mass Driver System

4.1 Overview of the Design Approach

A design goal of the OpenMDS is to create a prototype design that is interesting and feasible in order to attract a development community. The early results of the open engineering attempts (see Section 5.4) indicated a feasible ‘prototype’ OpenMDS design was required to attract volunteer developers. Therefore, the design goal for the project was to quickly identify a design for the mass driver that would be potentially feasible and have interesting design elements to attract the attention of potential volunteers. It was accepted that the proposed design may have errors but it is these errors that may entice a person with expertise to join the project to improve the design. OSS and open engineering projects are more tolerant to errors and more capable of taking risks and failing than commercial design projects because the cost of failure for OSS and open engineering projects is negligible. It is through these series of failures the path to a successful design may be found. The proposed design process is summarized by the IDEO motto, “Fail often to succeed sooner” [38].

The preliminary design presented in Chapter 3 has performance characteristics that are not practical for any near term space construction project. However, through the process of defining the preliminary OpenMDS of Chapter 3, a general approach to the design of the mass driver was realized. The design process can be simplified to specifying the design parameters of five sub-systems. These five sub-systems are depicted in Figure 4.1, as the launch trajectory, orbital trajectory, launcher mechanism, material package and the ‘catcher’ satellite (CatSat).

Figure 4.1 also depicts the design parameters that are either shared between the sub-systems or affect the sub-system. The overall design of the mass driver system is complicated by the interrelatedness of the sub-systems. Changing a design parameter to improve the design of one sub-system may have a negative effect on another sub-system.

The purpose of this chapter is to improve the design of the preliminary design presented in Chapter 3. This Chapter begins the design process by describing the evaluation and selection of the appropriate launch trajectory. From there, the orbital trajectory of the CatSat was considered.

Figure 4.1: Sub-Systems for Mass Driver Design

Once the launch and orbital trajectories are specified, the requirements from these trajectories are used to consider the design of the appropriate launcher mechanism, material package and CatSat. It must be indicated that the design detailed in Chapter four has been exclusively performed by the author under the guidance of Dr. F. A. Salustri. The design results of Chapter four are not the result of the Open Engineering collaboration.

4.2 Launch Trajectory Evaluation

The preliminary OpenMDS design proposed in Section 3.6.1 has minimum launch velocity requirements that are at the upper limit of demonstrated present launcher technology. If the launch corridor is not the ideal case of a perfect vacuum, then the launch velocity requirements increase and are satisfied only by proposed railgun designs that require further research and experimentation. Therefore, for the mass driver design to be practical in the near term, it would be prudent to reduce the range of required launch velocity to a level that is easily achievable with present technology. Since the ideal case assumes no aerodynamic drag is acting on the material package, the only means of reducing the launch velocity is by reducing the apogee height from GEO-SYNC to a lower orbit.

Two lower apogee altitudes are considered in the following analysis, one at 10 000 km and the second at 400 km altitude. The 10 000 km altitude is close to the 11 000 km altitude identified in the results for the 2D gravity only model of Section 3.51, as the altitude that corresponds to a zero velocity difference between the material package and CatSat for the 55 degree launch angle. Therefore, at the 10 000 km altitude, it is expected that a reasonable launch angle should minimize the velocity difference between the material package and the CatSat. The 400 km altitude represents a typical height for LEO. Since 400 km is very close to Earth compared to the GEO-SYNC altitude of 35 793 km, the launch velocities should be substantially reduced.

However, launching to a lower altitude complicates the design of the OpenMDS. Since at altitudes are lower than GEO-SYNC, the CatSat is no longer stationary with respect to the launcher on the ground. If the CatSat is located in an orbit lower than GEO-SYNC, then the CatSat has an orbital period less than the 24 hour rotation rate of Earth. Therefore, launching material packages to the CatSat can no longer be a continuous operation; there will be a period of time when the satellite is visible to the launcher and a period of time when the satellite is not visible to the launch site on Earth. The duration of these periods will depend on the type of orbit and the orbital altitude of the CatSat. The duration of time when launch of material packages is possible is known as the launch window. The following Section, Section 4.2, will investigate the launch window for the selected launch trajectory.

The results from the 2-D gravity only model discussed in Section 3.51, determined that there is a very large velocity difference between the material package at apogee and the orbiting CatSat for all launch angles except the 55 degree launch angle. These results indicate that at low launch velocities, thus altitudes less than GEO-SYNC, the velocity difference between the material package and a CatSat in a circular orbit (at the apogee altitude) is very high and range from about 1000 m/s (approximately the velocity of a bullet) to about 7000 m/s. Designing a capture system on the CatSat capable of stopping multiple high velocity bodies probably is not practical. If the design was possible, then the 'catcher' satellite would require a large amount of energy to maintain its orbit, due to the reaction force created by the capture event. The capture event will occur over a very short time interval due to the high velocity difference. During this brief time interval, the CatSat has only one attempt at capturing the material package since the material package would

return to Earth on the ballistic launch trajectory. Therefore mitigating the velocity difference is an important design consideration.

The traditional method of changing the velocity of spacecraft in space is with a rocket motor. Firing a rocket motor onboard the material package could reduce the velocity difference by increasing the velocity of the material package. The velocity of the material package could be increased enough to place the material package in a minimum energy elliptical orbit, with the perigee (lowest orbital altitude) just above Earth's sensible atmosphere (about 200 km). If the perigee altitude was set much lower than this, the material package orbit would rapidly decay due to atmospheric drag and the material package would return to Earth. Placing the material package in orbit is beneficial because the aforementioned velocity difference would be reduced and the CatSat has more than one attempt to capture the material package. A CatSat, located in a suitable orbit, could potentially rendezvous with the material package numerous times.

To evaluate launching material packages from Earth and firing a rocket motor for orbital insertion, the 1-D simulation spreadsheet described in Section 3.3.2, was modified to calculate a 2-dimensional launch from Earth's equator and includes a means of determining the effect and mass of a rocket motor burn at apogee. Figure 4.2 depicts the development process of the simulation spreadsheets created for the launch trajectory analysis. Note the 2-D gravity-only model, of Section 3.2.1, is omitted from this graphic, it should appear before the 1-D Launch Simulation spreadsheet. The next Section describes the functions of the new 2-Dimensional simulation models created.

Figure 4.2: Simulation Spreadsheet Development Process for Launch Trajectory Analysis

4.2.1 Improved 2-Dimensional Simulation Spreadsheet Models

The Earth_Sim spreadsheet simulates a 2-dimensional launch from Earth's equator with Earth and Moon gravity and aerodynamic drag acting on the material package. This spreadsheet is similar to the 1-D simulation spreadsheet, detailed in Section 3.3.2, except the launch scenario is extended to a 2-dimensional launch from Earth's equator. The user is required to input the material package mass, diameter, launch angle and launch velocity. These inputs are depicted as the blue boxes in Figure 4.2 for the Earth_Sim.xls spreadsheet in the upper left corner. The spreadsheet calculates the apogee altitude, launch energy and other resulting values for the material package. The output from the simulation is depicted as the green box in Figure 4.2. Like the 1-D simulation spreadsheet, the launch velocity was manually adjusted until the material package obtained the apogee altitudes of 10 000 and 400 km.

The Earth_Sim spreadsheet was revised to include a means of calculating the effect and size of an onboard solid rocket motor. The revised version is known as Earth_Sim_RevB.xls. The input parameters for the revised spreadsheet are the same as the previous version, except the user must also specify the velocity addition created by the firing of the rocket motor and the rocket motor performance parameters. The blue boxes in Figure 4.2 depict the input parameters to the Earth_Sim_RevB model. The output parameters are the same as the Earth_Sim spreadsheet except the mass of the required rocket motor is calculated and the resulting orbital trajectory is displayed in a graphic.

Figure 4.2 depicts that the Earth_Sim_RevB model was revised to include the effects of Earth's rotation. The new model, Earth_Sim_RevC, has the same input and output parameters as the Earth_Sim_RevB model.

The final revision of the simulation spreadsheet was done to approximate the effects of the launch corridor. Like the method used for the 1-D simulation spreadsheet of Section 3.3.2, the user specifies the percentage of drag reduction created by the launch corridor. The following Sections will discuss the calculation method implemented by the Earth_Sim spreadsheet and the subsequent revisions, the results and a discussion of hypersonic slip-stream effects in the launch corridor.

4.2.2 Calculation for Improved 2-D Simulation Spreadsheet Models

The Earth_Sim spreadsheet and its subsequent revisions perform the calculations in a similar manner to the 1D simulation spreadsheet of Section 3.4.2. The obvious difference between the calculations methods is the Earth_Sim models have a second dimension. The other differences are the rocket motor impulse calculations and the calculation for Earth's rotation. The following Sections describe the assumptions and input, output and constant parameters for the Earth_Sim spreadsheet model. The calculation method for Earth_Sim.xls and the revisions are detailed in Appendix B.

4.2.2.1 Assumptions

1. The rotation of the Moon and Earth is neglected. (*Only for Earth_Sim.xls and Earth_Sim_RevB.xls is Earth Rotation neglected*)
2. The launch plane (2-D) is the Earth-Moon equatorial plane.
3. The simulation is limited to three bodies, the Earth, material package and Moon.
4. The simulation model assumed constant acceleration during each time step.
5. The lower atmospheric density was assumed to be defined by a density function defined by data from the ICAO (International Civil Aviation Organisation) Standard Atmosphere [33].
6. The upper atmosphere was assumed to follow a density function defined by the 1976 US Standard Atmosphere [34].
7. The atmospheric temperature was defined by functions for each temperature region depicted in Figure B-2 [33].
8. All temperature functions between temperature regions were assumed to be linear.
9. Experimental total drag coefficient for a hypervelocity missile as a function of Mach number was used [31]. This drag co-efficient assumed a fin stabilised missile with an L/D of 24.
10. The material package was assumed to be composed of steel and the geometry was assumed to be a simple cylinder.
11. Velocity additions to "orbit" the material packages occur at apogee and utilise a solid rocket motor [39]. (*Does not apply to Earth_Sim.xls*)
12. Velocity additions to "orbit" the material packages occur during the time duration of one time step, an instantaneous rocket impulse. (*Does not apply to Earth_Sim.xls*)

4.2.2.2 Input, Output and Constant Parameters

Input Parameters Specified by User

- Time step (t_{step}) – the time duration between each calculation in the simulation (s)
Time step multiplier ($t_{\text{step Multiplier}}$) — effects the length of the time steps. It is used to provide short time steps during launch.
Launch Velocity (v_{Launch}) – the initial velocity of the material package at exit of launcher (m/s)
Material Package Mass (m_{package}) – total mass of material package (kg)
Material Package Diameter (D_{package}) – the diameter of the material package (m)
Material Package Density (ρ_{package}) – the density of the material package (kg/m^3)
Launch Frequency (ω_{launch}) – the launch frequency of material packages in launch corridor (Hz)
Propellant mass fraction - ($m.f._{\text{prop}}$) – for the rocket motor (specified by user – used 0.93 [39])
Specific Impulse (I_{sp}) – performance parameter of rocket motor (s) (specified by user – used 290s average from [39]).
Slip Stream Drag Reduction ($C_{\text{slip-stream}}$) – the percent drag reduction caused by the launch corridor (%)

Output Parameter of Simulation Spreadsheet

- Frontal Area (A_f) – frontal area of the material package (m^2)
Volume (V_{package}) – volume of material package (m^3)
Length (L_{package}) – length of material package (m)
Length to Diameter (L/D) Ratio – Length to Diameter ratio of material package
Launch Energy (E_{Launch}) – Launch energy of material package (MW)
Material Package Spacing (d_{spacing}) - Distance between material packages in stream
Mass Flow Rate (MFR) – the rate of mass transfer from the launcher mechanism (kg/s)
Launch Power (P_{Launch}) – the power used for the launch process
Maximum Altitude (h_{max}) – the highest altitude the material package obtains (m) (manually found in simulation table)
Mass fraction of the payload ($m.f._{\text{payload}}$) – the fraction of the material package mass that is devoted to construction material.
Mass of the rocket motor (m_{rocket}) – the mass of the rocket motor required to produce the specified velocity addition (Δv) (kg)

Constant Required

- Mass of Earth (M_{Earth}) = 5.976×10^{24} kg
Radius of a spherical Earth (r_{Earth}) = 6371 km = 6 371 000 m
Velocity of Earth's rotation (v_{Earth}) at the equator = 465.102 m/s
Gravitational Constant (G) = $6.67 \times 10^{-11} \text{ m}^3/\text{kg s}^2$ [3]
Specific Heat Ratio = 1.4 (for air)
Real Gas Constant (R) = 287 J/(kg K) [35]

4.2.3 Results for the Improved 2-D Spreadsheet Models

The final revision of the spreadsheet, *Earth_Sim_RevD.xls*, was used to generate the results detailed in this Section. To verify the operation of the *Earth_Sim.xls* spreadsheet, the vertical launch of a 10 gram material package with no drag reduction to a GEO-SYNC altitude was calculated. This launch scenario is identical to the scenario calculated using the 1-D simulation spreadsheet of Section 3.5.2, where the required launch velocity was determined to be 21.18 km/s. The required velocity for this launch scenario using the *Earth_Sim_RevD* simulation spreadsheet was determined to be 21.7 km/s. The two resulting launch velocities are very similar; therefore, the calculation method of the *Earth_Sim* spreadsheet seems to be valid for vertical launch scenarios. The validity for launching at angle was not explicitly determined; however, the results presented below correspond with the trend observed from the analytical model of Section 3.5.1, that is an increasing launch angle requires an increasing launch velocity to reach any given altitude. There is a slight anomaly in this trend for each of the results table listed below. The 0 degree launch angle has a higher launch velocity than the 10 and 20 degree launch angle. The cause of the anomaly has not been extensively investigated, but it is suspected that the probable cause is the averaging of the x and y accelerations in the time step calculation of Part R of Appendix B.

Table 4.1 represents the data obtained from the *Earth_Sim_RevD* spreadsheet, for the launch of a 10 gram material package from sea-level to a 400 km altitude with the launch angle varying from 0 degrees (vertical launch) to 50 degrees. Two launch velocities are listed in the table. The first velocity corresponds to the 0% drag reduction case and the second velocity corresponds to the 100% drag reduction (no drag) case. All launch velocity values in table 4.1, represent velocities that are attainable by present launcher technology (see Figure 3.13 in Section 3.6.1). The spreadsheet was used to calculate the mass of the rocket motor needed to place the material package in the minimum energy, elliptical orbit. The rocket motor performance was specified as a solid rocket motor with a specific impulse of 290s and a propellant mass fraction of 0.93, which correspond to values for typical high performance solid rocket motors [39].

Table 4.1: Results from a 2D Simulation Model for Launch of 10 Gram Material Packages to a 400 km Orbit at Launch Angles from 0 to 50 Degrees

Parameter						
Launch Destination (km)	400	400	400	400	400	400
Launch Velocity (km/s)	4.88	4.81	4.96	5.37	6.18	7.79
Launch Velocity - No Drag (km/s)	2.72	2.75	2.85	3.05	3.36	3.84
Velocity to Overcome Drag (km/s)	2.16	2.06	2.11	2.32	2.82	3.95
Launch Angle (from vertical) (degrees)	0	10	20	30	40	50
Launch Altitude (m)	0	0	0	0	0	0
Launch Longitude (degrees)	20	20	20	20	20	20
Pellet Mass (kg)	0.01	0.01	0.01	0.01	0.01	0.01
Pellet Diameter (m)	0.00409	0.00409	0.00409	0.00409	0.00409	0.00409
Launch Energy (MJ/Pellet)	0.12	0.12	0.12	0.14	0.19	0.3
Energy Saved if No Drag (MJ/pellet)	0.08	0.08	0.08	0.09	0.13	0.23
Percent Energy Savings (%)	69.17	68.49	66.16	66.78	70.29	75.42
Solid Rocket Specific Impulse	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)
Propellant Mass Fraction	0.93	0.93	0.93	0.93	0.93	0.93
Delta Velocity Added at Apogee to Orbit Payload* (m/s)	7050	6725	6450	6150	5850	5500
Total Impulse (N s)	70.5	67.25	64.5	61.5	58.5	55
Weight of Motor Required (kg)	0.027	0.025	0.024	0.023	0.022	0.021
Payload Mass Fraction	-1.67	-1.54	-1.44	-1.33	-1.21	-1.08
Time Step Multiplier**	13	175	225	300	500	1000
Max Time Step	0.5	0.5	0.5	0.5	0.5	0.5

* Delta Velocity for Elliptical Orbit Perigee at 200 km above Earth Surface

** Time Step Multiplier adjusted to allow 2000 cells before maximum time step is reached.

Figure 4.3 depicts a typical graphic for the launch trajectory and the elliptical orbit obtained from the Earth_Sim spreadsheet for launch to 400 km altitude. It can be observed that the shape of this orbit is nearly circular. Table 4.1 indicates that the mass of the rocket motor required to place the material package in this elliptical orbit was greater than the mass of the material package itself for all launch angles. Therefore, 400 km altitude is an unsuitable orbit, since the required rocket velocity addition to place the material package in a minimum energy elliptical orbit, requires a rocket motor mass greater than the mass of the entire material package.

The next step in the evaluation was to determine if a 10 gram material package could be launched to a 10 000 km altitude and placed into a similar elliptical orbit with a suitable sized rocket

Figure 4.3: Output Graphic for 400 km Elliptical Orbit

Figure 4.4: Output Graphic for 10 000 km Elliptical Orbit

motor. Figure 4.4 depicts the typical graphical output for the launch of the material package to 10 000 km altitude and insertion into an elliptical orbit with perigee altitude of 200 km. Table 4.2 summarizes the result data obtained from the simulation spreadsheet.

Table 4.2: Results from a 2D Simulation Model for Launch of 10 Gram Material Packages to a 10 000 km Orbit at Launch Angles from 0 to 50 Degrees

Parameter						
Launch Destination (km)	10000	10000	10000	10000	10000	10000
Launch Velocity (km/s)	15.76	15.2	15.21	15.9	17.4	20.71
Launch Velocity - No Drag (km/s)	8.74	8.69	8.69	8.71	8.76	8.83
Velocity to Overcome Drag (km/s)	7.02	6.51	6.52	7.19	8.64	11.88
Launch Angle (from vertical) (degrees)	0	10	20	30	40	50
Launch Altitude (m)	0	0	0	0	0	0
Launch Longitude (degrees)	20	20	20	20	20	20
Pellet Mass (kg)	0.01	0.01	0.01	0.01	0.01	0.01
Pellet Diameter (m)	0.00409	0.00409	0.00409	0.00409	0.00409	0.00409
Launch Energy (MJ/Pellet)	1.24	1.16	1.16	1.26	1.51	2.14
Energy Saved if No Drag (MJ/pellet)	0.86	0.78	0.78	0.88	1.13	1.75
Percent Energy Savings (%)	69.20	67.45	67.45	69.90	74.59	81.78
Solid Rocket Specific Impulse	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)
Propellant Mass Fraction	0.93	0.93	0.93	0.93	0.93	0.93
Delta Velocity Added at Apogee to Orbit Payload* (m/s)	3190	2720	2370	2070	1775	1470
Total Impulse (N s)	31.9	27.2	23.7	20.7	17.75	14.7
Weight of Motor Required (kg)	0.0121	0.0103	0.009	0.0078	0.0067	0.0055
Payload Mass Fraction	-0.21	-0.03	0.1	0.22	0.33	0.44
Time Step Multiplier**	10000	10000	10000	10000	15000	25000
Max Time Step	1	1	1	1	1	1

* Delta Velocity for Elliptical Orbit Perigee at 200 km above Earth Surface

** Time Step Multiplier adjusted to allow 2000 cells before maximum time step is reached.

Launching at 0 and 10 degree launch angles are not suitable since the required rocket motor mass is greater than the total material package mass of 10 grams. The launch angles from 20 to 50 degrees have rocket motor masses less than the 10 gram total. The payload mass fraction ranges from 0.1, that is 1 gram construction material and 9 grams rocket motor mass, for a 20 degree launch and 0.44, which is 4.4 grams construction material and 6.6 grams rocket motor mass, for a 50 degree launch angle. The launch velocities for launching 10 gram material packages to a 10 000 km altitude for the no drag reduction case, are greater than 15 km/s which exceeds the capabilities of present launcher technology, see Figure 3.13 in Section 3.6.1. The 10 gram material package was not further considered for two reasons. Firstly, the rocket motor mass estimate does not include the system required to determine when to fire the rocket; this system would likely require the remainder of the material package, even if this system was as simple as pyrotechnic fuse. Secondly, the launch velocity for the no drag reduction case exceeds the launch velocity capabilities of present launch technology; lower launch velocities are desired.

Since the 10 gram material package is unsuitable, the next step is to evaluate material packages with larger masses. Increasing the mass obviously aids the problem for the firing system; however, according to the Trend 6 in Section 3.5.2, increasing the material package mass also reduces the required launch velocity. The 1 kg material package mass was chosen to be evaluated next, for launch and insertion to an elliptical orbit with a perigee altitude of 200 km and an apogee altitude of 400 km or 10 000 km. The 1 kg material package represents the middle of the remaining material package mass range of 100 g to 10 kg; therefore, if the 1 kg material package is determined to be unsuitable, the evaluation would consider either a lighter or heavier material package.

Table 4.3 represents the results obtained for launching a 1 kg material package into an elliptical orbit with apogee and perigee of 400 km and 200 km altitude, respectively. The results are similar to the results for the 10 gram material package, in that for a launch angle from 0 to 50 degrees, the mass of the required rocket motor is greater than the mass of the entire material package. The evaluation was extended to launch angles of 60 and 70 degree and it was determined that at 70 degrees the mass of the rocket motor was 0.85 kg which leaves 150 grams for the rocket firing system and construction material. The launch velocity to accomplish the launch at 70

degrees is 9.18 km/s for the no drag reduction case and 5.59 km/s for the 100% drag reduction (no drag) case. The velocity to overcome drag is about 39% of the launch velocity; this is more than the 5% estimated for NASA nuclear waste disposal railgun of Section 2.3.6, but this is expected since the size of the NASA launch mass was greater than 2000 kg and Figure 3.11 of

Table 4.3: Results from a 2D Simulation Model for Launch of 1 Kg Material Packages to a 400 km Orbit at Launch Angles from 0 to 70 Degrees

Parameter								
Launch Destination (km)	400	400	400	400	400	400	400	400
Launch Velocity (km/s)	3.165	3.15	3.23	3.45	3.87	4.58	5.91	9.18
Launch Velocity - No Drag (km/s)	2.715	2.742	2.847	3.042	3.355	3.832	4.548	5.586
Velocity to Overcome Drag (km/s)	0.45	0.408	0.383	0.408	0.515	0.748	1.362	3.594
Launch Angle (from vertical) (degrees)	0	10	20	30	40	50	60	70
Launch Altitude (m)	0	0	0	0	0	0	0	0
Launch Longitude (degrees)	20	20	20	20	20	20	20	20
Pellet Mass (kg)	1	1	1	1	1	1	1	1
Pellet Diameter (m)	0.019	0.019	0.019	0.019	0.019	0.019	0.019	0.019
Launch Energy (MJ/Pellet)	5.01	4.96	5.22	5.95	7.49	10.49	17.46	42.14
Energy Saved if No Drag (MJ/pellet)	1.32	1.20	1.17	1.32	1.86	3.15	7.12	26.54
Percent Energy Savings (%)	26.43	24.21	22.36	22.24	24.86	30.01	40.77	62.98
Solid Rocket Specific Impulse	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)
Propellant Mass Fraction	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93
Delta Velocity Added at Apogee to Orbit Payload* (m/s)	7150	6750	6325	5880	5365	4725	3825	2250
Total Impulse (N s)	5878	5471.5	5032	4598.7	4118.1	4090.2	3825	2250
Weight of Motor Required (kg)	2.22	2.07	1.9	1.74	1.56	1.55	1.45	0.85
Payload Mass Fraction	-1.22	-1.07	-0.9	-0.74	-0.56	-0.55	-0.45	0.15
Time Step Multiplier**	13	13	13	13	13	15	25	100
Max Time Step	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5

* Delta Velocity for Elliptical Orbit Perigee at 200 km above Earth Surface

** Time Step Multiplier adjusted to allow 2000 cells before maximum time step is reached.

Section 3.5.2, depicted the launch velocity for the drag and no drag cases approach each other as mass approach 1000 kg.

The percent of energy used to overcome drag is 62.98% for the 70 degree launch angle and 26.43% for the vertical (0 degree) launch angle. In fact all angles below 50 degrees, require about one quarter of the launch energy to overcome drag. The energy lost to drag most likely would go to creating heat; therefore, launching at an angle of 70 degree would benefit most from the potential drag reducing effects created by the slip-stream launch corridor. The slip-stream drag effects could reduce the launch energy requirements for a material package launched at 70 degrees by over 60%. Therefore, the launch of a 1kg material package at 70 degrees launch angle into an elliptical orbit is a possibility; however, before selecting this scenario, the launch to 10 000 km is considered next.

Table 4.4 contains the results for the launch of a 1 kg material package to an elliptical orbit with apogee and perigee of 10 000 km and 200 km, respectively. From table 4.4 it is evident that 0 and 10 degree launch angles are not feasible since required rocket motor mass exceed 1 kg. At the other end of the launch angle range, the 70 degree launch requires a launch velocity that is at the upper limit of present launcher technology; however, the payload mass fraction is very high at 0.9, that is, 0.1 kg rocket motor mass and 0.9 kg for firing system and construction material. Again the launch at 70 degrees would benefit from the slip-stream reduction effects of the launch corridor since about 62.02% of the launch energy is required to overcome drag; therefore, a 62.02% energy savings is possible if the launch corridor was a perfect vacuum (no drag).

From inspection of Table 4.4, the results for launch angles from 40 to 70 degrees represent the region where the plausible design values for the mass driver system exist. At the low end of the launch angle range, the 40 degree launch angle requires a launch velocity of 10.4 km/s for the no drag reduction case and 8.75 km/s for the 100% drag reduction (no drag) case. The percent of energy associated with drag is 29.13%; therefore, a 29.13% energy savings would be possible if a perfect vacuum launch corridor condition (no drag) existed.

Table 4.4: Results from a 2D Simulation Model for Launch of 1 Kg Material Packages to a 10 000 km Orbit at Launch Angles from 0 to 70 Degrees

Parameter								
Launch Destination (km)	10000	10000	10000	10000	10000	10000	10000	10000
Launch Velocity (km/s)	9.94	9.82	9.81	9.96	10.4	10.92	12.05	14.51
Launch Velocity - No Drag (km/s)	8.736	8.688	8.678	8.702	8.755	8.824	8.898	8.964
Velocity to Overcome Drag (km/s)	1.204	1.132	1.132	1.258	1.645	2.096	3.152	5.546
Launch Angle (from vertical) (degrees)	0	10	20	30	40	50	60	70
Launch Altitude (m)	0	0	0	0	0	0	0	0
Launch Longitude (degrees)	20	20	20	20	20	20	20	20
Pellet Mass (kg)	1	1	1	1	1	1	1	1
Pellet Diameter (m)	0.019	0.019	0.019	0.019	0.019	0.019	0.019	0.019
Launch Energy (MJ/Pellet)	49.4	48.22	48.12	49.6	54.08	59.62	72.6	105.77
Energy Saved if No Drag (MJ/pellet)	11.24	10.48	10.47	11.74	15.75	20.69	33.01	65.59
Percent Energy Savings (%)	22.76	21.73	21.75	23.66	29.13	34.70	45.47	62.02
Solid Rocket Specific Impulse	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)	290 s (Avg)
Propellant Mass Fraction	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93
Delta Velocity Added at Apogee to Orbit Payload* (m/s)	3475	2900	2405	1920	1420	1065	655	275
Total Impulse (N s)	3475	2900	2405	1920	1420	1065	655	275
Weight of Motor Required (kg)	1.31	1.1	0.91	0.73	0.54	0.4	0.25	0.1
Payload Mass Fraction	-0.31	-0.1	0.09	0.27	0.46	0.6	0.75	0.9
Time Step Multiplier**	250	1000	1000	1000	1000	1000	1000	1000
Max Time Step	1	1	1	1	1	1	1	1

* Delta Velocity for Elliptical Orbit Perigee at 200 km above Earth Surface

** Time Step Multiplier adjusted to allow 2000 cells before maximum time step is reached.

Determining the optimum values for the launch trajectory requires a design trade-off analysis. It is desired that the launch velocity be reduced but reducing the velocity reduces the launch angle, payload mass fraction and the percent of launch energy associated with drag. It is desired that the payload mass fraction be maximized since the purpose of the mass driver system is

to transport material to space. The purpose of the proposed material stream mass driver is to exploit the drag reducing slip-stream effects of the launch corridor; therefore, it is also desired to maximize the energy savings for creating the ideal case of a no drag launch corridor.

At this stage of design, it is difficult to define if minimizing launch velocity is more important than increasing the payload mass fraction or maximizing the energy associated with drag (potential slip-stream energy saving). Therefore, to advance the design process further, the 50 degree launch angle to a 10 000 km altitude was selected as a design parameter since it is roughly the middle of the range of angles considered and represents a conservative set of design parameters. The launch velocity, for the 50 degree launch angle, ranges from 8.824 km/s for the 100% drag reduction (no drag) case and 10.92 km/s for the 0% drag reduction case. The payload mass fraction is 0.6, which is 0.4 kg for the rocket motor mass and 0.6 kg for rocket firing system and construction material. The energy associated with drag is 34.70%, which would represent a moderate energy savings for a perfect vacuum launch corridor.

Table 4.5 contains the estimated performance parameters for the selected 1 kg material package stream with either 0% or 100% drag reduction. Since the 10 diameter separation or spacing distance between the material packages, assumed in the results of Section 3.5.2, was merely an estimate, the separation distance range was extended to determine the effects separation has on the OpenMDS design. The separation distance was varied from the original 10 diameters to a distance of 1000 diameters which would likely represent the upper limit where any slip-stream effects would occur.

Table 4.5: Performance Values for the Selected 1 kg Material Package Stream

Case	Launch Frequency (Hz)	Mass Flow Rate (kg/s)	Percent of 2003 World Steel Production	Supply Power (GW)	No. of 500 MW Nuclear Reactor Units
0% Drag Reduction					
10 Dia Spacing	16850	10110	51.0	1004.7	2010
100 Dia Spacing	4640	2784	14.0	276.7	554
1000 Dia Spacing	560	336	1.7	33.4	67
100% Drag Reduction					
10 Dia Spacing	13620	8172	41.2	530.2	1061
100 Dia Spacing	3740	2244	11.3	145.6	292
1000 Dia Spacing	453	271.8	1.4	17.6	36

The values in table 4.5 range from the worst scenario^m case for launch corridor effects, where 10 diameter spacing creates no drag reduction to the performance of a best case scenario launch corridor effects where 1000 diameter spacing creates 100% drag reduction.

It can be seen in table 4.5, that the separation distance has a significant effect on the launch frequency and the supply power requirements of the OpenMDS design. The launch frequency ranges from a high of 16 850 Hz (half the oscillation speed of a quartz crystal in a wristwatch) to a low of 453 Hz (close to 100 Hz of conventional rotary cannon technology). The supply energy requirements range from a plausible 36 reactor units to a demanding 2010 reactor units. The mass transfer rates are not compared to the space construction supply times, since the duration of the launch window has not been determined yet. Section 4.2 will review the evaluation of the launch window.

From the results in table 4.5, it is obvious the design of the mass driver is dependent on determining the drag reduction effects associated with the slip-stream effects in the launch corridor. The next Section will review the work performed to estimate the likely effect of the slip-stream drag about hypersonic bodies.

4.2.4 Estimating the Hypersonic Slip-Stream Effects

It was determined early in this project that evaluating the actual hypersonic slip-stream effects which occur in a material package stream, operating at steady state, is beyond the scope of this project. Instead of directly determining the slip-stream effects, an appropriate goal for this project was to determine to be establishing an estimated range of slip-stream effects.

An extensive review of literature was performed to identify any previous work for the slip-stream, drafting or interference drag effects occurring about hypersonic bodies. The starting point was first to investigate the slip-stream effect about subsonic bodies.

Figure 4.5 depicts a schematic for the slipstream effect for two subsonic bodies. As the spacing between the bodies decrease, the pressure fields of the first and second body interfere with

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^m Note: if the worst case scenario exists then there is no purpose to establish a material package stream since no drag reduction occurs

Figure 4.5: Slip-Stream or Interference Drag About Two Bodies

each other. The second body's positive pressure field interferes with the first body's negative pressure field; the resulting interference pressure field could be positive or negative. The drag on the bodies is generally reduced because the magnitude of the pressure forces associated with the intermediate pressure region is reduced. Generally, the second body achieves a greater benefit from the slipstream effect; however, under certain conditions the opposite is true [40].

Figure 4.6 depicts the results for the slipstream of two circular disks in an axial laminar air flow field. In Figure 4.6, the leading disk, #1, has a constant drag co-efficient and therefore experiences a constant drag force. The drag co-efficient of the rear disk, #2 varies considerably with the separation distance between the disks. For small separation distances ($x < 2.2 D$) the drag coefficient is negative; this indicates that second disk would experience a force pushing it towards the leading disk since the pressure field in front of the second disk is negative. At separation distances greater than 2.2 Diameters ($x > 2.2 D$), the drag coefficient increases with increasing separation distance. The pressure field ahead of the second disk becomes increasing positive as the separation distance increases; therefore, the pressure force on the second disk also increases with increasing separation distance. As the separation increases past 8 Diameters

Figure 4.6: Interference Drag About Two Circular Disks [40]

($x > 8 D$), the drag coefficient is nearly equal to the leading disk. Therefore, at separation distances greater than 8 diameters, the slipstream effect for circular disks is minimal [40].

Figure 4.7 depicts interference drag about two streamlined bodies. The overall drag coefficient for these bodies is lower than the aforementioned circular disks because the bodies are slender and streamlined. The slipstream effect for streamlined bodies is different than the previous results. For separation distances smaller than 0.8 times the length ($x < 0.8L$), the drag coefficient on the leading body is negative and decreases to zero as the separation distance approaches $x = 0.8 L$. As the separation distance increases beyond $0.8 L$, the drag coefficient increases. An opposite effect occurs on the following second body, at small separation distances the drag coefficient is high and decreases as the separation distance increases. At large separation distances ($x > 4L$), the drag coefficient for both bodies is nearly equal. However, the drag coefficient for the second body is always greater than the drag coefficient for the leading body. Therefore, the following body is always pushing the leading body forward since, the positive pressure region created in front of the second body is larger than the negative pressure field behind the leading body [40].

Figure 4.7: Interference Drag About Two Streamlined Bodies [40]

These experimental results suggest that for the material package stream to be effective, the material packages should be blunt bodies with large negative base pressures. If a picture was taken of a steady state material package stream, it would look like a chain or cable of material packages extending out of the atmosphere, depicted as an exaggerated view in Figure 4.8. The leading material would have no positive pressure acting on the nose, since it is out of the atmosphere; therefore, no drag acts on this material package. Unlike the situation depicted in Figure 4.6, the leading material package would have a drag-coefficient of zero. It is assumed, the only drag that would be acting on the material package stream would be skin drag. It has been hypothesized that

a steady state material package stream would be similar to a very long cable traveling through a fluid. The movement of the material packages, like the movement of the cable displaces the fluid and the only force opposing the motion is the shear force along the outside of the material packages or cable.

Interference drag for hypersonic bodies is a field of study that is not well researched at the present. One numerical study done by V. Riabov et al. was identified [41]. Raibov investigated the drag effects between two cylinders placed one after another traveling at speeds up to Mach 5. The investigation was limited to separation distances of 1.5 diameters. Table 4.6 contains the drag coefficients determined by Raibov and the percent drag reduction. Raibov determined the drag reduction on the trailing cylinder for separation distances between 0.25 and 1.5 diameters ranged from about 81% to 75.6%.

The Raibov data was extrapolated by curve fitting in Excel to the 10, 100 and 1000 diameters used for the evaluation of the launch trajectory. The extrapolated data for 10, 100 and 1000 diameters was determined to

have drag reduction on the second cylinder of about 55%, 52% and 43% respectively. The

Figure 4.8: Material Package Stream As A Chain Through the Atmosphere

Table 4.6: Hypersonic Slip-stream Drag Co-efficient [41]

Spacing (Diameters)	Drag Co-efficient			Source
	$C_{x \text{ cylinder1}}$	$C_{x \text{ cylinder2}}$	% Reduction	
0.25	2.64	0.50	81.06	Riabov
0.5	2.66	0.55	79.32	Riabov
1	2.73	0.59	78.39	Riabov
1.5	2.87	0.70	75.61	Riabov
5	n/a	n/a	61.81	Extrapolated
10	n/a	n/a	55.83	Extrapolated
50	n/a	n/a	55.11	Extrapolated
100	n/a	n/a	51.62	Extrapolated
1000	n/a	n/a	42.83	Extrapolated

extrapolated data suggests a significant slip-stream reduction occurs at extended separation distances but the accuracy of the extrapolation data is questionable since, it is based on a very small range of experimental data. Operating the material package stream at 1000 diameter separation distances and obtaining a 43% reduction in drag would be beneficial since the required launch frequency and supply power would be significantly reduced.

4.3 Launch Window Analysis

With the analysis and selection of the launch trajectory completed, the next steps in the analysis, is to evaluate the launch window effects and the location of the CatSat. Figure 4.9

Figure 4.9: Overview of Launch Window and 'Catcher' Satellite Orbit Analysis

presents an overview of the analysis to be discussed in Sections 4.3 through to Section 4.5.

To effectively utilize the elliptical, material package orbit, described in Section 4.2.3, it is proposed that the elliptical orbit be used as a 'storage' orbit for material packages. The launcher will send up a series of material packages during a suitable launch period or launch window. The CatSat, located in an orbit that intersects the storage orbit, collects the material packages from the storage orbit. When a suitable launch window exists again, the launcher sends a stream of material packages to restock the storage orbit.

Due to Earth's rotation, there is only one time instant each day that the 10 000 km altitude apogee of selected launch trajectory overlaps the apogee of the equatorial elliptical storage orbit. Figure 4.10 depicts the scenario viewed along Earth's polar axis. The location corresponding to the minimum energy requirement for placing material packages in an elliptical orbit occurs at the intersection of the launch trajectory and elliptical orbit apogee. It is possible to place the material package into an elliptical orbit by firing the rocket motor at intersection locations other than the apogee intersection but this will require a larger rocket motor burn than calculated in Section 2.1.3. (These other intersection locations can be imagined by rotating the launch trajectory in Figure 4.9 about the centre of Earth.) The following Section will evaluate how the duration of the launch window affects the size of the rocket motor required for placing the material packages into the same elliptical orbit.

Figure 4.10: Material Package Apogee Intersection

4.3.1 Storage Orbit and Launch Trajectory Parameters

To evaluate the possible launch window, the size of the elliptical storage orbit and the launch trajectory ellipse, specified in Section 4.1.3, must be measured. The three parameters are required to specify the size of an ellipse, the semi-major axis (a), the semi-minor axis (b) and the linear eccentricity (c). These three parameters for the launch ellipse and the elliptical storage orbit were measured directly from the position data from the Earth_Sim_RevD.xls simulation spreadsheet and calculated as detailed below.

Figure 4.11: Elliptical Orbit Parameters

For an apogee of 10 000 km and perigee of 200 km above Earth's surface, the elliptical storage orbit parameters a , b and c , are depicted in Figure 4.11 and were determined to be as follows:

Semi-Major Axis - $a = 11\,478.14$ km

Where the semi-major axis measured directly from the simulation spreadsheet positional data.

Linear eccentricity, $c = 4900$ km

Where $c = a - R_{\text{Earth}} - 200$ km

Semi-Minor Axis, $b = 10\,379.67$ km

Where $b = (a^2 - 4c^2)^{1/2}$

The ballistic launch trajectory is an ellipse that intersects Earth and it is depicted in Figure 4.12. The elliptical parameters for the ballistic launch trajectory from Earth surface to a 10 000 km altitude were determined to be as follows:

Figure 4.12: Ballistic Trajectory Parameters

Semi-Major Axis, $a = 9560.65$ km

Where the semi-major axis measured directly from the simulation spreadsheet positional data.

Semi-Minor Axis, $b = 6699.8$ km

Where the semi-major axis measured directly from the simulation spreadsheet positional data.

Linear eccentricity, $c = 6820.57$ km

Where $c = a e$ where e (eccentricity) $= (1 - b^2/a^2)^{1/2}$

4.3.2 Evaluating an 80 Minute Duration Launch Window

The first step to evaluating the window was to determine how many degrees the Earth rotates during an 80 minute period. The rotation angle was determined as follows.

Degree Earth Rotation = Launch Window Duration $\times \omega_{\text{Earth}}$

$$= 80 \text{ minutes} \times 4.1667 \times 10^{-3} \text{ deg/s} = 4800 \text{ s} \times 4.1667 \times 10^{-3} \text{ deg/s}$$

$$= 20 \text{ degrees}$$

The next step was to create an AutoCAD model similar to Figure 4.13 with the launch trajectory apogee intersecting the apogee of the material package orbit. One copy of the launch trajectory was rotated 10 degrees clockwise about the centre of Earth to simulate the position of the beginning of the 80 minute launch of the material package stream. Another copy of the launch trajectory was rotated 10 degrees counter-clockwise about the Earth centre to simulate the position of the end of the 80 minute launch of the material package stream. Figure 4.13 depicts the 80 minute launch window model.

The intersection points P1, P2, and P3 depicted in Figure 4.13, represent the locations where the rocket motor must fire to place the material package in the same elliptical orbit. To calculate the mass of the required rocket motor, the velocity difference between the material package on the launch trajectory and the elliptical orbit at the intersection points is required. The velocity of the material on the launch trajectory was determined by measuring the intersection point in AutoCAD and using this position data to lookup the velocity in the simulation data. The velocity of the material package in the elliptical orbit was looked up in a position verses velocity

table created in Material_Package_Orbit.xls spreadsheet, which was programmed with analytical equations for position and velocity of elliptical satellites. See Appendix C for details.

Figure 4.13: Launch Trajectories for an 80 Minute Launch Window

Table 4.7 summarizes the velocities and the calculated velocity differences and the required rocket motor masses. The required motor mass was calculated using the equation in Appendix B - Part U assuming high performance solid rocket motor performance parameters.

Table 4.7: Material Package Velocities and Required Rocket Motor for 80 Min. Launch Window

	P1 (T=0 minutes)			P2 (T=40 minutes)			P3 (T=80 minutes)		
	v_x (m/s)	v_y (m/s)	v_R (m/s)	v_x (m/s)	v_y (m/s)	v_R (m/s)	v_x (m/s)	v_y (m/s)	v_R (m/s)
Ballistic Trajectory (before rocket burn)	-2643.68	824.64	2769.31	-2681.07	2.63	2681.07	-2653.92	696.59	2743.82
	Results obtained from Earth Sim-RevC.xls								
Elliptical Orbit (required velocity)	-3244.02	2484.09	4085.88	-3735.89	0	3735.89	-3696.52	-715.33	3765.1
	Results obtained from Material Package Orbit.xls								
Delta Velocity (required from rocket motor)	-600.34	1659.45	1764.705	-1054.82	2.63	1054.82	-1042.6	-1411.92	1755.145
	(Subtract Ballistic Velocity Components from Elliptical Orbit Velocity Components)								
Estimated Rocket Motor Mass (kg)	0.6672 kg			0.3988 kg			0.6636 kg		
Percent Change	26.84%			0%			26.48%		

The result is the rocket motor mass required for the material packages at the beginning and end of the 80 minute launch window require a mass 27% greater than the material package launched at the centre of the window. The other material packages between the middle and the beginning and end of the launch window would require some portion of the 27% increased mass depending on there position.

While it is possible to incorporate a varying rocket motor mass for each material package during the 80 minute window, it complicates the design and manufacturing of the material package. The intention was to mass produce the material packages; therefore, it would be easier to mass produce the material packages if the specifications for each package were the same. The next Section investigates shortening the launch window to 10 minutes, in an attempt to minimize the rocket motor mass difference thus making the design of each material package more 'mass produceable'.

4.3.3 Evaluating a 10 Minute Duration Launch Window

The evaluation of the 10 minute launch window was done in a similar manner to the 80 minute window, except using the following amount of Earth rotation.

$$\begin{aligned}\text{Degree Earth Rotation} &= \text{Launch Window Duration} \times \omega_{\text{Earth}} \\ &= 10 \text{ minutes} \times 4.1667 \times 10^{-3} \text{ deg/s} = 600 \text{ s} \times 4.1667 \times 10^{-3} \text{ deg/s} = 2.5 \text{ degrees}\end{aligned}$$

Figure 4.14 depicts the AutoCAD model created to determine the intersection locations. These intersection points are much closer together than for the 80 minute launch window. Similarly, the intersection positions where used to look up the corresponding velocities in the simulation data and the table created with Material_Package_Orbit.xls. Table 4.8 summarizes the velocities, the calculated velocity differences and the required rocket motor masses.

The resulting increase for the rocket motor mass is very small, on the order of less than 2%. Since the masses are nearly the same, it probably is possible to mass produce a material package with a rocket motor mass of about 1% greater than the middle material package (P2 T=5minutes) requirement. The difference in the orbit caused by a 1% difference in the required rocket motor mass could be compensated by the actions of the CatSat. The evaluation of the

OpenMDS will proceed assuming a 10 minute launch window with a constant orbital insertion motor mass of 0.3988 kg.

Figure 4.14: Launch Trajectories for a 10 Minute Launch Window

Table 4.8: Material Package Velocities and Required Rocket Motor for 10 Min. Launch Window

	P1 (T=0 minutes)			P2 (T=5 minutes)			P3 (T=10 minutes)		
	v_x (m/s)	v_y (m/s)	v_R (m/s)	v_x (m/s)	v_y (m/s)	v_R (m/s)	v_x (m/s)	v_y (m/s)	v_R (m/s)
Ballistic Trajectory (before rocket burn)	-2680.81	63.6	2681.56	-2681.07	2.63	2681.07	-2677.71	242.13	2688.63
	Results obtained from Earth Sim-RevC.xls								
Elliptical Orbit (required velocity)	-3735.64	56.33	3736.07	3735.89	0	3735.89	-3735.64	-56.33	3736.07
	Results obtained from Material Package Orbit.xls								
Delta Velocity (required from rocket motor)	-1054.83	-7.27	1054.86	-1054.82	2.63	1054.82	-1057.93	-298.46	1099.22
	(Subtract Ballistic Velocity Components from Elliptical Orbit Velocity Components)								
Estimated Rocket Motor Mass (kg)	0.3988 kg			0.3988 kg			0.4156 kg		
Percent Change	0.00%			0%			1.68%		

4.4 Determining the 'Catcher' Satellite Orbit

This Section is concerned with selecting an appropriate orbit for the CatSat that would permit the satellite to collect material packages from the elliptical material package storage orbit. Ideally, for capture to occur, the CatSat should have position and velocity values similar to the material package.

The first approach to determining the orbit of the CatSat was to create an Excel spreadsheet,

Satellite_Trajectory_Orbit.xls

which used the position dataⁿ, from the launch and orbit simulation data, to graphically depict and animate the location of the material packages and the CatSat. Figure 4.15 depicts the graphic generated by the Satellite_Trajectory_Orbit.xls spreadsheet. The user presses F9 key, to animate the motion of the material packages and CatSat. When F9 is pressed, the spreadsheet looks up the new position for the material packages and satellite in the table of simulation data.

The Satellite_Trajectory_Orbit.xls spreadsheet proved to be too cumbersome to use to evaluate the timing of the material capture event. A search was performed to determine if a freely available software tool exists for analyzing satellite trajectories. A commercial software product,

Figure 4.15: Snapshot of Satellite_Trajectory_Orbit.xls Spreadsheet Output

ⁿ The material package coordinates were obtained from Earth_Sim_RevD.xls simulation data and the coordinates for the 'catcher' satellite were generated with Material_Package_Orbit.xls spreadsheet.

AGI Satellite Toolkit (STK)^o, was discovered to be capable of simulating and visualizing the launch and orbit of satellites. The STK's basic satellite calculation engine and 2D visualization viewer is free on request from AGI; therefore, this software product is compatible with the open engineering approach where all software tools used should be freely accessible to all. The STK program also has a 3D visualization viewer; however, the 3D viewer is available for a 7-day free trial and afterwards must be purchased from AGI.

The STK software was used to evaluate the six possible scenarios where the 'catcher' satellite could be located to collect material packages from the elliptical orbit. The launch trajectory from the Earth_Sim_Rev.xls spreadsheet was imported into STK and the STK program was used to generate the material package and satellite orbits. The material package orbits created assume that the same orbital insertion rocket motor fires at apogee for all material packages launched during the 10 minute launch window (see Section 4.2.3). The result is a slight 'smearing' of the material package orbit due to the 2.5 degrees of Earth rotation during the 10 minute launch period. Figure 4.16 depicts the launch trajectories imported into STK and the resulting orbital smear for firing the rocket motor at launch trajectory apogee. The following six Sections present the six catcher satellite scenarios and the results obtained with STK. The author utilized the 7-day trial for the 3D viewer to create the graphics in the following Sections.

Figure 4.16: Snapshot of STK Visualization of Material Package Launch and Orbit

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^o AGI Satellite Toolkit Website - <http://www.stk.com/products/desktopApp/stkFamily/modules/core/stk/>

4.4.1 Scenario #1 - Catcher Satellite in 10 000 km Circular Orbit

Figure 4.17 depicts the overview of scenario #1. The 'catcher' satellite is situated in a 10 000 km (altitude) circular Earth orbit and intersects and collects the material packages at the apogee of the material package orbit.

Figure 4.17: Overview of 'Catcher' Satellite in 10 000 km Circular Orbit

The visualizations generated for this scenario, indicated that the CatSat travels faster than the material packages, so the CatSat approaches and collects the material stream from the last to the first material package launched. The position and velocity data for the satellite and the material packages^P at the closest approach during the apogee rendezvous were recorded in Table 4.9 and the associated screen shots at the capture events are depicted beside the table.

^P Only three material packages are considered in the analysis, first, middle and last material package, these represent the extreme points of the material stream; the data for the rest of the material stream will lie between the values for these extreme points.

Table 4.9: Position and Velocity for Material Package Capture Scenario #1

Capture Time of Last Package in Stream: 20:24:00			
	Last Package	Catcher Satellite	Delta
Position (km)	Launched		
X	11245.879	11265.226	-19.347
Y	11851.65	11888.567	-36.917
Resultant			41.679
Velocity (km/s)			
X	-1.6962	-2.714	1.0178
Y	1.9206	2.5717	-0.6511
Resultant			1.2082

Diagram illustrating the capture of the last material package by the CatSat. The CatSat is in a circular orbit, and the material package is in a lower, elliptical orbit. The diagram shows the relative positions and velocities at the capture time.

Left Material Package (1st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (2nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (3rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (4th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (5th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (6th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (7th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (8th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (9th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (10th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (11th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (12th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (13th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (14th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (15th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (16th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (17th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (18th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (19th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (20th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (21st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (22nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (23rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (24th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (25th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (26th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (27th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (28th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (29th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (30th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (31st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (32nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (33rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (34th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (35th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (36th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (37th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (38th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (39th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (40th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (41st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (42nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (43rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (44th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (45th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (46th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (47th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (48th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (49th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (50th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (51st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (52nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (53rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (54th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (55th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (56th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (57th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (58th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (59th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (60th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (61st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (62nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (63rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (64th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (65th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (66th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (67th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (68th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (69th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (70th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (71st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (72nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (73rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (74th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (75th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (76th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (77th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (78th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (79th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (80th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (81st Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (82nd Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (83rd Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (84th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (85th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (86th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (87th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (88th Package) Velocity:
 Time: 20:24:00
 X: 11265.226
 Y: 11888.567
 Z: 11888.567
 Vx: -2.714
 Vy: 2.5717
 Vz: 0.0000
 Vmag: 3.0398

Left Material Package (89th Package) Velocity:
 Time: 20:24:00
 X: 11245.879
 Y: 11851.65
 Z: 11851.65
 Vx: -1.6962
 Vy: 1.9206
 Vz: 0.0000
 Vmag: 1.2082

Left Material Package (90th Package) Velocity:
 Time: 20:24:00
 X

The position of the material packages and satellite are not as close as they appear in the Figure, the difference in distance is quite large, ranging from about 11.5 to 41.7 km. The velocity difference between the material packages and satellite is relatively large at about 1.2 km/s for the three material packages considered. The capture time is coincidentally the same duration as the launch window which is 10 minutes.

Locating the CatSat in a 10 000 km orbit seems plausible but the difference in position and velocity at capture between the satellite and material packages is a significant problem. The next Section will look at an alternative satellite location.

4.4.2 Scenario #2 - Catcher Satellite in 200 km Circular Orbit

Figure 4.18 depicts the overview of scenario #2. The 'catcher' satellite is situated in a 200 km (altitude) circular Earth orbit and intersects and collects the material packages at the perigee of the material package orbit.

Figure 4.18: Overview of 'Catcher' Satellite in 200 km Circular Orbit

The visualizations generated for this scenario, indicated that the material packages travel faster than the catcher satellite, so the material packages approach the satellite from behind; therefore, the satellite collects the material packages in the order of launch first to last. The position and velocity data for the satellite and the material packages, at the closest approach during the perigee rendezvous, were recorded in Table 4.10 and the associated screen shots at the capture events are depicted beside the table.

Like scenario 1, the distance between the satellite and the material packages at capture are large, ranging from about 1.5 to 46.3 km. The velocity difference is higher than scenario #1, at about 1.5 km/s over the complete capture event. The duration of the capture event is shorter than

Table 4.10: Position and Velocity for Material Package Capture Scenario #2

Capture Time of First Package in Stream: 8:09:10			
	First Package	Catcher Satellite	Delta
Position (km)	Launched		
X	4757.687	4740.671	17.016
Y	4603.514	4560.473	43.041
Resultant			46.283
Velocity (km/s)			
X	-6.4992	-5.0641	-1.4351
Y	5.8961	5.2642	0.6319
Resultant			1.5681

Capture Time of Middle Package in Stream: 8:10:40			
	Middle Package	Catcher Satellite	Delta
Position (km)	Launched		
X	4261.229	4262.006	-0.777
Y	5019.684	5010.707	8.977
Resultant			9.011
Velocity (km/s)			
X	-6.8602	-5.564	-1.2962
Y	5.5311	4.7327	0.7984
Resultant			1.5224

Capture Time of Last Package in Stream: 8:12:10			
	Last Package	Catcher Satellite	Delta
Position (km)	Launched		
X	3743.056	3740.809	2.247
Y	5412.017	5410.937	1.08
Resultant			2.493
Velocity (km/s)			
X	-7.1762	-6.0085	-1.1677
Y	5.1219	4.1539	0.968
Resultant			1.5168

scenario #1, at about 3 minutes; therefore, the satellite is required to collect material packages at a frequency greater than the launch frequency. A high capture frequency would complicate the design of the CatSat. Capturing the material package stream at perigee does not represent a better satellite location than scenario #1. The next Section will evaluate another alternative satellite location for material package capture.

4.4.3 Scenario #3 - Catcher Satellite in a Large Elliptical Orbit

In an attempt to reduce the position and velocity differences, scenario 3 attempts placing the satellite in an elliptical orbit that has an orbital period of exactly double the period of the material package storage orbit. Since the material packages have an orbital period of half the 'catcher' satellites, the material packages rendezvous with the 'catcher' satellite once every two revolutions of the orbit. Figure 4.19 depicts the overview of scenario #3. The orbital period of the material package orbit was determined with STK software to be 3.39 hrs. The parameters of an elliptical 'catcher' satellite orbit were adjusted until the period was 6.799 hrs and the trajectory was nearly tangent with material package orbit.

Figure 4.19: Overview of 'Catcher' Satellite in Large Elliptical Orbit

The visualizations generated for this scenario, indicated that the catcher satellite travels faster than the material packages, so the catcher satellite approaches and collects the material stream from the last to the first material package launched. The position and velocity data for the satellite and the material packages, at the closest approach during the apogee rendezvous, were recorded in Table 4.11 and the associated screen shots at the capture events are depicted beside the table.

Table 4.11: Position and Velocity for Material Package Capture Scenario #3

Capture Time of Last Package in Stream: 12:39:30			
	Last Package	Catcher Satellite	Delta
Position (km)	Launched		
X	-11966.736	-11945.467	-21.269
Y	10287.697	10299.395	-11.698
Resultant			24.274
Velocity (km/s)			
X	-2.4404	-3.2098	0.7694
Y	-1.4959	-2.7316	1.2357
Resultant			1.4557

Capture Time of Middle Package in Stream: 12:43:30			
	Middle Package	Catcher Satellite	Delta
Position (km)	Launched		
X	-12738.947	-12693.994	-44.953
Y	9635.257	9629.394	5.863
Resultant			45.334
Velocity (km/s)			
X	-2.182	-3.0273	0.8453
Y	-1.6908	-2.8491	1.1593
Resultant			1.4339

Capture Time of First Package in Stream: 12:47:30			
	First Package	Catcher Satellite	Delta
Position (km)	Launched		
X	-13433.73	-13398.321	-35.409
Y	8922.345	8933.106	-10.761
Resultant			37.008
Velocity (km/s)			
X	-1.9194	-2.8417	0.9223
Y	-1.8667	-2.9507	1.084
Resultant			1.4233

Like scenario 1 and 2, the distance between the satellite and the material packages at capture are large, ranging from about 24.3 to 45.3 km. The velocity difference is higher than scenario #1 but slightly lower than scenario #2, at about 1.4 km/s over the complete capture event. The duration of the capture event is shorter than scenario #1 but longer than scenario #2, at about 8 minutes. The satellite is required to collect material packages at a frequency greater than the launch frequency; therefore, this would complicate the design of the CatSat.

Capturing the material package stream with a CatSat located in an elliptical orbit tangent to the material package orbit does not represent a better satellite location than scenario #1. The next Section will evaluate another alternative satellite location for material package capture.

4.4.4 Scenario #4 - Catcher Satellite in Middle Material Package Orbit (capture at perigee)

Given that the position and velocity differences for the three previous scenarios have been large, a possible means of reducing the differences is to locate the catcher satellite in the same orbit as the middle material package. Therefore, the satellite could easily capture the middle material package, since it shares the same position and velocity. The remaining packages in the orbiting material package stream would fire a second rocket motor at perigee to propel the material package to the catcher satellite. The second rocket motor burn is assumed to propel the material package in a straight line with constant acceleration to the catcher satellite with the duration of the burn and transit to the catcher to occur over a 10 second period of time. Perigee was selected because the distances between the material packages are at a minimum at perigee; the material package stream squeezes together at this location. Figure 4.20 depicts an overview of scenario #4 at perigee of the material package orbit.

Figure 4.20: Overview of 'Catcher' Satellite in Scenario #4 Location

Table 4.12 contains the positions, velocities and the velocity differences between the first and last material packages with respect to the catcher satellite (middle material package) at 10 seconds past perigee.

Table 4.12: Position and Velocity for Material Package Capture Scenario #4

Distance between:	Δx (km)	Δy (km)	Resultant (km)
First Package at $t=x$ and Middle Package at $t=x+10$ s	2726.7	767.08	2832.54
Last Package at $t=x$ and Middle Package at $t=x+10$ s	2238.49	1369.24	2624.05
Velocity of:	v_x (km/s)	v_y (km/s)	Resultant (km/s)
First Package at $t=x$	8.518663	-1.135412	8.593997
Last Package at $t=x$	6.724228	-5.447245	8.653769
Middle Package at $t=x+10$ s	8.068438	-3.558507	8.818314
Velocity Difference between:	Δv_x (km/s)	Δv_y (km/s)	Δ Resultant (km/s)
First Package at $t=x$ and Middle Package at $t=x+10$ s	0.450225	2.423095	2.464567
Last Package at $t=x$ and Middle Package at $t=x+10$ s	-1.34421	-1.888738	2.318239

The rocket motor mass required to propel the last material package to the CatSat was determined to be 163.1 kg (see Appendix D for sample calculation). Obviously, this mass is much greater than the material package payload mass of 0.6 kg (mass after orbital insertion rocket burn); therefore, scenario #4 is not feasible.

4.4.5 Scenario #5 - Catcher Satellite in Middle Material Package Orbit (capture at apogee)

Scenario #5 is the same as scenario #4 except the second rocket motor burn and capture occurs at apogee. Figure 4.21 depicts an overview of scenario #5.

Note: $t = x$ is the time period when the middle material package and catcher satellite are at apogee.

Figure 4.21: Overview of 'Catcher' Satellite in Scenario #5 Location

Table 4.13 contains the positions, velocities and the velocity differences between the first and last material packages with respect to the CatSat (in the middle material package) at 10 seconds past apogee.

Table 4.13: Position and Velocity for Material Package Capture Scenario #5

Distance between:	Δx (km)	Δy (km)	Resultant (km)
First Package at $t=x$ and Middle Package at $t=x+10$ s	-519.562897	-525.616592	739.07
Last Package at $t=x$ and Middle Package at $t=x+10$ s	-478.048351	-629.703572	790.61
Velocity of:	v_x (km/s)	v_y (km/s)	Resultant (km/s)
First Package at $t=x$	-1.647937	1.934765	2.541459
Last Package at $t=x$	-1.414834	2.127243	2.554783
Middle Package at $t=x+10$ s	-1.656796	1.928123	2.542171
Velocity Difference between:	Δv_x (km/s)	Δv_y (km/s)	Δ Resultant (km/s)
First Package at $t=x$ and Middle Package at $t=x+10$ s	0.008859	0.006642	0.011072
Last Package at $t=x$ and Middle Package at $t=x+10$ s	0.241962	0.19912	0.313360

The rocket motor mass required to propel the last material package to the CatSat was determined to be 50.6 kg (see Appendix D for sample calculation). Obviously, this mass is much greater than the material package payload mass of 0.6 kg; therefore, scenario #5 is not feasible.

4.4.6 Scenario #6 - Catcher Satellite in Middle Material Package Orbit (manoeuvrable satellite)

The final scenario considers using a second rocket motor burn either at apogee or perigee to nudge all material packages into the same orbit, the middle material package orbit. Forcing all the material packages in the middle material package orbit removes the orbit smearing caused by the rotation of Earth during the 10 minute launch window. The second rocket motor burn is intended to occur at the locations where each material package intersects the middle material package orbit. Figure 4.22 depicts an overview of Scenario #6. The duration of the rocket motor burn is assumed to 1 second.

Table 4.14 contains the velocity differences between the first and last material package with respect to the middle material package at the intersection points. The table includes the

required acceleration, impulse and rocket motor mass to move the first and last material packages into the middle orbit (see Appendix E for sample calculation).

The required rocket motor masses are smallest for orbit changes at perigee; the first and last material package in the stream requires about a 5.5 gram and 1.6 gram rocket motor, respectively. At apogee, the required rocket

Figure 4.22: Overview 'Catcher' Satellite in Scenario #6 Location

motor mass for orbital change for the first and last material package in the stream was determined to be about 8.3 grams and 2.3 grams, respectively. Of course these required rocket motor masses

Table 4.14: Position and Velocity for Material Package Capture Scenario #6

Intersection of Middle Material Package and:	At Perigee Intersection		At Apogee Intersection	
	First Material Package	Last Material Package	First Material Package	Last Material Package
Velocity Difference:				
Δv_x	7.527 m/s	2.282 m/s	19.806 m/s	5.690 m/s
Δv_y	16.457 m/s	5.125 m/s	16.848 m/s	4.717 m/s
Δv_z	-0.1 m/s	0.077 m/s	0.022 m/s	-0.036 m/s
Required Acceleration* :				
a_x	7.527 m/s ²	2.282 m/s ²	19.806 m/s ²	5.690 m/s ²
a_y	16.457 m/s ²	5.125 m/s ²	16.848 m/s ²	2.757 m/s ²
Required Impulse:				
I_x	4.525 N s	1.334 N s	11.907 N s	3.325 N s
I_y	9.894 N s	2.995 N s	10.129 N s	2.757 N s
Required Rocket Motor Mass:	5.452 g	1.637 g	8.331 g	2.299 g

* (assume 1 sec rocket burn)

represent the maximum required rocket mass for the material package stream launched during the ten minute window. Material package orbits closer to the middle material package orbit require small rocket motor masses for orbital change to the middle material package orbit.

The results for scenario #6 are feasible; however, since the material packages and CatSat share the same orbit, the CatSat requires a means to move along the orbit to collect the material packages. The position of the CatSat, relative to the orbiting material package stream, can be manipulated by increasing or decreasing the apogee of the 'catcher' satellite orbit. Lowering the apogee will decrease the orbital period of the satellite; therefore, the satellite's position will advance relative to the material package stream. Conversely, increasing the apogee will increase the orbital period; therefore, the satellite's position will retard relative to the material package stream. The capture of material packages occurs when the catcher satellite transits between these upper and lower orbits. Figure 4.23 depicts this process. The satellite would oscillate between the upper and lower orbit until all the material packages are captured.

Figure 4.23: 'Catcher' Satellite Maneuvers for Capture

4.4.7 Selected 'Catcher' Satellite Orbit

Table 4.15 presents a summary of the results for the six scenarios.

Table 4.15: Summary of Results for 'Catcher' Satellite Scenarios

	Scenario #1	Scenario #2	Scenario #3	Scenario #4	Scenario #5	Scenario #6
Separation Distance from satellite and:						
First Material Package	23.006 km	46.283 km	37.008 km	2832.54 km	739.07 km	0 km
Middle Material Package	11.528 km	9.011 km	45.334 km	0 km	0 km	0 km
Last Material Package	41.769 km	2.493 km	24.274 km	2624.05 km	790.61 km	0 km
Velocity Difference from satellite and:						
First Material Package	1.2026 km/s	1.5681 km/s	1.4233 km/s	2.4646 km/s	2.5415 km/s	0 km/s
Middle Material Package	1.1990 km/s	1.5224 km/s	1.4339 km/s	0 km/s	0 km/s	0 km/s
Last Material Package	1.2082 km/s	1.5168 km/s	1.4557 km/s	2.3182 km/s	2.5422 km/s	0 km/s
2nd Rocket Motor Mass	N/A	N/A	N/A	≈ 163.1 kg	≈ 50.6 kg	≈ 0.005 kg

From inspection of table 4.15, Scenario #6 represents the best location for the CatSat. This Scenario eliminates the velocity difference because CatSat and material packages share the same orbit, the middle material package orbit. However, a small rocket motor mass is required to adjust the orbit the material packages into the middle material package orbit. From the results of scenario 6 (Section 4.3.6), the rocket motor mass required for orbital change of the first and last material package at apogee represents 2% and 0.6% of the mass of the orbital insertion rocket motor, respectively. This result is similar to the 1.68% increase in rocket motor mass determined in Section 4.2.3 for the last material package in the 10 minute launch window. Since the difference in rocket motor mass for perigee or apogee orbit changes is less than 3 grams, it would simplify the design of the material package if the orbit change was done at apogee with the orbital insertion rocket motor rather than incorporating a second rocket motor system. The manufacturing process for the orbital insertion rocket could incorporate a means of producing solid rocket motor grains with a slight variation in propellant mass to tailor the rocket motor thrust to insert all material packages into or at least very near the middle material package orbit.

As mentioned in 4.4.6, the CatSat for scenario 6 must manipulate its orbit to capture the material packages. A propulsion system capable of adjusting the orbit is discussed in the design of the CatSat in Section 4.8.

4.5 Summary of the Selected Launch Trajectory and 'Catcher' Satellite Orbit

Figure 4.24 depicts the launch and orbital trajectories selected for the design of the mass driver.

Figure 4.24: Launch and Orbital Trajectories for Selected the Mass Driver System

The selected launch trajectory parameters are as follows:

Material Package:	1 kg
Launch Angle:	50 degrees (from vertical)
Launch Velocity:	10.92 km/s (0% Drag Reduction – no drag reduction) 8.82 km/s (100% Drag Reduction – perfect vacuum)
Apogee Altitude:	10 000 km
Launch Window:	10 minute duration (2.5 degrees of Earth rotation)

The material package and catcher satellite orbits share the same orbit parameters. These parameters are as follows:

Apogee Altitude:	10 000 km
Perigee Altitude:	200 km
Orbital Period:	3.39 hours

The operation of the mass driver system can be summarized as the following three steps (also depicted in Figure 4.24):

1. Launch a stream of material packages from a launch mechanism located on Earth's surface.
2. When the material packages reach the apogee of the launch trajectory, a rocket motor fires to insert the material package into the middle material package orbit.
3. The material packages are collected by the 'catcher' satellite located in the middle material package orbit.

The following Sections discuss the design of the three remaining sub-systems, the launcher mechanism, material package and catcher satellite.

4.6 Preliminary Design for the Launcher Mechanism

The preliminary OpenMDS, proposed in Section 3.6.1, utilized railgun technology to accelerate the material packages to high velocities. Railgun technology has been demonstrated to launch small mass to high velocities; however, there has been significant technical difficulties developing suitable energy storage devices and durable launch rails capable of allowing more than one launch from a railgun. Each mass launched from a railgun requires an expendable armature assembly that separates from the launch body after exiting the launcher. Designing armature assemblies capable of handling high currents loads for high velocity launches has been a significant challenge. Many of direct space launch concepts, mentioned in Section 2.3, recommend the use of railguns but they also state that there are significant technical problems to solve before a practical design is possible.

Proposing solutions to the technical problems associated with railguns is beyond the scope of this project. Therefore, the following Sections will not focus on the design of a railgun launcher system, rather the feasibility of two original launcher concepts will be investigated. Since one of the purposes for the overall design for the mass driver system is to attract potential volunteers to the project, it is thought that a novel concept will draw individuals to read about the project simply from curiosity. Proposing a different concept than the mainstream idea of a railgun may at the very least create discussions why railgun technology is the best solution and should be implemented in the design.

However, before the launcher concepts are considered, the first step in designing any launch mechanism, for the mass driver system, is to consider where it is to be located. The next Section details the proposed launch site.

4.6.1 Proposed Launch Site Location

If the orbital insertion rocket motor fails to ignite, the material package will return to Earth following the reentry trajectory depicted in Figure 4.24. Therefore, the launch site location should locate the impact site for the reentry trajectory in a remote, sparsely populated region. By varying the launch site location in the STK software it was determined that a launch location in central Africa would have an impact site in an unpopulated location in the Pacific Ocean. The selected launch location is depicted in Figure 4.25 below.

Launch Site Location

Country: Democratic Republic of the Congo

Latitude: 0 degrees

Longitude: 21.32 degrees

Altitude: 0 m

Figure 4.25: Proposed Launch Site

4.6.2 Requirements for Launch Mechanism

From the launch trajectory and orbital analysis of the previous Sections, the following requirements for the launch mechanism have been determined.

Launch Mass: 1 kg

Launch Angle: 50 degrees from vertical (40 degrees from horizontal)

Launch Velocity: 10.92 km/s (0% Drag Reduction – full drag forces acting)
8.82 km/s (100% Drag Reduction – no drag forces acting)

Launch Energy: 59.62 MJ (0% Drag Reduction – full drag forces acting)
38.90 MJ (100% Drag Reduction – no drag forces acting)

Launch Frequency: 16 850 Hz (10 Dia. Spacing - 0% Drag Reduction – full drag forces acting)
4 640 Hz (100 Dia. Spacing - 0% Drag Reduction – full drag forces acting)
560 Hz (1000 Dia. Spacing - 0% Drag Reduction – full drag forces acting)
13 620 Hz (10 Dia. Spacing - 100% Drag Reduction – no drag forces acting)
3 740 Hz (100 Dia. Spacing - 100% Drag Reduction – no drag forces)
453 Hz (1000 Dia. Spacing - 100% Drag Reduction –no drag forces)

Supply Power:

1004.7 GW	≈2010 Nuclear Reactors	(10 Dia.. Spacing -0% Drag Reduction)
276.7 GW	≈554 Nuclear Reactors	(100 Dia. Spacing - 0% Drag Reduction)
33.4 GW	≈67 Nuclear Reactors	(1000 Dia. Spacing - 0% Drag Reduction)
530.2 GW	≈1061 Nuclear Reactors	(10 Dia. Spacing - 100% Drag Red.)
145.6 GW	≈292 Nuclear Reactors	(100 Dia. Spacing - 100% Drag Red.)
17.6 GW	≈36 Nuclear Reactors	(1000 Dia. Spacing - 100% Drag Red.)

Launch Duration: 10 Minute Launch Window

4.6.3 Circular Launch Mechanism Concept

The first concept for the launcher mechanism is a launcher that is similar to the cyclotron particle accelerators, where a mass is electromagnetically accelerated in a circular path. The concept is equivalent to a coilgun launch tube that is bent to form a circle. Figure 4.26 depicts the circular electromagnetic launch tube.

Figure 4.26: Circular Electromagnetic Launch Tube

The material package travels in the circular guide track through the centre of the drive coils. The magnetic fields created by the drive coils provide both a propulsive force to accelerate the material package and a force to constrain the material package to a circular path. A section of the guide track contains a very fast acting entrance and exit door mechanism to allow the launch package to enter and the exit the circular launcher.

The circular launch mechanism has two potential benefits. Firstly, the physical size of the circular launch mechanism would be much smaller than a linear coilgun mechanism operating at

the same acceleration rate and the design may be simplified. To obtain higher velocities for a linear coilgun, the length of the launch tube must be increased; therefore, the number of drive coils and switching components also increase with length. In contrast, to achieve a higher velocity with a circular electromagnetic launcher, the number of cycles around the circle is increased. The material package could traverse the circular path a number of times until the desired velocity is reached without requiring additional hardware. Secondly, the switching⁹ requirement for the circular launcher could potentially be reduced for high material package velocities by simply reducing the number of drive coils operating per revolution. For example if there were 12 drive coils spaced out equally around the circular track, like the hours on a clock, at low package velocities all drive coils can be turned on and off sequentially because the time period between the coils is sufficiently large to permit detection and electronic signaling. At high launch package velocities however, the time duration between each subsequent coil is small. Therefore, to increase the time period the number of drive coils operated could be reduced, such that only drive coils at the 12, 3, 6 and 9 o'clock positions would be used. The obvious disadvantages of the circular launcher is providing the additional force to constrain material package to a circular path and the need of fast acting entrance and exit doors.

The first step in the evaluation of the circular launcher concept was to evaluate the magnitude of the restraining force. Two forces were considered for the preliminary analysis, the centripetal force and the magnetic force required to oppose the centripetal force. These forces were calculated as follows,

Centripetal Force, F_c (N) [32],

$$F_c = m_{\text{package}} v_{\text{Launch}}^2 / r_{\text{Circ\_Launcher}} \quad (\text{eq. 4.1})$$

where

m_{package} – mass of material package (kg)
 v_{Launch} – Velocity of material package (m/s)
 $r_{\text{Circ\_Launcher}}$ – Radius of launcher (m)

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⁹ Turning on and off the drive coils

Magnetic Force, F_{mag} (N) [42],

$$F_{\text{mag}} = 0.5 B^2 A_{\text{perp}} / \mu_0 \quad (\text{eq. 4.2})$$

where

B – Magnetic Field Intensity (tesla)

A_{perp} – Area of launch package perpendicular to magnetic field (m^2)

μ_0 – Magnetic permeability

The Circular_Launcher_Design.xls spreadsheet was created to evaluate the magnitude of the required magnetic force to counteract the centripetal force, for circular launcher designs with radii between 25 m and 1000 m.

The Circular_Launcher_Design.xls spreadsheet assumed,

1. The total mass of the launch package was 1 kg.
2. The diameter of the launch package was 0.0019 m.
3. The length of the launch package was 0.46 m.
4. The magnetic force acts of half the surface area which is 0.0137 m^2 .
5. Magnetic saturation of launch package (rotor) not considered.
6. Maximum magnetic field strength of a super-conduction magnet is 14.7 Tesla^f.
7. Maximum magnetic field strength of an electromagnet is 2 Tesla^g.

Figure 4.27 contains the centripetal force vs. launch velocity data calculated for circular launcher radii ranging from 25 m to 1000 m with the Circular_Launcher_Design.xls spreadsheet.

The curves in Figure 4.27 correspond to the various radii of the circular launcher considered. The lower and upper horizontal lines in Figure 4.27 represent the theoretical maximum magnetic force that the conventional electromagnetic and super-conducting magnetic drive coil technology is capable of producing, respectively. It is evident, that conventional electromagnetic technology is only suitable for use with circular launchers with radii greater than 1000 m and launch velocities below 3 km/s since the higher performance values all are located above the lower horizontal line. Super-conducting drive coil technology seems suitable for a wide range of radii of circular launchers and launch velocities, since much of the curves lie below the upper horizontal line in Figure 4.27. Super-conducting technology, maybe capable of counteracting the centripetal forces, the centripetal forces acting on the material packages are large. For example, for a launcher with a radius of 500 m, the centripetal force acting on a 1 kg material

package traveling at 11 km/s (roughly the upper velocity required to launch to a 10 000 km apogee), is 242 000 N or about 27.2 tonnes. The amount of energy consumed to power the superconducting magnets to simply counteract the centripetal force, would likely lead to a very inefficient launcher mechanism. Expending large amounts of energy to confine the material package to a circle during launch probably is not worth the marginal benefits of a smaller launcher and reduced switching requirements gained. The development of this concept will not be further considered.

Figure 4.27: Comparison of Centripetal and Magnetic Forces vs. the Launch Velocity for Circular Launchers with Radii from 25 to 1000 meters

^r From Berkeley Lab Super-conducting Magnet Group - <http://www.lbl.gov/Science-Articles/Archive/14-tesla-magnet.html>

4.6.4 Distributed Oxidizer Launch Rail Concept

S. Molder's Distributed Propellant Launch Rail (DPLR), introduced in Section 2.3.7, is a unique approach for direct space launcher which utilizes essentially an externally fueled rocket sled. The DPLR offers higher performance than conventional rocket sled operations, such as the Holloman High Speed Test Track (HHSTT) [43], because DPLR rocket sled does not have to accelerate the propellant mass. However, the distributed propellant launch rail is not suitable for use with the material package stream launcher because the launch rail has to be reloaded with solid propellant between launches. Therefore, the distributed propellant launch rail is not capable of launch frequencies specified in Section 4.6.2.

The operation of the DPLR concept, inspired the author to propose a variant of the concept, the distributed oxidizer launch rail (DOLR). Instead of the launch rail containing the solid propellant, the launch rail in the DOLR dispenses an oxidizer gas, such as oxygen (O_2) or nitrous oxide (NO_4) to the launch sled as it passes over the Section of the rail. A series of fast acting valves would be required to control the oxidizer flow. The launch vehicle operates in a similar manner, except the combustion chamber of the launch vehicle contains a solid propellant grain material such as plastic or rubber. Figure 4.28 depicts a schematic for the distributed-oxidizer launch rail concept.

Figure 4.28: Schematic for the Distributed Oxidizer Launch Rail

The DOLR rocket sled is used to accelerate the material package attached to the rocket sled, see Figure 4.29, and release the material package at the end of the launch rail. The rocket sled is constrained to the rail with a component known as a slipper. The slipper and the collector nose

are used to direct oxidizer gas from the rail into the combustion chamber of the rocket sled. To accelerate the rocket sled, the rocket motor for the DOLR concept is proposed to operate similar to a hybrid rocket motor, unlike the rocket motor for the DPLR which operates like a solid rocket motor.

Figure 4.29: Schematic for Hybrid Rocket Sled

A conventional hybrid rocket propulsion system uses a gaseous or liquid oxidizer and a solid rocket fuel grain. The operation cycle of a hybrid rocket system is conceptual simple. An ignition system initiates the combustion of the fuel grain in the combustion chamber. A throttle valve opens and allows oxidizer gas to enter the

Figure 4.30: Hybrid Rocket Schematic

combustion chamber to support further combustion of the fuel grain. The hot combustion gases travel through the fuel grain, creating further combustion and finally, the gases exit through the rocket nozzle [39]. Figure 4.30 is a schematic of a hybrid rocket system.

The performance of the DOLR will be lower than the DPLR because the DOLR must carry the fuel mass onboard the rocket sled; however, the DOLR should be capable of high launch frequencies. The launch rate would be limited to how quickly the launch sled could be loaded on the rail. The DOLR concept would be safer to operate, since the oxidizer and fuel propellants are separated and the propellants themselves could be stable and easily stored.

Figure 4.31 below depicts an overview of the proposed DOLR site, for the OpenMDS. As shown in Figure 4.31, the launch site is to be located on a mountain side where the launch rail is inclined up the side of the mountain at a 40 degree angle (from the horizontal). The launch rail for the site forms a loop to allow the rocket sleds to be reloaded and reused. Figure 4.31 outlines the five major events that occur during launch.

Figure 4.31: Overview of the Proposed Distributed Oxidizer Launch Rail (DOLR)

The first event in the launch sequence is depicted in Figure 4.32. The rocket sleds with material packages are transferred from the storage facility to the distributed oxidizer launch rail along loading rails. Multiple loading rails are used to increase the launch frequency of the system. For example, if the loading time of the rocket sled was 1 second, then if only one loading rail was used, the launch frequency would be limited to 1 rocket sled per second. However, given the same loading time of 1 second, a system with four loading rails could launch at a rate of 4 rocket sleds per second. Four loading rails are shown for illustrative purposes, the final design will most likely

require more given that the lowest required launch frequency is 453 Hz and a 1 second loading time for the rocket sled simply represents an estimated time.

Figure 4.32: Hybrid Rocket Sleds Loading on Rails

The second event in the launch sequence is the acceleration of the rocket sled down the launch rail. Figure 4.33 depicts a schematic. The oxidizer gas is stored in large tanks and a network of valves is used to deliver the gas to the specific location of the rocket sled on the rail. The potential benefit the DOLR system has over electromagnetic launchers is the simplicity of the launch energy storage. The DOLR system stores the launch energy in the form of oxidizer gas located in conventional chemical storage tanks and the solid fuel stored separately inside the rocket sleds in the storage facility. Whereas, present electromagnetic launch systems and other large electrical power systems require the development of a simple, low cost, efficient energy storage media^s.

The DOLR system is also well suited to be used for the 10 minute launch window. The supply power for the 10 minute launch window is generated from depleting the supply of stored propellants. During the period when the launcher is not operating, the propellants can be delivered to replenish the chemical stockpile. Whereas, for electromagnetic launch systems, the

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^s If this electrical storage technology existed, one of the first applications would likely be buffering the present electrical grid for peak and off-peak electrical demand.

supply power for a 10 minute launch must either come from the previously mentioned storage media, or dedicated power generators such as nuclear reactors, or a combination of storage and generators may be used. During the launcher off time, the storage media is recharged from the output of the generators; in a well balanced system these generators would be operating continuously at a 100% capacity. However, storing large amounts of electricity and then quickly delivering it to a load is a technically challenging problem, whereas storing and delivering chemicals is a common practice.

Figure 4.33: Launch of Hybrid Rocket Sleds

The third event in the launch sequence is the release of the material packages while traveling up the hillside at a 40 degree angle. Figure 4.34 depicts the release event. The material packages travel the launch trajectory specified in Section 4.5.

The forth event in the launch sequence is the deceleration of the spent rocket sled. The kinetic energy of the rocket sled could be converted into electrical energy that could be sold commercially as a byproduct or used on-site for processes like generating oxidizer gas. The fifth

event in the launch sequence is the return of the rocket sleds to the storage facility. Once at this facility, the rocket sleds are reloaded with fuel grains and material packages. The next Section will detail the preliminary evaluation of the DOLR and the results.

Figure 4.34: Release of Material Packages

4.6.4.1 Evaluation of DOLR and Results

To investigate the performance of the DOLR concept a spreadsheet, LauncherSimulator.xls, was programmed to simulate the forces acting on the DOLR rocket sled. The simulation assumed two forces acting on the rocket sled, aerodynamic drag and a constant thrust rocket motor. The aerodynamic drag was based on the diameter of the rocket nozzle outlet and is calculated in a similar manner as previously described in Section 3.4.2.3 and Appendix B, the material package launch spreadsheets. The constant rocket thrust was assumed to be 0.934 MN which is equivalent the Super Road Runner (SRR) solid rocket motor used on the rocket sleds at HHSTT [43]. The specific impulse (319.2 sec), rocket nozzle outlet (0.4 m) and rocket nozzle mass (104.4 kg) of the SRR motor were used in the simulation. A conservative mixture ratio of 3 parts oxidizer and 1 part fuel mass was assumed for the simulation [39]. Appendix F details the calculation method.

Table 4.16a contains the original results for the DOLR rocket sled carrying either a 1 kg or 10 kg material package and operating in the open atmosphere with sea-level air density or in an evacuated launch tube with no air density. To obtain the results in Table 4.16 the fuel mass of the rocket sled was increased until burnout occurred at a distance of 20 km down the track. The 20

km track represents an extension of the current HHSTT facility with a track length of 15.48 km [43].

The results in Table 4.16a indicate that for launch in the atmosphere, the DOLR is capable of launching 1 kg and 10 kg material packages at a maximum velocity of 9.65 and 9.55 km/s, respectively at the end of a 20 km track. If the DOLR rocket sleds operate in an evacuated tunnel, the maximum velocity achieved is 15.04 km/s and 14.57 km/s, for rocket sled launches of the 1 kg and 10 kg material packages, respectively. The fuel mass required by these four cases is similar ranging from 247 kg to 300 kg. The launch of a 10 kg material package required about 2% more fuel mass than the launch of a 1 kg material package.

Table 4.16a: Results for the Original Distributed Oxidizer Launch Rail Analysis

Distributed Oxidizer Rail Parameters	Sea-Level Atmosphere		Perfect Vacuum	
	1 kg Material Package	10 kg Material Package	1 kg Material Package	10 kg Material Package
Specific Impulse (sec)	319.2	319.2	319.2	319.2
Air Density (kg/m ³)	1.19 (sea-level)	1.19 (sea-level)	0	0
Assuming SRR motor Performance Values:				
Thrust (N)	9.34E+05	9.34E+05	9.34E+05	9.34E+05
Exit Nozzle Dia. (m)	0.4	0.4	0.4	0.4
Estimated Mixture Ratio	3	3	3	3
Results				
Propellant Mass (kg)	1176	1200	988	1012
Fuel Mass (kg)	294	300	247	253
Oxidizer Mass (kg)	882 (from rail)	900 (from rail)	741 (from rail)	759 (from rail)
Material Package Mass (kg)	1	10	1	10
Rocket Nozzle Mass (kg)	104.4	104.4	104.4	104.4
Total Vehicle Mass (kg)	399.4	414.4	352.4	367.4
Launch Velocity at end of:				
1.5 km Rail (km/s)	2.764	2.712	3.051	2.983
5 km Rail (km/s)	5.214	5.114	6.019	5.877
10 km Rail (km/s)	7.340	7.215	9.232	8.988
15 km Rail (km/s)	8.736	8.611	12.147	11.798
20 km Rail (km/s)	9.655	9.546	15.036	14.567

However, as a result of the response from the open engineering approach, (see Section 5.4 for details), it was identified that this original evaluation of the DOLR concept excluded the effects of the oxidizer gas momentum. A modified version of the simulation spreadsheet,

LauncherSimulator_RevA.xls, was created to include accelerating the oxidizer gas to 80% of the velocity of the rocket sled. Accounting for only 80% of the oxidizer momentum was a suggestion from the open engineering approach detailed in Section 5.4. The modified results for the DOLR concept including the effect of oxidizer gas momentum are summarized in table 4.16b.

Table 4.16b: Results for the Modified Distributed Oxidizer Launch Rail Analysis

Distributed Oxidizer Rail	Sea-Level Atmosphere		Perfect Vacuum	
	1 kg Material Package	10 kg Material Package	1 kg Material Package	10 kg Material Package
Parameters				
Specific Impulse (sec)	319.2	319.2	319.2	319.2
Air Density (kg/m ³)	1.19 (sea-level)	1.19 (sea-level)	0	0
Assuming SRR motor Performance Values:				
Thrust (N)	9.34E+05	9.34E+05	9.34E+05	9.34E+05
Exit Nozzle Dia. (m)	0.4	0.4	0.4	0.4
Estimated Mixture Ratio	3	3	3	3
Results				
Propellant Mass (kg)	1456	1472	1256	1280
Fuel Mass (kg)	364	368	314	320
Oxidizer Mass (kg)	1092 (from rail)	1104 (from rail)	942 (from rail)	960 (from rail)
Material Package Mass (kg)	1	10	1	10
Rocket Nozzle Mass (kg)	104.4	104.4	104.4	104.4
Total Vehicle Mass (kg)	469.4	482.4	419.4	434.4
Launch Velocity at end of:				
1.5 km Rail (km/s)	2.384	2.351	2.594	2.547
5 km Rail (km/s)	4.221	4.165	4.749	4.661
10 km Rail (km/s)	5.638	5.576	6.742	6.613
15 km Rail (km/s)	6.438	6.385	8.279	8.125
20 km Rail (km/s)	6.873	6.837	9.590	9.409

Including the effect of the oxidizer momentum lowered the maximum velocity obtained by the DOLR. From table 4.16b, operating the DOLR in the atmosphere, the maximum velocity obtained at the end of a 20 km rail is reduced by about 28% to 6.87 and 6.84 km/s for the launch of a 1 kg and 10 kg material package respectively. If the DOLR rocket sleds operate in an evacuated tunnel, the maximum velocity achieved is reduced by 35% to 9.59 and 9.41 km/s for the launch of a 1 kg and 10 kg material package respectively. The required fuel mass for the modified analysis ranges from 314 kg to 368 kg. The required fuel mass for launching a 10 kg material package is less than a 2% increase over the mass required to launch a 1 kg material package.

From Section 4.6.2, the required launch velocity range to place a material package at an altitude of 10 000 km was determined to be 8.82 to 10.92 km/s corresponding to the 100% drag reduction and 0% drag reduction cases respectively. From the results in table 4.16b, the maximum velocity achieved by the DOLR for the 1 kg material package ranges from 6.87 km/s to 9.59 km/s for the sea-level and vacuum launch conditions respectively. Therefore, launching a 1 kg material package with the DOLR concept does not fulfill the complete range of required velocities. Since the DOLR concept cannot achieve the full velocity range requirement for the 1 kg material package mass and the fuel mass increase for launching a 10 kg material package is only 2%, the next step in the analysis was to determine the requirements for launching a 10 kg material package to a 10 000 km altitude. The required launch velocity range for the 10 kg material package is expected to be lower, since Trend 6 of Section 3.5.2, determined that an increase in the material package mass decreases the upper range of the launch velocity requirement.

The calculations of Section 4.2.2, were repeated for the launch of a 10 kg material package to a 10 000 km altitude and the results are represented in Figure 4.35. For a 10 kg material package with the launch conditions specified in Section 4.2.3, the required launch velocity was determined to be from 8.83 km/s to 9.8 km/s assuming 100% drag reduction and 0% drag reduction respectively. The reduction of the velocity range also reduced the energy savings between the 100% and 0% drag reduction cases to about 18.82%, which is approximately half the energy savings for the 1 kg material package.

Figure 4.35: Requirements for Launch of a 10 Kg Material Package to 10 000 km Altitude

Figure 4.36 was created to compare the launch velocity requirements to the estimated DOLR launch velocities. Figure 4.36 is a graph of the launch velocity requirements for the 1 kg and 10 kg material packages and the DOLR launch velocity values from tables 4.16a and 4.16b. The blue lines in Figure 4.36 depict the original DOLR performance results and the red lines in Figure 4.36 depict the modified DOLR results. It can be observed in Figure 4.36 that there is a very small velocity difference for launching a 1 kg and 10 kg material packages; in fact, the lines for 1 kg and 10 kg material package velocities overlap in Figure 4.36.

The original analysis (blue lines) for the DOLR concept suggested that the DOLR concept could fulfill the velocity requirements for both the 1 kg and 10 kg material packages if the DOLR operated in vacuum conditions. This result is depicted in Figure 4.36, since the two upper blue lines exceed the launch velocity requirements, or the design range, for the 1 kg and 10 kg material packages.

The modified results (red lines) for the DOLR concept, which account for the momentum of the oxidizer gas, are not as favorable as the original DOLR results. The DOLR concept is not capable of fulfilling the launch velocity design range for the 1 kg material package, since no red line in Figure 4.36 exceeds this design range. However, the DOLR concept is nearly capable of fulfilling the velocity design range for the 10 kg material package if the rocket sled operates in a vacuum. Launching a 10 kg material package in vacuum conditions will be considered as the launcher mechanism for the remainder of the launcher design process. However, operating the rocket sleds in an evacuated launch tunnel probably will not be practical since the rocket sleds generate combustion gases that would degrade vacuum conditions and likely could not be pumped out in a timely manner. Further work on the DOLR concept is needed to prove if the concept will be useful for the final design of the OpenMDS.

Figure 4.36: Comparison of Launch Velocity Requirements and DOLR Performance

The next step in the launcher design process was to consider the layout of the launcher site assuming a DOLR with a length of 20 km. Two layouts for the launch Section of the DOLR launch rail were considered. Figure 4.37 depicts the launch site located on the surface with a curved Section of track leading up a mountainside and an alternative launch site with a large subterranean curved Section leading in to an inclined launch Section. The depth of the subterranean curved Section was approximately 3500 meters, which is in the depth of the deepest mine[†]

Figure 4.37: Launch Site Layout for Distributed Oxidizer Launch Rail

The accelerations associated with both launch site layouts were calculated along the launch rail. The subterranean launch rail layout resulted in lower accelerations of about 230 to 720 g in the curved Section versus the 695 to 865 g experienced in the curved Section at the base of the mountain for the surface case. The subterranean launch rail reduces the acceleration experienced by about 145 g; however, the cost and complication of creating the subterranean tunnel would

[†] Western Deep Level Mine, South Africa - <http://www.southafrica-travel.net/north/a1johb04.htm>

offset the reduction in acceleration forces. It will be assumed that the launch site will be located on the surface.

4.6.5 Summary of the Proposed Launcher System

The launcher system selected is the DOLR concept launching 10 kg material packages located on Earth surface. This selection was made primarily because the velocity requirement range of the 10 kg package is close to the modified performance results of the DOLR concept. The operating parameters of the DOLR launcher are summarized in Figure 4.38.

Figure 4.38: Performance Parameters of Distributed Oxidizer Launch Rail (DOLR)

4.7 Preliminary Design for the Material Packages

Section 4.2.3 concluded a 1 kg material package mass would be usable; however, during the evaluation of the launcher mechanism, in Section 4.6.5, it was decided that a material package mass of 10 kg would be more suitable. The launch analysis of Section 4.2.3 determined the following requirements for the 10 kg material package:

Material Package Mass: 10 kg

Diameter: 0.0409 m

Length: 0.99 m ($L/D = 24$)

Orbital Insertion Rocket Motor Mass: 3.71 kg

Orbital Correction Rocket Mass: 0.0833 kg (maximum for last material package)

Payload Mass Fraction: 0.63

The following sub-systems have been identified for the material package along with the associated mass:

Payload System: This system consists only of the construction material required for the space project. All the analysis done has assumed a construction mass of solid steel.

Mass 6.3 kg

Orbital Insertion Rocket Motor: A solid rocket motor is fired, at material package apogee, to place the material package into an elliptical orbit. High performance values have been assumed (Specific impulse of 290s and Propellant Mass Fraction of 0.92).

Mass: 3.71 kg

Orbital Correction Rocket Motor: An additional rocket impulse is needed to change the position of the material package to the middle material package orbit. This rocket impulse counters the orbital 'smearing' caused by Earth's rotation during the launch widow. The middle material package requires no additional rocket impulse while the required rocket impulse increases with the distance from the middle material package.

Mass: 0.0833 kg

Guidance System/Rocket Firing System: A system is required to determine an appropriate time and location to fire the rocker motors. This system might be as simple as a time delay fuse that ignites the rocket motor at a specific time during launch. Alternatively the system could accept a signal from a satellite that monitors the position and attitude of the material package and issues appropriate instructions to the material package. The design goal for this system should be to reduce the 'smarts' that are onboard the material package.

Mass: To be determined

Attitude Control System: A system might be required to adjust the orientation of the material package prior to firing of the rocket motor. A conventional means for attitude control, is the use of cold gas thrusters; however, given the shape and mass constraints of the material packages it is not certain if this may be an option.

Mass: To be determined

Stability Control: Fins are assumed to maintain stable flight of the material package during the transit through the atmosphere. Instead of relying on fins, which increase drag and atmospheric heating, spin stabilization might be a possible solution with no mass penalty.

Mass: To be determined

Thermal Protection System: A system may be required to protect the material package from atmospheric heating and subsequent material ablation. The possible design solutions vary from sacrificing a portion of the payload to utilizing refractory materials.

Mass: To be determined

Figure 4.39 depicts a schematic for the design of the 10 kg material package. The overall design of the material packages leaves plenty of opportunity for further work to occur by the open engineering development group.

Figure 4.39: Schematic of Material Package

4.8 Preliminary Design for the 'Catcher' Satellite

The requirement of the CatSat is to collect all the material packages launched during the 10 minute launch window specified in Section 4.3.3. Section 4.4.7 concluded that all material packages would be located in the middle material package orbit and the CatSat would collect the material packages from the middle material package orbit by adjusting the satellites apogee altitude. The following Sections will discuss the requirements for the CatSat, the propulsion system and operation. The overall design of the CatSat leaves plenty of opportunity for further work to occur by the open engineering development group.

4.8.1 Requirements for Capture Mechanism

The requirements that follow are based on the 10 kg material package stream specified in Section 4.6.5.

Orbit: Elliptical Orbit (Apogee Altitude – 10 000 km, Perigee Altitude – 200 km)

Capture Mass: 10 kg

Capture Velocity: ≈ 0 m/s (since material packages and satellite share the same orbit)

Mass of Construction Material to be Collected per 10 Minute Launch Cycle

- 26 516.7 tonnes (10 Dia Spacing - 0% Drag Reduction – full drag)
- 7 291.6 tonnes (100 Dia Spacing - 0% Drag Reduction – full drag)
- 869.4 tonnes (1000 Dia Spacing - 0% Drag Reduction – full drag)
- 23 889.6 tonnes (10 Dia Spacing - 100% Drag Reduction – no drag)
- 2 789.6 tonnes (100 Dia Spacing - 100% Drag Reduction – no drag)
- 797.6 tonnes (1000 Dia Spacing - 100% Drag Reduction – no drag)

Capture Frequency: Assuming 1 day to capture a 10 minute launch cycle:

- 48.7 packages/s (10 Dia Spacing - 0% Drag Reduction – full drag)
- 13.4 packages/s (100 Dia Spacing - 0% Drag Reduction – full drag)
- 1.6 packages/s (1000 Dia Spacing - 0% Drag Reduction – full drag)
- 43.89 packages/s (10 Dia Spacing - 100% Drag Reduction – no drag)
- 12.1 packages/s (100 Dia Spacing - 100% Drag Reduction – no drag)
- 1.47 packages/s (1000 Dia Spacing - 100% Drag Reduction – no drag)

Assuming 1 week to capture a 10 minute launch cycle:

- 6.95 packages/s (10 Dia Spacing - 0% Drag Reduction – full drag)
- 1.91 packages/s (100 Dia Spacing - 0% Drag Reduction – full drag)
- 0.23 packages/s (1000 Dia Spacing - 0% Drag Reduction – full drag)
- 6.27 packages/s (10 Dia Spacing - 100% Drag Reduction – no drag)
- 1.73 packages/s (100 Dia Spacing - 100% Drag Reduction – no drag)
- 0.21 packages/s (1000 Dia Spacing - 100% Drag Reduction – no drag)

The following table, table 4.17, contains the estimated time to supply construction material to the space construction projects described in Section 2.7. The best and worst cases listed in table 4.17 are from the perspective of the best case for the launcher design, which is 100% drag reduction and 1000 diameter material package spacing for the best case and 0% drag reduction and 10 diameter spacing for the worst case launcher design. The best case for the launcher is desired because it minimizes the required launch velocity and frequency.

From inspection of the table, it is clear that the using the proposed material stream mass driver is not suitable for projects the size of MIR or International Space Stations. It seems the most probable construction project for the best case launcher scenario is the Island One Habitat, with a material supply time of about 36.5 years for daily launch and collection of all the material packages of the 10 minute launch window. The Island One Habitat is a 10 000 person habitat, so constructing it in a time period of 36.5 years seems plausible. The supply time period would be reduced if the launch conditions were not ideal. The other two projects have considerably longer supply time needs for the best case launcher scenario and daily launch and collection. The Island One habitat seems to be the most appropriately sized construction project.

Table 4.17: Time to Supply Selected Space Projects

Project	Total Mass of Project (tonnes)	Quantity of 10 Minute Launches (#)		Time to Supply (years)			
		Worst* to	Best** Case	1 Day Collection		1 Week Collection	
				Worst* to	Best** Case	Worst* to	Best** Case
Mir	67.5	<1	<1	<<1	<<1	<<1	<<1
International Space Station (ISS)	228	<1	<1	<<1	<<1	<<1	<<1
Island One Habitat	10.62×10^6	400.5	13314.9	1.1	36.48	7.7	256
Island Two Habitat	53×10^6	1998.7	66449.3	5.47	182	38.44	1277
Island Three Habitat	124×10^6	4676.3	155466.4	12.81	425.9	89.93	2989.7

* Worst Case - 0% Drag Reduction and 10 Diameter Spacing

** Best Case - 100% Drag Reduction and 1000 Diameter Spacing

4.8.2 Manipulating the ‘Catcher’ Satellite Orbital Position

In Section 4.4.7, the CatSat orbit was decided to be the middle material package orbit. In this orbit, the CatSat must be capable of moving around the orbit to collect all the material packages launched during the 10 minute window. Section 4.4.6 proposed manipulating the position of the CatSat by increasing and decreasing the apogee of the satellite orbit. Figure 4.23 depicts this proposal. To change the apogee altitude, a propulsion system for the CatSat is required. Capturing the material packages requires the CatSat to repeatedly use the propulsion system to adjust the apogee altitude; therefore, a propulsion system utilizing a propellant would probably not be practical. Two propellantless propulsion systems were considered for the catcher satellite, a solar sail and an electrodynamic tether.

The solar sail concept utilizes a lightweight, reflective sail material to harness the momentum of photons from the sun to propel the sail through space. The obvious advantage of a solar sail propulsion system is, it requires no onboard propellant or energy systems. However, from a literature search, it was discovered that atmospheric drag limits use of static solar sails to Earth altitudes above 1000 km [44]. Paul Fieseler proposed a solar sail design with either ramps (like a diffraction grating) or moveable slats which extend the operation range of solar sails to about 350 km altitude. The ramp or slat solar sail design reduces the atmospheric drag on the sail by reducing the cross-section exposed to the atmosphere [44].

Since perigee of the ‘catcher’ satellite orbit is 200 km, which is lower than the 350 km limit for the proposed solar sail design, solar sail technology is not useable for the design of the ‘catcher’ satellite. It also remains to be investigated if, solar sails could maneuver the ‘catcher’ satellite quickly enough to capture the material package stream in the one day and one week capture periods proposed. Additionally, implementing a solar sail propulsion system might not be possible because the solar sail would likely be damaged by material packages that are not collected during capture events. Finally, solar sail propulsion has yet to be demonstrated in space; significant design challenges exist for creating solar sail space structures that are lightweight and durable. The idea of using solar sail propulsion technology for the ‘catcher’ satellite was abandoned to investigate the suitability of another propellantless propulsion system, the electrodynamic tether propulsion system.

An electrodynamic tether system uses a long conductive wire to interact with the Earth's magnetic field to create small propulsive forces distributed over the length of the wire. Unlike solar sail propulsion that functions poorly at low attitude, the electrodynamic tether system's function improves as the attitude lowers. This occurs because the strength of the magnetic field of Earth increases as the altitude is reduced [45].

To reduce the apogee altitude of the CatSat, the electrodynamic tether generates a drag force and electrical current at perigee. The electrodynamic tether system operates on principle of induction where a conductor passed through a magnetic field generates a force that is perpendicular to the magnetic field and the conductor motion and a current flow in the conductor [42]. The orbital energy of the 'catcher' satellite is transformed into electrical current that can be used for the operation of the 'catcher' satellite or stored for reboost of the CatSat. To reboost the apogee of the 'catcher' satellite, an electrical current must be applied to the electrodynamic tether at perigee in an opposite direction to the induced current.

Therefore, to use an electrodynamic tether propulsion system for the CatSat, the satellite requires a system of storing and generating electrical current for the lowering and boosting of the apogee. Electrodynamic technology requires a significant amount of development before a practical system can be created. Presently the Tethers Unlimited^u firm is pioneering electrodynamic technology for deorbiting of satellites and propulsion for small satellites.

Electrodynamic tether technology would seem to be an appropriate propulsion system; however, a preliminary design for the electrodynamic tether system remains to be completed.

4.8.3 Capture Concepts for 'Catcher' Satellite

Section 2.6 discussed previous proposed capture concepts. The NASA concepts of Section 2.6.1 are the most applicable to the design of the CatSat; however, none of the concepts are suitable for collecting the quantity of material packages launched. These concepts inspired proposing three other possible capture concepts as a means to better suit the needs of collecting a large quantity of orbiting material packages.

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^u Tethers Unlimited Inc. - www.tethers.com

The first capture concept proposed is the inverse of an electromagnetic launcher, the electromagnetic decelerator. Figure 4.40 depicts a schematic for the electromagnetic decelerator. The electromagnetic decelerator catcher

Figure 4.40: Electromagnetic Decelerator Capture Concept [23]

mechanism is a coil gun operating in reverse [46]. A series of coils are stacked to create a deceleration tube. The trajectory of the material package must intersect the center of the deceleration tube. The coils are sequentially energised creating a magnetic that retards the motion of the pellet. The kinetic energy of the pellet is converted into electrical current and heat.

This concept has the benefit that the velocity difference between the material package could be converted into electrical energy. However, this concept requires precise alignment between the material package and CatSat and the material package must either be constructed of a conductive material or carry an electromagnet to interact with the coils of the decelerator. In the case where the material package is a conductive material, the current generated by the deceleration will be induced in the material package and cause heating of the material package. This concept was originally proposed for a CatSat in GEO-SYNC where the velocity difference at capture was high. Since the velocity difference for the selected CatSat orbit is very low, the potential energy production from the electromagnetic decelerator would be low. The benefits of this concept seem marginal compared to the complexity of the system; therefore, this concept was abandoned, to investigate a second concept.

The second concept proposed, a rotating drum, is a passive capture mechanism similar to the flat disk of crushable material proposed by NASA in Section 2.6.1.1. Figure 4.41 depicts the rotating drum capture concept. For this concept, the CatSat is a large drum of crushable material that rotates about its central axis. During the capture event, the drum positions itself such that the trajectory of the material packages in orbit pass through the drum. The thickness of the sidewall of the drum would be sufficient to allow the pellet to penetrate through the drum only once. The material packages are captured in the interior of the drum, where they may be collected by a

plunger or netting that travels along the interior of the cylinder for additional processing. The process is similar to shooting BB's into a pop can.

The advantage of the drum concept is the simplicity of operation. The position of the material packages and the catcher satellite are less critical and the size of the drum can compensate for positional error. The drum however would require regular service, since the capture event destroys the material of the drum. Building this rotating drum would require a significant supply of material. This system might be plausible; however, since it based on the destruction of the drum material it probably unlikely that the drum could survive the multiple impacts from even one ten minute launch cycle, given that at the very least there are 126 600 material packages in orbit per 10 minute launch.

Figure 4.41: Rotating Drum Capture Concept

The final concept proposed for the 'catcher' satellite is a design is similar to a Ferris wheel; Figure 4.42 depicts this Ferris wheel concept. The arms of the Ferris wheel are replaced by cables. The cables can be adjusted to adjust for positional error of the material package. The 'seats' of the Ferris wheel are replaced by a grapple mechanism, similar to the grapple mechanisms of Section 2.6.2 and 2.6.3, capable of quickly securing the material packages. The cable arms of the 'catcher' satellite rotate allowing the grapple mechanism to pickup a series of material packages. Once the material package is secured, the cable reels in the material package to the processing center. This concept requires greater positional accuracy than the rotating drum, but less accuracy than the electromagnetic decelerator. Figure 4.42 depicts the electrodynamic tether system for this capture concept.

Figure 4.42: Ferris Wheel Capture Concept

4.9 Summary of the Proposed Open Mass Driver System

Figure 4.43 summarizes the design of the OpenMDS as a flow chart of design values.

Figure 4.43: Summary of Open Mass Driver System Design

Chapter 5 – Open Engineering Organization, Tools, Attempts And Results

5.1 Open Engineering Organization

It is assumed that Open Engineering (OE) project organization should share most of the organizational characteristics of open source software (OSS) project. OSS projects require a founder to initiate the project. For this Open Engineering project, the author has assumed the role of founder of the project. Like an OSS project, the founding person or group for an OE project should have the following tasks to perform:

1. State the requirements for the 'engineered' hardware product and provide a prototype system.
2. Divide the hardware product into work/design modules.
3. Present and promote the project on the Internet.
4. Select the 'best' efforts (evaluation/design) of the design community for compilation into a functional design of the hardware product.

The requirements for task 1 have been fulfilled and are described in Chapter 3 and 4. The tools used to fulfill task 1 are described in Section 5.2. The requirements of task 2, defining work modules, has not been entirely fulfilled at the time of writing; however, the first level of the division of the work modules for the mass driver project is depicted in Figure 4.1, as the five sub-systems of the mass driver design. The fulfillment of task 3, promotion of the product, is detailed in Section 5.3. Task 4 requires the participation of a developer community; Section 5.4 will describe the results of the development community.

To promote courtesy and fairness among the development community the author adopted and improved the list of rules followed by the OSCar project^v. The rules for the Mass Driver Project are as follows:

- o Knowledge and material obtained from the project and contributed to the project is free.
- o Everyone has an equal voice.
- o If you have an idea, share it.

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^v Translated from OSCar Project Rules - <http://www.theoscarproject.org/de>

- If you can improve it, do it.
- If you can aid, please help.
- Mutual respect is the basis of our co-operation.
- Respect the original author of the material.
- Attach your name to material that you contribute.
- Netiquette Practices must be followed -
<http://www.albion.com/netiquette/corerules.html>
- If this project may interest somebody you know, please tell them.

The developer community for an OE project is responsible for a set of tasks that are similar to OSS developers. The tasks for OE developers are as follows:

1. Suggest and evaluate possible design solutions to work/design modules and select appropriate design parameters.
2. Create documents explaining assumptions, design decisions and systems design.
3. Create design specifications and technical/user manuals for hardware product.
4. Review the design work of others for errors in the design process of the hardware product.
5. Report errors in the hardware product to the design community.
6. Suggest new design solutions for reported errors.

The results of the developer community will be discussed in Section 5.4.

5.2 Open Engineering Tools

The fundamental tool for any OSS or OE project is a tool that will allow developers of the project to collaborate on the project. As mentioned in Section 1.2, OSS projects utilize the internet to collaborate on the software projects. Establishing an OSS project is relatively easy, since an internet host site, Sourceforge.net, exists and provides the collaboration tools and storage space to support OSS projects. This site, at the time of writing, hosts over 100 000 projects with over 1 million people registered as participates to these projects. The SourceForge.net web host could not be used to support the development of the mass driver project since the SourceForge web site restricts hosted projects only to OSS projects [47]. Section 5.2.2 describes the host site and collaboration tools utilized by the Mass Driver Project.

Developing an OSS project is relatively straight forward, since once the programming language for the project is defined all developers can use various editors to write programming

code and submit the code for compilation. Managing the information related to an OSS project is relatively simple; it is either programming code or text documents supporting the program. Managing the information regarding the design of a hardware product is a little more complicated than OSS projects, since the design information can exist in many forms such as CAD models, CAM programs, simulation/analysis models and data, spread sheets and text documents. To further complicate matters each type of information may have its own file type and many of the file types require commercial software to read and modify information and are not compatible with other software packages. To permit the greatest number of people to participate in an OE project, it was identified that software tools used by the project should be either products that are open source and freely available or at least a common tool that most computer users would have access to. Section 5.21 describes the software tools used to develop the prototype design for the OpenMDS previously described in Chapters 3 and 4.

5.2.1 Software Products

OSS tools for engineering design are virtually non-existent at this time. The software tools that do exist at this time mainly support the design of computer related hardware such as circuit layout and testing. Consequently, the design of the 'prototype' OpenMDS describe in Chapter 3 and 4 were performed with commonly available software.

5.2.1.1 Microsoft Excel and Open Office Calc

The software tool that was used the most was Microsoft's Excel spreadsheet product. As previously mentioned in Chapter 3 and 4, most of the evaluation of the launch trajectory was performed by creating Excel spreadsheets. This software tool was chosen since the author is more familiar with programming spreadsheets than other programming languages and the Excel program represents a program that is commonly installed on most home computers. There is an OSS alternative that is compatible with Excel program (and other Microsoft Office programs) known as Open Office-Calc^w. The advanced commands for the Calc program differ somewhat from some Excel commands, so when the Excel simulation spreadsheets were imported to Calc errors occurred. It is likely that learning the appropriate Calc commands and modifying the spreadsheet could correct these errors. However, since the priority was to evaluate and propose a

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^w Open Office web site - <http://www.openoffice.org>

prototype design for the mass driver, implementing the Calc program into the project was not completed.

5.2.1.2 AGI Satellite Toolkit (STK) Freeware

As mentioned in Section 4.4, evaluating the appropriate location of the CatSat was first attempted with an Excel spreadsheet; however, this method proved to be cumbersome. An internet search identified AGI's Satellite Toolkit (STK)^x core program as a freeware software tool that would simplified the evaluation of the 'catcher' satellite orbit. The core STK program is capable of calculating and displaying the time-dependent motion of multiple satellites and other space related objects on a 2D map or outputting tables of calculated data. Figure 5.1 depicts an example of a satellite's trajectory superimposed over a 2 dimensional map of Earth. A copy of the program can be obtained for no charge by requesting it from AGI; therefore, this core program is compatible with the Open Engineering initiative.

Figure 5.1: STK 2D Map^w

As mentioned in Section 4.4, the analysis and figures for the CatSat utilized a 3D visualization module for STK that was used under a 7 day evaluation license. The 3D visualization tool made evaluating the rendezvous or capture event of the satellite and material packages much easier since the position and velocity data can be viewed directly on screen when the satellite approaches the material packages. An open source module capable of 3D visualizations of the satellite and material package orbits would be very beneficial to the continued development of the mass driver project.

An additional module that would be beneficial for continuing the development of the mass driver project would be a module to simulate the motion of a satellite with alternative propulsion

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^x AGI Satellite Toolkit Website - <http://www.stk.com/products/desktopApp/stkFamily/modules/core/stk/>

systems like electrodynamic tethers and solar sails. As mentioned in Section 4.8.2, further investigation for using electrodynamic tether propulsion to maneuver the CatSat for the collection of orbiting material package stream needs to be completed. A search was made for a software tool capable of simulating electrodynamic tethers; the only program identified was a commercial product, TetherSim⁷, marketed by Tethers Unlimited.

5.2.1.3 Computer Aided Design (CAD) Software

Open Cascade² is an open source CAD package that is freely available and downloadable. It is capable of 3D modeling of surfaces and solids. The Open Cascade program was not used for the development of the prototype mass driver design since the version of the program available in 2004 was not compatible with Windows XP operating systems^{2a}.

Instead of using Open Cascade, the AutoCAD and SolidWorks2004 CAD packages available at Ryerson Computing labs were used. In addition to being available at the university, AutoCAD and SolidWorks were convenient for the author to use, since the author is very familiar with both programs from previous work experiences. The future development of the Mass Driver project and other OE projects will require developers to collaborate CAD information. Since the OSCar project utilized the Open Cascade program for vehicle design, it is likely the program will be suitable also for the design for the OpenMDS components. The program seems to be well documented and a large user manual exists; therefore, learning the program should not be too challenging or laborious for most potential developers.

5.2.2 Collaboration Tools

Since a preexisting host site for open engineering projects did not exist when this project commenced in September 2003, APD Ltd. provided a web server and the domain name www.massdriver.com to provide an internet home for the mass driver project. The web space for the mass driver project, of course had to be more than static web pages, the web site must provide a secure space to store and present information pertaining to the project as well as provide tools to

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⁷Tether Sim web site - <http://www.tethers.com/TetherSim.html>

² Open Cascade web site - <http://www.opencascade.org/>

^{2a} At the time of writing, the version of Open Cascade version is Windows XP compatible.

aid communication and collaboration between participants to the project. At the time of commencement of the mass driver project, the best suited software package for providing support to an open engineering project was PhProjekt^{bb}, which itself is an OSS product. APD Ltd. staff installed and maintains the PhProjekt program on the web server.

Figure 5.2 depicts the main page of the PhProjekt program implemented on the www.massdriver.com web site.

Figure 5.2: Main Page of PhProjekt Collaboration Web Page

The features of PhProjekt are arranged as a series of tabs along the upper edge of the window in Figure 5.2. The chat, forum and mail tabs provide the communication tools that are important for an OE project. These communication tools permit participants whom are remotely located to communicate and exchange design ideas with each other effectively. The files tab provides the space to upload and download files pertaining to the project. The file Section utilizes

^{bb} PhProjekt web site - <http://www.phprojekt.com>

user defined folders to organize the files located within. Figure 5.3 depicts the folders defined for the mass driver project.

Name	Date	Byte	Category	Comment
[-] Mass Driver	2004-02-11 09:31	D		These Files pertain to the Mass Driver project
[-] Background Knowledge Base	2004-02-11 10:49	D		Preliminary research for the MD design
[-] Cost analysis	2004-02-11 09:57	D		All files pertaining to the Economics of the MD project
[-] Design	2004-02-11 10:49	D		All files pertaining to MD design
[-] Catcher Satellite Analysis	2005-01-23 15:19	D		Documents related to the design of the 'Catcher' Satellite
[-] Conceptual Design	2004-02-11 09:55	D		
[-] Mathematical Models	2004-02-11 10:24	D		Folder Containing Mathematical Models to MD Project
[-] Ball Park Estimate	2004-03-05 12:27	D		
[-] Generation 1 Model	2004-03-05 12:29	D		Math Model - Gravity Only model
[-] Explanation and Results	2004-03-05 12:31	D		
[-] Maple Programs	2004-03-05 12:30	D		
[-] Generation 2 Model	2004-03-05 17:07	D		Math Model including Gravity and Atmospheric Friction
[-] Explanation and Results	2004-03-05 17:09	D		
[-] Maple Programs	2004-03-05 17:08	D		
[-] Generation 3 Model	2004-09-29 01:16	D		Simulation Model - Earth Gravity and Atmospheric Effect for a 2D Launch
[-] Launcher Mechanism	2004-10-20 22:47	D		Files that detail the mathematical designs of the launcher mechanism
[-] Pellet Stream Analysis	2004-05-14 00:59	D		Files for analysis and design of the pellet stream
[-] System Design	2004-07-23 18:27	D		Files for System Design
[-] Technical Specifications	2004-03-15 16:51	D		Technical specifications for the Mass Driver
Mass Driver Work Packages.doc	2005-02-27 20:07	25.50 k		A document summarizing work modules for the Mass Driver Project
[-] Misc	2004-02-11 09:53	D		Everything Else

Figure 5.3: Folder Structure for File Section of PhProjekt Collaboration Web Page

These folders contain 126 files totaling, totaling about 42 Mb of space. The files are all information that was generated at every stage of the design process for the prototype OpenMDS described in Chapter 3 and 4. All these files are available for download and modification by any member of the project. Visitors to the web site www.massdriver.com, may view these files by using Login: [guest](#) and Password: [guest](#).

The final feature that was important to the mass driver project was the Wiki tab. The Wiki feature is an environment allowing participants to create and modify static web pages relatively easily using only their internet browser and knowledge of a few simple formatting rules. The Wiki pages can be used to form a “living document”, a centralized document that is easily viewable and modifiable by all participants and defines the state of the project or sub-component of the project at that time. Much of this project’s information is organized as a series of linked Wiki pages that form a ‘living’ document; these Wiki pages describes the current state of the design of the mass driver project and can be updated at any time by any member of the project. The purpose of using

the Wiki environment was to ensure that all participants could quickly and easily view the current state of the design.

The only serious drawback of utilizing the PhProjekt software is, participants must login in with a user name and password to gain access to the project information. The login security measure protects the project information from malicious attack by non-interested parties; however, it prevents search engines and the general public from finding and viewing the project information. Therefore, the PhProjekt program hinders the 'openness' of the project. To ensure that the search engines and the general public can view information related to the project, static public web pages were created to advertise the project and summarize the current design of the mass driver system. The following Section details the attempts at advertising the project to potential participants.

5.3 Open Engineering Attempts

To attract people to the mass driver project, it is necessary to make the existence of the project known. The vast amount of space related material on the internet is a testament to the enthusiasm that exists for space applications. The following Sections describe the attempts and events made to promote the project, both in person and virtually on the internet, to potential participants. The results of these attempts are discussed in Section 5.4.

5.3.1 Public Web Site

As mentioned in 5.2.2, the PhProjekt does not allow search engines and the general public to view the information stored in PhProjekt. Obviously, having search engines locate the site is important since people searching the internet for an 'open source mass driver project' can locate the site. To allow search engines to find the site, a series of public web pages were created to describe the project and the present state of the OpenMDS design. APD Ltd staff provided the author with the template for creating the web pages. As of May 2004, the public web pages were composed of 7 pages describing the overall project and one page describing the prototype design of the OpenMDS described in Chapter 3. As of April 2005, the public web pages totaled nineteen web pages, with twelve of these pages displaying information about the design process and the design of the preliminary mass driver system described in Chapters 3 and 4. Figure 5.4 is a sample of the main public web pages available at www.massdriver.com.

Figure 5.4: Sample of the Public Web Pages

Starting in May 2004, the author submitted the www.massdriver.com URL directly to the Google search engine. Google did not locate the web site until about May 2005. This delay in locating the site might be related to the small size of the public website prior to updating the site with the summary of the mass driver design detailed in Chapter 4. Additionally, by May 2005, Dr.

F. A. Salustri's homepage and a few other internet sites had links to the project pages. These links improve the change of Google finding the web pages.

5.3.2 Public Events

The mass driver project was promoted publicly at a number of different venues. The following paragraph describes the promotional attempts. The results of these promotional attempts are described in Section 5.4.1.

The first efforts at promoting the project were two separate workshops hosted at APD Ltd. These workshops were intended as kick-off meetings to get people local to the Toronto area interested in the project. The work shops were similar to the public kick-off meeting Markus Merz of the OSCar project described in Section 2.2.1, hosted to attract developers to his project.

The first workshop was held in December 2003. At this time, very few details of the OpenMDS design were known; the general concept of launching a series of material packages one after another as a better method of transporting a large amount of material to space could be presented. APD Ltd and Prof. F. A. Salustri extended an invitation to clients and colleagues whom would be interested in space applications to attend the workshop. The second workshop was held near the end of April 2004 when the results for launching to GEO-SYNC without atmospheric drag were known (Section 3.5.1 mathematical model results). Similarly, APD Ltd. and Prof. F. A. Salustri extended an invitation to potentially interested parties.

The next public event, an Open Source Software conference hosted at the University of Toronto, occurred in May 2004. At this conference, a poster describing the goals of the project, the overall mass driver concept and the results at the time, which was the launch to GEO-SYNC without atmospheric drag, were displayed. The attendees were mostly people of the computer science background from all regions of Canada.

Following the open source conference, the next public event was the Materials and Manufacturing Ontario (MMO) Partnerships annual conference, held in Mississauga in June 2004. The purpose of the conference is to showcase the innovation available in Ontario. At this venue, a

booth with a poster similar to the poster for the open source conference was setup. The conference was attended by a mix of academic and industrial individuals from Ontario.

The final public event where the mass driver project was promoted was the International Astronautical Congress conference in Vancouver in October 2004. The author wrote a paper entitled “IAC-04-U.1.07 - Design of a mass driver system concept using open source principles”, which summarized the objectives, concept and present state of the mass driver project [48]. This paper summarized the design results of Chapter three of this manuscript. Prof. F. A. Salustri presented the paper at the conference in Vancouver. The conference was attended by academic and aerospace industry individuals from around the world.

5.3.3 Internet Postings

This Section summarizes the important internet postings that were issued to promote the project. An internet search was performed to target an audience that was more likely to be interested in space applications, like direct Earth launch systems.

The Space Studies Institute^{cc} (SSI) newsgroup was identified as a potential audience for the mass driver project. SSI was established at Princeton University by Dr. Gerard O’Neill as a non-profit education and research organization to study technologies to permit humanity to benefit from space technology. As previously mentioned, in Section 1.4, Dr. O’Neill researched the mass driver, an electromagnetic launcher operating on the Moon [49]. Given that the SSI studied mass driver technology, the SSI newsgroup, with its 400 members, was a logical audience to advertise for project participants or at the very least constructive advice. Two postings were made to advertise the project. The first posting was posted on March 16 2004, which was before preliminary OpenMDS design described in Chapter 3 was complete. At the time of posting, only the results from the gravity only launch to GEO-SYNC were known. The message posted introduced the newsgroup to the application of open source development principles to the design of space launch hardware. The message invited the readers to login to the www.massdriver.com site and review launch concept and present state of the mass driver system design. A copy of the

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^{cc} Space Studies Institute web site – www.ssi.org

message can be viewed in Appendix G and a summary of the response from the SSI newsgroup is detailed in Section 5.4.2.

A second message was posted to the SSI newsgroup, in May 2005, informing the newsgroup of the preliminary design of the mass driver project and the DOLR launcher concept. The message briefly described the material package stream launch concept and directed the readers to view the public pages which detail the present design of the OpenMDS. The readers were introduced to the DOLR concept and invited to view further details on the public web pages. Readers were asked to review the design work and provide comments. A copy of the message can be viewed in Appendix G and a summary of the response from the SSI newsgroup is detailed in Section 5.4.2.

To target an audience with the technical knowledge needed to evaluate the hypersonic slip-stream effects, two web sites providing free technical advice were identified, The Mad Scientist Network^{dd} and Eng-Tips^{ee} Forum.

The Mad Scientist Network is a web site hosted by the University of Washington that accepts science questions and directs the question to the appropriate panel of over 900 volunteer scientist and engineers for an answer [50]. The author posted a question in May 2004 to the Mad Scientist Network which simply asked, *“Is there a significant hypersonic slip-stream effect?”*. The question included a brief description of the mass driver project and the material package stream. The entire message to the Mad Scientist Network can be viewed in Appendix G and the response from the Mad Scientist Network is detailed in Section 5.4.2.

The Eng-Tips Forum’s purpose is to host dialogues between engineering professionals for work-related topics. The site hosts a number of forums organized by engineering discipline and then by topic. Two messages were posted to the Eng-Tips forum. The first message was posted in May 2004, and asked participants in the Aeronautic Engineering discipline if there would be a significant aerodynamic benefit to launching a stream of small launch bodies from Earth surface to orbit rather than a single large launch body.

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^{dd} Mad Scientist Network website - www.madsci.org

The second message was posted to Eng-Tips in May 2005. The message introduced participants of the Spacecraft Engineering forum (Aerospace Engineering discipline) to the DOLR concept. The message pointed readers to the public web pages on the www.massdriver.com website for additional information and asked readers to provide feedback on the DOLR concept. A copy of the messages posted on the Eng-Tips forums can be viewed in Appendix G and a summary of the responses from the Eng-Tips participants is detailed in Section 5.4.2.

5.4 Open Engineering Results

One of the research goals for this project was to document and observe how the design process of a complicated system occurs without the hindrance of traditional organizational structures, deadlines and financial constraints. Obviously, to fulfill this research goal, a developer group was needed to study. At the commencement of the project it was assumed that given the vast amount of internet users, there will be at least a few people interested in designing a material package stream mass driver. The difficulty of establishing a developer group was underestimated. Up until the design results of Chapter 4 were made public, the mass driver project did not attract any developers to the project. After the design results of Chapter 4 were made public, the mass driver project had seven people express an interest in joining the project. The interest and the response to the project tend to correlate with the state of the prototype OpenMDS design. At the beginning of the project the response from people was minimal likely because only a concept of the operation of the system was available. Whereas recently, people have expressed an interest in joining the project because presumably, the proposed prototype OpenMDS interests them. The following Sections summarize the responses the project received from the promotion attempts mentioned in Section 5.3.

5.4.1 Results from Public Events

The first work shop held at APD Ltd. unfortunately coincided with the busy holiday season, so only a few people attended the workshop. The attendees were interested in the concept and there was a good general discuss about the concept; the problem of material package 'pile up' during initialization of the material package stream was discussed.

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^{cc} Eng-Tips Forum web site - www.eng-tips.com

The second workshop held at APD Ltd. attracted approximately 30 people whom mostly were engineers from the Concord and Markham area. The attendees asked a few questions about the operation of the concept but were somewhat skeptical if the concept would be possible. The feedback from the attendees made it clear that a preliminary design of the mass driver system was needed to quantify the requirements of the design and ideally prove the physical feasibility of the proposed material stream mass driver concept. This preliminary design is thought to be the key to establishing a developer group.

The poster presentation at the Open Source Software conference attracted a couple of discussions about the project. However, the attendees were mainly of the computer science background; therefore, an open engineering project involving aerospace technology development probably was not of much interest to most attendees.

The booth and poster at the MMO Partnership similarly attracted a few discussions about the general concept of the project but failed to attract any developers for the project. This experience further indicated that the mass driver system design needed more work to make the project more enticing to potential developers.

Dr. F. A. Salustri presented the paper describing the project at the IAC conference. The response was similar to the previously mentioned conferences. The design presented was the preliminary of the OpenMDS described at the end of Chapter 3. It was stressed that this was not the final finding of the project, merely the state of the design at the time. As mentioned in Chapter 3, this design had challenging design requirements that are not practical for a near term application; it is likely the response to the paper was poor because the design did not seem practical.

5.4.2 Results from Internet Postings

It is common for people participating in virtual (internet) communities to protect their true identity by using usernames, since identity protection reduces the risk of personal information of the person to be stolen by individuals with malicious intent. However, over time, members of internet communities, such as open source development groups, tend to interact with each other and establish a certain amount of trust and friendship amongst each other. Overtime, the

anonymity of members of an internet community subsides and members of the internet community trust each other and know the true identity of the person behind the user name. Since this project is just beginning to establish an internet based development community, the identity of the people providing feedback to the project is limited to user names; the person providing the feedback is essentially anonymous and the credibility of the information provided is questionable. However, it is likely that overtime, if these people became actively involved with the project, the true identity of the person would be known and the credibility of the information could be verified. The following paragraphs summarize the responses received from the users in response to the message postings previously described in section 5.4.1.

The first posting on the SSI newsgroup in March 2004 received four responses. The first user, TangoMan, made a few technical suggestions. Firstly he suggested if the purpose of the system was to supply construction material to space projects, then the design of the proposed device would be much simpler if it was located on the Moon instead of Earth. If the proposed system operates from Earth surface, then there is the potential for the material package stream to destroy the satellites that presently occupy LEO. Relocating the 'catcher' satellite to LEO would lower the risk to existing spacecraft. Finally, this user indicated that kinetic energy difference between the material packages and the 'catcher' satellite would require a station keeping system.

The second user, Xenophile, also responded with technical suggestions. He suggested that possibly the material packages could be made of ice, or encased in ice as a thermal protection system. To reduce the velocity difference between the satellite and the material packages at capture, Xenophile suggested a rotating tether might be of use.

The third user, Mitchell James, responded with his experiences with collaborative engineering and open source projects. He provided a link to his web page for collaborative engineering projects^{ff}. From his experiences, he believes that all open source projects are required to provide the participating developer/users something useful. In the case of software, it is a software product. Successful projects, offer the developer/user something that is immediately useful to the developer/user, this encourages the developer/user to continue to improve the

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^{ff} Mitchell James Collaborative Engineering web site - www.Innertransit.org

product. According to Mitchell James, the mass driver project does not fulfill any of the above criteria; therefore, the project is unlikely to attract a developer/user group. He suggests that this is an issue for many open source engineering projects, how do you attract developers to a project that takes many years to complete?

The fourth user to respond to the first posting was RyanZ. This user also commented on the open source aspect of the project. The user was contemplating observing the design process in an open source project as his Master's degree thesis; however, he did not find a suitable project to observe. RyanZ agreed with Mitchell James statements. RaynZ stated that you cannot assume the ideology, "if you build it they will come". From his observations of open source, it seems that a project is more likely to be successful if it slightly extends the knowledge base rather than making a leap in knowledge.

The author responded to the responses by thanking them for the feedback. It was discouraging at this point that the project was not gathering any developers; however, this simply emphasized that a preliminary OpenMDS design was need to demonstrate the potential benefits of the mass driver system and attract people to the project. Therefore, at the time of the second posting on the SSI newsgroup a substantial amount of design work was completed and presented to the SSI newsgroup. The second posting received a response from five SSI users.

The first user, Kent Ogletree, responded with a brief technical suggestion. This user suggested that the concept could be improved by incorporating Mag-Lev technology to suspend the rocket sled over the rail.

The second user, Frank, praised the graphics and presentation of the public web pages. He was interested in obtaining the open source software used for the evaluation and design of the mass driver project. He provided a link to his ring spacecraft⁸⁸ that might be an idea for a 'catcher' satellite concept. I responded to Frank, indicating very little open source software exists for engineering design; therefore, much of the analysis was done by programming Excel spreadsheets. The ring spacecraft is like the name indicates a ring with netting stretched across the interior. The

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⁸⁸ Ring spacecraft www.solar-thruster-sailor.info/figs/fig2.html

spacecraft utilizes ion thruster for propulsion. I indicated to Frank that the project needs people to develop the 'catcher' satellite design and his concept might be a suitable design, particularly if he would like to participate in the project.

The third response came from Xenophile, a user that responded to the first SSI posting. He indicated he would be very interested in obtaining the spreadsheets created. The author directed Xenophile to log into the www.massdriver.com site and download the spreadsheet files and supporting documents from the files Section.

The fourth response came as an e-mail from Giorgio Gaviraghi, an industrial designer in Italy and the president of the Mars Society Italy. Mr. Gaviraghi expressed an interest in joining the Mass Driver project to help design the space construction project. He has previously proposed a design for modular habitat, named AstroHab which could be used for a lunar base. He was interested in modifying the AstroHab design for use for the mass driver project. The author had a lengthy discussion with Mr. Gaviraghi via e-mail exchanges and we identified that to make his modular concept feasible, the size of the material packages must be at least 100 kg. Since the project lacked a development group at this time, we agreed that the author would contact Mr. Gaviraghi when the mass driver project has a development group ready to consider the space construction mission details.

The fifth and final response to the second SSI posting was an e-mail from Mr. Kumar Bharath, an aeronautics graduate located in India. Mr. Bharath expressed an interest in joining the project but he wanted to review more information related to the project and he wanted to know the size of the present developer community. Mr. Bharath did not reply to a follow up e-mail sent by the author, it is assumed that he was not interested in the project at this time.

Comparing the responses from the first and second SSI postings, it seems there is a correlation between the prototype design and the response from the audience. At the time of the first posting, there was not much more than the concept of the mass driver system and consequently nobody was willing to become a developer for the project. At the time of the second posting, the prototype mass driver design detailed in Chapter 4 was available for review; therefore, more details about the mass driver system were known and two people expressed an interest in

joining the project. It is expected that this trend would continue, the more detailed the prototype design becomes the more people will be attracted to the project.

The following paragraphs discuss the results of the technical questions posted on the Mad Scientist and Eng-Tips forums. The question regarding hypersonic slip-stream effect was sent to the Mad Scientist Network in May 2004. At this time the author was having a difficult time locating any previous work on slip-stream effects for hypersonic flows. A physicist for the Mad Scientist Network, John Link responded to the question in September 2004. Mr. Link indicated that many of the scientists of the Mad Scientist network declined to answer the question because it is a difficult and complicated subject of ballistics and it is beyond the usual scope of questions on the Mad Scientist network. Mr. Link also did not have an answer to the question; however, he suggested a possible means of estimating the effect. Mr. Link suggested I find information regarding the aerodynamics effects about shotgun shot in the initial stages of the ballistic trajectory either from shell manufacturers or traditional journals. The information could be used to program a spreadsheet to calculate the trajectory of inline BB shots. Mr. Link provided a link to a spreadsheet he created to simulate the trajectory of a single BB from a shotgun shot and suggested this could be the starting point for my analysis. Mr. Link indicated the spreadsheet is a simplified version of the calculation method that aerospace scientists use to calculate the ballistics of rockets, missiles and other spacecraft. These accurate computer models require computation power beyond that of a personal computer; an Excel spreadsheet provides a convenient means of performing these calculations on a personal computer.

By the time the response from Mr. Link was received, the author independently located the hypersonic interference drag information previously detailed in Section 4.2.4. Therefore, research into ballistics of shotgun shells was only briefly investigated but no significant information was identified. The link Mr. Link provided to his spreadsheet calculation was reviewed. Mr. Link's spreadsheet operated in a similar manner to the spreadsheets created by the author; therefore, the evaluations performed in Chapters 3 and 4 are similar to those used by aerospace scientists.

The first posting on the EngTips forum regarding the benefits of launching a series of inline bodies slip-streaming each other to space received one response. The user IRstuff responded

suggesting that for any slip-stream drag reduction to occur, the bodies would almost be in contact with each other. If these bodies were traveling one behind another then creating a guidance control system to maintain the position would be challenging. Finally vortex shedding and other disturbances from the leading bodies might cause significant problems to the trailing bodies. Since, the responses to this question was limited to one, it is assumed that the users on the EngTips forum do not have experience with hypersonic slip-stream effects. The comment from this user suggests there are many aerodynamic issues that occur around the launch bodies that must be evaluated.

The second posting on the EngTips forum was posted in May 2005 and asked users to provide comments regarding the DOLR concept. Four users provided a response to this posting.

The first response came from the same user as the first posting, IRstuff. The user pointed out that aerodynamic heating of the oxidizer gas at high velocities could cause pre-ignition of the fuel on the rocket sled and uncontrolled combustion would occur.

The second user, Greg Locock, suggested that shock wave effects created by the rocket sled would cause problems with combustion. He also suggests that the initial performance of this concept would be better than a conventional rocket sled but at high velocities there would be little difference between this concept and a traditional rocket sled. This user suggested that the better way to get to orbit is to fly the payload to high altitude and use a rotating tether to pick the payload up and place it in orbit.

The third user, btrueblood, suggested the concept is similar to a RAM jet or the RAM tube accelerator (Section 2.5.13) of Washington University. The user had first hand experience with the RAM tube because he worked on the concept as a graduate student. This user indicated that the statement made by the user Greg Locock, was not completely correct. The performance of a RAM jet is better than a rocket because it accelerates the oxidizer only to about 80 to 90% of the velocity of the RAM jet, whereas a rocket accelerates the oxidizer to 100% of the velocity of the rocket. Therefore, the concept could perform better than a conventional rocket sled. The user also suggested that the fuel for the proposed hybrid rocket motor could be made from recycled tires.

The final user, nhughes1, commented that the mechanics of the system is similar to a space launch concept he evaluated for development. The concept involved accelerating a launch body down a rail until it reached a velocity of about Mach 1. At this point, the launch body would ignite a RAM jet and fly to 55 km altitude. At this altitude, a rocket propulsion system places the body in orbit. The user indicated that the proposed DOLR combustion method is similar to a RAM jet, so theoretically the operation of the DOLR should be possible.

The author posted a response thanking the users for the feedback. These responses received from the EngTips forum were very helpful. The idea of using old tires as a fuel for the rocket sleds could make the launch system economically viable since the energy source, used tires, is relatively cheap, abundant and presently an environmental problem. The author did not consider the DOLR concept as being similar to a RAM jet on a rail until the responses to the idea were posted. The revelation that this concept was similar to a RAM jet immediately identified to the author that there was a problem with the evaluation of the DOLR concept of Section 4.6.4.1. The author did not consider the momentum of the oxidizer gas in the initial analysis. The results of the modified DOLR analysis, including momentum, were presented previously in Section 4.6.4.1.

During the writing of this document (June 2005), the author was contacted via e-mail by a Mr. Joeri van Zuilen, a graduate of civil engineering at Delft University in the Netherlands. Mr. Zuilen found the site as a result of a Google search. Coincidentally, he was interested in setting up an open source mass driver project and was researching if a similar project exists. He has four other friends whom are graduates from Delft University in aerospace engineering and physics disciplines that are interested in starting an open source mass driver project. Mr. Zuilen indicated they are considering the design of a mass driver capable of launching satellite payloads, since this would be of greater near term and economic purpose. Mr. Zuilen and his associates were interested in starting a project in the fall of 2005 and would create a web site to host their project. Mr. Zuilen and the author agreed to remain in contact regarding future developments of either of the open source projects.

Chapter 6 - Conclusion and Recommendations

6.1 Open Mass Driver System Design

Thus far, the evaluation and design process for the OpenMDS has not identified any 'show stopper' physical effects that suggest the design of the OpenMDS would be impossible. Since directly investigating the aerodynamic flow effects occurring around the material package stream was determined by Dr. F.A. Salustri and Dr. J. Lassoline to be beyond the scope of this project, the evaluation of the OpenMDS system determined the range of performance requirements for the OpenMDS based on assuming best and worst case scenarios and a range of material package separation (spacing) distances. The best case scenario corresponds to the ideal situation where aerodynamic effects of material package stream creating an evacuated tunnel (100% drag reduction) through the atmosphere. The worst case scenario corresponds to aerodynamic effects of the material package stream that results in no change in atmospheric properties (0% drag reduction). The conditions for worst scenario case are no different than launching a single material package through the atmosphere; therefore, if aerodynamic conditions are determined to be the worst case then, the material package stream is not beneficial. Figure 4.42 of Section 4.9 depicts the summary of the current 'prototype' OpenMDS design and organizes the design results for the best and worst cases and material package spacing into a flow chart.

The evaluation and design of the OpenMDS thus far has proven that, if a steady state material package stream could be established, then a system to launch and capture the material packages is plausible; however, a practical OpenMDS requires significant further research and design work to make the launch system, material package system and 'catcher' satellite systems feasible. The further design of these systems is highly dependent on evaluating the actual aerodynamic effects that are occurring about the material package stream. If the aerodynamic effects are known, then specific design requirements from the range of performance values in Figure 4.42 can be chosen and detailed evaluation and design of the sub-systems could occur.

The purpose of establishing the material package stream is to exploit slip-stream drag effects which have been determined to potentially decrease the launch energy requirements over 80% (see table 4.2); however, the amount of slip-stream drag reduction is dependant on the mass

of the material package, the launch angle and the apogee altitude of the material package (or orbital altitude of the CatSat). The proposed OpenMDS, summarized in Figure 4.42, potentially could benefit from an 18% reduction in launch energy if best case scenario of 100% drag reduction occurs for the 10 kg material packages launched at a 50 degree angle (from vertical).

The proposed OpenMDS design, summarized in Figure 4.42, is likely not the optimal design and leaves plenty of potential for others to optimize the design. The interrelatedness of the sub-systems and design parameters of the OpenMDS (see Figure 2.1) requires a compromise between the competing design parameters to be determined. Throughout the evaluation of the OpenMDS, the design goal was mainly to reduce the launch velocity to levels that were achievable by current or near-term launch technology. To lower the launch velocity, the altitude of the CatSat was reduced; however, this had a negative effect of increasing the velocity difference between the CatSat and material packages. Lowering the altitude of the CatSat complicated the operation of the OpenMDS by introducing a launch window and an orbital insertion rocket motor onboard each material package. The mass of the material package was increased to reduce the upper limits of the range of required launch velocities and increase the payload mass fraction of the material package. However, increasing the material package mass caused a significant decrease in potential energy savings of the material package stream (from 34% for the 1 kg material package to 18% for the 10 kg material package). Lowering the launch velocity has a beneficial effect of reducing the launch frequency requirements. The launch frequency requirements are dependant on the launch velocity and the spacing between the material packages. The limiting factor for creating a practical OpenMDS design will probably be the launch frequency of the launcher. The launch frequency requirement for the 'very' best case scenario for the proposed OpenMDS was determined to be 211 Hz which is about double the frequency of the state of the art rotary cannon, GE M61 Vulcan. The requirements for the worst case scenarios are an order of magnitude greater than the current state of the art.

Determining an optimal OpenMDS design requires identifying an appropriate material package mass, slip-stream drag reduction effect and material package spacing that reduces the launch velocity and launch frequency to values obtainable by current or near term launch technology. The selected design values also must maximize the launch energy savings created by

the material package stream and maximizing the payload mass fraction of the material package to maximize the usefulness of the OpenMDS. To determine an appropriate compromise between the design parameters, evaluating the actual slip-stream drag reduction created by the material package stream is required. This evaluation should include how spacing between the material packages affects the slip-stream drag reduction.

A detailed investigation of the aerodynamic effects about the material package stream will determine if the OpenMDS concept is feasible. If the drag reduction proves to be negligible, then the OpenMDS concept is irrelevant. If drag reduction proves to be favorable at extended separation distances, then the investigation should evaluate the thermal loads on the material packages. If the thermal loads are manageable then the next step of the investigation should consider if it is possible for the material packages to maintain a trajectory (flight path) in-line with each other without causing the material packages to contact each other or 'pile up'. If the material packages can maintain an in-line flight path, then the last step of the investigation should consider how to initiate the material package stream and the aerodynamic effects of the transient material package stream.

If the investigation of the aerodynamics effects of the material package stream is determined to be favorable then the design of the OpenMDS should continue. The following areas of the OpenMDS design could be further designed:

- DOLR system could be further evaluated to determine if the concept is capable of achieving the required launch velocity and frequency. The investigation of the DOLR concept should include the effects of friction between the slipper of the rocket sled and the launch rail. If the DOLR concept proves not feasible, then railgun technology could be further considered for the launcher application.
- The suitability of fin stabilization for the material packages could be investigated and the size and material for the fins could be specified.

- The requirements for guidance, attitude control, and rocket motor systems of the material packages capable of firing the rocket motor at the correct position and time could be evaluated.
- The feasibility of using an electrodynamic tether propulsion system to maneuver the CatSat for the purpose of collecting material packages could be evaluated. If the electrodynamic tether system is feasible, then the design of the tether and energy storage and generation systems could be considered.
- The space construction mission that the OpenMDS would supply should be considered. The design of the construction mission would require in-space manufacturing process to convert the raw material supplied by the OpenMDS into the final form required by the design of the space project. Proposing a feasible construction project is just as important as the design of the OpenMDS because without a credible and feasible space construction project there is no need for the OpenMDS.

Ultimately, the further design of the OpenMDS depends on the evaluation of the aerodynamics of the material package stream. If the drag on the material packages is only marginally reduced, then the usefulness of the OpenMDS concept is questionable. The design process presented in this manuscript should provide an adequate roadmap for future investigation, design and optimization of the OpenMDS concept.

6.2 Open Engineering

Establishing a developer group for the open engineering (OE) of the OpenMDS concept was more difficult than initially expected at the commencement of this project. The OE results of Section 5.4, suggest that to attract potential developers to the project, a preliminary design of the system is required. Presumably the preliminary design demonstrates to potential participants that the concept is feasible and the present development team is committed to completing the project. The Open Engineering (OE) aspect of this project did not begin to develop until the design results of Chapter 4 were posted on the public web pages of the www.massdriver.com in May 2005 near the end of the project. Until May 2005, nobody expressed an interest in the project, presumably because the concept was not proven to be either feasible or beneficial. There was simple not

enough information about the project to draw the attention of any potential participants. After the results were posted in May 2005, seven people have expressed an interest in joining the project. These results suggest that for an OE project to begin to attract a development group, a significant amount of design work must be completed to prove the proposed concept is feasible, beneficial and the present participants are dedicated to the project.

The low level of OE participation in the project might have occurred because the expertise needed to evaluate and design OpenMDS, particularly the aerodynamic effects of the material package stream, requires the participation of highly skilled aerospace professionals. The potential pool of aerospace experts willing to participate in the OpenMDS, is likely smaller than potential pool of programmers willing to work on OSS projects. OSS projects usually has an expert or a group of experts to divide the project into a number of work modules, with the skill level required to complete these modules ranging from expert to novice developers. At the present time, all the design work for the OpenMDS, requires the participation of aerospace experts, since the design of the OpenMDS depends on the aerodynamic effects that occur about the material package stream. The design of the OpenMDS has not been reduced to the level where novice developers from diverse disciplines can participate by performing design of small components or simple calculations.

This is where the challenge exists, how do you attract experts to volunteer their time and effort to evaluate these aerodynamic effects? Evaluating these aerodynamic effects may require basic research and experimental testing such a hypersonic wind tunnel tests; these tasks are difficult to do from a home computer on a no cost basis. This aerodynamic problem probably would be best addressed by a PhD project.

Developers of OSS projects volunteer for OSS project because at the end of the process they are compensated for their effort with a copy of a software product that satisfies their collective needs [4]. OE projects have less tangible commodities to offer as compensation. At the end of the design process the end result is a set of documents specifying the design and possibly instructions for the construction of the design. The end result of the OSCar project (Section 2.21) was a prototype vehicle and a set of documents specifying the construction of the vehicle. Participants of the OSCar project did not receive any compensation other than the satisfaction of

being involved in the project. It can be concluded that a tangible commodity is not essential for the success of an OE project, but it probably would attract participants.

The OSCar project (Section 2.2.1) and OpenCores (Section 2.2.2) projects have had some success at adapting open source principles to the design of physical devices. The goal of both the OSCar and OpenCores projects is to improve the design of a well known technology. In the case of OSCar, the goal is to design an improved vehicle design and the OpenCores project is to design integrated circuits with improved function and performance. Neither project requires deriving much new knowledge or much physical experimentation to permit a successful design. It is a matter of rearranging how present technology is applied. The design of the OpenMDS concept is a little more demanding than the design goals of the OSCar and OpenCores projects because the design process is pioneering a new concept not merely improving an existing design or technology. The design of the OpenMDS is similar to ThinkCycle projects which are pioneering new designs. ThinkCycle projects use open source collaboration to identify potential projects. The design of the solution to these problems utilize the effort of student design teams rather than internet volunteers; therefore, ThinkCycle projects rely more on the traditional academic development model where basic research into new knowledge is done at the university level and then offered to industry for further development. The OpenMDS project probably represents the first project to attempt deriving new knowledge from open source collaboration.

A research goal for this project was to evaluate the OE design process. This project did not attract an active developer group; therefore, this research goal remains unfulfilled. If a future project has a similar mandate to evaluate the OE design process, the subject of the design process should be carefully selected. Selecting a challenging design problem to solve using open source principles could limit the potential pool of developers because there are a limited number of people with the required expertise. The design project must be interesting to attract the attention of potential participants yet present a non-trivial challenge. Improving the design of a present device could fulfill these aforementioned requirements. Additionally, such a project would be a good research project because the open source development process can be easily compared to the conventional design process and benefits and drawbacks can easily be identified.

The fact that this project attracted the attention of seven people^{hh} suggests that support for the project exists; however, it remains to be determined if the open source development model will be successful at deriving new knowledge and pioneering new designs. One sure way of determining if this project will be successful is by continuing the project. The author intends to continue the project as a hobby, either continuing to work on the www.massdriver.com site if it is maintained or joining Mr. Joeri van Zuilen's project (see Section 5.4.2).

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^{hh} Four of these people found the project because they were specifically looking (googling) for a project to design a mass driver space launcher using open source principles.

Appendix A – Launcher Technology Comparison Data

Table A-1: Launcher Technology Comparison

	Payload (kg)	Velocity (km/s)	Energy (MJ)	Reference
Conventional Cannon Launcher				
Paris Gun	120	1.4	120	[2]
HARP	205	2.2	500	[2]
HARP	68	3.6	440	[2]
Light Gas Launcher				
Sharp Light Gas Gun	4.9	4	39.2	[51]
Light Gas Gun	0.18	7.7	4.4	[2]
Distributed Light Gas Gun**	113	7	2768.5	[14]
RAM Launcher				
Washington University	0.1	2.7	0.36	[24]
Railgun Launcher				
ISL Pegasus	1	2.5	9	[12]
Railgun	0.001	12	0.06	[2]
Railgun	1.1	3.5	6.7	[2]
Railgun	2.4	2.6	8.1	[2]
NASA-Lewis Research	0.00035	3.5	0.0021	[52]
Lawrence Livermore Lab	0.001	15	0.11	[52]
Los Alamos Lab	0.02	7	0.49	[52]
Westinghouse	0.5	6	9	[52]
ESA Sub-orbital**	4	2.15	30	[12]
ESA LEO**	60	6	3500	[12]
Railgun**	1000	7.5	35000	[13]
Railgun**	1264	5.5	19118	[53]
Railgun**	15	4	40	[2]
Railgun**	2	10	100	[2]
Railgun**	18.6	5.5	1400	[15]
Railgun**	6500	5	81250	[16]
Railgun**	2055	20	411000	[16]
Railgun**	1000	12.3	75000	[54]
Coilgun Launcher				
Coilgun	7	0.6	1.3	[2]
Coilgun	0.18	1	0.08	[2]
Coilgun	5	0.3	0.23	[2]
MIT Coilgun	10	0.2	0.2	[52]
Coilgun**	200,000	8	648000	[19]
Coilgun**	0.15	4	1.2	[2]
Coilgun**	1100	0.3	50	[2]
Novel Launcher				
Molder Rail Launcher**	700	8.12	23077	[17]
Lofstrom Launch Loop**	15,600,000	14	1500000	[20]
Lofstrom Launch Loop**	5000	10.5	275625	[20]

** Proposed

Appendix B - Calculation Method for EarthSim.xls and Revisions

Figure B-1 depicts the three bodies, the Earth, material package and Moon, considered in the Earth_Sim simulation spreadsheet.

Figure B-1: Overview of 2-D Earth Launch Simulation Model Geometry

Figure B-2 depicts the ballistic trajectory of the launch through the four regions of Earth atmosphere. If no rocket impulse occurs at apogee, the material package follows the ballistic trajectory depicted in Figure B-2 back to Earth.

Figure B-3a depicts the definition of the launch angle for the simulation spreadsheet. Figure B-3b depicts the direction of Earth's rotation and the x and y velocity components for an arbitrary angle of longitude.

Figure B-4 is a free body diagram of the forces, force components and angles of force acting on the material package.

Figure B-2: Atmospheric Regions Consider in 2D Earth Launch Simulation Model

Figure B-3: Earth Launch Locations

Figure B-4: Free Body Diagram (FDB) for the Pellet (material package)

Spreadsheet Simulation Operations

The spreadsheet simulation performs the following steps:

1. The user inputs the desired time step multiplier, maximum time step, launch velocity, launch frequency, launch altitude (above Earth surface), launch longitude, launch angle, material package mass, material package diameter and material package density. For Earth_Sim_RevB, RevC and RevD the user must also specify the rocket velocity addition and the rocket motor specific impulse and propellant mass fraction. For Earth_Sim_RevD, the user must also specify the percent slip-stream drag reduction. The user may also change the constant properties for the air, the Earth and the Moon.
2. The frontal area, volume, length, L/D (length to diameter) ratio, launch energy and spacing of the material package is calculated. The mass transfer rate and the launch power of the system is calculated. See Part A for the calculations.

NOTE: It is important that the L/D ratio close to 24, since the drag coefficient used by the spreadsheet is based on this value (see Part F). L/D ratio is adjusted by varying either the diameter of the material package or the density.

3. At time = 0s, the initial launch position (X and Y co-ordinates calculated – Part B), launch altitude (above Earth surface) and launch velocity inputted by the user are entered into the simulation table.
4. The angles, α and β , is calculated; these angles define the line of action of the gravity forces of the Earth and Moon. See Figure A-4 and equations in Part C.
5. The X and Y components of the launch velocity, $v_{Launchx}$ & $v_{Launchy}$, is calculated according to equations in Part D (*Earth_Sim_RevC and RevD include velocity of Earth's rotation*).

6. The **Earth's total gravity force**, F_{GE} , is calculated based on the material package's position. See Part E for equation.
7. The **X and Y components of the Earth's total gravity force**, F_{Gex} & F_{Gey} , are calculated according to the equations in Part F.
8. The **Moon's total gravity force**, F_{GM} , is calculated based on the material package's position according to the equation in Part G.
9. The **X and Y components of the Moon's total gravity force**, F_{GMx} & F_{Gmy} , are calculated according to the equations in Part H.
10. The altitude for the time step is used to calculate the **density of air**, ρ_{atm} , according to the density functions given in Part I.
11. The altitude for the time step is used to calculate the **temperature of air**, T_{atm} , according to the temperature functions given in Part J.
12. The **speed of sound**, a_{ss} , at the altitude of the time step is calculated according to the equation in Part K. The speed of sound is required to calculate the Mach number of the material package.
13. The **Mach number**, Ma , of the material package for the time step is calculated according to the equation in Part L. The Mach number is required to determine the drag co-efficient, Cd .
14. The **drag coefficient**, Cd , for the material package, with a given Mach number, is determined from experimental data detailed in Part M.
15. The **atmospheric drag** (F_{Dx} & F_{Dy}) are for the material package is calculated according to equations in Part N.
16. The **X and Y resultant forces**, F_{Rx} & F_{Ry} , acting on the material package are calculated according to equations in Part O.
17. The **X and Y average acceleration**, a_{AVGx} & a_{AVGy} , experienced by the material package during the time step, is calculated according to the equations in Part P.
18. The **resultant average acceleration**, a_{AVG} , is calculated. See Part Q.
19. The magnitude of the resultant average acceleration is used to determine the length of the **subsequent time step** according to the calculation in Part R.
20. The **X and Y position and altitude** of the material package for the subsequent time step is calculated using the average acceleration according to equations in Part S.
21. The **X and Y components of velocity**, v_x and v_y , for the subsequent time step is calculated according to equations in Part T.
22. Steps 5 through 21 are repeated for the following time steps.

NOTE: The simulation table should be copy and pasted down to row 15005. The length of the simulation is adjusted by changing the time step multiplier and maximum time step values. The time step multiplier should be adjusted to allow approximately 2000 rows to calculate launch

parameters through the atmosphere (up to 300 km above Earth surface).

23. The required mass of the rocket motor, m_{rocket} and the payload mass fraction, $m.f._{\text{payload}}$, of the material package is calculated according to the equations in Part U.

Part A – Calculated Material Package Parameters

The following design parameters are calculated using the material package mass (m_{package} in kg), material package diameter (D_{package} in meters), material density (ρ_{package} in kg/m³), launch velocity (v_{launch} in m/s) and launch frequency (ω_{launch} in Hz) entered by the user.

See Equations 3.16 to 3.24

Part B – Initial Launch Position

The initial launch position, X and Y in meters, of the material package can be calculated as follows:

$$X = R_{\text{Earth}} \cos \theta_{\text{Longitude}} + A_{\text{Launch}} \cos \theta_{\text{Longitude}} \quad (\text{eq. B.1})$$

$$Y = R_{\text{Earth}} \sin \theta_{\text{Longitude}} + A_{\text{Launch}} \sin \theta_{\text{Longitude}} \quad (\text{eq. B.2})$$

where:

R_{Earth} – Radius of Earth (m)

A_{Launch} – Altitude of launch above Earth Surface

$\theta_{\text{Longitude}}$ – Angle of launch longitude (°) – see Figure 3

Part C – Angles α and β

The angles from vertical to the Earth and Moon gravity forces are α and β (°), respectively (see FigureB-1 and B-4). The angles can be calculated as follows:

$$\alpha = \tan^{-1} [x(t) / y(t)] \quad (\text{eq. B.3})$$

$$\beta = \tan^{-1} [(384\,401\,000 - x(t)) \setminus y(t)] \quad (\text{eq. B.4})$$

where:

$x(t)$ – x position (m) of the material package at time t.

$y(t)$ – y position (m) of the material package at time t.

Part D – X and Y components of Initial Launch Velocity, $v_{Launchx}$ & $v_{Launchy}$

The X and Y components of the initial launch velocity, $v_{Launchx}$ & $v_{Launchy}$ (m/s), for Earth_Sim.xls and Earth_Sim_RevB.xls were calculated as follows:

$$\begin{aligned} v_{Launchx} &= v_{Launch} \cos (\theta_{Longitude} + \phi_{Launch}) \\ v_{Launchy} &= v_{Launch} \sin (\theta_{Longitude} + \phi_{Launch}) \end{aligned} \quad (eq. B.5)$$

where:

v_{Launch} – Initial launch velocity (m/s)

$\theta_{Longitude}$ – Angle of launch longitude (°) – see Figure 3a and 3b .

ϕ_{Launch} – Angle of launch measured from vertical (°) – see Figure B-3a and B-3b.

The X and Y components of the initial launch velocity, $v_{Launchx}$ & $v_{Launchy}$ (m/s), for Earth_Sim_RevC and RevD were calculated as follows:

$$v_{Launchx} = v_{Launch} \cos (\theta_{Longitude} + \phi_{Launch}) - v_{Earth} \sin (\theta_{Longitude}) \quad (eq. B.6)$$

$$v_{Launchy} = v_{Launch} \sin (\theta_{Longitude} + \phi_{Launch}) + v_{Earth} \cos (\theta_{Longitude}) \quad (eq. B.7)$$

where:

v_{Launch} – Initial launch velocity (m/s)

v_{Earth} – Velocity of Earth's rotation at the equator (465.102 m/s) [3]

$\theta_{Longitude}$ – Angle of launch longitude (°) – see Figure B-3a and B-3b .

ϕ_{Launch} – Angle of launch measured from vertical (°) – see Figure B-3a and B-3b.

Part E – Total Earth Gravity Force, F_{GE} , Acting on the Material package

The total Earth gravity force, F_{GE} (N), acting on the material package can be calculated as follows:

$$F_{GE} = [G \times M_{Earth} \times m_{package}] / [[384\,401\,000 - x(t)]^2 + y(t)^2] \quad (eq. B.8)$$

where:

G - Gravitational Constant = $6.67 \times 10^{-11} \text{ m}^3 / \text{kg s}^2$

M_E – Mass of the Earth = $5.976 \times 10^{24} \text{ kg}$

$m_{package}$ – Mass of material package (kg)

$x(t)$ – x position (m) of the material package at time t.

$y(t)$ – y position (m) of the material package at time t.

Part F - X and Y components of Total Earth Gravity Force, F_{GEx} & F_{GEy}

The components X and Y of the total Earth gravity force, F_{GEx} & F_{GEy} (N), can be calculated as follows:

$$F_{GEx} = F_{GE} \sin \beta \quad (eq. B.9)$$

$$F_{GEy} = F_{GE} \cos \alpha \quad (eq. B.10)$$

where:

F_{GE} – Total Earth gravity force (N)

α - Angle from the vertical to the Earth gravity force (°) – see Figure B-4

β - Angle from the vertical to the Moon gravity force (°) – see Figure B-4

Part G – Total Moon Gravity Force, F_{GM} , Acting on the Material package

The total Moon gravity force, F_{GM} (N), acting on the material package can be calculated as follows:

$$F_{GM} = [G \times M_{\text{Moon}} \times m_{\text{package}}] / [x(t)^2 + y(t)^2] \quad (\text{eq. B.11})$$

where:

G - Gravitational Constant = $6.67 \times 10^{-11} \text{ m}^3 / \text{kg s}^2$

M_{Moon} – Mass of the Moon = $7.35 \times 10^{22} \text{ kg}$

m_{package} – Mass of material package (kg)

$x(t)$ – x position (m) of the material package at time t.

$y(t)$ – y position (m) of the material package at time t.

Part H - X and Y components of Total Moon Gravity Force, F_{GMx} & F_{GMy}

The components X and Y of the total Moon gravity force, F_{GMx} & F_{GMy} (N), can be calculated as follows:

$$F_{GMx} = F_{GM} \sin \alpha \quad (\text{eq. B.12})$$

$$F_{GMy} = F_{GM} \cos \beta \quad (\text{eq. B.13})$$

where:

F_{GM} – Total Moon gravity force (N)

α - Angle from the vertical to the Earth gravity force (°) – see Figure B-4

β - Angle from the vertical to the Moon gravity force (°) – see Figure B-4

Part I – Air Density Functions

See Equations 3.24 to 3.28

Part J – Air Temperature Functions

See Equations 3.29 to 3.33

Part K – Speed of Sound

See Equations 3.34

Part L – Mach Number

See Equations 3.35

Part M – Drag Coefficient

Experimental data for the drag coefficient, C_d , vs. Mach number for a finned missile is depicted in Figure 3.7 [31]. The experimental data is valid for a body with an L/D of 24. This data is used for the drag co-efficient of the material package.

The spreadsheet contains a table of C_d vs. Mach number values obtained from Figure 3.7. The spreadsheet uses the Mach number of the material package to 'look up' the appropriate drag coefficient, C_d .

Part N - Atmospheric Forces Acting on the Launch Body

Figure B-5 is a free body diagram (FBD) that depicts the atmospheric drag force acting on a 2-D launch body.

The drag force acting in the x-direction, F_{Dx} (N), on the launched body for all spreadsheets except Earth_Sim RevD are calculated as [35]:

$$F_{Dx} = 0.5 \times C_d \times \rho_{atm} \times v_x^2 \times A_F \quad (\text{eq. B.14})$$

where:

C_d – drag coefficient for launched body – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D=24$ is used for this simulation (see C_d Part M).

ρ_{atm} – density of local atmospheric air the body is traveling though – air density is a function of altitude – See Part I.

v_x – velocity of launched body in x-direction – See Part D.

A_F – frontal area of the launched body – see Part A.

Figure B-5: FBD for 2-D Launch Body

The drag force acting in the y-direction, F_{Dy} (N), on the launched body for all spreadsheets except Earth_Sim_RevD are calculated as:

$$F_{Dy} = (0.5 \times C_d \times \rho_{atm} \times v_y^2 \times A_F) \times (1 - C_{slip-stream} / 100) \quad (\text{eq. B.15})$$

where:

C_d – drag coefficient for launched body – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D=24$ is used for this simulation (see C_d Part M).

ρ_{atm} – density of local atmospheric air the body is traveling though – air density is a function of altitude – See Part I.

V_y – velocity of launched body in y-direction – See Part D.

A_F – frontal area of the launched body – see Part A.

The drag force acting in the x-direction, F_{Dx} (N), on the launched body for Earth_Sim_RevD is calculated as:

$$F_{Dx} = (0.5 \times C_d \times \rho_{atm} \times v_x^2 \times A_F) \times (1 - C_{slip-stream} / 100) \quad (\text{eq. B.16})$$

where:

C_d – drag coefficient for launched body – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D = 24$ is used for this simulation (see C_d Part M).

ρ_{atm} – density of local atmospheric air the body is traveling through – air density is a function of altitude – See Part I.

v_x – velocity of launched body in x-direction – See Part D.

A_F – frontal area of the launched body – see Part A.

$C_{slip-stream}$ – Factor inputted by user as the percent slip-stream reduction

The drag force acting in the y-direction, F_{Dy} (N), on the launched body for Earth_Sim_RevD is calculated as:

$$F_{Dy} = (0.5 \times C_d \times \rho_{atm} \times v_y^2 \times A_F) \times (1 - C_{slip-stream} / 100) \quad (\text{eq. B.17})$$

where:

C_d – drag coefficient for launched body – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D = 24$ is used for this simulation (see C_d Part M).

ρ_{atm} – density of local atmospheric air the body is traveling through – air density is a function of altitude – See Part I.

V_y – velocity of launched body in y-direction – See Part D.

A_F – frontal area of the launched body – see Part A.

$C_{slip-stream}$ – Factor inputted by user as the percent slip-stream reduction

The calculation method for Earth_Sim_RevD is a simplistic means of evaluating a complex gas dynamic process occurring in the launch corridor. A slip-stream reduction factor of one hundred percent drag reduction is equivalent to the ideal case where the launch corridor conditions are a perfect vacuum. A slip-stream reduction factor of zero percent is equivalent to the case where the launch corridor is the full density of the atmosphere at each region. These two cases represent the range of the expected drag effect, therefore the actual drag reduction effect exists at some point in this range.

Part O - Resultant X and Y Forces Acting on the Material package, F_{Rx} & F_{Ry}

The resultant X and Y forces acting on the material package, F_{Rx} & F_{Ry} (N), can be calculated as follows:

$$F_{Rx} = F_{GEx} + F_{GMx} + F_{Dx} \quad (\text{eq. B.18})$$

$$F_{Ry} = F_{GEy} + F_{GMy} + F_{Dy} \quad (\text{eq. B.19})$$

where:

F_{GEx} – Earth gravity force in X direction (N)

F_{GMx} – Moon gravity force in X direction (N)
 F_{Dx} – Atmospheric drag force in X direction (N)
 F_{GEy} – Earth gravity force in Y direction (N)
 F_{GM_y} – Moon gravity force in Y direction (N)
 F_{Dy} – Atmospheric drag force in Y direction (N)

Part P - Average Acceleration in the X and Y Directions, a_{AVGx} & a_{AVGy}

The average acceleration in the X and Y directions, a_{AVGx} & a_{AVGy} (m/s²), can be calculated as follows [32]:

$$a_{AVGx} = F_{Rx} / m_{package} \quad (\text{eq. B.20})$$

$$a_{AVGy} = F_{Ry} / m_{package} \quad (\text{eq. B.21})$$

where:

F_{Rx} – Resultant force in the X direction (N)

$m_{package}$ – Mass of material package (kg)

F_{Ry} – Resultant force in the Y direction (N)

Part Q - Resultant Average Acceleration, a_{AVG}

The resultant average acceleration, a_{AVG} (m/s²), can be calculated as follows:

$$a_{AVG} = [a_{AVGx}^2 + a_{AVGy}^2]^{0.5} \quad (\text{eq. B.22})$$

where:

a_{AVGx} – Average acceleration in the X direction (m/s²)

a_{AVGy} – Average acceleration in the Y direction (m/s²)

Part R – Subsequent Time Step Length

The length of the subsequent time step, t_{step} (s), is based on the resultant average acceleration, user defined time step multiplier and user defined maximum time step length. Basing the time step duration on the average acceleration was done to shorten the time step during the launch phase of flight to when the forces are changing quickly on the material package. The shorter time step enhances the accuracy of the simulation.

The length of the subsequent time step is calculated as follows:

$$\begin{aligned}
 &\text{if } (1 / a_{AVG}^2 \times t_{step \text{ Multiplier}}) < t_{step} \\
 &\text{then } t_{step} = (1 / a_{AVG}^2 \times t_{step \text{ Multiplier}}) \\
 &\text{else } t_{step}
 \end{aligned} \quad (\text{eq. B.23})$$

where:

a_{AVG} – Resultant average acceleration (m/s²)

$t_{step \text{ Multiplier}}$ – Time step multiplier value (user defined) – effects the length of the time

steps. It is used to provide short time steps during launch. i.e. larger $t_{\text{step Multiplier}}$ value equals a larger time step

t_{step} – Maximum length of time step (s) - (user defined – controls length of time step when the resultant average acceleration value becomes very small).

Part S - Altitude and X and Y Position of the Material package for a Subsequent Time Step

The X and Y position, $X_{(t+1)}$ and $Y_{(t+1)}$ (m), of the material package can be calculated for a subsequent time step as follows:

$$X_{(t+1)} = X_{(t)} + [v_{x(t)} \times t_{\text{step}}] + [0.5 \times a_{\text{AVGx}} \times (t_{\text{step}})^2] \quad (\text{eq. B.24})$$

$$Y_{(t+1)} = Y_{(t)} + [v_{y(t)} \times t_{\text{step}}] + [0.5 \times a_{\text{AVGy}} \times (t_{\text{step}})^2] \quad (\text{eq. B.25})$$

where:

$X_{(t+1)}$ – X position of the material package for the next time step (m)

$X_{(t)}$ – X position of the material package in the present time step (m)

$Y_{(t+1)}$ – Y position of the material package for the next time step (m)

$Y_{(t)}$ – Y position of the material package in the present time step (m)

$v_{x(t)}$ – Velocity of the material package in X direction for the present time step (m/s)

$v_{y(t)}$ – Velocity of the material package in Y direction for the present time step (m/s)

a_{AVGx} – Average acceleration in the X direction during the present time step (m/s²)

a_{AVGy} – Average acceleration in the Y direction during the present time step (m/s²)

t_{step} – length of time of the time step (s)

The altitude, $A_{(t+1)}$ (m), of the material package can be calculated as follows:

$$A_{(t+1)} = [X_{(t+1)}^2 + Y_{(t+1)}^2]^{0.5} - R_{\text{Earth}} \quad (\text{eq. B.26})$$

where:

$X_{(t+1)}$ – X position of the material package for the next time step (m)

$Y_{(t+1)}$ – Y position of the material package for the next time step (m)

R_{Earth} – Radius of Earth (m)

Part T - X and Y Components of Velocity, v_x and v_y , for a Subsequent Time Step

The X and Y components of velocity, v_x and v_y (m/s), for a subsequent time step can be calculated as follows:

$$v_{x(t+1)} = v_{x(t)} + [a_{\text{AVGx}} \times t_{\text{step}}] \quad (\text{eq. B.27})$$

$$v_{y(t+1)} = v_{y(t)} + [a_{\text{AVGy}} \times t_{\text{step}}] \quad (\text{eq. B.28})$$

where:

$v_{x(t+1)}$ – Velocity of the material package in the X direction for the next time step (m/s)

$v_{x(t)}$ – Velocity of the material package in the X direction for the present time step (m/s)

$v_{y(t+1)}$ – Velocity of the material package in the Y direction for the next time step (m/s)

$v_{y(t)}$ – Velocity of the material package in the Y direction for the present time step (m/s)

a_{AVGx} – Average acceleration in the X direction during the present time step (m/s²)

a_{AVGy} – Average acceleration in the Y direction during the present time step (m/s²)

t_{step} – Length of time of the time step (s).

Part U – Rocket Motor Mass, m_{rocket} , and Payload Mass Fraction, $m.f_{\text{payload}}$, Calculations
(Not calculated in the *Earth_Sim.xls* spreadsheet).

To calculate the mass of the rocket motor required, the x and y components of the rocket motor velocity addition, Δv_x and Δv_y , specified by the user, are calculated as,

$$\Delta v_x = \Delta v \sin (90 - \alpha) \quad (\text{eq. B.29})$$

$$\Delta v_y = \Delta v \cos (90 - \alpha) \quad (\text{eq. B.30})$$

where:

Δv_x – Velocity addition of rocket motor in the x-direction at apogee (m/s)

Δv_y – Velocity addition of rocket motor in the y-direction at apogee (m/s)

Δv – Velocity addition of rocket motor at apogee specified by user (m/s)

α - Angle from the vertical to the Earth gravity force (°) – see Figure B-6

Figure B-6: FBD for Rocket Motor Burn

The rocket motor thrust forces in the x and y-directions, $F_{\text{thrust}_x(\text{apogee})}$ and $F_{\text{thrust}_y(\text{apogee})}$, are calculated as follows,

$$F_{\text{thrust}_x(\text{apogee})} = (m_{\text{package}} \times \Delta v_x) \div t_{\text{step\_apogee}} \quad (\text{eq. B.31})$$

$$F_{\text{thrust}_y(\text{apogee})} = (m_{\text{package}} \times \Delta v_y) \div t_{\text{step\_apogee}} \quad (\text{eq. B.32})$$

where:

$F_{\text{thrust}_x(\text{apogee})}$ – Thrust force of the rocket motor in the x-direction at apogee (N)

$F_{\text{thrust}_y(\text{apogee})}$ – Thrust force of the rocket motor in the y-direction at apogee (N)

m_{package} – Mass of material package (kg)

Δv_x – Velocity addition of rocket motor in the x-direction at apogee (m/s)

Δv_y – Velocity addition of rocket motor in the y-direction at apogee (m/s)

$t_{\text{step\_apogee}}$ – Length of time of the time step at apogee (s)

The total impulse, I_{total} , generated by the rocket motor is calculated as follows,

$$I_{\text{Total}} = (F_{\text{thrust}_x(\text{apogee})}^2 + F_{\text{thrust}_y(\text{apogee})}^2)^{0.5} \times t_{\text{step\_apogee}} \quad (\text{eq. B.33})$$

where:

I_{Total} – Total impulse of the rocket motor burn (N s)

$F_{\text{thrust}_x(\text{apogee})}$ – Thrust force of the rocket motor in the x-direction at apogee (N)

$F_{\text{thrust}_y(\text{apogee})}$ – Thrust force of the rocket motor in the y-direction at apogee (N)

$t_{\text{step\_apogee}}$ – Length of time of the time step at apogee (s)

Finally the required mass of the rocket motor, m_{rocket} , to produce the velocity addition, Δv , specified by the user is calculated as follows [39],

$$m_{\text{rocket}} = \frac{I_{\text{Total}}}{m.f._{\text{prop}} \times I_{\text{sp}} \times g} \quad (\text{eq. B.34})$$

where:

m_{rocket} – Mass of the rocket motor required to produce the velocity addition (Δv) (kg)

I_{Total} – Total impulse of the rocket motor burn (N s)

$m.f._{\text{prop}}$ – Propellant mass fraction for the rocket motor (specified by user – used 0.93 [39])

I_{sp} – Specific Impulse performance parameter of rocket motor (s) (specified by user – used 290s average from [39]).

g – gravitational constant -9.807 m/s^2

The payload mass fraction, $m.f._{\text{payload}}$, for the material package is calculated as follows,

$$m.f._{\text{payload}} = (m_{\text{package}} - m_{\text{rocket}}) \div m_{\text{package}}$$

where:

$m.f._{\text{payload}}$ – Mass fraction of the payload for the material package.

m_{package} – Mass of material package (kg)

m_{rocket} – Mass of the rocket motor required to produce the velocity addition (Δv) (kg)

Appendix C - Material Package Orbit.xls Calculations

Figure C-1 depicts the typical output table of the Material Package Orbit.xls spreadsheet.

Figure C-1: Typical Output for Material_Package_Orbit.xls Spreadsheet

The Material Package Orbit.xls spreadsheet requires the following input data:

Semi-Major Axis (a) in km – length of the elliptical orbit (see Figure C-2)

Semi-Minor Axis (b) in km – width of the elliptical orbit (see Figure C-2)

Increment (degree) – selects the calculation interval in the table

Standard Gravitational Parameter (μ) – constant value for Earth ($3.9886 \times 10^{14} \text{ m}^3/\text{s}^2$)

The spreadsheet uses the input data to calculate the following parameters [3]:

Orbit Eccentricity (e),

$$e = \left(1 - \frac{b^2}{a^2}\right)^{1/2} \quad (\text{eq. C.1})$$

Semi-Latus Rectum (P),

$$P = \left(\frac{a}{2} [1 - e^2]\right) \quad (\text{eq. C.2})$$

The input and calculated parameters are used to calculate the position and velocity of the material packages at the angular interval specified using the following equations:

The radius (r) of the ellipse measured from the focus on Earth,

$$r = \frac{\frac{a}{2}(1 - e^2)}{1 + e \cos(\phi)} \quad (\text{eq. C.3})$$

where ϕ is the angle measured counterclockwise from the perigee, see Figure C-2

The x-position (x) measured from the focus on Earth,

$$x = r \cos(\phi) \quad (\text{eq. C.4})$$

The y-position (y) measured from the focus on Earth,

Figure C-2: Elliptical Orbit Parameters

$$y = r\sin(\phi) \quad (\text{eq. C.5})$$

The x-velocity component was calculated as follows,

$$v_x = \frac{u^{1/2}}{P} [(\cos(\phi) + e) \times -\sin(\phi)] \quad (\text{eq. C.6})$$

The y-velocity component was calculated as follows,

$$v_y = \frac{u^{1/2}}{P} [(\cos(\phi) + e)] \quad (\text{eq. C.7})$$

Appendix D - Sample Calculation for Second Rocket Motor Mass for Scenario #4

(similar calculation was used for scenario # 5)

For the last material package in material stream (10-minute duration) for x-component,

From simulation data,

Initial x-velocity of Last Material Package, $v_{ox} = 6724.23 \text{ m/s}$

Delta x-distance for Last Material Package $\Delta x = 2238.49 \text{ km}$ (to catcher satellite at $t=x+10$)

Duration of rocket motor burn, 10 sec

The average acceleration, a_x , is [32],

$$\Delta x = v_{ox} t + 0.5 a_x t^2$$

$$a_x = (\Delta x - v_{ox} t) \div 0.5 t^2$$

$$a_x = (2238.49 \text{ km} - 6724.23 \text{ m/s} \times 10\text{s}) \div 0.5 (10 \text{ s})^2$$

$$a_x = 43\,424.954 \text{ m/s}^2$$

For the last material package in material stream (10-minute duration) for y-component,

From simulation data,

Initial y-velocity of Last Material Package, $v_{oy} = -5447.24 \text{ m/s}$

Delta y-distance for Last Material Package $\Delta y = 1369.24 \text{ km}$ (to catcher satellite at $t=x+10$)

Duration of rocket motor burn, 10 sec

The average acceleration, a_y , is [32],

$$\Delta y = v_{oy} t + 0.5 a_y t^2$$

$$a_y = (\Delta y - v_{oy} t) \div 0.5 t^2$$

$$a_y = (1369.24 \text{ km} - (-5447.24 \text{ m/s}) \times 10\text{s}) \div 0.5 (10 \text{ s})^2$$

$$a_y = 28\,474.25 \text{ m/s}^2$$

The rocket forces are,

$$F_x = m a_x = 0.6 \text{ kg} \times 43\,424.954 \text{ m/s}^2 = 26\,055 \text{ N}$$

$$F_y = m a_y = 0.6 \text{ kg} \times 28\,474.25 \text{ m/s}^2 = 17\,085 \text{ N}$$

Where m is the mass of the last material package after the first rocket motor burn has been completed.

The impulses are,

$$I_x = F_x t = 26\,055 \text{ N} \times 10 \text{ sec} = 260\,550 \text{ Ns}$$

$$I_y = F_y t = 17\,085 \text{ N} \times 10 \text{ sec} = 170\,850 \text{ Ns}$$

Mass of Second Rocket Motor [39],

$$m_{2\text{nd rocket}} = (I_x + I_y) \div (\text{Propellant Mass Fraction} \times \text{Specific Impulse} \times g)$$

$$m_{2\text{nd rocket}} = (260\,550 \text{ Ns} + 170\,850 \text{ Ns}) \div (0.93 \times 290 \text{ s} \times 9.807 \text{ m/s}^2)$$

$$m_{2\text{nd rocket}} = 163.1 \text{ kg}$$

Appendix E - Sample Calculation for Second Rocket Motor Mass for Scenario #6

For first material package in material stream at perigee intersection

From simulation data

Velocity Difference – x-component, $\Delta v_x = 7.527 \text{ m/s}$

Velocity Difference - y-component, $\Delta v_y = 16.457 \text{ m/s}$

Duration of rocket motor burn, $t = 1 \text{ sec}$

The average acceleration, a_x and a_y , is [32],

$$a_x = \Delta v_x \div t = 7.527 \text{ m/s} \div 1 \text{ sec}$$

$$a_x = 7.527 \text{ m/s}^2$$

$$a_y = \Delta v_y \div t = 16.457 \text{ m/s} \div 1 \text{ sec}$$

$$a_y = 16.457 \text{ m/s}^2$$

The rocket forces are,

$$F_x = m a_x = 0.6012 \text{ kg} \times 7.527 \text{ m/s}^2 = 4.5252 \text{ N}$$

$$F_y = m a_y = 0.5844 \text{ kg} \times 16.457 \text{ m/s}^2 = 9.8939 \text{ N}$$

Where m is the mass of the last material package after the first rocket motor burn has been completed.

The impulses are,

$$I_x = F_x t = 4.5252 \text{ N} \times 1 \text{ sec} = 4.5252 \text{ Ns}$$

$$I_y = F_y t = 9.8939 \text{ N} \times 1 \text{ sec} = 9.8939 \text{ Ns}$$

Mass of Second Rocket Motor [39],

$$m_{2\text{nd rocket}} = (I_x + I_y) \div (\text{Propellant Mass Fraction} \times \text{Specific Impulse} \times g)$$

$$m_{2\text{nd rocket}} = (4.5252 \text{ Ns} + 9.8939 \text{ Ns}) \div (0.93 \times 290 \text{ s} \times 9.807 \text{ m/s}^2)$$

$$m_{2\text{nd rocket}} = 5.4516 \text{ g}$$

Appendix F - Explanation for the Simulation of the Rocket Launch Rail Launcher_Simulator.xls

Purpose: To explain the calculations used for the Rocket Launch Rail simulation spread sheet. The simulation is used to calculate the velocity and acceleration of three different launch rails; each launch rail has a different propulsion method. The first launch rail has a rocket sled carrying the fuel and oxidizer (a conventional rocket motor – see Figure F-1). The second rocket sled carries the fuel only and the oxidizer gas is collected from the launch rail (distributed oxidizer launch rail – see Figure F-2). The third rocket sled carries no propellant, both fuel and propellant is collected from the launch rail (S. Molder’s distributed propellant launch rail – see Figure F-3 [17]).

Figure F-1: Conventional Rocket Sled Launch Rail Schematic (Hybrid Rocket Shown – Solid or Liquid rocket motors possible)

Figure F-2: Distributed Oxidizer Launch Rail Schematic

Figure F-3: Distributed Propellant Launch Rail Schematic [17]

Assumptions

1. The launch motion is assumed to be along a straight line.
2. Gravity forces are ignored.
3. Friction forces with the rail and the sled are ignored.
4. No thermal heating effects are considered.
5. The simulation model assumed constant acceleration during each time step.
6. The rocket sled is assumed to have the shape of a cylinder.
7. Experimental total drag coefficient for a hypervelocity missile as a function of Mach number was used [31]. This drag co-efficient assumed a fin stabilised missile with a L/D ratio of 24.
8. The lower atmospheric density was assumed to be defined by a density function defined by data from the ICAO (International Civil Aviation Organisation) Standard Atmosphere [33].
9. The upper atmosphere was assumed to follow a density function defined by the 1976 US Standard Atmosphere [34].
10. The atmospheric temperature was defined by functions for each temperature region depicted in Figure 2. [33]
11. All temperature functions in temperature regions were assumed to be linear.

Spreadsheet Simulation Operations

The spreadsheet simulation performs the following steps:

1. The user inputs the desired time step multiplier, maximum time step, initial velocity, percent air reduction (vacuum), launch rail angle, rocket sled mass, rocket sled diameter, rocket thrust, rocket motor specific impulse, mixture ratio (of propellants), mass of fuel, estimated density of rocket sled, characteristic velocity, combustion chamber pressure and combustion efficiency. The user may also change the constant properties for air.
2. The frontal area, volume, length and L/D (length to diameter) ratio of the rocket sled is calculated. See Section A for the calculations.

NOTE: It is important that the L/D ratio close to 24, since the drag coefficient used by the spreadsheet is based on this value (see Section F). L/D ratio is adjusted by varying either the diameter of the rocket sled or the density.

First Time Step

3. At time = 0s, the initial X and Y co-ordinates, X_i & Y_i , are both set to zero (start position) in the simulation table. The user inputted initial launch velocity is entered into the simulation table and the initial X and Y velocity components, v_{Intx} & v_{Inty} are calculated. See Section B for calculations.
4. The Y-position value for the time step is used to calculate the density of air, ρ_{atm} , according to the density functions given in Section C.
5. The Y-position value for the time step is used to calculate the temperature of air, T_{atm} , according to the temperature functions given in Section D.
6. The speed of sound, a_{ss} , at the Y-position value of the time step is calculated according to the equation in Section E. The speed of sound is required to calculate the Mach number of the pellet.
7. The Mach number, Ma , of the rocket sled for the time step is calculated according to the equation in Section F. The Mach number is required to determine the drag co-efficient, C_d .
8. The drag coefficient, C_d , for the rocket sled, with a given Mach number, is determined from experimental data detailed in Section G.
9. The atmospheric drag (F_{Dx} & F_{Dy}) are for the pellet is calculated according to equations in Section H.
10. The rocket motor thrust force, F_{rocket} , inputted by the user, is entered into the simulation table and the X and Y force components, $F_{rocketx}$ & $F_{rockety}$, are calculated. See Section I for calculations.
11. The mass of fuel, m_{fuel} , inputted by the user, is entered into the simulation table and the mass of the oxidizer, m_{ox} , is calculated. See Section J for calculations.

12. The **X and Y resultant forces**, F_{Rx} & F_{Ry} , acting on the rocket sled are calculated according to equations in Section K.
13. The **X and Y average acceleration**, a_{AVGx} & a_{AVGy} , experienced by the rocket sled during the time step, is calculated according to the equations in Section L.
14. The **resultant average acceleration**, a_{AVG} , is calculated. See Section M.
15. The magnitude of the resultant average acceleration is used to determine the length of the **subsequent time step**, t_{step} , according to the calculation in Section N.

Subsequent Time Steps

16. The **X and Y position**, X & Y , of the rocket sled for the subsequent time step is calculated using the average acceleration according to equations in Section O.
17. The **X and Y components of velocity**, v_x and v_y , of the rocket sled for the subsequent time step is calculated according to equations in Section P.
18. The Y-position value for the time step is used to calculate the **density of air**, ρ_{atm} , according to the density functions given in Section C.
19. The Y-position value for the time step is used to calculate the **temperature of air**, T_{atm} , according to the temperature functions given in Section D.
20. The **speed of sound**, a_{ss} , at the Y-position value of the time step is calculated according to the equation in Section E. The speed of sound is required to calculate the Mach number of the pellet.
21. The **Mach number**, Ma , of the rocket sled for the time step is calculated according to the equation in Section F. The Mach number is required to determine the drag co-efficient, C_d .
22. The **drag coefficient**, C_d , for the rocket sled, with a given Mach number, is determined from experimental data detailed in Section G.
23. The **atmospheric drag** (F_{Dx} & F_{Dy}) are for the pellet is calculated according to equations in Section H.
24. The **rocket motor thrust force**, F_{rocket} , inputted by the user, is entered into the simulation table and the X and Y force components, $F_{rocketx}$ & $F_{rockety}$, are calculated. See Section I for calculations.
25. The **mass of propellant**, m_{prop} , burned during the time step is calculated and subtracted from the total propellant mass. The new mass of fuel and oxidizer, m_{fuel} and m_{ox} , is calculated. See Section Q for calculations.
26. If the oxidizer is collected from the launch rail, then the inertial force to accelerate the oxidizer gas to the speed of the rocket sled is calculated. See Section R for calculations.
27. The **X and Y resultant forces**, F_{Rx} & F_{Ry} , acting on the rocket sled are calculated according to equations in Section R.

28. The **X and Y average acceleration**, a_{AVGx} & a_{AVGy} , experienced by the rocket sled during the time step, is calculated according to the equations in Section L.
29. The **resultant average acceleration**, a_{AVG} , is calculated. See Section M.
30. The magnitude of the resultant average acceleration is used to determine the length of the **subsequent time step**, t_{step} , according to the calculation in Section N.
31. Steps 16 through 31 are repeated for the following time steps.

NOTE:

- The simulation table should be copy and pasted down to row 6500. The length of the simulation is adjusted by changing the time step multiplier and maximum time step values.
- Using the rocket propellant mass burn rate and user inputted rocket motor characteristics, estimated rocket nozzle throat and exit diameter are calculated. Theoretical exit diameter can be used to compare to the diameter of the rocket sled inputted by the user.

Section A – Calculated Rocket Sled Parameters

The following rocket sled parameters are calculated using the rocket sled mass (m_{sled}), diameter (D_{sled}) and material density (ρ_{sled}) entered by the user.

See Equation 3.16 to 3.19

Section B – Initial Position and Velocity of the Rocket Sled at Start of Launch

The initial launch position, X_{intx} and Y_{inty} in meters, of the rocket sled are set to zero:

$$X_{intx} = 0$$

$$Y_{inty} = 0$$

The X and Y components of the initial velocity of the rocket sled, v_{intx} & v_{inty} are calculated as follows:

$$v_{intx} = v_{int} \cos (\theta_{rail}) \quad \text{(eq. F.1)}$$

$$v_{inty} = v_{int} \sin (\theta_{rail}) \quad \text{(eq. F.2)}$$

where:

v_{int} – Initial velocity of the rocket sled inputted by the user in m/s.

θ_{rail} – Angle of the launch rail inputted by the user in degrees.

Section C – Air Density Functions

See Equations 3.24 to 3.28

Section D – Air Temperature Functions

See Equations 3.29 to 3.33

Section E – Speed of Sound

See Equations 3.34

Section F –Mach Number

See Equations 3.35

Section G – Drag Coefficient

Experimental data for the drag coefficient, C_d , vs. Mach number for a finned missile is depicted in Figure 3.7 [31]. The experimental data is valid for a body with an L/D of 24. This data is used for the drag co-efficient of the rocket sled.

The spreadsheet contains a table of C_d vs. Mach number values obtained from Figure 3.7. The spreadsheet uses the Mach number of the pellet to 'look up' the appropriate drag coefficient, C_d .

Section H - Atmospheric Forces Acting on the Launch Body

Figure F-4 is a free body diagram (FBD) that depicts the total atmospheric drag force, F_{Drag} , acting on the rocket sled.

Figure F- 4: Drag and Rocket Thrust Force on Rocket Sled

The drag force acting in the x-direction, F_{Dx} (N), on the rocket sled is calculated as [32]:

$$F_{Dx} = 0.5 \times C_d \times \rho_{atm} \times v_x^2 \times A_F \quad (\text{eq. F.3})$$

where:

C_d – drag coefficient for the rocket sled – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D = 24$ is used for this simulation (see C_d Section G).

ρ_{atm} – density of local atmospheric air the rocket sled is traveling through – air density is a function of altitude – See Section C.

v_x – velocity of the rocket sled in x-direction – See Section B.

A_F – frontal area of the rocket sled – see Section A.

The drag force acting in the y-direction, F_{Dy} (N), on the rocket sled can be calculated as:

$$F_{Dy} = 0.5 \times C_d \times \rho_{atm} \times v_y^2 \times A_F \quad (\text{eq. F.4})$$

where:

C_d – drag coefficient for the rocket sled – an experimentally obtained graph of C_d vs. Mach number for a finned missile with $L/D = 24$ is used for this simulation (see C_d Section G).

ρ_{atm} – density of local atmospheric air the rocket sled is traveling through – air density is a function of altitude – See Section C.

V_y – velocity of rocket sled in y-direction – See Section B.

A_F – frontal area of the rocket sled – see Section A.

Section I - Rocket Motor Force, $F_{rocket(x)}$ & $F_{rocket(y)}$

Figure F-6 is a free body diagram (FBD) that depicts the total rocket thrust force, F_{rocket} , acting on the rocket sled. Note: The total rocket thrust force, F_{rocket} , is specified by the user.

The rocket thrust force acting in the x-direction, $F_{rocket(x)}$ (N), on the rocket sled is calculated as:

$$F_{rocket(x)} = F_{rocket} \cos(\theta_{rail}) \quad (\text{eq. F.5})$$

where:

F_{rocket} – Rocket thrust force specified by the user (N).

θ_{rail} – Angle of the launch rail inputted by the user in degrees.

The rocket thrust force acting in the y-direction, $F_{rocket(y)}$ (N), on the rocket sled is calculated as:

$$F_{rocket(y)} = F_{rocket} \sin(\theta_{rail}) \quad (\text{eq. F.6})$$

where:

F_{rocket} – Rocket thrust force specified by the user (N).

θ_{rail} – Angle of the launch rail inputted by the user in degrees.

Section J - Initial Rocket Motor Fuel Mass, $m_{\text{fuel\_int}}$ and Oxidizer Mass, $m_{\text{oxidizer\_int}}$

The initial rocket motor fuel mass, $m_{\text{fuel\_int}}$, is specified by the user.

The initial rocket motor oxidizer mass, $m_{\text{oxidizer\_int}}$, is calculated as follows [39]:

$$m_{\text{oxidizer\_int}} = m_{\text{fuel\_int}} \times r_{\text{mix}} \quad (\text{eq. F.7})$$

where:

$m_{\text{fuel\_int}}$ – initial rocket fuel mass specified by the user.

r_{mix} – fuel to oxidizer mass ratio specified by the user.

Section K - Resultant X and Y Forces Acting on the Rocket Sled, F_{Rx} & F_{Ry}

The resultant X and Y forces acting on the rocket sled, F_{Rx} & F_{Ry} (N), can be calculated as follows for the three different launch cases:

For the rocket sled carrying fuel and oxidizer (conventional rocket motor),

$$F_{\text{Rx}} = F_{\text{Dx}} + F_{\text{rocket}(x)} \quad (\text{eq. F.8})$$

$$F_{\text{Ry}} = F_{\text{Dy}} + F_{\text{rocket}(y)} \quad (\text{eq. F.9})$$

For the rocket sled carrying fuel and obtaining oxidizer from the rail (distributed oxidizer launch rail),

$$F_{\text{Rx}} = F_{\text{Dx}} + F_{\text{rocket}(x)} \quad (\text{eq. F.10})$$

$$F_{\text{Ry}} = F_{\text{Dy}} + F_{\text{rocket}(y)} \quad (\text{eq. F.11})$$

For the rocket sled obtaining both fuel and oxidizer from the rail (distributed propellant launch rail),

$$F_{\text{Rx}} = F_{\text{Dx}} + F_{\text{rocket}(x)} \quad (\text{eq. F.12})$$

$$F_{\text{Ry}} = F_{\text{Dy}} + F_{\text{rocket}(y)} \quad (\text{eq. F.13})$$

where:

F_{Dx} – Atmospheric drag force in X direction (N)

$F_{\text{rocket}(x)}$ – Rocket thrust force in X direction (N)

F_{Dy} – Atmospheric drag force in Y direction (N)

$F_{\text{rocket}(y)}$ – Rocket thrust force in Y direction (N)

Section L - Average Acceleration in the X and Y Directions, a_{AVGx} & a_{AVGy}

The average acceleration in the X and Y directions, a_{AVGx} & a_{AVGy} (m/s^2), can be calculated as follows for the three different launch rail cases:

For the rocket sled carrying fuel and oxidizer (conventional rocket motor),

$$a_{\text{AVGx}} = F_{\text{Rx}} / (m_{\text{sled}} + m_{\text{fuel}} + m_{\text{oxidizer}}) \quad (\text{eq. F.14})$$

$$a_{\text{AVGy}} = F_{\text{Ry}} / (m_{\text{sled}} + m_{\text{fuel}} + m_{\text{oxidizer}}) \quad (\text{eq. F.15})$$

For the rocket sled carrying fuel and obtaining oxidizer from the rail (distributed oxidizer launch rail),

$$a_{AVGx} = F_{Rx} / (m_{sled} + m_{fuel}) \quad (\text{eq. F.16})$$

$$a_{AVGy} = F_{Ry} / (m_{sled} + m_{fuel}) \quad (\text{eq. F.17})$$

For the rocket sled obtaining both fuel and oxidizer from the rail (distributed propellant launch rail),

$$a_{AVGx} = F_{Rx} / (m_{sled}) \quad (\text{eq. F.18})$$

$$a_{AVGy} = F_{Ry} / (m_{sled}) \quad (\text{eq. F.19})$$

where:

F_{Rx} – Resultant force in the X direction (N)

F_{Ry} – Resultant force in the Y direction (N)

m_{sled} – Mass of rocket nozzle (kg)

m_{fuel} – Mass of rocket fuel (kg)

$m_{oxidizer}$ – Mass of oxidizer for rocket motor (kg)

Note: for the first time step the initial fuel and oxidizer masses, m_{fuel_int} and $m_{oxidizer\_int}$ are used.

Section M - Resultant Average Acceleration, a_{AVG}

See Equation B.22

Section N – Subsequent Time Step Length

See Equation B.23

Section O - X and Y Position of the Rocket Sled for a Subsequent Time Step

See Equation B.24 and B.25

Section P - X and Y Components of Velocity, v_x and v_y , for a Subsequent Time Step

See Equation B.27 and B.28

Section Q – Mass of Propellant Consumed During Each Time Step

The mass of propellant consumed during each time step is calculated as follows [39]:

$$m_{prop(t+1)} = m_{prop(t)} - [(F_{rocket} \times t_{step}) \div (I_{sp} \times g)] \quad (\text{eq. F.20})$$

where:

$m_{prop(t+1)}$ – mass of propellant (fuel and oxidizer mass) in next time step (kg).

$m_{prop(t)}$ – mass of propellant (fuel and oxidizer mass) in the present time step (kg).

F_{rocket} – Rocket thrust force specified by the user (N).

t_{step} – length of time of the time step (s).

I_{sp} – Specific impulse of rocket motor specified by user (s).

g – Earth gravity constant, 9.807 m/s^2 .

The fuel mass for the subsequent time step, $m_{fuel(t+1)}$, can be calculated as:

$$m_{fuel(t+1)} = m_{prop(t+1)} \div (r_{mix} + 1) \quad (\text{eq. F.21})$$

where:

$m_{prop(t+1)}$ – mass of propellant (fuel and oxidizer mass) in the next time step (kg).

r_{mix} – fuel to oxidizer mass ratio specified by the user.

The oxidizer mass for the subsequent time step, $m_{oxidizer(t+1)}$, can be calculated as:

$$m_{oxidizer(t+1)} = m_{prop(t+1)} \times r_{mix} \quad (\text{eq. F.22})$$

where:

$m_{prop(t+1)}$ – mass of propellant (fuel and oxidizer mass) in the next time step (kg).

r_{mix} – fuel to oxidizer mass ratio specified by the user.

Appendix G – Internet Postings

The following message was posted by the author on the Space Studies Institute (SSI) newsgroup in March 2004.

Message: 3

Date: Tue, 16 Mar 2004 04:08:01 -0000

From: "dean_gerl" <dgerl@ryerson.ca>

Subject: The Mass Driver Project

Hello,

This is my first time posting on a message board, so forgive me if I violate any rules (and tell me what i have done wrong). I am trying to organize a new open source project, the Mass Driver Project. www.massdriver.com - Use, login: guest password: guest, to review the site.

Open source is an internet collaboration method that has successfully developed software like Linux Operating System. These open source programs are developed by people sharing their time and talents over the internet.

The www.massdriver.com has been setup as a space on the web where interested people may exchange ideas, share and post knowledge, identify and work through technical problems and eventually produce a successful mass driver concept.

Open source projects are successful because it allows many different people to work together towards a common goal. The goal is set and modified by the people participating in the project. Large projects like mass driver technology cannot be developed by one individual successfully; however connecting many individuals together can achieve such large tasks.

If you have a moment, please review the website. (www.massdriver.com login: guest password: guest) - this gives read access.

Click on the "Wiki" tab, OpenEngineeringProjects and MassDriverProject. Feel yourself around in the rest of the site.

If you are interested in joining, please contact me.

This project is purely an academic exercise sponsored by Ryerson University in Toronto Canada. I am a graduate student at Ryerson University. All the information obtained is public information.

The web site is young, and not so pretty, so please bear with us.

Thank you in advance,

Dean Gerl

The following message was the second message posted to the SSI newsgroup in May 2005.

Date: Sun, 01 May 2005 03:15:24 -0000
From: "deangerl" <deangerl@hotmail.com>
Subject: Launch Rail Concept

Hello,

I put a few web pages summarizing my work to date for a graduate thesis on a material transport system from Earth surface to orbit. Material is sent to orbit as a series of small masses following on after another (a material package stream, if you will). The system overview can be found at www.massdriver.com/massdriver.html.

The launcher system proposed I think is unique. Instead of using an electromagnetic system, which has technical problems with energy storage and distribution, the technology is based on a novel chemical rocket sled launch rail. (I don't think the concept has been publicly proposed before.)

The proposed rocket sled traverses a launch rail carrying its launch payload and half a hybrid rocket motor that is the nozzle and the solid fuel grain. As the rocket sled traverses the launch rail, it collects an oxidizer gas dispensed from the launch rail. A schematic of the concept can be found at www.massdriver.com/distributed_oxidizer_launch_rail.htm.

The performance of the rocket sled is improved since the oxidizer mass (and storage structure) does not need to be stored onboard and accelerated.

Your feed back on the idea would be helpful,

Dean Gerl

The following question was posted to the Mad Scientist Network in May 2004.

Question: Is there a significant hypersonic slip-stream effect?

Mass drivers have been proposed for transporting lunar material. A mass driver operating on Earth must overcome the larger gravity well plus the atmospheric drag.

We are proposing that drag could be minimized by rapidly launching a stream of material pellets. It is intended that the pellets would slip-stream each other, sort of forming a tube through the atmosphere. We are looking for feedback on the idea or references for hypersonic slip-stream studies.

I am a mechanical engineer studying design theory, open source software development applied to the design of physical products. This idea is one of the proposed projects to develop using open source principles. In short, evaluating this pellet stream is not my thesis.

Thank you for any feedback you can provide.

Dean Gerl

The following message was posted to the Eng Tips Aeronautic Engineering forum in May 2004.

3 May 04 14:00

Is there an aerodynamic advantage to launching a stream (series) of small hypervelocity cone-like projectiles (say 1 kg total mass) from Earth surface to Earth orbit vs. one large (1000 kg) mass? Machine gun vs. cannon ball scenario.

The idea is that the initial projectiles will reduce the drag for the following projectiles. It is envisioned that this would sort of form a tunnel up through the atmosphere.

Any comments on this situation would be helpful; this idea is from a few mechanical engineers, so forgive us if we are violating something obvious.

Dean Gerl

The following message was posted to the Eng Tips Spacecraft Engineering forum (Aerospace discipline) in May 2005.

3 May 05 22:09

I am a mechanical engineer working on the system design of a material transport system for space construction "i.e. an Earth based mass driver".

Instead of the electromagnetic launcher traditionally proposed I have proposed a rocket sled that traverses the launch rail and collects an oxidizer gas dispensed from the launch rail. The oxidizer gas is combusted in a solid fuel grain similar to a hybrid rocket motor, onboard the rocket sled. Therefore the performance of the rocket sled is improved because the oxidizer gas and storage structure is not accelerated with the rocket sled.

A schematic of the proposed system can be viewed at http://www.massdriver.com/distributed_oxidizer_launch_rail.htm.

My question is what is the typical pressure oxidizer gas is stored on a hybrid rocket?

Also I am not an aerospace expert, more of an abstract mechanical thinker so is there any significant technical problems with this launch rail concept?

I know rail friction is a real problem which might be solved by some form of MagLev technology, like Inductrak.

This space system design is an attempt at a collaborative (or Open Source) design of a hardware product. All information related to the project is public knowledge.

Thanks,
Dean Gerl

REFERENCES

- [1] NASA; "US Launch Vehicle Data", NASA Cost Estimating Web Site;
http://www.jsc.nasa.gov/bu2/ELV_US.html ; Accessed on July 12, 2004.
- [2] Palmer, M. R., Lenard, R.X.; "A revolution in access to space through spinoffs of SDI technology", IEEE Transactions on Magnetics, Vol.27, No. 1, pgs. 11-20, January 1991.
- [3] Logsdon, T.; Orbital Mechanics - Theory and Applications, John Wiley & Sons Inc., New York, 1998.
- [4] Feller, J., Fitzgerald, B.; Understanding Open Source Software Development; Addison-Wesley, London, 2002.
- [5] News.com; "Image: Rock 'n' roll heaven for XM"; News.com;
http://news.com.com/Image+Rock+n+roll+heaven+for+XM/2009-1027_3-5551064.html, Accessed on December 5, 2003. Original image downloaded from site and modified by D. Gerl.
- [6] Linux.org; "What is Linux?"; Linux.org, <http://www.linux.org/info/index.html>; Accessed on June 1, 2005.
- [7] theOSCARProject.org; "News", theOSCARProject.org; <http://www.theoscarproject.org/de/> ; Accessed on March 13, 2004.
- [8] Blankman, E. *et al.*; "OSCAR – The Open Source Car Project", Stern School of Business, New York University, 2000.
http://pages.stern.nyu.edu/~sjournal/articles_02/os_car_project.pdf ; Accessed on March 15, 2004.
- [9] OpenCores.org; "Frequently Asked Questions", OpenCores.org;
<http://www.opencores.org/faq.cgi/Section/2/2.1#2.1> ; Accessed on March 13, 2004.
- [10] ThinkCycle.org; "About ThinkCycle", Thinkcycle.org; <http://www.thinkcycle.org/about> ; Accessed on April 23, 2004.
- [11] Bishop, F.; "South Pole Accelerator"; Institute of Atomic-Scale Engineering, unpublished work at <http://www.iasc.cc/accelerator.htm>; Accessed on February 13, 2004.
- [12] European Space Agency; "Electromagnetic railgun technology for the deployment of small sub-/orbital payloads", ESA Contract No. 13420, ESA, September 2003;
http://esamultimedia.esa.int/docs/gsp/completed/comp_i_03_N02.pdf ; Accessed on June 6, 2004.
- [13] McNab, I.R.; "Launch to Space with an Electromagnetic Railgun", IEEE Transactions on Magnetics, Vol. 29, No. 1, January 2003, pg. 295 - 304.
- [14] H. Gilreath, A. Driesman, W. Kroshl, M. White, H. Cartland, J. Hunter; "The feasibility of launching small satellites with a light gas gun", In *Proceedings of the 12th Annual ALAA/USU Conference on Small Satellites*, Logan, Utah, August 31 - September 3, 1998. - <http://www.smallsat.org/proceedings/12/ssc98/3/sscii6.pdf>

- [15] M. R. Palmer, A. E. Dabiri; "Electromagnetic space launch: a re-evaluation in light of current technology and launch needs and feasibility of a near term demonstration", IEEE Transactions on Magnetics, Vol.25, No. 1, pgs. 393-399, January 1989.
- [16] Rice, E.E., Miller, L.A., Earhart, R.W; "NASA Final Technical Report - Preliminary Feasibility Assessment for Earth-to-Space Electromagnetic (Railgun) Launcher", Contract Number NAS3-22882, Battelle – Columbus Laboratories, June 1982. Pg. 1 – 432. Accessed on September 9, 2004 at:
http://ntrs.nasa.gov/index.cgi?method=display&redirect=http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19820021469_1982021469.pdf&oiID=oi:casi.ntrs.nasa.gov:19820021469
- [17] Molder, S.; "The Distributed Propellant Launch Rail", unpublished work obtained from author.
- [18] Graf, R.; "Gun Launched", Encyclopedia Astronautica;
<http://www.astronautix.com/lvfam/gunnched.htm>, Accessed on May 22, 2005.
- [19] Powell J. R., Maise G., Paniagua J., Rather J.; "StarTram: a new approach low-cost Earth-to-orbit transport", IEEE 0-7803-6599-2/01, 2001. Pg. 2569 – 2590
- [20] Lofstrom, K.; "The launch loop", AIAA-85-1368, 21st Joint Propulsion Conference, 1985.
- [21] Spies J.; LV Hopper: consolidated system concept", Acta Astronautica, Vol. 53, 2003. Pg. 709-717.
- [22] Bogar T.J., Bangham M.E., Forward R.L., Lewis M.J.; "Hypersonic airplane space tether orbital launch (HASTOL) system: interim study results", AIAA-99-4802, 9th International Space Planes and Hypersonic Systems and Technologies Conference, November 1999.
- [23] Advanced Propulsion Concepts; "Launcher Technology", Jet Propulsion Laboratory.
<http://sec353.jpl.nasa.gov/apc/catapults/00.htm>, Accessed on Dec 3, 2003
- [24] Knowlen C.; "RAM accelerator overview", University of Washington – Department of Aeronautics and Astronautics;
<http://www.aero.washington.edu/AERP/ramac/detailed/abstract.html>, Accessed on May 5, 2005.
- [25] Space Studies Institute, "SSI Research", SSI.org; <http://www.ssi.org/research.html>, Accessed on May 23, 2005.
- [26] Tidman, D.A.; "Constant-frequency hypervelocity slings", Journal of Propulsion and Power, Vol. 19, No. 4, July-August 2003. pg. 581-587.
- [27] NASA: "1977 Summer Study Mass Driver III Engineering", Ames Research Center, Moffett Field, California.
<http://www.nas.nasa.gov/Services/Education/SpaceSettlement/spaceres/III-3.html> ; Accessed on April 20, 2004.

- [28] Sorensen K.; "Momentum eXchange Electrodynamic Reboost (MXER) tether technology assessment group final report", In-Space Propulsion Technologies Program, NASA Marshall Space Flight Center, July 2003.
- [29] Tennessee Technological University; "TTU MXER research", Tennessee Technological University; <http://mxer.tntech.edu>, Accessed on April 4, 2005.
- [30] Global Security.org; "US Warships", Global Security.org; <http://www.globalsecurity.org/military/systems/ship/index.html>, Accessed on May 21, 2004.
- [31] Hoerner, S.F.; Fluid-Dynamic Drag; Hoerner Fluid Dynamics, Brick Town, New Jersey, 1965. Pg. 18-6.
- [32] Riley, W, Sturges, L.; Engineering Mechanics – Dynamics; Second Edition, John Wiley & Sons, New York, 1996.
- [33] Bray, D.; "E338 – Weapon Systems Course Notes", Cranfield University, United Kingdom, October 2003. Exterior Ballistics Section. - <http://www.rmcs.cranfield.ac.uk/acroextra/e338.htm> ; Accessed on April 29, 2004.
- [34] United States Committee on Extension to the Standard Atmosphere, "U.S. Standard Atmosphere, 1976", National Oceanic and Atmospheric Administration, National Aeronautics and Space Administration, United States Air Force, Washington D.C., 1976.
- [35] White, F.; Fluid Mechanics – Fourth Edition, McGraw-Hill Inc., Boston, 1999
- [36] MESTeel; "2003 World Crude Steel Production", MESTeel; http://www.mesteel.com/cgi-bin/w3-msql/goto.htm?url=http://www.mesteel.com/info/world_cs_prod.htm, Accessed June 24, 2004.
- [37] Pollock W.; "World Electric Power Production" Encarta encyclopedia; http://encarta.msn.com/text_761566999_1/Electric_Power_Systems.html; – Accessed on July 9, 2005
- [38] IDEO; "Prototyping"; IDEO; <http://www.ideo.com/about/methods/info.asp?x=3> ; accessed on August 12, 2004.
- [39] Sutton, G.P., Biblarz, O.; Rocket Propulsion Elements, Seventh Edition; John Wiley & Sons Inc., New York, 2001.
- [40] Hucho, W.H.; Aerodynamics of Road Vehicles: From Fluid Mechanics to Vehicle Engineering; Fourth Edition; Society of Automotive Engineers; Warrendale, PA, 1998.
- [41] Raibov, V., Yegorov, I.V., Yegorova, M.V.; "Analysis of hypersonic viscous flow about cylinders placed one after another": American Institute of Aeronautics and Astronautics, AIAA- 98-0171, 36th Aerospace Sciences Meeting & Exhibit, January 1998.
- [42] Boctor, S., Ryff, P, Hiscocks, P., Ghorab, M., Holmes, M.; Electrical Concepts and Applications; West Publishing; New York, 1997.

- [43] Lu, F., Marren, D.; Advanced Hypersonic Test Facilities; Progress in Astronautics and Aeronautics, Volume 198; American Institute of Aeronautics and Astronautics Inc., Beston Virginia, 2002.
- [44] Fieseler, P.D; “A method for solar sailing in low Earth orbit”; Acta Astronautica, Vol. 43, No. 9-10, 1998; pg. 531-541.
- [45] Hoyt, R.P.; “Design and simulation of a tether boost facility for LEO to GTO transport”, American Institute of Aeronautics and Astronautics; AIAA 2000-3866, July 2000.
- [46] Salustri, F.A.; Personal conversation, Ryerson University, June 2004.
- [47] SourceForge: “SourceForge - Hosting Requirements”, SourceForge.net; http://sourceforge.net/docman/display_doc.php?docid=14027&group_id=1 accessed on July 27, 2005.
- [48] Gerl, D., Salustri, F.A.; “Design of a mass driver system concept using open source principles”; 55th International Astronautical Congress; IAC-04-U107, October 2004.
- [49] Space Science Institute; “SSI: History”; Space Science Institute – Princeton University; <http://www.ssi.org>; Accessed on 17 July, 2005.
- [50] Mad Scientist Network; “About the Mad Scientist Network” , Mad Scientist Network – Washington University; <http://www.madsci.org/info/intro.html>, Accessed on July 17, 2005.
- [51] Furniss, T.; “Sharp shooter aims for space”, Professional Engineering, March 1993 pg. 14.
- [52] Hull, J.R., Carney, L.M.; “Application of superconducting technology to earth-to-orbit electromagnetic launch systems”, IEEE Transactions on Magnetics; Vol. 25, No.1, January 1989. pg. 243-248.
- [53] Brown, J.L., Jamison, K.A., Johnson, N.E., Rolander, G.E., Scanlon, J.J.; “Earth-to-orbit railgun launcher”; IEEE Transactions on Magnetics; Vol. 29, No.1, January 1993. pg. 373-378.
- [54] Kolm, H., Mongeau, P., Williams, F.; “Electromagnetic launchers”; IEEE Transactions on Magnetics; Vol. MAG-16, No.5, September 1980. pg. 719-721.

GLOSSARY

Apogee – The trajectory point that is furthest from the centre of the orbiting body.

CatSat – ‘Catcher’ Satellite is the orbiting component of the Material package Stream Transport System (MPSTS) that collects the Material Packages (MP)s in orbit.

Direct Space Launch – a means of launching a body from Earth surface such that the vehicle is imparted with the momentum required to reach orbital altitudes without an onboard propulsion system.

DOLR – Distributed Oxidizer Launch Rail is the modified launched rail concept proposed by Dean Gerl.

DPLR – Distributed Propellant Launch Rail is Professor S. Molder’s externally fuelled rocket sled concept.

GEO-SYNC – Geosynchronous Earth Orbit = 35 790 km altitude

GTO – Geo-synchronous Transfer Orbit

Lagrange points - are five locations in space where the Earth and Moon gravity forces are equal and opposite in direction, therefore a mass located at a Lagrange point is stationary relative to the Earth and Moon.

LEO – Low Earth Orbit $\approx$ 200 km – 1000 km altitude

MP – Material Package is the launch body of the proposed mass driver material package transport system.

MPSTS – Material Package Stream Transport System (also known as OpenMDS) is the direct space launch concept proposed by Mr. Ed Panzeria and the subject for design and evaluation of this thesis. The proposed benefit of the concept is aerodynamic drag is minimized by launching a series of small mass to exploit the slip-stream effect.

OpenMDS – Open Mass Driver System (also known as the MPSTS) is the direct space launch concept proposed by Mr. Ed Panzeria and the subject for design and evaluation of this thesis.

OSS – Open Source Software is any software product that has the original source code for software available to the public for use and/or modification at no cost.

Perigee – The trajectory point that is nearest to the centre of the orbiting body.